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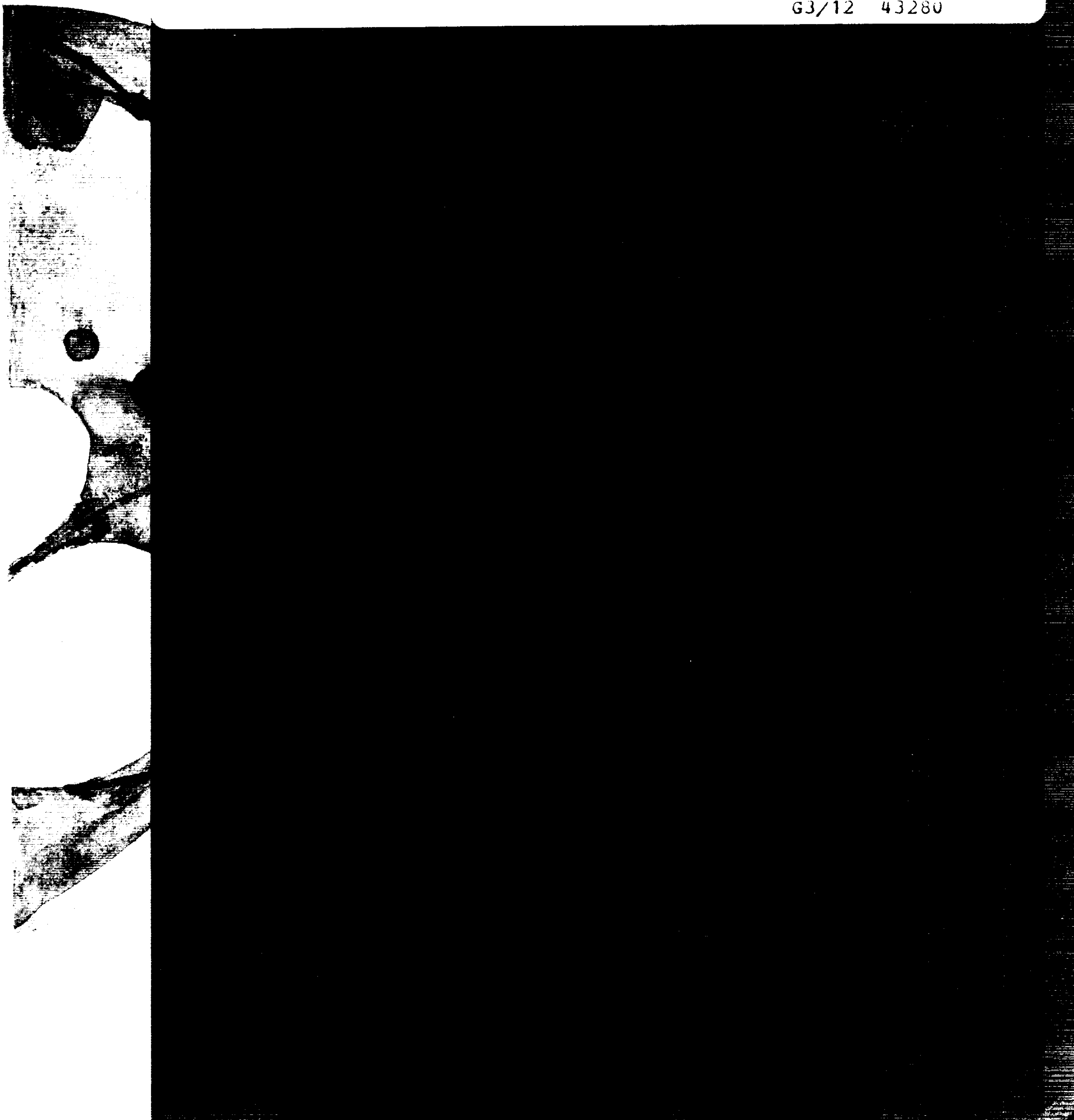

MANNED MARS MISSIONS
WORKING GROUP PAPERS

A WORKSHOP AT
MARSHALL SPACE FLIGHT CENTER
HUNTSVILLE, ALABAMA

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PREFACE

In 1984, three important factors modified the NASA planning environment. That year the Space Shuttle became operational, the Space Station program received strong presidential support, and Congress mandated the creation of a National Commission on Space to survey the space program and recommend future strategies and missions. In this environment, a study of manned Mars missions was initiated at the suggestion of former astronaut, H. H. Schmitt.

A study of approximately five (5) months' duration was undertaken by NASA centers and the Los Alamos National Laboratory (LANL), assisted by a few experts from university and other governmental organizations. The purposes were to update earlier Mars missions study data, to examine the impact of new and emerging technologies on Mars mission capabilities, and to identify technological issues that would be useful in projecting scientific and engineering research in the coming decades. In the first half of 1985, the study team held meetings at Los Alamos National Laboratory, Johnson Space Center, Kennedy Space Center, and Marshall Space Flight Center. Michael Duke served as Chairman of the steering committee for the study, with membership consisting of representatives from NASA centers and LANL (including H. H. Schmitt as a consultant). Barney Roberts provided study coordination and integration.

The final meeting was held at the Marshall Space Flight Center (MSFC), June 10-14, 1985, as a workshop entitled "Manned Mars Missions." A few additional outside experts participated in the workshop, and a total of over 90 invited and contributed papers were presented there. This report contains papers from the workshop. The papers and authors are listed in the Table of Contents; the authors are listed alphabetically, along with their organizational affiliations, in Appendix A.

The papers were grouped into nine (9) sections at the workshop, and the same grouping format has been followed in this report. Each section had an editor who was responsible for a major part of the editing process. The section and editors were: Rationale, Michael Duke; Transportation Trades and Issues, Barney Roberts; Mission and Configuration Concepts, John Butler; Surface Infrastructure, James Blacic; Science Investigations and Issues, Paul Keaton; Life Science/Medical Issues, Joseph Sharp; Subsystems and Technology Development Requirements, James French; Political and Economic Issues, Kelley Cyr; and Impact on Other Programs, Barbara Askins. Overall editing of the report was done by John Butler and S. T. Wu. MSFC and personnel of the University of Alabama in Huntsville hosted the workshop and provided logistics support for the report.

Some of the data provided herein may have become slightly outdated since the workshop. This is probably more likely to be the case for some of the data on the assumed "then-existing infrastructure" for the timeframe of the manned Mars missions, since the activities from which such data were obtained are on-going and dynamic processes. Most notable of such cases might be the Space Station data, and in particular, its configuration. However, it is believed that such changes would not significantly alter the concepts and conclusions presented in this report.

Many unanswered questions remain, and much work must yet be done in many areas. It is hoped that this report might provide a basis and a stimulus for furthering this process.

A summary report has been published separately as NASA Report M001, Manned Mars Missions Working Group Summary Report, May 1986.

SECTION I

RATIONALE

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SCIENCE OBJECTIVES

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ABSTRACT

This paper traces briefly some of the more significant findings about Mars since its discovery. It discusses the key Mars science objectives, such as the biological, planetological and climatological objectives, and the history of Mars' interaction with the external space environment. It then discusses the types of measurements required to accomplish these objectives.

INTRODUCTION

Mars has long been an object of fascination for Mankind. Its red color makes it readily identifiable in the night sky and to the ancients the planet came to symbolize the carnage and destruction of war. More recently, the fascination of Mars stemmed mainly from the possibility that the planet might harbor life. For three centuries following the invention of the telescope in the early 1600's Mars was perceived as Earthlike. It was thought to have oceans and continents, weather patterns similar to those here on Earth, and prolific vegetation. A major change took place in 1877 when the Italian astronomer Schiaparelli published maps showing linear markings or canals. Subsequently, observers all over the world strained to see the markings, and drew even more elaborate maps of the canal system. Speculation focussed on the possibility that Mars might have intelligent life, and on the liklihood that the planet-wide network had been built by an advanced civilization trying to survive in face of progressive dessication of their planet.

The perception of Mars as the planet with canals persisted until 1965 when the Mariner 4 spacecraft gave us the first close-up view. It revealed an ancient cratered surface, somewhat like the Moon's, but of the canals, there was no sign. They appear to have been an imaginary perception of the surface gained by observers who were straining to view the features at the limit of telescopic resolution. Two subsequent missions to the planet in 1969 also sent back pictures that resembled those of the Moon. However, Mars continued to surprise us, for when the Mariner 9 spacecraft started to systematically map the surface in 1972 it revealed a very non-Moonlike planet, one with huge active volcanoes, deep canyons, and enormous dry river beds, a planet periodically engulfed in vast dust storms, subject to varied weather patterns and having experienced long-term climatic changes. The previous missions had all presented a misleading view by fortuitously passing over the most ancient and most Moon-like parts of the surface.

The most recent episode of martian exploration was the landing of two Viking spacecraft on the surface in 1976 and the monitoring of activity around the landing sites during the succeeding four years. The main purpose of the Viking mission was to search for life. While none was detected at the two sites sampled, the Viking mission returned valuable information on the peculiar chemistry of the Martian soil, confirmed and added to the impression of geologic variety, and sent back new evidence for less severe climatic conditions in the past.

MANNED EXPLORATION

The main motivation for manned exploration of Mars is not scientific, although science is a major beneficiary. The exploration of space is one of those vase inexorable movements of the human race, like the westward expansion of the United States. It is our manifest destiny. We will explore space for the same reasons that Scott and Amundsen raced to the south pole, and Hillary climbed Mt. Everest. Space is the remaining unconquered frontier. The planets will ultimately be explored, and Mars will almost certainly be the first. It is the most hospitable and one of the easiest to get to. The only uncertainity is the timing.

Because of the long communication links, the exploration of of Mars presents problems not encounted on the Moon. The round trip communication length to Earth can be as long as 40 minutes. Thus exploration by unmanned vehicles presents severe logistical problems. Traverses of any significant length will require a high degree of automation to avoid hazards. In order to assess potential dangers and scientific opportunities, the vehicle will be required to pause repeatedly as new information is relayed back to Earth, digested, then commands returned to tell the vehicle how to proceed. Progress will be hesitant and time-consuming. Many of these problems will be solved by having astronauts present. However, in order to capitalize on their presence, the astronauts would have to be trained to make independent science judgements since interaction between humans on the surface and mission control back on Earth will be ponderously slow. Moreover, staytimes at the planet are likely to be measured in months or years rather than days thereby allowing far more detailed and varied science than was possible during the Apollo missions.

SCIENCE OBJECTIVES

Although science will not be the primary motivation for going to Mars, a manned mission to Mars will have several major science objectives. These can be conveniently categorized as follows:

Biological Objectives

Although the Viking landers failed to detect living organisms or any complex organic matter, many biologists think that these results are inconclulsive in that only two locations were sampled and neither was optimum for sustaining life. A primary objective of a manned Mars mission will therefore be to extend the search for life to more appropriate locations, specifically UV-protected, water rich locations, possibly near volcanic fumaroeles or other energy sources.

Conditions in the Martian past may have been much more conducive to initiation of life than are present conditions. Three conditions currently mitigate against life--the lack of liquid water, the intense UV radiation at the surface, and lack of protection against solar flares. However, geologic evidence suggests that climatic conditions in Mars' distant past were sufficiently benign that water could flow across the surface. This observation together with the isotopic composition of the present atmosphere implies that the atmosphere was substantially thicker than the present one, and it may have provided significant UV protection. In addition, the interior of Mars was almost certainly hotter, possibly allowing circulation within the core, generation of a magnetic field, and so providing protection against solar flares. Thus the three conditions that currently render Mars inhospitable may not have prevailed early in the planet's history.

Martian life forms, should any have developed, are likely to have been very primitive. It took billions of years for complex life forms to develop on Earth, and conditions hospitable for life probably persisted on Mars for no more than a billion years. The search would therefore be mainly for bacteria-like forms rather than macrofossils. Man could play a crucial role in this search by first being able to recognize potential host rocks, such as lacustrine sediments, and second, by being able to examine samples while the mission is in progress and so modify subsequent activities.

Many of the conditions necessary for growth and photosynthesis are present on Mars. Sunlight, carbon dioxide, appropriate minerals, and probably water are all readily available at the surface, and ultravoilet light can readily be filtered out. One major biologic objective will therefore be to assess the ability of terrestrial life forms to survive there. Such experiments would have a profound effect on subsequent Mars exploration by providing an indication of the degree to which Man could sustain himself at Mars, independently of resources brought from Earth.

Planetological Objectives

Theories about how the planets formed are based largely on the Earth and the Moon, which formed in the same part of the solar system, and on meteorites, whose place of formation is unknown. Mars formed in a different part of the solar system from the Earth so it provides a means of testing different theories on the condensation and fractionation of materials within the early solar system, and their accumulation into the planets. One objective of any mission to Mars will be to refine our ideas on how the planets form by testing different theories against what is found on Mars.

We have little information on how Mars evolved to its present configuration after it formed. Major questions are: (1) When was the global dichotomy into uplands and plains established, and what caused the dichotomy? (2) When did the planet differentiate into crust mantle and core, assuming that it did? (3) What is the composition of the crust mantle and core? (4) How have temperatures in the interior changed with time? (5) What have been the principle mechanisms of heat dissipation? (6) What has been the history of volcanic and tectonic activity? (7) How has the structure and thickness of the lithosphere changed with time? A major objective of manned missions to Mars will therefore be to reconstruct the geologic history of the planet.

Climatological Objectives

Mars appears to have undergone both secular and periodic changes in its climate. Crucial to understanding these changes are the total volatile inventory of the planet, its outgassing history, and the history of fixation of the volatiles in sinks within the crust. We need to know where the volatiles are now, in what form they are, and how readily they can be exchanged with the atmosphere in response to changes. In addition we need to better understand the dynamics of the present atmosphere so that global circulation models (GCM) can be refined. Present models for the Earth are artifically forced to fit the observed rather narrow statistical climatic variations that occur on Earth, but they are suspect when used to predict major changes such as would result from a long term increase in the CO_2 content. A third major set of scientific objectives is thus to reconstruct the climatic history of the planet and better understand current atmospheric dynamics.

<u>History of Interaction with the External Space Environment</u>

It is generally assumed that the impact histories of the various bodies within the inner solar system are very similar. However, this has not yet been tested. One science goal is therefore to establish the impact history of Mars and compare it with that of the Moon, the only other body for which the history is reasonably well established. In addition, the energy output of the Sun may change periodically and in the long term, and evidence of such changes may be preserved in the various sedimentary stacks that occur on the surface.

MEASUREMENT REQUIREMENTS

The general objectives just outlined can be accomplished only by a broad based effort involving determination the internal structure of the planet, the detailed chemistry of its various components, and the history of redistribution of materials on the planet. Clues for answering any one question generally come from a variety of sources. Information on the thermal history of the interior, for example, is obtained from seismic data on the structure of the interior, the present heat flow, the chemistry of volcanic rocks, the geologic record of volcanic activity, the history of the magnetic field and so forth. The following discussion is accordingly organized around characteristics that have to be determined rather than the questions that need to be answered since knowledge of any one characteristic generally contributes toward answering a range of questions.

Geophysics

Very little is known about the internal structure of Mars. It is generally assumed that it is, like the Earth, divided into crust, mantle and core but their dimensions and composition are unknown. In addition the dynamics of the interior are unknown. The Earth's heat is lost largely through the action of plate tectonics. Mars has no plate tectonics and its volcanic activity is highly localized, being mostly in the Tharsis region. This suggests that the Mars mantle is thermally and, possibly chemically, inhomogeneous. Internal structure can be inferred from heat flow, seismic, and magnetic measurements, and from direct samples of the mantle as inclusions in volcanic rocks. Widely spaced arrays of geophysical instruments, possibly grouped in compact packages similar to the ALSEP concept during the Apollo missions, will be required to probe the interior.

Seismic Measurements

Passive seismometry involves monitoring the natural seismicity of the planet. The internal structure of the planet can be determined from its natural seismicity. Widely spaced seismic stations must be established, each equipped with both short-period and broad-band three-axis seismometers, and designed to last for many years. This global array should be supplemented by more dense arrays at locations where more intense seismic activity is expected, such as around the large volcanoes, and adjacent to the canyons, or where specific problems, such as the size and location of the magma chambers beneath the volcanoes, or variations in the thickness of the crust, need to be addressed.

Active seismometry involves detection of seismic signals artifically generated as with explosions or a "thumper". Such measurements provide information on near surface structures (up to a few km depending on the magnitude of the signal generated). These techniques could be used for measuring thicknesses of lava flows, detection of subsurface ice, detecting the base of the permafrost, determining the thickness of the polar layered deposits.

Heat Flow

Determination of heat flow is a first order requirement for assessing the thermal state of the interior. The measurements are difficult to make, requiring drilling, emplacement of sensors at various depths and monitoring of the temperature variations over a long period of time. In addition to determination of the equilibrium temperature gradient, the relaxation of the temperatures after perturbation by the drilling must be monitored in order to assess thermal conductivities. Heat flow measurements should be made in several different geologic environments such as deep within the ancient catered terrain and on the more recently active Tharsis ridge. While the main reason for making heat flow measurements is assessment of the thermal state of the interior, such measurements will also lead to an improved understanding of other factors such as the thickness of the permafrost and the absorptive capacity of the deep regolith for volatiles.

Magnetic measurements

Mars presently has no magnetic field or only a very small one. However, the planet may have had a stronger field in the past, which would have left a record in the remanent magnetism of igneous rocks. A prime objective will therefore be to take oriented samples of volcanic rocks with a wide array of ages and locations so that the history of the magnetic field can be reconstructed.

Sample Analysis

Chemical and mineralogical analysis of primary igneous rocks will be the main source of information on the geologic evolution of the planet. Analyses of primary igneous rock, for example, provide indications of when the rock formed, what conditions were at the depths where the magma originated, what the chemical composition of the source region was, what their fractionation history had been, the extent to which there had been mixing of mantle and crustal rocks at the time the magma formed, whether the magma was contaminated with rocks above the source regions in its passage to the surface, the nature of any such contaminants and a wide range of other factors. Similarly, analyses of altered rocks or sedimentary rocks will give indications of the past availability of water at and near the surface, the thickness and composition of the atmosphere at various times in the past, previous surface temperatures, the composition of near-surface waters, and so forth.

Extraction of all the information embedded in samples requires application of a wide variety of sophisticated analytical techniques. In the years following return of lunar samples, a substantial fraction of geosciences' analytical and intellectual capabilities were used in their interpretation. Going to Mars is such a major endeavor that every effort must be made to maximize the science return on each mission. This will inevitably involve doing a substantial amount of analytical work at Mars, so that the sampling program can respond in an informed way to information that the samples contain. Sampling guided by the appearance of hand specimens, such as was done on the Moon, is very inefficient. Sampling should be an iterative process with the emphasis shifting as the meaning of each set of the results becomes better appreciated. Such an interactive sampling program has become more practical in recent years with the miniaturization of analytical instruments. It is also a practical goal for a manned Mars mission in that stay times of several months will provide time for analysis and interpretation while the mission is in progress.

Mars' geology is far more complex than the Moon's. The rocks are likely to have a wide spread of ages possibly from around 4 billion years ago up to the present. They are also likely to have a wide range of origin, including a variety of different kinds of igneous rocks, lake sediments, fluvial sediments, eolian debris, and glacial deposits. Sampling must be done in a informed way as so to include the widest range of possible types and ages, to avoid undue emphasis on highly fractionated rocks such as eolian sands or evaporites, and to recogize the kinds of rocks most likely to yield information on broad global scientific goals. Clearly such a program could only be conducted by trained scientists with substantial analytical support at Mars.

Geologic Analysis

An understanding of the planet's geology is an essential requisite for intelligent sampling and for interpretation of data from the samples. Rocks of a wide range of origin (igneous, impact, lacustrine, fluvial, glacial, eolian) are probably exposed at the martian surface. The sequence of events that led to the present configuration can be reconstructed and understood by determination of where and in what sequence the different rocks were laid down. Stratigraphic analysis and detailed characterization of sample location must therefore accompany the sample acquistion. Determination of vertical sections through the crust is particularly important. Such sections are accessible in a variety of locations such as canyon walls, channel walls, and escarpments around volcanoes and along the plains/upland boundary. To ensure optimum samples will require careful traverse planning, partial sample analysis en route, the ability to adapt new findings and the capability of obtaining drill cores where appropriate.

Climatology

To improve our knowledge of the general atmospheric circulation, a global network of weather stations needs to be established. Each station should be capable of monitoring vertical profiles of temperatures, pressure, water content, dust content, wind directions and magnitudes, and composition. If possible, such local monitoring should be supplemented by satellite observations to provide global sounding, monitoring of the global cloud patterns, and following of the advance and retreat of the ice caps. The weather stations would serve not only scientific purposes but could also be used to warn those present on the surface of potentially hazardous conditions such as dust storms.

Clues of past climates on Mars will be provided by deposits that are the result of climate sensitive processes such as weathering, and the action of wind, water and ice. The most obvious example of climate sensitive deposits are the stack of layered deposits at the poles. The deposits are believed to be mixtures of dust and ice that record climatic changes in the recent geologic past. Although these deposits are relatively young, their precise age is unknown, and they could be as old as a few hundred million years. They are therefore somewhat analogous to continental ice sheets on Earth. Vertical sections through the deposits are well exposed in valleys that spiral out from both the poles. Climatic changes should be recorded in the variations in composition and lithology, so sampling of the layered deposits should lead to elucidation of climatic changes in the recent geologic past. Climatic conditions in the distant past will be more difficult to assess. However, clues as to past climates will be provided by the record of weathering, soil formation, eolian activity, and the action of water and ice. Interpretation of the record will require both field interpretation and detailed sample analysis.

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THE POLITICS OF MARS

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ABSTRACT

This paper provides a discussion comparing past and present major accomplishments of the U.S. and the Soviet Union in space. It concludes that the Soviets are presently well ahead of the U.S. in several specific aspects of space accomplishment and speculates that the Soviet strategy is directed towards sending a man to the vicinity of Mars by the end of this century. The paper briefly reviews a major successful multinational space endeavor--INTELSAT--and suggests that the manned exploration of Mars offers a unique opportunity for another such major international cooperative effort. The paper assesses the current attitude of U.S. leadership and the general public as uniformed or ambivalent about the perceived threat of Soviet dominance in space.

INTRODUCTION

As we approach the turn of the Third Millennium, the rate at which the Soviet Union is creating new space capabilities is three to four times that of the United States. These capabilities include those necessary to put cosmonauts in the vicinity of Mars by the year 2000 as well as those necessary to dominate human activities in near-Earth space. This looming dominance must be countered in order to preserve the scientific, economic and political competitiveness of the free world. A national and, if possible, international program to explore and settle Mars is required as the focus of a long-term commitment by the United States to space stations, lunar bases and the human settlement of space.

The last quarter century has witnessed three key events in the evolution of the human species into space. These events mark both physical and political milestones in that evolution. Although discussed below in a different order, the events are, chronologically: August 20, 1964, the signing of the INTELSAT agreements; December 24, 1968, the entry into lunar orbit by Apollo 8; and July 20, 1969, the landing on the Moon by Apollo 11. Other events, such as those marking early human flights in Earth orbit, were important in and of themselves, but were in reality a continuation of many steps that led to these more fundamental events.

EVOLUTION OF THE HUMAN SPECIES

December 24, 1968. Human evolution, rapidly enhanced by modern technology resulting from that evolution, made the terrestrial planets an accessible and survivable part of human kind's sphere of activity. The commitment of the Apollo 8 spacecraft and its crew to an orbit around the Moon marked the modern culmination of the evolution of the human mind and body. With great confidence, but without an absolute guarantee of return, members of the species were committed to a planetary environment entirely different from that in which the species had evolved. From that time on, many of the planetary shores of the solar system's sea came to fall psychologically and technically within the envelope of potential human activities.

How humankind will utilize this new evolutionary status is not yet clear, however, it is clear that many of the young people of the Earth with whom I have spoken believe that the next great human adventure will take place at the space frontier, and that the planet Mars will be the focus of that adventure. There are strong indications that the growth of human politics and emotions, the advance of space technology, and the increase in understanding of human physiology are such that this adventure will begin around, or soon after, the turn of the Third Millenium: the year 2000 A.D.

This "tide in the affairs of men" is the ultimate and inevitable rationale for the exploration and settlement of Mars. This tide will be "taken at the flood" and "will lead on to fortune" for those who recognize it[1].

EVOLUTION OF FREEDOM

July 20, 1968. The evolution of human freedom reached the surface of the Moon as the United States of America placed the flag of that nation at Tranquility Base. The crew of Apollo 11, representing 500,000 Americans motivated by the belief that this was the most significant contribution they would make with their lives, established the beginnings of a tradition of freedom in the solar system sea and on its planetary islands. When faced with a modern challenge of uncertain dimensions from the Soviet Union, these men and women demonstrated, once again, the psychological and technological power of freedom to act on behalf of humankind.

As has always been the case, to the great suffering of vast numbers of human beings, the forces of freedom have slept between great challenges. They are aroused only when once again clearly threatened. While asleep, these forces have been nourished frontiers of exploration and settlement, enterprise and industry, intellect and science, and compassion.

Today, the forces of freedom are dozing off. Neither the threat of dominance by the forces of oppression nor the opportunities of the space frontier have yet significantly disturbed their rest. However, as was the case half a millenium ago in the New World, the political imperative to compete in a new arena is clear. Mars has become the focus of that competition whether or not the political leadership of the United States and the Free World currently choose to recognize this fact.

INTERNATIONAL THREAT

December 24, 1968. With Apollo 8 in orbit around the Moon, the leadership of the Soviet Union began the process of developing a strategy to become the politically dominant power in the solar system sea. The presence of American astronauts around the Moon meant the "Moon Race" was over. The Soviet leadership was embarrassed. Having challenged the United States and its society to the race, and having reaped the heady political and technical benefits of Sputnik and Gagarin, the Soviets found they were not yet a match for the aroused emotions, technology, and industry of Americans. Americans were already orbiting the Moon. There was not much political benefit to being second after having before tasted the sweet wine of being first.

With this bitter lesson understood, I strongly suspect a strategy was devised along several lines. First, continue to publicly emphasize Soviet activity in near-Earth space that would divert the primary attention of the U.S. toward civilian space stations. Such Soviet activity incidentally would lead to the development of capabilities supportive of military dominance in this arena.

Second, provide conflicting public information (or disinformation) about Soviet interest in the Moon, in Mars, and in human exploration of deep space in order to dilute the competive instincts of Americans.

Finally, undertake the deliberate step by step development of the technical capabilities to put cosmonauts in the vicinity of Mars by the end of the 20th Century and, preferably, at a time tied politically to 1992. This year will herald the 75th anniversary of the Bolshevik Revolution and, in a perverse twist of history, the 500th anniversary of the discovery of America by Columbus.

If this is the Soviet strategy, it has been implemented well. Look at the evidence:

The only large U.S. civil space program is the Space Station and even its development is being stretched out into the mid 1990's, if then, due to the lack of Executive and Congressional will.

The Soviets are rapidly approaching a permanent human presence in near-Earth space and are accumulating experience in manned spaceflight at a rate far in excess of that of the U.S. (3700 man-days in space versus 1300 for the U.S. as of mid-1985).

The Soviets' capabilities for direct tactical and strategic defense action in and from space exist and are increasing rapidly. The U.S. has no such capabilities and has made no firmly funded commitment to create them.

The Soviets are on the verge of testing a sophisticated heavy-lift launch vehicle, possibly larger than the Saturn V[2]. It is of the class that can support the Earth-orbital construction and launch of a manned Mars spacecraft as well as a rapid expansion of their space station and strategic defense systems. This activity is supported by the construction of several new launch facilities which will greatly extend their already impressively high rate of space launches.

The Soviets are developing and assimilating the technologies necessary for successful manned interplanetary flight, including those for life support, spacecraft maintenance, deep space navigation and scientific activities[3]. One also must assume that they picked up and matured the cancelled U.S. space nuclear program.

The Soviets have, most significantly, extended their tests of human physiological and psychological adaptation to long duration space flight beyond times necessary or desirable for the efficient operation of space stations. These times are steadily approaching the 250 days required for most one-way flights to Mars.

In short, the Soviets are creating new capabilities related to space in general and Mars in particular at a rate many times that of the United

States. For all intents and purposes, as it did in the 1950's, the U.S. is once again standing still in a much expanded and much more critical space race.

INTERNATIONAL OPPORTUNITY

August 20, 1964. One hundred and nine nations began a unique experiment in international cooperation when the INTELSAT agreement was signed[4]. Through this new entrant on the scene of international organizations, these nations, now one hundred and nine strong, agreed to share both the benefits and responsibilities of managing the technology and opportunities of international telecommunications satellites. This experiment has worked.

The human and technical opportunities that will come with sailing the solar system sea, as well as the political threat posed by the Soviet Union, encompass an even more remarkable opportunity for international cooperation. The turn of the Third Millenium presents an increasingly responsive environment for young men and women from all nations to join in an enterprise unique to our times: a project to establish a permanent human outpost on Mars by the end of the first decade of the new Millenium.

The essential ingredient of such a project is an unequivocal commitment by the United States to undertake the project with or without international cooperation.

With such a commitment, cooperation will follow. Astronauts and cosmonauts from all nations can join hands in this evolutionary and potentially moderating leap into a bright and exciting future.

Without such a commitment, efforts toward cooperative ventures in space will shift from those based on the collaboration of independent peoples to those based on a dominance of Soviet culture and technology.

The unequivocal commitment to this Millenium Project, which is required of the United States, will not come about under present circumstances. Due to the failure of most of our national decision-makers to comprehend either the opportunity or the threat, and the failure of the national media to adequately and regularly report about space, the spectrum of tangible and historical benefits coming from the space frontier goes largely unperceived by the American public. Although excited and occasionally entertained by major events or mishaps in space, the American public is ambivalent about space as a significant arena for national commitment[5]. When the American public is ambivalent about anything, modern political decision-makers know that they do not have to make commitments. In such an environment, statesmanship becomes an increasingly rare commodity. Past political history would indicate that the unequivocal commitment of the United States to participating in human and political evolution in space depends on the development of an interested, informed, and active public constituency: a constituency every decision-maker will see when looking over his or her political shoulder.

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S E C T I O N   I I

TRANSPORTATION TRADES AND ISSUES

N87-17724

AEROBRAKING

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ABSTRACT

This paper presents a discussion of the basic principles of aerobraking. Typical results are given for the application of aerobraking to orbital capture at Mars, descent to the Mars surface and orbital capture on return to Earth.

AEROBRAKING

Introduction

Aerobraking is the use of a planet's atmosphere to dissipate an entry vehicle's orbital energy to achieve a new orbital state or to descend to the planet's surface.

Numerous planetary descents have been successfully executed; however, aerobraking to a new orbit has not been attempted. A reason for this lack of attempts is that is was believed to be extremely difficult, if not impossible. With recent technology advances, aerobraking is still considered difficult, but it is more promising as a useful technology for space missions.

Many parameters with complex interactions must be considered with design of aerobraking systems and it is difficult to say which are the more important. An iterated approach is used in defining complex algorithms to achieve aerobraking trajectories.

Entry State

The entry state is one of the more important factors. The range of acceptable entry states leading to a successful braking is very limited and is nominally set after a study of the factors shown in Figure 1.1.

The basic parameters of entry state are time, latitude, longitude, altitude, velocity azimuth and flight path angle, the entry vehicle's aerodynamic characteristics and physical constraints (atmospheric structure).

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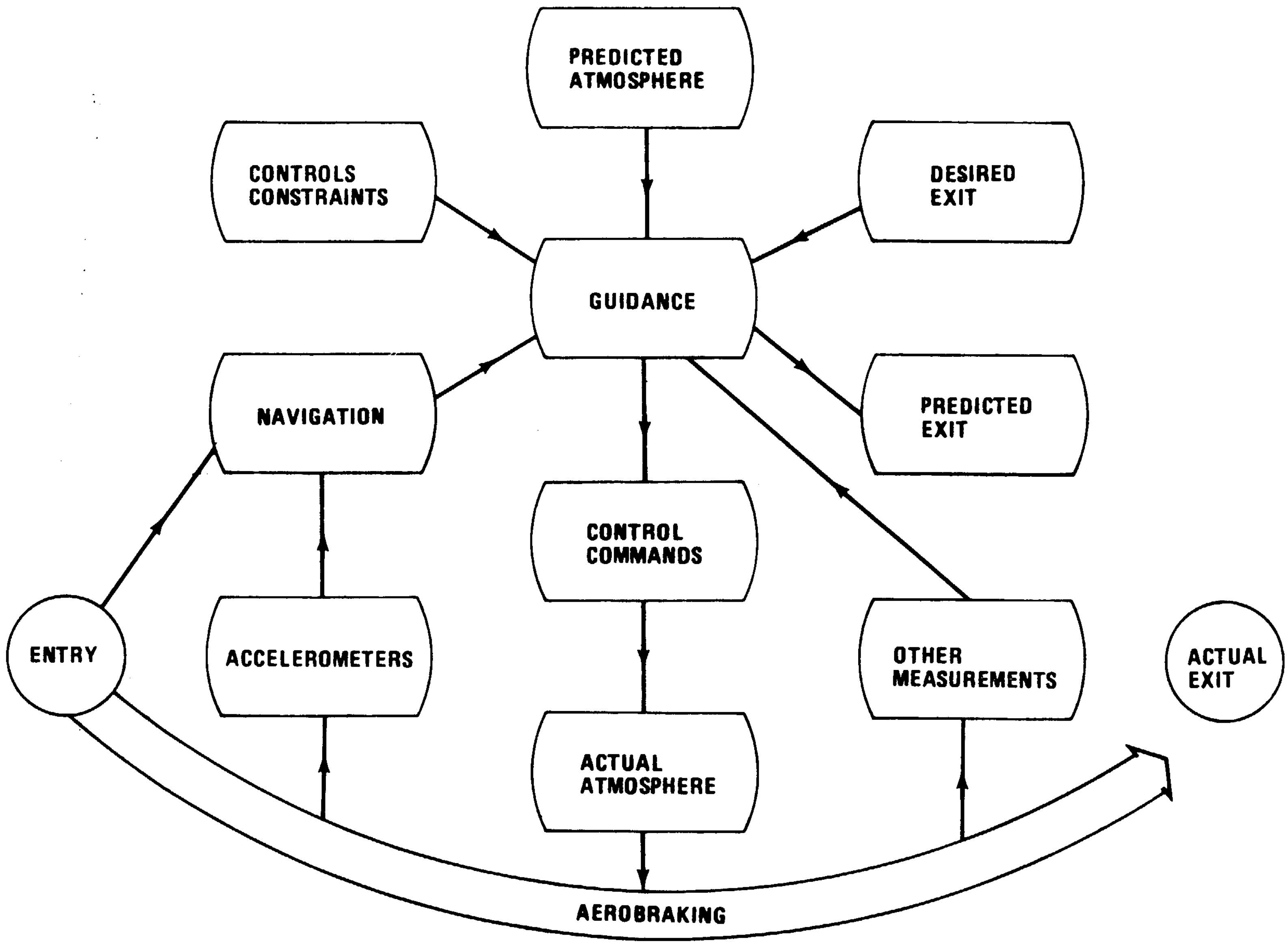

FIGURE 1.1 AEROBRAKING SCHEMATIC

The kinetic and potential energy per unit mass (E) of a vehicle on entry to the atmosphere is expressed as:

$$E = V^2/2 - \mu/R$$

where V is the entry velocity, and R is the radius with respect to the planet's center, and μ is the gravitational constant.

Keplerian equations can be used to calculate the entry orbit apogee, perigee, and mean motion. A time of passage from entry to exit (without an atmosphere) can be calculated. This is a lower bound on the actual passage time. In a similar manner, an upper bound can be calculated from the exit state.

Perigee altitude is a major parameter. The actual perigee, in the atmosphere, will be very near this prediction; usually within two nautical miles. Most of the aerobraking will occur in this region. Atmosphere perturbations in this altitude range can have a very large effect on the trajectory.

Exit State

The exit state conditions are usually specified as an altitude leaving the atmosphere, a desired apogee, and in most cases, a desired flight plane. The other orbit parameters can be approximated if the semimajor axis is known. The actual trajectory perigee will be near the entry perigee, and a crude approximation for the exit orbit perigee will also be near the entry perigee. Then the exit apogee and perigee will define an eccentricity, a semimajor axis, the orbit's angular momentum, and energy level. From the energy equation, an approximate exit velocity can be determined.

Aerobraking Time Limits

Once the entry orbit is known and the exit orbit has been approximated, a lower and upper limit for the aerobraking passage time can be estimated. For aerobraking at Earth, the time in general will be between 3 to 12 minutes.

The aerodynamic characteristics of this vehicle, the vehicle's controls, the predicted atmosphere, the physical constraints and the desired exit conditions are used to design the nominal entry state, and therefore, the aerobraking time. The range of the controls limit the allowable perturbation about this nominal trajectory.

Aerobraking

The aerodynamic forces are the forces that accomplish aerobraking. These are derived from the atmosphere density the velocity with respect to the atmosphere, the angle of attack, the angle and direction of bank, the lift and drag coefficients, and the vehicle's aerodynamics area and weight. It must be emphasized that once an entry has commenced, the actual passage through the atmosphere is within a narrow corridor and a slight deviation up or down in altitude can change the exit apogee drastically. See Figure 1.2 for a graph and table of density changes with altitude.

TRAJECTORY DESIGN

Goals and Physical Constraints

The goals of aerobraking are mission dependent. In both the aerobraking at Mars and at Earth, the desired exit state is an orbit around the planet, with a specified apogee. Typically, the desired orbit must be compatible with that of a transfer vehicle to return to a space station or planetary surface. During the aerobraking phase, physical constraints of aerodynamic heating, aerodynamic pressure and deceleration must be observed.

The deceleration profile is generally bell shaped and follows the atmosphere density profile encountered in the trajectory down and back up through the atmosphere. An approximation for the average acceleration (a) can be obtained from:

$$\Delta V = V\ exit - V\ entry$$

$$a = \Delta V/(\text{time of passage})$$

The maximum is about two and one-half times the average. The dynamic pressure and heating rate profiles are also similar to the density profile. The dynamic pressure (P) is estimated by:

$$P = \rho V^2/2$$

where ρ , and V are the values near perigee.

The heating rate may be approximated by:

$$\dot{Q} = \frac{k}{\sqrt{R_n}} \left(\frac{\rho}{\rho_{SL}}\right)^{\frac{1}{2}} \left(\frac{V}{V_{ref}}\right)^{3.15}$$

where $\dot{Q}$ is heating rate, ρ is k, ρ_{SL} and V_{ref} are derived from those values in Reference 1, and are k = 17600., ρ_{SL} = .076474,

NEAR AEROBRAKING PERIGEE

ALTITUDE KM	DENSITY KGN/M3	RELATIVE RATIO
75	4.3×10^{-5}	2.1
80	2.0×10^{-5}	1.0
85	8.0×10^{-6}	.4

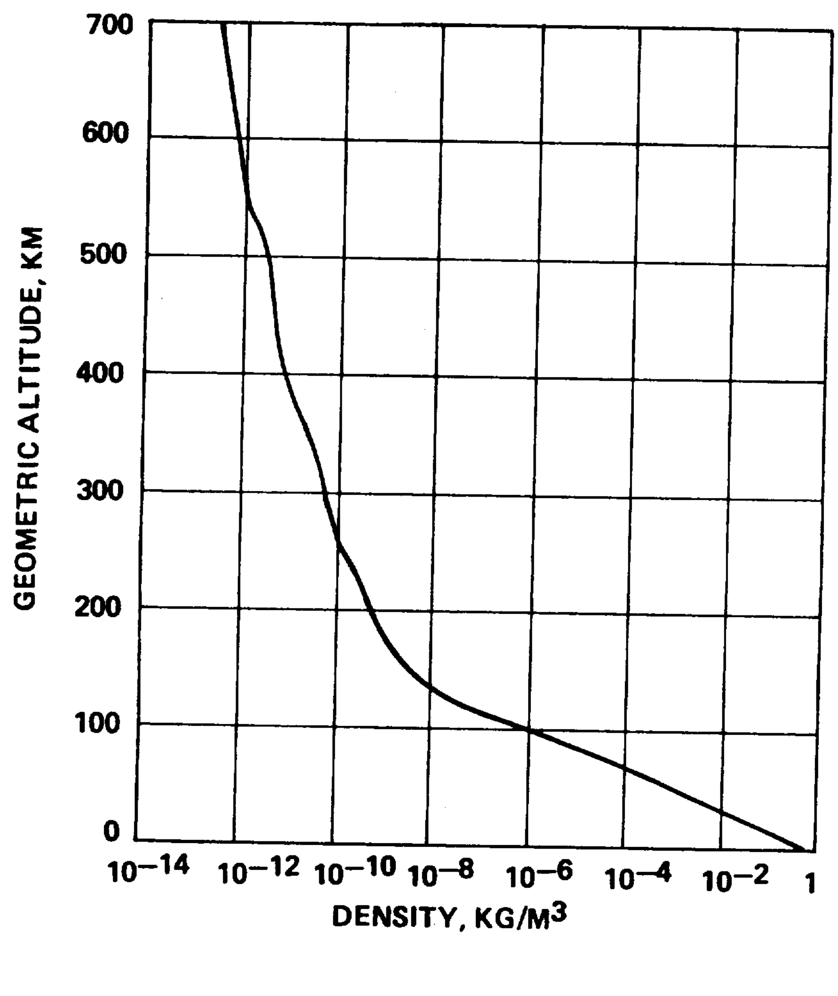

FIGURE 1.2 US62 STANDARD ATMOSPHERE

and V_{REF} = 26000 ft./sec. Limits to P and $\dot{Q}$ can be calculated from the entry orbit perigee velocity and the expected density at perigee.

Representative maximum design values are:

P 50 lbs/ft^2

and

$\dot{Q}$ 30 $BTU/ft^2/sec$ for a flexible TPS

$\dot{Q}$ 50 $BTU/ft^2/sec$ for fixed TPS

Guidance and Controls

Various guidance algorithms have been and are being investigated. See references 2 and 3. Among the algorithm's under study are: a predictor-corrector that guides to the desired apogee using a deceleration profile; a type which adds prediction of the apogee rate; types that utilize bank angle and also predict the final flight plane; types that use numerical integration of the equations of motion; and others that use closed form analytical approximations. All are designed after a consideration of the entry vehicle and it's aerodynamic characteristics and controls.

With the aerodynamic parameters, the direction of bank (L-R), the reversals of bank direction, reversal rates and reversal times (RRT) can be used as control candidates for the guidance algorithm. In designing an algorithm, three types of entry craft may be considered:

I. A variable area vehicle that can fly a deceleration profile but does not have any lateral plane control. Its ability to adjust to the desired deceleration profile is limited by the physical limits of its maximum and minimum area available. Current limits are less than a ratio of 2 to 1.

II. A fixed area vehicle, but with variable angle of attack, angle of bank and RRT. A typical example of this vehicle is the Space Shuttle. It can fly a predetermined profile within its control limits and flight plane control is achieved with the angle of bank and RRT.

III. A fixed area and angle of attack vehicle, with variable angle of bank and RRT. Since $C_D = C_D(\alpha, M)$ and α is fixed, it can only indirectly fly a deceleration profile. Lift must move the craft to a lower (higher) density region to affect drag. RRT does provide a measure of flight plane control.

All of these are feasible for both Martian and Earth aerobraking. The last concept is particularly interesting and is currently being investigated by personnel at MSFC, JSC, C.S. Draper Laboratories and others.

A simple numerical integration predictor corrector algorithm is being used at MSFC to obtain representative trajectories. It iterates the angle of bank, the reversals, and reversal times to obtain the desired exit apogee and flight plane. However, it is not a flight candidate as it takes too long to converge to acceptable values.

TYPICAL RESULTS

Figure 3.1 shows some of the features of the MSFC simple "bang-bang" algorithm for entry and capture at Mars and at Earth. In figure 3.2, representative graphs of altitude, velocity, density, dynamic pressure, acceleration and heating rates are given for a 3 reversal capture profile.

Mars Aerobraking Capture

Figure 3.3 and Table 3.1 present results obtained from a 14 reversal entry into the Martian atmosphere. The initial entry is in a medium to high energy, C_3 = 30 km^2/sec^2, approach orbit. The final orbit is a Molniya type orbit with a 24 hour period. Two assumed Martian atmospheres are given in Table 3.2.

Mars Descent

Results of a ballistic entry to the Martian surface are given in Table 3.3. No controls were assumed. Deboost at the apoapsis of the parking orbit described in Section 3.1 was assumed.

Earth Capture

Figure 3.4 and Table 3.4 give results from an entry into the Earth's atmosphere for capture. The initial orbit is a high energy, C_3 = 81 $km2/sec_2$, return orbit from Mars. If aerobraking were used with this high energy orbit, the peak deceleration would be in excess of 5g for over 2 minutes. Therefore, a braking burn 1 hour before entry is used to slow the entry craft. The final orbit shown is 10 nm above the Space Station orbit for rendezvous with an orbital transfer vehicle.

SUMMARY

Aerobraking to dissipate an entry craft's energy to achieve a new orbital state is difficult but possible. Aerobraking time from entry to

2984–85

- APOGEE CONTROL
  - ANGLE OF ATTACK α
    - FIXED
  - ANGLE OF BANK ϕ
    - ITERATED

- BANG–BANG PLANE GUIDE
  - PREDICTED PLANE ERROR δ
  - DEAD BAND LOGIC
    - INSIDE – HOLD DIRECTION
    - OUTSIDE – SET DIRECTION (L/R)

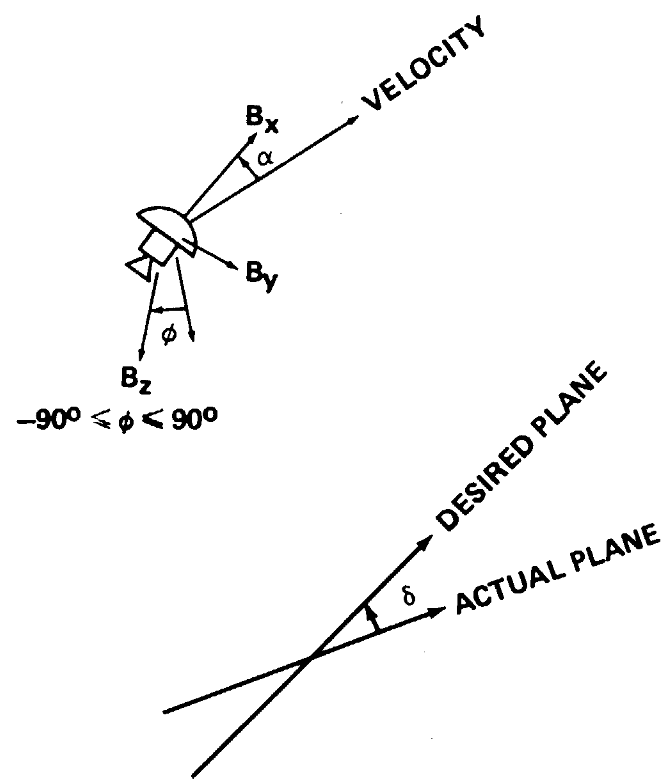

FIGURE 3.1 A SIMPLE GUIDE ALGORITHM

2988–85

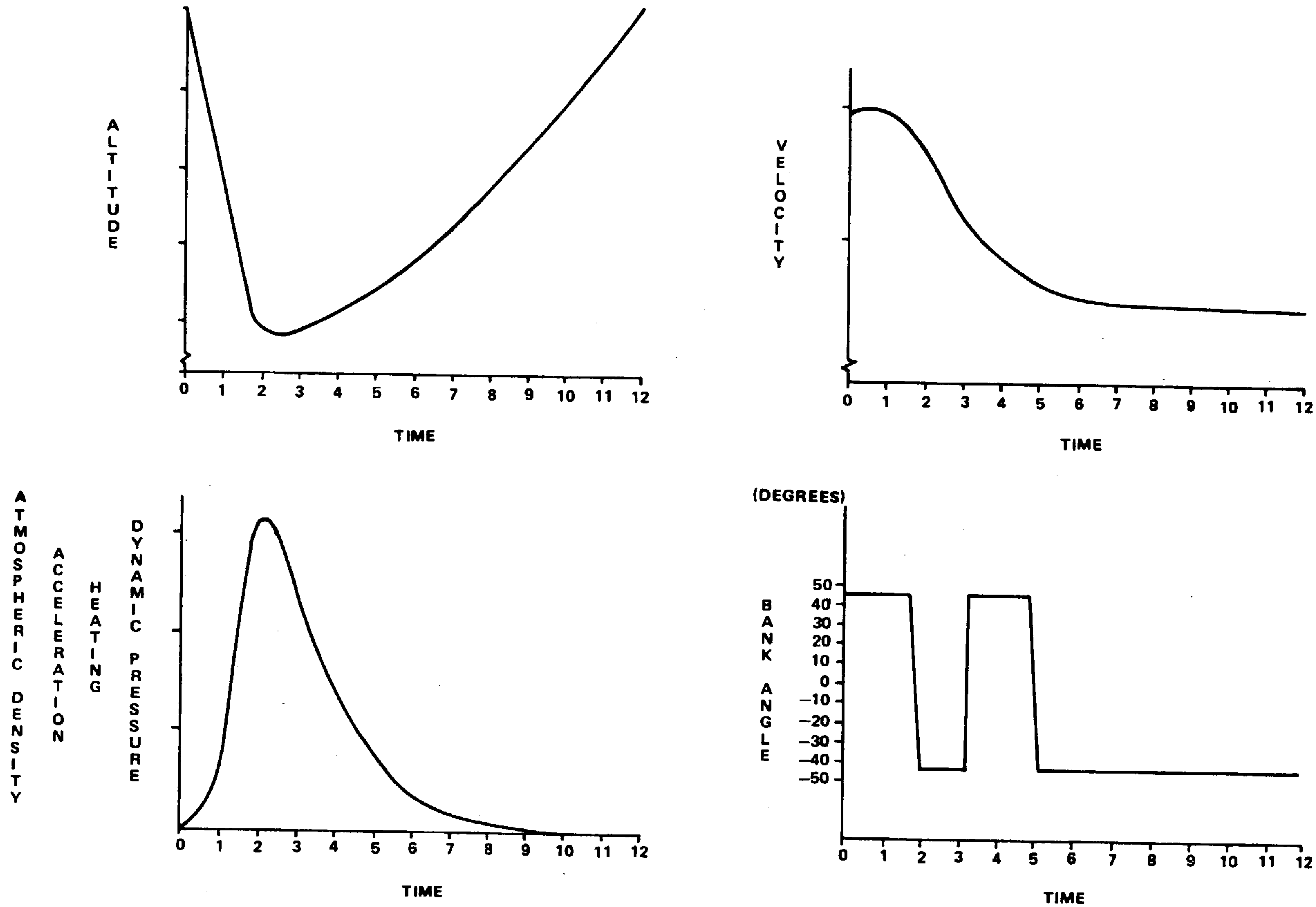

FIGURE 3.2 REPRESENTATIVE CAPTURE PROFILE

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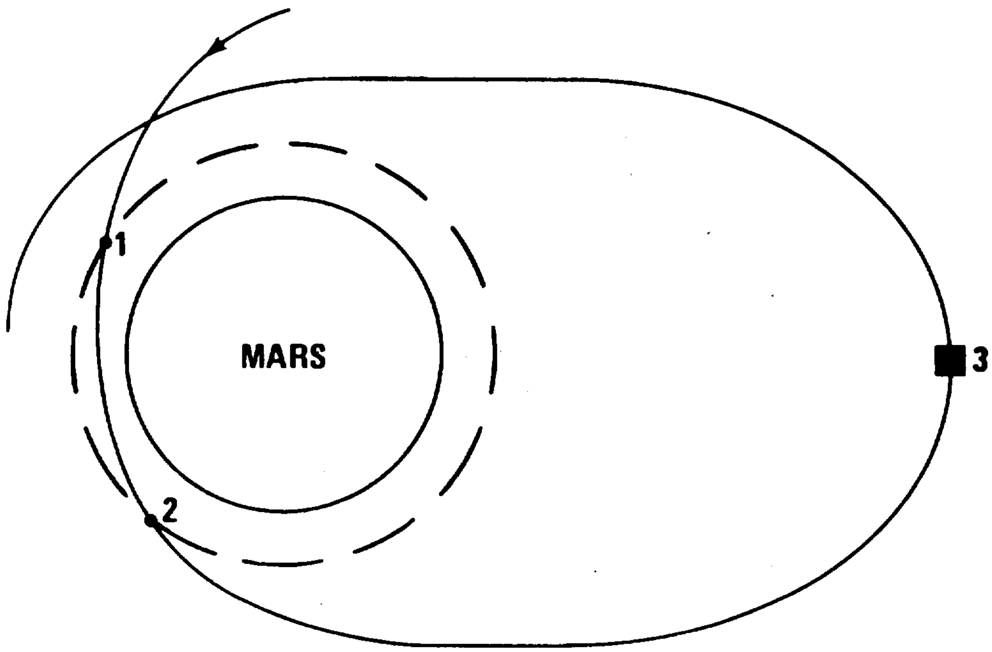


1. ENTER ATMOSPHERE V_R = 23422 FT/SEC

   C_3 = 30 KM^2/SEC^2

   PERIAPSIS = 24 NM

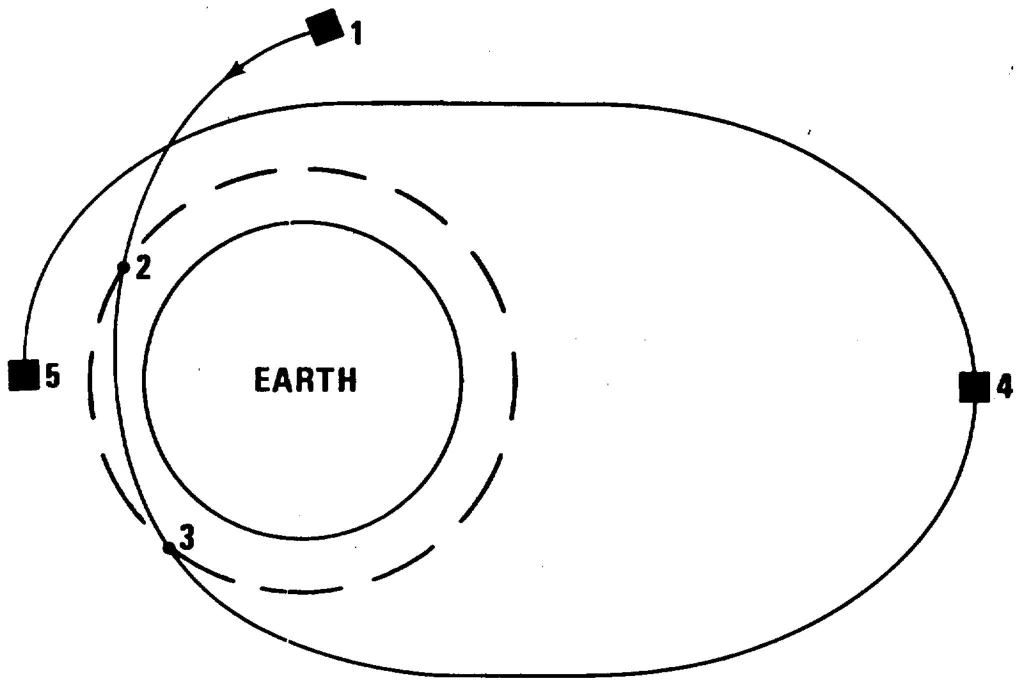
2. LEAVE ATMOSPHERE V_R = 14708 FT/SEC

   ORBIT 24 X 17814 NM

3. BURN TO RAISE PERIAPSIS ΔV = 85 FT/SEC

   ORBIT 268 X 17814 NM
   24 HOUR PERIOD

FIGURE 3.3 MARS AEROBRAKING CAPTURE



1. BRAKING BURN
   $C_3 = 81\ KM^2/SEC^2$

2. ENTER ATMOSPHERE — $V_R = 36297$ FT/SEC

   $C_3 = 9\ KM^2/SEC^2$

   PERIGEE = 45.2 NM

3. LEAVE ATMOSPHERE — $V_R = 24802$ FT/SEC

   ORBIT 44 X 350 NM

4. BURN TO RAISE PERIGEE — $\Delta V = 406$ FT/SEC

   ORBIT 280 X 350 NM

5. BURN TO CIRCULARIZE — $\Delta V = 118$ FT/SEC

   280 X 280 NM

FIGURE 3.4 EARTH AEROBRAKING CAPTURE

TABLE 3.1

MARS CAPTURE DATA

Entry Parameters	
Weight	415000 lbs
W/CDA	61 lbs/ft_2
Altitude	54 nm
Inertial Velocity	24225.7 ft/sec
Flight Path Angle	-9.1328 deg
Orbit C_3	30 km_2/sec_2
Inclination	1 deg
Periapsis	23 nm
Aerodynamic Parameters	
C_L	.405
C_D	1.35
Heat Shield	
Diameter	80 ft
Curvature	50 ft
Atmosphere	Mars Low Density
Controls - Bank Angle - Reversals - Times of Reversal	
Maxima	
Heating Rate	20.5 $BTU/ft_2/sec$
Dynamic Pressure	134 lbs/ft_2
Deceleration	2.4 g's
Orbit Leaving the Atmosphere	24 x 17814 nm
Time in the Atmosphere	380 sec
Apoapsis Burn to Raise Periapsis to	268 nm
ISP	482 sec
Propellant	2280 lbs
Delta - V	85.4 ft/sec
Final Orbit	268 x 17814 nm
Inclination	1 deg
Period	24 hours

TABLE 3.2

SUMMER, MID-LATITUDE, DAILY-MEAN ATMOSPHERE OF MARS
(COOL AND WARM MODELS)

	Cool, Low Pressure Model				Warm, High Pressure Model			
z, k	T. ^oK	p. mb	p. kg/m^2	P/P_o	T. ^oK	p. mb	p. kg/m^3	p/p_o
0	204	5.9	1.51×10^{-2}	1.000	224	7.8	1.82×10^{-2}	1.000
4	204	4.03	1.03	0.683	224	5.51	1.29	0.706
8	199	2.74	7.20×10^{-3}	0.464	219	4.09	9.77×10^{-3}	0.524
12	191	1.84	5.04	0.312	211	2.85	7.07	0.366
16	185	1.22	3.45	0.207	205	1.97	5.01	0.252
20	178	7.96×10^{-1}	2.34	0.135	198	1.34	3.54	0.172
24	173	5.13	1.55	8.70×10^{-2}	193	9.03×10^{-1}	2.45	0.116
28	168	3.27	1.02	5.54	188	6.03	1.68	7.73×10^{-2}
32	163	2.06	6.60×10^{-4}	3.49	183	3.99	1.14	5.11
36	158	1.28	4.23	2.17	178	2.61	7.67×10^{-4}	3.35
40	152	7.81×10^{-2}	2.69	1.32	172	1.69	5.13	2.16
44	148	4.70	1.66	7.96×10^{-3}	168	1.08	3.36	1.38
48	144	2.79	1.01	4.73	164	6.83×10^{-2}	2.18	8.75×10^{-3}
52	140	1.64	6.12×10^{-5}	2.78	160	4.28	1.40	5.48
56	137	9.49×10^{-3}	3.62	1.61	157	2.65	8.84×10^{-5}	3.40
60	134	5.44	2.12	9.22×10^{-4}	154	1.63	5.55	2.09
64	132	3.09	1.22	5.23	152	9.99×10^{-3}	3.44	1.28
68	130	1.74	7.01×10^{-6}	2.95	150	6.08	2.12	7.79×10^{-4}
72	129	9.76×10^{-4}	3.96	1.65	149	3.68	1.29	4.72
76	129	5.47	2.22	9.27×10^{-5}	149	2.23	7.83×10^{-6}	2.86
80	129	3.07	1.24	5.20	149	1.35	4.75	1.73
84	129	1.72	6.99×10^{-7}	2.92	149	8.21×10^{-4}	2.88	1.05
88	129	9.70×10^{-5}	3.93	1.64	149	4.99	1.75	6.39×10^{-5}
92	129	5.46	2.22	9.26×10^{-6}	149	3.03	1.07	3.89
96	129	3.08	1.25	5.22	149	1.85	6.49×10^{-7}	2.37
100	129	1.74	7.00×10^{-8}	2.95	149	1.13	3.96	1.44

TABLE 3.3

MARS DESCENT DATA

Deboost at Apoapsis (From Capture Orbit)	
Weight	135000 lbs
ISP	293 sec
Propellant	1228 lbs
Delta-V	85.4 ft/sec
Entry Parameters	
Weight	133770 lbs
W/CDA	45 lbs/ft_2
Altitude	54 nm
Inertial Velocity	15515.16 ft/sec
Flight Path Angle	-7.1518 deg
Orbit	22 x 17814 nm
Inclination	1.0 deg
Aerodynamic Parameters	
C_L	0
C_D	1.0
Heat Sheild Area	
Diameter	50 ft
Curvature	50 ft
Atmosphere	Mars Low Density
Controls - None - Ballistic Entry	
Maxima	
Heating Rate	4.4 $BTU/ft_2/sec$
Dynamic Pressure	64 lbs/ft_2
Deceleration	1.4 g's
Time to an altitude of 1 nm	593 sec
Velocity at 1 nm	1980 ft/sec

EARTH CAPTURE DATA
TABLE 3.4

Initial State	
Weight	40795 lbs
Altitude	17580.8 nm
Inertial Velocity	33049 ft/sec
Flight Path Angle	-76.3196 deg
Orbit C_3	81 km^2/sec^2
Inclination	28.5 deg
Perigee	65.8 nm
Braking Burn	(C_3 = 78 km^2/sec^2)
ISP	482 sec
Propellant	25795 lbs
Entry	
Weight	15000 lbs
W/C_DA	8.84 lb/ft^2
Altitude	65.8 nm
Inertial Velocity	37652 ft/sec
Flight Path Angle	-4.5442 deg
Orbit C_3	9 km^2/sec^2
Perigee	45.2 nm
Aerodynamic Parameters	
C_L	.405
C_D	1.35
Heat Shield Diameter	40 ft
Curvature	50 ft
Atmosphere	US 62
Controls - Bank Angle - Reversals - Times of Reversal	
Maxima	
Heating Rate	22 $Btu/ft^2/sec$
Dynamic Pressure	21 lbs/ft^2
Deceleration	2.9 g's
Orbit Leaving the Atmosphere	46.6 x 350 nm
Time in Atmosphere	330 seconds
Apogee Burn to Raise Perigee to	280 nm
ISP	482 sec
Propellant	380 lbs
Delta V	406 ft/sec
Perigee Burn to Circularize at	280 nm
ISP	482 sec
Propellant	110 lbs
Delta V	118 ft/sec

exit is less than 15 minutes in most cases. Deceleration forces, dynamic pressure, and heating rates are basically a function of the energy to be dissipated, the time of dissipation and the aerodynamic characteristics of the entry craft. Guidance algorithms are still being investigated but are beginning to show great promise.

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N87-17725

COMPARISON OF MISSION DESIGN OPTIONS FOR MANNED MARS MISSIONS

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ABSTRACT

A number of manned Mars mission types, propulsion systems, and operational techniques are compared. Conjunction and opposition class missions for cryogenic, hybrid (cryo/storable), and NERVA propulsion concepts are addressed. In addition, both Earth and Mars orbit aerobraking, direct entry of landers, hyperbolic rendezvous, and electric propulsion cases are examined. A common payload to Mars was used for all cases. The basic figure of merit used was weight in low Earth orbit (LEO) at mission initiation. This is roughly proportional to launch costs.

INTRODUCTION

There are many ways to design a manned Mars mission. The optimum design depends a great deal on the long and short term goals of the program. These are at present officially undefined, but range from beating the Russians to Mars with a one landing program to permanent colonization. A program to carry large quantities of material to Mars over a long period of time will tend to settle on designs with minimum initial mass in LEO (includes vehicles and propellants) since Earth launch costs will eventually overwhelm development costs. A short term, one or two mission program, perhaps schedule driven, could concentrate on minimum development costs rather than minimizing LEO mass. The best design depends on the program. In the absence of clear direction, mission designers will produce designs that tend to fulfill their own personal view of what a manned Mars program should be. Since the authors of this paper favor a long term program and would like to see propulsion technology advance, minimum LEO mass is emphasized. Others may have different, but not at all incorrect views.

SCENARIOS

The basic scenario advanced in this paper is a Mars mission carrying two aerobraking landers/ascent stages of 62 metric tons totai mass each, one Mission Module (MM) of 53 metric tons, and one Orbital Transport

Vehicle (Mars-OTV) of 31 metric tons. The spacecraft leaves a 500 km circular low Earth orbit, the basic Space Station orbit, and transfers to Mars. At Mars it boosts into a 24 hr ellipse (500 x 33,000 km) at the proper inclination so that perigee precesses to be lined up correctly for departure to Earth at the proper time. Once in Mars orbit the two manned landers descend to the surface while the MM and propulsion stages remain in elliptical orbit. The Mars-OTV is used by the crew to rendezvous with and explore the two Martian moons. At the end of this surface exploration, the two ascent stages (one on each lander) launch to low Martian orbit where the Mars-OTV meets them and transfers crew and samples up to the MMM. The ascent stages and the MOTV are then discarded. The propulsion stage(s) then return the MM to a 24 hr Earth ellipse (500 x 72,000 km) where it is met by an OTV from the Space Station.

MISSION TYPES

The above scenario was examined for a generic conjunction mission and opposition type Venus swingby missions for the years 1999, 2001, and 2005, as defined in Reference 3. In addition, an electric propulsion case and two hyperbolic rendezvous cases were included.

The conjunction mission uses a near Hohmann transfer from Earth to Mars, a one and one-half year wait at Mars for proper planetary phasing, and a near Hohmann transfer back to Earth. This is the minimum-energy mission with a total mission time of approx. 1000 days and flight opportunities every two years. Delta-V requirements vary somewhat between mission opportunities, but remain constant enough so that a generic Delta-V budget can be constructed for planning purposes.

The opposition missions require transfer to Mars, a stay time of 30 to 60 days, then a transfer back. Because of the phasing, non-Hohmann, high-energy transfers must be used. It has been found that a Venus swingby, either outbound or inbound, can substantially reduce the total energy requirements. Such a swing-by exists for virtually every mission opportunity every two years, but the variation in the three-body relationships creates large Delta-V variations between missions. Thus, each opportunity must be addressed as an entirely separate mission. These missions typically take around 700 days.

The electric thruster case gives high ISP but very low thrust. For low thrust the system (unmanned) spirals out from LEO to some high orbit such as the L2 Lagrangian point. The crew is then transported to the spacecraft via a high thrust OTV flight from LEO. The manned Mars stack then spirals out to Mars and slowly spirals down to low Mars orbit. The landers are dispatched and when the phasing is suitable the process is reversed to return to Earth.

When the power supply is sufficiently large, this reduces to a conjunction type mission with spirals at both ends. The time at Mars including spiral down, orbit operations, and spiral back up becomes the year and a half Mars stay time of the conjunction missions. Electric thruster mission times vary from a minimum of 3 years upward depending on the power source. Practical manned missions will require one megawatt or more of electrical power.

The hyperbolic rendezvous concept requires a launch from Earth carrying the landers and a MM. When Mars is reached, the system does not deboost into Mars orbit; instead, the landers separate and perform hyperbolic aerobraking entry maneuvers to landing sites while the Mission Module flies by Mars and is discarded. A second spacecraft with a second Mission Module leaves Earth at nearly the same time as the first spacecraft, but on a year and a half period trajectory that passes Mars 30 days after the first vehicle. The ascent stages that were landed from the first vehicle launch as the new MM passes by and perform hyperbolic rendezvous maneuvers with it. The crew must then ride the MM for one and a half orbits until it reintersects Earth. Mission time is three years, almost all of it in transit.

A modified version of this, the hyperbolic exchange, assumes a continuing manned base on Mars. The original vehicle with MM and landers is launched into the one and one-half year orbit, passing Mars. As it passes Mars the landers separate and do a hyperbolic entry and landing while, simultaneously the crew that had landed on the previous mission two years before launches to a hyperbolic rendezvous with the MM for the orbit and one-half flight back to Earth. In effect, a crew exchange takes place. Total mission time for a crew with this scenario is at least 5 years. Delta-V's for the various missions are given in Table 1.

TABLE 1

MISSION DELTA-V'S M/SEC

Mission Type	TMI	MOI	TEI	EOI
Conjunction Generic	3808	1666	1490	967
Opp. 1999 In-bound Swingby	4489	2757	1628	3725
Opp. 2001 In-bound Swingby	3792	1798	3633	1252
Opp. 2005 Out-bound Swingby	4400	3543	1673	1198
Low Thrust	13300	2600	8300	0
Hyperbolic Rend. Launch	3799	0	0	0
Hyperbolic Rend. Pickup	3843	0	81	1474
Hyperbolic Rend. Exchange	3843	0	81	1474

PROPULSIVE SYSTEMS

Hybrid

The hybrid system was used as a baseline. It consists of cryogenic liquid oxygen-liquid hydrogen (LO2/LH2) stages for trans-Mars injection (TMI) and Mars orbit insertion (MOI) and a LO2/propane "space storable" stage for trans-Earth injection (TEI) and Earth orbit insertion (EOI). This eliminates the problem of storing liquid H2 in the high heat environment of Mars planetary orbit, where additional cooling equipment to reduce propellant boiloff would be required.

All-Cryogenic

This system uses LO2/LH2 for all stages. This assumes that insulation and refrigeration are developed to allow long term (2 to 3 year) H2 storage.

NERVA

This nuclear rocket system uses nuclear engines with hydrogen as a reaction mass. Three engines of 75,000 lb. thrust each were used. All three are used for TMI to get the thrust/weight up to around .1 in order to keep gravity losses from being excessive. After TMI, one engine and all the empty hydrogen tanks are discarded. Engines 2 and 3 are used together to perform MOI. Engine 2 and the tanks emptied during MOI are then discarded. Engine 3 then performs TEI and EOI. Again, long term hydrogen storage is required. This also assumes that the NERVA engines can be started, shut down, and restarted several times while still maintaining their 10 hour total thrusting lifetime.

Electric Propulsion

High power, low thrust, high Isp ion engines are used for this system. Isp's from 3,000 to 20,000 seconds were examined requiring power supply sizes from .2 to 6 megawatts. Though ion engines with nuclear electric power is a reasonably well known case, any thruster and power processing system with specific mass in the 10 kg/kw range and primary power supply with specific mass as shown in Table 2 will provide equivalent performance. The stage characteristics and other parameters used are shown in Table 2. The electric propulsion design used only a single stage. The delta Vs shown in Table 1 for Low Thrust assume a spiral out to L2 and a transfer to Mars vicinity summed together as TMI, a spiral in to Mars (MOI), and a spiral out from Mars and transfer to Earth-Moon

TABLE 2

PROPULSION STAGE CHARACTERISTICS

Stage Type	All-Hybrid	Mer. Cryo	Ces. Nerva	Ion	Ion
Stage # 1					
Isp	468	468	825	3,000	20,000
A	0	0	11.5	*	*
B	0.0811	0.0811	0.15	0.1	0.1
M.R. O2/Fuel	7	7	0	0	0
Stage # 2					
Isp	480	480	825	0	0
A	0	0	11.5	0	0
B	0.1765	0.1765	0.18	0	0
M.R. O2/Fuel	7	7	0	0	0
Stage # 3					
Isp	370	480	825	0	0
A	0	0	11.5	0	0
B	0.0638	0.1765	0.18	0	0
M.R. O2/Fuel	3.5	7	0	0	0

Stage inert weight = A + B x (Propellant wt.)
A = Mass of power and propulsion system
B = Structure and tankage factor (dimensionless)
All masses in metric tons

Note: For large chemical propulsion stages such as these, the weight of the engines and control systems can be included in the massless parameter B. This assumes lthe number and/or size of the engines increases with increases stage size so that a constant thrust to weight is maintained.

* For electric propulsion, A = power parameter + power processing & thruster parameter)x(electric power). The power processing and thruster mass parameter used for all cases was 10 kgm/kw. An overall conversion efficiency of .7 was also used for all cases. The power parameter as a function of total power is shown below:

Power, kw electric	200	600	1000	3000	6000
Power parameter kg/kw	40	30	15	10	10

L2 (TEI). The spent stage is left at L2, and the crew is transfered back to Earth with an OTV.

FLIGHT OPTIONS

The software built for this study allows us to stack any given mission (opposition, conjunction, etc.) with any propulsive system and payload configuration and combine these with any of a large number of flight case options. These include:

- All propulsive four stage operations
- All propulsive three stage operations
- All propulsive two stage operations
- All propulsive one stage operations
- Aerobraking at Mars--two stage
- Aerobraking at Earth--one, two, or three stage
- Aerobraking at Mars and Earth--two stage

(Note: The above three aerobraking cases consider aerobrake weight as a % of braked cargo to be percentage is a variable parameter.)

- Separation of landers before MOI with the landers performing hyperbolic aero entry--three stage

The cases using aerobraking at Mars can reflect aerobraking to different Mars apoapses by simply changing the TEI delta V to reflect the lower ellipse.

RESULTS

The bulk of the study concentrated on the generic conjunction and the three opposition opportunities with the three standard propulsion systems--hybrid, all-cryo, and NERVA. Figure 1 shows the mass required in LEO for each of these three propulsion systems applied to all four of the standard missions. These were all-propulsive cases, each carrying the same reference cargo set. This chart immediately yields the following results:

- All-cryo does not yield substantially better performance than the more conservative hybrid case.

- With chemical propulsion, the all propulsive opposition missions are significantly more expensive than the conjunction missions. Aerobraking reduces this disparity in cost.

- The NERVA system shows a clear performance advantage for Mars planetary missions. This advantage becomes more and more marked as the mission energy requirements go up. Consequently, the NERVA system could offer a reasonably practical option of flying some of the short stay opposition missions during the early phases of Mars exploration.

O Provided multi-megawatt power supplies are available, electric propulsion is competitive with NERVA and high thrust conjunction class missions, but not as flexible.

Figure 2 shows the impact of discarding part of the MM before the EOI burn. Again, the impact is greater on the high energy missions. This is not generally a major impact but the savings in launch costs (at approx. $1 million per metric ton) warrant examination of the reuse value of the MM parts.

Figure 3 shows the impact of aerobraking at Mars if the vehicle is aerobraked to the same 24 hr period ellipse as in the propulsive case. Various values of aerobrake mass as a percentage of mass to be carried are shown. Only the hybrid propulsion system was examined. The non-aerobraked references are shown as marks on the y-axis. These data show that the overall performance is relatively insensitive to the aerobrake mass in the range considered.

Aerobraking yields substantial gains; the greatest gains being shown for the outbound Venus swingby cases, where encounter (MOI) velocities at Mars are high. Aerobraking can bring some opposition missions down to a reasonable departure weight. (The problem encountered is high acceleration during braking and its effect on the crew).

Figure 4 shows the impact of aerobraking as the apoapsis of the post-aerobrake orbit is reduced. For this comparison, only the conjunction and the 2005 opposition missions with hybrid propulsion were examined. The aerobrake weight used is 15% of the mass carried. Targeting an aerobrake to a very high apoapsis ellipse is difficult because the target velocity is so near escape that even a relatively small aero-exit error could cause loss of the vehicle. The apoapsis may have to be targeted to as low as 2000 km (500 x 2000 km) to guarantee a safe capture.

Nearly all of the aerobraking advantage for the conjunction mission is lost if a low Mars orbit is used (because of the required delta V increase for TEI). However, the absolute change with apoapsis altitude

Figure 1

BASELINE MISSIONS

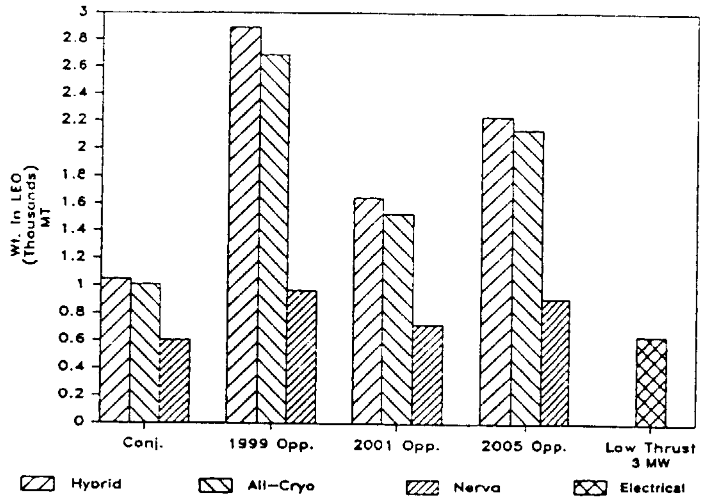


Figure 2

EFFECT OF DISGARDING MMM

50% OF MMM DISCARDED

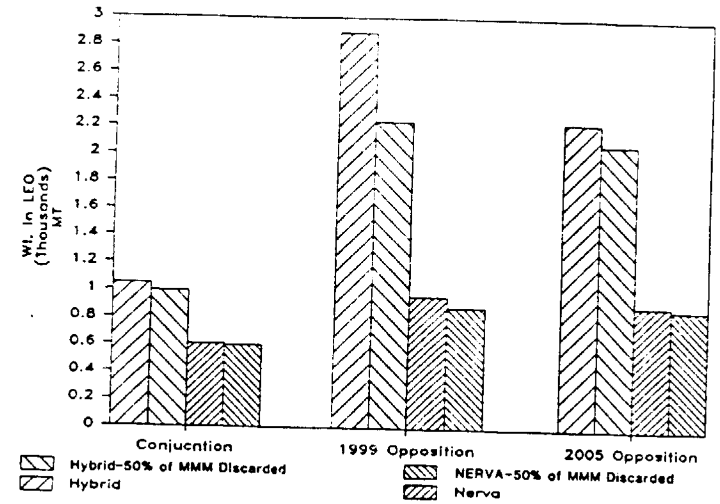


Figure 3

AEROBRAKE TO 24 HR MARS ORBIT

Hybrid Propulsion System (2 Stage)

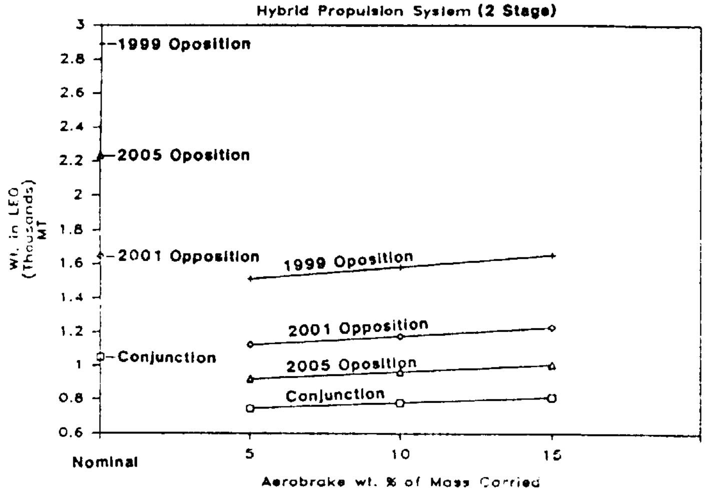
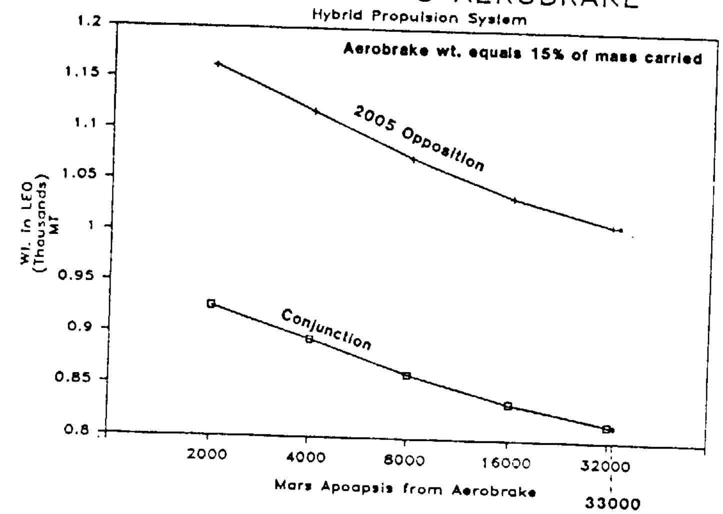


Figure 4

WT TO LEO VS AEROBRAKE

Hybrid Propulsion System



is nearly constant for both missions, so the 2005 opposition mission still shows a massive reduction from the all propulsive case.

Figure 5 shows aerobraking for different Mars apoapses, using a NERVA propulsion system. Again, the gains for the conjunction mission are minimal. The mass for the 2005 case is reduced by about a third; however, the potential advantage of aerobraking is not so great for the NERVA cases, which are already very efficient.

Figures 6, 7, and 8 show the sensitivity of the various missions to changes in lander weight (or cargo carried to Mars orbit and left). The three charts are for the three propulsion systems, hybrid, all-cryo, and NERVA.

Figures 9, 10, and 11 show the sensitivity of the missions to Mission Module mass (or mass carried round trip). The results of these figures for all 12 combinations are summarized in Table 3 as equations of the form: Initial weight in LEO = A + B x (Lander & Mars-OTV Weight) + C x (Mission Module Weight).

Figures 12 and 13 compare various aerobraking modes for the conjunction and the 2005 opposition cases with hybrid and NERVA propulsion. The most notable item is the relative effectiveness of releasing all landers pre-MOI and letting them aerobrake either to direct landing or to a low orbit to await landing site availability. Since the landers are designed for aero-entry already, it may prove relatively inexpensive to do this. Entry g levels may be high however.

Figure 14 shows the crew time, or the time the crew spends in the spacecraft from L2 departure to L2 return, versus power supply for the electric propulsion case. This defines the power requirement for each case since flight times should be kept below four years. Combined with Figure 15, which shows initial mass in LEO versus power, the two figures show that more than one megawatt of electric power will be needed. The lowest Isp cases have short trip times for low power, but Figure 14 shows their LEO masses are approaching the NERVA (600 metric ton) and conventional chemical conjunction (1,000 metric ton) cases. One 3,000 second case with a reduced payload of one lander and no MOTV might be performed with 600 kw. The low thrust cases must provide substantial LEO mass savings to offset the additional development costs; however, if large

Figure 5

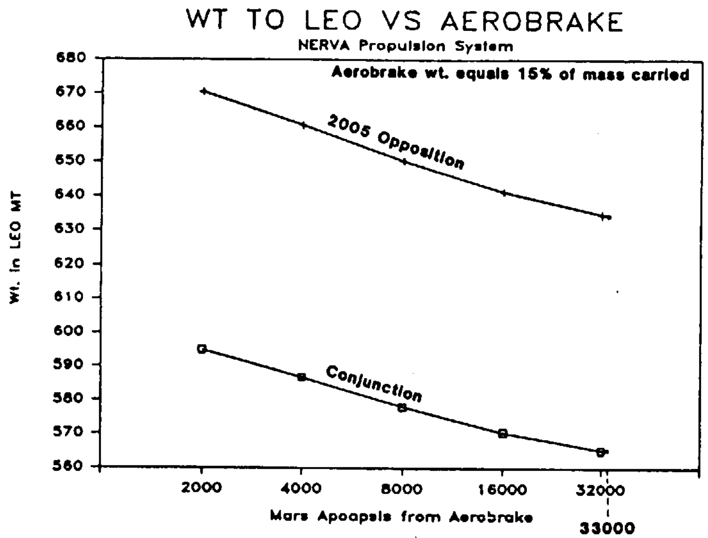


Figure 6

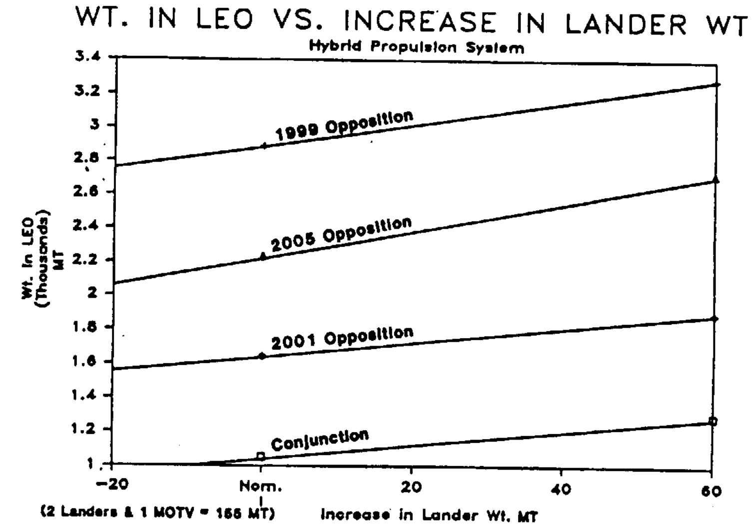


Figure 7

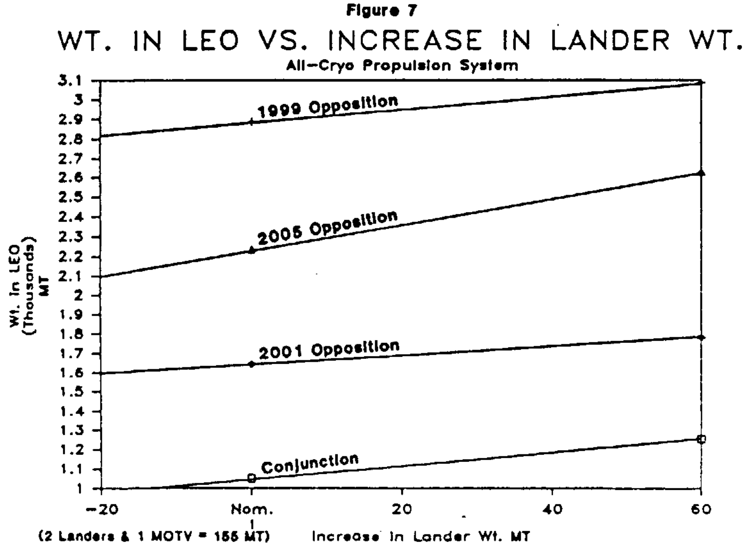
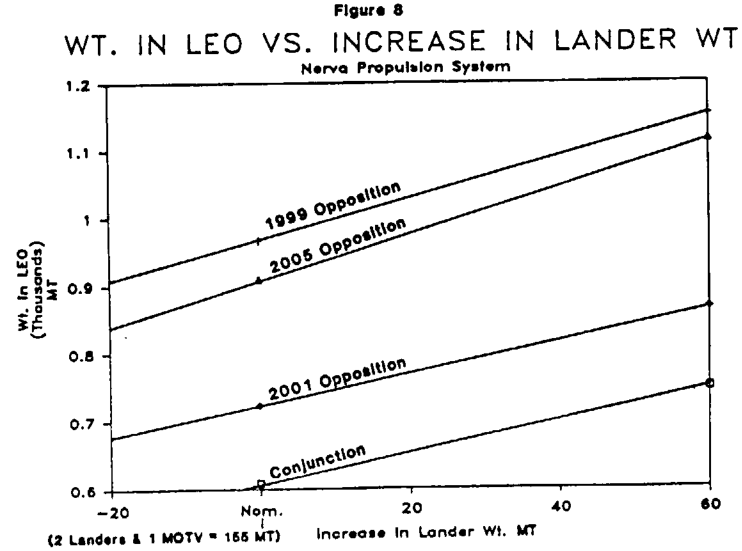


Figure 8



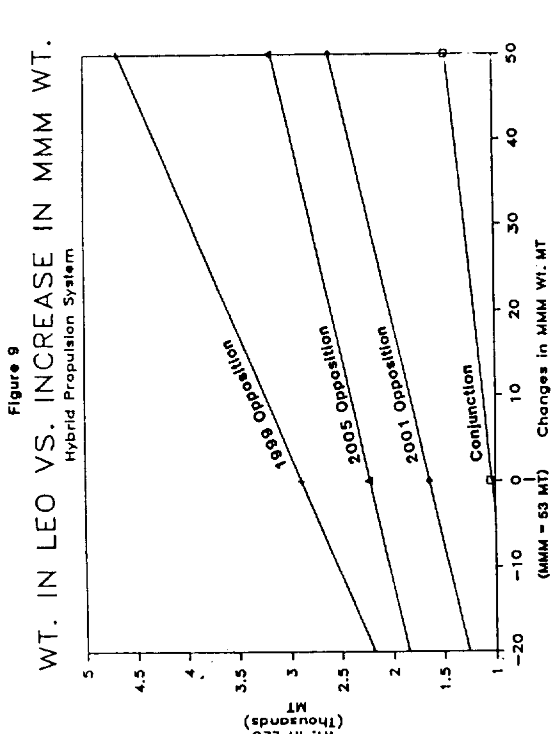


Figure 9
WT. IN LEO VS. INCREASE IN MMM WT.
Hybrid Propulsion System
1999 Opposition
2005 Opposition
2001 Opposition
Conjunction
Wt. in LEO (Thousands) MT
Changes in MMM Wt. MT
(MMM = 53 MT)

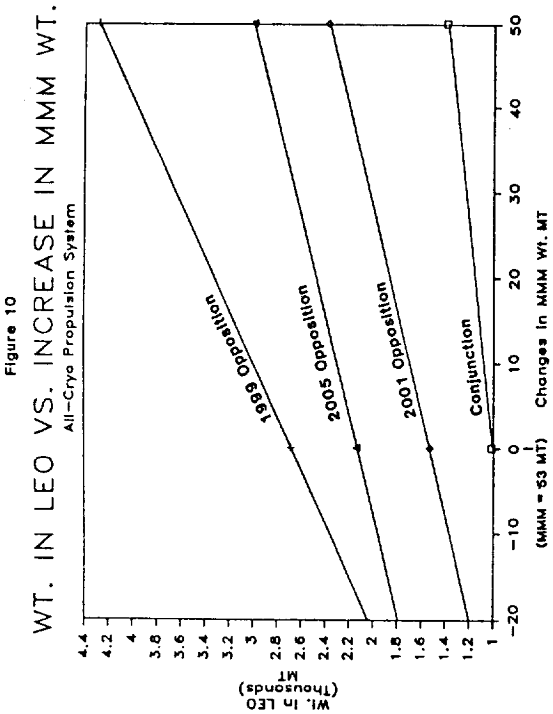


Figure 10
WT. IN LEO VS. INCREASE IN MMM WT.
All-Cryo Propulsion System
1999 Opposition
2005 Opposition
2001 Opposition
Conjunction
Wt. in LEO (Thousands) MT
Changes in MMM Wt. MT
(MMM = 53 MT)

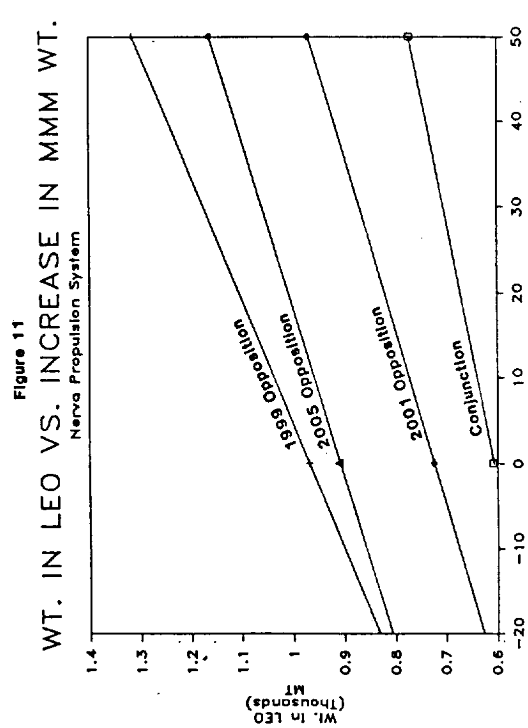


Figure 11
WT. IN LEO VS. INCREASE IN MMM WT.
Nerva Propulsion System
1999 Opposition
2005 Opposition
2001 Opposition
Conjunction
Wt. in LEO (Thousands) MT
Changes in MMM Wt. MT
(MMM = 53 MT)

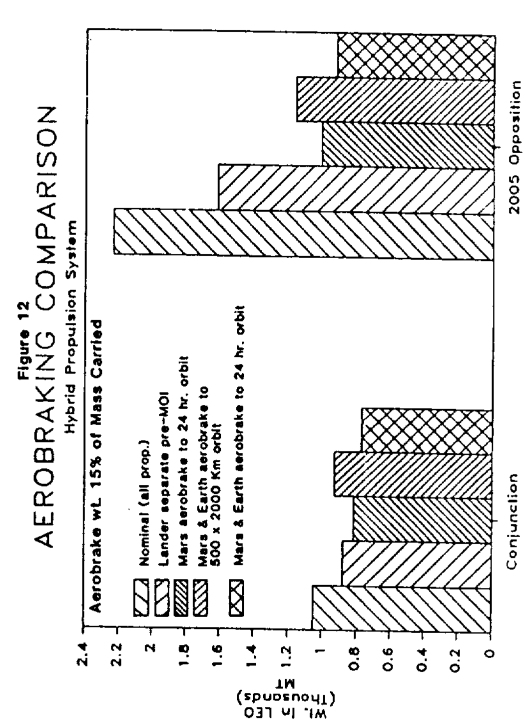


Figure 12
AEROBRAKING COMPARISON
Hybrid Propulsion System
Aerobrake wt. 15% of Mass Carried
Nominal (all prop.)
Lander separate pre-MOI
Mars aerobrake to 24 hr. orbit
Mars & Earth aerobrake to 500 x 2000 Km orbit
Mars & Earth aerobrake to 24 hr. orbit
Wt. in LEO (Thousands) MT
Conjunction
2005 Opposition

TABLE 3

WEIGHT IN LEO AS A FUNCTION OF PAYLOAD
TO MARS AND MM ALL RETURNED

Wt. in LEO = Empirical A ÷B x (lander & Mars-OTV) + (C x MM)

	Parameters	A	B	C
Conjunction Missions	Hybrid	A = 0	B = 3.94	C = 8.28
	Cryo	A = 0	B = 3.94	C = 7.56
	Nerva	A = 86	B = 2.25	C = 3.26
1999 Opposition	Hybrid	A = 0	B = 6.42	C = 35.73
	Cryo	A = 0	B = 6.42	C = 31.94
	Nerva	A = 140	B = 2.97	C = 6.93
2001 Opposition	Hybrid	A = 0	B = 4.07	C = 19.06
	Cyro	A = 0	B = 4.07	C = 16.92
	Nerva	A = 105	B = 2.30	C = 4.93
2005 Opposition	Hybrid	A = 0	B = 7.93	C = 18.96
	Cyro	A = 0	B = 7.93	C = 17.14
	Nerva	A = 100	B = 3.32	C = 5.12

A = Parameter relating required LEO Weight to NERVA systems Wt.

B = Parameter relating required LEO weight for systems carried one way.

C = Parameter relating required LEO weight for systems carried on round trip to Mars.

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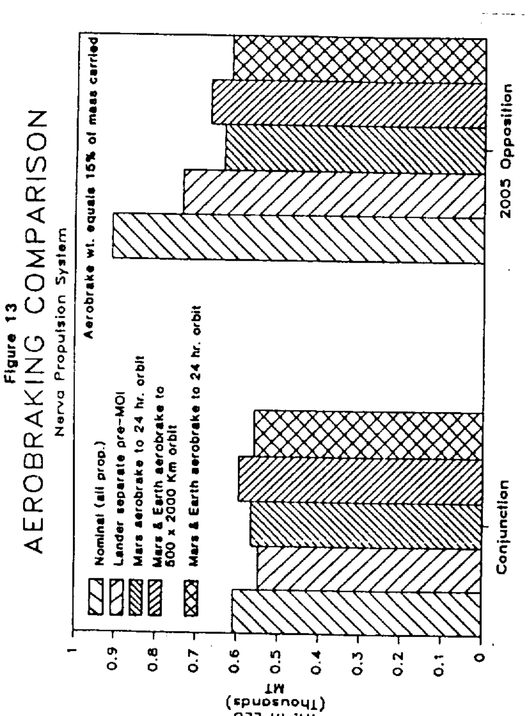

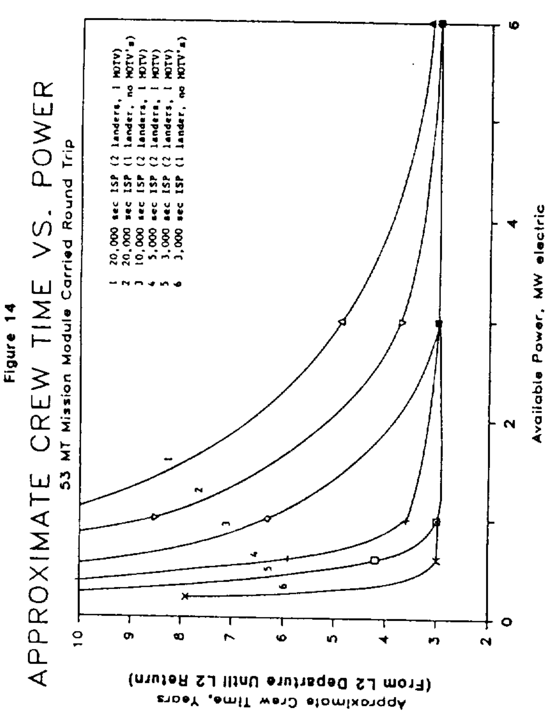

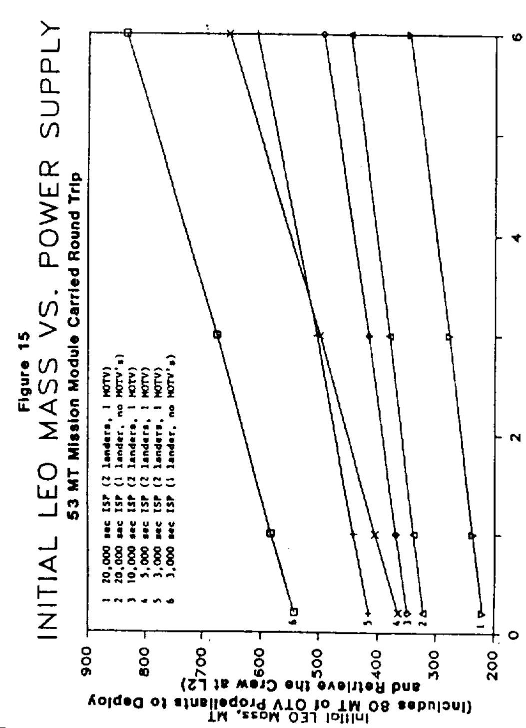

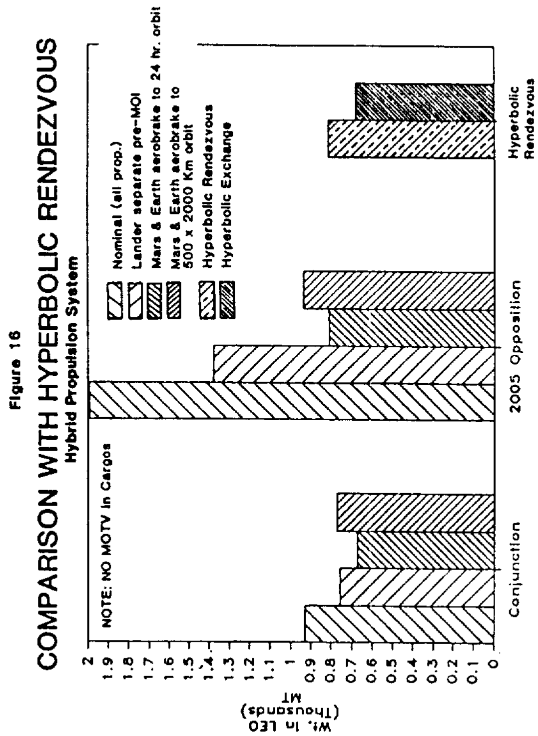

power supplies are developed separately, the low thrust opportunities will be highly competitive.

Figure 16 compares several aerobraking cases with the hyperoblic rendezvous schemes for hybrid propulsion. For this figure the Mars-OTV was removed from all cases to make a one-to-one comparison possible and the hyperbolic rendezvous landers were increased from 62 metric tons each to 90 metric tons (Ref. 1) each to account for the extra propellant required in the ascent stages to reach the hyperbolic outbound velocities. The hyperbolic case requires less mass than the opposition mission, but the comparison should be made with the conjunction missions since the total mission times are nearly the same (3 years). For hyperbolic rendezvous, nearly all the time is in interplanetary transfer, while for the conjunction missions, half of the time is at Mars. Hyperbolic rendezvous shows some weight advantage; however, nearly the same gain can be achieved in the conjunction case by simply staging the landers pre-MOI and doing a hyperbolic entry. This is much simpler than the hyperbolic landing and ascent required of the other case. Significant risk may be associated with the hyperbolic ascent and rendezvous.

GENERAL CONCLUSIONS

Advanced technology propulsion should be pursued vigorously to support a long term Mars program. Given the assumptions used in this paper, NERVA appears to yield an advantage even in the minimum energy cases and may provide the flexibility of flying the higher energy mission options. This advantage may become more pronounced as high energy missions to destinations past Mars are contemplated. This conclusion was also reached by workers of the late 60s (Ref. 1). Reference 1 documents the last large, overall systems level study done on a manned Mars mission/program on NASA contract.

The NERVA program, canceled in 1970, was designed with a manned Mars mission in mind. However, there were several problems which are assumed solveable in this paper.

O The old NERVA specific impulse estimate of 900 seconds was degraded to the 750 second region by erosion problems of the graphite core elements and by the propellant losses needed to cool the reactor after each burn. This paper assumes an Isp of 825 seconds.

O The inert shielding mass was high. This paper assumes a shield and reactor mass of 11.5 metric tons per stage. Changes in this can significantly alter the results. Formidable operations problems for manned operations in the vicinity of NERVA also would exist.

O The low density of the hydrogen propellant (4.4 lbm/ft^3) compared to O_2/H_2 (22-25 lbm/ft^3) resulted in higher cost per unit mass for delivery.

O No mission model large enough to absorb the development costs and still make the old NERVA program pay existed.

O Environmental and political/emotional impact of testing were severe.

O A "nuclear safe altitude" is not well defined. This paper assumed the NERVA could depart from a 500 km circular orbit. If this changes radically, the results may also change.

Aerobraking is worth continued investigation, particularly if no advanced space propulsion is available.

Conjunction class missions can be flown for reasonable weights even with chemical all-propulsive cases. However, either the NERVA or aerobraking is necessary to make the opposition missions a practical alternative.

Electric propulsion also offers weights in the NERVA range, but with less flexibility. Its feasibility hinges on the practicality and cost of megawatt level electric power supplies, which need to be determined.

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2. Integrated Manned Interplanetary Definition Study (IMIS), 1968, Boeing, Seattle, Contract No. NAS1-6674, 6 volumes.

3. Young, A. C., Mars Mission Concepts and Opportunities: In Manned Mars Missions, NASA Report M0002.

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EARTH VICINITY TRADES AND OPTIONS

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ABSTRACT

The options for recovering a returned manned Mars spacecraft are surveyed. Earth parking orbits from libration point to low circular are discussed, with a 500 km perigee, 24 hour period elliptical orbit chosen as a baseline for further calculation. Several techniques for recovering up to 100 metric tons of returned spacecraft are investigated, including recovery by a LEO based OTV pushing the spacecraft to LEO, an OTV transporting an aerobrake to the spacecraft, and an OTV delivering propellant to the spacecraft. Methods utilizing OTVs result in less total mass in LEO, but may not be the minimum cost solutions if significant development and testing are required.

INTRODUCTION

A number of methods exist for recovering a manned Mars mission crew and spacecraft in or near Earth orbit. The parking orbit, mass, and volume of the returned spacecraft must first be determined, then a technique can be chosen to return this mass to low Earth orbit (LEO) for refurbishment.

PARKING ORBITS

Options for Earth parking orbits on return of a manned Mars mission range from high circular, perhaps including a libration point and high elliptical; with periods on the order of 48 hours, to low apogee elliptical and low circular; or direct entry into the Earth's atmosphere. All these options, with the exception of the last, assume propulsive insertion.

The high circular parking orbits are most appropriate for electric propulsion stages. References 1 and 2 discuss these mission scenarios. If multimegawatt power supplies are available, electric propulsion may prove to be attractive. It is a special case, apart from high thrust propulsion, however.

Electric propulsion trajectories consist of many-revolution spirals, due to the low, usually continuous thrust levels, and are thus constrained for all practical purposes to circular orbits. A manned electric propulsion stage cannot spiral up or down through the radiation belts with a crew aboard because of the many months required and high radiation dose involved. Also, radiation-sensitive equipment (including integrated circuits sensitive to logic level upsets, etc.) may not be able to stand such radiation levels unless protective shielding is provided. A high-thrust boost through the belts is possible, but much of the performance advantage of electric propulsion may be negated. The high thrust delta V to geosynchronous orbit (4.2 km/sec, 3.82 with no plane change) is more than a typical trans-Mars insertion burn from the Space Station orbit for a conjunction class trajectory (3.8 km/sec). The electric propulsion stage must therefore either spiral up through the belts unmanned or be based beyond them. In either case, the crew must be brought up and retrieved from the interplanetary spacecraft parked in high circular orbit.

The altitude of this high circular orbit requires some study. Geosynchronous orbit (GEO) is a candidate. The 42 metric ton propellant capacity Orbital Transfer Vehicle (OTV) described later in this paper (Figure 5) can carry a 6 metric ton crew module round trip from the Space Station orbit to GEO and back.

The L2 libration point (the one behind the Moon, see Ref. 3) and low lunar orbit, have also been proposed as staging points for repeated Mars missions that would use lunar-derived propellants. L2 has also been proposed as a staging point for missions that might use a largely reusable chemical stage or electric propulsion. The high thrust delta V from the Space Station orbit to L2 (approx. 3.5 km/sec) is less than the delta V to GEO. It is not much less than the conjunction class trans-Mars injection delta V from LEO however. L2 staging will probably require substantial infrastucture in high orbits and may therefore be viewed as a longer term option that still requires study. Use of lunar-derived propellants (Ref. 4) will depend on the ratio of lunar to Earth launch costs and is still under study.

Delta V from LEO to low lunar orbit (4.13 km/sec) is almost the same as the LEO to GEO delta V (4.2 km/sec). As a first order approximation, we can therefore assume that a LEO based spacecraft that can retrieve a

Mars mission crew from GEO can also retrieve one from low lunar orbit or L2.

The high elliptical parking orbit requires the minimum insertion burn of a returning Mars spacecraft. The higher the apogee, the less the burn. Table 1 shows the insertion burns required for a number of orbits for conjunction and opposition missions. The best high thrust way to get to a high circular orbit is first to do an "Earth flyby" or insert into an ellipse with apogee at the desired circular altitude. Table 1 illustrates this, showing insertion delta Vs with and without flybys for a number of cases.

Figure 1 shows initial LEO mass versus round trip mass for a number of mission configurations. One extra ton carried round trip requires from 3.3 to 31.9 extra tons initially in LEO, depending on the mission trajectory and propulsion type. Recovery from a 24 hour ellipse without plane change,using LEO- based OTVs, costs roughly 2 metric tons for every ton recovered to 500 km circular LEO, depending on the scheme. It therefore pays in terms of initial mass in LEO to carry as little propellant and stage as possible for the Earth orbit insertion burn. To reduce overall mass in LEO, the parking orbit with the minimum insertion delta V requirement should be used. This means using as high an apogee as possible. How high this can actually be requires more study. The stability of the longer-period ellipses has been questioned. The maximum may be somewhere around a 48 hour period ellipse with perigee at 500 km.

The radiation belts may cause problems for high elliptical parking orbits. Only a limited number of passes through the belts can be tolerated by a crew at the end of a long mission during which high level radiation exposure may have already occurred. If the "storm shelter," needed during interplanetary flight for protection from solar flares, is placed in the ellipse, it may protect the crew during passage through the belts. This requires more study.

Figure 2 plots initial mass in LEO versus elliptical orbit apogee and period for a number of configurations. The knee in the curve is around the 12 hour period orbit for chemical propulsion. The nuclear propulsion (NERVA) cases are relatively flat for the entire range. All the curves are flat beyond 12 hour periods. The 24 hour period ellipse,

Fig. 1

Inital Leo Mass versus round trip Payload Mass

All cases-3 stages (TEI and EOI combined) except for aerobraking case. Hybrid-first two stages CRYO (O2/H2), last Propane/O2, 24 hour, 500KM perapsis parking orbit at Earth and Mars. All cases carry two landers (124mt) and one Mars OTV (31mt).

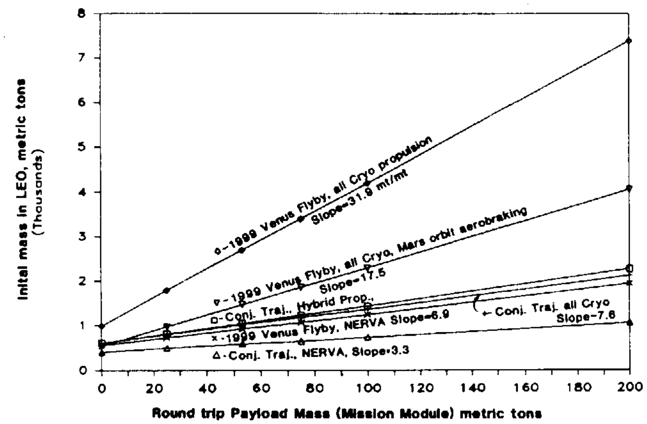


Fig. 2

Initial Mass in LEO versus return orbit

All cases-3 stages (TEI and EOI combined) except for aerobraking case, return 100% of 53mt mission module, carry two landers (124mt) and one MOTV (31mt), use 24 hour 500KM periapsis Mars parking orbit. Hybrid-TMI and MOI burns are all Cryo. (O2/H2), last two burns are O2/propane.

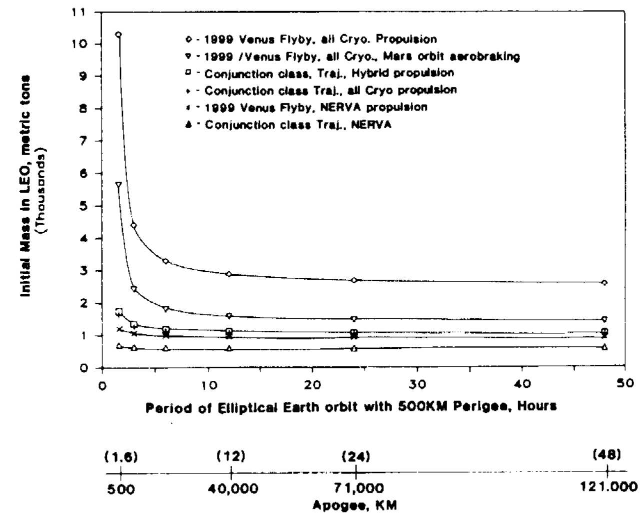
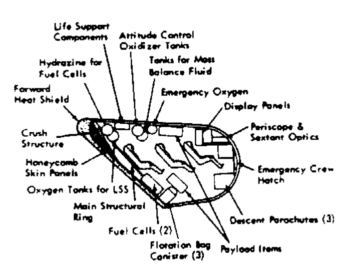


Fig. 3



Estimated Weights	
Crew and Seats	1,362 pounds
Controls	270
Guidance and Navigation	300
Communications	185
Science	912
Life Support	732
Electrical Power	659
Attitude Control	1,120
Recovery	870
Heat Shield	4,340
Structure	4,160
Growth & Contingency (15%)	2,240
Total	17,150 pounds (7,779 kg)

Biconic Earth Entry Module (from ref. 5)

SIX-MAN CREW
V_e = 60,000 fps

Fig. 4

Initial mass in LEO versus mass returned

All cases-3 stages (TEI and EOI combined) except for aerobraking case 24 hour, 500KM periapsis parking orbit at Earth and Mars. All cases carry two landers (124 mt) and one Mars OTV (31 mt) Hybrid-TMI and MOI burns are all Cryo. (O2/H2), last two burns are O2/propane.

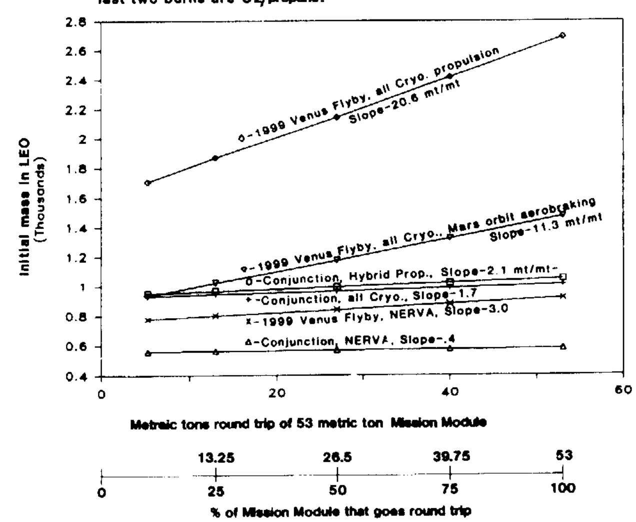


TABLE 1

DELTA V's FOR EARTH ORBIT INSERTION AND RETRIEVAL
1999 CONJUNCTION AND OPPOSITION TRAJECTORIES

Final Destination Orbit - 500 km (270 nm) circular, 28.5 deg., 1.58 hour period.

Insertion Orbit			Delta V's		
500 Km perigee, 28.5 deg.			1	2	3
	Apogee Km	Period Hours	99 Opp. Insert. Delta V km/sec	99 Conj Insert. Delta V km/sec	Delta V from Ins. Orb. to Dest.km/sec
	121,000	48.00	3.55	0.91	2.87
	71,000	24.00	3.72	1.08	2.70
	40,000	12.00	3.99	1.35	2.43
	20,000	6.00	4.44	1.80	1.98
	7,870	3.00	5.2	2.56	1.22
	500	1.58	6.42	3.78	--
Direct Insertion into Circular Orbit					
	Altitude Km	Period Days			
(L2)	443,000	34.66	8.16	3.28	3.50
	121,000	5.23	7.57	2.95	4.07
(GEO Alt)	35,900	1.00	6.92	2.83 (28.5 deg. inclin.)	3.82
	20,370	0.5	6.67	2.91	3.37
(Space Station)	500	0.07	6.42	3.78	--
Insertion into Circular via Earth Flyby (and burn) at 500 km altitude					
(L2)	443,000	34.66	3.27	0.63	3.5
	121,000	5.23	4.75	2.11	4.07
(GEO Alt)	35,900	1.00	4.05	1.41 (28.5 deg. inclin.)	3.82

Note:
For 1999 Opposition, C3 inbound = 81; for 1999 Conj., C3 = 16 $(km/sec)^2$

with perigee at 500 km, is well beyond the knee in the curve, and has been used in a number of reference missions.

Direct entry into the Earth's atmosphere from the interplanetary trajectory requires no burn. Figure 3 shows a concept for a 7.8 metric ton direct entry capsule taken from reference 5. The large crew compartment flies on by Earth. The crew is only in the small capsule for a day or so. This approach results in the lowest initial mass in LEO of all and should not be discarded lightly. Its disadvantages include potential high g loads for a crew that may have just spent 2 to 3 years in zero g, no capability to quarantine the crew in the perhaps unlikely event Martian life is found and proves to be infectious on the long trip home, no capability for reuse of the large crew compartment or Mission Module, and the requirement to develop an additional entry vehicle.

Aerobraking into low Earth orbit avoids all but two of these problems. Initial studies indicate the g levels must still be high for a crew that has just experienced two to three years of zero g, and pre-entry burns are probably not a practical way to keep them down. If the crew habitat has significant artificial g, the g loads may not be a problem. The aerobrake, which may weigh 5 to 15 % of the aerobraked mass, must still be carried round trip, however, and will require significant additional development work. This aerobrake might also be used for Mars entry. The aerobraking option requires more study, and will be addressed in other papers.

Propulsive insertion into a high ellipse avoids all these problems at the cost of an Earth orbit insertion stage and the requirement to go after the crew and spacecraft with OTVs. It is therefore the leading contender at present.

HOW MUCH TO RECOVER

How much of the interplanetary spacecraft to recover? The options range from recovery to a refurbishment facility of an entire propulsion and crew module capable of single stage round trips, to direct entry into the Earth's atmosphere of a small crew module only as shown in Figure 3. Single stage options will probably require aerobraking at least at Mars and Mars orbit refueling, and are therefore longer-term options. The pros and cons of direct entry capsules are noted in the previous paragraphs.

Propulsive insertion of some fraction of the Mission Module and a trans-Earth/Earth orbit insertion stage into a 24 hour ellipse is considered in Figure 4, which shows the effect of inserting various masses for several reference missions. The increase in initial LEO mass/increase in inserted mass or slope of the lines in Figure 4 is not as great as the increase in LEO mass/total round trip mass (Figure 1). How much of the Mission Module is inserted into Earth orbit is not as important as how much the complete Mission Module and other round trip mass weighs. This other round trip mass could be propellant to lower the apogee of the ellipse. It must be carried round trip and inserted into the ellipse and is therefore very expensive, which makes it attractive to consider delivering it with an OTV to the returned spacecraft in high elliptical Earth orbit.

Since the actual Mission Module mass recovered is more a function of the economics of reuse than anything, it is beyond the scope of this work to define. This recovered mass will almost certainly be no more than 100 metric tons however, so a range from zero to 100 metric tons will be assumed.

METHODS OF RECOVERY FROM HIGH ELLIPTICAL EARTH ORBIT

Given the assumptions of a 24 hour period elliptical parking orbit and a mass range of zero to 100 metric tons, several methods for recovering this mass to the Space Station orbit can be proposed: 1) An unmanned OTV can dock with the spacecraft and propulsively return it to the Space Station orbit; 2) A manned or unmanned OTV can bring up an aerobrake to attach to the spacecraft, which then lowers apogee by aerobraking; 3) A manned or unmanned OTV brings up propellant to refuel the Earth orbit insertion stage and the spacecraft comes down propulsively; and 4) A manned OTV recovers the crew and mission artifacts and the spacecraft is left in orbit or deorbited to a controlled re-entry.

In the following analysis, a space-based aerobraked OTV, as shown in Figure 5, is assumed. This OTV has an empty weight of 7 metric tons, carries 42 metric tons of liquid hydrogen and oxygen that is burned at a specific impulse of 480 seconds, and carries an 8 metric ton crew module capable of carrying a crew of 8. It is assumed to be reusable and stackable as shown in Figure 6.

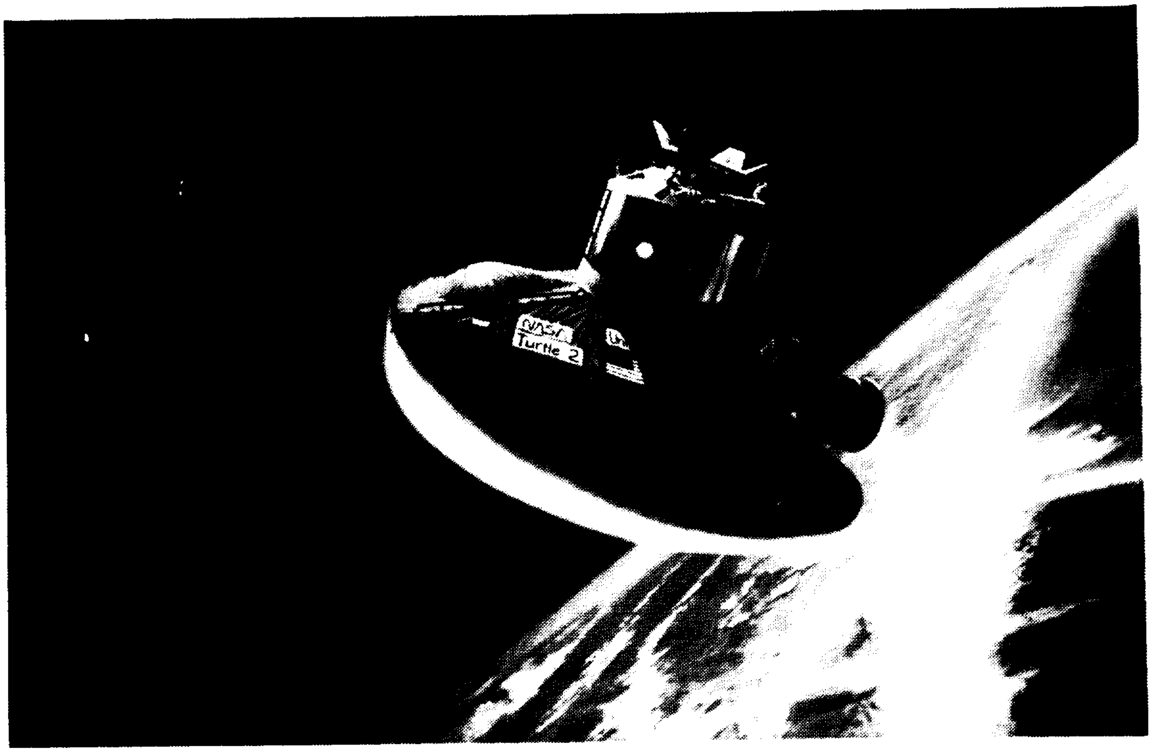


Fig. 5 Aerobraking OTV

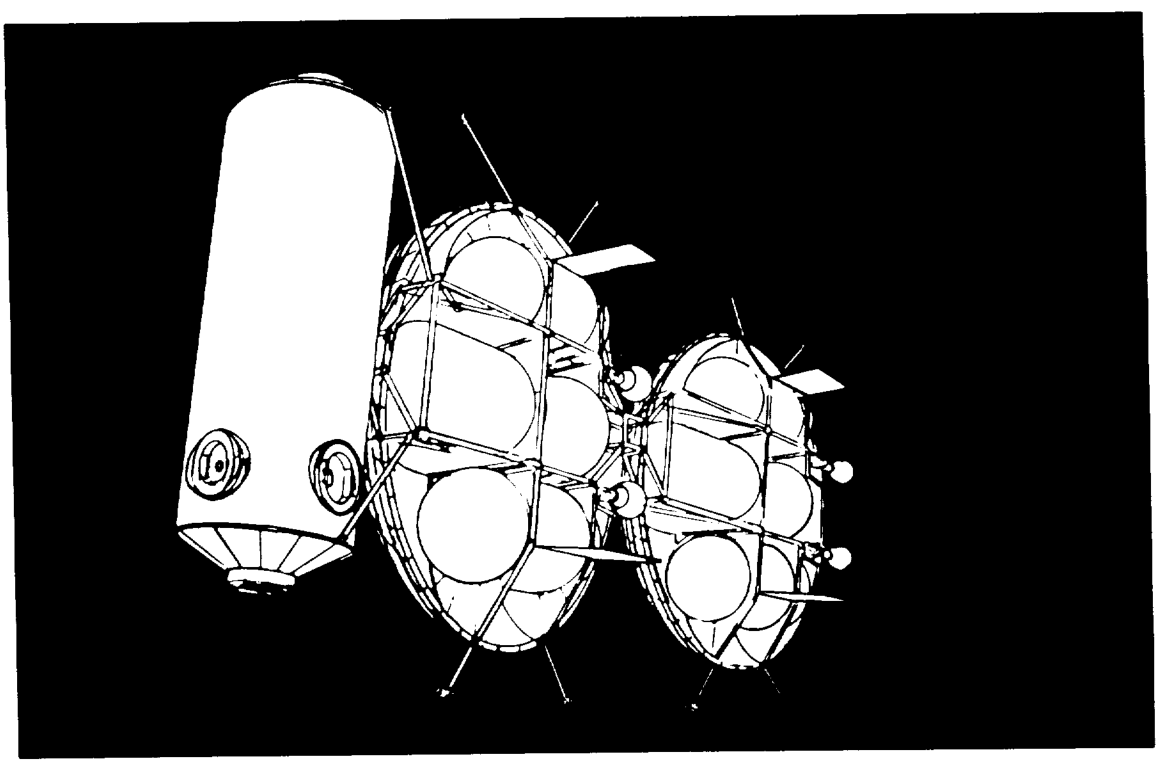


Fig. 6 Stacked OTVs

Table 2 shows a range of numbers for an unmanned OTV(s) docking with the Mars spacecraft and pushing it to LEO. One OTV uses 21 metric tons of fuel to deliver itself and a maximum of 20 additional metric tons of propellant in its own tanks from the Space Station orbit (500 km, 28.5 deg. circular) to the 24 hour ellipse (71,000 km x 500 km, 28.5 deg). One OTV can also deliver a second OTV with a maximum of 39 metric tons of propellant in its tanks to the 24 hour ellipse. The first stage OTV then aerobrakes back to LEO.

The last row in Table 2 shows the OTV propellant needed in LEO over the returned mass. For the heavier masses, this number is constant around 2.0. This means 2.0 metric tons of OTV propellant are needed in LEO for every 1.0 metric ton of Mars Mission Module brought back to LEO with the OTVs. Each metric ton of propellant placed in the 24-hour orbit can return approximately one metric ton of Mission Module to LEO from the 24 hour orbit. If this metric ton of propellant had to go round-trip to Mars it would have cost between 3.3 and 31.9 metric tons in LEO. By using the OTV-delivered propellant we are thus saving between 3.3-2 = 1.3 and 31.9-2 = 29.9 metric tons in LEO per metric ton of Mission Module recovered to LEO with this technique. This can be a good mass trade, particularly for the opposition class missions. The OTV sorties are not free however. A cost analysis is required.

The case in which a manned or unmanned OTV brings up an aerobrake to attach to the spacecraft has an even better mass trade, but introduces additional operational complexities and costs. One OTV can deliver an 8 metric ton (8 person) crew module, a 15 metric ton aerobrake (capable of aerobraking an entire 100 metric ton spacecraft), 7 metric tons of oxygen and hydrogen propellant for the Mars spacecraft or Mission Module to do perigee lower/raise manuevers, and an additional tank of 12 metric tons of propellant to bring itself and the crew module back propulsively to keep the returning Mars crew from experiencing high acceleration loads.

One OTV can handle the worst case aerobrake situation. The Mars spacecraft must be compact enough to be aerobraked however, and the aerobrake must be assembled in LEO. The total payload mass of the OTV is 42 metric tons. To deliver this the OTV uses 39 metric tons of fuel. For 100 metric tons recovered, the OTV LEO mass over recovered mass is

TABLE 2

UNMANNED OTV DOCKS WITH SPACECRAFT AND
PROPULSIVELY RETURNS IT TO STATION

No. OTVS Required	1 (single stage)	1 stack of 2 (two stage)	1 stack of 2 plus 1	3 stacks of 2
Inserted Mass (MT)	7	42	50	100
Prop. to Return (MT)	11	39	50	95
OTV Prop. in Leo (MT)	31	81	106	203
OTV Prop. over Ins. Mass	4.43	1.93	2.13	2.03

24 hour ellipse parking orbit (71,000 x 500 km, 28.5 deg.)
500 km circular, 28.5 deg. destination orbit

TABLE 3

PROPELLANT A 42 MT CAPACITY OTV CAN DELIVER TO THE
24 HOUR ELLIPSE. OTV AEROBRAKES BACK TO LEO

	MANNED	UNMANNED
Delivered Prop in OTV Tanks	16	20
All delivered Prop in 2 mt mass external tank (not part of OTV)	35	43

roughly .8, better than 2.0 in the previous case. The cost to develop the aerobrake may be significant, however.

Table 3 shows the propellant which a manned or unmanned OTV can deliver to the Mars spacecraft, such that it can return itself propulsively to a space station compatible orbit. An extra (external) tank will be required for most cases. Table 4 shows the propellant that must be delivered for both manned and unmanned OTVs and for cryogens and storables. The manned LEO OTV propellant divided by the recovered mass ranges around 1.8 to 2.0 for cryogens and around 2.8 for storables. In terms of mass gain in LEO it is similar to the case where the OTV pushes the Mars spacecraft. Propellant transfer and tankage requirements will probably make it cost more however.

A single manned OTV can easily recover the crew and artifacts only, bring them back propulsively, and send a 100 metric ton spacecraft in the 24 hour ellipse to a controlled re-entry with a 200 m/sec push. It requires a full 42 metric tons of propellant.

In summary, the baseline case of a 50 metric ton Mission Module can be entirely recovered in several ways. It can be done with one OTV flight that delivers an aerobrake to it and recovers the crew. One OTV could also recover the crew and deorbit the spacecraft. Two OTV flights can deliver enough propellant to the mission module to allow it to utilize its own propulsion system to return to LEO. Three OTV flights (one stack of two plus one) can push it to LEO.

TABLE 4

OTV DELIVERS PROPELLANT

INSERTED (returned) MASS (MT)	7	42	50	100
CRYOGENS				
PROP. REQ. TO RETURN (MT) (480 ISP)	6	33	39	79
UNMANNED				
NO. UNMAN. OTV FLIGHTS TO DELIVER	1	1*	1*	2*
TOT. OTV PROP. MASS REQ.	16	65	75	155
MANNED				
NO. MAN. OTV FLIGHTS TO DELIVER	1	1*	2*	3*
TOT. OTV PROP. MASS REQ.	24	72	95	181
STORABLES				
PROP. REQ. TO RETURN (MT) 340 ISP	9	53	64	127
UNMANNED				
NO. UNMAN. OTV FLIGHTS TO DELIVER	1*	2*	2*	3*
TOT. OTV PROP. MASS REQ.	23	109	128	249
MANNED				
NO. MAN. OTV FLIGHTS TO DELIVER	1*	2*	2*	4*
TOT. OTV PROP. MASS REQ.	29	122	141	281

*Delivered Propellant is in extra external tank.

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LIBRATION-POINT STAGING CONCEPTS FOR EARTH-MARS TRANSPORTATION

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ABSTRACT

The use of libration points as transfer nodes for an Earth-Mars transporation system is briefly described. It is assumed that a reusable Interplanetary Shuttle Vehicle (ISV) operates between the libration point and Mars orbit. Propellant for the round-trip journey to Mars and other supplies would be carried from low Earth orbit (LEO) to the ISV by additional shuttle vehicles. Different types of trajectories between LEO and libration points are presented, and approximate delta-V estimates for these transfers are given. The possible use of lunar gravity-assist maneuvers is also discussed.

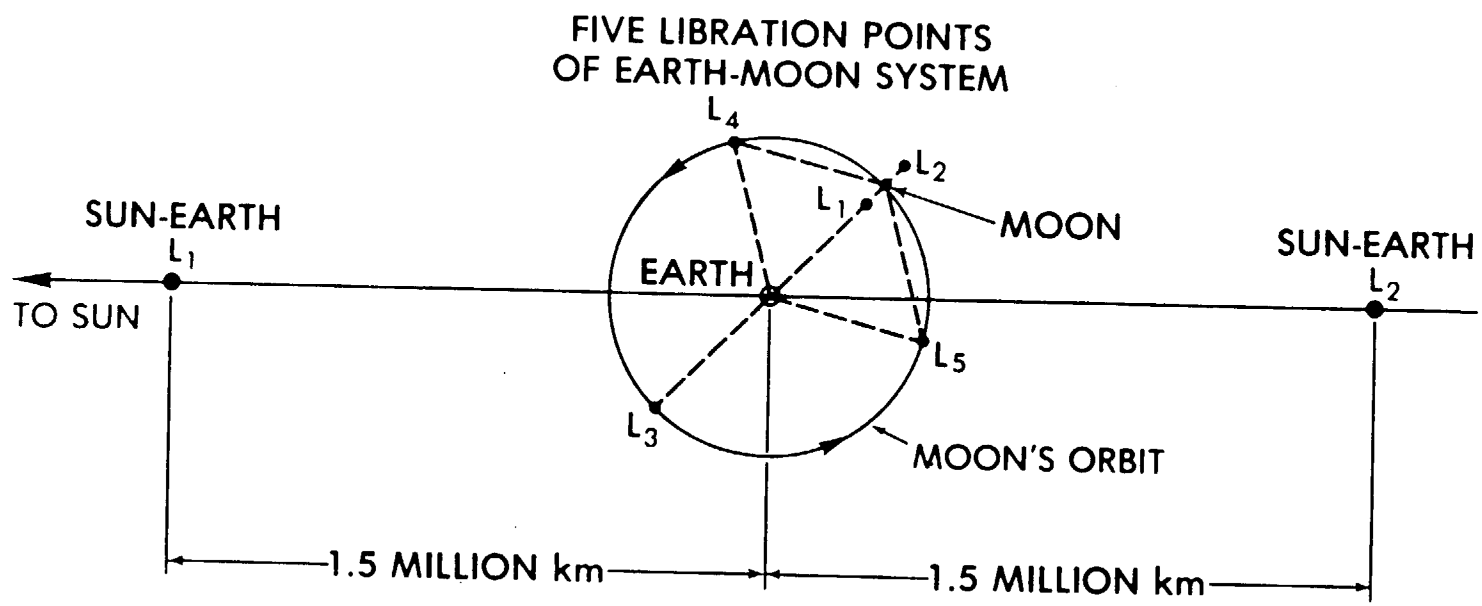
LIBRATION-POINT STAGING CONCEPTS FOR EARTH-MARS TRANSPORTATION

The existence of five positions of equilibrium in the gravitational field of an isolated two-body system (e.g., Earth-Moon or Sun-Jupiter) is well known. As shown by the French mathematician, J. Lagrange in 1772, these "libration points" have the interesting property that if a third body were placed at one of them with the proper velocity, the centripetal acceleration of the third body would be perfectly balanced by the gravitational attractions of the two primary bodies. Three points are situated on a line joining the two attracting bodies, while the other two form equilateral triangles with these bodies. Although the three collinear points are inherently unstable and the two triangular points are only quasi-stable, the stationkeeping cost to maintain a spacecraft at or near one of these points for an extended period of time is very small [1].

A total of seven libration points are located in the Earth's neighborhood (see Figure 1). Five of them are members of the Earth-Moon System and two belong to the Sun-Earth System. In the reference frame shown in Figure 1, the Sun-Earth line is fixed and the Earth-Moon configuration rotates around the Earth. From the standpoint of potential applications to astronautics, the L1 and L2 points of both systems are noteworthy. It is anticipated that some or all of these points will be

FIGURE 1

LIBRATION POINTS IN THE VICINITY OF THE EARTH



utilized as transportation nodes in future manned expeditions to the Moon and Mars [2].

Spacecraft trajectories from low Earth orbit (LEO) to collinear libration points are difficult to analyze because these trajectories spend considerable time in a region where the gravitational effects of the two primary bodies are comparable. In this situation, standard analytic approximations such as the patched-conic technique break down, and numerical integration must be employed.

