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(NASA-TM-89321-Vol-2) MANNED MARS MISSION. WORKING GROUP PAPERS, VOLUME 2, SECTION 5 - APPENDIX (NASA) 559 p CSCL 22A

N87-17760 THRU N87-17808

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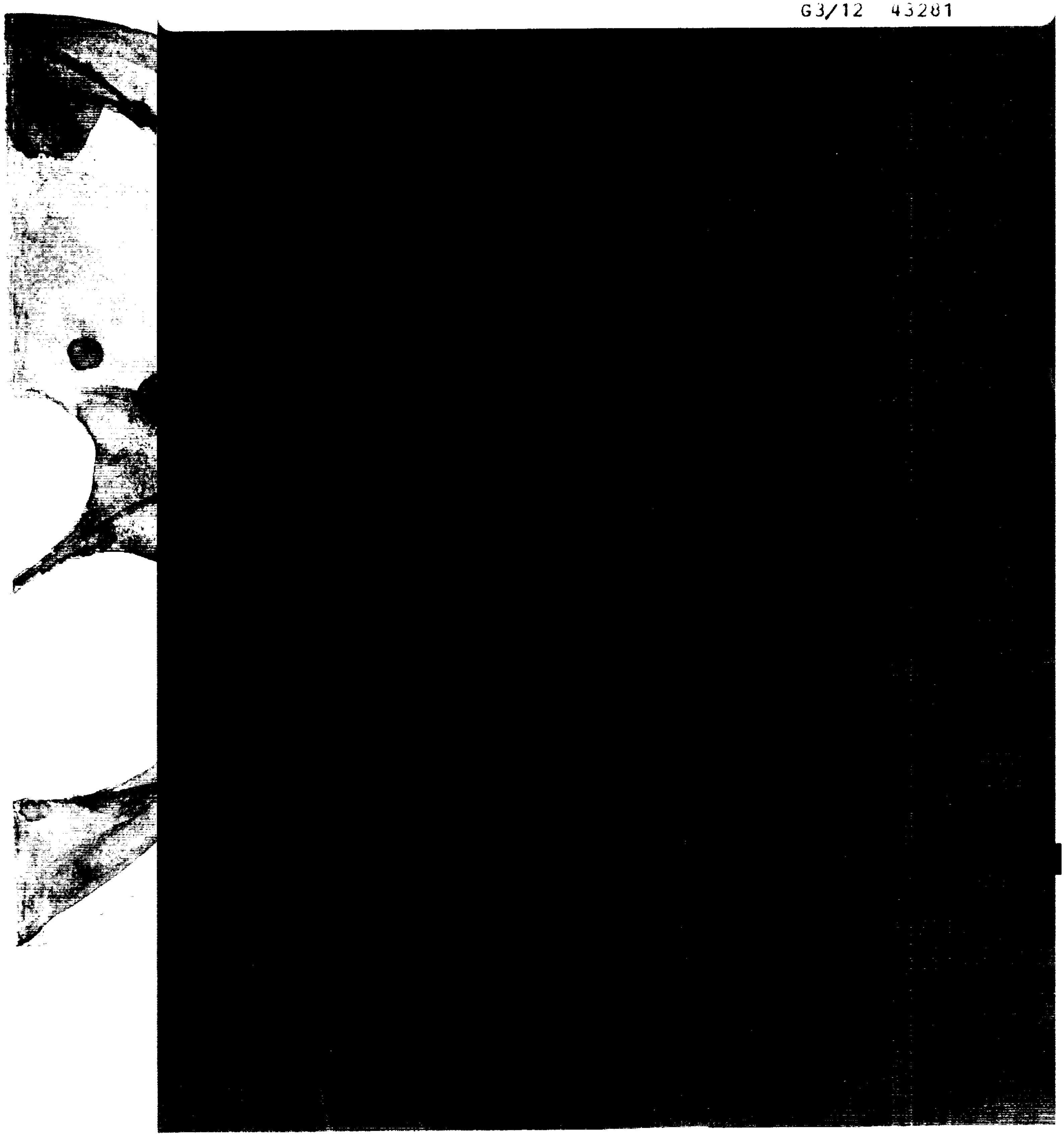

MANNED MARS MISSIONS
WORKING GROUP PAPERS

A WORKSHOP AT

MARSHALL SPACE FLIGHT CENTER

HUNTSVILLE, ALABAMA

VOLUME II OF II
SECTION V - APPENDIX

JUNE 10-14, 1985

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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.

S E C T I O N   V

SCIENCE INVESTIGATIONS AND ISSUES

ASTRONOMY ON A MANNED MARS MISSION

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ABSTRACT

Three extra-solar-system astronomical experiments aboard a manned Mars mission are proposed. First, a modest, 50-cm aperture optical-uv-IR telescope (or pair used as an interferometer) coupled with the Mars-Sun baseline would increase the number (by a factor of 3.4) and a volume of stars with accurately measured distances via stellar parallax and, therefore, greatly improve upon the cosmic distance scale; the darker sky at Mars would also provide nearly a full astronomical magnitude deeper images of distant and low brightness objects (limited by zodiacal light). Second, a gamma-ray burst detector coupled with similar detectors in other parts of the solar system will be used to reduce the position error boxes and to study the nature of these energetic sources. Third, the long baselines on a Mars mission radio interferometer will provide a view of the radio universe at unprecedented resolution, 4×10^{-9} arcsec at 1-mm wavelength, which can potentially resolve the "engine" in nearby active galaxies. Each of these experiments is relatively inexpensive, taking advantage of the human presence for operation and maintainance, and the long Earth to Mars baselines.

INTRODUCTION

The duration of a manned mission to Mars may be anywhere from one to three years. The majority of this time will be spend in transit, going to and returning from the planet. What scientific activities will occupy the attentions of the crew during the long voyage? One can envision several useful and possibly unique astronomical experiments that could be performed during the flight with minimal additional cost to the mission.

The trip to Mars should not be viewed simply as a long wait before the commencement of productive scientific endeavors. To the contrary, a manned Mars mission could present us with an unprecedented opportunity to study the cosmos, free of the confinement of the Earth-Moon system. The spacecraft could serve as an interplanetary platform to house important scientific experiments not currently possible in our local environment,. These experiments could provide data for scientists back on Earth as well as challenge the scientists and engineers aboard the spacecraft.

We propose three criteria for selecting astronomical experiments. First, they must be in some sense superior to what could be performed on the Earth, in Earth orbit, or on a future lunar base. Therefore, the Mars mission scientific station could genuinely add to our knowledge of the Universe. Second, the experiments should represent a minimal additional cost to the mission. A significant cost savings can be realized over that of completely automated space probes by using the crew to operate and maintain the telescope instrumentation. Third, where possible, they should allow (or even require) human judgement and possible human intervention to maximize the available science and to take advantage of unexpected research opportunities in flight. The experiments could be more complex and ambitious than those on an unmanned probe since human beings will be present to operate and adapt the equipment to the environment or unanticipated changes in the experimental goals.

OPTICAL-UV-IR OBSERVATIONS

The Hubble Space Telescope is the first major optical-uv-near-IR spaceborne observatory. The primary mirror has a diameter of 2.4-meters and the principle imaging detector (wide-field camera) operates over the wavelength range of 1155 nm to 1100 nm. Such a wavelength range is unprecendented because of the previous atmospheric constraints of ground-based optical telescopes. This facility is the first major optical astronomical telescope that will reach the diffraction limit of resolution. The wide-field camera and the faint object camera will have point response functions for as little as 0.04 arcseconds (FWHM) in the middle of the optical band in the longest focal ratio mode; the faint object camera can achieve a resolution of 0.007 arcsec in very narrow fields.

One might envision a more modest aperture telescope, say 50-cm in diameter, operating over about the same wavelength range, for astronomical observations on the Mars mission. At a wavelength of 200 nm, this telescope has an impressive diffraction-limited resolution of 0.08 arcsec. Astrometric centroid positions of stellar objects can be measured nearly a factor of ten more accurately. A pair of such telescopes, operating as an interferometer, could achieve astrometric position accuracy similar to that of the star tracking telescopes on the Space Telescope (ST), namely 0.002 arcsec. The cost of this telescope (or pair) would be quite small in comparison to ST. Once again, the vast majority of the $\$10^9$ cost of ST is due to the required unmanned automation and great weight of this free-flying observatory. On the other hand, a 50-cm telescope is cheap to manufacture and very inexpensive to operate on a manned mission since astronauts will point the telescope, change and maintain detectors, record the data, and provide in-flight calibration of this data. With the new lightweight mirror designs, launch costs will also be minimal.

What purposes will be served by having a moderate-sized optical telescope aboard the spacecraft? First, it represents the only astronomical experiment that we will propose which allows astronauts to visually inspect the fields at which they are pointing. The astronaut-observers will be able to visually roam across a magnificently dark sky. In the vicinity of Mars, the flux of energy from the Sun is decreased by a factor of about 2.3, which corresponds to nearly one astronomical magnitude, in comparison to that near the Earth. Currently, the Space Telescope is projected to have a sensitivity of about 28^m in the visual band for point sources, which corresponds to a flux of 2.2×10^{-19} Watts m^{-2}. This limit is due primarily to the readout noise of the charge-coupled device detector in the wide-field camera. If the detector technology continues to improve at the rapid pace of the past decade, then we will quickly become limited by the sky brightness in space (zodiacal light). Even the modest aperture Mars telescope offers an advance in terms of reduced sky background over the terrestrial environment. In addition, the on-board scientists will have some degree of freedom to select their own observations such as monitoring galaxies for supernova explosions or accurately tracking stellar positions and/or velocities for signs of perturbations by other planetary systems.

Second, such a telescope will be needed to monitor the Sun for potentially lethal solar flare activity. This will be one of the most important safety features of the mission (see Hathaway in this volume).

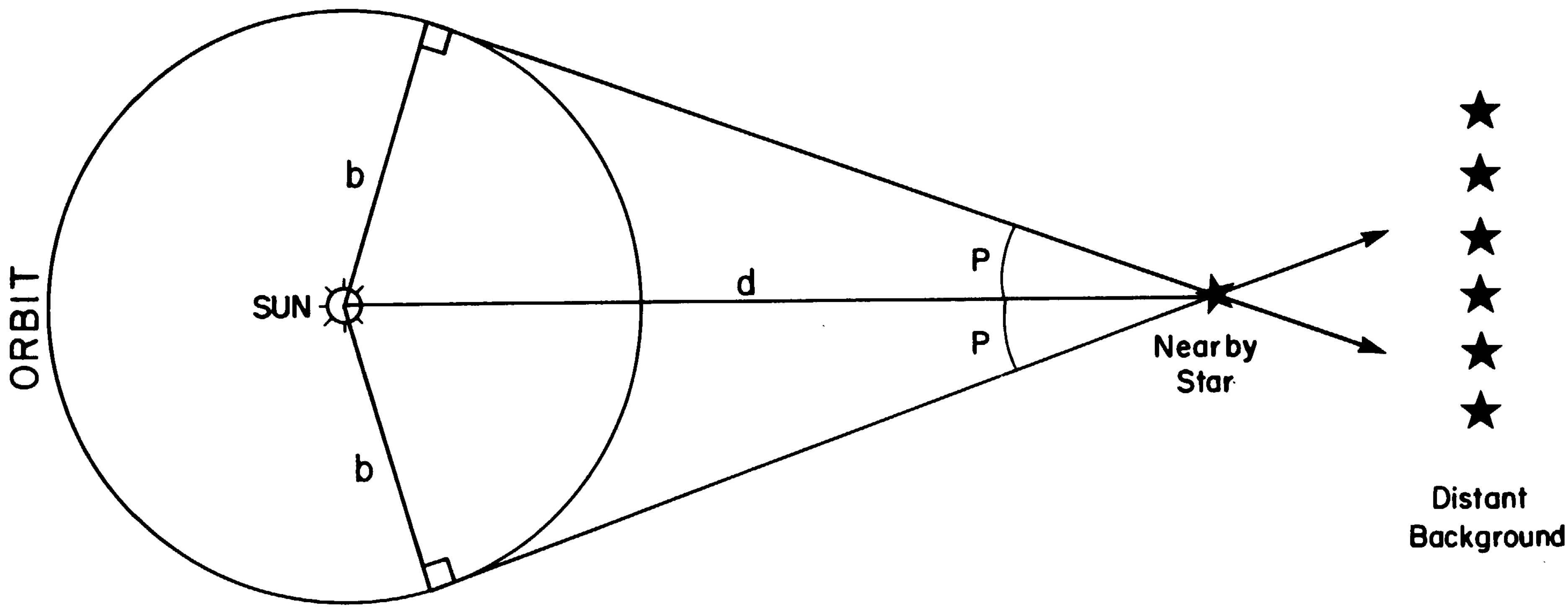
Third, important observations utilizing the longer Earth-Mars or Mars-Sun baselines could also be performed. One of the most important involves stellar parallax studies of more distant stars. Our knowledge of the size and future evolution of the Universe hinges strongly on how accurately we know the distance to the stars in our local neighborhood of the Milky Way Galaxy. This provides the foundation upon which the cosmic distance ladder is constructed. The classical technique used to determine distances to nearby stars, stellar parallax, is illustrated in Figure 1. The longer the baseline and the smaller the seeing disk for stars, the further away one can directly measure the distance. The smallest parallax angle, p, which can be measured from the ground is about 0.05 arcsec. This corresponds to a maximum distance of about 65 lightyears. This situation has remained nearly constant since the middle of the 19th century. The first major advance will come with the launch of Hipparchus satellite by the European Space Agency. This telescope will be capable of measuring stellar parallaxes to ± 0.002 arcsec, similar to the Space Telescope. An optical telescope in orbit about Mars will increase the baseline by about a factor of 1.5. If we conservatively assume the same parallax measurement uncertainty (presently limited by instrumental jitter), then the maximum distance becomes 2500 ly. The number of stars accessible for parallax study increases by a factor of 3.4 over that possible in LEO. This will represent a major advance in calibrating the extragalactic distance scale. (The above assumes a relatively clean environment near the telescope/spacecraft.)

<u>HARD X-RAY AND GAMMA-RAY BURST EXPERIMENT</u>

During the early and middle 1970's, enigmatic, extraterrestrial sources of hard x-ray (>10 keV) and gamma-ray (100 keV to 1 MeV) radiation were discovered by a set of Vela satellites which were launched by the Department of Defense. These sources are characterized by a brief (0.01 to 80 seconds) burst of emission which rises hundreds to thousands of times higher than the quiescent background. The distribution of these sources of energetic photons suggest that they are confined primarily to our Galaxy.

Our understanding of these sources has remained poor during the last decade and a half. The best theoretical models suggest that the gamma-

FIGURE 1 - DISTANCE DETERMINATIONS USING STELLAR PARALLAX



In the simplest case, an image of a relatively nearby star is recorded on opposite sides of the Sun during the orbit of a planet or spacecraft. In comparing the two images, the foreground star will appear to move with respect to the distant stellar background. The distance to the foreground star is simply $d = b/p$ where b is the sun-planet (or spacecraft) baseline, and p is the parallax angle (radians). The Hipparchus satellite will be capable of measuring p to within 0.002 arcsec for stars of visual magnitude <17 ray bursts originate from the vicinities of neutron stars. The emission may be produced by matter accreting onto the surfaces of the compact objects or may be due to magnetic field line reconnection resulting from instabilities in the magnetic field geometry. These models are consistent with the size estimates from the burst durations and from the one optical identification made with a supernova remnant in the Clouds of Magellan (Neutron stars are believed to be superdense remnant cores of massive stars which have exploded spectacularly in supernova; e.g., the Crab pulsar).

Information on the gamma-ray bursters has remained sparse for three reasons. First, the hard x-ray and gamma radiation is absorbed by the Earth's atmosphere. Therefore, all observations must be performed from space. Second, the sensitivity of the present detectors is relatively poor, partly because they must also cover a large area of sky. Third, there is at present no method for imaging these very high energy photons. The detection of a strong gamma-ray source with a single detector will typically have a position uncertainty of 1^{o}-2^{o}. With these error boxes, it is impossible to make optical identifications of the sources of the radiation.

However, it is possible to pinpoint the location of these sources using several detectors located at different positions in the solar system. By comparing the arrival times of the bursts in the different detectors, one can determine the direction of the incoming photons. The longer the baselines and the larger the numbers of spacecraft, the more accurately the source position can be determined. The Soviet Konus experiment using eleven separate detectors spread throughout the inner solar system, including four on the Venus Venera probes, was able to reduce the position error box of the March 5, 1979 event so that an optical ID with the supernova remnant in the Magellanic Clouds became possible.

We often find that there are too few spacecraft available to perform these coincidence experiments or even to confirm a gamma-ray burst. The situation will improve with the 1988 launch of the Gamma-Ray Observatory. The detectors will be 10-100 times more sensitive than those on previous spacecraft and will cover a wider range of energies.

A gamma-ray burst detector on the Mars mission would add significantly to our understanding of these sources. The substantially longer baseline could reduce position uncertainties by factors of 2 to 10 when coupled with other detectors in the inner solar system. The sensitivity will certainly be much greater than is currently possible, so that accurate measurements of line radiation within the burst can be performed. There have been suggestions of lines in the range of 40 to 70 keV in the Konus data; the most popular interpretation involves cyclotron radiation.

In any event, if such a detector were on board the manned Mars mission, it would almost certainly add to the interpretation of the physics of these energetic sources. Again, the human presence will reduce the cost and increase the flexibility of these relatively simple detectors.

A MARS MISSION RADIO INTERFEROMETER

Because of the relatively long wavelengths (millimeters to meters), single antenna radio telescopes have poor resolutions in comparison to even modest sized (e.g., 6-inch diameter) ground-based optical telescopes. For example, the largest single dish telescope in the world, the Arecibo radio telescope in Puerto Rico (1000-ft diameter), has a FWHM beam of about 2.2 arcmin at a wavelength of 20-cm. This is comparable to the resolving power of the human eye or 180 times worst than the optical telescope noted above.

A multiple-antenna radio interferometer, on the other hand, can achieve resolutions superior to that of the best ground-based optical telescopes. Each antenna baseline samples one Fourier component of the radio source brightness distribution at a given instant. It is desireable to sample as many of these components (i.e., many baselines at many different position angles) as possible for proper reconstruction of the radio source structure. The goal here is to synthesize an aperture using individual antennas. Martin Ryle was the first to recognize that the rotation of the Earth could greatly assist in the aperture synthesis by sampling many Fourier components per baseline as the source is tracked across the sky. This basic idea is illustrated in Figure 2.

The most sophisticated example of this synthesis technique is the Very Large Array (VLA) radio interferometer located in west central New

APERTURE SYNTHESIS

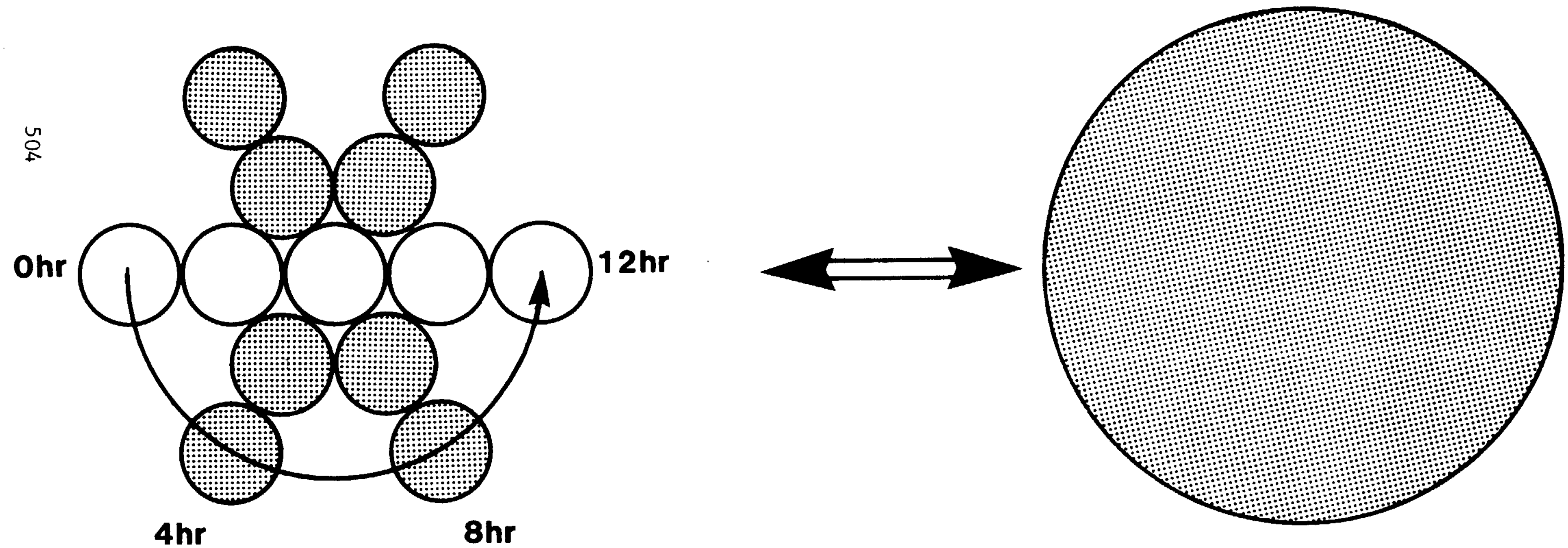


FIGURE 2 - THE PRINCIPLE BEHIND EARTH ROTATION APERTURE SYNTHESIS IN RADIO INTERFEROMETRY

Imagine that an observer is stationed above the North Pole of the Earth looking down upon a linear array of 5 antennas. As the Earth rotates, the line sweeps out portions of a filled aperture. In 12 hours, the line has synthesized a circular aperture with diameter equal to the maximum baseline between the outmost antennas. This is equivalent (in terms of resolution) to observing a radio source with a single very large antenna.

Mexico and operated by the National Radio Astronomy Observatory. It consists of 27 individual antennas distributed in a Y-configuration. The resolution of this telescope is 0.1 arcsec at 2-cm wavelength in its longest baseline mode (maximum baseline of 35-km).

During the past decade, this synthesis technique has been applied to even longer baselines, stretching across the U.S. and in Europe. Very Long Baseline Interferometry (VLBI) differs from the VLA in that the data is recorded on Tape at individual antennas (using accurate time Marks generated from hydrogen maser clocks) and later correlated with data from other telescope stations. Using hybrid mapping techniques to partially recover phase information in the data, VLBI maps of radio sources are now being made with dyanamic ranges rivaling those of short-integration VLA maps, but at a resolution of less than a milliarcsecond.

There are proposals to extend the radio interferometry baselines to radio telescopes in Earth orbit and on the Moon. A 10-meter radio antenna will be deployed from the cargo bay of the Space Shuttle to test the feasibility of space VLBI during the next several years. A joint European-American consortium has proposed a free-flying 15-meter VLBI radio antenna called Quasat (Quasar satellite) to be launched in the early to middle 1990's. Finally, I have suggested that a relatively simple antenna built as part of a permanent colony on the Moon could effectively serve as a long baseline component of a spaceborne VLBI network. In this case, the orbit of the Moon around the Earth would aid in the aperture synthesis. The resolution of such a telescope will be 30 microarcsec at 6-cm wavelength and improve as a direct proportionality with wavelength at shorter wavelengths.

It is an exciting possibility to extend the baselines even further by carrying a 15-meter class radio antenna on the manned Mars mission. Such a deployable antenna is lightweight and could be folded into the cargo bay of the spacecraft. At the maximum Earth-Mars separation of 3.8×10^8 km, the diffraction limited resolution at a given wavelength will be a factor of 1000 times better than that of the Moon-Earth Radio Interferometer and a factor of 10^7 times that of the VLA!

However, it is unlikely that we will be able to achieve the full diffraction limit at centimeter wavelengths due to the scattering of radio waves by electrons in the turbulent fluctuations at high galactic

TABLE 1

THEORETICAL RESOLUTIONS OF A MARS MISSION INTERFEROMETER*

OBJECT	DESCRIPTION	DISTANCE (LY)	LINEAR RESOLUTION (KM)	COMMENTS
Proxima Centauri	Nearest Star	4.3	0.8	Resolve active regions and star-spots for stars in local neighboor
Sagittarius A	Galactic	32,000	6000	Resolve a 10^4 M black hole+
M31	Andromeda	$2.2x10^6$	$4x10^5$	Resolve individual stars in Local Group Galaxies
Centaurus A	Nearest Active Galaxy	$1.6x10^7$	$3x10^6$	Resolve Accretion Disk and 10^6 M black hole.+
3C273	Quasar	$1.7x10^9$	$3x10^8$	Resolve Accretion Disk and 10^8 M black hole.+

*These calculations assume a resolution of 4 nanoarcsec at 1-mm wavelength. Actual detection of source components on these scal sizes will depend upon the sensitivity of the radio telescope and the strength of the components.