Figure 2 depicts fuel-optimal examples of the two principal classes of transfers between LEO and the Sun-Earth L1 point. Optimality has been determined on the basis of the terminal maneuver at L1 because the injection delta-V at LEO is virtually identical for all cases. Although the delta-V requirement is higher for the fast transfer, the flight time is less than one-third of that needed by the slow transfer. Smaller delta-V costs can be achieved by using transfers with much longer flight times and/or by including a lunar gravity-assist maneuver. However, a more effective way to reduce the delta-V cost is to simply target to an orbit around the L1 point instead of the point itself [3]. This method was used to place the International Sun-Earth Explorer-3 (ISEE-3) spacecraft into a large "halo orbit" around the L1 point [4] (see Figures 3 and 4). The retro delta-V for ISEE-3 was essentially the sum of delta-V2 and delta-V3 (i.e., 36.3 m/sec).

Two types of trajectories between LEO and the Earth-Moon L2 point are shown in Figure 5. In both cases the delta-V at LEO is roughly 3.15 km/sec. Notice that the two-impulse transfer is almost 5 days faster than the three-impulse example. However, the retro delta-V for the three-impulse transfer is smaller by about 900 m/sec. This comparison demonstrates that the identification of an efficient trajectory to or from the vicinity of a libration point can be a rather subtle exercise. The use of a powered lunar swingby to reduce the retro delta-V at L2 was certainly not obvious.

The three-impulse trajectory of Figure 5 is a key element of a lunar transporation concept that uses the Earth-Moon L2 point as a staging location. In this concept, a large chemical orbit transfer vehicle (OTV) carries payloads between LEO and L2 point. At the L2 point, the payload is transferred to a smaller OTV that operates between L2 and low lunar

FIGURE 2

TRAJECTORIES TO THE SUN-EARTH L_1 LIBRATION POINT

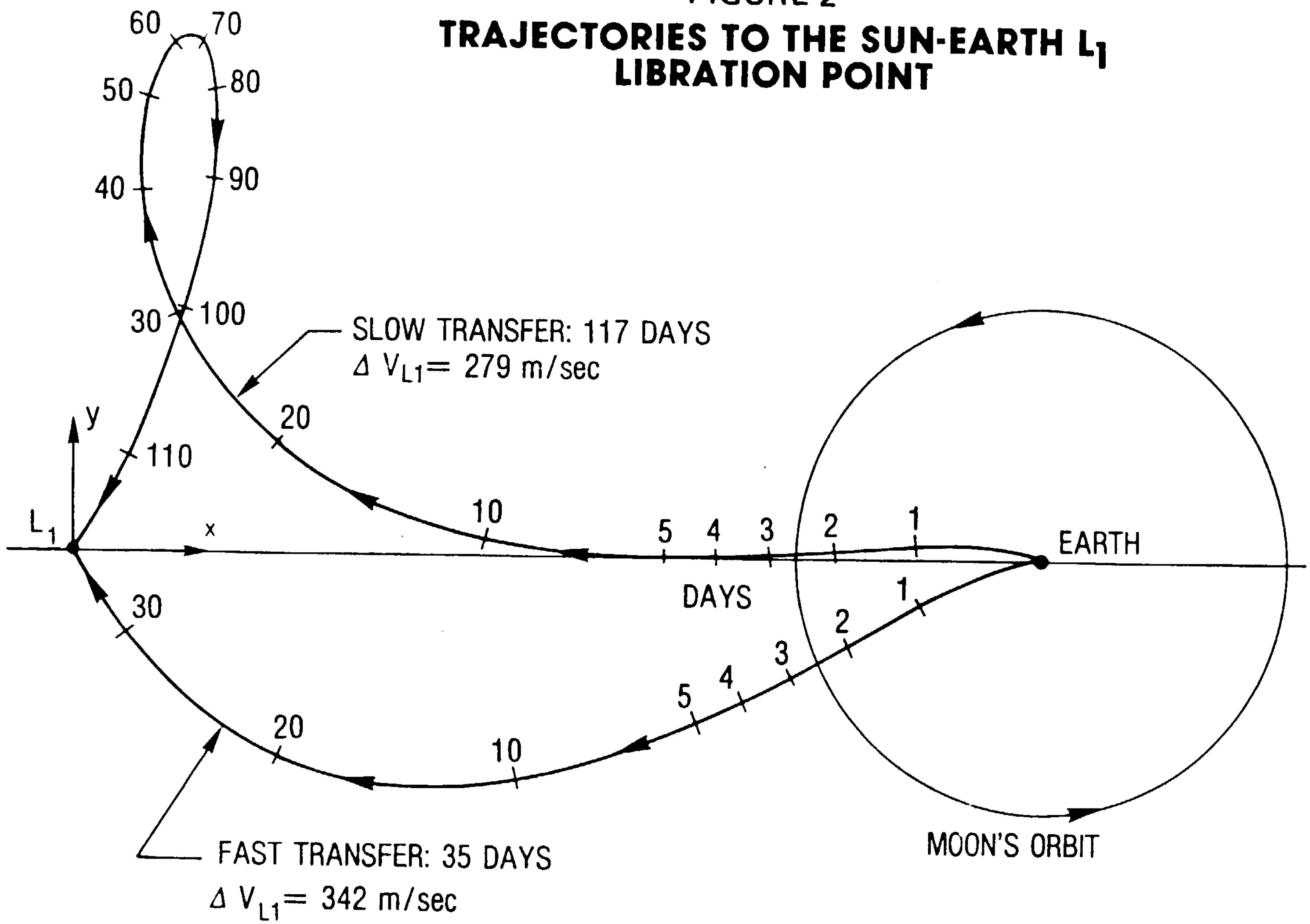


FIGURE 3

HALO ORBIT AROUND SUN-EARTH LIBRATION POINT

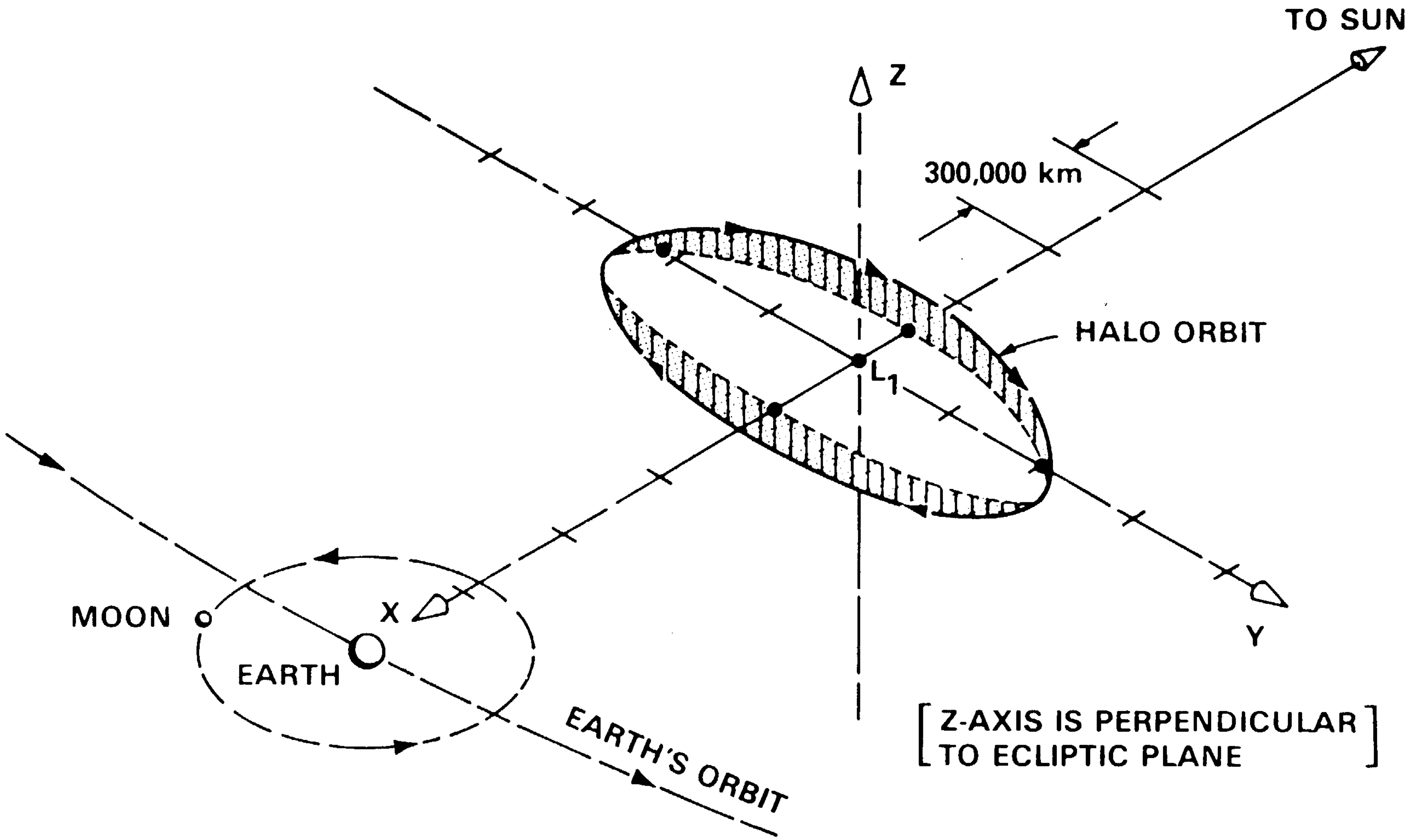


FIGURE 4

ISEE—3 TRANSFER TRAJECTORY

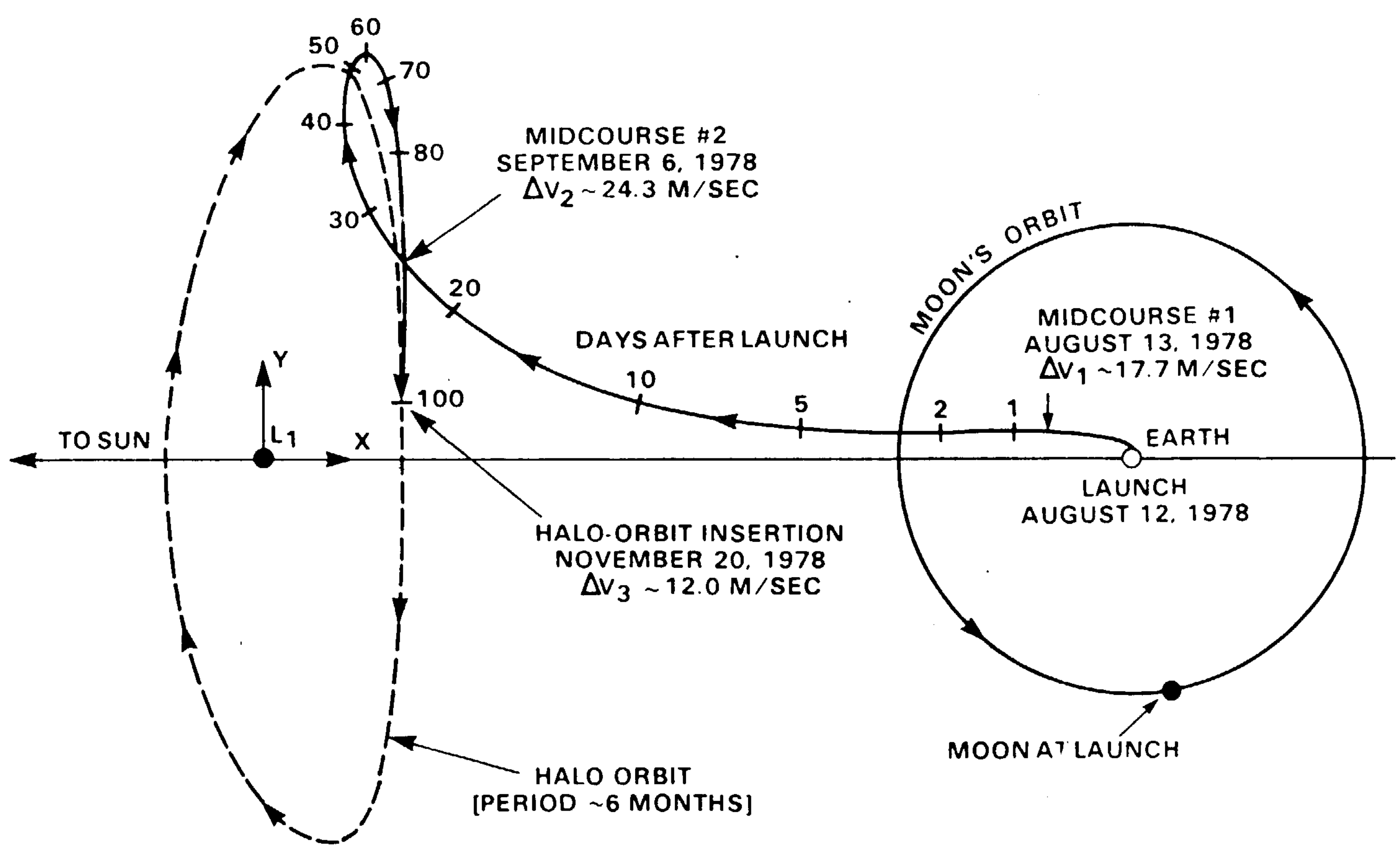
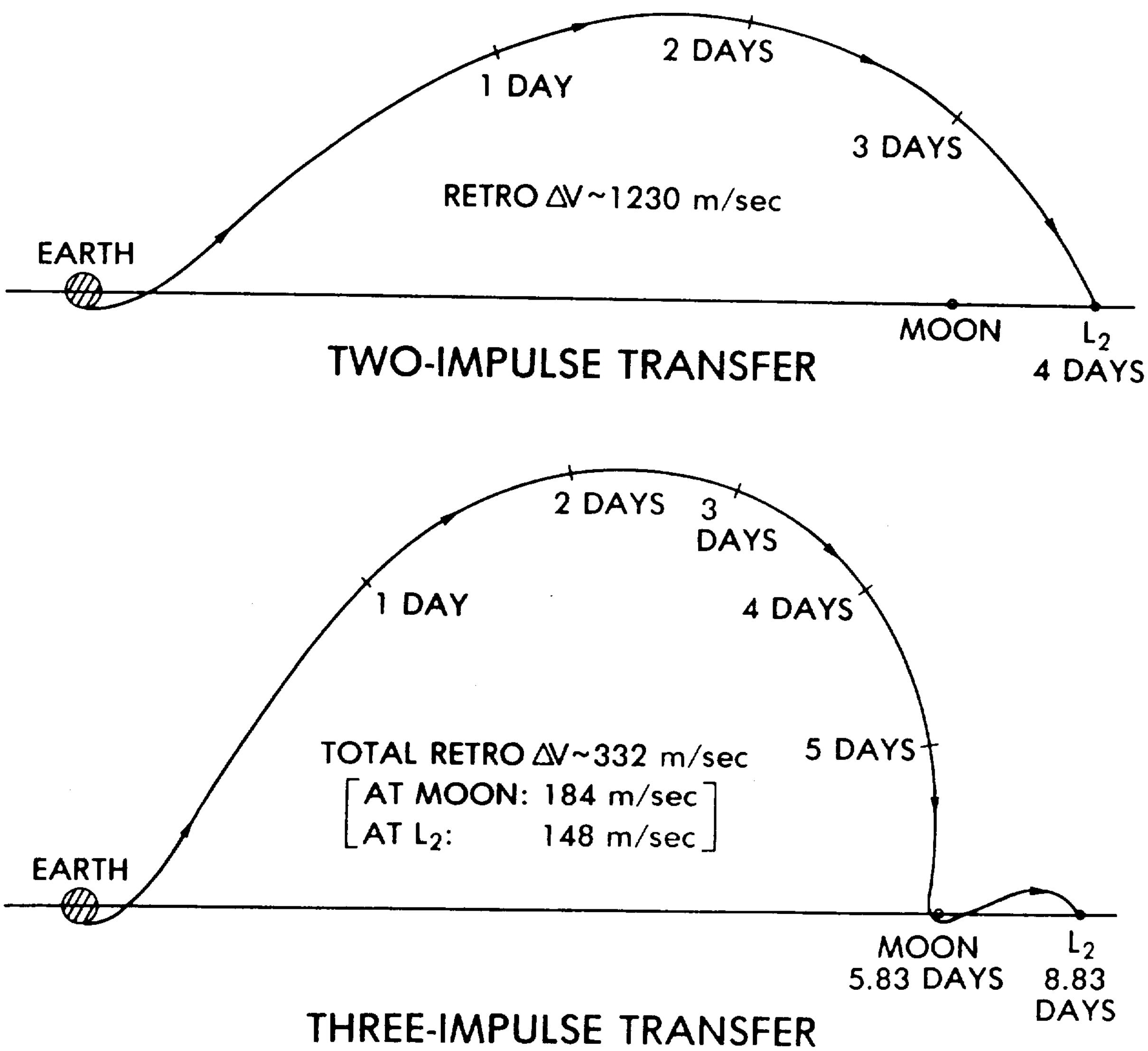


FIGURE 5

TRAJECTORIES TO VICINITY OF EARTH-MOON L_2 POINT



orbit (LLO). Comparison of this scheme with the more conventional techniques of using a single OTV between LEO and LLO showed that a significant performance advantage could be gained by using L2 staging [5].

Libration point staging may also be advantageous for Earth-Mars transportation. In this case, there are six potential locations for transfer nodes. They are the L1 and L2 points of the Sun-Earth, Earth-Moon, and Sun-Mars Systems. One or all of these points could be used. The L1 and L2 points of the Sun-Mars system average about 1.08 million kilometers from Mars, but their distance varies by more than 10% due to the eccentricity of Mars' orbit.

For instance, consider a reusable stage that is stationed in the vicinity of the Sun-Earth L1 point. This vehicle would operate between the L1 point and Mars orbit (or possibly a Sun-Mars libration point). The transfer would be initiated by applying a small impulse at L1 to bring the interplanetary shuttle vehicle (ISV) close to the Earth. Near perigee, a larger delta-V maneuver would be used to place the ISV into the proper trans-Mars trajectory. The ISV would also be used to achieve Mars orbit (either by aerocapture or propulsive maneuver). A reverse procedure would be used to return the ISV to the Sun-Earth L1 point. Resupply of the ISV would be accomplished by OTV's that travel between the L1 point and LEO. In all likelihood, these would be the same OTV's that would be used for lunar transportation.

Preliminary delta-V estimates for transfers that begin or end in a halo orbit around the Sun-Earth and Earth-Moon libration points (L1 and L2) are given in Figure 6. The second delta-V for the escape case is applied near the Earth, at the perigee of a highly eccentric transfer orbit whose initial apogee is at the departure halo orbit. These data can be used to obtain a coarse measure of the performance of the libration-point staging concept. However, as noted earlier, delta-V costs for transfers to libration-point orbits are sensitive to variations in flight time and the type of trajectory that is employed. It is hoped that a more accurate and complete summary of these delta-V costs will be available in the near future.

Lunar gravity-assist maneuvers can be used to improve performance, reduce flight times, and ease launch-window restrictions [6, 7, 8].

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FIGURE 6

PRELIMINARY ΔV ESTIMATES (M/SEC)

- LOW EARTH ORBIT → HALO ORBIT

	SUN–EARTH	EARTH–MOON
ΔV_1	3200	3150
ΔV_2	50	300

- HALO ORBIT → ESCAPE [$C_3 \sim 10\ KM^2/SEC^2$]

	SUN–EARTH	EARTH–MOON
ΔV_1	50	300
ΔV_2	500	550

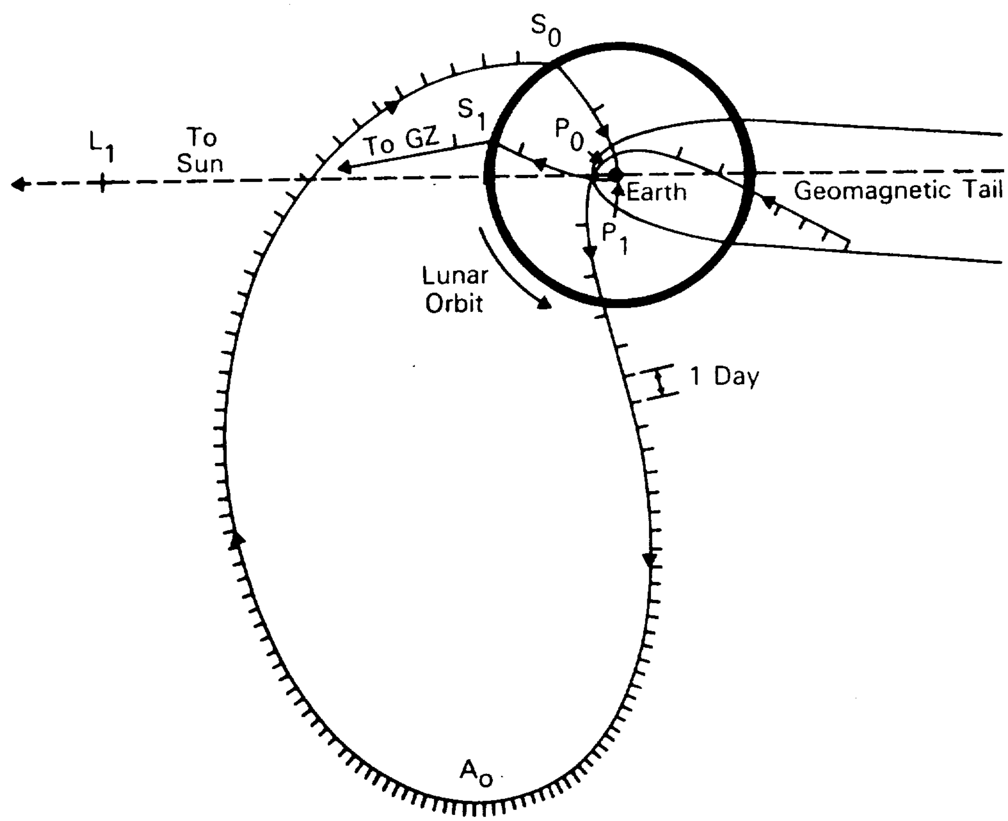
- OTHER TRANSFERS

LUNAR ORBIT → SE HALO	850
LUNAR ORBIT → EM HALO	750
SE HALO → EM HALO	300

These maneuvers are expected to play an important role in shaping the ISV and OTV flight profiles. An example of how lunar swingby maneuvers can be used to augment orbital energy is shown in Figure 7. Notice that the two lunar maneuvers have increased the C3 value from -0.5 to +4.5 km2/sec2. It may be possible to add a third swingby maneuver to attain sufficient energy to reach Venus and then on to Mars. The flight times for this scenario might be too long for crew transfers, but should be satisfactory for cargo missions.

The main idea of the transportation concept outlined here is to use the libration point region as a stepping stone to get to Mars. By starting the Mars journey from a location at the rim of the "energy well" instead of LEO, the delta-V requirement for the ISV is considerably lower. However, performance is not the only relevant factor. Tradeoffs involving flight time, launch-window flexibility, rendezvous operations, abort modes, propulsion options, etc. should be included in comparison studies of alternative mission modes for Earth-Mars transportation. A thorough system study of the competing concepts is needed to identify a baseline plan.

FIGURE 7
HYPERBOLIC DOUBLE LUNAR SWINGBY TRAJECTORY TO COMET GIACOBINI-ZINNER



EVENT	1984 DATE	G.M.T.	DISTANCE	SHADOW	C_3 (km/sec)^2
P_0	Sept. 5		9.4 R_e		
A_0	Oct. 24		307 R_e		
S_0	Dec. 17	$16^h\ 23^m$	2283 km	46^m	−0.46
P_1	Dec. 19	10 23	1.8 R_e	29	+1.20
S_1	Dec. 21	4 20	1800 km	29	+4.49

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MARS ORBIT SELECTION

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ABSTRACT

Parking orbits for a manned Mars mission are examined for ease of access to the Martian moons. Delta V plots for a variety of burns versus elliptical orbit apoapsis are included. A high elliptical orbit (24 hour period, 500 km periapsis, 20 to 30 deg. inclination) minimizes delta V to the Martian moons and Mars orbit insertion (MOI) and trans-Earth injection (TEI) delta Vs.

MARS ORBIT SELECTION

Use of an elliptical Mars orbit has been suggested by mission designers for years. It reduces both MOI and TEI delta Vs by the same amount: the difference between circular velocity at periapsis and elliptical velocity at periapsis.

Figure 1 plots MOI and TEI delta V versus apoapsis altitude (500 km periapsis) for a 1999 conjunction trajectory. MOI and TEI both continue to decrease as apoapsis increases, however, after a 48 hour period orbit is reached (500 x 57,000 km), a 1,240 m/sec reduction in both MOI and TEI has been achieved and less than 150 m/sec additional gain is possible. Figure 2 shows the same plot as Figure 1 with a different scale that makes this flattening of the MOI and TEI curves more apparent. Figures 3 and 4 show the same plots for a 2001 Venus swingby trajectory.

The next step beyond the extremely high ellipse is to let the Mission Module (the large crew module that might not enter Mars orbit at all and fly on by. The lander then enters directly from the interplanetary trajectory and ascends to rendezvous with another Mission Module flying by. The National Commission on Space has recently studied this option in some detail. Several Mission Modules will be required, depending on the scenario.

If the Mission Module is parked in Mars orbit, the parking orbit should have a periapsis as low as possible without encountering atmospheric drag. This minimizes deorbit delta V (for the lander) and for the same apoapsis also minimizes MOI and TEI.

Fig.1

MARS MOON VISITS FROM ELLIPTICAL ORBIT

1999 CONJ. MISS.(500 KM.PERIAPSIS ,0° INCLINATION)

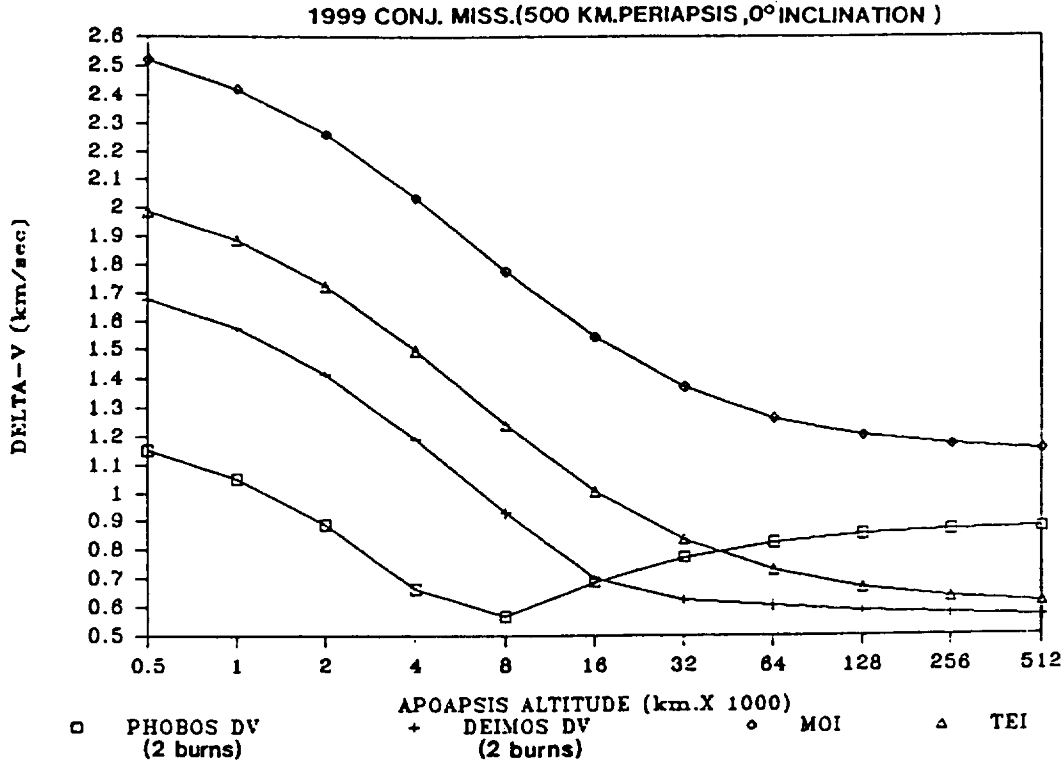


Fig.2

MARS MOON VISITS FROM ELLIPTICAL ORBIT

1999 CONJ. MISS.(500 KM.PERIAPSIS ,0°INCLINATION)

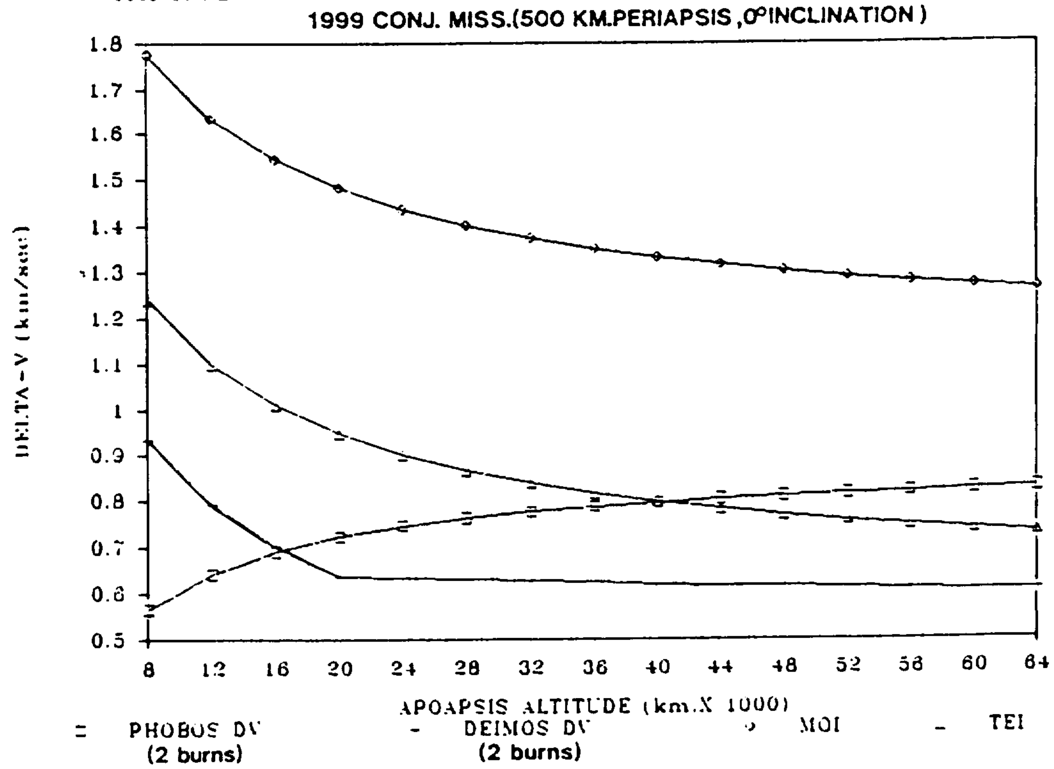


Fig.3

MARS MOON VISITS FROM ELLIPTICAL ORBITS

2001 VENUS SWINGBY MISS.(500 KM.PERI., 0° INCLINATION)

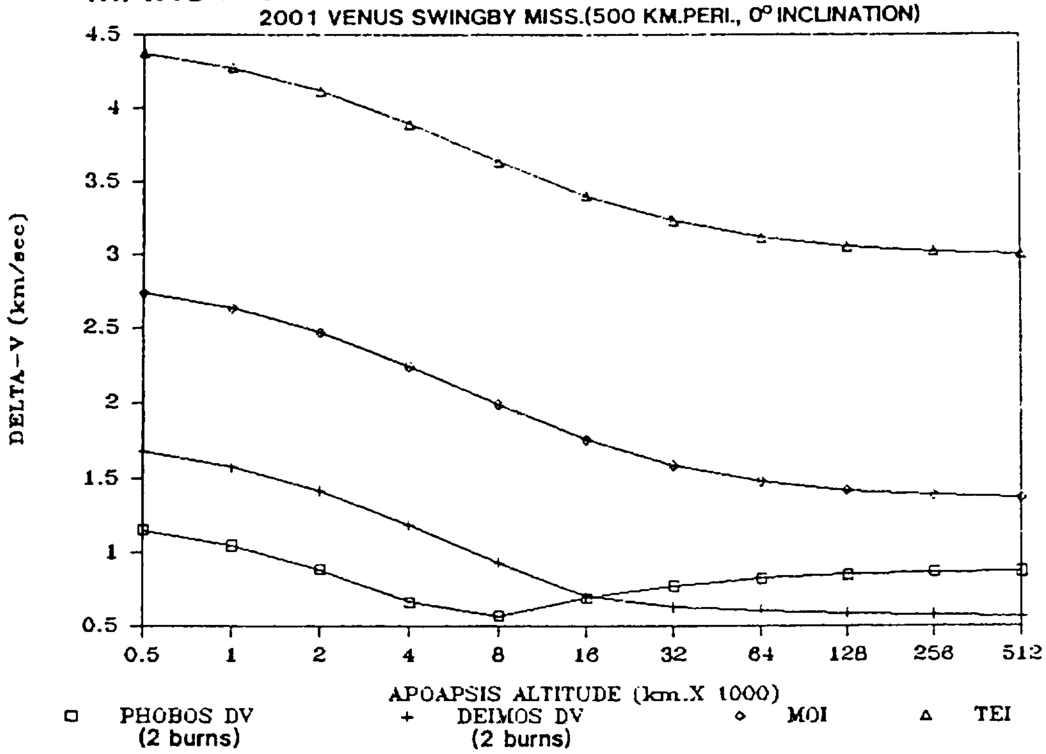
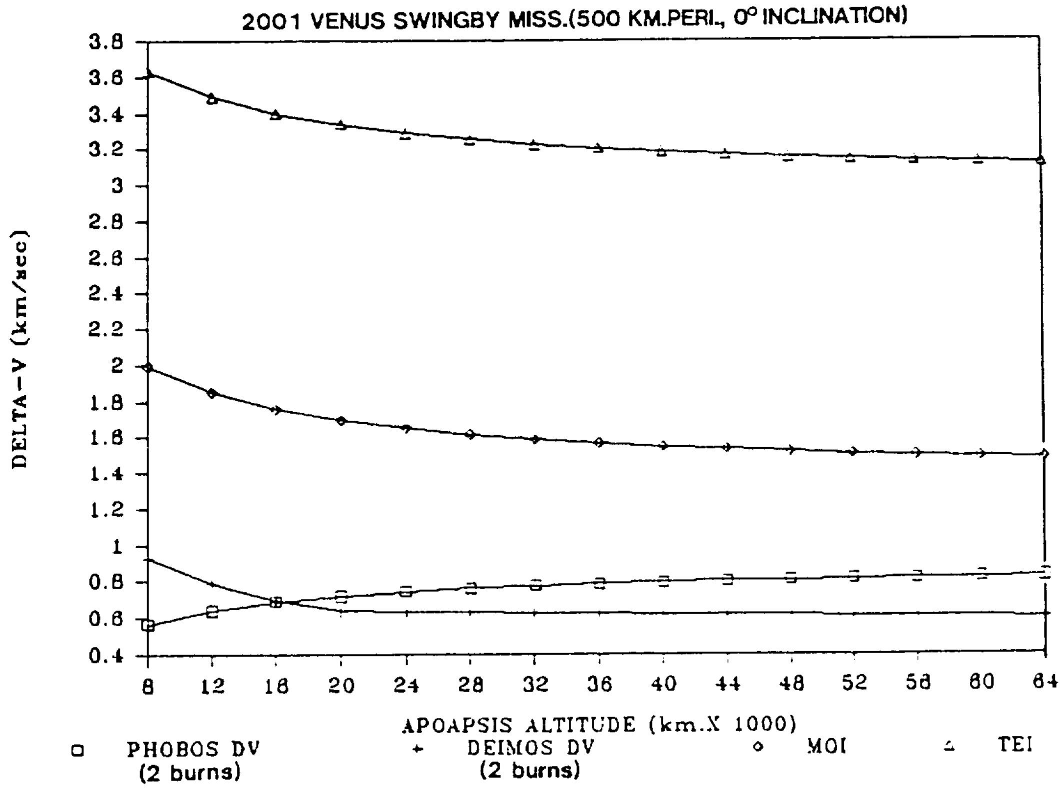


Fig.4

MARS MOON VISITS FROM ELLIPTICAL ORBIT

2001 VENUS SWINGBY MISS.(500 KM.PERI., 0° INCLINATION)



The lander ascent stage pays a penalty for high elliptical orbit. Its ascent delta V is increased by the same amount as the TEI savings. Lander deorbit is essentially aerobraked and is not penalized significantly so long as the periapsis is low.

Reference 1 plots lander mass and initial mass in low Earth orbit (LEO) versus apoapsis altitude for a variety of lander designs and overall mission propulsion and trajectory options. In general, lander mass is increased 30 % or so going from a 500 km circular to a high elliptical orbit. The effect of this small increase (a lander will mass 40 to 80 metric tons, depending on the design) on initial mass in LEO is swamped by the effect of increasing MOI and TEI by one km/sec or more each. Low circular Mars orbit therefore results in an increase in initial LEO mass over high elliptical from 30 to 100 % depending on the trajectory and propulsion scheme.

MARTIAN MOON ACCESS

Low delta V from the parking orbit to the two moons of Mars is highly desired. Both moons are in near circular, almost equatorial orbits (Phobos - 6,068 km alt., 1.02 deg. inclination, Deimos - 20,168 km alt., 1.82 deg. inclin.). Figures 1 through 4 show the in-plane transfer from various parking orbits to Phobos and Deimos. In these figures it is assumed that the line of apsides of the elliptical orbit is in the plane of the moon's orbit. The validity of this assumption for various missions requires more study.

The delta V to Phobos reaches a minimum of approximately 600 m/sec at an apoapsis of 6,000 to 8,000 km and grows thereafter to a fairly steady value of about 850 m/sec for apoapsis above 40,000 to 50,000 km. The delta V to Deimos decreases steadily to a virtually constant minimum of 650 m/sec for apoapsis above 20,000 km.

In-plane operations to the moons of Mars will not be the normal situation however. Geometry forces the parking orbit to have an inclination at least as great as the declination of both the MOI and TEI V-infinity vectors. These declinations are typically on the order of 15 to 20 degrees from the equator. In addition, some inclination is necessary to provide parking orbit precession so as to achieve a correct plane for TEI. The moons are in essentially equatorial orbits so a plane change is necesary for transport from an inclined parking orbit.

Figures 5, 6, and 7 show the delta Vs to Phobos and Deimos from ellipses of variable apoapsis inclined 30, 60, and 90 degrees to the equator respectively. All the plots show a steady, sharp reduction in moon visit delta V as apoapsis increases, indicating, the higher the ellipse, the better. Figures 8, 9, 10, and 11 show moon visit delta V from a 72, 48, 24, and 14 hour ellipses as a function of required plane change or inclination of the parking orbit. The plots are all similar. Plane change from high elliptical orbit is not expensive if it can be made at apoapsis. These figures assume the elliptical orbit line of apsides is in the plane of the moons' orbit. If approach and departure asymptotes prevent this, then these conclusions may not be applicable.

CONCLUSIONS

Orbits in the range of 48 to 24 hour periods allow plane changes to be made quite inexpensively at apoapsis and minimize moon visit, MOI, and TEI delta Vs. The 24 hour orbit (500 x 32,963 km), chosen as a baseline by many mission designers, does not have an excessive period and is not so high that serious stability problems would be expected.

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Fig.5

MARS MOON VISITS FROM ELLIPTICAL ORBITS

1999 CONJ. (500 KM.PER.,inc.=30 deg.)

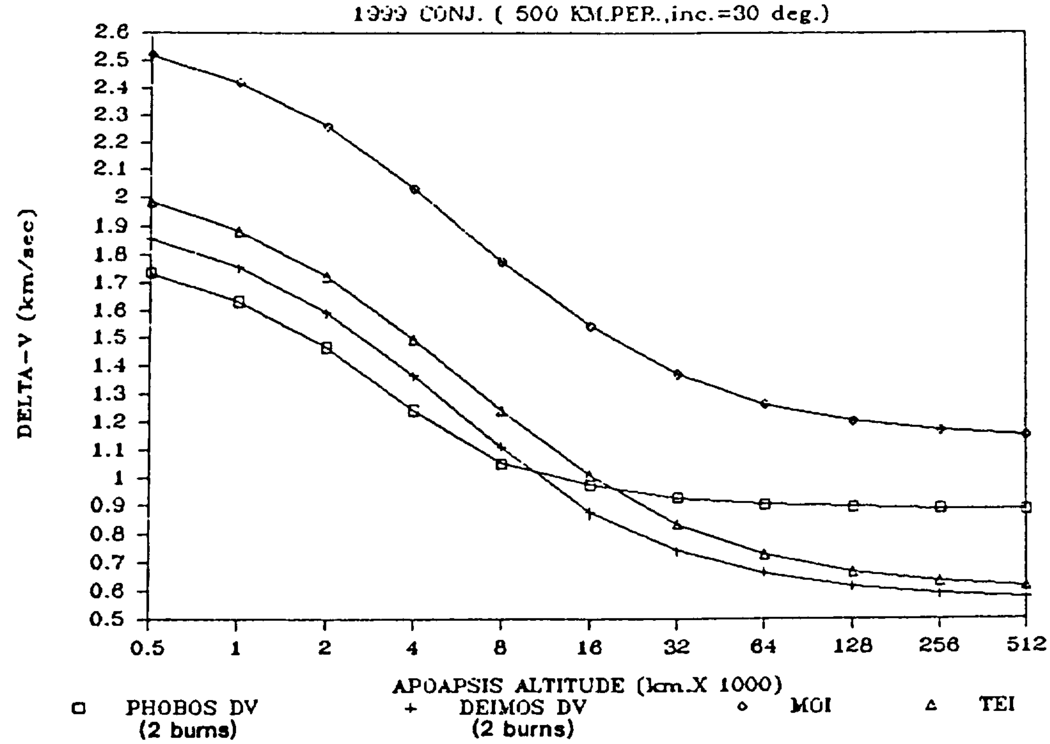


Fig.6

MARS MOON VISITS FROM ELLIPTICAL ORBITS

1999 CONJ. (500 KM.PER.,inc.=60 deg.)

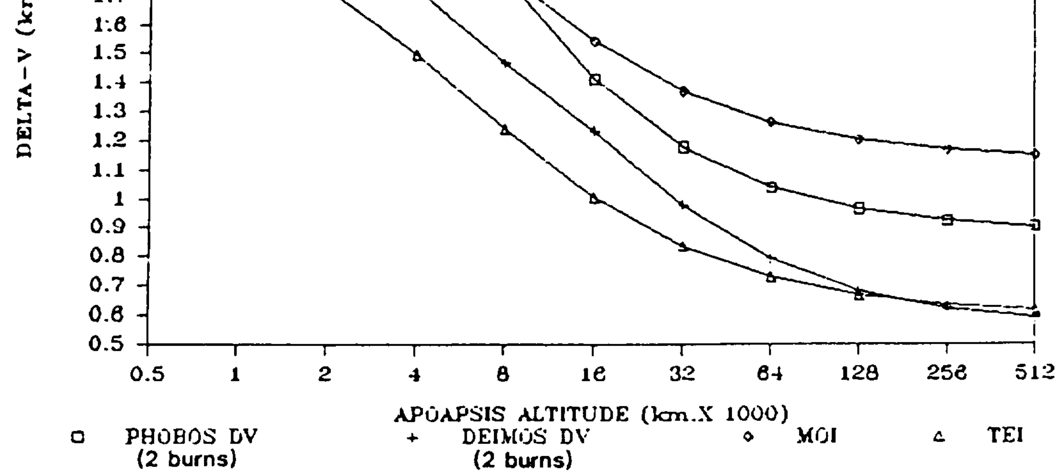


Fig.7

MARS MOON VISITS FROM ELLIPTICAL ORBITS

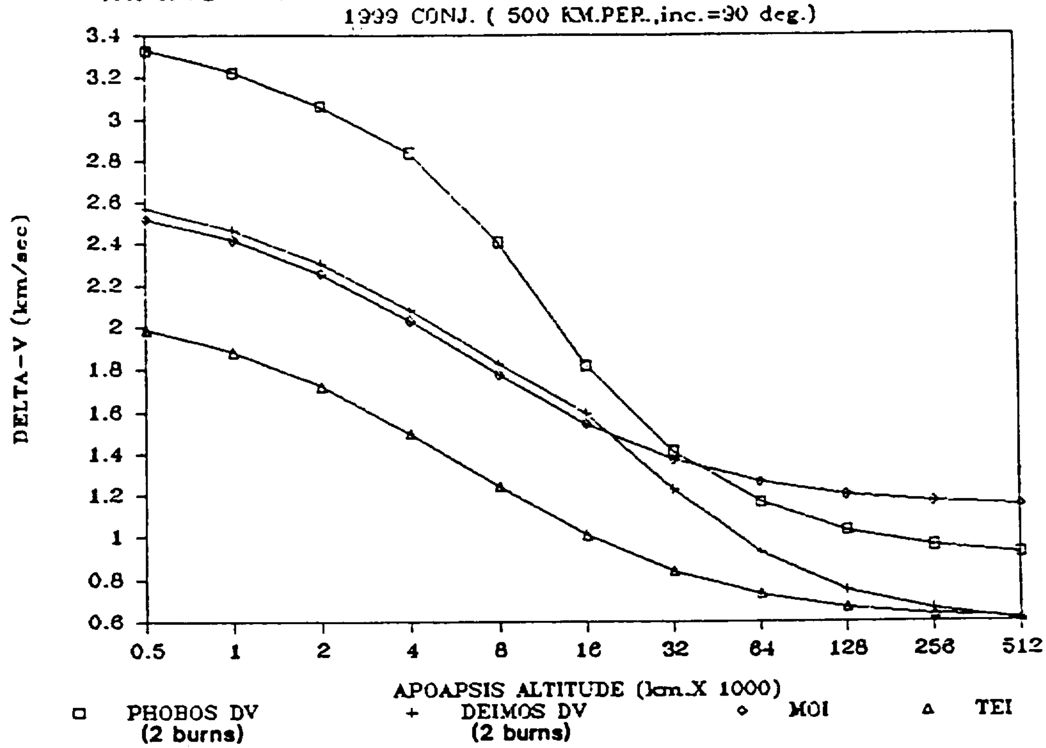


Fig.8

MARS MOON VISITS FROM ELLIPTICAL ORBITS

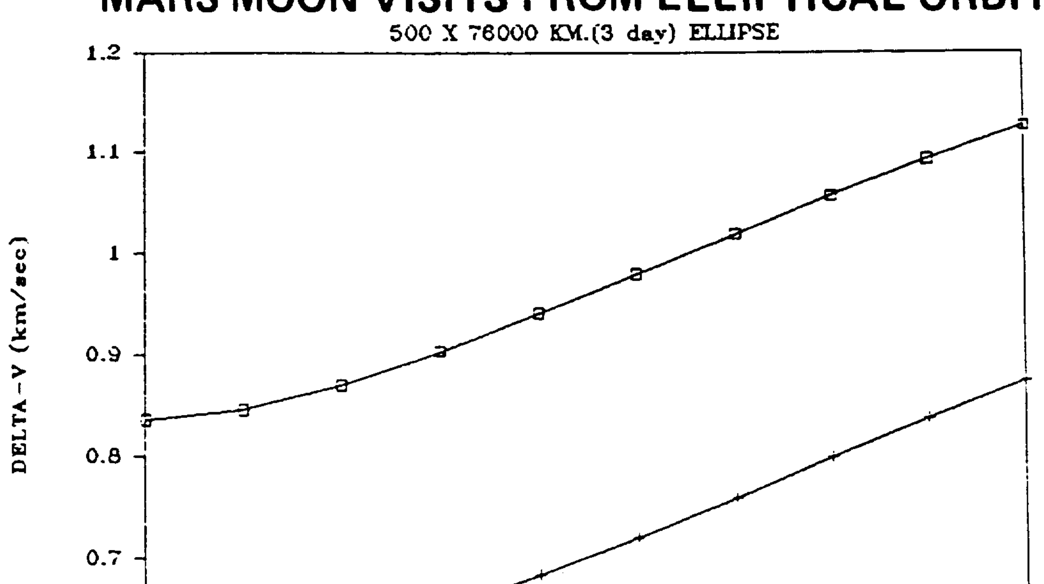


Fig.9

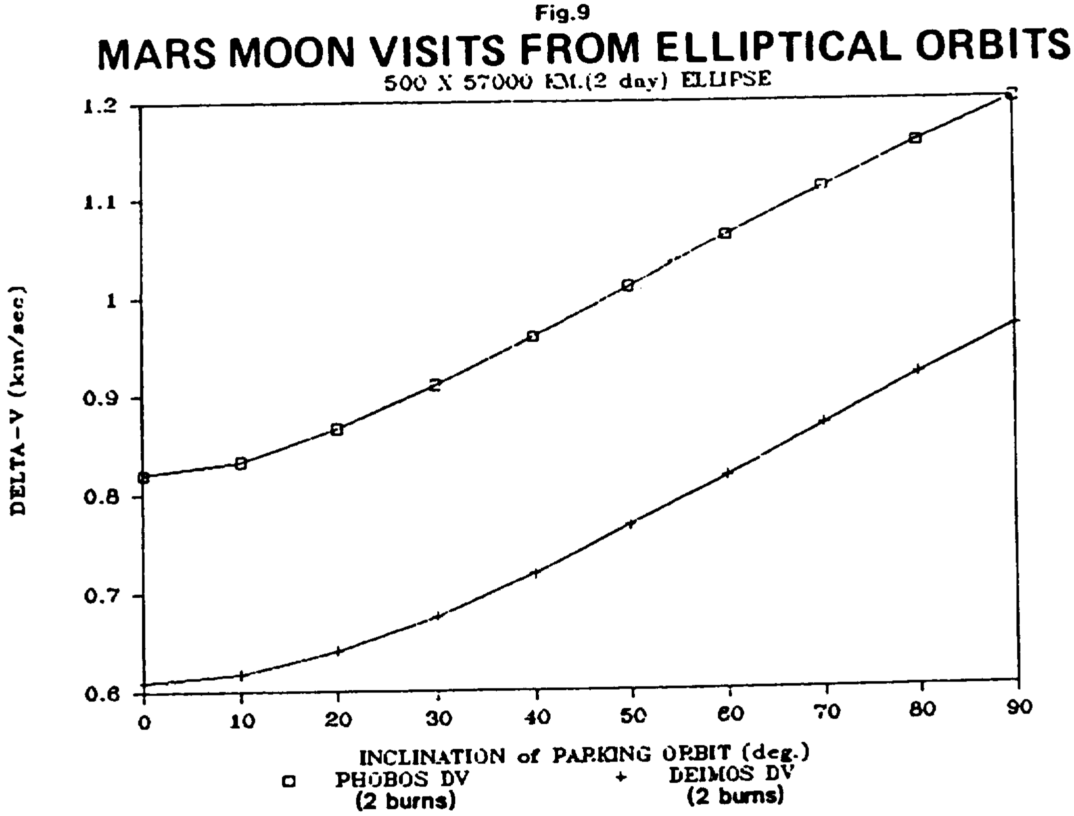

MARS MOON VISITS FROM ELLIPTICAL ORBITS
500 X 57000 KM.(2 day) ELLIPSE
1.2
1.1
1
0.9
0.8
0.7
0.6
DELTA-V (km/sec)
0
10
20
30
40
50
60
70
80
90
INCLINATION of PARKING ORBIT (deg.)
PHOBOS DV
(2 burns)
DEIMOS DV
(2 burns)

Fig.10

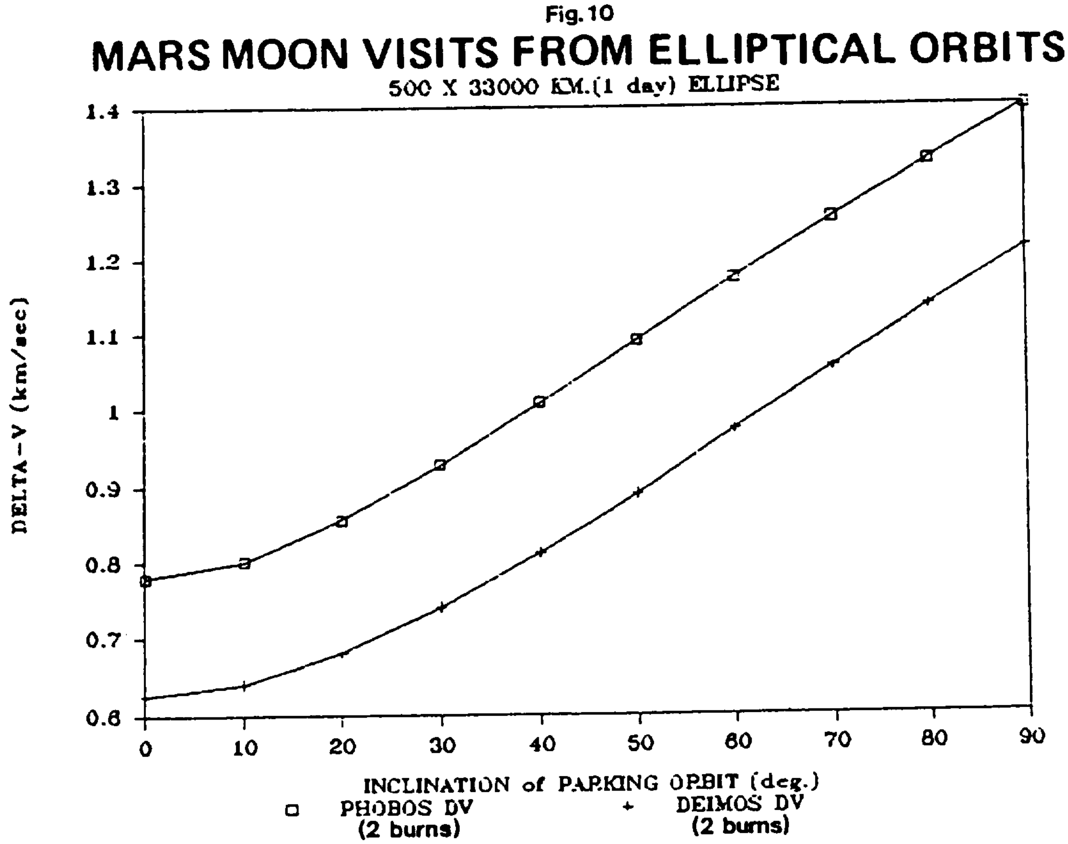
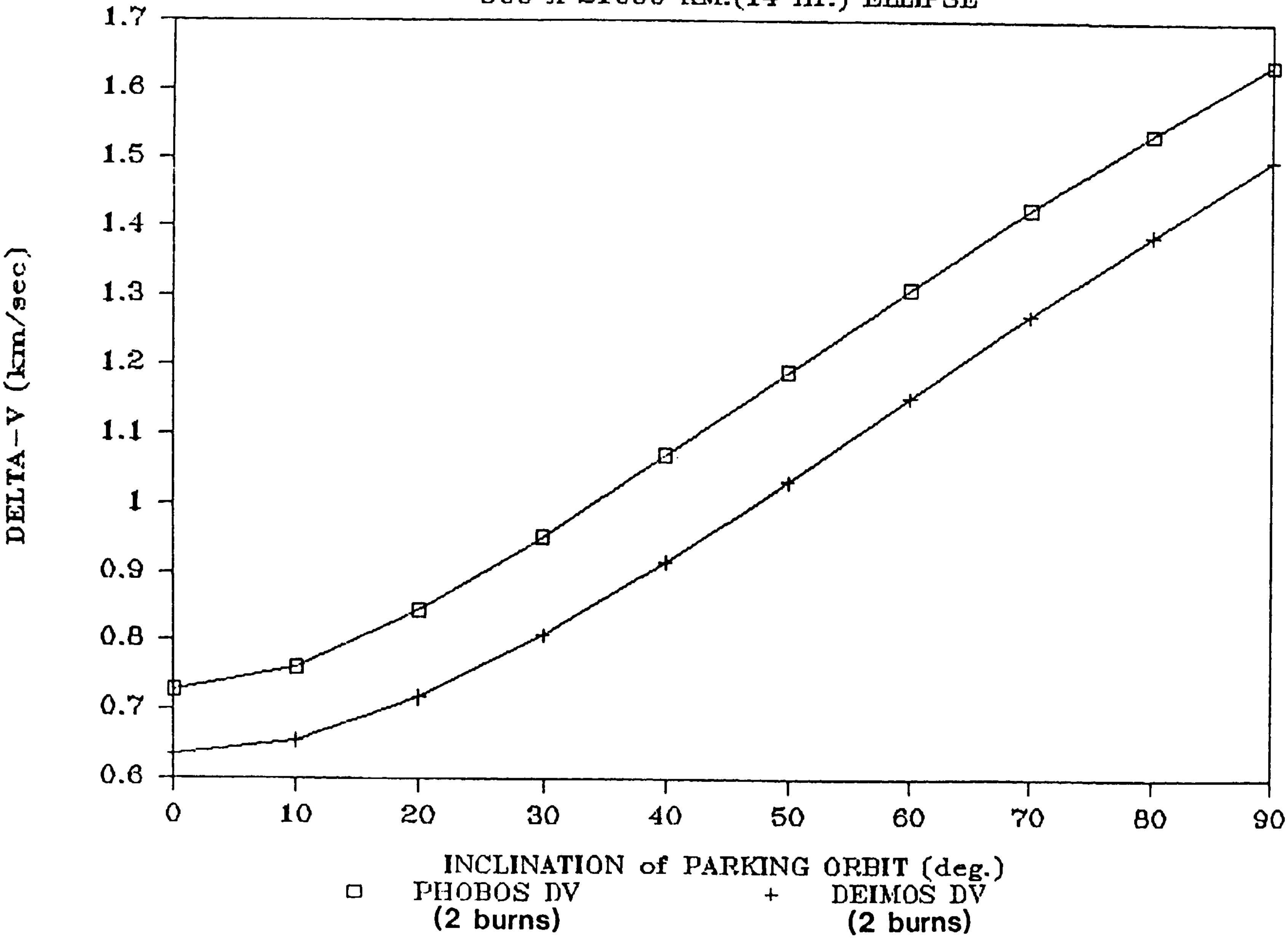

MARS MOON VISITS FROM ELLIPTICAL ORBITS
500 X 33000 KM.(1 day) ELLIPSE
1.4
1.3
1.2
1.1
1
0.9
0.8
0.7
0.6
DELTA-V (km/sec)
0
10
20
30
40
50
60
70
80
90
INCLINATION of PARKING ORBIT (deg.)
PHOBOS DV
(2 burns)
DEIMOS DV
(2 burns)

Fig.11

MARS MOON VISITS FROM ELLIPTICAL ORBITS

500 X 21000 KM.(14 hr.) ELLIPSE



N87-17729

MISSION AND VEHICLE SIZING SENSITIVITIES

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ABSTRACT

Representative interplanetary space vehicle systems are sized to compare and show sensitivity of the initial mass required in low Earth orbit to one mission mode and mission opportunity. Data are presented to show the requirements for Earth-Mars opposition and conjunction class roundtrip flyby and stopover mission opportunities available during the time period from year 1997 to year 2045. The interplanetary space vehicle consists of a spacecraft and a space vehicle acceleration system. Propellant boil-off for the various mission phases is given for the Lox/LH (Liquid Oxygen / Liquid Hydrogen) propulsion systems. Mission abort information is presented for the 1999 Venus outbound swingby trajectory. transfer profile.

INTRODUCTION

This paper presents information on performance and operational requirements and their sensitivity to flyby, Venus swingby with stopover, and conjuction class missions to Mars with stopover missions. The time period considered in developing this information is 1996 to 2045. The initial mass required in low Earth orbit was determined for each launch opportunity associated with the three classes of missions. The Mars flyby is a nonstop encounter with Mars; the Venus swingby mode opposition-class mission is a mission of less duration than the conjunction class mission but only allows a short stopover time of 60 days at Mars. Conjunction class missions require longer stopover times, up to 550 days, at Mars, but require less propellant.

Information developed in this paper is not final, as configurations of the transportation vehicles are not firm. Different values of the Mission Module (MM), Mars Excursion Module (MEM), and Mars probes may appear. The important thing to note is the relative comparative values presented for the different mission modes.

ASSUMPTIONS

Pertinent assumptions used in this study are given for the departure and capture orbit parameters, propulsion stages and planetary spacecraft elements (Figure 1). The interplanetary space vehicle was assumed to be assembled in, and depart from the 270 nm altitude, 28.5 degrees inclination, Space Station circular orbit. For the all propulsive case, required interplanetary velocity increments are achieved by three propulsive stages. The first propulsion stage effects the Earth escape maneuver, the second stage brakes the spacecraft and Earth braking stage into the Mars elliptical capture orbit and effects the escape maneuver from the Mars elliptical orbit. The third propulsion stage brakes the MM into a 24-hr elliptical orbit at Earth return. Each of the three propulsion stages' mass fractions were developed using scaling equations. For the Mars aerocapture and Earth return aerobraked case, the interplanetary velocity increments are achieved by two propulsive stages. The first and second stages were used to effect the Earth and Mars escape maneuvers, respectively.

Venus swingby, outbound, inbound, or double swingby, was used to lower the energy required for the Mars opposition class missions. The Venus closest approach distance was constrained to be equal to or greater than 0.1 planet radii (330 nm).

For the conjunction class missions, type I (<180 deg) or type II (>180 deg) Hohmann transfer trajectories were used. The Mars stopover time was optimized to achieve minimum initial weight in Earth orbit.

Interplanetary trajectory parameters (launch dates, trip times, heliocentric transfer angles, etc.) have been determined which result in a minimum total initial weight to be assembled in the Space Station's orbit. The variable propulsion stages were sized using general scaling weight laws which are dependent upon propellant loading. These coefficients are input to the interplanetary trajectory shaping program. Up to six major interplanetary maneuvers can be optimized.

INTERPLANETARY SPACE VEHICLE

The spacecraft is made up of a MM (the living and work area for the crew), a MEM and experimenter accommodations. A number of unmanned probes and orbiters are included to complement the manned activity. Major elements of the spacecraft are interconnected by pressurized tunnels allowing shirt sleeve passage between them. A minimum crew of 6 is necessary to operate the space systems and perform a reasonable scientific exploration program.

3067–85

MARS EXPLORATIONS
POST SPACE STATION MISSIONS

STUDY ASSUMPTIONS

FOR VENUS SWINGBY MISSION MODE

TIME PERIOD OF CONSIDERATION: YEAR 1997 TO 2045

PLANET DEPARTURE AND CAPTURE ORBIT PARAMETERS

EARTH DEPARTURE	CIRCULAR ORBIT ALTITUDE = 270 N. MI
MARS CAPTURE	24 HR ELLIPTIC ORBIT PERIAPSIS ALTITUDE = 270 N. MI
MARS ESCAPE	24 HR ELLIPTIC ORBIT PERIAPSIS ALTITUDE = 270 N. MI
EARTH CAPTURE	24 HR ELLIPTIC ORBIT PERIAPSIS ALTITUDE = 270 N. MI

HELIOCENTRIC PROFILE

VENUS SWINGBY MODE (OUTBOUND, INBOUND OR DOUBLE SWINGBY)

VENUS MINIMUM CLOSET APPROACH EQUAL 0.1 PLANET RADII (330 N. MI)

CONJUNCTION CLASS MISSION USES TYPE I OR TYPE II TRAJECTORIES

INTERPLANETARY SPACE VEHICLE (1)

			MARS FLYBY	OPPOSITION MISSION	CONJUNCTION MISSION
SPACECRAFT:	MISSION MODULE WEIGHT	=	88,500 (2)	113,633	160,000
	MARS EXECURSION MODULE WEIGHT	=	N/A	133,047	133,047
	PROBES WEIGHT	=	20,000	44,523	60,000

PROPULSION STAGES	FIRST STAGE	SECOND STAGE	THIRD STAGE
MASS FRACTION (λ)	S. EQ.	S. EQ.	S. EQ.
I_{SP} (SEC)	461	482	482
PROPELLANT	LOX/LH_2	LOX/LH_2	LOX/LH_2
I_{SP} (SEC)	331.6	345.4	345.4
PROPELLANT	N_2O_4/MMH	N_2O_4/MMH	N_2O_4/MMH

(1) POST YEAR 2000 USES DIFFERENT WEIGHTS THAN THESE

(2) INCLUDES A 7,500 LB EARTH RETURN MODULE

FIGURE 1

Two interplanetary space vehicle configurations for the opposition class mission via an outbound Venus swingby for the year 1999 opposition opportunity are given in Figure 2. Information for each of the propulsion stages and the total interplanetary vehicle weight is given. The total initial mass required in the Space Station orbit for the all propulsive configuration is 3,575,321 lb; for a configuration that utilized aerobraking at Mars capture and Earth return, the total initial mass required in the Space Station orbit is 1,433,294 lb.

Earth return with aerobrake entry has been analyzed and results show that with an Earth return C_3 greater than 25 km^2/sec the g-load will be in excess of 5 g's. This high g-load probably cannot be tolerated by the crew. Earth return with C_3 greater than 25 km^2/sec^2 will require propulsive braking in order to stay within the g-load constraint.

MISSION AND VEHICLE SIZING SENSITIVITY

In mission profile design and vehicle sizing there are many variables which influence the resultant mission profile and space vehicle configuration. Some of the more significant variables include: (1) Earth launch window duration, (2) Stay time at Mars, (3) MM weight, and (4) MEM weight, including Mars lander capsule weight and Mars ascent capsule weight.

Sensitivities to the Earth launch window duration and Mars stay time for the 1997 and 1999 opportunity Venus swingby mission profiles is given in Table 1. For the 1997 opportunity, a 40 day stay time at Mars and an Earth launch window of 10 days requires 1,591,700 pounds initial weight in low Earth orbit to perform the mission. A 60 day stay time and an Earth launch window of 30 days will require 1,949,700 pounds of initial weight in low Earth orbit to perform from the 1997 launch opportunity; this weight is an increase of 22 1/2 percent over a 40 day stay time and 10 day launch window case. The 1999 launch opportunity is not as sensitive as the 1997 opportunity. A 60 day stay time and a launch of 30 days requires an initial weight in low Earth orbit of 1,434,200 pounds; this weight is an increase of 6 3/4 percent over a 40 day stay time at Mars and a 10 day Earth launch window.

The interplanetary space vehicle sensitivity to changes in MM and MEM weight is shown in Table 2 for an aerobrake (at Mars capture and Earth return) space vehicle configuration. An initial weight of

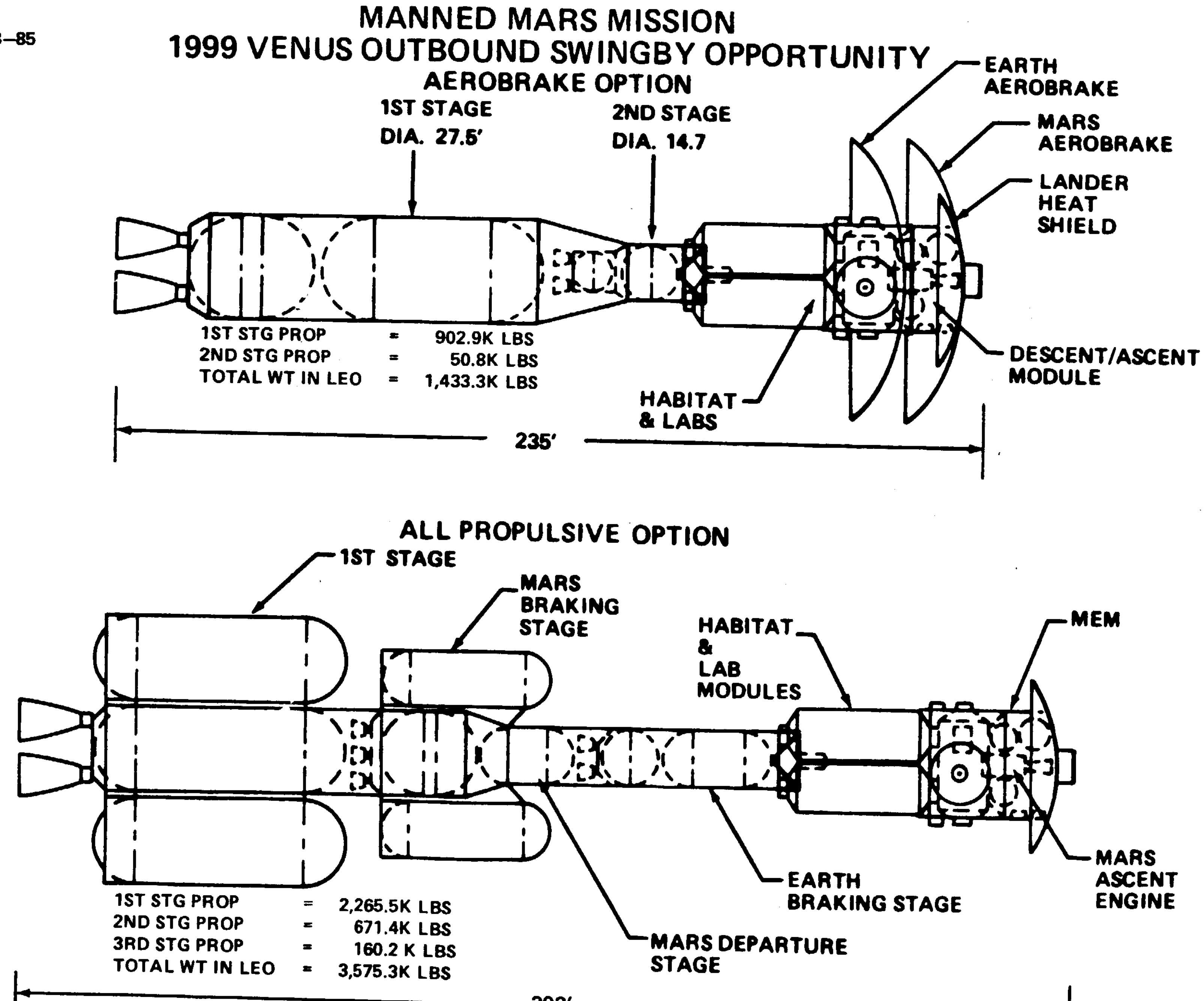

FIGURE 2 INTERPLANETARY VEHICLE CONFIGURATION

3059–85

MANNED MARS EXPLORATION
ALL CHEMICAL PROPULSION MANEUVERS
AEROBRAKE AT MARS AND EARTH RETURN
WEIGHT REQUIRED IN 1000'S OF LBS

DOUBLE SWINGBY FOR 1997 OPPOSITION OPPORTUNITY

MARS STAY TIME ↓ / EARTH LAUNCH WINDOW →	10	30
40	116.6 (1)	125.8
	1008.5 (2)	1115.5
	1591.7 (3)	1712.0
60	179.5	200.0
	1129.4	1270.4
	1782.6	1949.7

OUTBOUND SWINGBY FOR 1999 OPPOSITION OPPORTUNITY

MARS STAY TIME ↓ / EARTH LAUNCH WINDOW →	10	30
40	60.0	60.0
	825.3	901.1
	1343.2	1421.5
60	64.6	64.6
	833.3	908.8
	1356.3	1434.2

(1) 2ND STAGE PROPELLANT
(2) 1ST STAGE PROPELLANT
(3) TOTAL WEIGHT IN EARTH ORBIT

TABLE 1

MANNED MARS EXPLORATION
1999 VENUS OUTBOUND SWINGBY OPPORTUNITY
WEIGHT SUMMARY (IN LBS)
AEROBRAKE AT MARS AND EARTH RETURN

	NOMINAL CASE	Δ MM BY 15%	Δ MEM BY 15%
2 STAGES DRY WEIGHT	89,529	91,739	90,779
2 STAGES PROP WEIGHT	974,064	1,037,407	1,012,306
MISSION MODULE WEIGHT	134,585	154,773	134,585
MARS EXCURSION MODULE WEIGHT	137,881	137,881	158,563
PROBE & MARS AEROBRAKE WEIGHT	98,156	98,156	98,156
TOTAL WEIGHT IN LEO	1,434,215	1,519,956	1,494,389

TABLE 2

3061–85

MANNED MARS MISSION

EXCHANGE FACTOR

TYPE PROPELLANT	$\frac{\partial \text{ MEM WT}}{\partial \text{ LANDER WT}} \left(\frac{\text{LB}}{\text{LB}}\right)$	$\frac{\partial \text{ MEM WT (LB)}}{\partial \text{ ASCENT WT (LB)}}$
N_2O_4/MMH	1.37	8.30
LOX/MMH	1.33	7.08

TABLE 3

1,434,215 pounds is required in low Earth orbit for the nominal case. If the MM weight is increased by 15 percent, the initial weight in low Earth orbit is increased by 6 percent over the nominal case. If the MEM weight is increased by 15 percent, the initial weight is an increased by 4.2 percent over the nominal case.

The MEM initial weight sensitivity to variation in Mars lander capsule and Mars ascent capsule weights is given in Figure 3. The exchange factors are given in Table 3 for two different types of propellants, N_2O_4/MMH and LOX/MMH.

The initial mass required in low Earth orbit for each mission opportunity is given in Table 4 and Figures 4 and 5. The initial mass required ranges from 1,280,001 to 3,575,321 lb for LOX/LH propellant. These values do not include propellant boiloff in low Earth orbit during orbital assembly time. The initial mass required in low Earth orbit for the 1999 opposition outbound Venus swingby using N_2O_4 /MMH propellant is 8,869,090 lb. The initial mass in Earth orbit can be equated to cost and used to determine the most favorable mission opportunities and the most effective type of propellant for the propulsive stages.

PROPELLANT BOILOFF

The Mars mission is characterized by different mission environments including LEO buildup, interplanetary transit, and Mars orbit. The passive thermal protection on the cryogenic propellant tanks consists of 1 to 4 inches of MLI on the first stage and 4 inches MLI on the second and third stage tanks. Vapor cooled shields are utilized on all tanks. Table 5 relates cryogenic boiloff rate ranges for the different mission environments using the all cryogenic vehicle configuration.

The boiloff rates for LEO were calculated with 1 and 4 inches of MLI on the first stage. The boiloff rates in LEO are relatively high due to large tank areas and albedo (reflected thermal energy from the Earth) heating effects. The interplanetary transit mission phase is characterized by relatively low boiloff because of reduced vehicle tank area (stages two and three) and lower environmental heating. The lower heating is contributable to transit vehicle orientation during flight to minimize solar flux on tank wall areas and greater distance from the Earth. Any deviation from the preferred orientation will result in increased boiloff. The Mars orbit stages experience medium boiloff rates

1567–85

MARS EXPLORATION MEM DESCENT AND ASCENT FOR 165° INCLINATION SENSITIVITY DATA

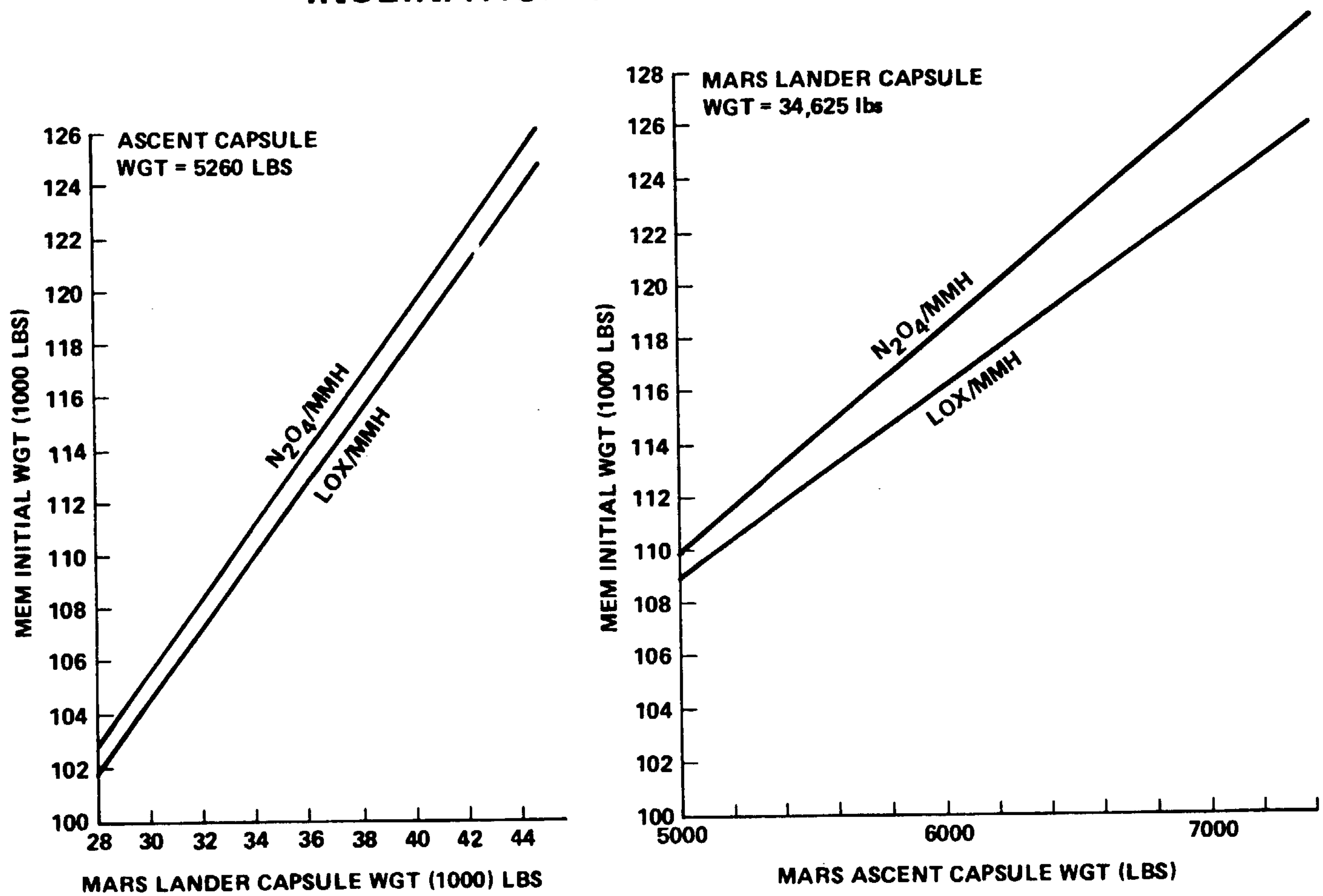

FIGURE 3

2846–86

MARS EXPLORATION
MASS IN EARTH ORBIT REQUIREMENTS

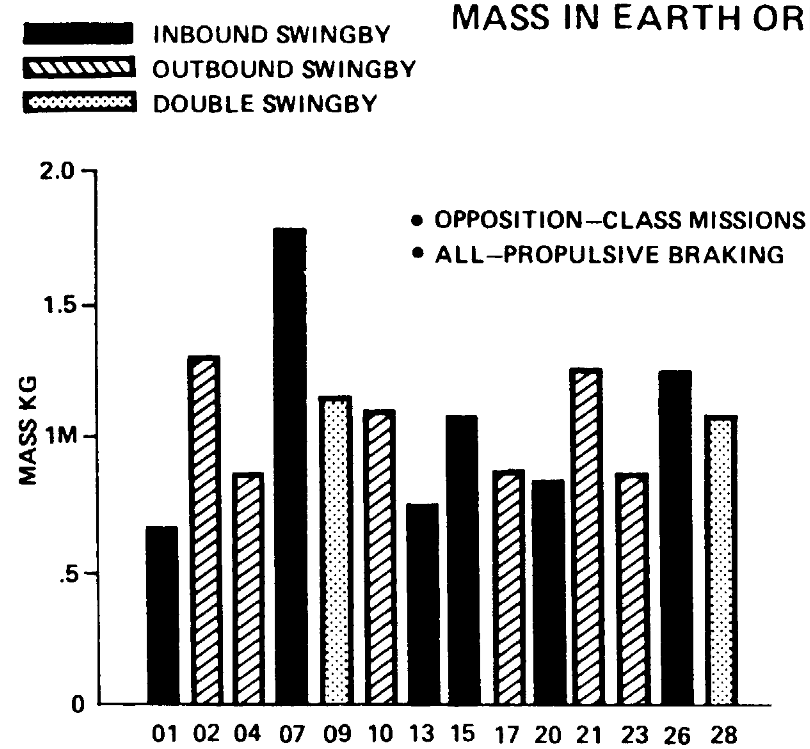

FIGURE 4

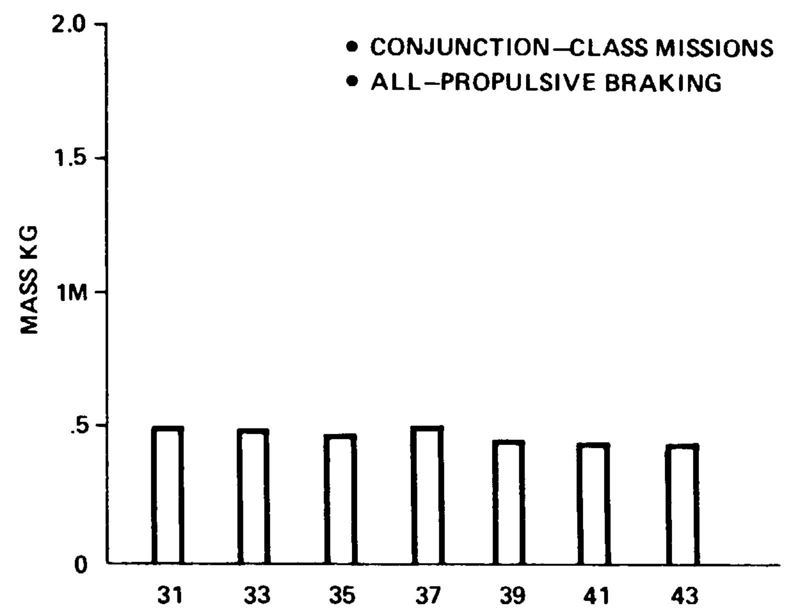

FIGURE 5

MANNED MARS EXPLORATION
PROPULSION STAGE SIZE AND LEO WEIGHT REQUIRED

MISSION/ VEHICLE	ΔV EARTH ESCAPE (KM/SEC)	ΔV MARS CAPTURE (KM/SEC)	ΔV MARS ESCAPE (KM/SEC)	ΔV EARTH RETURN (KM/SEC)	PROPELLANT LOADING (1000 LBS) 1ST STAGE	2ND STAGE CORE	2ND STAGE TANKS	3RD STAGE	TOTAL WEIGHT IN LEO (1000 LBS)
1999 ONE YR FLYBY ALL PROPULSION	7.0690	0	0.406	6.1636	1,075.2 (1)	15.9	N/A	45.8 (2)	1,346.4
1999 VENUS SWINGBY ALL PROPULSION	4.4289	2.7569	1.6238	3.7246	2,265.5	594.7	76.8	160.2	3,575.3 (3)
1999 VENUS SWINGBY AERO CAPTURE @ MARS AND EARTH	4.4289	0	1.6238	0	902.9	50.8	N/A	N/A	1,433.3
1999 CONJUNCTION ALL PROPULSION	3.7169	1.4365	0.9294	1.1135	1,004.9	211.2	26.4	47.6	1,764.7
1999 CONJUNCTION AEROCAPTURE @ MARS AND EARTH	3.7169	0	0.9294	0	724.7	43.7	N/A	N/A	1,280.0
1999 VENUS SWINGBY ALL PROPULSION (4)	4.4289	2.7569	1.6238	3.7246	6,673.2	1,436.0	N/A	262.6	8,869.1
2001 VENUS SWINGBY ALL PROPULSION	4.1535	1.6744	3.2567	1.4963	1,483.3	426.6	37.5	48.1	2,431.6
2001 VENUS SWING BY AERO CAPTURE @ MARS AND EARTH	4.1535	0	3.2567	0	977.3	157.8	N/A	N/A	1,602.0

(1) THIS VALUE COULD BE REDUCED CONSIDERABLE BY OPTIMIZING MISSION TIME

(2) EARTH RETURN WITH RETURN CAPSULE ONLY

(3) EQUALS 3,772.1 FOR REUSABLE FIRST STAGE

(4) N_2O_4/MMH PROPELLANT ALL OTHER CONFIGURATIONS USES LOX/LH_2 PROPELLANT

TABLE 4

MANNED MARS MISSION
1999 VENUS OUTBOUND SWINGBY
TOTAL VEHICLE BOILOFF RANGES ~ LB/HR

ENVIRONMENT	BOILOFF RANGE LH_2	BOILOFF RANGE LO_2
LOWER EARTH ORBIT – 1 IN MLI ON FIRST STAGE	16.2 ~ 18.3	10.4 ~ 12.6
LOWER EARTH ORBIT – 4 IN MLI ON FIRST STAGE	6.8 ~ 9.0	3.8 ~ 5.0
*VEHICLE TRANSIT	.06 – .10	.04 – .08
MARS ORBIT	.7 ~ 1.0	.5 ~ .8

ASSUMPTIONS

- ALL CRYO CONFIG.
- STAGE 1 W_p = 2266K
- STAGE 2 W_p = 671.4K
- STAGE 3 W_p = 160.2K
- NO ACTIVE TPS
- VCS UTILIZED
- 4 IN. MLI ON 2ND & 3RD STAGES

* BASED ON PREFERRED VEHICLE ORIENTATION. DEVIATION FROM ORIENTATION WILL RESULT IN INCREASED BOILOFF RATES.

TABLE 5

through potential environmental heating due to vehicle orientations driven by mission requirements. Preferred orientation in Mars orbit to reduce environmental heating would lower the boiloff rate.

1999 VENUS OUTBOUND SWINGBY MISSION ABORT

In the final selection of trajectories for the manned Mars stopover missions, many factors other than vehicle weight must be considered. Abort capability of the vehicle is one of these factors. It is, therefore, necessary to plan and prepare for the possible irreparable failures at some point during the mission.

Abort situations can be characterized as occuring in two different phases of the mission which can be defined as (1) Earth departure phase and (2) Heliocentric orbit phase. If abort maneuvers are executed within 30 minutes after trans-Venus injection, return to low Earth orbit can be achieved within two days. The interplanetary vehicle is within Earth's gravity sphere of activity up to 1 3/4 days after trans-Venus injection; if abort maneuvers are undertaken within this time span, an elliptical orbit return to low Earth orbit can be achieved within 18 days.

Heliocentric orbit phase is reached 1 3/4 days after trans-Venus injection. The interplanetary vehicle (aero capture at Mars and aero brake at Earth return) delta V capability after trans-Venus injection is in excess of 9 km/sec for a small Earth return capsule; the Mars excursion module has a 7.2 km/sec delta V capability and the second stage main propulsion system has a 1.6 km/sec delta V capability with the total mission module weight of 113,633 lb. If mission abort is executed sometime less than 40 days into the mission, an Earth return rendezvous trajectory can be achieved which returns back to low Earth orbit within 80 to 250 days. After 180 days into the mission, the interplanetary vehicle is committed to a Mars flyby which would return to Earth in 560 days.

The above description of recovery from orbit conditions emphasized minimum delta V requirement for the return to Earth trajectory. Other abort situations (i.e., abort after 40 days, braking into orbit at Mars, Mars landing, Mars escape, etc.) need to be studied in more detail.

CONCLUSION

Comparative and sensitivity data have been developed for an opposition class Mars flyby and 60 day stopover missions to Mars. Also, data were developed for conjunction class stopover missions. The 60 day stopover mission utilized the.Venus swingby mode in order to reduce the propulsive energy required.

There is a great variation in initial mass required in low Earth orbit for the all propulsive interplanetary space vehicles over a number of mission opportunities. This variation is due to the eccentricity of Mars orbit which has a perihelion distance of 1.38 A.U. and an apahelion distance of 1.66 A.U. The wide variation in initial mass may be reduced by aerocapture at Mars and Earth return or by only returning to Earth capture orbit with a small Earth return module and leaving the heavier Mission Module in an Earth-Mars periodic orbit. The variation in initial mass for the conjunction class mission over a number of mission opportunities is relatively small because there is more freedom to optimize the outbound transfer to Mars and the return transfer to Earth.

Mission abort capability, for the 1999 Venus outbound opportunity, can extend out to 40 days after trans-Venus injection. In order to minimize required weight in low Earth orbit, 4 inches of MLI on all stages seems to be the most effective.

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N87-17730

MARS MISSION CONCEPTS AND OPPORTUNITIES

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ABSTRACT

Trajectory and mission requirement data are presented for Earth - Mars opposition and conjunction class roundtrip flyby and stopover mission opportunities available between 1997 and 2045. The opposition class flyby mission uses direct transfer trajectories to and on return from Mars. The opposition class stopover mission employs the gravitational field of Venus to accelerate the space vehicle on either the outbound or inbound leg in order to reduce the propulsion requirement associated with the opposition class mission. The conjunction class mission minimizes propulsion requirements by optimizing the stopover time at Mars.

INTRODUCTION

Ballistic mission profiles are convenient flight path approximations based on the use of instantaneous velocity impulses (ΔV) near the planetary bodies to enter free-fall (coasting) trajectory segments between the planets. The free-fall segments are represented by "two-body" equations that result from integration of the differential equations describing the motion of a space vehicle in the force field of a control gravitational body. To achieve the velocity impulse, high thrust chemical or nuclear propulsive systems were assumed with initial thrust acceleration > 0.1g.

Data are presented for the Mars opposition and conjunction class mission profiles. These profiles are pictorially described in Figure 1. Two categories of the opposition class profiles were considered: a Mars flyby with no landing or stay at Mars; and a Mars stopover mission with a short stay time of 60-80 days. These are relatively high energy missions, either at departure from or arrrival at one of the planets. The conjunction class mission profile requires low Hohmann energy transfer trajectories which are achieved by optimizing the stay time, from 300 to 550 days, at Mars. Another type of Earth-Mars-Earth trajectory is the free-fall approximately 2 year periodic orbit which may find use as an orbiting connecting node.

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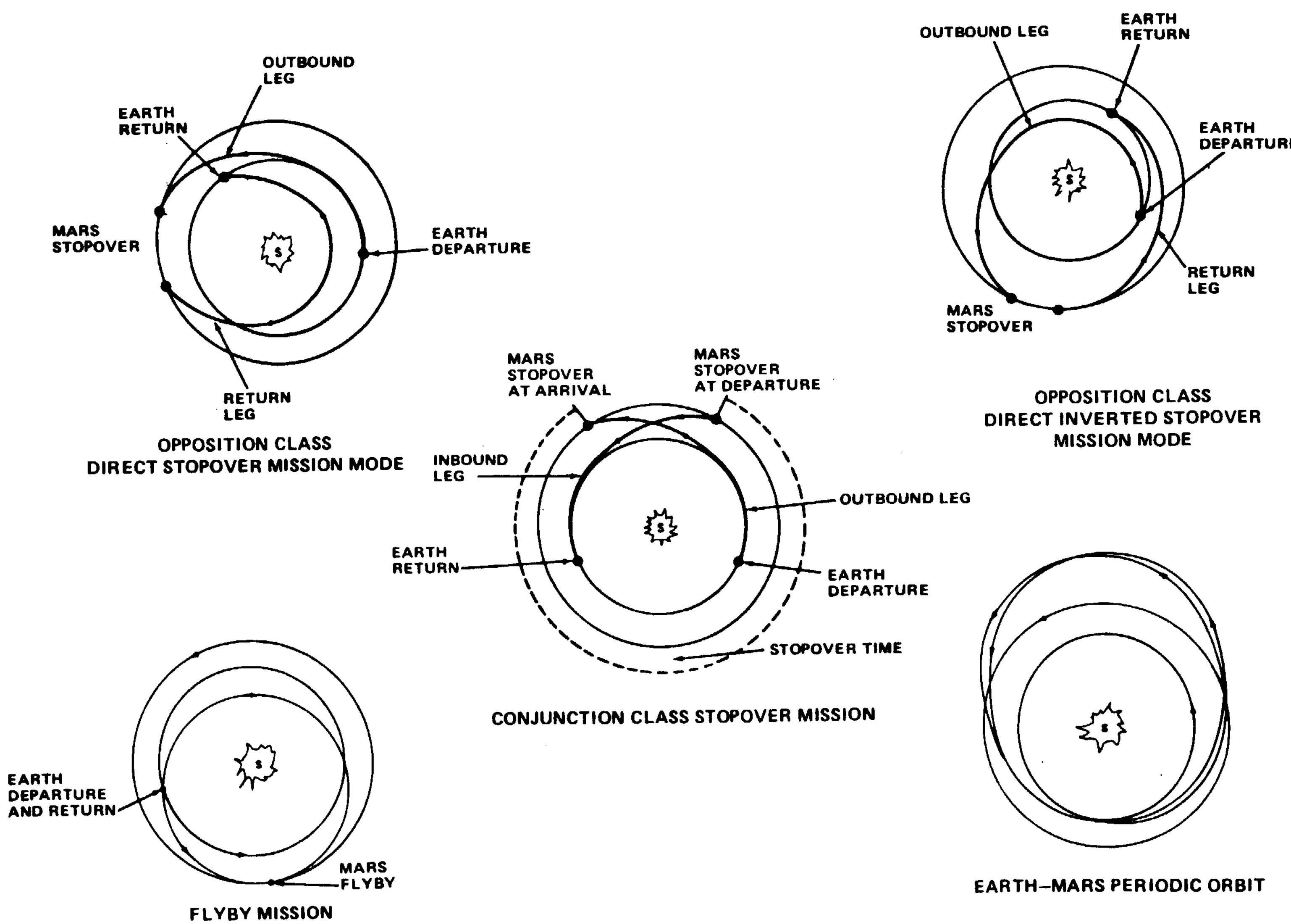

FIGURE 1. EXAMPLE MISSION PROFILES

For opposition-class missions, a Venus swingby utilizes the gravitational field of Venus to either accelerate or decelerate the space vehicle as it passes by the planet, thus reducing the high energy requirements. An acceleration effect is desired for an outbound Venus swingby enroute from Earth to Mars and a deceleration effect is desired for an inbound Venus swingby enroute from Mars to Earth. The time contained in this paper is year 1997 to year 2045.

MARS MISSION PROFILES

Mars round-trip flyby trajectories are the Martian counterpart of lunar flyby return flight paths. A round-trip flyby may be attractive as an early manned mission to Mars, which would reconnoiter the planet at close range. In order to construct a flyby trajectory, three requisite characteristics of the outbound and inbound transfer trajectories are as follows: (1) the outbound arrival and inbound departure dates at Mars must be the same, (2) the hyperbolic excess speed ($V\infty$) at Mars on the inbound and outbound legs must be equal, and (3) the angle between the hyperbolic excess speed of the approach and departure must be less than a certain critical value in order not to require an excessive amount of powered flyby maneuver. The Venus swingby profile involves one or more gravitational encounters with Venus and often requires significantly less ΔV's than direct trajectories to Mars and return.

MISSION OPPORTUNITIES

Mission opportunities for standard direct flights to Mars will occur near the Earth-Mars opposition, and precede by 90 to 180 days the opposition dates which will occur on the average every 26 months. Because of the eccentricity of Mars orbit, the mission trajectory profile changes from one opposition to the next. The cyclic pattern of mission profile variation repeats every 15 years or every 7 oppositions [1]. The relative positions of the Earth-Mars oppositions are indicated in Figure 2 for two periodic cycles of oppositions from year 1997 to 2031. The slight inclination of the Mars orbit with respect to the ecliptic plane causes an interplanetary transfer trajectory also to be inclined to the ecliptic, but this effect is small compared to the effect caused by the eccentricity. The relative position of Earth and Mars for an opposition class mission causes the energy requirement to be excessive because the flight time for a near-Hohmann outbound leg is such that, at Mars

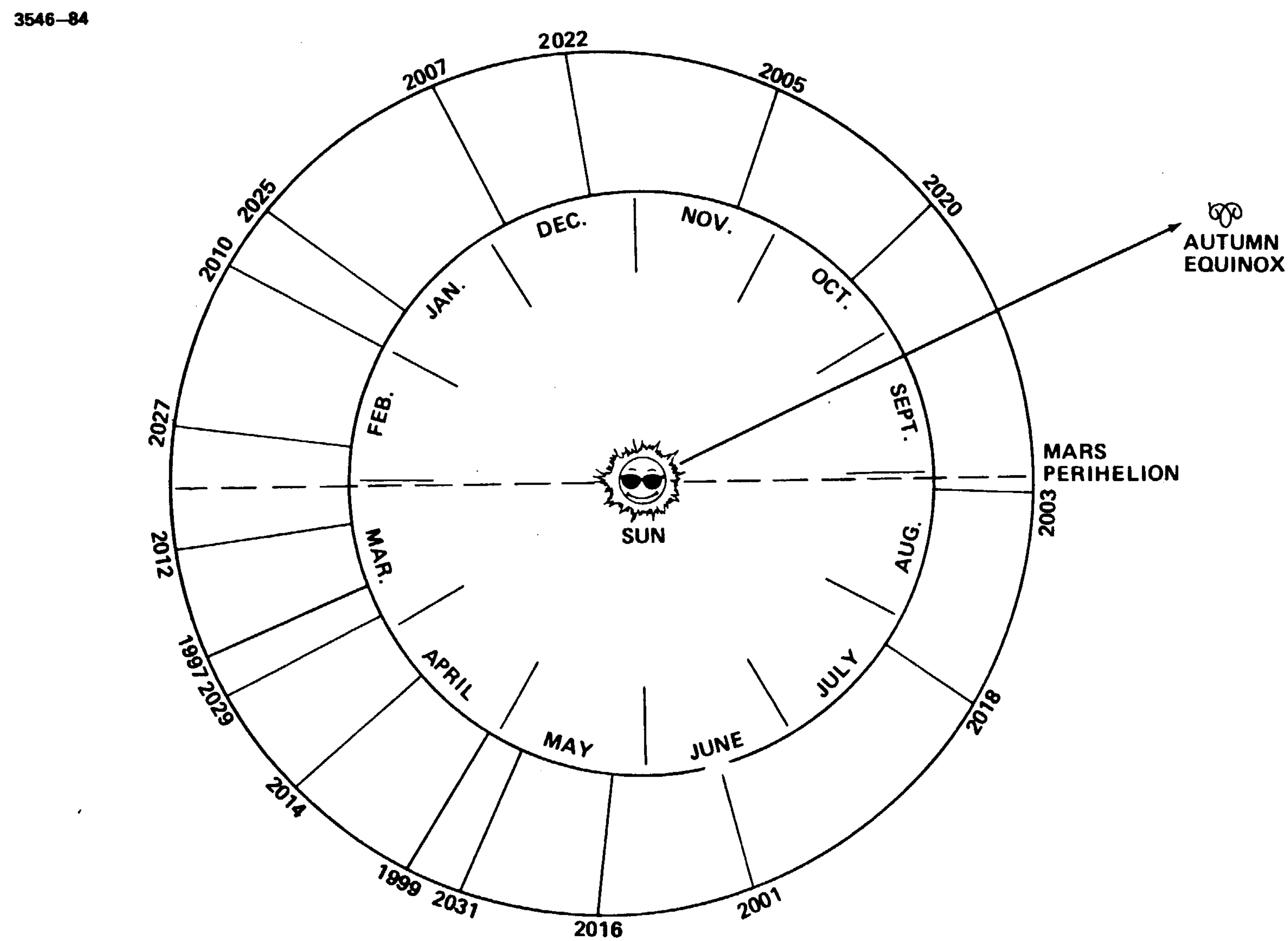

FIGURE 2. EARTH–MARS OPPOSITION FOR YEARS 1997–2031

arrival, Earth is ahead of Mars in heliocentric longitude, i.e., Mars arrival occurs after opposition. This makes it impossible to employ a near-Hohmann transfer for the inbound leg; the required heliocentric transit angle must greatly exceed the Hohmann transfer angle of 180 deg. Thus, it is never possible to leave Earth on a minimum energy inbound leg. The relative position of Earth at Mars arrival can be adjusted with a swingby of Venus enroute to Mars on an outbound leg or swingby of Venus enroute to Earth on an inbound leg. The major advantage of making a swingby of Venus is that the hyperbolic encounter with the planet changes the velocity of the space vehicle relative to the Sun. The magnitude of the velocity change can be large enough to make a significant desirable change in the heliocentric trajectory. The high energy level required can be avoided in the conjunction class mission mode where near-Hohmann transfers can be used on both the outbound and inbound leg by adjusting the stay time at Mars appropriately.

The availability of a Venus swingby mode can be determined by the following facts [1]: (1) The space vehicle will normally pass inside or near the orbit of Venus either on the outbound leg or on the inbound leg of a direct roundtrip mission to Mars. Figure 3 illustrates these conditions for an outbound leg and an inbound leg. (2) The gravity field of Venus is sufficiently powerful to significantly shape the interplanetary transfer trajectory in a desirable way. (3) The angular rate of Venus orbit is large compared to that of Mars, so that Venus is generally available either on the outbound leg or on the inbound leg. The initial step in determining a Venus swingby trajectory profile for a given mission opportunity is the determination of the relative heliocentric position of the three planets, Venus, Earth, and Mars.

INTERPLANETARY TRAJECTORY CALCULATIONS

The computer program used in this work to compute the interplanetary trajectory characteristics is based on the restricted two-body (patched conic) approximation of the interplanetary space vehicle trajectory. While the vehicle is within the sphere of influence of Venus or Mars, the swingby planet or flyby planet respectively, it is assumed to be on a free-flight hyperbolic trajectory about Venus or Mars, and gravitational effects of all other bodies are neglected. There is no change of energy with respect to the swingby or flyby planet, Venus or Mars. Conservation

2966–85

MARS FLYBY FOR 1999 OPPOSITION ONE YEAR MISSION

1 EARTH DEPARTURE, APRIL 2, 1999
2 MARS PASSAGE, AUGUST 8, 1999
3 EARTH ARRIVAL, APRIL 2, 2000

OUTBOUND VENUS SWINGBY 1999 OPPOSITION

1 EARTH DEPARTURE, JAN. 26, 1998
2 VENUS PASSAGE, JULY 9, 1998
3 MARS ARRIVAL, JAN. 16, 1999
4 MARS DEPARTURE, MARCH 17, 1999
5 EARTH ARRIVAL, NOV. 18, 1999

CONJUNCTION CLASS MISSION 1999 OPPOSITION

1 EARTH DEPARTURE, DEC. 17, 1998
2 MARS ARRIVAL, SEP. 28, 1999
3 MARS DEPARTURE, JAN. 25, 2001
4 EARTH ARRIVAL, SEP. 2, 2001

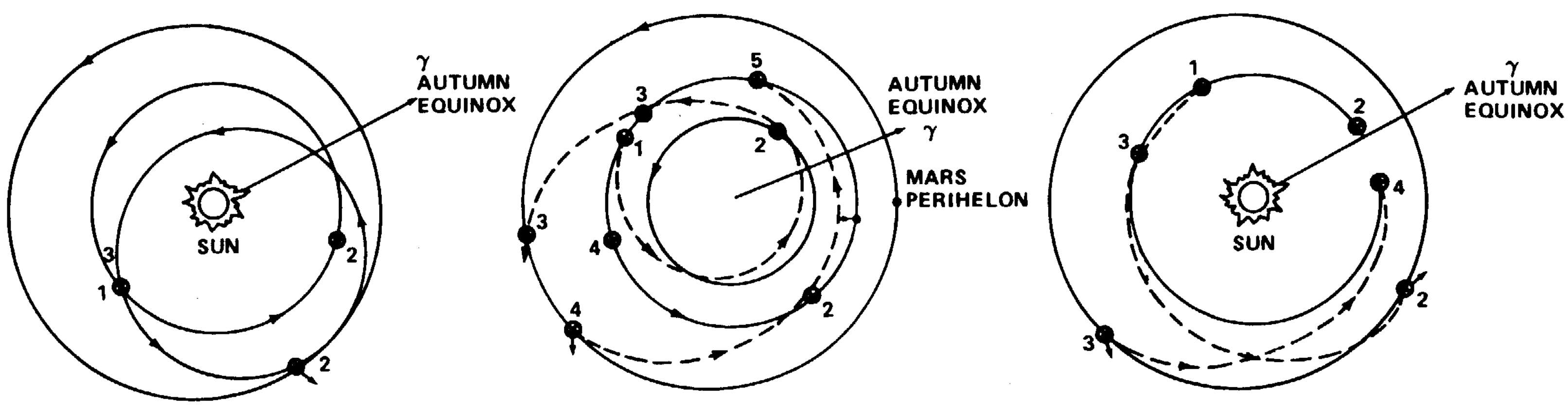

FIGURE 3. REPRESENTATIVE MISSION PROFILES FOR 1999 OPPOSITION

of energy requires that the magnitude of the vehicle's velocity, relative to Venus or Mars, as it leaves the sphere of influence of Venus or Mars must equal to the magnitude of its velocity as it enters the sphere of influence approaching Venus or Mars. If the required angle of deflection, bend angle, at Venus or Mars is too large to be achieved by constraining the periapsis altitude to one-tenth of the planet radii, a propulsive maneuver is effected in conjunction with the Venus or Mars gravity field to give the required bend angle.

Independent optimization of each leg is possible when the conjunction class roundtrip mission is considered. The outbound leg takes place near one opposition and by adjusting the stopover time at Mars appropriately, the inbound leg will take place near the following opposition. Examination of single leg trajectory data [2] indicates that if the outbound and inbound legs of a roundtrip mission could be optimized separately, then departure and arrival hyperbolic excess speeds at both Earth and Mars of less than 0.10 to 0.15 EMOS (Earth Mean Orbital Speed of 97,700 ft/sec) could be attained. The total mission time for conjunction class missions is greater than the mission time of the Venus swingby opposition class mission (950 to 1004 days for conjunction class compared to 558 to 737 days for Venus swingby).

REPRESENTATIVE MISSION PROFILES

Tables 1, 2 and 3 present summary data for the Mars flyby, opposition class stopover mission with Venus swingby, and conjunction class missions for missions between 1998 and 2045. Representative profiles are presented for the three missions described in Figure 3.

The one year flyby mission departs Earth April 2, 1999 with excess hyperbolic velocity, C_3, of 99.5 km^2/sec^2. A flight time of 128 days brings it to a Mars flyby date on August 8, 1999. A propulsive maneuver, requiring a ΔV of 0.406 km/sec, is made at Mars to achieve the necessary turn angle at Mars for the Earth return trajectory. The Earth return date is April 2, 2000 with the interplanetary trajectory having a hyperbolic energy of 156 km^2/sec^2. The Earth departure and return C_3's of 99.5 and 156 km^2/sec^2, respectively, are very high for a Mars mission. However, these C_3 values can be reduced by optimizing the total mission time and by making efficient midcourse maneuvers.

2964-85

MARS 1-YR ROUND-TRIP MISSIONS (OPPOSITION CLASS)*

LAUNCH DATE	C_3 (km/SEC)^2	$\Delta V_{@}$ MARS (km/SEC)	C_3 @ EARTH RETURN (km/SEC)^2	ΔV_{TOT} (km/SEC)
2/28/97	159.6	0.802	237	18.239
4/2/99	99.5	0.406	156	13.639
5/22/01	63.5	0.425	108	10.846
6/8/03	71.6	1.723	134	13.299
10/15/05	122.6	3.806	253	20.518

* DATA FROM REFERENCE 6

TABLE 1. MARS FLYBY MISSION

2965-85

STOPOVER TIME EQUAL 60 DAYS
TIME PERIOD 1996 TO 2031

MISSION	EARTH LAUNCH DATE	TOTAL TRIP TIME (DAYS)
DOUBLE SWINGBY	MARCH 1996	733
OUTBOUND SWINGBY	JANUARY 1998	666
INBOUND SWINGBY	JANUARY 2001	708
OUTBOUND SWINGBY	AUGUST 2002	610
OUTBOUND SWINGBY	JUNE 2004	659
INBOUND SWINGBY	SEPTEMBER 2007	558
DOUBLE SWINGBY	JANUARY 2009	736
OUTBOUND SWINGBY	NOVEMBER 2010	650
INBOUND SWINGBY	NOVEMBER 2013	634
INBOUND SWINGBY	NOVEMBER 2015	577
OUTBOUND SWINGBY	APRIL 2017	638
INBOUND SWINGBY	JUNE 2020	594
OUTBOUND SWINGBY	OCTOBER 2021	636
OUTBOUND SWINGBY	SEPTEMBER 2023	614
INBOUND SWINGBY	NOVEMBER 2026	570
DOUBLE SWINGBY	MARCH 2028	737
OUTBOUND SWINGBY	JANUARY 2030	654

TABLE 2. MARS STOPOVER MISSION WITH VENUS SWINGBY.

2962–85

MARS STOPOVER TIME OPTIMIZED FOR MINIMUM ENERGY

DATE OF OPPOSITION		EARTH LAUNCH DATE		MARS STOPOVER TIME (DAYS)	TOTAL MISSION TIME (DAYS)
MARCH	1997	NOVEMBER	1996	485	1025
APRIL	1999	DECEMBER	1998	485	1005
JUNE	2001	JANUARY	2001	530	1020
MAY	2031	DECEMBER	2030	500	998
JUNE	2033	APRIL	2033	550	950
SEPTEMBER	2035	JUNE	2035	530	1004
NOVEMBER	2037	AUGUST	2037	340	986
JANUARY	2040	SEPTEMBER	2039	340	984
FEBRUARY	2042	OCTOBER	2041	340	990
MARCH	2044	NOVEMBER	2043	340	996

TABLE 3. MARS CONJUNCTION CLASS STOPOVER MISSION.

The 1999 opposition outbound Venus swingby is characterized by a hyperbolic transfer angle between Earth and Venus of over 180 deg, with the transfer angle between Venus and Mars of less than 180 deg. The total transfer angle of the two trajectory transfers is slightly greater than 360 deg. Of paramount importance is the fact that the average angular rate of the outbound leg is much greater than that of Earth in its orbit. Thus, Earth is behind Mars at Mars arrival, i.e., Mars arrival occurs much sooner than oppositions. This situation permits, as shown, a near-Hohmann type Mars-Earth trajectory to be utilized on the inbound leg. However, the Earth return hyperbolic energy, C_3 , is slightly high with a value of 81.52 km^2/sec^2 . This C_3 level could be lowered by effectively applying a propulsive midcourse maneuver on the Mars-Earth transfer leg. The total mission time for the year 1999 outbound Venus swingby opposition opportunity is 661 days.

Aerobraking is commonly asserted to be a means of reducing propulsion requirements for Mars missions. Earth return with aerobrake entry has been analyzed and results show that with an Earth return C_3 greater than 25 km^2/sec^2 the g-load will be in excess of 5 g's. This high g-load cannot be tolerated by the astronauts. Earth return with C_3 greater than 25 km^2/sec^2 will require propulsive braking in order to stay within the g-load constraint.

CONCLUSION

Optimum trajectory transfers for opposition class mission to Mars for flyby and stopover missions have been computed for attractive launch and arrival dates between years 1997 and 2031. Also, Optimum transfer for conjunction class missions to Mars have been computed for attractive opportunities for years 1997, 1999, 2001, and 2030 to 2045.

It is possible to employ an outbound or inbound Venus swingby for every Earth-Mars opposition; oppositions occur approximately every 26 months. Venus swingby permits the heliocentric transfer trajectory to be nearly tangential relative to Earth and Mars orbit upon planet departure and arrival, thus reducing the required propulsive maneuver energy requirement. The mission time is increased from 20 to 50 percent employing the Venus swingby mode over the direct flights to Mars.

Optimum roundtrip trajectories for the conjunction class mission to Mars and return can be achieved by adjusting the stopover time at Mars.

Near-Hohmann type trajectories can be employed both on the outbound and inbound leg with the conjunctions class mission. Data have been developed for years 1997, 1999, 2001 and one Earth-Mars synodic period between years 2030 and 2045 which consists of seven launch opportunities associated with the oppositions occuring during this time period.

Free-fall periodic orbits which travel back and forth between Earth and Mars on a scheduled interval may be attractive for use as a regularly scheduled transportation system between Earth and Mars.

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MANNED MARS MISSION VEHICLE DESIGN REQUIREMENTS FOR AEROCAPTURE

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ABSTRACT

The goal of this study was to define vehicle design requirements of a reusable system for manned Mars missions which employ aerocapturing techniques to obtain desired orbital velocities. Requirements for vehicle L/D and ballistic coefficient are determined for expected aerocapture velocities. This paper presents conclusions concerning g-loads environment and TPS requirements for a vehicle that aerocaptures at Mars and Earth. Although the goal of a reusable system (based on current state-of-art technologies) was not obtained, the viability of aerocapture at Mars and Earth was established.

INTRODUCTION

The deceleration of a vehicle from hyperbolic approach velocities to orbital velocity at Mars and Earth can be accomplished by propulsive braking or atmospheric braking (aerocapture). Many authors have shown that aerocapture is more advantageous than propulsive braking in terms of initial departure mass in low-Earth-orbit (LEO). Therefore, to take advantage of aerocapture at Mars and Earth for a manned Mars mission, vehicle design requirements must be defined in terms of external configuration (L/D), size and mass (m/CDA), entry velocity, aerodynamic heating, and g-loads. The goal of the aerocapture analysis was to define vehicle design requirements for a reusable aerocapture system.

MARS AEROCAPTURE

Trajectory analyses of Earth to Mars transfers for arrival dates from 1999 to 2028 have determined the entry velocity requirement to be approximately 17,700 ft/sec to 30,000 ft/sec. This velocity range corresponds to two classes of missions: conjunction class (<20,500 ft/sec) and opposition class (>20,500 ft/sec).

In order to minimize the scope of the entry trajectory analysis, the analysis of external configuration and mass requirements made use of recent and previous Mars mission studies. Raked-off elliptical cone configurations provide a range of L/D's which were assumed to be adequate for aerocapture. Previous Mars mission studies provided estimates of vehicle mass. With these estimates, an aerocapture analysis was conducted with a modified version of the guidance logic from reference 1. The aerocapture vehicles were assumed to be trimmable within $\pm$ 4.0 degrees of the desired angle-of-attack.

The aerocapture guidance was required to achieve the target apoapsis altitude in the presence of all combinations of the following system dispersions: (1) Flight path angle dispersion of $\pm$ 0.30 degree; (2) Angle of attack dispersion of $\pm$ 4.00; and (3) Mars atmosphere density models from reference 2. A minimum altitude constraint of 100,000 feet at Mars was utilized.

An aerocapture is a guided deceleration through an entry corridor in a planet's atmosphere to achieve a desired orbital velocity. The entry corridor is defined by those trajectories which have flight path angles steep enough to avoid skipping out of the atmosphere (remaining at hyperbolic velocity) and shallow enough to achieve a desired apoapsis while maintaining desired g-load and aerodynamic heating levels. The vehicle L/D is the parameter which controls the width of the entry corridor for a vehicle using lift vector modulation for control. Figure 1 shows the required vehicle L/D to meet the aerocapture velocity requirements at Mars. An L/D of 0.6 is required to satisfy the complete aerocapture velocity range requirement. Within the aerocapture corridor the minimum altitude of a trajectory is important for control of aerodynamic heating, g-loads and other considerations such as obstacle avoidance. For a specified guidance logic, the vehicle ballistic coefficient, m/CDA, is the primary driver of the minimum altitude of an aerocapture trajectory (Figure 2). The aerocapture analysis demonstrated that a ballistic coefficient greater than 100 lbm/sq ft would violate the minimum altitude constraint at Mars (Figure 3). Therefore, the vehicle design requirement for external configuration, size and mass is an L/D of 0.6 with a ballistic coefficient less than 100 lbm/sq ft. The effect of these conclusions on the stagnation heat flux and g-load environments must also be studied to determine thermal protection system requirements and crew environment.

Figure 4 presents the reference stagnation heating rate for a one foot radius sphere as function of ballistic coefficient and entry

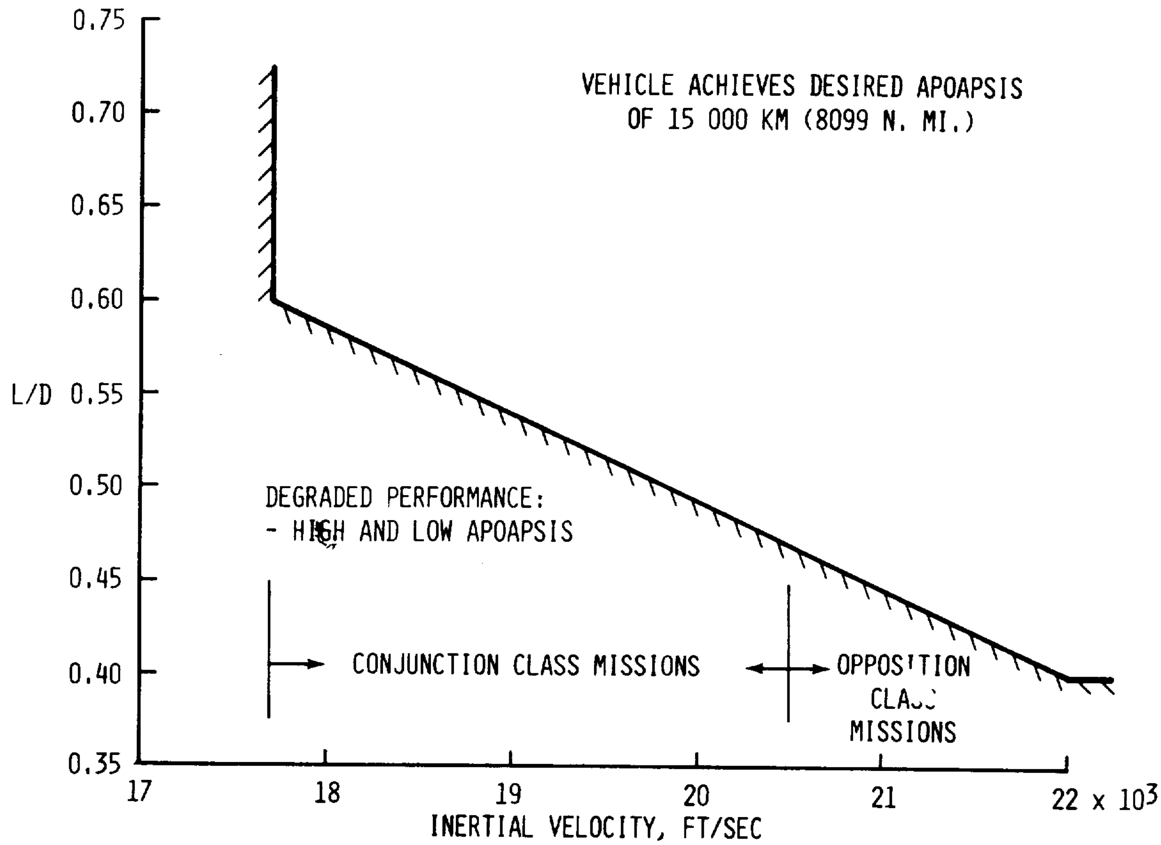

Figure 1.- Mars aerocapture L/D requirements.

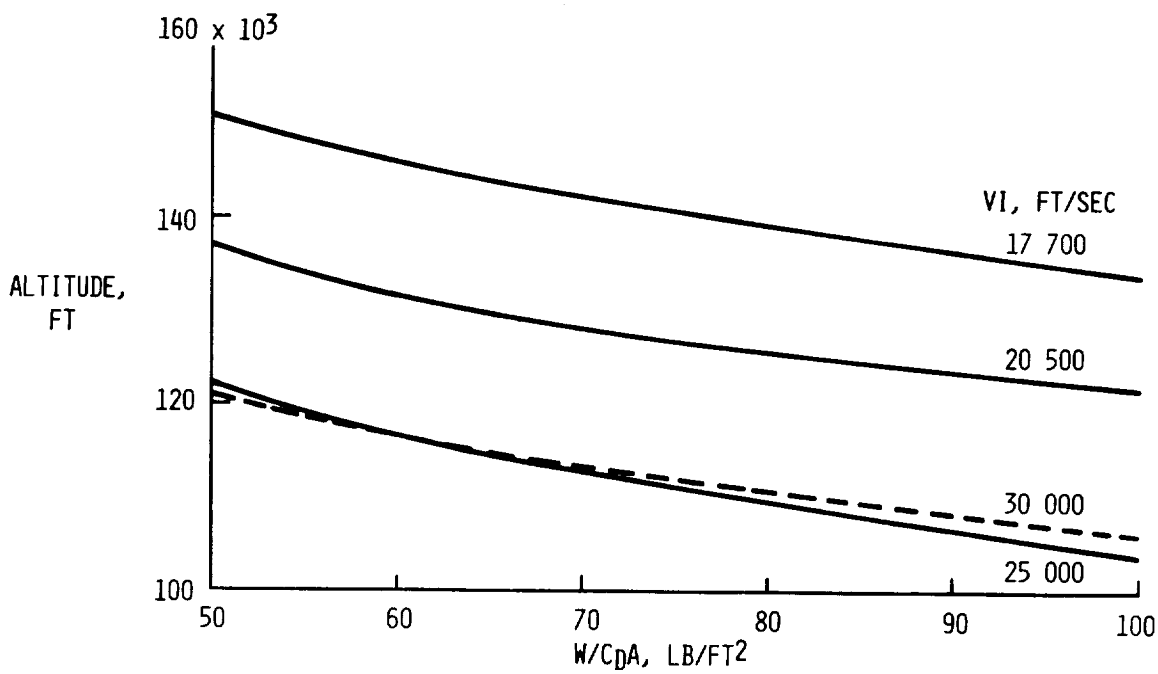

Figure 2.- Minimum altitude during aerobraking at Mars.

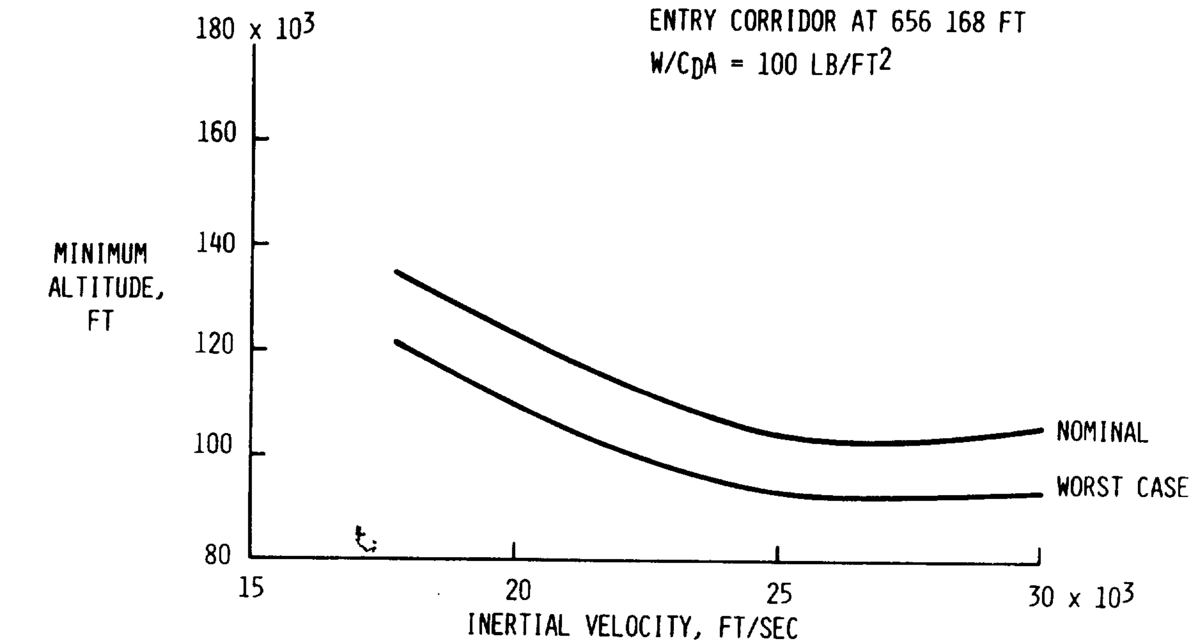

Figure 3.- Minimum altitudes for Mars aerocapture maneuver.

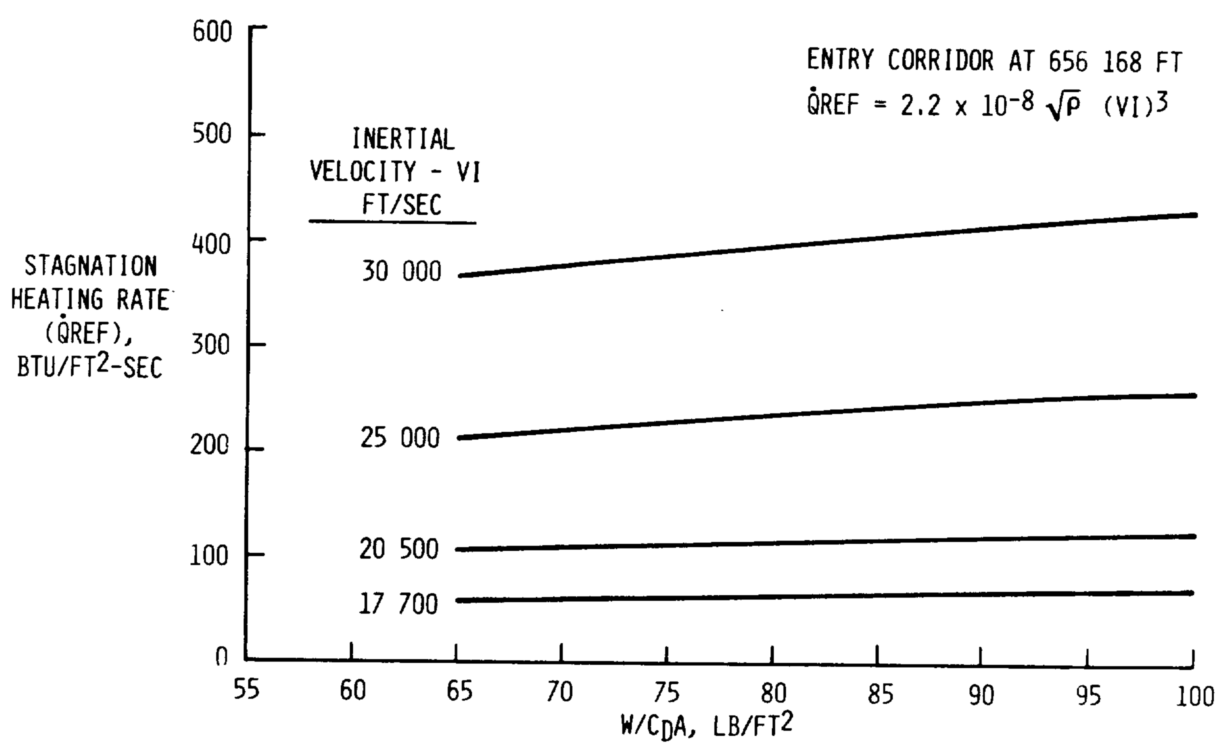

Figure 4.- Mars stagnation heating rates versus ballistic number.

velocity for aerocapture at Mars. When these reference heating rates are assessed for an 85 foot diameter aerobrake, the conclusion can be drawn that an ablative or advanced state-of-the-art TPS is required for opposition class missions and may be required for conjunction class missions.

Figures 5 and 6 present the expected g-load for conjunction and opposition class missions, respectively, within the acceptable Mars entry corridor. The expected g-loads for conjunction class missions appear to be acceptable, while the g-loads for opposition class missions approach intuitively unacceptable values. However, life scientists will have to identify acceptable g-load requirements.

The most severe conditions for the aerocapture maneuver are produced by analyzing a vehicle which has a ballistic coefficient of 100 lbm/sq ft. Tables 1 through 4 present the detailed results of the Mars aerocapture analysis for the complete range of approach velocities which cover conjunction, opposition and Venus swingby missions.

EARTH AEROCAPTURE

Trajectory analyses of Mars to Earth transfers have determined that the maximum expected entry velocity for conjunction class missions is 38,000 ft/sec and that opposition class entry velocities significantly exceed 38,000ft/sec. The aerocapture analysis at Earth was limited to vehicles that satisfied the Mars aerocapture requirements because the same vehicle was assumed to perform the Mars and Earth aerocaptures. The analysis was also limited to conjunction class missions because the conclusions drawn from this conjunction class analysis would only be amplified by the more severe vehicle environment of opposition class missions. Figures 7 and 8 present the g-load and reference stagnation heating rates across the aerocapture corridor for a vehicle which has an L/D of 0.6 and ballistic coefficient of approximately 55 lbm/sq ft (greater than expected ballistic coefficients for actual vehicle designs). From the calculated g-load environment and extrapolations to opposition class entry velocities, it can be concluded that the crew would experience intuitively unacceptable g-loads. Furthermore, when thermal protection system requirements are assessed using the data on Figure 8 for a vehicle with an 85 foot diameter aerobrake, the conclusion can be drawn that an ablative or advanced state-of-the-art TPS is required. Since g-loads and a reusable TPS appear unacceptable, a

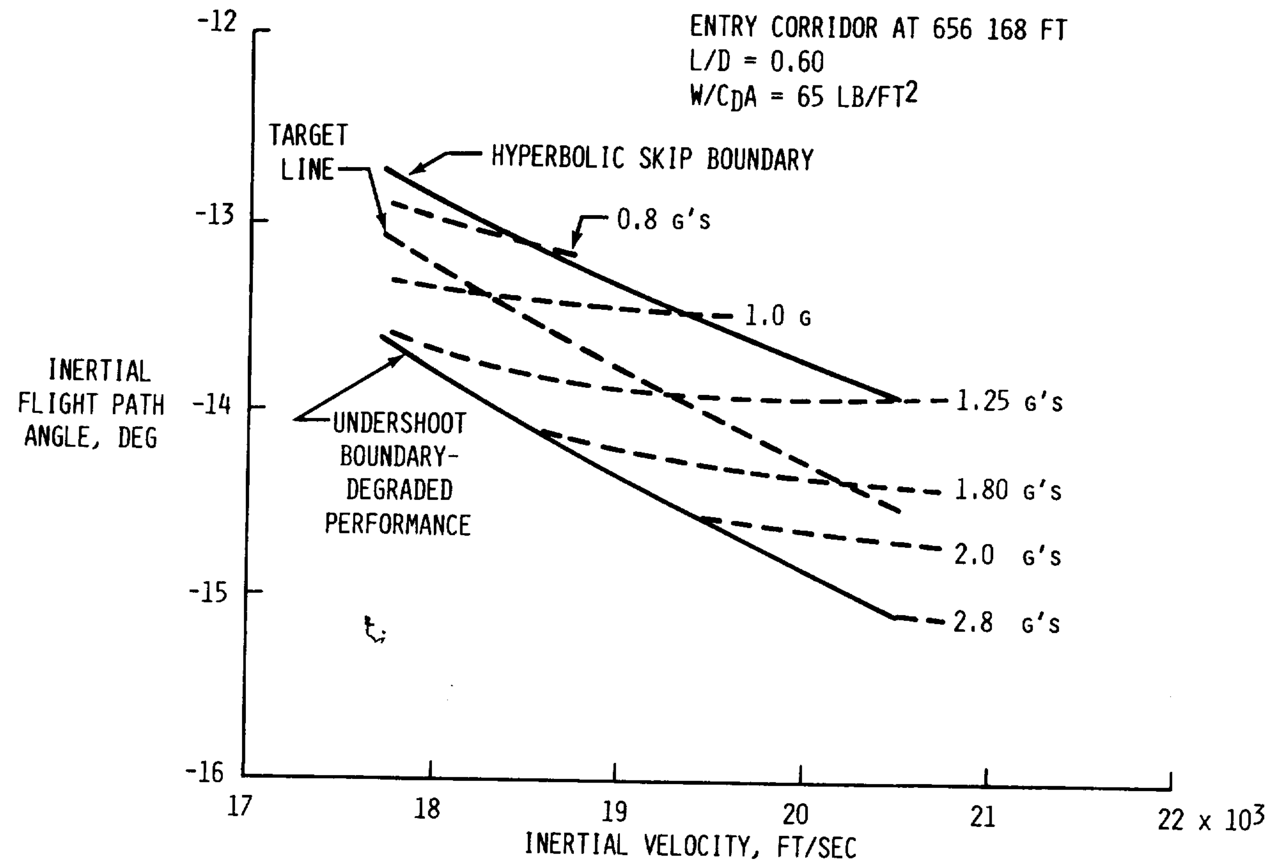

Figure 5.- Mars entry corridor for conjunction class missions.

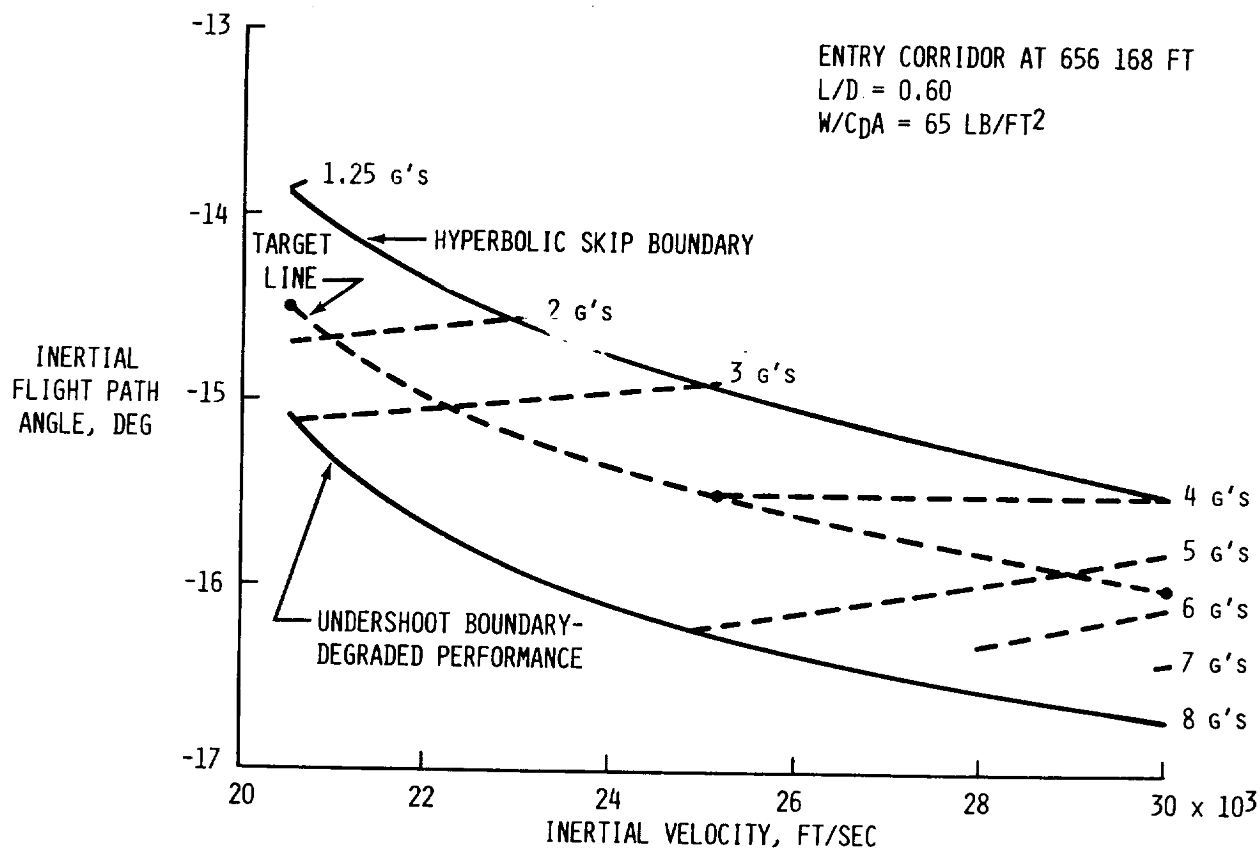

Figure 6.- Mars entry corridor for opposition class missions.