+Resolve a Schwarzschild radius.

and extragalactic radio sources using the Mars Mission Radio Interferometer is illustrated. The potential science is indeed impressive.

The radio telescope will function during the entire trip to and from Mars as well as in orbit. The motion of the spacecraft with respect to the Earth will provide the aperture synthesis necessary to crudely map radio sources at very high resolution. The theoretically expected fraction of the Fourier transform plane (i.e., the aperture) which will be sampled for 1999 Mars opposition profile is shown in Figure 3. In this plot, the interferometer is assumed to be centered on the Earth and the source is perpendicular to the ecliptic plane. Although the coverage is not spectacular, it compares favorably with fractional coverages for present ground-based VLBI experiments. The inner portion of the transform plane could be filled more completely by linking the Mars mission radio telescope with a ground-based VLBI network, an orbiting radio antenna, and a Moon base radio dish.

As in the two previously proposed telescopes, the human crew will greatly simplify the operation and maintenance of the radio telescope. A very simple and inexpensive pointing scheme could be envisioned for this human-operated telescope, unlike the elaborate schemes needed for remote operation. Furthermore, the antenna could be constructed in flight, much like that planned for the space station, thus saving space and reducing costs on the mission.

CONCLUDING REMARKS

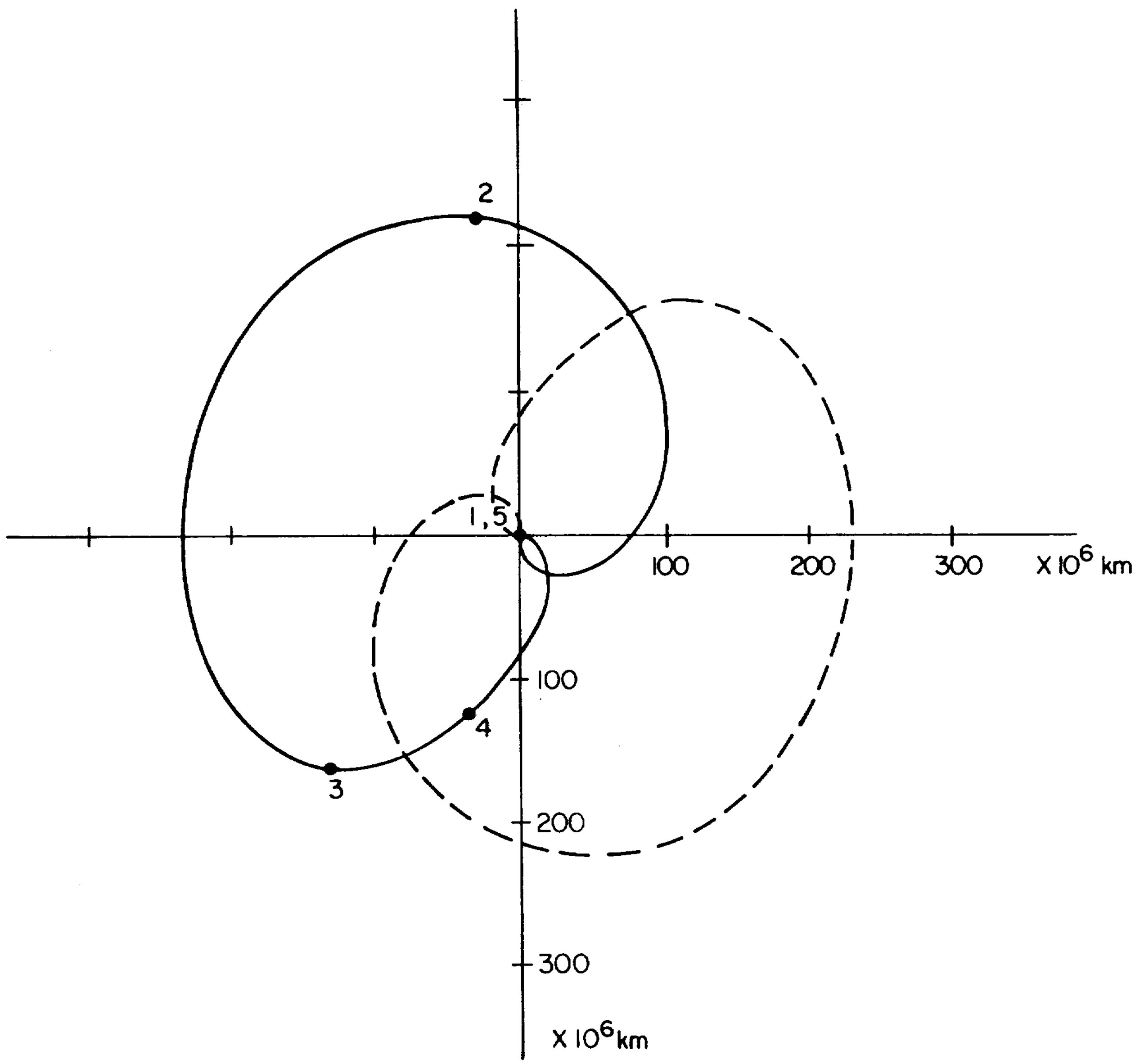
The three projects described above merely scratch the surface of possible astronomical experiments that could be performed during a manned mission to Mars (including a reduced magnetic field and particle flux from the solar wind). The radio telescope may also be operable at both longer wavelengths (for scintillation studies of compact radio sources) and at millimeter wavelengths (for spectral line studies of molecular clouds). In a related vein, fundamental experiments on inertial mass and the gravitational constant (G) could be performed to test for variations in different regimes of gravity.

Because these telescopes will be operated and maintained by a human crew on the mission, the cost savings over completely automated robot probes will be enormous. The weight of each telescope is small and thus

FIGURE 3

THE PORTION OF AN APERTURE SYNTHESIZED

BY A MARS MISSION RADIO INTERFEROMETER



The array is assumed to be centered on the Earth. The coverage is based upon a trajectory which will include a Venus outbound swingby on route to Mars. Earth launch occurs on 1/27/99 at 3, Mars departure at 4, and Earth return on 11/19/99 at 5. The dashed curve represents the additional sampling produced by the Hermitian property of the Fourier transform plane.

will not significantly impact on the launch costs. Thus, for a relatively low expenditure, major new astronomical telescopes can "piggy-back" on the Mars mission. The long Earth-Mars baselines can be used to gather data that is beyond the scope of telescopes in the Earth-Moon Environment.

N87-17762

EXOBIOLOGY ISSUES AND EXPERIMENTS AT A MARS BASE

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ABSTRACT

Research in Exobiology, the study of the origin, evolution and distribution of life in the universe, may be a major component of the science activities at a Mars Base. Exobiology activities would include: continuing the search for life on Mars; searching for evidence for ancient life from a warmer Martian past; research into the chemistry of the biogenic elements and their compounds; and other related activities. Mars provides a unique opportunity in Exobiology, both for immediate study and for long range and possibly large scale experimentation in planetary biology.

INTRODUCTION

The goal of Exobiology is to understand the origin, evolution, and distribution of life and life-related compounds on Earth and throughout the universe (DeVincenzi, 1984). To accomplish this goal, exobiological studies have been, and continue to be, carried out on missions to the other planets. Clearly Exobiology, as a scientific discipline, is inextricably tied to space and space missions.

Outside of the Earth, Mars has the most clement environment for biology in the solar system, and it naturally holds a particular fascination for exobiologists. It is not surprising that many opportunities for exobiological experiments exist within the context of a Mars Base. In fact, exobiology may be a major element of the Martian surface science activities conducted at a Base. However, a Mars base also focuses an important issue, of a fundamentally exobiological nature: the search for indigenous life on Mars. There will be strong pressure to conduct the search before humans land.

Humans add a new and vigorous dimension to the exploration of the surface of Mars, particularly in exobiology. For example, the presence of humans would ensure much better site selection and sample acquisition for exobiology investigations.

Equally important, the quality and quantity of scientific observation would be enormously increased.

The issues and opportunities for exobiology on Mars are discussed in this paper and the unique capabilities provided by humans on the surface are considered. The biological results of the Viking missions to Mars are briefly reviewed. The wealth of information obtained from the Viking missions generated a new set of questions and new lines of inquiry regarding exobiology and the question of the existence of life on Mars.

VIKING

In addition to the lander cameras, which would show the presence of any obvious macroscopic life-forms, the Viking landers contained three experiments specifically designed to search for indications of life on Mars:

1) The Gas Exchange Experiment (Oyama and Berdahl, 1977), designed to determine if martian life could metabolize and exchange gaseous products in the presence of a nutrient solution.
2) The Labeled Release Experiment (Levin and Straat,1977), which sought to detect life by the release of radioactively labeled carbon initially incorporated into organic compounds in a nutrient solution.
3) The Pyrolytic Release Experiment (Horowitz and Hobby, 1977), based on the assumption that martian life would have the capability to incorporate radioactively labeled carbon dioxide in the presence of sunlight (photosynthesis).

The results of all three experiments showed definite signs of chemical activity, but this was probably non-biological in origin (Horowitz, 1977; Klein, 1978, Mazur et al., 1978). In addition, the negative results of the Gas-Chromatograph/Mass-Spectrometer (GCMS) search for organic compounds places severe restrictions on the probability of life on Mars. The GCMS failed to detect organic material at levels less than parts per billion for heavy organics and parts per million for lighter ones (Biemann et al., 1977). It is relevant to note that these concentrations are considerably lower than would be found on the lunar surface. The conclusion seems to be that organics are not created but are actually destroyed on the surface of Mars. This is a strong indication that there is no life in the sands of Mars. The similarity of the elemental composition between the two Viking lander sites may be an indication that the top layers of martian regolith is a aeolian mantle of sand that has been reworked repeatedly (Carr, 1981). The Viking landers sampled in this mantle and conditions at greater depths may be different.

While in general finding no indication of the existence of life on Mars, the equivocal results of the biology experiments and the detection of all the elements necessary to support Earth-type life has led to much speculation concerning martian biology (Friedmann and Ocampo, 1976; Foster et al., 1978; Clark, 1979; Kuhn et al., 1979).

The properties of the Martian environment which seem most unfavorable to the sustenance of life are:

1) The general scarcity of water
2) The cold temperatures and extreme temperature variations which occur both diurnally and seasonally
3) Penetration to the surface of ultraviolet radiation between 190 and 300 nm and particle radiation
4) The apparent presence of strong oxidants in the soil
5) The low atmospheric pressure and, as a direct result, the exclusion of liquid water as an equilibrium state
6) The low concentration of nitrogen in the atmosphere and the apparent absence of nitrogen, in any form, in the top-regolith

With the possible exception of the final point, all of the unfavorable aspects of the Martian environment have sufficient variations such that any one of them can be eliminated by suitable site selection. Neither the Viking 1 nor the Viking 2 lander site was chosen based on this consideration.

CONTINUING THE SEARCH FOR LIFE

While the Viking missions gave us valuable insight into the biological potential on Mars, a more extensive search for extant Martian life forms needs to be done. The discovery of an indigenous Martian biota would profoundly effect the planning for a human presence on the planet's surface. An important question integral to any plans for a Mars Base program is: What steps and precursor missions should be undertaken to continue the search for life on Mars before humans land on the surface.

In approaching this question we must realize that the absence of life cannot be conclusively demonstrated by robotic missions, and possibly not by human biologists on the surface. The possibility that life could exist in an undiscovered cryptic niche will continually plague those who desire absolute answers from biology. The resolution of this dilemma will have to be in a two-step process. First, defining the biological impact of a localized human presence on Mars, and then determining what level of search is required to insure that any undiscovered cryptic Martian community of organisms is sufficiently well hidden that it is unlikely to be affected by such a presence.

Clearly, this approach points toward the necessity to understand the strategies of terrestrial organisms that occur in cryptic niches. As part of NASA's exobiological research effort, work in this area has been conducted with respect to the crypto-endolithic microbial communities of the Antarctic dry valleys (Friedmann, 1982; McKay and Friedmann, 1985). Studies of groups of organisms which grow in small depressions on glaciers, termed cryoconite holes, are also applicable (Wharton et al., 1985). Do these represent analogs for life in the present polar caps of Mars? Many other examples of cryptic life can certainly be found that would be applicable. By examining these communities on Earth we may be able to determine the relationship between the cryptic nature of a microhabitat and its degree of isolation from the environment. Often the very reason the organisms are in a cryptic niche is to isolate themselves from adverse environmental conditions. The more isolated a martian biological community is, the less likely it is to be affected by a human base.

The problem of searching for life on Mars with a rover or other sampling device is primarily centered around how to select the sample. It is probable that the development of artificial intelligence systems could result in a rover with the skills to navigate in rough terrain as well as a rudimentary form of biological and geological decision-making. The search for hidden life may require such sophistication, and preliminary studies should be started in the near future.

Once humans have landed, the search for extant life forms will certainly continue. Life is unique in that it is so diverse that it can only be loosely defined in terms of a set of attributes, rather than as a chemical or physical entity. Thus, human intelligence and the ability to make decisions, on the spot, could be critical for the recognition of Martian life forms, particularly as the search extends into environments beyond our terrestrial experience.

EARLY MARS

In looking at the profile of Mars drawn by the Viking results, it is hard to find any evidence that extant life forms exist on such a small, cold, dry planet. However, there is considerable evidence that at some time in the past, conditions on Mars were quite different. Large valley networks and outflow channels attest to the fact that copious amounts of liquid water once flowed on the martian surface (e.g. Carr, 1981). This in turn implies that the surface temperature was considerably warmer than it is today with concomitant high pressures. Some of these fluvial features occur in terrain that is heavily cratered, indicating that this warmer climatic regime probably dates back over four billion years (e.g. Carr, 1981).

This view is supported by theoretical considerations which suggest that during the first million years after the formation of Mars and Earth, the surface conditions of both planets were similar. During this time period, the atmospheric composition and pressure on these planets was determined primarily by outgassing of juvenile material and the surface may have been dominated by the processes of crust formation (e.g. Pollack and Yung, 1980). In fact, Earth, Venus and Mars may have all undergone initial periods of outgassing and crust formation that resulted in similar surface conditions on all three of the terrestrial planets. We know from the fossil record that life on Earth evolved and reached a fair degree of biological sophistication in the first 800 million years. The time interval was probably much shorter, but the absence of a suitable fossil record prevents that determination. It is entirely possible, then, that life also arose on Venus and Mars during an early clement epoch on these planets. Subsequent planetary evolution, however, seems to have favored only the Earth. The record of the origin and early evolution of life on Earth, and certainly on Venus, has been obscured by extensive surface erosion, while on Mars the situation is quite different and large fractions of the surface date back to this early time period (Carr, 1981). Hence, it is entirely possible that while no life exists on Mars today, it holds the best record of the chemical and biological events that led to the origin of life.

There are many areas in which a search for microfossils of an ancient martian biota could be conducted. One such area would be the valley networks that lace the ancient terrain and the bottoms of the outflow channels. Interesting lake sediments may exist on the floors of some of the martian equatorial canyons (Lucchitta and Ferguson, 1983). McKay et al. (1985) have shown that, similar to the Antarctic dry valley lakes, paleolakes on Mars could have extensive liquid water under a relatively thin ice cover, even under current cold martian conditions. Clearly, studies of these sediment beds, on site, by interdisciplinary teams of scientists could yield detailed information on past Martian biota.

THE BIOGENIC ELEMENTS

In addition to the search for life and past life, a goal of exobiology on Mars will certainly be to study the distribution and evolution of the biogenic elements (C,H,N,O,P,S) and compounds (e.g. water, carbon dioxide). The cycling of these elements is key to the sustenance of a biosphere. Even in the absence of life, studies of these cycles will provide a point of comparison and contrast for the biogeochemical cycles of these elements on Earth.

Information on the global scale cycling of the biogenic elements can be collected, to a large extent, by robotic observer spacecraft. It is the collection of more specific information on fine scale gradients, isotopic variations, and elemental ratios that would be greatly facilitated by a human presence. Such detailed studies might be required in areas with interesting mineralogy or salt composition. Interesting mineral phases, such as clay, may play an active role in processing the biogenic elements and compounds. Areas of rich salt concentration may be sites of enhanced water activity (Huguenin, et al., 1979). The study of this natural cycling of biogenic material would be useful in establishing the role of abiotic synthesis in the origin of life on Earth.

Planetary evolution and the interaction of global reservoirs of volatiles is an area of interest for the study of the evolution of life on planetary surfaces. If life did originate on Mars and subsequently became extinct, this event may be associated with changes in the cycling of the biogenic elements. Evidence for such changes may still be present in terms of the detailed elemental and isotopic structure of the layered terrain. Relatively recent information on the global cycles of carbon dioxide and water on Mars may be obtained from studies of the polar laminated terrain.

DISCUSSION

As one of the four main goals of planetary exploration through the year 2000, the Solar System Exploration Committee included the following, "Understand the relationship between the physical and chemical evolution of the solar system and the appearance of life" (SSEC, 1982). Studies of the surface of Mars can contribute to this goal. Many of the key research areas are dependent upon a human presence and a continually operated Mars Base.

Specific areas in which exobiological investigations on Mars would greatly benefit from, and possibly require, a human presence include:

1) Search for extant life
2) Search for evidence of ancient life
3) Examination of the chemistry of the biogenic elements and their compounds
4) Study the environmental parameters associated with each of the above
5) In situ exobiology experiments.

Exobiological investigations on Mars that would benefit from, but not require, a human presence include:

1) Observational Exobiology (e.g. studies of biogenic elements in the interstellar medium via VLBI techniques)
2) Interplanetary and interstellar dust collection for studies of the biogenic elements in primordial matter

In addition, a human base on Mars could serve as a springboard for exobiological research in the outer solar system. The research into the adaptive strategies of terrestrial organisms and an extensive understanding of the geochemical cycles on Mars may also lay the foundation for large scale exobiological experiments on Mars investigating the prospects for planetary ecosynthesis (Averner and MacElroy, 1976). It is the human component of the Mars Research Base that motivates these long term goals.

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N87-17763

MARS RESOURCES

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ABSTRACT

The most important resources of Mars for the early exploration phase will be oxygen and water, derived from the martian atmosphere and regolith, which will be used for propellant and life support. Rocks and soil may be used in unprocessed form as shielding materials for habitats, or in minimally processed form to expand habitable living and work space. Resources necessary to conduct manufacturing and agricultural projects are potentially available, but will await advanced stages of Mars habitation before they are utilized.

INTRODUCTION

In the forseeable future, the transportation cost of sending people or material to Mars will be high. Thus, for even the first manned missions to Mars, an effort should be made to utilize the indigenous materials of the planet, and any sustained presence must be supported by utilization of Martian resources.

Mars resembles the Earth enough in its geological history and current state that one can imagine the availability of practically anything that can be found on Earth, in terms of inorganic mineral resources (Cordell, 1984). It's biotic history is uncertain, certainly much less extensive than the Earth and perhaps totally absent, so concentration of elements and compounds by biogenic means is not expected. It has polar ice caps and a thin atmosphere. Surface alteration processes due to the interaction of atmosphere and surface are expected. These are the potential sources of useful products. Solar energy flux is approximately one quarter that of the Earth; geothermal heat may be tappable in some places, but early missions will probably carry nuclear energy supplies.

With adequate supplies of energy, it is probable that any substance available on Earth can be obtained from the materials of Mars. However, this paper focuses on our current state of knowledge of Mars resources and their potential early uses.

IMPORTANT EARLY USES OF MARS RESOURCES

The earliest uses of Mars resources will come where: 1) the product is required in substantial quantities and would otherwise be transported from Earth; 2) the mass of product desired is significantly larger than the mass of the equipment which must be transported to Mars to produce it; 3) the raw materials are readily available; and 4) the energy necessary for manufacturing the product is available. By-products of primary processes may find uses that do not meet these criteria. Among uses which may fit this category are: 1) extraction of propellant (H_2, O_2, CO, CH_4, etc.) for Mars-Earth space transportation and trans-Mars transportation; 2) shielding of habitats and equipment from radiation, and insulation of structures for thermal control; 3) expansion of living and work space; 4) makeup of consumables in life support systems and expansion of controlled life support systems.

PROPELLANT

Propellant produced at Mars or on its moons could greatly reduce Mars-orbit and Mars-Earth transportation (see Ref. this volume). In the past, proposals have been made to extract CO and O_2 for propellants from abundant CO_2 in the atmosphere. This is a predictable and accessible resource. Liquid hydrogen and oxygen also could be prepared from water, which exists as ice in polar regions and probably as permafrost elsewhere. Propellants could also be produced from carbonaceous-chondrite-like material, which probably exists on Phobos and Deimos.

SHIELDING AND INSULATION

The radiation environment at the surface of Mars is sufficiently harsh that long term habitation requires substantial shielding, by up to the equivalent of 3 meters of rock or mineral material. Surface temperatures at equatorial latitudes are approximately $240^{o}K$ in daytime and $190^{o}K$ at night. Loose soil or rock can probably be used for thermal insulation of habitats; however, for specific applications, it may be necessary to develop better insulators than those obtainable from the unprocessed rock and soil materials.

EXPANSION OF LIVING AND WORK SPACE

The first visitors to Mars will be severely cramped in their living and working space. Extended staytimes will be much easier if ways are available to expand the pressurizable volume that can be occupied for the crew. Because of the need for shielding, these expansions must be subsurface structures. Thus, there will be a requirement for structural materials that can support the necessary loads. This requirement could be met with non-metallic materials (fused or sintered silicates, or concretes) or metals. Seals for enclosures and interconnections to the outside environment will require materials from Earth at first, but local production will undoubtedly receive significant attention because these structures will control the rate of habitat expansion.

LIFE SUPPORT SYSTEM REQUIREMENTS

An abundant supply of water is essential for many aspects of life support systems, for direct use by humans, for agriculture, and for manufacturing. A large reservoir of water must be available to a permanent base. Oxygen and nitrogen are principal components of the atmosphere in which humans can exist, and CO_2 is the principal constituent of botanical interest. To the extent that closed ecological life support systems can be established, the resupply requirement for these compounds can be reduced; however, some irretrievable losses or at least sinks with very long residence times will be present, and in order to expand the initial base replenishment of these elements and compounds will be necessary from indigenous sources. It would be useful also to have sources of the principal nutrients for plants, although life support system closure can reduce the need for these materials.

MARS MATERIAL RESOURCE AVAILABILITY

The principal potential sources of materials on Mars can be categorized into four general groups: 1) atmosphere; 2) ground ice (permafrost, polar caps); 3) primary (igneous) rocks; and 4) regolith (dust and altered igneous rock) and sedimentary deposits formed by redistribution of the regolith. The moons of Mars represent another distinct category of material with potential resource implications.

ATMOSPHERE

The atmosphere of Mars consists of CO_2 (95.3%), N_2 (2.7%), Ar (1.6%), O_2 (0.1%), CO (0.1%), and small amounts of water and noble gases (Owen et al, 1977). Total surface atmospheric pressures measured by Viking are in the range of 7-8 millibars (Hess et al, 1977). A seasonal reduction is correlated with precipitation of CO_2 at the South pole and the pressure rises as CO_2 is sublimated. Peak surface temperatures at the Viking sites were 240^oK and lowest temperatures were slightly less than 190^oK.

The amount of water in the atmosphere is small, on the order of 100 precipitable microns (Farmer, et al, 1977). The entire atmosphere contains approximately the equivalent of $1.3 km^3$ of ice.

From time to time, the Martian atmosphere develops dust storms, which raise substantial quantities of what must be very fine dust which can remain aloft for weeks at a time. Most dust storms begin in the southern hemisphere when the planet is near perihelion and is receiving maximum solar energy. The dust is carried northward and longitudinally, but the continual supply of dust suggests that much of the material is eventually returned to the southern hemisphere. The Hellas basin, a large low area in the southern hemisphere, may be a sink for much of the dust, and possibly a source for new dust storms (Mutch et al, 1976).

ICE IN THE SUBSURFACE AND POLAR CAPS

Mars' atmosphere has small amounts of water vapor. A perennial ice layer of a few meters thickness has been inferred at both poles, extending to about 10 degrees equatorward. This overlays a series of layered deposits that apparently consist of a mixture of dust and ice. The total thickness is perhaps 1-2 km in the south and 4-6 km in the north, with perhaps 85 percent of the material consisting of ice. In the south, there is a seasonal cap of dry ice.