TABLE 1.- MARS AEROCAPTURE PERFORMANCE FOR VI = 17 700 FT/SEC

V_I = 17 700 FT/SEC   L/D = 0.60   TARGET APOAPSIS ALTITUDE = 8099 N. MI.
γ_I = -13.22 DEG   656168 FT W/C_DA = 100 LB/FT^2

CASE		ALTITUDE MINIMUM FT	LOAD FACTOR g's	$\dot{q}$ STAG BTU/FT^2-SEC	HA N. MI.
NOMINAL		134 642	0.86	69.8	8104.0
: ATM - COOL LOW PRESSURE					
$\Delta\gamma$ = -0.30 DEG	$\Delta\alpha$ = -4.0 DEG	126 222	1.02	73.0	8346.1
	$\Delta\alpha$ = 4.0 DEG	122 485	0.96	78.3	8295.3
$\Delta\gamma$ = 0.30 DEG	$\Delta\alpha$ = -4.0 DEG	137 240	0.65	57.0	8121.1
	$\Delta\alpha$ = 4.0 DEG	136 721	0.57	59.8	7925.3
: ATM - WARM HIGH PRESSURE					
$\Delta\gamma$ = -0.30 DEG	$\Delta\alpha$ = -4.0 DEG	138 094	1.27	79.6	8025.7
	$\Delta\alpha$ = 4.0 DEG	137 699	1.10	83.6	8119.7
$\Delta\gamma$ = 0.30 DEG	$\Delta\alpha$ = -4.0 DEG	152 147	0.80	63.8	8090.1
	$\Delta\alpha$ = 4.0 DEG	146 353	0.80	69.4	8085.9

TABLE 2.- MARS AEROCAPTURE PERFORMANCE FOR VI = 20 500 FT/SEC

V_I = 20 500 FT/SEC   L/D = 0.60   TARGET APOAPSIS ALTITUDE = 8099 N. MI.
γ_I = -14.5 DEG 656168 FT W/C_DA = 100 LB/FT^2

CASE		ALTITUDE MINIMUM FT	LOAD FACTOR g's	$\dot{q}$ STAG BTU/FT^2-SEC	HA N. MI.
NOMINAL		120 702	1.69	124.4	8082.9
: ATM - COOL LOW PRESSURE					
$\Delta\gamma$ = -3.0 DEG	$\Delta\alpha$ = -4.0 DEG	113 368	1.92	129.4	8837.3
	$\Delta\alpha$ = 4.0 DEG	108 225	1.90	139.0	8194.9
$\Delta\gamma$ = 3.0 DEG	$\Delta\alpha$ = -4.0 DEG	124 222	1.26	102.9	8393.1
	$\Delta\alpha$ = 4.0 DEG	117 808	1.29	111.1	8344.9
: ATM - WARM HIGH PRESSURE					
$\Delta\gamma$ = -3.0 DEG	$\Delta\alpha$ = -4.0 DEG	130 277	2.10	135.1	8244.4
	$\Delta\alpha$ = 4.0 DEG	125 529	1.95	144.7	7981.7
$\Delta\gamma$ = 3.0 DEG	$\Delta\alpha$ = -4.0 DEG	138 194	1.56	114.1	8114.9
	$\Delta\alpha$ = 4.0 DEG	133 196	1.52	123.1	8067.1

TABLE 3.- MARS AEROCAPTURE PERFORMANCE FOR VI = 25 000 FT/SEC

VI = 25 000 FT/SEC L/D = 0.60 TARGET APOAPSIS ALTITUDE = 8099 N. MI.
γI = -15.6 DEG 656168 FT W/C_DA = 100 LB/FT^2

CASE		ALTITUDE MINIMUM FT	LOAD FACTOR g's	$\dot{q}$ STAG BTU/FT^2-SEC	HA N. MI.
NOMINAL		104 362	3.87	258.7	8087.2
: ATM - COOL LOW PRESSURE					
$\Delta\gamma$ = -3.0 DEG	$\Delta\alpha$ = -4.0 DEG	98 260	4.23	259.8	8246.9
	$\Delta\alpha$ = 4.0 DEG	93 168	4.26	285.6	8211.5
$\Delta\gamma$ = 3.0 DEG	$\Delta\alpha$ = -4.0 DEG	106 735	3.12	218.3	8295.6
	$\Delta\alpha$ = 4.0 DEG	99 508	3.34	243.7	8235.8
: ATM - WARM HIGH PRESSURE					
$\Delta\gamma$ = -3.0 DEG	$\Delta\alpha$ = -4.0 DEG	113 351	4.41	270.9	7997.1
	$\Delta\alpha$ = 4.0 DEG	107 954	4.48	293.6	8089.2
$\Delta\gamma$ = 3.0 DEG	$\Delta\alpha$ = -4.0 DEG	120 239	3.68	240.0	8086.5
	$\Delta\alpha$ = 4.0 DEG	114 810	3.65	265.6	8079.2

TABLE 4.- MARS AEROCAPTURE PERFORMANCE FOR VI = 30 000 FT/SEC

V_I = 30 000 FT/SEC   L/D = 0.60   TARGET APOAPSIS ALTITUDE = 8099 N. MI.
γ_I = -16.05 DEG 656168 FT W/C_DA = 100 LB/FT^2

CASE		ALTITUDE MINIMUM FT	LOAD FACTOR g's	$\dot{q}$ STAG BTU/FT^2-SEC	HA N. MI.
NOMINAL		106 283	5.07	431.1	8097.3
: ATM - COOL LOW PRESSURE					
$\Delta\gamma$ = -0.30 DEG	$\Delta\alpha$ = -4.0 DEG	98 731	5.86	441.2	8348.4
	$\Delta\alpha$ = 4.0 DEG	93 464	5.90	484.3	8345.2
$\Delta\gamma$ = 0.30 DEG	$\Delta\alpha$ = -4.0 DEG	118 689	2.93	338.6	8351.2
	$\Delta\alpha$ = 4.0 DEG	101 632	3.12	359.8	8161.4
: ATM - WARM HIGH PRESSURE					
$\Delta\gamma$ = -0.30 DEG	$\Delta\alpha$ = -4.0 DEG	109 743	6.89	469.2	8315.0
	$\Delta\alpha$ = 4.0 DEG	105 189	6.83	510.3	8135.6
$\Delta\gamma$ = 0.30 DEG	$\Delta\alpha$ = -4.0 DEG	122 084	4.78	400.0	8118.6
	$\Delta\alpha$ = 4.0 DEG	113 481	4.80	439.1	8106.2

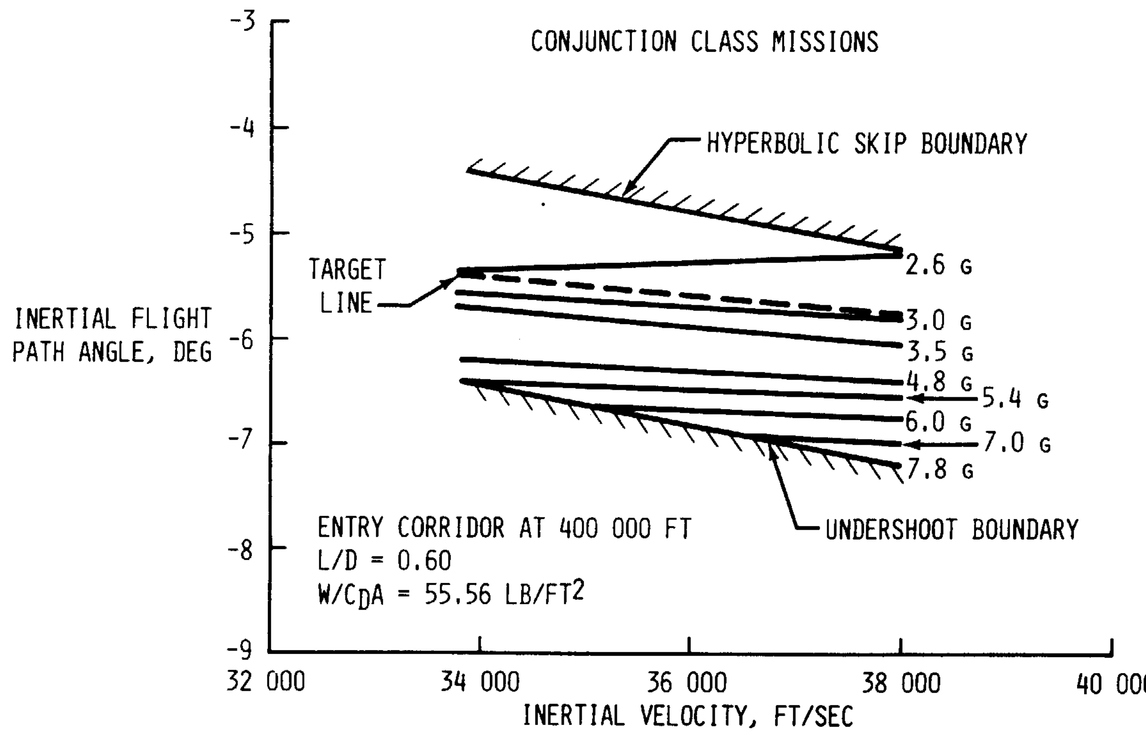

Figure 7.- Earth aerocapture corridor with g-loads.

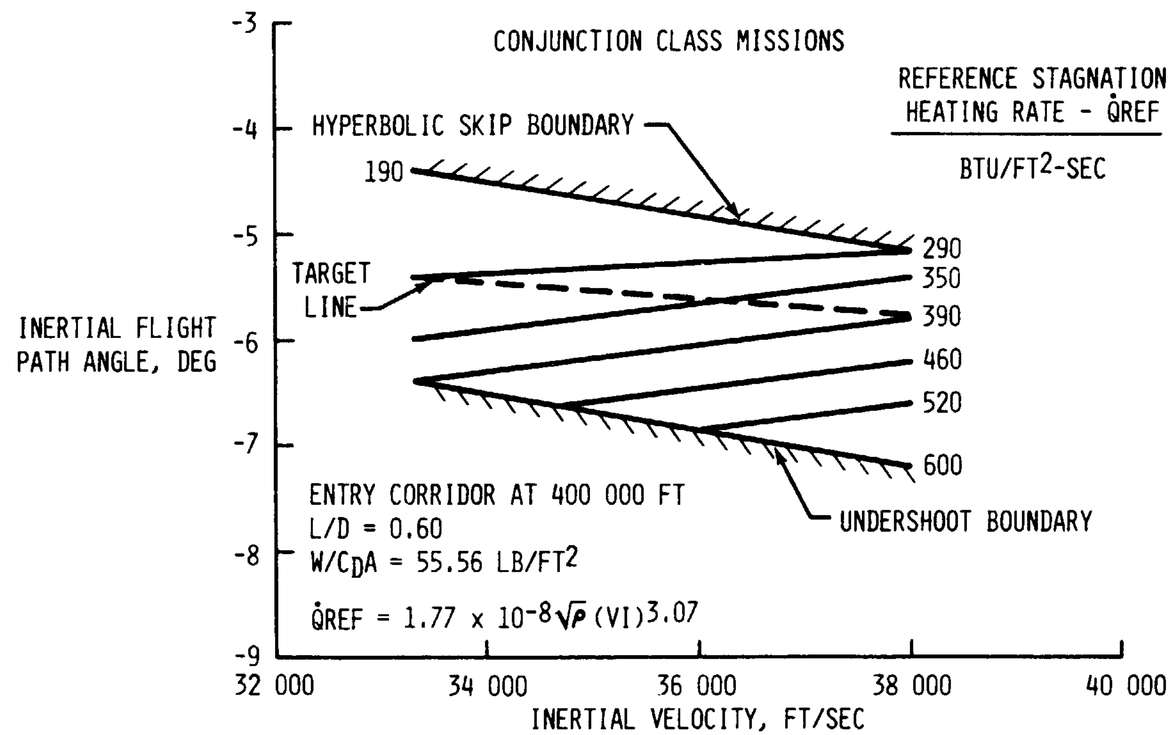

Figure 8.- Earth aerocapture corridor with stagnation heating rates.

propulsive braking system is required to augment the aerocapture system to reduce the aerocapture velocity and, thereby, relieve g-load and aero-heating environments of the aerocapture system.

Another approach to aerocapture at Earth is to aerocapture only part of the Earth return vehicle. A "small" crew and Mars sample module could be designed into the Earth return vehicle which would have a small ballistic coefficient. The advantage of this approach is that the minimum altitude during entry would be increased which would decrease the amount of aerodynamic heating. Figures 9 and 10 present the g-load and reference stagnation heating rates across the aerocapture corridor for a vehicle which has an L/D of 0.6 and ballistic coefficient of 10 lbm/sq ft. Several conclusions can be drawn from these plots. Propulsive braking may still be required for g-load control of the small module. However, the mass of propellent required to perform the braking of the small module would be less than the mass of propellent required to perform the same function for the complete Earth return vehicle. Also, reusable TPS may be acceptable only for conjunction class entry velocities for the small module.

CONCLUSION

The initial goal of the aerocapture analysis was to derive vehicle design requirements for a reusable system that could aerocapture at Mars and Earth. The aerocapture analyses have determined that a vehicle with L/D of 0.6 and ballistic coefficient less than 100 lbm/sq ft can be aerocaptured at Mars and Earth. However, the goal of a reusable system may be unrealistic. The TPS requirements point to non-reusable TPS or an advanced state-of-the-art TPS. Also the expected g-load environment at Earth points to aerocapture systems which have some propulsive braking capability for control of the vehicle g-loads. Since TPS requirements are affected by vehicle ballistic coefficient, reduction in ballistic coefficient can be obtained by studying separate aerocaptures at Mars of the Mars transfer vehicle and staged Mars landers; and at Earth by considering aerocapturing only a small crew/sample module.

RECOMMENDATIONS

The approach to this study was to make use of previous Mars mission studies and recent raked-off cone vehicle studies. The next step will be to take a more parametric approach to vehicle design requirements defini-

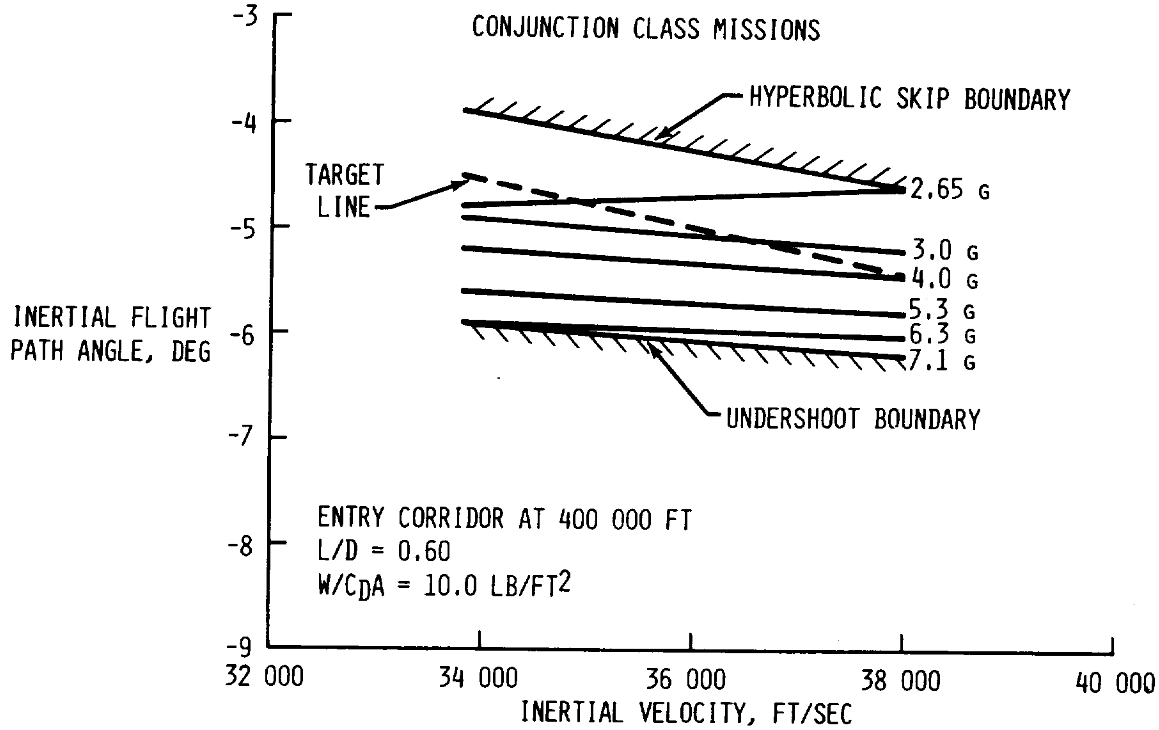

Figure 9.- Earth aerocapture corridor with g-loads.

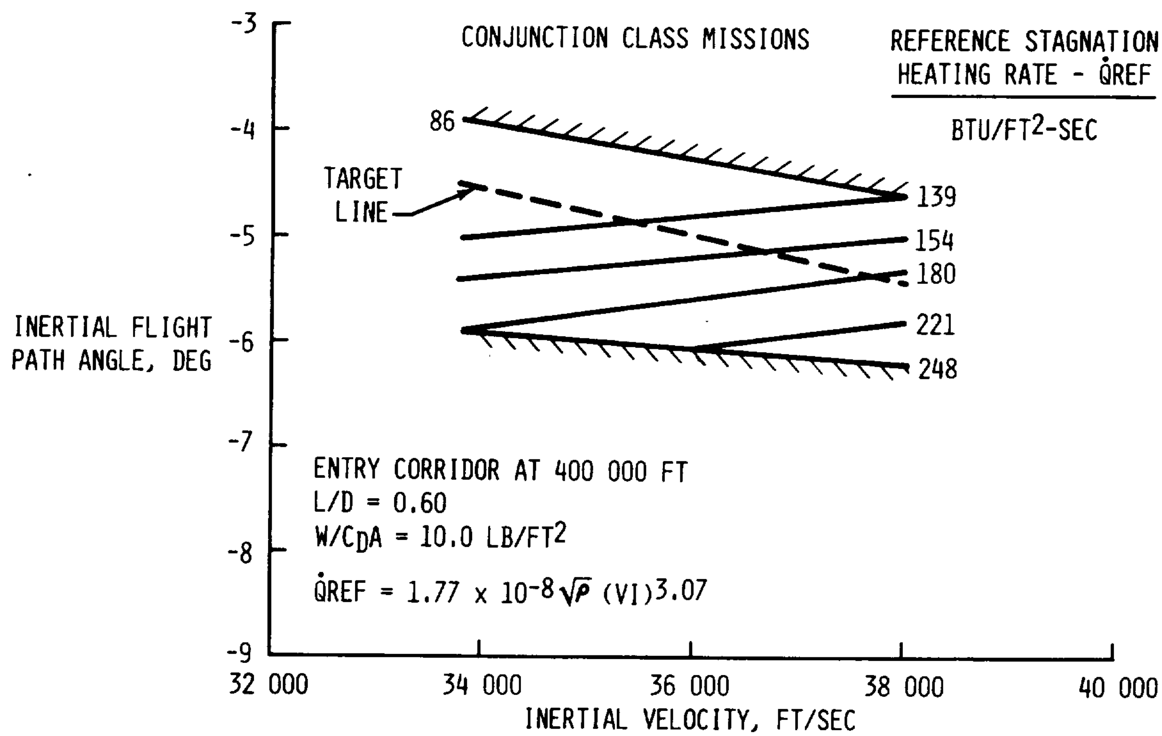

Figure 10.- Earth aerocapture corridor with stagnation heating rates.

tion by assessing a larger range of L/D, ballistic coefficient, and external configuration. Preliminary analyses indicate that an advancement in the state-of-the-art TPS technology is required to make a reusable system possible. Therefore, further TPS studies are recommended. Finally, the allowable crew entry g-load levels require definition for the case of long exposure to zero g or low level g. Physiological tests could be performed during an Apollo type entry from Space Station for a crew made up of personnel who have had long exposure to zero g and personnel who have not.

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N87-17732

NUCLEAR ELECTRIC PROPULSION

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ABSTRACT

We investigate the feasibility of using nuclear electric propulsion (NEP) for slow "freighter" ships traveling from a 500 km low Earth orbit (LEO) to the Moon's orbit about the Earth, and on to Mars. NEP is also shown to be feasible for transporting people to Mars on long conjunction-class missions lasting about nine months one way, and on short "sprint" missions lasting four months one way. Generally, we have not attempted to optimize ion exhaust velocities, but rather we have chosen suitable parameters to demonstrate NEP feasibility. Various combinations of missions are compared with chemical and nuclear thermal propulsion (NTR) systems. Typically, NEP and NTR can accomplish the same lifting task with similar mass in LEO. When compared to chemical propulsion, NEP was found to accomplish the same missions with 40% less mass in LEO. These findings are sufficiently encouraging as to merit further studies with optimum systems.

INTRODUCTION

Space propulsion systems can be placed into two broad categories: (1) "impulse" rockets, which produce large accelerations for short periods of time, typically several minutes, and (2) "low-thrust" rockets, which produce small accelerations for long periods of time, typically several months. All of today's operational rockets are of the impulse type. Usable low-thrust engines have been developed in the laboratory.

We address here a specific low-thrust rocket by assuming the engines to be 30 cm diameter mercury ion thrusters with characteristics that exist in the laboratory today. A specific thruster power of 125 w/kg is assumed (see Table I). The thrusters are powered by a nuclear reactor

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NOTE TO THE READER: As the Manned Mars Mission Workshop approached, the authors were asked to investigate the feasibility of using nuclear electric propulsion in a manned Mars program. The present paper constitutes our preliminary findings as of June, 1985. Because low-thrust propulsion showed such promise with this first investigation, more careful studies involving numerical integration techniques were subsequently undertaken and the findings were published as two Los Alamos reports (Refs 8,9). The conclusions have not changed significantly.

TABLE 1

PROJECTED NUCLEAR REACTOR CHARACTERISTICS

	REF 1	REF 2	REF 3	TAKEN HERE
ELECTRIC POWER (Mw_e)	8.5	1	10	3
MASS (metric tons)	26	4	75	20
SPECIFIC POWER (w/kg)	327	250	133	125

PROJECTED ION THRUSTER CHARACTERISTICS

	Ar MPD	Xe ION	Hg ION	CURRENT Hg ION
SPECIFIC IMPULSE (s)	5,000	5,000	4,250	3,000
THRUST PER ENGINE (n)	14.7	13.4	0.63	0.132
DIAMETER (cm)	3	30	30	30
SPECIFIC POWER (w/kg)	300,000	7,500	1,900	125
SYSTEM EFFICIENCY	0.5	0.78	0.7	0.7

TABLE 2

FOUR MONTH "SPRINT" TO MARS WITH NEP

	MASS
MISSION MODULE (3 people)	
(tons)	28
(k lbs.)	62
CONSUMABLES	
(tons)	6
(k lbs.)	13
STRUCTURE (k = 0.05)	
(tons)	2
(k lbs.)	4
REACTOR ($3Mw_e$, 8 kg/kw)	
(tons)	24
(k lbs.)	53
ENGINES	
(tons)	24
(k lbs.)	53
PROPELLANT	
(tons)	34
(k lbs.)	75
TOTAL MASS IN EARTH-MOON ORBIT	
(tons)	115
(k lbs.)	253

supplying 3 megawatts of electrical power. In addition, we have conservatively assumed a specific power of 125 w/kg to describe the power source reactor, shielding, and electrical conversion system. (Ref. 1-4) Low-thrust propulsion relying on nuclear reactors for electrical energy - which is then used to accelerate ions - is referred to as nuclear electric propulsion (NEP).

Specific impulse, I_{sp}, which relates directly to exhaust velocity, c, is used to characterize rocket engines. Ideally, the specific impulse is given by

$$I_{sp} = c/g_o$$

were g_o is the gravitational acceleration at the Earth's surface. Here we take $g_o = 9.8\ m/s^2$ and for our purposes, we characterize chemical, nuclear thermal, and nuclear electric propulsion systems by I_{sp} = 460 sec, 850 sec, and 3,000 sec, respectively.

The purpose of this paper is to examine the feasibility of using nuclear electric propulsion for slow "freighter" ships traveling from a 500 km low Earth orbit (LEO) to the Moon's orbit about the Earth, and on to Mars. We also show that NEP is feasible for transporting people to Mars on long conjunction-class missions, lasting about 9 months one way, and on short "sprint" missions, lasting 4 months one way. Various combinations of missions are compared with chemical and nuclear thermal propulsion systems.

Our study shows that NEP matches with Nuclear thermal performance about evenly. However, when we compared NEP with chemically fueled impulse rockets, we found NEP could accomplish the same missions with 40% less mass. We arrive at these factors by comparing the amount of mass that must be delivered initially from the surface of the Earth to low Earth orbit. When other criteria are used, such as obtaining reusable ships, low-thrust rockets become even more attractive. In short, we believe the best rocket propulsion system for most situations is a hybrid system combining the best features of impulse rockets and low thrust rockets.

WHY CONSIDER LOW THRUST ROCKETS

In its simplest form, the fundamental rocket equation relates M_p, the mass of propellant required to change the rocket velocity by delta v, with the constant propellant exhaust velocity, c. The equation may be written

$$M_p/M_i = [1 - e^{-(\Delta v/c)}]$$

where M_i is the initial rocket mass. Since the exhaust velocity of ion engines is extremely high, less propellant is required than for a purely chemical rocket. This illustrates just one of the advantages of a low thrust propulsion system.

Another advantage of low-thrust propulsion is illustrated in Figure 1. Here an NEP rocket is slowly spiraling out from low Earth orbit. (It should be mentioned that this process is not drawn to scale, i.e. there would be many more turns of the spiral at low altitudes.) For small accelerations ($a/g_o << 1$), the ship velocity will be nearly equal to the velocity, V_c, required for a circular orbit at each point along the trajectory. This means that $V(r) \approx V_c$. When the ship reaches the moon's orbit, for example, it can have nearly zero hyperbolic velocity relative to the Moon. The same can be true of a ship traveling to Mars, where little or no braking maneuvers are required. This gives NEP the advantage that a ship can either choose to spiral slowly into Mars orbit, or be captured into a highly elliptical orbit with a small (chemical) delta v of, say, 200 m/S applied at periapsis.

NEP ORBITAL CALCULATIONS

The calculations for this paper, except for the last section, follow those of Jones (Ref. 5), where the initial mass and trip time are parameterized in terms of specific impulse, power, thruster efficiency, tankage fraction, specific reactor power, specific thruster power, delta v, and payload mass. In this work, we have taken thruster efficiency to be 0.7, the tankage fraction to be 0.05, the specific reactor power to be 125 w/kg, and the specific thruster to be 125 w/kg. Specific impulses ranged from 3,000 sec to 10,000 sec, and the power ranged from 3 Mw_e to 30 Mw_e. The delta v used for LEO to Moon's orbit was 6.93 km/s and included a 28.5^o orbital plane change. The delta v used for the Earth to Mars mission was 5.82 km/s and included a 1.85^o orbital plane change. The payload mass was either adjusted to make the trip time about one year, or was fixed to compare NEP with some mission using chemical propulsion. In addition, a factor of 0.05 times the reported payload mass was subtracted from the calculated payload mass to account for the

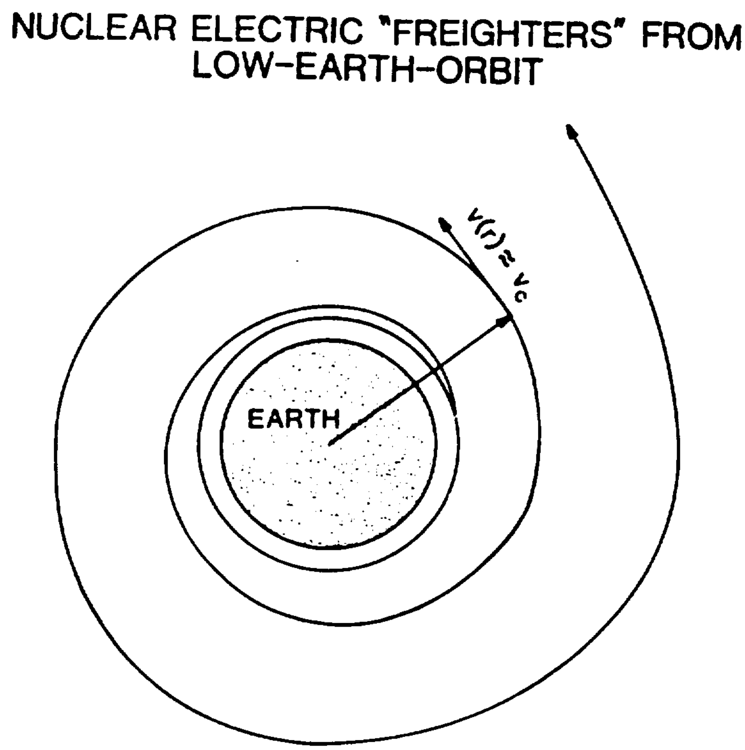

Figure 1. The low thrust-spiral of nuclear electric propulsion (NEP) rocket leaving low Earth orbit (LEO).

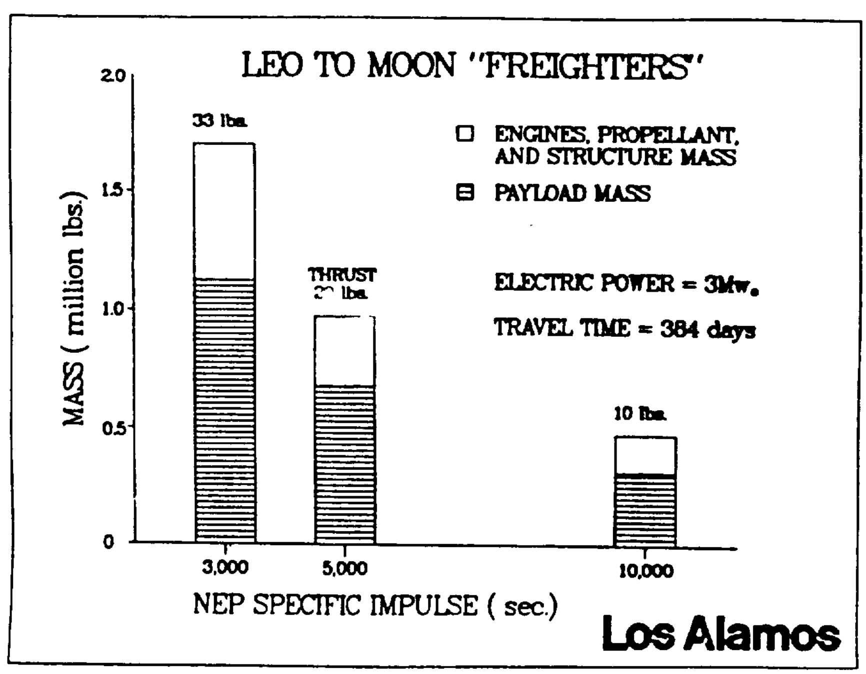

Figure 2. Payload capabilities of NEP freighters going from LEO to the Moon's orbit.

payload structure mass. The equations reported by Jones are valid for $a/g_0 \ll 1$ and a tangential thrust, provided the polar coordinate angle of the trajectory is small (See Ref. 6). Initially, the rocket must increase its velocity by accelerating away from its host planet to develop enough centrifugal acceleration to increase its radius vector. Subsequently, as the radius vector increases, the ship's velocity decreases, and it falls behind its host planet. This initial process is not addressed in our calculations. Based on Irving's report (Ref. 7), we have verified that our calculations are valid for the long-duration missions to Mars reported here, but not for times much smaller than 9 months.

We used the result of Irving's work to derive our 4 month sprint mission to Mars. Irving formulates low thrust propulsion in terms of a fundamental integral

$$\gamma^2 = \frac{\alpha}{2} \int_0^T a^2(t)\, dt$$

here α is one divided by the specific power and the thrust acceleration, $a(t)$, varies with time. Irving then shows how to optimize reactor mass, payload mass, and propellant mass one γ^2 is known.

For the last section of this study, we used α = 8kg/kw and a 3 Mw_e power supply to address a 4-month one-way mission to Mars. The remainder of the ship components were optimized accordingly.

NEP FREIGHTERS

We began our study by noticing that months are usually required for NEP to lift a large payload from payload from LEO to the Moon's orbit. Consequently, we focused first on unmanned freighters where long transfer times are not critical. By extending the transfer time to a year, freighters make use of the large mass carrying capability of NEP. Figure 2 shows the payload mass which can be delivered to the Moon's orbit about the Earth from LEO for three specific impulses. Notice that when trip time and electrical power are held constant, the payload decreases as the specific impulse increases. More detailed information is given in Appendix A, Table A1.

Once the freighter is in the Moon's orbit, gravitational assists from the Moon can be used to direct the ship's velocity toward Mars, as illustrated in Figure 3. We now concern ourselves with the cargo we wish to take to Mars. Figure 4 shows the payload mass that can be transported

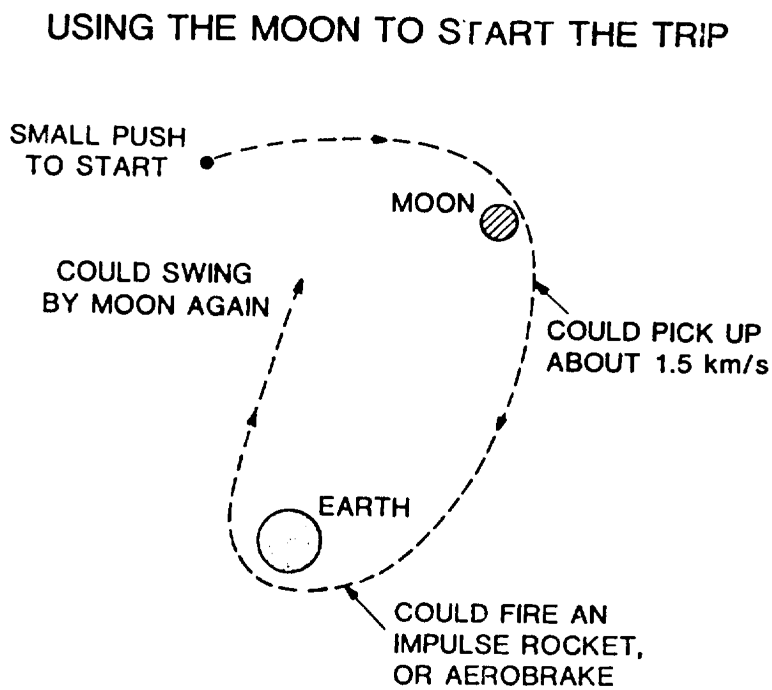

Figure 3. Gravitational assists from the Moon can start an NEP rocket to Mars.

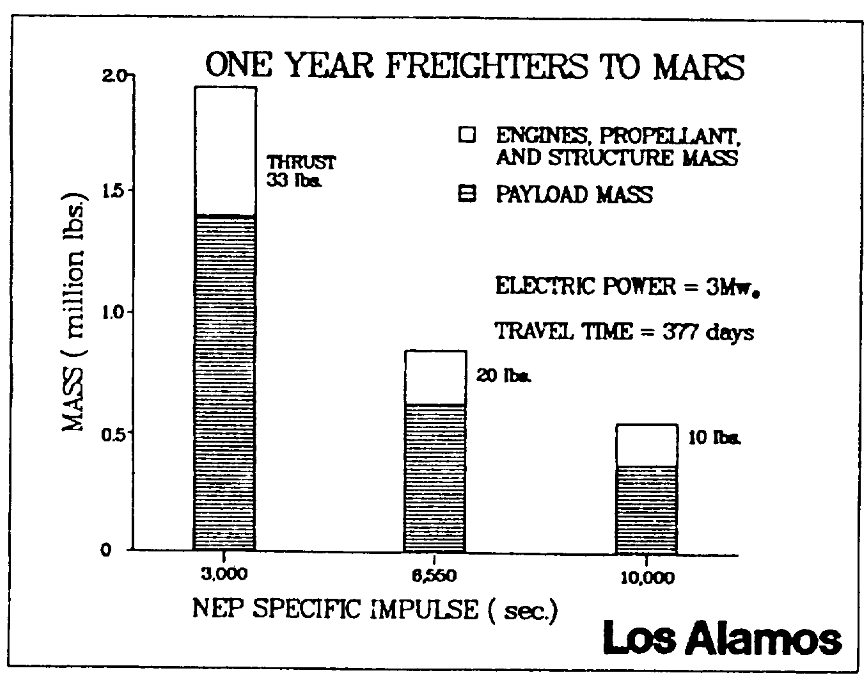

Figure 4. Payload capabilities of NEP freighters going from the Earth's orbit around the Sun to Mars' orbit around the Sun.

for three specific impulses. Notice that the same inverse relationship holds between payload and specific impulse as in traveling from LEO to the Moon's orbit. However, more importantly, for the same reactor power and approximate trip time, more payload can be taken from the Earth-Moon system to Mars than from LEO to the Moon's orbit (see Figure 2). In short, it is cheaper to take cargo to the Moon from Mars than from LEO. This fact is extremely interesting if a lunar base already exists. A further analysis is provided in Table A2.

HYBRID NEP VERSUS IMPULSE ROCKETS

We now address the issue of sending a manned mission to Mars using NEP. To make such a comparison with impulse rockets, we have first identified a "hybrid" rocket combining NEP and chemical propulsion. We consider the 1999 opposition class mission with Mars and Earth aerobraking as described by the Marshall Space Flight Center for a chemical rocket. In the hybrid rocket, we have kept the mass of all the chemical rocket components the same, except for the first stage, which we replaced with an NEP system in LEO. The NEP freighter is used to lift the chemical rocket to lunar orbit. At that point, the crew joins the ship. From there, the Moon is used for gravitational assist, as stated earlier, and the chemical engine is fired at perigee. Otherwise, the I_{sp} = 460 (chemical) and I_{sp} = 3,000 (NEP) systems shown in Figure 5 are the same. As another comparison, I_{sp} = 850 (nuclear thermal reactor, NTR) delivering the same payload to Mars and back to Earth is shown in Figure 5. Again, more detail is given in Appendix A, Table A3.

Another mission scenario involves a conjunction-class trajectory. In Figure 6, NEP is compared with NTR and chemical rockets for conjunction class missions. The NEP system here is a different type of hybrid rocket. Four 15,000 lb. thrust chemical engines with storable propellant and I_{sp} = 345 sec are contained within the NEP system. These chemical engines are used so that small velocity changes of about 200 m/s can be made quickly for escaping from and braking into Earth and Mars orbits. Table A4 gives more specific information about this mission.

FOUR MONTH "SPRINT" TO MARS WITH NEP

Lastly, we consider getting a fast manned mission to Mars from the Moon's orbit about the Earth, using NEP - a "sprint" mission in effect. Table 2 shows an initial rocket mass of 252,000 lbs. that delivers a

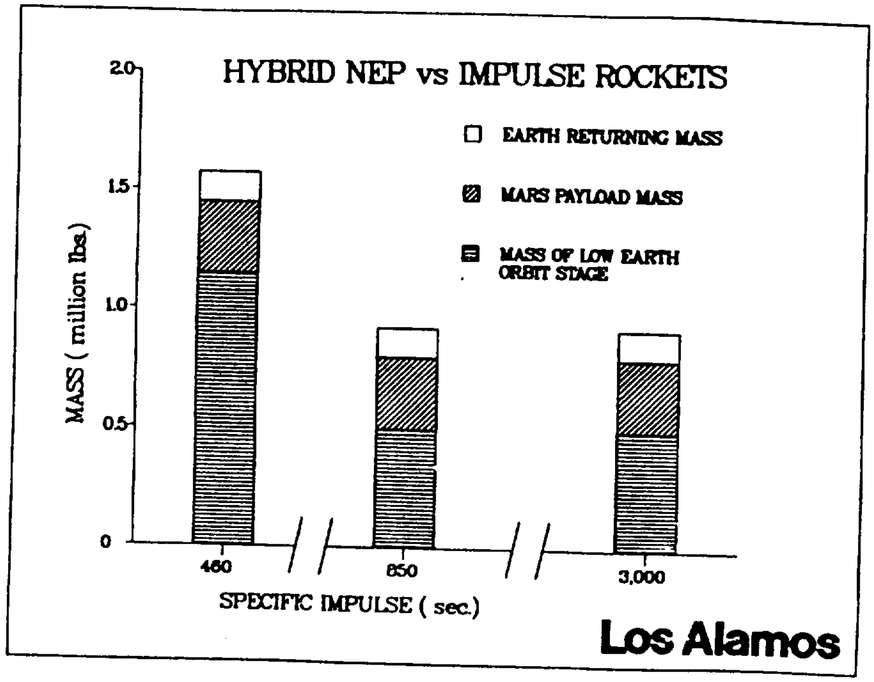

Figure 5. A 1999 opposition-class mission from Earth to Mars. Specific impulses of 460, 850, and 3,000 represent chemical propulsion, nuclear thermal rocket propulsion, and hybrid NEP rockets, respectively. The second and third stages of all three rocket systems are kept the same.

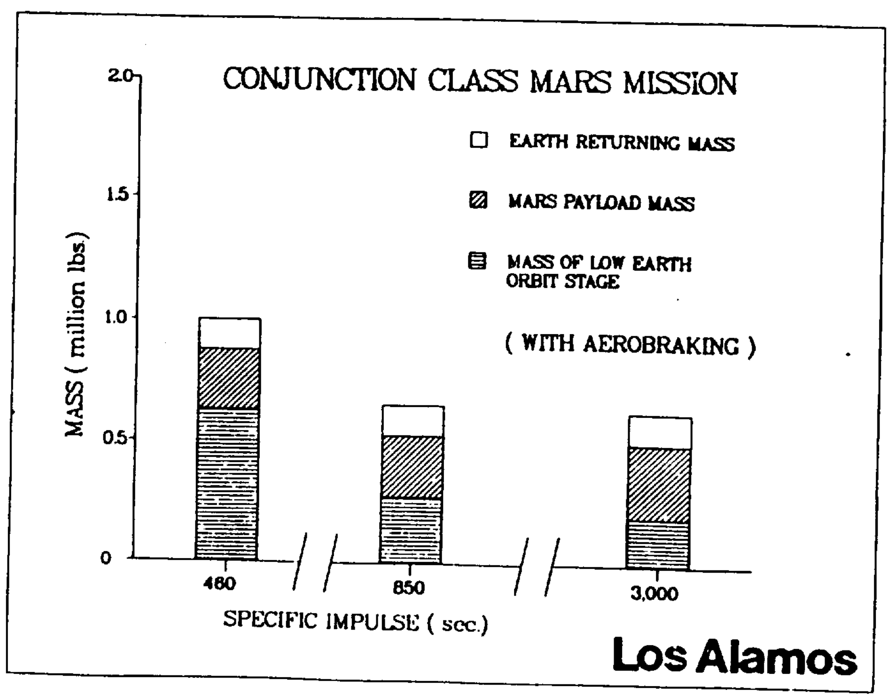

Figure 6. A typical conjunction-class mission from Earth to Mars. See the figure caption for figure 5.

three person crew to Mars in four months. These numbers are taken from Reference 7, as stated earlier. Reference 7 uses a variable thrust rather than the constant NEP thrust assumed in all other calculations for this study. However, this establishes the feasibility of a four-month "sprint" mission to Mars, which would be very difficult with chemical propulsion.

ACKNOWLEDGEMENTS

The authors wish to express their appreciation to W. Stump of Eagle Engineering and to R. Jones of the Jet Propulsion Laboratory for their support by apprising us in a short time of the current literature on nuclear electric propulsion.

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APPENDIX - TABLE 1

LOW EARTH ORBIT (LEO) TO MOON "FREIGHTERS"

	CASE 1	CASE 2	CASE 3	CASE 4
SPECIFIC IMPULSE (sec.)	3,000	5,000	10,000	5,000
ELECTRIC POWER (Mw_e)	3	3	3	30
TRAVEL TIME (days)	384	383	386	383
THRUST (n)	143	86	43	857
(lbs.)	32	19	10	192
MASS IN LEO (tons)	768	439	214	4,388
(k lbs.)	1,690	966	471	9,654
PROPELLANT MASS (tons)	161	58	15	579
(k lbs.)	354	128	33	1,274
ENGINES, STRUCTURE (tons)	48	48	48	480
(k lbs.)	106	106	106	1,056
PAYLOAD MASS (tons)	514	311	142	3,105
(k lbs.)	1,131	684	312	6,831

APPENDIX - TABLE 2

ONE YEAR FREIGHTERS TO MARS

	I_{sp}=3,000 sec.	I_{sp}=6,550 sec.	I_{sp}=10,000 sec.
TRAVEL TIME (days)	377	375	378
THRUST (n)	143	65	43
(lbs.)	32	15	10
MASS LEAVING E/M (tons)	881	381	247
(k lbs.)	1,938	838	543
PROPELLANT MASS (tons)	158	33	14
(k lbs.)	348	73	32
MARS PAYLOAD (tons)	633	283	175
(k lbs.)	1,393	623	385

APPENDIX - TABLE 3

HYBRID NEP vs IMPULSE ROCKETS FOR MARS MISSIONS

AEROBRAKING (PROPULSIVE BRAKING)

	CHEMICAL I_{sp} = 460 sec.	NTR I_{sp} = 850 sec.	CHEMICAL + NEP I_{sp} = 3,000 sec.
EOI (tons) (k lbs)	60 (150) 133 (329)	60 (115) 133 (254)	60 (150) 133 (329)
TEI (tons) (k lbs)	101 (246) 222 (541)	94 (151) 207 (332)	101 (246) 222 (541)
MOI (tons) (k lbs)	187 (632) 412 (1,390)	179 (329) 394 (723)	187 (632) 412 (1,390)
TMI (tons) (k lbs)	715 (2,121) 1,574 (4,667)	421 (757) 926 (1,666)	281 (900) 618 (1,979)
LEO (tons) (k lbs)	715 (2,121) 1,574 (4,667)	421 (757) 926 (1,666)	422 (1,216) 929 (2,675)

APPENDIX - TABLE 4

CONJUNCTION-CLASS MARS MISSION WITH NEP

	CHEMICAL I_{sp} = 460 sec.	NTR I_{sp} = 850 sec.	CHEMICAL + NEP I_{sp} = 5,660 sec.
EOI (tons) (k lbs)	60 133	60 133	112 247
TEI (tons) (k lbs)	85 187	84 185	130 286
MOI (tons) (k lbs)	170 375	169 371	201 442
TMI (tons) (k lbs)	453 996	300 660	287 631

N87-17733

MANNED MARS MISSION SUNLIGHT AND COMMUNICATION OCCULTATIONS

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ABSTRACT

Calculations are presented for the 1999 opposition class mission and a procedure for obtaining similar occultation data for any other given Mars mission is given. Occultation data for a Mars orbiter in a 24.5 hour parking orbit and a Mars base have been calculated for: sunlight occultation - the time in darkness; and radio communication occultation - the communication losses between the lander and the orbiter, the lander and Earth, and orbiter and Earth.

CALCULATIONS

Mars Orbiter Sunlight Occultation

To find the time in darkness for a Mars Orbiting Spacecraft it is necessary to determine the orientation of the parking orbit with respect to the Sun. This is done by finding the angle between the semi-major axis of the orbit and the Mars to Sun line. This angle, α, (see Figure 1), is found using the following equation:

$$\alpha x = RAP - VE - P - L$$

The values of RAP, VE, P and L are found using the trajectory data (1) and the Planetary Handbook (2).

Once α is known it is possible to find the points on the orbit corresponding to the beginning and end of occultation, defined as ψ_1, and ψ_2. This is done using the equation of the parking orbit and a transformation between a reference frame centered on the orbit (x, y) and one that lies along the Mars-Sun line (x', y'). These reference frames are shown in Figure 2. The second frame, (x', y') is defined as a rotation of the orbit centered frame (x, y) through an angle α followed by a translation d, where d is defined as:

$$d = ae \sin \alpha$$

The quantity ae represents the distance from the center of the orbit to the focus of the orbit which is the center of Mars.

The values of ψ_1 and ψ_2 are found by finding the values of x' that correspond to y' equal to $\pm$ the radius of Mars (RM), then con-

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FIGURE 1
GEOMETRY FOR DETERMINING MARS ORBITER OCCULTIONS

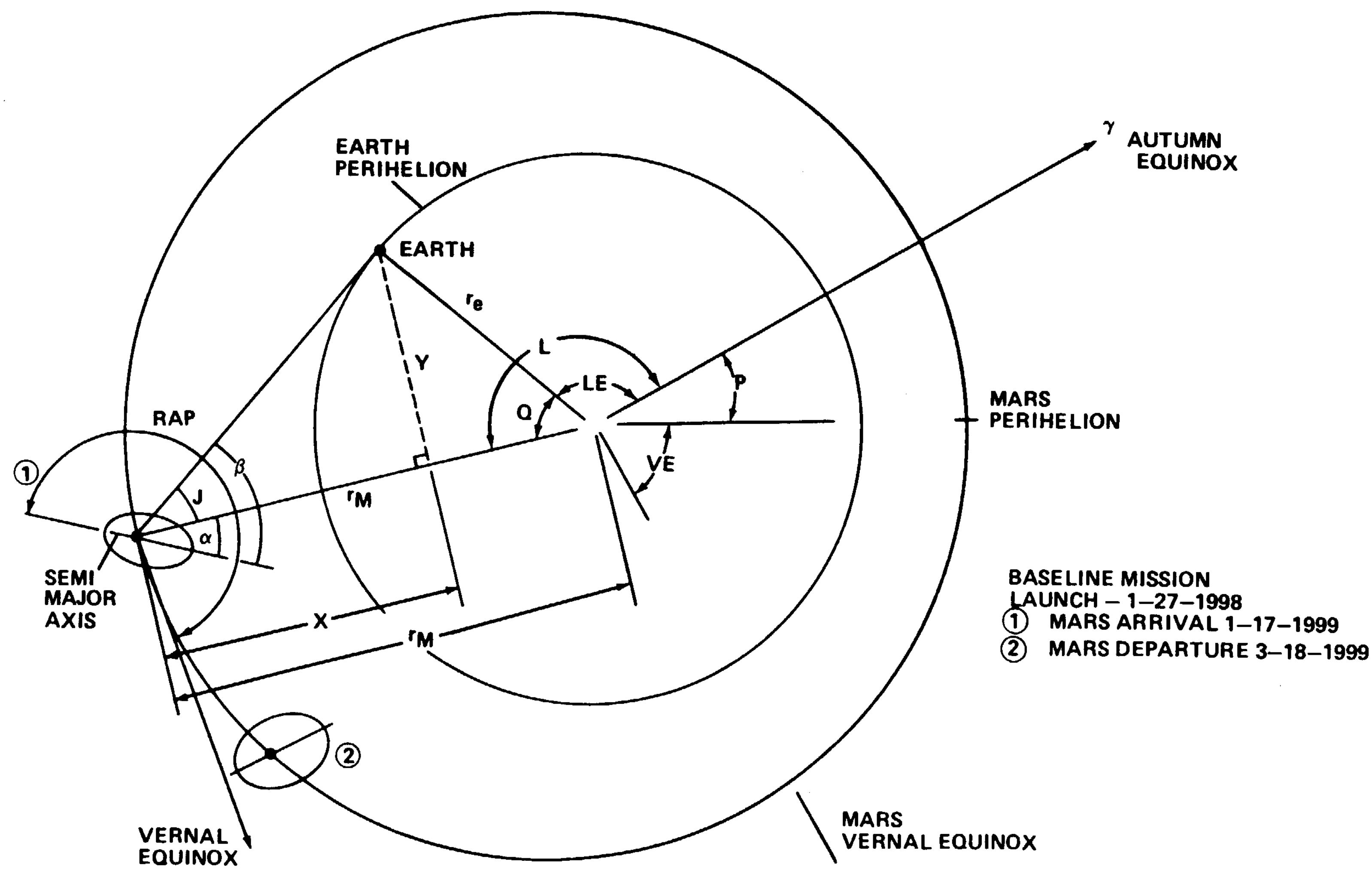

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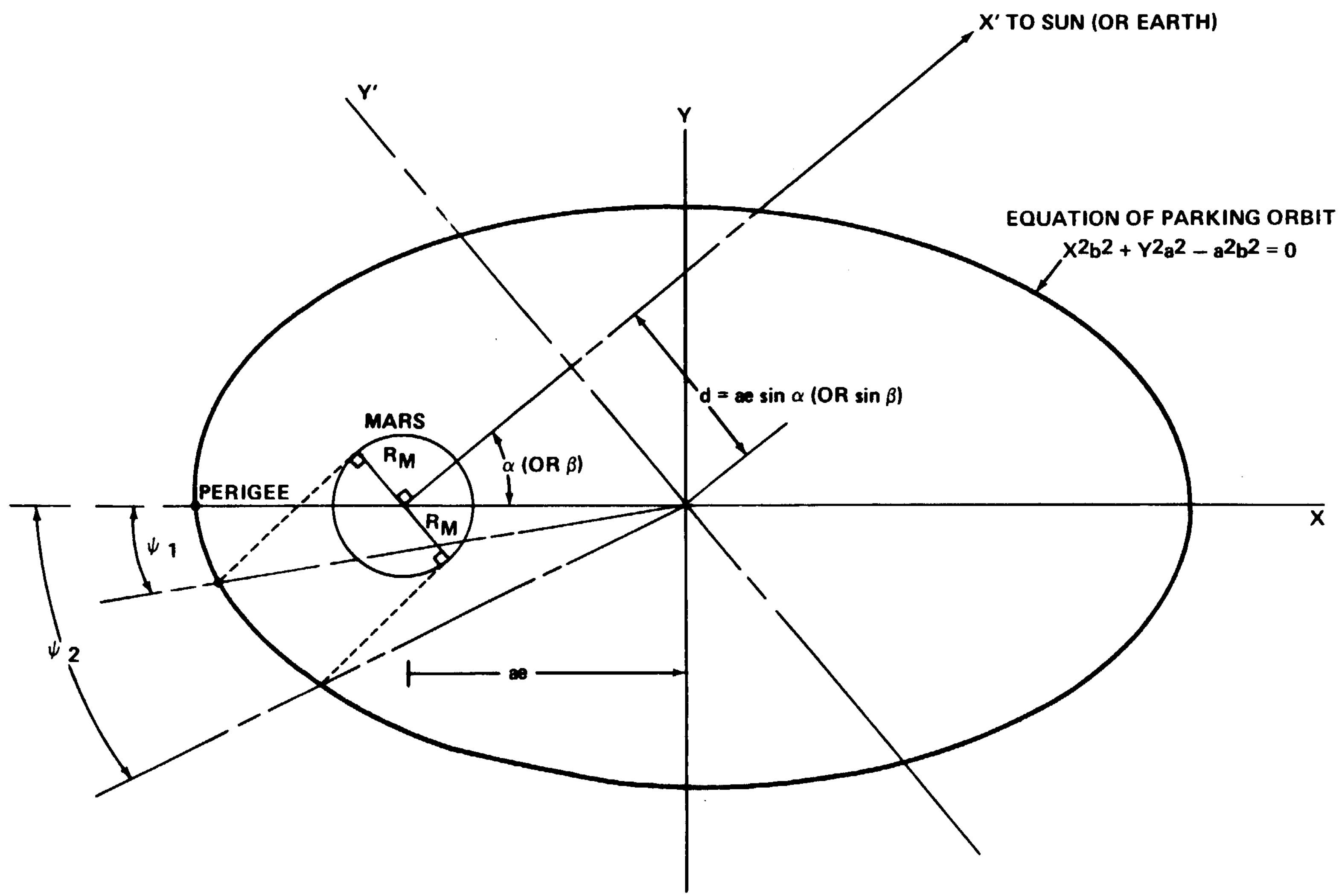
FIGURE 2
GEOMETRY FOR CALCULATING MARS ORBITER OCCULTATIONS



verting to the x,y frame. [Once ψ_1 and ψ_2 are known,] Kepler's time equation then can be used to find the time from perigee to ψ_1 and ψ_2. The difference in the two times represents the duration of the occultation.

The transformation between the two coordinate frames are found in the following manner:

First, a rotation of the x,y frame through an angle α to yield the x_1, y_1 axes (see Figure 1).

$$x_1 = x\cos\alpha + y\sin\alpha$$
$$y_1 = -x\sin\alpha + y\cos\alpha$$

Next, a translation along the y axis a distance d to yield the x', y' axes.

$$x' = x_1 = x\cos\alpha + y\sin\alpha$$
$$y' = y_1 - d = -x\sin\alpha + y\cos\alpha - d$$

In matrix notation the transformation between the x,y frame and the x', y' frame can be written as:

$$\begin{bmatrix} x' \\ y' + d \end{bmatrix} = \begin{bmatrix} \cos\alpha & \sin\alpha \\ -\sin\alpha & \cos\alpha \end{bmatrix} \begin{bmatrix} x \\ y \end{bmatrix}$$

or:

$$\begin{bmatrix} x \\ y \end{bmatrix} = \begin{bmatrix} \cos\alpha & -\sin\alpha \\ \sin\alpha & \cos\alpha \end{bmatrix} \begin{bmatrix} x' \\ y' + d \end{bmatrix}$$

To find the values of x' corresponding to $y' = \pm$ RM the equation of the orbit must be found in terms of x' and y'. The equation of the orbit in the x,y frame is:

$$\frac{x^2}{a^2} + \frac{y^2}{b^2} = 1$$

or:

$$x^2b^2 + y^2a^2 - a^2b^2 = 0$$

using the transformation:

$$x = x'\cos\alpha - (y' + d)\sin\alpha$$
$$y = x'\sin\alpha + (y' + d)\cos\alpha$$

The equation of the orbit in terms of x' and y' is:

$$x'^2\,[b^2\cos^2\alpha + a^2\sin^2\alpha] + x'\,[2(y' + d)(a^2 - b^2)\sin\alpha\cos\alpha] + [(y' - d)^2\,(b^2\sin^2\alpha + a^2\cos^2\alpha) - a^2b^2] = 0$$

Given $y' = \pm RM$ the corresponding values of x' can be found using the quadratic formula, then these values of x' and y' can be converted to the x,y frame using the transformation matrices. The values of ψ_1 and ψ_2 are found using the x and y coordinates of the orbit at the beginning & end of occultation.

$$\psi_1 = \tan^{-1}\left(\frac{y}{x}\right)_{\text{beginning}} \qquad \psi_2 = \tan^{-1}\left(\frac{y}{x}\right)_{\text{end}}$$

The duration of the occultation is found by using Kepler's time equation to calculate the time from perigee at ψ_1 and ψ_2.

$$t_\rho = \sqrt{\frac{a^3}{\mu}}\,(\psi - e\sin\psi)$$

The duration of the occultation is:

$$\Delta t = t_{\rho_2} - t_{\rho_1}$$

The orbiting spacecraft is occulted for a period of Δt once during each orbit. The value of Δt changes during the staytime as the longitude of Mars changes and the orbit precesses. To obtain the minimum and maximum occultations, Δt should be calculated on Mars arrival and departure.

<u>Mars Lander Sunlight Occultation</u>

The time in darkness that a Mars lander would be subjected to is highly dependant on the latitude of the landing site and the heliocentric longitude of Mars. The amount of daylight varies on Mars just as it does on Earth, since its equator is inclined to its orbit by 23.984 degrees. To calculate the time in darkness, the following calculations are required.

The geometry shown in Figures 3a and 3b represents the orientation of the axis of rotation of Mars with respect to the Sun. η represents the angle between the polar axis and the vertical as viewed perpendicular to the Mars-Sun line. Using Figure 3a, η can be found:

FIGURE 3
GEOMETRY FOR FINDING η

FIG. 3a.

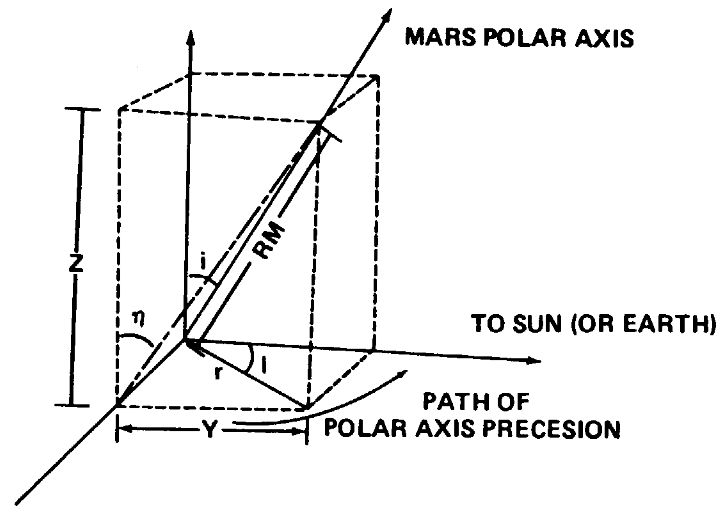
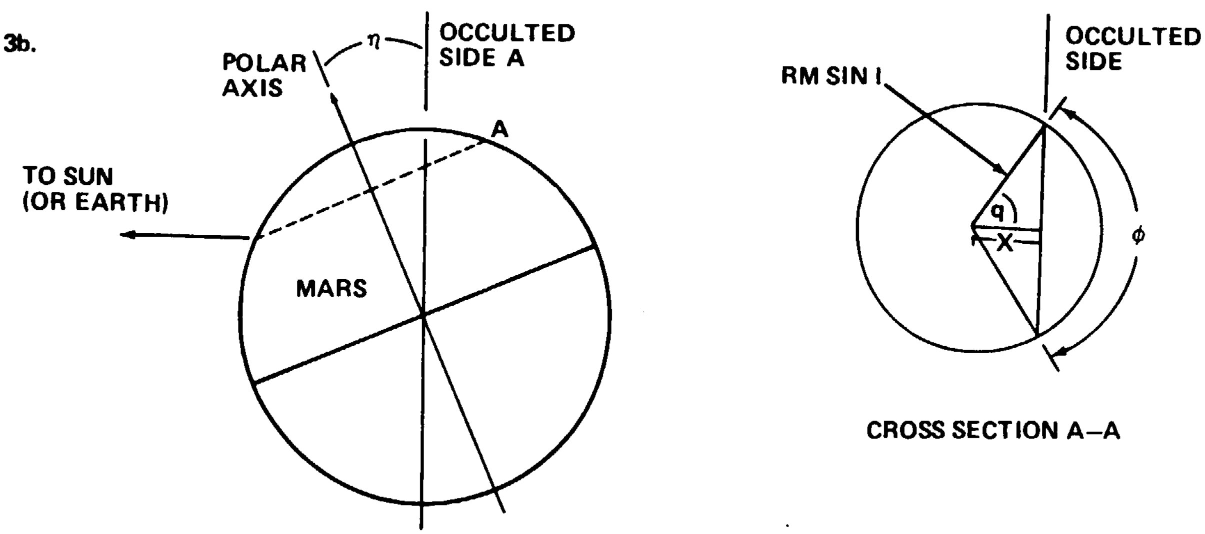


FIG. 3b.



$$i = 23.984 \text{ DEG}$$

$$r = RM \sin i$$

$$y = M \cos L = RM \sin i \cos L$$

$$z = RM \cos i$$

$$\tan \eta = \left(\frac{\cos i}{\cos L} \right)$$

Figure 3b shows the orientation of Mars with respect to the Sun and a cross section of a particular latitude shows how much of that latitude is in the sunlight. The time in darkness is found using the following calculations and Figure 3b.

$$\tan \eta = \left(\frac{RM \sin \ell}{x} \right)$$

$$x = \left(\frac{RM \sin \ell}{\tan \eta} \right)$$

$$q = \cos^{-1} \left(\frac{x}{RM} \right) = \cos^{-1} \left(\frac{\sin \ell}{\cos \eta} \right)$$

$$\emptyset = 2q = 2 \cos^{-1} \left[\left(\frac{\sin \ell}{\cos L} \right) \tan i \right]$$

$$\omega = \frac{2 \pi}{24.5 \text{ hrs}}$$

$$\Delta T = \frac{\emptyset}{\omega} = \frac{1}{\pi} \cos^{-1} \left[\frac{\sin \ell \tan i}{\cos L} \right]$$

<u>Communication</u> <u>Occultation</u> <u>Between</u> <u>The</u> <u>Mars</u> <u>Orbiter</u> <u>And</u> <u>Earth</u>

This calculation is performed using the same procedure as the Mars orbiter sunlight occultation except instead of α being used in the calculations, a different angle, β is used. β is defined as the angle between the semi-major axis of the parking orbit and the Mars to Earth line. The following calculations are required (see Fig. 1).

$$Q = \left| L - LE \right|$$

$$r_e = \left[\frac{a\ (1-e^2)}{1 + e \cos\theta} \right] \text{ Earth}$$

$$r_m = \left[\frac{a\ (1-e^2)}{1 + e \cos\theta}\right] \quad \text{Mars}$$

$$X = r_m - r_e \cos Q$$

$$Y = r_e \sin Q$$

$$J = \tan\left(\frac{Y}{X}\right)$$

$$\beta = \alpha - J$$

Once β is found the procedures for finding the communication occultations are identical to the sunlight occulations. Starting with equation 1, β is substituted for α .

<u>Communication Occultation Between The Mars Lander and Earth</u>

The communications occultation between the Mars lander and the Earth is dependent on the same parameters that influence the sunlight occultation. Since the plane of the Earth's orbit is inclined only 1.849 degrees to the Mars orbit plane, the value of the duration of the communications occultation would be essentially equal to the duration of the sunlight occultation for a given latitude.

<u>Communication Occultation Between The Mars Lander and The Mars Orbiter</u>

The communication occultation of the Mars lander and the Mars orbiter is obtained by finding when the angle between the local vertical at the landing site and the position of the orbiter is greater than 90 degrees. The geometry for this calculation is shown in Figure 4. It was assumed that the orbiter is directly above the lander when it is at perigee. The angle c represents the angle between the vertical and the orbiter. Communication is occulted as long as c is greater than 90 degrees. Using Figure 4, c was obtained with the following calculations. Given a, e, μ and ψ.

$$\Theta = 2 \tan^{-1}\left[\sqrt{\frac{1 + e}{1 - e}} \tan\left(\frac{\psi}{2}\right)\right]$$

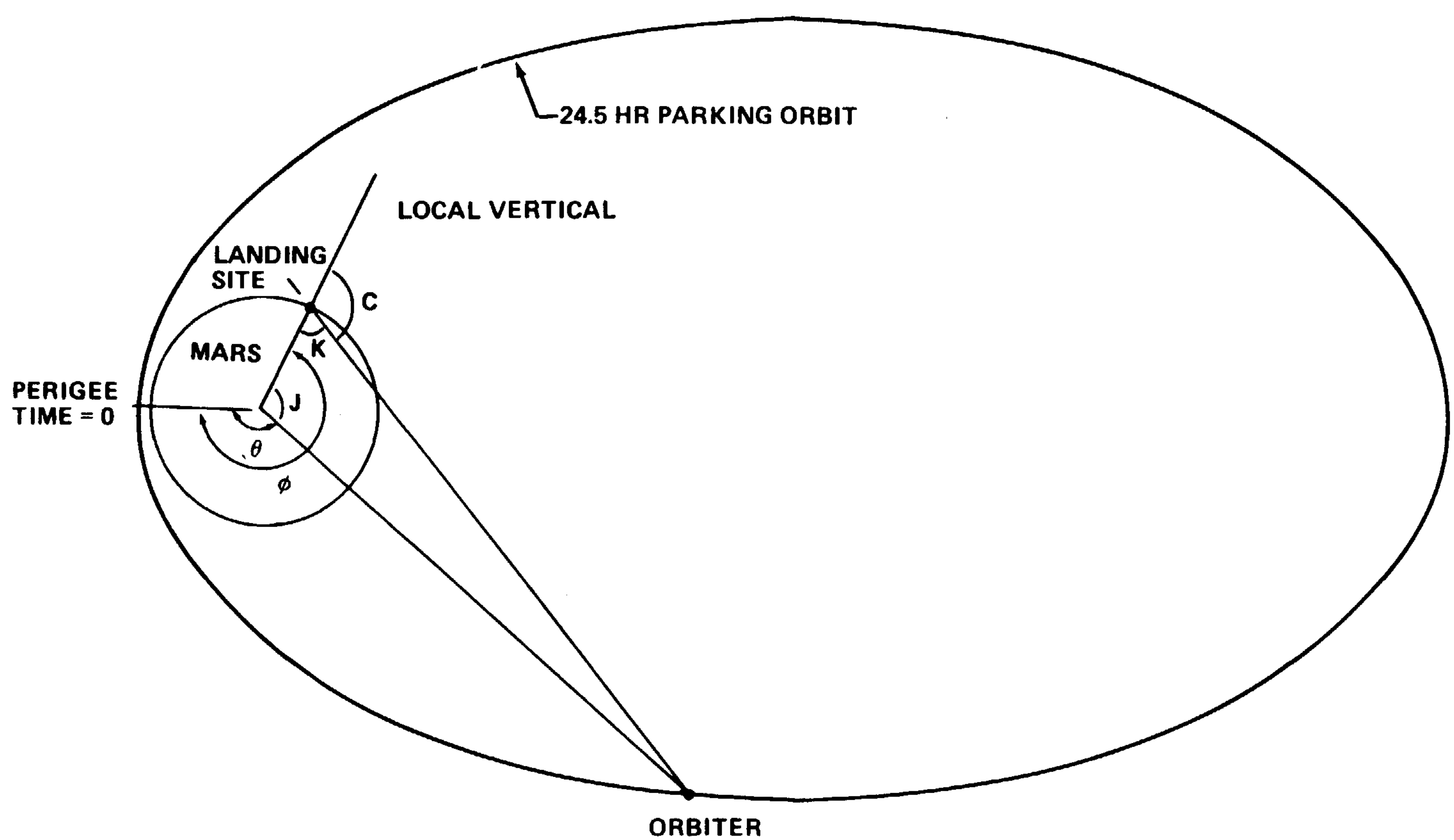
$$t = tp = \sqrt{\frac{a_3}{\mu}} (\psi - e \sin \psi)$$

$$\omega = \frac{2\pi}{24.5} \frac{\text{RAD}}{\text{HRS}}$$

3217–85

FIGURE 4

MARS LANDER – ORBITER COMMUNICATION GEOMETRY



$\Phi = 2\pi - \omega t$ NOTE: Φ is equal to this only when the parking orbit is retrograde, i.e., direction is opposite Mars's rotation.

$$J = \left| \Theta - \Phi \right|$$

$$h = \sqrt{RM^2 + r^2 - 2\,RM\,r \cos j}$$

$$k = \sin^{-1} \left[\frac{r \sin J}{h} \right]$$

$$c = \pi - K;\ h \geq r$$

$$c = K\ ;\ h < r$$

DATA FOR BASELINE, 1999 MISSION

24.5 hour parking orbit

r_p = 3900 km r_a = 36829.2 km

a = 20364.63 km

e = .8084915

MARS ORBITER SUNLIGHT OCCULTATION

	MARS ARRIVAL	MARS DEPARTURE
RAP =	273.816 deg	275.242 deg
VE =	-67.01 deg	-67.01 deg
P =	335.323 deg	335.323 deg
L =	168.915 deg	195.73 deg
α =	16.588 deg	8.801 deg
ΔT =	.267 hr (16.06 min.)	.264 hr (15.85 min.)

MARS LANDER SUNLIGHT OCCULTATION

No landing site has been baselined so, assume occultation of 1/2 Mars day,

$$\Delta t = 12.25 \text{ hrs}$$

COMMUNICATION OCCULTATIONS

ORBITER TO EARTH

	MARS ARRIVAL	MARS DEPARTURE
L =	168.91	195.73
LE =	115.	176.
Q =	53.91	19.73
Θ_E =	12.747	73.477

Θ_M =	193.588	220.41
r_e =	.98367 AU	.99498 AU
r_m =	1.6604 AU	1.62526 AU
X =	1.0809 AU	.68449 AU
Y =	1.315 AU	.33589 AU
J =	50.59 deg	26.138 deg
β =	34.002 deg	17.337 deg
ΔT =	.2823 hr (16.93 min.)	.2677 hr (16.06 min.)

LANDER TO EARTH

Since no landing site has been chosen, assume communication occultation occurs for half of the Mars day.

$$\Delta T = 12.25 \text{ hr}$$

LANDER TO ORBITER

These calculations were made by calculating c for ψ = 0 to 360 degrees, The results are shown below:

tp (hr)	ψ(deg)	c (deg)
0	0	0
.3	18	79.2
.4	24	108.7 occultation begins
7.5	140	89.2 occultation ends
12.25	180	0
17.0	220	89.2
17.2	222	93.2 occultation begins
24.1	336	108.7 occultation ends

ΔT = 7.1 hrs occuring twice every orbit.

REFERENCES

1. Young, A., "Mars Mission Concepts and Opportunities"; MSFC paper in Section II.
2. Planetary Flight Handbook, Volume 3, NASA SP 35 Part 1, 1963.

LIST OF SYMBOLS

a = semi-major axis of an orbit

b = semi-minor axis or an orbit

c = angle between local vertical at landing sight and the Mars orbiter

d = distance from origin of x,y reference frame to the origin of the x_1, y_1 frame

e = orbit eccentricity

i = inclination of Mars' equator to the ecliptic

l = latitude of landing site

L = heliocentric longitude of Mars

LE = heliocentric longitude of Earth

P = heliocentric longitude of Mars perihelion

RM = radius of Mars

RAP = right ascension of perigee of Mars parking orbit

r = distance from center of Mars to orbiter

r_e = distance from center of sun to center of Earth

r_m = distance from center of sun to center of Mars

tp_1 = time from perigee that occultation begins

tp_2 = time from perigee that occultation ends

Δt = duration of occultation

VE = heliocentric longitude of Mars vernal equinox

x,y = coordinates in the x,y reference frame (Mars parking orbit)

x', y' = coordinates in the x', y' reference frame

α = angle between semi-major axis of parking orbit and Mars-Sun line

β = angle between semi-major axis of parking orbit and Mars-Earth line

ψ = eccentric anomaly

η = angle between vertical and the Mars polar axis as seen perpendicular to the Mars-Sun line

μ = gravitational constant km_3/sec_2

Θ = true anomaly

N87-17734

MANNED MARS MISSION TRANSFER FROM MARS PARKING ORBIT TO PHOBOS OR DEIMOS

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ABSTRACT

This paper addresses the problem of orbit transfers from a Mars parking orbit with an inclination of 165 degrees to the Mars moons. The transfer can be accomplished using a three impulse transfer.

The current 1999 baseline manned Mars mission requires a Mars parking orbit with an inclination of 165 degrees. This orbit inclination is necessary due to the direction of the Mars arrival and departure asymptotes of the interplanetary trajectory. The selection of this inclination for the parking orbit minimized the delta velocity requirements at Mars arrival and departure. This presents a problem in making transfers from this orbit to either Phobos or Deimos since it is a retrograde orbit. It is possible to make this transfer efficiently using a three impulse transfer and an intermediate transfer orbit with a very large apogee altitude. This paper will show how the intermediate transfer orbit apogee can be determined based on a preselected transfer time, the delta velocities required as a function of transfer time, and the propellant required as a function of mission module weight for a transfer time of 5 days. The data presented in this paper is specifically for the 1999 opposition class mission but the methods outlined are applicable to any other mission which requires a high inclination parking orbit.

DISCUSSION

The three impulse transfer begins with a propulsive burn at the apogee or perigee of the parking orbit which puts the spacecraft into an orbit with a very high apogee. The apogee of this intermediate orbit is selected on the basis of a desired transfer time. When the spacecraft reaches the apogee of the intermediate orbit, its orbital velocity is at its minimum value. At this point, the second impulse is made to perform the desired plane change. The second propulsive burn puts the spacecraft into a posigrade transfer orbit to Phobos or Deimos which is in the plane of the moons' orbit. The third impulse is made when the spacecraft reaches Phobos or Deimos. This propulsive burn puts the spacecraft into the moons' orbit. To return from Phobos or Deimos to the original parking orbit, the sequence is reversed.

The calculations required to determine the altitude of the intermediate transfer orbit are as follows:

$$a_x = \frac{r_1 + r_2}{2}$$

$$a_x = \frac{r_2 + r_3}{2}$$

$$\Delta V_1 = \left| \sqrt{\mu\left(\frac{2}{r_1} - \frac{1}{a_1}\right)} - \sqrt{\mu\left(\frac{2}{r_1} - \frac{1}{a_{x_A}}\right)} \right|$$

$$\Delta V_2 = \sqrt{V_{x_A}^2 + V_{x_B}^2 - 2V_{x_A} V_{x_B} \cos(\Delta i)}$$

where:

$$V_{x_A} = \sqrt{\mu\left(\frac{2}{r_2} - \frac{1}{a_{x_A}}\right)}$$

$$V_{x_B} = \sqrt{\mu\left(\frac{2}{r_2} - \frac{1}{a_{x_B}}\right)}$$

$$\Delta V_3 = \left| \sqrt{\mu\left(\frac{2}{r_3} - \frac{1}{a_3}\right)} - \sqrt{\mu\left(\frac{2}{r_3} - \frac{1}{a_{x_B}}\right)} \right|$$

$$t_x = \pi\left(\sqrt{\frac{a_{x_A}^3}{\mu}} + \sqrt{\frac{a_{x_B}^3}{\mu}}\right)$$

The value of r_2 can be found by iteration of the above calculations until the desired transfer time, t_x is achieved.

Phobos and Deimos could be visited sequentially during the same mission. The delta velocity required between the orbits of Phobos and Deimos is 2,460 feet per second. The total delta velocity for the sequential visit is obtained by adding this value to that for a one way transfer from the parking orbit to the first moon plus the delta velocity for a one-way transfer from the second moon back to the parking orbit. Figure 1 shows a profile of the three-impulse transfer from the parking orbit to either Phobos or Deimos. Figure 2 shows the altitude of the apogee of the transfer orbit as a function of transfer time. Figure 3 shows the one-way delta velocity requirement as a function of transfer time. Figure 4 shows the propellant required for a 5 day transfer to Phobos or Deimos as a function of mission module weight. These data are based on the assumption of a mass fraction of .84 and an I_{sp} of 370 seconds.

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FIGURE 1.
TRANSFERS TO MARS MOONS
3–IMPULSE TRANSFER

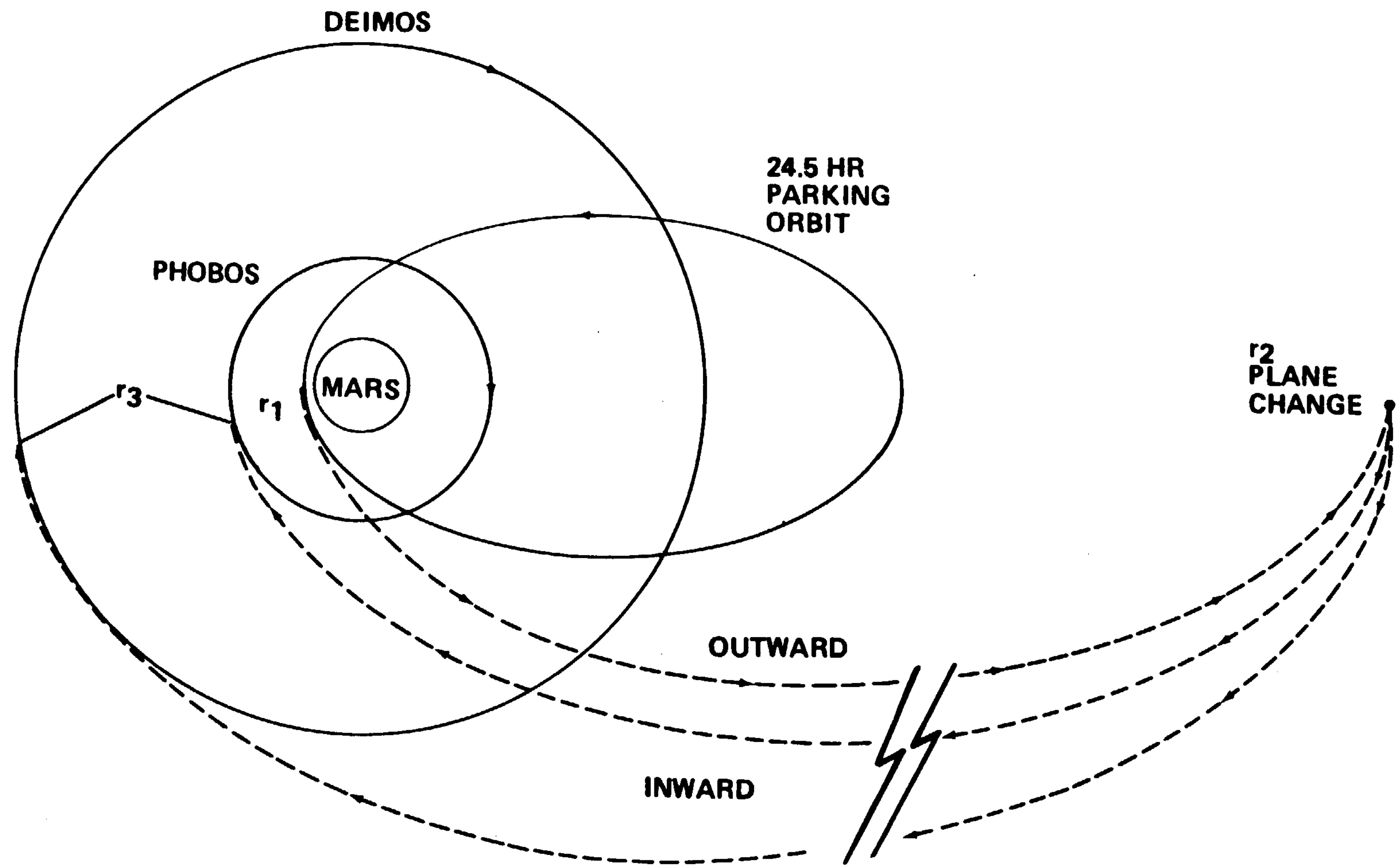

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FIGURE 2
TRANSFER ORBIT APOGEE AS A FUNCTION OF TRANSFER TIME FOR TRANSFERS FROM MARS PARKING ORBIT TO PHOBOS OR DEIMOS

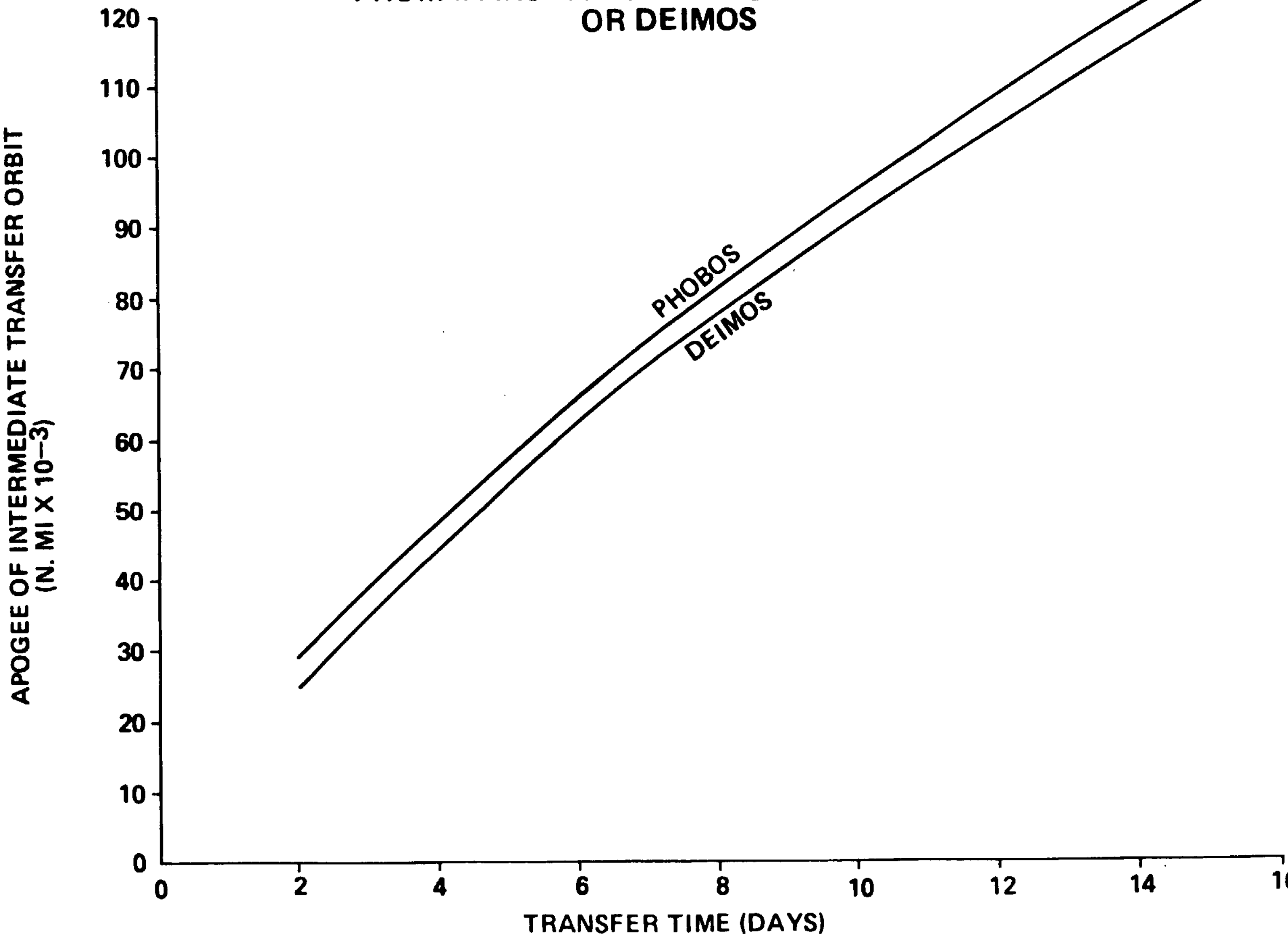

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FIGURE 3
DELTA VELOCITY AS A FUNCTION OF TRANSFER TIME FOR TRANSFERS FROM MARS PARKING ORBIT TO PHOBOS OR DEIMOS

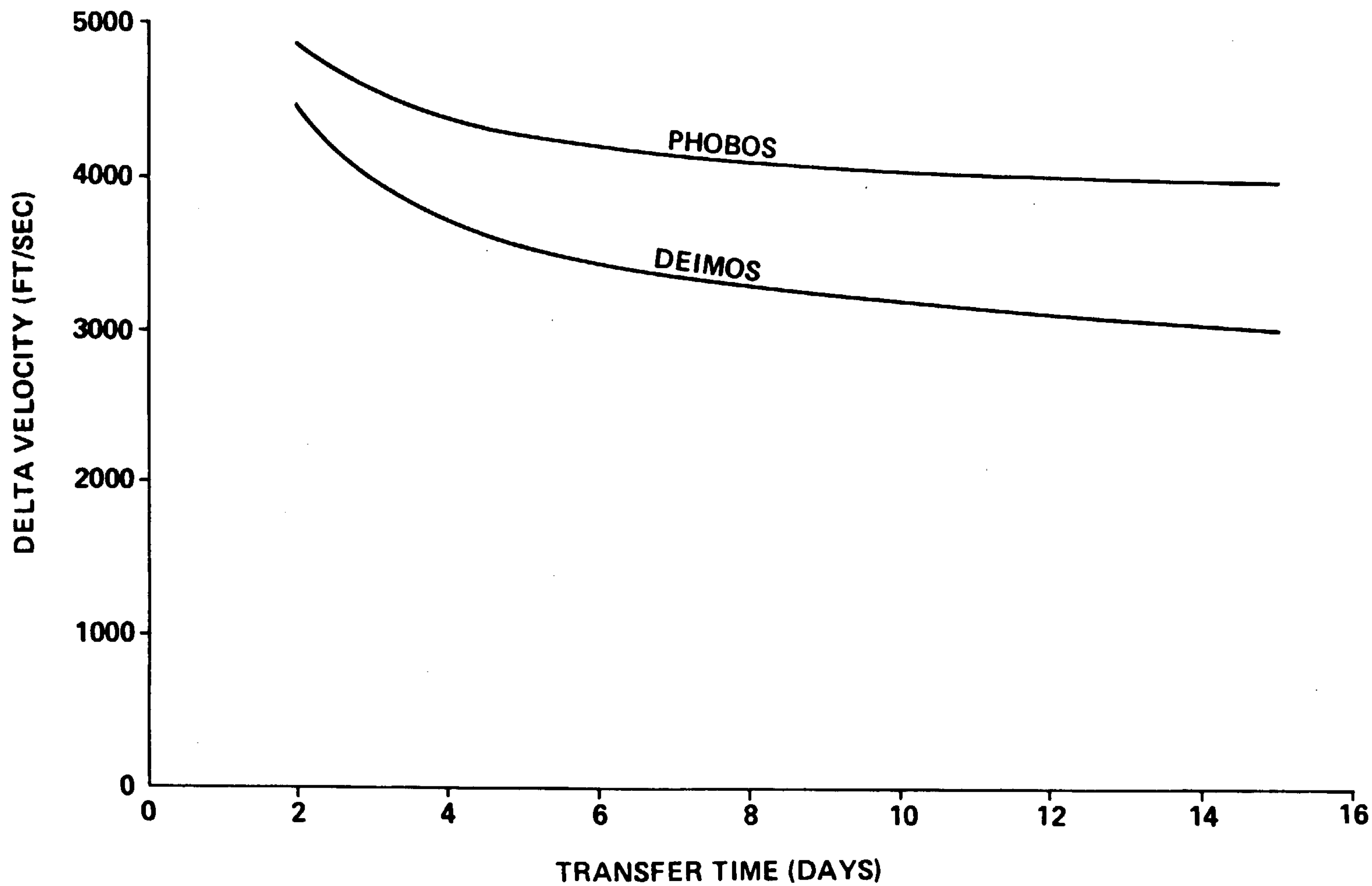

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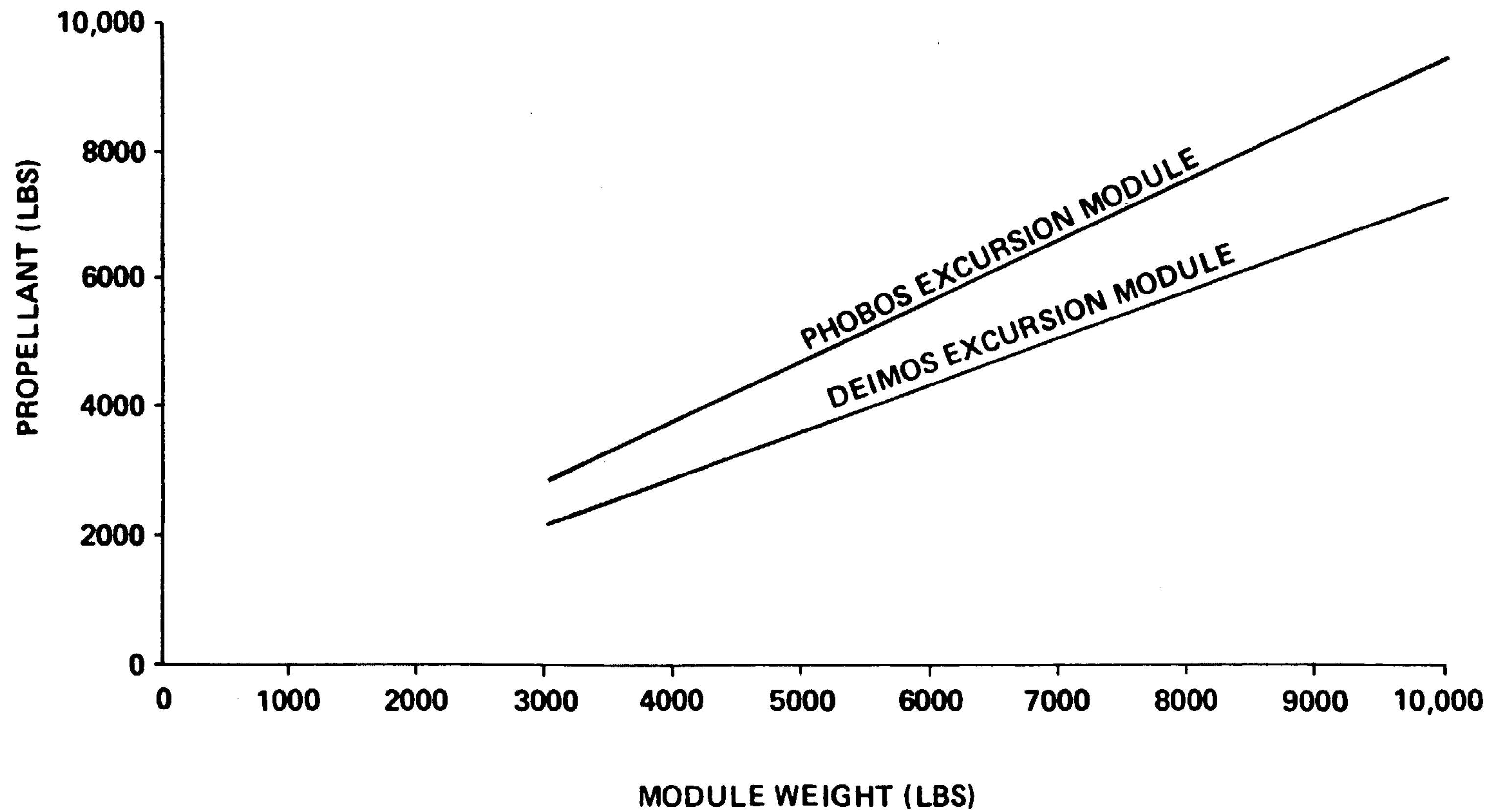
FIGURE 4

PROPELLANT WEIGHT REQUIRED AS A FUNCTION OF EXCURSION MODULE WEIGHT FOR A ROUND TRIP FROM A 24.5 HR. PARKING ORBIT TO THE MARS MOONS.

– PARKING ORBIT: 270 X 18050 AT 165° INCL.
– 3 IMPULSE TRANSFER
– ISP = 370



LIST OF SYMBOLS

a_1 = semi-major sxis of Mars parking orbit

a_3 = semi-major axis of Mars moon orbit

a_{XA} = semi-major axis of first transfer leg

a_{XB} = semi-major axis of second transfer leg

Δi = plane change angle

ΔV_1 = first delta velocity

ΔV_2 = second delta velocity

ΔV_3 = third delta velocity

V_{XA} = apogee velocity of transfer orbit before the second impulse

V_{XB} = apogee velocity of transfer orbit after the second impulse

r_1 = orbit radius at first impulse

r_2 = apogee of intermediate transfer orbit

r_3 = orbit radius at third impulse

t_x = transfer time

μ = gravitational constant for Mars = 42,860 $\frac{km^3}{sec^2}$

I_{sp} = specific impulse

N87-17735

THE EFFECT OF MARS SURFACE AND PHOBOS PROPELLANT PRODUCTION ON EARTH LAUNCH MASS

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ABSTRACT

Fuel and oxidizer produced on the surface of Mars and on the Martian moon Phobos can reduce the cumulative mass of fuel and oxidizer which must be launched to low Earth orbit for Mars exploration missions.

A scenario in which ten conjunction class trajectory missions over a twenty year period land a surface base and propellant production facilities on the Martian surface and on Phobos was examined. Production of oxygen on Phobos provides the greatest benefit. If all the propellant for Mars operations and Earth return is produced at Phobos and on Mars, a 30% reduction in cumulative LEO mass can be achieved at the end of the 20 year period.

INTRODUCTION

Manned missions to Mars utilizing cryogenic oxygen/hydrogen or oxygen/propane engines can benefit from the production of propellants on one of Mars' moons (Phobos or Deimos) or on the surface of Mars, to provide propellant for the return trip. Cases where either oxidizer or oxidizer and fuel are produced on Phobos (or Deimos) and or Mars are presented here. The mission concept utilized is a conjunction class mission, described in Reference 2, utilizing a 500 km, 24 hr elliptical parking orbit with a 500 km periapsis at Earth and Mars. A small Mars-orbit transfer vehicle Mars-OTV is utilized between the elliptical Mars orbit and low circular Mars orbit, Phobos or Deimos. Table 1 gives delta V requirements for various legs of the trip. A conjunction class opportunity is available on approximately 2-year centers (each round trip requires three years). As requirement for conjunction class missions do not vary much from opportunity to opportunity, a generic set of delta Vs was used here. A base building scenario requiring 10 missions over a 20 year period was examined.

Table 2 describes mission components and delivery capabilities. Each mission delivers 44.7 MT of payload which remains on Mars. In ten

TABLE 1

DELTA V's AND PROPULSION CHARACTERISTICS

		ISP	PROP.	MASS FRACT.
Trans Mars Injection (TMI) (departing from 500 km circular Earth orbit)	- 3.808 km/sec	468	LO_2/H_2	.925
Mars Orbit Insertion (MOI) (into 500 x 32,963 km, 24 hour ellipse)	- 1.666 km/sec	370	LO_2/H_2	.85
Trans Earth Injection (TEI) (departing from 24 hour ellipse)	- 1.490 km/sec	370	LO_2/prop	.94
Earth Orbit Insertion (EOI) (into 500 x 71,00 km, 24 hour ellipse)	- .967 km/sec	370	LO_2/prop	.89
Mars 24 hour, 30 deg. inclination ellipse to Deimos, one way	- .900 km/sec	460	LO_2/LH_2	.68
Mars 24 hour, 30 deg. inclination ellipse to Deimos, one way	- .750 km/sec	460	LO_2/LH_2	.68
Deorbit from 24 hr Mars ellipse	- .100 km/sec	360.5	LO_2/MMH	
Landing on Mars surface	- 1.000 km/sec	360.5	LO_2/MMH	
Ascent from Mars surface to 500 km	- 4.500 km/sec	360.5	LO_2/MMH	

TABLE 2

SPACECRAFT WEIGHTS AND PROPULSION AND DELIVERY CHARACTERISTICS

Each Baseline Mission Consists of:

One Mission Module (or round trip crew compartment)	-	53 M. tons
Three expendable landers	-	62 M. tons each
Two manned landers carry ascent stages and	-	9.1 M. tons cargo (each)
One unmanned lander for cargo (descent stage only)	-	26.5 M. tons cargo
One (loaded with 21 metric tons of propellant) expendable Mars OTV	-	31.00 M. tons
Each Baseline mission delivered cargo	-	44.7 M. tons

Lander Characteristics:

Manned Lander ascent inert	-	3.8 M. tons
Manned Lander total ascent propellant (oxygen/propane)	-	13.6 M. tons
Manned Lander total ascent oxygen	-	8.4 M. tons
Manned and Cargo Landers total descent propellant (oxygen/propane)	-	20.7 M. tons
Manned and Cargo Landers descent oxygen	-	12.8 M. tons

missions, approximately 447 MT could be delivered to Mars, which could emplace a base with the characteristics shown in Table 3.

DEVELOPMENT SCENARIOS

In order to assess the effect of producing propellant at Mars the following scenario were assumed.

Baseline Reference

No Mars propellant was assumed. All fuel and oxygen were brought from Earth. One mission was flown every conjunction opportunity (every 2 years) for 20 years. Each mission carried one manned mission module (MM) plus 3 expendable landers to Mars orbit. The three landers are alike and all weigh the same. Two of the landers carry manned ascent stages plus consumables to the surface. The third lands unmanned carrying 26 tons of Base elements for the permanent Martian Base. The MM is returned to low Earth orbit at the end of the mission.

Each mission also carries a fueled Mars orbital transfer vehicle (Mars-OTV) which allows exploration of the Martian moons, Mars orbital mapping, and in-orbit rescue, etc. Throwaway propulsive stages were sized for each mission. Table 3 shows the base masses landed on Mars surface. The masses are the same as for a lunar base previously developed (Ref 3).

In-Situ Propellant Production (ISPP) Scenarios

Scenarios were investigated in which oxygen-only and oxygen-plus-fuel were produced by delivery of production plants to Phobos and Mars. The Mars surface base buildup progresses at the same pace for all the scenarios. The ISPP scenarios thus require increased mass during the early missions to deliver the propellant production plants.

Missions 1 and 2 would deliver the Phobos O_2 or O_2 and fuel plants <u>in addition</u> to the normal mission cargo. The Phobos O_2 plant is estimated at 50 metric tons. These missions would also have to carry a total of 12 extra tons of Mars-OTV fuel (above baseline missions) to transport the plant to Phobos. A Phobos plant which could produce both oxygen and fuel is estimated at 75 tons plus 18 tons <u>extra</u> Mars-OTV fuel. These weights are carried in addition to the reference mission weights. Mission 3 and subsequent missions are then refueled from this plant.

TABLE 3

MARTIAN BASE ELEMENTS (DERIVED FROM LUNAR BASE ELEMENTS)

0	Habitats - 5 X 17.5 M. tons each (13 or 26 M. ton units)	- 87.5 M. tons
0	Power units - 3 X 17.5 M. tons each	- 52 M. tons
0	Earthmover/Crane - 1 at 26 M. tons	- 26 M. tons
0	Surface 02, pilot and production plants = 3 X 17.5 M. tons each	- 52 M. tons
0	Pressurized mobility unit 3 X 17.5 M. tons	- 35 M. tons
0	Geo/Chem lab - 2 X 17.5 M. tons	- 35 M. tons
0	Workshops - 2 X 17.5 M. tons	- 35 M. tons
0	Ceramics & metalurgy plants 2 X 17.5 M. tons each	- 35 M. tons
0	Misc. mobility - 2 X 17.5 M. tons	- 35 M. tons
0	Total	- 392.5 M. tons

Figure 1 shows a low-g Phobos propellant production plant concept and an Mars-OTV delivering propellant.

The Mars surface O_2 production plant weighs 16 metric tons, to be delivered on the third mission. Another O_2 plant is already in place, landed on the first two missions as part of the base. The surface O_2 and fuel plant combined would weigh 56 metric tons. This combination would be landed on mission 3 and 4. These plants would be landed in the place of the normally scheduled base elements. The replaced cargo would be brought down on later missions after propellant production has started.

MISSION DESCRIPTION

The reference mission at departure from Earth consists of the MMM, 3 Mars landers, 1 Mars-OTV, two LO2/propane propulsive stages for return from Mars and two LO2/LH2 propulsive stages for transport for Mars.

The first LOX/LH2 stage performs the Trans Mars Injection (TMI) burn and is then discarded. When Mars is reached several hundred days later, the second LO2/LH2 stage is used for Mars Orbit Insertion (MOI) placing the stack into a 24 hour elliptical (500 kmx 3 3,000 km) parking orbit around Mars at an inclination of around 30%. The landers are separated and aerobrake to low circular parking orbits to await proper alignment and phasing for precision landing at the base site. Meanwhile, the MOTV is used to visit and explore the Martian moons and for detailed Mars in-orbit mapping at the end of the mission (1.5 years later) the ascent stages bring the crew back up to the MMM. They are then discarded. The MOI stage is discarded and the first LO2/propane stage performs the trans-Earth injection burn (TEI). This stage is then discarded. The original Mars parking orbit was selected so that natural precession will have so placed the orbit so that this TEI departure burns at periapsis.

When Earth is reached all that remains is the MM plus the final LO2/propane stage which provides Earth orbit insertion (EOI) into a 24 hour (500 km x 71,000 km) ellipse.

If oxygen alone is produced on Phobos the scenario is the same except that the Earth return stages (LO2/prop.) and the landers leave Earth with empty oxygen tanks. After Mars orbit is reached, the MOTV flys to Phobos and brings back oxygen to fill these tanks before continuing the mission. If oxygen and fuel (most probably Hydrogen) are both available at Phobos, the LO2/prop stages are not carried at all and

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Figure 1

Phobos Propellant Plant

the landers propellant tanks are carried empty. At MOI the MOTV flys to Phobos and returns with fuel for the landers and also refuels the stage which was used for Mars orbit Insertion. This stage is no longer discarded but instead is used to return the MMM to Earth (both TEI and EOI burns).

GROUNDRULES

1. Conjunction missions are used throughout.
2. All interplanetary maneuvers are propulsive. No aerobraking capability is assumed except for the landers.
3. Earth departure is from 500 km circular LEO.
4. Mars parking orbit is a 500 x 33,000 km 24 hr. ellipse.
5. This Mars parking ellipse can be positioned at Mars insertion so that natural precession effects will align the orbit properly for departure to Earth.
6. The spacecraft returns to a 24 hour ellipse at Earth.
7. Transport of fuel, mining plants, etc. in Mars orbit will be provided by the Mars-OTV.
8. LO2/LH2 propellants were used for transport to Mars and LO2/propane were used for return because of the difficulty of storing LH2 for long periods in Mars orbit. When propellant was produced at Mars the appropriate tanks were simply carried empty from Earth and filled at Mars. It was assumed that the stages could be altered to burn whatever fuel was available at Mars, ie., the ascent stages would be altered to burn LO2/LH2 if H2 is available on the Martian surface.
9. Propellant produced on the surface of Mars is only used for fueling the ascent stages.

RESULTS

Figure 2 shows the case where all stages are loaded with fuel and oxidizer at Phobos or Mars wherever they arrive empty. The scenario requires more mass in LEO in the early years than the baseline which assumes no Phobos or Mars propellant production, as these early missions must transport the machinery or propellant to Mars. After the second mission, cumulative gains in performance are realized. Extrapolating the results beyond the 20 year period of Figure 1 gives the results of Table 4. The longer the program, the greater the benefit of producing

TABLE 4

Years Since Program Start	Percent Reduction in Cumulative LEO Mass at the given year	
	O2 and Fuel Production	O2 Only
20	31	23
40	42	32
60	46	35
80	48	36

propellant at Mars. Improvement in performance (weight required in LEO) from 23% to nearby 50% in a very long program are possible.

Figure 3 shows the cumulative weight reduction versus year for the best case, with propellants provided to all stages, and for a case with propellants provided to all stages except the lander descent stage. Landers may not initially be designed for propellant loading in space. The payback for designing in this feature is shown.

Figure 4 shows the cumulative weight reduction if only oxygen is produced for all stages except the lander descent stages. Phobos oxygen for the lander descent stages results in a savings of 7% more over a twenty year period than with LEO delivered descent stage oxygen.

Figure 5 shows the effect of only producing oxygen on Mars and for producing oxygen and fuel on Mars Oxygen production alone results in a 5.5% savings over a twenty year period and oxygen and fuel saves 7.5% of the no-ISPP total LEO mass. Figure 5 shows no initial gain in LEO mass because early optional cargo mass is just replaced with plant mass, and the initial cargo is then brought down later, after propellant production has started.

Figure 6 shows the effect of oxygen, and oxygen and fuel production on Phobos. The (Mars-STS) lander ascent and descent stages, are loaded with propellant at Phobos. Phobos propellant production alone produces a 25% savings over a twenty year period.

Figure 6 shows the effect of using Phobos produced oxygen and fuel in the Mars-STS and descent stages and using them only in the Mars-STS. Figure 5 shows a roughly 15% gain at the end of twenty years, if the descent stages are loaded with propellant at Phobos.

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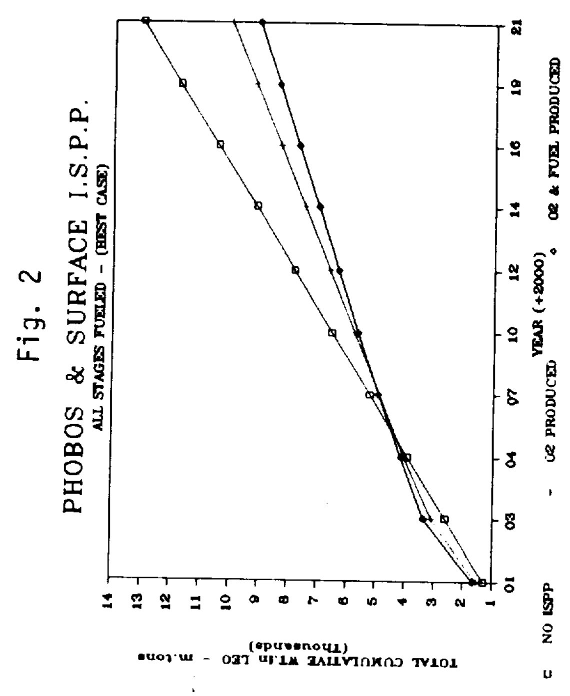

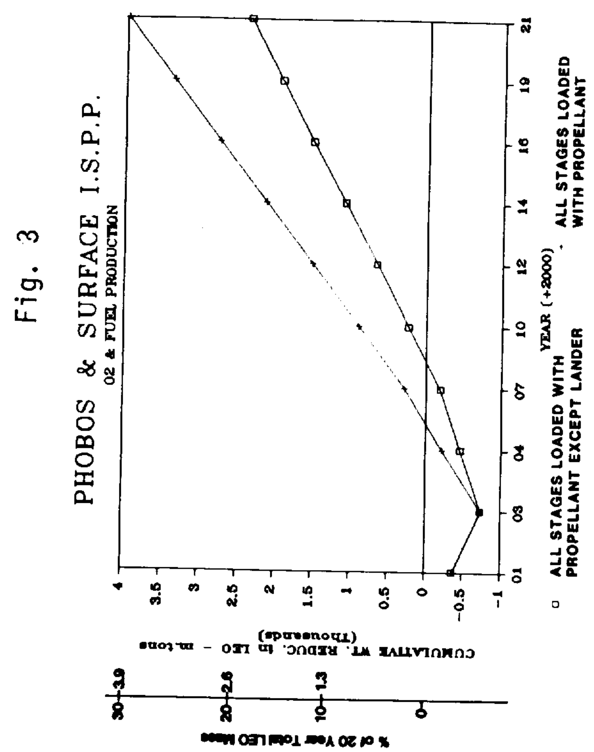

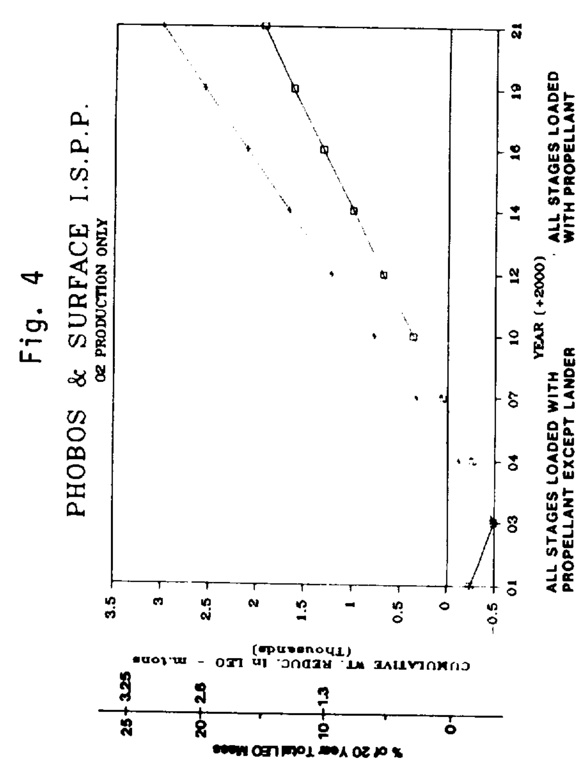

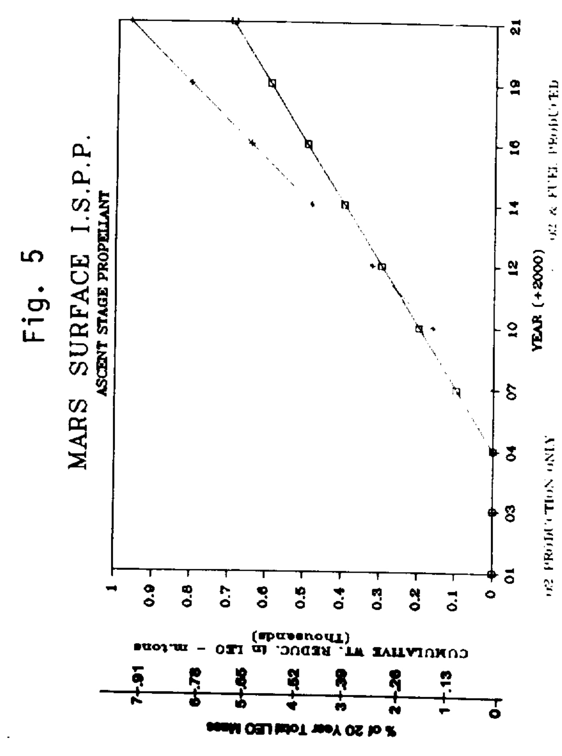

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Fig. 6

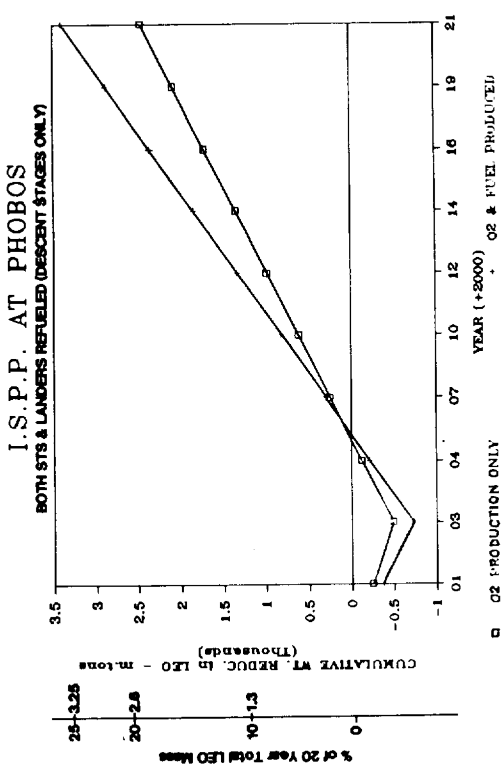


Fig. 7

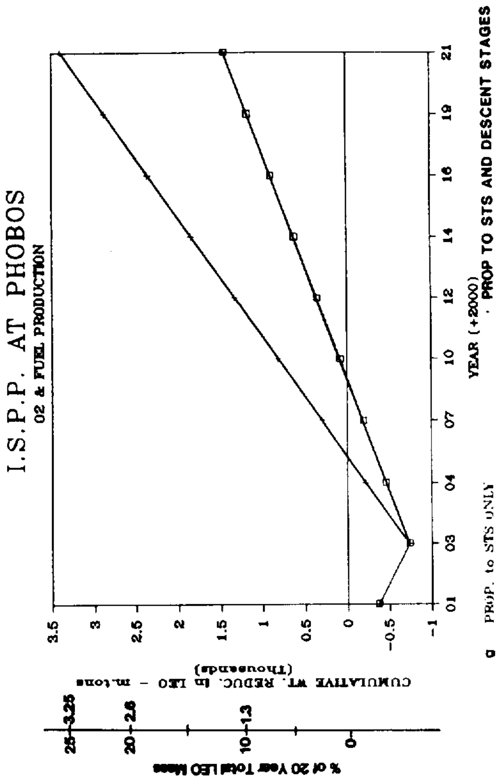


Fig. 8

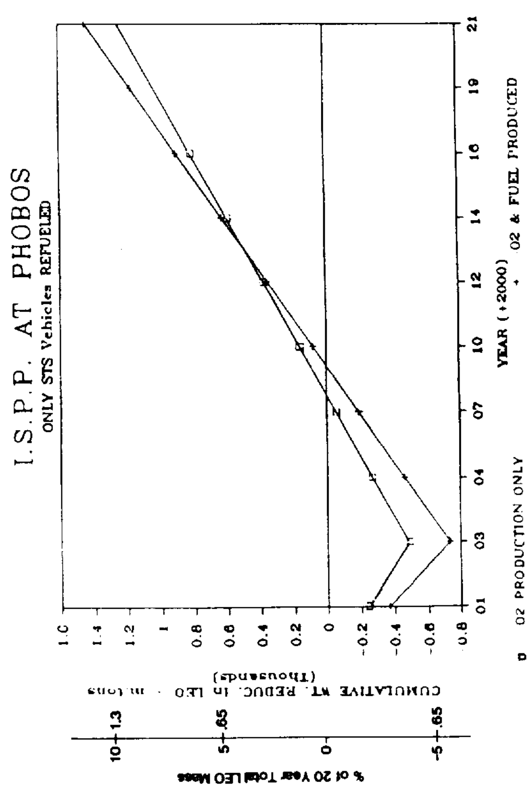
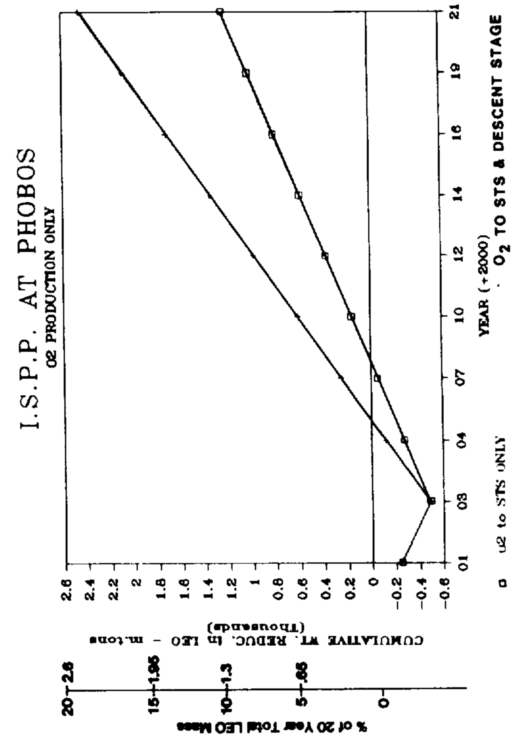


Fig. 9



C-3

Figure 7 compares the effect of producing all propellant on Phobos, or oxygen only, if the Mars-STS is loaded with propellants. The benefit of producing fuel is small; almost all the gain comes from the production of oxygen.

Figure 8 shows the effect of oxygen only production for the Mars-STS only and Mars-STS and the descent stages. Loading the descent stages with oxygen results in a roughly 10% gain at the end of twenty years.

TMI PROPELLANTS FROM PHOBOS

There is one other technique that may decrease the LEO mass requirement: return propellant from Phobos or Deimos to Earth orbit to be used in the initial trans-Mars injection burn, where most of the total propellant is consumed.

Studies of lunar derived oxygen (Ref. 4) have shown it possible to return more oxygen from the lunar surface to LEO than the required hydrogen sent to LEO, even if all hydrogen must come from Earth. Ref. 5 addresses the use of lunar derived propellants for a manned Mars program. The economics of such an operation are still being studied. The mass payback ratio (propellants returned from the Moon over propellants sent from the Earth) ranges from just over one if all hydrogen must be transported from Earth to as high as 20, if hydrogen can be produced on the Moon. This mass payback ratio is sensitive to aerobrake mass and boiloff and very sensitive to whether lunar hydrogen can be used.

It requires less delta V to get from LEO to Phobos and return than that required for a round trip from LEO to the lunar surface (Table 5).

Thus, there is a performance advantage to using propellants from Phobos delivered to LEO. However, Phobos propellant production for Earth return will almost certainly require 1,000 days round trip for the transportation return, and the large problems of large scale low-g mining may be significant. Thus, the technology and economics are not clear and the concept requires more study.

CONCLUSION

In a long term exploration of Mars with frequent repeated missions, propellant production at Phobos and on the Mars surface offer sufficient performance gains to warrant further study.

TABLE 5

(both cases use Earth aerobraking, all delta Vs in km/sec)

LEO-Mars Orbit-LEO			LEO-Lunar Surface-LEO		
TMI	-	3.7	TLI	-	3.3
MOI	-	1.1 (without aerobraking)	LOI	-	1.0
	-	.1 (with aerobraking)			
To Phobos Orbit	-	.8	Lunar Descent	-	2.1
From Phobos Orbit	-	.8	Lunar Ascent	-	1.9
TEI	-	.9	TEI	-	1.0
EOI	-	.2	EOI	-	.1
TOTAL	-	7.5 (without aerobraking)	TOTAL	-	9.4
	-	6.5 (with aerobraking)			

Most of the gain is realized by simply having a Phobos oxygen plant and in-orbit refueling. This has the advantages of not requiring a single permanent Mars surface base. Each mission could land at a different spot for wide-spread exploration and still realize the gain from a Phobos plant.

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EXCERPTS FROM SOLAR SAIL CONCEPTS AND APPLICATIONS

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ABSTRACT

This paper excerpts material applicable to Mars missions from an earlier study covering a broader range of applications of solar sails. The basic principles of solar sail operation are provided, and the implications on trajectories and missions ate discussed briefly. Concepts of solar sails and interplantary vehicles are described and discussed. Some of the important solar sail material considerations are presented and some selections criteria are provided.

INTRODUCTION

Most of the mission analysis work on solar sails has been done since 1975, yet it has never been collected for publication. This memorandum is a revision and update of a 1976 draft report.

Most of the work presented herein was done at the Jet Propulsion Laboratory, California Institute of Technology, and was sponsored by the National Aeronautics and Space Administration.

In order to minimize the size of this section, all information not directly related to solar sail technology and Mars missions has been excised. The mission analysis is clearly out of date and not applicable to mission opportunities at which a manned Mars mission might be flown. However, the data will suffice to give insight as to the general capabilities of a solar sail vehicle to support Mars missions. The purpose for including this information is to provide some data on possible alternative approaches to a manned mission.

The solar sail is a means of using solar radiation directly as a method of propulsion. The sail is a large, flat, lightweight, highly reflective first-surface mirror. Mission applications for the solar sail range from probes to the Sun to trips to all of the planets and escape from the solar system. The solar sail concepts currently considered the most promising are based upon supporting the sail by means of spars and, alternatively, by centrifugal force. Astronaut assistance in the testing, development, and operation of solar sails may become very desirable.

PRINCIPLES

Reflection

Photons carry momentum, therefore when they are reflected they experience a change in momentum and a force is exerted against the reflecting surface. This resulting force is proportional to the incident solar radiation power. It is inversely proportional to the square of the solar distance and is proportional to the cosine squared of the angle between the sail and the direction of the Sun. This force is also proportional to the reflectivity of the mirror surface and, therefore, performance of the solar sail is also proportional to the surface reflectivity. This case of the ideal sail is illustrated in Figure 1.

Solar Wind

The solar wind is composed of electrons, protons, and heavier charged particles. The solar wind particles which impact a sail will exert a very light force which is several orders of magnitude less than the pressure from solar radiation. The solar wind may have a degradation effect upon the reflectivity of the solar sail because of erosion of the reflecting surface by the particles.

Performance

For a given reflectivity, the inherent performance of a solar sail is a function of the total unit loading on the sail, that is , the total mass of the sail plus supporting structure and mass of the spacecraft divided by the total sail area. Most solar sail missions can be flown with a wide range of total unit loads on the sail. A heavier payload necessarily means a heavier unit load on a particular sail and a longer trip time. Missions to Mercury, for example, may have sail loadings as much as $50g/m^2$ or greater while the requirement for a rendezvous with Halley's comet may be as low as $6.1g/m^2$. The mission to Halley's comet has the most demanding requirement in terms of the sail unit load of any of the missions so far identified for the 1980's. If a sail were constructed of currently available materials, the resulting total unit load might range from about 7 to 10 g/m^2. Thus the mission to Halley's comet may require some improvement in the current technology of materials processing; whereas for other missions which are less demanding, currently available materials may be quite satisfactory.

Trajectories

The sunlight acting upon the sail results in a component of force continuously acting in the radially outward direction from the Sun, unless the sail were turned edge-on to the Sun. The sail may be tilted so as to have a force component perpendicular to the solar radius line. This component may be directed along the velocity vector to increase the energy and angular momentum of the vehicle, moving the vehicle outward, or it may be directed against the velocity vector, reducing energy and angular momentum and allowing it to spiral in toward the Sun. This lateral force component may also be directed out of the plane of the vehicle's velocity vector, thereby changing the inclination of its orbit. In spite of the continuous existence of the radially outward force component, the solar sail is very versatile and can probably be directed to any destination in the solar system envisioned as a target in this century.

MISSIONS

Inner Planets and Solar

An interesting comcept for a solar sail vehicle is that of the role of an inner planet shuttle. This vehicle is envisioned as being a reuseable solar sail which would have the role of delivering spacecraft to various inner planets or solar orbit. The sail may carry multiple payloads on a single mission, and after completing all of its deliveries would return to an Earth parking orbit for its next mission. While the sail is in this orbit, it may undergo any necessary repairs or refurbishment prior to its next mission. If a solar sail is developed for use with the Halley's comet mission, it may be feasible to design the sail module in such a manner that it can readily be adapted to a reuseable configuration.

The sail would enable the return of a sample from Mercury, and if used at Mars, could probably provide for the return of a sample significantly greater than what could be achieved by purely ballistic means. A Mars lander of 5 to 6 tons might be delivered by a sail of the design used for a Halley rendezvous.

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FIGURE 1. SOLAR PRESSURE ON SAIL

$$\text{PRESSURE} = 0.90 \times 10^{-5} \times \frac{\cos^2\theta}{R^2} \quad \text{N/m}^2$$

TOTAL SAIL LOADING

σ_T = TOTAL MASS/SAIL AREA, g/m^2

CHARACTERISTIC ACCELERATION

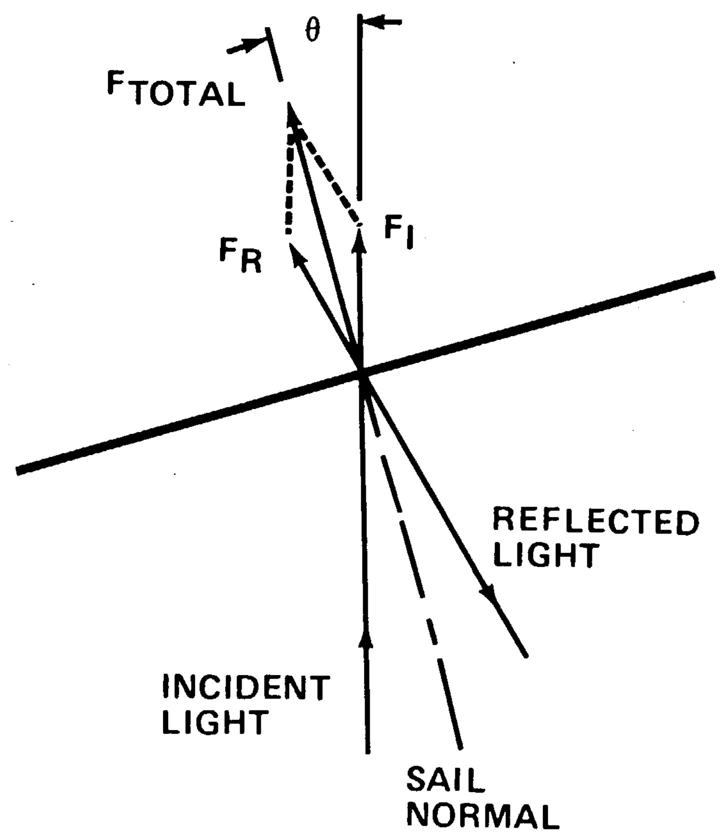
$a_c = 9 \times \eta/\sigma_T$, mm/sec^2

(η = SAIL EFFICIENCY $\approx$ 85%)

TYPICAL VALUES

$0.2 \leqslant a_c \leqslant 1.3$

$40 \geqslant \sigma_T \geqslant 6$



CONCEPTS

Many concepts for solar sail configurations have been considered since the sail first appeared in the literature. These concepts have included a parachute type, the heliogyro, and others. All of these concepts are still being considered; however, the following concepts are those which appear to be the most promising at the present time.

Square Sail

The square sail and the heliogyro were studied extensively for the Halley Rendezvous mission (Friedman, 1978). Although they are very different design concepts, they were found to have essentially the same performance capability for that mission. Both designs were found to be workable, but the heliogyro was selected for that mission.

The square sail is supported by spars extending to the corners of the sail. For a large sail it is necessary to stabilize the spars with tension lines to avoid massive spars. This would mean using a mast and numerous mast-spar and spar-spar lines. Although the design may be intricate, it has a low structure-mass-to-sail-area ratio. Automated deployment is possible but entails high risk. This is responsible for the decision against the square sail for the Halley's comet mission.

The spacecraft is 3-axis stabilized, with attitude control provided by solar pressure venes (small solar sails themselves), or by a center-of-mass shift mechanism, or both. Once the sail is deployed, the structure can remain essentially dynamically inert. The spacecraft is easy to control and can be balanced in the desired attitude. Attitude changes typically require less than one hour, and up to a few hours for large changes.

Sail area can be up to about 10^6 m^2 with automatic deployment, and several times that if erected in space.

Heliogyro

The heliogyro has a shape and dynamic function like a helicopter rotor. It can have 3 or more blades; the Halley's comet design had 12, in 2 banks of 6. The blades form the reflective surface with the sail material supported by edge tendons.

The blades are stored on rollers for launch. After the spacecraft receives an initial spin-up, the blades are partly unrolled. The blades are given a collective pitch to add more angular momentum as deployment continues - a process requiring about 2 weeks. The deployment process is relatively simple and reliable compared to the square sail.

The thrust vector can be changed and directed to some extent by collective and cyclic pitch changes, as with a helicopter. Cyclic pitch changes can be made in less than one hour, but a major reorientation of the spacecraft can require more than one day. Cruise operation of the spacecraft is more complex than with the square sail.

Sail area can be up to about 10^6 m^2. The Halley's comet design had blades about 8 m wide by more than 6 km in length.

MATERIALS

Sail Sheets

There are four principle materials which appear suitable for use as a solar sail sheet. These are known by trade names Kapton, Paralene, B-100, and Mylar. These materials differ principly in the maximum temperatures at which they may be used. Kapton appears to be serviceable at temperatures up to 700° F or above, while paralene is useable up to slightly lower temperatures, B-100 is also good at high temperatures. Mylar is serviceable only up to 300 to 350° F. Considerable testing must be done to determine the capabilities of these materials to withstand the intense ultraviolet radiation to which they would be subjected in space. In addition, tests must be run to determine the rate at which rips would propagate in the material once the material was punctured. Tests must also be conducted on suitable methods of fastening seams, whether by chemical bonding or heat welding. Paralene and Mylar are commercially available in thicknesses very near the minimum requirement for solar sail sheets. Kapton is presently available in material about three times the thickness needed for solar sails.

Reflective Coatings

There are presently two known coatings which appear to best meet the needs of solar sail applications; these are silver and aluminum. Silver has a higher overall reflectivity than aluminum but it has an abrupt transparent window in the ultraviolet region. this would allow ultraviolet radiation to penetrate the silver coating with the danger of degradation occurring in the material below the silver. An additional concern with silver is its tendency to oxidize into a dark coating in the prescence of atmospheric oxygen. While aluminum has only a slightly lower reflectivity than silver, it has a full spectrum response to solar radiation and appears to be the best overall choice. Other possible materials would include gold and other metals or possibly a combination of aluminum and silver.

ENVIRONMENTAL EFFECTS

Pressure Load

A solar sail would approach close to the Sun in its trajectory. The total pressure load upon the sail would increase by the inverse square of the distance from the Sun. The increase in pressure would cause a greater deflection in the sail and in any supporting spars, which would lower the overall efficiency of the sail. This results from the fact that the local angle of incidence with respect to the Sun would increase at some points on the sail. Since the pressure force is a function of the cosine^2 of the local angle, this would cause a lower total force upon the sail; thus, the sail will have a somewhat lower efficiency as it gets nearer the Sun. However, this is more than offset by the increase in pressure which results from the decrease in solar distance.

Temperature

The front surface of the sail is highly reflective, turning away approximately 90% of the incident solar radiation. The backside of the sail will have a reasonably high emissivity value, which will result in the backside of the sail acting as a huge radiator surface. As a result, the sail will achieve equilibrium temperatures which are rather moderate considering some of the approach distances to the Sun. At a distance of 0.3 a.u., sail equilibrium temperatures may range from 250 to 400^{o} F while at 0.2 a.u. the equilibrium temperatures may range from 500 to 700^{o} F. These resulting temperatures are within acceptable ranges for at least some of the potential sail materials. However, this will remain true only as long as the sail front surface maintains a high value for its reflectivity.

Aging

The aging effects on the solar sail are a definite matter of concern, but the magnitudes of the effects are not yet known. There are at present known processes which could contribute to aging effects of the sail material. The first of these is erosion, which is caused by dust and solar wind particles. Since this is basically an impact phenomenon, the effect will probably be localized around the area of the impact. The effect will principally be physical damage resulting from a puncture or cratering of the coating or sail material. Breaks in the reflective coating could lead to localized degradation of the substrate material behind the coating. Another factor is outgassing from the plastic, causing local eruptions, with results similar to particle impacts. Another aging factor is that of ultraviolet radiation passing through the reflective coating. The prime effect of the radiation is to change the molecular structure of the sail material substrate, which can lead to embrittlement of the material. The degree to which this embrittlement occurs and the resulting problems have not yet been quantified. It appears likely that the effect of the radiation can be controlled to some degree by the proper selection of the reflective coating and the thickness to which it is applied.

Photoelectric Effect

A significant photoelectric effect is expected to occur with the solar sail. The front surface of the sail is exposed to the incident photons from the solar radiation. These photons will strike the surface of the sail. As this positive charge builds up, it will influence the components of the solar wind striking the sail. That is, protons in the solar wind will tend to be deflected and electrons attracted, with the result that charge would probably build up on to some equilibrium value. It will be possible to control the degree of this charge by the use of electron or proton emitters.

Tear Resistance

The sail materials which have been identified to date are all relatively tough materials with good stress properties. However, when these materials are subjected to high tension and then punctured in such a manner as to leave a sharp cut in the material, tears will readily propagate through the material. For this reason, it is thought that rip-stoppers will be necessary on the sail sheet. Seams in the sail sheet may serve as rip-stoppers in one direction and the addition of special rip-stoppers would thus be required in only the remaining direction. The network of rip-stoppers is not expected to add greatly to the overall weight of the sail but the effect will nonetheless probably be significant. Sail configurations which have lower stress values in the sail sheet may have a much reduced requirement for the presence of rip-stoppers.

INTERPLANETARY SHUTTLE

Concept

The Interplanetary Shuttle is a recoverable solar sail vehicle capable of returning samples from planets and small bodies. The vehicle itself may be reusable for subsequent missions. It would use either a Shuttle/Capture launch or a spiral escape from Earth and a spiral capture upon return. The sail vehicle may interface with Earth-based vehicles at an orbital space dock facility. This facility may be located at an altitude of about 1000 km or at a higher altitude above Earth's radiation belts. The vehicle would be based upon designs developed for the first solar sail mission applications, in particular, the Halley's comet rendezvous and the Mars Surface Sample Return.