Morphological evidence (canyons, channels) strongly suggest the former presence of liquid water. In order to explain the quantities of water that would be required, a reservoir other than the polar caps must have existed in the past and may exist now. This is generally believed to be permafrost, and morphological observations have been used to infer its distribution (Squyres, 1985). Chaotic terrain, overlapped lobate deposits of crater ejecta, and patterned ground all have been interpreted in terms of the former presence of water. Creep deformation of ice in a matrix of silicate deposits, similar to rock glaciers on Earth, has been cited by Squyres and Carr (1984) as evidence for the current presence of ground ice. The regions of high ice content may extend to depths of 1 km or more, which would make ground ice the largest water reservoir on the planet.

The creep features indicative of ground ice are notably absent between 30 degrees north and south on the planet, suggesting that the regolith has outgassed in equatorial regions.

<u>IGNEOUS</u> <u>ROCKS</u>

Mars currently exhibits the largest volcanic landforms of the solar system. These features are inferred to be basaltic from their morphology. Features such as Olympus Mons are believed to be relatively young, based on the distribution of impact craters, and may have been highly active until a billion years ago. In all other terrestrial planets studied intensively to date (Earth, Moon, Venus), basaltic volcanism is a consistent feature (Basaltic Volcanism Study Project, 1981). Earlier internal igneous activity may have been basaltic or possibly more primitive (ultrabasic, komatiitic), but in any case can be inferred to be extensive.

Few direct data exist on the compositions of Martian volcanic rocks. Analysis of the Viking X-ray fluorescence experiment data indicated that the surface fine material, dominated by weathering products, was derived primarily from mafic (basaltic ?) material, low in trace elements, alkalis and aluminum (Toulmin et al, 1977).

Recently, strong evidence has accumulated that the Shergottite meteorites (and perhaps the Nakhlites and Chassignite) are of Martian derivation (Bogard and Nyquist, 1983). Among other evidence, the presence of trapped noble gases similar to those measured by Viking in the Martian atmosphere and different from all other terrestrial or meteoritic noble gas patterns is compelling. The crystallization age of the shergottites is on the order of one billion years, approximately the inferred age of the large volcanic landforms. The meteorites were probably expelled from Mars by one or more impact events that introduced substantial shock damage.

Minerals in the shergottites are principally the silicates olivine, iron-calcium-magnesium pyroxenes (augite and pigeonite), intermediate plagioclase feldspar (sodium-calcium aluminosilicate), and small amounts of silica, with titano-magnetite (titanium-bearing iron oxide), with up to 1 percent of a calcium phosphate mineral, whitlockite, and minor amounts of iron sulfides. This composition is consistent with inferences from the Viking analyses. No primary mineral containing water of crystallization (e.g. amphibole, mica) has been observed. However, these rocks are more highly oxidized than any other differentiated (igneous) meteorites.

If shergottite-like basalt is common on Mars, other rock compositions may have also developed locally. In slow-cooling (subsurface) environments, separation of olivine, pyroxene and feldspar could have led to significant enrichments in silica, alkalis and alumina to form granitic or rhyolitic rocks. Where volcanic materials came into contact with ice, a variety of unique rock types may have formed, some of which may have produced hydrothermal deposits. Cordell (1985) has speculated that Mars may have as diverse a set of igneous and hydrothermal deposits as the Earth. Although speculative, these represent targets to be sought in further exploration of the planet.

REGOLITH AND SEDIMENTARY MATERIAL

The principal means of rock degradation on Mars appears to be impact cratering. Locally, comminution through the action of fluids has certainly occurred, as the material eroded from the major canyons attests. Chemical weathering is probably a minor source of physical degradation of primary rock. At both Viking landing sites, abundant fine-grained materials are present, and the occurrence of dust storms indicates the existence of substantial quantities of ultrafine material.

Following physical degradation, chemical weathering may have been effective in changing the chemical composition of the particles. Viking inorganic analyses are consistent with a soil composition containing aluminum-poor iron-rich clay minerals and iron oxides, plus sulfates, carbonates, and chlorides developed by leaching and precipitation of water-soluble phases by liquid water or possibly by thin films of moisture (Toulmin et al, 1977). Surface reflection spectra are dominated by Fe^{3+} + $-O^{2-}$ absorption bands, but are more consistent with spectra expected from altered volcanic glass than from specific clay minerals (Singer, 1982). If the original rock composition was similar to that of the shergottites, one might expect the soil to contain minor amounts of free silica, unreacted feldspar, and phosphate. It is of interest that soils present in the dry valleys of Antarctica, where liquid water is generally absent, are similarly enriched in chlorine, sulfate and carbonate(Gibson et al, 1983), suggesting concentration mechanisms for these elements by soil capillary action.

PHOBOS AND DEIMOS

Phobos and Deimos were imaged by the Viking orbiters. Phobos was shown to have complex surface features and both bodies are saturated with respect to impact craters greater than 300m (Veverka and Duxbury, 1977). Both bodies are spectrally uniformly gray and both have low mean density (Duxbury et al, 1979). For these reasons, it has been speculated that they are similar in composition to carbonaceous chondrites, a class of meteorites (asteroids) rich in clay minerals (hydrated silicates) and organic compounds (Pollack et al, 1973).

TYPICAL RESOURCE EXTRACTION PROCESSES

Table 1 gives estimates of the mass, power and productivity of extraction of some useful resources from Martian and Phobos/Deimos materials. In general, these processes will include mining (or concentration from the atmosphere), separation of beneficial minerals, chemical reactors, and storage. No subsequent processing is described here.

TABLE 1

TYPICAL CHARACTERISTICS OF MATERIAL PROCESSING SYSTEMS

PROCESS	FEEDSTOCK	PLANT MASS (KG)	PLANT POWER (KW)*	PRODUCT	REFERENCE
CO_2 Extraction from Atmosphere	Atmos.	250	6	1000 KG/day	Ash (1978) Frisbee (1983)
Electrolysis of CO_2 to O_2 +CO	1000 KG CO_2/Day	1000	10	750 KG O_2/D. 150 KG CO/D.	Ash (1978) Frisbee (1983)
Electrolysis of CO to O_2 +C	250 KG CO/Day	250	75	100 KG C/D. 150 KG O_2/D.	Ash (1978) Frisbee (1983)
Melting Ice for Water	Subsurface Ice	50	0.2	10 KG/Day	R. Williams, JSC (Pers. Comm.)
Extract Water from Clays	Clays, with 5% Water	130	5	10 KG/Day	C. Lin, JSC (Pers. Comm.)
Carbon Reduction of Iron From Oxides	Conc. Oxide 90% FE2O3	700	25	1000 KG/Day	R. Gertsch, Co Sch Mines (Pers. Comm.)
CAO Production from CA Carbonate	Conc. CACO3	550	80	1000 KG/Day Port. Cement	R. Gertsch, Co Sch Mines (Pers. Comm.)
Glass Production	FSPAR, Silica, Clay	5450	23	1000 KG/Day Glass	M. H. McDonald Corning Ind. (Pers. Comm.)
Water Extraction from Carb. Chondrite	Unproc. Typ II Carb. Chond.	150	5.5	10 KG/Day	C. Lin, JSC (Pers. Comm.)
Electrolysis and Liquefaction of O_2, H_2	10 KG Water/ Day	120	25	10 KG/Day (O2 & H_2, Liq.)	C. Lin, JSC (Pers. Comm.)

*Power requirements are based on conventional technology. Recent advances in use of microwave heating (Meek et al, 1978) may allow significant reduction in the power and mass of thermal power supply systems. JPL technology research is substantially reducing the mass and power requirements for CO_2 reduction and H_2 and O_2 liquifaction (E. Lawton, JPL, communication.

Extraction from Atmosphere

CO_2 can be concentrated from the atmosphere by compression. The CO_2 can be reduced electrolytically to CO and O_2. Ash (1978) proposed such a process as a source of rocket propellant. CO_2 can be reduced further to O_2 and elemental carbon. This requires significant quantities of electrical power. It is possible to condense water from the atmosphere; however, the abundance is so small that large systems will be required for extraction of significant quantities, making water extraction from ice or possibly clay minerals more practical.

Extraction from Ice

Water can be obtained from ice or permafrost by melting. A typical soil/ice permafrost may contain 25 - 35 percent water. As drilling through ice or permafrost may be difficult, it is recommended that ice be melted in the subsurface and pumped to the surface. Only low grade thermal energy is required.

Extraction from Clay Minerals and Martian Dust

If hydrated minerals are present in the regolith or in ancient sedimentary deposits, water may be extracted by simple heating. The amount of water in clays is probably less than 5 percent, and temperatures of 120 - 300 degrees centigrade may be necessary. Thermal energy may be available as a byproduct of other surface activities, such as the Mars base power system.

The Martian dust appears to be rich in iron oxides. It may be possible to extract metallic iron from these minerals, using electrolytically produced carbon as the reactant. Primary iron oxides in igneous rocks are an alternative source. Carbonates are believed to be present in the Martian regolith. If calcium carbonate is present, it may provide a source for the production of CaO to be used in portland cement for construction purposes. Phosphates (whitlockite) or feldspars may also be sources for CaO.

Utilization of Primary Rocks and Their Constituent Minerals

The major minerals of the igneous rocks of Mars are likely to be olivine, pyroxenes and feldspar. Silica (quartz) may be abundant locally. These rocks will find their first use as materials of construction, for example, as shielding materials or as aggregate in concrete. This will require mining, but not significant thermal or chemical processing. The abundance of feldspar, silica and clay minerals may provide ready materials for production of glass and ceramics, which could find use in construction and manufacturing. Data for a ceramics plant are included in Table 1.

Extraction of Propellant from Carbonaceous Chondrite (Phobos?)

Typical water contents of carbonaceous chondrites are : Type I, 20.8 percent; Type II, 13.4 percent; and Type III, 1.0 percent. Extraction of water can be achieved by heating to temperatures of 300 - 650 degrees centigrade. Production of cryogenic propellants would require electrolysis and liquification systems and significant electrical energy.

Better information on the nature and distribution of Mars resources is necessary in order to determine the accessibility of usable resources and in order to develop specific processes for their utilization. (This is unnecessary only for atmospheric CO_2). For those processes that involve chemical reactions with solid minerals of rocks and soil, mineralogical and chemical compositional data must be obtained by analysis of enough precision to identify the main minerals and determine their composition. This can be accomplished by in-situ or returned sample analysis of geologically characterized terrain on Mars. Similar information is needed for Phobos and Deimos. For those processess that require locating a specific resource, for example water ice, global orbital data may suffice, which will also allow the extension of detailed mineralogical and compositional data to other regions of the planet. This information will be significant in determining the rate of expansion of a human foothold on Mars, so should probably be obtained on precursor science missions, unless early manned missions are contemplated.

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N87-17764

MARS SURFACE SCIENCE REQUIREMENTS AND PLAN

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ABSTRACT

We analyze the requirements for obtaining geological, geochemical, geophysical, and meteorlogical data on the surface of Mars associated with manned landings. We identifiy specific instruments and estimate their mass and power requirements. A total of 1-5 metric tons, not including masses of drill rigs and surface vehicles, will need to be landed. Power associated only with the scientific instruments is estimated to be 1-2 kWe. We define some requirements for surface rover vehicles and suggest typical exploration traverses during which instruments will be positioned and rock and subsurface core samples obtained.

INTRODUCTION

The purpose of this paper is to present an analysis of desirable physical science activities (geology, geophysics, meteorology) associated with manned Mars landings. The scientific rationale and objectives for Mars investigations are discussed in detail in previous studies (e.g., [1]); differences associated with the manned aspects are discussed in references [2] and [3]. As a context for the plan, we assume a multiple-landing mission scenario that leads to a permanent manned surface base called "Columbus Base"[4]. In this approach, during each of the first three missions a crew of four lands at different sites and performs scientific investigations for about two months. On these first three landings, the crew has the aid of an extra vehicular activity (EVA) rover vehicle with a range of about 10km. On the fourth mission, one of the previously visited sites is selected for development as a base from which more extensive explorations will take place. More capable surface transportation is assumed to be available at this point, namely a shirt sleeve (SS) rover with a range of about 100km. A remotely piloted air-plane of the type suggested by Clarke, et al [5] is also assumed to be available by the fourth landing as an instrument carrier for long range (1000+km) geochemical, geophysical, and atmospheric surveys as well as visual reconnaissance.

We define instrumentation needs and give estimates of power and mass requirements in order to help estimate the total landed payload from which propulsion and other requirements can be calculated. Our mass and power estimates represent upper limits because they are based on present technology. We expect that advances in instrument technology stimulated by mission requirements such as those proposed here will greatly reduce the ultimate payload mass and power values. The main science questions we address here are the composition and structure of the solid planet and the nature of geological and atmospheric processes. As a consequence, we emphasize geologic sampling and geophysical and meteorology observations. We do not discuss life science or operational engineering science requirements.

ROCK COMPOSITION AND PROPERTIES

Samples

Obtaining rock and soil samples will be a primary function of the landing team. We suggest that a total sample mass of 100-500kg should be returned to Earth from each of the first three landing sites. Of these amounts, about 25% should be returned in sealed, refrigerated storage containers so as to retain volatile materials and heat-sensitive structures in as close to a pristine martian environment as possible. Samples will consist of hand-sized rocks, 50-100g soil samples (including special samples to investigate microenvironments), and core samples obtained with the aid of drilling equipment. The surface sampling, cataloging, and documentation procedures will be derived from those developed during the Apollo lunar explorations [6] and are not discussed further here.

We suggest that each landing team take one 100m-long core at the landing site and numerous 10m-long cores at remote sites during rover traverses. These cores will be essential for analysis of near surface physical properties and geologic stratigraphy. The cores will be examined on the surface and portions selected for environmental storage and return. The remaining cores can be studied by the crew while present on the surface and/or stored locally at ambient conditions for future retrieval. Judging from experience on Earth, hole diameters of 15cm and core diameters of 5-10cm seem appropriate. Details of the equipment that will be required to perform these coring operations, including masses and power requirements, are given in reference [7]. It is assumed that the landing craft and rover vehicles will be capable of supporting the respective drilling and sample storage equipment.

Petrology/geochemistry:

The landing craft should have analytic equipment capable of determining rock compositions quickly to help guide the sampling and geologic exploration. These instruments will remain behind on the lander for future use. Instrumentation, with estimated mass and overall power requirements, will include the following:

1. A combined x-ray fluorescence/diffraction instrument (exchangeable anode, with capability of synchronized movement of tube, sample, and detector). This instrument is for major-element chemistry and for mineral analysis. MASS: 70 kg

2. An electron beam instrument optimized for imaging (scanning electron microscope), but equipped for energy-dispersive analysis (microprobe). This instrument is for microfossil exploration and for mineral analysis. MASS: 80 kg

3. A combined thermogravimetric/differential scanning calorimeter instrument for hydrous mineral analysis. MASS: 15 kg

4. Sample powdering, dissolution, and optical analysis equipment. MASS: 40 kg

5. A gas and water analysis system based on one or more of atomic absorption, gas chromatography, laser emission spectroscopy, and mass spectroscopy. MASS: 50 kg

PETROLOGY/GEOCHEMISTRY SYSTEM - TOTAL ESTIMATED MASS: 255 kg
TOTAL ESTIMATED POWER: 2 kw

Rock Physical Properties:

Rock physical properties will be observed directly during rover traverses, in the immediate vicinity of the lander, and remotely by geophysical means (discussed below). Suggested requirements for the direct observations are listed below.

Soil: Core penetrometer and plate bearing tests will be performed automatically at every rover sampling stop. MASS: 10kg

Core holes: Each of the 10m-long core holes will be used for an in situ seismic Q and P-wave velocity measurement by deploying a reusable acoustic probe in the hole and hitting the nearby surface with a spring-loaded or chemically-propelled impacting source.

MASS: 15 kg

POWER: 1 w

To determine basic rock mechanics properties, about ten hand-sized rock samples at each EVA site will be crushed in a simple point load press to obtain strength data under martian conditions. No sample preparation is required for these tests but after crushing, samples can then be used for petrology/geochemistry analysis after further preparation.

MASS: 20 kg

POWER: 1/2 w

Pieces of the 10m core and surface samples will be used to measure dielectric constant under in situ atmospheric conditions in order to interpret radar absorption data.

MASS: 3 kg

POWER: 1 w

SURFACE SCIENCE TELEMETER STATION

Conceptual Design and Requirements

We propose that multiple, long-duration science stations be deployed by a two-man crew operating from a rover vehicle. A number of identical stations, shown schematically in fig. 1, will telemeter their data to the landing base for up-link to the main craft. In the initial landings, a maximum of four stations will be deployed from an EVA-type rover. In later landings, more stations will be deployed at larger ranges using the SS rover. The stations will be powered by radioisotope thermal generators (RTGs) for an operational lifetime of at least 10 years. The stations will have the following instrumentation, consisting of separate functional packages interconnected by cable.

SEISMOMETER:

A 3-axis, broad-band (0.1-50 Hz), high sensitivity seismic unit that will be well coupled to ground by installation in a drilled, cored, and backfilled hole (see fig. 1).

MASS: 1 kg

POWER: 1/2 w

ELECTROMAGNETIC SYSTEM:

A permanent, passive electromagnetic (EM) data acquisition system will be installed at each Surface Science Telemeter Station (SSTS). This equipment will be similar to tensor magnetotelluric (MT) systems widely used in Earth applications. The system we propose consists of a three-

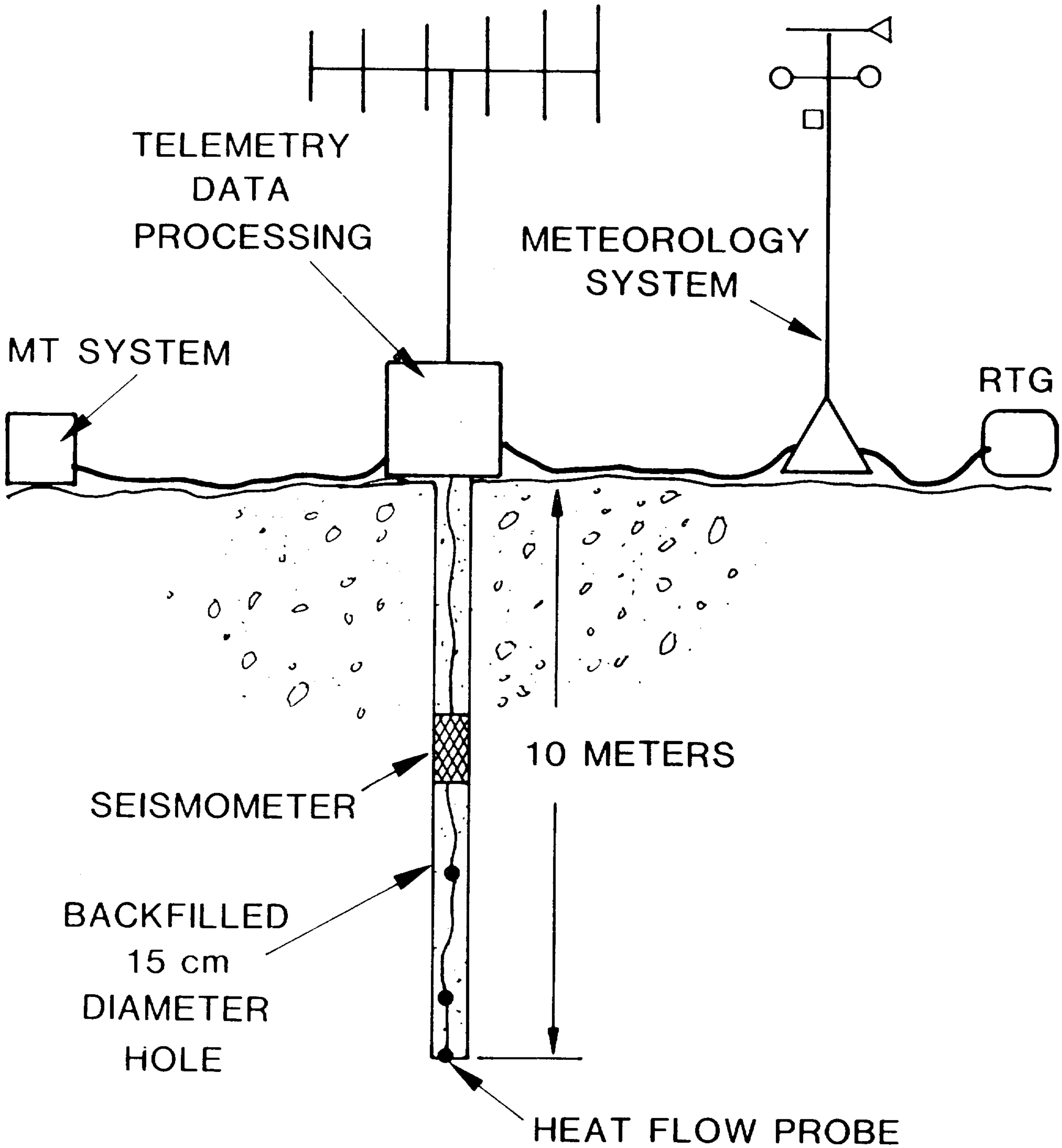

FIGURE 1

Schematic Illustration of the proposed
Surface Science Telemeter Station.
Instrument Modules are explained in the text.

axis fluxgate magnetometer for magnetic (H) field measurements less than 10^{-3}Hz, a three-axis coil magnetometer for H-field measurements from 10^{-3} to 10^{2}Hz, and a two-axis horizontal electric (E) field dipole for measurements from 10^{-3} to 10^{2}Hz. Thus, with this equipment, the martian magnetic spectrum and it's time variations may be studied below 100Hz, and subsurface electrical resistivity estimated to great depths in order to help determine radial structure and thermal state. The presence of the E-field dipoles will permit estimates of the tensor electrical resistivity from 10^{-3} to 10^{2}Hz. Deep magnetic and electric field sounding. DC to 100 Hz: MASS: 100kg

POWER: 1/2 w

METEOROLOGY SYSTEM: Instruments to measure: Temperature, wind speed and direction, barometric pressure, aerosol content (mini-LIDAR?), and composition using a mass spectrometer: MASS: 20 kg

POWER: 10 w

HEAT FLOW PROBE: MASS: 1/4 kg

POWER: 1/2 w

DIGITAL TELEMETRY/DATA PROCESSING SYSTEM - 15 channel (AM) with microVax -equilvalent or better processing capacity: MASS: 15 kg

POWER: 25 w

RTG POWER SUPPLY (50 watts): MASS: 25 kg

TOTAL ESTIMATED SSTS INSTRUMENT MASS, PER STATION - 162 kg

TOTAL ESTIMATED POWER, PER STATION - 37 watts (50 w-class RTG)

EXPLORATION TRAVERSES

Rover surface vehicles

Exploration from the initial three landing sites will consist of about ten one-day-long traverses using the EVA rover out to a range of about 5 km. Primary exploration from the permanent base will consist of four approximately 5-day-long SS rover traverses out to a linear range of 30-40 km. Schematic plan views of rover traverses and typical placements of instrumentation stations and explosive seismic sources are shown in figs. 2 and 3. The EVA rover will carry modules for installation of Surface Science Telemeter Stations (SSTSs, described above) on at least three of the traverses, and explosives (100 kg) for at least two seismic-source stations. The advanced SS rover will carry sufficient modules for

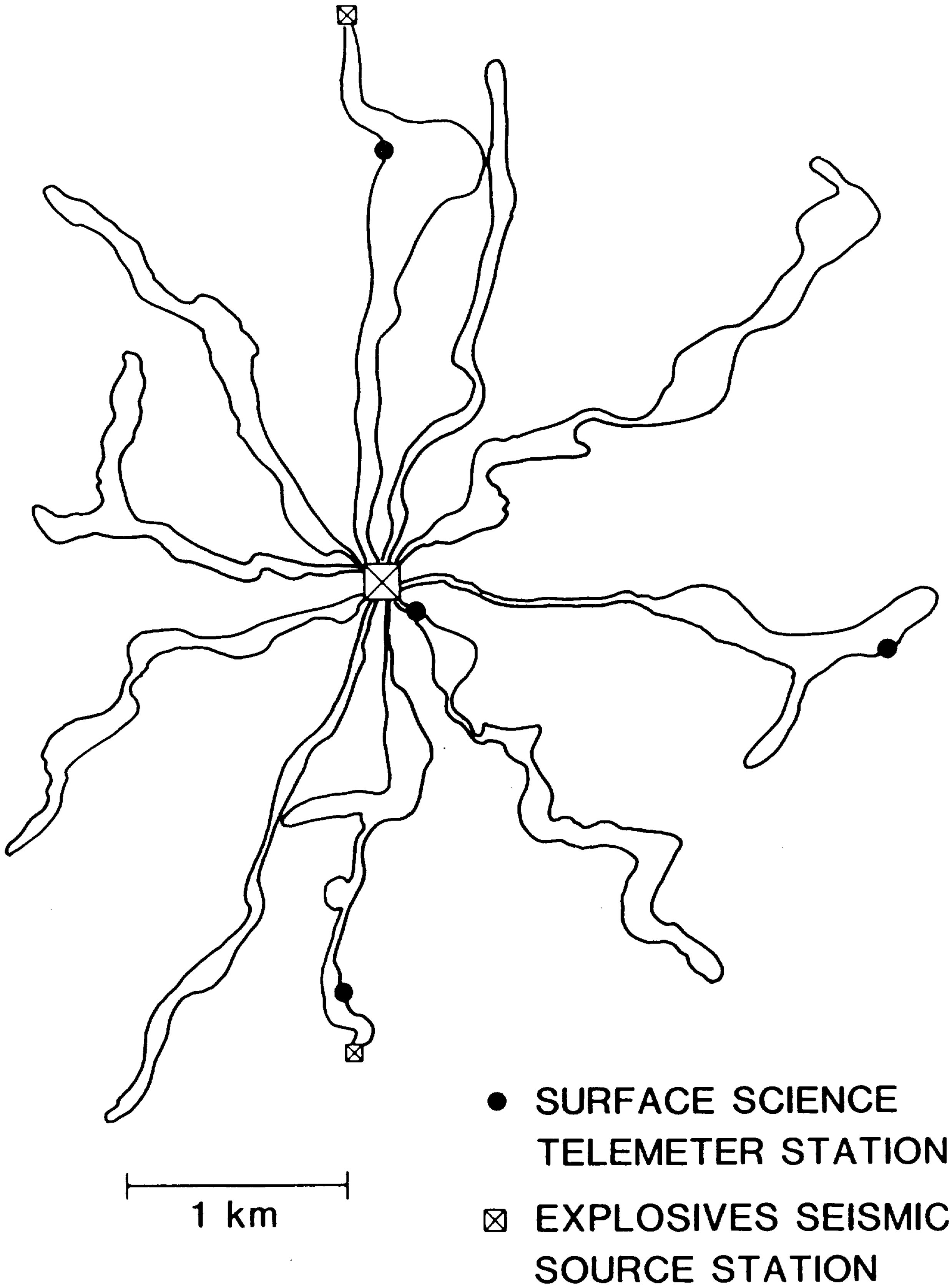

FIGURE 2. Schematic plan view of proposed EVA rover science traverses. Possible placements of Surface Science Telemeter Stations and explosive sources for a seismic refraction line are indicated.

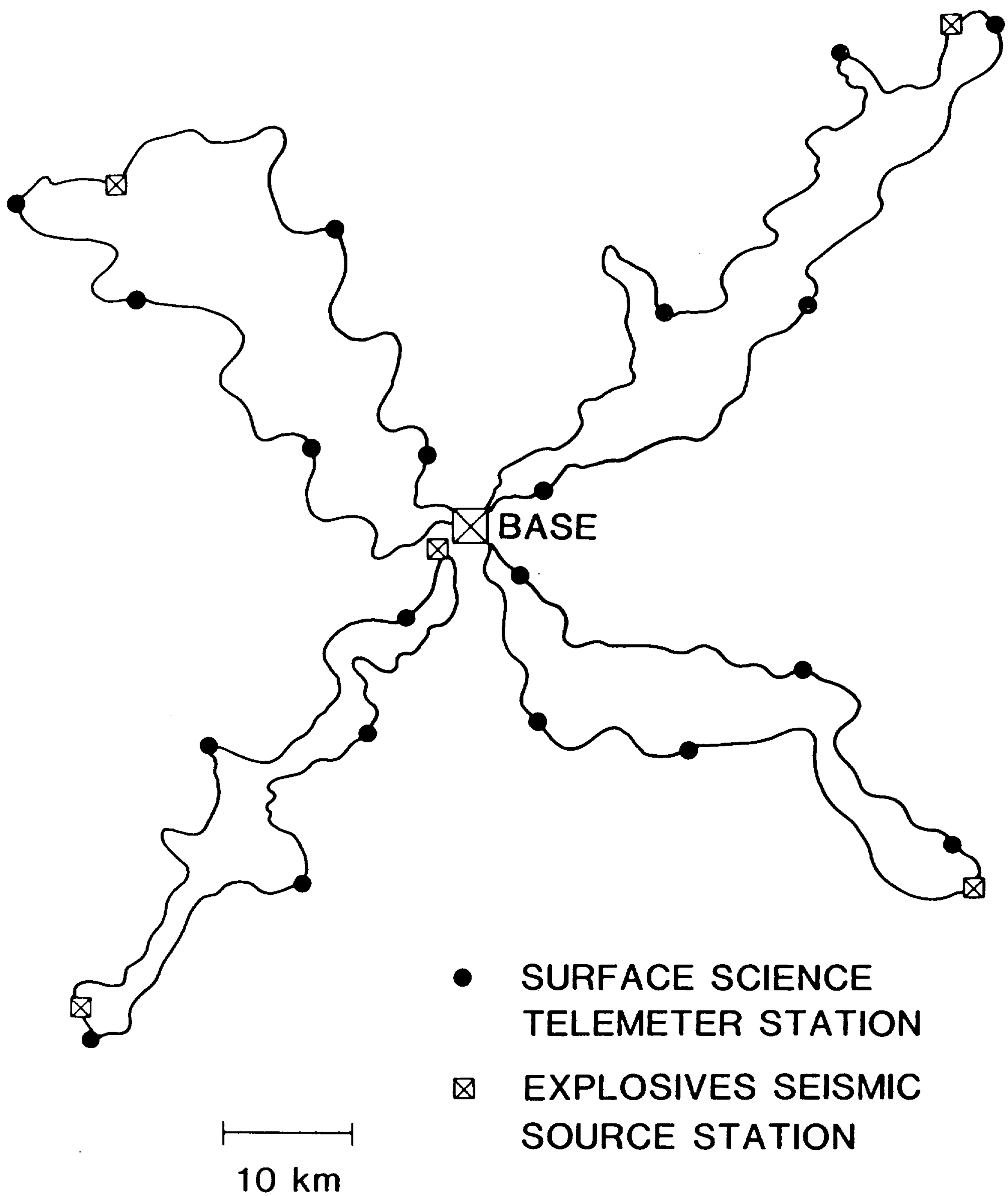

FIGURE 3. An expanded science station network deployed with the aid of an extended range SS rover. Two, approximately perpendicular seismic lines are shown.

the installation of five SSTSs and explosives (100 kg) for one radio armed and detonated seismic-source hole located at the extreme range of each traverse. Both rover types (EVA and SS) will be manned by a crew of at least two. In addition to adequate life support consumables and motive fuel, the rovers must transport a drill rig and compressor for the station holes. The SS rover will also carry a self-leveling gravimeter that will automatically make a gravity measurement at each stop (typically every several hundred meters).

Gravimeter: <u>MASS</u>: 10 kg
<u>POWER</u>: 1/2 w

A separate, portable passive EM system that measures three components of the H-field and two components of the E-field from 10^1 to 10^4Hz will be carried on the rovers. This system will be used for rapid reconnaissance around each SSTS to measure very near surface tensor electrical resistivity that will yield information on geologic structure as well as the possible presence of ground ice.

Portable EM system: <u>MASS</u>: 75 kg
<u>POWER</u>: 0.5w

The crews will alternate for each traverse with the main base crew who will be performing other activities such as monitoring the traverse and station installation, examining samples from previous traverses, and performing long-range remotely-piloted-vehicle (Mars airplane) surveys.

<u>Mars airplane</u>

A remotely-piloted airplane or drone will be assembled by the crew at the permanent base and used to perform long range (1000+km) airborne geophysical surveys from about 100-1000m altitude. The surveys will include magnetic, photographic, low resolution gravity, atmospheric composition, and gamma spectrometric measurements. On-board TV will help guide the vehicle to interesting surface features. At the farthest distance from the lander, the drone will be landed to deploy a seismology/meteorology station.

Mars Airplane [5]: <u>MASS</u>: 300 kg

<u>SUMMARY</u>

A summary of the mass and power requirements for the major elements of the surface science equipment and sample collection is given in Table 1. We estimate that 1-5 metric tons, including mass for generation of

TABLE 1

SUMMARY OF MASS AND POWER REQUIREMENTS
FOR SURFACE SCIENCE ACTIVITIES

ELEMENT	MASS (kg/landing)	POWER (kwe/landing)	DEPLOYMENT	COMMENTS
Rock, Soil, and Core Samples	500	1 (refrigeration)	EVA, Rovers	25% of total will be kept in environmental storage.
Petrology/ geochemistry system	255	2	Lander	Analytical equipment will be left on the surface.
Rock Properties	48	0.0025	Rovers	———
SSTS	162*	.037*	Rovers	* Mass & power are for a single station. 3-20 stations landing are envisioned.
Explosives	100-150*	0.001*	Rovers	* Mass is for 1 seismic source event. 2-5 events per landing are envisioned.
Drill Rigs	3800/400	37/5	Lander/Rovers	Large rig for 100m-deep hole smaller rig for 10m-deep hole [7].
Mars Airplane	300	--	Lander	40-100kg of total mass is for instruments.

1-2kw electrical power, will need to be landed on the martian surface to support the basic physical science activities. To these must be added masses and powers for the drilling equipment, rover vehicles, and airplane plus any strictly operational equipment (e.g., propellant manufacturing plant).

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N87-17765

MANNED MARS MISSION

ASTRONOMY OPTIONS

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ABSTRACT

Astronomical observations during the transit phase, in orbit about Mars, and from the surface present important scientific objectives. Primary astronomical objectives are being summarized by J. Burns (Univ. of New Mexico). Additional or alternative options will be introduced here, together with their strengths, weaknesses, viability, and value. It is important to note at the outset that not all possible options are necessarily important or viable[1].

Options

Potential experiments are limited only by imagination. Several options are listed in Table 1 and discussed individually below. Ultimately, in addition to weight, power, and volume limitations, the selection of experiments will be based on research interest and the value of anticipated scientific return.

Radio Astronomy

Prime radio astronomy experiments and observations that can be made from the manned Mars mission are those that require the radio quiet conditions found far from the neighborhood of the Earth. Some frequencies of interest here are quite low so that dipole arrays, in addition to parabolic dish antennas, would be used.

In addition, there are several active and passive radio measurements that can be made of the Martian in situ plasma wave detector. Many of these would make use of the same equipment used for the radio astronomy experiments. The plasma wave detector would be on the free-flying spacecraft in low Martian orbit.

Finally, the radio telescopes may be used for further monitoring of solar activity from the surface of Mars.

Solar Astronomy

Solar optical observations show the photospheric structure of the Sun and, with simple filters, map the large scale magnetic field structure, sunspots, filaments, and flares. During a large part of the transit

TABLE 1

OPTIONS

Experiment	Transit/In Orbit/Surface (X:yes, -:no)
Radio telescope	-/-/X
Met wave dipole array radio telescope	-/-/X
Solar astronomy	X/X/-
Optical observations of cataclysmic variables	X/-/-
UV all-sky mapping	X/-/-
Lyman-alpha all-sky mapping	X/-/-
Cosmic ray detectors/telescope	X/X/X
Gamma ray telescope	X/X/X

TABLE 2

SOLAR INSTRUMENTATION

Hydrogen-alpha telescope		25- 40-cm optics Birefringent filter 1000 x 1000 CCD detector Digitally stored images Digital display
Weight	50 kg including multi-use display terminal	
Power	< 1 kw	
Whole-disk x-ray monitor		0.5- 8 Angstrom sensitivity 1 second time resolution
Weight	< 10 kg	
Power	< 0.1 kw	

and while at Mars, the side of the Sun exposed to the spacecraft will be invisible from the Earth. Observations of the Sun in the Hydrogen-alpha spectral line will therefore be necessary for solar flare/erupting filament prediction and warning. These observations would have to be made during all three phases of the mission, but would be done from orbit during the surface excursions. More information on this is given in the white paper on Solar Physics: Solar Activity Monitoring and Prediction. A minimum instrument package is outlined in Table 2.

Additional solar observations are possible with more complex instruments - including magnetic field and velocity measurements. However, it is probable that these observations would be more efficiently and accurately made from Earth orbiting spacecraft.

The solar telescope will be used for other observations[2] with supporting equipment such as special filters. Planets will be observed to determine their albedo in the UV and visible ranges, and the same UV filters will be used for an all-sky UV survey. Opportunities will occur for the observation of stellar occultation by the outer planets, and the ephemerides should be developed for these observations. Other opportunities may occur for the observation of cataclysmic variables[2].

Planetary Astronomy

Optical observations of the planets and asteroids are possible during all three phases of the mission. The main advantage of doing these from the manned Mars mission is that during the transit phase the spacecraft will approach significantly closer to the main asteroid belt than is ever done by the Earth and will have uninterrupted viewing during the transit phases. However, it is likely that Space Telescope observations would supersede any information that would be gained from the size and quality of a telescope that could be carried to Mars, except for those types of observations noted above in the solar astronomy section. Therefore, it is suggested that the solar telescope will be sufficient for planetary astronomy in visible, ultraviolet, and infrared wavelengths.

These low spatial and spectral resolution, uninterrupted measurements will supplement the extremely high spatial and spectral resolution observations that will be made in the near future from Earth orbit.

One opportunity is that stereo observations of the planets or asteroids would be possible using Earth and mission based telescopes. However, there has been no stated need for such experiments.

Backscattered Solar Lyman-alpha

An opportunity exists to make high spatial resolution observations of Lyman-alpha solar emission that is backscattered from interstellar neutral gas that is entering the solar system. Low resolution observations have been made from unmanned spacecraft and have provided important information of solar spectral emission variability and the properties of the local interstellar medium. These observations are best made far from the Earth because of contamination by the Earth's hydrogen geocorona.

Optical, Infrared, Ultraviolet, and X-ray Extra-Solar System Astronomy

Observations of stars, the Milky Way, nebulae, pulsars, extra-galactic objects, etc. will be receiving particular emphasis with Earth-orbiting spacecraft. The Space Telescope and AXAF are only the first of a variety of missions planned by NASA and ESA. It is extremely unlikely that a large variety of useful information would be gained from manned Mars mission observations. However, there are a few narrowly selected specific observations that suggest potential benefits from being made on the manned Mars mission.

Galactic and Solar Cosmic Rays

Cosmic ray detectors and telescopes will be used for monitoring solar energetic particles. In addition, the data from these instruments will show galactic cosmic ray intensities and their variation with solar activity and the solar cycle. The modulation mechanism of cosmic rays in the solar system is not understood and data on variations at Mars, with correlative data from solar observations, will help solve this problem. The observations can be made from the surface of Mars, as well as from orbit and during transit phases.

Gamma Rays

Gamma rays are emitted impulsively during very short astronomical events. The location of these events is determined through timing of the detection of the gamma ray pulse at widely spaced spacecraft. Placing a gamma ray event detector on Mars would help with these calculations by making it easier to do the timing calculations.

Summary

All types of astronomical observations are technically possible from the manned Mars mission, and it is desirable to go into space for many of these observations. However, the advantages of going into space are equalled or even exceeded by placing many of the instruments in low Earth orbit (LEO), rather than sending them to Mars. The reason for this is that the main purpose for going to space is usually to avoid atmospheric interference and the day/night cycle. These goals are best achieved from LEO or, to avoid the day/night cycle, from a polar orbit or near-Earth stable position.

A more distant observing position is required by four types of observations. The first is any measurement which might be obstructed by the Earth's hydrogen geocorona - which extends out to many Earth radii. The second is any radio observation of low magnitude objects which might be obstructed by artificial radio sources on and near the Earth. This second source of pollution also extends to several Earth radii at some wavelengths and has led to suggestions for observatories on the opposite side of the Moon[1]. The third is a cosmic ray observatory/detector on the Mars orbiter and on the surface, to monitor solar cosmic rays and provide a more global view of how solar disturbances and evolution modulate galactic cosmic rays. The fourth is a gamma ray detector to give a long baseline for gamma ray event timing and triangulation. A minimal instrument package to meet these goals is outlined in Table 3.

Also, it will be essential to conduct solar observations from the manned Mars mission for the purpose of activity monitoring and prediction, complimented by solar whole-disk x-ray detectors and energetic particle monitors.

TABLE 3

OTHER INSTRUMENTATION

<u>Backscatter Solar Lyman Alpha</u>

	Utilizes solar telescope		
	Special filter		
	Weight	< 5 kg	
	Power	No additional	

<u>UV All Sky Map</u>

	Utilizes solar telescope		
	Special filter		
	Weight	< 5 kg	
	Power	No additional	

<u>Radio Astronomy</u>

Dish antenna			
	Weight	< 100 kg	
	Power	< 1 kw	receiving
		< 10 kw	transmitting
Dipole array			
	Weight	< 100 kg	
	Power	< 1 kw	receiving
Ionosonde			
	Weight	<100 kg	
	Power	< 1 kw	receiving
		< 10 kw	transmitting

<u>Cosmic Ray Detectors/Telescope</u>

	Weight	< 10 kg
	Power	< 1 kw

<u>Gamma Ray Detector</u>

	Weight	< 20 kg
	Power	< 1 kw

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N87-17766

MANNED MARS MISSION AND PLANETARY QUARANTINE CONSIDERATIONS*

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ABSTRACT

A short review of the history of planetary quarantine, the issues and changes in official advisory groups' pronouncements are presented. Then a discussion of the current situation and some ideas on how best to address them are outlined. Both manned and unmanned or automatic missions are discussed and their advantages and impediments outlined.

The first, and probably the most vexing aspect of this issue is the insufficiency of data that are both conclusive and relevant. Data are needed both about the presence (or its historical existence) of life on Mars, and about the conditions on Mars that may support "foreign" life forms. As a consequence of this paucity of data, proponents of any one side of this multifaceted issue have and will continue to profess the probity of their beliefs. More, better and germane data will tend to lessen the intensity of the discussions.

A little background and a review of the history of Mars Planetary Quarantine will be useful to those unfamiliar with the issues. When exploration of the solar system started to become practical back in the 1960's, there was concern that some terrestrial organisms might be carried to a planet and thereby establish themselves in their new environment. Once established on this non-terrestrial planet or satellite, it was feared that terrestrial organisms would upset the natural environment there and destroy or modify it irrevocably. The subsequent study of such a "contaminated" body would, therefore, become much more complicated and confusing. This would be especially true if the objective was the study of extraterrestrial biology. For these reasons, there was general agreement among scientists that solar system research should be conducted in ways that virtually precluded earthly organisms from "contaminating" the target body. This principle was discussed on an international level by delegates to the Committee on Space Research (COSPAR). These discussions resulted in a resolution establishing a criterion of 10^{-3} chance of contaminating a planet like Mars during the period of "biological exploration." The time period of "biological exploration" was at first assumed to be 20 years. More recently, the period has been extended to 50 years.

The United States and the Soviet Union approached the problem differently. The United States used an analytic approach; that is, all known (or assumed) factors that could lead to "contamination" were considered. Some of these factors were: determination of the "biological load" of a spacecraft (what numbers and kinds of organisms were launched with the vehicle), detailed assessment of the probability of survival of the organisms during the flight to the planet and during entry into the planetary atmosphere, and most important, the probability of growth (P_G) in the organisms' new environment (assuming viable organisms reach the planetary surface - or atmosphere). Of course, there was and is, no way to accurately calculate P_G. It's estimate was based on what we knew of the particular solar system body in question, and in the case of Mars, upon simulation experiments to determine the viability of terrestrial organisms in the Martian environment. The actual setting of P_G on Mars was based on a study of all relevant information available at the time by the Space Science Board of the National Academy of Science. For most solar system bodies, the estimate of P_G was so low that the COSPAR criteria could be met by simply sending a reasonably clean spacecraft. The P_G for Mars was estimated to be high enough to require positive measures to drastically reduce the load at launch. This led to a requirement that the Viking landers be heat "sterilized", as well as protected from later contamination during passage through the Earth's atmosphere.

The Soviet Union implemented the COSPAR resolution by a combination of heat and chemical "sterilizations", followed by an actual determination, of a duplicate spacecraft literally ground up and cultured for all possible organisms. The results claimed to show that no organisms survived these procedures, and hence, there was no chance to contaminate Mars (Vashkov, _et al_., in "Life Sciences and Space Research", XII, 199, 1974).

NASA has recently developed a new strategy to comply with the COSPAR guidelines on out-bound spacecraft. At COSPAR's last session, this new strategy was accepted. This strategy no longer requires an estimate of P_G. The new proposal suggests establishing, _a priori_, five categories of solar system missions, and for each category indicating what level of concern exists, and what quarantine measures would be activated for each category. This determination of categories does not, in my view, change the fundamental problem; the stipulation of what category and particular mission will be assigned will be based upon "advice from the scientific community" (in the United Stated probably the Space Science Board). For Mars, some sort of collective judgement will still have to be made, taking into account the planet's "friendliness to terrestrial organisms". This process of determining a judgement is, essentially, what went into establishing a P_G for Mars in the first place.

What is the current status of our knowledge about the environment on Mars relative to growth of terrestrial organisms? As the consequence of a post-Viking assessment of Viking data, the Space Science Board has reduced the P_G for Mars (NASA Publ., "Recommendations on Quarantine Policy", 1975). These recommendations were largely based upon finding no detectable organic compounds at the two landing sites (even in a protected area under a rock); the extremely oxidizing nature of the surface material; and the very high UV flux at the surface. All these facts point to an extremely harsh environment for living organisms from Earth. A note of caution however, viable cocci (bacteria) were brought back in Surveyor equipment by the Apollo 12 crew after several years on the Moon. The environment on the Moon is considered to be far more severe than that of Mars! It would be relatively easy to agree that some terrestrial organisms might survive (not necessarily reproduce) for a long time in some protected niche on, or in, Mars. As an extension of this line of reasoning, most scientists probably would agree the chances of terrestrial organisms eventually growing on Mars is exceedingly low, but their growth cannot be ruled out. If one must be sure of no growth, we would introduce no organisms into the atmosphere, and especially onto the surface of Mars.

What then are the quarantine issues of landing people on Mars? Assume the landing would accur prior to obtaining any relevant and substantially new Martian data. For example, data from the proposed MGCCO mission would alter thinking on this matter by providing a more detailed understanding of the water budget on Mars.

Two or more decades from now, will anyone care whether Mars is contaminated with terrestrial organisms? Almost certainly! While there appears to be a lessening in the fervor of those concerned with this problem, when the time comes, they will probably make an issue of any contamination. Some scientists, truly interested in comparative planetology, will not want to take the risk of introducing terrestrial organisms into the Mars environment. Finally, there will still be open the most fundamental questions, to laymen and scientists alike, of whether indigenous life exists on, or in, Mars. Scientists will probably attempt to insist on an exhaustive test of this idea, and to do so without introducing terrestrial organisms into the environment, assuming none will have been introduced prior to the manned mission.

In order to eliminate or to minimize the risks of contamination of Mars by a manned mission to that planet, should that be our policy, two approaches are available. First is absolute containment of all terrestrial biology while at Mars, and second is obtaining the requisite information prior to sending people. In principle, it would be possible to provide adequate technologies to achieve the former. People do work with very dangerous and highly infectious agents on Earth. An entire technology has been developed to contain these agents. Using a "sterilized" lander (as done with Viking), with adequate filter, vents, pressure regulators, etc., to prevent the escape of spacecraft atmospheric particulars upon human egress and during EVA on Mars. The EVA systems could not leak, as do all current systems. All this would be terribly expensive, but in the long run it may be the only sure approach and it will work only if no failures occur. An intermediate approach would be the use of automated or telepresent devices in place of the humans. The people might be kept in orbit or in a sterilized lander. The rovers and science instruments would, of course, all be sterilized. Since people on or near Mars are going to have to carry their own life support systems with them (either as spacecraft, EVA suits, landers, rovers, etc.) the design of all such systems have to incorporate this very stringent specification.

The second approach to helping eliminate the risk of contaminating Mars is fraught with serious difficulties. Prior to sending people to the surface, we must obtain the necessary information to assure that contamination cannot occur. In the past, many investigators have performed simulations to determine if organisms can grow in the Martian environment, both in the United States and in the Soviet Union. These efforts have shown the UV flux on Mars was the single most potent deleterious agent to terrestrial organisms. It could always be argued that almost any thin layer of shielding material could protect terrestrial organisms, even on Mars. In this connection, it may be of some use to again consider simulation studies. These should be done in the light of Viking data from Mars (e.g., if there is actually no organic material in the Martian environment, which terrestrial organisms could possibly maintain themselves there? What would they eat? If they were photosynthetic, how could they obtain their radiant energy while protected from the UV, etc.?) Some in-depth studies might be useful when the time comes to place a Mars mission into one of the five NASA planetary quarantine categories. In this regard, more information about what makes the Martian "soil" could be extremely useful. We do not know which, if any, nitrogen-containing compounds are in the surface material. If all the nitrogen on Mars is in the atmosphere, this would drastically reduce the kinds of terrestrial organisms that could grow there to a few species of blue-green algae and bacteria. What is the nature and distribution of the postulated oxidizing matter on Mars? As mentioned above, a thorough knowledge of where the water is on Mars, and what translocations of water occur would help immensely in putting limits on the prospects of contaminating Mars.