The vehicle is envisioned as being one which is relatively autonomous. The economics of returning a vehicle require low mission operations costs. The vehicle would determine its own trajectory in a simplified manner, computing and maintaining the proper sail angle to reach its destination. The computer program constants would be updated and special commands sent periodically. In this manner, the vehicle would be making simplified computations allowing it to follow trajectories close to the optimum. Earth-based mission control will assume command near the vehicle's destination, removing residual errors (although a fully automated terminal sequence may be possible by the time the solar sail vehicles are operational). The vehicles would be self-monitoring and report any detected problems or anomalies.

Capabilities

The capabilities of the Interplanetary Shuttle summarized in Table 1 are based upon the use of the square sail configuration.

Performance and Cost

The different solar sail concepts under consideration are expected to have some what differing values of sail loading (total mass/sail area). These differences are most prominent when lightweight payloads are being carried; when heavier payloads are carried the percentage differences in sail loading becomes much smaller. These differences can always be expressed as differences in flight times to the destination for the given payload. This allows differences in sail loading to be expressed as cost differences for specific missions because of the difference in total mission operations cost due to differences in flight times. The differences in total mission costs are a function of the sail vehicle costs, costs resulting from differences in the mission operation cost rates, and differences resulting from the times of flight.

Operating Range

The region of space in which a sail vehicle operates can have a strong influence on its design. This will generally show up in terms of the sail loading and the thermal characteristics of the sail. It is expected that the design of the first sail vehicle will be such as to allow subsequent vehicles of the same design to operate anywhere in the solar system beyond a minimum solar distance of about 0.3 a.u.. If a specific vehicle is built to operate only in a restricted region, such as that between Earth and Mars, then that vehicle may follow the general design of a Halley's Comet Rendezvous vehicle, but some aspects of the design may be altered to take advantage of the more benign environment in which it would operate. Based upon present knowledge, it seems reasonable to impose the requirement upon the sail vehicle design that it be capable of operating anywhere in the solar system outside of 0.3 a.u..

Commonality

Once a solar sail vehicle becomes operational, subsequent mission applications may follow fairly quickly. Time and founding constraints will probably not allow the development of new solar sail designs for each mission application. Careful attention should then be given to making the first sail design be capable of carrying a wide range of payloads to destinations located as described in the preceding paragraph.

TABLE 1

INTERPLANETARY SHUTTLE SIZE AND PERFORMANCE FOR MARS

Square Sail Size (meters)	Outbound Trip Time (days)	Payload (metric tons)
700	400	1.8
	500	3.9
	700	6.0
1000	350	1.6
	400	3.7
	500	8.0
	700	12.0
2000	350	6.4
	400	15.0
	450	25.0
	500	32.0
1000*	350	3.4
	400	5.5
	500	9.8
	700	14.0
2000*	350	14.0
	400	22.0
	450	32.0
	500	39.0

Notes: Based upon total mass excluding payload
- Sail efficiency 85%
- Baseline 1982 sail (sail loading = 4.8 g/m^2)
- Advanced sail (sail loading = 3.0 g/m^2)
- Based upon total mass excluding payload

* Advanced sail

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USE OF LUNAR PRODUCED PROPELLANTS FOR MANNED MARS MISSIONS

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ABSTRACT

Manned Mars Mission departures from low lunar orbit (LLO), L2, and low Earth orbit (LEO), using oxygen or oxygen and hydrogen produced on the Lunar surface; or Phobos produced propellants; are compared to departures from LEO using Earth produced propellants. The economy of a given scheme is a function of the ratio of Earth launch to lunar launch costs per unit mass. To achieve savings on the order of 40% of total Earth launch costs for steady state operations requires the availability of both oxygen and hydrogen on the Moon and launch per unit mass costs of lunar surface to LLO in the range of 25% of Earth to LEO costs.

INTRODUCTION

A manned lunar base capable of producing propellants on the lunar surface has been the subject of a number of recent studies (References 1, 2, & 3). Lunar oxygen propellant production for lunar landers appears to be economical if a large base is operated. Similar propellant production capability can be postulated for the Martian moons, Phobos and Deimos. This paper discusses the conditions under which propellant for manned mars missions could be economically produced off-Earth. Regular departure of manned missions to Mars will require roughly 1,000 metric tons of propellant, mostly oxygen, every two years.

COMPARATIVE SCENARIOS

Propellants produced on Earth, Phobos or Deimos, or the Lunar surface can be ferried to a Mars spacecraft and loaded in a number of different orbits. Three propellant loading points for the trans-Mars injection (TMI) burn were considered: LEO (500 km circular); LLO (500 km circular); and L2 (the Lagrangian point behind the Moon). Reference 4 discusses L2 in more detail. Spacecraft departing from the Earth-Moon system can also be loaded with propellants at a Martian moon for the return trip. There are many options and combinations of options. Table 1 shows the combinations that are considered in this paper, which does not include all combinations or options. Departure from geo-synchronous

TABLE 1

CASES PLOTTED

Case #	Departure Point			Propellant from Dep. Point on Produced at							
	LEO	LLO	L2	Earth		Luna		Mars		Phobos	
				O2	H2	O2	H2	O2	H2	O2	H2
1	X			X	X						
2		X			X	X					
3		X				X	X				
4			X		X	X					
5			X			X	X			X	
6	X			X	X					X	
7	X			X	X					X	X
8		X			X	X				X	
9		X				X	X			X	X
10	X				X	X					
11	X				X	X	X*				
12	X					X	X				

*Lunar produced with hydrogen used in LLO-LEO OTVs only. Not in Mars stack.

orbit is not addressed and the possibility of returning Martian moon produced propellants to LEO is not considered.

Table 2 shows the delta V, propulsion, and spacecraft mass assumptions for the cases considered. The baseline case (#1) departs from 500 km circular LEO with Earth produced propellants on a generic conjunction class trajectory to Mars. This trajectory favors optimum performance over speed. Twenty-four hour period, 500 km periapsis, Earth (on return to Earth) and Mars parking orbits are assumed. The baseline trajectory includes 5% delta V reserves, 10% added to C3's for windows, and 100 m/sec midcourse corrections.

The baseline spacecraft, derived from the configuration described in reference 6, uses three stages for LEO departure; the first two (TMI and MOI) use O_2/H_2 propellant. The last stage makes two burns (TEI and EOI), uses drop tanks, and O_2/propane propellant. The baseline propulsion is sized to deliver a large load to Mars (3 landers and a Mars orbital transfer vehicle), and is the type of design that might be appropriate for a 10 mission, 20 year base-building scenario.

All the other options also use this baseline spacecraft with some modifications. For the LLO departure scenarios, trans-lunar injection (TLI), lunar orbit insertion (LOI), and trans-Mars injection (TMI) are all done with the first stage. The spacecraft departs LEO, is loaded with propellants again in LLO, and then goes to Mars. The TLI and LOI burns size the first stage. The oxygen tank must be large enough to supply TLI and LOI burns and then be filled for TMI. The hydrogen tank must supply all three burns, if no lunar hydrogen is available. L2 departure works the same way, with all the burns up to and including TMI done with the first stage.

BASELINE (CASE NO. 1)

Case No. 1, the baseline, masses 1,300 metric tons in LEO and is described in detail in reference 5 and the tables. It is a three stage, conjunction clase, base-building design, which is all expended except the 53 metric ton mission module which is returned to Earth. It and all the other cases carry three landers and a small Mars orbital transfer vehicle (MOTV).

TABLE 2

MASS, PROPULSION, AND ORBITAL MECHANICS ASSUMPTIONS

BURN	DELTA V (KM/SEC)	ISP (SEC)	PROPELLANT	MASS FRACTION
Baseline LEO Departure:				
TMI	3.808	468	L02/H2	.925
MOI	1.666	480	New L02/H2	.850
TEI	1.490	370	L02/Methane (Mixture = 3.5:1)	.940
EOI	0.967	370	L02/Methane	.890
Low Lunar Orbit Departure:				
TLI	3.1555	468	L02/H2	.925
LOI	0.975	468	L02/H2	.925
TMI (2 burns - TEI & burn at earth flyby)	1.628	468	L02/H2	.925
L2 Departure:				
TL1I	3.150	468	L02/H2	.925
L20I (2 burns - lunar flyby & at L2)	0.350	468	L02/H2	.925
TMI (2 burns - L2 departure & earth flyby	1.008	468	L02/H2	.925
LLO to L2 (2 burns - LLO departure & L2 arrival	0.800	480	L02/H2	.850

Payload Mass (delivered by each mission):

Item	--	mass, metric tons
Mission Module (all returned to earth)	--	53
Mars Landers (3)	--	62 each (186 total)
Mars Orbit Transfer Vehicles	--	31

LLO DEPARTURE WITH LUNAR O2 (CASE NO. 2)

Case No. 2 assumes a modified baseline stack is launched from LEO to LLO carrying all its own hydrogen and methane, but with only enough oxygen for TLI and LOI. The Mars spacecraft is then filled with lunar produced oxygen in LLO. Figure 5 shows the Mars spacecraft and a lunar orbit propellant depot. An Earth flyby is used during TMI. Two burns, one in LLO, and one at Earth flyby are required.

Figure 1 shows that case No. 2 reduced Earth launch mass around 23% compared to the baseline (case No. 1) Lunar launch requirements are not insignificant however. Figure 3 indicates total cost savings of 10% if launch costs of Earth to LEO are 25% of launch costs from the lunar surface to LLO. The payoff would be greater if more of the post-LOI mass was oxygen or some lunar produced propellant or material because TMI from LEO (3.8 km/sec) is less than TLI and LOI (3.155 + .975 km/sec). The payoff might be greater if the outbound C3 was much higher [80 to 100 $(km/sec)^2$].

LLO DEPARTURE WITH LUNAR O2 AND H2 (CASE NO. 7)

Case No. 7 is the same as case No. 2, except lunar oxygen and hydrogen are provided to the Mars spacecraft in LLO. The TMI and MOI stages are filled with lunar derived hydrogen and oxygen. The TEI/EOI stage carries its own pwopane, but uses lunar oxygen. Figure 2 shows a 46% reduction compared to baseline Earth launch mass. Figure 4 shows a 38% reduction in total launch costs is a ton can be launched from the lunar surface to LLO for 25% of the cost of launching it from the Earth's surface to LEO.

L2 DEPARTURE WITH LUNAR O2 (CASE NO. 3)

Case No. 3 is similar to case No. 2 except L2 is used as the propellant loading point instead of LLO. The Mars spacecraft carries all its own hydrogen and propane. A small OTV delivers oxygen from LLO to the Mars spacecraft at L2. Hydrogen for the lunar landers and small OTV, and propellant to get this hydrogen to LLO is also charged to the LEO mass of the Mars spacecraft. The oxygen for the small OTV is charged to the lunar surface to LLO launch mass.

Case No. 3 is slightly better in terms of LEO mass reduction and cost than case No. 2. This is because TL2I + L2OI + TMI = 4.508 km/sec (3.150 + .350 + 1.008), is less than TLI + LOI + TMI = 5.758 km/sec

Fig. 1 Launch Requirements Versus Scenario

Oxygen Production Only

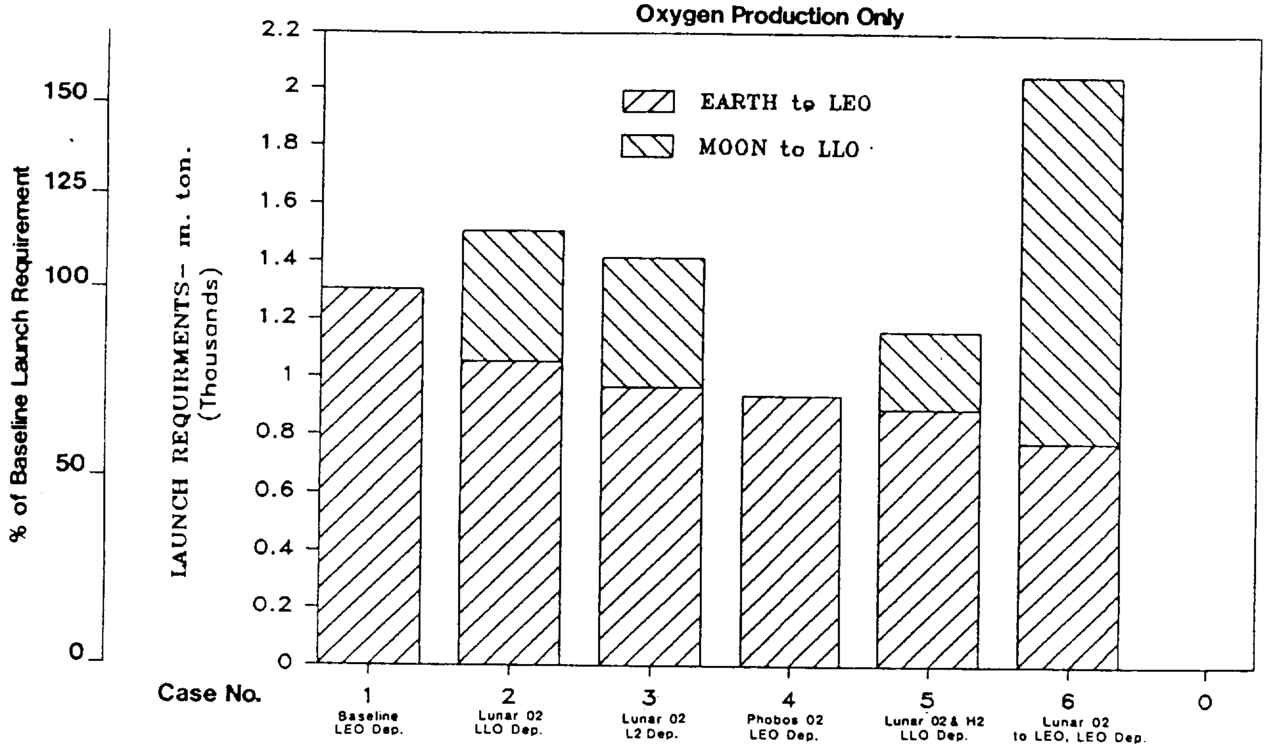

Fig. 2 Launch Requirements Versus Scenario

Oxygen and Hydrogen Production

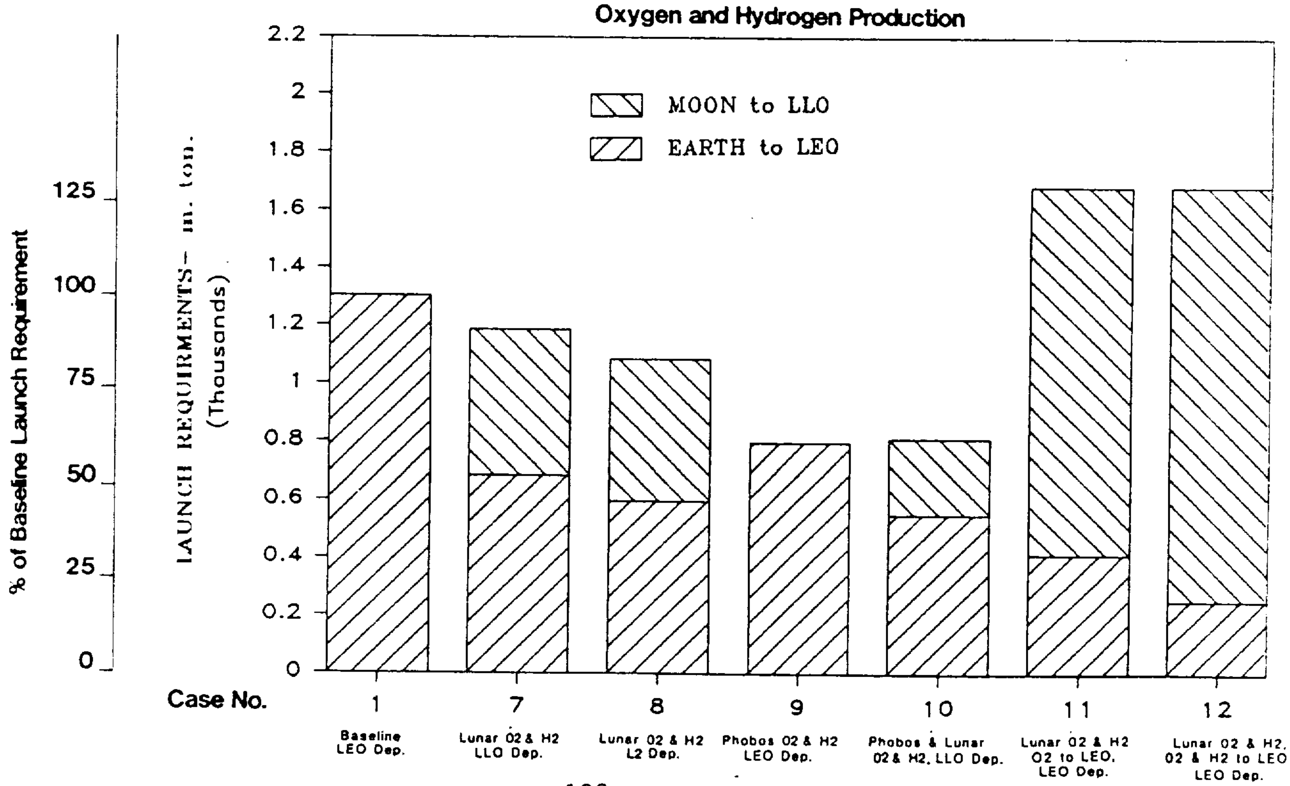

Fig. 3 Launch Cost Ratios

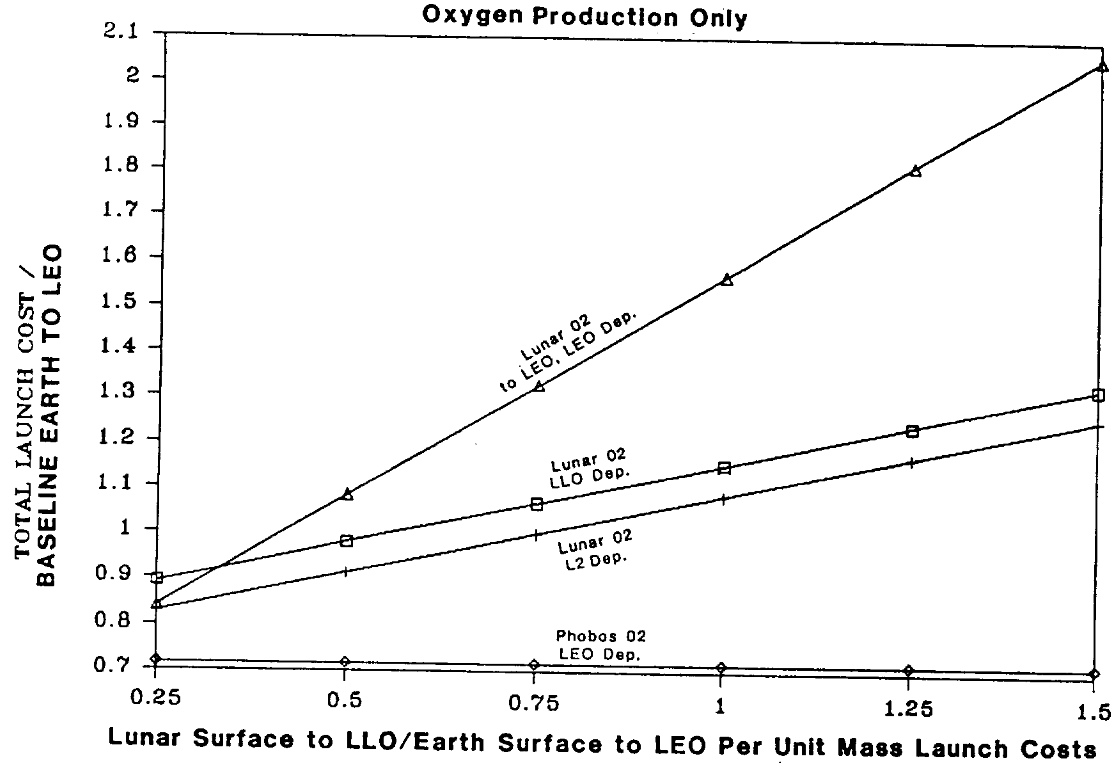

Fig. 4 Launch Cost Ratios

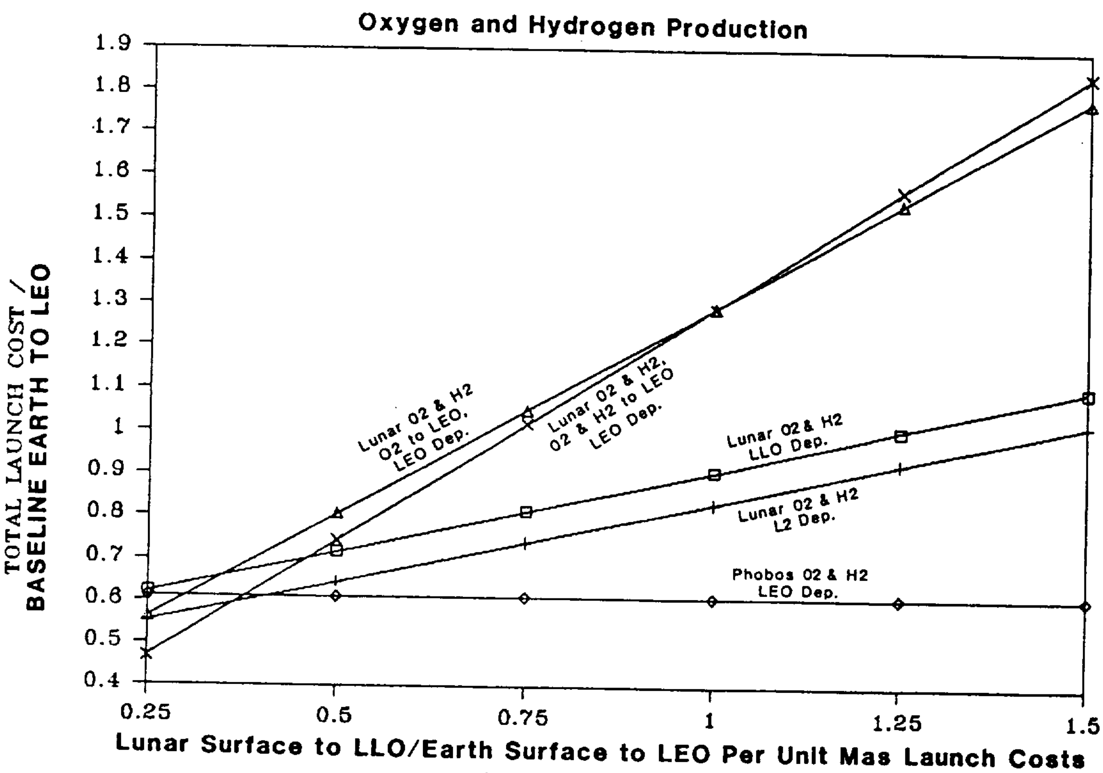

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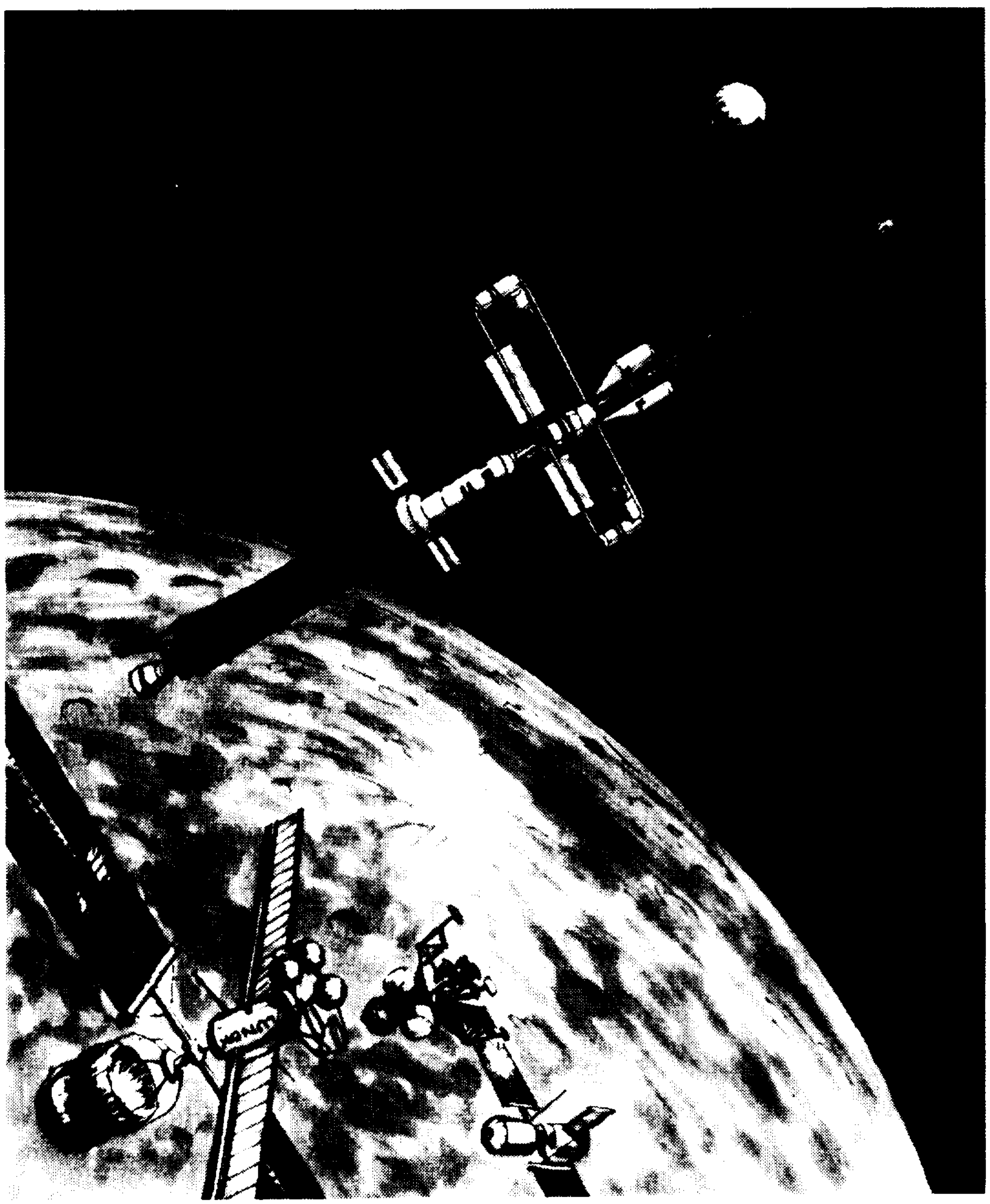
Fig. 5 Mars Spacecraft in Low Lunar Orbit



(3.155 ÷ .975 ÷ 1.628). This is due to not having to go into lunar orbit. Propellant does have to be carried up further out of lunar orbit, and the extra stage (the small OTV) needed to do this may negate the cost savings over LLO departure.

L2 DEPARTURE WITH LUNAR O2 AND H2 (CASE NO. 8)

Case No. 8 is the same as case No. 3 except lunar produced hydrogen as well as oxygen is provided. Propellant is delivered to L2 from LLO with a small OTV. The Mars spacecraft carries only its own propane for the TEI and EOI burns. As with cases 2 and 3, case 8 is slightly better than case 7 in terms of Earth launch mass and cost. However, both cases 7 and 8 (with hydrogen) are dramatically better than cases 2 and 3 (oxygen only). Hydrogen does not have to be brought from Earth for landers and OTVs for cases 7 and 8. OTV hydrogen and oxygen is charged to lunar launch mass however.

LEO DEPARTURE WITH PHOBOS O2 (CASE NO. 4)

This case is similar to the baseline (case No. 1), except Phobos produced oxygen is delivered with a small OTV to the TEI and EOI stages and Mars landers in 24 hour elliptical Mars orbit. This case is slightly better than LLO and L2 departures with lunar oxygen, but the Earth launch requirement is not as great. Figure 3 implies the cost curve is essentially independent of Earth launch costs. This is not precisely true. Transfer of propellants from Phobos orbit (6,068 km circular) to the Mars spacecraft parking orbit (500 x 32,963 km, 24 hour period) is not free (800 to 900 m/sec one way), but may be less difficult and expensive than lunar ascent/descent (roughly 2.0 km/sec each way).

High elliptical Mars parking orbits are best for scenarios without Mars propellant production. The parking orbit for the Mars spacecraft needs to be optimized for scenarios with Mars propellant production. The parking orbit for the Mars spacecraft needs to be optimized for scenarios with Mars propellant production. If oxidizer and propellant are both available, it may be optimum to park in Phobos orbit.

LEO DEPARTURE WITH PHOBOS O2 AND H2 (CASE NO. 9)

Case No. 9 is the same as case No. 4 (LEO departure with Phobos O_2) except hydrogen is also assumed to be available at Phobos. Phobos produced hydrogen and oxygen are used in the TEI and EOI stages and the landers. This results in a 38% reduction in LEO launch mass compared to the baseline. Oxygen alone at Phobos results in a 29% reduction in launch mass. Hydrogen at Phobos does not make as dramatic a difference as it does on the Moon.

LLO DEPARTURE WITH PHOBOS AND LUNAR O2

Case No. 5 is the same as case No. 2 (LLO departure with lunar O_2) except oxygen is now provided at Phobos. The TMI and MOI stages are filled with lunar produced oxygen in LLO and TEI and EOI stages and Mars landers are filled with Phobos produced oxygen in Mars orbit. The Mars spacecraft carries its own hydrogen and propane. The hydrogen required for the lunar landers and propellant to get the hydrogen to lunar orbit is charged to the LEO mass.

This produces almost no improvement over Phobos O_2 or lunar O_2 alone. Since the delta V to get from LEO to LLO is more than LEO TMI, unless considerable propellant for later burns or payload is loaded in LLO, the scenario will not pay.

LLO DEPARTURE WITH PHOBOS AND LUNAR O2 DELIVERED TO LEO (CASE NO. 6)

Case No. 6 assumes lunar produced oxygen is delivered by aerobraked OTV to LEO at a mass payback ratio of 2.45 (Ref. 1). The mass payback ratio is the oxygen returned to LEO over hydrogen sent out from LEO for a given lunar oxygen production scheme. Ref. 1 explains such a scheme in detail. The oxygen is used to fill all stages of the Mars spacecraft. Hydrogen delivered to LLO for the OTVs and landers, and the hydrogen used in the OTVs to get it there is charged to the LEO launch mass.

This effectively reduced the LEO launch originally dedicated to launching oxygen in the baseline by 2.45. The mass payback ratio is highly sensitive to aerobrake and boiloff parameters, so this scenario could easily change. As it is, a 40% reduction in LEO launch mass is predicted, but the lunar launch requirements are now more than Earth launch requirements and, not surprisingly, Figure 3 shows this scenario highly sensitive to lunar/Earth launch cost ratio.

LEO DEP. WITH LUNAR O2 DEL. TO LEO, LUNAR H2 AVAIL. (CASE NO. 11)

Case No. 11 is the baseline case with lunar produced oxygen delivered by aerobraked OTV to LEO at a mass payback ratio of infinity, that is, nothing must be sent out to get oxygen back. All the Mars spacecraft stages are filled with lunar produced oxygen. Earth launched hydrogen and propane are used in the Mars spacecraft however. Lunar produced oxygen and hydrogen are used in the LEO to LLO OTVs and in the lunar landers. The Earth launch requirement is now 70% less than the baseline but the lunar launch requirements are not as much as the entire baseline LEO mass. Figure 4 predicts a 45 % reduction in launch costs if lunar launch per unit mass costs are 25% Earth to LEO costs.

LEO DEPARTURE WITH LUNAR O2 AND H2 DELIVERED TO LEO (CASE NO. 12)

Case No. 12 is the "best" case for lunar produced propellants with all the Mars spacecraft oxygen and hydrogen delivered in LEO at a mass payback ratio of infinity. Except for the propane, all propellants for all vehicles are lunar produced. This results in an 80% reduction in Earth launch requirements, a large lunar launch requirement, and a possible over 50% reduction in costs if the lunar to Earth launch cost ratio is 25%.

RESULTS AND CONCLUSION

Figures 1 and 2 show the launch requirements from Earth and the Moon for the twelve cases examined. Figure 2 cases, which assume lunar or Phobos hydrogen as well as oxygen production, show a substantial reduction in Earth launch mass. The launch requirements from the lunar surface are not trivial however, and it is clear that the lowest cost solution will depend on the ratio of Earth launch to lunar launch costs.

Figures 3 and 4, for lunar oxygen, and oxygen and hydrogen production respectively, show total launch cost (normalized to baseline Earth to LEO launch costs) as a function of the relative launch costs per unit mass from the lunar surface to lunar orbit to be cost effective. This lunar to Earth launch ratio must be low enough to drive the total cost below the baseline, to be cost effective.

For a continuing Mars program, O_2 production at Phobos shows the most cost gain for the least investment and with virtually no infrastructure required. The only real problem is whether O_2 in significant amounts is easily available at either martian moon. (The result would be essentially the same if Deimos were the O_2 source.)

Lunar production of H_2 (or any other fuel) as well as O_2 appears to be necessary for profitable lunar support of Mars missions. Without lunar produced fuel, much of the potential weight savings in LEO is used in transporting H_2 to the lunar surface to launch the O_2.

A prediction of the actual lunar surface to LLO/Earth surface to LEO per unit mass launch cost ratio is needed. Briefly comparing the delta Vs and mass ratios provides some insight: (1) Earth surface to LEO delta V = 8 km/sec, mass ratio = 5.9; and (2) Lunar surface to LLO, one way delta V= 2 km/sec, mass ratio = 1.6. An extremely crude estimation of the cost ratio is therefore 1.6/5.9 = .26. The mass ratios assume 460 second Isp, single stage propulsion. The lunar lander requires another 2 km/sec to descend, probably with a much smaller load however, and refurbishment in the lunar vicinity must be accounted for.

Looking at Figures 3 and 4, it can generally be concluded, that to effect a 20% - 40% reduction in total costs, lunar launch costs must be 25% or less of Earth launch costs if only oxygen is available and 50% or less if oxygen and hydrogen are available. Assuming launch costs of 1 million/metric ton, from the Earth's surface to LEO 1,300 metric ton mission would cost 1.3 billion to place in LEO. For a 10 mission program, 20% cost savings amounts to approximately 2.6 billion dollars; 40% amounts to 5.6 billion dollars. These must be large enough to pay for the extra infrastructure needed to operate the propellant production system. If no infrastructure had been emplaced for other purposes, even saving the total launch cost of a 20 year Mars program (13 billion) probably would not be enough to finance a Phobos or lunar base/propellant plant/OTV/lander infrastructure. However, if a lunar base has been established for other purposes and it is possible to produce hydrogen as well as oxygen, the non-terrestrial propellant production scenarios may be cost-effective.

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S E C T I O N   I I I

MISSION AND CONFIGURATION CONCEPTS

N87-17738

CONCEPT FOR A MANNED MARS FLYBY

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ABSTRACT

A concept is presented for a three man crew to fly by the planet Mars. The groundrule for the study is to execute the mission as quickly as possible which dictates using late 1990's technologies and space infrastructure. The proposed mission described herein uses a preliminary concept for the agency's Manned Orbit Transfer Vehicle (MOTV) and proposed Space Station elements. The space vehicle will depart from the LEO Space Station and is delivered to Low Earth Orbit (LEO) by a future launch vehicle of a Shuttle Derived Launch Vehicle (SDV) class. The trajectory parameters are chosen such that the mission duration is on the order of one year, with a two and one-half hour period within ten planetary radii of Mars. If the issues of acceptable crew "g" loads and entry vehicle heat load can be resolved, then the returning vehicle can aerobrake at Earth into a Space Station compatible orbit. Otherwise, a propulsive maneuver will be required to reduce vehicle velocity prior to Earth entry interface. It is possible to execute a mission of reasonable capability at an initial LEO departure weight of 716,208 pounds for the aerobraked case of 1,350,000 pounds for the propulsive case.

INTRODUCTION

The collection of rationales for a manned Mars mission divides into categories of: (1) science and exploration, (2) the manifest destiny of man in space, (3) the benefits of technology spin-offs, and (4) geopolitical issues such as national pride and prestige. A manned flyby mission is a mission that principally responds to the last category; probably such a mission would arise in an atmosphere of competition with the Soviets in a race in the geopolitical arena where the prize is an addition to the trophy case of national pride and prestige. Although the intangible benefits can be significant, a flyby mission should be carefully balanced between the perceived "value" of national pride and prestige, the value of the scientific return, mission costs, mission timeliness, and usability of the hardware for follow-on missions. Timeliness is addressed in reference 1 and indicates that the preponderence of the evidence, as based on the activities within the Soviet Union, point to a Soviet manned Mars flyby mission in the late 1990's. If the U. S. is to respond, existing or near term, vehicles and space infrastructure must be used in order to save, or at least share, development costs and assemble and execute a mission as quickly as possible.

ASSUMPTIONS

Transfer Vehicle

The civilian space agency is in the early phases of defining the next generation of vehicles for space transportation. In pursuit of this goal, the NASA Marshall Space Flight Center is managing the Phase A studies for the Orbit Transfer Vehicle (OTV). One of the competing vehicle configurations under study is a manned OTV of lunar and geosynchronous delivery capability. A description of this vehicle is given in NASA technical memorandum number (TM) 58264 (reference 2). This vehicle is adopted as the basic transportation unit for the Manned Mars Flyby Mission described herein. Figure 1 is an artist's concept of this vehicle at the Space Station with a Mars sample return mission payload being attached. Figure 2 is a sketch of the vehicle. Additional "droptanks" will be required in order to increase the propellant capacity of the vehicle. These tanks, and possibly some advanced power systems, are the only unique developments for this mission as outlined herein.

Mission Module

The concept for the Mission Module (MM) was taken from a Lunar Base Study performed by the Johnson Space Center. The basis for the data in that report came from NASA TND-6349. The MM is a Space Station derivative and is fully equipped with life support systems, health maintenance facilities, galley, and sleeping areas. It will contain, or have attached to it, a solar flare storm shelter scaled down from the design in reference 4 and contains an assumption that 1/2 the required shielding is contained within the vehicle mass. Figures 3 and 4 are from the Lunar Base Study and define the mass and geometry of the MM.

Command Module

The Command Module (CM) proposed for the flyby mission is based on a design for a manned geosynchronous sortie vehicle. The conceptual design for the geosynchronous CM was accomplished by the Johnson Space Center

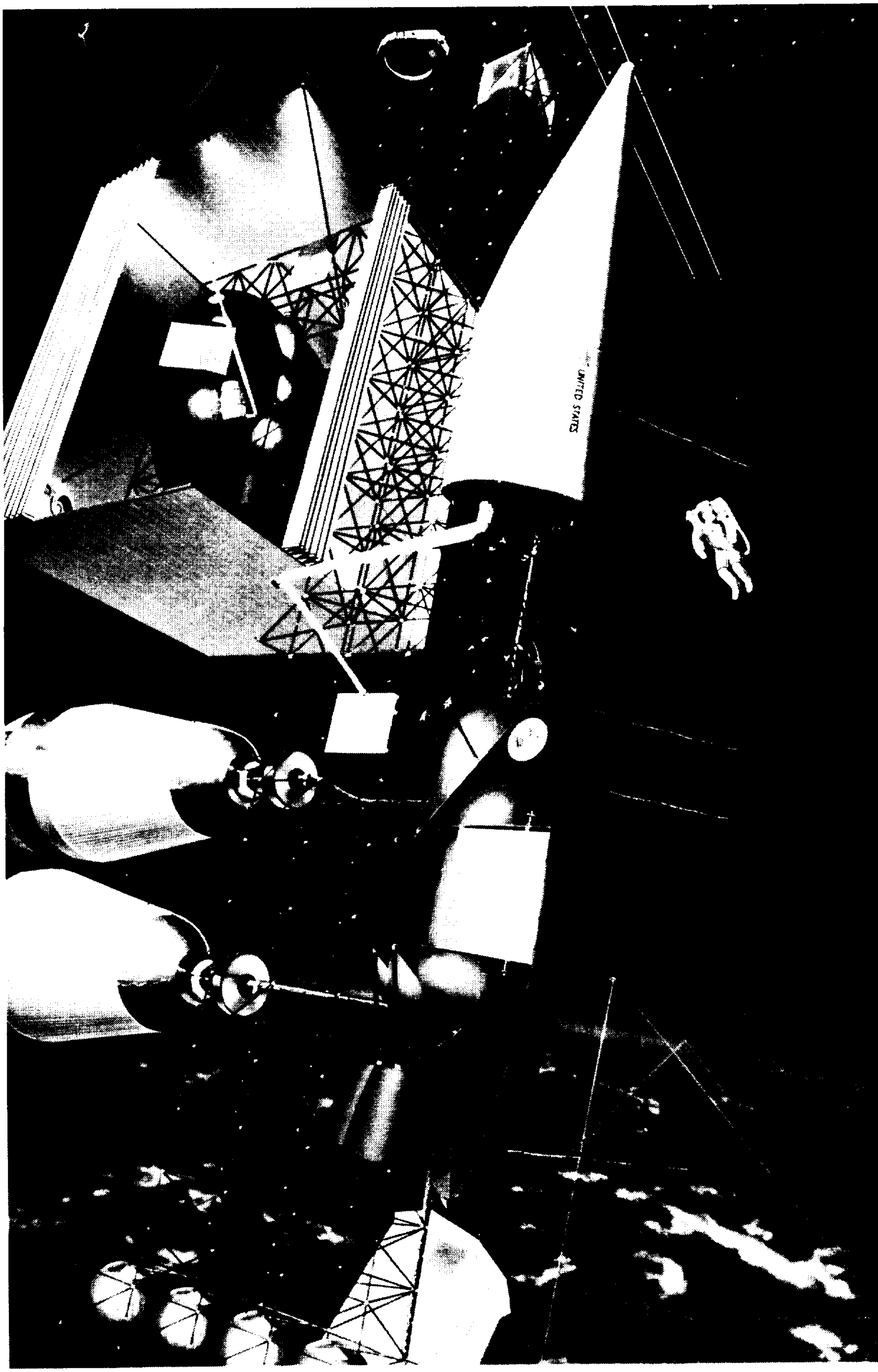

FIGURE 1 - ORBIT TRANSFER VEHICLE BEING MATED WITH A MARS SAMPLE RETURN MISSION

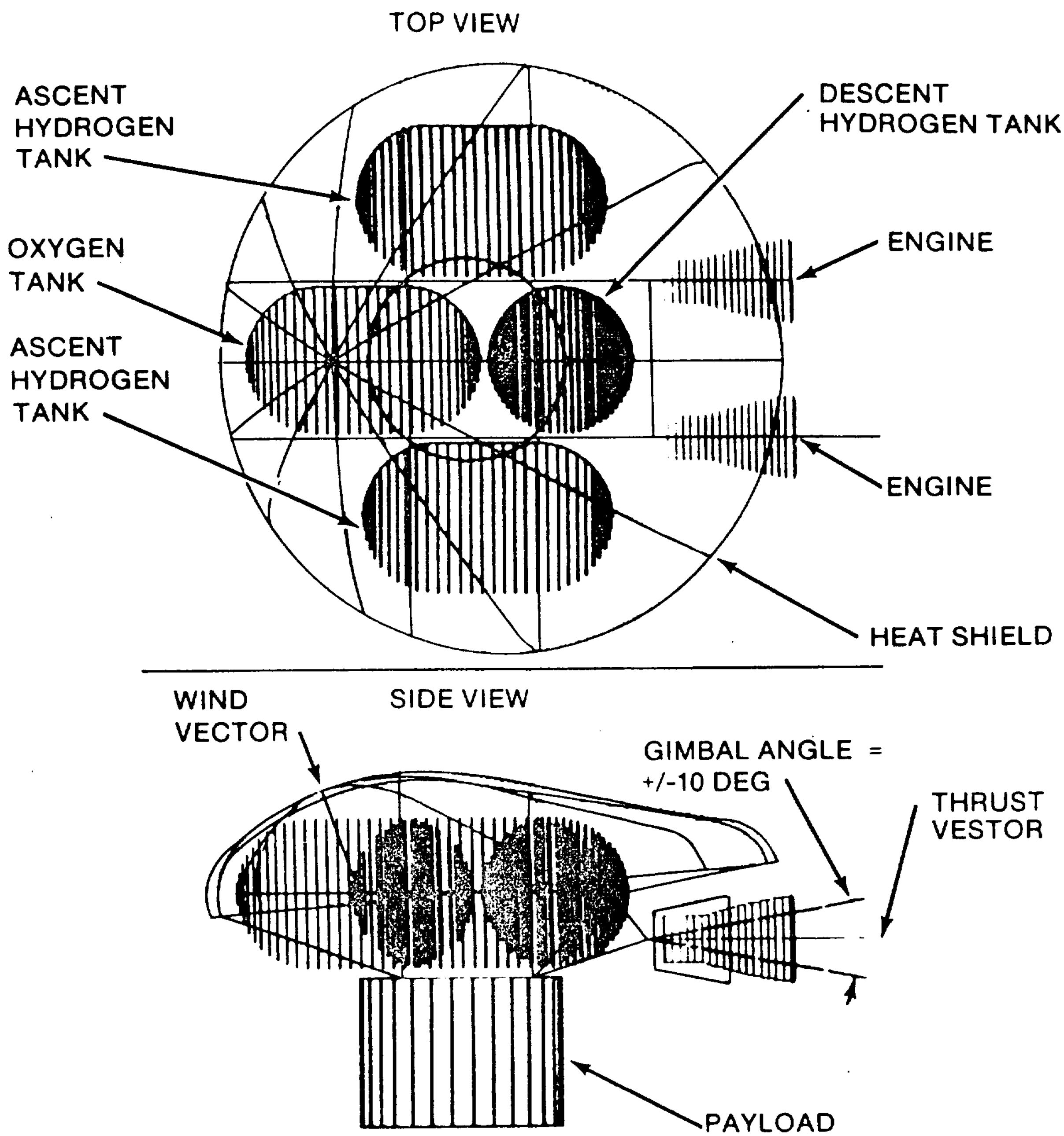

Figure 2. - Integrated AOTV concept with 12-m (40-ft) diameter heat shield.

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FIGURE 3 - MODULE SIZING	ADVANCED PROGRAMS OFFICE (ED13)	
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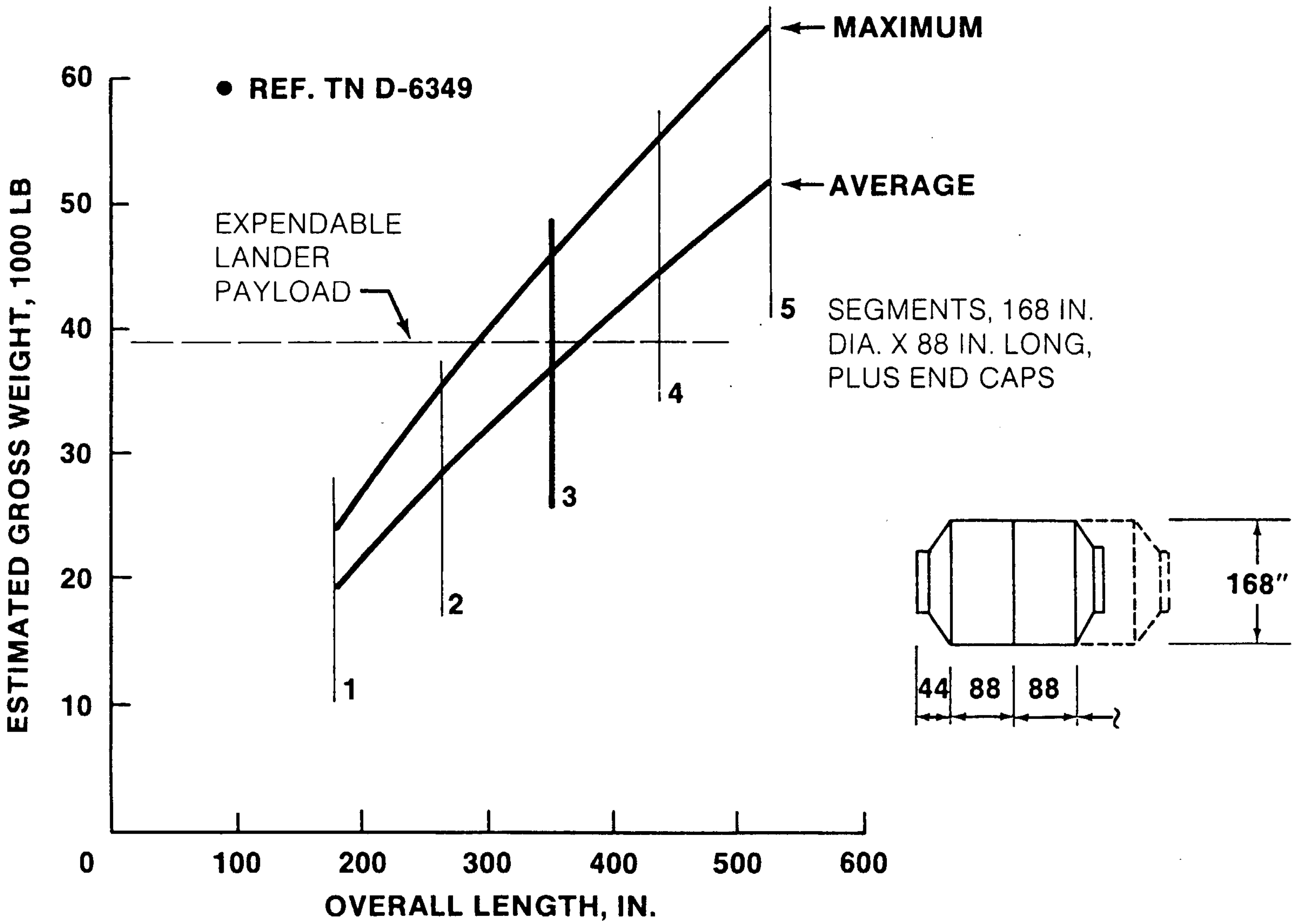

FIGURE 4 - MODULE SIZING	ADVANCED PROGRAMS OFFICE (ED13)	
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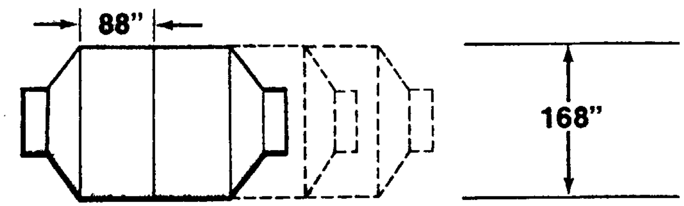
ASSUME 88" SEGMENTS PER TIGER TEAM STUDY:



NO. OF SEGMENTS	VOLUME, FT^3	GROSS WT., LB (AVG)	GROSS WT., LB (HIGH)
2	2718	27,900	34,900
3	3847	36,200	45,300
4	4976	43,900	54,900
5	6105	51,200	64,000

- IF OTV PAYLOAD IS 38,600 LB, THREE-SEGMENT MODULES ARE MAXIMUM USABLE SIZE
- IF MODULES ARE APPRECIABLY HEAVIER THAN AVERAGE CURVE (E.G., FOR STRUCTURAL BEEF-UP TO RESIST REGOLITH - BURIAL LOADS), THREE SEGMENTS (264" CYLINDER) ARE TOO LARGE
- IF SPACE STATION STRUCTURAL ELEMENTS ARE NOT USED, DIAMETER IS STILL CONSTRAINED BY ORBITER PAYLOAD BAY. NEW STRUCTURAL DESIGN WILL NOT APPRECIABLY ALTER THE ABOVE RESULTS

in 1983. It can carry three men and has all the necessary systems for command, communication, control, and life support.

Trajectory Data

Trajectory data were taken from reference 5. There are two key variables that determine the propulsive requirements for this mission. They are mission duration and mission date (planetary alignment). Reference 5 has a table of delta velocities as a function of the mission variables: duration and launch date. For the case discussed in this paper, a representative set of delta velocities was chosen for a one year mission. They are: (1) Earth depart - 28,200 ft/sec, and (2) Earth return - either zero for advanced Thermal Protection System (TPS) systems or non-reusable ablative systems or 20,000 ft/sec to reduce the vehicle's energy to parabolic with respect to the Earth.

Obviously, some additional comments are necessary to explain the choice of velocity change for Earth return. The velocity at perigee of the returning vehicle is approximately 55,000 ft/sec. At these velocities, the aeroheating to the returning vehicle will most likely exceed the limits of state-of-the-art reusable TPS (see reference 2) available for the entry heatshield. To aerocapture the returning vehicle at these velocities will require advanced TPS or ablative systems. Also, the g-levels experienced by the crew may be exhorbitant at the aerobraking re-entry velocities shown. Reference 5 has incorporated, as an option, an impulsive rocket burn that will place the return vehicle in a parabolic orbit. This maneuver should reduce the aerothermal and g-level environments to a level that current state-of-the-art TPS and crew can withstand. Thus, the choice of technology for the heatshield and crew g-level considerations will affect the main rocket impulse requirements which in turn greatly impact the initial weight in LEO.

Configuration and Mission Scenario Configuration

Figure 5 shows the configuration in LEO at departure for a one year flyby mission with drop tanks sized for the case of no propulsive burn on Earth return. Two lunar OTVs, from reference 2, are mated at a docking ring on the MM. At the forward end of the MM, the CM is mated at a docking ring. Two propellant tanks are attached to the trunnion pins on the MM and are dropped prior to Earth entry.

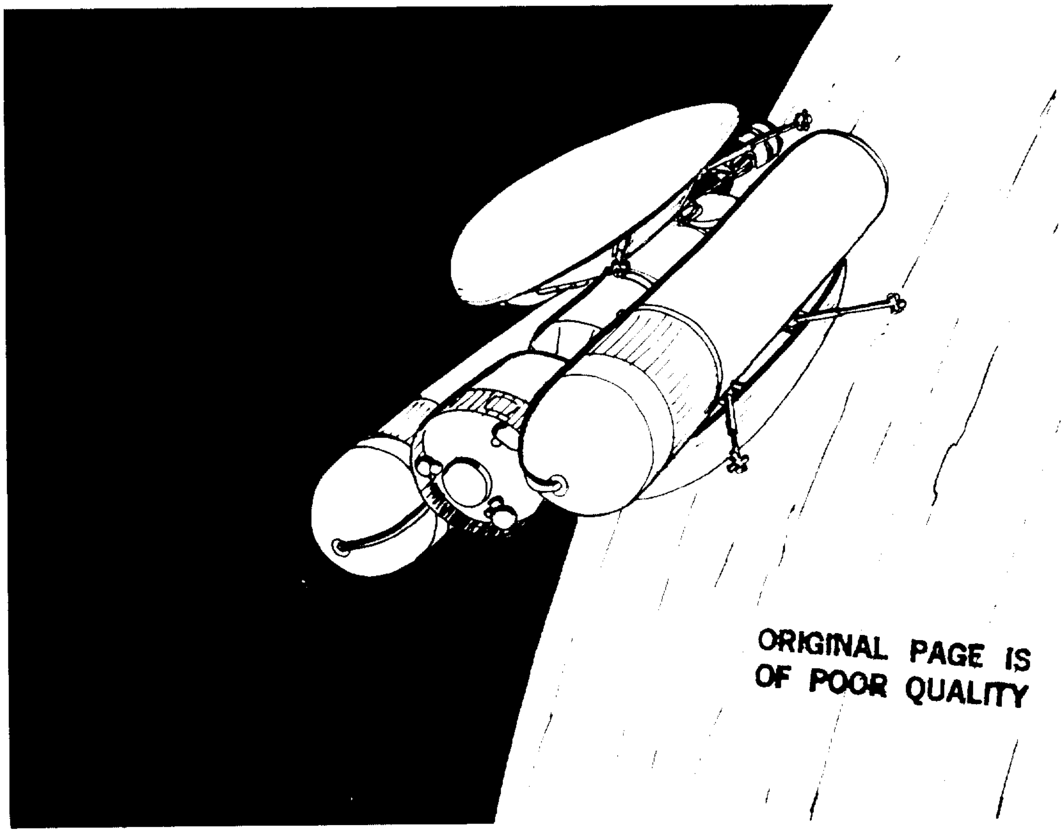


FIG. 5 - START BURN IN LEO

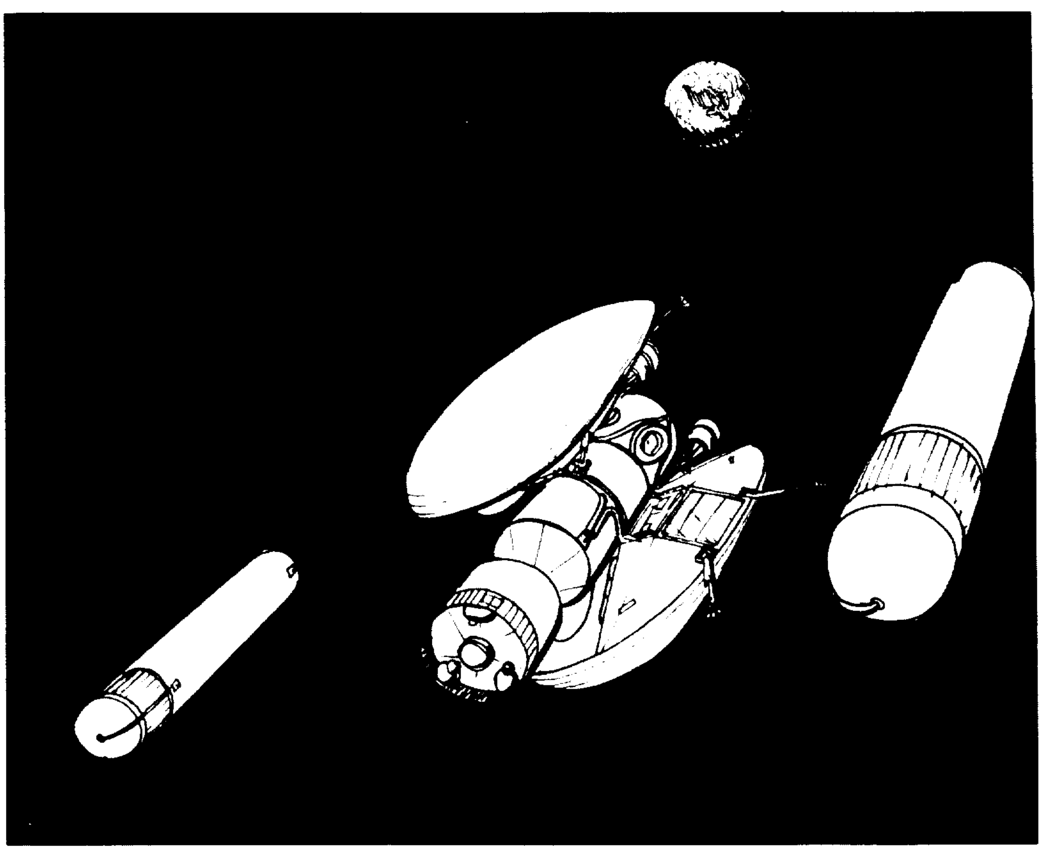


FIG. 6A - PREPARATION FOR ENTRY, DROP PROPELLANT TANKS

WEIGHT STATEMENT FOR ONE YEAR MISSION WITH NO EARTH RETURN BURN

1.	OTVs (2) @ 11,500 each		23,000
2.	Mission module		36,000
3.	Storm shelter		3,000
4.	Food & water (closed LSS)		2,300
5.	Scientific equipment		7,000
6.	Command module		12,000
7.	Drop tanks (mass fraction = .95)		23,245
8.	Main propellants	441,663	609,663
	- Drop tanks	273,663	
	- OTV tanks	168,000	
			716,208

Mission Scenario

The following brief notations describe the mission scenario:

Assembly at the Space Station

The MM and two drop tanks are delivered to LEO. The CM and manned OTVs are assumed to be operational space-based elements of the transportation system and available for this mission. All elements of the configuration mate at docking rings except the drop tanks, which will require mating to the trunnion pins on the MM and connection of umbilicals for propellants and electrical signals.

Trans-Mars Injection

Propellants are delivered by a SDV and transferred to the stacked configuration. At the start of the burn, the thrust-to-weight ratio is on the order of .1, and the total burn time is approximately one hour. To keep gravity losses to a minimum, the burn may be split into two burns if necessary. The start burn configuration is shown in Figure 5.

Trans-Mars Coast

The propellant tanks can be dropped at this time, however, since they can provide some additional shielding to the MM for meteoroids and solar storms, it might be advisable to keep them attached until just prior to Earth entry.

Mars Encounter

The encounter period (within 10 planetary radii) will be approximately 2 1/2 hours and the periapsis velocity at Mars will be approximately 26,000 ft/sec at an altitude of 160 n.m. (reference 5).

Return to Earth

As the vehicle returns to Earth, the OTVs are uncoupled from the MM, and the CM is docked with one OTV for aerocapture at the Earth (Figures 6A and 6B). The second OTV is jettisoned unless the heating problem is resolved, because no propellants have been saved for return of this vehicle. As an alternate, this vehicle does have Mars entry capability and could be used to place a payload on the Martian surface. The MM is jettisoned. If a burn is needed to reduce the velocity for TPS heating constraints or to meet g-level constraints, it will be done at this time. The OTV returns to the Space Station after passing through the atmosphere and performing some orbit adjustments. Figure 7 is an artist's concept of entry.

PERFORMANCE CONSIDERATIONS

The OTVs, storm shelter, and CM are fixed weights that cannot be manipulated; however, the MM (which would include choices on open or closed life support system), scientific equipment, and consequently, the drop tanks, are parameters that can be varied to perform some sensitivity studies. The payoff function for these sensitivity studies will be weight in LEO (W_{LEO}) at Earth departure, since this parameter has been generally accepted as an economic indicator of mission cost. Using the rocket equation, a relationship can be established for the weight in LEO for this mission, it is:

$$WLEO = \frac{\lambda_T}{T^{-1+e^{-a_1}}} \quad \frac{S \div CM}{e^{-a_2}} \div W_S \;\hat{-}\; \frac{2\ (1 - T)}{T} W_{PF} \div W_{MM} \div W_{BO} \quad (1)$$

where:

λ_T = mass fraction for the drop tanks; $.90 < \lambda_T < .96$

$a_{1,2} = \frac{\Delta V_{1,2}}{Ig_o}$, where g = 32.174 ft2/sec2

ΔV = velocity change, ft/sec

I_{sp} = specific impulse, sec.

W_{PF} = Full propellant load of the OTV described in reference 2 = 84,000 lbs.

W_{BO} = Propellant boil-off prior to the Earth entry burn, lbs.

W_S = Stage weight = 11,500 lbs.

W_{CM} = Command module weight = 12,000 lbs.

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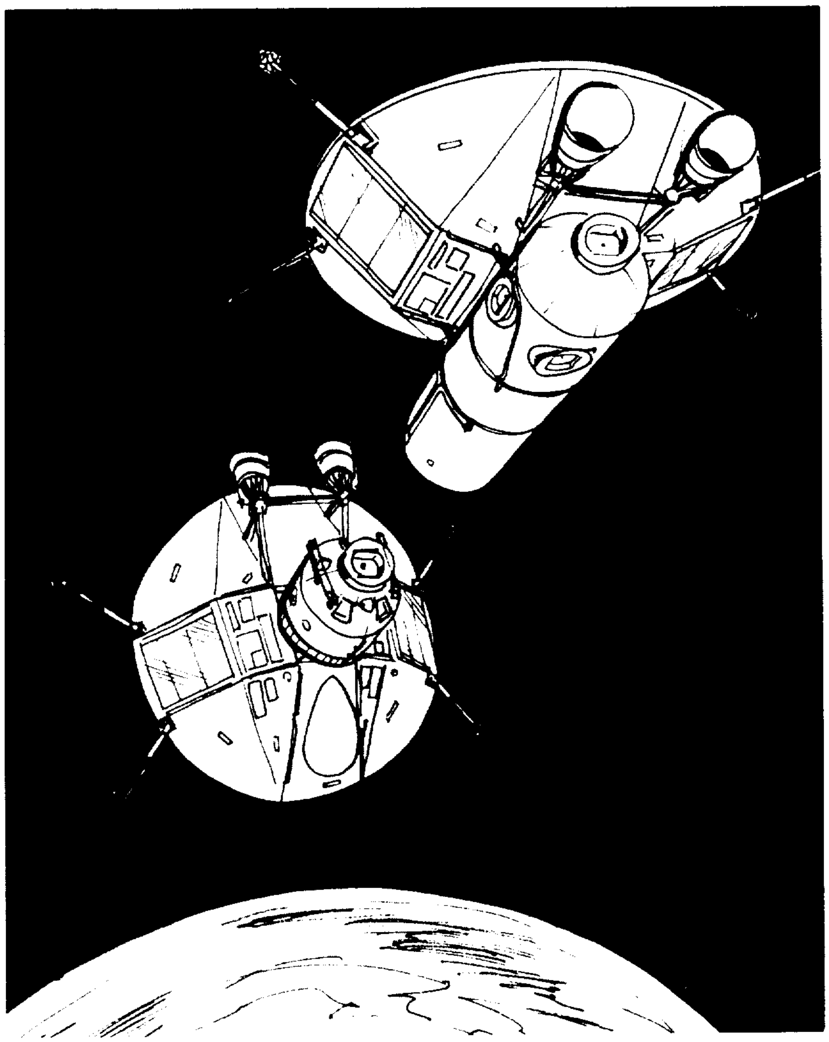

FIG. 6B - PREPARATION FOR ENTRY; ATTACH COMMAND MODULE AND DROP MISSION MODULE

FIGURE 7

ARTIST'S CONCEPT OF ENTRY OF OTV

MM = Weight, in lbs., of MM including:

- o Solar storm shelter
- o Scientific equipment
- o Consumables
- o Life support systems

If it is assumed that:

1. Boil-off can be reduced to one pound per hour
2. I_{SP} = 460 sec. (RL-10 IIB)
3. The ΔVs are as stated earlier, and
4. λ_T = .95,

then the variation of weight in LEO as a function of W_{MM} is as shown in Figure 8. The design point for the weight statement given in the section "Configuration and Mission Scenario" is indicated on the plot. The impact of making the second burn to parabolize the Earth relative trajectory is also shown. Note that when all other considerations are equal, the decision to include this second burn impacts the LEO weight by approximately half a million pounds (all in propellant and larger drop tanks), which at forecast heavy lift vehicle delivery costs of $500 per pound, equates to an additional mission cost of $250 million (approximately $750 million for shuttle delivery). Three other points are indicated on the plot in addition to the previously discussed "design point"; one of these is an indicator of what might represent the absolute minimum mission. This point is for a mission in which the MM is replaced with a small (10,000 lbs) logistics module, principally designed for food and water storage but also providing some minimum increase in living space. Health maintenance equipment and science equipment are the most notable omissions. This minimum configuration will have a LEO depart weight of 465,000 lbs. It should be noted by the reader that although the physical relationship of W_{LEO} to W_{MM} is precise, there is not much rigor in weight estimates presented in this paper for the MM. The OTV stage weight, CM, and propellant tank weights are higher quality and would change little with detail design. However, the bottom line is probably valid and it is that a manned flyby of Mars of minimum capability can be executed with late 1990's technologies and potential space infrastructure for a LEO weight of 500-750 thousand pounds.

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FIGURE 8 W_{LEO} VS W_{MM}	ADVANCED PROGRAMS OFFICE (ED13)	
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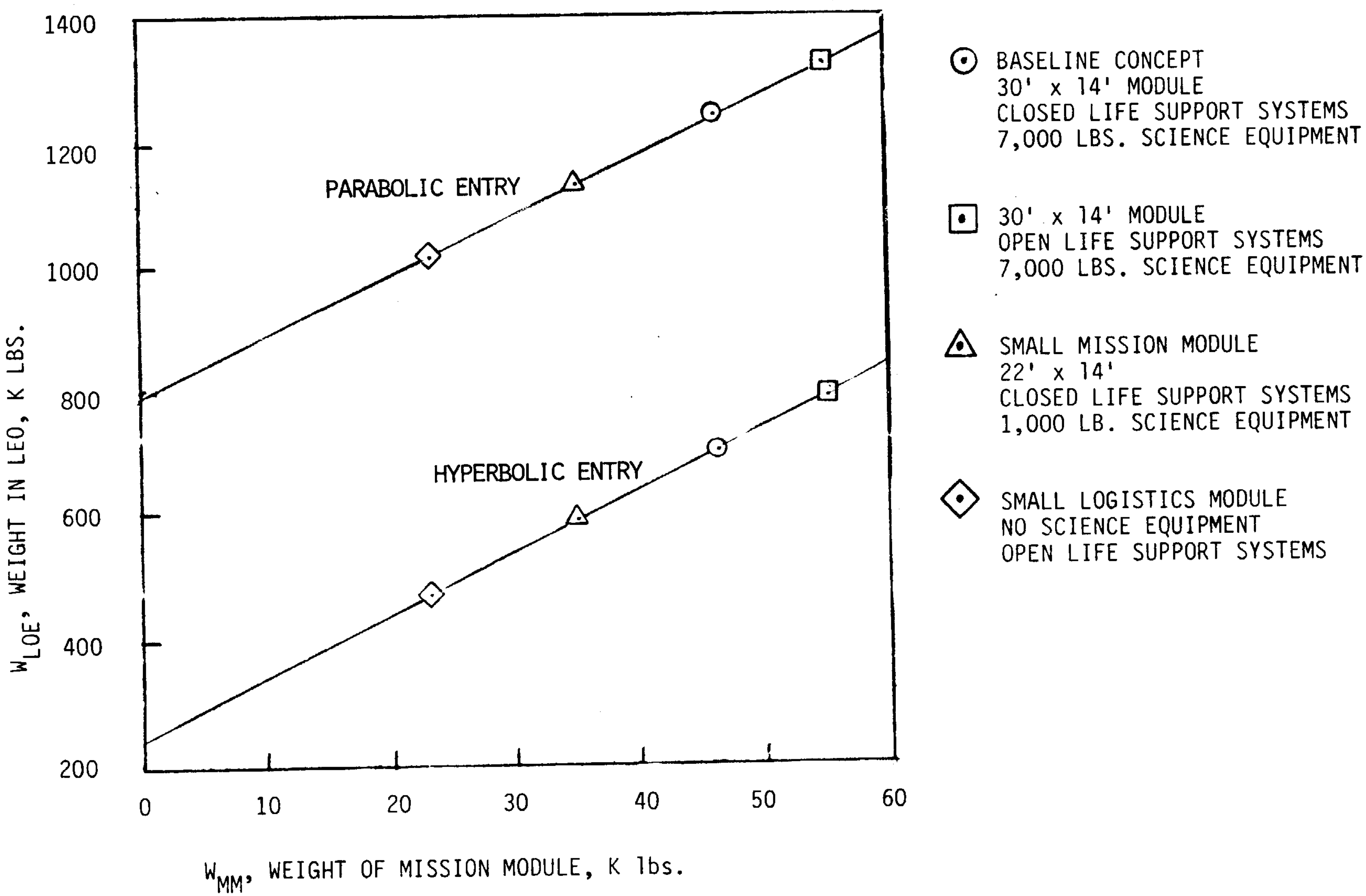

SUMMARY AND CONCLUSIONS

Summary

(1) A manned flyby of Mars would most likely be conceived in a competitive environment and mandate use of late 1990's technologies and space infrastructure.

(2) Proposed concepts for advanced space transportation system elements along with a Space Station derivative MM would satisfy the requirements for a vehicle for this mission.

(3) It is most likely impractical to utilize the reusable TPS planned for the proposed OTV for this mission due to the significant weight increase in LEO required for the Earth arrival burn. Ablative systems or advanced TPS concepts are required.

(4) A minimum mission can be performed for an initial weight in LEO of 500-750 thousand pounds.

Conclusions

It should be noted that there are no firm plans by the agency to put any of the elements discussed herein into development. The only element that is beginning to solidify is the Space Station module. The real value of this paper is to put in place the special requirements of a Manned Mars Flyby Mission using these elements such that if, and when, their development is approved, decisions will be made so as not to exclude the opportunity to use these elements to configure for this mission. On the other hand, should the decision for a Mars flyby preceed the development of the needed elements, the flyby mission components should be designed to support the Manned GEO and Lunar Base objectives.

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MANNED MARS FLYBY MISSION AND CONFIGURATION CONCEPT

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ABSTRACT

A concept is presented for a flyby mission of the planet. The mission was sized for the 2001 time period, has a crew of three, uses all propulsive maneuvers, and requires 442 days. Such a flyby mission results in significantly smaller vehicles than would a landing mission, but of course loses the value of the landing and the associated knowledge and prestige. Stay time in the planet vicinity is limited to the swingby trajectory but considerable time still exists for enroute science and research experiments. All propulsive braking was used in the concept due to unacceptable g-levels associated with aerobraking on this trajectory. LEO departure weight for the concept is approximately 594,000 pounds.

MISSION DESCRIPTION

The Mars round-trip trajectories are the Martian Counterpart of lunar free-return flight paths, with the exception that when the mission time is optimized a powered maneuver is required during Mars passage in order to achieve the desired return trajectory to Earth. A round-trip flyby may be attractive as a possible early manned mission to Mars. The basic objective for such a mission would be to reconnoiter the planet at close range, to monitor scientific probes during this atmospheric entry and landing, and to perform scientific experiments enroute to and return from Mars. The gravitation encounter with Mars plus a required powered maneuver must necessarily cause a significant alteration of the interplanetary vehicle's heliocentric trajectory. Within the activity sphere of Mars, planets sphere of influence, the trajectory is approximated by a planetocentric hyperbolic that serves as a transition segment between the outbound and inbound heliocentric trajectories. Therefore, the characteristic of the Martian encounter trajectory, i.e., passage altitudes, passage speed, orientation relative to the sunline and planet equator, powered maneuver, etc. are unique functions of the Earth departure, Mars encounter and Earth return dates.

The three requisite characteristics of the two heliocentric transfer trajectories which make up a round-trip Mars flyby mission are as follows: (1) the outbound arrival and inbound departure dates must be the same; (2) the hyperbolic excess speed at Mars, V_∞, on the inbound and outbound trajectories must be within some tolerance range with respect to each other, and (3) the angle between the V_∞'s of approach and departure must be less than a certain critical value in order to keep the required power maneuver and passage distance to the planet within an acceptable range.

The propulsive energy required to achieve the Mars flyby mission is highly dependent on time of mission opportunity because of Mars' elliptical orbit about the Sun; where Mars' distance from the Sun varies from 1.38 to 1.66 a.u. The year 2001 opportunity requires less propulsive energy than any other opportunity within a plus or minus 15 year span about the year 2001 because Mars is at its closest position from the Sun during the mission's Mars passage date. The optimum launch date for the 2001 Mars flyby opportunity is March 9, 2001, with a flight time to Mars of 172 days and a total mission time of 442 days. The Mars flyby date is August 20, 2001, and the Earth return date is May 25, 2002. The Earth departure trajectory has a C_3 value of 10.1 km^2/sec^2. A propulsive maneuver, requiring a ΔV of 1.281 km/sec, is made during Mars' flyby to achieve the necessary turn angle at Mars to connect to the Earth return trajectory. The Earth return trajectory C_3 at Earth is 117 km^2/sec^2. The Earth return braking maneuver must be achieved propulsively in order to stay with in g-level constraints required for a manned mission; braking the Earth return spacecraft aerodynamically would result in g-level greater than 4 g's.[1] An Earth return module, which is separated from the interplanetary vehicle just before Earth braking maneuver, of 7,500 lbs is decelerated propulsively into a 24 hour capture ellipse at Earth. Figure 1 gives the mission profile for the 2001 opportunity.

A weight of 594,000 lbs is required to be assembled in low Earth orbit to achieve the 2001 flyby opportunity. The 594,000 lbs assembled weight can be accomplished with 4 Shuttle-Derived Vehicle (SDV) flights and 3 Shuttle flights.[2]

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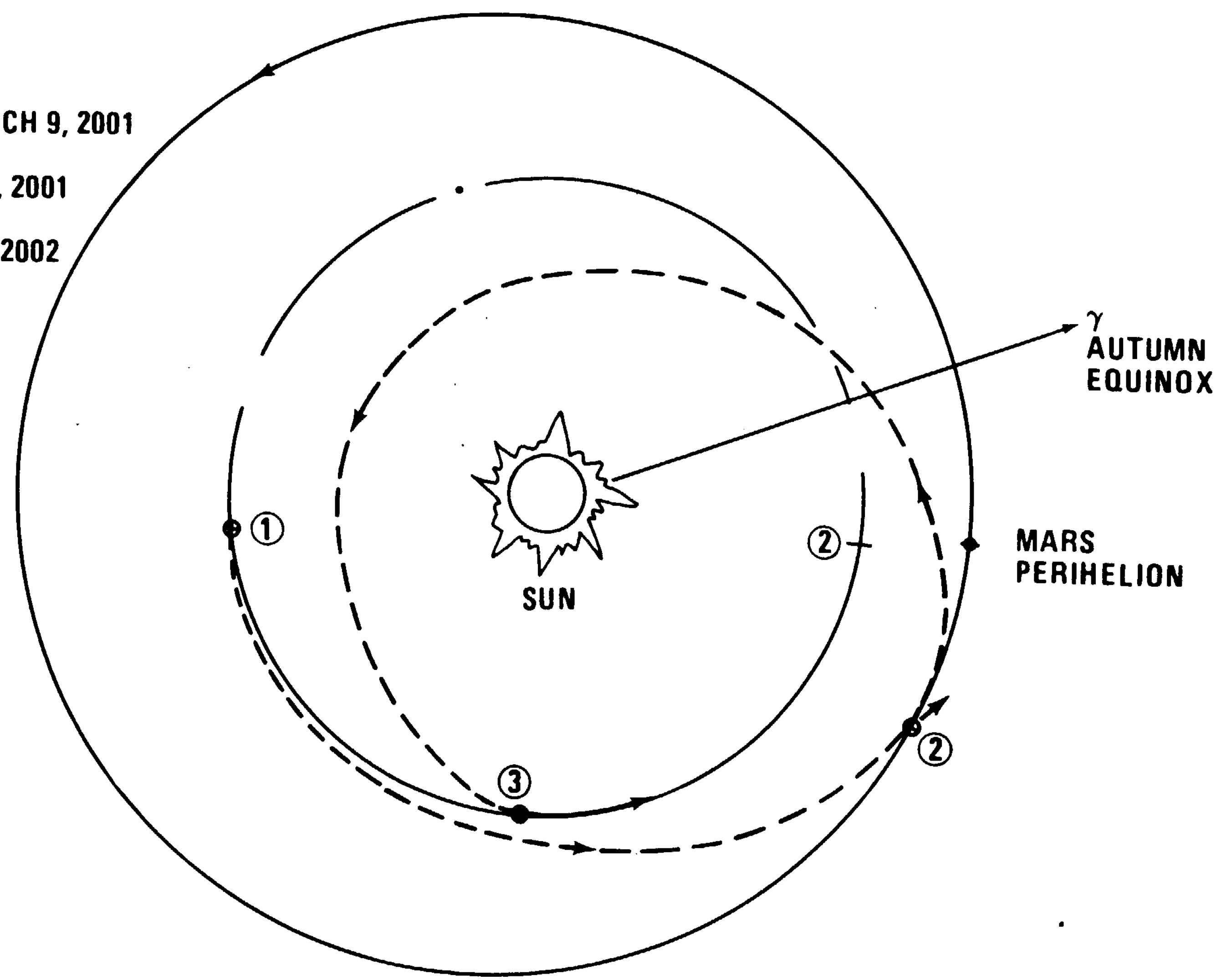
FIGURE 1 MANNED MARS FLYBY 2001 OPPORTUNITY

1. EARTH DEPARTURE, MARCH 9, 2001
2. MARS FLYBY, AUGUST 28, 2001
3. EARTH RETURN, MAY 25, 2002



A C_3 level less than 117 km^2/sec^2 for Earth return can be obtained by performing optimum midcourse maneuvers on the outbound and inbound legs.[3] A C_3 level of less than 25 km^2/sec^2 can be realized; however, the outbound and inbound midcourse correction maneuver would have to be performed with the heavier interplanetary vehicle, thereby requiring a larger initial mass in low Earth orbit than the 594,000 lbs asssociated with Earth return C_3 of 117 km^2/sec^2. AC_3 value of less than 25 km^2/sec^2 would allow aerobraking for capture into the Earth return orbit.

CONFIGURATION

Figure 2 shows a concept for the all propulsive maneuver mission. The configuration consists of two LO_2/LH_2 propulsion stages, a spacecraft, and experiments (probes, etc.). The configuration is assembled and prepared for the mission in LEO and is sized for the swingby mission of the planet and return to LEO (24 hour elliptical orbit) using propulsive energy for departure from LEO, maneuver at the Mars vicinity, and braking for Earth orbit capture.

The propulsive stages are sized for a 6:1 propellant mixture ratio, with both stages using OTV engines as shown in the figure. The first stage is separated after the burn for LEO departure. The remaining energy requirements for a maneuver at Mars and subsequent braking at Earth required most of the energy at Earth. Therefore, one stage was chosen to perform both of these burns rather than two stages or a drop tank option. This stage may have potential commonality with Orbital Transfer Vehicles developed for other programs. The sizing of the second stage also was based on returning only a portion of the spacecraft to LEO in order to reduce total propellant requirements. The recoverable portion is returned to a 24 hour elliptical orbit and would require support from an auxiliary stage (such as the planned Orbital Maneuvering Vehicle) for recovery.

The spacecraft is sized using Space Station diameter modules (approximately 14 feet). A criteria used in the design was provision of two separately pressurizable modules for safety consideration in the event one module were to become uninhabitable during the mission. Since one of these modules was to then be jettisoned on Earth return, they were sized unequally in order to return the minimum mass. This then led to

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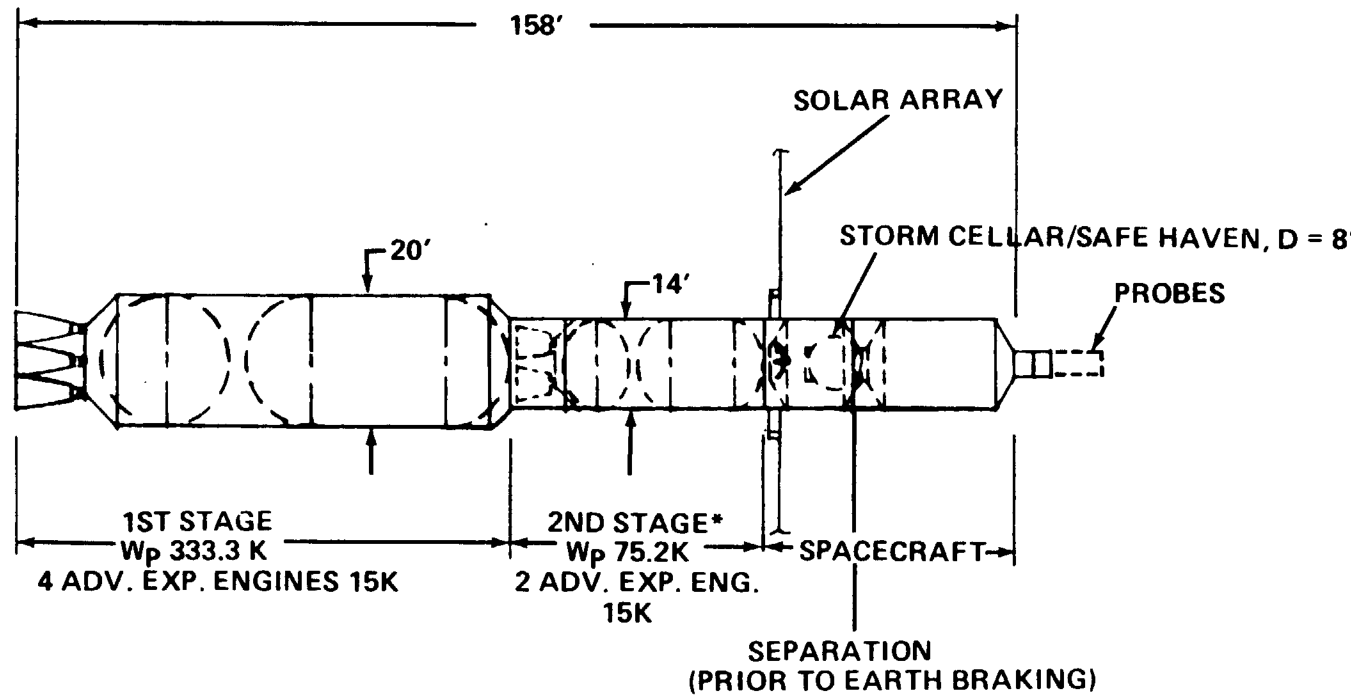
FIGURE 2 MARS MANNED FLYBY CONCEPT
ALL PROPULSIVE, 2001 OPPORTUNITY, 442 DAY MISSION



* SECOND STAGE PERFORMS PROPULSIVE MANEUVER AT MARS & BRAKING AT EARTH RETURN.

inclusion of a separate pressurized compartment within the Earth return module to serve in the event that Earth return module was the one that had become uninhabitable. This provides redundancy within the Earth return portion and is also used as a storm cellar during the mission (packaging of equipment, etc. around the compartment provides shielding). Internal layouts of these modules were not evaluated and size was estimated. A solar array system is shown for the power system during the mission. The vehicle is oriented to minimize solar array pointing requirements and to minimize heating of the propellant tanks.

External experiments were not evaluated but a weight allowance was included for them. These would include probes attached to the modules.

WEIGHT SUMMARY

Weight summary for the all-propulsive cryogenic manned Mars flyby vehicle for 2001 opportunity is presented in Table 1. The interstages and payload adapter weights are included with the structures. The number of engines (OTV type) in the propulsion system is shown in parentheses for each stage. The thermal control system includes the heavy vapor cooled shield which allows less than 500 pounds boiloff in the 2nd stage, and none in the 1st stage after departure from LEO. The avionics system for the propulsive stages are minimal since the main avionics system is in the spacecraft. A 15% contingency is added to all the dry weights, since most of the hardware is new and considered to be current technology equipment. The usable propellants (consumables) for the propulsive stages were determined by performance analysis as shown in Table 1. The stage launch weight at LEO as the vehicle departs is shown for each propulsive stage.

The weights for the Earth Entry Module and spacecraft are shown together in the third column. The weights for the avionics, ECLSS, crew systems, consumables, and mission equipment were estimated using data from[4]. The configuration is shown in Figure 2 with the propulsive stages attached. The pressurized modules including the safe haven are included in the structures. One airlock is also included with the structures in addition to the micrometeroid shield and outer insulation weights. The main avionics and power for the vehicle are shown in the spacecraft. The consumables for the spacecraft include food, water, oxygen, nitrogen, clothes, power system reactants, and other crew systems

TABLE 1

WEIGHT SUMMARY (POUNDS)

ALL-PROPULSIVE CRYOGENIC VEHICLE FOR 2001 OPPORTUNITY MANNED MARS FLYBY

	Earth Departure 1st Stage	Mars Maneuvers & Earth Braking 2nd Stage	Earth Entry Module & Spacecraft
Structures	12,592	4,017	21,275
Thermal & Insulation	5,543	1,992	2,354
Propulsion System	(4 Eng) 4,358	(2 Eng) 2,253	-
Avionics	500	300	8,373
ECLSS	-	-	10,986
Crew Systems	-	-	8,419
Contingency (15%)	3,449	1,258	7,711
Residuals	2,560	1,011	295
Consumables	332,340	76,060 (w/boiloff)	17,749
Mission/Science Equipment	-	-	6,645
Science/Mars Probes	-	-	20,000
Crew (3)	-	-	1,140
Stage Launch Weight (LEO)	361,342	86,891	104,947
Total Vehicle Weight (LEO)		553,180	

expendables (closed-loop ECLSS). The mission/science equipment and science/Mars probes are only representative and would change as requirements are established. The crew weights include three men with flight suits. The total vehicle weights are for a 442-day mission at launch from LEO.

SUMMARY

A manned Mars flyby mission can be achieved early with inplace resources and facilities and would utilize high heritage from other space programs; i.e., Shuttle, Space Station, Shuttle Derived Vehicle (SDV), and the Orbit Transfer Vehicle. The launch opportunity of March, 2001, will be the least demanding launch opportunity through launch opportunities up to year 2016. The objectives of an early Mars flyby mission would be to conduct scientific experiments enroute to and return from Mars, observe scientific probes sent through the Martian atmosphere and probes which accomplish surface landings and mapping of the planet at close range (180 n.mi.)

The 594,000 lbs weight required in low Earth orbit (LEO) to achieve this mission can be assembled with 4 SDV flights and 3 Shuttle flights. The 594,000 lbs required in LEO for the 2001 Mars flyby mission compares to 1,602,000 lbs required in LEO for the 2001 Mars landing mission with a 60-day stay time at Mars[5]. A Venus inbound swingby is used to reduce the propulsion requirement for the 2001 Mars landing mission. Other alternatives to the Mars direct flyby mission would be to (1) flyby Venus on the inbound leg, however, this mission profile would require an increase in mission time of about 200 days over the 442 days direct mission profile;[6]; or (2) make midcourse maneuvers both on the outbound trajectory and inbound trajectory in order to be aerodynamically captured at Earth return. However, this option would increase the initial weight required in LEO above the 594,000 lbs.

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EARTH-TO-ORBIT LAUNCH VEHICLES FOR MANNED MARS MISSION APPLICATION

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ABSTRACT

Manned Mars missions (MMMs) will require payloads to low Earth orbit (LEO) much heavier and larger than can be accommodated with the Shuttle. Three typical launch vehicles are described that could possibly satisfy the MMM needs. The vehicle concepts include Shuttle Derived Vehicles (SDVs), which are composed essentially of Shuttle components, and Heavy Lift Launch Vehicles (HLLVs), which utilize new and improved technologies and require additional development.

EARTH TO ORBIT LAUNCH VEHICLES

MMMs will create requirements for cargo sizes and weights that are greater than the current Space Transportation System (STS) can accommodate (see references 11 and 12). It may be possible to divide MMM payloads into smaller and lighter units, but with the division comes the requirements for additional launches and on-orbit-assembly. This will increase the cost and complicate the operations of the missions.

Several types of advanced, partially and fully reusable ETO launch vehicles are under study by NASA and the Department of Defense. Both manned and unmanned vehicle concepts are being studied, including multistage and single stage configurations. Payload delivery capabilities for these advanced concepts range from about 10,000 lb. to about 400,000 lb, and propulsion includes rocket and air breathing varieties. Vehicles at the lower end of the payload range would be primarily "people carriers" and those at the higher end would be primarily cargo vehicles. Figure 1 shows sketches of some of the concepts presently being studied.

Three classes of the heavy-lift systems are discussed in this paper, and a specific vehicle within two of those classes was selected as a reference vehicle in the study. The STS was also used as a reference vehicle, but is not discussed here.

Combinations of Shuttle components can be used to configure SDVs with greater launch capabilities than the Shuttle. New configurations using more advanced state-of-the-art technologies have been investi-

FIGURE 1. TYPICAL EARTH–TO–ORBIT VEHICLES

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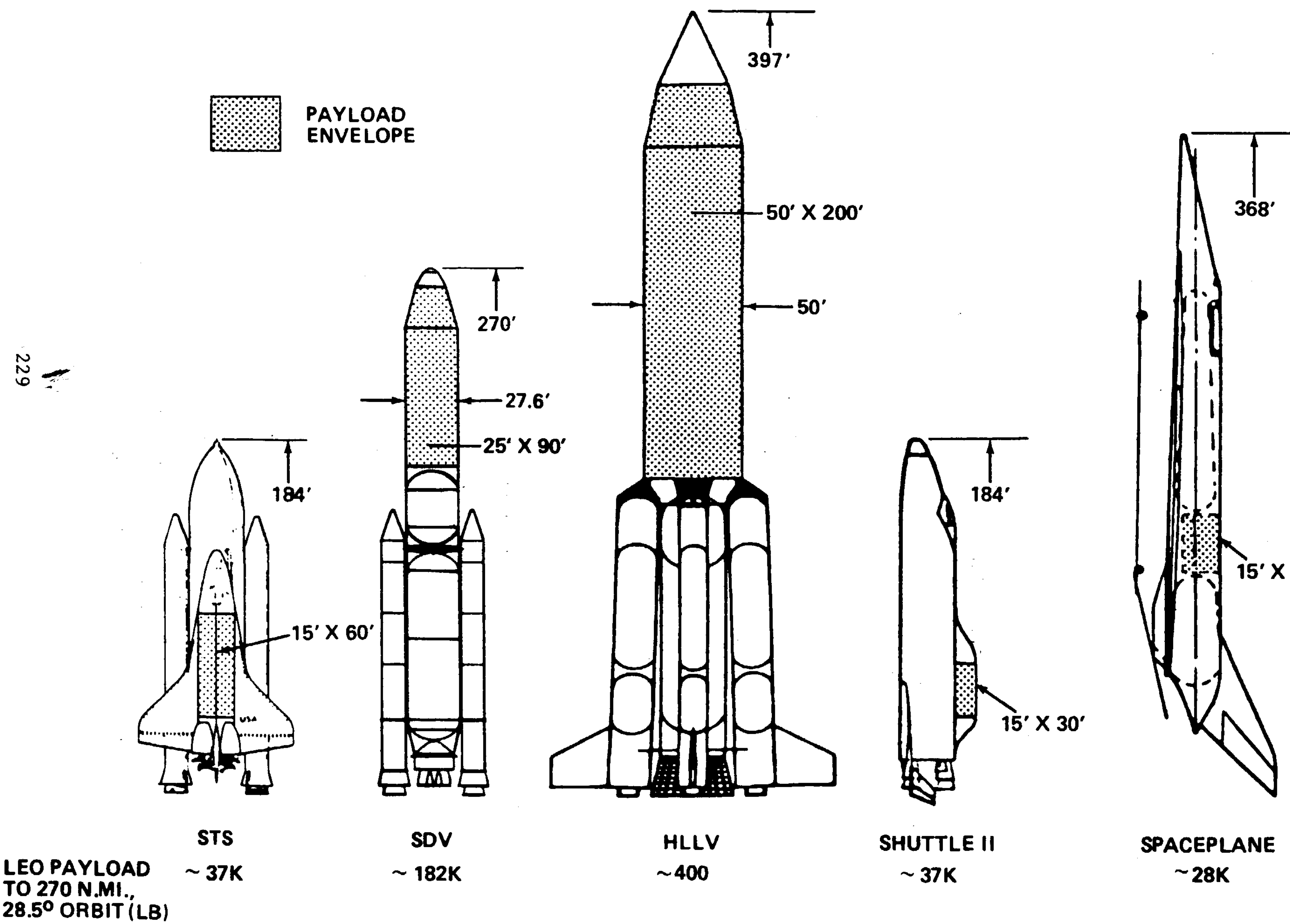

gated which could provide greater lift capacity with improved operations and costs. Evolution from the "smaller" SDVs to "larger" HLLVs have been investigated as a logical path to satisfying the 1990's and 2000's payload requirements.

SHUTTLE DERIVED VEHICLE

One potential vehicle for MMMs is the SDV-3R. The "3R" denotes three Space Shuttle Main Engines (SSMEs) in a recoverable propulsion/ avionics (P/A) module as shown in Figure 2. The SDV-3R was used as a reference vehicle in the study.

Vehicle Description

The SDV-3R consists of components and systems entirely from the present Shuttle program with the exception of the payload shroud and recoverable P/A module.

The first stage, or booster stage, uses the standard STS Solid Rocket Booster (SRB). The standard SRB uses a steel Solid Rocket Motor (SRM) case; however, a lighter weight Filament Wound Case (FWC) is being developed for the Shuttle to increase the vehicle payload capability, and can be used interchangeably with the standard steel case. The second stage, or core stage, uses the Shuttle's External Tank (ET). The ET will require slight modification to accommodate the P/A module installation at the base of the tank and the payload mounted on top of the tank. The ET is near-standard but has a flatter top to permit inline stacking of the payloads and upper stages (if required). Three standard SSMEs and the vehicle avionics are incorporated into a recoverable module located under the ET which will permit the recovery/ reuse of the SSMEs and avionics.

The SSMEs are the same as used in the Shuttle and are arranged in the same order as the Shuttle engines and use the same plumbing configuration. The engines plus the avionics are included in a recoverable P/A module that uses ballistic reentry from orbit, with ballute and parachute landing on land or water. The SSMEs, avionics, and auxiliary equipment are refurbished and reused in future flights. A Centaur G Prime third stage, which is located within the payload shroud, can be used for high energy missions such as Geosynchronous Earth Orbit (GEO) or MMM Missions. A larger stage designed specifically for the SDV-3R could

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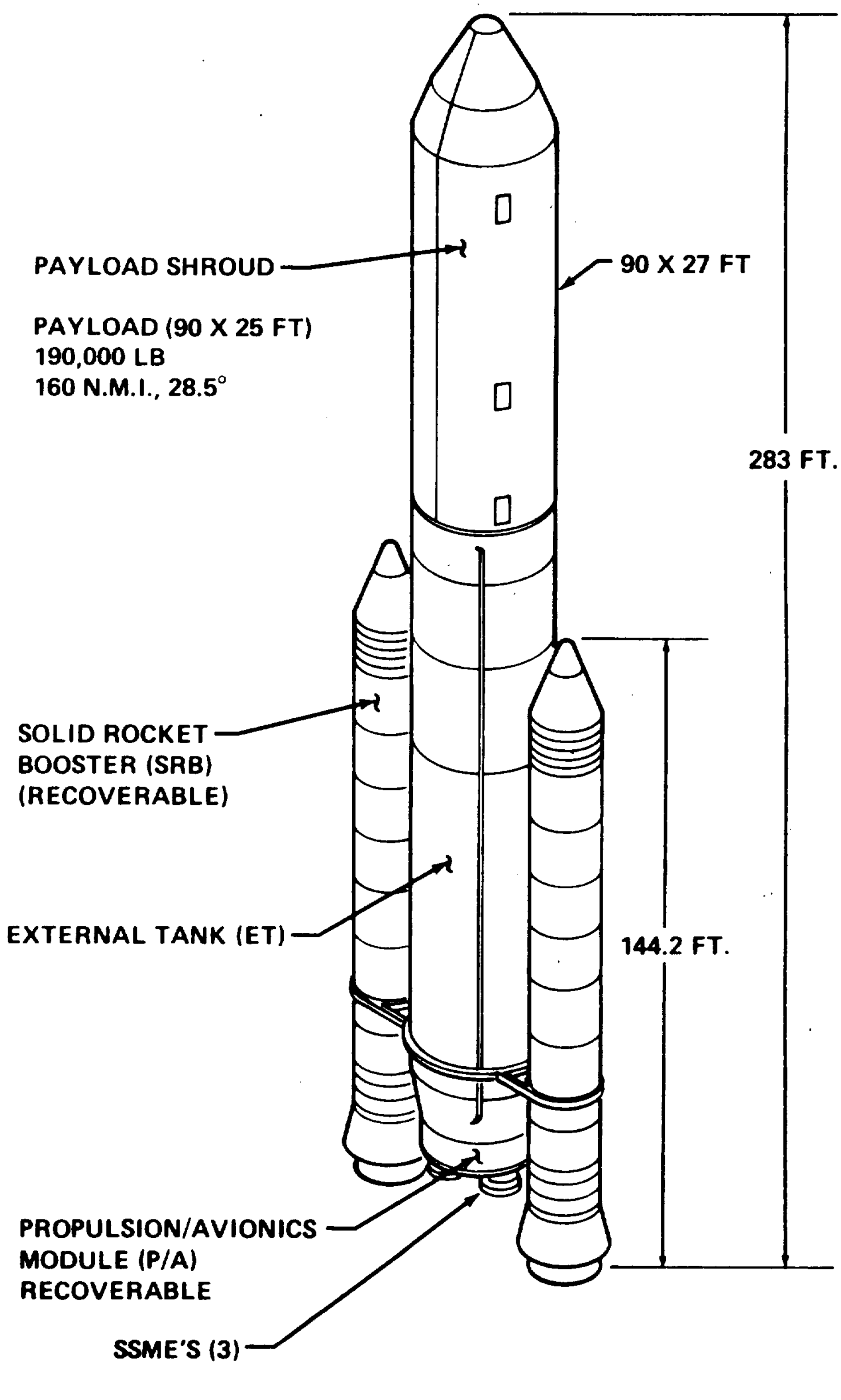
FIGURE 2
SHUTTLE DERIVED VEHICLE
THREE SSME'S, RECOVERABLE
(SDV–3R)



also be used with more than twice the performance of the Centaur G Prime.

Performance

The SDV-3R offers a wide range of performance. The two-stage vehicle has the capability of placing 190,000 pounds into a 160 n.mi., 28.5 degree inclination orbit, 182,000 pounds into a 270 n.mi, 28.5 degree orbit (that presently planned for the Space Station), and 159,000 pounds into a polar orbit. The SDV-3R can place 19,000 pounds into GEO by using the Centaur G Prime as a third stage. This payload weight is the maximum that a Centaur G Prime can take from LEO to GEO. A larger upper stage for the SDV-3R could permit payloads to GEO to increase to 50,000 pounds.

Launch Facilities/Operations

The SDV-3R will use the STS assembly and launch facilities with slight modifications. A new Stacking Integration Building (SIB) and Mobile Launch Tower (MLT) may be located at Kennedy Space Center (KSC) to provide redundancy in facility capabilities, where redundancy does not already exist in basic Shuttle facilities. This combination of existing and new facility elements can greatly enhance launch assurance and can be made available as an option. Additional facilities requirements will depend upon the launch rate required to meet the needs of MMMs.

Schedule

First flight of the SDV-3R vehicle can occur after a five (5) year development program.

SHUTTLE DERIVED/HEAVY LIFT VEHICLE (SD/HLV)

Requirements for payload weights to LEO greater than the capability of the SDV-3R will require a larger SD/HLV. This larger vehicle could evolve from the SDV-3R through normal growth or could be developed as the basic launch vehicle of the MMM. If the vehicle is developed directly for MMMs, the components/systems inherited from the SDV-3R will require development under the HLLV program, thereby adding to the development time.

Growth from the SDV-3R type of configuration to larger-lift capability could be achieved by any one or more of several means. The one shown in Figure 3 uses reusable liquid rocket boosters in lieu of the

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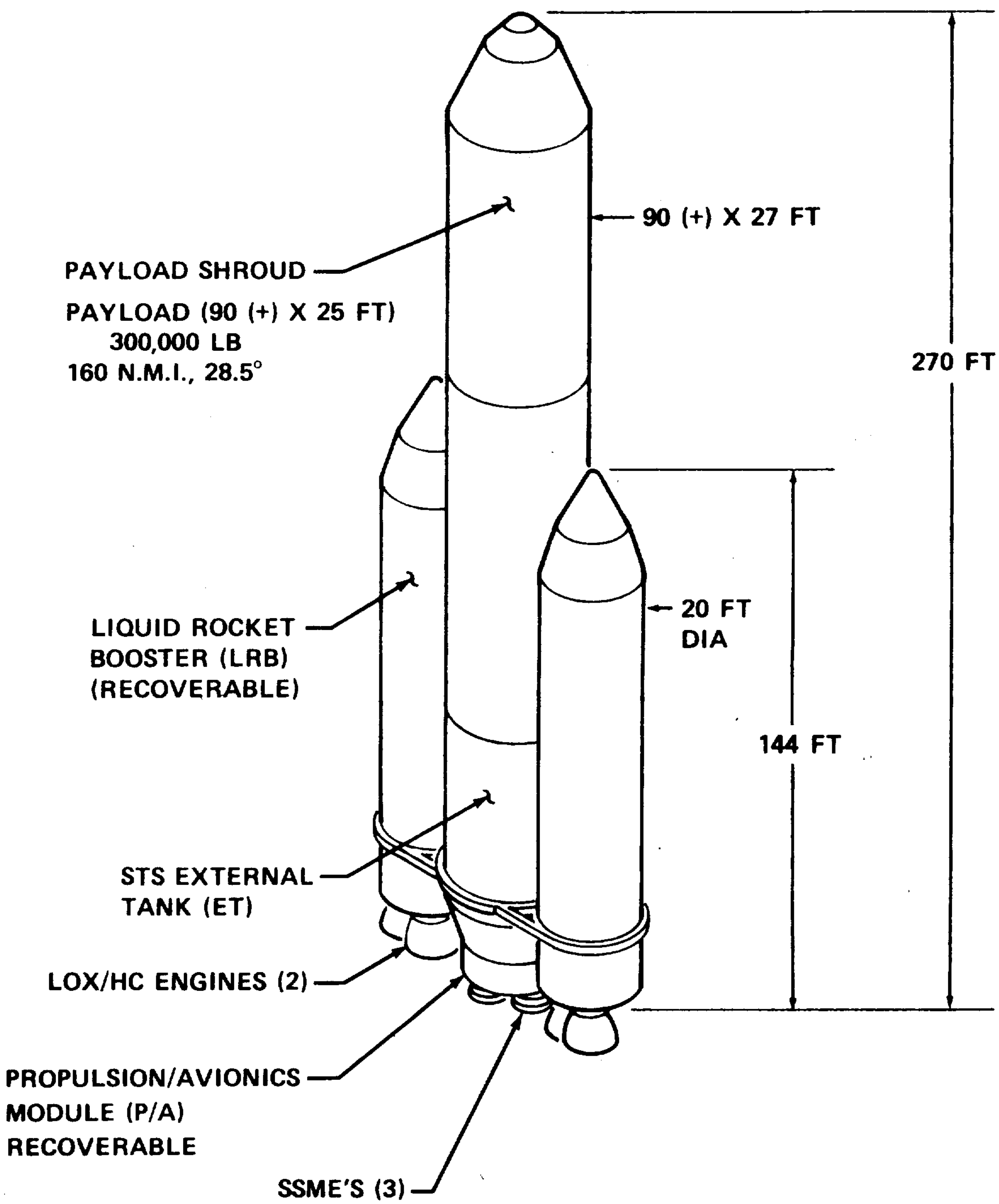
FIGURE 3
SHUTTLE–DERIVED HEAVY LIFT VEHICLE (SD/HLV)



current solid rocket boosters. The "core stage" is retained essentially as utilized in the SDV-3R vehicle.

Vehicle Description

The first stage consists of two reusable liquid rocket boosters, each equipped with two LOX/hydrocarbon rocket engines of approximately 1.6 million pounds thrust. These boosters are 20 feet in diameter, approximately 150 feet in length, and would be recovered by parachute/paraglider types of devices of advanced design, in a manner similar to SRB recovery on the Space Shuttle. The LOX/hydrocarbon boost engine would be developed for this and other applications and could be described as an advanced technology version of the F-1 engine used in the Saturn-Apollo program.

The core stage or second stage consists of an ET with the main engines and avionics installed at the base of the ET in a recoverable P/A module. This stage is retained in essentially the same form as used in the SDV-3R vehicle. A payload shroud of the same diameter as the ET (as shown in the illustration) would allow accommodation of payloads up to 25 ft. x 90 ft. Payloads of larger dimensions can be accommodated without placing undue demands upon vehicle control and dynamics.

A third stage using a single SSME can be employed for intermediate destination orbits beyond the efficient range of the basic two-stage vehicle.

The lower turnaround cost for the reusable liquid rocket booster, due to refurbishment and lower propellant cost than for the SRMs, combined with P/A modules recovery and reuse, will allow per flight costs even lower than an SDV-3R type vehicle of comparable size. The ET and payload shroud are the only expendable items with this arrangement.

Inheritance

The ET for the core stage will be inherited directly from the Space Shuttle and the SDV-3R vehicle, along with production, test, and logistics support capabilities. The recoverable P/A module will likewise be retained directly from the SDV-3R vehicle and SSMEs from both predecessor vehicles. The payload shroud for payloads up to 25 ft. x 90 ft. could be used as is from the SDV-3R vehicle; shrouds for larger payloads can be developed and built as they are needed. The LOX/hydrocarbon booster engines and the booster stages share heritage from F-1 engines and Saturn V boosters, as well as vehicle arrangement and booster recovery methods from the Space Shuttle.

HEAVY LIFT LAUNCH VEHICLE (HLLV)

Advancement in technologies and state-of-the-art may make it advantageous to design and develop a HLLV that is independent of the Shuttle. This "new design" HLLV could use the LOX/HC Booster Engines employed in the SD/HLV and new LOX/LH_2 Engines for upper stages. Techniques involving propellant cross-feed from Booster to Center Core Stage during the boost phase of the flight would enhance the performance of the vehicle. A new advanced recovery system, advanced avionics/software and improved operations could make a new advanced configuration economical.

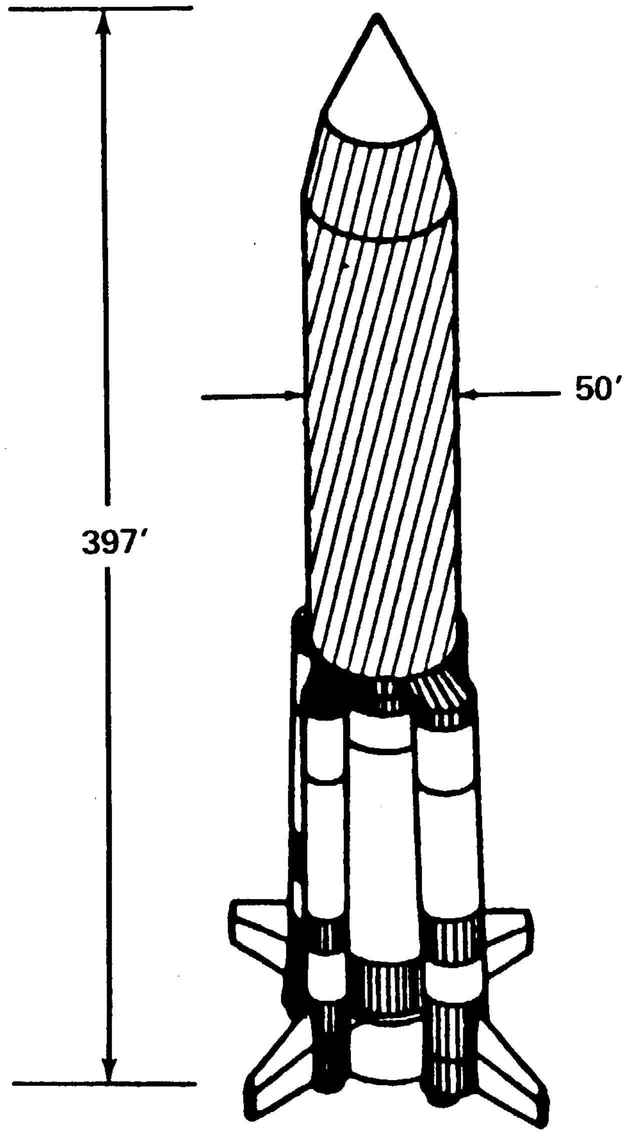
Vehicle Description

Since this vehicle definition is still in the early stage, only basic concepts and descriptions are possible at this time. Figure 4 shows a typical HLLV concept. The first stage could consist of two to four reusable liquid rocket boosters or of boosters with reusable P/A modules similar to that used for the SDV-3R core stage. Each booster is equipped with two LOX/Hydrocarbon rocket engines of approximately 1.6 million pounds thrust or two of the boosters are equipped with two LOX/Hydrocarbon engines and the other two boosters are equipped with one LOX/Hydrocarbon engine. The LOX/Hydrocarbon boost engines would be developed for this and other applications and could be described as an advanced technology version of the F-1 engine used in the Saturn-Apollo program. The booster or the booster boat-tail containing the engines and avionics would be recovered and reused.

The core stage or second stage consists of a propellant tank and a recoverable P/A Module containing five LOX/LH_2 engines. The boosters and core stage use engines which are burned in parallel. The boosters include an auxiliary liquid hydrogen tank which permits cross feed of LH_2 and LOX into the core LH_2 and LOX tanks, which permits the core stage to have a full complement of propellant at booster separation, resulting in higher vehicle performance.

A third stage using SSME or an Advanced Cryogenic Engine (ACE) can be employed for intermediate or high energy missions beyond the efficient range of the basic two-stage vehicle. The payload shroud to accommo-

FIGURE 4. HEAVY–LIFT LAUNCH VEHICLE (HLLV)
(RECOVERABLE PROPULSION AND AVIONICS (P/A) MODULE)



- REUSABLE P/A MODULE
  - 5 X STME'S @ 100%
- 4 X LIQUID BOOSTERS (RECOVERABLE)
  - 2 W/2 X 1.616 MLBF ENGINES
  - 2 W/1 X 1.616 MLBF ENGINE
- 33 FT. DIA. EXPENDABLE CORE STAGE TANKAGE
- USABLE P/L ENVELOPE – 45' X 200'
- PERFORMANCE*
  - 270 NM/28.5° – 401K
  - 160 NM/28.5° – 418K
  - 540 NM/90° – 302K
- IOC 1995

* ORBITS CIRCULARIZED WITH KICK STAGE

date payloads of about 45 feet in diameter by about 200 feet in length will require development.

Performance

The basic two-stage vehicle can place approximately 408,000 pounds to a 160 n.mi., 28.5 degree inclination orbit, approximately 401,00 pounds into a 270 n.mi., 28.5 degree orbit, and approximately 302,000 pounds to a 540 n.mi., polar orbit.

Payloads of approximately 120,000 pounds to high energy orbits or to GEO are possible depending on the size of the third stage.

Launch Facilities/Operations

New launch facilities and launch sites must be investigated. Specifics will be dependent upon vehicle configuration, logistics, launch rates expected and mission requirements.

Schedule

The schedule will, of course, be dependent on the mission requirements. Ten to twelve years is normally required for a new vehicle development.

SUMMARY

Manned Mars missions will require launch vehicles with considerably larger capability than the present STS. Launch vehicles evolving from the Shuttle can be made available in the early years to meet MMM goals. Also, larger vehicles can be made available in the later years using new and improved techniques. Economic analyses need to be made to determine the best vehicle for the mission and the time period the mission is accomplished.

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12. Paper in Section III of this report, entitled "MMM ETO Delivery and On-Orbit Assembly", by B. Barisa and G. Solmon of MSFC.

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MARS LANDER SURVEY

N87-17741

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ABSTRACT

The requirements, issues, and design options are reviewed for manned Mars landers. Issues such as high l/d versus low l/d shape, parking orbit, and use of a small Mars orbit transfer vehicle to move the lander from orbit to orbit are addressed. Plots of lander mass as a function of Isp, destination orbit, and cargo up and down, plots of initial stack mass in low Earth orbit as a function of lander mass and parking orbit, detailed weight statements, and delta V tables for a variety of options are included. Lander options include a range from minimum landers up to a single stage reusable design. Mission options include conjunction and Venus flyby trajectories using all-cryogenic, hybrid, NERVA, and Mars orbit aerobraking propulsion concepts.

REQUIREMENTS

A manned Mars lander or Mars Excursion Module (MEM) will be one of, if not the major cost item in a manned Mars mission program. The nature of the program will determine the requirements for the lander. The major questions are: 1) How many landings or missions are to be flown, or what is the overall scope of the program? 2) How long must the lander support a crew on the surface? and 3) Must major cargo items be landed?

A short program with only two or three Apollo style landings would be required to support a crew for only a few weeks or a month on the surface, and land only a small amount of cargo. Cost would probably be the major driver. Only approximate guidance and navigation might be adequate.

A 20 mission program might require a lander that could spot-land, grow to support a crew for 100s of days on the surface, take advantage of surface propellant production, and perhaps land significant cargos, such as a surface base. Performance, which would be important in long term costs, might well be the driver.

The program is not defined at present, so we must look at all the options. The lander will be expensive and we only want to design one, and may only get the chance to design one, so the program must be carefully defined at the start.

It may be possible to design a Mars lander that can also be used on the Moon[1].

	Lunar	Mars
Descent Delta V, km/sec	2.08	1.23
Ascent Delta V, km/sec	1.91	4.84 minimum
		6.00 typical

Since the Mars lander ascent tanks will not be full when landing on the Moon, the descent tanks, sized for a Mars landing, may be able to handle lunar descent. Reference 1 proposed a lunar surface landing as part of a MEM test program.

ISSUES

The lift/drag shape of the lander is a major issue. Two basic families of shapes have been proposed, the low lift/drag (l/d) ratio or Apollo Command Module shape, and the high l/d or lifting body shape. Figures 1 through 4 show proposed low l/d shapes. Figures 5, 6, 7, and 8 show different high l/d shapes.

The low l/d shape is roughly 10 % lighter (Ref. 1) than typical high l/d designs. The low l/d lander is easier to build and test and therefore less expensive, and can accommodate growth more easily. The low l/d shape may be more easily built to land on the Moon. The low l/d shape may not be capable of direct entry into the Mars atmosphere from a trans-Mars trajectory (if this is a desired requirement), and may be more difficult to spot-land. Landing accuracy problems may be overcome to some extent by additional hover propellant.

Figure 9 shows a concept for a Mars base in a water-eroded canyon that would require spot-landing capability. Such a difficult landing site may be a desired target, because of the possibility of fossils or other evidence of life in those locations.

The high l/d shapes have a wider entry corridor, a much bigger footprint, and may be easier to spot-land. There is a problem keeping the g forces on the crew "eyeballs in" during both entry and ascent, however, without drastic measures. The high l/d shapes can enter directly from the interplanetary trajectory to the surface.

Fig. 1 Rockwell low l/d MEM
l/d-.5

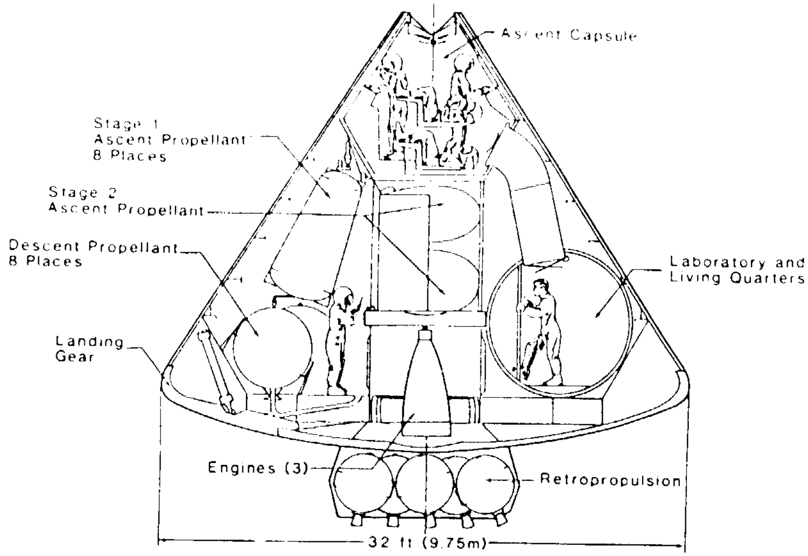

Fig. 2 Rockwell low l/d MEM,
Mission Phase Configurations
(from Ref.1)

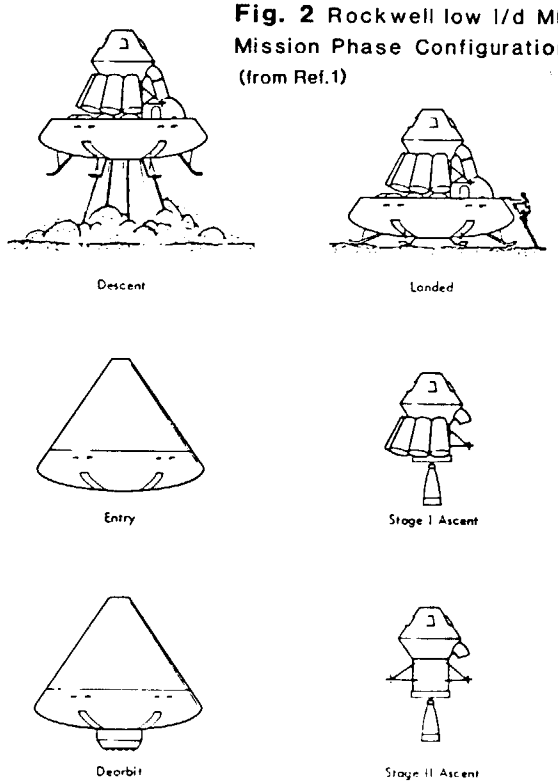

Fig. 3 Rockwell low l/d MEM
with MSFC updates, Three View

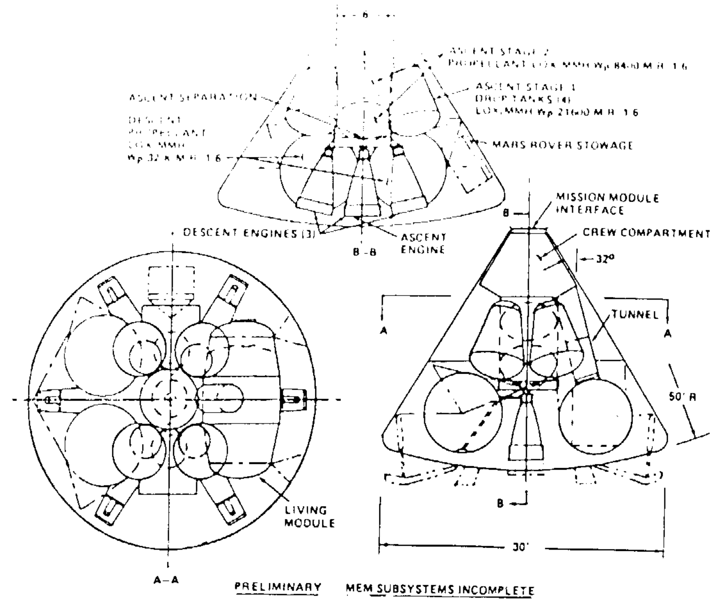

Fig. 4 MSFC low l/d concept
with open afterbody

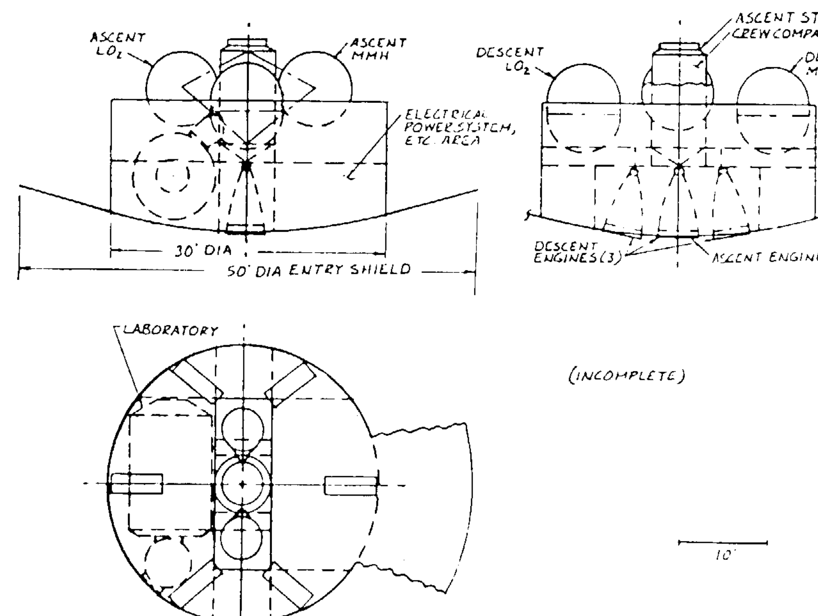

Fig. 5 Rockwell lifting body MEM l/d-1.0, wings drop off before landing. (from Ref.1)

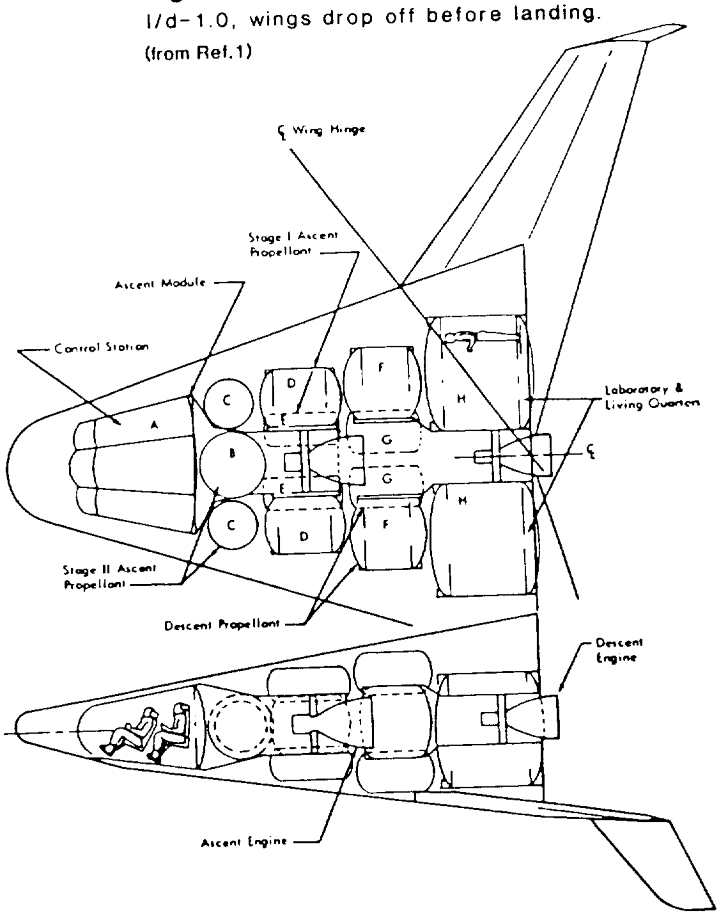

Fig. 6 Rockwell lifting body MEM ascent (from Ref.1)

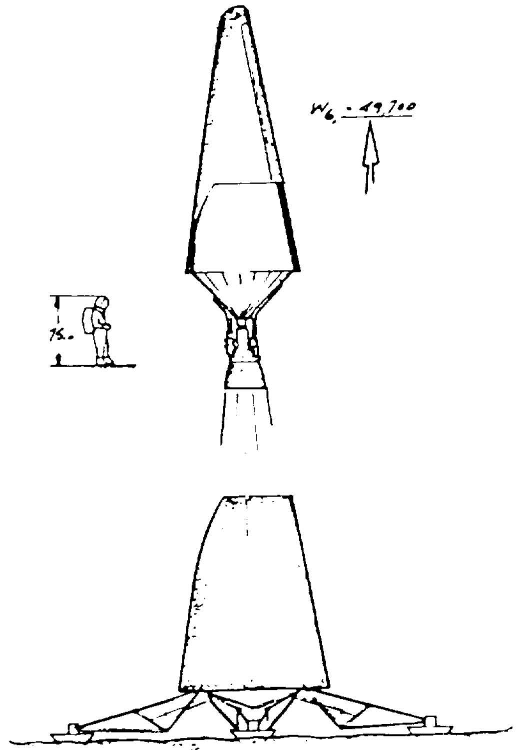

Fig. 7 Case for Mars II Bent Biconic Concept-uses surface produced propellants. (from Ref.4)

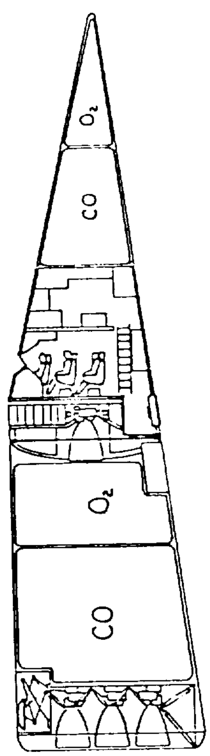

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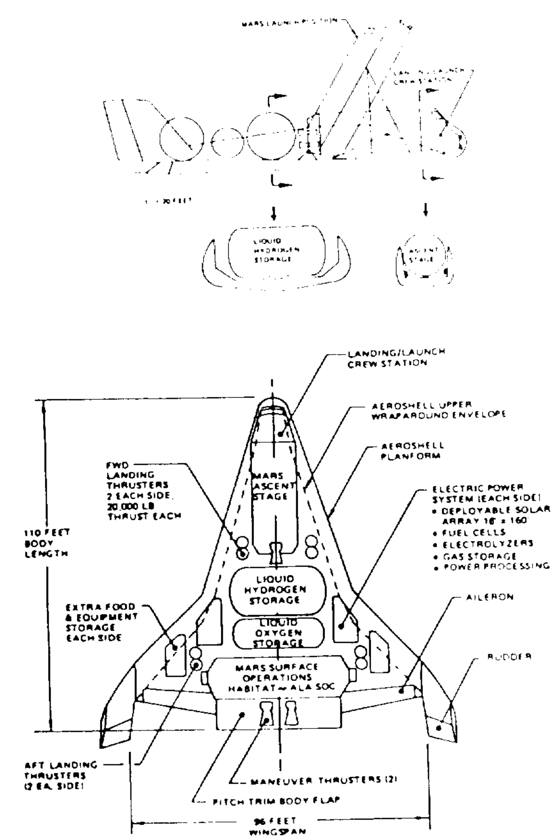
Fig. 8 Open Afterbody high l/d MEM (from Ref. 2)



The most comprehensive study of manned Mars landers to date (Ref. 1, 1967), which did comparison designs of both high and low l/d shapes (Figures 1 and 5), chose the low l/d as a baseline. This was based on cost, testing requirements, and simplicity, and the absence of mission requirements that might dictate another choice (such as a requirement for direct entry). Since the body of data Rockwell subsequently generated (Ref. 1) on a low l/d design is extensive, and the mission requirements have not been defined much better since 1967, this paper uses the low l/d shape as a baseline for calculation purposes. To get high l/d numbers, add roughly 10% to the gross weights in the graphs and tables.

Another issue of significance is Mars parking orbit: low circular (500 km), high elliptical (24 hour), or none (direct entry from the interplanetary trajectory for the lander, and hyperbolic rendezvous with a passing interplanetary spacecraft at departure). The lander is insensitive to entry parking orbit (given a low perigee or a low circular orbit; this is not true for high circular orbit), in terms of mass, since it uses essentially an aerobraked entry. G levels for direct entry and entry from the elliptical parking orbits may be high, however. Ref. 1 predicts g levels of 4.5 for high elliptical versus 2 for low circular entry. This may make a significant difference for a crew that has been in zero g for six months or more.

The higher the orbit the lander must ascend to, the greater its initial mass. Figure 10 plots lander entry mass versus destination orbit for a variety of possible landers. The difference between low circular and hyperbolic escape values is only a factor of two or so. Figure 11 shows the effect of high elliptical and low circular parking orbit on initial mass in LEO for a variety of propulsion and trajectory schemes. The high elliptical parking orbit reduces Mars orbit insertion and trans-Earth insertion burns by over a km/sec each. This vastly overwhelms the effect of lander mass changes and can lead to a reduction in initial mass in LEO by factors of 1.3 to 2.0, depending on the mission propulsion and trajectory. So, based on LEO mass, the high elliptical parking orbit is better than a low circular orbit.

A small Orbital Transfer Vehicle (OTV) can also be used to ferry the MEM ascent stage from low circular Mars orbit to high elliptical Mars orbit. This small stage could result in savings of 10 to 20% of initial

Figure 9

Mars Base in a Canyon, spot landings required

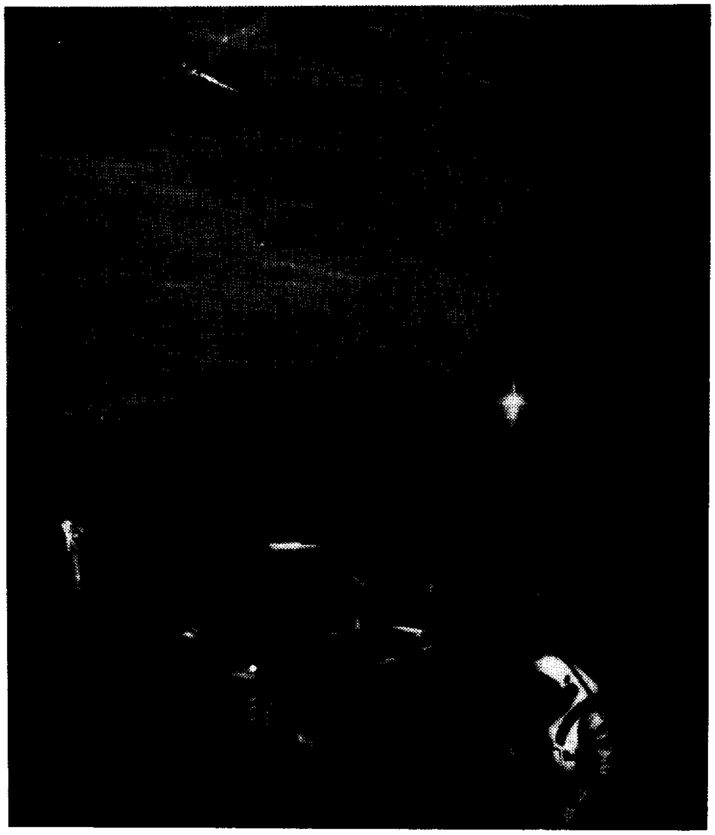


Figure 10

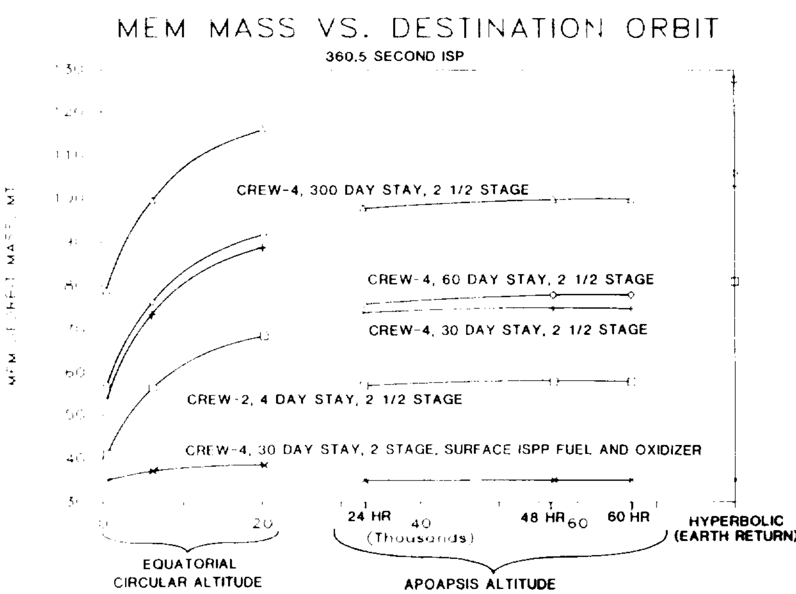
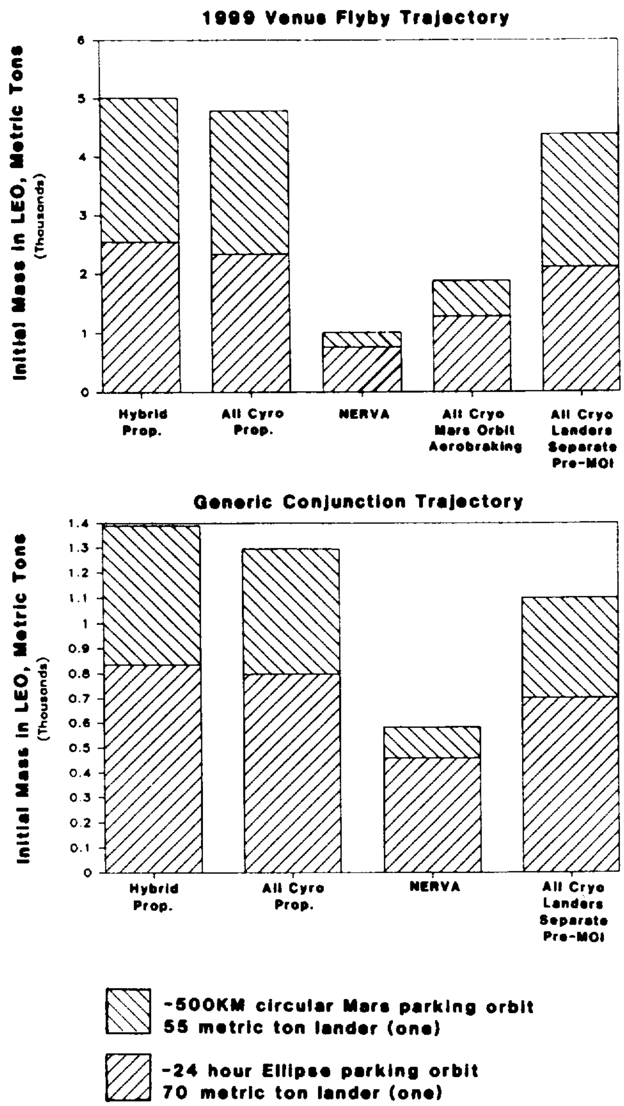


Figure 11

Initial Mass in LEO for 500KM circular and 500KM X 32,963KM (24 hour) Mars parking orbits.