For these and similar reasons, the more information about Mars that can be obtained on precursor missions the easier the design specification for the manned missions would be. A series of carefully thought-through precursor missions designed to glean data to better assess the probability of contaminating Mars would probably be money and talent well spent. In the final analysis, it must be recognized that all data collected about Mars will serve for ever more accurate analytic assessment of whether or not terrestrial organisms can survive, and grow on Mars, thereby "contaminating" it.

In the end, the best case that can be made to allay the concerns of whose who would protect Mars from terrestrial organisms will be the design of a system that contains <u>all</u> terrestrial organisms. It is quite certain that analytic methods will never give the confidence that well-developed systems and carefully thought-through procedures will give. The pragmatic issue will ultimately be a weighing of the costs versus some ill-defined confidence level. It seems this sort of trade is the forte of NASA and its associated "advisors."

*Many of the ideas and issues in this paper are taken from the work of H. P. Klein. His help in formulating the positions taken herein are appreciated by the author.

N87-17767

MANNED MARS MISSION
SOLAR PHYSICS:
SOLAR ENERGETIC PARTICLE PREDICTION AND WARNING

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ABSTRACT

There are specific risks to the crew of the manned Mars mission from energetic particles generated by solar activity. Therefore, mission planning must provide for solar monitoring and solar activity forecasts. The main need is to be able to anticipate the energetic particle events associated with some solar flares and, occasionally, with erupting filaments.

A second need may be for forecasts of solar interference with radio communication between the manned Mars mission (during any of its three phases) and Earth.

These two tasks are compatible with a small solar observatory that would be used during the transit and orbital phases of the mission. Images of the Sun would be made several times per hour and, together with a solar x-ray detector, used to monitor for the occurrence of solar activity. The data would also provide a basis for research studies of the interplanetary medium utilizing observations covering more of the surface of the Sun than just the portion facing the Earth[1].

THE RISKS

Severe injury or even death can result from exposure to solar energetic particles. The Earth's magnetosphere and atmosphere protects us on the surface of the Earth. In space, beyond the Earth's magnetosphere, there is no equivalent protection except for that offered by the spacecraft itself.

Energetic particles exist in negligible fluxes at all times. However, occasionally the Sun emits bursts of these "solar cosmic rays" in fluxes sufficient to cause the above-mentioned risks. When this happens, the particles travel outward into the solar system at a large fraction of the speed of light, with a flux that falls off inversely with the square of the distance from the Sun. It takes at least 15 to 20 minutes for them to reach the Earth, corresponding approximately to at least 30 minutes to reach Mars. The "events" can last up to several days.

THE SOLAR PHENOMENA

The solar events are flares and erupting filaments.[2,3,4] A flare occurs in a solar active region around a sunspot group. A filament lies along a neutral line dividing regions of oppositely directed photospheric large scale magnetic field.

Flares can be directly observed in the hydrogen-alpha spectral line and regular solar flare patrols are made at several observatories around the world. These same observatories monitor the appearance, maturation, and death of sunspots and the general evolution of the sunspot cycle. The relationships between flares and sunspots is well enough understood so that flare forecasts and warnings now are issued daily, in the form of a prediction of the probability that a flare of a particular magnitude will occur and whether there will be energetic particles. These predictions are significantly better than guesswork.

Erupting filaments normally produce fewer energetic particles than do flares - although not always. It is only in recent years that the role of erupting filaments in the overall picture of solar activity has been even appreciated. Filaments are monitored along with sunspots but, although it is possible to assign a probability to filament eruption, no predictions are presently being made for filament eruptions.

Filaments and sunspots are rooted in the photosphere of the Sun so that they rotate with the Sun - which goes through one complete revolution every 25.5 days (sidereal). Thus, the activity would only be on the disk of the Sun for 12.75 days out of any given solar rotation - or half the time. This reduces the risk from isolated activity because the energetic particles are beamed out into space but cannot travel around the limb of the Sun. Unless extremely close to the limb, any activity behind the limb would have no life-endangering effects. However, there is often more than one activity center on the Sun so that the sum of risks from each individual center needs to be considered.

In conjunction with the monitoring of filaments and flares, the entire menagerie of solar phenomena is observed and mapped daily. Most data is collected from the Earth's surface, although energetic particles and whole-disk x-ray emission are observed from space. By the time of the manned Mars mission, there will be x-ray telescopes and coronagraphs in orbit that are dedicated to synoptic solar observations for the purpose of monitoring and forecasts.

SOLAR ASTRONOMY

Solar optical observations show the photospheric structure of the Sun and, with simple filters, map the large scale magnetic field structure, sunspots, filaments, and flares. During a large part of the transit, the side of the Sun exposed to the spacecraft will be invisible from the Earth. Observations of the sun in the hydrogen-alpha spectral region would therefore be necessary for solar flare/erupting filament prediction and warning. These observations would be made during all three phases of the mission. A second option would be to place at least two spacecraft in 1 AU orbits but evenly separated in longitude with respect to the Earth. These two options are evaluated in Table 1, with a preference being indicated for placing a solar telescope on the Mars Mission spacecraft due to transmission delays in the second option.

Additional solar observations are possible with more complex instruments - including magnetic field and velocity measurements. However, it is probable that these observations would be more efficiently and accurately made from Earth orbit spacecraft.

The inclusion of a 25- to 40-cm telescope for solar observations with supporting equipment such as special filters will make possible many interesting non-solar observations. The position of several of the solar system bodies may be tracked with exceptional accuracy. Planets can be observed to determine their albedo in the UV and visible ranges. Opportunities may occur for the observation of stellar occultation by the outer planets, and ephemerides should be developed for these observations. Other opportunities may occur for the observation of cataclysmic variables.

CONCURRENT OBSERVATIONS

In order to utilize the solar observations for research purposes, concurrent observations of other solar and interplanetary processes will be made from the manned Mars mission. The intent is to make a coordinated set of scientific observations aimed at solving specific problems. The problems that can be addressed deal with the interaction between solar particulate and electromagnetic emissions and the dynamics and evolution

TABLE 1

OPTIONS

Requirement	Observation and prediction of solar activity that may affect the Manned Mars Mission
Solution	Observation and monitoring of the portion of the Sun facing towards the spacecraft at all times during mission
Options	1. Solar observatory on spacecraft 2. At least two other solar observatories in orbit around the Sun at 120 and 240 degrees away from the Earth, at 1 AU.
Recommendation	Option 2 would permit observation of the entire Sun at all times. It would also require at least a 30 minute delay time between detected activity and notification of the Mars Mission. Therefore, option 1 seems to be the only viable choice.

of the Martian atmosphere. Of particular interest are upper atmospheric dynamics and chemistry because this will show what is happening to the water in the atmosphere.

Some coordinated experiments are shown in Table 2[1].

SPACE ENVIRONMENT SERVICES CENTER (DOC/NOAA/ERL/SEL)

Space environment services are provided by this agency (SESC) for the entire U.S. -- both civilian and military. Their activities are briefly summarized in Tables 3 through 7. Their services are essential to the Mars mission because they provide ongoing interpretation of the state of the Sun and both short and long term predictions. They would use the observations made from the Mars mission to assess current risks. They are analogous to the Weather Bureau providing weather conditions and forecasts to the whole continent as a supplement to the observations made and broadcast by a local TV station. The TV station can tell if there is a tornado right now, but the Weather Bureau provides warnings to be on the lookout for a tornado in 6 hours[5].

TABLE 2

COORDINATED EXPERIMENT PACKAGE (INCOMPLETE)

Instrument	Location
Magnetometer and plasma detector to measure interplanetary parameters	Orbiter
Low energy plasma analyzer to measure ionospheric dynamics/constituents	Free Flying Low Orbiter
Ionospheric topside sounder	Orbiter
Ionosonde	Surface
Lidar	Surface
Meteorological Instruments	Surface

TABLE 3

ABBREVIATED SUMMARY OF DATA SOURCES USED BY SESC

Type	Primary Source
Solar Patrol	
x-rays, 1 minute averages	Geostationary satellite
Hydrogen alpha, continuous	Ground observatories
Radio, 10.7 cm, 1 minute	Ground observatories
Solar Synoptic	
Hydrogen alpha	All are ground based
White light images	
Ca K-line images	
Helium 10830 images (shows coronal holes)	
Magnetograms (full disk and regional)	
Sunspot reports	
Solar mean field	
10.7 cm radio flux	
Energetic particle patrol	
Protons to 500 MeV, 1 minute averages	Synchronous orbit satellite
Miscellaneous	
Neutron monitor, 15 minute averages	Ground based
High latitude riometers (energetic protons), 15 minute averages	
Ionosondes, hourly (solar ionizing radiation)	

TABLE 4

SESC DATA DISTRIBUTION SYSTEMS

Telephone:

- FTS (Federal Telephone System)
- WATS (Wide Area Telephone Service)
- Commercial Telephone Service
- Dedicated Telephone Lines (Hot Lines)
- Recorded Information Numbers

Teletype:

- ATN (Astro-geophysical Teletype Network)
- AUTODIN (U.S. Government Teletype Service)
- Commercial Teletype Services
- Secondary Networks

Computer Links:

- Space Environment Laboratory Data Acquisition and Display System (SELDADS) Public User
- Dedicated Data Links

WWV Shortwave Broadcasts

Mail

TABLE 5

SESC OBSERVED INDICES AND ACTIVITY SUMMARIES

- Solar Active Region Summary Report
- Sunspot Number
- Flare (and Other Event) Lists
- Solar Neutral Line Analysis and Synoptic Maps
- Ten Centimeter Flux
- Solar Proton Events and Proton Flux
- SST Radiation Levels
- Geomagnetic A- and K- indices
- Substorm Log
- Sector Boundaries (at 1 AU)

TABLE 6

SESC ALERT CATAGORIES

SOLAR FLARES

- X5 (1-8 Angstrom X-ray Classification)
- X1
- M5
- 3B (Optical Classification)
- 2B
- 1B

MAGNETIC DISTURBANCES

- A >= 50 (real time A measured at Boulder)
- A >= 30
- A >= 20
- K >= 6 (real time K measured at Boulder)
- K >= 5 observed in successive three-hour intervals
- K >= 5
- K >= 4
- Sudden Commencement

RADIO BURSTS/NOISE STORMS

- 10 cm Radio Burst Greater Than 100 Flux Units
- 245 MHz or Noise Storm
- Type II and/or Type IV Decametric Emission

PROTON EVENTS

Proton Flux (E > 10 MeV) > 10 $cm^{-2}sec^{-1}sterad^{-1}$

TABLE 7

SESC PREDICTIONS

LONG TERM SOLAR ACTIVITY AND SOLAR RADIATION LEVELS

- Smotthed sunspot number (1 month - 10 years)
- Geomagnetic activity and ten-centimeter flux (1 month - 10 years)
- General level of solar activity (27 days)

SOLAR ACTIVITY - SHORT TERM

- Solar Flares (1, 2, 3 days)
- Solar proton events (1, 2, 3 days + post flare prediction)

SOLAR RADIATION LEVELS - SHORT TERM

- Ten-centimeter flux (1, 2, 3 days)

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N87-17768

PLANETARY SCIENCE QUESTIONS FOR THE MANNED EXPLORATION OF MARS

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ABSTRACT

A major goal of a manned Mars mission is to explore the planet and to investigate scientific questions for which the intensive study of Mars is essential. The systematic exploration of planets has been outlined by the National Academy of Science. The nearest analogy to the manned Mars mission is the Apollo program and manned missions to the Moon, but the analogy is limited. The case is argued here that Mars may have to be explored far more systematically than was the pre-Apollo Moon to provide the detailed information necessary if we plan to use any of the resources available on Mars. Viking missions provided a wealth of information, yet there are great gaps in our fundamental knowledge of essential facts such as the properties of Martian surface materials and their interaction with the atmosphere. Building on a strong data base of precursor missions, human exploration will allow great leaps in our understanding of the Martian environment and geologic history and its evolutionary role in the solar system.

INTRODUCTION

The exploration of the planets is among the most exciting and challenging endeavors of science and the human curiosity. The ability to send spacecraft to other bodies of the solar system is a recent development in human history. Surely the astronomers of many centuries ago would be envious of our opportunities.

A major stated goal of a manned Mars mission is to explore the planet and to investigate scientific questions for which the intensive study of Mars is essential. The orderly exploration of the solar system proceeds within a logical framework which has been carefully debated and is well documented.[1,2,3,4] But, that logical plan for the orderly exploration of the Moon and the planets and their moons can be disrupted by external events and budgetary contingencies. Still, for whatever the reason we are allowed to follow our fancy to Mars, the fundamental scientific problems and questions should be kept in the forefront of our planning. This mission is an integral part of the overall strategy for the study of Mars, building on the knowledge from preceding missions and forming the foundation for subsequent missions.

This paper will look at the logical plan for planetary exploration and compare that to a sketch of the exploration of the moon which culminated in the landing of twelve men on the Moon's surface. It will look at what we have learned from previous missions to Mars and discuss some of the specific scientific priorities for the exploration and study of Mars.

THE PLANETARY EXPLORATION MODEL

> The primary scientific goals in exploring the solar system are to determine the composition, structure, and environment of the planets and their satellites in order to define the present morphology and dynamics of the solar system and with the purpose of making major steps in understanding the processes by which planets formed from the solar nebula and how they evolved with time and how the appearance of life in the solar system is related to the chemical history of the system. The investigation of the interplanetary and interstellar medium is considered an intrinsic part of such an endeavor.
>
> Space Science Board 1975[1]

Models for the exploration of the planets have been presented by the SSB (Space Science Board of the National Academy of Sciences)[1,3] and Complex Committee on Planetary and Lunar Exploration[2] of the Space Science Board. The models identify three stages in the unmanned exploration of any planetary body: (1) Reconnaissance stage of flyby and hard lander missions; (2) Exploration stage of orbiter and entry probe missions; and (3) Intensive study stage of soft landers and sample returns. Missions involving humans represent a special case of intensive study as well as an advanced stage in themselves.

The solar system exploration strategy as stated by the SSB[1] follows the principle that the overall exploration of the solar system should advance more or less evenly for all of the planets, sending "reconnaissance" missions to all of the planets before we move into the "exploration" and "intensive study" of any of them. The principle is admittedly tempered by the technical difficulty of even the most rudimentary mission to distant outer planets. Consequently, the plan calls for advancing the exploration of selected inner planets while continuing the reconnaissance of the outer planets. A manned mission to Mars is a significant anomoly to this orderly exploration plan. The decision to undertake this mission will be made on many considerations in addition to purely scientific issues.

MANNED PLANETARY EXPLORATION BASED ON THE LUNAR EXPERIENCE

Human events and budgets affect the orderly course of planetary exploration. The Apollo program of lunar exploration is an example in which political events strongly influenced both the objectives and the pace. It is likely that the commitment to send humans to explore Mars will have much in common with the commitment to send humans to explore the Moon. It may be fruitful to examine the lunar exploration experience for lessons that can be applied to the manned mission to Mars.

The race to the Moon was conducted by the U.S. and U.S.S.R. along roughly equivalent developmental paths (Tables 1 and 2). Both programs progressed through the reconnaissance phase using flybys and hard landers, and through the exploration phase with orbiters and soft landers. Finally, the U.S.S.R. returned samples with three automated landers, and the U.S. returned samples and explored the Moon with six landing teams.

The exploration stage of lunar science was truncated by the political urgency to land men on the Moon before 1970. As a result, we still do not have high quality global maps nor do we have global mineralogical and chemical data. In some respects we have more complete global coverage of Mars than we do of the Moon because of the extended coverage of Mariner 9 and the two Viking orbiters. Only now, with attention returning to the Moon and with a lunar base a possibility, is a more systematic exploration of the whole Moon getting started with the proposed Lunar Geochemical Observer (LGO).

There are important lessons to be learned from the Apollo experience. Scientifically and developmentally, the Apollo missions were an anamoly. As long as the Apollo missions were totally self-sufficient and the objectives of the missions were primarily engineering objectives, there was little need for more detailed exploration. A lunar base will not be a totally self-sufficient system. It will need to use some of the resources available on the Moon. As a result, there is a need to resume the detailed global exploration of the Moon with LGO and to follow with

TABLE 1

PROFILE OF LUNAR EXPLORATION

YEAR	MISSION	FLYBY	HARD LANDING	ORBITER	SOFT LANDING	RETURN	MANNED FLYBY/ ORBITER	MANNED LANDING	COMMENTS
1959	Luna (2)	1	1						
o									
o									
o									
1964	Ranger (1)		1						
1965	Ranger (2)		2						
	Zond (1)	1							
1966	Luna (5)			3	2				
	Lunar Orbiter (2)			2					
	Surveyor (1)				1				
1967	Explorer (1)			1					
	Lunar Orbiter (3)			3					
	Surveyor (1)				2				
1968	Apollo (1)						1		
	Surveyor (1)				1				
	Zond (1)	1				X			
1969	Apollo (3)			1			1	2	
	Zond (1)			1		X			
1970	Apollo (1)						1		
	Luna (2)				2	X			One Rover
1971	Apollo (2)							2	One Rover
	Luna (1)			1					
1972	Apollo (2)							2	Both Rovers
	Luna (1)				1	X			
1973	Luna (1)				1				Rover
o									
o									
o									
1976	Luna (2)				1	X			
o									
o									
o									
199X	LGO								

TABLE 2

PROFILE OF MARTIAN EXPLORATION

YEAR	MISSION	FLYBY	HARD LANDING	ORBITER	SOFT LANDING	RETURN	MANNED FLYBY/ ORBITER	MANNED LANDING	COMMENTS
1965	Mariner	1							
o									
o									
o									
1969	Mariner	2							
o									
o									
o									
1971	Mariner			1					
	Mars (2)		2	2					Landers Failed Orbiters Failed
o									
o									
o									
1973	Mars (4)			4	1				One Orbiter Failed One Lander Missed
o									
o									
o									

1992	MGCO			X					
1996?	Sample Return				X	X			
?	Russian Mars Probe			?	?	X	X ?		
o									
o									
o									
?	Manned Mars Mission			X	X	X	X	X	

intensive study of selected areas, with possibly automated sample returns.

The exploration phase for Mars may have to be far more systematic and thorough than was the pre-Apollo exploration of the Moon. The thoroughness of the prelanding exploration will depend on the extent to which the proposed manned Mars mission will use systems that depend on being recharged using locally available resources on the Martian surface. The Mars manned exploration scenario is made somewhat simpler if it can regenerate at least some of its simpler systems at Mars. To enable any such simplification, we will need a detailed knowledge of the Martian surface and certain knowledge of the regions of Mars that will have proven resource potential. The pace of Martian exploration may be somewhat slower than was possible for the Moon because of the infrequent opportunities and the technical difficulty. Nevertheless, as evidenced by Mariner and Viking orbiters, well planned capable precursor missions such as those planned for the observer class missions of the planetary exploration program can add immensely to our knowledge base.

OUR PRESENT KNOWLEDGE OF MARS

The reconnaissance stage of the exploration of Mars was accomplished by the Mariner flyby missions of the 1960's and Mariner 9 Mars orbiter in 1971. After several unsuccessful Russian attempts to visit the planet, the U.S. continued systematic exploration with the Viking landers and orbiters in the late 1970's. With the data from these exploratory missions, our understanding has progressed to the level to support the beginning of the detailed exploration of the planet.

With Viking, a more intensive investigation of Mars began. It was discovered that there are only very low levels of hydrocarbons and no direct evidence of life(4) (Table 3). The tenuous atmosphere is depleted in nitrogen and generally enriched in the heavy isotopic species of the atmospheric gases. We have only the simplest idea of the composition of the materials on the Martian surface(5) (Table 4). The polar caps appear to have permanent water ice and seasonal carbon dioxide ice. Water, while present in the atmosphere, is present only in small amounts. A major scientific question is the evolution and fate of the Martian atmosphere and the history of the volatiles(5) (Table 5).

TABLE 3

VIKING MARS SCIENCE HIGHLIGHTS

-- No definite evidence for biological activity in soil, despite unusual chemical reactions produced in life detection experiments.
-- Surface rocks resemble basalt; surface chemistry resembles altered basalt.
-- Polar cap in North made largely of water ice, with lesser amounts of solid carbon dioxide.
-- Isotope ratios of carbon and oxygen in the atmosphere resemble those in the Earth's atmosphere.
-- Loss of nitrogen to space has produced isotopic ratios on Mars that are different than those on Earth; the heavy isotope of nitrogen (^{15}N) has been preferentially retained.
-- Abundant erosional channels on surface suggest that Mars could have had a denser atmosphere in the past and may have had liquid water on its surface.
-- Noble gas abundances (Ar and Ne) suggest that Mars has a lower volatile content than either Venus or Earth.
-- Red color on the surface is due to oxidized iron.
-- Soil is fine grained and cohesive, like firm sand or soil on Earth.
-- Water compounds and sulfur compounds are present in soil.
-- Small-scale land forms formed by aeolian (wind) processes.
-- Typical surface temperatures range from about -84° C at night to -29° C in the afternoon.
-- Surface pressure of the atmosphere (only 0.8 percent of the Earth's) varies seasonally in accordance with the sublimation of the polar caps.
-- Martian moons (Phobos and Deimos) are grooved, indicating that incipient fracturing has occurred; they are heavily cratered and may be captured asteroids.

TABLE 4

COMPOSITION OF MARTIAN SURFACE MATERIALS[5]

	CHRYSE FINES	CHRYSE DURICRUST	CHRYSE DURICRUST	UTOPIA FINES	ESTIMATED ABSOLUTE ERRORS
SiO_2	44.7	44.5	43.9	42.8	5.3
Al_2O_3	5.7	n/a	5.5	n/a	1.7
Fe_2O_3	18.2	18.0	18.7	20.3	2.9
MgO	8.3	n/a	8.6	n/a	4.1
CaO	5.6	5.3	5.6	5.0	1.1
K_2O	<0.3	<0.3	<0.3	<0.3	
TiO_2	0.9	0.9	0.9	1.0	0.3
SO_3	7.7	9.5	9.5	6.5	1.2
Cl	0.7	0.8	0.9	0.6	0.3
	----		----		
SUM	91.8	n/a	93.6	n/a	

TABLE 5

COMPOSITION OF ATMOSPHERE AT MARTIAN SURFACE[5]

Carbon dioxide	95.32%
Nitrogen	2.7%
Argon	1.6%
Oxygen	0.13%
Carbon monoxide	0.07%
Water vapor	0.03%
Neon	2.5 ppm
Krypton	0.3 ppm
Xenon	0.08 ppm
Ozone	0.03 ppm

ISOTOPE RATIOS

RATIO	EARTH	MARS
C^{12}/C^{13}	89	90
O^{16}/O^{18}	499	500
N^{14}/N^{15}	277	165
Ar^{40}/Ar^{36}	292	3000
Xe^{129}/Xe^{132}	0.97	2.5

C-2

MARS SCIENCE GOALS AND PRIORITIES

COMPLEX(2) divides the strategies for the exploration of the inner planets into two groups, one for bodies without atmospheres and one for bodies with atmospheres. Their recommendation was to make the triad of planets with atmospheres the focus of the exploration attention in the period of 1977-1987.

The three planets Earth, Venus, and Mars were seen by COMPLEX to represent a "natural experiment in planetary evolution." The first experiment produced Earth with its abundant volatiles and free water in the oceans and atmosphere. Water on the Earth plays a central role in the morphology of the planet and in the origin and sustenance of life. The second experiment produced Venus. Venus has been nearly completely degassed but has very little water in its atmosphere. Presumably, many of the light volatiles, including water, have been lost to the intense heating that characterizes the Venus surface. Mars is the product of the third experiment. Mars has lost much of its atmosphere but has not been thoroughly degassed as a planet. Substantial amounts of liquid water have clearly played a role in the formation of the surface morphology of the Martian surface, yet no liquid water is known to exist on the surface today (and probably none has been on the surface since very early in the planet's evolution). Is Venus a more thoroughly evolved Earth? Has Mars been cut short in its evolution and does it still retain the potential to develop into an Earth-like planet that in some future era may also be hospitable to life?