All cases use a 53 metric ton Mission Module, 360.5 sec ISP landers and all carry a 31 metric ton MOTV. All cases are 3 stage, last stage does TEI and EOI. Hybrid-first two stages are all cryo. (H2/O2), last stage-O2/ Propane

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MEM + OTV mass in high elliptical Mars orbit compared to a one and one half stage MEM capable of ascending directly from the surface to high elliptical orbit. The cost of the OTV would probably overshadow the mass savings however, unless the OTV was required for another purpose, such as to visit Phobos and Deimos.

The Ref. 1 design uses no chutes or ballutes. That report concludes that this reduces the development cost substantially, but makes the lander 5 to 10% heavier. Figure 12 plots initial stack mass in LEO as a function of one-way payload mass to Mars (MEM + OTV mass) for a variety of cases. Note the slopes. One extra metric ton of lander and/or OTV mass costs 2.3 to 6.4 metric tons in LEO, depending on the propulsion and trajectory scheme.

Figure 13 plots lander mass versus specific impulse for a variety of cases. The cargo lander is insensitive to specific impulse, indicating a one way lander using solids might be possible. The MEM using surface-produced-propellant is also insensitive, indicating the proposed CO/O_2 propellant, whose Isp may be less than 300 seconds is feasible. The CO/O_2 propellant may be easy to produce from the carbon dioxide atmosphere of Mars.

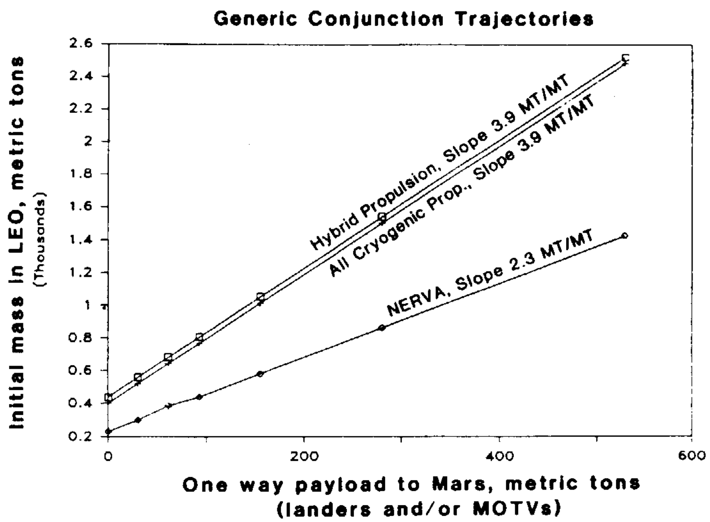
Figure 14 plots MEM deorbit mass versus cargo mass down. The problem of a cargo lander will be packaging in an aeroshell. Figure 15 shows a lunar cargo lander unloading an 18 metric ton Space Station Common Module, postulated to be the largest and heaviest cargo to be landed on the Moon (Ref. 3). Figures 4 and 8 (from ref. 3) show low and high l/d concepts with open afterbodies that could accomodate such a cargo.

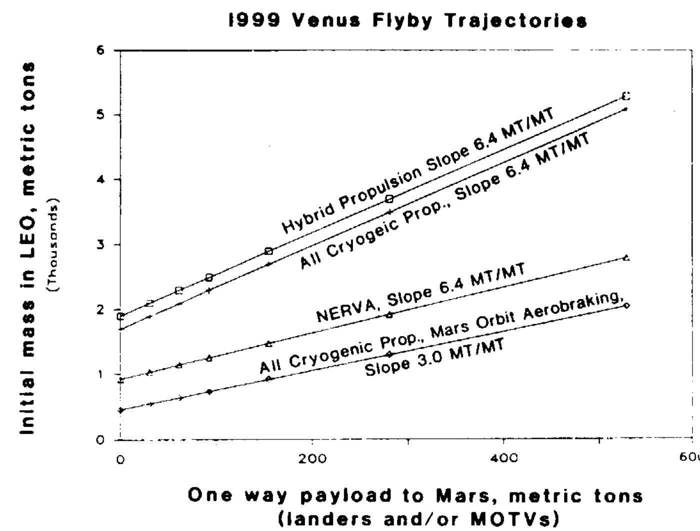
Figure 16 shows MEM deorbit mass versus ascent cargo mass for several cases. To lift tens of tons off the surface will strongly drive the design towards surface propellant production. Table 1 shows the delta Vs used to produce the plots discussed below.

CONFIGURATIONS

Figure 3 shows the 1967 Rockwell low l/d design with recent updates provided by the Marshall Space Flight Center (MSFC) group, which includes a different engine design and propellant. The weight statement provided in reference 1 with MSFC updates was extrapolated with scaling equations and other software to produce Tables 2 and 3 and Figures 11 through 16.

Initial Mass in LEO versus one-way payload to Mars
24 hour parking orbit at Mars and Earth

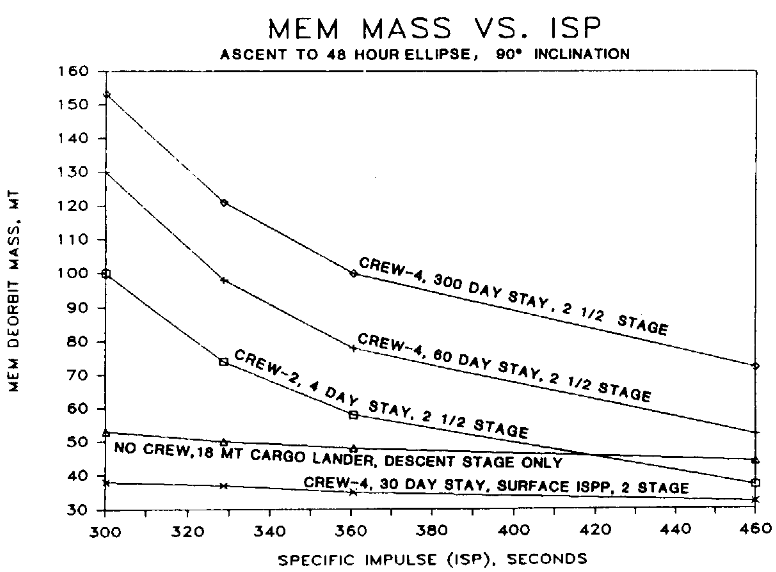




All cases use a 53 metric ton Mission Module:
All cases are 3 stage, except last stage does TEI and EOI. Hybrid-first two stages are all cryogenic (H_2/O_2), last stage-O_2/Propane.

Fig. 13

Fig. 14



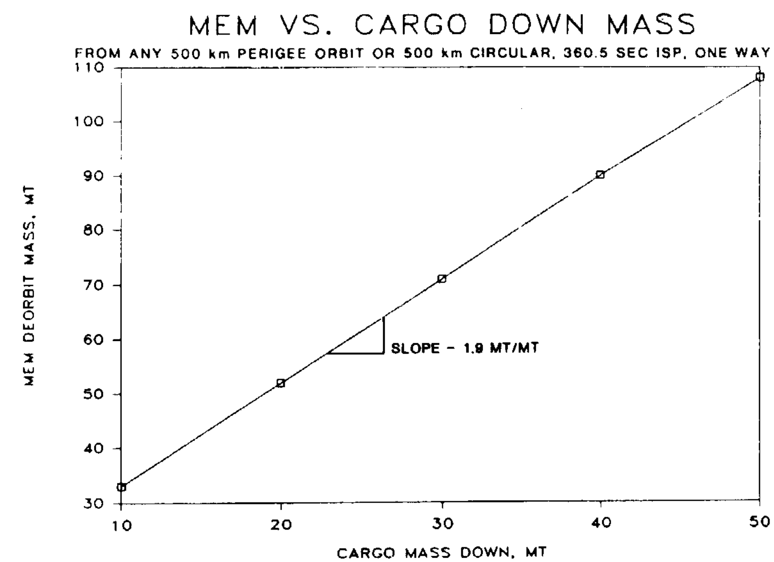


Figure 15

Cargo lander on the Lunar Surface

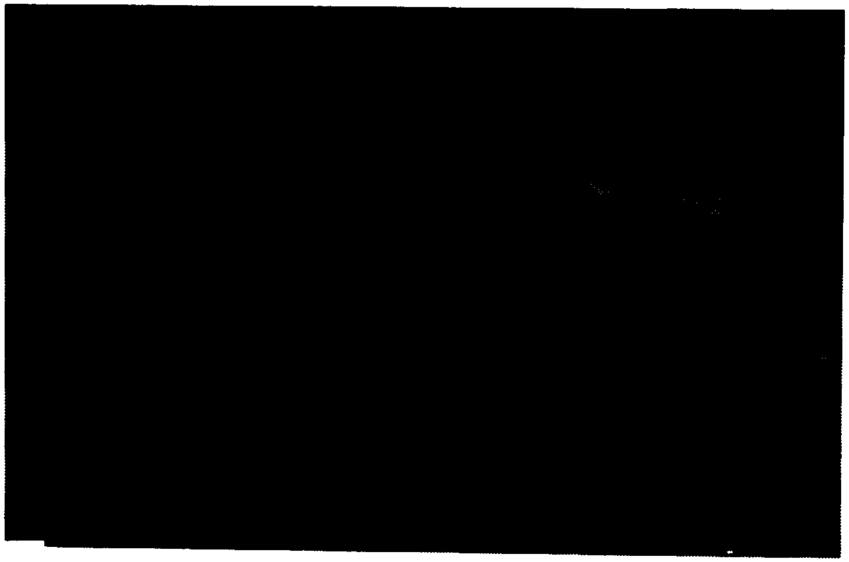


Figure 16

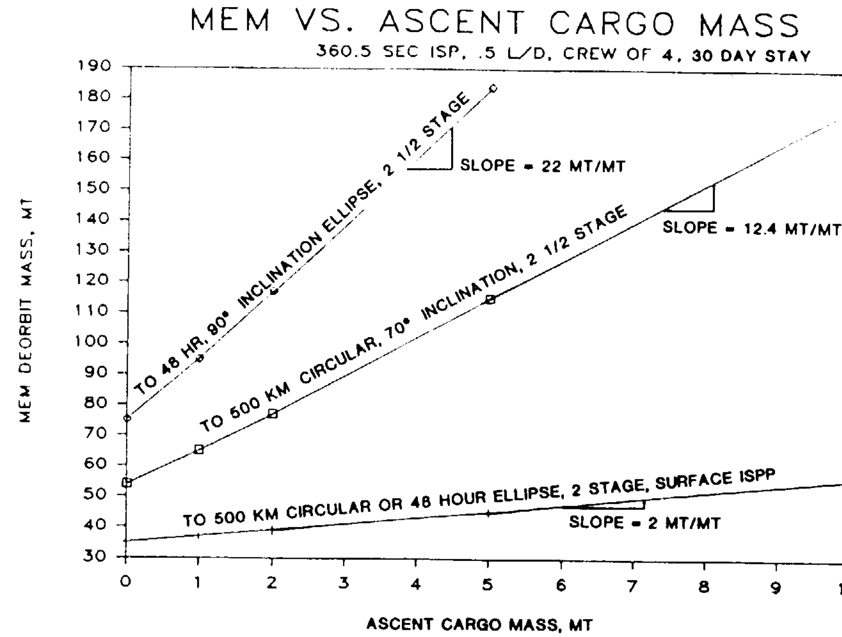


Table 1

MEM DELTA V TABLE

MM ORBIT	HYPERBOLIC (EARTH RETURN) km/sec	60 HOUR 300 x 66,900 km km/sec	48 HOUR 500 x 57,000 km km/sec	24 HOUR 500 x 32,963 km km/sec	DEIMOS 20,168 km circular km/sec	PHOBOS 6,068 km circular km/sec	LOW CIRC. 500 km circular km/sec
DEORBIT DELTA V	0.2	0.2	0.2	0.2	0.667	0.574	0.2
LANDING DELTA Vs							
*BRAKE (Max. delta V, no chute)	0.92	0.92	0.92	0.92	0.92	0.92	0.92
*WIND (52 m/sec)	0.08	0.08	0.08	0.08	0.08	0.08	0.08
*HOVER (1 min., 1 nm range)	0.23	0.23	0.23	0.23	0.23	0.23	0.23
LANDING TOTAL*	1.23	1.23	1.23	1.23	1.23	1.23	1.23
ASCENT TO 185x93 km	3.9	3.9	3.9	3.9	3.9	3.9	3.9
DEST. ORB. INCLIN.	15	90	90	90	1.8	1.1	70
INCLIN. EFFECT	0.008	0.242	0.242	0.242	.000	.000	0.159
DRAG LOSS	0.15	0.15	0.15	0.15	0.15	0.15	0.15
CIRC. AT 185 km	0.02	0.02	0.02	0.02	0.02	0.02	0.02
TRANSFER TO FINAL ORBIT	2.405	1.32	1.3	1.226	1.75	1.257	0.17
ASCENT SUBTOTAL	6.48	5.63	5.61	5.54	5.82	5.33	4.40
10% CONTINGENCY	0.65	0.56	0.56	0.55	0.58	0.53	0.44
ASCENT TOTAL	7.13	6.20	6.17	6.09	6.40	5.86	4.84
TRANSFER FROM 500 KM CIRCULAR TO	2.38	1.25	1.23	1.14	1.66	1.13	-

km x .54 = nm km x 3280 = ft

Table 2

MEM WEIGHT STATEMENT SUMMARY

MEM OPTION	MIN. MEM	30 DAY	60 DAY	300 DAY	CARGO MEM	SURFACE ISPP MEM, 2 STGE	REUSABLE MEM (SING. STAGE)
CREW	2	4	4	4	0	4	4
LIFE SUP. DURATION (days)	4	30	60	300	0	30	30
AERO L/D	0.5	0.5	0.5	0.5	0.5	0.5	0.5
ASCEND TO	500x32963km (24 hour)	500x32963km (24 hour)	500x32963km (24 hour)	500x32963km (24 hour)	500x32963km (24 hour)	500x32963km (24 hour)	500x32963km (24 hour)
PRIMARY PROPULSION Isp, sec	360.5	360.5	360.5	360.5	360.5	360.5	460
LENGTH, m	8	8	8	8	8	8	-
DIAMETER, m	9	9	9	9	9	9	-
STORM SHELTER ?	NO	YES	YES	YES	NO	YES	NO
LANDED CARGO (MT)	1.91	1.91	1.91	1.91	18.00	1.91	1.91
ASCENT CARGO (MT)	0.14	0.14	0.14	0.14	0.00	0.14	2.05
DEORBIT MASS (MT)	56.52	73.71	76.38	98.30	48.31	35.09	451.05

TABLE 3

MEM WEIGHT STATEMENT

ASCENT TO 24 HOUR, 500 KM PERIAPSIS ELLIPSE.

MEM OPTION	MIN. MEM	30 DAY	60 DAY	300 DAY	CARGO MEM	SURFACE ISPP MEM, 2 STGE	REUSABLE MEM (SING. STAGE)
(ALL MASSES IN KGMS UNLESS OTHERWISE NOTED)							
ASCENT CAPSULE							
PRIMARY STRUCTURE	255	255	255	255	255	255	510
COUCH, RESTRAINTS	18	36	36	36	0	36	36
HATCHES, WINDOWS	55	55	55	55	55	55	55
DOCKING PROVISIONS	77	77	77	77	77	77	77
PANELS, SUPPORTS	23	23	23	23	23	23	23
BATTERY	123	123	123	123	123	123	123
EPS DISTRIBUTION	105	105	105	105	105	105	105
COMMUNICATIONS	95	95	95	95	95	95	95
GUIDANCE AND NAV.	102	102	102	102	102	102	102
CONTROLS & DISPLAYS	91	91	91	91	0	91	91
INSTRUMENTATION	86	86	86	86	0	86	86
LIFE SUPPORT SYS.	236	432	432	432	0	432	432
RCS - DRY	107	133	133	133	0	133	151
RCS - PROPELLANT	89	110	110	110	0	110	125
RETURN PAYLOAD	136	136	136	136	0	136	136
CREW	159	318	318	318	0	318	318
CONTINGENCY	195	242	242	242	93	242	274
ASCENT CAPSULE TOTAL	1,953	2,419	2,419	2,419	928	2,419	2,738
ASCENT PROPULSION							
STAGE 2 DELTA V, km/sec	2.66	2.66	2.66	2.66	0.00	0.00	0.00
TANK MASS/PROP. MASS	0.07	0.07	0.07	0.07	0.07	0.07	0.07
2ND STAGE ISP, sec	360.5 (LO2/MMH)	360.5 (LO2/MMH)	360.5 (LO2/MMH)	360.5 (LO2/MMH)	360.5 (LO2/MMH)	360.5 (LO2/MMH)	460 (LO2/H2)
2ND STAGE MASS RATIO	2.12	2.12	2.12	2.12	1.00	1.00	1.00
TANKS & SYSTEM	243	294	294	294	0	304	0
ENGINE & INSTAL.	253	253	253	253	0	253	0
CONTINGENCY	50	55	55	55	0	56	0
BOILOFF & ULLAGE	316	382	382	382	0	0	0
USABLE 2ND STGE PROP	3,162	3,823	3,823	3,823	0	0	0
2ND STAGE PROP. WITH BOILOFF & ULLAGE	3,478	4,205	4,205	4,205	0	0	0
2ND STAGE PROPULSION SYSTEM MASS TOTAL	4,025	4,807	4,807	4,807	0	613	0
2ND STAGE IGNITION MASS	5,978	7,226	7,226	7,226	928	3,032	2,738
1ST STAGE DELTA V km/sec	3.43	3.43	3.43	3.43	0.00	0.00	0.00
TANK MASS/PROP. MASS	0.07	0.07	0.07	0.07	0.07	0.07	0.07
1ST STAGE ISP, sec	360.5 (LO2/MMH)	360.5 (LO2/MMH)	360.5 (LO2/MMH)	360.5 (LO2/MMH)	360.5 (LO2/MMH)	360.5 (LO2/MMH)	460 (LO2/H2)
1ST STAGE MASS RATIO	2.64	2.64	2.64	2.64	1.00	1.00	1.00
TANKS & SYSTEM	1,083	1,309	1,309	1,309	0	1,382	0
ENGINE & INSTAL.	0	0	0	0	0	0	0
CONTINGENCY	108	131	131	131	0	138	0
BOILOFF & ULLAGE	1,407	1,700	1,700	1,700	0	0	0
USABLE 1ST STGE PROP	14,066	17,004	17,004	17,004	0	0	0
1ST STAGE PROP. WITH BOILOFF & ULLAGE	15,473	18,704	18,704	18,704	0	0	0
1ST STAGE PROPULSION SYSTEM MASS, TOTAL	16,664	20,144	20,144	20,144	0	1,520	0
1ST STAGE IGNITION MASS (TOT. ASCENT)	22,642	27,370	27,370	27,370	928	4,552	2,738

TABLE 3

MEM WEIGHT STATEMENT (CONT'D.)

MEM OPTION	MIN. MEM	30 DAY	60 DAY	300 DAY	CARGO MEM	SURFACE ISPP MEM, 2 STGE	REUSABLE MEM (SING. STAGE)
DESCENT STAGE							
JETTISONED STRUCTURE	2,114	2,114	2,114	2,114	2,114	2,114	0
RETAINED STRUCTURE	2,477	2,477	2,477	2,477	2,477	2,477	7,500
SEC. STRUCTURE	409	409	409	409	409	409	409
LAB STRUCTURE	477	3,810	3,810	3,810	0	3,810	477
ELECTRICAL PWR SYS.	253 (2kw fcell)	1,009 (2kw fcell)	1,882 (2kw fcell)	8,864 (2kw fcell)	0	1,009 (2kw fcell)	1,009 (2kw fcell)
POWER DISTRIBUTION	182	182	182	182	0	182	182
COMMUNICATION	168	168	168	168	0	168	168
GUIDANCE & NAV.	5	5	5	5	0	5	5
CONTROLS & DISPLAYS	5	5	5	5	0	5	5
INSTRUMENTATION	114	114	114	114	114	114	114
LIFE SUPPORT SYS. (open loop)	22 (2kw fcell)	621 (2kw fcell)	1,169 (2kw fcell)	5,555 (2kw fcell)	0	621 (2kw fcell)	621 (2kw fcell)
RCS - DRY	441	575	596	767	376	273	3,613
RCS - PROPELLANT	912	1,191	1,234	1,588	780	566	7,484
LANDING GEAR	991	991	991	991	991	991	991
NET LANDED PAYLOAD	1,909	1,909	1,909	1,909	18,000	1,909	1,909
CONTINGENCY	1,164	1,731	1,896	3,217	2,807	1,628	10,494
DESCENT SUBTOTAL	11,643	17,310	18,960	32,175	28,068	16,281	34,981
DESCENT PROPULSION							
DESCENT DELTA V, km/sec	1.23	1.23	1.23	1.23	1.23	1.23	7.32
TANK MASS/PROP. MASS	0.07	0.07	0.07	0.07	0.07	0.07	0.06
DES. STAGE ISP, sec	360.5 (LO2/MMH)	360.5 (LO2/MMH)	360.5 (LO2/MMH)	360.5 (LO2/MMH)	360.5 (LO2/MMH)	360.5 (LO2/MMH)	460 (LO2/H2)
DES. STGE MASS RATIO	1.42	1.42	1.42	1.42	1.42	1.42	5.07
TANKS & SYSTEM	1,144	1,493	1,547	1,991	978	710	21,961
ENGINE & INSTAL.	504	704	704	1,000	704	704	2,000
CONTINGENCY	165	220	225	299	168	141	2,396
BOILOFF & ULLAGE	925	1,207	1,251	1,610	790	574	20,718
USABLE DES STGE PROP	15,418	20,116	20,847	26,839	13,175	9,563	345,304
DES. STGE PROP. WITH BOILOFF & ULLAGE	16,344	21,323	22,097	28,449	13,965	10,136	366,022
DESCENT STAGE PROPULSION MASS	18,156	23,740	24,573	31,740	15,815	11,691	392,379
DES. STAGE IGNITION MASS (ENTRY MASS)	52,442	68,420	70,904	91,285	44,811	32,524	430,100
DEORBIT PROPULSION							
DEORBIT DELTA V, km/sec	0.20	0.20	0.20	0.20	0.20	0.20	0.20
DEOR. TANK/PROP MASS	0.07	0.07	0.07	0.07	0.07	0.07	0.06
DEORBIT ISP, sec	300 (GOOD SOLID)	300 (GOOD SOLID)	300 (GOOD SOLID)	300 (GOOD SOLID)	300 (GOOD SOLID)	300 (GOOD SOLID)	460 (LO2/H2)
DEORBIT MASS RATIO	1.07	1.07	1.07	1.07	1.07	1.07	1.05
TANKS & SYSTEM	260	339	352	453	222	162	1,174
ENGINE & INSTAL.	100	100	100	100	100	100	200
CONTINGENCY	0	0	0	0	0	0	0
BOILOFF & ULLAGE	0	0	0	0	0	0	0
USABLE DEORBIT PROP	3,717	4,847	5,023	6,465	3,177	2,308	19,574
DEORBIT PROP. WITH BOILOFF & ULLAGE	3,717	4,847	5,023	6,465	3,177	2,308	19,574
DEORBIT STAGE	4,077	5,287	5,475	7,017	3,500	2,569	20,948
DEORBIT IGNITION MASS (MEM TOT. MASS)	56,519	73,707	76,378	98,302	48,310	35,094	451,048

Table 3 and the plots use the basic Rockwell design, first stage descent and second stage ascent concepts with drop tanks, and an open loop life support system, using 2 KW fuel cell power. No life support volume calculations were performed. No chutes or ballutes were included. 10% ascent delta V and 10% dry mass contingency numbers were used. A 3.3 metric ton storm shelter for solar flares was used for all configurations except the four day stay and reusable, single stage MEM. Boiloff was limited to 10% of usable stage propellant for the ascent stages. This assumption may not be realistic for the longer surface stays.

Seven different vehicle designs were addressed: (1) A minimum MEM (4 day stay for a crew of two), (2) 30 day stay MEM, (3) 60 day stay MEM, (4) 300 day stay MEM, (5) A cargo lander, (6) Surface-produced-propellant using MEM (in situ propellant production, or ISPP), and (7) A reusable single stage MEM. Table 2 summarizes their characteristics for one case for which a weight statement (Table 3) is included.

The single stage reusable MEM numbers in the tables should be viewed with caution because they are a distant extrapolation from the original Rockwell vehicle. All structural mass was doubled, and a 30% contingency on dry mass was added (up from 10%). Iterative calculations assuming two metric tons payload up and down plus a crew of four and 30 days consumables resulted in the following numbers for a single stage reusable MEM:

Case		Mars Entry Mass
To a 60 hour ellipse, 360.5 sec. Isp	-	1,206 m. tons
To 500 km circular, 360.5 sec. Isp	-	300 m. tons
To 500 km circular, 460 sec. Isp	-	157 m. tons
Surface ISPP for ascent stage only, 300 sec. Isp, to any orbit	-	83 m. tons
Surface ISPP for ascent stage only, 460 sec. Isp, to any orbit	-	69 m. tons

At least in terms of simple mass calculations, a single stage reusable MEM does not appear to be out of reason. A substantial infrastructure in Mars orbit or on the surface will be needed to maintain it, however.

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MARS BASE BUILDUP SCENARIOS

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ABSTRACT

Two Mars surface base build-up scenarios are presented in order to help visualize the mission and to serve as a basis for trade studies. In the first scenario, direct manned landings on the Martian surface occur early in the missions and scientific investigation is the main driver and rationale. In the second scenario, early development of an infrastructure to exploit the volatile resources of the Martian moons for economic purposes is emphasized. Scientific exploration of the surface is delayed at first in this scenario relative to the first, but once begun develops rapidly, aided by the presence of a permanently manned orbital station.

INTRODUCTION

In order to place the manned Mars mission studies on a more firm conceptual basis, I believe that it is helpful to establish one or more specific mission scenarios. This makes it possible to more clearly visualize the context of the overall mission. Base build-up scenarios can serve as a consistent basis for back calculation (e.g., propulsion requirements) and form a common ground for trade studies, costing, etc. The evolutionary scenarios I propose are two, by necessity, somewhat arbitrary cases selected from a potentially large set of reasonable alternatives. Nevertheless, I believe they perhaps represent "end member" cases that emphasize national political and basic science goals on the one hand versus operational and economic motivations on the other (see refs. [1], [2], and [3] for discussions of the rationales for a manned Mars mission). The scenarios arbitrarily extend over five manned missions and twenty years from the start date. These numbers could easily be extended by factors of two or more but with, in my opinion, considerable less impact and likelihood of sustained funding. On the other hand, it seems unlikely that anything less than three manned missions could achieve the ambitious overall goals.

COLUMBUS BASE SCENARIO

Objective

The overall objective of this scenario is to establish a manned outpost on the surface of Mars to serve as a base for the scientific exploration of the planet.

Time-line

The missions begin with an unmanned precursor approximately four years before the first manned landing on the Martian surface (the individual missions are discussed in detail below). It is assumed that mission opportunities occur approximately every 2 years and are of the "opposition" type (ref.[4]). The first three landings are spaced 4 years (2 opportunities) apart and are essentially identical explorations of three sites on the planet (designated sites A, B, and C, Table 1). The fourth landing two years later returns to one of the previous landing sites that has been selected as the site at which to begin establishment of the permanent base. Two years later the fifth mission lands an expanded crew to complete construction of the base. When a portion of the crew of the fifth mission leaves some months later, a hold-over crew is left on Mars until relief at the next opportunity. This ends the first phase of the exploration of Mars and assumes a second phase (not discussed) that continues and expands permanent human occupation of the planet.

Unmanned Precursor Mission

The purpose of an unmanned precursor mission is to obtain information about potential landing sites that will reduce the risk of the first manned landing, position essential assets in the Martian vicinity for future missions, and determine the feasibility of processing resources contained within the Martian moons. These important operational objectives will be supplemented by a considerable increase in basic scientific knowledge about Mars and it's moons.

I envision the spacecraft to position a satellite in a low-altitude, high-inclination orbit from which optical imagery of the surface will be acquired with a per pixel resolution of about one meter. This would allow discrimination of boulders down to a dimension of about three meters, the smallest size object likely to represent a serious landing hazard. Resolution of Viking imagery is about ten meters at best at a small number of sites and is more like 100 meters or more over most of

TABLE 1

COLUMBUS BASE SCENARIO

Year	-4	0 (start date + 8)	+4	+8	+10	+12
Mission	precursor	1st Landing	2nd Landing	3rd Landing	4th Landing	5th Landing
Crew (total/land/stay)	unmanned	6/4/0	6/4/0	6/4/0	12/12/0	15/15/10
Landing Site	orbital	A	B	C	one of A, B or C (BASE)	Columbus Base
Mars Optical Comsat	*	*	*	*	*	*
EVA Rover (10 km range)		*	*	*	earth-mover	agritractor
Shirt-Sleeve Rover (100 km range, the "Winnebago")					*	*
RPV (mars airplane)					*	*
Manned Scout Rocket (uses ISPP, 20,000 km range)						(uses ISPP) *
ISPP		Test/PP	Test/PP	Test/PP	Production	Modification & Production
ISWP		Test/PP	Test/PP	Test/PP	Production	Modification & Production
Permanent Shelter		Test/RS	RS	RS	HAB Production	Habitat Production
Agriculture		Test	Test	Test	Test Enclosure	
SSTT Network (#/dimension)		4/5 km	4/5 km	4/5 km	20/30 km	50/10,000 km

ISPP = In situ propellant production

ISWP = In situ water production

SSTS = Surface science telemeter station (station number/network dimension)

PP = Pilot production

RS = Radiation shelter

the planet. If the Viking data is the best that we have as the basis for picking landing sites (the Mars Observer is not planned to include high resolution imagery), the first landing crew could well encounter house-size hazards too extensively distributed to be evaded using the few kilometer lateral hovering capability of a landing craft. This possibility seems like an unnecessary risk to me. It is true that the first crew could scrutinize the surface from orbit and select a landing site at that time, but I argue that it would be safer and more productive to extensively preplan and prioritize a number (say, ten) of landing sites on the basis of high resolution images and then have the crew validate and possibly reprioritize these sites based on orbital observation.

I propose that the mission also install a very high data rate (laser) communication satellite in Mars orbit to transmit the large amount of data required by the high resolution imagery. This comsat should be designed for a long operational life so that it can be used by all of the subsequent manned missions. It is highly likely, in my opinion, that TV coverage of the the manned missions will be a required feature and this plus the large amount of scientific and operational data transmission will necessitate an optical bandwidth communication capability.

Finally, it is possible that the Martian moons Phobos and Deimos contain relatively large amounts of water and carbonaceous materials [5]. If so, these materials represent important resources that could be processed for use by the missions. For example, rocket propellant or life support consumables could be manufactured to lessen the amounts needed to be transported from Earth with potentially very large savings. This possibility and it's economic exploitation forms the basis of the second scenario presented below. Consequently, I propose that the precursor mission also rendezvous with one or both of the moons and determine with certainty their compositions.

First Landings

As noted above, I propose that the first three manned landings be at three different sites preselected using the precursor results and validated by a crew upon arrival in Mars orbit. The sites will be selected on the basis of a balance of scientific and operational criteria. For example, a landing on Tharsis or even Olympus Mons would be exciting and valuable from a scientific viewpoint, but the thinness of the already tenuous Martian atmosphere would probably preclude in-situ propellant or water production (ISPP, ISWP) and increase the severity of cosmic ray and solar flare irradiations. Thus, some compromise will be established for initial landing sites after extensive analysis of all mission goals.

I envision a crew size of six, four of whom will land on the Mars surface and two of whom will remain in orbit. The total time in the vicinity of Mars will be about two months with part of the crew on the surface for at least six months. The orbital crew will monitor and support the surface activities, perform orbital scientific investigations of Mars, and visit and investigate the Martian moons with probable installation of pilot processing plants there. The prime goal of the surface crew will be to intensively investigate the immediate vicinity of the landing site with the aid of an extravehicular activity (EVA)-type rover vehicle similar to the Apollo rover vehicle. Detailed proposals for surface science investigations are presented elsewhere [6]. An important operational as well as scientific goal will certainly be to determine the presence or absence of water within the Martian surface materials down to depths of several kilometers. The presence of exploitable quantities of water will be a prime selection factor for siting of the permanent base, and it is presumed that with three different landing sites there is a reasonable likelihood of success in attaining this important goal.

In addition to the scientific investigations, the crew will establish important operational assets and carry out investigations in addition to the water evaluation. The crew will construct a radiation shelter, possibly using explosive tunnel driving techniques [7], after first performing some excavation and basic rock mechanics tests. Tests will be performed to evaluate in-situ propellant and water production techniques with actual small scale production on the second or third landings, if possible. Tests will be performed to evaluate the possibility of growing plants for human consumption, since it will be desirable to gain as much self sufficiency as possible by the time the permanent outpost is established.

The surface crew will return scientific samples and data plus operational data and experience, and leave behind a radiation shelter, rover, scientific equipment, and possibly propellant and water manufacturing facilities to form the start of a permanent base (if the site is selected) or a "line shack" if the site is revisited later for scientific purposes.

<u>Establishing the Base</u>

On the fourth manned mission, an expanded crew of twelve will land at one of the previously visited sites to begin construction of a permanent base and to expand the scientific exploration in the vicinity of the base. A second EVA-type rover will be landed that is specially designed for "earth" moving activities. This will be used to expand the surface facilities at the base. The originally constructed radiation shelter will be expanded and modified for permanent habitation. A test enclosure will be constructed to further evaluate agricultural techniques. Sustained production of fuels and water will begin and inventories will be accumulated.

Scientific exploration of the region around the base will expand and become more sophisticated with the aid of a shirt-sleeve roving vehicle with a range of about 100km [6]. In addition, long range geophysical and meteorological investigations will be aided by deployment of a remotely piloted airplane [8] that has a range of several thousand kilometers.

<u>Columbus Base</u>

The fifth landing will occur at the new base some twelve years after the initial manned landing on the surface. Fifteen people will land along with additional vehicles, equipment, supplies, and, by this time if not before, a nuclear power plant. Habitats will be expanded along with ISPP, ISWP, and food production. The new vehicles will use ISPP and the old vehicles will be modified to do so. A new, long range vehicle will be introduced that can reach any point on the planet with men and equipment. This will be something like a manned scout rocket or air vehicle.

At this point, about a third of the crew will return to Earth and the rest will stay over until relieved by a resupply ship at the next opportunity. The permanent scientific exploration and exploitation of Mars will then begin.

PHOBOS STATION SCENARIO

An alternative approach to direct Martian surface exploration emphasizes development of Mars orbital infrastructure before extensive surface activities are attempted. I call this approach the "Phobos Station" scenario. The idea behind this approach is that the Martian moons may contain very valuable resources whose exploitation will be the driver for missions to Mars based on a largely economic rationale as opposed, or in addition, to scientific and political reasons [3]. If the suggested carbonaceous chondrite compositions of Phobos and Deimos are correct, then they may contain as much as 10^{15}kg of water [9] plus large amounts of other volatile elements such as C, N, and alkali metals. All of these volatile elements are rare on the Moon, but are essential ingredients of future large-scale space industrial activities. Furthermore, delivering these valuable resources to the Moon or lunar orbit from Mars is only half as expensive, in delta-V terms, as supply from Earth [10] which is the main alternative source besides Earth-crossing asteroids. The latter are more difficult to visit for sustained periods and do not appear to have any advantages over the Martian moons as sources of volatiles for near-Earth space industrial activities. Therefore, I believe that these facts may form the basis of an economic rationale for manned Mars missions that is equally, if not more, compelling than scientific curiosity.

Objective

The overall objective of this scenario is the establishment of the infrastructure to support the economic development of Phobos/Deimos resources. This Mars-orbital infrastructure would then be a way-station for manned scientific exploration of the Martian surface.

Time-line

The missions begin with an unmanned precursor to Mars orbit similar to that proposed in the Columbus Base scenario (Table 2). However, in this case the emphasis will be placed on observation and sampling of the Martian moons with essentially no activities aimed at the Martian surface. Two years later, the first manned mission to the Mars vicinity will be launched. This mission will have as it's goals the detailed scientific investigation and resource assessment of the Martian moons, and the establishment of pilot ISPP and ISWP plants on or near Phobos. Two years after this, an unmanned mission will be launched to position

TABLE 2

PHOBOS BASE SCENARIO

Year	-2	0 (start date + 8)	+2	+4	+6 to +10	+12
Mission	precursor	Ph/D landing	precursor	Phobos Station	1st Surface Landings	Surface Base
Crew (total/land/stay)	Unmanned	6/6/0	Unmanned	6/0/6	12/4/0	15/15/10 (c)
Landing Site	orbital	Ph/D		Permanent Orbital Station	multiple sites plus teleoperated vehicles from	Surface Base
Mars Optical Comsat	*	*	*	*	*	*
EVA Rover (10 km range)					*	additional vehicles
Shirt-Sleeve Rover (100 km range, the "Winnebago")					*	*
RPV (mars airplane)					*	*
Manned Scout Rocket (uses ISPP, 20,000 km range)						(uses ISPP) *
ISPP		Test/pp(a)		Production	Production (d) and Export	Surface Production
ISWP		Test/pp(a)		Production	Production and Export	Surface Production
Permanent Shelter			Station Elements		RS	Habitat
Agriculture		Test(b)		PP	Test Enclosure	Production
SSTT Network (#/dimension)					4/5 km	50/10,000 km

ISPP = In situ propellant production
ISWP = In situ water production
SSTS = Surface science telemeter station (station number/network dimension)
PP = Pilot production
RS = Radiation shelter
RPV = Remotely piloted vehicle
Ph/D = Phobos and/or Deimos

(a) On Phobos
(b) On spacecraft using Phobos resources
(c) Permanently manned surface base, may exchange crew with Phobos Station
(d) Production for export to lunar space, and LEO with logistic support for surface landings.

near Phobos* the structural and support elements of a permanent, artificial gravity habitat from which mining and processing of volatiles from Phobos will be controlled. This large cargo can perhaps utilize advanced propulsion capabilities such as nuclear-electric low thrust propulsion which would appear to be ideally suited to this type of freight mission. At the next opportunity, a crew will be sent to assemble and begin operation of the station. Volatiles mining and ISPP production will then be established and expanded over the next few years with crew rotations and resupply at each opportunity. By year +8 or +10 I expect that substantial, essentially routine, unmanned tanker traffic would be established from Phobos Station to lunar space or surface and thence to low Earth orbit. However, before then, probably by +6, the infrastructure would be in place at Phobos Station from which to launch the first Mars surface explorations. With the aid of Phobos Station, the surface exploration could develop at a more rapid pace than with the Columbus Base approach, probably by means of unmanned, teleoperated roving vehicles. By +12 (the same time as for the Columbus Base scenario) it should be possible to establish a permanent manned base on the Martian surface from which to explore the planet. From then on, exploration and development should proceed similarly although the added benefit of the Phobos Base facilities, and resources would seem to offer an advantage for continued development compared to the direct approach in which the surface landings come first.

<u>Establishing a Manned Orbital Station</u>

I will not discuss in detail the unmanned precursor or manned surface landings. These should be similar to those proposed for the Columbus Base scenario and any differences can be seen in Table 2. Instead, I focus on the one element that is decidedly different in this approach - the manned, artificial gravity, Mars orbital station. I envision the station as a rotating structure approximately 600m in diameter providing

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*I have assumed Phobos as the site for an orbital station, but an equally good case can be made for Deimos. Phobos is closer to the Martian surface which would facilitate activities there. On the other hand, Deimos is more loosely bound so that reduced delta-V's would be required relative to Phobos for frequent interplanetary insertion maneuvers. Some balance of these and other issues will need to be struck before a final decision on the orbital station location can be made.

about 1/3 Earth gravity at 1RPM. This gravity value is chosen to be similar to that of the Martian surface so that crews adapted to the station would also be adapted to Mars. Initially, the station should adequately house about 6 people and be expandable to a crew two or three times that amount. The primary function of the station will be to provide a habitat for personnel engaged in operating the mining and refining operations on Phobos and, eventually, Deimos (see footnote, p.9). Secondarily, the station will function as a research station for remote investigation of the Martian surface and as a staging base for manned expeditions to the surface. I expect that teleoperation of vehicles and facilities on the Martian surface will be quite effective and will strongly supplement, but not replace, manned operations on the surface.

SUMMARY

I have outlined two approaches to the establishment of a permanent manned base on the Martian surface. If achieving scientific and political (i.e., being the first to land men on Mars) goals are paramount, then the direct mission scenario I call "Columbus Base" (or something similar to it) seems to be the most logical. If, driven by space industrialization in the 21st century, the economic demand for the extensive volatile element resources probably contained in the Martian moons becomes as strong as I think it will, then the second scenario I propose looks more appropriate and effective. In this "Phobos Station" approach, manned exploration of the Martian surface is delayed somewhat in order to develop the infrastructure needed to exploit the Martian moon resources. However, once surface landings and scientific investigations begin, they appear to do so from a much stronger infrastructure base and thus this may be the more powerful and fruitful approach in the long run.

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MISSION & SPACE VEHICLE SIZING DATA FOR A CHEMICAL PROPULSION/AEROBRAKING OPTION

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ABSTRACT

This paper presents sizing data for various combinations of Mars missions and chemical-propulsion/aerobraking vehicles. Data is compared for vehicles utilizing opposition (2-year mission) and conjunction (3-year mission) trajectories for 1999 and 2001 opportunities, for various sizes of vehicles. Payload capabilities for manned and unmanned missions vehicles and for propulsive-braking and aerobraking cases are shown. The effect of scaling up a reference vehicle is compared to the case of utilizing two identical vehicles, for growth in payload capability. The rate of cumulative build up of weight on the surface of Mars is examined for various mission/vehicle combinations, and is compared to the landed-weight requirements for sortie missions, moving-base missions, and fixed-base missions. Also, the required buildup of weight in low Earth orbit (LEO) for various mission/vehicle combinations is presented and discussed.

REFERENCE VEHICLE

A typical chemical propulsion/aerobraking Space Vehicle (SV) for a manned Mars landing mission is shown in Figure 1, along with the key assumptions and parameters associated with the mission. The vehicle utilizes cryogenic propellants in its propulsive stages, aerocapture at Mars and Earth, and aerobraking plus propulsive burns during the descent to the Martian surface. The mission for which this vehicle is sized is an opposition mission which arrives at Mars in 2001. The total mission time is 780 days, including a stopover time of 60 days at Mars. In this mission, three of the crew members remain in Mars orbit, and the other three descend to the surface. This mission and vehicle are described more fully in references 3, 4, and 5.

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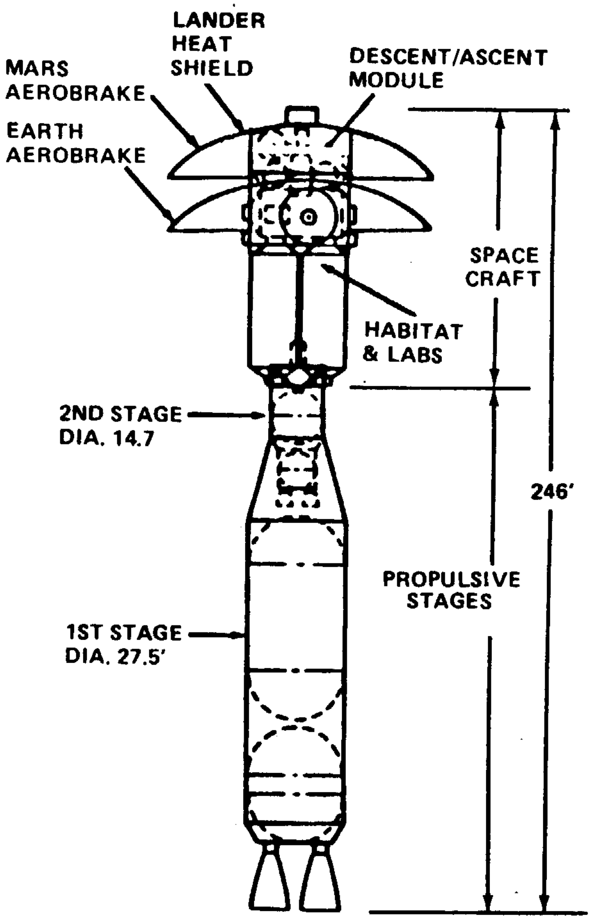
FIGURE 1. TYPICAL CHEMICAL (CRYO) PROPULSION/AEROBRAKING SPACE VEHICLE CONCEPT FOR MANNED MARS MISSION



KEY MISSION & VEHICLE PARAMETERS

OPPOSITION CLASS MISSION
MARS ARRIVAL IN 2001

TOTAL MISSION DURATION 780 DAYS

MARS STOPOVER TIME: 60 DAYS

AEROBRAKING & PROPULSIVE DESCENT AT MARS

TOTAL VEHICLE WET WEIGHT: 1.57M LB.

1ST & 2ND STAGE CRYO PROPELLANT WEIGHT: 1.14 M LB.

AEROCAPTURE AT MARS

AEROCAPTURE AT EARTH

CREW SIZE: 6 (3 TO SURFACE)

USES SS HAB, LAB, & LOG MODULES

1ST STAGE IS ET DERIVATIVE, W/2 EACH STME CRYO ENGINES

2ND STAGE IS OTV, W/2 EACH RL 100 CRYO ENGINES

LANDER HAS 3 EACH NEW ENGINES (LOX/MMH)

SPACE VEHICLES SIZING SENSITIVITIES

Using this mission and vehicle as a reference, parametric data have been developed for various other missions and vehicles. The left side of Figure 2 shows how the SV low Earth orbit (LEO) weight would change as this mission and vehicle are scaled from a 2-year to a 3-year mission. The data shown for the crew consumables, science equipment, and spacecraft subsystems is shown as a linear function of time, and is independent of the mission date. The additional science equipment would have to be provided in order to make better use of the additional time at Mars, and a rough estimate of weight for this equipment has been made here. Spacecraft subsystems weight would increase as shown to accommodate the increased volume of consumables and experiments and to provide additional systems lifetime. The total SV weight is dominated by the weight of the propulsive stages, so the increase in spacecraft weight is more than offset by the decrease in propulsion weight for the 3-year mission, compared to the 2-year mission.

In actuality, there is no continuum in mission possibilities between the 2-year and the 3-year data points. The 2-year data point corresponds to an opposition-type mission arriving at Mars in 2001, which has about a 60-day stopover time; the 3-year data point corresponds to a conjunction-type mission arriving at Mars in 1999, which has a stopover time of about 1 year. There are no realistic choices of missions in the region between these data points. The propulsive vehicle weights vary considerably from opportunity to opportunity, as discussed in reference 1, with the opposition-class missions varying much more than the conjunction-class missions. The conjuction missions require less propellant than the opposition missions. More discussion on these is provided in references 3 and 5.

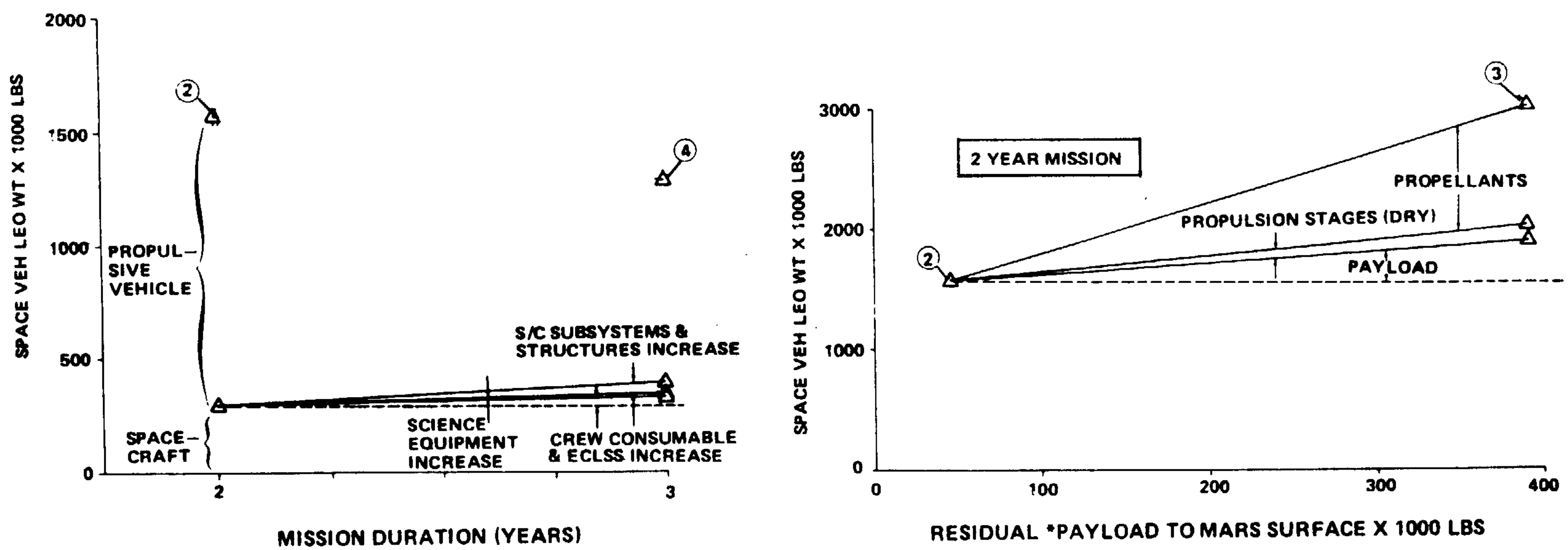
The right-hand side of Figure 2 gives an idea of the sizing sensitivity associated with scale-up of the reference vehicle to a vehicle with greater payload capability. In this case, the term "residual payload" implies the payload delivered to the surface of Mars and left there (excludes the ascent stage on manned landing missions). There is a pound-for-pound increase in the SV LEO weight for each payload pound added to the SV. In addition, the weight of the propulsive stages must increase as shown to deliver the additional payload weight. Increasing

FIGURE 2. SPACE VEHICLE SIZING SENSITIVITES

- CRYO PROPULSION
- AEROBRAKING FOR MARS CAPTURE/DESCENT & EARTH CAPTURE ("ALL – AEROBRAKING")
- 6–MAN CREW

SPACE VEHICLE SENSITIVITY TO MISSION DURATION

SPACE VEHICLE SENSITIVITY TO PAYLOAD GROWTH



*PAYLOAD DELIVERED TO MARS SURFACE AND LEFT THERE

◯ DATA POINTS FOR BARS ON SUBSEQUENT BAR CHART the residual payload to the surface of Mars by a factor of 8 only costs an increase in SV LEO weight of about a factor of 2, providing a net 4-to-1 benefit-to-cost ratio. Flying 2 of the initial SV's would result in only a 1 to 1 ratio; hence, a growth version of the SV's appears to be much more efficient than 2 SV's for transporting payload to Mars. The circled numbers denote data points corresponding to bars on Figure 3.

MISSION/VEHICLE COMPARISONS

Figure 3 is a bar graph showing the total SV LEO weight for several types of SV's across a large portion of the spectrum of possibilities for cryo propulsion systems.

Bars #1-3 are for 2-year missions and #4 is for a 3-year mission. Bar #1 is for an "all propulsive" SV (although aerobraking is used here during part of the Mars descent), and bars #2-4 are for "all-aerobraking" SV's (although retro propulsion is used here during the final descent to Mars). Bars #1 and #2 show the savings on propulsion system weight which is possible with an aerobraking vehicle compared to an all-propulsive vehicle, for the same size payload.

Bar #2 is for the reference SV mentioned previously (Figure 1). This bar corresponds to the 2-year data point in Figure 2 (left-hand side of both graphs), and bar #4 is for the 3-year data point (right-hand side of the left graph) on Figure 2. Bar #3 is for the growth version of the 2-year SV shown in the right-hand graph of Figure 2.

Each bar is divided into subelements to show which portion of the total weight represents the SV propulsion stages' dry weight, propellant weight, and payload (spacecraft or other) weight. Two cases are shown for the residual payload weight for each bar (residual payload weight here means weight delivered to and left on the Martian surface). One case ("A") is representative of payload for a manned mission, wherein additional elements and propellants must be provided to return the crew to Earth. The other case ("B") is a preliminary estimate of payload for an unmanned one-way delivery mission, which allows greater payload weight to be delivered and left on the surface, since no crew or equipment have to be returned to Earth. The unmanned payload numbers represent merely a estimate (essentially the total spacecraft weight from the manned landing cases), but these numbers are believed to be fairly accurate. There are intermediate cases, not shown, of missions having the spent propulsive

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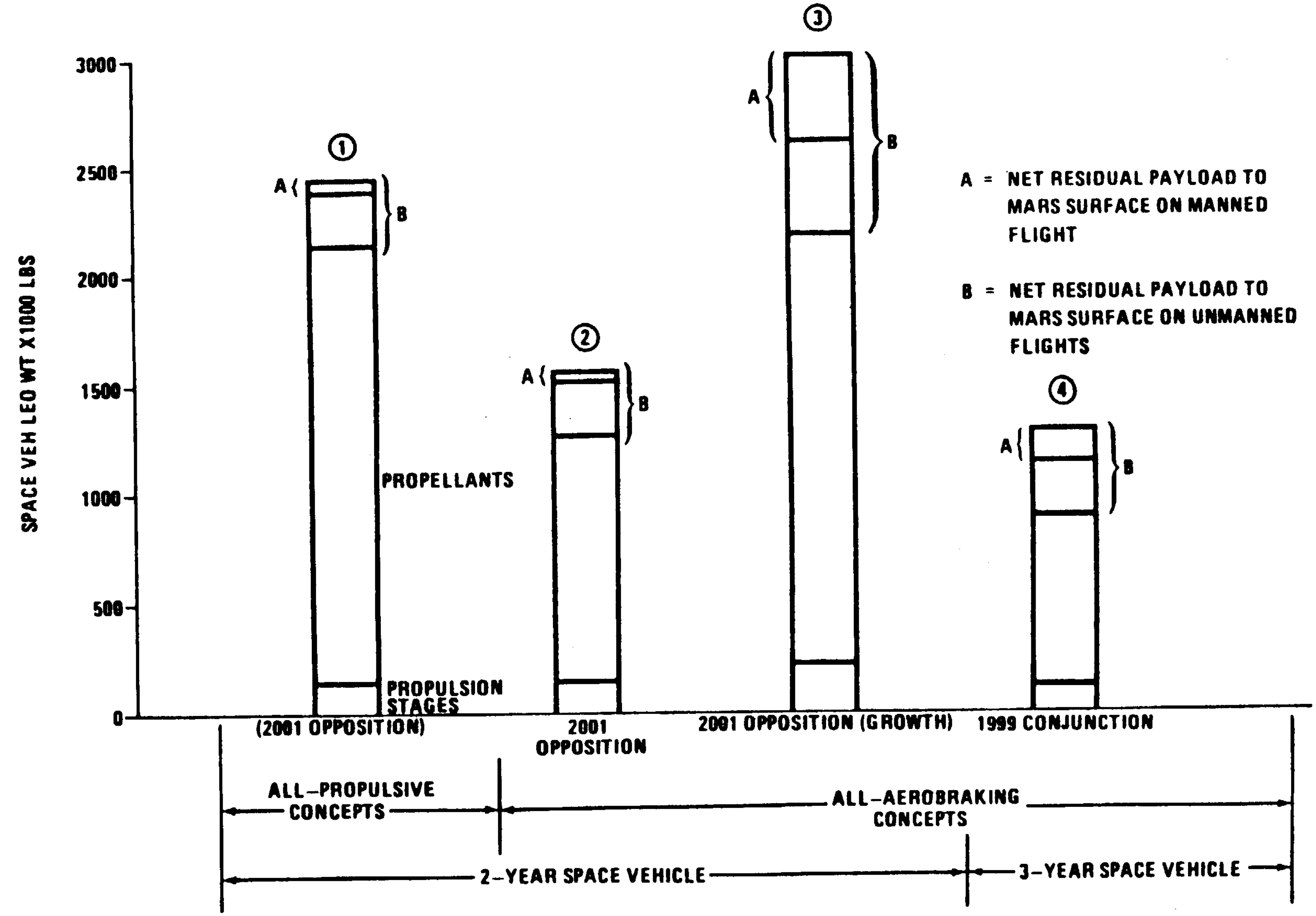
FIGURE 3. SPACE VEHICLE WEIGHT BREAKDOWN (CRYO PROPULSION)



stages returned to Earth for reuse. This is an issue of considerable interest to NASA, and further study must be done to determine its cost-effectiveness.

CUMULATIVE BUILDUP AT MARS

Figure 4 shows the potential cumulative buildup of weight of equipment left on the surface of Mars for manned and unmanned missions, using different propulsive vehicles of the types shown on previous charts. The circled numbers refer to the bars on Figure 3, and indicate which type of vehicle and mission was used for each line of Figure 4. The degree of improvement in buildup rate can be seen for cases using growth versions of the propulsive vehicle compared to cases using two vehicles, and compared to cases using just the basic propulsive vehicle. Assumptions were made here that launches occur at every opportunity and that propulsion requirements for every opportunity are the same. As previously mentioned, the latter assumption is not the true situation, and considerable differences may exist between opportunities. Hence, the launch vehicle sizes and/or payload capabilities would vary from one opportunity to another, and the curves would not be as smooth as shown. Trends, however, should be roughly the same. The horizontal lines shown on Figure 4 represent amounts of weight necessary to be delivered to Mars and left there to achieve weight buildups equivalent to those required for 5 different types of bases, as identified in reference 6. As can be seen, the manned landing case which uses the basic propulsive vehicle and the case which uses 2 vehicles both require a signficant number of missions before meeting the required levels of buildup for bases. The growth SV and/or combinations of manned and unmanned launches allow implementation of the bases in much more reasonable time spans.

An example of the variation in overall SV LEO mass from one opportunity to another (over different years than those discussed thus far) can be seen in Figure 5, which plots all-propulsive vehicle data from reference 1. The corresponding variation in mission time for those years is shown in Figure 6.

CUMULATIVE BUILDUP IN LEO

Figure 7 is similar to Figure 4, except that it shows the cumulative weight buildup required in LEO to accomplish the launches to Mars for the mission and vehicle options previously mentioned. Here, the

3185–85

FIGURE 4. CUMULATIVE BUILDUP OF PAYLOAD WEIGHT ON MARS SURFACE

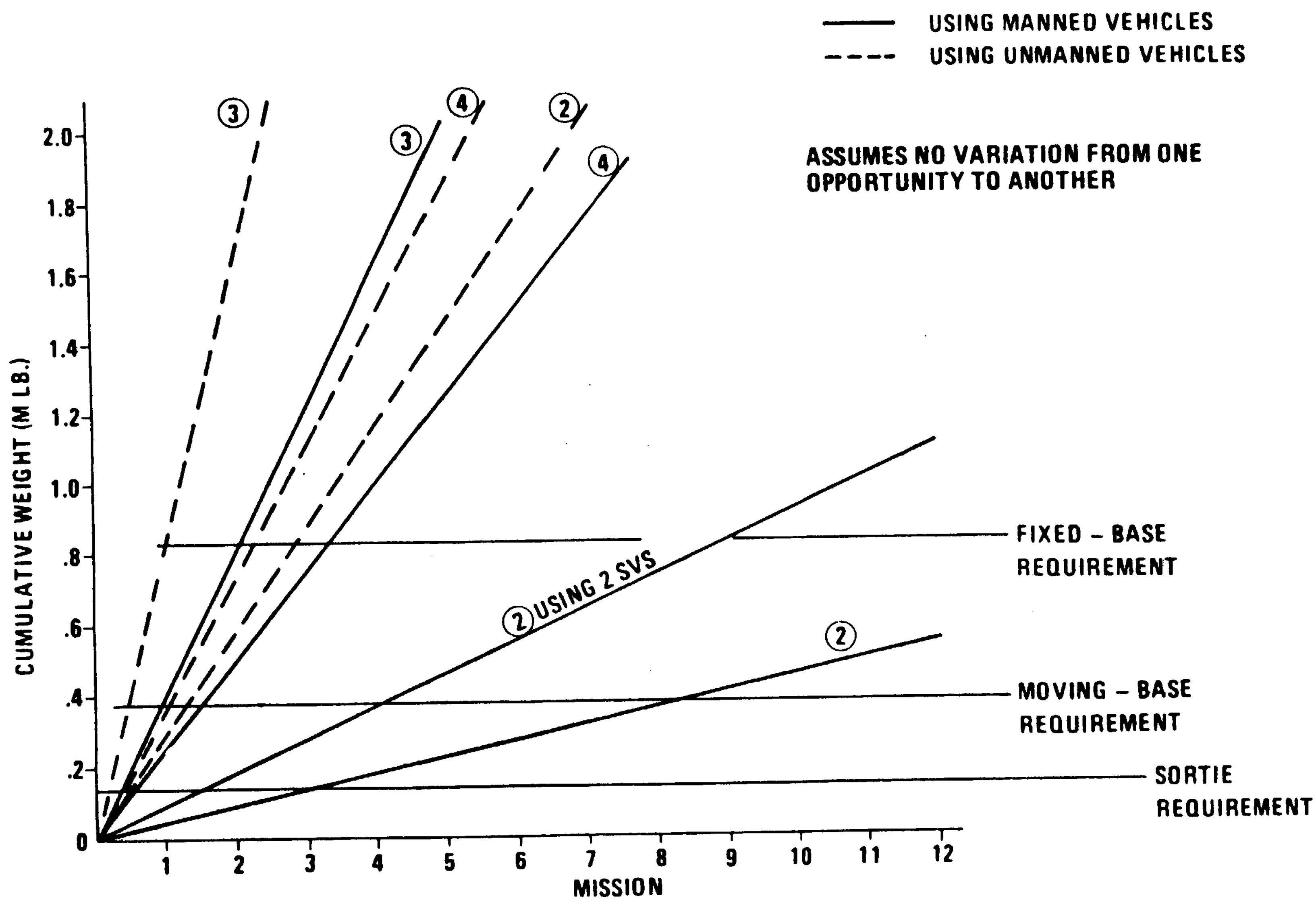


FIGURE 5. TYPICAL LEO MASS REQUIREMENTS FOR MARS MISSIONS

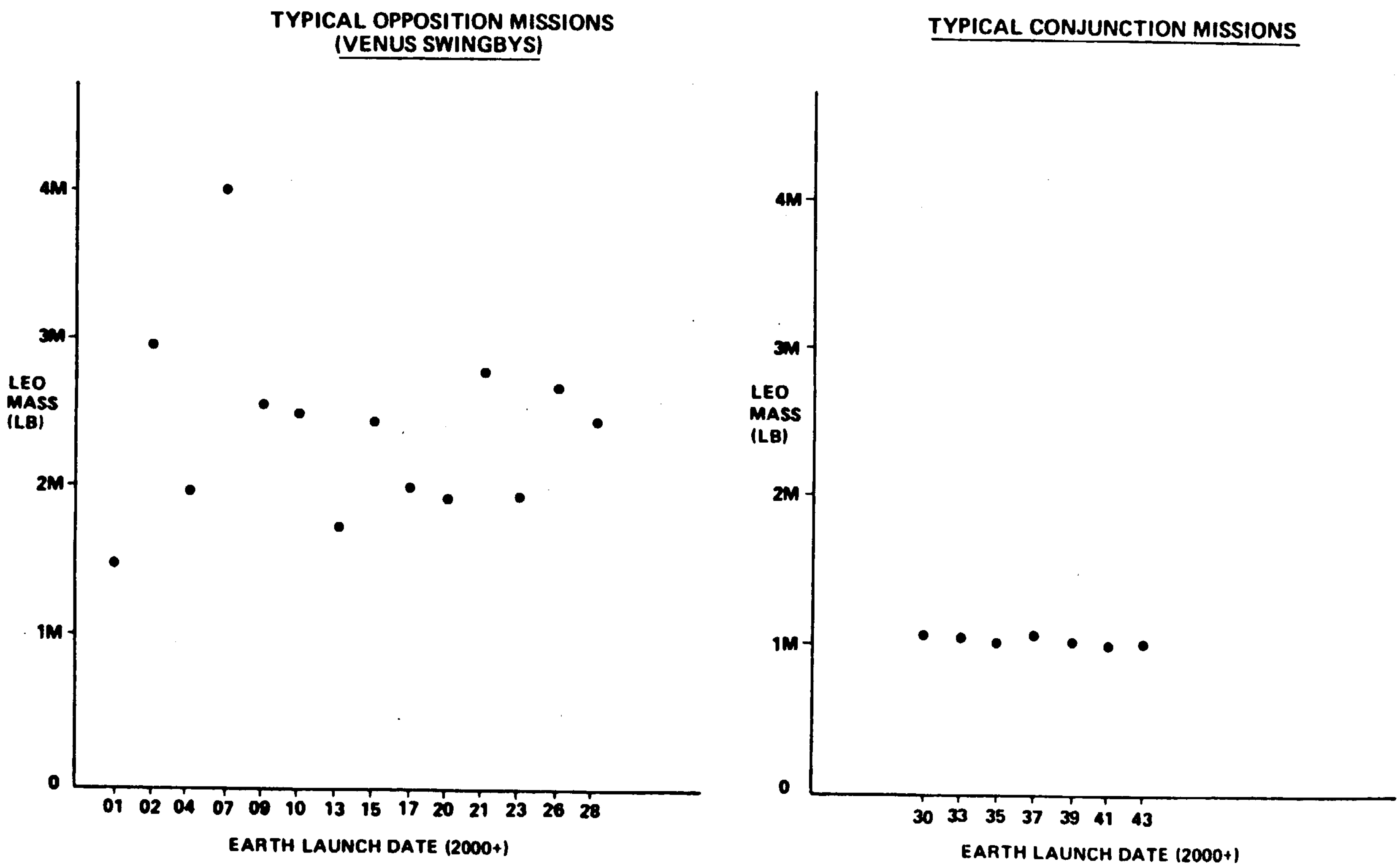


FIGURE 6. TYPICAL MARS MISSION DURATIONS

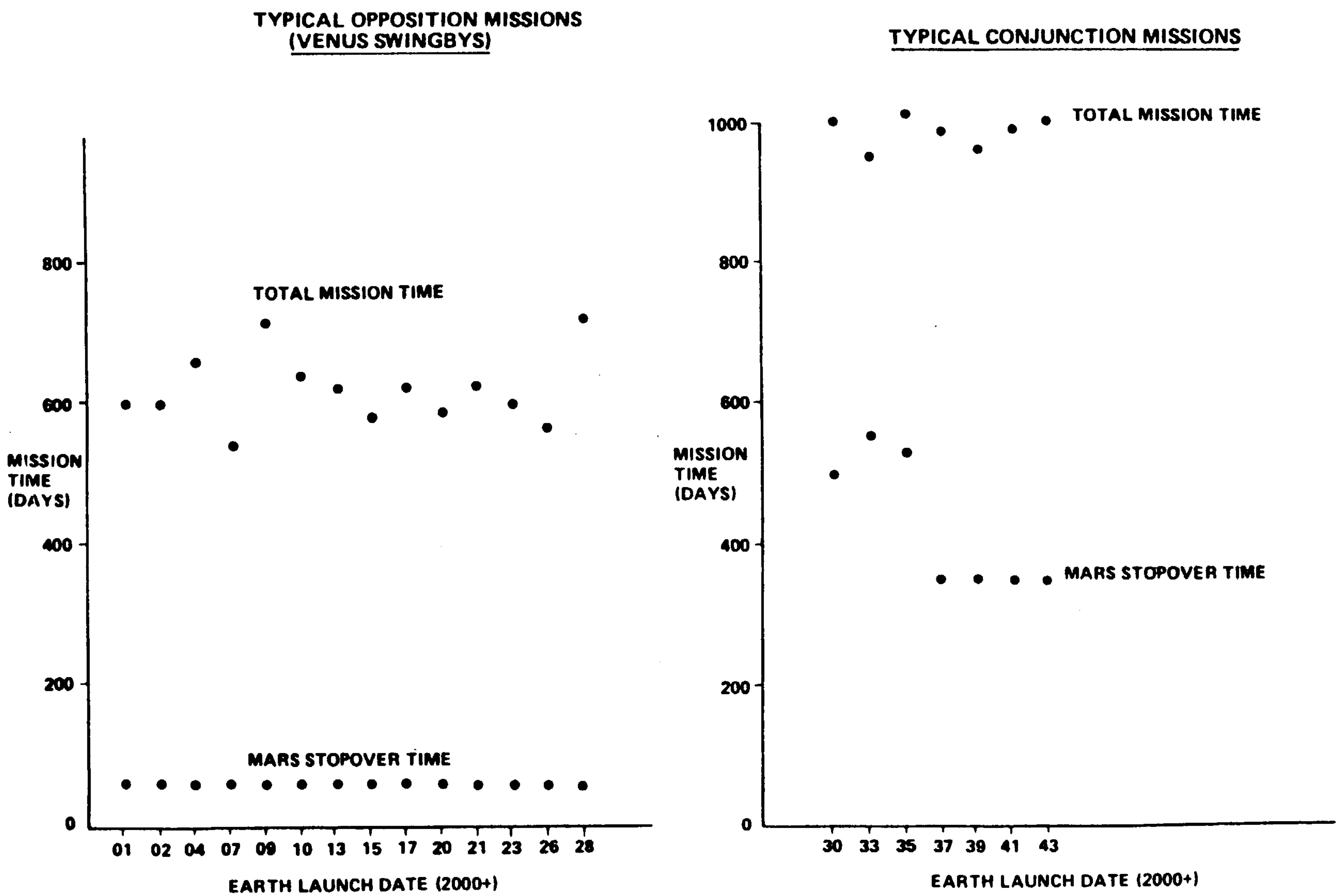
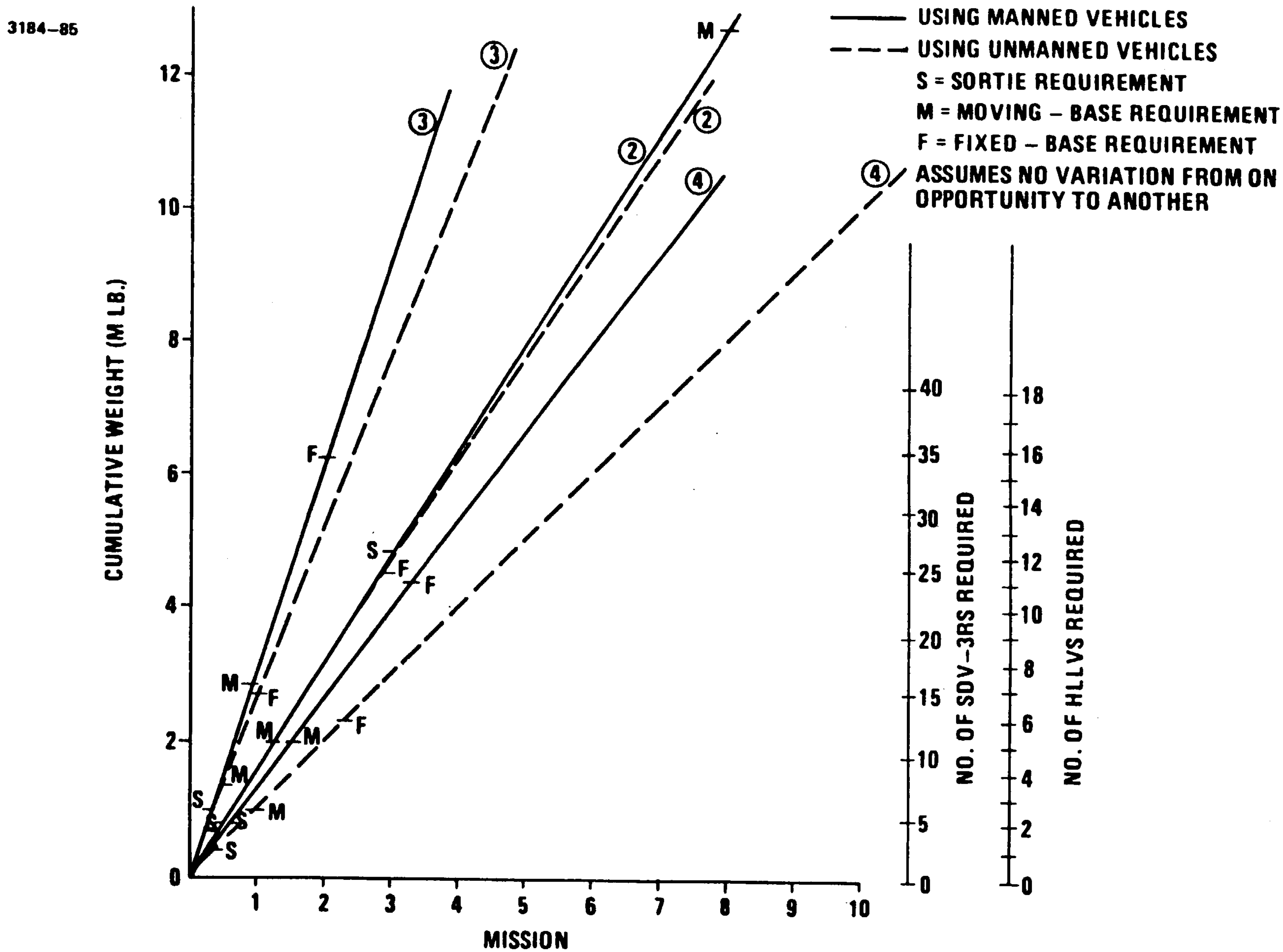


FIGURE 7. CUMULATIVE BUILDUP OF LEO WEIGHT



effect can be seen of the more efficient trajectory of the 3-year conjuction mission (curve #4) compared to the 2-year opposition mission (curves #2 and 3). As discussed in references 4 and 7, both types of missions will probably be desired as part of a Mars program. The ordinate axes on the right-hand side of this chart show the quantity of Shuttle-Derived Vehicles (SDV's) or Heavy-Lift Launch Vehicles (HLLV's) required, depending on which of these concepts is used. Here, the SDV-3R and the HLLV of the type defined in reference 2 were assumed. These vehicles would have launch capabilities of about 182K pounds and about 400K pounds, respectively, to the Space Station (SS) orbit (assumed to be 270 nautical miles altitude and 28.5 degrees inclination). No detailed "capture" analysis was done here, so the data shown on these axes may be overly optimistic in terms of estimates of packaging efficiency in the SDV-3R and HLLV.

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MISSION AND SPACE VEHICLE CONCEPTS

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ABSTRACT

This paper discusses a number of top-level considerations which affect Mars and vehicle selection. Indications are provided of the nature and severity of the impact of these considerations on missions and vehicles. The paper identifies and discusses various types of missions, such as Mars fly-bys, Mars orbiting and landing missions, and missions to the moons of Mars. Mission trajectories and opportunities are discussed briefly.

The paper also discusses the different types of vehicles required in a Mars program. Discussion includes several potential Earth-to-Orbit (ETO) vehicles, Mars surface vehicles, and 2 types of Orbit-to-Orbit (OTO) vehicles. Indications are provided as to preference for some of the concepts discussed.

OVERALL CONSIDERATIONS

The exploration of Mars will require multiple manned (and/or unmanned) missions. Furthermore, the utilization of Mars as a science outpost, a resource production site, or as a site for colonization experiments, etc., adds a significant level of increase in quantity and sophistication of missions. The initial Mars mission usually receives the greatest interest and definition activity, but this mission should not be considered an end in itself. The technology and design concepts selected for the initial mission should be chosen so as to allow their utilization and evolution to occur in subsequent missions.

Some of the key top-level considerations which will determine the nature of mission and vehicle concepts for a manned mission to Mars are 1) the desired launch timeframe, 2) the desired stopover time at Mars, 3) the nature and location of the science to be conducted, 4) design implications implied by the physiological effects of long-term zero-g environments, 5) contamination considerations, and 6) cost.

Launch Timeframe

The two launch timeframes of interest for study activities have been specified broadly as an "early" (pre-2000) timeframe and a later (post-2000) timeframe. The main effects of specifying the earlier launch timeframe are to constrain technology selection to that which is more near-term and to restrict more severely the options for shaping the cost envelope. Also, the scope and complexity of the science associated with the initial mission would probably be more limited if the mission were in an early timeframe rather than in a later one. For one thing, earlier technology would be less efficient, making weight more critical and hence, not as much science (or other) equipment could be transported. Also, any international prestige factor ("race to Mars" context) associated with an early mission might be a forcing function towards ensuring that mission (and science) complexity remained low, lest it jeopardize the schedule.

Mars Stopover Time

Within either of the broad launch timeframes, there are only a limited number of practical opportunities for launch, due to the severity of the energy requirements for a launch at any but the optimal planetary alignments (References 9 & 10). These practical opportunities occur roughly every 2 years, but the energy requirements can vary by a factor of 2 to 1 between successive opportunities for some trajectories. Hence, selection of a specific launch date can have significant implications for sizing of the propulsive vehicle. The vehicle size is fairly sensitive to launch window size, with a 30-day launch window requiring about a 6-10% increase in propellant, compared to a 10-day window.

The choice of stopover time at Mars is pre-set by the selection of the trajectory to be used, and vice versa. There are basically two choices of stopover times: 1) about 60 days, and 2) about a year, corresponding to total mission times of 1) about 2 years ("opposition"-type trajectory), and 2) about 3 years (conjunction-type trajectory). The wide variation in these times can have a significant effect on the mission and vehicle concepts. There are systems technologies and concepts which might be usable for a 60-day stopover, but which might not be usable for the longer stopover. The longer mission time also implies the need for greater lifetime and reliability of systems, for more expendables, and for more science equipment (to make the longer stopover productive).

Science Activity Nature and Location

The nature and location of the science activity to be conducted has a fairly significant bearing on the mission and vehicle concepts. Science activities are planned for all phases of the missions (in transit, in the Earth vicinity, in the Mars vicinity, and on the Mars surface), but that planned for the Mars surface is likely to be the most demanding and to also have the greatest implications for mission and vehicle concepts. For example, some form of surface traverse capability will be necessary for efficient exploration. Concepts vary from short-range lunar-rover-type vehicles to mobile laboratories with ranges up to hundreds of kilometers and several days' duration. The location of the desired surface science activity can vary from the polar regions to the equator, from rocky fields to sand dunes, and from mountainous regions to smooth plains. Each of these imposes some different requirements on the mission (particularly the trajectory) and on the vehicles and equipment (particularly the surface infrastructure elements). Ideally, the mission and vehicle concepts should be able to accommodate any of the desired landing locations and science activities, since separate locations will probably be desired on different missions (particularly the early missions).

Physiological Effects

Physiological considerations (particularly the long-term zero-g effects which can incapacitate astronauts) can have significant impacts on mission and vehicle concepts. Research must be done to understand more fully the physiological mechanisms involved, and to discover preventive or corrective measures. It is possible that diet supplements can offer significant help in this regard, for example, in aiding fixation of calcium in the bones. Exercise, also, will probably be part of the solution. Major questions remain, however, in regard to 1) whether there must be a gravity field provided during the long transit periods or not, 2) the level of the g-forces required, 3) the consistency of the g-forces required (constant vs. intermittent, and unidirectional vs. reciprocating), etc. The greatest impact on the vehicle design would occur if there were a requirement to spin the entire vehicle, or a major portion thereof. A lesser impact would occur if, for example, a reciprocating sled arrangement might be available for occasional astronaut use. For spinning vehicles, arrangements must be made to despin some science equipment and any vehicle systems equipment needing preferential orientations (solar arrays, radiators, antennas etc.). Some vehicle system concepts must be able to operate in the LEO environment (during assembly), in the Earth-Mars transit phase, in Mars orbit, and on the Martian surface; the g-levels vary from zero-g to about one-third g across these mission phases, even before consideration of any additional effects due to spinning vehicles. Reference 5 and papers in Section VI provide further discussion of this subject.

Contamination

Contamination considerations can be major drivers of mission and vehicle concepts. In addition to the usual concerns of contamination due to the natural and induced environments associated with the mission and vehicle, there are two special areas of concern which can have far-reaching impacts. One is the potential for biological contamination of Mars and Earth. Some of the more significant potential impacts are sterilization of equipment, use of bio-locks and facilities, and quarantine periods. The other special area of concern is the potential for radiological contamination of Earth and Mars, if nuclear power and/or propulsion concepts are used. There are reasons to believe that these concerns might not result in major impacts, but considerable attention must be given to them in future studies to further determine this. Convincing the general public of their safety is a major part of considerations in this area.

Cost

Cost will be one of the most important governing paramenters of a Mars mission. We are, in the respects of knowledge, proven technology, and flight experience, well ahead of the place where we were when we began the Apollo lunar landing program. There will be a significant base of Space Station technology, designs, hardware, and operations experience, and even an in-orbit Space Station at Earth, for potential support of a Mars landing program. Also, there will likely be an Earth-to-orbit heavy-payload-capability vehicle available for use. Many of the challenges of a Mars mission (long durations, great distances, difficult environments, and more sophisticated science requirements) will be demanding, on but by comparison to our situation at initiation of the Apollo program, they are less demanding than the challenges were then. Reference 11 discusses this subject, also.

MISSION TYPES

The simplest and nearest-term type of manned mission to Mars which might be envisioned is a manned fly-by of Mars, in which case there is no injection into Mars orbit, nor landing, of any manned elements (although unmanned probes would probably be ejected from the passing Space Vehicle (SV) to do both of these things). Such a mission could be accomplished, using then-existing technology, in the late 1990's. A short mission duration (about a year) would probably be required for such a mission. This would require a "hot" trajectory, and the total delta velocity from LEO to Mars and return would be about 13.64 km/sec. A preliminary estimate of the total SV weight in LEO (assuming cryogenic chemical propulsion) would be about 1.35M lbs., but this might be reduced by as much as 50% if mission time is extended by about 20% (these weights assume that only a small module is returned to Earth orbit).

The next easiest type of manned mission to accomplish, and one which could also be done before the end of the century, would be a manned mission to Mars orbit, with an alternate mission being a manned landing on one of the moons (Phobos or Deimos) of Mars. Practical trajectories for this type of mission fall into two categories, depending on planetary alignments: 1) conjunction-type missions, which have a total mission time of about 3 years (including a 1-year stopover), and 2) opposition-type missions, which have a mission time of about 2 years (including a 60-day stopover). Depending on the type of trajectory and the type of braking (aero or propulsive), these missions require a total delta velocity of about 4.65 to 12.53 km/sec., and a total SV weight in LEO of 1.3M (conjunction/aerobraking) to 3.6M (opposition/propulsive) on some of these missions, all habitable modules could be returned to Earth orbit for re-use.

The manned Mars landing type of mission is more complex and costly than either of the others mentioned previously, but it provides greater science return, a greater capability for buildup of Mars surface elements towards a Mars base capability, greater international prestige, etc.

This type of mission, like the others previously mentioned, could be accomplished before the end of the century. The mission trajectory and duration options would be the same as for the Mars orbit missions. The total delta velocity requirements would be about 7.2 km/sec higher than those, to effect the descent and ascent at Mars. Both oppostion and conjuction types of missions might be desireable during a Mars program, the opposition type for early low-risk missions and/or for later unmanned cargo missions the conjuction type for more extensive science/exploration and/or for Mars base activities. As mentioned previously, the energy requirements vary considerably from one opportunity to another for opposition trajecories. The 2001 opportunity (Mars arrival date) offers considerable improvement in energy requirements over earlier or later opportunities, and would be an attractive year if an early opposition mission were desired. References 9 and 10 provide more details on performance analyses of these missions.

TRANSPORTATION APPROACHES

For the initial manned mission to Mars, no matter what type of mission is chosen, it would seem that the simplest, cheapest, and most reliable way to transport the people and equipment would be to transport them all together in one vehicle. Another possibility is to utilize two or more separate vehicles which are very similar and which would travel along together; this has some advantages but also adds some complexity and cost to the mission, and so would probably be best considered for later missions. Data applicable to this concept are provided and discussed briefly in reference 1.

A variation of the multiple-vehicle, simultaneous-travel approach is to have separate vehicles for cargo and for people. Some parametric sizing data for such vehicles have been generated and are discussed in reference 1. A fourth approach is to have separate vehicles for cargo and people, but to not constrain the vehicles to travel together. This allows for utilization of a "slow freighter" cargo vehicle concept and a "fast-track" manned vehicle concept, although when practical constraints are imposed, this approach may evolve back towards the third approach. A fifth option (reference 14) is a "loop vehicle" approach, wherein a large transportation vehicle continuously traverses a loop between Earth and Mars, on a fly-by trajectory at each planet. Smaller crew and cargo "commuter" vehicles would ascend to and descend from the loop vehicle at Earth and Mars proximities. Several (3-5) of such loop vehicles might be necessary to provide adequate encounter opportunities without exhorbitant gaps in the program. One of the potential difficulties associated with this concept would be that the need to occasionally replace/refurbish systems hardware on the loop vehicle might necessitate periodically returning it to Earth orbit for a "dry-dock" period, which might cause the Earth departure dates to get out of synchronization with the planetary alignments. Rendezvous windows would also be very critical with the loop vehicle concept.

A loop-vehicle concept has been proposed for the Earth-Lunar system and was assessed briefly by MSFC (reference 15). In that case, the loop mission time is only a few days, whereas in the Earth-Mars case, loop mission times of 2 - 3 years would be minimum. Due to these longer mission times, a dry-dock operation would probably be necessary after each loop, which would necessitate having a second loop vehicle available to alternate missions with the first vehicle. In this event, the loop vehicle approach essentially evolves back to the dedicated mission approaches discussed previously.

VEHICLE CONCEPTS

The basic types of vehicles required for a manned Mars mission are an ETO vehicle, an SV, Mars surface vehicles (included as part of the SV), and OTO vehicles. The ETO vehicle is utilized to launch the SV elements into low Earth orbit (LEO) in the vicinity of the Space Station. Because of the size of the SV (greater than 1M lbs.), it will be necessary to assemble it in orbit, and a number of flights of ETO vehicles will be required to deliver it there (reference 12). An assembly system may be required for on-orbit buildup of the SV. A concept of such an element is discussed in references 2 and 5.

Earth-To-Orbit Vehicles

The Space Transportation System (STS) would be utilized for launch of the crew and some of the smaller elements of the Mars SV. ETO's of the proposed Shuttle-Derived Vehicle (SDV) class (<200K lbs. of payload to LEO) and the Heavy Lift Launch Vehicle (HLLV) class (about 400-500K lbs. of payload to LEO) would be candidates for Mars missions. These have been studied extensively by MSFC and others for a number of years and a considerable amount of work is still in progress in this area. Reference 3 provides some updated data on vehicles. Figure 1 shows the concepts which were utilized in this study as typical ETO's. More than likely, some such vehicles will already exist, having been developed by NASA or DoD (or jointly) by the time frame being discussed for Mars missions.

Space Vehicle

The Space Vehicle as discussed herein is the vehicle which travels to Mars. Figure 2 shows a typical SV. It consists of a Transportation Vehicle and a Spacecraft. Their key elements and different options for each are discussed briefly below.

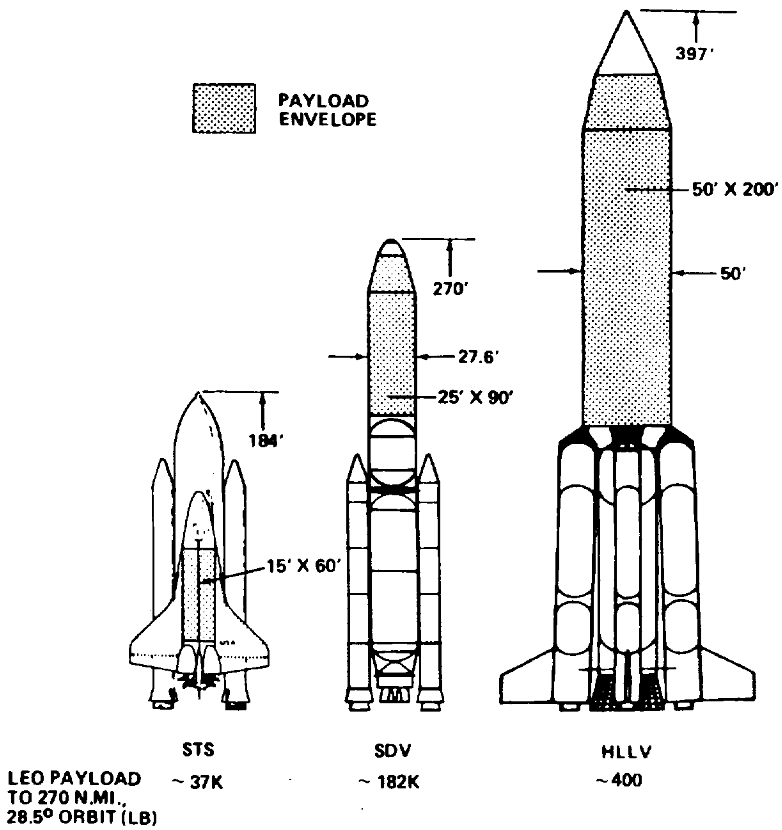
Transportation Vehicle

The types of propulsion which have most often been suggested are chemical (cryogenic, liquid storable, or solid storable), ion-drive (solar-electric or nuclear-electric), nuclear-thermal, solar sail, and hybrids of these. Each of these has been studied in the past, and a discussion and comparison of some of them is provided in reference 4 and in several papers in Section II of this report. Chemical propulsion with aerobraking is presently the most developed technology, and would probably be the choice for an early Mars mission. More data and discussion are provided on chemical propulsion concepts than on others in this paper.

The very-low-thrust systems (nuclear-electric, solar-electric, solar sail, etc.) can spiral out of LEO, given sufficient time (months), but they spend a significant amount of time in the trapped radiation belts, in addition to adding significantly to the mission time. This approach would not be acceptable for manned travel. Even for "cargo ships", the radiation is detrimental to some systems hardware, such as solid state electronics and solar arrays (if used). Shielding of sensitive systems against trapped radiation would have to be provided in the very-low-thrust systems' designs. Practical consideration of very-low-thrust systems should probably be as a part of a hybrid system, with chemical stages used to deliver the crew to Earth-departure nodes (such as Earth-Moon libration points) beyond the belts. Nuclear-thermal systems (such as the NERVA) several standpoints, but their development

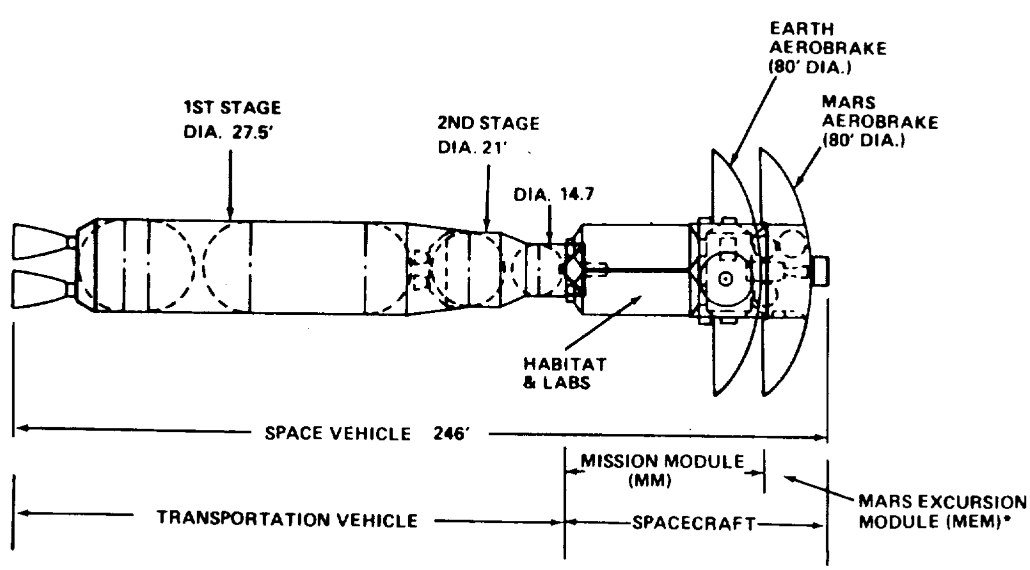
4289–85

FIGURE 1. TYPICAL EARTH–TO–ORBIT VEHICLES



4585–85

FIGURE 2. TYPICAL CHEMICAL PROPULSION/AEROBRAKING SPACE VEHICLE FOR MANNED MARS MISSION 2001 OPPOSITION



* CONTAINS ASCENT/DESCENT PROPULSIVE SYSTEMS FOR MARS LANDING

appear to be further downstream and more costly than chemical propulsion systems.

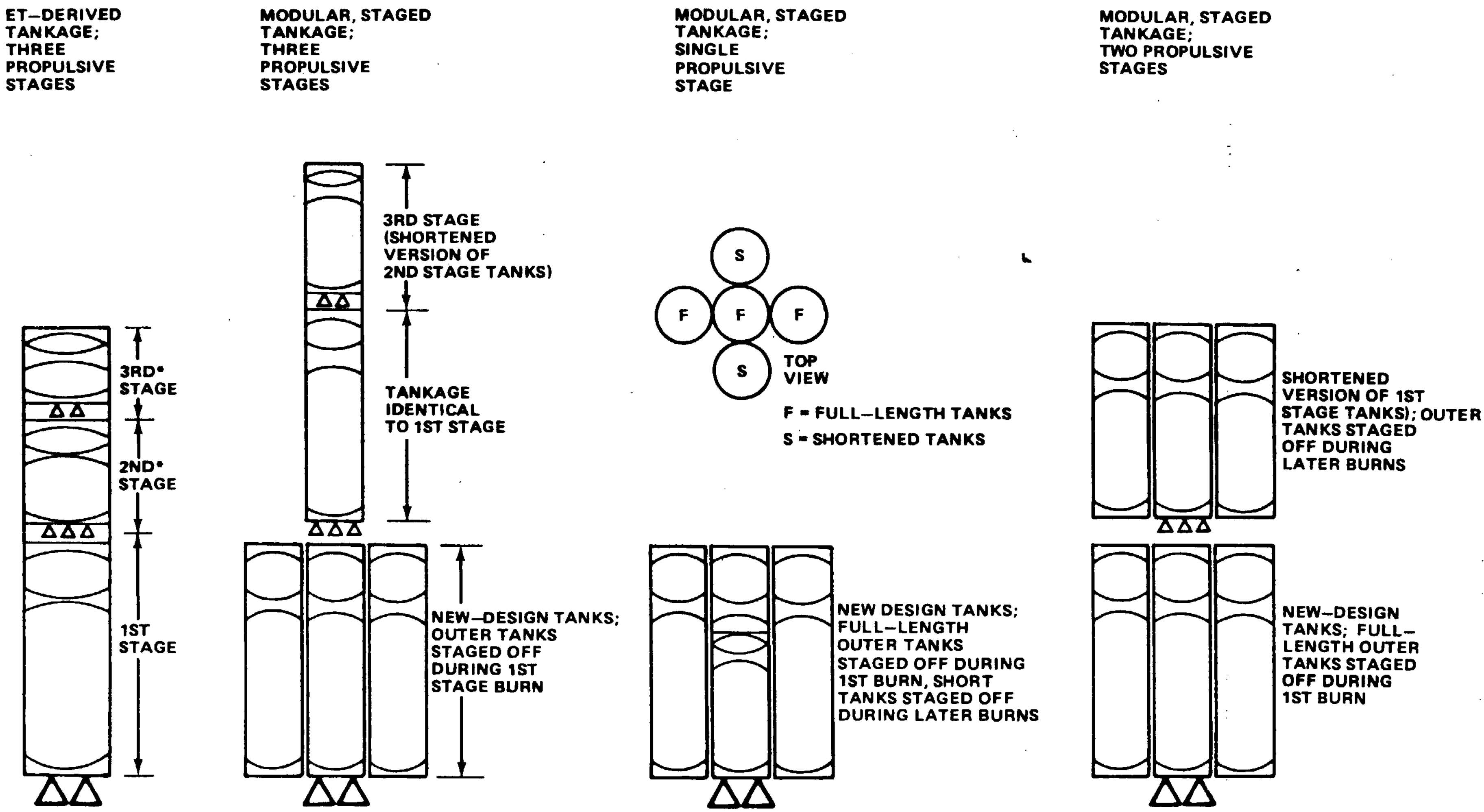
Some of the options for a chemical all-propulsive transportation vehicle are shown in Figure 3 (not to scale). These concepts vary from single-stage to 3-stage vehicles. One of the features stressed in these concepts is commonality of design among the stages, with tank length being a variable to accommodate differences in sizing.

On the STS External Tank (ET)-derived vehicle, it probably would be difficult to design the third stage tanks with as large a diameter as the ET, since the required propellant quantity may not be that large. The first and second stages, however, could probably make use of this commonality. The single-stage concept does not appear to be as good as some of the others from several standpoints. For one thing, it would be difficult to cover the required thrust range with only one engine concept. The engines would have to be fairly large and heavy (approximately 7,000 lbs. each) to accommodate the first stage requirements, and would have to be carried along for the entire mission, which adds a significant weight penalty. The 2-stage and 3-stage vehicles alleviate these problems, but at the expense of some cost and complexity. On these concepts, empty tanks and/or expended stages are jettisoned to save weight. There is a tradeoff between the propellant weight savings accrued by jettison of dead weight, and the cost, complexity, and weight associated with the additional stages. A preliminary design was developed for a modified version of the 3-stage concept shown here, and is described in reference 5.

An all-propulsive vehicle would probably not be utilized, especially for opposition missions, due to excessive propellant weight penalties; a more attractive approach would be to utilize a vehicle capable of aerobraking at Earth and Mars. Research and development is already underway on aerobraking concepts, as part of OTV technology work, and the technology should be supportive of Mars vehicle needs and should be available in the timeframe needed for Mars applications. As mentioned previously, multiple missions will be needed for Mars exploration and utilization. The variation of energy requirements across the opportunities of interest implies that the SV must have the capability of accomplishing missions across the range of worst to best-case

655-85

FIGURE 3. CHEMICAL–PROPULSION TRANSPORTATION VEHICLE OPTIONS FOR MANNED MARS MISSIONS



* SHORTENED VERSIONS OF 1ST STAGE TANKS (LIGHT–WEIGHTED ET DERIVATIVES)

opportunities. For maxium versatility and cost-effectiveness, a transportation "system" should be developed which allows accomplishment of a wide range of missions over a wide range of opportunities. One approach to such a system is described in reference 5. In this system, an aerobraking, cryogenic-propulsion SV is used for either opposition or conjunction missions at any opportunity. Elements of this system can be used for an early Mars fly-by mission as well as for more demanding later landing missions, with modular additions to the elements. No costly dead-ended concepts would be involved in this type of approach. The elements and associated systems would incorporate "technology transparency" to the degree feasible, for efficient upgrading of capability over long time periods.

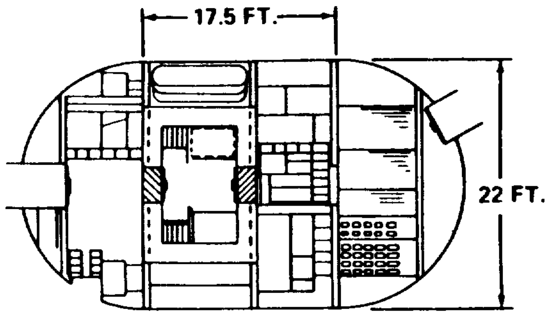
Spacecraft

The nature of the spacecraft is dependent on the nature of the mission. Some missions would have only an orbiter, some only a lander, and some both. For unmanned "cargo" missions, no habitable elements would be necessary. Some of the concepts which have been proposed as orbiters are shown (not to scale) in Figure 4. The terminology most frequently used for this element is "Mission Module" (MM).

The MM concepts could be elements derived from Space Station (SS) modules (14 ft. diameter X 35-45 ft. long) or could be larger-diameter modules of a new design. The former approach would have cost, experience, and logistics advantages. The latter approach may have internal packaging and weight advantages. Multiple pressurizable habitable volumes will probably be necessary for safe-haven reasons, hence a large-diameter module will probably need to have separate pressurizable compartments. There are some limitations on the MM configuration, but generally, these are not as restrictive as those on the Mars Excursion Module (MEM) discussed later. Since the MM can be assembled in orbit, it does not have to withstand (as a whole) the ETO launch environment nor be constrained to the ETO shroud dimensions. A large-diameter (approximately 80 ft.) aeroshell will probably be needed for aerocapture at Mars, and this also permits a good bit of freedom in configuration of the equipment (MM and other) located behind the aeroshell (the areoshell would be assembled or deployed in LEO, because of its large size). Some

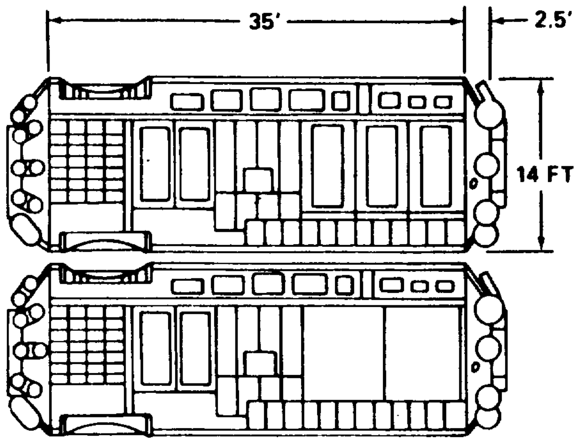
3808–85

FIGURE 4. MISSION MODULE OPTIONS



LARGE MODULE

VOLUME = 12,250 FT^3
*WEIGHT = 13,050 LB.**



SPACE STATION MODULES

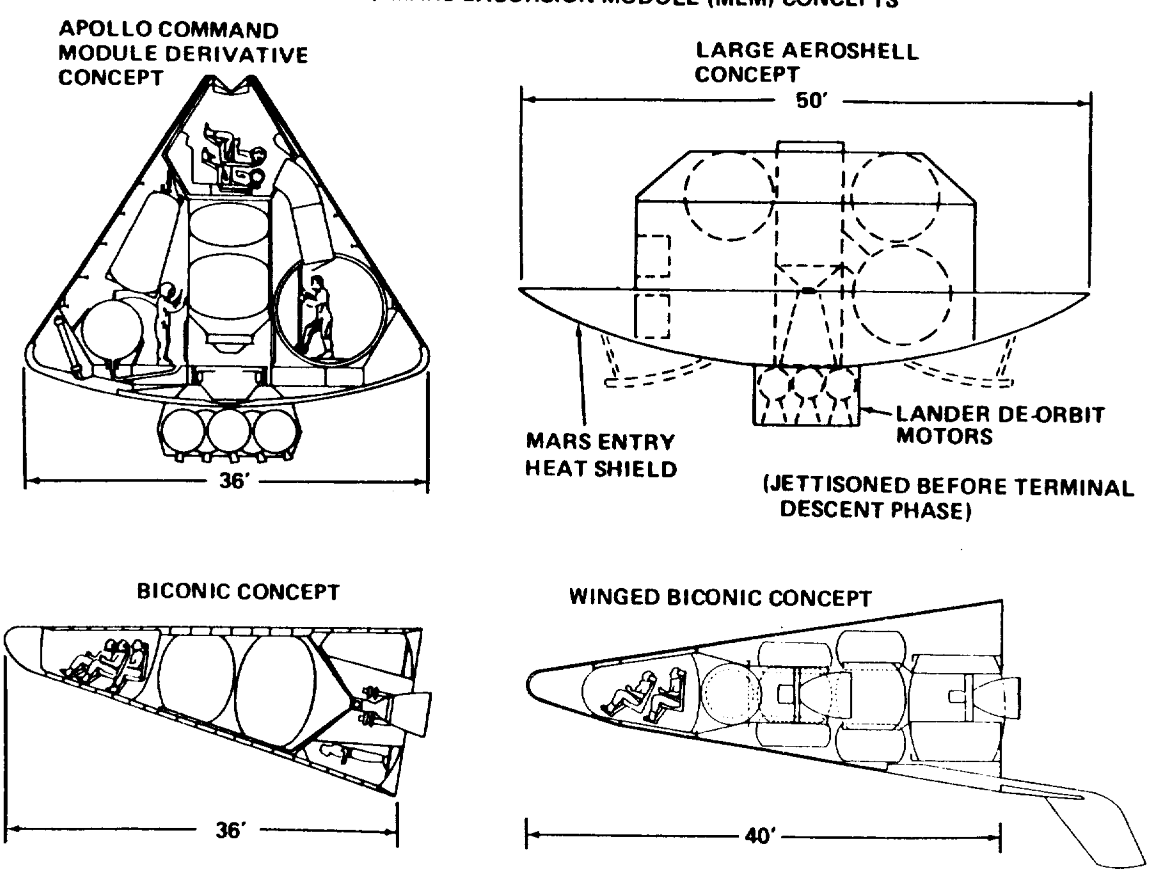
VOLUME (2 MODULES) = 11,488 FT^3
* WEIGHT (2 MODULES) = 22,762 LB.**

** WEIGHTS ARE REFERENCED TO EARTH

* PRIMARY STRUCTURE ONLY

4167–85

FIGURE 5 . MARS EXCURSION MODULE (MEM) CONCEPTS



of the concepts which have been proposed as landers, or MEM's, are shown in Figure 5. Some of these are discussed in reference 13.

The MEM design is heavily dependent on the concept of entry into the Mars atmosphere. Most concepts have utilized aerobraking for partial descent. In addition, some have utilized parachutes and some have utilized propulsive braking. Some MEM concepts have utilized a biconic shape, and others have utilized a conical shape. Both of these approaches impose rather severe limitations on the configuration and quantity of equipment which can be taken to the surface, since the equipment must be conformable to the conic or biconic envelope dimensions. A large diameter (approximately 50 ft.) aeroshell seems to be required for aerobraking of the MEM during descent to the surface. Such a large diameter shell would probably allow freedom to package equipment of various sizes and shapes behind it if the MEM configuration were not constrained to a conical envelope. This allows development of a delivery "system" concept, in which the size and shape of the equipment behind the aeroshell can vary considerably from mission to mission, affording a high degree of adaptability and versatility for surface delivery of men and equipment. Such a concept is discussed more fully in reference 5.

Mars surface transportation vehicles (such as land rovers, "pogo" propulsive vehicles, airplanes, etc.) would be transported to the Martian surface in the MEM. Concepts of these are discussed more fully in reference 6.

<u>Orbit-To-Orbit Vehicles</u>

<u>Orbital Transfer Vehicles</u>

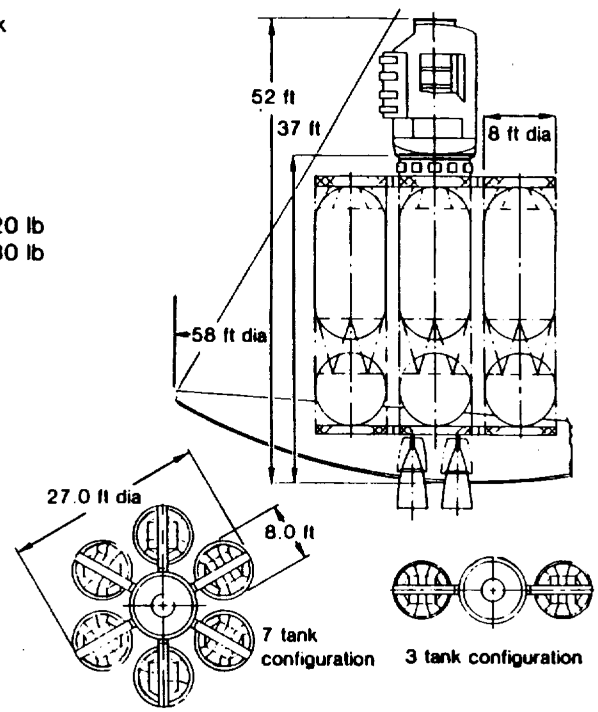
The Orbital Transfer Vehicle (OTV) (reference 7) should be an operational vehicle in the mid-to-late 1990s. One or more orbit-based OTVs is planned to be a part of the advanced SS infrastructure. OTV studies are in progress, and no selection has yet been made of a preferred concept. However, one concept is shown in Figure 6 to aid familiarization with this class of vehicle.

For all Mars mission options, a LEO-based OTV (possibly one on loan from the SS) can be used to circularize the orbit of the elements returned from Mars (which would probably have been injected into an elliptical Earth orbit having a perigee equal to the SS orbit). Compared

4196–85

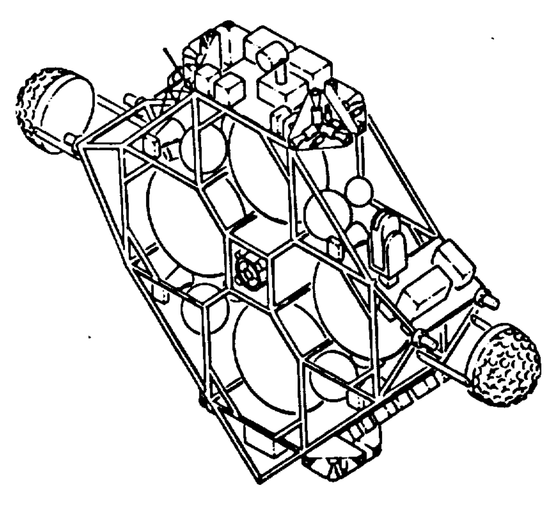
FIGURE 6.
S-4 CORE PROPELLANT MODULAR OTV THREE & SEVEN TANK ASSEMBLIES

	3 tank	7 tank
Propellants	O_2/H_2	
Engine(s)/no.	Adv Eng/2	
Isp & mixture ratio (sec)	482/6:1	
Thrust level	7.5 Klb/engine	
Avionics	Baseline	
Payload available length	—	
Propellant capability	63,780	148,820 lb
Burnout weight	7,290	12,530 lb
ASE weight	TBD	
Payload capability	—	
• GEO transfer	TBD	
• GEO circular delivery		
— Expendable	TBD	
— Reusable	27	72 Klb
• GEO roundtrip	14	37 Klb
• 12 hour, 63 deg	TBD	
• C_3 of 85 km^2/sec^2	TBD	
Source of data	GDC-1985	



4166–85

FIGURE 7. TYPICAL OMV



DIMENSIONS:

37 X 178 INCHES

WEIGHT LOADED:

10,496 LBS.

PROPELLANTS:

6700 LBS. NTO/MMH
200 LBS. GN_2

to the case of having to transport a circularization stage to Mars and back, this would allow significant savings of weight on the SV.

For Mars missions using very-low-thrust vehicles, a new orbit-to-orbit vehicle development would be required for the chemical portion of the hybrid propulsion system, in addition to the new development require for the very-low-thrust portion. This vehicle could possibly be a derivative of the OTV.

Orbital Maneuvering Vehicles

The Orbital Maneuvering Vehicle (OMV) (reference 8) should be an operational vehicle in the early-to-mid 1990's. OMV studies are in progress, and no selection has yet been made of a preferred concept. A generic concept is shown in Figure 7 to aid familiarization with this class of vehicle.

One or more orbit-based OMVs is planned to be a part of the early SS infrastructure. An OMV (possibly on loan from the SS) will be useful in on-orbit assembly of the SV, and in ferrying men and equipment between the STS, SV, and SS (especially if the SV and SS are co-orbiting with each other in the SV assembly phase).

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N87-17745

A MANNED MARS MISSION CONCEPT WITH ARTIFICIAL GRAVITY

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ABSTRACT

A series of simulated manned Mars missions was analyzed by a computer model developed by the author under contract to NASA - JSC. Numerous mission opportunities and mission modes were investigated. Sensitivity trade studies were performed of the vehicle all-up mass and propulsion stage sizes as a function of various levels of conservatism in mission velocity increment margins, payload mass and propulsive stage characteristics. This study emphasized the longer duration but less energetic type of conjunction class mission. The specific mission opportunity reviewed was for a 1997 departure.

From the trade study results (some 300 separate mission simulations), a three and one-half stage vehicle concept evolved, utilizing a Trans-Mars Injection (TMI) first stage derived from the Space Shuttle External Tank (ET). The vehicle was completely ground assembled but required propulsion system reconfiguration, refueling with liquid hydrogen and oxygen, and payload mounting in Low Earth Orbit (LEO), utilizing the services available from the LEO Space Station. The second stage, used solely for propulsive Mars Orbit Insertion (MOI) of the space vehicle into a 24 hour period orbit about Mars, utilized cryogenic propellants (O_2/H_2) and advanced, active thermal control features to preserve the liquid hydrogen over the 9 month duration journey from Earth orbit to Mars. The final "stage and one-half" propellants chosen were liquid oxygen and liquid propane in recognition of the formidable problem of retaining liquid hydrogen during the 15 month loiter in the vicinity of the heat-emitting planet Mars. Following the "Trans-Earth Insertion (TEI)" burn, the spent TEI propellant tanks were jettisoned and the remaining tankage provided for propulsive return of the "Command Module (CM)" with the crew into a 24 hour elliptical orbit of Earth. Final crew recovery was accomplished by a man-rated Orbit Transfer Vehicle (OTV).

Artificial gravity was provided for 40 metric tons of living quarters in two "Mission Modules (MMs)" mounted on outrigger tunnels extended from the spinning central core which contained a heavily-shielded, despun CM of 40 metric tons mass serving as both command station for the

mission and "storm shelter" to allow survival of the crew during solar flare events.

Two 75 metric ton landers were provided, permitting crews of four persons two surface stays of two to four weeks duration each at separate locations on the surface of Mars. Four "Mars Maneuvering Vehicles (MMVs)" were also provided to permit manned sorties from the Mars orbit to both Phobos and Deimos during the Mars orbit staytime. The MMVs were also employed to recover the ascent stages of the two landers from low (500 km) circular Mars orbits.

The aggregate payload mass for this mission was 287 metric tons and the departure mass from Earth orbit was 1,254 metric tons, over 60% liquid oxygen. Generous electrical power service was provided for the mission by a cluster of "SP-100" class nuclear-electric generators energized after the TMI burn and left in solar orbit at mission's end.

TRAJECTORY SELECTION

A round trip mission from Earth orbit to the planet Mars is not necessarily more energetic than other missions contemplated as precursor activities. A round trip mission from Low Earth Orbit (LEO) to the geo-stationary Earth orbit requires a total propulsive velocity change of about 8.5 km/sec. A round trip journey to the Earth's moon requires about 5.5 km/sec if aerobraking is used for Earth return as was done on the Apollo missions. The "Conjunction Class" missions to Mars described in this paper require 6 to 7 km/sec, not including mission reserves. The mission would return to a high elliptical Earth orbit accessible from the Space Station by means of a man-rated OTV. The four major maneuvers--Trans-Mars Insertion, Mars Orbit Insertion, Trans-Earth Insertion and Earth Orbit Insertion (EOI)--of the mission are relatively invariant with the specific bi-annual mission opportunity used for the mission, permitting a single design to serve a series of mission opportunities.

If low energy "Hohmann transfer" heliocentric orbits are employed, the time of flight is seven or more months each way. Awaiting favorable alignment of the planets requires stay times in orbit about Mars on the order of 15 months. Total elapsed time for the Conjunction Class missions is therefore over 2 1/2 years.

Many schemes have been proposed to reduce these flight times. However, each scheme has drawbacks, mainly, a large increase in required mission energy. Use of a Venus Swingby approach is one means of reducing this mission energy penalty. This strategy brings the interplanetary vehicle much closer to the Sun than do the Hohmann transfers and poses thermal management problems that must be solved.

The "Opposition Class - Venus Swingby" missions discussed in the literature have total mission durations of about 23 months-- with limited time (1 to 2 months) available to orbit or land on Mars and accomplish the mission before the return trip to Earth. Mission energy ranges from 9 to 14 km/sec and is highly variable in the magnitude of its four components, each being highly dependent upon the specific mission opportunity selected.

The type of mission chosen is driven by the relative importance of mission energy magnitude and variability versus mission elapsed time. In the era of "permanent human habitation of space", the 30% increase in mission duration of the Conjunction Class mission is of lesser importance than in the Apollo era, when most of the Mars literature was written. Rather than being a detriment, the long stay time at Mars of the Conjunction Class mission can be productive and increase the science return of the mission.

Previous space transportation studies (NASA-Boeing FSTSA, 1976, NASA-DOE SPS, 1977) indicate that the cost of launching the vehicle components and propellants into Low Earth Orbit dominate the total mission cost for advanced space missions. A good indicator of a Mars mission cost is thus the "Initial Weight in Low Earth Orbit (IWLEO).

Sensitivity studies presented at this conference show that the Conjunction Class mission can deliver four times the useful payload round trip Earth-Mars-Earth or six times the payload from Earth orbit to orbit around Mars than can the Opposition Class mission for a given propulsion technology and IWLEO. These factors may be different for advanced propulsion technologies or with the development of "in-situ propellant production", but the trend remains - a large increase in payload mass is available to the Conjunction missions compared to the Opposition, even Swingby, missions. This added payload capability will allow for more experimentation equipment to be used during the mission's extended time in the vicinity of Mars.

Increased payload capability can also be used to enhance crew comfort. The requirement to spend more than two years in space, remote from the conveniences and companionship of Earth means more attention must be paid to human needs. No degree of training and motivation will allow a crew member to endure this long without the ability to briefly get away from others, to be frequently stimulated by interesting events, to have a change of environment and to enjoy a diet and personal services more Earth-like than heretofore provided to astronauts. For this reason, it is suggested that each mission carry two Mars landing vehicles and the vehicular capability to visit the moons of Mars during the long stay time at Mars. It is also suggested that multi-compartment living quarters, with artificial gravity and separated from the workplace, are required even though these conveniences may not be clearly necessary from purely physiological considerations.

For these reasons, a large space vehicle, aggregating some 1250 metric tons and carrying 287 metric tons of useful payload is proposed to accomplish one or a series of missions. The intent of this mission concept is to provide a point of departure and framework for future discussion and study.

PRIMARY PROPULSION (TMI) SYSTEM OPTIONS

The manned Mars mission was a feature of the post-Apollo "Integrated Plan" for proposed future NASA space activities. At that time, about 1970, it was envisioned that Saturn-derived launch vehicles, the "Int-20 and Int 21" would continue to be launched immediately following the Apollo program and later co-exist with a vehicle similar to but smaller than today's Space Shuttle to provide crew launch services. With a capacity of almost 110 metric tons per launch (compared to the Shuttle's 29.5 metric tons), the Int-21 represented a powerful instrument for proposed space activities.

A large, 10 m diameter Space Station was envisioned in Low Earth Orbit as the home base for a man-rated, multi-purpose "Space Tug" using hydrogen/oxygen propellants at a specific impulse of 460 seconds and a "Reusable Nuclear Shuttle (RNS)" for ferry missions to lunar orbit and to Mars. The RNS was to use the "NERVA" thermal reactor power plant heating 136 metric tons of liquid hydrogen to provide thrust of up to 1/3 megaNewton (75,000 lbf) at a peak specific impulse of 800 seconds.

The Tug was to have modular add-ons to permit landing on the Moon and to carry humans to the Geostationary Orbit and return.

The NERVA engine and the M-1 engine, a gas generator cycle hydrogen/oxygen engine rated at 4.5 MN (1,000,000 lbf) enjoyed vigorous development activity. None of these plans reached fruition.

Aside from the magnificent Space Shuttle Main Engine (SSME) and modest improvements in bi-propellant liquid propellant engines, little of real significance has been accomplished in high thrust space propulsion in the intervening 15 years. A "Spacecraft Propulsion Systems" seminar held by the AIAA in May 1984 predicted that, with concentrated effort, improvements of 4 orders of magnitude in specific impulse might be attained in the next 25 years, culminating with anti-matter propulsion at a specific impulse of 1,000,000 seconds by 2010 or so. Although such developments may possibly occur and would assuredly enable space travel for the masses, recent history and the dearth of true breakthroughs do not lend high confidence to these optimistic forecasts.

The U.S. DOD "Strategic Defense Initiative (SDI)" may revitalize this field and produce, for civil use, new and markedly improved space propulsion systems. For planning circa 2000 manned Mars missions, however, only two high thrust propulsion technologies now appear to be available that quickly - chemical propulsion, as exemplified by the SSME, and nuclear thermal propulsion of the NERVA type. An assortment of other, smaller bi-propellant liquid rocket engines will be needed to achieve the total mission objectives which can have small but cumulatively significant improvements in performance when compared to the rocket engines available today. Given the 6 to 12 year gestation interval for new space propulsion elements, serious development must soon begin in order to meet the turn of the century goal for the manned Mars mission.

Since the 2.2 MN (500,000 lbf) thrust class SSME, as applied to the Shuttle, must safely operate at sea level atmospheric pressure, the extent to which the exhaust gases can be expanded in the nozzle is necessarily constrained. If future variants of the SSME are intended to only be used in the space environment, this constraint is removed, and a larger bell nozzle can be used to improve the SSME performance to the 470 second specific impulse range. With an inert mass of less than 4 tons, a high expansion SSME is an attractive choice for the first and most difficult maneuver of the mission - Trans-Mars Insertion.

It's competitor is a latter-day variant of the NERVA engine. Improvements in the carbon core material of NERVA have been proposed which might permit specific impulse levels as high as 850 seconds. This type of engine, with the necessary shadow shield to protect the crew from the radiation field of the engine, is inherently massive as compared to combustion engines of higher thrust such as the SSME. This inert mass, coupled with the high inert mass of the hydrogen fuel storage vessels, degrades the apparent advantage over the SSME type engine. When operational constraints and penalties due to low thrust and the radiation field and safety measures required to protect both the space operations and the population of Earth are accommodated, the apparent advantages further decline. If used at all, a nuclear engine must be initially energized or stored for reuse only at "nuclear safe" orbital altitude (defined as that altitude which would permit natural decay to safe levels of the radioactive products generated within the engine during its operation, before atmospheric drag would cause re-entry of the vehicle). This constraint could force final assembly of the Mars space vehicle to take place at altitudes sufficient to induce another radiation problem - encounter with the intense natural radiation field of the trapped radiation belts around Earth.

Future studies will be needed before the relative merits of the two competing propulsion technologies are fully understood. Vital to the trade study is the cost of placing propellants into Low Earth Orbit. Therefore, such a study must consider not only the technical and operational factors described above but also the characteristics of the launch capability available for use. Monetary tradeoffs must also have available a reliable "mission model", as apparent savings in mission costs may be overwhelmed by disparities in development cost between the alternative space vehicles and their supporting infrastructure. Thus, an indefinite series of missions can better justify a larger development cost for primary propulsion systems than can a single mission or a short series of missions.

PROPULSIVE VEHICLE STAGING

The mission simulation routine used for this analysis was set up to permit the use of two propulsive stages for each of the major maneuvers, or "burns", including perigee raise in Earth orbit, to permit the computational routine to be employed for "aerobraking" upon return to Earth. Since the Conjunction Class of mission was chosen as the "baseline" for this study, the velocity increments for each of the four major burns were not sufficient to justify two stages for any of them.

For other reasons, staging was found to be called for just prior to intiation of each of the last three major burns. The TMI stage carried 716 tons of oxygen/hydrogen propellants (57% of total space vehicle mass) at an Oxidizer-to-Fuel (O/F) ratio of 6.0:1 and was powered by a single engine derived from the SSME which delivered 468 seconds of specific impulse. This stage performs a single burn of about 25 minutes duration within a few hours of separation from the Low Earth Orbit Space Station (LEO SS) and has then completed its principal function. Thrust-to-mass ratio is 0.17 at ignition and 0.40 at burnout. Inert mass of this stage is estimated by two independent methods to be about 43 tons. As a consequence of its short mission life, the cryogenic insulation provisions necessary to retain propellant servicing at the LEO SS are expected to be fully adequate for the flight mission. To provide shielding from natural radiation and to permit, if desired, use of residual fluids by the spacecraft, the spent TMI stage is retained until Mars approach.

To acquire the 24 hour period orbit at Mars, the second stage of the space vehicle is used. This stage also uses oxygen/hydrogen propellants, carries 102 tons of propellant in heavily insulated and actively refrigerated tanks and produces a 0.20 thrust-to-mass ratio at ignition by use of a 920 kN (207,000 lbf) thrust engine of new design with high chamber pressure and an ultra-high expansion ratio nozzle to deliver 480 seconds of specific impulse. Smaller, multiple engines may be found to be preferable on further analysis. Including the insulation and active refigeration provisions, this stage has an estimated inert mass of 18 tons. As the outbound mission flight time is in the vicinity of nine months, propellant conditioning technology will be challenged to minimize or eliminate hydrogen loss through tank venting. It is expected that this can be done by the time this mission is to be dispatched. This stage is assumed to provide a 100 m/sec orbit adjust burn shortly after arrival in Mars orbit. The spent MOI stage is retained until shortly before the space vehicle is to begin the return to Earth.

Since the vehicle remains in orbit around Mars for about 15 months and is subjected to a higher heat load there than in free space due to the albedo of the planet, use of liquid hydrogen as the Earth return propulsion fuel is not considered to be a likely nor prudent choice. Instead, the higher boiling point fuel, liquid propane, is selected to be burned with oxygen for this third stage of the Mars space vehicle.

Oxygen and propane have an overlapping liquid range and therefore do not require thermal isolation from one another. Two propellant tanks with a capacity of 36.5 metric tons supply propellant to begin the return flight. Multiple engines (three?) of new design, with a total thrust of 300 kN (65,000 lbf) produce a thrust-to-mass ratio of 0.20 and deliver 373 seconds of specific impulse over a burn interval of a bit less than 8 minutes. Spent mass associated with the TEI burn totals 3.6 metric tons. The engines are retained for later use in acquiring Earth orbit.

Reuse of the TEI propulsion system occurs following the approximately seven month trans-Earth coast to acquire a highly elliptical 24 hour period Earth orbit in the orbital plane of the LEO SS. Only the central Command Module is present as the payload for this maneuver - the artificial "g" living quarters, stores modules, nuclear power supply and supporting structure are staged and left in the heliocentric Earth-Mars orbit. Approximately 19.5 metric tons of propellant are required, stored in partly-filled tanks of the same design as those used for TEI.

Inert mass of this propulsion system is estimated to be 2.4 tons. A thrust-to-mass ratio of 0.70 is experienced near end of burn, unless fewer than three engines ignite or the engines are throttled to lower than full rated thrust. A single engine of the cluster of three is adequate to safely complete this final maneuver.

Secondary propulsion systems of the Mars space vehicle include a pair of RL-10 engines on the large TMI stage to permit roll control, a gas oxygen/hydrogen reaction control system to provide 100 m/sec mid-course correction and attitude change during trans-Mars flight. About 12.5 metric tons of cryogenic propellant are used at a specific impulse of 400 seconds for these maneuvers. These propellants can be tank vent gas from the MOI stage, relieving somewhat the thermal control difficulty. Inert mass of this RCS is estimated to be 8.2 metric tons.

A separate oxygen/propane RCS is used for the 100 m/sec of maneuvers assumed necessary during inbound flight. 3.5 tons of oxygen/propane propellant are consumed at a specific impulse of 315 seconds by an RCS of 2.27 tons inert mass which is integral with the Command Module.

Many other combinations of staging arrangement and propellant selection for the manned Mars missions are possible. One or more combinations different from those described above may be found, on more detailed analysis, to be preferred. The ones described above were selected with operational suitability in mind and other candidate systems will have to provide at least equal attention to operability as well as performance. Electric propulsion is a provocative alternative for at least some of the propulsive functions, and deserves more attention if a power-rich environment may be economically provided.

HABITATION CONSIDERATIONS

Several important principals need to be observed in design of the manned Mars mission and its space vehicle. First, there must be sufficient human resources and skills present for this microcosm of society to be entirely self-sufficient (except for information) for the two to two and one-half year journey. This will require bakers and barbers as well as planetary scientists and pilots. New and unexpected maintenance and repair tasks will require tools, supplies, and information adequate to the potential tasks. Equally unexpected science opportunities are apt to present themselves for exploitation. What this indicates regarding minimum crew size remains to be determined. It is likely that the estimated minimum crew will increase rather than decrease with time as the studies unfold. Perhaps design accommodations for 12 to 20 persons is not excessively conservative if 6 to 8 are planned today.

A favorable mix of fully committed personalities will also be necessary, as, even if spacious by contemporary spacecraft standards, the Mars vehicle will become a confining and overly intimate place before the mission ends. A key to rendering this close proximity of people tolerable may be to permit interruption of their intimacy. The space vehicle should have the workplace and living quarters sufficiently apart to require a conscious "going to work" personal journey from private quarters each day. The Mars excursion modules should be pressurized and, although a bit remote from the normal living and working quarters, should be accessible via pressurized compartments so that periodic subsystem checks may be periodically performed - the principal beneficiary being perhaps the human subsystem.

While in orbit around Mars, the excitement and newness of personal, on-site science experimentation will no doubt assure complete harmony among the entire crew. If there can be, on each Mars mission, two landings on the surface and visitation to both of the Martian moons, the crew will always have either a task underway, data to reduce or planning to review for the next critical event. Care should be taken that useful work is reserved or provided for the 9 month return journey.

One final consideration relates to the provisioning of a heavily shielded "storm shelter" for the crew to gather in during solar flare events. This place must be occupied for the duration of the atypically high natural radiation levels which may persist several days or possibly weeks. This close confinement may not come at all during a mission or it may be repeated. Rather than making a very small volume dedicated to this purpose, which requires total inactivity of the occupants, it will prove more acceptable to the crew to arrange provisions, including propellants, around the Command Module so that, even though normal living quarters are inaccessible during these intervals, useful work continues. If necessary, additional shielding mass beyond theoretical minima should be provided to assure that adequate volume for near-normal activity is maintained during these stays.

These considerations, coupled with the need for an artificial gravity field previously discussed, have led to the suggested general arrangement of an artificial gravity manned Mars space vehicle illustrated by Figures 1 through 4. As these design matters will continue to be highly subjective, different people will have different and strongly defended views as to what is "correct" for the mission.

Throughout these discussions, it must be remembered what people on this journey are asked to do--remain in close quarters functioning at peak efficiency with no possibility of altering their circumstance for a very long time.

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MANNED MARS MISSION

CONJUNCTION CLASS
NEAR TERM TECHNOLOGY OPTION
HYBRID CHEMICAL PROPULSION
1400 METRIC TON CLASS

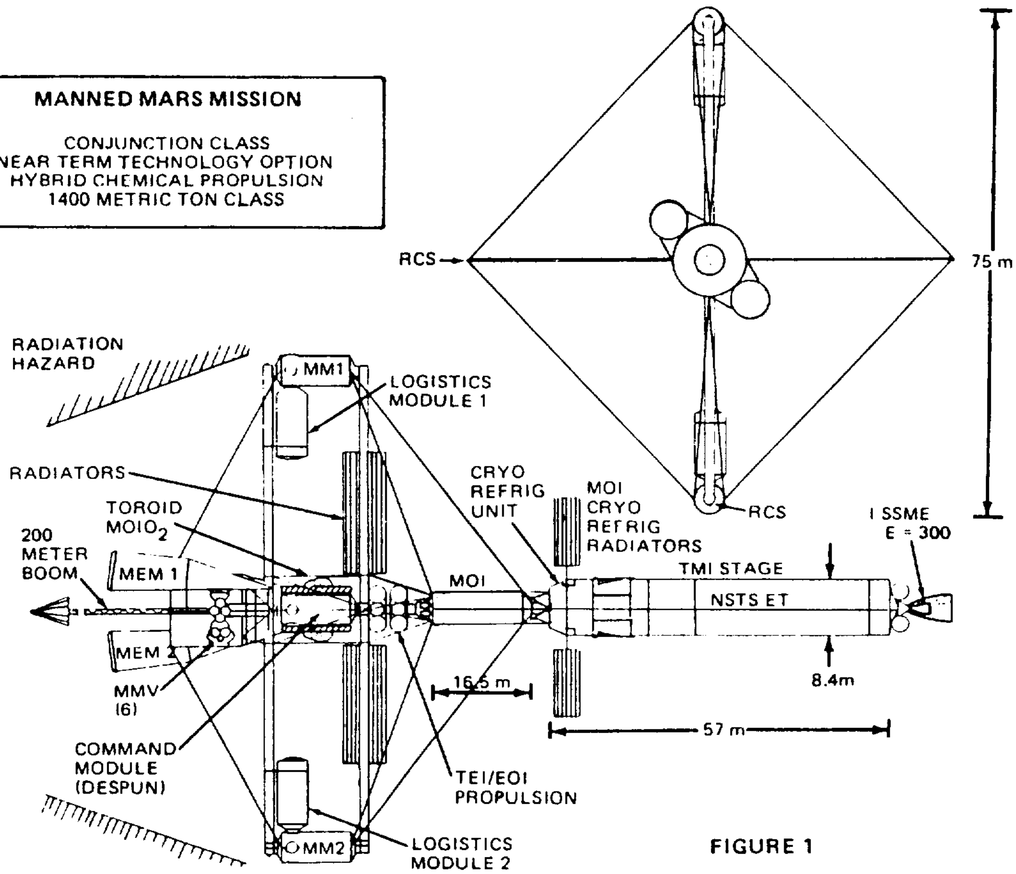


FIGURE 1

INITIAL MARS ORBIT CONFIGURATION

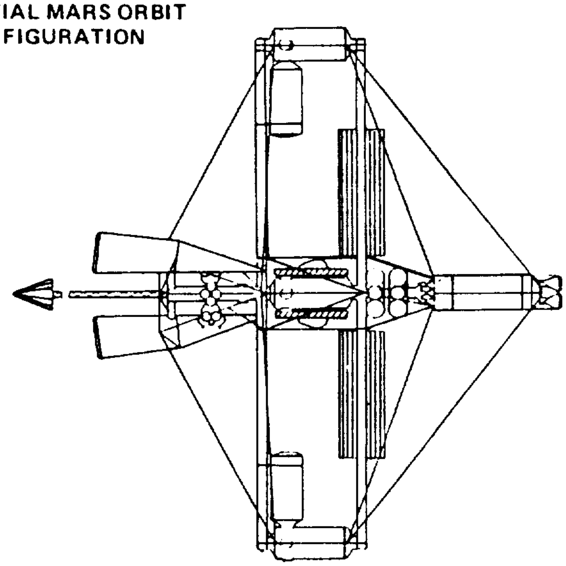


FIGURE 2

TRANS–EARTH CONFIGURATION

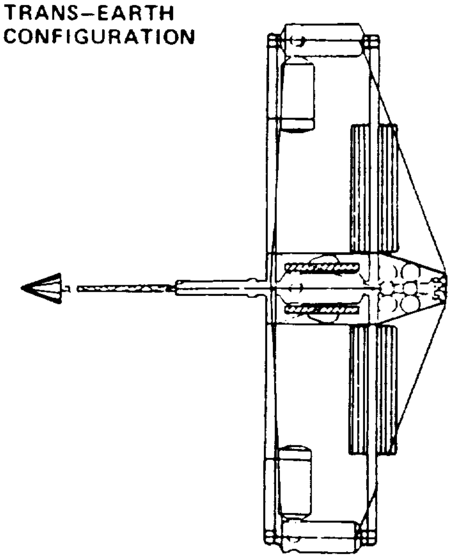
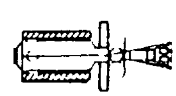


FIGURE 3

FIGURE 4



EARTH ORBIT CONFIGURATION

CANDIDATE SPACE VEHICLE GENERAL ARRANGEMENT

The general arrangement of the candidate manned Mars mission space vehicle in the Earth departure configuration is illustrated by Figure 1. It is a three and one-half stage propulsion vehicle system with a large spacecraft "cluster" aggregating almost 300 metric tons mounted forward of the three stages. The spacecraft cluster is comprised of: (a) Two "Mission Modules (MM's)", which serve as living quarters for the crew for the two and one-half year duration mission; (b) The core "Command Module (CM)" which serves the purposes of on-board command and control, a central meeting place for conferences, meals and recreation and for a radiation-hardened "storm shelter" for crew habitation and support during the abnormally high natural radiation levels occurring infrequently as a consequence of solar activity; (c) Two "Mars Excursion Modules (MEM's)" which accomplish the descent from the 24 hour period Mars orbit of the "mother ship", support the crew for their two week surface stay, and return the crew and mission artifacts to a 500 km altitude circular Mars orbit; (d) Four "Mars Maneuvering Vehicles (MV's) which provide mobility to the crew in the vicinity of the high Mars orbit, transport a "Crew Module" to soft land on both moons of Mars - Phobos and Deimos - and return to the "mother ship", and finally to recover the ascent stages/crew compartments of the MEMs from their 500 km circular orbits to the high ellipse; (e) A nuclear-electric power supply, made up of a cluster of "SP 100" or later nuclear reactor/thermal cycle power plants with the associated space radiator, control and power conditioning subsystems; (f) Four tunnels interconnecting the MM and CM; (g) An extensible boom interconnecting the CM and power supply; (h) Structural ties to lend rigidity to the spacecraft cluster; (i) Stage separation equipment to permit the spacecraft to shed all appendages from the CM just before the final propulsive maneuver into a 24 hour period Earth orbit.

Figure 2 illustrates the vehicle configuration as it appears just before insertion into Mars orbit and as it will remain during MEM and MMV operations during the 15 month stay in the Mars orbit. Overall dimensions of the departing space vehicle are approximately 120 m long and 75 m in platform span. The total mass at Earth departure is approximately 1250 metric tons.

The large "Trans-Mars Insertion (TMI) Stage" is jettisoned just prior to the Mars Orbit Insertion (MOI) burn. It remains with the space vehicle for the trans-Mars journey to provide radiation shielding mass and to permit possible recovery and use of the stage residual propellants and subsystems capabilities. In a similar fashion, the MOI stage remains with the space vehicle after it has performed its primary mission until preparations begin for return to Earth.

Figure 3 illustrates the post-TEI stage configuration which remains in this arrangement for the trans-Earth flight for the same reasons that the TMI stage was retained for the outbound leg. In both cases, a penalty is paid in attitude control and mid-course correction propellant in exchange for the utility these spent stages may possess. More detailed studies will be necessary to determine whether or not retention is an effective and economical choice.

In the last day of the mission before beginning the Earth Orbit Insertion (EOI) maneuver, the crew gathers in the CM with all of the science yield and mission documentation. The MM's, nuclear powerplant, tunnels and supporting structures are then jettisoned to remain in the heliocentric orbit. Post-jettison propulsion may be required for the larger masses, particularly the spent nuclear reactors, to assure no future contact with the biosphere of Earth or interference with future space missions. Batteries power the CM during the 12 to 36 hours necessary for recovery of either the entire CM or only the crew and science yield from the 24 hour period ellipse to the LEO Space Station. Future Orbit Transfer Vehicle (OTV) capabilities and mission costs will determine the recovery scenario. Figure 4 illustrates this final mission configuration. Dimensions are roughly the same as the Space Station "Common Modules", about 4.5 m diameter by 12 to 15 m in length, and mass is about 40 metric tons. Thus, only a bit over three percent of the mass dispatched from the LEO Space Station for this mission will be recovered.

STAGE TANKS

The mission simulation studies mentioned earlier consistently indicated TMI propellant quantity required in the 500 to 750 metric ton range. As the Space Shuttle External Tank (ET) has a capacity of 707 metric tons, it was elected to conceptualize a means of achieving Trans-Mars Insertion by using the ET as the propellant container for the TMI stage. The TMI stage would thus be completely assembled before launch. It would be used as a standard STS ET for launch of a shuttle, carrying useful payload to the Space Station and placing the TMI stage into orbit. The "ET Stage" was therefore conceptually defined, utilizing a single modified SSME for primary propulsion. This general concept (the "OIS" studies by MSFC, RI, and MDAC) has been reviewed earlier by NASA MSFC and others, circa 1970.

The second stage of the space vehicle also utilizes Oxygen/Hydrogen propellants, but is a much smaller stage than the 1st stage - (requiring a propellant load available at arrival at Mars) of 102 metric tons.

The final stage serves the dual purposes of departing from Mars orbit and placing the CM into high Earth orbit to complete the mission. Since its first use is almost two years after launch, the less volatile fuel--liquid propane--was selected to ease the problem of boil-off loss or reliquefaction. The propellant is contained in eight spherical tanks, each less than 2.6 m in diameter. This stage may either be assembled in space or launched as a single Shuttle payload fully assembled and loaded with propellant in LEO.

N87-17746

MANNED MARS MISSION EARTH-TO-ORBIT (ETO) DELIVERY AND ORBIT ASSEMBLY OF THE MANNED MARS VEHICLE

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ABSTRACT

The contents of this section contain the initial concepts developed for the in-orbit assembly of a Manned Mars Vehicle and for the Earth-to-Orbit (ETO) delivery of the required hardware and propellant. Two (2) Mars vehicle concepts (all-propulsive and all-aerobrake) and two (2) ETO Vehicle concepts were investigated. Both Mars Vehicle concepts are described in Reference 1, and both ETO Vehicle concepts are described in Reference 2. The all-aerobrake configuration reduces the number of launches and time required to deliver the necessary hardware/propellent to orbit. Use of the larger of the 2 ETO Vehicles (HLLV) further reduces the number of launches and delivery time; however, this option requires a completely new vehicle and supporting facilities.

INTRODUCTION

Two (2) Mars vehicle concepts were investigated. An "all-propulsive" vehicle (i.e , one using propulsive braking for capture at Mars and Earth) concept (Figure 1) was analyzed and found to require twenty-five (25) Shuttle-derived (SDV-3R) Vehicle (Figure 2) launches to deliver the required hardware and propellant to Earth orbit. The SDV-3R vehicle is described in Reference 2. An additional Space Shuttle launch is required for the delivery of the supporting equipment (Assembly System plus associated equipment) and crew. Most of this study was performed on the all-propulsive vehicle; however, the same assumptions were applied to an all-aerobrake concept (Figure 3). This second configuration requires nine (9) SDV-3R hardware and propellant deliveries to orbit and two (2) Space Shuttle deliveries. Additional crew deliveries would be required if the crew is rotated. The assumptions and description of the operations are presented below, followed by a KSC ground flow concept (Figure 4) for the processing of the SDV-3R vehicle and payloads. Data is also provided for utilization of the Heavy Lift Launch Vehicle (HLLV) for delivery of the Mars vehicle elements to LEO.

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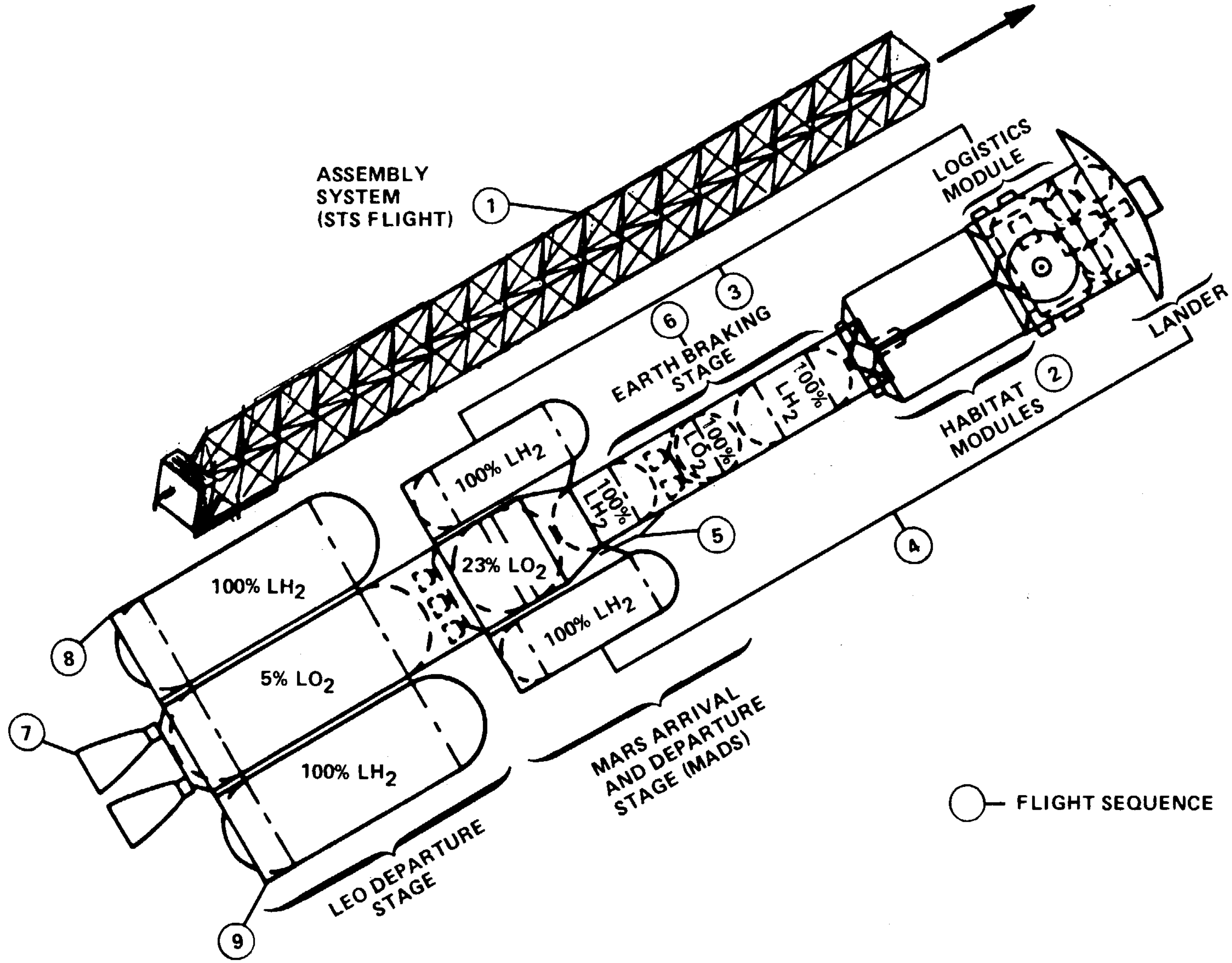
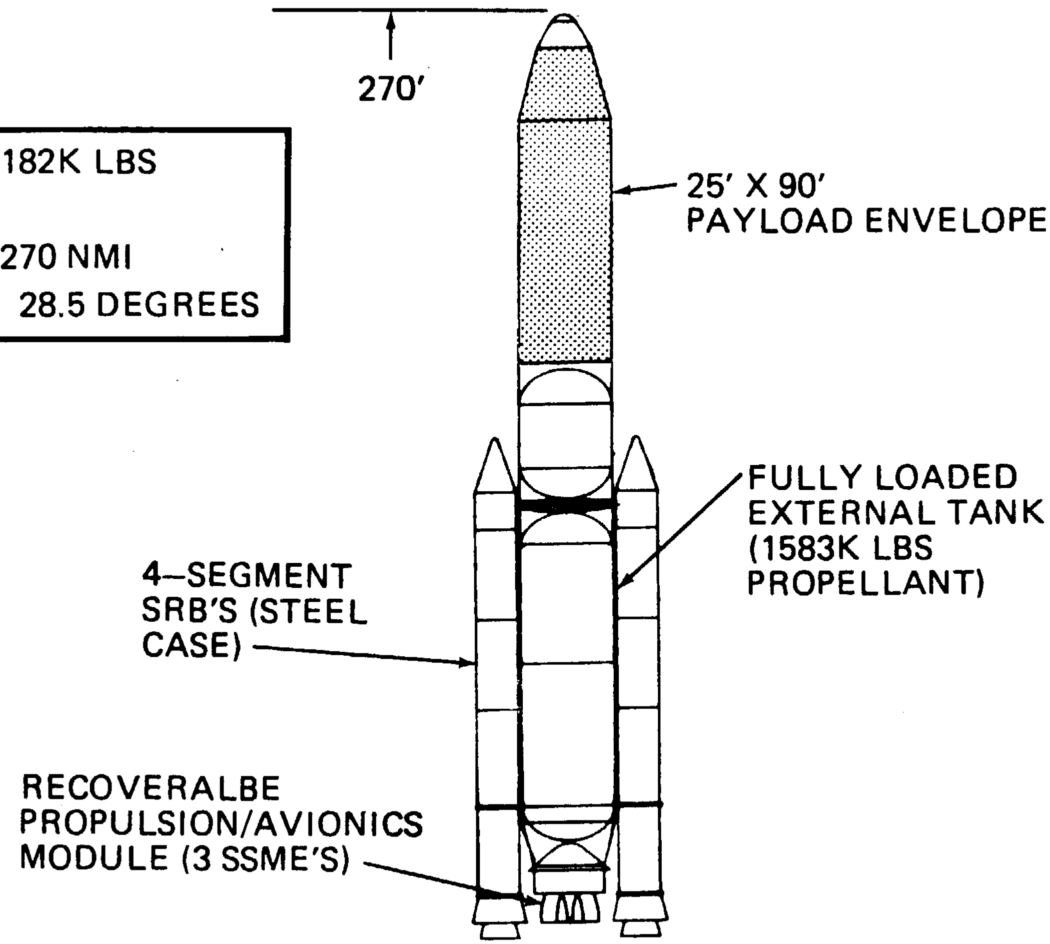


ALL–PROPULSIVE CONCEPT
FIGURE 1

693–85

FIGURE 2
IN–LINE SHUTTLE DERIVED VEHICLE

PAYLOAD	182K LBS
ORBIT	
ALTITUDE	270 NMI
INCLINATION	28.5 DEGREES



SDV–3R

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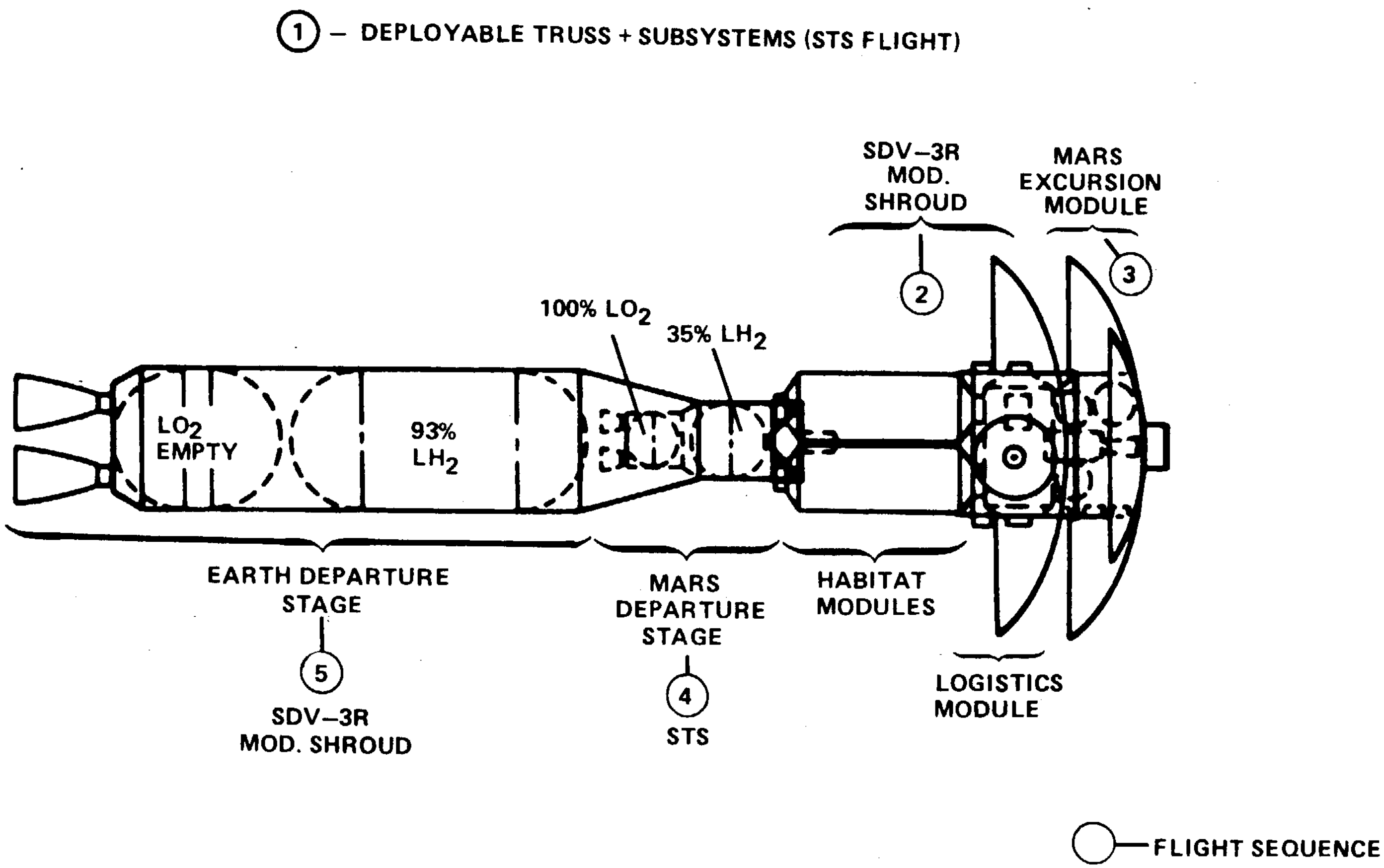
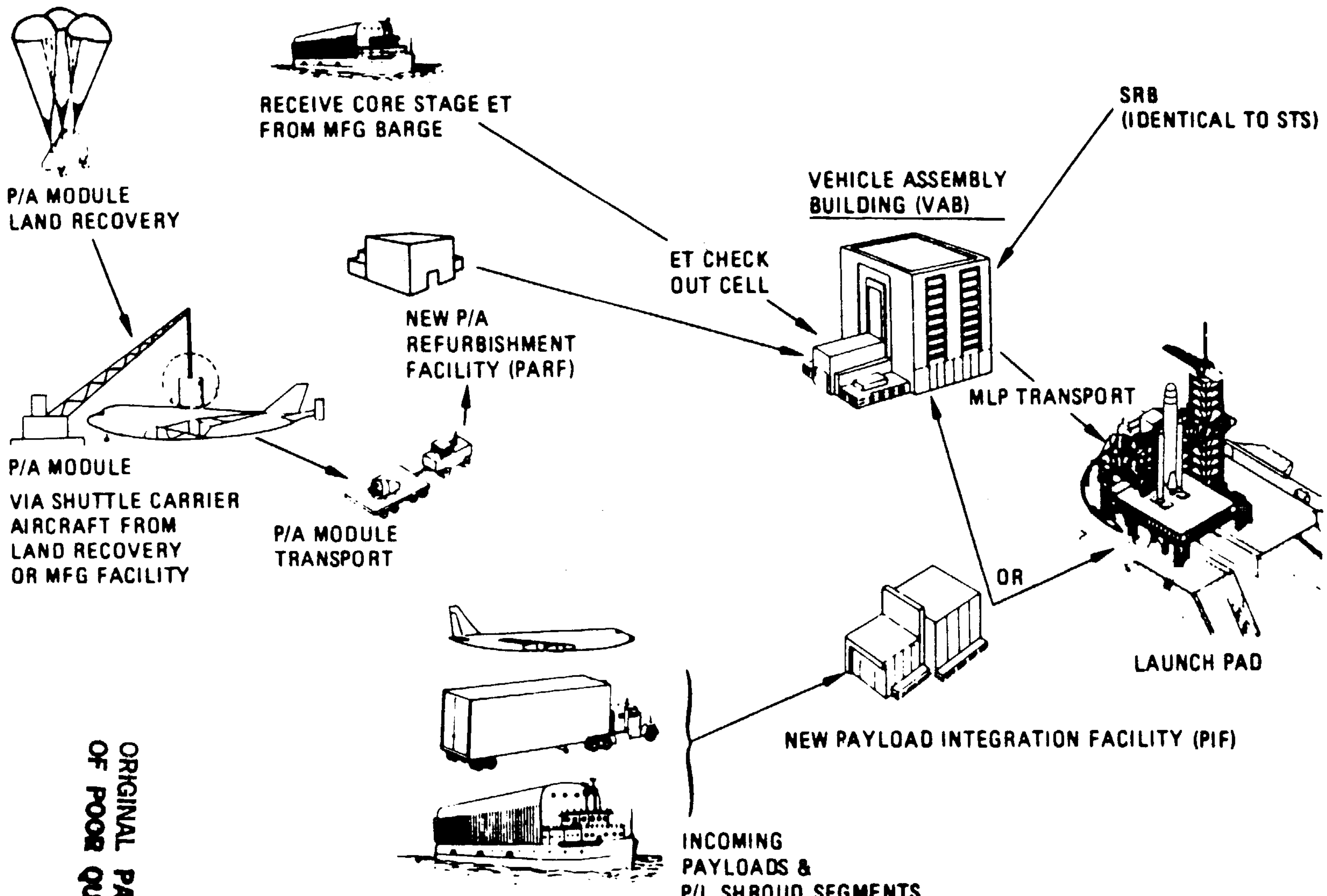


ALL AEROBRAKE CONCEPT
FIGURE 3



MANNED MARS KSC GROUND OPERATIONS FLOW
FIGURE 4

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ASSUMPTIONS

The assumptions given here are applicable to both configurations. The vehicle buildup crews would be transported to orbit and returned by the Space Shuttle. More than one crew may be necessary due to specialized requirements, such as propellant transfer, electrical or mechanical operations and possiby due to fairly long assembly times. The habitat modules would be used by the buildup crews and would be refurbished for the flight mission, if assembly times can be kept reasonable.

The Shuttle-derived Vehicle (SDV-3R) would be the primary vehicle for hardware and propellant deliveries. The aerobrake(s) would be deployable for SDV-3R payload integration and ETO delivery.

Based on the studies performed by the Martin Marietta Corp. (May 1985) on the KSC ground operations (Reference 3), the launch frequency of the SDV-3R is six (6) per year for minimum impact to the KSC operations. An increased launch frequency would require facilities beyond those presented (Figure 4).

The facility (Assembly System) for orbital assembly of the Mars vehicle was conceptually viewed as an erectable or deployable structure with integral subsystems capabilities, derivable from the Space Station (SS) as discussed in Reference 4. The subsystems required are: (a) Atitude stabilization, (b) Communication and data handling*, (c) Electrical Power*, (d) Mobil RMS (MRMS) or equivalent, and (e) Crew aids (lighting, restraints, tools, etc.) The post-assembly disposition alternatives for the Assembly System and associated equipment are: (a) Leave in orbit for future applications (e.g., other Mars vehicles or growth station), (b) Transfer to Space Station via Orbital Maneuvering Vehicle/Orbital Transfer Vehicle (OMV/OTV), and (c) Return to Earth (Requires disassembly if >32,000 lbs). Potential uses of the SS to augment the Mars vehicle assembly are discussed in Reference 4.

ALL-PROPULSIVE CONFIGURATION

The all-propulsive configuration is illustrated in Figure 1. The concept of SDV-3R delivery for the propellant and hardware for the Mars mission vehicle buildup consists of: (a) One (1) Space Shuttle (STS) flight, (b) Eight (8) hardware flights (SDV-3R), and (c) Seventeen (17) propellant flights (SDV-3R).

* Possibly supplied by the Mars vehicle.

Ideally, the vehicle elements would all be delivered "dry" to LEO, would be assembled into the Mars vehicle, then would be loaded with propellant just prior to departure. However, efficient use of the SDV-3R requires "wet" and partially wet launches of these elements. The flight sequence is defined in Figure 1. The STS flight would carry the Assembly System and associated equipment to orbit. The two (2) habitat modules on the SDV-3R would follow or be launched concurrently with the STS flight. These modules would be used for the buildup-phase crew quarters and would later, if necessary, be refurbished prior to the scheduled mission. The STS crew could assist the buildup crew in the initial setup of the Assembly System. The remaining seven (7) illustrated hardware/propellant deliveries have been derived based on the SDV-3R capability and are listed as follows: (1) Logistic module + one (1) fully loaded LH_2 tank; (2) Lander + one (1) fully loaded LH_2 tank; (3) Mars Arrival/Departure Stage engine + one (1) fully loaded LH_2 tank and one (1) partially loaded (approx. 23%) LO_2 tank; (4) Earth braking stage (fully loaded LH_2 + LO_2 tanks); (5) Partially loaded (5%) LO_2 tank + engines (LEO Departure Stage); (6) Fully loaded LH_2 tank for Lower Earth Orbit (LEO) Departure Stage; and (7) Fully loaded LH_2 tank for LEO Departure Stage. The seventeen (17) propellant flights (163,800 lbs/flight) required to fill and replenish boiloff of the vehicle tanks may be meshed with the above hardware delivery flights for optimization. As previously stated, the maximum launch rate of SDV-3R vehicles on a minimum impact basis to the KSC facilities is 6 vehicles per year. Hence, delivery to LEO of the Mars vehicle hardware elements alone would require 14 months. Based on 17 required propellant flights, an additional 32 months would be required, but so much additional boiloff would occur over this time period, the vehicle may never get fully loaded. Obviously, this is not a viable approach.

ALL AEROBRAKING CONFIGURATION

The all aerobraking configuration concept is illustrated in Figure 3. This configuration saves approximately 2 million pounds over the all-propulsive configuration. The concept for ETO delivery of hardware and propellant consists of: (a) Two (2) Space Shuttle (STS) flights; (b) Three (3) hardware flights - SDV-3R (2 of the 3 flights will have a modified shroud to accomodate the larger diameter/length of the payload);

and (6) Six (6) propellant flights - SDV-3R. The Assembly System would be delivered to orbit by the STS as in the all-propulsive configuration. The second STS flight would deliver the second stage (OTV size). The SDV-3R would require (2) flights with a modified shroud to deliver the habitat module, logistics module and one aerobrake as one flight, and the first stage of the Mars vehicle as the other. The Mars excursion module (2 aerobrakes) would be the third SDV-3R flight.

This configuration can be delivered to orbit in approximately 1 1/2 years as compared with approximately 4 years for the all-propulsive configuration, based on the limitaion of 6 SDV flights per year. This analysis includes 172,800 lbs of boiloff propellant. Some expansion of the facilities at KSC and acquisition of additional SDV hardware could increase the launch frequency. If an HLLV, as described in reference 2 (see Figure 5) is used instead of the SDV-3R, the situation would be further improved. Using HLLV's, the total number of flights to deliver the Mars vehicle elements (all-aerobrake) to LEO would be 4 flights, of which 2 are for hardware/propellant and 2 are for propellant only. If one flight were available every 2 weeks, the delivery time spans would be 6 weeks for hardware and propellant. The significant time advantage of using an HLLV is readily apparent from these figures. Other related advantages are that larger segments of the Mars vehicle can be delivered at a time, reducing the on-orbit assembly, integration, and checkout effort and time required. The developement of a completely new vehicle and related facilities may be required, however, unless these were developed as part of other NASA programs or other agencies' activities.

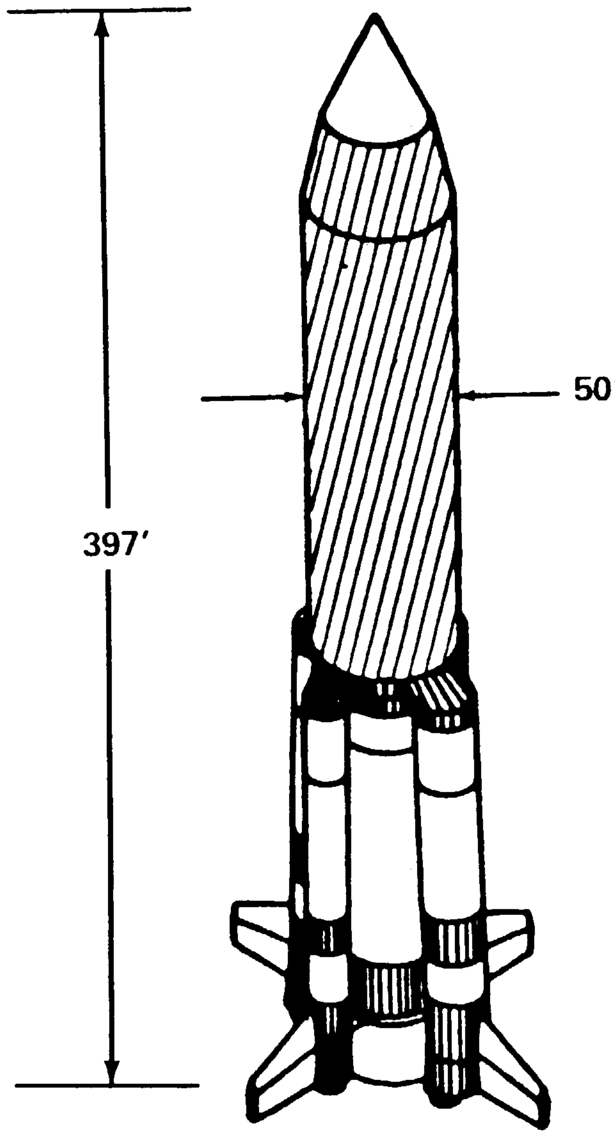
GROUND OPERATIONS

A conceptual ground operations flow is established for the SDV-3R vehicle. This concept is based on a minimum impact to KSC, avoiding new launch facilities. Six (6) SDV flights per year can be accomplished, resulting in approximately 4 years for hardware delivery for the all-propulsive concept and 1 1/2 years for the all aerobrake concept. The ground flow requires a new P/A facility and payload integration facility.

FUTURE STUDY CONSIDERATION

Items which require future study are: (a) Methods/procedures for propellant transfer from the ETO vehicle payload tanks to the Mars vehicle; (b) Disposition of the ETO vehicle (Mars vehicle propellant)

FIGURE 5. HEAVY–LIFT LAUNCH VEHICLE (HLLV)
(RECOVERABLE PROPULSION AND AVIONICS (P/A) MODULE)



- REUSABLE P/A MODULE
  - 5 X STME'S @ 100%
- 4 X LIQUID BOOSTERS (RECOVERABLE)
  - 2 W/2 X 1.616 MLBF ENGINES
  - 2 W/1 X 1.616 MLBF ENGINE
- 33 FT. DIA. EXPENDABLE CORE STAGE TANKAGE
- USABLE P/L ENVELOPE – 45′ X 200′
- PERFORMANCE*
  - 270 NM/28.5^{o} – 401K
  - 160 NM/28.5^{o} – 418K
  - 540 NM/90^{o} – 302K
- IOC 1995

* ORBITS CIRCULARIZED WITH KICK STAGE

tanks; (c) Assembly system and it's subsystems configuration; (d) Disposition of the assembly system after Mars mission departures; (e) Vehicle assembly optimization and procedures; (f) Berthing procedures; (g) Procedures to transfer payload from ETO delivery vehicle (SDV-3R) to Assembly System; (h) Increased launch frequency impact on KSC; (i) Schedule for buildup crews (may not be required for duration between deliveries); and (j) Trades of on-orbit-deployable vs. on-orbit-assembleable aeroshells.

SUMMARY

The all-propulsive Mars vehicle is not pracical to utilize if the SDV-3R ETO vehicle must be used, due to the extensive number of ETO delivery flights for propellants and hardware and the time it would take to assemble and load the vehicle. Obviously, a prefered approach for ETO delivery and on-orbit assembly of the Mars vehicle would be to use an all-aerobraking vehicle and deliver its elements to LEO with the SDV-3R. The ETO delivery of the Mars vehicle concepts could be shortened by expansion of the KSC facilities. Use of the HLLV for the ETO delivery appears most desireable except that a new vehicle would need to be developed with costly new facilities. However, if the HLLV vehicles and facilities costs could be shared with other programs, it would be of significant benefit for the Mars mission ETO delivery.

REFERENCES

1. Paper in Section III of this report, entitled "Space Vehicle Concepts," by M. Tucker, O. Meredith, and B. Brothers of MSFC.
2. Paper in Section III of this report, entitled "Earth-to-Launch Vehicles for MMM Application," by M. Page of MSFC.
3. Final review of "Advanced Space Transportation Systems Ground Operations Study Extension," by Martin Marietta for KSC Operations, May 1985, Contract # NAS.10-10572.
4. Paper in Section IX of this report, entitled "Space Station Utilization and Commonality," by J. Butler of MSFC.

N87-17747

SPACE VEHICLE CONCEPTS

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Oliver Meredith
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ABSTRACT

This paper presents several concepts of chemical-propulsion Space Vehicles (SVs) for manned Mars landing missions. For vehicle sizing purposes, several specific missions were chosen from opportunities in the late 1990's and early 2000's, and a vehicle "system" concept is then described which is applicable to the full range of missions and opportunities available. In general, missions utilizing planetary opposition alignments can be done with smaller vehicles than those utilizing planetary opposition alignments (reference 1) . The conjuction missions have a total mission time of about 3 years, including a required stay-time of about 60 days. Both types of missions might be desirable during a Mars program, the opposition type for early low-risk missions and/or for later unmanned cargo missions, and the conjunction type for more extensive science/exploration missions and/or for Mars base activities. Since the opposition missions appeared to drive the SV size more severely, there were probably more cases examined for them.

Some of the concepts presented utilize all-propulsive braking, some utilize an all aerobraking approach, and some are hybrids. Weight statements are provided for various cases. The aerobraking cases have significant advantages in size and weight. Cryogenic propellants were used for the main propulsive elements in all cases, due to their significant weight advantage over storable propellants (reference 1). Extensive use is made of existing propulsive elements and other systems.

Most of the work was done on 0-g vehicle concepts, but partial-g and 1-g concepts are also provided and discussed. A recommendation is added that efforts be made to find ways to offset the long-term 0-g effects on the crew, other than providing a g-field for the total SV or spacecraft, since this causes significant design and operations impacts.

Several options for habitable elements are shown, such as large-diameter modules and Space Staion (SS) types of modules. The latter were used as a reference because of their cost advantage as existing elements.

Several options are shown for the Mars landing vehicle, and a landing "system" is recommended which makes use of a large aeroshell to allow landing of payloads of various sizes and shapes over the course of a multi-year program.

Because of the large size and weight of the SV it will be necessary to launch individual elements and assemble them in low Earth orbit (LEO). A configuration of one potential assembly concept is provided.

ALL-PROPULSIVE OPTION

Figure 1 illustrates an all-propulsive option which is sized for propulsive braking maneuvers (no aerobraking at Mars or Earth return) using LO_2/LH_2 propellants. This vehicle is sized for the 1999 mission opportunity, using an opposition-type trajectory. The concept utilizes 3 propulsion stages for the mission which accomplish LEO departure, Mars arrival and departure, and Earth braking, respectively. The stages are jettisoned after use, including jettison of external hydrogen tankage prior to departure from Mars. This figure also provides the terminology used for the configuration elements. Figure 2 illustrates the concept at different stages during the mission. In the Earth-Mars transit phases, the normal vehicle orientation is with its long axis towards the sun, to minimize propellant boiloff losses. Other orientations can be effected occasionally, as long as they are kept within reasonable limits.

The stage sizing and tank arrangements were influenced by the size and delivery capability of the launch vehicle used for delivery of elements to LEO, with a significant amount of on-orbit propellant transfer necessary to fill the propellant tanks.

The engines for the first stage are Shuttle-derived Space Transporation Main Engines (STME's), as defined in reference 2. The first stage tanks are derivable from the SDV-3R Earth-to-orbit (ETO) vehicle (see reference 3) or from the Shuttle External Tank (ET). The second and third stage engines are Orbital Transfer Vehicle (OTV) - derived RL-10 engines, as defined in reference 2. The second stage tankage should be derivable from the SDV-3R (or ET) and the OTV, and the third stage tanks should be derivable from the OTV. The tanks are insulated with 4 inches of multilayer insulation and are outfitted with vapor-cooled shields, to

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FIGURE 1
MANNED MARS MISSION
1999 OPPOSITION
ALL PROPULSIVE OPTION

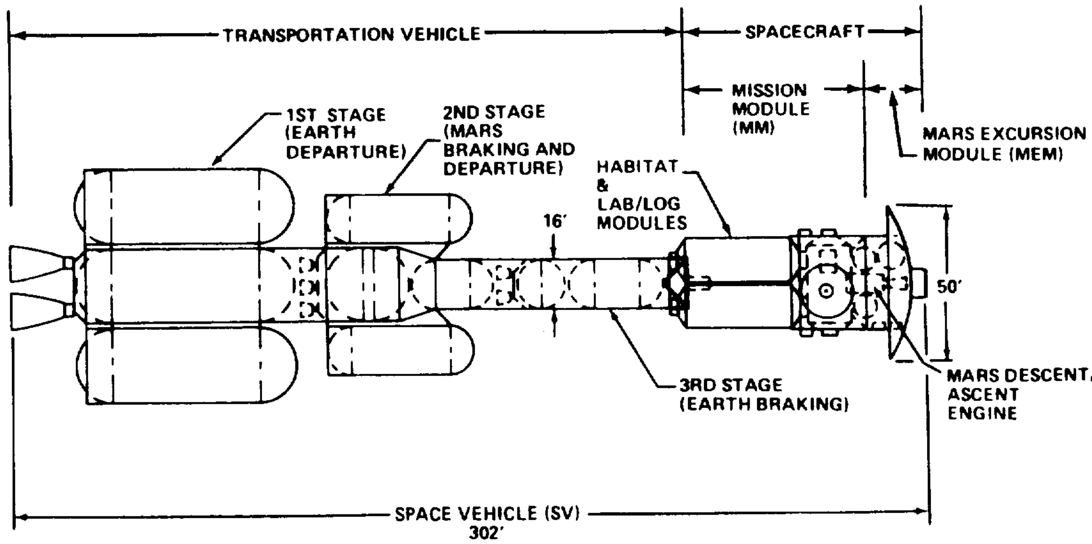

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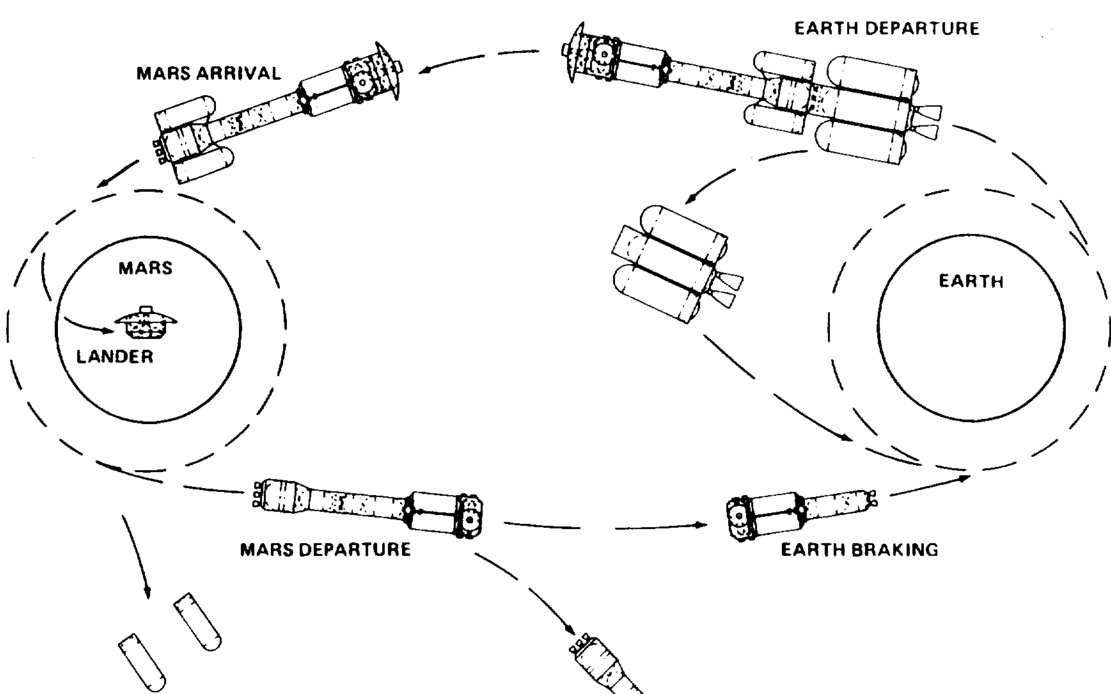
FIGURE 2.
1999 OPPOSITION
MANNED MARS MISSION
ALL PROPULSIVE STAGING



minimize cryogen boiloff for each stage. A discussion on insulation thickness trades is provided in reference 4.

The spacecraft portion of the vehicle consists of the Mission Module (MM), (which includes 3 Space-Station (SS)-type modules), the Mars Excursion Module (MEM), (which consists of a lander and ascent stage for the Mars surface), and experiments and experiment probes for deployment during the mission. Weight of these elements is important because of the effect it has on propulsive stage sizing (particularly the round trip portion). The SS-type modules shown in the MM include 2 Habitability Modules and a Laboratory/Logistics Module, as modified for the Mars mission. The MM remains in Mars orbit with a crew of 2 persons, while the MEM descends to the surface with a crew of 4, during opposition missions; all 6 crewmen would descend to the surface during a conjunction mission.

Most Spacecraft subsystems technology/designs were assumed to be the SS-type, for sizing and costing purposes. Although SS modules and subsystems are still in a very early stage of definition, it appears that a closed-loop (except for the food loop) ECLSS will be used there. The Spacecraft power source was asssumed to be a Radioisotope Thermoelectric Generator (RTG) - type (non-SS), operating at a power level of 25kw during the transit phases (MEM and MM systems active) and having 10kw for the surface phase (MEM).

The spacecraft concept shown is based on a "0-g" in-transit environment for the crew, which provides the simplest configuration approach. Several options considered for the MM are illustrated in Figures 3 and 4. Figure 3 is provided primarily to show the relative size comparison of a single module concept from reference 5 with a twin SS module concept having approximately equal volume. The volumes shown here are not adequate for the Mars mission currently being discussed. Also, the single module from reference 5 provides no safe haven volume in case of emergency. A large tunnel could be installed down the center of the single module to provide such a region. The larger-diameter module has advantages in volumetric and weight efficiency, and probably allows better utilization of the basic equipment weight for radiation shielding. However, it would be a new design, and would not allow as much cost-savings benefit as the concept which utilizes SS modules.

FIGURE 3. MISSION MODULE OPTIONS

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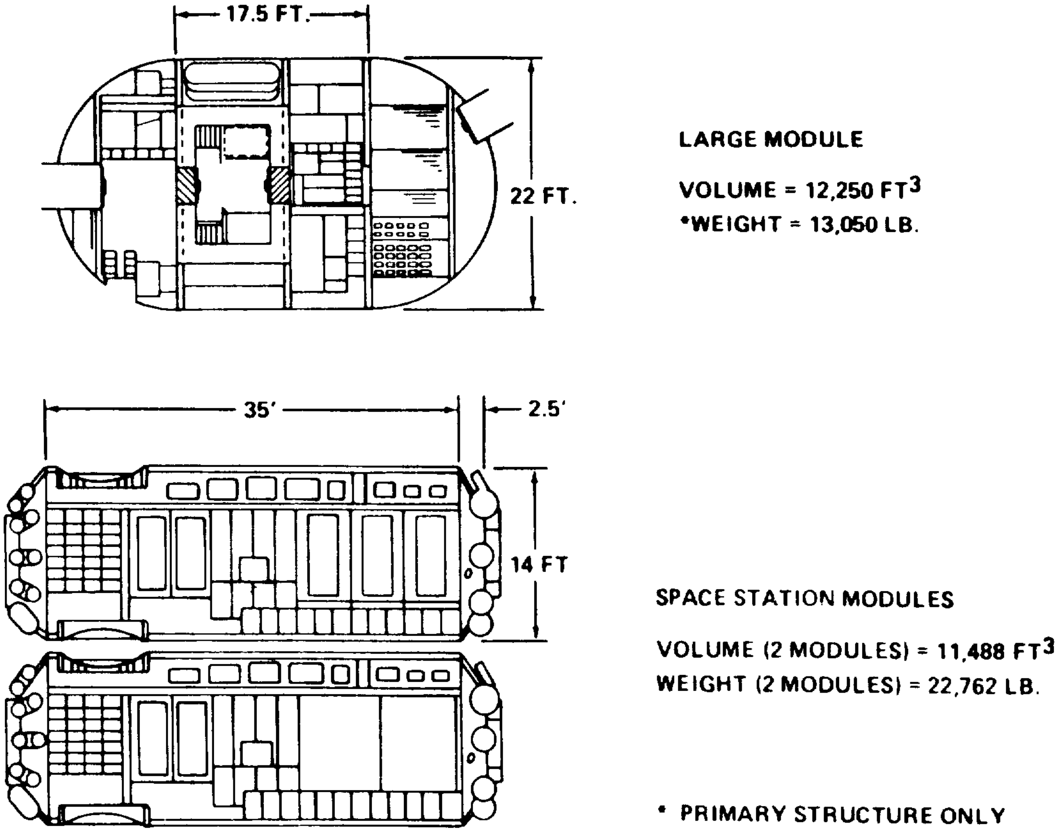

FIGURE 4. MANNED MARS HAB MODULE CONCEPT

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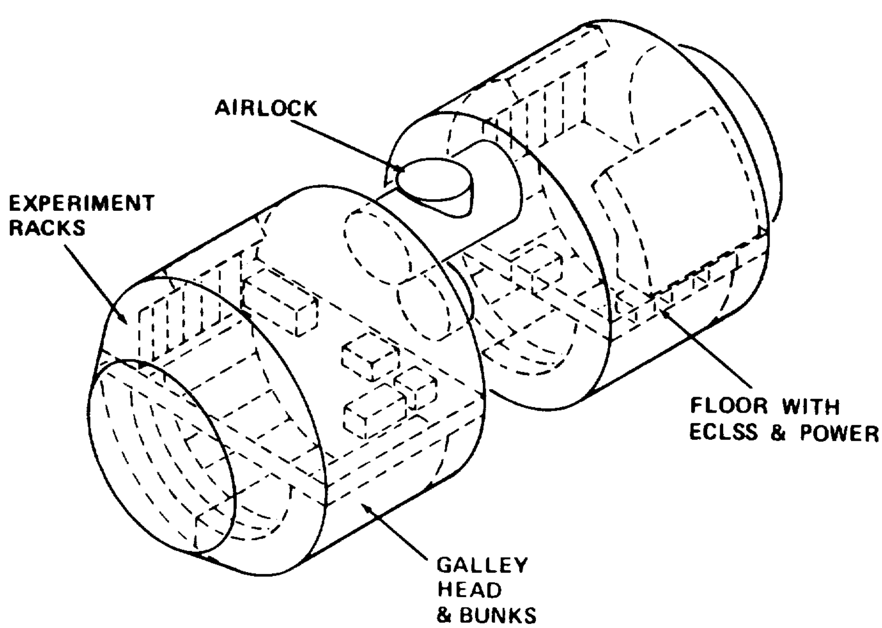

Figure 4 provides a concept which uses 2 end-to-end large-diameter modules. The modules shown here utilize a floor across their midsections which would house much of the ECLSS, power and other required equipment, leaving the cylindrical walls free for experiments, bunks, and other facilities. The EVA airlock rests between the two modules, granting access from both. For all options, it was assumed that a minimum of 2 separate pressurized compartments was necessary in case of an incident that required evacuation and isolation of an area.

As discussed later, preliminary calculations showed that the total spacecraft systems mass should be sufficient to provide adequate protection from background radiation and solar flares, if its distribution could be effected properly. Such detailed layout activity was beyond the scope of this study, but such an approach seems feasible. This consideration would necessitate packaging most of the spacecraft equipment around the walls of the pressure vessels, for maximum shielding effectiveness. Retention of expended propulsive stages during the long coast phases of the mission may also benefit the radiation protection for the crew. Packaging of fluids such as propellants (especially H_2) and water around the habitable modules would add significantly to the radiation protection, but no viable concept of this sort has been developed yet. Boil-off, tank weight, interfaces/integration, and module visibility are difficulties associated with such a concept. Figure 5 depicts the spacecraft used as a reference for this study. It provides more details on the MM concept utilizing SS modules. Three modules are required to provide the necessary volume for the Mars mission. Figure 5 also provides details of the MEM. The MEM consists of a descent stage which stays on the Mars surface and an ascent stage for return of the crew and samples to Mars orbit for rendezvous with the MM. Existing solid rocket de-orbit motors as defined in reference 2 are used for de-orbiting the MEM prior to Mars landing. An entry heat shield is provided for deceleration and protection during entry, and propulsive braking and attitude control are used for landing. The pressurized portion of the ascent stage is occupied by the 4-person crew during the Mars entry and landing. Descent engines are arranged such that one is subsequently reused for ascent. (Liquid oxygen and monomethylhydrazine (LO_2/MMH) propellant is used.) These engines are defined in reference 2, and would be a new

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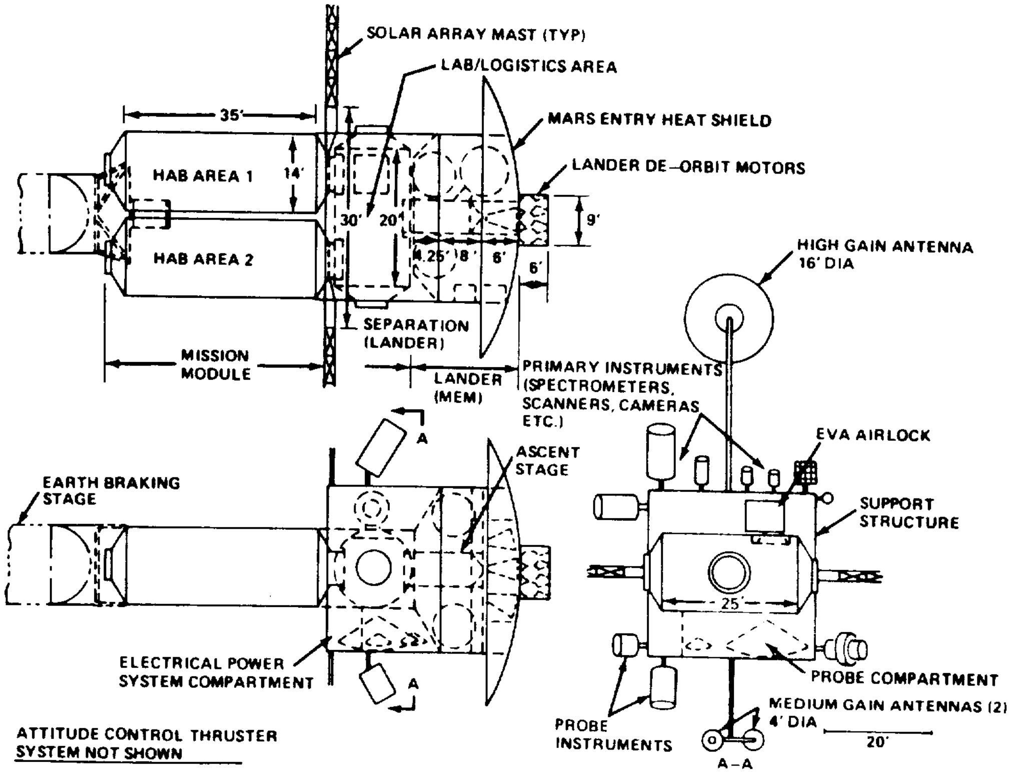
FIGURE 5. MANNED MARS MISSION SPACECRAFT CONCEPT (SPACE STATION HABITAT MODULES)



ATTITUDE CONTROL THRUSTER SYSTEM NOT SHOWN

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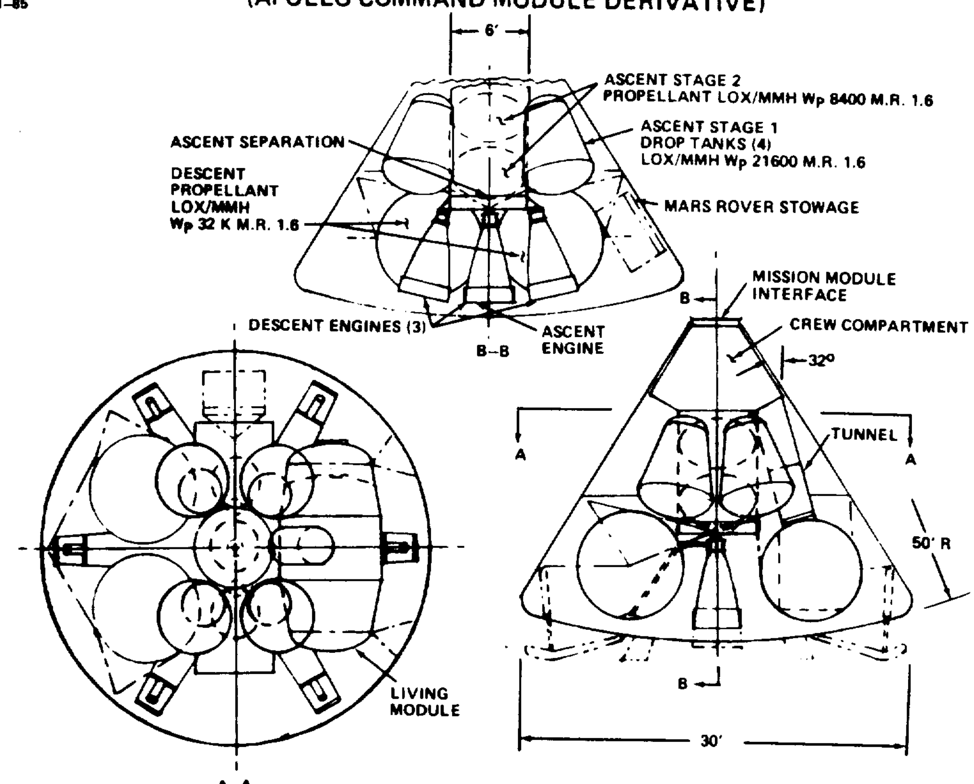
FIGURE 6 MARS EXCURSION MODULE CONCEPT (APOLLO COMMAND MODULE DERIVATIVE)



PRELIMINARY MEM SUBSYSTEMS INCOMPLETE

design. The lander portion of the vehicle includes a pressurized crew module/laboratory, experiments, and exploration provisions (including surface mobility provisions such as a rover vehicle having power, communications, and thermal control capability). EVA capability is provided from the crew module. Upon completion of the surface mission, the crew and samples return to Mars orbit in the ascent stage, leaving most of the landed mass on the surface. After rendezvous with the orbiting vehicle, the crew and samples are transferred and the ascent stage is jettisoned prior to Mars orbit departure.

Figure 6 depicts a MEM option which is a derivation of the Apollo Command Module, and is a modified version of a concept from reference 6. This concept imposes severe packaging shape and size/weight constraints on the equipment and habitability volumes necessary to be transported to the surface, particularly that for longer-duration missions. Such a concept might suffice for very limited early missions, but would be dead-ended from a growth standpoint.

In contrast, the large aeroshell approach previously showm (Figure 5) allows implementation of a surface delivery "system" concept, wherein the aeroshell is used to accommodate small or large payloads, with minimum impact on their shape, size, or weight. A cylindrical shell is shown behind the aeroshell to serve as a heat shield, but this item may not be required.

ALL-AEROBRAKE OPTION

An all-aerobrake option of the Manned Mars Space Vehicle is shown in Figure 7 for the 2001 opportunity, using an opposition-type trajectory. This concept utilizes the same spacecraft as the all-propulsive versions, but uses aerobraking instead of propulsive braking for Earth and Mars capture. This design, therefore, uses much less propellant and has a much lower weight (discussed later) at Earth departure than the all-propulsive version. Aerobraking concepts were assumed to be derivatives of those utilized for the OTV and STS concepts. The OTV is expected to be operational in the mid-to-late 1990s.

The first stage is expended after departure from Earth and is returned to LEO (Figure 8). The propellant tanks of the first stage were sized to take advantage of current hardware; the diameter and bulkheads have commonality with the STS External Tank. The second stage can also

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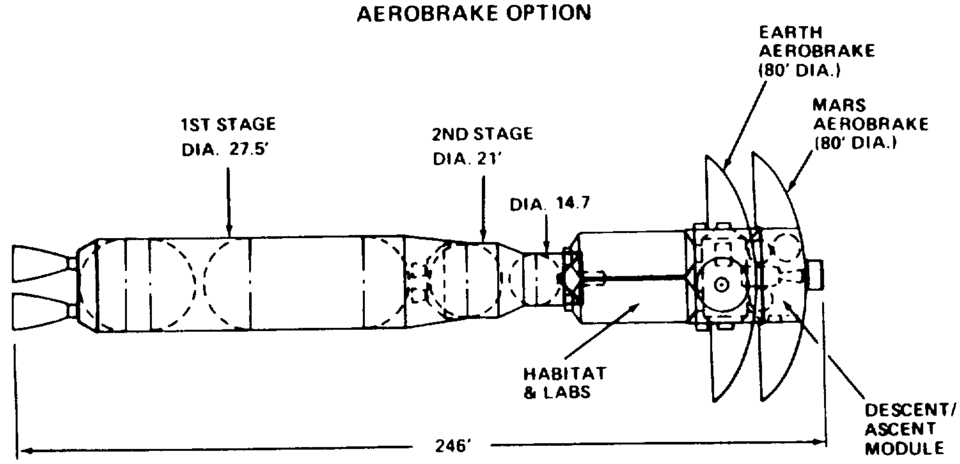
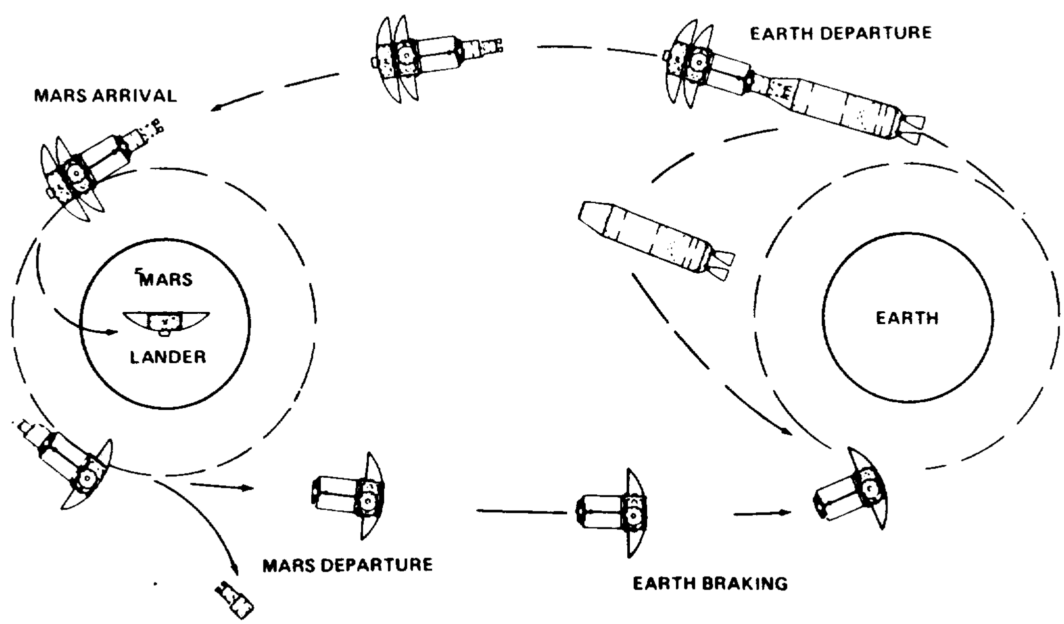
FIGURE 7.
2001 OPPOSITION
MANNED MARS MISSION
AEROBRAKE OPTION



3574-85

FIGURE 8.
2001 OPPOSITION
MANNED MARS MISSION
AEROBRAKE STAGING



make use of then-existing designs, specifically OTVs. Of course, both stages can grow by adjustments to their cylindrical lengths. As with the all-propulsive vehicle, the first and second stages utilize engines derived from existing (or then-existing) vehicles (Shuttle and/or SDV-3R, and OTV). An 80 ft. diameter aerobrake provides the braking for Mars arrival. This aerobrake can be jettisoned, revealing a separate 50 ft. heat shield for the MEM, or only part of the aerobrake may be jettisoned reducing it to a reusable 50 ft. diameter heat shield for the MEM. Another option is to reuse the entire 80 ft. diameter aeroshell for the MEM heat shield. A third option is to reuse the 80 ft. aeroshell for Earth braking, and provide a separate 50 ft. heat shield for the MEM.

As shown in Figure 8, once the MEM ascent stage returns to the MM in Mars orbit, the crew and cargo are transferred, and the ascent stage is jettisoned. The second propulsive stage provides Mars departure velocity and is discarded. The vehicle then attains Earth orbit with the use of the 80 ft. diameter Earth-braking shield.

HYBRID OPTION

Another option is a hybrid vehicle which uses aerobraking at Mars and then propulsive braking for Earth return (Figures 9 and 10). The same spacecraft as utilized in the other options was also used here, except as noted below. This vehicle is sized for the 1999 opportunity, using an opposition-type trajectory. Utilizing an opposition-type trajectory at this opportunity results in an energy level which will produce a high g-level if the total spacecraft is aerobraked into Earth orbit. The crew may be especially susceptible to g-level effects if they have been in a reduced-g or 0-g field for a long period of time. To keep the g-level within acceptable bounds (estimated to be about 3g to 5g) for the crew, it is necessary to do propulsive braking just prior to Earth orbit entry. However, if the entire spacecraft is propulsively braked, the addition of a fairly large 3rd stage and significant growth in the first and second stages would be required. An alternative approach, used for this concept, was to retain the MEM ascent stage, to jettison the MM near Earth, then propulsively brake only the MEM ascent stage using MEM engines or a small third stage. Once the energy level is reduced to this acceptable limit, very little additional propulsive braking would be required to brake into Earth orbit. This approach was selected rather

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FIGURE 9.
1999 OPPOSITION
MANNED MARS MISSION
AEROBRAKE OPTION

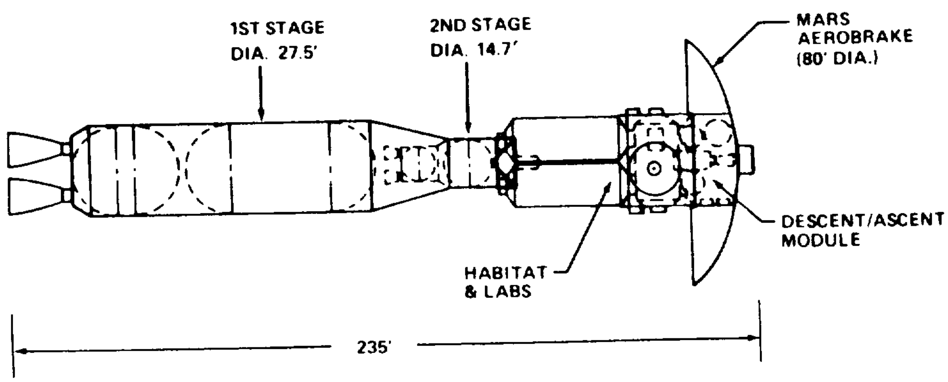

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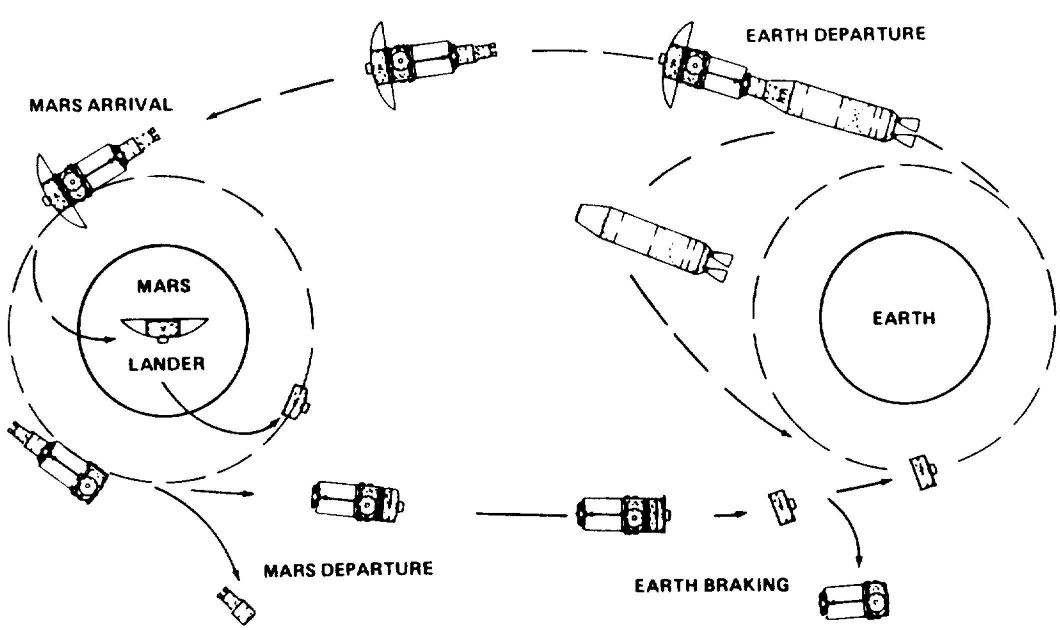
FIGURE 10.
1999 OPPOSITION
MANNED MARS MISSION
AEROBRAKE STAGING



than aerobraking for this configuration. Weights are considerably lower using this option than using the all-propulsive vehicle.

SV "SYSTEM"

The concepts described above for the 1999 and 2001 missions are summarized graphically in Figure 11. The conjunction-type missions are generally easier to accommodate configuration-wise than opposition missions. (See references 1, 7, 8 and 9). This is especially true for all-propulsive vehicles. However, the use of aerobraking concepts allows much easier accommodation of opposition missions, and allows development of a vehicle "system" which can perform either oppostion or conjunction missions at any opportunity and which can be used for manned or unmanned payloads (see references 8 and 9). About 65-70% of the opposition missions do produce acceptable g-levels when aerobraking is used at Earth.

The large aeroshells delivered to the Martian surface may provide useful structures for habitation or storage. Much of the aerobraking technology required should be developed as part of the OTV program, now in progress.

The 3-year (conjunction) missions allow a one year or so stay at Mars, which offers science benefits and may be more useful for more mature, Mars-base-era operations. However, the 2-year (opposition) missions, with their 60-day or so stay time at Mars, may be more attractive for earlier and/or simpler missions, or for unmanned cargo or other flights in the later timeframes. The "system" identified herein appears to offer a good bit of versatility to the user, for any of these applications.

The greatest contribution that the vehicle designer might make to the program is to provide a high degree of versatility to accomodate various mission and program options, at reasonable cost. Thus, an early flyby mission might be accomplished readily, and yet, the elements selected for such a mission would not be dead-ended, but would serve efficiently for follow-on exploration and utilization.

Some of the critical ingredients of such a vehicle system will be modularity and technology transparency. Vehicle designs must have multiple stages, add-on tanks, etc., to be able to accommodate greater payloads (or the same size payloads in years having less favorable opportunities), and must be able to incorporate newer technology systems as

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FIGURE 11.
AEROBRAKING OPTIONS
FOR 1999 AND 2001 OPPORTUNITIES

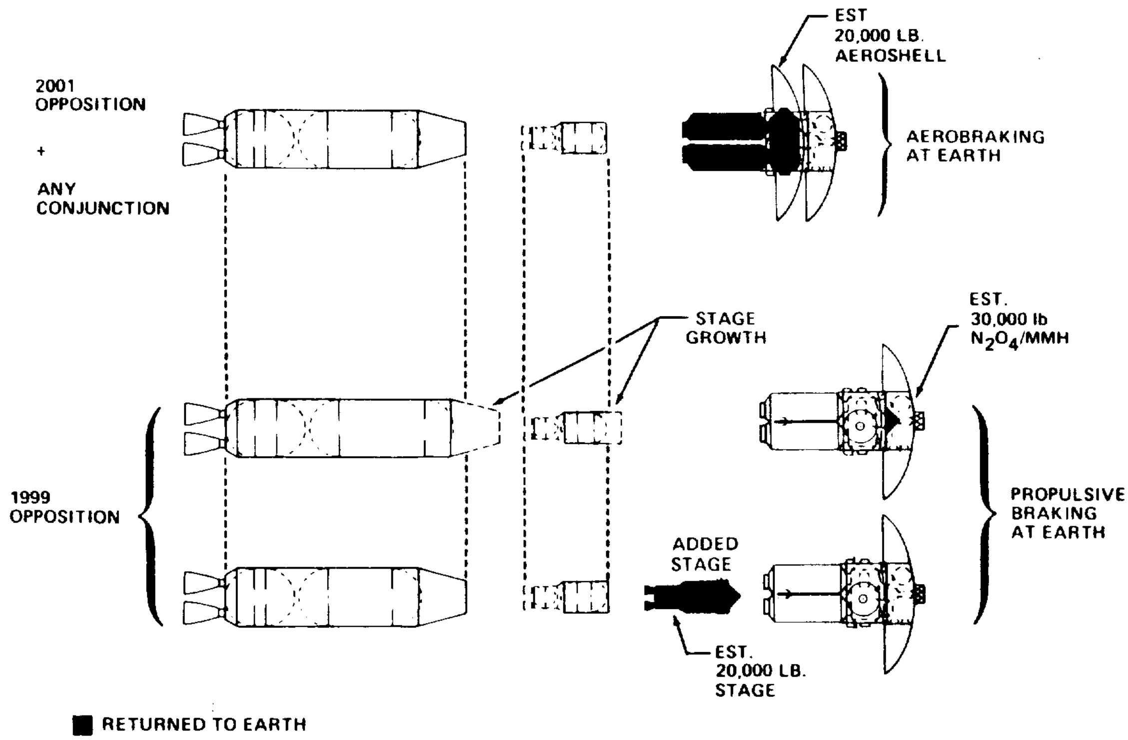

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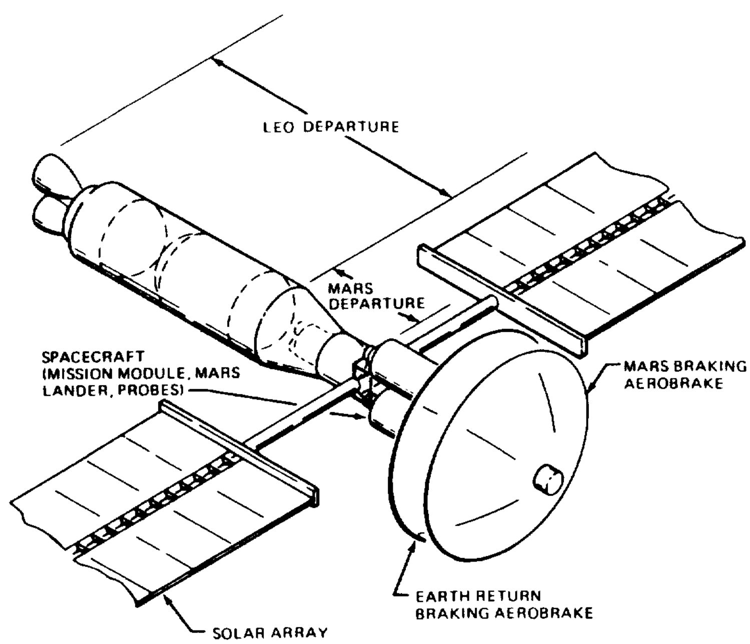
FIGURE 12.
ALL–AEROBRAKING MARS SPACE VEHICLE
UTILIZING SOLAR ARRAY



they become available with minimum impact on the rest of the vehicle. The vehicle should have adaptability to either manned or unmanned (cargo) missions, with minimum impact.

Figure 12 depicts an all-aerobraking concept which makes use of a solar array as part of the MM. The relative size of the solar array wings compared to the other elements can be seen here.

ON-ORBIT ASSEMBLY

Figure 13 dispicts one potential configuration of the SV undergoing on-orbit assembly in LEO. Here, a free-flying assembly "system" is being used, but other options range from using no assembly system to using the SS as the assembly system. References 12 and 14 provide futher discussion of assembly options. The assembly system shown here consists of a piece of the SS truss structure, including SS Attitude Control System elements and the Mobile RMS (MRMS).

GRAVITY-FIELD CONCEPTS

Some solution must be found to ameliorate the deleterious effects on the crew of long-term weightlessness. Hopefully, solutions to this problem will not require the total SV to provide a gravity field. While not impossible to do, this adds complexity to the SV which should be avoided unless absolutely necessary. If artificial-g is required, it might be acceptable to have less than 1g, but this is unknown. Configurations providing several different g-levels have been investigated, and some of these are discussed below.

Physiological constraints limit the rotation rate to a maximum of 4 RPM (reference 10). The spacecraft must thus have a radius of rotation of 200 ft. in order to obtain 1g acceleration (see Figure 14). This vehicle is based on the all-propulsive version, with the addition of two 200 ft. arms to support the MM and MEM. These arms would most likely be deployable beams such as those utilized as Space Station structure. Tunnels would probably be desired between modules, and would be a major difficulty due to their length. Environmental Control and Life Support System (ECLSS) control for the tunnels could be a significant problem. The 2 modules at the end of the 200 ft. arms must be fairly close to the same weight for good balance. The entire spacecraft or just the habitat section could be spun up, but if the entire vehicle is spun, the communication antennas, some science equipment, and possibly the solar arrays

FIGURE 13. ON-ORBIT ASSEMBLY OF MANNED MARS SPACE VEHICLE

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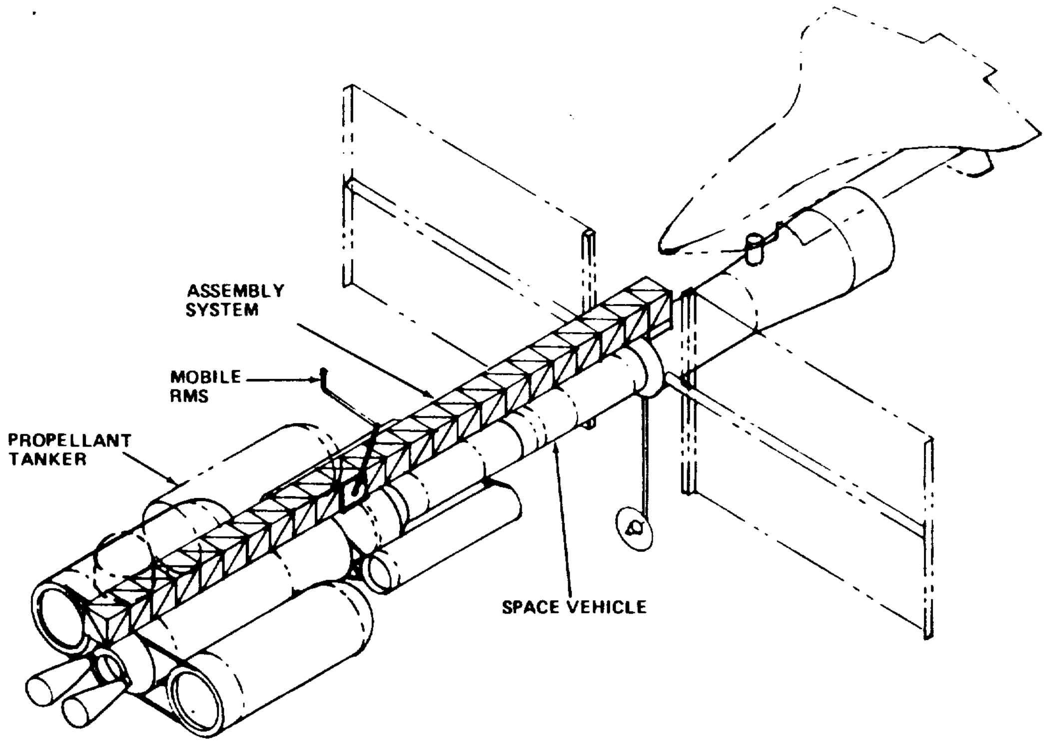

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FIGURE 14
MANNED MARS MISSION
1-G OPTION

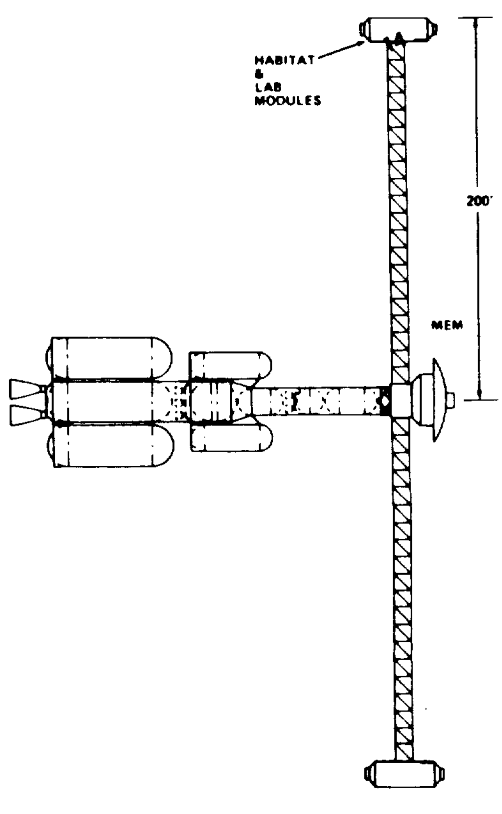
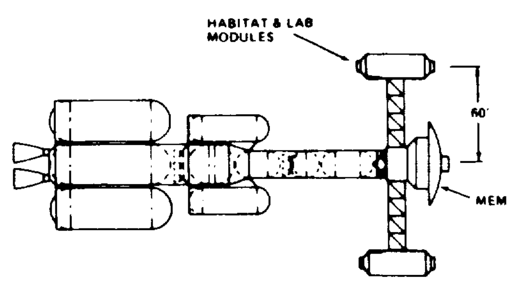


FIGURE 15
MANNED MARS MISSION
0.4-G OPTION



(if used) would have to be despun. Figure 15 depicts a vehicle option designed to generate .4 g radial acceleration. This vehicle is derived from the 1g design, the only change being the shorter 60 ft. radius of rotation.

Mass must be added to the SV for: (1) the RCS system required for spinup and maintenance of the spin rate; (2) the truss structure supporting the modules; and (3) the tunnels and their ECLSS equipment, additional shielding weight, etc.

Design and operational complexities are introduced since: (1) efficient utilization of the habitable environment is difficult due to the distances involved; (2) frequent traversing between modules would tend to produce sickness due to the varying g-levels experienced, (3) systems and living quarters would have to operate and be functional in 0g, partial g, and 1g environments, with the latter two involving two different g-force directions (ground and on-orbit); (4) some of the modules and other elements would have to be relocated to the region behind the aeroshell of an all-aerobraking concept for capture at Mars and Earth; and (5) EVA activities would necessitate stopping the rotation. The booms may have to be adjustable length-wise to balance the changing masses as the configuration changes over the two-or three-year length of the mission.

Some elements of the SV (astronomy instruments, guidance sensors, etc.) would have to be de-spun to allow their proper operation and others (appendages, etc.) would have to be stiffened to withstand the g-forces.

WEIGHTS

Weight summaries for four different manned Mars propulsion vehicles are shown in Tables 1 through 4. Propellant weights are from reference 1. Weights are included for interstages and payload adapters to connect stages together as well as for the spacecraft propulsive vehicle and crew. The number of engines in the propulsion system is shown in parentheses for each stage. The avionics weights for the propulsive stages are minimal, since the main avionics system would be in the spacecraft. A fifteen percent contingency is added to all the dry weights, since most of the hardware is new and considered to be current technology equipment. Boiloff propellants are included for the vehicle after Earth departure only, since it was assumed the propellants could be "topped off" just prior to Earth departure. The aerobrake/heat shield weight for the MEM

3300–85

TABLE 1
WEIGHT SUMMARY (POUNDS)
ALL PROPULSIVE CRYOGENIC VEHICLE FOR 2 YEAR 1999 OPPOSITION MISSION

	1ST STAGE EARTH DEPARTURE	2ND STAGE MARS ARRIVAL & DEPARTURE	3RD STAGE EARTH BRAKING
PROPELLANT TANKS	37529	13083	3650
STRUCTURES	11436	7524	2030
INSULATION & VAPOR COOLED SHIELDS	23996	12865	3520
ENGINES & PROPULSION SYSTEM	(2) 24903	(5) 6737	(2) 2293
AVIONICS (MINIMAL ONLY)	800	500	200
CONTINGENCY (15%)	14800	6106	1754
RESIDUALS	8948	4101	1278
SUBTOTAL BURNOUT WEIGHT	122412	50916	14725
BOILOFF PROPELLANTS	–	1680	704
USABLE PROPELLANTS	2265472	671420	160222
STAGE LAUNCH WEIGHT (LEO)	2387884	724016	175651
SPACECRAFT (LAUNCH)		291,203	
TOTAL SPACE VEHICLE AT LEO LAUNCH		3,578,754	

3301–85

TABLE 2
WEIGHT SUMMARY (POUNDS)
AEROBRAKING CRYOGENIC VEHICLE FOR 2 YEAR 1999 OPPOSITION MISSION

	1ST STAGE EARTH DEPARTURE	2ND STAGE MARS DEPARTURE	3RD STAGE EARTH BRAKING
PROPELLANT TANKS	21991	1404	267
STRUCTURES	14631	2150	925
INSULATION & VAPOR COOLED SHIELDS	10303	1521	773
ENGINES & PROPULSION SYSTEM	(2) 24213	(2) 1939	(2) 1773
AVIONICS (MINIMAL ONLY)	800	200	200
CONTINGENCY (15%)	10791	1082	590
RESIDUALS	4334	901	425
SUBTOTAL BURNOUT WEIGHT	87063	9197	4953
BOILOFF PROPELLANTS	–	335	105
USABLE PROPELLANTS	902938	50830	15000
STAGE LAUNCH WEIGHT (LEO)	990,001	60,362	20,058
AEROBRAKE FOR MARS ARRIVAL (80 FEET DIA)		38,893	
SPACECRAFT (LAUNCH)		291,203	
TOTAL SPACE VEHICLE AT LEO LAUNCH		1,400,517	

3302–85

TABLE 3
WEIGHT SUMMARY (POUNDS)
AEROBRAKING CRYOGENIC VEHICLE FOR 3 YEAR 1999 CONJUNCTION MISSION

	1ST STAGE EARTH DEPARTURE	2ND STAGE MARS DEPARTURE
PROPELLANT TANKS	18234	1334
STRUCTURES	14631	2150
INSULATION & VAPOR COOLED SHIELDS	8680	1470
ENGINES & PROPULSION SYSTEM	(2) 24115	(2) 1914
AVIONICS (MINIMAL ONLY)	800	200
CONTINGENCY (15%)	9969	1060
RESIDUALS	3730	876
SUBTOTAL BURNOUT WEIGHT	80159	9004
BOILOFF PROPELLANTS	–	1600
USABLE PROPELLANTS	724706	43528
STAGE LAUNCH WEIGHT (LEO)	804,865	54,132
AEROBRAKE FOR MARS & EARTH ARRIVAL (80 FEET DIA.)	38,893	
SPACECRAFT (LAUNCH)	383,510	
TOTAL SPACE VEHICLE AT LEO LAUNCH	1,281,400	

3303–85

TABLE 4
WEIGHT SUMMARY (POUNDS)
AEROBRAKING CRYOGENIC VEHICLE FOR 2 YEAR 2001 OPPOSITION MISSION

	1ST STAGE EARTH DEPARTURE	2ND STAGE MARS DEPARTURE
PROPELLANT TANKS	24381	3959
STRUCTURES	15222	2697
INSULATION & VAPOR COOLED SHIELDS	10734	3273
ENGINES & PROPULSION SYSTEM	(2) 24265	(2) 2287
AVIONICS (MINIMAL ONLY)	800	200
CONTINGENCY (15%)	11310	1862
RESIDUALS	4568	1269
SUBTOTAL BURNOUT WEIGHT	91280	15547
BOILOFF PROPELLANTS	–	705
USABLE PROPELLANTS	977280	157804
STAGE LAUNCH WEIGHT (LEO)	1,068,560	174,056
AEROBRAKE FOR MARS & EARTH ARRIVAL (80 FEET DIA.)	38,893	
SPACECRAFT (LAUNCH)	291,203	
TOTAL SPACE VEHICLE AT LEO LAUNCH	1,572,712	

is included in the MEM weights. The eighty foot reusable aerobrake weight shown for the aerobraking vehicles was estimated and includes heat tiles (Orbiter type). This eighty foot aerobrake could be constructed so that the outer section could be jettisoned and left at Mars, and the remaining part used for Earth aerobraking if a smaller aerobrake is desired.

The MEM propulsion systems are shown in Table 5 for two different concepts. The N_2O_4/MMH (storable) concept is shown as the reference and includes the descent and ascent stages. The number of engines which are included in each stage are shown in parenthesis. All three engines are used during descent to the Mars surface, but only one is used for the ascent phase of the mission. The LOX/MMH option shows a large boiloff of LOX during the 60-day stay on the Mars surface. This boiloff of LOX could possibly be used by the ECLSS or the power system if fuel cells were used, but mission time would be limited. The total MEM propulsion system weights and stage weights are shown at launch from LEO. The deorbit propulsion system (solids) are not included on this chart, but they are included with the spacecraft and payload weights in Table 7.

Preliminary weight estimates for crew consumables are provided in Table 6; totals are given for an opposition (approximately a 2-year mission). The weight summary for the spacecraft for two and three year missions are shown in Table 7; for the 3-year mission, all 6 men go to the surface. The weights are shown separately for the Habitability Module #1, Habitability Module #2, Laboratory/Logistics Module, the MEM, and the Science Probes. The micrometeoroid shield and outer insulation weights are included with the structures. An airlock weight is shown for the Lab/Log Module, and on the same line, an aerobrake/heat shield is shown for the MEM. The main avionics, power, and ECLSS are shown in the Habitability Modules and the MEM. The Lab/Log Module would be supplied power and ECLSS from the Habitability Modules. A fifteen percent contingency is included on all the dry weights, since most of the hardware is new and considered current technology equipment. Spares are included for non-structural weights at three percent per year. Further study and analysis should be done in estimating spares. Fluids, consumables, and propellants are shown separately for each module. The deorbit propulsion system includes extra propellants for limited plane changes and landing

TABLE 5

3304—85

MEM PROPULSION SYSTEM WEIGHT SUMMARY (POUNDS)

	REFERENCE N_2O_4/MMH SYSTEM		OPTIONAL LOX/MMH SYSTEM	
	DESCENT STAGE	ASCENT STAGE	DESCENT STAGE	ASCENT STAGE
PROPELLANT TANKS	287	346	305	420
STRUCTURES	700	350	700	350
INSULATION	173	187	181	222
ENGINES & PROPULSION SYS	(2) 2014	(1) 1115	(2) 1906	(1) 1055
AVIONICS (MINIMAL ONLY)	100	100	100	100
CONTINGENCY (15%)	491	315	479	322
RESIDUALS	434	294	426	306
SUBTOTAL BURNOUT WEIGHT	4199	2707	4097	2775
BOILOFF PROPELLANTS	–	–	–	7200
USABLE PROPELLANTS	34000	38400	31250	35250
STAGE LAUNCH WEIGHT (LEO)	38,199	41,107	35,347	45,225
PROPULSION SYSTEM WEIGHT (LAUNCH)	79,306		80,572	

1184–85

TABLE 6
PRELIMINARY WEIGHT ESTIMATE OF CREW CONSUMABLES (POUNDS)

1. BASIC WEIGHTS						
AMBIENT FOOD	2.6 #/MD	608D	6 MEN	9,500 #		
FROZEN FOOD	1.0 #/MD	608D	6 MEN	3,600		
CLOTHING	.2 #/MD	608D	6 MEN	700		
MISCELLANEOUS	1.6 #/MD	608D	6 MEN	5,800		
2. Δ FOR ORBITING CREW				19,600 #	19,600 #	
AMBIENT FOOD	2.6 #/MD	60D	2 MEN	310 #		
FROZEN FOOD	1.0 #/MD	60D	2 MEN	120		
CLOTHING (REUSABLE)	.2 #/MD	60D	2 MEN	20		
MISCELLANEOUS	1.6 #/MD	60D	2 MEN	190		
				640 #	640 #	
SUBTOTAL (STOWED IN MISSION MODULE)					20,240 #	20,240 #
3. Δ FOR SURFACE CREW						
AMBIENT FOOD	2.5 #/MD	60D	4 MEN	600 #		
CLOTHING (EXPENDABLE)	1.2 #/MD	60D	4 MEN	290		
MISCELLANEOUS	1.6 #/MD	60D	4 MEN	380		
SUBTOTAL (STOWED IN MEM)				1,270 #	1,270 #	1,270 #
TOTAL						21,510 #

MD= MAN–DAY

3303-85

TABLE 7
WEIGHT SUMMARY (POUNDS)
MANNED MARS SPACECRAFT FOR 2 & 3 YEAR MISSIONS

SUBSYSTEMS	HAB MOD # 1 (LBS)	HAB MOD # 2 (LBS)	LAB/LOG MOD (LBS)	MEM (LBS)
STR. MECHANISMS	1500	1500	1000	1500
PRESS. STRUC. (3)	5250	5250	4750	4125
SECONDARY STRUC.	1500	1500	1000	1500
MICR/INSULATION	900	900	700	470
INTERFACE STR/SHELLS	1200	1200	6800	4100
AIRLOCK/HEAT SHIELD	–	–	1500	4000
STRUCTURES SUBTOTAL	10350	10350	15750	15695
THERMAL CONTROL	1177	1177	50	1527
ELECTRICAL POWER	3000	3000	120	5475
COMM. & DATA	2027	2027	150	2220
GN&C	833	–	–	833
CREW SYSTEMS	5482	2937	4260	6545
ECLSS	7324	7324	233	2733
PROPULSION SYSTEM W/CONTIN.				6956
CONTINGENCY (15%)	4529	4022	3084	5254
SPARES (3%/YEAR) (NON–STRUCT.)	1369	1136	332	1334
SUBTOTAL (DRY)	36091	31,973	23,979	48572
FLUIDS, THERMAL	140	140	–	140
FLUIDS, ELECTRICAL	55	55	–	–
ECLSS CONSUM.	5394	5394	–	1920
CREW SYS. COMSUM.	4800	4800	9715	1140
PROPULSION DEORBIT & PLANE CHANGE CAPABILITY				7791
PROPELLANTS DESCENT & ASCENT				72004
MISSION/SCIENCE	4430	4430		1480
CREW (6)	2280			
TOTAL (LAUNCH)	53190	46792	33694	133047

SCIENCE PROBES	24480
TOTAL MISSION MODULE (LAUNCH)	133676
TOTAL MEM (LAUNCH)	133047
TOTAL SPACECRAFT (LAUNCH) 2 YEAR MISSION	291,203 LBS
ADDITIONAL MISSION/SCIENCE EQUIPMENT	10,920
ADDITIONAL CREW SYSTEMS, ECLSS, & CONSUMABLES	51,825
ADDITIONAL STRUCTURES AND SUBSYSTEMS	29,562
TOTAL SPACECRAFT (LAUNCH) 3 YEAR MISSION	383,510

site selection capability. The mission/science weights are only representative and would change as requirements are established. The crew weights include six men with flight suits. The total launch weights are for a two year mission at launch from LEO. Additional equipment, consumables, structures, and subsystems would need to be added (mostly to the MEM) for a three year mission, as shown. Shielding could be provided in the modules, mostly from the equipment and consumables shown on this chart, provided that the layout of each module is carefully done with shielding as the driving requirement. The effective thickness of aluminum for shielding of each module has been estimated to be approximately 1.5 inches for the Habitability Modules and 1.86 inches for the Lab/Log module, assuming even distribution of equipment throughout each module.

Reference 11 indicates that 1.75 inches is required. Hence, a primary challenge for spacecraft designers is to package equipment sufficiently densely, in at least a "storm shelter" region, so that no additional weight will have to be added for shielding. In addition to the SV elements, other items must be transported to LEO for the Missions to Mars. Some of these are listed in Table 8. If an assembly system is required in LEO, for the Missions to Mars, it must be transported there. Propellant which boils off during the assembly period must be placed. Assembly can last several months to a year or more, for some cases considered (see Reference 12), and boiloff can amount to half a million pounds or so, as shown in Table 8. Aerobraking vehicles, of course, would suffer much less boiloff of propellants than the all-propulsive case shown here. Ideally, the SV elements would be launched and assembled dry, then propellants would be added. This would minimize boiloff. However, to gain maximum efficiency from the ETO launch vehicles (see reference 3), the SV elements must be launched "wet", or at least partically wet.

The crew consumables used during on-orbit assembly must also be replenished, and the SV must be re-boosted occasionally in LEO to offset orbit decay and/or to maintain proper phasing with respect to the co-orbiting SS. Reference 13 discusses potential roles of the SS in more detail. If the assembly period lasts a long time, there will probably need to be a crew rotation every 3 months or so. Weights are not shown for this.

TABLE 8
TOTAL WEIGHT * TO BE TRANSPORTED FROM EARTH TO EARTH ORBIT

● SPACE VEHICLE WEIGHT		2,701,810
– DRY	284,939	
– FLUIDS, CONSUMABLES, PROPELLANTS, ETC	2,416,871	
● ASSEMBLY SYSTEM (W/CMG'S AND MRMS)		16,000
● PROPELLANT BOILOFF REPLENISHMENT		528,258
– LEO ASSEMBLY	469,362	
– 30–DAY DEPARTURE WINDOW	58,896	
● CREW CONSUMABLES REPLENISHMENT (LEO ASSEMBLY)		32,165
– GN_2	3,752	
– FOOD, MEDICAL, PERSONAL, HOUSEKEEPING, ETC.	28,413	
● REBOOST PROPELLANT DURING ASSEMBLY		7,000
		3,285,233 LB.

* FOR 1 SPACE VEHICLE, 1999 OPPOSITION MISSION, ALL–PROPULSIVE CONCEPT

In some options, the SS may serve as the assembly system, and may also provide the crew, related resources, and possibility the reboost propellants during LEO assembly. If so, these would all be subtracted from the list of items (Table 8) that must be furnished by the Mars program separately.

REFERENCES

1. Paper in Section II of this report, entitled "Mission and Vehicle Sizing Sensitivites", by Archie Young of MSFC.
2. Paper in Section VII of this report, entitled "Propulsion System Issues, Options and Trades", by Douglas Forsythe of MSFC.
3. Paper in Section III of this report, entitled "Earth to Orbit Launch Vehicles", by Milton Page of MSFC.
4. Paper in Section VII of this report, entitled "Mars Transit Vehicle Thermal Protection System: Issues, Options, and Trades," by Norman Brown of MSFC.
5. Final Oral Report of the "Integrated Manned Interplanetary Spacecraft Concept Definition (IMISCD), D2-113543-3, January 1968, by the Boeing Co., NASA Contract NAS1-6774.
6. Volume II (design) of the Final Report of the "Definition of Experimental Tests for a Manned Mars Excursion Module", SD 67-755-2, 12 January 1968, by the Space Division of North America Rockwell Corp., NASA Contract NAS9-6464.
7. Paper in Section II of this report, entitled, "Mars Mission Concepts and Opportunities," by Archie Young of MSFC.
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10. Space Systems Technology, edited by Regis D. Heitchue, Jr., published by Reinhold Book Corp., N.Y., Copyright 1968, pp. 148-150.

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12. Paper in Section III of this report, entitled "Manned Mars Mission ETO Delivery and On-Orbit Assembly", by B. Barisa and G. Solmon of MSFC.

13. Paper in Section IX of this report, entitled "Space Station Utilization and Commonality", by John Butler of MSFC.

N87-17748

THE "CASE FOR MARS" CONCEPT

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ABSTRACT

The Case for Mars workshops conducted in 1984 (Ref.1) dealt with a program to establish a permanent scientific research base at Mars. The participants, some of whom are listed in Appendix A, viewed a Mars base as the much needed long-term focus for the space program. A permanent base was chosen rather than the more conventional concept of a series of individual missions to different sites because the permanent base offers much greater scientific return plus greater crew safety and the potential for growth into a true colony. This paper summarizes the results of the workshops.

The Mars base will strive for self-sufficiency and autonomy from Earth. Martian resources will be used to provide life support materials and consumables. The Martian atmosphere will provide a convenient source of volatiles: CO_2, N_2, and water. Rocket propellant, fuels for surface and air vehicles and possibly power plants, breathable air, and fertilizers will be manufactured from the Mars atmosphere. Food will be grown on Mars using Martian regolith as a growth substrate.

A permanent human presence will be maintained on Mars beginning with the first manned landing via a strategy of crew overlap. This permanent presence will ensure safety and reliablity of systems through continuous tending, maintenance, and expansion of the base's equipment and systems. A permanent base will allow the development of a substantial facility on Mars for the same cost (in terms of Earth departure mass) as a series of temporary camps. A base equipped with surface rovers, airplanes, and the ability to manufacture consumables and propellant will provide far more extensive planetary exploration over a given period of years than would an Apollo-style approach.

SCIENCE AND EXPLORATION

A human presence on Mars will accelerate and enhance scientific exploration of the planet. Humans have unique capabilities which are difficult or impossible to automate. These features, along with the inherent flexibility of people, make the in-situ human the best possible tool for Mars science.

Prior to a manned landing, automated precursor missions are required to investigate the Mars environment and select an optimal location for the permanent base. The base must be located in an accessible area suitable for a landing field, and must be near areas of scientific interest. Martian resources will be used for base operations. Thus the chemistry, mineralogy, and the state and distribution of volatiles on the Martian surface, particularly water, must be assessed globally and locally. The meteorological environment of Mars must be studied to forecast the likelihood of dust storms in the base location, and characterize the local, regional and global weather.

A precursor program to accomplish these objectives includes the planned Mars Geoscience Climatology Orbiter (MGCO). An orbiter mission to provide high resolution images of candidate base site areas is also needed. A network of surface weather stations supported by low resolution orbital imaging of cloud features is desirable for several Mars years in advance of the manned mission. A series of unmanned rover and sample return missions is needed to collect samples of Mars materials from prospective base sites and bring them to Earth for analysis.

An alternative possibility is for the precursor missions to be manned. The crew for the first few (say three) landings would evaluate the most promising sites and bring back samples. The next mission (fourth?) would then return to the best site to begin base establishment. In this scenario, unmanned rover/sample return would probably be unnecessary since the manned missions would do the same thing. A high resolution orbital precursor might be sufficient to choose the first landing sites.

Assuming unmanned precursor missions, the initial human landing at the base site will certify the safety and habitability of the base location, provide ground truth about the presence of water and other raw materials for base operations, set up resource extraction equipment, and establish meteorological stations in support of future manned landings. Permanent scientific research facilities will be the next priority, after these survival technologies are deployed. Facilities for research in atmospheric science will provide weather observation and reporting as well as climate, atmospheric dynamics, and atmospheric chemistry studies.

Geoscience research capabilities will include surface exploration, seismic and drilling equipment, manned and teleoperated rover vehicles, and laboratory equipment for geochemical and petrological study of samples. Life science research on Mars will search for present or past life, supported by appropriate laboratory capabilities.

MISSION STRATEGY

The mission strategy is directed toward support of a permanently inhabited Mars base with crew rotation and resupply at each Earth-to-Mars launch opportunity. In order to minimize the total mass departing Earth orbit to support the base and to provide Earth return capability for the crew being rotated home, a Mars powered flyby and return to Earth is performed by the Deep Space Habitat vehicle (Figures 1a and 1b). Arriving crew members separate from the habitat in Mars Shuttle vehicles (Figure 2) while on the approach leg. The Shuttles proceed to Mars and land at the base using a combination of aerodynamic braking and rocket thrust (Figure 3). To get into an Earth - return trajectory, the deep space habitat vehicle performs (unmanned) a propulsive maneuver as it flies by the planet. Returning crewmembers depart Mars in their Shuttles which rendezvous with the Habitat vehicle on the outbound leg departing Mars. In preparation for the next habitat flyby (two years later), the Mars Shuttles at the base are refueled using $CO\text{-}O_2$ propellant manufactured from Mars CO_2 (Figure 4).

While the newly arrived crew takes up its duties at the base, the returning crew rides back to Earth in the Deep Space Habitat. Arriving at Earth, the crew enters the Mars Shuttles and aerobrakes down to the Space Station, and the Habitat vehicle makes a final use of its propulsion system to enter a loose elliptical orbit around Earth from which it is later recovered for refurbishment and reuse.

Each mission of the Habitat/Mars Shuttle assembly delivers fifteen crew members to Mars. In the early stages of the program, a lesser number (say nine) of the base crew will return to Earth. This will not only provide growth but also a highly desirable continuity in base operation.

VEHICLES

Three new major vehicles are involved in execution of the mission strategy developed in this paper. These are: Mars Shuttle, the Deep Space Habitat, and the Earth Departure Stage.

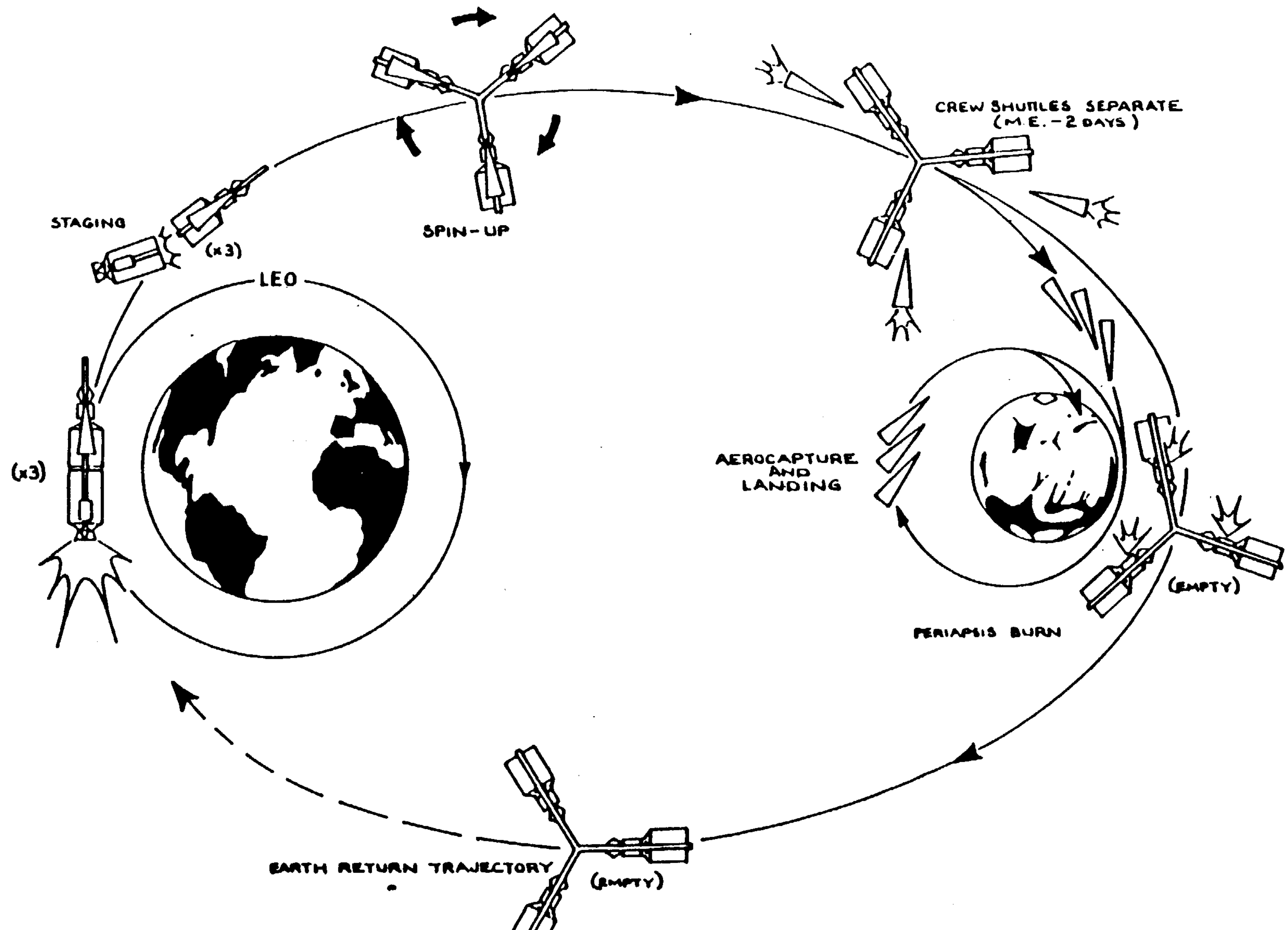
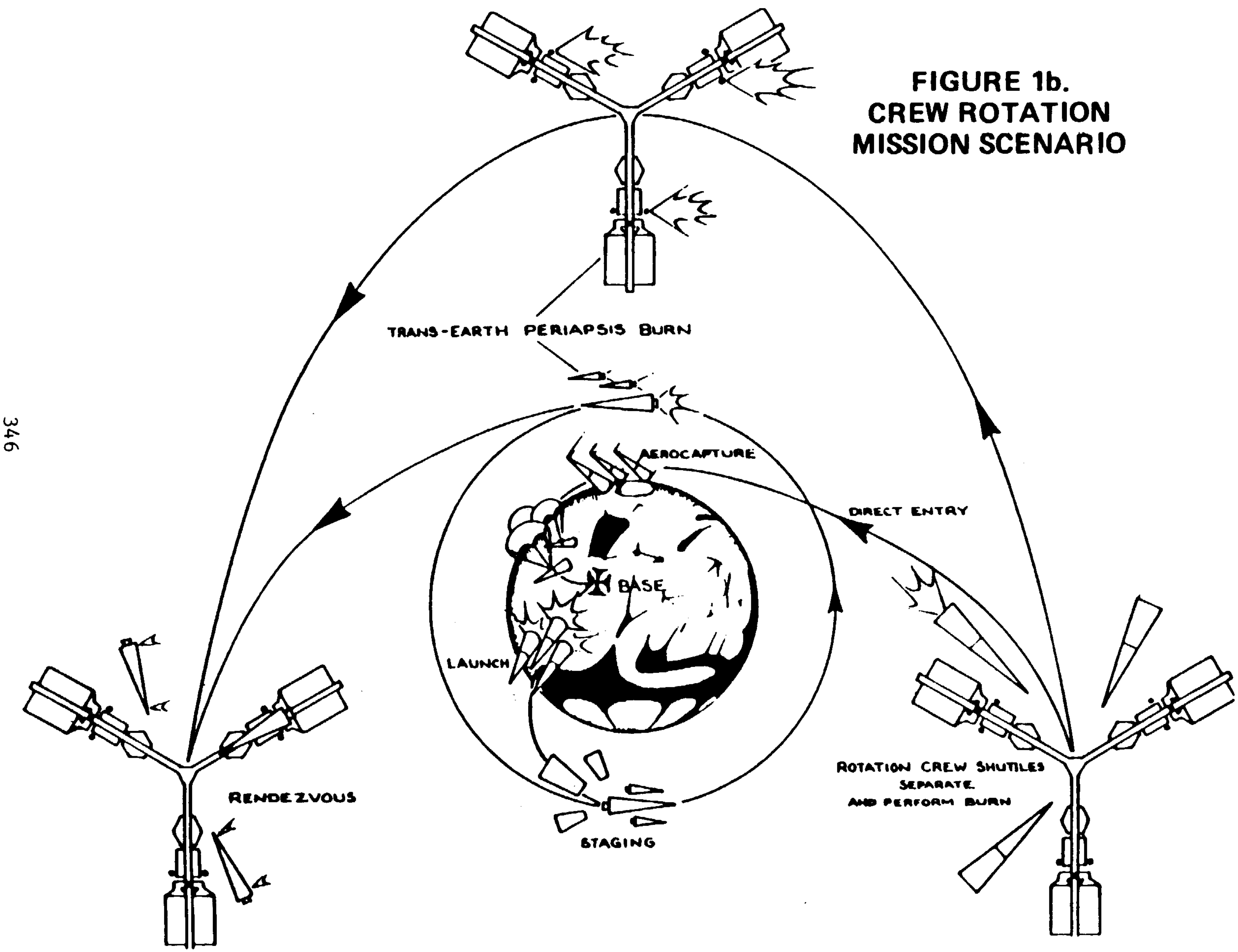


FIGURE 1a
INITIAL MISSION SCENARIO

FIGURE 1b.
CREW ROTATION
MISSION SCENARIO



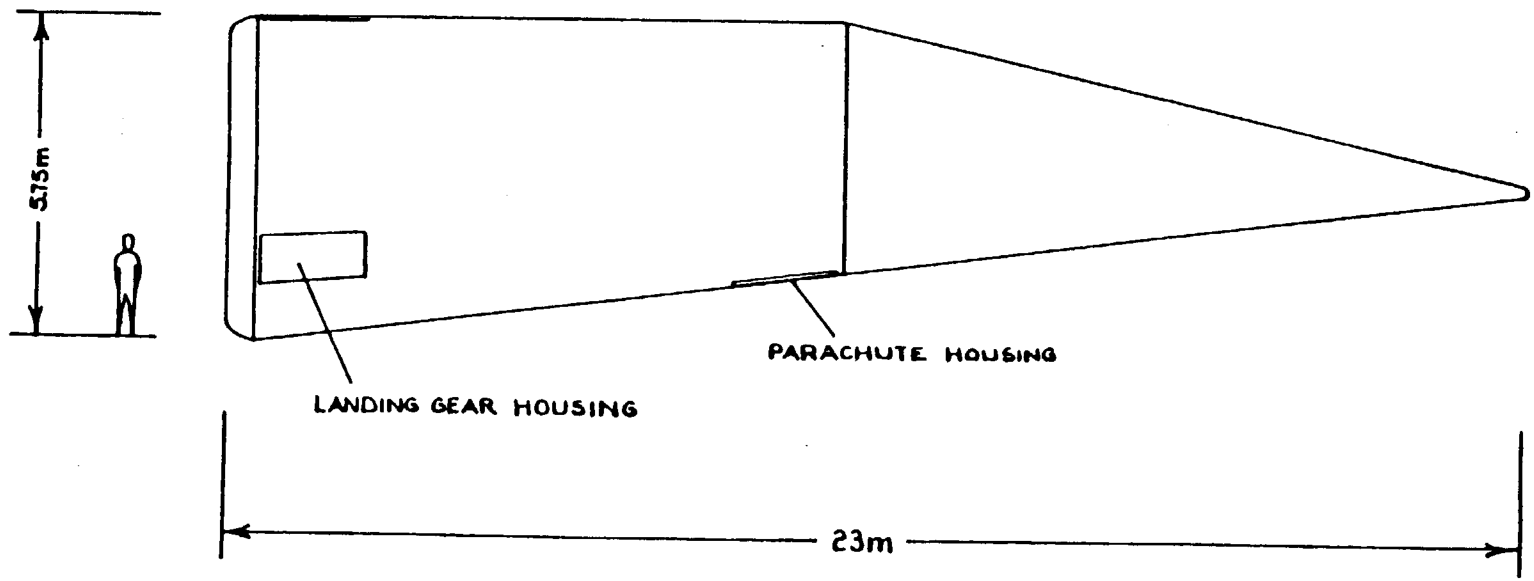


FIGURE 2
MARS SHUTTLE VEHICLE

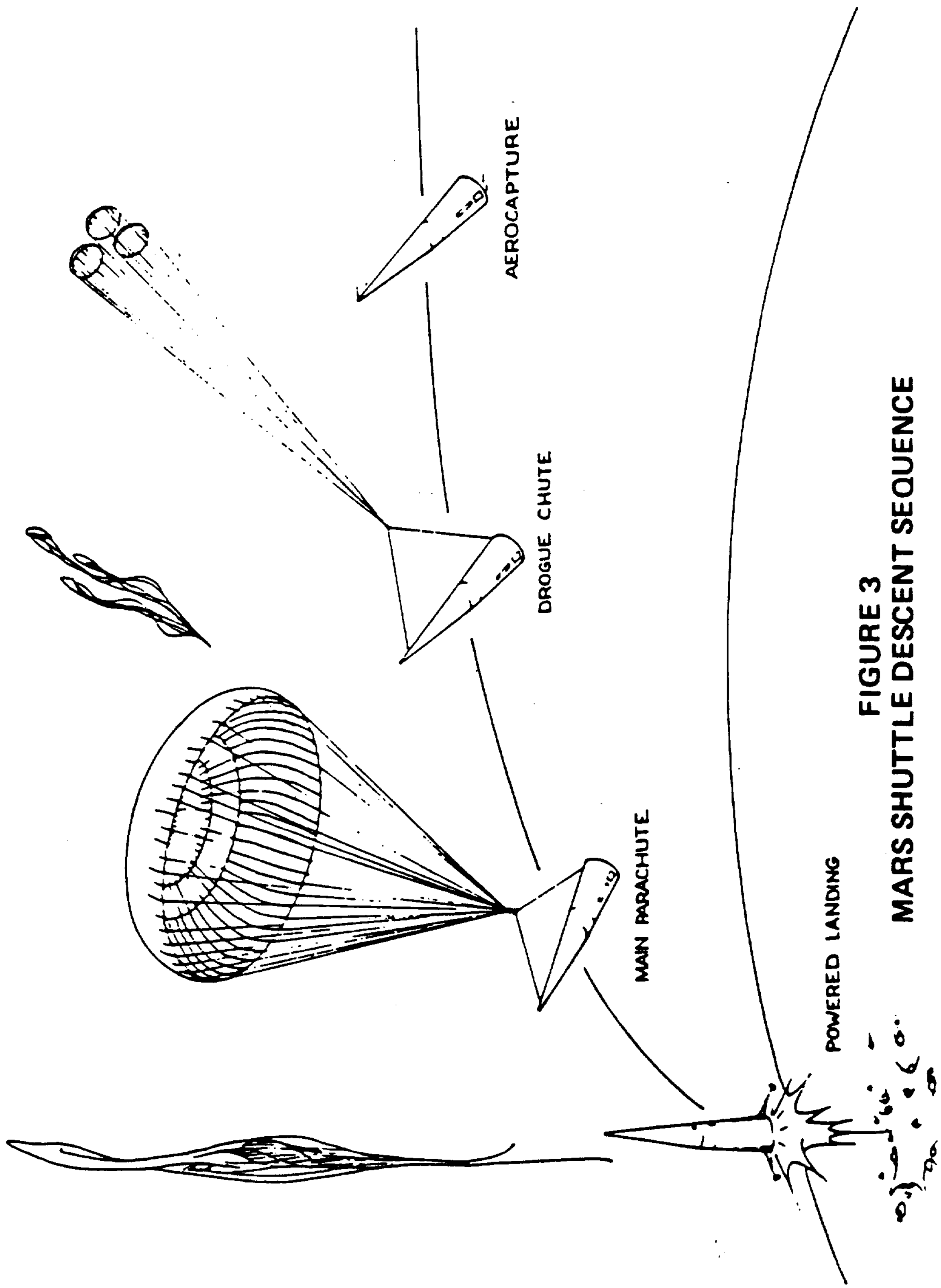
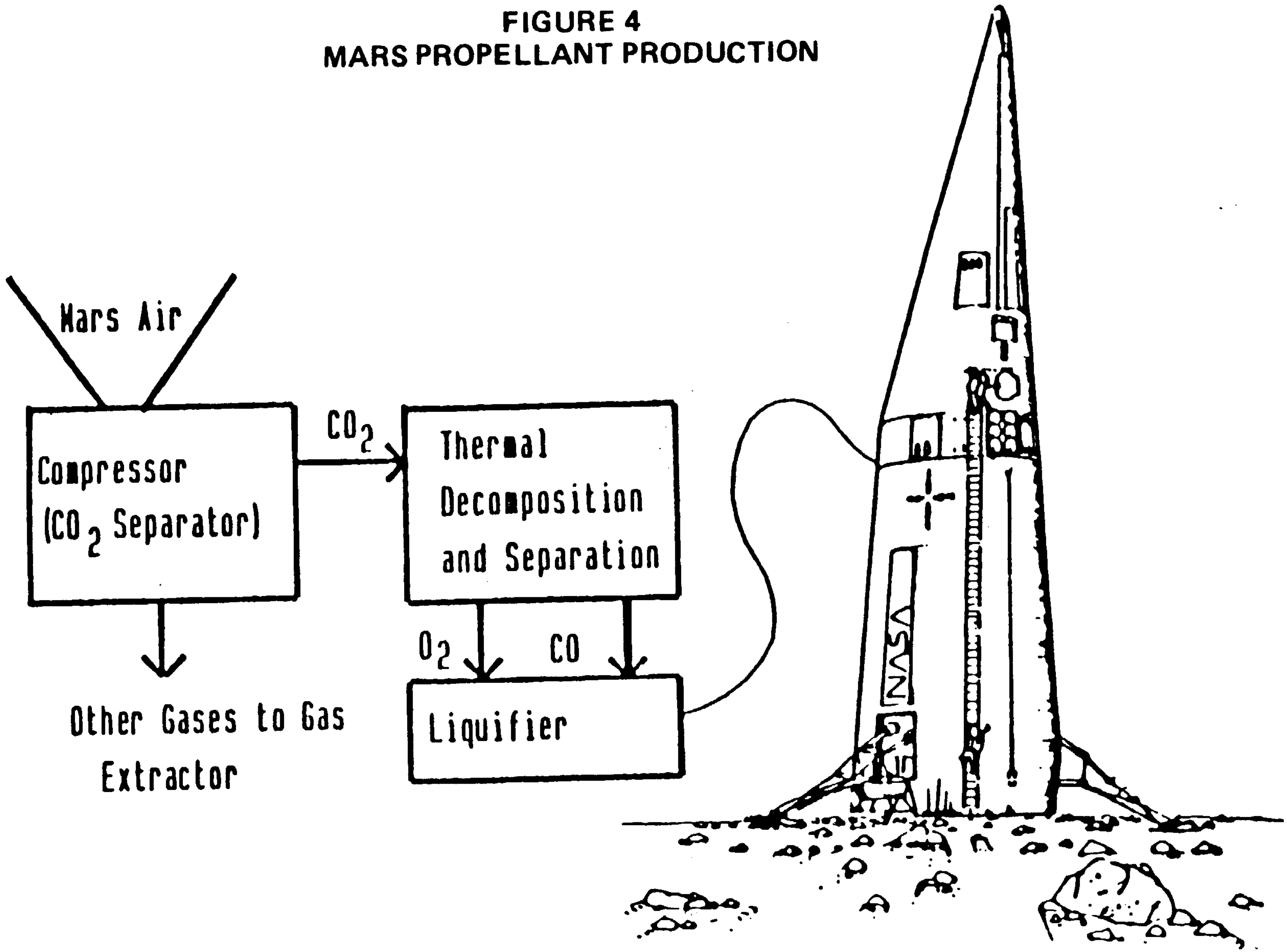


FIGURE 3
MARS SHUTTLE DESCENT SEQUENCE

FIGURE 4
MARS PROPELLANT PRODUCTION



The Mars Shuttle vehicles, as the name implies, are used to transport arriving crew members to the Martian surface from the Deep Space Habitat and to bring homeward bound crew members from Mars to the Habitat. At the end of the return journey, they are also used to bring the crew to the Space Station. For the descent to Mars, the Shuttles depend upon aerodynamic braking to slow them from an entry velocity of 5 to 6 km/sec down to a velocity suitable for parachutes. To provide the required accuracy and control, a relatively high lift - to - drag ratio is needed. A biconic airframe (shaped like a slightly crooked cone) provides this capability. Two versions of the Mars Shuttle are needed; one is a one - way unmanned cargo vehicle (Figure 5) the other a manned version which can be reloaded with propellant on Mars for the return (Figure 6). The initial manned version will be a two stage vehicle, since the $CO\text{-}O_2$ propellant manufactured on Mars is of low performance. Later in the program, higher Isp propellants may allow a single stage vehicle. Protection from aerodynamic heating would be provided by a reusable heat shield similar to that used for the Space Shuttle.

The Deep Space Habitat (Figure 7) is composed of three identical sections. Each section is assembled at the Space Station, and consists of two Space Station modules, life support system, consumable storage, and a propulsion system. All this is attached to a boom and tunnel assembly, terminating in a docking adapter which allows the three sections to dock into a pinwheel configuration that is rotated to provide artificial gravity. A crew - type Mars Shuttle is docked along each boom. Each section (with its Mars Shuttle) is boosted separately on a Mars-bound trajectory from low Earth orbit. The three sections rendezvous and dock on the way to Mars, remaining linked for the remainder of the mission.

The Trans-Mars injection stage (Figure 8) is used to boost the Habitat/Mars Shuttle assemblies out of Earth orbit and into the Mars transfer trajectory. It uses adaptations of the Space Shuttle Main Engine for thrust. Tankage for the liquid oxygen/liquid hydrogen propellant is modular to allow each tank to be launched in the Space Shuttle for on-orbit assembly of the stage.

Cargo versions of the Mars Shuttle travel to Mars without being attached to the Habitat. To provide power and other services to the

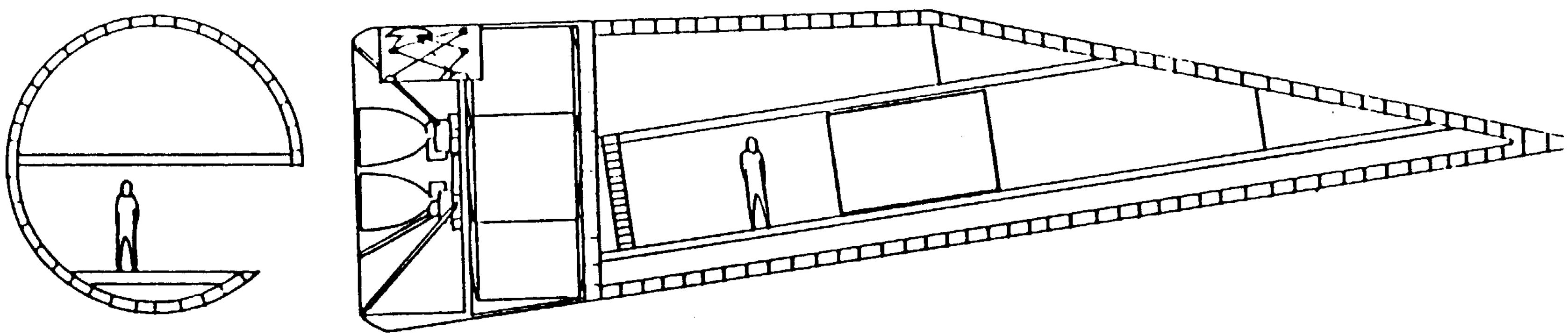


FIGURE 5
CARGO VEHICLE

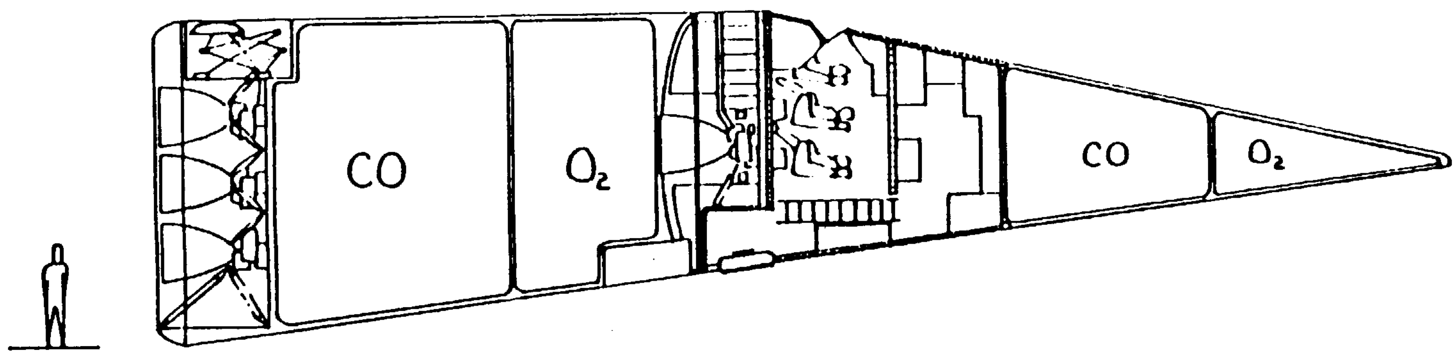


FIGURE 6
CREW SHUTTLE

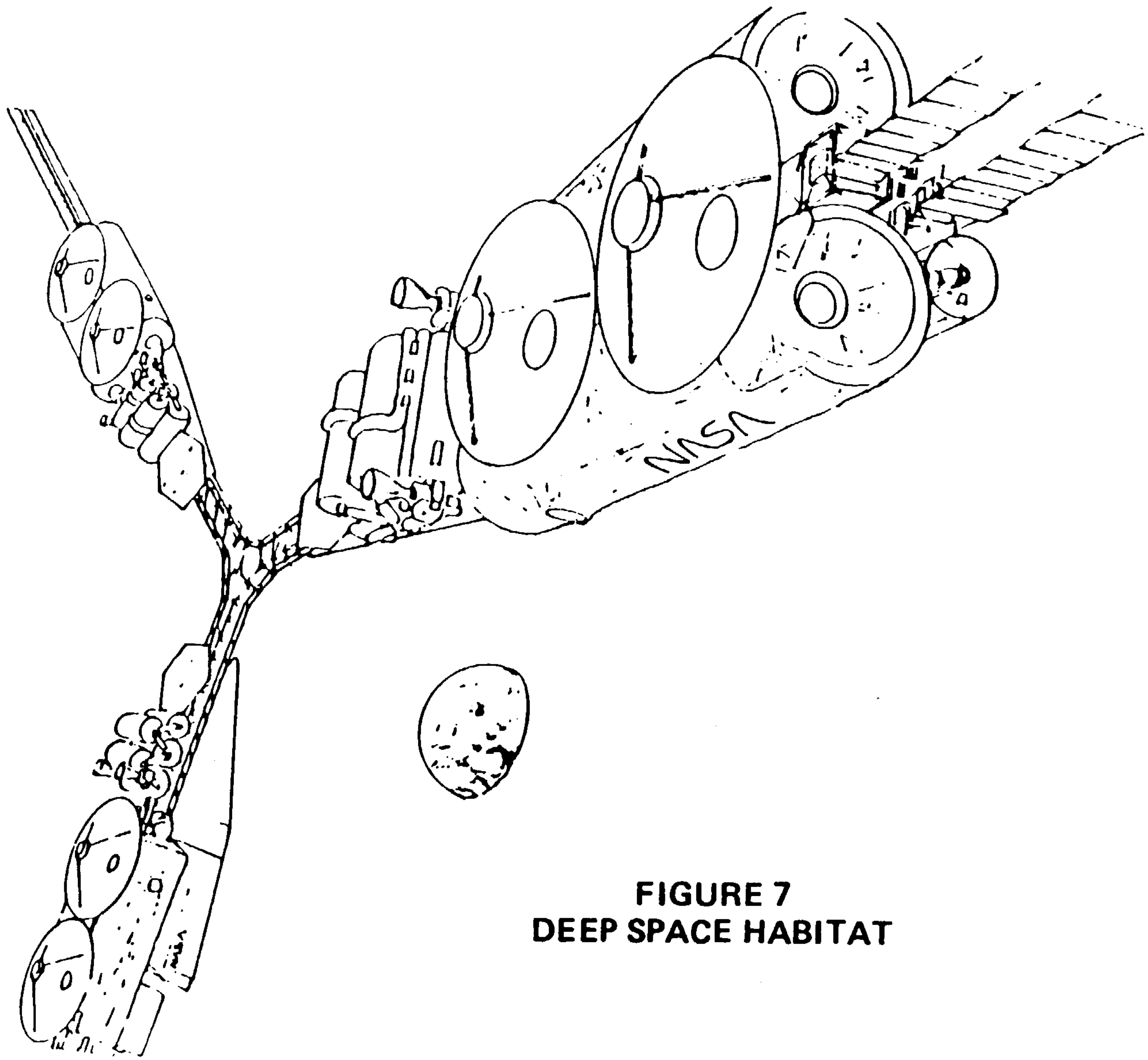
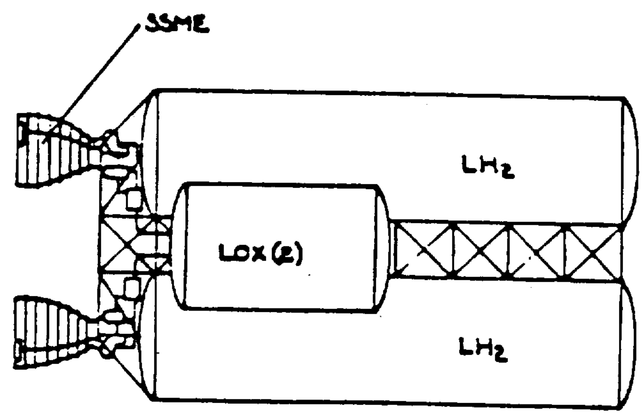


FIGURE 7
DEEP SPACE HABITAT

FIGURE 8
TRANS–MARS INJECTION STAGE



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vehicle during interplanetary flight, a jettisonable service module will be attached. Appendix B provides some characteristics of the vehicles.

HUMAN FACTORS

Human factors encompasses those facts of mission planning and design which affect the physiological and psychological condition and the performance of the Mars Base crewmembers.

Life support facilities must be provided for long-duration spaceflight with primary considerations being mass, volume and reliabiltiy. Recycling of water and breathable gases is essential. Food is primarily transported, with possibly some supplementary food production in flight. Organic waste can be stored for later use as an agricultural commodity at the Mars Base. Development of long - duration life support is seriously lagging behind other technologies relevant to human missions to Mars.

Life support at the Mars Base involves a program of gradually expanding food production and gas recycling capability. Martian water, gases, and possibly regolith will provide most of the raw consumable materials. Greenhouses are used to provide the basic foodstuffs for the Mars Base food chain. Optimum use of organic recycling is encouraged, and the feasibility of microbial processing to provide a variety of biological products and enhance nutritional value and palatablity of food is suggested. The overall facility is envisioned as a managed ecological system relying on biological cycling of materials when possible. This is augmented with chemical and physical subsystems to provide buffering capability against system oscillations. Emergency food supplies are cached in case of system failure.

Medical care must provide for the normal needs of crewmembers over 5 years' mission duration plus the ability to address a variety of foreseeable problems in unknown and hazardous environments of space and the Martian surface. The likelihood of accidents requires the capability to perform at least limited surgical procedures. A carefully selected pharmacopoeia must be included to cover a reasonable range of disease and accident treatments. All crewmembers must be trained in basic rescue and emergency medicine. At least one physician must be included in the crew. Relevant medical questions to be pursued prior to a Mars mission include effects of zero and fractional gravity over long periods of time and ameliorating drugs or techniques, and development of medical devices and techniques appropriate to the space extraterrestrial environments.

Psychological considerations are involved at all stages of mission planning including crew selection and training, selection of command protocols, scheduling of work loads, provision of recreational facilities, ergonomics and the Mars Base design, rotation of crews from mission to mission, mission continuity with changing personnel, and interpersonal relationships.

MARS BASE

The primary function of the Mars Base is to support a continued human presence on the surface and to achieve self - sufficiency through the use of Martian resources. This provides the security and home base from which to conduct the scientific exploration that will become the main thrust of activities on the surface.

The major components and requirements for the Mars Base are shown in Figure 9. The initial crew size is 15 people, with incremental growth over time. Major components include: Habitats derived from cargo vessels; air shells/greenhouses which are lightweight erectable structures which can be pressurized with Mars air; power supplies to provide power to the base and to the resource extraction equipment (the largest power user); rovers, trucks, and other mobility units for construction and field experiments; habitat life support systems which can have considerable inheritance from Space Station Systems and from the resources available on the Martian surface; gas extractors which would obtain breathable air and water from the Mars atmosphere or surface.

Breathable air could include an Ar/N_2 buffer gas mixture. These elements together comprise over 5% of the Martian atmosphere and can be obtained by condensing out the CO_2. Oxygen can be obtained by reducing the atmospheric CO_2. The Mars atmosphere contains water (nearly at saturation), which can be extracted with compression and cooling equipment. Water may also be available from the Mars regolith. Rocket fuel can be made from the CO_2 itself (CO and O) or in combination with water (CH_4 and O_2). An active research program must be established to look at the use of gases and minerals available on Mars in support of the exploration effort.