The science objectives for the study of Mars in the post-Viking era are primarily geological and geophysical. With the first order knowledge that present life on Mars is unlikely, the objectives for further study are defined by the need for primary information about the planet Mars and its atmosphere which is essential to understand its place in the evolution of the solar system. This same information is important in the debate over the probability of past life on Mars. After the Earth, Mars is the next-most-likely planet for the support of life. Although the essential elements for life are present, no proof of present life has been forthcoming.

Careful studies of the surface materials and the evidence therein for interactions between the atmosphere and the Martian surface may prove extremely valuable for understanding the evolution of the Martian atmosphere. Precise dating of the surface rocks and soils will allow the major geologic processes to be put into chronological sequence. Stratigraphy of carefully recovered cores and samples from layered deposits will give insight into the more recent effects of geological processes and the nature of the processes themselves.

A list of science objectives has been assembled by the Solar System Exploration Committee (SSEC)[4] for the exploration of Mars. The list (Table 6) defines the basic science tasks on Mars. Many of the tasks require global perspective and are appropriate for precursor orbiters and soft landers. Other tasks are clearly appropriate for and could benefit greatly from the human presence on the surface of Mars. The missions to Mars defined for the NASA Core Program of Planetary Missions address the objectives for global exploration of Mars. Mars Geochemical Climatological Orbiter (MGCO) will gather data on the global chemistry and the global atmospheric circulation. The Mars Aeronomy mission will explore the intricacies of the upper atmosphere and its interaction with the solar wind. The Mars Network Mission will collect fundamental geophysical information on the planet as well as surveying the composition of surface material and taking meteorological data using multiple penetrators and their surface stations.

Clearly, the intensive study of the local materials is the area of science most greatly aided by the presence of human explorers. The global scale objectives are best done with orbiting instruments and observers, but they too are greatly aided by having "ground truth" established for them by human explorers.

CONCLUSIONS

Comparison of the Apollo and manned Mars missions leads to the conclusion that their analogy is a limited one. We will need to do a far more thorough exploration of Mars at a global level and intensive study at a local level before we launch a manned mission than was done for the Apollo missions. The chief discriminator is the need to depend on any of the Martian resources as an essential element of the manned Mars mission plan. While there is a good start in uncovering the mysteries of Mars, there are many fundamental pieces of information that are lacking, especially concerning the exact nature of the surface materials and their

TABLE 6

PRIMARY SCIENCE OBJECTIVES FOR THE EXPLORATION OF MARS[4]

-- Characterize the internal structure, dynamics, and physical state of the planet.

-- Characterize the chemical composition and mineralogy of surface and near surface materials on a regional and global scale.

-- Determine the interaction of the atmosphere and the regolith.

-- Determine the chemical composition, distribution, and transport of compounds that relate to the formation and chemical evolution of the atmosphere.

-- Determine the quantity of polar ice and estimate the quantity of permafrost.

-- Characterize the dynamics of the atmosphere at a global scale.

-- Characterize the planetary magnetic field and its interaction with the upper atmosphere and the solar wind.

-- Characterize the processes that have produced the landforms of the planet.

-- Determine the extent of organic chemical and biological evolution of Mars and explain how the history of the planet constrains these evolutionary processes.

-- Search for evidence of the signature of the early atmosphere in the ancient sediments.

interaction with the atmosphere. Manned exploration of Mars will be most helpful for the detailed understanding of these phenomena at various local areas of Mars.

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N87-17769

SITE SELECTION FOR MANNED MARS LANDINGS: A GEOLOGICAL PERSPECTIVE

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ABSTRACT

Issues relating to the selection of initial landing sites for manned Mars missions are discussed from a geological viewpoint. The two prime objectives for initial manned exploration should be the youngest, unambiguous lava flows (to tie down the late end of the cratering history curve for Mars) and old highland crust, which is best sampled and studied through the use of large impact basins as natural, planetary drill-holes. Exploration of these two sites will provide data on martian chronology, volcanism, impact processes and gross chemical structure that will enable a first-order global synthesis through integration of these results with the global remote-sensing data already in hand from Viking and that to be provided by the Mars Observer Mission.

INTRODUCTION

A system to deliver men to the surface of another planet implies scientific capabilities many times greater than that of an automated, unmanned exploration spacecraft (see Taylor, 1975). Although site selection for the initial manned landings on Mars will be guided by many complex factors, the geological perspective is the purpose of this contribution. Other scientific disciplines, such as geophysics, may be interested in different sites. For the purpose of this discussion, I will concentrate on potential landing sites that will fundamentally contribute to deriving a detailed knowledge of martian geologic history. This involves selecting landing sites that span the vast ranges of time and processes that we observe on the surface of Mars.

A GEOLOGIC RATIONALE FOR MARTIAN LANDING SITE SELECTION

Although numerous studies are conducted during mannned missions, from a geological point of view, we are interested primarily in: 1) absolute ages of regional stratigraphic units; and 2) the composition, lithology, and possible paleontology chemistry, of rocks that make up the martian surface. Geological mapping based on returned photographs (e.g. Scott and Carr, 1978) has shown that Mars is a complex, heterogeneous planet, with regional geologic units that span the range from heavily-cratered terrain (representing the oldest units) to very sparsely-cratered lava flows (representing some of the youngest units).

In selecting landing sites to address the global history of Mars, a general strategy might be based on establishing two end points for martian geologic history. First, we would like to know the absolute age of the youngest martian lava flows. This would answer the questions: When did martian volcanism cease? By calibrating the lower end of the planet-wide crater-frequency curve, the absolute age of most martian geologic units could be derived. Moreover, sampling lava flows not only gives us direct information concerning the composition of martian surface units, it also indirectly provides data on the probable chemical and petrologic nature of the martian mantle. Second, a landing site to sample and investigate the oldest martian geologic units would provide data at the opposite end of the age spectrum. This is best accomplished on Mars, as it is on the Moon, by sampling the rims of multi-ring basins, which are large impact craters that have excavated many kilometers into the crust of Mars. We therefore, have an opportunity not only to obtain samples of the ancient martian crust for age dating, chemistry and petrology, but also the potential to establish any vertical stratigraphy that may exist within the crust by reconstructing the basin impact target. Additionally, all martian basin landing sites appear to be partially embayed by numerous geologic units of diverse ages. Thus, a manned mission to one of these sites could not only provide data for early martian history, but also fill in gaps by sampling and dating some intermediate age units as well.

These two prime objectives, to investigate both the latest and earilest martian geologic units, will enable global extrapolations that should give us a fairly complete understanding of martian geologic history. The intermediate phases of martian history could be reconstructed by carefully integrating global photographic and remote-sensing compositonal data (to be provided by the MGCO mission), with the results of manned sample return and geologic exploration. However, detailed knowledge of martian geology will probably come only after many generations of surface exploration. Such a long range plan is beyond the scope of this paper; the following section will briefly describe some selected landing sites that will maximize the geologic return of brief series of manned missions and will give a broad knowledge of the geologic history of Mars and the processes that have shaped its surface.

SOME RECOMMENDED MARTIAN LANDING SITES

Mars is such a geologically diverse and complex planet (e.g. Mutch et. al., 1976; Carr, 1981), that to compile a list of geologically interesting landing sites to inventory all the processes that have operated during the planet's history would be an exercise in futility. Instead, this discussion will be confined to the two prime objectives listed above; some additional geologic "targets of opportunity" are presented, in addition to the prime sites, in Table 1.

Prime Objective 1 - The youngest martian lava flows

The Tharsis province of Mars possesses some of the most spectacular volcanoes observed in the solar system. It was recognized early in martian exploration that vast regions of this area contain few superposed impact craters, indicating a geologically-young age (Carr,1973; BVSP, 1981). Through detailed mapping and crater-counting of lava flows in the Tharsis region (Schaber et. al., 1978; Plescia and Saunders, 1979; Morris, in press), the youngest flows may be recognized (Fig. 1).

The smooth lava plains of the uppermost member of the Olympus Mons Formation (Scott and Tanaka, 1985) have the lowest cumulative crater density of all Tharsis flows (Number of craters $\geq$ 1 km diameter = 78 / 10^6 km^2; Morris, in press). Moreover, they are unambiguous lava flows, displaying flow lobes and pressure ridges (Fig. 2). A mission to this site would also have the opportunity to sample the basal scarp of Olympus Mons, the youngest shield volcano of the Tharsis province. The elevation of this site is between 2 and 3 km above the mean planetary level; if this elevation is too high for a spacecraft to obtain the necessary aerobraking capability, an alternate site exists at about 20^oN, 150^o (Table 1; Fig.2). This site is near the 0 km contour on the global topographic map. It consists of lava flows only slightly older than the previously mentioned Olympus flows ($N \geq$ 1 km ~ 100-200/ 10^6 km^2; Morris, in press). In addition to these young lavas, a carefully selected site at this locality could investigate both the distal margins of an ejecta-flow impact crater and the enigmatic aureole deposits of Olympus Mons (Fig. 2), for which diverse, and mostly unconvincing, origins have been proposed (see review in Carr,1981).

TABLE 1

GEOLOGICALLY-PRIORITIZED LIST OF LANDING SITES FOR MANNED MARTIAN EXPLORATION

SITE	OBJECTIVES	COMMENTS
1. Base of Olympus Mons[1]	a. youngest recognized martian lava flows b. Olympus Mons flows; basal scarp on mountain	. elevation may be too high for sufficient aerobraking . lava flows are somewhat older than site 1
2. West of Auureole[1] (20^{o}N, 150^{o})	a. young lava flows b. aureole deposits c. ejecta-flow impact crater	. lava flows are somewhat older than site 1
3. Argyr basin[1] (47^{o}S, 30^{o})	a. basin massifs and ejecta b. intercrater plains material c. eolian deposits and landforms	. wide diversity of features to investigate . geophysical net for Tharsis plateau
4. Isidis Basis[1]	a. basin massifs and ejecta b. intercrater plains; basin-fill lavas c. distal edges of Syrtis Major shield flows d. small channels (fluvial drainage?)	. good for establishing global, geophysical net in conjunction with sites 1 and 2
5. Chasma Boreale[2]	a. water ice and other polar volatiles b. layered terrain; nature and origin c. northern plains material-periglacial processes and volcanic deposits	. address climatic problems
6. Canyonlands[2]	a. stratigraphy in walls b. debris and superposed deposits on canyon floor c. processes of canyon formation	. complex site, probably requiring many weeks of surface stay time
7. Outwash channels[2]	a. fluvial (?) deposits b. origin and process	. ability to make long traverses (several 100 km) desirable
8. Fretted terrain[2]	a. process and origin b. possible section into Martian crust	. ability to make long traverses (several 100 km) desirable
9. Highland Patera[2]	a. highland volcanism (ash shields?)	. limited objectives site
10. Mangala Vallis region[2]	a. massive, unconsolidated deposits that straddle hemisphere boundary; origin under debate (ash flow?; polar deposits?)	. relevant to climatic questions or recent volcanism problems

NOTES:
1. Numbers along landing sites localities indicate relative priority, except that sites 1 and 2, and sites 3 and 4 are interchangeable, depending upon operational constraints.
2. No specific sites are given for these targets of opportunities, could be sites for follow-up missions after initial landings.

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Figure 1. Young lava flows embaying the distal flanks of Olympus Mons (O). A site selected in a carefully chosen region (X) could sample both the young flows and portions of the Olympus Mons basal scarp. North at top; Viking subquad MC-9 NW.

Figure 2. Alternate site for young lava flows. Impact crater at top (C) displays ejecta flow morphology. Young lavas embay aureole deposit (A); this is a multi-objective site. North at top; Viking subquad MC-8 NE.

Either of these two sites would give us samples of the youngest unambiguous martian lava flow. As such, they would calibrate the planet-wide crater-frequency curve and enable us to extrapolate the results from this site to volcanic plains across the martian surface, which cover over 60% of the surface area of the planet (Greeley and Spudis, 1981).

Prime Objective 2 - The Ancient Martian Crust

Experience with Apollo lunar surface exploration has shown that investigations of multi-ring basins and their ejecta provide good strategies to reconstruct the composition and structure of planetary crusts. The cratered terrain hemisphere of Mars displays numerous basins, in preservation states ranging from near-pristine (e.g. Lowell; Wilhelms, 1973) to almost totally-obliterated (Schultz et. al., 1983). By investigating and sampling these basins, we can learn about the processes involved in basin formation, the age and composition of the martian highlands, and crustal stratigraphy and structure.

The Argyre basin in one of the best preserved, large (800 km diameter) martian multi-ring basins (Fig. 3; Table 1). A landing in this location would have several objectives. The prime sampling objective would be the basin massifs (Fig. 3). These mountains consist of both uplifted and rotated crustal blocks and /or basin ejecta, excavated from many kilometers depth (a model calculation suggests maximum depths of excavation for an Argyre-size basin at 40- 50km, extrapolated from the relation for lunar basins given in Spudis and Davis, 1985). Additionally, knobby-deposits (Fig.3) may well consist predominantly of primary basin ejecta, by analogy with similar deposits observed around the lunar Orientale basin (e.g. Head, 1974). Old plains material partially embays Argyre basin terrain; these units may consist of old volcanic flows that have resurfaced almost 50% of the martian cratered terrain hemisphere (Greeley and Spudis, 1981). Finally, a variety of eolian features, such as dune fields and etched terrain, occur within Argyre; both the morphology and process of eolian activity could be investigated at this site.

An alternate highland/basin site is the Isidis basin (Fig. 4; Table 1). This basin (1500 km diameter) may have been excavated to depths of 70 to 80 km into the martian crust. Objectives at this site consist of basin massifs as described above, basin-filling lavas, and the distal

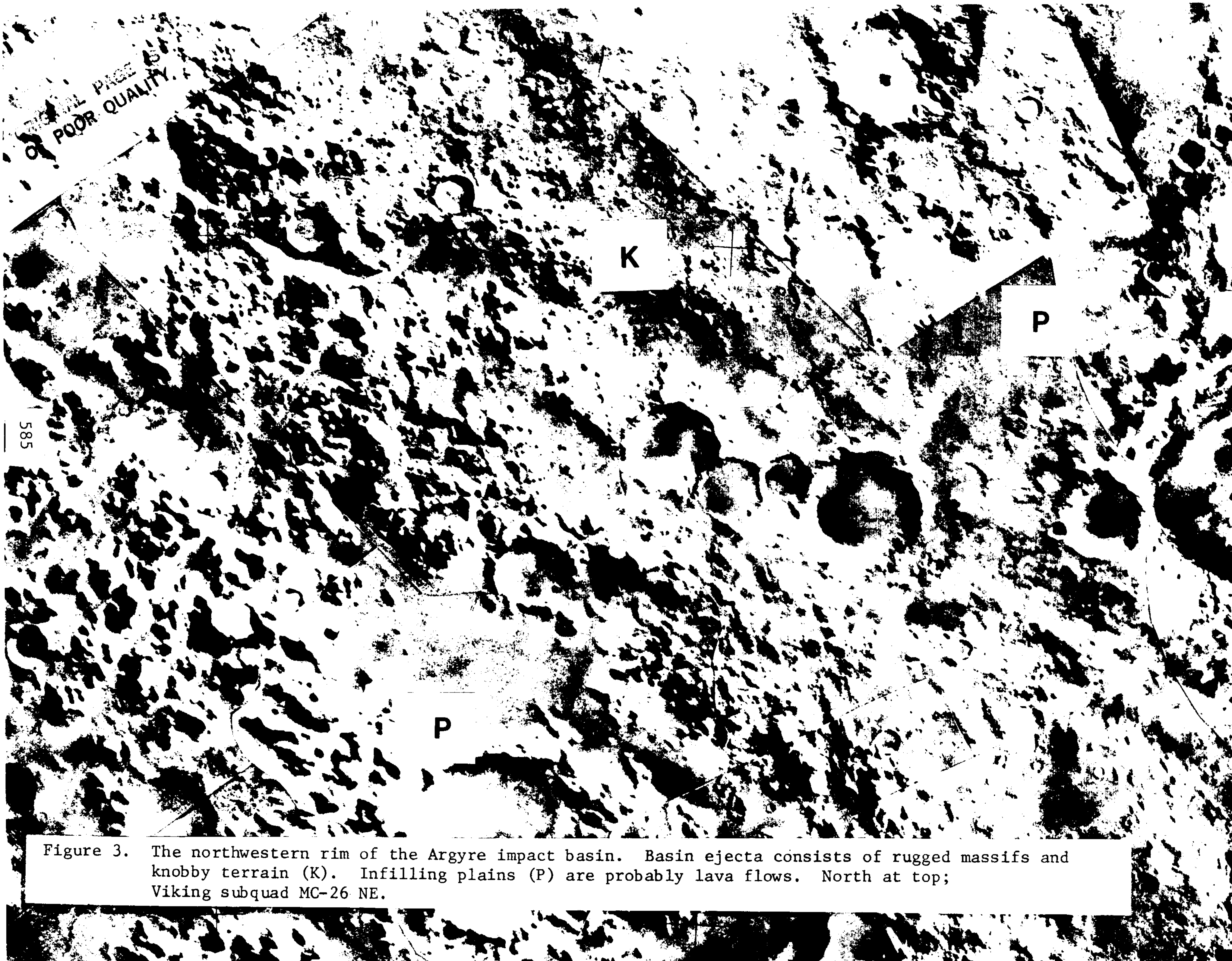

Figure 3. The northwestern rim of the Argyre impact basin. Basin ejecta consists of rugged massifs and knobby terrain (K). Infilling plains (P) are probably lava flows. North at top; Viking subquad MC-26 NE.

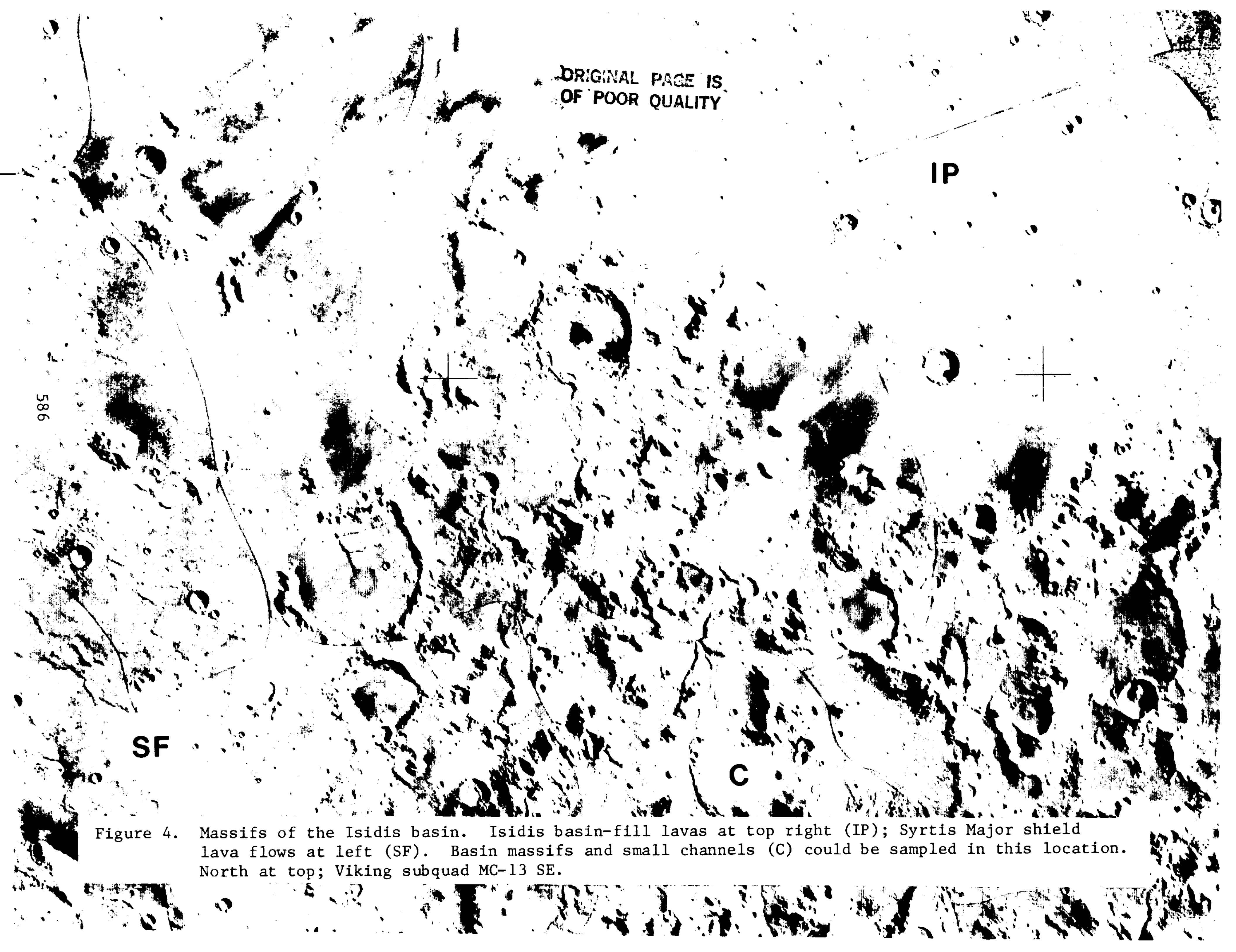

Figure 4. Massifs of the Isidis basin. Isidis basin-fill lavas at top right (IP); Syrtis Major shield lava flows at left (SF). Basin massifs and small channels (C) could be sampled in this location. North at top; Viking subquad MC-13 SE.

flows of the Syrtis Major shield volcano (Schaber, 1982). Additionally, some drainage channels occur within the rugged basin terrain (Fig. 4); another goal of this site would be to establish the nature of these channels, which may be of fluvial origin (Carr, 1981). An advantage of the Isidis site over the Argyre basin site described above is its near-antipodal location to the young volcanic sites described earlier; the placement of a geophysical station in both the Tharsis and Isidis regions might enable a determination of the existence and properties of a martian core.

Additional Sites of Geologic Interest

Six additional regions on Mars of geologic significance are listed in Table 1. As mentioned previously, Mars is such a complex planet, that a list hundreds of entries long could easily be given. In this tabulation, I have attempted to rank other targets only in terms of how they will help us address key issues in martian geologic history. After satisfying the two prime objectives, perhaps the most interesting site from both a geological and resources viewpoint is the north polar region (Table 1). Geologically, the polar layered deposits contain a record of alternating deposition and quiescence that is invaluable in terms of recent martian history. In terms of resources, the permanent polar cap is composed of water ice (Kieffer et. al., 1977). This resource is directly available at this site for life support at a permanent base and for propulsion uses.

The list presented in Table 1 is not meant to be definitive in any way. This is only an outline of site selection targets that will provide answers to several key questions regarding Mars. If the lunar experience is a guide, this initial exploration plan will probably raise many more questions than it answers.

CONCLUSIONS

The site selection strategy proposed here will address two key fundamental issues in martian geology: 1) the timing and composition of martian volcanism; and 2) the nature of the martian highland crust. Although detailed knowledge of martian geologic history will take decades of manned surface exploration, these initial manned landings will, at the very least, enable a formulation of the proper questions and provide a framework within which the evolution of Mars as a terrestrial planet can be understood.

ACKNOWLEDGEMENTS

I thank E. C. Morris, D. H. Scott, and G. A. Swann for helpful comments on this paper.

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SECTION VI

LIFE SCIENCES / MEDICAL ISSUES

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N87-17770

ADAPTATION AND READAPTATION MEDICAL CONCERNS OF A MARS TRIP

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ABSTRACT

This paper deals with the ability of the human body to adapt to microgravity environments and to later readapt to a gravity environment. Issues specifically relating to the effects of long-duration space flight on the adaptation/readaptation process are discussed. The need for better health prediction techniques is stressed in order to be able to better anticipate crew health problems and to perform corrective actions. Several specific examples are discussed of latent diseases which could occur during a long duration space mission, even after having subjected the crew to thorough pre-mission checkups.

The paper also discusses the need for learning how to prevent or ameliorate such problems as space adaptation syndrome, bone and muscle (and possibly tissue) atrophy, immune system atrophy, and heart arrythmias. It briefly addresses the implications of the age of the crew, the influence of an on-board low level gravity field, and drugs as factors in the adaptation/readaptation process.

INTRODUCTION

Anticipating a trip to Mars will first engender excitement and then concern as the enormity of the task is realized. As this is written no human has lived in a microgravity environment for a period as long as it will take to reach and return from Earth's sister planet.