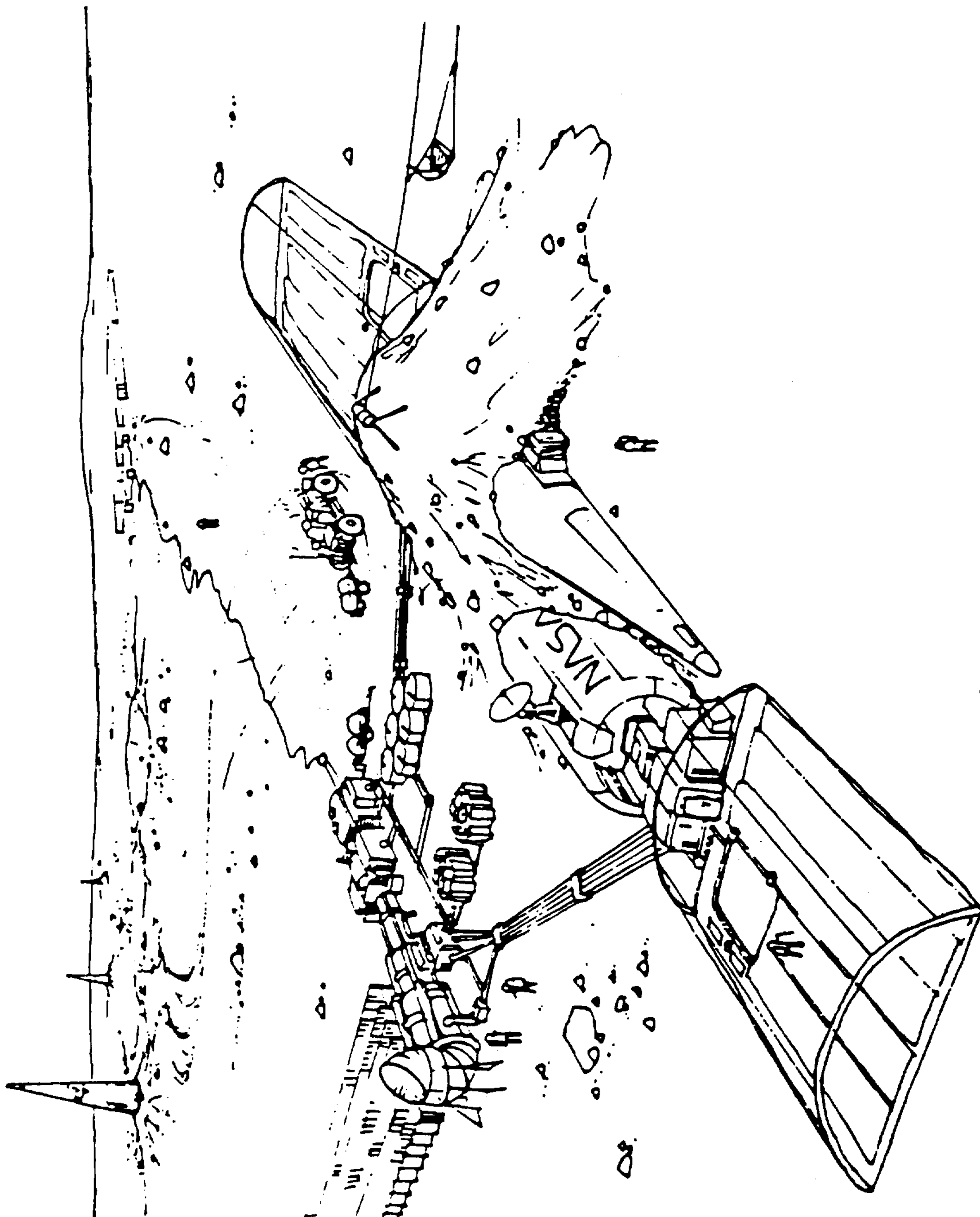

FIGURE 9
MARS BASE

The initial focus of activities at the Mars base must be the development of resource utilization technologies, since the continued presence of the base and the long range science goals are contingent on establishing the resource base.

Some areas that need development in connection with the Mars base include: 1) Power supply suitable to provide the approximately 200 - 400 kwatts needed; 2) Mars suit design; 3) small engines to run on fuel made in-situ; 4) the study of life support and resource utilization.

<u>REFERENCES</u>

1. Preliminary draft report " The Case for Mars: Concept Development for a Mars Research Station " 5 December 1984.

APPENDIX A

AUTHORS OF THE CASE FOR MARS: CONCEPT DEVELOPMENT FOR A MARS RESEARCH STATION

EDITOR
S. M. Welch

PROGRAM JUSTIFICATION DESIGN	MARS SCIENCE	MISSION/PROFILE SPACECRAFT
S. M. Welch	C. R. Stoker	J. R. French
C. R. Stoker	R. L. Grossman	R. L. Staehle
R. B. Wilson	P. J. Boston	S. M. Welch
T. R. Meyer		

HUMAN FACTORS/ LIFE SUPPORT	MARS BASE DESIGN	PHOBOS/DEIMOS MISSIONS
P. J. Boston	C. P. McKay	T. Caudill
T. R. Meyer	D. Jones	

ARTIST

C. Emmart

APPENDIX B

VEHICLE CHARACTERISTICS

<u>MARS SHUTTLES</u>:

LENGTH: 23m

BASE DIAM: 5.75m

MASS AT EARTH DEPARTURE: 50 TONNES
(includes landing propellant)

MASS AT TOUCHDOWN: 30 TONNES

MANNED VERSION:
CARGO - CREW OF 5 and 5 - 6 TONNES EQUIPMENT
LIFT-OFF MASS: 215 TONNES (PROPELLANT CO/O_2, Isp = 260 sec)

CARGO VERSION:
CARGO - 24 TONNES

<u>DEEP SPACE HABITAT</u>:

EACH MODULE SUPPORTS 5 CREW (10 EMERGENCY)

DEPARTURE MASS PER MODULE 100 TONNES (220,000 lbs.) PLUS 50 TONNES MARS SHUTTLE NORMAL ASSEMBLY THREE COMPLETE MODULES

<u>EARTH DEPARTURE STAGE</u>:

LOADED MASS: 300 TONNES (660,000 LB.) LO_2/LH_2 Isp=465 sec

EMPTY MASS: 20 TONNES (44,000 LB.)

PAYLOAD - ONE MODULE OF DEEP SPACE HABITAT OR 3 CARGO SHUTTLE DELTA V CAPABILITY: 4.4 km/sec

SECTION IV

SURFACE INFRASTRUCTURE

CELSS AND REGENERATIVE LIFE SUPPORT FOR MANNED MISSIONS TO MARS

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NASA Ames Research Center

ABSTRACT

In the mid 1990's, the Space Station will become a point from which inter-planetary vehicles can be launched. The practicalities of a manned Mars mission are now being studied, along with some newer concepts for human life support. Specifically, the use of organisms such as plants and algae as the basis for life support systems is now being actively considered. A Controlled Ecological Life Support System (CELSS) is composed of several facilities: (a) to grow photosynthetic plants or algae which will produce food, oxygen and potable water, and remove carbon dioxide exhaled by a crew; (b) to process biomass into food; (c) to oxidize organic wastes into CO_2; and (d) to maintain system operation and stability. Such a system, when compared to using materials stored at launch, may have distinct weight and cost advantages, depending upon crew size and mission duration, as well as phychological benefits for the crew. The use of the system during transit, as well as in establishing a re-visitable surface camp, will increase the attractiveness of the CELSS concept for life support on interplanetary missions.

INTRODUCTION

A manned mission to Mars has been a human dream ever since Percival Lowell (1) first began to popularize the planet as a place where highly advanced civilizations built canals to bring water from the polar regions to service cities at the equator. Unfortunately, no evidence of the "canali" sketched by Schaparelli in 1877 and by Lowell were revealed during the intensive, planet-wide scanning performed by the Viking orbiters; and no evidences for life, or even significant amounts of organic carbon, were detected by the Viking landers. Nevertheless, the Red Planet will probably be the first object that humans will visit outside of the Earth-Moon system.

Life support considerations for manned missions to Mars should include transit to and from the planet, the period of visit on the surface, and the possibility of leaving behind structures and equipment for subsequent visits, thus eventually making colonies easier to achieve. For these reasons, life support systems based on the use of biological components (primarily plants and algae) are discussed in this paper. These systems are generally termed Controlled (or Closed) Ecological Life Support Systems (CELSS).

CELSS CONCEPTS

The concept of CELSS is to provide for humans in space by regenerating life support materials as they are needed. A CELSS relies on photosynthetic organisms to regenerate food and oxygen from carbon dioxide and other waste materials. The reason for using such a system is to decrease the amount of material that must be launched from Earth for life suport purposes.

Biogeneration depends on the absortion of energy (light) by photosynthetic organisms (e.g. higher plants or algae). In the presence of light, plants absorb the principal human metabolic waste product, carbon dioxide, and elaborate it into materials that can be used as human food (Figure 1). At the same time, plants produce oxygen. For simplicity, the system is usually described as involving only plants or algae, however, the use of animals, ranging from shellfish through fish, birds and other small vertebrates, is not excluded.

The development of a CELSS solely for use on the Martian surface for a short (60 day) residence period is not likely to be economically justifiable. However, it is reasonable to expect that by the time a manned Mars mission is scheduled, a CELSS will have been developed independently for use on a "growth" Space Station (2), a second space platform, or for a Lunar Base (3). Moreover, if a CELSS were to be used in transit to and from Mars, and if a CELSS system were left in place for subsequent missions and visits, the economics of a CELSS system for a manned Mars mission would be positive.

CELSS FUNCTIONS

The major human life support requirements are well known. Figure 2 lists tha major input/output masses for one person (4). Food requirements are not just caloric, but must include specific nutrients: carbohydrates, protein, lipids, fibre, minerals and vitamins in acceptable ratios (5). Requirements for water include low salt content, as well as freedom from toxic materials and microorganisms. The demands for food

FIGURE 1. THE MAJOR MATERIAL FLOWS IN A CELSS.

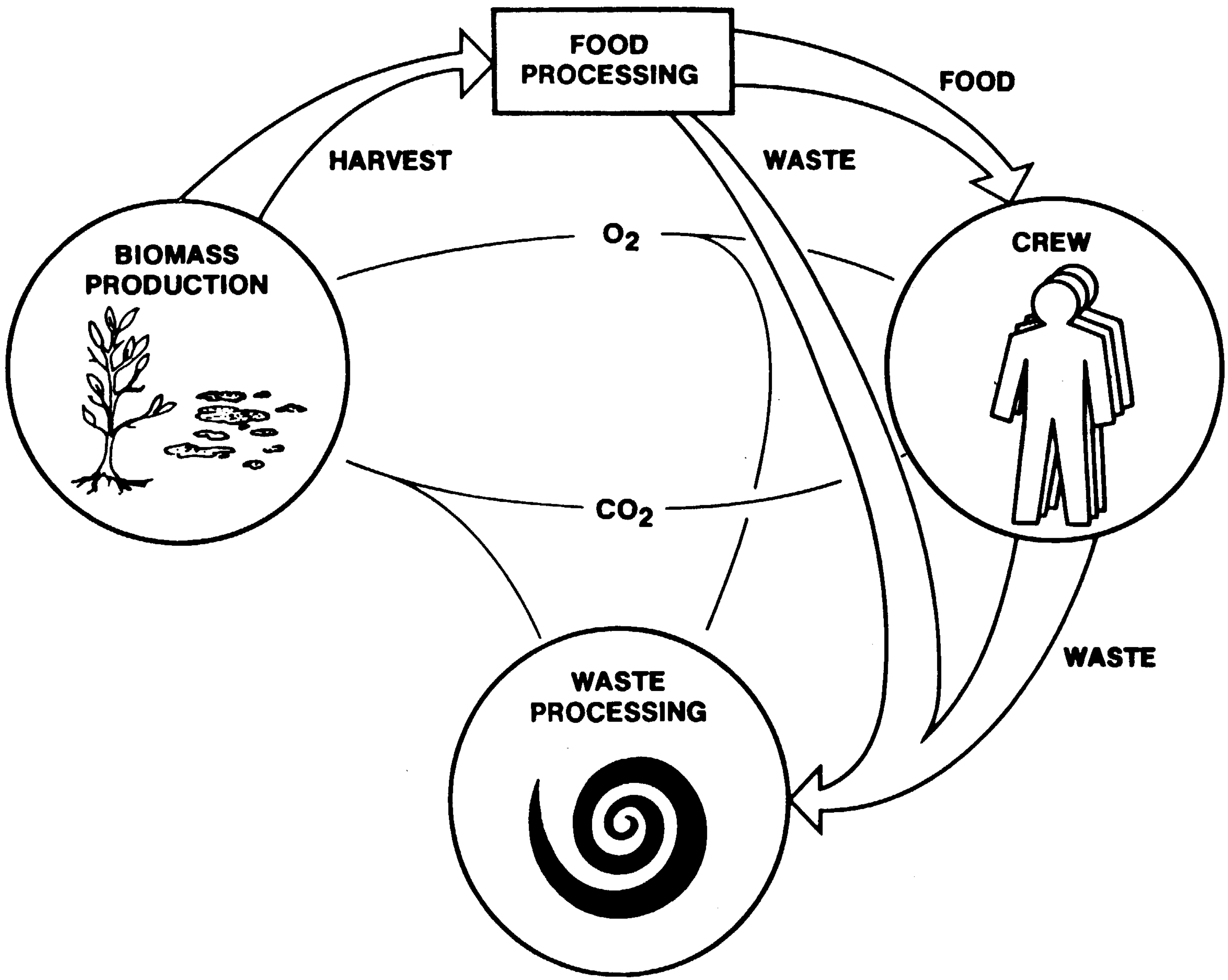
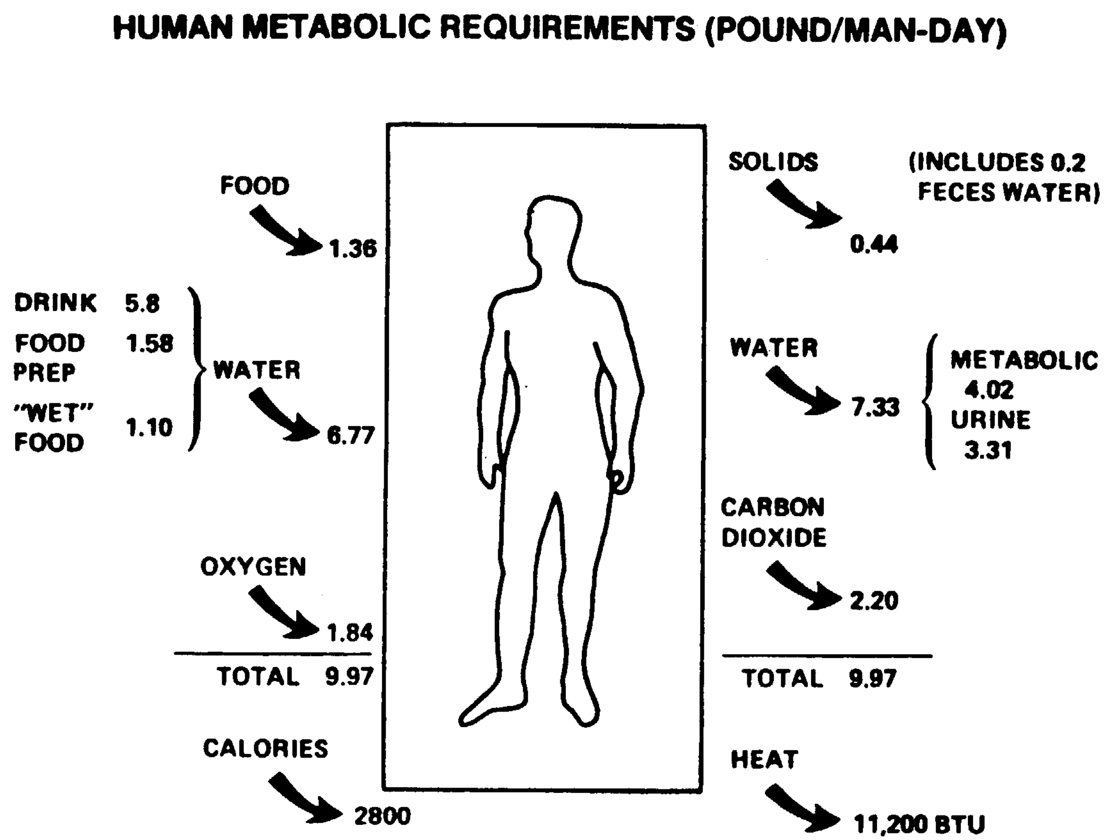


FIGURE 2. THE PRIMARY INPUT/OUTPUT MASSES OF A PERSON UNDER A MODERATE WORKLOAD, REQUIRING APPROXIMATELY 2800 CAL/DAY (AFTER OLSEN, ET AL., 1982, p 2). THE OXIDATION OF FOOD RESULTS IN THE PRODUCTION OF WATER AS WELL AS ENERGY.



FROM: GUSTAN, E. AND VINOPAL, T., CONTROLLED ECOLOGICAL LIFE SUPPORT SYSTEM: A TRANSPORTATION ANALYSIS. NASA-CR-166420. NASA-Ames Research Center MOFFETT FIELD, CA (1982) and water, as well as for oxygen, will vary according to the amount of crew activity.

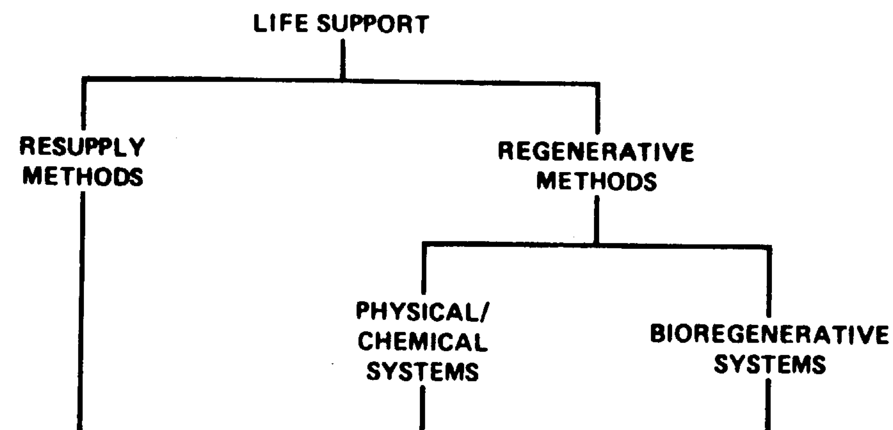
At the present time, life support requirements for Shuttle crews are met by taking as cargo the necessary materials: Food, water, and oxygen. Liquid and solid waste materials are collected in various ways and stored. Carbon dioxide is absorbed by lithium hydroxide and stored. This method of life support is very appropriate for small crews which are in space for relatively short periods of time. As crew sizes and/or mission durations grow, the cost of life support will become an increasingly significant fraction of total launch costs (6). Further, materials intended for life support will compete in weight and volume with other essentials such as equipment and fuel.

There are two options available to meet crew life support requirements (Figure 3). The first option ("resupply") allows for including all of the required materials at launch, or the establishment of unmanned resupply depots containing life support materials, a scenario that is unlikely on the first voyages to Mars. The second option is to regenerate the necessary materials partially or fully from waste materials. Depending upon the extent of recycling and regeneration, the last option can offer considerable savings in launch costs and in space habitat volumes.

Life support technology presently under development uses physical and chemical techniques to partially regenerate oxygen and potable water from waste materials. The carbon dioxide produced by the crew, instead of being absorbed by lithium hydroxide as it is at present, will be concentrated and processed to release the oxygen it contains. Used water, particularly wash water and exhaled vapor, will be reclaimed by removing materials dissolved or suspended in it. The equipment necessary for these processes has been developed under programs operating through NASA/Ames and NASA/Johnson, (see Quattrone (7), 1984 for a thorough review), and by several private companies (8).

Recycling part of the water and regenerating part of the oxygen needed for life suport will go a long way to decrease the mass and volume of materials required for life support. However, because the recycling of materials is incomplete, and because food is not regenerated, a significant mass of life support material will have to be launched from Earth.

FIGURE 3. CLASSIFICATION OF VARIOUS LIFE–SUPPORT TECHNIQUES.



O_2	STORED	FROM CO_2	FROM PLANTS
CO_2	LiOH ADSORBED	1. AMINE ADSORBED 2. CONC./REDUCED	TO PLANTS
H_2O	STORED	1. DISTILLATION 2. MEMBRANE/EVAP.	PLANT TRANSPIRATION
FOOD	STORED	STORED	PRODUCED
WASTE	STORED	STORED	RECYCLED

Methods that rely solely on chemical or physical means for regenerating food are unlikely to be practicable before the turn of the century, and may never supply all human nutritional needs. It is of interest, therefore, to use methods that were evolved by the photosynthetic organisms that are the fundamental suppliers of all of the food and oxygen that we use on Earth. Photosynthesis has the advantage that it simultaneously accomplishes three tasks necessary for human sustenance. (1) It directly uses the major human metabolic waste product, carbon dioxide; (2) it chemically reacts CO_2 with water to create the organic materials that we use as food; and (3) it produces essential oxygen. In addition, since water is the transporter of materials in vascular plants, and is rapidly passed from the plant to the atmosphere, higher plants can act to purify water.

SYSTEM CONTROL

An engineered, biogenerative life support system, such as CELSS, will depend upon the same biological processes that support life on Earth. However, it is obvious that the collection of plants, bacteria and animals on Earth are controlled in some way, so that there is a regulation of the abundance of different kinds of organisms, and consequently, in the concentrations of gases in the atmosphere and of solutes in the water. In essence, there are controls that maintain the stability of the environment. The kinds of controls that are operative in the natural environment are the objects of study of the discipline of ecology.

The distinction between the functioning of Earth's life support system and that of a smaller-scale, engineered life support system is primarily complexity. Each living organism in the natural system is "connected" with many others through a large number of interfaces, and controlled by activities such as access to nutrient supplies, competition for light, space or nutrients, predation, etc. A CELSS in space will have some of the same interrelationships, and many of the same physical structures and processes as the massive terrestrial life support system. But to a significant extent, the interfaces and system processes will have to be identified and stringently controlled. The reason is that an engineered bioregenerative system will be very small compared to terrestrial systems, and it will have to operate productively at a much higher rate, yet it will have to be at least as stable as a terrestrial system. Achievement of long-term stable and productive CELSS operation will require system control at levels not yet generally practiced (9, 10). This is a primary engineering problem for CELSS, and one which many Soviet scientists and technicians have been working on steadily for the past decade in their human-scale BIOS series of experimental life support systems.

CELSS COMPONENTS

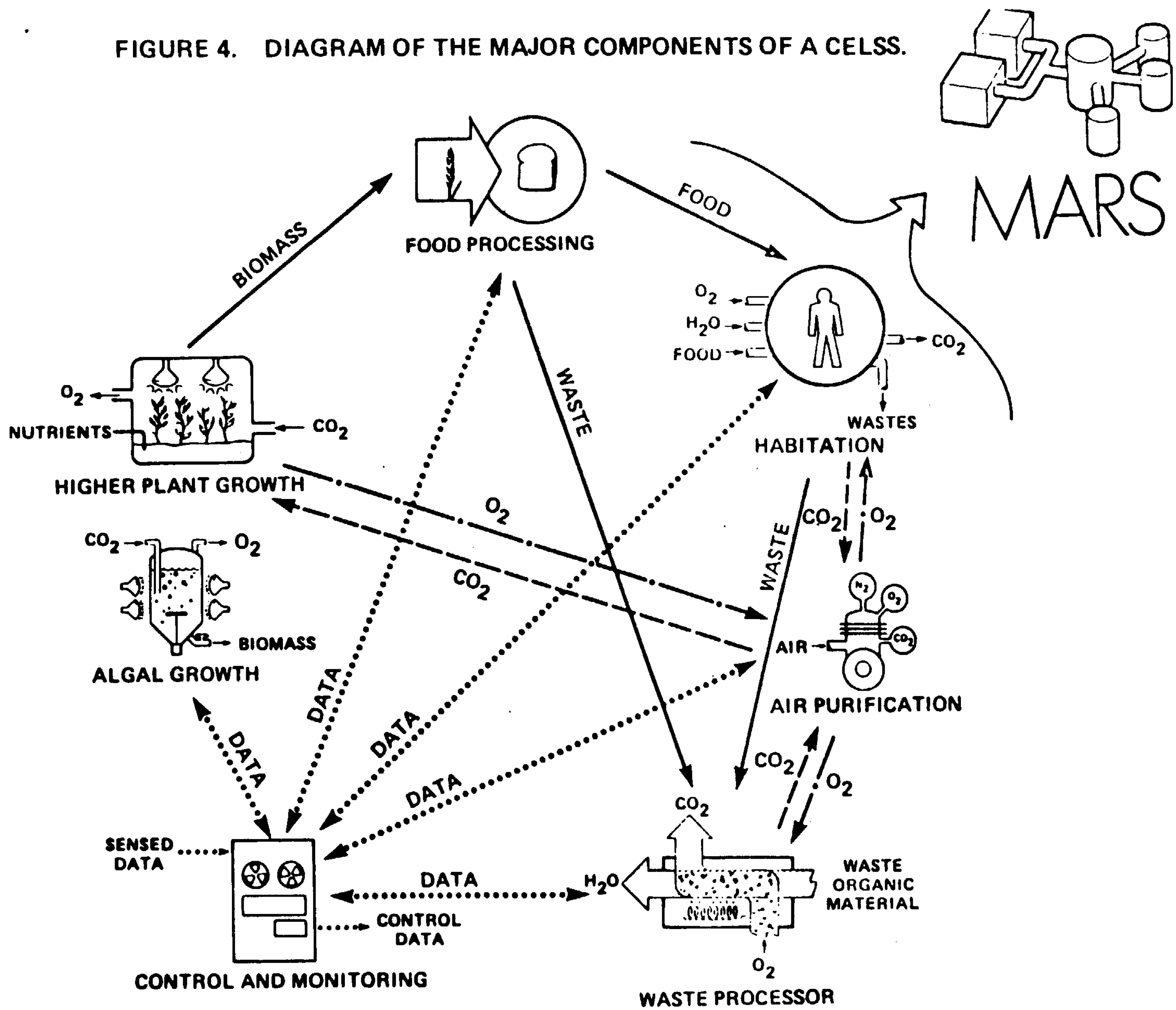
Figure 4 is a diagram of the components of a CELSS. The system will require modules for growing photosynthetic organisms, for processing food from plants, for processing waste materials, for treating water (removal of salts and micro-organisms), for separating gases, for storage of gases, liquids and solids, and for computer control of the system.

Of these components, the largest and the one requiring most power, is the plant or algal growth system. A plant growth system will require lighting with intensities between 10 and 1200 micro-Einsteins/m^2/sec, over a wavelength range from 400 nm to 800 nm. Because, in practice, less than 20% of incident radiation is utilized for chemical reactions, 80% of the incident energy must be removed as heat. Therefore, cooling devices must be incorporated into the growth systems. Cooling surfaces are also needed to maintain humidity between 60 and 95% (relative), as well as to collect water transpired by the plants.

Plant roots will be supplied with nutrients dissolved in water, and maintained at required levels by automated machinery. The plants' roots must be supplied with oxygen, and the stem portions of the plants must be supplied with carbon dioxide. Since during photosynthesis the plants produce oxygen, a gas separation system must be developed to "harvest" O_2 and supply to the plant growth units concentrations of CO_2 higher than is comfortable for a crew. Automated plant cultivation techniques will be required, as will automated food harvesting and food processing techniques, to conserve valuable crew time.

Between 20% and 60% of a plant's mass (depending on the species, and the plant's age) is material that is normally considered to be inedible. However, this material contains nutrients valuable in the human diet, if extracted properly. Cellulose can be converted to sugars, and high

FIGURE 4. DIAGRAM OF THE MAJOR COMPONENTS OF A CELSS.



quality protein can be easily extracted. The remainder of the material can be considered waste, and along with solid and liquid human waste, can be completely oxidized to produce CO_2. Several kinds of waste processors have been investigated: the one that is apparently most efficient (the super-critical water reactor) operates continuously to raise the temperature of a very small volume (about 10 ml) of waste slurry to about 500 C at a pressure of about 250 kg/cm^2 (about 3500 psi). Oxidation is complete in less than 1 second.

PHYSICAL REQUIREMENTS: MASS, POWER, VOLUME

The largest mass requirements in a CELSS are for water, and for plant growth and food processing equipment. Recycling machinery, such as waste and gas processors, constitute a smaller fraction of the total required mass. The masses involved have relatively low densities and will pose no problems for terrestrial lift-off vehicles.

More significant than mass is the volume required for the placement and operation of a CELSS. Volume is dependent on the biological productivity of the system. At the present time, sufficient food can be produced by higher plants, growing and being harvested continously, in an area of about 20 m^2/person. Such a cultivated area is able to supply 2800 calories/day. A mix of plant species can provide the variety of nutrients required in the human diet, but it is likely that preserved foods, such as meats, will supplement diets. An area 20 m^2 will require a height that is dependent on the species and on the growth phase of the plant. Young plants are short, and can be grown in smaller volumes than mature plants; wheat, particularly the short cultivars, can be grown in smaller volumes than soybeans or potatoes. Based upon current area requirements, and assuming dynamic changes in the growth support structures and equipment as the plants mature, a total volume estimate has been made.

PACKING

Recent work by Mel Oleson of Boeing Aerospace (11) has involved the packing of some unique designs for CELSS plant growth equipment within a "standard" Space Station module. The concept is that the system is an experimental one which would be used to investigate micro-gravity effects on all of the component's operations, including the plants. The sizing of the system is based on laboratory data for continuous production of sufficient wheat to meet daily caloric requirements (20 m^2 per person). The plant growth units, waste processing system and the storage reservoirs, sufficient to support 100% of all food requirements for 2.5 crew members, would occupy 4.6 to 5.2 meters of a module 4.5 meters in diameter (72 to 81.7 m^3).

DISCUSSION

It is anticipated that NASA's CELSS program will construct a series of increasingly automated and closed ecological systems during the next decade. It is further anticipated that a small experimental CELSS will be flown on the Space Station to determine the effects of fractional and micro-gravity on both the organisms and the devices that compose the system. Similar directions have been followed by Russian space scientists for the past two decades (12), and European (13) and Japanese (14) scientists have evidenced considerable interest in the problem. The literature in the field is growing rapidly, and the assumption is readily made that practical development of a system is based on sound theoretical grounds. The problems remaining are primarily technological, and their solution appears to be well within grasp.

The critical issues that must be addressed are: The production by organisms of sufficient food, water and oxygen for crews within the mass, power and volume constraints posed by space flight; the stability of a large system whose dynamics are dependent on a variety of organisms; the effects of fractional- and micro-gravity on the higher plants (see, for example, 15) that will probably form the primary source of food; and the extent to which human involvement will be required for system maintenance, or can be effectively replaced by automation and robotics.

The use of a CELSS for human life support during a flight to Mars appears to be within the constraints of the mission, particularly if it were designed to be functional during transit, and was then dropped for use on the Mars surface. Although it would be useful on the surface only for the short period of the human visit, it is one of the major items, requiring many years of lead-time for development unique to manned missions. Once on the Martian surface, it can be re-used by subsequent landing parties.

The existence of a CELSS on the surface would stimulate extensive scientific investigation of the utility of Martian materials in supporting terrestrial organisms. The most abundant gas in the martian atmosphere is CO_2, which is required by a CELSS. One of the least abundant gases is oxygen, which is produced by a CELSS. With properly developed scenarios, an automated CELSS, operating even after the departure of the human crew, might function to accumulate stores of oxygen and biomass useful to crews on subsequent visits to the Martian surface.

Long-term planning and international coordination by life science researchers can efficiently distribute the effort of developing a CELSS among technologically advanced nations, and can create a spirit of co-operation in an essentially non-sensitive area of technology development. Such a cooperation is a logical first step to synchronize a common human effort to visit, for the first time, another planet-the Red Planet, Mars.

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CHEMICAL PRODUCTION ON MARS USING IN SITU PROPELLANT PRODUCTION TECHNOLOGY

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ABSTRACT

In situ propellant production (ISPP) has been examined in terms of its applicability to a manned Mars mission. Production of oxygen from Martian atmosphere was used as the baseline system for ISPP technology assessment. It was concluded that production of oxygen was an important element in a manned Mars mission which could be developed in terrestrial laboratories. Expert system methodology will be required to enable reliable, autonomous production of oxygen. Furthermore, while no major technical breakthroughs are required, this research requires a long lead time to permit its systematic evolution.

INTRODUCTION

It situ propellant production (ISPP) was described initially in 1978 (Ref. 1) as a method for producing rocket fuel for a Mars sample return mission. Use of ISPP resulted in significantly less Earth launch mass than by other concepts. The original concept proposed utilization of atmospheric carbon dioxide and absorbed water in a simple chemical processor to produce methane and oxygen. A major constraint in that study was the availability of water, but an important finding was that primitive chemical processors could be operated at very low throughputs and produce very large quantities of chemicals in time intervals of one year. Subsequently, the technology was investigated for possible applications at other planetary bodies (Ref. 2) and recently, a more comprehensive investigation of the technology required to produce oxygen from the Martian atmosphere for a low mass sample return was reported (Ref. 3). In situ production of chemicals is a logical element in an overall manned Mars program. The purpose of this brief report is to place the technological issues before the manned Mars mission working group.

The production of oxygen from Martian atmosphere is an important process both for life support and as an oxidizer source for ascent vehicle propellant. The processor technology is also an important element in a variety of other scientific and propulsive systems. Since that technology is understood sufficiently to permit specific identification of future research needs and programmatic emphases, it will be described in some detail. The broader issue of production of fuels and other chemicals will be discussed briefly in terms of its potential for future enhancements in an overall Mars exploration program.

ISPP is an important technology in the evolution of a Manned Mars Mission because it exploits the following advantages: (1) Substitution of power generating equipment for chemical mass results in a more flexible system, (2) Radioisotope sources produce much higher energy densities (several orders of magnitude) than conventional rocket fuels when the radioiosotope is used several hundred days, (3) Up to fifty percent reduction in Earth launch mass is possible by offloading the return propellant, (4) Autonomous production of oxygen at Mars is an important element in a manned Mars mission for life support, regardless of return vehicle or surface stay time, (5) ISPP can enable the return vehicle to be sent to the Martian surface well in advance of the manned landing, thereby allowing the Earth return vehicle to be certified prior to sending people to the surface, and 6) The ISPP system can be developed and tested in terrestrial laboratories.

Depending upon constraints, ISPP is an enabling technology. Furthermore, using the oxygen production technology as an example, it is possible to show that ISPP is not a radical departure from presently understood terrestrial systems. The idea of depending on the resources of an unexplored landing site for the ultimate success of a mission is both logical and consistent with historical precedent. Technological issues do remain and will be outlined subsequently.

OXYGEN PRODUCTION

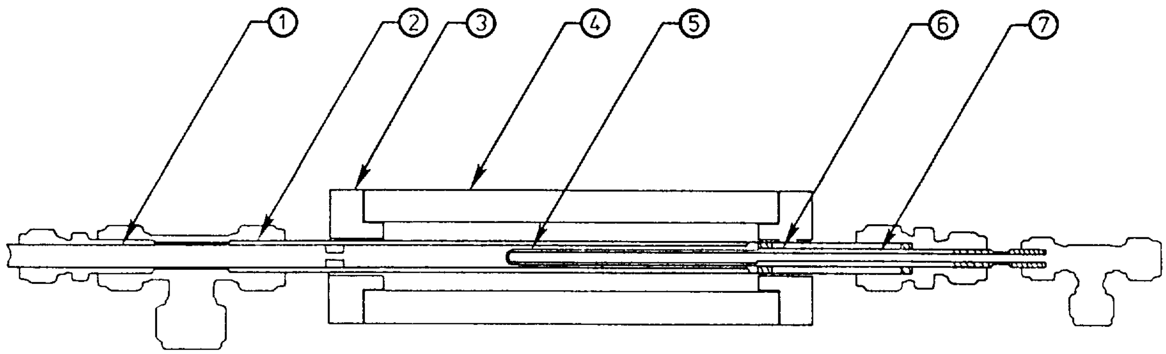
The composition of the Martian atmosphere is well documented (Ref. 4). While there may be slight variations in composition due to location and season, the availability of relatively pure carbon dioxide (95.32%) as a feedstock for oxygen production is insensitive to landing site selection. It is tempting to consider atmospheric water vapor as an equally available feedstock since the atmosphere is relatively humid (Ref. 5), but the low atmospheric pressure and temperatures never allow water vapor to represent more than a few hundredths of a percent by volume. The low density of the atmosphere (on the order of 0.02

kg/m^3) means that relatively large volume flow rates of atmosphere are required to produce useful carbon dioxide flow rates, and the volume flow rates that would be required for water collection would be staggering--to say nothing of the refrigeration requirements that would be imposed for water condensation. Water in small amounts could be collected from the atmosphere for life support.

A nominal Mars environment was suggested in Ref. 3 for oxygen processor design considerations. An atmospheric temperature of 200 K, a barometric pressure of 6.8 mb, an average solar load of 140 W/m^2, a density of 0.018 kg/m^3, and a wind speed of 1.5 m/s were assumed. Using those data, rather detailed thermomechanical designs were developed for a system that could produce 10 kg of oxygen per day. That system was assumed to have a carbon dioxide conversion efficiency of 25 percent which meant that for every mole of Martian atmosphere that passed through the oxygen processor, approximately 0.12 moles of molecular oxygen were produced.

Oxygen collection was accomplished using an electrochemical pump. Essentially, a voltage can be applied to porous platinum electrodes on both sides of a yttria stabilized zirconia membrane to selectively conduct (pump) oxygen ions across the electrolyte. By heating the collected Martian atmosphere to approximately 1270 K, sufficient carbon dioxide dissociation can occur to permit the oxygen collection to occur. That system has been studied extensively by Richter (Ref. 6), and a schematic cross section of the cell is shown in Figure 1. A schematic diagram of the oxygen processor system developed in Reference 3, is shown in Figure 2.

In order to scale up the system described for Mars sample return to manned mission size, the system mass (less electric power generator) can be scaled, to a first approximation, by multiplying the ratio of atmospheric flow rates, raised to the 2/3 power, by the baseline mass. Baseline mass is affected by trades between ascent vehicle mass and refrigeration system/electric power generator masses. However, if the baseline mass was 300 kg for a production rate of 10 kg/day with a conversion efficiency of 25 percent, a 100 kg/day system with a 20



1. INCONEL 600 TUBING 9.5mm × 0.9mm WALL
2. INCONEL 600 TUBING 12.7mm × 0.9mm WALL
3. BORON NITRIDE SLEEVE (2)
4. HEATER, LINDBERG MODEL 50012 TYPE 74-KS (2)
5. 8% YTTRIA STABILIZED ZIRCONIA TUBE WITH 0.1mm PLATINUM COAT ON BOTH SIDES 6.6mm × 1.5mm WALL
6. INCONEL 600 TUBING 9.5mm × 1.5mm WALL
7. INCONEL 600 TUBING 4.2mm × 0.1mm WALL

Figure 1. Electrolytic Cell Design

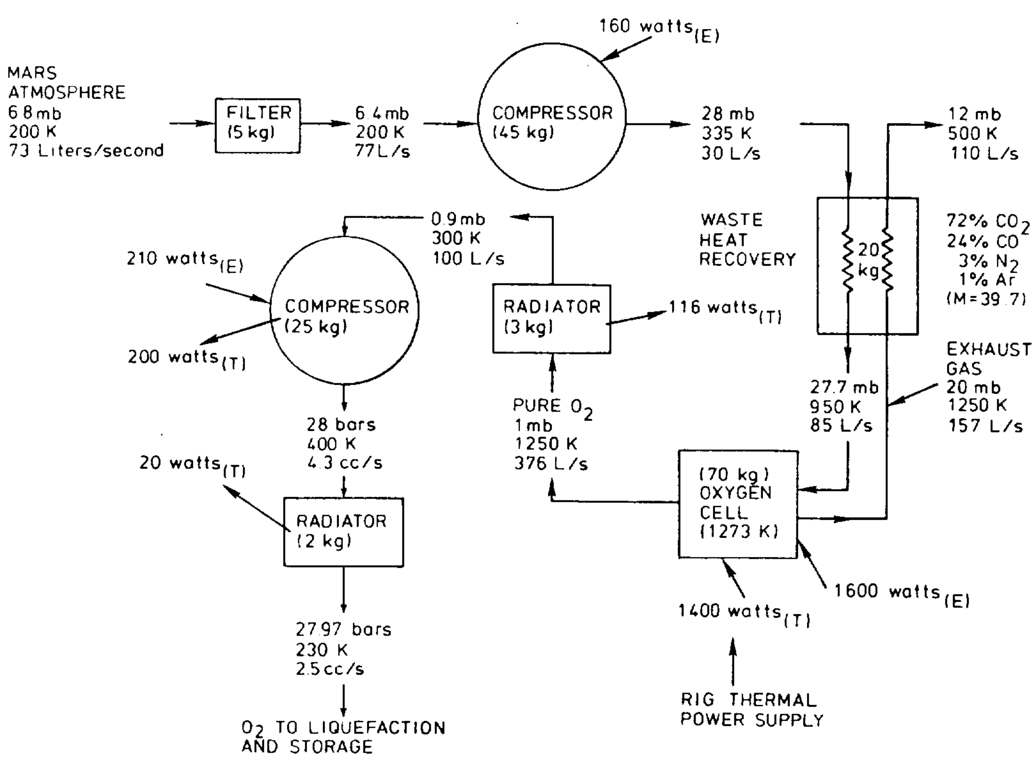


Figure 2. Schematic diagram of 10kg/day Mars oxygen production system.

percent conversion efficiency would have an estimated volume flow rate which is 12.5 times the baseline and an estimated mass of 1630 kg. The electric power requirement scales almost linearly with the throughput. Hence, if the baseline power requirement was 3000 Watts electric, the manned system would require approximately 30 kWe. That system could produce, liquefy and store 10,000 kg of oxygen in 100 days.

The technology issues identified in previous studies will be discussed briefly, and then other issues which relate to ISPP systems for production of other chemicals will be disussed.

Expert Systems

In order to minimize the possibility of a single point design failure in the system, it will be necessary for ISPP hardware to monitor itself and anticipate pending system failure. Proper design of system elements and software should enable these machines to identify pending problems and take evasive action. The nuclear and chemical industries are developing such technology at this time, but they are using massive amounts of historical data and experience to develop these systems. It will be necessary to develop sufficient long term operating histories on prototype ISPP machines to enable them to distinguish between normal degradation and pending failure.

Repair vs. Redundancy

When system elements have characteristics masses and/or volumes which are large, it is not feasible to carry duplicate or parallel elements through the system to avoid single point failures. When considered in the context of the expert system strategy, a system design that exchanges an increase in power requirements or decreased efficiency for repairability using computer controlled manipulators and common components becomes potentially a more reliable and lower total mass system. These systems are very desirable for manned missions in order to keep routine maintenance time to a minimum. Space Station experiments which are designed to develop and test autonomous/repairable chemical processor systems should be given high priority.

Mobility vs. Fixed Site

If ISPP is included on a manned mission, the efficiency and risk related to separating the processor system from the human must be addressed. It does not make sense to carry a 10,000 kg oxygen processor system over large distances. It may make sense to move these systems for short distances in the reduced gravitational environment, to reduce the risk to manned habitat and transportation vehicles.

Filter Systems

A Mars dust filter design has been discussed in Reference 3. Based upon data returned from Mars, the dust does not appear to be a serious problem. However, filter designs which minimize inlet pressure drops and are relatively insensitive to unusual or unexpected particulate loading are needed. These systems can be designed and tested in terrestrial laboratories. The ability to remove the accumulated particulates periodically should be incorporated into the design.

Pumps and Compressors

Pumps and compressors operating at Mars will be in an operating regime which is similar to roughing pumps in vacuum facilities. Compressors will be required to elevate fluid pressures from a few millibars to a few bars. Mechanical stresses will be low, but tolerances and efficiencies of these devices will require a systematic research and development program. This program should be started early enough to permit selection of a set of generic devices which will enable the evolution of a set of common components amenable to self diagnosis and repair. The cryogenic refrigeration components will be similar.

Fault Tolerant Electrolytic Networks

The electrolytic cell system is likely to be a large matrix of cells of the type shown in Figure 1. Based on experience to date, one or more of these cells is likely to fail during an extended operation. It will be very desirable to design this system in a manner which will permit either passive tolerance of cell failures or active alteration of the flow network. This research program could greatly improve system performance and reliability.

Oxygen Distribution and Storage

It will be desirable to store oxygen and other cryogenic liquids in more than one tank. Since these liquids will likely be recycled as they vaporize due to heat exchange with the surroundings, it will be desirable to develop passive fluid management systems which use Mars gravity and density gradients to move fluids to desired locations.

Propulsion Systems

Methane/Oxygen rockets have been built and tested. However, they have not been designed for either Mars sample return or a manned mission. Those engines and a variety of other propellant and propulsion combinations should be investigated to optimize opportunities for manned exploration.

Electric Power

Power generators were not studied here, but the Galileo RTG's are sufficient for sample relturn. SP-100 greatly exceeds anticipated requirements for manned ISPP.

Packaging and Deployment

Depending on the power generation system selected for a manned mission, the packaging problem can be a serious problem. One advantage of ISPP is that a large, potentially hazardous power generating system can be sent in an unmanned mission in advance of the manned mission. Either way, thermal and radiation problems will require careful examination.

Radiators

While the Mars atmosphere is thin, the wind appears to blow nearly all of the time (Ref. 7). The increased energy exchange is very important for radiator surfaces with temperatures approaching Mars ambient conditions.

Trace Contaminants

A research program which identifies potential contaminants that can damage elements in the ISPP system should be undertaken. Simultaneously, realistic probabilities of contaminant existence should be developed based on current knowledge of the solar system. Study of samples collected on a precursor or early mission will be essential.

Complementary System Elements

The machinery required for in situ propellant production can be used intermittently for a variety of manned or scientific systems. This opportunity has been given little attention by the scientific community. However, the electric power and cryogenic cooling capabilities of the ISPP system can enhance many activities ranging from water collection to sophisticated chemical analyses.

Acceptance

While ISPP technology requires a different perspective for manned missions to Mars, it does not require major scientific break throughs. In fact, the research required to place ISPP on equal footing with other options can be accomplished at modest total costs if the program is spread over a long enough period of time. Not only does ISPP become an accepted option with increasing time, but the historical data required to develop expert system-based machinery becomes economical. However, compressing a decade of machine history into less than a year can be very expensive.

OTHER ISPP OPTIONS

I have attempted to use the oxygen production system as a base from which to identify technology issues related to in situ propellant production. It is important to realize that the simplest system is the oxygen production system, since it uses a simple chemical processor operating on an abundant raw material. If a manned station were established in the north polar region of Mars, where there is known water ice, use of water and carbon dioxide to produce methane and oxygen becomes very attractive. In addition, carbon monoxide can be recovered from the oxygen processor system and used as a fuel. Both of these systems involve more than one chemical process. The methanation process was described in Reference 1, but the extraction of carbon monoxide was not. Commercial recovery of carbon monoxide is common, but the systems require thermal energy and relatively complex flow networks (Ref. 8).

Ultimately, all manned extraterrestrial stations will likely require autonomous production of fuels and oxidizers for continued operation. Production of methane and carbon monoxide are both important resource options that should be studied in greater detail.

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CONCEPTUAL DESIGN STUDIES FOR SURFACE INFRASTRUCTURE

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ABSTRACT

The ultimate design of a manned Mars base will be the result of considerable engineering analysis and many trade studies to optimize the configuration. Many options and scenarios are available and all need to be considered at this time. Initial base elements, two base configuration concepts, internal space architectural concerns, and two base set-up scenarios are discussed in this paper. There are many variables as well as many unknowns to be reckoned with before people set foot on the red planet.

INTRODUCTION

The design process begins with some initial requirements. These requirements will inevitably change and increase in number and scope as various concepts are generated, evaluated and refined. This cycle of design and refinement continues until acceptable conceptual designs are defined and detailed design can begin. We are now in the first iteration of this process on the manned Mars mission.

REQUIREMENTS

The requirements we are now considering for the surface infrastructure on Mars are as follows:

Overall

(1) Use proposed and existing equipment to keep down cost--Space Shuttle and Space Station modules, and (2) Provide adequate radiation protection--daily and solar events, in transit and on surface.

Base Elements

(1) Provide habitat(s) for four people initially with future add-on capability, (2) Provide laboratory, both stationary and mobile, (3) Provide means of surface transportation, EVA (extra vehicular activity), and shirtsleeve, (4) Provide vehicle capable of moving modules on the surface, (5) Provide capability to move Martian soil, to clear landing fields, and bury modules if required, and (6) Set up for habitability in minimum number of missions.

Base Elements

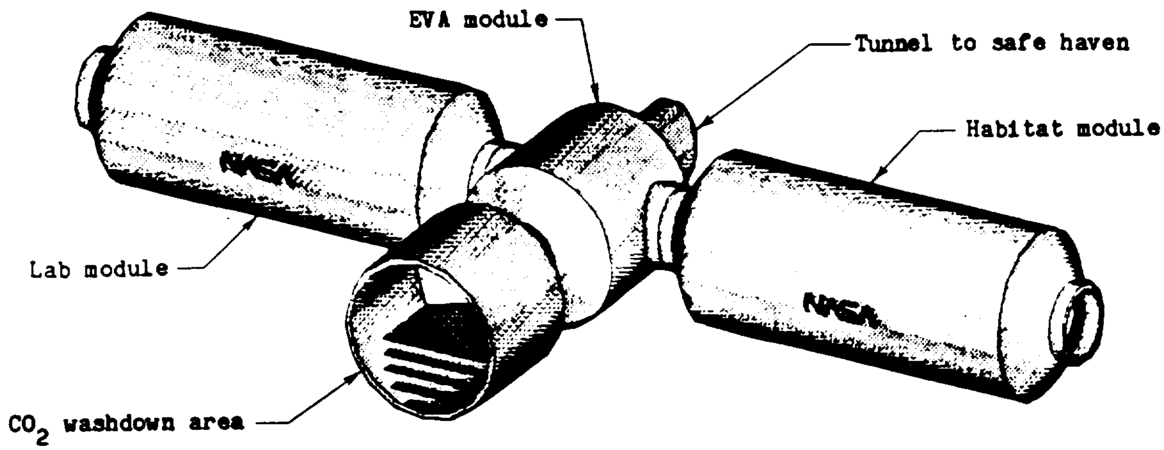
Two basic configurations are being studied for feasibility at this time. They are the "T" configuration and the "little b" configuration, both named for their shape (see figures 1 and 2). These configurations are considered to have the bare necessities for an operational base. It will take perhaps two-three missions to achieve the operational phase with two landers per mission. These configurations are similar in that they both use the same basic elements and are open to the same options, which will be discussed later.

Both the "T" and "little b" configurations contain the following elements: (1) One self-sufficient habitation module: contains bunks, ECLSS, galley, etc., (2) One laboratory module: contains various experiments in materials processing, geology, etc., (3) One EVA module: contains EVA suits, tools, and other equipment for EVA; can be used as emergency pressure chamber, (4) One CO_2 wash down area: pressurized Mars atmosphere is used to remove most of the dust from the EVA suits, (5) One tunnel to base safe-haven (radiation): constructed using shaped charges or other method, and (6) One or more vehicles for moving modules, towing the lab to a new study area, moving soil, or just moving people around the planet.

The habitation and lab modules could be modified Space Station modules. The interior configuration concepts are based on designs to be used on Space Station, modified for 0.4-g. The first iteration of a Mars habitat is shown in figure 3. These designs and architectural concerns are discussed in the "Infrastructure- Interior Space" section below. The lab interior has not been studied yet. These designs are being driven by requirements developed by Fairchild (Ref. 2).

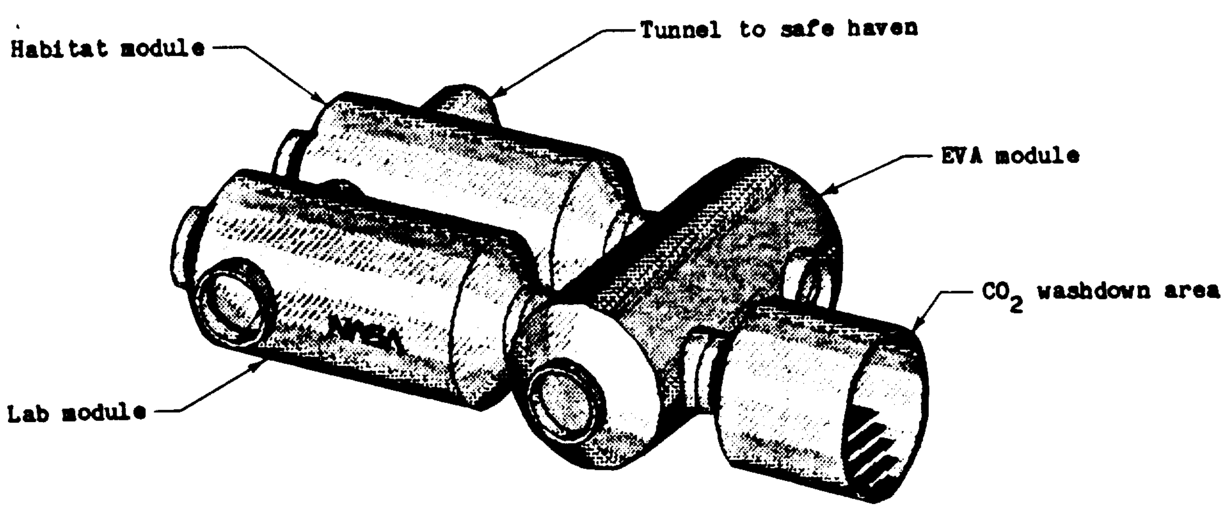
The exteriors will include hatches and docking equipment for mating to other modules. Leveling equipment with some lateral adjustment will be necessary for all of the modules.

The EVA module of the "T" configuration is smaller than a habitat or lab module. It can be sent with either the lab or habitat module to LEO in the cargo bay of the Space Shuttle. The EVA module of the "little b" configuration may also use a modified Space Station module. This allows for docking of two modules to either side of it. The larger EVA module allows the crew more room for suiting up, maintenance of suits, and



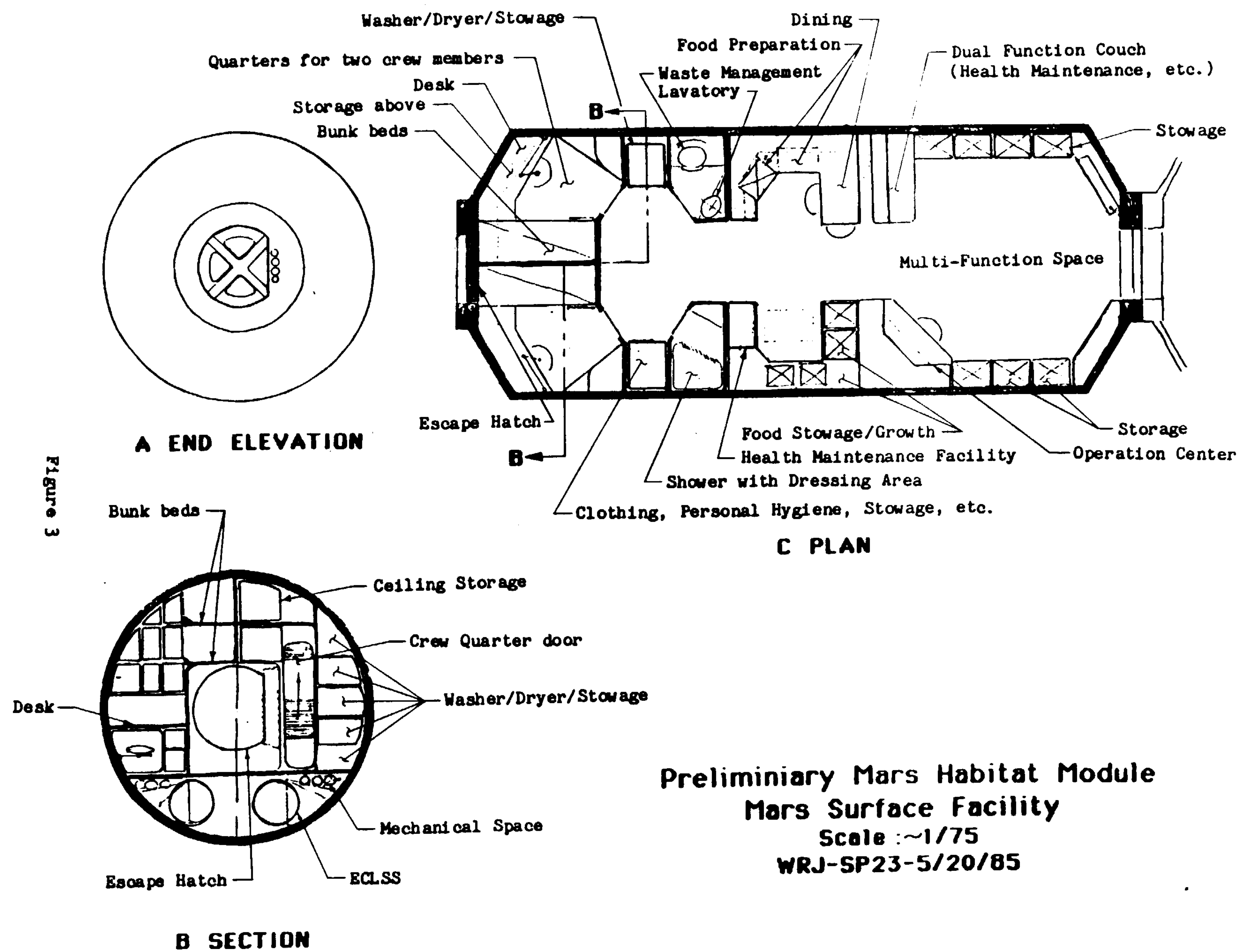
The "T" Configuration

Figure 1



The "Little b" Configuration

Figure 2



Preliminiary Mars Habitat Module
Mars Surface Facility
Scale :~1/75
WRJ-SP23-5/20/85

Figure 3

stowage. This module also provides a second path between habitat and lab modules.

The wash down area will have grated stairs leading to a grated platform raised above the surface. On the walls of this area, shower heads will be mounted for spraying suited crew members. Also, a flexible shower head may be desirable.

A solar event radiation safe-haven could be located through the tunnel shown. The safe-haven could be constructed using an inflatable structure installed in the side of a mountain or buried. The hole in the mountain as well as the burying system could use shaped explosive charges to remove or move dirt. Los Alamos National Laboratories (Ref. 5) is working out the explosive techniques that could be used. Solar events can last several days, so the safe-haven will have to provide ECLSS and contain water and food rations for this time period.

Several vehicles will be necessary on the surface for many different tasks. One vehicle could be developed for most or all of the tasks. But, assuming long treks out of the walking range for a suited crew person, at least one other vehicle will be needed for rescue purposes or as backup for most tasks.

INFRASTRUCTURE-INTERIOR SPACE ARCHITECTURAL CONCERNS

Architecture and Habitability as it Relates to Micro-G and 0.4-G

There has been and is considerable effort in developing habitability requirements such as the current effort of developing these requirements for a Space Station in low Earth orbit with a micro-gravity environment. Such an environment offers unique opportunities in the architectural utilization of space by re-examining the anthropometric requirements for the human body in the neutral body position. The lessons learned with relation to long duration in total man-made environments will be invaluable; however, the derived architectural solutions will not be applicable to the 0.4-gravity environment found on Mars. In general, the architectural environment will be more Earth-like in terms of orientation, proportion, and anthropometric criteria allowing more accurate verification of potential configurations than is possible with the Space Station.

General Concerns

(1) Circulation spaces will possibly have to be designed with slightly higher ceilings than past vehicle designs to accommodate added spring in walk, (2) Openings (i.e., doors and hatches) will be more Earth-like to allow for a more erect posture when passing through them, and (3) In flight optimum man-machine interfaces will differ from those on the surface due to differences in the micro-gravity neutral body position and a full stature standing position.

Possible Solutions for this

(1) To provide totally separate and different architecturally configured transportation modules and surface facility modules, (2) To provide equipment that can be adjusted and/or reconfigured (i.e., adjustable work station heights, movable walls and ceilings). The ability to move heavier objects would help to support this approach, and (3) The ability to move heavier objects, on the surface of Mars, than we are accustomed to on Earth will require equipment to have hold-down mechanisms to prevent inadvertent movement.

Structure

In considering the integration of all the various systems, subsystems, and components, there would be advantages in having these components interchangeable from place to place and from module to module. This can be done by developing a range of standard volumes with similar attachment mechanisms and common system interface connections (i.e., universal power connectors). The advantages of developing this modular infrastructure are: (1) Conversion of stowage space (supplies required for the flight to Mars) into habitable and/or work space, (2) Addition of new equipment without the need to increase the existing facility volume by removing nonessential or inoperative equipment as mission goals change and technology advances, (3) Ability to redefine space use as the facility evolves (i.e., crew quarters could be added near existing ones) thus ensuring controlled growth, and (4) Forces commonality so that equipment components might be usable from one device to the next (i.e., cannibalizing equipment for repairs, etc.).

Functional Layout

As indicated previously, the presence of gravity on Mars drives the character of the environment closer to that of Earth. Therefore, models of existing buildings will provide the support needed to define the optimum configuration for a Mars surface facility.

General Concerns

(1) Minimize the presence of support systems by placing them in remote locations and/or under visual concealment and sound insulation. At all times, this equipment should be accessible without moving unrelated equipment or furnishings. A prime location for this common module equipment will be in the floor cavity under the circulation space.

(2) Separate and dedicate space for significantly different tasks. Although volume can be saved by allowing a space to serve dual purposes (i.e., the galley table doubling as a worktable), the penalties that arise from scheduling to prevent task interference and the inability of designing an object to serve two purposes well outweigh these savings.

(3) In general, if the volume to be inhabited is to be a long cylindrical object, then the functional organization of the space should be as follows: (a) The initial entry from EVA or lab module should be located at one end to act as a buffer between work areas and private areas, (b) The next area should be the galley and dining facilities. Again, acting as a buffer from the working environment to a private environment, (c) The crew quarters should be placed in the furthest and most removed area from daily activities, providing the privacy required for crew quarter activities, (d) The personal hygiene facilities are best located between crew quarters and the public spaces to reduce interference when in use by either group, and (e) Equipment and stowage should be located around the perimeter of the volume so that the operational space required by a user can be shared, with general circulation free space creating a perceived larger overall volume and to take advantage of any additional radiation shielding the equipment may provide.

BASE SET-UP SCENARIOS (OPTIONS)

When the landers reach the surface, there is no doubt they will not be very close together or close to the desired base location. Therefore, the need for vehicle(s) to move the modules is apparent. Also, the modules may need to be buried to provide radiation protection. At this time, it is believed that this will not be necessary, but soil will have to be moved to create landing areas and level the ground to place the modules in an assembled configuration.

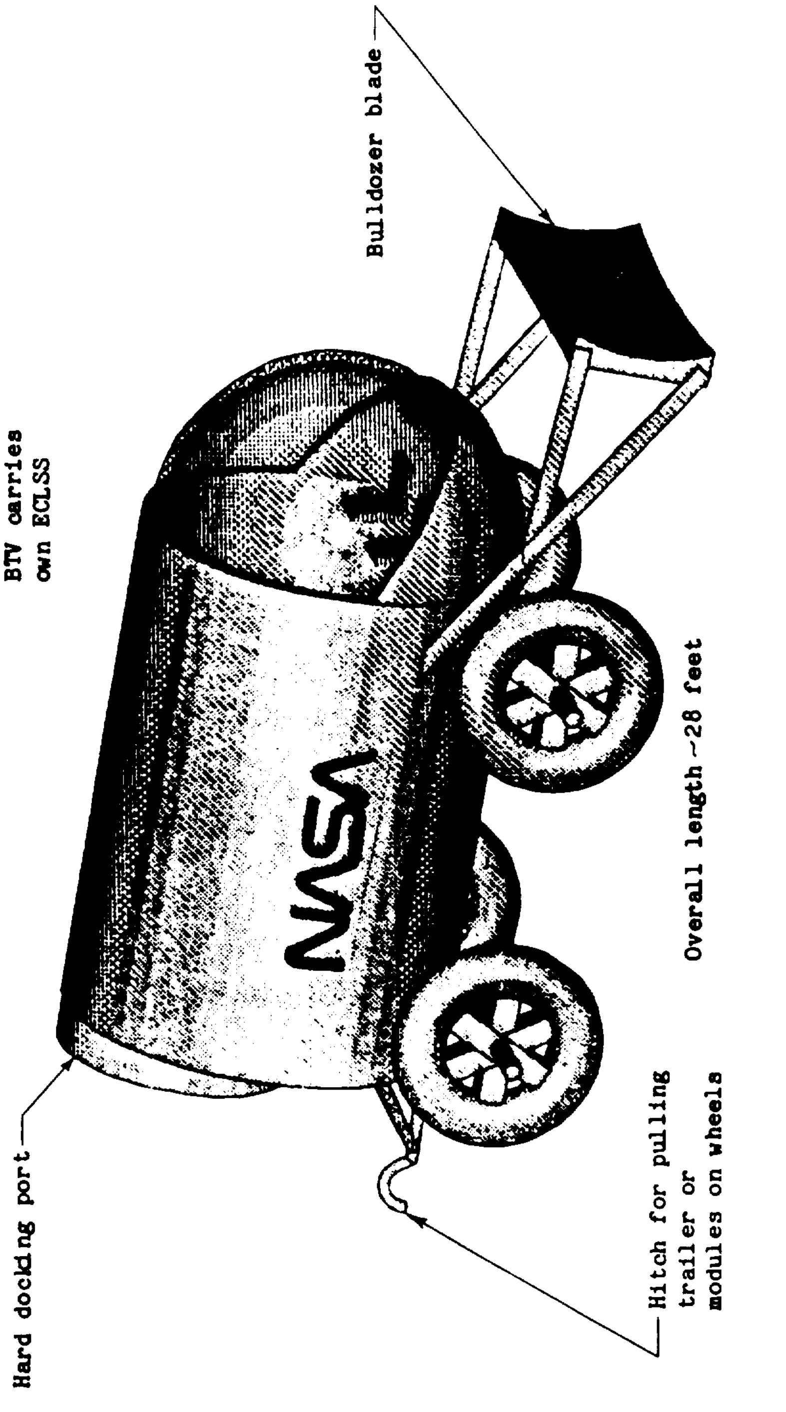
Two options are now being considered: (1) Bulldozer type vehicle with hitch for pulling a module (figure 4), and (2) Crane with a drag bucket and hitch (figure 5).

In both cases, the modules must be moved to the base location. This could be accomplished by putting wheels on all the modules and towing them. Another solution would be to use one trailer to move all the modules. The landers could have leveling and lateral adjustment equipment built in, with detachable descent engines and tanks. Once the engines and tanks are detached and dragged away, the trailer is positioned under the lander stand. Using the leveling equipment or jacks on the trailer, the module and stand are supported by the trailer and moved to the base. The modules could be located and docked to one another, one by one.

The need for a mobile lab could be satisfied by this same method. The lab located at the base could be undocked and towed to a new study area as described in the next paragraph. Another option being considered is a separate mobile lab.

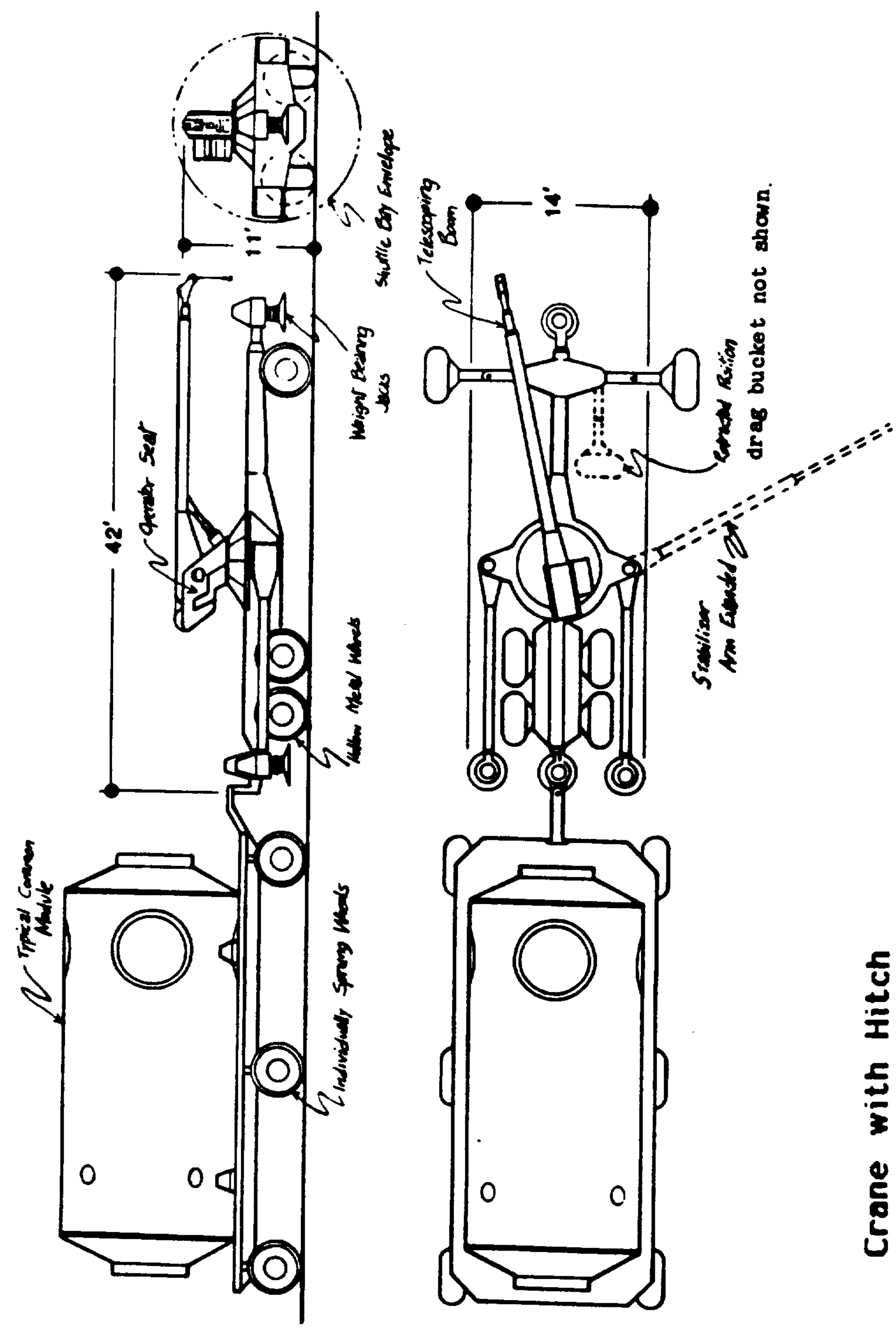
The bulldozer type vehicle (BTV) will carry its own ECLSS on board, capable of supporting the lab module. In this situation, the habitat module would have its own ECLSS also capable of supporting the lab when in hard docked mode at the base. When in transit to a new study location, the lab would be secured to the BTV with a trailer hitch and be docked through a flexible duct. When the new study location is reached, the lab and BTV perform a hard dock, providing a shirt sleeve environment. If EVA's are necessary, the hatch between the two could be sealed off and the BTV depressurized. Having a bulldozer attachment in the front of this vehicle will enable it to get past objects that may cause the crane to go the long way.

The crane will probably be an EVA operated vehicle and have no ECLSS capability. This vehicle may be easier to use for putting the modules on a trailer, without the lander stand or special jackup equipment becoming factors. If this is the vehicle for moving soil as well, a drag bucket would be included. This method of clearing and leveling the land may be a more tedious process than with the BTV.



Bulldozer Type Vehicle (BTV)

Figure 4



Crane with Hitch
(Illustration copied from page 12 of reference 4)

Figure 5

CONCLUSIONS AND RECOMMENDATIONS

(1) The "little b" configuration appears to be more attractive for the addition of future modules because of its compact size, i.e., less land to clear and level, (2) Both the BTV and the crane could be made to work for all the tasks necessary, but perhaps a crane with a bulldozer attachment is preferable and (3) As far as power is concerned, batteries could be used to run the surface vehicles, but some other propulsion form should be developed, perhaps an engine that runs on super oxides or regenerating fuel cells. Power for the station itself could be nuclear (SP-100), solar, etc. This will be the subject of further study.

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LONG RANGE INHABITED SURFACE TRANSPORTATION SYSTEM POWER SOURCE FOR THE EXPLORATION OF MARS (MANNED MARS MISSION)

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ABSTRACT

A hydrogen-oxygen fuel cell system is identified as a viable power source for a long range inhabited surface transportation system for the exploration of Mars. Power system weights and power requirements are determined as a function of vehicle weight. For vehicles weighing from 2700 to 7300 kg in LEO, the total power system weight ranges from 1140 to 1860 kg, with the reactants and energy conversion hardware (fuel cells, reactant storage, and radiator) weighing 430 to 555 kg and 610 to 1110 kg, respectively. Vehicle power requirements range from 45 kw for a 2700 kg vehicle to 110 kw for a 7300 kg vehicle. Power system specific weights and power profiles for housekeeping and the operation of scientific equipment such as coring drills and power tools are also specified.

INTRODUCTION

The extensive and sustained exploration of Mars, once a manned base has been established, will require an inhabited transportation system to explore the planet. This vehicle will require a power system capable of being recharged at the base in order to carry out continuing missions. A hydrogen-oxygen fuel cell may be a candidate for such a power system. The oxygen storage tanks may be integrated with the life support system, with significant weight savings. The waste heat from the hydrogen-oxygen fuel cell may also be used for internal environmental control of the vehicle.

The vehicle weights reported in this study are based on the following mission profile: velocity 10km/hr, range 100 km, duration 5 days, occupancy 5 persons, and slope climbing capacity of 30 degrees uphill for 50 km. Exact vehicle weights will be determined when an actual mission is defined.

POWER SYSTEM DESIGN

In order to define a power system and determine appropriate weights, both power and energy needs must first be determined.

Power and energy must be produced by the on-board system to counteract rolling resistance, carry out a slope climbing function, and operate internal and external equipment required for the mission. The rolling resistance of the Long Range Inhabited Surface Transportation Vehicle (LRIST) is determined for a 32 inch diameter Lunar Rover-type wheel in loose sand (reference 1,2) for the Mars surface gravity environment. The energy budget, which determines the reactant requirements and tank sizes, consists of the reactants needed to overcome the rolling resistance, the increase in potential energy due to slope climbing, and the operation of internal and external functions. A 25% reactant reserve is added for contingency reasons.

The vehicle power requirements are determined by the rates of energy expenditure to meet the rolling resistance and the slope climbing requirements, in addition to the internal power requirements while the vehicle is underway. A 50% power reserve is added to the fuel cell to accommodate a reactant trailer, which may be used to extend either mission range or duration by an additional 100 km / 5 days if desired. An outline of the power requirements and mission requirements is given in Tables 1 and 2.

Seven categories of weights are considered. They are the power dependent hardware, energy dependent hardware, waste heat rejection, radiator, reactants (not trailer), power management and distribution system (PMAD), and electric drive motors. Hydrogen-oxygen fuel cells representing year 2000 technology are used to determine power system hardware weights, including the energy dependent hardware such as Kevlar filament-wound pressure vessels. Table 3 gives the fuel cell and related power system parameters. Figure 1 is a schematic of the fuel cell/electrolyzer system.

TABLE 1

LONG RANGE INHABITED SURFACE TRANSPORTATION SYSTEM
POWER SOURCE FOR THE EXPLORATION OF MARS

Operational Power Requirement

Externally Mounted Corning Drill	10 kw - 3 hrs/day
External Power Tools	2 kw - 4 hrs/day
Housekeeping - Internal Power	5 kw - continuous
Power Reserve	50% (kw)
Energy Reserve	25% (kw-hrs)

Extended Range/Duration

Excess reserve power is provided in order that a "reactant trailor" could be towed to extend the range/duration by another 100 km / 5 days. If desired.

TABLE 2

LONG RANGE INHABITED SURFACE TRANSPORTATION SYSTEM
POWER SOURCE FOR THE EXPLORATION OF MARS

Mission Profile

Range	100 km
Speed	10 km/hr
Duration	5 days
Terrain	

30^0 for 50 km at 10 km/hr (rolling resistance = 0.32 - loose sand)

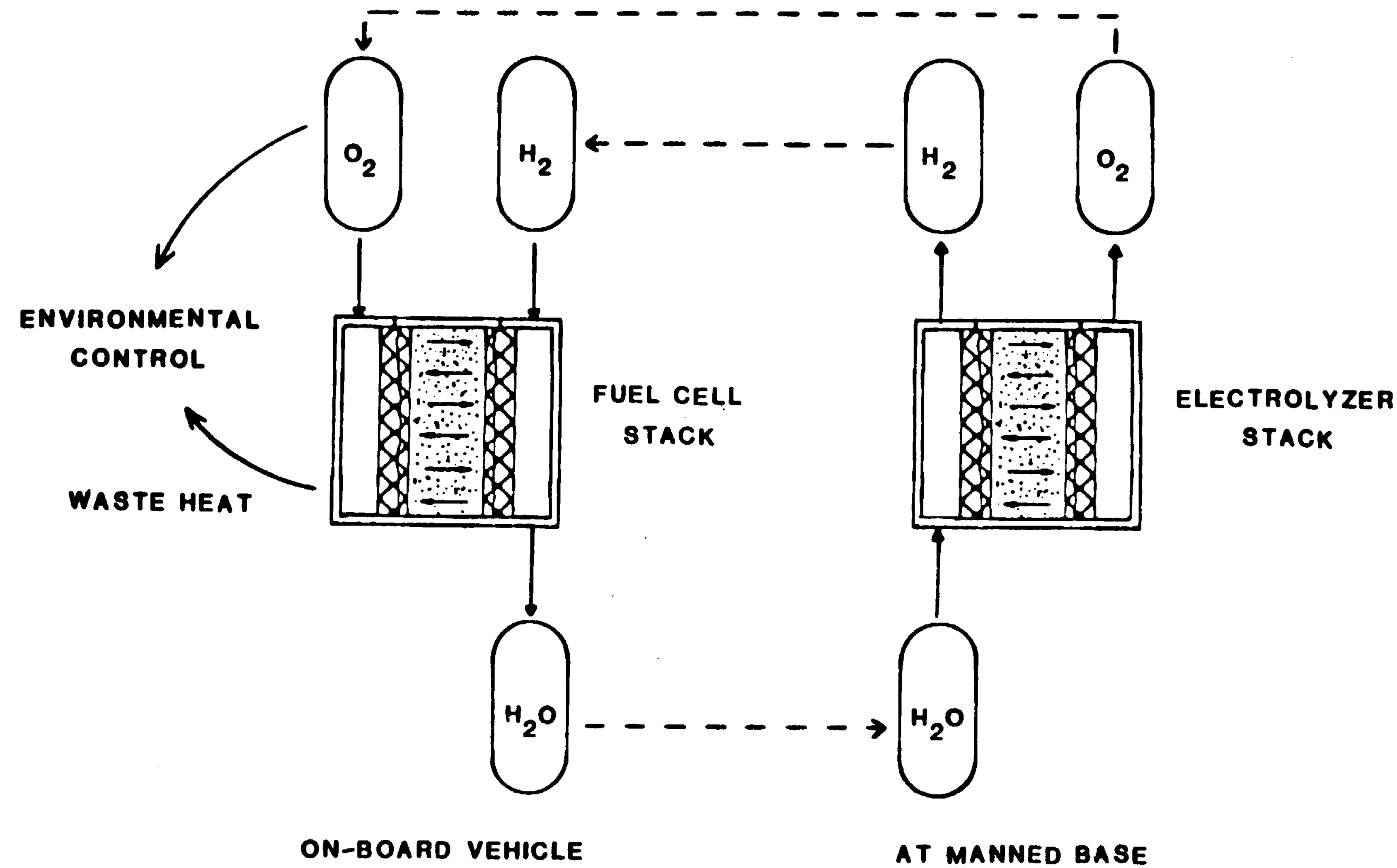

FIGURE 1. - SCHEMATIC OF LRIST HYDROGEN-OXYGEN FUEL CELL SYSTEM.

TABLE 3

LONG RANGE INHABITED SURFACE TRANSPORTATION SYSTEM
POWER SOURCE FOR THE EXPLORATION OF MARS

<u>Power System Specific Weights</u>

H_2-O_2 Fuel Cell	
Power Dependent Hardware (Cells)	2 kg/kw
Energy Dependent Hardware (Tanks)	0.3 kg/kw-hr
Reactants	0.36 kg/kw-hr
Efficiency (Discharge)	75%
Radiator	5 kg/kw
Power Management and Distribution	$\frac{5.31}{P_{KW_e}^{0.25}}$
Electric Motors	1 kg/kw

The operation of this system is outlined in the section describing the mission logistics.

RESULTS

The on-board power system weights are shown in Figure 2. Power requirements, reactant requirements, and energy conversion hardware weights are given as a function of LRIST vehicle weight. The energy conversion hardware consists of the fuel cells, reactant storage tanks, and radiator. The total power system weight, which includes the weight of the power management and distribution system and electric drive motors, as well as the energy conversion hardware and reactant weights, is also represented as a function of total vehicle weight.

The fraction of the total vehicle weight that can be attributed to the power system is given in Figure 3. As the figure shows, the power system represents a smaller percentage of the LRIST vehicle weight as the weight of the vehicle increases.

MISSION LOGISTICS

The LRIST reactant tanks are fully charged at the Mars base prior to the mission. As the mission proceeds, hydrogen and oxygen are combined in the fuel cell to produce electricity and water. The water is stored in a tank for reuse in recharging the vehicle reactant tanks upon return to the base. The life support system may be integrated with the oxygen reactant tanks to provide breathing oxygen and cabin make-up gas. This type of life support/fuel cell integration would give some benefit by reducing the overall vehicle system weight. Upon returning from the mission, the water is electrolyzed at the electrolysis facility (Figure 1) on the base. After electrolysis, the hydrogen and oxygen are pumped into the respective reactant tanks either on the LRIST or the support trailer. The concept shown here would require an additional support system mass delivered to the Mars surface to electrolyze the water and recharge the vehicle reactant tanks. The design of the support system was not considered for this report.

CONCLUSION

This study shows the viability of a hydrogen-oxygen fuel cell power system for a long range inhabited Mars surface transportation vehicle. To provide additional benefits, the power system can be integrated with the life support system to provide breathing oxygen for the crew and

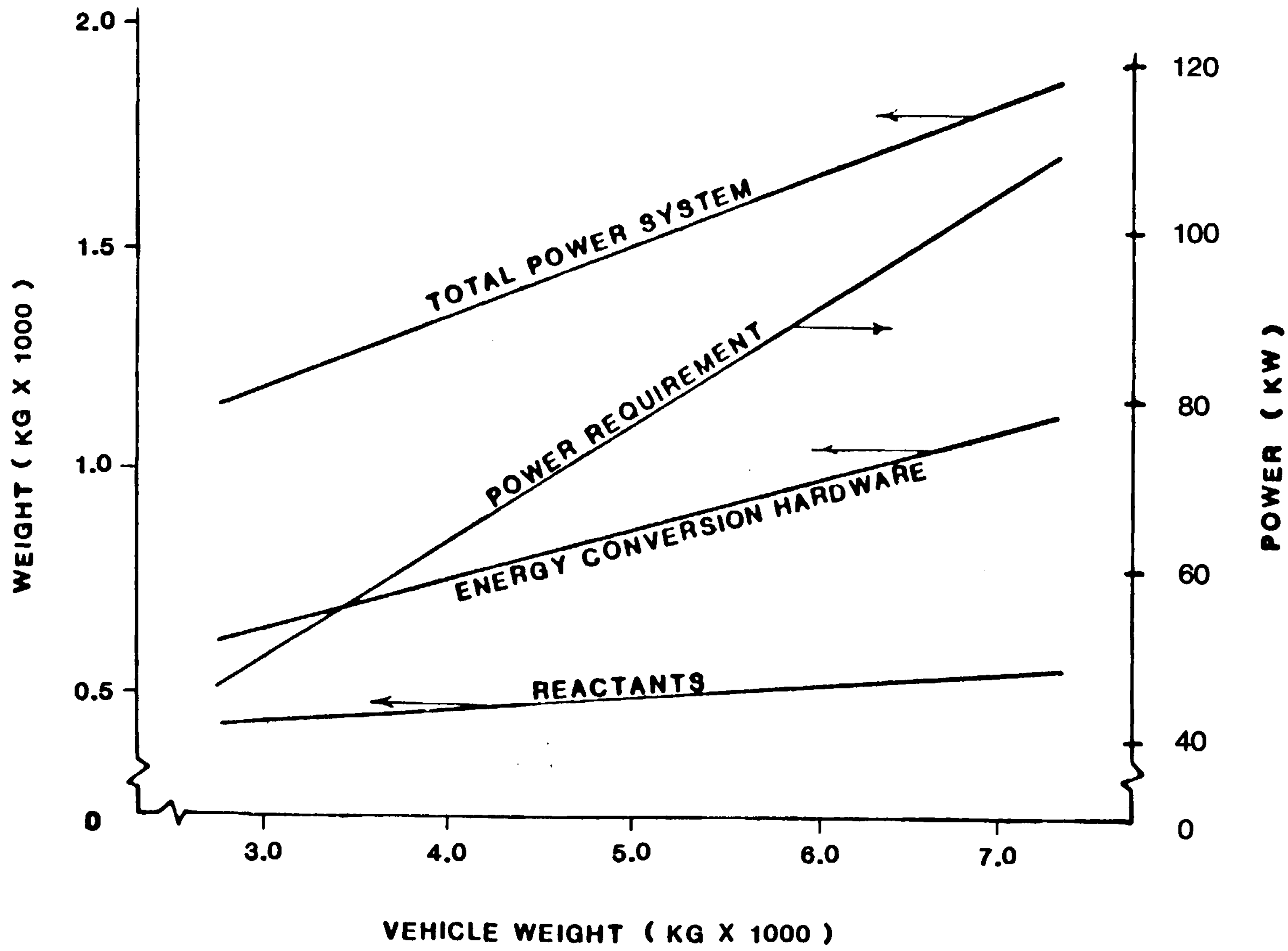

FIGURE 2. - LRIST POWER SYSTEM WEIGHT EFFICIENCY

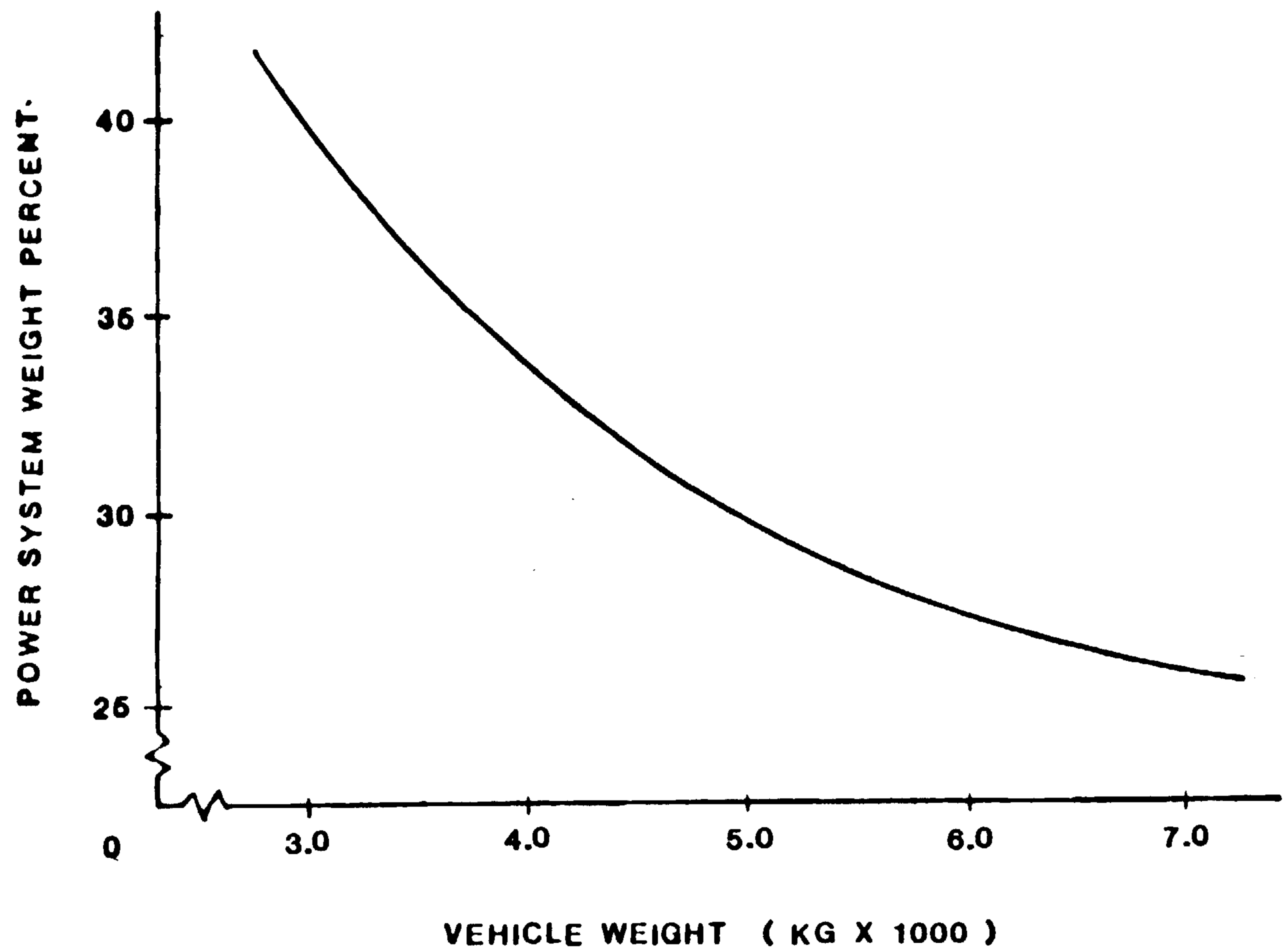

FIGURE 3. - POWER SYSTEM WEIGHT PERCENT AS A FUNCTION OF TOTAL VEHICLE WEIGHT

thermal environment control using the fuel cell waste heat . As Figure 3 shows, the fuel cell power system comprises 41% of the vehicle weight for a light (2700 kg) vehicle, but drops to 25% for a heavier (7300 kg) vehicle.

The mission profile and other parameters used in this study are indicative of those that would result from an actual mission design process. The actual vehicle design weight will depend upon the final mission definition.

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N87-17753

THE MARS AIRPLANE

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ABSTRACT

The concept of the Mars Airplane was developed as a potential vehicle for unmanned Mars exploration. This paper suggests that its most appropriate use would be as an unmanned adjunct to a manned mission. Functions such as reconnaissance, exploration, remote delivery of instruments, etc. are possible. Several operational aspects of such a vehicle are unique compared to Earth operating aircraft.

BACKGROUND

The Mars Airplane concept was developed by JPL and Dryden Flight Research Center personnel in 1977/78 as a potential system for unmanned exploration of Mars. The concept grew out of studies at Dryden of an unmanned aircraft capable of operating for long periods at altitudes near 30.5 km (100,000 ft). Such altitudes at low airspeeds dictated a non-airbreathing engine. This together with the fact that atmospheric density at 30.5 km on Earth is much like that near the Martian surface suggested the possibility of a Mars airplane. Initial concepts by JPL and Dryden looked promising and small contracts were let in 1977 and 1978 to Developmental Sciences Inc. (DSI) to study Mars Airplane design (Ref. 1). In addition, an ad-hoc science working group was convened to study what the Mars Airplane might do (Ref.2).

This paper summarizes the results of those studies and adds some additional thoughts of the author. Two versions of the airplane were studied: 1) the cruiser, which could not land, but flew, taking data until its fuel was exhausted and, 2) the lander which was capable of repeated controlled landings and takeoffs. Only the latter version is considered here since, for a manned mission, reusability of even robot equipment is desirable.

ENGINEERING ASPECTS

Because of the limited data rate capability and desire to minimize power requirements, the Mars Airplane was designed for a cruise speed of 90 m/sec (175 kt). The low atmospheric density (<1% Earth) dictated an airfoil with high cruise lift efficiency and a high aspect ratio wing.

This combination together with the low gravity yields excellent range capability as will be discussed later.

To minimize both mass and drag as well as for convenience in stowage, an inverted Vee tail was used. The overall configuration appears in Figure 1 (from Ref. 1). The resemblance to a modern high performance sailplane is obvious. The dimensions are similar as well. The propeller is quite large by Earth standards in order to perform efficiently in the thin atmosphere. The tail is also large to allow for large center-of-mass travel.

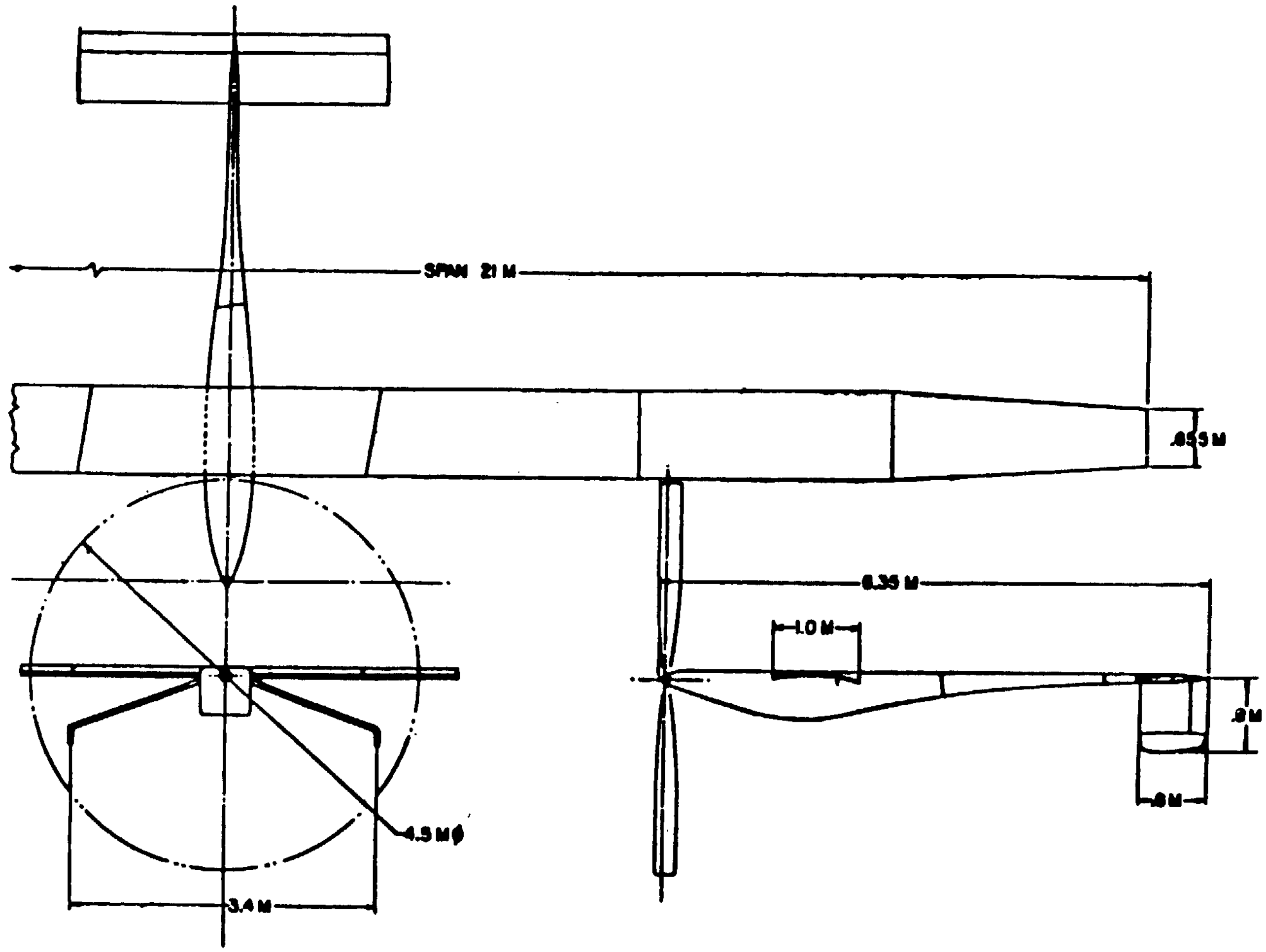
Martian conditions at these airspeeds result in a very low Reynolds number for a vehicle of this size. Specifically, the Reynolds number for the Mars Airplane at cruise is of the order of 4.5×10^4 compared to 3×10^6 for a typical light aircraft. Most experience in this range is with free-flight model aircraft. Thin high efficiency airfoils derived from the work of Eppler at the University of Stuttgart were used in the DSI design Ref.1. Since the time of that design (1978) there has been increased interest in low Reynolds number airfoils and some of this new work may be applicable.

The baseline powerplant for the Mars Airplane was a hydrazine-fueled reciprocating piston engine developed by Jim Akkerman of NASA Johnson Space Center. This engine functions much like a reciprocating steam engine. The hydrazine decomposes into a hot gas in a catalyst bed similar to that used in monopropellant rocket engines. This gas is then valved into the cylinder of the reciprocating engine which vents it overboard following the power stroke. This engine was flown successfully on the Dryden "Mini-Sniffer" aircraft. The horsepower requirement for the Mars Airplane as designed is 15HP. The complete engine when developed for Mars should have a mass of about 13kg. The cruise specific fuel consumption is expected to be 2.2 kg/HP-hr (4.85 lb/HP-hr).

An alternative powerplant was also investigated. This consisted of a light-weight samarium-cobalt electric motor with a gearbox and a solid state inverter. The unit was expected to weigh roughly the same as the hydrazine engine, assuming very high performance of the Sa-Co motor. At the time of the design, this was considered speculative by some and needs to be reinvestigated. Electrical power was provided by primary batteries, probably lithium thionyl chloride. The energy density usually

FIG. 1

BASIC AIRPLANE CONFIGURATION



quoted for these batteries is about 66 watt-hr/kg (300 watt-hr/lb). For this study a value of 649 watt/hr-kg (295 watt-hr/lb) was used. A much higher energy density of 1199 watt-hr/kg (545 watt-hr/lb) was also evaluated. This required heroic weight reduction measures in the battery package and must be considered speculative.

Landing and takeoff were to be done vertically using variable thrust monopropellant rocket engines derived from the Viking lander. The fuel for these engines is hydrazine. In the case of the hydrazine engine airplane, the rocket engines and the reciprocating engine would draw propellant from a common supply, whereas the electric vehicle required a hydrazine supply just for the rockets. (Note that a disadvantage of the electric airplane in regard to landing and takeoff is that it never becomes lighter than it was initially, while the chemically powered version loses mass as propellant is burned, thus making landing and takeoff toward the end of a particular trip much less costly in fuel.) For a manned mission, the airplane would presumably be reused in a manner similar to remotely piloted vehicles (RPVs) on Earth. In this case, the chemically fueled version may be more desirable, since rechargeable batteries have much lower energy density than the primary batteries originally postulated. If the capabilities of the manned surface base included In-Situ Propellant Production (ISPP), the airplane could be supplied with these propellants. Otherwise, residual propellant from the lander stage should supply ample quantities. Engine types which might be considered include reciprocating, gas turbine, and electric motors driven by fuel cells. (As an aside, if propellant manufacturing capability exists to generate CO and O_2 from the Martian atmosphere, a fuel cell operating on these materials would be most useful for airplanes, rovers and other portable power needs rather than using the combination in a combustion mode. This concept deserves further attention.)

PERFORMANCE

The hydrazine powered version of the Mars airplane was estimated to have an operating ceiling of 15 km on a Mars standard day at minimum mass (150 kg.) At maximum gross mass of 300 kg, the ceiling would be about 8 km, from which altitude the power-off glide range would be over 250 km. Initial rate of climb at low altitudes is estimated to be 12.7 m/sec .PA

(2500 ft./min.), which is quite respectable and bodes well for terrain avoidance capability and ability to cope with downdrafts.

Figure 2 (Ref. 1) presents range performance. The numbers are quite respectable even if one ignores the rather debatable upper curve. These numbers make no allowance for landing propellant, and would be typical of a reconnaissance sortie with landing on a prepared surface at the base rather than vertically. Maximum sortie radius would be half the range. An early landing at high mass could reduce the range by 30-40%, while retaining fuel for a final vertical landing at low mass would have a lesser penalty.

OPERATIONS

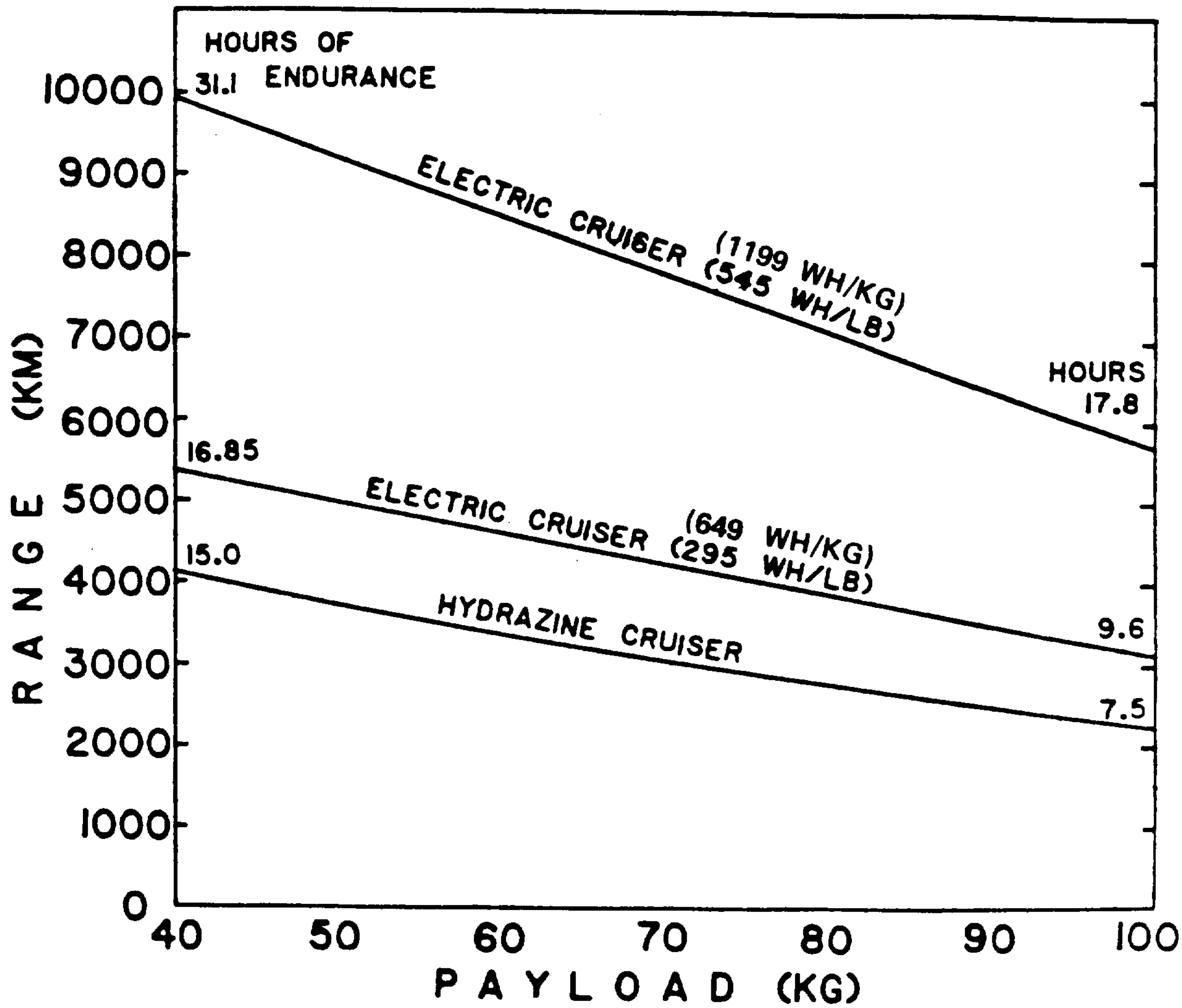
The vehicle would be unstowed after landing and assembled by the crew much in the manner of sailplanes on Earth. (This avoids one of the serious drawbacks of the unmanned version; namely, how to achieve self-deployment of a very complex folded structure while dangling from a parachute.) Launch might be via catapult or the internal rockets. Guidance and navigation would be preprogrammed, probably using inertial systems with landmark identification for updates. Upon its return to the vicinity of the base, control would be assumed by a crew member on the ground who would control the landing, following normal Earth RPV practice.

Figure 3 shows a possible vertical takeoff profile suggested by DSI. It is the author's opinion that propellant could be saved by starting the propeller as soon as ground clearance is adequate and accelerating directly into wing-supported flight rather than following the lofted trajectory shown. Landing would invoke a technique called "stable stall".

Stable stall is a technique originally developed for the recovery of free-flight model aircraft. Briefly, it involves deflecting the horizontal stabilizer to a very high angle, placing the aircraft in a deep stall. The aircraft descends vertically in a flat attitude at a modest and quite predictable rate. NASA studies show that the technique can be satisfactorily applied to larger aircraft as well. In the case of the Mars Airplane, the descent terminates in a rocket-braked landing. Figure 4 shows the profile of such a descent. Creation of a runway at the base site would allow for conventional landing thus eliminating the need to

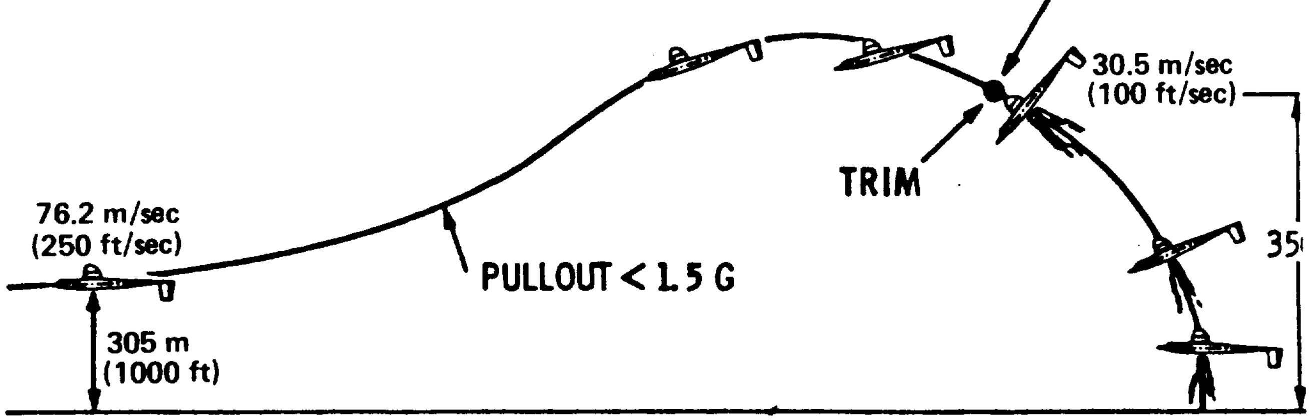
FIG. 2

CRUISE PERFORMANCE



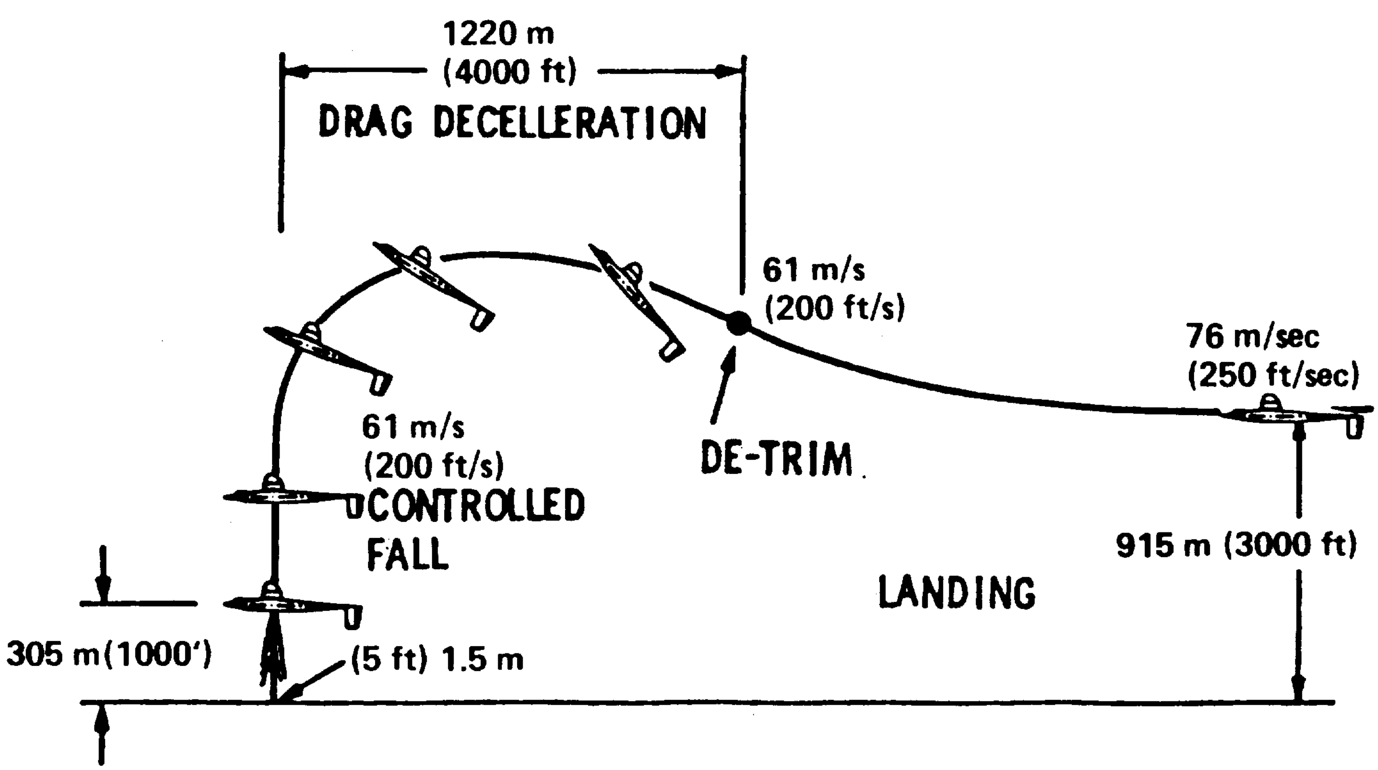
AUW	300 KG	PROP EFFICIENCY	.85
WING SPAN	21 M	HYDRAZINE ENGINE	
WING AREA	20 M^2	CRUISE SFC	4.85
CRUISE ALTITUDE	1 KM	ELEC. MOTOR EFFICIENCY	.85
LIFT/DRAG AT 300 KG	27.75	AUX. POWER CONSUMPTION	.4 KW

FIG. 3



VERTICAL TAKEOFF

FIG. 4



VERTICAL LANDING

carry propellant for a final vertical landing, on each sortie. To minimize length of landing gear, all landings would be made with the propeller stopped in the horizontal position. Mass could be saved by using skids rather than wheels if only vertical or catapult takeoff is used.

FUNCTIONS OF A MARS AIRPLANE

An unmanned Mars Airplane could perform a variety of useful functions in support of a manned Mars mission. Examples include:

1) Reconnaissance sorties to provide detailed route maps in support of surface traverses by the crew in rover vehicles.
2) Scientific surveys of large regions or particular sites distant from the base or otherwise difficult to reach.
3) Deployment of an array of remote observing stations (penetrators, surface packages, or both) either by air drop or by landing.
4) Delivery of high priority hardware to a crew far from base on a surface sortie and/or return of priority samples, etc. from the rover crew to the base.

Other functions will probably arise as the capability of the Mars airplane becomes better understood and the mission definition improves. Even the set of functions listed above could be of substantial benefit. For example, detailed aerial maps will allow more rapid cross-country traverses and warning of possible hazards. Optimum routes can be selected ahead of time. Large area aerial surveys supplementing work done from orbit can be of great geological significance. The ability to deliver science instrument packages to remote locations will be of substantial benefit, since it could not be done by surface rover but would have to be done from space using individual entry packages. The ability to deliver supplies of one sort or another to a rover crew could be vital in case of hardware failure or a medical emergency.

MANNED MARS AIRPLANE

The question will inevitably arise as to feasibility of a manned airplane for Mars. While (to the author's knowledge) this has not been studied, there seem to be no technical reasons to prevent it. In fact, the Mars Airplane described herein has adequate payload mass capacity to carry any normal human. However, it could not carry a human in a spacesuit and full complement of life support equipment without exceeding the design mass. Also, the compact fuselage volume lacks room for such a payload. A realistic manned Mars airplane would have to be considerably larger than the vehicle described here, especially since it would probably be desirable to carry a crew of two and a payload. The mental picture that develops is that of a vehicle of the general size and appearance of the U-2, except being propeller driven.

The technical difficulties involved in creating such a vehicle do not appear any more formidable than those involved in the small unmanned craft. Some practical difficulties are stowage (especially for atmospheric entry), assembly and handling on Mars, and propellant consumption.

CONCLUSION

Based upon the work summarized in this paper, there appear to be no serious technical difficulties involved in designing and operating a Mars Airplane. It further appears that such a vehicle could be most useful in extending the capability of the Mars surface crew and possibly enhancing safety.

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MARS SURFACE TRANSPORTATION OPTIONS

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ABSTRACT

As the number of scientific experiments for the surface of Mars grows, the need for effective surface transportation becomes critical. Because of the diversity of the experiments proposed, as well as the desire to explore Mars from the equator to the poles, the optimum surface vehicle configuration is not obvious. Five candidate vehicles are described, with an estimate of their size and performance. In order to maximize the success of a manned Mars mission, it appears that two vehicles should be designed for surface transportation: an advanced long-range rover, and a remotely-piloted airplane.

INTRODUCTION

In order to maximize the usefulness of a manned Mars base, surface transportation vehicles are required. These vehicles would transport both men and instrumentation to sites not within walking distance of the landing craft or home base, and also expedite the return of samples. Since a large number of scientific missions are envisioned, several types of vehicles should be considered in an effort to determine the optimum configuration. Consideration must be given to the size and weight of the vehicle, since it must be transported to the Martian surface by a landing craft.

It is assumed that a Mars surface transportation vehicle will be operating from a Mars base located within plus-or-minus 30 degrees latitude from the equator. Scientific experiments will be done in locations ranging from near the Mars base to the nearest pole. The most likely candidates for a Mars surface vehicle are: 1) a lunar-type rover; 2) an advanced rover equipped with a life support tent; 3) a large-scale mobile lab; 4) a robotic walker; and, 5) a remotely-piloted airplane. Following is a description of these vehicles and an estimate of the performance one might expect to obtain from such vehicles. Note that the capabilities and size are rough estimates; no detailed design has been done on any vehicle. Also, these options do not include quantum advancements in technology which might make another type of vehicle possible.

LUNAR-TYPE ROVER

The first option is a two-manned, four-wheeled rover similar to that used in the Apollo program. A direct derivative of the lunar rover could be very effective on Mars, with only minor design changes needed. Such a vehicle would weigh more than the lunar rover because of the requirement to operate in the increased gravitational environment on Mars, perhaps weighing about 600 kg. It could be transported in a container about 3.5-by-2-by-1.2 meters in size, and carry a payload of 680 kg.

Such a vehicle could be expected to have a range of 40 km round-trip, while cruising at a speed of about 7-9 meters per second. An important consideration for any Mars surface vehicle is its ability to operate on the rough terrain without excess maneuvering. The lunar-type rover would be able to drive over a rock with about a .15 meter diameter, and climb a 20 degree grade. This vehicle would satisfy a good deal of the scientific requirements, but would lack the long-range capability needed for polar exploration. Also, each scientific expedition would be necessarily short since the scientists would be restricted to their life support suits. However, this design requires the least amount of technological development; since it is derived directly from a proven vehicle, the design need only be optimized for operation on Mars.

ADVANCED LONG-RANGE ROVER

The second option is similar to the lunar rover but includes enhancements to improve its capabilities. This vehicle may be 4- or 6-wheeled, and will be larger than the lunar-type rover. It will include the capability to plug life support suits directly into the rover power supply, thereby increasing the range limit imposed by the life support systems. Also, the rover will carry an inflatable life support tent to be used for sleep periods. The scientists will be able to remove their life support suits for cleaning during this sleep period.

This vehicle will weigh about 770 kg., and will be packaged in a container about 4.5-by-2.5-by-2 meters in size. A range of 125 km round trip at a speed of 7-9 meters per second can be expected. The rover will be able to carry a payload of about 800 kg., including the life support tent. The rover should have the same terrain capabilities as the lunar-type rover (drive over .15 meter rock, climb a 20 degree slope), but may improve upon this with an advanced design incorporating several wheels and an active control system.

Obviously, this design requires more advanced technology than the lunar-type rover. A better power supply is required to power both the rover and the scientists' life support suits. Weight saving techniques need to be employed in order to reduce the burden on the landing vehicle. Also, the life support tents needed to allow long range need to be developed. This design should significantly improve upon the capabilities of the lunar-type rover without an undue weight penalty or unattainable technological advancements.

<u>LARGE-SCALE MOBILE LAB</u>

Given that the Mars base will have a fully-instrumented scientific laboratory, there are two options for lab configurations which include a large-scale vehicle. These configurations are the Mars Autonomous Research Vehicle (MARV), and a complete laboratory on wheels. In either case, the vehicle will be capable of maintaining life support for 5-day scientific excursions. The complete lab on wheels concept is one in which the entire lab module is mobile; the lab is driven to the site of the scientific experiment, and all the analysis is done there. Obviously, this vehicle will be quite large. Only the details of the MARV will be discussed here.

The MARV is a self-contained life support vehicle which has a limited laboratory capability. This vehicle would travel to a desirable site and perform scientific experiments within the scope of the lab's instrumentation. Samples would also be returned to the permanent, complete lab at the Mars base for further, more extensive testing. The MARV would be smaller than the complete lab on wheels; it would be transported to the surface in a 9-by-9-by-3 meter container. Weighing about 4500 kg., it would be capable of carrying a 1800 kg. payload of scientific equipment, life support suits, and so on. At a speed of 7-9 meters per second, its range over the course of a 5-day trip would be about 600 km. As with the two rover concepts, the MARV's terrain capabilities include climbing a 20 degree hill and negotiating a .15-.3 meter rock.

Although the first-cut design of the MARV might resemble a Winnebago, a significant amount of work can be done to optimize a configuration for operation on the surface of Mars. Also, much needs to be done in the area of life support systems in mobile, self-contained vehicles.

ROBOTIC WALKER

Because recent experience with walking robots seems to indicate an increasing capability of robotic walking vehicles, it is worthwhile to consider a walker among the possibilities for Mars surface transportation vehicles. Such a vehicle would be smaller than the previously-described vehicles, but its short range capabilities are similar. It would most likely be remotely-piloted (unmanned).

A walker large enough to carry some instrumentation would weigh about 225 kg., but because its 'legs' will fold up significantly, its transportation size would be about 2-by-2-by-2 meters. Because the vehicle lacks a range-constraining life support system, it would likely be able to cover 125 km. at a speed of about 4.5 meters per second. The walker would be able to carry about 225 kg. of payload, and advanced robotics would make up somewhat for the lack of scientific personnel in the operation of experiments. Current walking robots have shown the capability to climb a 45 degree slope, and to walk over a 1 meter rock. This, obviously, is a significant improvement over conventional wheeled vehicles. An effective walking vehicle for use on Mars would depend on significant improvements in remote-control capabilities. Robots have only recently, through the use of on-board computers, been able to negotiate difficult terrain and to master an efficient gait. The most effective load-carrying design is certainly not obvious at this point. Despite the walker's lack of long-range capabilities, its maneuverability suggests it may yet find a place on the Mars base.

REMOTELY-PILOTED AIRPLANE

The final concept for consideration as a Mars surface transportation mode is an unmanned, remotely-piloted airplane. This airplane would fly from the Mars base, land occasionally to pick up samples or drop off instrumentation, survey the area traversed, and return to the base. Since it is assumed that the base will be within 30 degrees latitude of the equator, and it is desired to explore the poles, the airplane must be able to travel from the base to the nearest pole and back without refueling. Although some work has been done on the design of an airplane that is deployed from orbit, the constraints put on the configuration from the reentry phase make a Mars-based vehicle more practical.

The aircraft would be designed to fly at altitudes less than 6 km., and at a speed of about 75 meters per second. The range will be at least 4500 km., in order to reach the nearest pole. It is expected that such an airplane will weigh about 900 kg., and will be packaged in a 6-by-1.5-by-1.5 meter container. The useful load of the airplane will be about 225 kg. Although current technology suggests that a hydrazine engine may be the best power plant, high-density electric batteries or solar cells should also be considered.

The two most significant stumbling blocks to the successful design of a Mars airplane are the aerodynamic configuration and an accurate remote control system. Because of the low air density on Mars, a conventional airplane will necessarily have a large wing span. This presents a problem in efficient storage and transportation. Besides a conventional, fixed wing configuration, others that should be investigated include lighter-than-air, rotary wing, and propulsive lift (vertical takeoff and landing) configurations. Because of the great distance involved, the design of an accurate remote control mechanism may be difficult. Perhaps a satellite in stationary orbit may prove useful in this area.

The remotely-piloted airplane obviously shows some advantages over ground-based transportation modes. It appears to be the only configuration that will achieve the goal of travel to the poles in a timely manner. The only difficulty caused by the rough terrain occurs in takeoff and landing, and this problem may be overcome by vertical takeoff and landing capability. However, because the aircraft will be unmanned, the success of some experiments may be compromised.

CONCLUSION

A surface transportation vehicle is essential to the success of future manned Mars missions. Although there are several vehicle configurations which will accomplish many of the desired objectives, none will do it all. Because the remotely-piloted airplane is the only vehicle studied which has sufficient range to reach the poles, it should be strongly considered. However, because the airplane may not be as efficient in conducting research near the base, a rover may also be necessary. It appears that an advanced technology rover equipped with a life support tent will successfully fill the need for manned, near-base explorations. It should be emphasized that no detailed design was done; the values given for vehicle size and performance are estimates of what one might expect from such a vehicle.

As a result of this investigation, it is recommended that a detailed design study be undertaken on two Mars surface vehicles. To completely explore the area near the Mars base, a rover should be designed to travel about 125 km. while providing power for two life support suits. Also, a detailed investigation of the feasibility of life support tents should be performed. A Mars-based airplane should be designed to fly at least 4500 km. at an altitude of less than 6 km. As previously mentioned, the configuration studies should not be limited to conventional, fixed-wing designs. The use of these two types of vehicles will make the extensive scientific exploration of Mars a successful program.

SUMMARY TABLE
MARS SURFACE TRANSPORTATION OPTIONS

VEHICLE	WEIGHT (KG)	PACKING SIZE (METERS)	RANGE (KM)	PAYLOAD (KG)	SPEED (M/S)	SLOPES/TERRAIN CAPABILITIES	TECH. ADVANCEMENTS REQUIRED
Rover	600	3.5x2x1.2	40	680	9	20 deg/.15m. rock	refinements
Advanced Rover	770	4.5x2.5x2	125	800	9	20 deg/.15m. rock	life support tents
Mobile Lab	450	9x3x3	600	1,800	9	20 deg/.15m. rock	life support
Walker	225	2x2x2	125	225	4.5	45 deg/1m. rock	control, load-carrying
Airplane	900	6x1.5x1.5	4,500	225	75	---	RPV control, aero configuration

VEHICLES -
1) Rover ------- Lunar-type rover
2) Rover w/tent ------- Advanced rover with inflatable life support tent (extended range)
3) Mobile lab ------- Large scale enclosed, pressurized rover
4) Walker ------- Small (manned or unmanned) ambulatory robot
5) Airplane ------- Remotely-piloted airplane

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MISSION AND SURFACE INFRASTRUCTURE CONCEPTS

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ABSTRACT

This paper identifies and discusses several types of manned Mars surface missions, including sorties, fixed-base, and hybrid missions, which can be envisioned as potentially desirable approaches to the exploration and utilization of Mars. Some of the advantages and disadvantages of each type are discussed briefly. Also, some of the implications of the types of missions on the surface elements' designs are discussed briefly. Typical sets of surface elements are identified for each type of mission, and weights are provided for each element and set.

INTRODUCTION

The types of surface infrastructure elements which are needed are heavily dependent on the nature, duration, and timeframe of the mission. For manned Mars flyby missions or manned Mars orbiter missions, no habitable surface elements would be necessary, but unmanned probes and/or robotic surface explorer vehicles would no doubt be required.

For manned landings (on Phobos, Deimos, or Mars), the types of required infrastructure elements can vary significantly with several factors. One is the timeframe of the mission. For early missions, there is likely to be less emphasis on "permanent" types of infrastructure elements and more emphasis on the elements which are "bare essentials" for landing men and returning them safely. Technology levels will be lower on early missions, and hence equipment on early missions will be less efficient than that on later missions. Hence, weight, volume, power, and other resources will be more critical, which will allow less infrastructure equipment to be taken per flight than on later missions. The only practical types of manned Mars landing missions are those which can be done during favorable planetary alignment periods. The favorable alignments (reference 6) are of either the conjunction or opposition type, and occur about every 2 years. The conjunction-type opportunities require about a 1-year stopover time at Mars, and the opposition-type missions require about a 60-day stopover. The energy requirements for longer or shorter stopover times increase severely for even a few days' change from the optimum times stated. The initial manned Mars landing mission may be of the type having a 60-day stopover, to minimize cost, risk, and complexity of the mission. The opposition-type missions usually have a Space Vehicle (SV) weight penalty compared to the conjunction-type missions, but this is not too great for all-aerobraking concepts.

SURFACE MISSION OPTIONS

There are at least three types of surface exploration/utilization options which are possible (Figure 1): (1) sortie; (2) moving-base; and (3) fixed-base options. In the sortie approach, each mission is directed to a different landing site, with short-distance, limited-round-trip surface traverses being made in that general vicinity for exploration and science investigation purposes. In the fixed-base mode, successive missions are directed to the same site, with fairly extensive round-trip surface traverses being made from the base. The moving-base mode is a hybrid of the other two modes, wherein two or more missions may be directed to one location, then the entire base is moved to another location, etc.

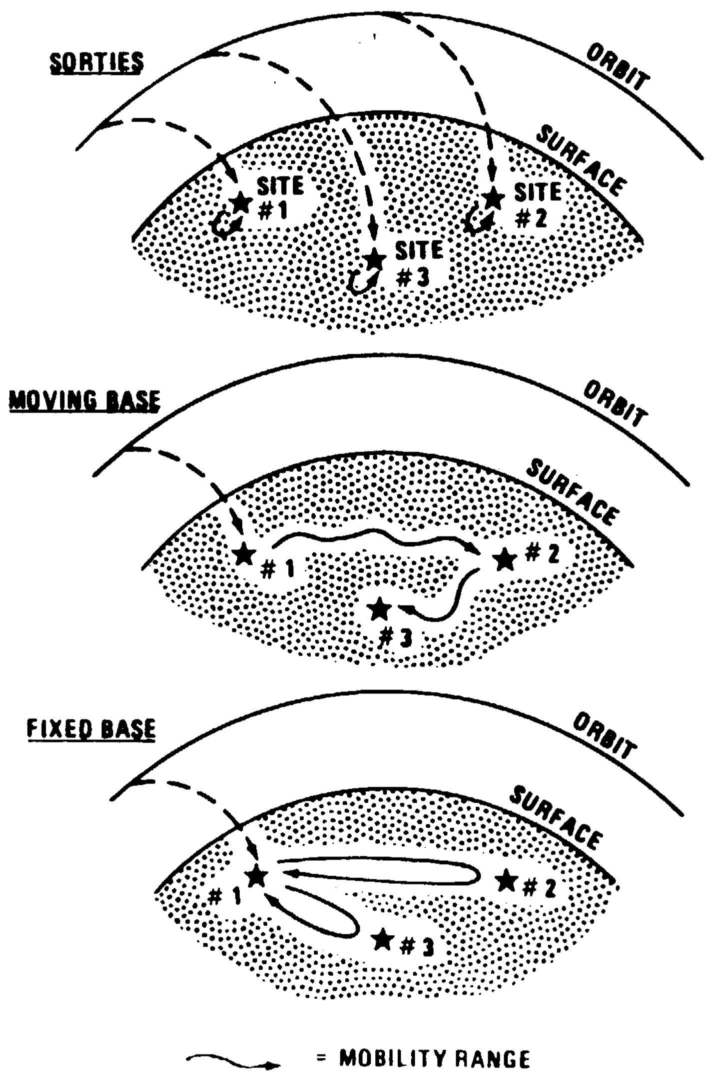
The sortie approach provides flexibility for exploration of surface areas having widely different terrain, climate, etc., on different missions, since widely separated landing locations can be chosen each time. Sortie missions would be more limited in scope and duration than the other missions, since each mission must furnish all its own equipment and resources (no carry-overs from previous missions). The variety of surface features which can be explored during each sortie mission is limited by the landing location, the range and capability of the surface traverse vehicle, and the duration of the mission. The mission complexity of sortie-type missions is probably lower than the others (especially if the scope and duration of the mission is more limited), and the equipment complement is smaller.

The fixed-base concept provides the least variety of surface features across missions (unless surface traverse distances can be extremely great). It does offer significant advantages, however, in the buildup and re-use of equipment from mission-to-mission. It would be

4776-86

FIGURE 1. MARS SURFACE OPTIONS & COMPARISONS

MARS SURFACE OPTIONS



SURFACE OPTION CONSIDERATIONS

- MISSION COMPLEXITY
- SCIENCE VARIETY & VALUE
  - PER MISSION
  - ACROSS MISSIONS
- SAFETY
- RE-USE OF SURFACE EQUIPMENT
- SUPPORT OF MFG./CONSTR.
- COST

necessary to use this mode for any long-term construction or manufacturing activity.

The moving-base mode lies somewhere between the other two modes in almost every respect, having some of the advantages and disadvantages of both.

In actual practice, the missions may shift from one option to another, and occasionally back again. For example, the earliest missions will probably be of the sortie variety, with later missions trending towards the moving-base or fixed-base variety. It is likely, however, that an occasional sortie mission to a different location might be desired, even after a fixed-base was established at one location. This might be desirable for science/exploration reasons, or to begin establishment of another base.

Many factors will help determine the selection of the surface options to be used. The total number of missions in the program and the flight frequency will have a significant bearing on this. The availability of systems and resources (e.g., flying vehicles and in-situ-manufactured propellants) to allow rapid and easy movement of equipment over great distances would strongly influence selection of surface options. For cases where there is a gap between successive habitation or use of surface equipment previously landed, the advantage of buildup and reuse of such equipment must be traded against the possibility that the equipment might have become damaged or otherwise become inoperable during the interim period.

SURFACE INFRASTRUCTURE ELEMENTS

Design of the surface infrastructure elements must be closely coupled to design of the other elements of the SV (e.g., Mission Module) in some cases. This is strongly dependent on Mars surface stopover duration. For example, on a mission which only has a 60-day stopover, if the lander (e.g., Mars Excursion Module (MEM)) equipment were designed independently from the orbiter (e.g., MM equipment), the lander would only operate for 60 days out of a total mission time of 2 years. It would be a much better use of the lander systems to utilize them for a greater part of the 2-year mission. If, however, the mission were one having a 1-year stopover, there might be more concern about the lifetime of the lander systems if they were operated for the full mission duration.

The division of the crew between Mars surface and Mars orbit operations will be a factor in design of the lander. On an early sortie mission of the 60-day-stopover variety, half the crew may be sent to the surface in the lander and the other half may stay in the orbiter. On a 1-year-stopover mission, the entire crew may be sent to the surface. Obviously, the split of the crew accommodations equipment between lander and orbiter would vary significantly between these two types of missions.

An artist's concept of a Mars base is shown in Figure 2. Some of the infrastructure elements shown here (greenhouses, Habitability Modules, etc.) are more applicable to the fixed-base surface option, but other equipment (rover, lander/departure stages, etc.), are applicable to any of the surface options. More discussion is provided on this subject in later paragraphs. Several of the infrastructure elements are depicted with the large-diameter aerobraking shells still attached, but these shells could be removed if necessary. It might be desirable to remove these large structures for potential use as living quarters, storage shelters, etc. An artist's concept of living quarters made from such structures is shown in Figure 3.

Table 1 identifies a set of typical surface elements for each type of surface option. As shown, the sortie concept would be the most simplistic of the three, the fixed-base concept would be the most complex, and the moving base concept would lie somewhere between the other two concepts in terms of the amount and complexity of equipment required. Where items have checkmarks enclosed in parentheses, an early version of the item would probably be needed or desired as an element of that type of surface option.

The lander/departure element would be the MEM, or a growth version of it. A number of different concepts of the MEM have been defined in past studies, including Apollo Command Module derivatives, biconic vehicles, etc. Data for some of these are shown, along with the MEM defined in this study, in references 1, 7, and 8. In the fixed-base mode of operation (and possibly the moving-base mode), the spent MEM descent stages could be used as storage areas, or could possibly be joined together to serve as a habitability volume.