Before crews are selected for the historic event, Aerospace Medicine will need to develop much more skill in the art and science of predictive medicine. In no other situation has medicine been called upon to certify that an individual will be healthy enough to perform full duty for two years following the pre-flight examination. The annual physical examination is well ingrained in the practice of aerospace medicine and is used to certify pilots. But, it is not unknown for a seemingly healthy pilot to develop an incapacitating illness between examinations. Generally, illness presents itself with warning signs, often first realized on awaking in the morning so that the pilot can call in sick and in that way avoid illness during the trip. Most airplane flights are relatively short so their duration often does not exceed the prodromal phase and minor symptoms of an illness. NASA and the USSR have had experience certifying astronauts and cosmonauts for mission periods up to eight months. In orbital space flights, any mistake could be rectified by having the crew return to Earth after the onset of symptoms from a dangerous illness or accident. On a Mars mission, the medical luxury of returning for hospitalization will not be possible. Therefore, a predictive system which will foretell illness two years in advance should be developed even through significant health maintenance capability should exist in-flight.

LONG TERM HEALTH PREDICTION TECHNIQUES MUST BE DEVELOPED USEFUL FOR A MISSION WHICH COULD LAST OVER A YEAR AND FOR WHICH EARLY RETURN TO EARTH IS NOT FEASIBLE

When the Mars craft launch acceleration slows, the crew will sense weightlessness (in reality, microgravity) which could continue until they begin the braking maneuvers needed to obtain a Mars orbit. The sudden unloading of the otolith as well as other physiological changes is followed by what is know as space adaptation syndrome (SAS). Crews report that the acute symptoms of SAS disappear in two to three days. Thus it will not be a major problem for a long duration Mars mission. However, long duration USSR crew members apparently report annoying returns at random times of the disorientation which they felt on entering weightlessness. It is probable that when NASA counters the acute symptoms, these later manifestations will be prevented. It is possible also that if the craft has some acceleration, none of the space adaptation syndrome symptoms will appear.(1)

SOLVING THE SPACE ADAPTATION SYNDROME IS NOT A PREREQUISITE TO A MANNED MARS MISSION

During a prolonged mission bone and muscle atrophy will occur if not prevented. In the case of muscles and bone, proper loading, stretching and use will prevent this problem. At present a low cost method to accomplish this has not been found. The USSR reportedly requires long duration crews to spend several hours each day in body conditioning maneuvers. We have no proof that these work or are even helpful. However, neither the U.S. nor the U.S.S.R. has flown people for long durations without the (presumed) countermeasures being taken.

Bed rest studies suggest that it takes four hours of vigorous walking each day to prevent negative calcium balance. Four hours daily is obviously too long a duration for a physical fitness program. Drugs are being looked at as a possible way to prevent bone atrophy. In micro-gravity, if a drug enhanced bone formation it could stimulate areas where increased calcium deposition is harmful, e.g. skull where calcium addition could cause damage to the cranial nerves. Nature relies on the stress and strain of everyday activity in gravity to signal the location of bone formation. Any failure to faithfully reproduce the gravity conditions could encourage bone formation in the wrong areas. Thus, development of a drug to correct this problem may be difficult. An ideal drug would keep the skeleton exactly as it was before the mission. This might interfere with the other major function of bone. Bone is used as a source of metabolic calcium for periods when calcium is not being added to the system from food. Additionally, the rate of calcium absorption is never ideal and calcium blood levels would go too high after a calcium containing meal if there were no way to rid the plasma of the calcium entering from the gastrointestinal tract. The kidney can do part of the job but its responses are relatively slow. The skeleton is used for this purpose. Any drug which hinders bone metabolic activity would likely result in hypercalcemia during the gastrointestinal absorptive periods and hypocalcemia when gastrointestinal calcium absorption ceases.(2)

NASA researchers are actively investigating the muscle and bone changes produced by microgravity. It can be anticipated that significant progress will be made before it is time for the Mars mission if a systematic investigative road-map is implemented. Unfortunately, these atrophies are best studied in microgravity and there is no plan for long duration U.S. missions until the Space Station. U.S. researchers are thus confined to ground based studies or experiments which look at the problem for periods of only about a week, the maximum planned duration of Shuttle flights. The USSR has longer missions. A cooperative program using the advanced technology and equipment of U.S. researchers during USSR Space Station missions would be mutually advantageous.

LEARNING HOW TO PREVENT OR AMELIORATE MICROGRAVITY INDUCED BONE AND MUSCLE ATROPHY IS AN IMPORTANT MEDICAL PREREQUISITE TO A MARS MISSION

NASA is just now learning the extent of the heart mass changes produced by the changed cardiac dynamics of microgravity living.(3) It is known that the microgravity cardiovascular adaptation causes undesirable symptoms on return to gravity. The cardiovascular changes are, in one sense, normal adaptation to the microgravity environment and, again a normal adaptation upon return to a gravity environment. NASA is beginning to learn also that the spontaneous adaptation to microgravity which causes changes in heart size may be associated with cellular changes in the heart which make the heart less stable electrically and thus more sensitive to arrythmias. EVA crewmen who never before showed a tendency to develop cardiac arrythmia can spontaneously develop them during or following an EVA. Whether this would be true also in the low G of Mars is unknown. This phenomena can be studied during the Shuttle era if plans are made to include regular cardiac monitoring during EVA.

LEARNING MORE ABOUT THE ELECTRICAL INSTABILITY OF THE HUMAN HEART ASSOCIATED WITH EVA ACTIVITIES IS AN IMPORTANT NASA GOAL

There are other atrophies associated with microgravity living. An interesting example is the decrease and leveling out in the number of circulating red blood cells which occurs early during a space flight and seems to continue throughout missions as long as six months(4). The reasons for this atrophy is unknown. The operational impact of a decrease in the circulating red blood cell mass is not very great, but unknown now is whether the bone marrow should respond normally if a crew member were to have a hemorrhage great enough to require a response in the bone marrow, although clearly the marrow is replenishing red cells once the lower level of red cell mass is reached. A large hemorrhage might come from the laceration of an accident or from a bleeding ulcer. Either way, the bone marrow would have to respond to prevent serious consequences. There are some who believe that the red cell decrease may be the most obvious example of a more extensive atrophy of body cells brought on by exposure to microgravity. Whether microgravity induced atrophy of other areas is of importance to the well being of a crew member is unknown at this time.

A GENERALIZED ATROPHY OF TISSUE CELLS MAY BE PRESENT DURING MICROGRAVITY LIVING. THE OPERATIONAL SIGNIFICANCE OF THIS NEEDS TO BE LEARNED FOR CONTINUED SUCCESS OF MANNED SPACE ACTIVITIES

Malignancies are another class of diseases for which medicine has yet to develop good predictive techniques, with the possible exception of cervical carcinoma. Certainly no one thinks it unusual when an individual develops clinical cancer a few months after being called physically fit by the examining physician. A single example makes this point. There existed an individual who as a research subject did much to enhance medical science. His esophagus was destroyed by lye ingestion during childhood. To allow him to eat, an accessible gastrostomy was placed on the abdomen. The physician/scientist caring for this patient used the gastrostomy for research studies several times each year. After a series of studies, the subject returned home for periods up to six months. Two months after a study series, the subject called the scientist to report bleeding that day from the exposed gastric mucosa. This had not previously happened. Brought immediately to the medical center, a small ulcer was visible. On biopsy it proved to be gastric cancer. Practicing physicians can document similar situations with cancer of the breast and lung appearing soon after negative chest x-rays and mammograms. Thus, it is desirable to have ways to predict cancer well before it is currently clinically recognizable. The potential for malignant change could be greater during a space flight because of increased exposure to radiation.

PREVIOUSLY UNDIAGNOSED MALIGNANT DISEASE HAS TIME TO BECOME CLINICALLY SIGNIFICANT DURING A MARS TRIP

The crew living in what amounts to an isolated state would have less problem with infectious disease except from those pathogens brought with them. The human adapts to isolation by a gradual decrease in the immune surveillance system since it is not called upon daily to respond to new disease threats. The immune system atrophies much as an unused muscle atrophies. It has been learned from the Antarctic isolations that symptomatic respiratory virus infections regularly appear among the station complement months after the start of the isolation.(5) It is generally believed that these viruses have been sequestered in one of the crew. This is similar to the reappearance of the chicken pox virus as herpes zoster years later. Some infectious diseases, for example acquired immune deficiency syndrome, have incubation periods over one year in length and during the incubation period diagnosis is not now possible. It is possible also that illnesses appear because the virus has been present in food since its preparation. Presterilization of the food supplies has been part of the space program since the Mercury missions. An additional factor are aerosols and dust which in a weightless environment do not settle quickly to the floor and thus continue to be rebreathed unless the air is well filtered.

The same situation exists for bacteria and bacterial diseases. Most of these will be brought with the crew in the GI, GU and respiratory tracts. Depending upon the body area and severity of pathological changes, these bacteria may become important. As an example, poor drainage of the renal pelvis, made worse by weightlessness, might allow bacteria to produce a pyelonephritis. Similarly, trichomonas residing asymptomatically in one female crew member might cause disease in a second. For most bacterial diseases there are easily administered antibiotics provided that resistant strains are not developed by transfer of plasmids.

Research must be done so that reappearance of infectious viruses from an unsuspecting crew member will not cause illness among the other crew members. Ways must be found to prevent adaptation of the immune system to an isolated environment. By the time of the Mars trips, immunization and/or effective antibiotics for virus infections should have been developed. An active reimnmunization program might be useful.

<u>THE CREW WILL NOT BE FREE OF INFECTIOUS DISEASES AND OPPORTUNISTIC INFECTIONS BUT IMMUNE SYSTEM ATROPHY MAY OCCUR</u>

Unshielded radiation exposure is of great importance to the crew members; this topic is discussed elsewhere. Certain features of this exposure are important. First we might send a crew who no longer wish to produce children and in that way avoid concerns about genetic effects from the radiation. Crew members relatively advanced in age might be used and in that way these individuals would die of other natural causes before radiation induced malignancies become significant clinically.

On the other hand, NASA data indicate there are significant decreases in the response of the crew member's lymphocytes to stimulation by foreign proteins after a space flight and there is some evidence from similar lymphocyte culture experiments on Spacelab #1 that gravity may be necessary for proper functioning of lymphocytes. If this is true, the combination of decreased lymphocyte function and increased radiation exposure would be predicted to increase the rate of clinically significant malignancies induced by the radiation exposure during the trip (6,7).

RADIATION EXPOSURE COMBINED WITH DECREASE FUNCTION OF THE SPACEFLIGHT ADAPTED IMMUNE SYSTEM MAY BE OF CRITICAL IMPORTANCE TO A MARS CREW SINCE THERE COULD BE AN ACCELERATION IN THE RATE OF FORMATION AND GROWTH OF MALIGNANCIES. NASA MUST VIGOROUSLY STUDY THIS FEATURE OF SPACE FLIGHT.

Some of the changes usually associated with aging may become more symptomatic in the spacecraft environment and the long duration of a Mars trip would allow these changes to become more symptomatic. A simple example makes the point. A crew member over 45 years of age could suddenly find that there is trouble with near vision due to symptomatic presbyopia which some crew members report is made more symptomatic in microgravity. Happily this is easily treated with glasses if these are available on board. Other aging symptoms might appear. This is particularly true of osteoarthritis of the spine which is made more symptomatic because of the changes in the spinal dynamics resulting from microgravity.

DISABILITIES OF AGING WILL HAVE TIME TO BECOME SYMPTOMATIC DURING THE TRIP. SOME OF THESE ARE MADE MORE SYMPTOMATIC BY MICROGRAVITY.

Gravity is required for physiologic processes to work efficiently. In microgravity, adaptations of physiology are required. Most of these adaptations are perceived as minor by the individual except during a short period following a sudden change from one gravitational state to the other. At the present time there is no information available which suggests that the adaptation process would be simplified or prevented by a gravity level somewhere between one G and zero G. Until this is known any attempt to design a spacecraft with partial gravity must rest on other than medical reasons.

AT THIS TIME IT IS NOT DETERMINED WHETHER PARTIAL ARTIFICIAL GRAVITY BUILT INTO THE MARS SPACECRAFT WOULD HAVE A POSITIVE EFFECT ON THE CREW'S HEALTH

SUMMARY

This has been a short review of some of the microgravity adaptation processes that Flight Medicine must take into account before NASA can certify man for a trip to Mars. Certainly none of these are show stoppers and it is certainly possible to send a crew to Mars even if none of these physiologic adaptations and pathologic changes were prevented or solved. Happily, healthy middle agéd humans tend to be relatively free of disease processes for long periods of time and the problems they do develop are usually not so severe that they can not continue to function for a period of time long enough to complete an important task.

Additional medical research is necessary to help man's adaptation to the flight environment and on return to the Earth environment after a Mars mission.

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N87-17771

HUMAN ADAPTATION AND READAPTATION FOR MARS MISSION

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ABSTRACT

Human adaptation and readaptation in space appears to involve complex physiological and psychological interactions and adjustments. There has been no comprehensive clinical characterization of the symptoms of these interactions, much less a comprehensive examination and testing of appropriate measures to counteract their near and long term adverse consequences. The variety of credible potential countermeasures is great; however, a systematic clinical research program for Shuttle and Space Station must be implemented as an early part of a Mars Mission strategy.

INTRODUCTION

The current situation we face relative to human adaptation and readaptation to various environments is more one of ignorance than of knowledge. On the one hand we know that physiological adaptive responses take place when human beings are exposed to weightlessness. All current evidence indicates these responses are reversible upon re-exposure to a gravitational field. We know that the obvious neurololgical symptoms of the first stage of that adaptation process varies in severity from individual to individual; the process seems to be complete within three to four days in all but very few cases.

We know that more subtle symptoms of full cardiovascular adaptation, and probably of most biochemical adaptations, stabilize after several weeks of exposure to weightlessness. The important known exceptions to this are mineral balance and long term vestibular response, both of which have shown continous adverse change during flights of durations approaching those necessary for missions to Mars.

On the other hand, we do not know the basic causative mechanisms of these adaptation symptoms except that vestibular agitation (head motion) aggravates the severity of symptoms in the few days of the first stage, but does not prevent initial neurological adaptation. We do not know which, if any, of the many potential countermeasures against space adaptation symptoms will work. We do not know how, or if, the <u>re</u>adaptation process can be accelerated so that crews exposed to very long periods of weightlessness can rapidly regain function in a gravitational environment.

The development of the capacity to conduct a mission to Mars must include: first, the complete clinical characterization of the physiological and psychological basis for space adaptation and readaptation; second, the clinical or flight testing of countermeasures and readaptation strategies.

Near-term Physiological Adaptation

The presently available anecdotal information on space adaptation symptoms have been summarized by Schmitt (in press) and Oman, et. al. (1984). The operational data in hand relative to these symptoms is grossly incomplete; they do suggest however, that the basic cause of the symptoms is probably a neurological conflict resulting from a wide variety of incompatible signals being received by the balance, vision and orientation processing centers of the brain and from superimposed physiological adaptive responses.

Most of the overload appears to come from visual disorientation cues combined with head motion; however, the full effects of multi-sensory conflict, autonomic dysfunction, hemodynamic alterations, and the absence of the Schumann electromagnetic resonance field have yet to be evaluated. Prolonged exposure to this overload apparently results in a loss of initiative and a general malaise (parasympathetic neural response) in some individuals. In some cases, unexpected aggravation of the overload causes the rapid onset of a single episode of unexpected vomiting which temporarily provides relief from intense symptoms. In other cases, vomiting can be prolonged and potentially detrimental to health and performance.

The known symptoms of space adaptation syndrome (SAS) resemble those of increased intracranial pressure, high altitude sickness and, possibly, other clinical problems observed on Earth aggravated by sensory conflict within the autonomic nervous system rather than symptoms associated with terrestrial motion sickness.

SAS symptoms vary in nature and intensity from person to person and from mission to mission; however, four general levels of severity can be defined as follows: (1) Fullness of the head with other associated symptoms (all crewmen); (2) Slight stomach awareness and/or slight frontal headache (about 75% of crewmen); (3) Strong stomach discomfort and/or severe headache combined with a general loss of initiative or malaise (about 40% of crewmen); (4) Intermittent, single episode vomiting (combined with level 3 above) that, temporarily at least, reduces the level of other symptoms (about 40% of crewmen); and (5) Frequent vomiting with prolonged adaptive period (about 5% of crewmen).

The process of adaptation resulting in these symptoms is, apparently, the brain learning to ignore the inputs from various sources which conflict with visual inputs. This adaptation process generally takes one to four days. The adaptation may be accelerated by pushing oneself up to detectable symptoms and then backing off from them by stopping head and body motion and strong visual orientation changes.

Significant SAS symptoms can be delayed, but probably only delayed, by highly challenging first day activities in which the crew is emotionally involved (sympathetic neural response). This does not include, however, just a full timeline that allows no time for adaptation by those crewmen who need it.

The strong effects of spatial disorientation and of head motion in inducing symptoms are clear. Methods should be explored to reduce crew visual dependency on "learned" orientations acquired during training and piloting experience in a one-gravity environment. Development of individual "egocentric" orientation references may be helpful. Considerations also should be given to using variable orientations with respect to gravity for Shuttle and Spacelab simulators and to increased visual and VFR instrument aerobatic maneuvers that give variable orientation of Earth horizon references.

There are many human physiological changes induced by a weightless environment any or all of which may play a role in inducing SAS symptoms. Among these are the following: (1) Multi-sensory conflicts, including head movements, proprioception and vision; (2) Autonomic dysfunction, including desynchronosis; (3) Hemodynamic alterations including cerebral spinal fluid shift, autonomic baroreceptor change, and hydrodynamic pressure; (4) Cardiovascular adaptation; (5) Head, neck, and spinal column position and length changes; (6) Reduction in kinesthetic sensitivity; (7) Increased cardiovascular efficiency in transport and metabolism; (8) General electrolyte, fluid, endocrine and other chemical balances; (9) Hydrogen in the water supply (intestinal gas buildup); (10) Diet and olfactory sensitivity; (11) Shift of internal organs; (12) Sleep deprivation; and (13) Absence of Schumann electromagnetic resonance field.

The bottom line is that we must understand the clinical characteristics and baseline of this early phase of the adaptation process if we are to successfully counter its long-term, more serious effects. The Space Shuttle should be more aggressively utilized to this end.

TRANSITIONAL PHYSIOLOGICAL ADAPTATION

The Skylab missions enabled us to get a general feeling for the transitional adaptation processes that take place in all individuals over a few weeks to a few months. Although it is clear that given a proper exercise regime the cardiovascular system stabilizes, it is not clear what happens to various biochemical and cellular balances. This transitional adaptation process also must be far better understood before the correct mix of countermeasures can be formulated. The early availablilty of the Space Station is crucial to such understanding.

LONG-TERM PHYSIOLOGICAL ADAPTATION

There are two known physiological adaptive responses to prolonged weightlessness that may be highly detrimental to the success of Mars surface activities. These are (1) the apparent loss of mineral mass from at least the more dense skeletal elements and from the otolith and (2) the apparent gradual decline in sensory perception related to upright activities in a subsequently imposed acceleration environment.

If not countered, either of these adaptive responses might seriously impair the efficiency, if not the feasibility, of human activity on Mars, at least for the first several weeks after arrival. On the other hand, the range of potential countermeasures is large and most can be verified during the early years of Space Station operations.

The most obvious countermeasure is to provide some form of artificial "gravitational" acceleration. However, the most commonly proposed means of doing so, namely spacecraft rotation, not only makes for very complex and costly design trade-offs, but it might create as many adaptation problems as it would solve. A more prudent approach would be to provide a means for regular exposure to appropriate levels of linear acceleration. The duration and magnitude of such acceleration could be determined by experiments at the Space Station, probably with no more than one to three months of experimentation. The resulting protocols also probably could be tailored to individual crew members after each had completed a three month tour at the Station,

The specific anti-mineral loss protocol also may be enhanced and/or simplified by diet and exercise adjustments or by use of mineral fixing drugs and electromagnetic stimulation, varieties of which are currently in clinical use here on Earth. Early tests of these approaches, looking at electrolyte and biochemical balances, could be performed during Space Shuttle flights, while testing the complete protocols would require use of the Space Station.

LONG-TERM PSYCHOLOGICAL ADAPATION

The toughest area to research and to do something about relative to long-term spaceflight is that of psychological adapation and compatibility. It is probably safe to say that history tells us that most human beings can get along in close quarters for long periods of time if they are motivated and productively active. History also tells us that there are exceptions.

The incorporation of major science, training and recreational activities into each mission to Mars should solve most potential psychological problems. Individual hide-aways, hobbies and counseling should help as well. However, a precursor visit to the Space Station by each flight crew as a unit may well be a desirable means of sorting out any individual or group problems.

CONCLUSION

Many physiological and psychological unknowns remain relative to long duration missions to Mars and to subsequent activities on the Martian surface. However, the potential of the Shuttle and Space Station for systematic clinical studies and clinical tests of countermeasures and the range of potential options are adequate to prepare for such missions. Unfortunately, NASA has not yet begun the process of developing a full clinical understanding of human adaptation to space and readaptation for Mars missions, nor has NASA begun the organized clinical testing of potential engineering and biomedical countermeasures.

N87-17772

MANNED MARS MISSION CREW FACTORS

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ABSTRACT

Crew factors include a wide range of concerns relating to the human system and its role in a Mars mission. There are two important areas which will play a large part in determining the crew for a Mars mission. The first relates to the goals and priorities determined for such a vast endeavor. The second is the design of the vehicle for the journey. The human system cannot be separated from the other systems in that vehicle. In fact it will be the human system which drives the development of many of the technical breakthroughs necessary to make a Mars mission successful. As much as possible, the engineering systems must adapt to the needs of the human system and its individual components.

INTRODUCTION

By far the most complicated and perhaps the most unpredictable system involved in a manned Mars mission will be the HUMAN system. This paper will discuss some considerations relating to that system: that is, crew selection, size, and composition and other relevant crew factors for a mission to Mars.

The human system cannot function without considering other systems that would be involved in such a complicated mission. For that reason, the makeup of the crew for the Mars journey will largely depend on a number of important factors. Perhaps the most important consideration will be: what will be the overall goals of a Mars Mission? Are we going just "to get there and back again safely"? This is obviously not a trivial goal, but it is a different matter if the goal were to be "to get there and accumulate as much data, do as many experiments, and explore as much of the Martian surface as possible or lay ground work for a future outpost, then return safely to Earth. In the former case, a smaller and less scientifically trained crew might suffice, while in the latter situation a larger more diverse crew with different scientific specialties would probably be required.

Prior to landing on the Moon, NASA was able to "practice" sending crews nearly to the Lunar surface without landing them just to test all the necessary maneuvers that were required for a Lunar landing. It was also known in advance that there would be other Apollo missions (after Apollo 11) landing on the Moon and therefore, the scientific objectives could be split up among all the missions. But what if we had only one chance--one mission to land on the Moon? While the Moon is hundreds of thousands of miles away from our Earth, compared to millions for a mission to the Martian surface, it is literally, "next door."

HABITABILITY

The importance of designing efficient man-machine systems has long been accepted since it was recognized that optimal integration increased the efficiency of the human's interaction with the machine and therefore decreased error (1-4). These issues of habitability become extremely important on longer duration space endeavors. Attention to environmental detail and real productivity may be an important factor in helping a Mars crew deal with the stress inherent in a Mars mission.

Jones [5] and Johnson [6] have discussed the major aspects of habitability which will be relevant to the design of any vehicle intended to take humans to Mars. These include: (1) environment--including atmosphere, temperature, lighting, and radiation levels; (2) architecture--how the living space is arranged; (3) mobility, restraints, and equipment handling; (4) food--i.e., preparation and storage; (5) clothing; (6) personal hygiene--body waste collection and grooming; (7) housekeeping--refuse, cleaning, and laundering; (8) communication--intravehicular only; and (9) off-duty activity provisions--exercise, Earth contact, and entertainment. Santy [7] has suggested that continued communication between crew members and their families, allocation of space for individual mementos (e.g. pictures and items with sentimental attachment), time for off-duty creative activities (such as painting, writing or gardening), and the option of being alone, (i.e. having space designed to allow individuals to be by themselves) will be extremely important for crewmembers on a long-duration space voyage.