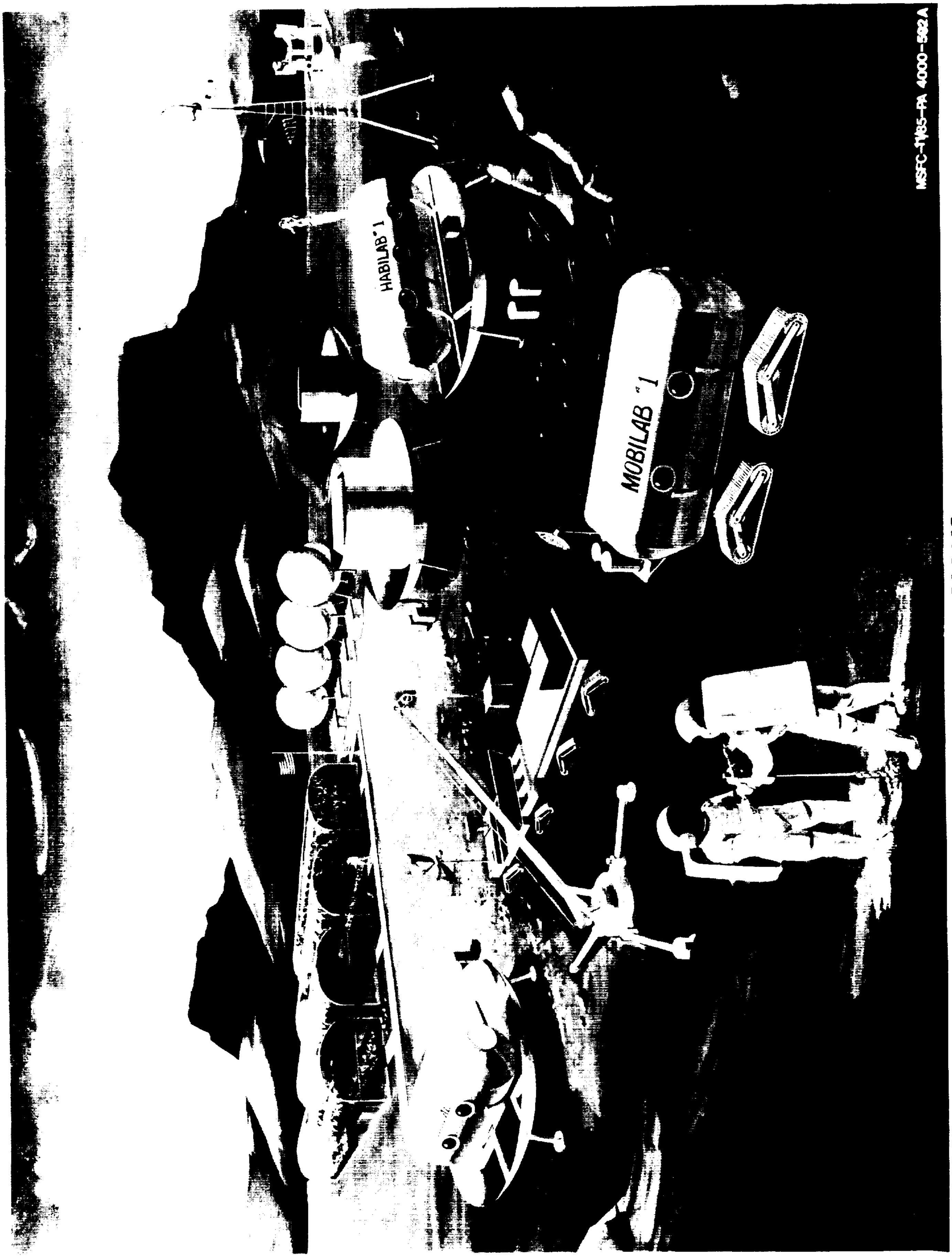


FIGURE 2. MARS BASE

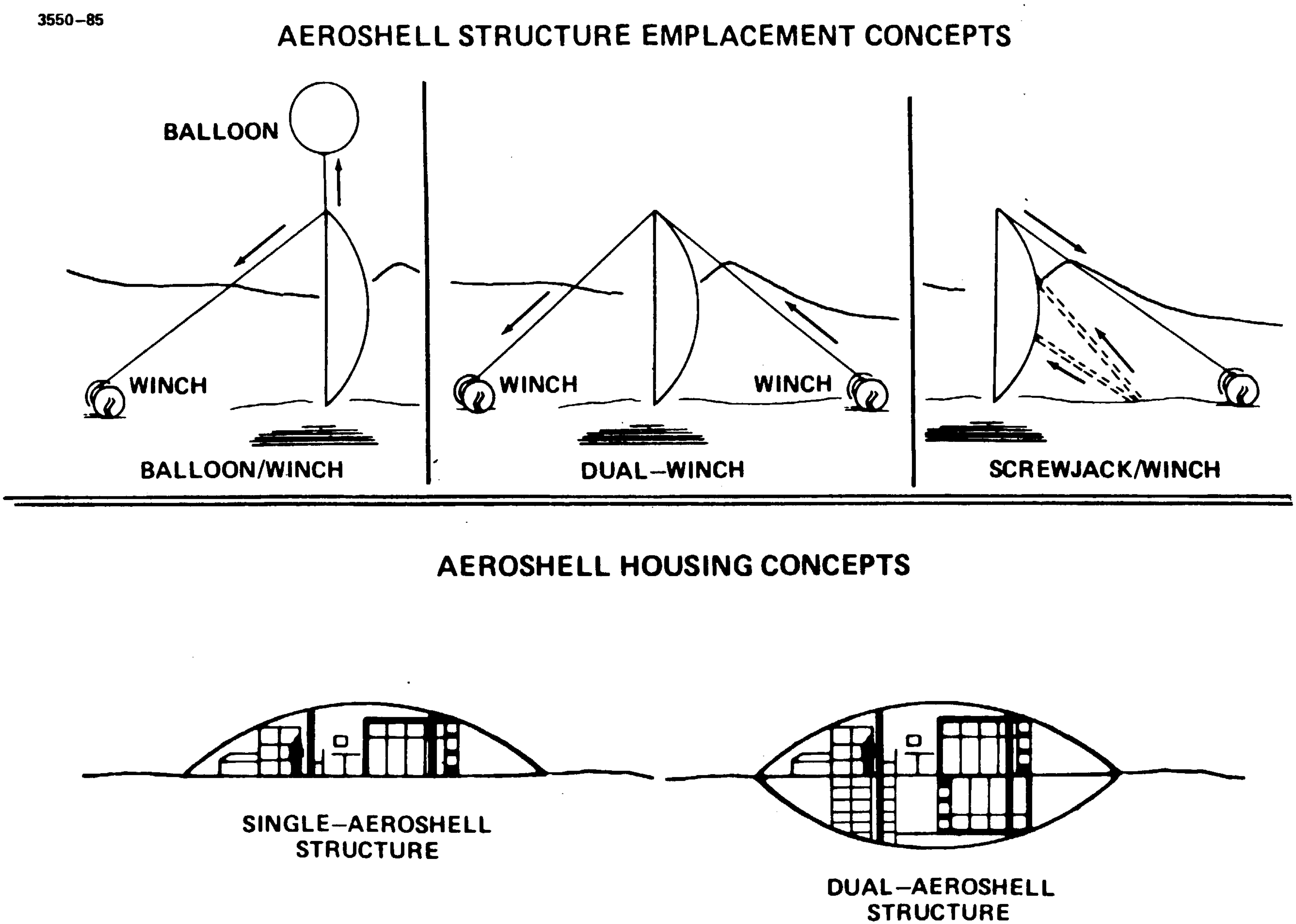


FIGURE 3. AEROSHELL UTILIZATION CONCEPTS

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TABLE 1. SURFACE COMPLEMENTS

TYPICAL SURFACE ELEMENTS	SORTIE	MOVING–BASE	FIXED–BASE
• LANDER/DEPARTURE	√	√	√
• HAB FACILITY		(√)	√
• LAB FACILITY		(√)	√
• POWER FACILITY			√
• GREENHOUSE			√
• IRPU'S		(√)	√
• ROVER			
– SMALL	√		
– LARGE		√	√
• "POGO" VEHICLE	?	√	√
• AIRPLANE			?
• DRILLS/MINING	(√)	(√)	√
• CONSTRUCTION		(√)	√
• SOIL MOVER		(√)	√
• CRANE	(√)	(√)	√
• PORTABLE SCIENCE	√	√	√
• COMMUN. RELAY	√	√	√

(√) LIMITED OR EARLY VERSIONS

The early habitability and laboratory facilities might be modules derived from Space Station (SS) modules, but later ones may be made from other elements such as the large aerobraking shells as previously noted (Figure 3) or from in-situ-produced materials ("concrete", etc.) The power facility item might be a nuclear reactor or nuclear isotope power generator; other possibilities would include fuel cells operated from in-situ-produced reactants and some sort of solar-energy system. Reference 2 describes some of these options in more detail.

The greenhouse would be an element only of the fixed-base surface option. A definition of it is provided in reference 3. For the greenhouse, an inflatable plastic structure on a pad could be used. The structure would be optically transparent with a UV filter. It would be pressurized and would require a night-time cover. Due to the thin atmosphere, no support structure would be required, even during high winds.

The In-situ Resource Production Units (IRPU's) are elements which would produce such products as propellants, breathable gases, fuel cell reactants, or water. Typical units have been defined in references 2 and 3. The small rover is an upgraded version of the MSFC-developed Lunar Roving Vehicle (LRV) which was used on several Apollo missions. It is discussed in reference 4. This vehicle requires the passengers to wear space suits, and it has a limited traverse range and cargo capacity.

The large rover is essentially a small Hab/Lab Module on a tracked undercarriage. It has a traverse capability on the order of 100 km and 30 days, and is piloted by the crew from within the module. The Molab was a vehicle of this sort, which is discussed in reference 4. The "pogo" vehicles are propulsively-powered vehicles which can vary in size from a 1-man backpack to a platform capable of transporting modules or other large elements. These are discussed in reference 4. These elements have the advantages of being insensitive to obstacle size during traverses, require no horizontal takeoff and landing strip, and can traverse great distances in a short time. They will require a large amount of propellant, however, and are thus more practical if a local source of propellant can be utilized.

The airplane is a remotely-piloted vehicle which will contain science equipment and will be used to explore regions which would be difficult or impossible for man to explore directly. One disadvantage it has is the requirement for a takeoff and landing strip. The airplane is discussed in references 4 and 9.

The "drills/mining" equipment item listed in Table 1 is intended to include only the larger size equipment of this nature. Smaller drills are included under the "portable science" item. The larger equipment would be used for taking deep core samples, for implanting deep seismic charges and sensors, etc. The mining equipment would be used for digging tunnels, for extracting minerals, etc.

The construction item includes equipment necessary to manufacture building materials as well as equipment needed for erecting or emplacing structures. A soil-mover of some sort will be needed for the fixed-base missions, to support habitability element emplacement, construction activities, road-building, trench digging and filling, etc. A limited-capability version would be desired on moving-base missions. Types of equipment which have been suggested in past studies for this category are draglines, road-graders, backhoes, etc. A crane could be used to lift and emplace any of the larger elements (Hab Modules, etc.) delivered to the surface and would be used in the construction activity, as required. The crane would be used on fixed-base missions, with smaller versions used on other missions.

The portable science equipment includes a myriad of small items of equipment which might be carried in the small rover vehicle or used in the vicinity of the lander, to gather and analyze geological samples, to make weather or environment measurements, etc.

The communications relay is not really a surface element, but is an element which may be required in orbit to support the surface activities. In manned missions to the surface, some elements will be left in Mars orbit for the return trip to Earth. These elements will have communications equipment built in, and can serve as the communications relay for the surface activities when needed. Unmanned missions may or may not have such equipment left in Mars orbit, and so may require that a separate element be provided.

WEIGHTS

In order to estimate weights for the total complements of equipment for the various surface options, assumptions were necessary in a few key areas, and are listed below:

(1) Sortie: 3 men/60 days surface stay/10 kw elect. power

(2) Moving-base: 6 men/1 year surface stay/25 kw elect. power

(3) Fixed-base: 12 men/1 year surface stay/100-200 kw elect. power

Table 2 provides weight data for some of the key elements previously discussed. The top part of the table summarizes the portable science equipment items, some of which might be taken along on surface traverses. The bottom part of this table lists weights for miscellaneous larger elements.

Table 3 provides a weight summary of the equipment necessary to be delivered to the surface of Mars for each of the three surface mission options. Reference 5 uses these weights as requirements for delivery of equipment to the Martian surface, and shows the rates of buildup of cumulative delivered weight to the Martian surface and to LEO as a function of time, for various SV options.

TABLE 2. EQUIPMENT WEIGHTS

PORTABLE SCIENCE EQUIPMENT	WT (LBS)
GRAVIMETER	26
X–RAY DIF/X–RAY FL	99
ELECTRON MICROSCOPE	115
GAS CHROMATOGRAPH	33
SPECTROMETER	33
MAGNETOMETER	64
SMALL DRILL	81
CENTRIFUGE	32
POLARIMETER	13
pH METER & REAGENTS	32
REFRACTOMETER	9
THERMOMETERS	2
SCALES	12
REFRIGERATOR	27
INCUBATOR	21
OVEN/STERILIZER	42
WORK BENCH	55
MICROMANIPULATOR	19
ULTRASONIC CLEANER & SOLVENTS	106
AGITATORS & BLENDORS	8
HAND TOOLS	22
SAMPLE HOLDERS & CONTAINERS	49
ANEMOMETER	15
EXPLOSIVES	344
MICROTOME	19
RTG POWER SUPPLY	87
BAROMETER	15
SEISMOMETER	8
SOIL BEARING STRENGTH	25
POWDERING, DISSOLUTION, OPTICAL ANAL	70
EM PROPERTIES	52
THERMAL PROPERTIES OF SOIL	24
IONOSPHERE STRUCTURE PROPERTIES	88
SOIL SAMPLE BOX	17
TOTAL	1664

MISCELLANEOUS ITEMS	
LARGE DRILL	910
ROVER	600
AIRPLANE	660
MOLAB	3400
CRANE	450
EARTH MOVER	450

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TABLE 3. SUMMARY WEIGHTS OF SURFACE MISSION OPTIONS

TYPICAL SURFACE ELEMENTS	TYPICAL WEIGHTS REQUIRED (LBS) SORTIE	MOVING–BASE	FIXED–BASE
LANDER/DEPARTURE	128000	200000	256000
HAB FACILITY	--	112000	162000
LAB FACILITY	--	50000	50000
POWER FACILITY	--	8730	350000
IN SITU RES PROD UNIT	--	750	2250
PORTABLE SCIENCE	832	1664	1664
TRANSPORTATION ELEMENTS ROVER			
– SMALL	600	600	600
– LARGE	--	--	3400
AIRPLANE JETPACK DERIVATIVE	--	660	660
LARGE DRILL	910	910	910
SOIL MOVERS	--	--	900
GREENHOUSE	--	--	500
TOTALS	130342	375314	828884

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MANNED MARS MISSION SURFACE TRANSPORTATION ELEMENTS

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ABSTRACT

The necessity and advantage of surface transportation was well demonstrated by the Apollo 15, 16, and 17 missions. Baseline surface transportation elements for further studies are Lunar Rover, Elastic Loop Mobility System, Mobile Laboratory, Airplane, and Rocket Powered Flying Vehicles.

INTRODUCTION

Metabolic expenditures required for walking and working are predicted to be nearly the same on Mars as Apollo missions were on the lunar surface. The supporting evidence for this is that most of the effort for movement was exerted in simply overcoming the suit resistance. The difference in gravity (Mars vs. Moon) will be equalled by the less resistive suits being developed. For the lunar surface, normal walking required an average expenditure of 950 BTUs per hour. Fast walking required 1400-1500 BTUs per hour. These rates increased when coupled with even slight hill climbing or obstacle negotiation. A more desirable expenditure would be approximately 550 BTUs per hour. The desire for lower metabolic rates and additional speed, range, and science equipment for data gathering indicate the need for surface transportation.

DISCUSSION

Surface Rovers

The most developed form of surface transportation is the surface rover. The advantage of surface rovers was well demonstrated by the Apollo 15, 16, and 17 missions. Two classes of surface rovers are discussed.

Small Rovers

The small two-man type rover would be applicable to all types of missions (Sortie, Mobile-base, Fixed-base). A candidate small rover is obviously the MSFC-developed wheeled Lunar Roving Vehicle[1] (LRV). See Figure 1. The LRV specifications are: (1) 1014 kilograms (460 pounds), (2) 2381 kilograms (1080 pounds carrying capacity), (3) 78 hours life-

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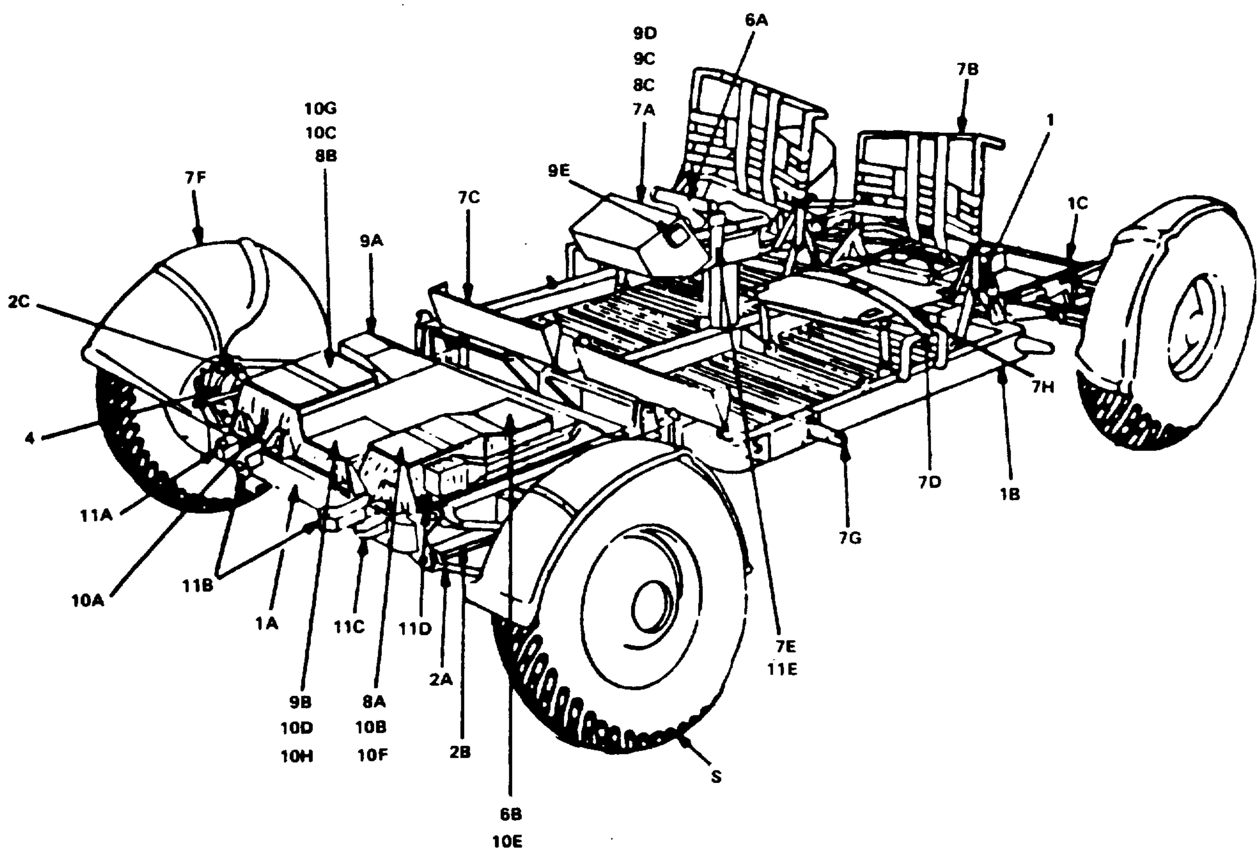

1 CHASSIS

A. FORWARD CHASSIS
B. CENTER CHASSIS
C. AFT CHASSIS

2 SUSPENSION SYSTEM

A. SUSPENSION ARMS (UPPER AND LOWER)
B. TORSION BARS (UPPER AND LOWER)
C. DAMPER

3 STEERING SYSTEM (FORWARD AND AFT)

4 TRACTION DRIVE

5 WHEEL

6 DRIVE CONTROL

A. HAND CONTROLLER
B. DRIVE CONTROL ELECTRONICS (DCE)

7 CREW STATION

A. CONTROL AND DISPLAY CONSOLE
B. SEAT
C. FOOTREST
D. OUTBOARD HANDHOLD
E. INBOARD HANDHOLD
F. FENDER
G. TOEHOLD
H. SEAT BELT

8 POWER SYSTEM

A. BATTERY #1
B. BATTERY #2
C. INSTRUMENTATION

9 NAVIGATION

A. DIRECTIONAL GYRO UNIT (DGU)
B. SIGNAL PROCESSING UNIT (SPU)
C. INTEGRATED POSITION INDICATOR (IPI)
D. SUN SHADOW DEVICE
E. VEHICLE ATTITUDE INDICATOR

10 THERMAL CONTROL

A. INSULATION BLANKET
B. BATTERY NO. 1 DUST COVER
C. BATTERY NO. 2 DUST COVER
D. SPU DUST COVER
E. DCE THERMAL CONTROL UNIT
F. BATTERY NO. 1 RADIATOR
G. BATTERY NO. 2 RADIATOR
H. SPU THERMAL CONTROL UNIT

11 PAYLOAD INTERFACE

A. TV CAMERA RECEPTACLE
B. LCRU RECEPTACLE
C. HIGH GAIN ANTENNA RECEPTAC
D. AUXILIARY CONNECTOR
E. LOW GAIN ANTENNA RECEPTAC

LRV WITHOUT STOWED PAYLOAD
FIGURE 1

time, (4) 92 Km total range, (5) Two 36-volt silver-zinc batteries, (6) Obstacle negotiation: (a) 30 centimeters (one-foot) high from standing start with both front wheels in contact, (b) 71 centimeters (28-inch) crevasse, and (c) 25 degree slope.

The obstacle-negotiation limits are prohibitive, especially for surfaces similar to the Viking I and II landing sites. A redesign using the LRV as a baseline would be prudent. Changes would need to include wheel size and power requirements.

Another candidate for the small rover is the MSFC-developed Elastic Loop Mobility System[2] (ELMS), a tracked vehicle without the conventional "tracks" shortcomings of high internal losses, mechanical complexity, and heavy weight. See Figure 2. The advantages over wheeled vehicles are: (1) High static stability through low c.g. location, (2) Better traction in soft soil which results in better slope climbing capability, (3) Reduced drive torque requirements for obstacle negotiation, (4) Simpler stowage and deployment concept, and (5) Smoother ride characteristics due to large footprint.

ELMS obstacle negotiation: (1) 30 degree slope, (2) 46 centimeters (>18 inch) step obstacle, and (3) 102 centimeters (40 inch) crevasse.

Further development is desirable for manned expeditions with surface conditions similar to the Viking I and II landing sites.

Large Mobile Laboratories

The mobile laboratory[3] (MOLAB), whether two-man or three-man, would be applicable to the fixed-base mission. The MOLAB should be capable of traversing a relatively smooth surface. The small rover would be used to gather specimens and data from the less friendly regions. The MOLAB should also be capable of maintaining astronaut life support and science equipment, including a mini-laboratory, for 30 days with a range of 100 Km.

Atmospheric Rovers

Greater range is a desirable for exploration of the Martian surface. Range extension can be achieved by taking advantage of the atmosphere and low gravity. A probable requirement for an atmospheric vehicle would be the vertical take-off and landing (VTOL) capability (this requirement could perhaps be eliminated for the fixed-base mission).

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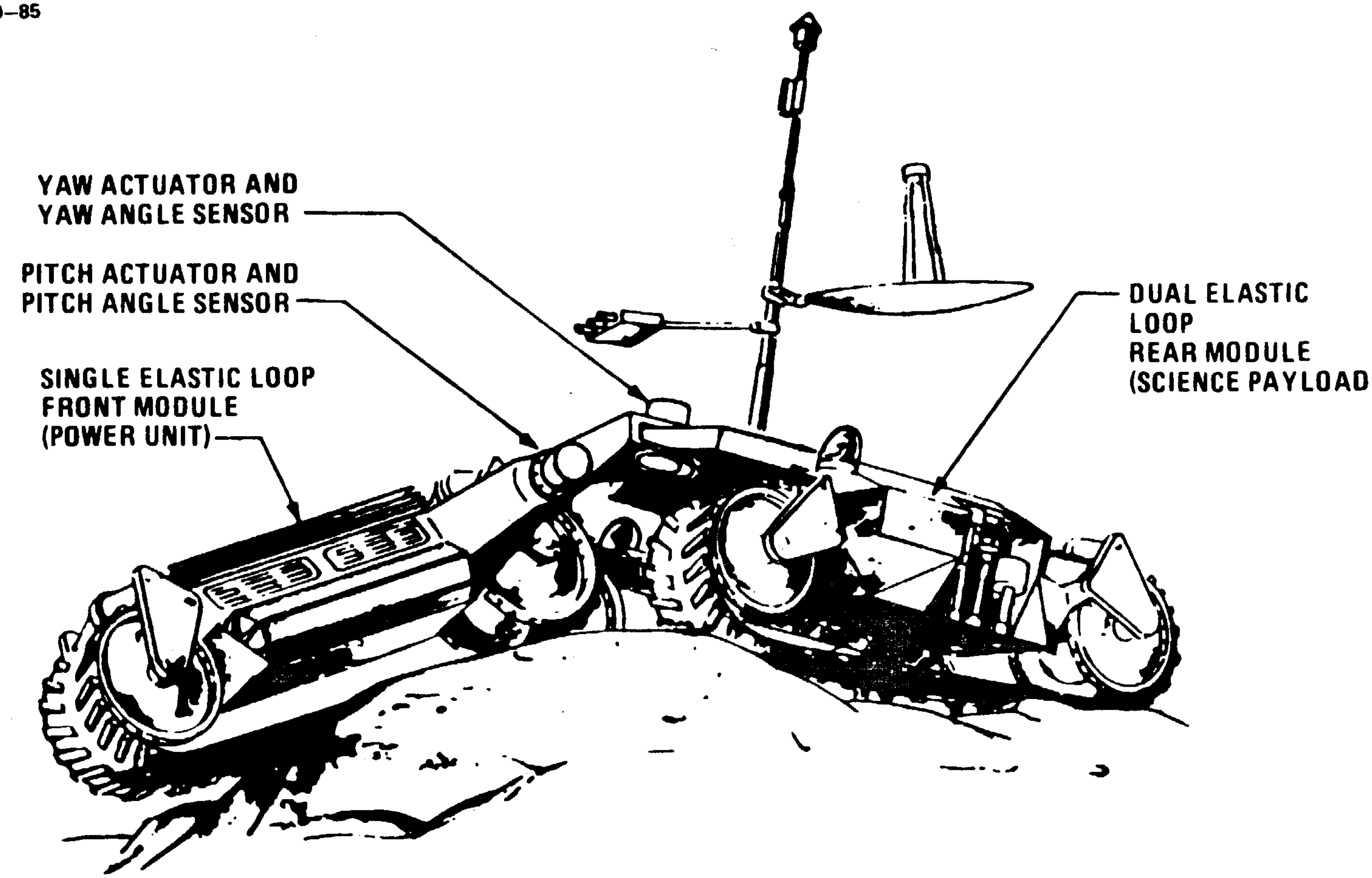

FIGURE 2
ELASTIC–LOOP MOBILITY SYSTEM (ELMS)

Helicoptor

A Martian helicopter was investigated and deemed inappropriate due to basic aerodynamic lift requirements and thin Martian atmosphere.

Airplane

A baseline has been established in the JPL design[4]. Some changes to be considered are: (1) Manned operability, (2) Load carrying weight, and (3) VTOL capability.

Preliminary missions to determine atmospheric conditions at various altitudes would be required.

Rocket Powered Flying Vehicles

Rocket powered flying vehicles offer some advantages over surface vehicles since they do not have to contend with many of the obstacles on the rugged martian surface. This type of vehicle has many applications and can range in size from one-man platforms to mobile bases.

A one-man vehicle similar to a one-man flying vehicle shown in Figure 3 could aid in increasing the mobility of the astronauts in the vicinity of the Mars base. This vehicle is propelled by two side mounted rockets and is controlled manually by the pilot. The graph in Figure 3 shows that this type of vehicle would have a payload of several hundred pounds and a range of 1 to 7 kilometers.

A larger rocket powered flying vehicle could be designed to carry two astronauts over greater distances. Such a vehicle could be patterned after the Apollo Lunar Ascent Module[6]. It would have a dry weight of about 11,000 kilograms (5,000 pounds) and a gross weight on the order of 22,000 kilograms (10,000 pounds). This type of vehicle would have a round trip range of 20 to 100 kilometers as shown in Figure 4.

A final option for rocket powered flying vehicles would be to provide mobile bases on the surface of Mars. These vehicles would be fairly large, with a dry weight of about 88,100 kilograms (40,000 pounds) and a one way range on the order of 500 to 800 kilometers as shown in Figure 5. This type of vehicle would require large amounts of propellants and would have a gross weight near 220,400 kilograms (100,000 pounds). As manned presence on Mars increases and propellant is manufactured on Mars this option may prove beneficial.

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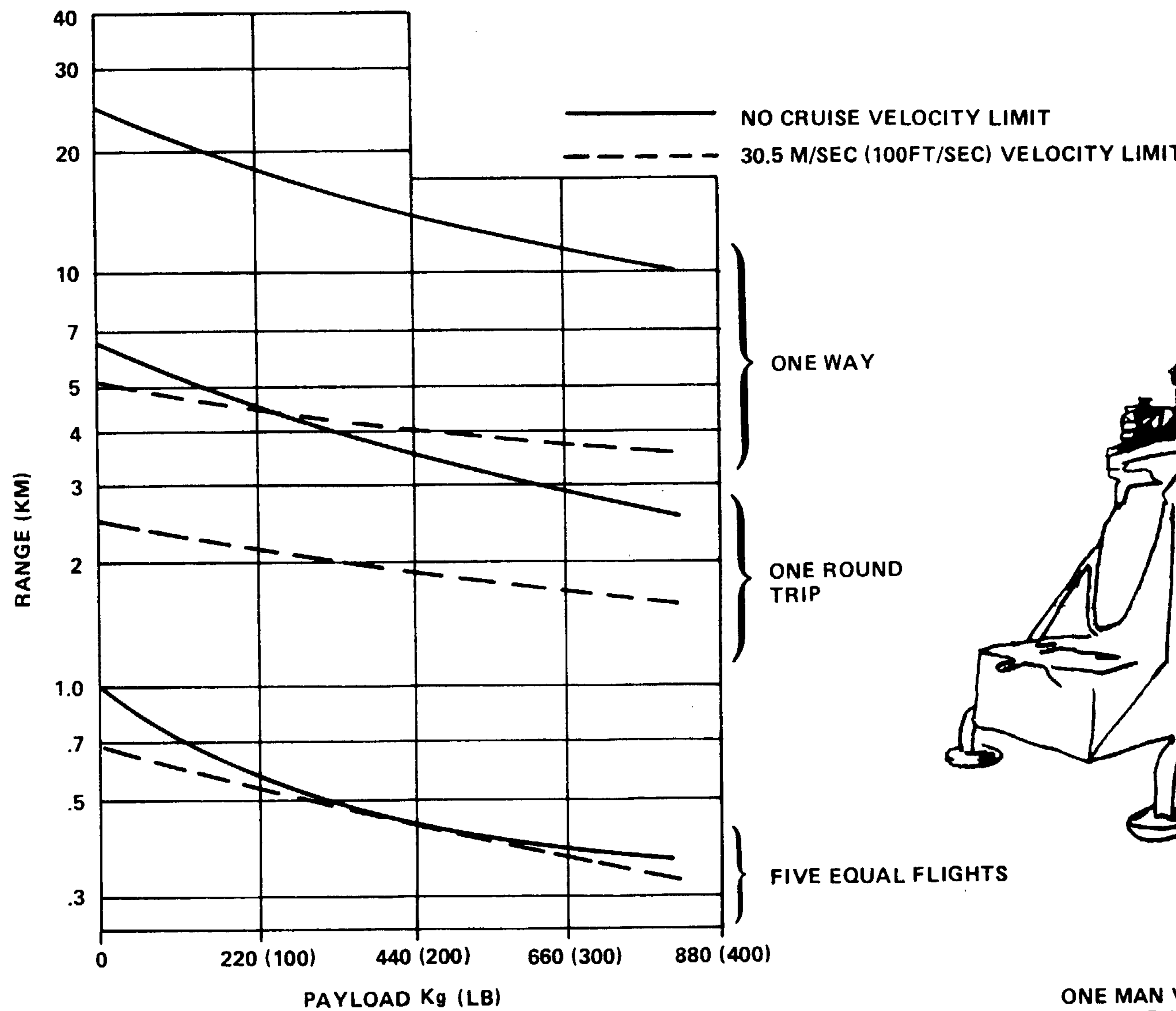
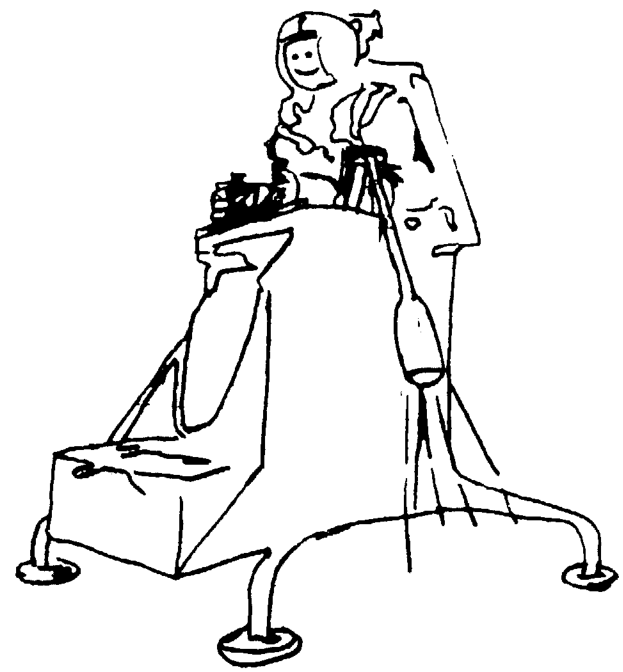
ROCKET POWERED FLYING VEHICLE PERFORMANCE





ONE MAN VEHICLE
VEHICLE DRY WGT = 650 Kg (294 LB)
PROPELLANT USED =590 Kg (270 LB)

FIGURE 3

5136–85

VEHICLE GROSS WEIGHT AS A FUNCTION OF VEHICLE DRY WEIGHT FOR A ROCKET POWERED MARS FLYING VEHICLE

- I_{SP} = 300 SEC
- ASSUME BALLISTIC TRAJECTORIES
- ROUND TRIP

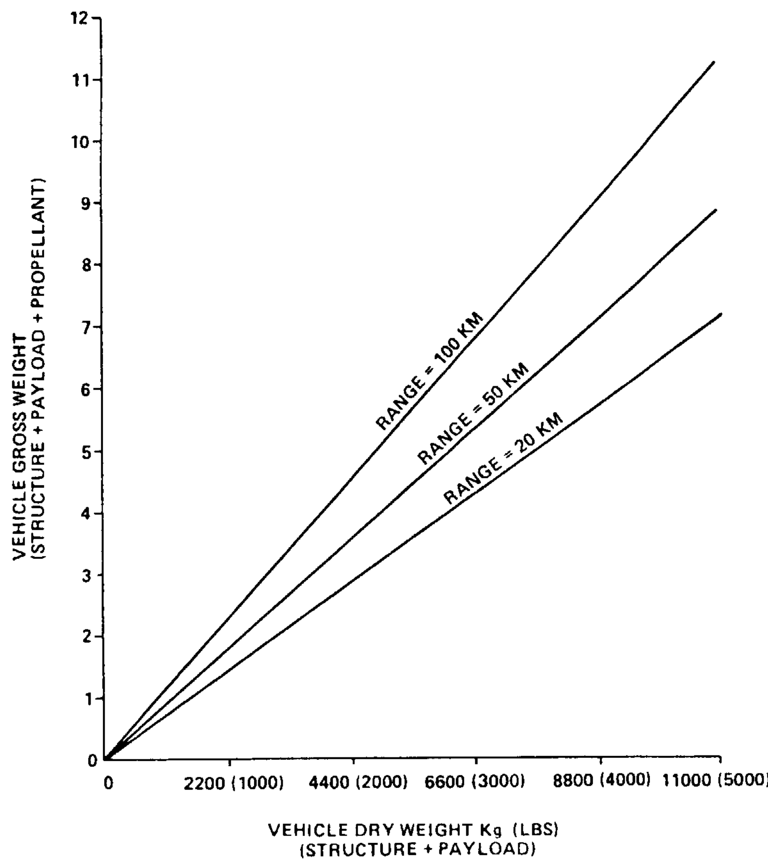


FIGURE 4

5135–85

VEHICLE GROSS WEIGHT AS A FUNCTION OF VEHICLE DRY WEIGHT FOR A ROCKET POWERED MARS FLYING VEHICLE

- I_{SP} = 300 SEC
- ASSUME BALLISTIC TRAJECTORY
- ONE WAY TRIP

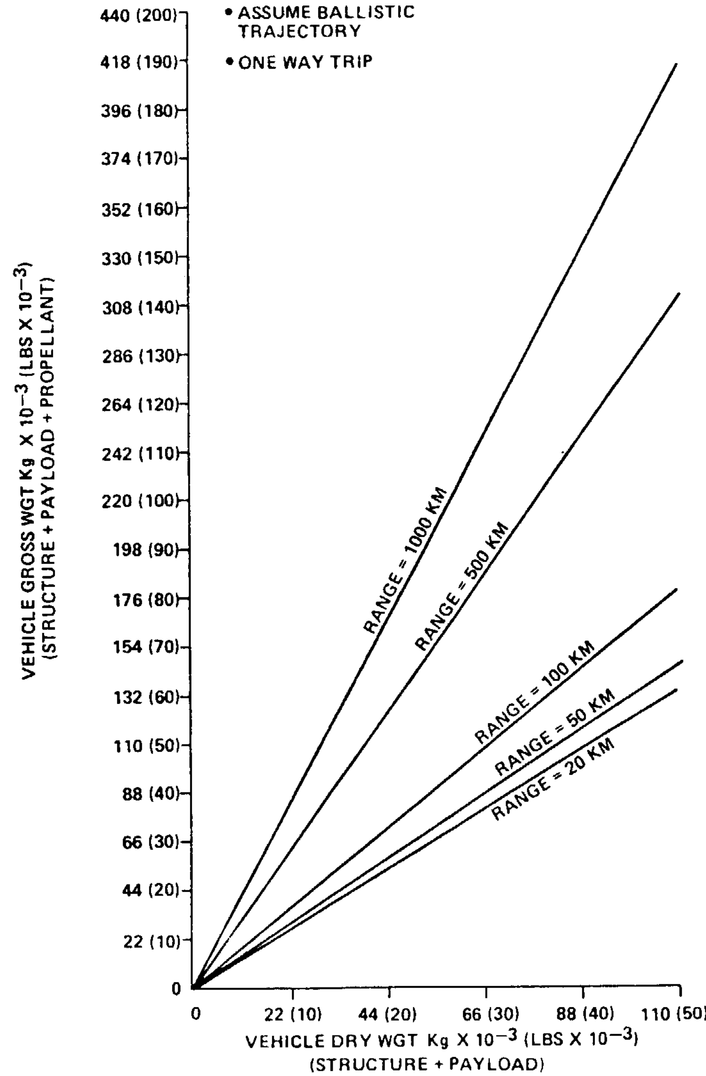


FIGURE 5

CONCLUSION

Starting points for further in-depth studies of surface transportation elements have been identified. For ground rovers, tracked vehicles of the ELMS nature look promising. For atmospheric rovers, Rocket Powered Vehicles with VTOL capabilities could prove quite beneficial.

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N87-17757

ROVER TECHNOLOGY FOR MANNED MARS MISSIONS

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ABSTRACT

A set of Roving vehicle design requirements were postulated by JSC, corresponding to an idealized Mars transport vehicle operational scenario which could serve as a reference for a manned Mars mission. The ability of conventional vehicles to satisfy these requirements were examined. The study indicated that no conventional vehicle could satisfy all of the requirements, as the vehicles are presently configured. Consequently, the requirements have to either be relaxed (as will be proposed in a section of this report) and/or an alternative, less conventional vehicle design will have to be developed. A possible unconventional vehicle design which has received considerable attention for DARPA and the Army is the walker vehicle. The design issues associated with this vehicle will be presented in this paper, along with a comparison of the performance capabilities of this technology vs. conventional vehicle technology.

INTRODUCTION

In the last year the U.S., Japan, and European nations have committed hundreds of millions of dollars to developing computers that can "think" more like humans, moving and acting independently according to what their electronic senses tell them. For now, these mobile thinking manned transport vehicles will have to serve the planetary mission designers on wheels or tracks, and depend on human operators for major decisions. However, DARPA is currently funding work at Ohio State University on a six-legged robot which is aimed at achieving mobility closer to that of humans and animals than to conventional vehicles. This will allow manned vehicles to venture into cluttered environments, steep slopes, and areas accessible to animals or humans but not to wheeled vehicles.

In recognition of the above circumstances, this paper is devoted to a summary of the design comparisons of legged versus traditional mobility systems for manned transport on Mars.

APPROACH

A number of Rover vehicle point design configurations have been proposed over the years which appeared to have the potential for providing Mars surface operations of high science yield. However, the analytical tools did not exist for comparing these designs. Thus, it was impossible to select an optimal vehicle configuration for the mission options of interest. To eliminate this difficulty, an attempt has been made to generate some preliminary rover vehicle requirements, for comparison with a compilation of the capabilities of existing rover vehicle point designs. This information was then used to eliminate all but the most promising rover vehicle design concepts. For the remaining vehicle candidates, a comparison was made of their predicted performance capabilities. Each of these issues will be addressed in more detail below.

VEHICLE PERFORMANCE REQUIREMENTS AS DEFINED BY THE JOHNSON SPACE CENTER

Table 1 outlines the mobility requirements for a manned Mars rover vehicle capable of performing a site traversal on the Mars surface. The following traverses were selected as the basis for the definition of these requirements: a traverse for a Mars operational scenario which is equivalent to an idealized Lunar Appollo 15 scenario, the traverses planned for the Candor Chasma region of Mars, and the Viking Lander 1 and 2 geologic sites.

A survey was conducted to identify the performance characteristics of all existing rover vehicle point designs documented in the current literature. These vehicle performance characteristics were compared against the Mars rover vehicle requirements, as presented in Table 2 (Refs. 1-14). Based upon this comparison, only three vehicles appeared as candidates for mars surface operations: (1) a six-wheel rover (ex. Lunar Rover Vehicle (LRV), (2) an ELMS (Lockheed Loopwheel Vehicle), and (3) a walker.

CONVENTIONAL VEHICLE PERFORMANCE COMPARISON SUMMARY

A performance comparison of walker technology versus alternative concepts will be deferred until the following section. Empirical data on component performance characteristics is required as input into analytical models describing the performance of the wheel and loopwheel vehicles. Thus, comparisons of vehicle performance could only be found for the point design concepts identified above. A discussion of the perfor-

TABLE 1

MOBILITY CHARACTERISTICS

CRITERIA	REQUIREMENT
1. Maximum slope capability (Affects: wheels, drive, wheelbase, tread)	45 deg, soft soil
2. Ground clearance (Affects: suspension, wheels, wheelbase, tread)	(A) Straddle a 35 deg-wedge formed by two intersecting crater walls (B) Undercarriage clearance 16 in. (approx) (Within central compartment area)
3. Maneuverability (Affects: wheels, suspension, steering, tread, wheelbase)	(A) Turning radius 10-15 ft (approximately) (B) Front and rear steering (C) Reverse drive
4. Stability (Affects: wheel suspension, tread, wheelbase)	Approximately 40-50 deg for traversing crater walls of soft soil and providing for some wheel sinkage
5. Obstacle capability (Affects: wheel, suspension, wheelbase, tread)	3 ft (approx)
6. Crevasse capability	2-3 ft (approx) (Not critical)
7. Roving route capability (Drag, torque, power)	
(A) General slopes	5 deg (approx) continuous over a considerable route length
	(B) Local slopes 20% of route assumed to be 30-deg crater walls

TABLE 2
COMPARISON OF ROVER PERFORMANCE

Rover	Ref.	Type	Number of Wheels	Max Slope Capability deg	Ground Clearance m $(x10^{-2})$	Turning Radius m $(x10^{-2})$	Stability Pitch deg	Stability Roll deg	Obstacle Capability m $(x10^{-2})$	Crevasse Capability m $(x10^{-2})$	Mass kg	Range m $(x10^{3})$	Velocity m/hr	Remarks
LRC A	1	Wheel	4	7.5	12.7	90			5	18	20-35	.050		
LRC B	1	Wheel	4	7.5	12.7	90			5	18	35-60	.020		
LRC C	1	Wheel	4	7.5	12.7	90			5	18	60-90	1.0		
RPI	2	Wheel	4	40.0	73.7	456		25	14	76				90% of Wheel Diameter
McD	3	Wheel	4-2	24.0	43.0	290			50					
GM (JPL)	4	Wheel	6	20°	24.0	120	45	45	53	60				
Lunokhod	5	Wheel	8											
MM (LRV)	6	Wheel	4	25	12.0		45	45	10 (μ=0.6)	24 (μ=0.6)	66.8	0.05-0.1	56	
JPL - Lockheed	7	Loop-Wheel	4	32.0 (on grass)	125.0				56		290			Scuff Steering
Lockheed	8	Loop-Wheel	3	36.5		1.4 Lv			85% of ELMS-24	90% of ELMS-25			0.5-2	
Mobil Lander	9	Loop-Wheel	2-1	36.0*	28				29-39 (μ=0.6)		675			
MBB				20.0	25				8.1 (μ=0.6)		180			
Dual LRV	10	Wheel	6	35.0*			57	45	100	100	449	1000		
MM74	11	Wheel	4	5.0*	21	Scuff	40	60	5.7	31	127	10-50	80	
Grumman	12				30						20			
	12				30						56			
	12				40						100			
Mars Ball	13	Wheels	2	10-20 (Max)	185									
LRV	14	Wheels	4	<30*	35 (crater ledge)	0 or 457	40-45	40-45	8.10"	27"	200	1500	$1-1.5x10^3$	
Mission Requirements				45	40.6	305-457	45	45	3'	2-3'	400	100	576	

LV = length of vehicle

*analytical only

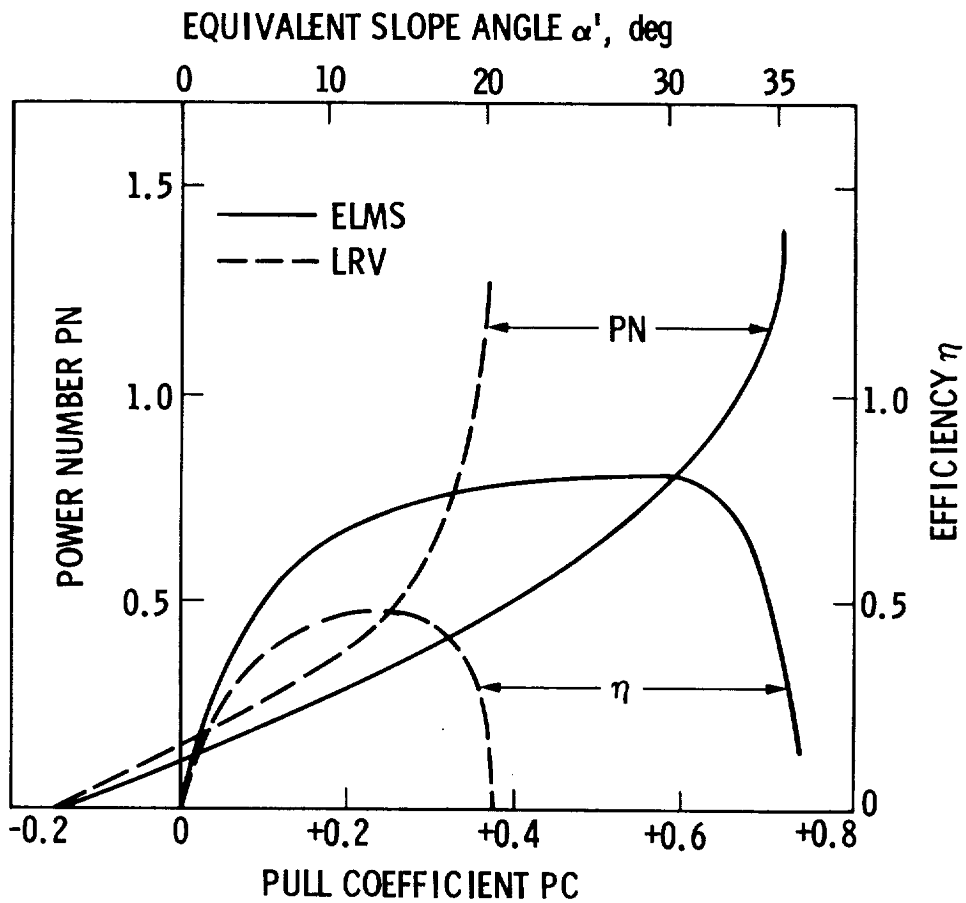
mance capabilities for these two vehicles is provided in Refs. 15-16, and a comparative summary will be outlined below. This comparison is not satisfactory from a mission/system engineering perspective, since it is necessary to examine the entire range of performance and packaging capabilities of these vehicles. Consequently, a comprehensive examination will still be required to assess which vehicle design can best satisfy the manned Mars operational scenairos and mission launch mass contraints.

Figure 1 shows a comparison of the performance characteristics of a large-scale, single 3 x 3 loop wheel (ie. 3 wheels with all 3 wheels driven) Elastic Loop Mobility System (ELMS) concept and a 6 x 6 wheeled Lunar Rover Vehicle (LRV) concept in loose, air dry soil. The Pull Coefficient (PC) and the Power Number (PN) can be considered to represent respectively the specific energy output by the system and the specific energy input to the system, both normalized with respect to the applied normal load and distance traversed by the rover unit. This plot should be indicative of the soft-soil slope angle that can be negotiated by the rovers at a given energy input. Higher slip values developed on slopes at the same thrust and torque level tend to indicate a relative increase in the specific energy consumption of the rover compared to its performance on level ground. This relative performance degradation increases with increasing PC values until a 100-percent-slip failure condition is reached at which the system is immobilized.

In addition to the vehicle's power efficiency, the following performance characteristics must be included in the assessment of an optimal vehicle design for the manned Mars mission: obstacle negotiation, ride quality, and maneuvering capabilities. We note that the 3 x 3 loop wheeled vehicle has been shown to have an obstacle climbing capability which is equivalent to the 6 x 6 wheeled vehicle. For climbing large obstacles (ex., 3-foot obstacles), both the six wheeled vehicle and the 3 x 3 loop wheeled vehicle will display a substantial angular displacement of its rigid frame, as shown in Figure 2. Both vehicle designs are maneuverable enough to enable them to navigate either over or around the boulder fields associated with the Viking Lander 1 and 2 geologic sites (Ref. 17). It is believed that vehicle traversals associated with alternate sites may be less abundant in rocks, but still subject to opera-

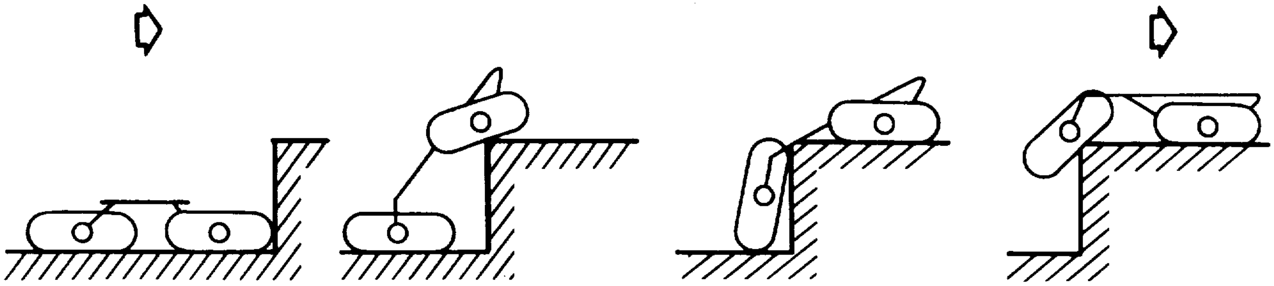
COMPARISON OF PERFORMANCE CHARACTERISTICS OF LARGE-SCALE, SINGLE ELMS UNIT AND LRV WHEEL IN LOOSE, AIR DRY SOIL

Figure 1



OBSTACLE NEGOTIATION BY 3 x 3 ELMS VEHICLE MODEL AND 6 x 6 "ELASTIC-FRAME" WHEELED VEHICLE

Figure 2



tional restrictions due to the presence of the sandy, sloping soil encountered along the traverse.

Clearly, the above two vehicles cannot satisfy all of the requirements outlined in Table 2. Thus, these requirements may have to be relaxed. It should be noted, however, that the power required for obstacle negotiation may represent a constraint on vehicle selection. For climbing over obstacles, for moving around very tight spaces, and for platform stability during drilling operations, the walker technology (discussed below) offers a potential advantage over conventional vehicle designs.

UNCONVENTIONAL LEGGED TECHNOLOGY FOR A ROVER VEHICLE

In the above discussion, no assessments have been made of the wheeled and loop wheeled vehicle technology performance capability in comparison with walker technology. To this end, Odetics Corp. was asked to generate the design of a walker vehicle which could be compactly stowed within a $1m^3$ volume and which could satisfy the Mobility characteristics outlined in Table 1. This vehicle has a variable stance and gait, and omnidirectional movement capability (Ref. 18).

Figure 3 shows the vehicle in its fully deployed configuration, traversing a 1 m wide trench. In this configuration, the vehicle design is inherently stable, having a large base with a low center of gravity. In Figure 4, the vehicle is shown traversing a 1 m boulder. Comparison of Figure 3 with Figure 4 shows that the main body frame of the vehicle has now been elevated to facilitate large boulder traversal while maintatining platform stability. The stresses experienced by the payload are thereby minimized with this design.

UNCONVENTIONAL ROVER LEGGED TECHNOLOGY VERSUS ALTERNATIVE CONVENTIONAL ROVER TECHNOLOGY COMPARISONS

A preliminary performance evaluation has been made of wheel, loop wheel, track, and walker vehicle technologies. For this comparison, the specific resistance of these vehicles was plotted against each other as a function of speed, as shown in Figure 5. The specific resistance, e (Ref. 19), is defined as: $e = P / (WV)$ where P is the mechanical power input to the vehicle--that is, the output power of the prime mover; W is vehicle weight; and V is vehicle velocity. Specific resistance can also be thought of as the inverse of the lift-to-drag ratio, where "drag" is

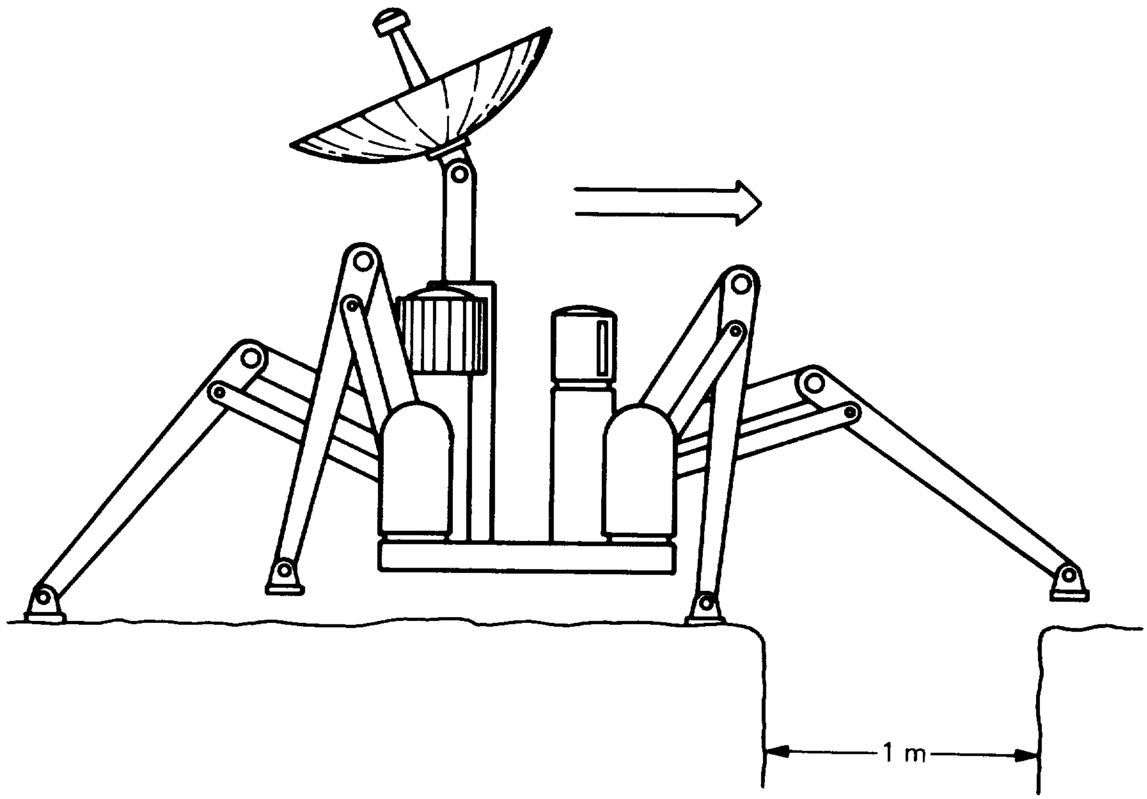


Figure 3. Mars "Rover" Traversing 1 meter Wide Trench
(Four legs shown)

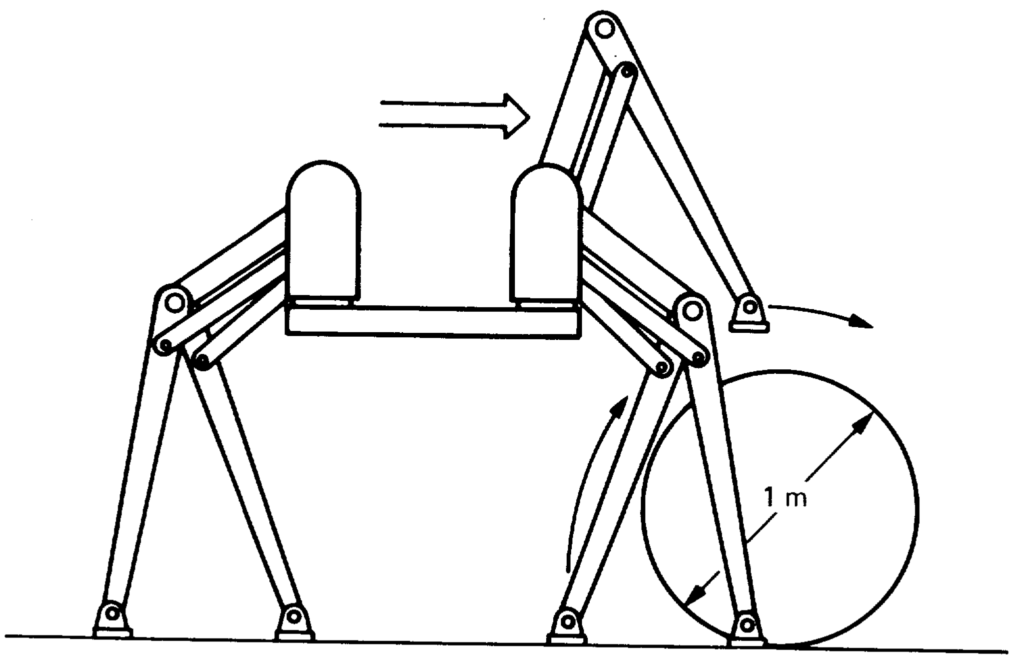
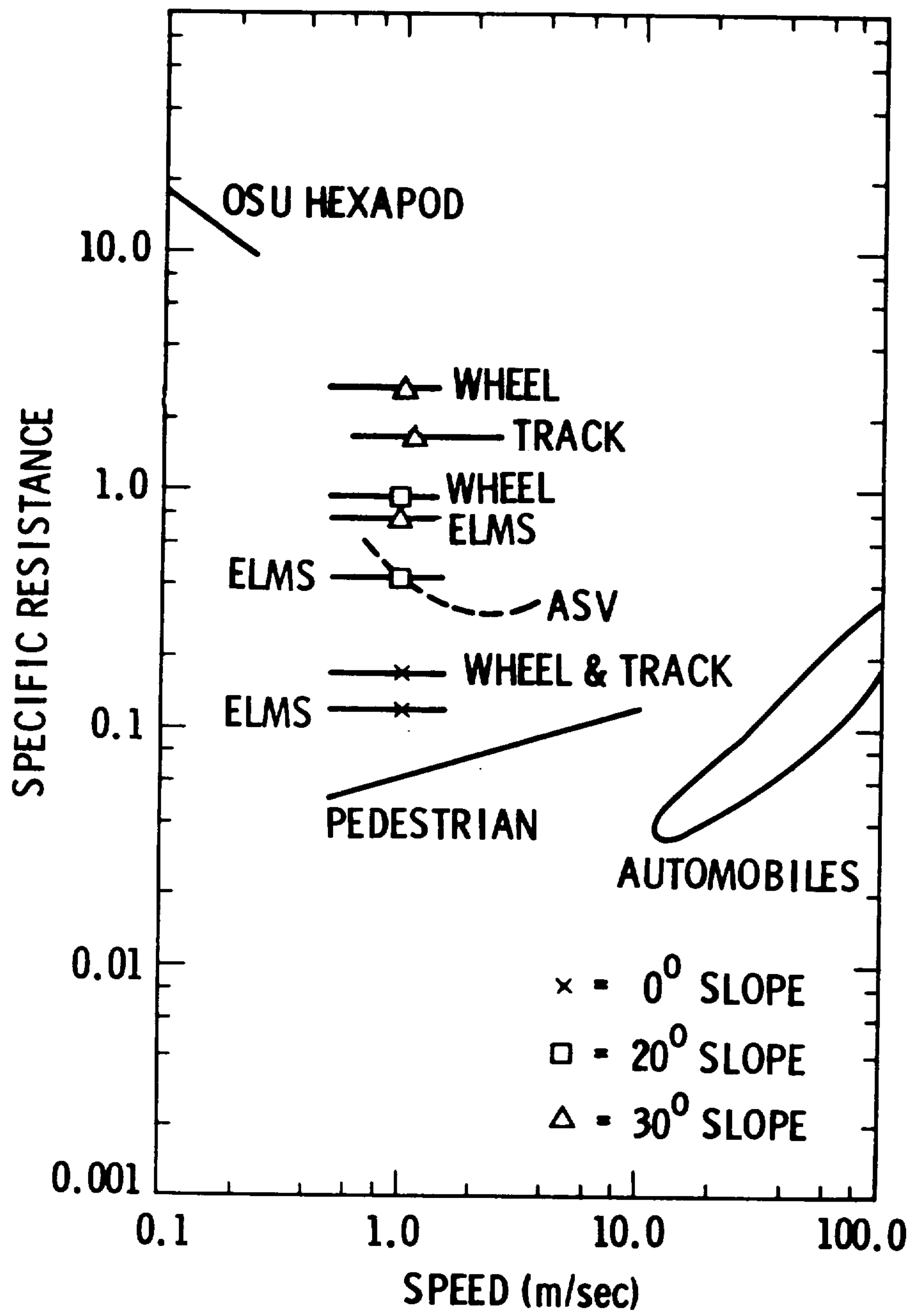


Figure 4. Mars "Rover" Raising One Leg Over An Obstacle
(Four legs shown)



SPECIFIC RESISTANCE $\left\{ \epsilon = \frac{\text{POWER}}{\text{WEIGHT} \cdot \text{VELOCITY}} \right\}$ AS A FUNCTION OF SPEED [NOTE: SPECIFIC RESISTANCE IS THE INVERSE OF THE LEFT-TO-DRAG RATIO, WHERE "DRAG" IS AN EFFECTIVE DRAG INCLUDING ALL ENERGY-DISSIPATION MECHANISMS]

(FIG. 5)

an effective drag including all energy-dissipation mechanisms. From this plot, it may be seen that recent advances in legged locomotion (i.e. the Adaptive Suspension Vehicle ASV) currently make this technology competitive with wheel, track, and loop wheeled systems operating on prepared surfaces. It should be noted that the ASV speed has been optimized for over 2 m/sec and the leg has been designed to support loads far greater than those required for currently envisioned manned or unmanned Sorties on Mars. Thus, it is anticipated that the power consumption of the vehicle should improve with reoptimization of the vehicle's leg design for the lower speeds and reduced loads.

The walker's design is flexible enough to provide for the integration of claws, picks, or alternative grappling devices with removable treaded forrt designs, in order to prevent foot slippage. Furthermore, the vehicle's design offers limited foot contact with the soil, as compared to wheels which are continually compressing the soil surface and pushing sand out of the way as they go. Thus, this vehicle should be able to succesfully negotiate 45 degree slopes in air dry soil simulant (Ref. 18). Contrary to the walker described above, the relative performance of wheeled vehicles and loop wheeled vehicles degrades rapidly for increasing slope angles. If the energy performance of the walker can be improved to a state roughly equivalent to that of 6 x 6 wheel or 3 x 3 loop wheel vehicles, it is anticipated that this vehicle will out-perform alternative concepts on the steep slopes and rugged terrain conditions which are anticipated to be encountered at the geology sites of current mission interest.

Before any final vehicle selection can be made, a model of the terrain-vehicle system for off-road locomotion must be developed. This type of analysis is critical to the optimal selection of a vehicle concept, and will ultimately provide a considerable cost savings in the final phase of the vehicle's engineering design and development.

DARPA UNCONVENTIONAL LAND VEHICLE PROGRAM

Currently, the Defense Advanced Research Projects Agency (DARPA) has an unconventional land vehicle program which is focused on the development of a walking machine. However, most of the program's effort is directed toward the solution of the complex issues associated with the walking machine's control, in order to provide a field test of a large scale version of this machine in FY '86. A well-focused research and development program for the transfer of this technology to space applications must be directed toward improving the vehicle's power efficiency, stability, and control.

ROVER VEHICLE DESIGN ASSESSMENT SUMMARY

A preliminary examination has been made of existing rover vehicle concepts in comparison with a proposed set of Mars rover operational requirements. The 6 x 6 wheeled vehicle, 3 x 3 loop wheeled vehicle, and walker vehicle technologies were analytically compared for the following point design concepts: Lunar Rover Vehicle, Elastic Loop Mobility System, OSU Hexapod, and Adaptive Suspension Vehicle. Based upon this comparison, the 3 x 3 loopwheel vehicle showed equivalent stowage and step climbing capability, as well as improved slope climbing performance and efficiency charcteristics over a 6 x 6 wheel vehicle. However, neither vehicle can satisfy the 45 deg Mars obstacle negotiation requirements. Furthermore, both vehicles suffer in the area of platform stability during traversal of rugged terrain and exhibit some difficulty in negotiating around obstacles. On the other hand, the hexapod vehicle offers excellent platform stability and it can currently satisfy all postulated Mars rover operational requirements (i.e., step climbing, obstacle traversal and negotiation, and slope climbing). Walking vehicles show an energy cost problem in comparison with the more conventional rover technologies. This issue must be addressed if this technology is to ever be employed for Mars rover applications.

ACKNOWLEDGEMENT

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SURFACE DRILLING TECHNOLOGIES FOR MARS

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ABSTRACT

We propose rock drilling and coring conceptual designs for the surface activities associated with a manned Mars mission. Straightforward extensions of equipment and procedures used on Earth are envisioned for the sample coring and shallow high explosive shot holes needed for tunneling and seismic surveying. A novel rocket exhaust jet piercing method is proposed for very rapid drilling of shot holes required for explosive excavation of emergency radiation shelters. Summaries of estimated equipment masses and power requirements are provided, and the indicated rotary coring rigs are scaled from terrestrial equipment and use compressed CO_2 from the martian atmosphere for core bit cooling and cuttings removal. A mass of 120 kg and power of 3 kW(e) are estimated for a 10 m depth capability. A 100 m depth capacity core rig requires about 1150 kg and 32 kw(e). The rocket exhaust jet equipment devised for shallow (3m) explosive emplacement shot holes requires no surface power beyond an electrial ignition system, and might have a 15 kg mass.

INTRODUCTION

Achievement of manned Mars mission scientific exploration and permanent human occupation of the planet will require drilling and coring operations associated with subsurface exploration and facility construction [1,2]. These operations will include: (1) subsurface geologic sample coring, (2) geophysical instrument emplacement, (3) seismic source (explosive shot hole) emplacement, and (4) shot hole drilling for explosive excavations. Successful execution of these drilling operations will be essential for energy, water, and mineral resource assessment and for understanding the origin, evolution, and present structure of the planet. We suggest that rather straightforward extensions and adaptations of terrestrial equipment are possible to effectively solve the required subsurface sampling and shot-hole formation problems. The suggested systems were developed on Earth in response to much the same needs as will exist for the Mars exploration scientific efforts.

DRILLING AND CORING REQUIREMENTS

We will assume a Mars surface exploration scenario[1] consisting of five landings at three different sites. The ultimate objective of the missions will be to establish a permanently manned outpost to serve as a base for the scientific exploration of the planet. This implies drilling through a wide range of rock and soil types for both scientific and construction purposes. In all cases, the drilling and coring operations should be as automated as possible (where consistent with reliability and mobility) to minimize the expenditure of valuable crew time.

Scientific Drilling and Coring

Drilling and direct sampling of the uppermost materials of the martian surface will be essential to the emplacement of instruments, determination of near-surface stratigraph, and interpretation of geophysical measurements. Core samples should be large enough to encompass anticipated textural inhomogeneities and the holes should be as deep as possible. Because little is known of the materials that are likely to be encountered, arbitrary decisions on the drilling parameters are inevitable and final details will be largely controlled by anticipated power and mass availabilities on the martian surface. Consequently, as outlined by Blacic *et al* [2], we propose that two basic scientific drilling and coring capabilities be developed: (1) capability to drill and core a single ~ 100m "deep" hole at each landing site with support (e.g., power) provided by the landing craft, and (2) a highly mobile drilling and coring capability for many ~ 10m deep "shallow" core holes supported by roving exploration vehicles. In both cases, we suggest hole diameters of about 15cm and oriented cores of 7cm diameter. Furthermore, since volatile materials are likely to be contained in the rocks, refrigerated storage of a substantial portion of the core (say, 25%) should be provided.

Explosive Shot Holes

Mars exploration will need extensive drilling of shallow, noncored holes for the emplacement of explosives in support of both scientific[2] and operational[3] objectives. In most cases, shot holes for explosive excavations need only be a few meters deep and a few centimeters in diameter. The holes can be drilled as rapidly as possible with no regard to preservation of the host rock or samples. In the case of the remote, rapidly excavated radiation emergency shelter[3] that will be needed, the explosive emplacement shot holes must be drilled in a matter of minutes. Emplacement of explosives for active seismic surveys can be in the same shallow core holes drilled for the rover vehicle geologic and resource explorations.

CORING AND DRILLING APPROACHES

We assume that coring hardware is required that is relatively insensitive to rock and soil type. The device should reliably yield high quality cores at a high recovery rate. Limited manpower requirements and restrictions on mass and power are anticipated. The major problem to be addressed is the cooling of the core bit and clearing of rock chips and cuttings from the core holes. We suggest that an electric powered, rotary driven core rig is appropriate. The optimum fluid for core bit cooling and hole cleaning appears to be compressed CO_2 from Martian atmosphere. To achieve cores in permafrost-like material will require a reverse CO_2 fluid circulation with cold CO_2 flowing in contact with the core hole wall. A stock of a variety of core bit types and configurations will be needed to achieve the desired core quality and recovery because of the expected wide variability in rock and soil conditions. These bits will be the major expendable items needed for the proposed core rigs. Characteristics and descriptions of the core rigs are summarized in Table 1.

The second type of hardware we envision is designed to drill small diameter shot holes for emplacement of high explosive charges for tunneling and other excavation tasks. The best choice for this application would appear to be a percussion drill powered by compressed CO_2. Hole cleaning of these relatively shallow, small diameter holes would be accomplished by exhausting the CO_2 drive gas into the bottom of the hole to lift the cuttings. The percussion drill approach would provide rather rapid production of holes in a wide variety of media. This type of equipment is well developed and widely used for similar applications on Earth. The additional use of this tool as a jack-hammer for construction purposes is possible. Table 2 summarizes the descriptions and characteristics for these two shallow shot hole drilling techniques.

TABLE 1

MARTIAN CORE RIG TYPES AND CHARACTERISTICS

Hole Type[a]	Depth Capability (m)	Hole Diameter (cm)	Core Diameter (cm)	Average Coring Rate[b] (m/hr)	Deployment Mode
Deep	100	15	7	8	From landing craft.
Shallow	10	15	7	2	From rover vehicle, possibly towed.

(a) Visualized is rotary drive by electric motor with compressed CO_2 for bit cooling and hole cleaning.

(b) Core rig concepts wireline type core run-in and retrieval capability with a 2-meter core tube length for ease of core handling and equipment mobilization.

TABLE 2

MARTIAN SHOT HOLE DRILLING TECHNIQUES AND CHARACTERISTICS

Application or Hole Type	Depth Limit (m)	Hole Diameter (cm)	Drilling Time (min)	Deployment	Technique
Explosive Tunneling or Seismic Shot Hole	3	5	30	From rover and manual.	Percussion drill w/ star drill bit & CO_2 hole cleaning
Emergency Shot Hole	3	5	5	From rover with manual set up and firing.	Solid rocket exhaust jet; hole cleaning by exhaust gas

Finally, we consider equipment that could be deployed very rapidly to create high explosive shot holes almost instantaneously. These holes are required for explosive excavation of emergency shelters from solar flare radiation[3]. The approach that is envisaged uses solid rocket exhaust penetrators[4] that can produce holes in any rock or soil type in a matter of seconds. Such solid rocket ground piercing units appear optimum for this application for reasons of drilling speed, long-term storage, mobility, rapid deployment, safety, and simplicity of set-up and firing.

EQUIPMENT CONCEPTS

We now turn to some specifics of the requirements for coring and drilling on the martian surface and to preliminary descriptions of equipment. The following drill rig and equipment descriptions are based on analogous terrestrial drilling applications. The major constraints in the selection of the approaches, concepts, and initial designs we present are the need for (1) simple and reliable technology, (2) drilling and coring in variable ground (hard rock, soils, and frozen rock and soil), (3) use of an expendable fluid for bit cooling and hole cleaning, (4) minimum mass and power consumption, (5) rapidity of progress, (6) possibility of automation, as a trade-off against simplicity and mobility, and (7) safety and reliability of equipment and procedures.

Scientific Drilling and Coring

As discussed above, two types of coring equipment are proposed based on analogies to exploration activities routinely conducted on Earth. Our concept is illustrated in Figure 1 where the rig with deep coring capability is depicted. A direct adaptation from terrestrial hardware is envisaged, with compressed martian atmosphere (CO_2) used as a core bit cooling and hole cleaning fluid. Core drilling can be extensively automated. However, we believe that manual set-up, core barrel run-in and retrieval, and core removal operations are likely to be more reliable, at least initially. The unit is shown trailer-mounted, so that deeper holes away from the lander site might be planned if found to be needed and if justification is developed by the shallow core hole traverses or other exploration activities. Power can be supplied by cable for relatively short distances from the landing craft. Preliminary selection of the features of this core rig are indicated in Table 3. All parameters,

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TABLE 3

CORE RIG EQUIPMENT SPECIFICATIONS

Core Rig Type	Derrick Height (m)	Core Barrel Length(m)	Rig Mass[a] (kg)	Rotary Power (kwe)	CO_2 COMPRESSOR Pressure (MPa)	Power (kwe)	Flow Rate (m^3/s)	Comments
Deep (100m)	12	3	1150	32	2	5	0.004	Mass estimate includes 100m length of drill string.
Shallow (10m)	6	2	120	3	1	2	0.002	Mass estimate includes 10m length of drill string.

a Includes draw works, CO_2 compressor, and rotary drive. Note that draw works are required for core-barrel wireline (cable) retrieval and run-in. Mass estimates are scaled from terrestrial equipment both structural and material optimization included.

TABLE 4

SUMMARY OF SHOT HOLE DRILLING EQUIPMENT CHARACTERISTIC

Hole Type	Depth Capability (m)	Mass[a] (kg)	Mobility Mode	Drilling Time (min)	Power Sources	CO_2 Compressor Pressure (kPa)	Power (kWe)	Comments
Percussion	2-3	90	skid Mounted	30	Electric Motor & Compressed CO_2	0.7	2	Hole cleaning injected through hollow drill rods.
Solid Rocket Jet	2-3	30	Pallet & Tripod	5	Solid Rocket Propellant	N/A	N/A	Reverse reaction jets reduce together and guide tripod requirement. Hole cleaning by high exaust velocity and vaporization of rock "cuttings"

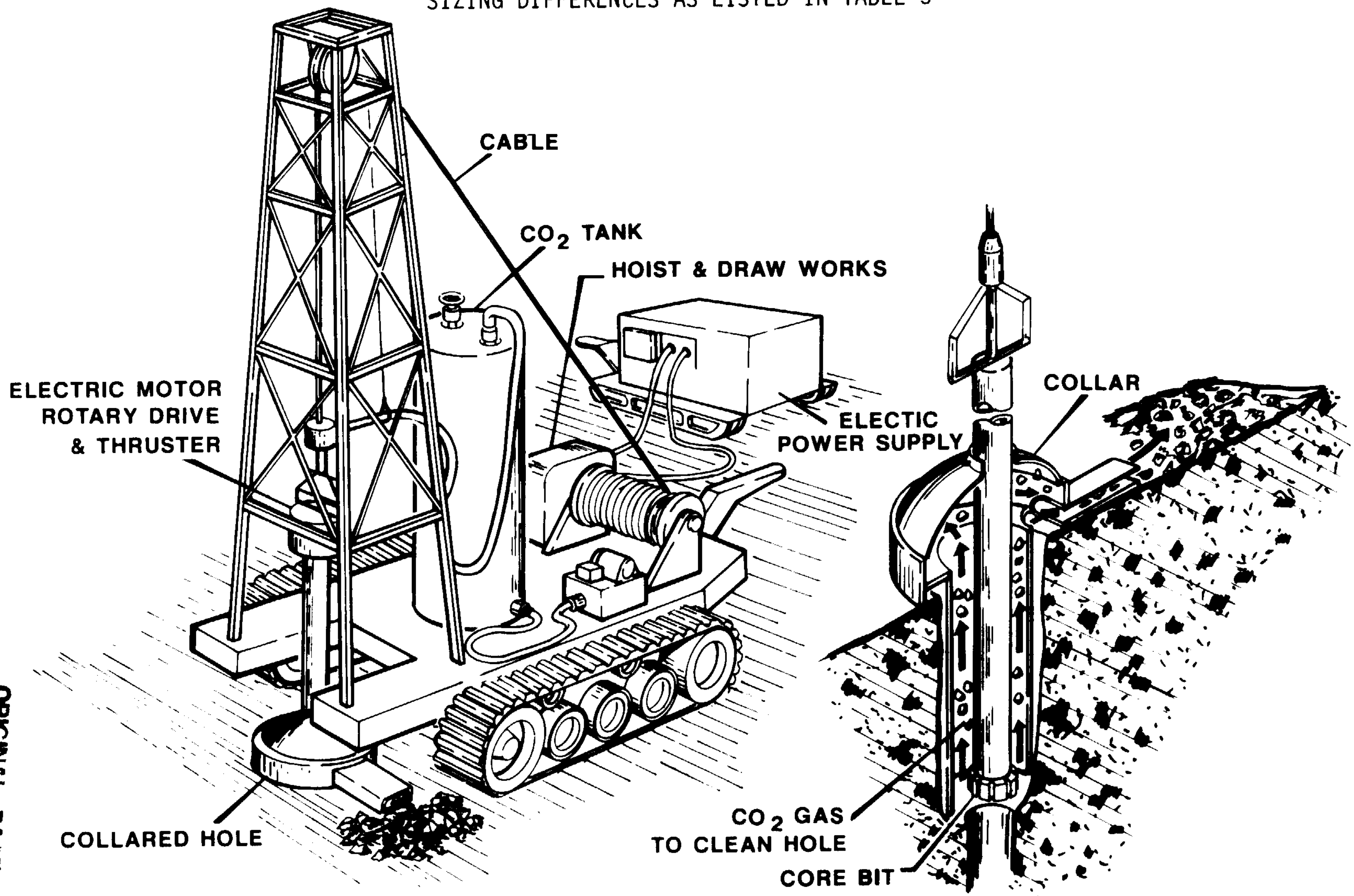
a Scaled from similar terrestrial equipment.

FIGURE 1

GENERAL CONCEPT FOR A MARS CORING RIG

THE BASIC LAYOUT WOULD BE THE SAME FOR BOTH THE DEEP AND SHALLOW DRILLING RIGS:

SIZING DIFFERENCES AS LISTED IN TABLE 3



sizes, power requirements, and flow capacities were scaled from existing equipment[5]. Considerable optimization should be possible with detailed design, trade-off analysis, and efficiency enhancement studies.

The shallow core rig design is also specified in Table 3, and is visualized as a highly mobile rig that can be mounted on skids or trailer to be towed by a rover vehicle or even manually. It should also be designed for ease of disassembly and assembly into light weight sub-components so that it can be "back packed" into rugged and remote areas.

The core drilling procedures and operations are completely analogous to those on Earth. One person operation is visualized with attention and activity requirements mainly focused at core barrel handling intervals. This approach is suggested because of the anticipated complexity and high mass requirements of an automated system. Also, consideration must be given to maintaining core quality during the crucial stage of removal of core from the barrels.

Percussion Drill

The conceptual sketch in Figure 2 indicates our suggested approach for a mobile, shallow depth capacity shot-hole drill. The concept is a direct analogy to the terrestrial jack hammer or pneumatic percussion shot-hole drill widely used in mining. The equipment requirements are illustrated in Table 4 and are scaled from typical hardware currently in use[4]. The compressed atmospheric CO_2 is used both to drive the oscillating impact mass and to clean chips from the hole. A hollow-shaft drill rod is used with a star drill bit. Slight rotary motion of the drill rod is provided to enhance cutting rate and cleaning. We estimate that a 3m-long, 50mm diameter hole can be cut in one-half hour in hard rock and more rapidly in loose soils or gravel. Table 4 records the estimated characteristics for this shot hole drilling device. The mobility is by a light weight sled on skids. Manual operation is considered optimum due to the many potential applications of such a device. We anticipate wide use of this tool concept in a variety of construction chores. Also, the same basic approach can be adapted for repetitive, specialized applications such as an automated drill, blast, and muck tunneling machine[3].

FIGURE 2

CONCEPT FOR A MOBILE ROTARY PERCUSSION DRILL FOR PRODUCING SHOT HOLES

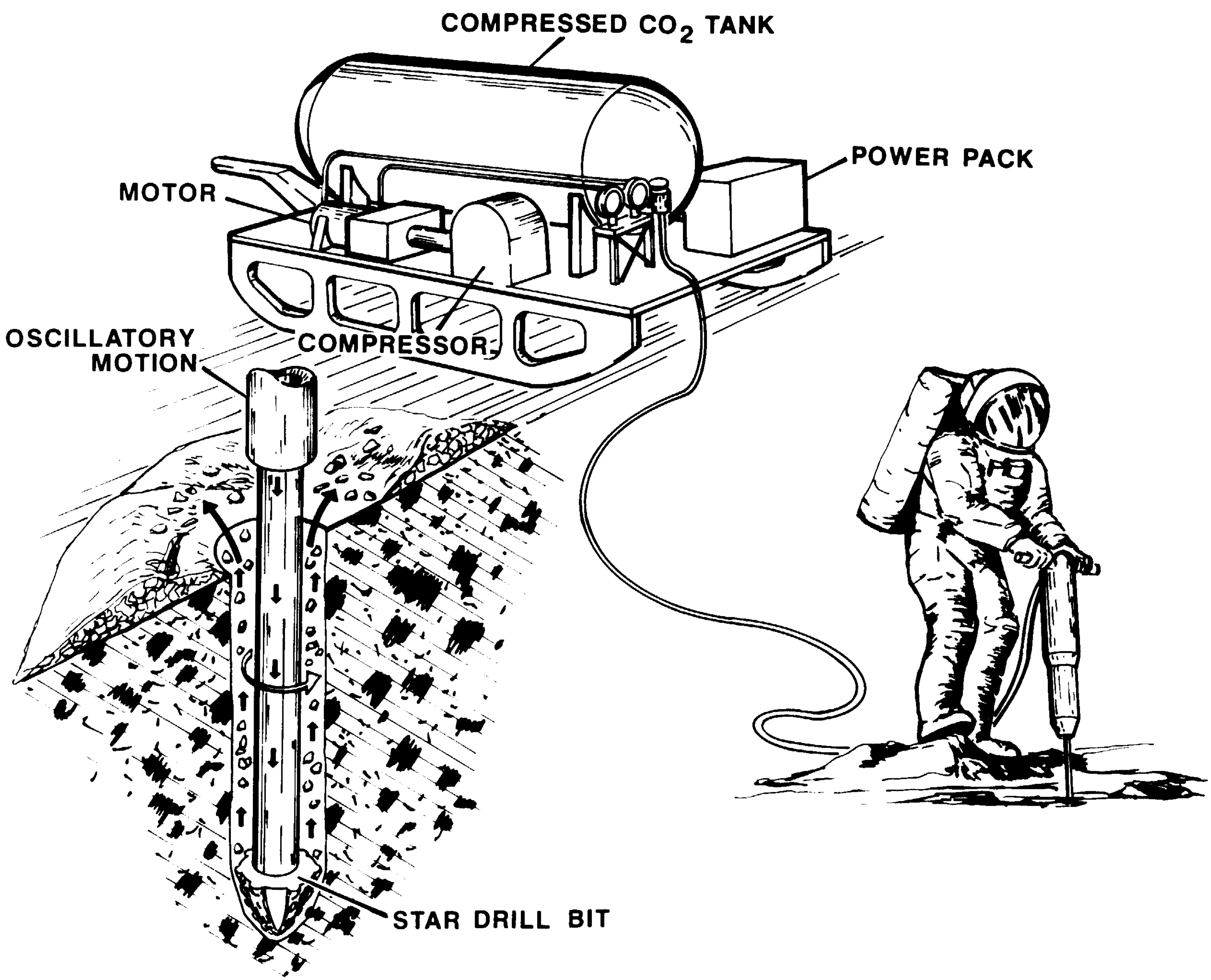
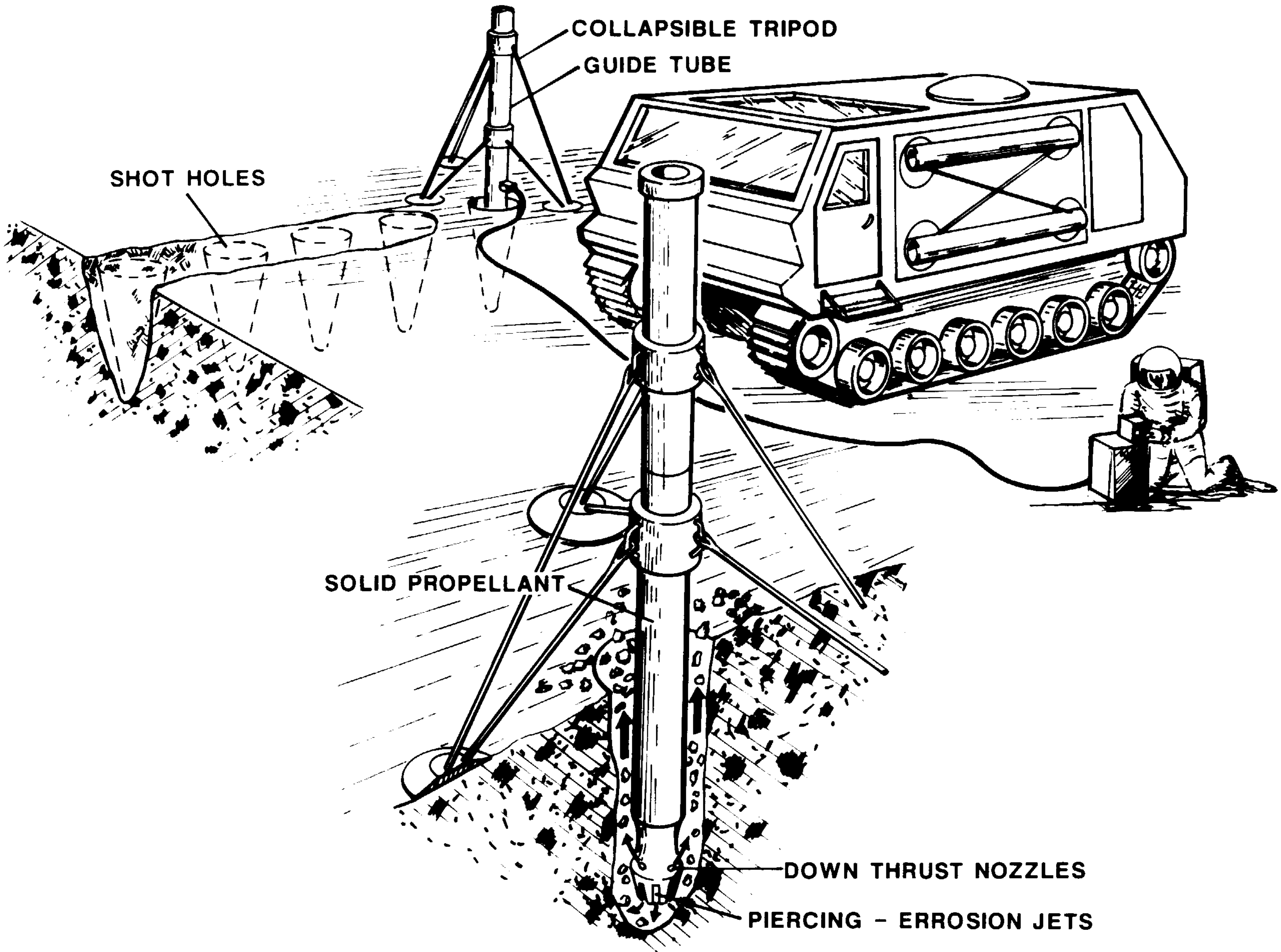


FIGURE 3

RAPID DRILLING CONCEPT USING THE ROCKET EXHAUSE JET PIERCING TECHNIQUE
SOME SIMPLE SET-UP METHODS ARE INDICATED IN THE UPPER LEFT
SCHEMATIC DETAILS OF THE ROCKET NOZZLE ARE SHOWN IN THE LOWER RIGHT



Rocket Exhaust Drill

Extremely rapid shot hole drilling for emergency shelter construction can be achieved by use of tethered, solid rocket exhaust jet piercing technology[4]. In this approach, a section of tubing and a following guide tube are erected with a light weight tripod. Figure 3 shows the proposed equipment configured to be used in conjunction with a rover vehicle. A single shot hole can be made in a few seconds after a few minutes to deploy, assemble and fire (ignite) the solid propellant. A row of holes can be made in sequence if only one tripod is provided, or a mutiple set-up is possible. This is the most rapid drilling method (including set-up) we know of for shallow shot holes.

CONCLUSIONS

The manned Mars mission drilling applications of geologic sampling, emplacing scientific explosive sources, producing shelters and other constructions, and rapid excavation of remote emergency shelters are projected to be rather straightforward adaptations of terrestrial equipment and procedures. The proposed approaches rely on established technologies and should be safe, reliable, easily automated to the degree deemed desirable, and adaptable to a wide range of anticipated applications on Mars. The concepts feature manual operation of essential activities where its employment can minimize mass, power, and complexity. Equipment designs can be accomplished and optimized for martian conditions. Design of the required CO_2 compressors should be a priority task, but can rely on the extensive Earth-bound experience in this area. All the designs outlined can be built in prototype hardware forms and tested at atmospheric pressures, temperatures, and compostions expected on Mars and in simulated materials likely to be encountered in the martian subsurface.

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SURFACE INFRASTRUCTURE FUNCTIONS, REQUIREMENTS, AND SUBSYSTEMS FOR A MANNED MARS MISSION

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ABSTRACT

Planning and development for a permanently manned scientific outpost on Mars requires an in-depth understanding and analysis of the functions the outpost is expected to perform. The optimum configuration that accomplishes these functions then arises during the trade studies process.

In a project this complex, it becomes necessary to use a formal methodology to document the design and planning process. The method chosen for this study is called top-down functional decomposition. This method is used to determine the functions that are needed to accomplish the overall mission, then determine what requirements and systems are needed to do each of the functions. This method facilitates automation of the trades and options process. In the example, this was done with an off-the shelf software package called TK!Solver.

The basic functions that a permanently manned outpost on Mars must accomplish are: 1) Establish the Life Critical Systems, 2) Support Planetary Sciences and Exploration, and 3) Develop and Maintain Long-term Support Functions, including those systems needed towards self-sufficiency.

The top-down functional decomposition methodology, combined with standard spreadsheet software, offers a powerful tool to quickly assess various design trades and analyze options. As the specific subsystems, and the relational rule algorithms are further refined, it will be possible to very accurately determine the implications of continually evolving mission requirements.

INTRODUCTION

Large scale systems involve a large number of often abstract variables, changing conditions and system requirements, as well as varying interpretations of definitions. It rapidly becomes difficult to assess the entire system without a formal documented process. Often, many solutions turn out to be counter-intuitive.

A carefully documented methodology also facilitates automating many facets of the process, particularly the computation of overall system parameters, subsystem by subsystem. These parameters include weight, volume, geometry and power requirements. By incorporating a set of rules that define the subsystems and their interactions with each other, the computer can be used to quickly assess the effects of various design changes, working towards the optimum configuration for a given set of mission requirements.

TOP DOWN FUNCTIONAL DECOMPOSITION

This method starts with the overall Gross System Requirement for the mission to be accomplished. The functions that need to be done to accomplish that goal are then carefully outlined in the order that they would occur. Each of these 1st level functions are broken down further into 2nd, 3rd, etc., level functions until all the necessary detail is defined, Next, the specific requirements needed to accomplish each of those functions are determined. Finally, the hardware or subsystems that are needed to meet these functional requirements are determined. This hardware has associated mass property and power requirements that can be put into a functional matrix to determine overall mass property and power requirements. These matrices, combined with the input/output interactions between the subsystems and the functional groupings, can be used to assess the affects of various mission requirements on the needed system parameters.

MAIN FUNCTIONAL GROUPS

It's clear from looking at the overall mission requirement, "To Establish and Maintain a Permanently Manned Outpost on Mars", and the functional decomposition, that the Mars surface infrastructure is driven by four main areas - Life Critical Systems, Planetary Science and Exploration Systems, Mission Support Systems and Long-Term Self Sufficiency Systems. These are defined as follows:

Life Critical Systems

These are those systems necessary to ensure survival on Mars. Currently, these systems include: Environmental Control and Life Support Systems (ECLSS), Thermal Control Systems, Crew Systems, Nutritional Needs, Radiation Exposure Protection and Monitoring, Health Maintenance, Electrical Power Processing, and Extravehicular Activity (EVA) Capability.

Planetary Science and Exploration Systems

This includes Martian chemical, physical, biological and magnetic field phenomena. Specifically, and in order of priority, the following areas are included: Local Chemical and Physical Phenomena, Local Biological Phenomena, Martian Atmosphere, Geological Phenomena, Martian Magnetic Field, Global Chemical and Physical Phenomena, and Global Biological Phenomena.

Mission Support Systems

Construction - Habitat Assembly and Protection

Construction subsystems will be used mainly for the initial establishment of the life critical systems. The major concerns in this area are design, assembly, growth flexibility, safety, and maintenance. The need for Galactic Cosmic Ray and Solar Event protection must also be looked into.

Power - As required for the entire outpost

Dependable and safe power generation must be investigated for use by the entire outpost. It also must be sufficiently flexible to allow growth, as the outpost expands. Power requirements will be determined by the needs of the other systems. As the design is further refined, power requirements can be expected to increase.

Transportation - Sample collection, experiment deployment, maintenance

Various modes of transportation must be investigated. In order to make adequate trade studies, detailed information concerning vehicle range, mass properties, payload capacity, dependability, etc., must be determined. The options should include lunar-type rover, mobile pressurized lab, rover with inflatable shelter, and remotely piloted vehicle, as well as other vehicle concepts.

Long-Term Self Sufficiency

The most economically viable scenarios are those that make use of existing resources, and recycle them as much as possible. This is considered a key area for permanent human presence in space. This section also includes systems needed for habitat expansion.

In-Situ Resources Utilization

The Martian environment contains most of the resources needed to provide complete self-sufficiency. These resources can be utilized with food production facilities such as greenhouses, hydroponics, aquaculture, etc.; an atmosphere reduction facility to produce fuel, water, air, energy storage, fertilizer and other chemicals; and a materials processing facility to make metals, glass, cement, and other structural materials.

Habitat Growth - Configurations, including Habitat Construction from Martian materials

This includes techniques such as explosives, inflatable shelters and spray sealants for the creation of pressurized shelters.

ESTABLISHING GROUNDRULES

One of the most difficult problems at this point is establishing clear guidelines without restricting promising avenues of investigation. However, some decisions will have significant impact on surface infrastructure synthesis. Three such areas are mentioned here.

Space Station Common Modules

It is cost-effective to use as much existing technology as possible. Using the proposed Space Station Common Modules could significantly bring down the cost of a Manned Mars Mission and improve system reliability. It was decided to investigate using these modules to meet mission requirements on the surface of Mars. Preliminary evidence suggests these modules will prove quite sufficient for these requirements. Unfortunately, the parameters have not been completely fixed for the Space Station Common Module. If there is much change from the reference configuration, the decision to use them on Mars will have to be reevaluated. Table 1 shows the first order weight and volume requirements for the habitation module (HAB1). It's also possible to modify an additional module to be used as the scientific laboratory (LAB1). Some redundancy of Life Critical Systems could then be integrated into the design, eliminating single point failure areas.

Mission Modules - In-transit/Surface

It is not yet clear whether the surface mission modules should be used by the crew in-transit. If so, subsystems will have to be flexible

TABLE 1

SAMPLE OF TOP LEVEL SUBSYSTEMS MASS BALANCE VARIABLES

St	Input	Name	Output	Unit	Comment
					********** MISSION PARAMETERS ******
	4	P			NUMBER OF CREW MEMBERS
	1.5	D		yr	LENGTH OF MISSION
		PD	2190		CREWSIZE * MISSION LENGTH
	0	WATRECY			PERCENT WATER RECYCLABILITY
	.2	FOODHYD			PERCENT FOOD HYDRATION
	0	FOODSIT		lb/d	IN-SITU FOOD PRODUCTION
	0	WATSIT		lb/d	IN-SITU WATER PRODUCTION
	0	POWSIT		w	IN-SITU POWER PRODUCTION
					********** MISSION PARAMETERS ******
					***********************************
		WMISS	15834.843	lb	TOTAL WEIGHT REQUIRED TO SURFACE
		VMISS	1260	ft3	TOTAL VOLUME REQUIRED TO SURFACE
		PMISS	2322	w	TOTAL SURFACE POWER REQUIREMENTS
					***********************************
					********* MISSION SUBSYSTEMS *******
		VLABEQ	181.9	ft3	LAB VOLUME FOR EQUIPMENT
		WLABEQ	749	lb	WEIGHT OF LAB EQUIPMENT
		PLABEQ	1312	w	POWER FOR LAB EQUIPMENT
					***********************************
		VLCS	4503.2824	ft3	******** LIFE CRITICAL SYSTEMS *****
	0	WLCS		lb	
	0	PLCS		w	
		VECLSS	742	ft3	ECLSS
	0	VTCS		ft3	Thermal Control System
		VCS	2811	ft3	Crew Systems
		VNUTRI	470.28239	ft3	Nutritional Needs
		WNUTRI	17958	lb	
	0	VRADEXP		ft3	Radiation Exposure
	480	VHMF		ft3	Health Maintenance Facility
	1800	WHMF		lb	
	0	VHABPOW		ft3	Electrical Power and Processing
	0	VEVACAP		ft3	EVA Capability
					************************************
		VLCSS	0	ft3	*** LIFE CRITICAL SUPPORT SYSTEMS ***
		WLCSS	3000	lb	
		PLCSS	0	w	
		VCONSTR	0	ft3	Construction
		WCONSTR	0	lb	
		PCONSTR	0	w	
		VPOWER	0	ft3	Power
		WPOWER	3000	lb	
		PPOWER	0	w	
					************************************
		VPS	1260	ft3	********* PLANETARY SCIENCES ********
		WPS	2005	lb	
		PPS	2322	w	
					Chemical,Mineralogical,Petrological &
		VCHEM	1162.5	ft3	Envir. Interact. In Local Area (F2.1)
		WCHEM	1666	lb	
		PCHEM	2060	w	

enough to adapt to gravity differences that may exist between in-transit and surface environments. Having the same effective gravity in both the in-transit and Mars surface phases could solve this problem. However, the obvious question is once the modules are deployed on the surface, how does the crew return to Earth.

Radiation Considerations

Initially, it was thought that to be protected from Galactic Cosmic Radiation (GCR), the habitat would need to be buried under at least two meters of Martian soil. Recent data indicate that satisfactory short term (up to 4 years) radiation protection from GCR can be achieved with no external shielding. Any intermediate amount of shielding is unsatisfactory. This counter-intuitive development is due to the heavily ionizing heavy particles that are produced as secondary emissions as the lesser ionizing protons and electrons pass through the shielding. The GCR radiation dosage is approximately 50 REMS/yr in-transit and 25 REMS/yr on the surface of Mars (unprotected) during Solar minimum. On a 3 year mission with 1 1/2 years on the surface, this would give 75 REMS in transit and 38 REMS on the surface, for a total of 118 REMS, well below the current limit of 400 REMS career exposure limit. These numbers would be lower during Solar maximum (20 and 10 REMS/yr) since the increased magnetic field of the sun keeps out more of the non-solar cosmic rays. If this assumption remains valid, much of the construction and assembly equipment can be scaled down or eliminated.

The exposure dosages above assume no solar events (solar flares) during the entire mission. For the long transit and surface stay time involved, this assumption is not reasonable. Short-term solar event protection must be provided. For example, the August 1972 solar event would have given an unprotected astronaut in free space a lethal dose of 150,000 REMS. Fortunately, this extremely high dosage is very short term. Radiation protection to withstand this dosage need only be provided for about 12 hours. The equivalent of 4 1/2 inches of aluminum shielding would bring the dosage down to under 4 REMS (this corresponds to general shielding requirements of 30 grams/sq. cm.).

FUNCTIONAL BREAKDOWN

Table 1 shows a mass balance example of the top level subsystems. The infrastructure system is nowhere near completion, but the basic framework is established. This framework is useful in showing the subsystems developed using the functional analysis methodology. Of less importance are the present values of the variables. In many cases, the values were not known. In this case, a zero will appear either in the input or output column. Input variables are assigned by the user. Output variables are computed from the input variables, according to the rules on the rules sheet. The rules sheet is used to express the interactions among the various subsystems. As the method and subsystems are further refined, these will be reflected by additions and modifications to the rules sheet.

The top of the variables sheet shows the overall mission parameters. These input variables can be changed dynamically to show the total changes to the mission mass properties (volume, mass, and power requirements). These variables can be adjusted for changes in mission length, number of crew, as well as the percent of water recycling and food hydration. Variables can also be added to account for in-situ food, water or power production.

In establishing the functional framework, much effort was given to keeping it as general as possible. No assumptions have been made regarding for example, construction or transportation trade options. Specific trades will be studied via the various sets of inputs that can be used. This general approach has the added benefit that this framework can be used to examine trade options of any surface infrastructure system (lunar, for example). There is no breakdown for making flight manifest assignments for multiflight scenarios. This is a relatively easy addition and can be made when needed.

CONCLUSIONS

The most critical mission elements are those that involve the Life Critical Systems. Although the numbers shown are only the first rough pass, they do answer some fundamental questions. The proposed Space Station Common Module can be used to meet basic mission requirements for a permanently manned outpost on Mars. The module has a usable volume of 3980 cubic feet. The basic volume requirements for 4 crew members, 1 1/2 years on the surface, is about 4500 cubic feet. The additional needed volume can either be taken care of by modifying requirements or can be contained in the lab module, which will have excess volume, according to current science requirements.

The top-down functional decomposition methodology, combined with standard spreadsheet software, offers a powerful tool to quickly assess various design trades and analyze options. As the specific subsystems and the relational rules algorithms are further refined, it will be possible to very accurately determine the implications of continually evolving mission requirements.

USE OF CHEMICAL EXPLOSIVES FOR EMERGENCY SOLAR FLARE SHELTER CONSTRUCTION AND OTHER EXCAVATIONS ON THE MARTIAN SURFACE

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ABSTRACT

The necessity to shelter people on the Martian surface from solar flare particles at short notice and the need for long-term habitats with thick cosmic ray shielding suggests that explosives could be used effectively for excavation of such structures. Modern insensitive high explosives are safe, efficient, and reliable for rock breakage and excavation. Extensive Earth-bound experience leads us to propose several strategies for explosively-constructed shelters based on tunneling, cratering, and rock casting techniques.

INTRODUCTION

Extended duration manned surface exploration and, ultimately, permanent human presence on Mars will require protection from the constant galactic cosmic ray and intermittent solar flare irradiations. For the relatively short exposures on the Martian surface in the exploratory phase prior to a permanent outpost, the high energy proton flux associated with large, relatively unpredictable solar flares is the largest source of danger. It will be expensive and cumbersome if shelters transported from Earth are used to protect personnel at every point of their potential exposure to these lethal events. However, if indigenous rock materials could be used instead, then large savings in the mass required to be landed on the Martian surface are potentially possible. On the other hand, this approach implies an excavation capability for which the mass of the required construction equipment may negate any savings relative to bringing a preconstructed shelter. In this paper, we call attention to the fact that explosives are very efficient rock-movers. Modern explosive excavation technology can be used to safely, efficiently, and quickly construct a variety of structures that will be required as part of any realistic operations on the Martian surface.

In our discussion, we assume that approximately 50 gm/cm^2 shielding for a few hours during the intense phase of a large solar flare and 500

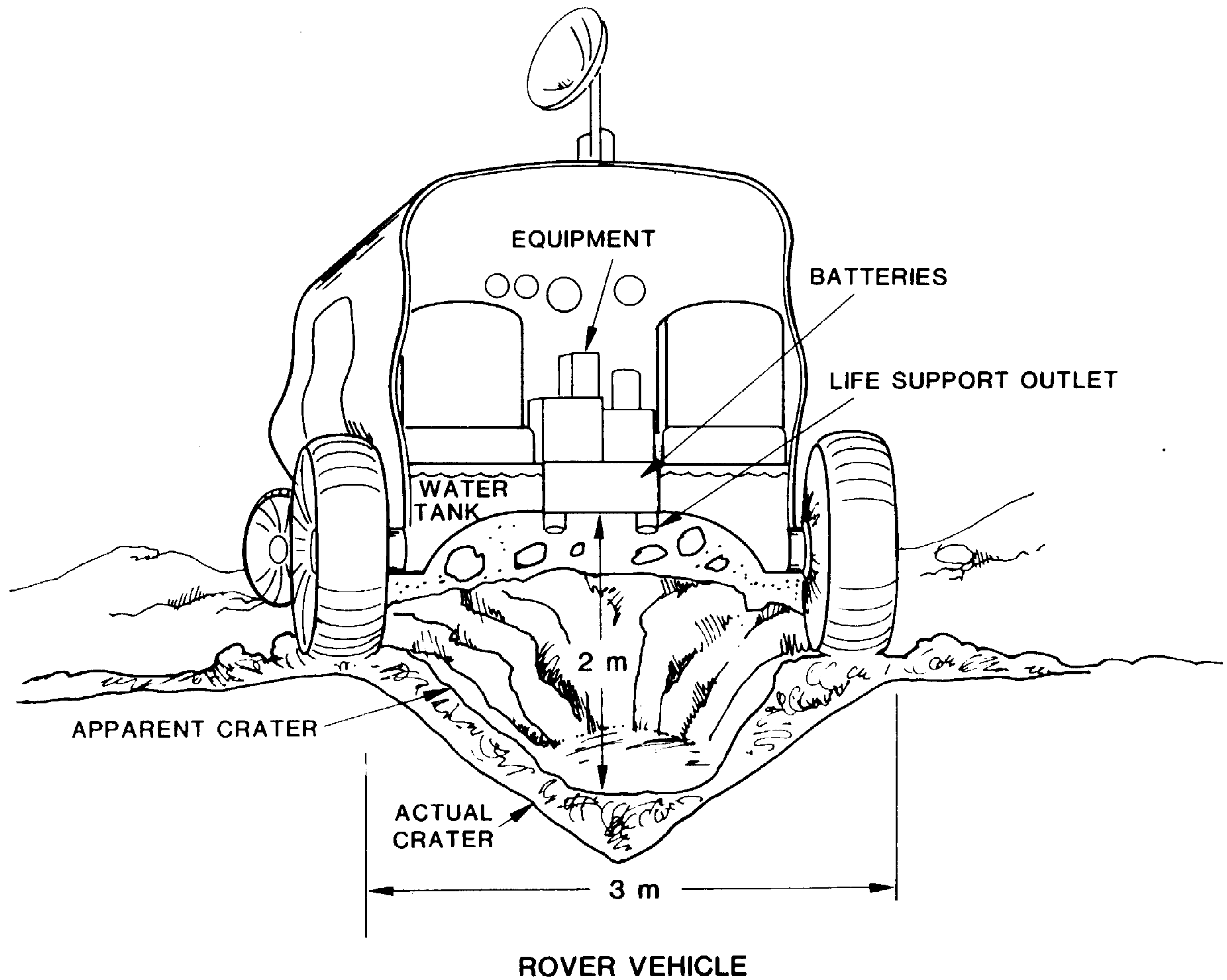
gm/cm^2 for long duration exposure to cosmic rays are required. This translates to rock thicknesses of about 20cm and 200cm, respectively. We further assume that 2 Pi shielding is necessary and that we cannot expect more than about one hour warning before the effects of an intense solar flare would be felt on the Martian surface. Finally, we assume an operational scenario consisting of five manned landings involving extensive surface explorations using rover vehicles and leading to a permanent manned outpost (the "Columbus Base" scenario, [1]).

EXPLOSIVE EXCAVATION STRATEGIES

Explosives are a safe, efficient, and practical means of cratering and tunnel driving to provide protective shelters as well as being useful for scientific purposes. Four areas for which explosives are useful on the manned Mars mission are (1) remote shelter construction to protect the rover vehicle crew from intense solar flare protons, (2) construction of the main base shelter such as a tunnel or a rock-covered module placed in an explosively formed trench, (3) providing a tunnel or crater to bury a main base nuclear power reactor for shielding, and (4) an energy source for active seismic experiments as part of the scientific exploration of Mars.

The rover vehicles should be configured so that the floors contain materials such as batteries, water, wastes, and other equipment useful for shielding the crew (Figure 1). During rover traverses away from main base shelters, a remote shelter large enough for two people could be constructed in less than an hour by producing a trench in the Martian surface using explosives, driving the rover with shielding in the floor over the trench, and then "sandbagging" around the edges of the rover with thrown-out debris for side protection. The crew (in EVA suits) then takes shelter during the intense phases of the solar flare (2-10 hours). Life support and communication outlets in the rover floor could be provided for the crew to plug into for increased comfort during their stay. At the main base, a permanent shelter could be constructed by tunneling into a nearby rock face using drilling and blasting methods. A more useful shelter could be constructed from this simple tunnel by either placing an inflatable envelope within the tunnel or by closing the entrance with a bulkhead and airlock, sealing the tunnel walls with insulating foam, and then pressurizing the inclosed volume. In our

FIGURE 1. CONCEPT FOR A REMOTE SOLAR FLARE SHELTER.



opinion, the latter approach is probably the best method of producing large habitable volumes for a permanent manned base. Alternatively, or in addition, appropriately sized modules brought from Earth could be placed in explosively produced trenches and covered with rock and dirt by explosive casting techniques. These same techniques could be used to bury a nuclear reactor to shield against its radiation.

EXPLOSIVES AND INITIATORS

The explosives will of necessity need to be transported to Mars from Earth, at least for the first manned mission, since it is not certain whether all the ingredients needed to manufacture explosives on Mars are present in usable quantities. (If nitrate salts are found or if nitrogen can be extracted from the small amount present in the Martian atmosphere, then in situ explosives production is possible and ultimately desirable.) Some requirements of explosives to be used on Mars are (1) insensitivity to detonation from impact over a wide range of impact velocities, (2) safe to transport, store, and handle, (3) availability in convenient sizes and shapes, (4) chemically and mechanically stable over large temperature and pressure ranges, (5) high energy content per unit volume to effectively blast craters, trenches, and tunnels, (6) detonatable in 3 to 5m lengths and 25 to 50mm diameters, (7) detonatable at very low temperatures in a safe and reliable manner, and (8) easily loaded in uneven boreholes. Explosives (military and commercial) vary greatly in energy content, density, sensitivity of initiation, and detonation pressure. Table 1 is a list of a few representative military and commercial explosives in common use for munitions and blasting. Explosive 9502, composed of 5% Kel-F plastic binder and 95% TATB* (item 2 in Table 1), is a high-energy, insensitive military explosive. The other two military explosives, PETN and TNT, are much more sensitive. The next three items in Table 1 are commercial blasting agents that are insensitive to initiation by impact and are less energetic than the military explosives. Explosive 9502 may be a good choice to perform the excavation on Mars, but no data exists on it's blasting capability. The commercial explosives are used extensively for Earth-bound excavations. Item 7 in Table 1 represents a speculative suggestion that hydrogen peroxide might

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* 1,3,5 triamino-2,4,6-trinitrobenzene

TABLE 1

COMPARATIVE PROPERTIES FOR SEVERAL MILITARY AND COMMERCIAL EXPLOSIVES

Explosive	Density (g/cm3)	Detonation Velocity (m/s)	Energy (cal/g)	Detonation Pressure (GPa)
1. PETN	1.7	8800	1510	35.0
2. TATB (9502)	1.89	7600	1200	30.0
3. TNT	1.65	7000	1090	19.0
4. Atlas 840 Powermax	1.34	6000	940	12.0
5. IREGEL 1175C Emulsion	1.25	5000	890	8.0
6. ANFO	0.85	3500	900	3.0
7. Hydrogen Peroxide	1.45	7000	690	7.0

have some attractive features as an explosive for use on Mars. It is less energetic than the military explosives listed but is comparable to many commercial blasting agents. It's main attraction is that it could very likely be easily manufactured on Mars from indigeneous water. Poured as a liquid into irregular boreholes, it would quickly freeze and couple well to the rock and detonator. More information on its explosive properties under Martian conditions is needed before it can be further evaluated.

Initiators or detonators need the following requirements: (1) safe to transport, store, and handle; (2) storable separate from the explosive charges; (3) easily and securely attachable to the charge; (4) sufficiently energetic to detonate insensitive explosives such as 9502 or a blasting agent through a booster arrangement; and (5) must be reliable at very low temperatures, stable chemically and mechanically, and very easy to connect and use in a shot situation. Since insensitive explosives will likely be used for rock removal, a booster explosive will be required between the initiators and the main charges for reliable detonation to take place. Any booster charge used needs to have similar reliability, stability, and ease of use requirements as the detonators. There are several types of electric detonators - standard blasting caps, exploding bridgewire (EBW), and minislappers. Another type of initiator is the nonelectric cap, widely used in the blasting industry. The difference between the electric detonator types is the application of the electrical energy. For EBW or minislapper systems, a large energy density is applied to a small diameter wire (EBW) or foil (minislapper) in less than a microsecond causing the detonation of a primary explosive which in turn detonates the booster charge. The actuation energy for these detonators is 1-3J at several thousand volts. The standard blasting cap is a low energy device that also has a bridgewire, but is not exploded. Instead the wire is heated to the ignition temperature of the primary explosive in contact with the bridgewire. The firing conditions are approximately 5A and 450V. The electric detonators can be fired from small portable firesets. Blasting caps are produced with a large variety of time delays while EBWs and minislappers are instantaneous and require any delays to be built into the firing circuits. The nonelectric system consists of a nonelectric detonator connected to a plastic tube coated with PETN powder on the inside surface that is connected to a detonator. A "starter" (safety fuse or electric detonator) ignites the powder causing the detonator to fire after the burn front propagates the length of the tube. Various delays are also available. The EBW-minislapper systems or the Nonel (trade name for Nitro-Nobel nonelectric detonator system) initiators are very safe and convenient to use, even with military explosives. Many of the systems discussed above (e.g., TATB and EBWs) have been used reliably at temperatures down to -50C but would need to be tested at still lower temperatures for Mars use.

METHODS

Cratering

For blasting a crater or trench, the following steps are necessary: (1) select a depth-of-burial based on the general type of material to be blasted and the blasting application; (2) drill the borehole(s) to the selected depth; (3) load the initiation device, booster charge, and explosive to the desired depth-of-burial; (4) connect the detonator/booster assembly to the fire set; and (5) fire the shot(s) after retreating from the explosive site a distance sufficient to prevent damage to people and equipment from fly rock (in the Mars 1/3 gravity, rocks with the same initial velocity will fly three times farther than on Earth). The cratering shots for the remote shelter must be designed to throw as much rock as possible to eliminate the need to muck the crater.

Since the remote shelter is basically a conically shaped crater or string of connected craters (trench) with the rover over it, the parameters for the blast must be chosen to provide a crater with an aspect ratio (crater diameter to crater depth) on the order of 2:1 or less in order to maintain adequate head room under the rover. Fig. 1 is an illustration of the shelter concept. Based on previous cratering test data [2,3], a 100kg charge in alluvium or a 150kg charge in solid rock buried at a depth of 2m will produce an apparent crater 2.5m deep and 5.0m diameter. In 1/3 gravity, this apparent crater depth [4] will likely be greater on Mars than on Earth for the same surface material and explosive loading. Cratering from charges placed on the alluvium or rock surfaces is very inefficient! Even shallow burial of the charge greatly enhances the crater volume. A preliminary study of cratering (on Earth at least) indicates that an adequate shelter remote from the base could be constructed quickly using explosives buried at 2 to 3m depth. Additional factors that need to be investigated are the drilling equipment, methods for quick set-up, and reliable operation at very low temperatures.

Tunneling

At the main base, tunneling by the drill, blast, and muck technique [5] appears to be an efficient means to construct a shelter. The tunnel driving methods are highly developed and seem adaptable to tunneling in Mars rocks. We have chosen a tunnel size of 2.1m square and 10m long (Figure 2) as adequate for each of the landing site bases for the first three manned missions. This size requires the removal of $44m^3$ of rock. Using the industry's experience in blasting on Earth, the powder factor, PF (mass of explosive needed to remove one cubic meter of rock), can be calculated from the empirical relation [5]

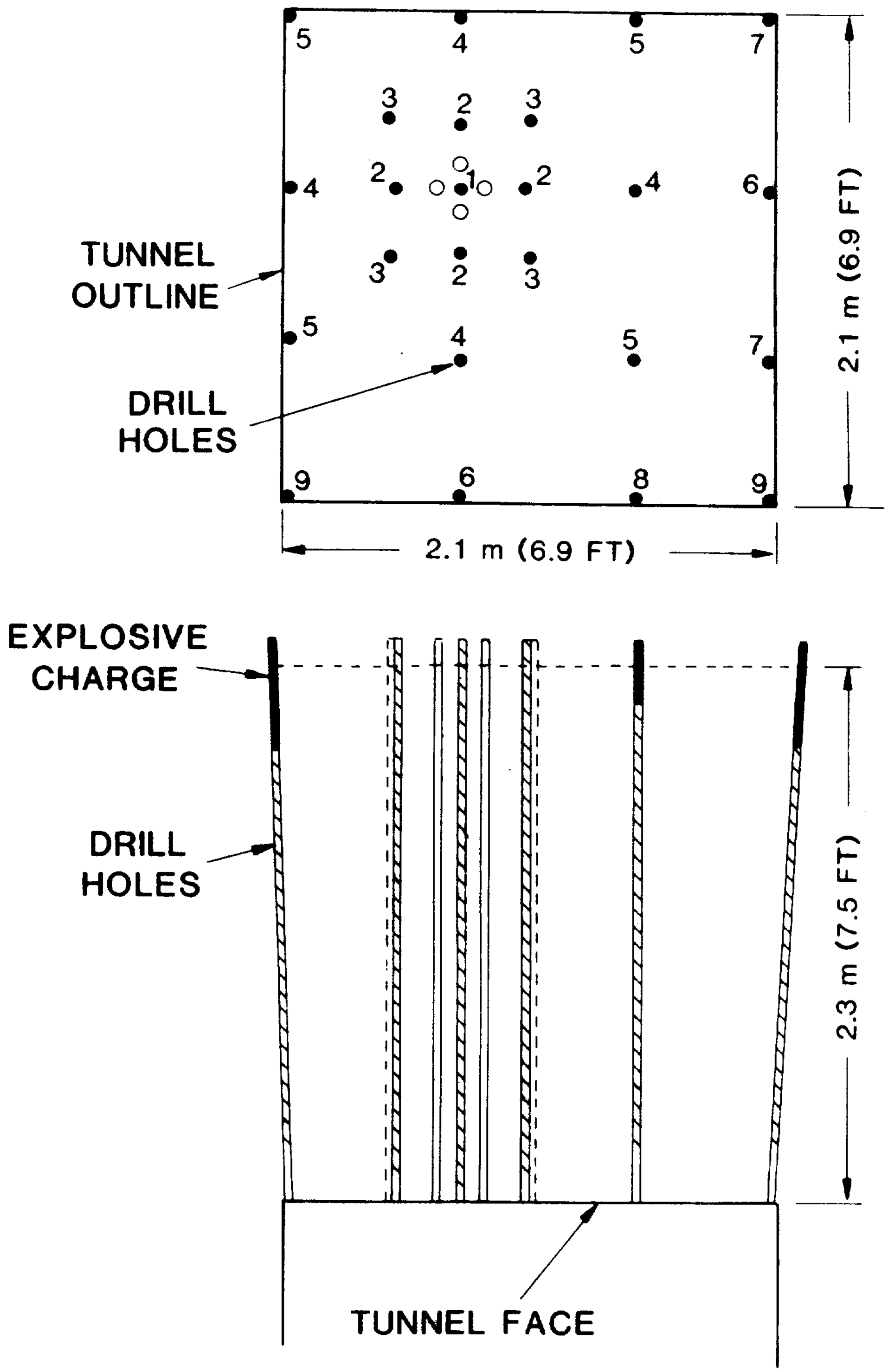
$$PF = 14/s + 0.8$$

where s = area of the tunnel face. PF for a 2.1m square tunnel is 4 kg/m^3. Hence, 175kg of explosive is necessary to remove the required volume of rock. To maximize the usage of this explosive, several tunnel driving parameters need to be included in a predetermined blast plan such as the drill hole pattern at the tunnel face, drill hole diameter, strength properties of the rock, degree of explosive packing in the holes, and the ignition sequence of the round. An example of a drilling pattern for a smooth wall tunnel with a $4.4m^2$ face is given in Figure 2. This blast pattern produced an advance of 2.3m per round, so a 10m long tunnel can be blasted with four rounds. Muck removal after each round can be accomplished with a dragline powered by the rover vehicle, electric power winch, or by hand. A crude time estimate for each round is 16 hours, including drilling, loading, mucking, and equipment setup and teardown. We believe this construction time could be substantially reduced by utilization of a specially designed tunneling machine that would combine drilling, blasting, and muck removal in a nearly continuous, semi-automated operation (e.g., ref. [6]).

Trenching and Casting

Trenching to form a protective shelter at the main base is an exten-

FIGURE 2. AN EXAMPLE OF A DRILLING PATTERN FOR A SINGLE EXPLOSIVE ROUND FOR A 4.4m2 TUNNEL FACE (REF. [6]). THERE ARE 28 DRILL HOLES EACH 32mm IN DIAMETER.



sion of the cratering process discussed above. A v-shaped trench 2.5m deep, 5m wide, and 15m long could be produced in soil by sequentially firing six row charges spaced 2.5m apart [2]. The charge burial depth is 2m and each charge is 30kg for a total mass of 180kg. In rock, the charge mass is approximately 225kg to form a similar size trench. The shelter module is then placed in the trench and covered by either using machinery or using explosives to cast [7] the soil and rocks from a nearby bench. Figure 3 is a schematic diagram of the technique.

Approximately $65m^3$ of material is needed to provide a 2m thickness of material over a module that is 2m diameter and 10m long. Assuming a PF of $3kg/m^3$ and assuming 25% of the material is lost due to excessive flyrock and dispersion, nearly 250kg of explosive is needed to produce the cast material to cover the module. A total mass of 450kg of explosive appears to be sufficient to bury the module in a trench. If a natural ravine near to a bench or cliff could be found in which to place the module, then the explosive usage could be reduced by one half.

SAFETY ISSUES

The development of modern insensitive high explosives has largely removed the danger of transport and use of these materials. Reference [8] describes the many tests that are performed to characterize the sensitivity of explosives and assure their safe use. In our opinion, an explosive based on TATB is capable of surviving a launch pad explosion and fire without detonation. To illustrate this, Figure 4 shows a missile containing explosive 9502 impacting a target at high velocity without detonation. The only event that we can conceive of on a manned Mars mission that could unintentionally detonate an explosive like 9502 is the impact of a gram-size meteoroid traveling at several tens of meters per second. This (unlikely) eventuality could be rather easily guarded against by storing the explosive inside a container with shock absorbing walls such as metal-epoxy honeycomb, double-wall, or similar material. A layer about 10cm thick would be adequate to stop a 1 cm diameter meteoroid without propagating a shock wave into the explosive. Normal safety practice would result in separate storage of detonators and initiators in similar containers.

FIGURE 3. SCHEMATIC ILLUSTRATING THE CONSTRUCTION OF A PROTECTIVE SHELTER BY USING EXPLOSIVES TO PRODUCE A TRENCH AND THEN COVER A PRE-FABRICATED MODULE.

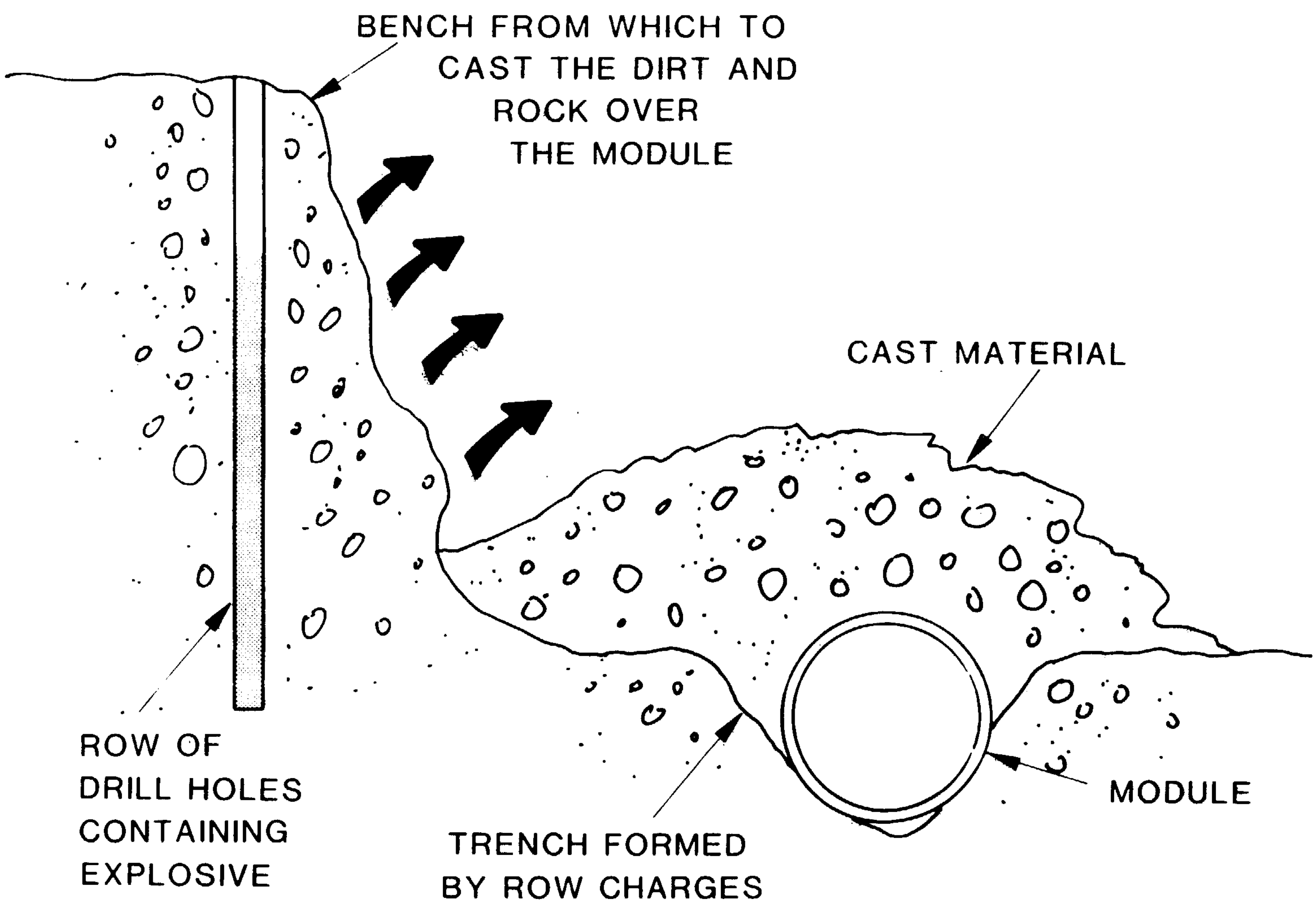
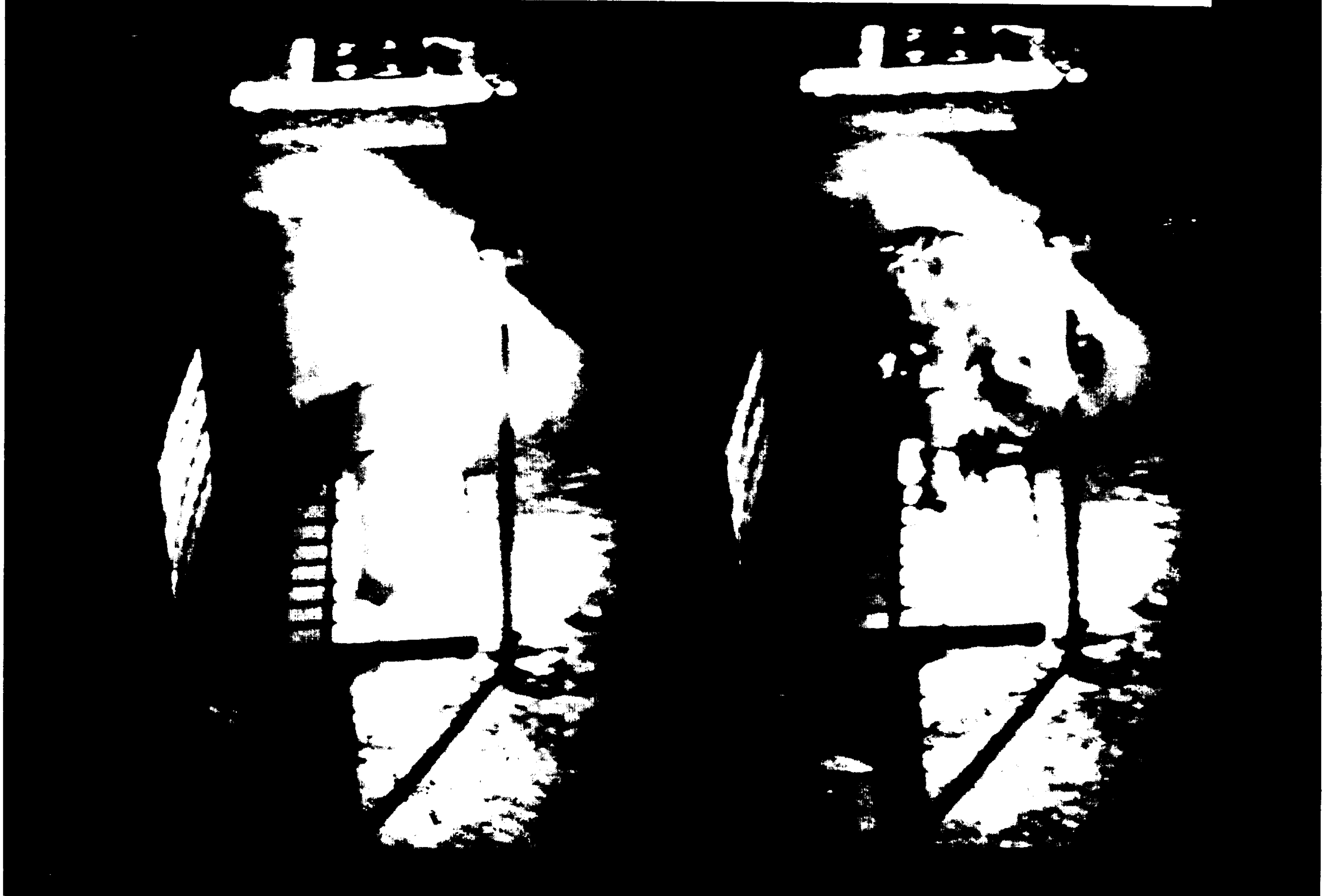


FIGURE 4. IMPACT AT 150m/s OF A TEST MISSILE CONTAINING INSENSITIVE HIGH EXPLOSIVE 9502. THE EXPLOSIVE DISINTEGRATES DUE TO IMPACT BUT DOES NOT DETONATE. (LOS ALAMOS NATIONAL LABORATORY).



CONCLUSIONS

The applications for the use of explosives on Mars are extensive. There is probably no better source of stored energy in a small volume and mass than explosives. They are safe, easily handled in the field, and their usage requires very little specialized apparatus. Based on Earth-bound blasting operations, much of the design and planning of the particular blast applications on Mars could be accomplished in the mission planning phases and even tested in rocks and soils simulating materials expected on the Martian surface. Blasting information needed by the crew includes the type of explosive, initiators for the charges, drilling patterns, depth-of-burial of charges for cratering, powder factors, drill hole diameters, spacings for row charges, and delay timings. Once on the surface of Mars, a cratering test in soil and hard rock should be conducted using the 100kg of explosive designated in Table 2 for testing. This proof test would be conducted on the first manned mission to validate the blasting designs conceived during the planning stages. Parameters to be evaluated in these tests are powder factor, crater size and shape, and effective strengths of Martian rock materials. This information would then be used to produce final designs for cratering, trenching, and tunnel driving.

TABLE 2

EXPLOSIVE (TNT) ESTIMATES FOR FIVE MANNED LANDINGS ON MARS

	Site A	Site B	Site C	Initial[a] Base	Columbus[b] Base
* Remote Shelter					
Hard Rock	150kg	150kg	150kg	150kg	300kg
Soil	100kg	100kg	100kg	100kg	200kg
* Base Station					
Tunnel (rock)	175kg[c]	175kg	175kg	300kg	500kg
Trench (rock)	225kg[d]	225kg	225kg	225kg	550kg
(soil)	180kg	180kg	180kg	180kg	360kg
Casting (rock)	250kg	250kg	250kg	250kg	500kg
Reactor (rock)				150kge	
(soil)				100kg	
*Seismic	100kg	100kg	100kg	200kg	500kg
*Testing (rock)	50kg				
(soil)	50kg				
MISSION TOTALS	525-825kg	425-725kg	425-725kg	800-975kg	1300-1850kg

(a) - Enlargement of Site A, B, or C by a factor of 2.

(b) - Enlargement of initial base by a factor of 2.

(c) - Initially, 2.1m square by 10m long; extended in length and diameter for permanent base.

(d) - 2m deep by 10m long trench for placement of module that is subsequently covered by casting.

(e) - Crater 2m deep by 8m diameter.

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