MEDICAL FACTORS IN CREW SELECTION

Certainly all individuals selected to be crewmembers on a Mars voyage would have to be in excellent health. However, the unique aspects of a mission of such long duration have certain other implications relating to health. Since the mission will most likely be two years or longer, careful consideration will have to be given to those disease processes which a specific individual might have a higher than usual chance of developing, particularly those which might incapacitate the individual or possibly prove fatal in the space environment. On the long journey to Mars, even simple medical problems which are easily treated here on Earth, might become potentially life-threatening. An example of this would be appendicitis, in which simple surgical removal of the inflamed appendix is usually curative. But in space, when the individual is hundreds of thousands of miles away from an operating room, there is no agreed upon way at the present time that such surgery can be done in space; nor could the ill crewmember be treated palliatively and brought home. Our screening process then for the Mars trip might well include a requirement that the individual have already had his appendix removed. Other medical considerations for exclusion might include positive family histories of certain diseases (which at the moment are not disqualifying) such as myocardial infarction, alzheimers, diabetes, and certain types of cancer. At the very least, individuals with these family histories should be more carefully looked at and their own risk factors determined. It might be important also that women who are being considered should be sufficiently well-protected against the possibility of becoming pregnant, or be beyond her childbearing years.

Of course we cannot anticipate all potential problems, but there are some which we might avoid altogether if we have the proper screening.

Another area of medical considerations for a manned Mars mission are the specific psychological factors which might help an individual endure the long journey. A high degree of maturity and experience would definitely be desirable. In this context, the psychological profiles of the early space pioneers may not be the best of psychological guidelines for determining crew selection on an extended-stay mission. Those persons who need constant stimulation might not necessarily perform well when confined to a small, isolated environment for long periods of time [8-9]. On the other hand, it has been suggested that chosen individuals should have to deal with a minimum of separation stresses; i.e., that they not have minor dependent children (10). Again all of these points argue for individuals over the age of 45 as being ideal candidates for an early Mars crew.

COMPOSITION OF CREW

A lunar landing was planned around a crew of three---all of them pilots. Certainly several pilots and engineers will need to be on the Mars crew. In addition, probably a physician/life scientists will be required. Then several physical scientists--possibly a geologist, geophysicist or planetologist. Possibly a physicist, astronomer, or astrophysicist would also be good choices.

Individuals will have to have some overlap in knowledge. This will be particularly important in trans-Mars science activities and in the operational area--such as the piloting/navigation duties, since if one person became incapacitated there would be someone else to perform those assignments crucial to bringing the crew back to Earth.

Should there be "mixed" crew--that is both men and women? Having crewmembers of both sexes could potentially raise some problems on a long-duration mission, specifically regarding issues of sexuality. Of course this is a very individual and personal area, but there are no research data to guide us in decisionmaking. The experience at the U. S. Antarctic camp is also very limited since only a few women have stayed over during the long winter when the camp is completely isolated from the outside world [11]. It would be worthwhile to contact these women and their male associates and obtain their perspective on the specific stresses or problems that arose for them in that isolated environment, since they may well be relevant to mixed crews on a Mars mission.

There is no reason why a Mars mission should not have women crewmembers working side by side with their male colleagues. Although this is a sensitive area, there is time to look into it and develop some recommendations. One possible idea is that the Mars crew be made up of married couples. However, this only looks at one part of the sexuality issue [12]. McGuire [13] has suggested that on any crew for a long-duration mission there should be at least two women (or two men), and not just one since the stresses imposed (including further psychological isolation) might be very difficult to deal with. This might be a useful principle to apply in other areas rather than just for the gender issue. For example, it is possible that a Mars mission might be an international venture and a multinational crew might be considered. Psychologically speaking, it might be wise to always have at least two individuals from any country so as to decrease the problem of severe psychological isolation. On the other hand, it would not be useful or efficient to have a crew which is broken up into small cliques. Both of these factors would have to be weighed carefully.

Other considerations for crew composition will have to depend on the specific mission goals.

LEADERSHIP ISSUES

Some decision will have to be made regarding how leadership is to be structured on a Mars mission. First, who will have ultimate authority--the ground control or the individuals in space? As far as the crew structure, is democracy the best policy? Or is some kind of authoritarian/military system more efficient? There will probably be a mixture of political, military, and scientific goals for a Mars mission. Should there be separation of "military" and "scientific" personnel [8]?

If a small crew (i.e. less than five) were chosen to go to Mars, it is possible that a more "military" style of leadership might work best. On the other hand, a larger crew might require a more democratic style. Obviously, arguments can be made for exactly the reverse of the above. The Soviet experience in this regard is interesting. Several cosmonauts have commented that they got along much better interpersonally if leadership was shared [14]. This was in situations where there were only two or three cosmonauts together on the Russian Space Station. Clearly, this is an area which requires further study. It would be important to look at groups with different leadership styles to see which ones are able to perform group goals in the most efficient and harmonious manner in other isolated environments. While efficiency is very important, the ability of the crew to get along with each other will be crucial for a long journey. In the space environment, there will be no place to go to "get away from it all," and interpersonal conflict may result in behavior that threatens the entire crew. This must be prevented if at all possible.

The type of leadership style which might be best for a Mars mission will also depend on the specific mission goals for such a mission.

SIZE OF CREW

Again, the number of individuals on a Mars crew will in large part depend on the specific mission goals and on the size of the vehicle(s) that will be used. However, there are a number of points to consider in this area. Bluth [15] for example, has suggested that there should be an odd number of crewmembers for such a voyage. "Experience has shown that even numbers of people under stress tend more often to split into two equal and opposing camps, unable to reach a democratic solution to urgent mission decisions."

Too many individuals may not be efficient or economically feasible. However, too few might lead to overwork and a lack of necessary overlap in crew duties which would increase the hazards of the mission quite extensively. However many individuals are on the mission, it is essential to keep in mind the habitability factors mentioned previously. Individuals can deal with severe discomfort, crowding and lack of privacy for reasonable periods of time if their motivation remains high, but even short intervals of time in those conditions will take a great physical and psychological toll. The Mars crewmembers must be able to expect a reasonable degree of comfort and privacy during their two year voyage to Mars.

SUMMARY

Specific crew considerations for a mission to Mars will depend on the goals that are set for the mission and on the limitations of the vehicle designed for the journey. The human system and its unique problems and potentials must be integrated into the vehicle engineering systems in order to maximize the ability of the crew to carry out mission goals.

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N87-17773

MANNED MARS MISSION HEALTH MAINTENANCE FACILITY

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ABSTRACT

This paper addresses the Health Maintenance Facility (HMF) requirements which enable/enhance manned Mars missions (MMMs). It does not attempt to resolve any issues that may affect the feasibility of any given element in the HMF. The paper makes reference to the current work being conducted in the design of the Space Station HMF. HMF requirements are discussed within the context of two distinctly different scenarios: (1) HMF as part of the Mars surface infrastructure, and (2) HMF as part of the nine months translation from low Earth orbit to Mars orbit. Requirements for an HMF are provided, and a concept of an HMF is shown.

PART 1: HMR AS PART OF THE MARS SURFACE INFRASTRUCTURE

Objective

To define the requirements for a module dedicated to crew medical support and medical research on the surface of Mars. The assumption is that four (4) individuals will remain on the surface for 60 - 90 days.

Background

The establishment of a permanently manned Mars Station creates an unprecedented state of crew isolation with neither immediate nor near-term return capability to Earth. The situation created is, in some ways, similar to Antartica expeditions in which the people live in a state of near isolation for a period of nine months at the South Pole. The Mars crew, however, will be substantially more dependent upon life support systems since atmospheric oxygen and probably water are not available on Mars. The crew will also be months or years from reaching Earth; this represents a significant extension of Antartica isolation.

Thus, it is clear that a crew of four people detailed to Mars will not have access to the full spectrum of health care support and the same standards of health care available on Earth. In other words, certain risks will have to be accepted by the program and the crew. The medical screening of the crew participating in such a mission will need to be far more extensive than any such screening previously conducted, possibly to include prophylactic appendectomies and cholecystectomies.

At least one of the four crewmembers will need to be a surgically trained medical generalist. Fortunately, the Antartica experience (as well as NASA'S own experience) has demonstrated that medical problems are relatively infrequent among properly screened individuals. There seems to occur two situations, the first being one in which the medical contingency is of such benign nature (e.g. colds in a crewmember who is relatively immunosuppressed) as to present no significant health hazard, and the second being of such catastrophic dimensions (usually secondary to accidental trauma) as to result in death even if it did have Earth-bound medical support.

Since actual medical events are so infrequent, a substantial portion of the resources in the HMF and of the time spent by the physician crewmember will be directed to the long term medical monitoring of the crew and the practice of preventive medicine in the form of exercise, education and entertainment.

The presence of one-third (1/3) gravity on the surface of Mars will facilitate the use of off-the-shelf medical hardware in the HMF. It will also simplify medical procedures such as surgery which would otherwise be very difficult to perform in microgravity. The crew will also have the possibility, with the adjunct of exercise, to remain in a much more well conditioned state than if exposed to microgravity for a similar length of time. In brief, the gravity of Mars is definitely a positive feature for both design of the infrastructure HMF and for the overall health of the crew.

Requirements and Design of the HMF

The therapeutic/diagnostic modalities of the HMF must be such that the following general requirements may be satisfied: (1) The Mission Surgeon and HMF can reasonably handle most minor common non-surgical medical problems, and (2) The Mission Surgeon and HMF can reasonably handle minor surgical problems and possess limited capability to deal with major surgical events.

The preventive modalities of the HMF should satisfy the following requirements: (1) The Mission Surgeon and the HMF can obtain a predefined (as well as unscheduled) array of medical data on the crewmembers in order to follow the effects of long term exposure on the surface of Mars, and (2) The Mission Surgeon and the HMF can provide a scheduled conditioning program in order to maintain cardiovascular and musculoskeletal function at optimum levels while on the surface.

The JSC Task Force in charge of designing the Space Station HMF has subdivided its various components under prevention, diagnostic, and therapeutic classifications. The group has proceeded to identify state-of-the-art hardware which would make up the various components of the Space Station HMF. The dimensions of the Space Station (SS) HMF are estimated at 320 cubic feet in equipment and workspace (6' x 6' x 9') and 1500 pounds in weight or the approximate equivalent of four (4) single racks, Spacelab style. Figure 1 shows a schematic picture of the SS HMF valid as of May 1, 1985.

The manned Mars mission (MMM) HMF is envisioned as a larger facility in order to provide more supplies which will be needed for a much longer mission as well as increased capabilities to satisfy more extensively the requirements listed under preventive/diagnostic/therapeutic categories. The dimensions of the MMM HMF are estimated at least at 480 cubic feet in equipment and workspace (9' x 9' x 6') and 2000 pounds in weight. There are no schematics of the MMM HMF at this time.

A preliminary analysis of the various functional requirements for the HMF is shown and is identical to the currently planned Space Station HMF. It is felt that increased quantity of supplies and capabilities for more extensive procedures such as surgery will be definitely required for the MMM HMF.

PART 2: HMF AS PART OF THE TRANSLATION FROM LOW EARTH ORBIT TO MARS ORBIT

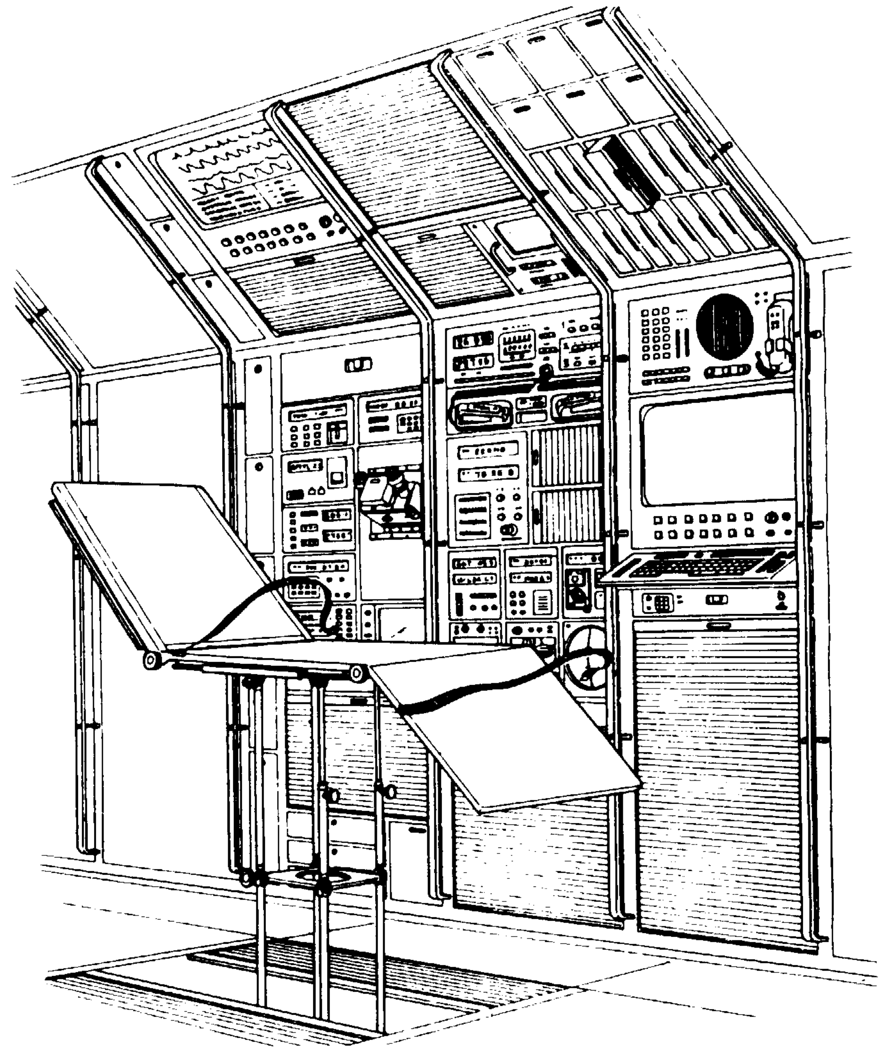
Objective

To define the requirements for a module dedicated to crew medical support and medical research during the transit from low Earth orbit to Mars orbit.

Background

The facts pertaining to a manned outpost on Mars apply equally to the crew during transit from Earth to Mars. A major environmental difference lies in the existence of a one-third gravitational field on the surface of Mars whereas none is present during translation.

Since many months exposure to microgravity is likely to result in severe deconditioning and, further, would require a prolonged adaptation



HEALTH MAINTENANCE FACILITY (HMF) FOR SPACE STATION. HMF FOR MARS MISSION WILL CONSIST OF A SIMILAR MODULAR ASSEMBLY OF VARIOUS UNITS FOR DIAGNOSIS AND TREATMENT. ADDITIONAL SPACE WILL BE REQUIRED FOR STORAGE OF SUPPLIES FOR A LONGER MISSION. ALSO, THE MARS MISSION HMF WILL REQUIRE A SUBSTANTIAL AMOUNT OF SPACE FOR EXERCISE EQUIPMENT NOT INCLUDED IN THE 3-RACK SYSTEM FOR SPACE STATION

FIGURE 1

of the crew landing on Mars, the case is made here for an artificial gravitational field of at least one-third (1/3) G during translation.

Requirements and Design of the HMF

Assuming that an artificial field of one-third (1/3) G is available during transit, all requirements and design elements proposed here remain identical to those previously described for the manned outpost on Mars.

If artificial gravity is not available, then the design elements for the HMF during transit should incorporate hardware and techniques which will function in microgravity. Such considerations are being taken for the design of the Space Station HMF.

TABLE I

FIRST LEVEL FUNCTION -

Establish Life Critical Systems

SECOND LEVEL FUNCTION -

Health Maintenance (MMM)

THIRD AND LOWER LEVEL FUNCTIONS -

COUNTERMEASURES:

1. Functional Requirements: To include aerobic/anerobic equipment, upper and lower body capability interactive with computer exercise protocols.
2. Hardware Recommendations: Treadmill, bicycle ergometer, nordic track, rowing device, hill climber.

TOXICOLOGY:

1. Functional Requirements: Capability to measure O_2, N_2, CO, CO_2 and TBD compounds.
2. Hardware Recommendations: Mass Spectrometer, gas chromatograph, laser fingerprint I.D.

HYPERBARIC:

1. Functional Requirements: Two person hyperbaric treatment facility (HTF) capable of generating 6 atmosphere.
2. Hardware Recommendations: Modify airlock.

HEMATOLOGY/IMMUNOLOGY:

1. Functional Requirements: Complete blood count with differential, hematocrit, hemoglobin, platelet count, prothrombin time, partial thromboplastin time, fibrinogen, and C reactive protein.
2. Hardware Recommendations: QBC system, Flow cytometry, Digital microscopy.

CLINICAL CHEMISTRY:

1. Functional Requirements: Sodium, potassium, chloride, bicarbonate, CO_2, urea, calcium, phosphate creatinine, Glucose, triglycerides, cholesterol, ammonia, amylase, lipase, total and direct bilirubin, alkaline phosphatatase, SGPT, SGDT, GGPT, creatine phosphokinase and isoenzymes, albumin, total protein, alanine, valine, isolevcine, phenylalanine, tyrosine, 3-methyl-histidine, and tryptophan.

2. Hardware Recommendation: Dry chemistry, ectachem, reflotron, ion sensitive electrodes, ion sensitive field effect transducer.

MICROBIOLOGY:

1. Functional Requirements: Rapid identification and AB sensitivity of Medical/environmental pathogens.
2. Hardware Recommendations: Automated microbial system.

URINALYSIS:

1. Functional Requirements: Specific gravity, Ph, quantitative protein, glucose, ketones, cell count, sodium, potassium, chloride, bicarbonate, urea, 3-methylhistidine, calcium, phosphate, myglobin, creatinine.
2. Hardware Recommendations: Stand-alone vs piggy back with other systems.

IMAGING:

1. Functional Requirements: A low radiation digital diagnostic imaging system with Earth transmission capability.
2. Hardware Recommendations: Digital radiography, miniaturized CAT, computerized ultrasound.

PHYSICIAN'S EQUIPMENT:

1. Functional Requirements: Standard physical exam equipment including stethoscope, otoscope, ophthalmoscope, visual acuity apparatus, and measurement equipment for height, weight, and blood pressure.
2. Hardware Recommendations: Physician's "Black bag".

IV-HYPERAL:

1. Functional Requirements: A rehydratable intravenous administration system utilizing stanard physiologic intravenous solutions and peripheral hyperalimentation.
2. Hardware Recommendations: Purification system administration system (portable).

CARDIOVASCULAR/LIFE SUPPORT:

1. Functional Requirements: Capability to monitor systolic and diastolic blood pressure, heart rate, electrocardiogram with digital output for arrmythmia detection, cardiac output, ejection fraction, peripheral vascular integrity, peripheral PO_2, PCO_2, PH. A cardiac defibrillator is required. Capability to measure body surface temperature, core temperature, ambient temperature, and metabolic rate.

2. <u>Hardware Recommendations</u>: Modular unit for ADV. cardiac life support and critical care capability. Modular unit incorporating capnograph, breathing gas mixture with different CO_2 concentration for obtaining mixed venous PCO_2 to determine cardiac output non-invasively.

RESPIRATORY/VENTILATOR:

1. <u>Functional Requirements</u>: Capability for measuring respiratory pressures, flows, minute and alveolar ventilations, deadspace and tidal-ventilation, respiratory quotient, O_2 consumption, CO_2 produced, pulmonary capillary blood flow, and pulmonary function tests, capability to measure respiratory volume/flow relationship. A programmable positive pressure ventilation with positive and expiratory capability, blood gas analysis including Ph, PAO_2, $PaCO_2$, $A\text{-}VO_2$ difference, right atrial pressure.
2. <u>Hardware Recommendations</u>: Small programmable positive pressure entilation with peep capability. Small blood gas analyzer incorporating a minimum of blood handling procedures.

PHARMACY/SUPPLIES:

1. <u>Functional Requirements</u>: A supply of necessary pharmaceuticals, bandages, and splints to support the designated crew size and duration. Emergency medical supplies will be stored in the safe haven to allow for a 28 day self-contained survival period.
2. <u>Hardware Recommendations</u>: Pharmaceutical and supply modules in which constituent items are organized by function. Small contingency pharmacy and supply kit for safe haven.

SURGERY/ANESTHESIA:

1. <u>Functional Requirements</u>: Capabilities for surgery, local and regional anesthesia, and dental intervention.
2. <u>Hardware Recommendations</u>: Portable surgical module incorporating surgical supplies, restraint systems, lighting, electrocautery, medical surgical suction device, and dental kit.

MANNED MARS MISSION PSYCHOLOGICAL ISSUES

N87-17774

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ABSTRACT

A manned Mars mission would undoubtedly be the most ambitious undertaking mankind has ever attempted, but it is a logical extension of NASA'S Space Shuttle and Space Station programs. Many of the technical and engineering problems inherent in such a long journey have already been, or are being solved at this time. What may be some of the more important limiting factors of such an historic mission, however, are the potential psychological and social problems which might develop on such a long-duration space journey.

Many studies done over the last twenty years in environments that have similarities to the space environment have demonstrated clearly that it is not a question of "if such problems develop" but "when". All types of groups studied (and they include submariners, Antarctic expeditioners and others) had significant decrements in performance over time as well as increased social conflict and incidence of somatic complaints, all of which indicates that such environments take their psychological toll on both individual and group functioning. What unique factors the space environment may introduce into this picture is not yet well-defined, but a manned Mars mission will certainly be an unprecedented stressful psychological milieu for the human organism.

It is reasonable to assume that those issues which have been found to adversely effect isolated groups in other extreme environments will likely be present on the voyage to Mars. With careful planning, these problems can be minimized for the Mars voyager.

INTRODUCTION

A manned mission to Mars poses some very real psychological challenges to the individuals selected to carry out the plan, as well as to their families. Those issues which have been found to adversely affect isolated groups in other extreme environments will likely be present on any manned voyage to Mars.

Oberg (1) has pointed out that "if most of the great exploratory expeditions of the past. . .had been required to meet the safety and comfort standards expected today from space expeditions, they never could have been made. Their ship losses and personnel losses were substantial, even--on tragic occasions--total.

"That could not be allowed to happen on the first expeditions to Mars, whatever the cost. . .the man-to-Mars effort must guarantee that the crew will stay alive the whole trip, without untoward medical or psychological complications, and perform their duties as well as the hardware allows them to". The purpose of this paper is to suggest that with careful planning, many of the potential psychological complications can be minimized or at least more effectively dealt with by the Mars mission crew members.

ISOLATION/CONFINEMENT

The problem of isolation and sensory deprivation was felt to be one of the more serious psychological issues early in the history of the Space Program. This problem has been shown to have serious effects on isolated individuals and groups in Earth-bound environments. Numerous studies done on groups such as submariners (2-8), Antarctic journeyers (3,4,8) and volunteers in simulated environments (9-16) reported remarkably consistent findings regarding man's response to isolation. Reported symptoms varied only slightly from study to study, and all studies recorded the following symptoms: boredom, restlessness, anxiety, sleep disturbances, somatic complaints, temporal and spatial disorientation, anger and (most important), deficits in task performance over time. Furthermore, in studies of isolated groups of men (4,5,7), researchers had to take into account such factors as group and social influences, individual and leader roles, and individuals' personality coping mechanisms. For example, Gunderson (4) and Gunderson and Nelson (5) studied groups in isolated and remote Antarctic stations over several months and followed subjective evaluation by the men of their symptoms. By far the most common complaint was sleep disturbance (reported by 72% of individuals), followed by depression, headache, irritability, and other somatic complaints. It was noted that all the symptoms increased over time. Even when such symptoms (especially depression and irritability) occur among only a few members of a small isolated group, they could pose serious survival problems for the rest of the group under certain circumstances. Consistant in all these group studies was evidence that group compatibility and performance typically declined over prolonged isolation. On the other hand, Earls (6), who studies submariner groups, was struck by "the relative absence of overtly expressed hostility" on the part of the submariners. This conflict--between the individual's unwillingness to alienate the group (upon which one's individual survival depends) and the increasing and inevitable normal kinds of tension that must remain undischarged (no overt agression)--can arise in any group situation. Often the unresolved conflict can lead to depression and somatic complaints.

Berry (17) pointed out that sensory deprivation/isolation issues have (at least for the American Space Program) not been a significant issue. However, for a Mars mission, manned by individuals with heterogeneous background, personality styles, and scientific objectives etc., isolation issues will be extremely relevant and will require further study(18).

GROUP/SOCIAL INTERACTION

Since the beginnings of manned flight, the trend has been to increase the number of persons on each mission. The Space Station is expected to have up to 10-12 or more persons living and working together. A mission to Mars will of necessity have a diverse selection of individuals with different scientific and or mission objectives. It will require pilot astronauts who will need to fly the vehicle to and from the Red Planet; it will require scientists with different specialties (such as geology, physics, medicine). It is not known at this time the number of individuals which would make up the crew of the Mars mission, but any interplanetary mission will require numerous individuals working in a close knit, efficient team manner for a long period of time (possibly up to two years, or more). Obviously, psychological compatibility and the methods used to determine the selection of crew members will play a role in the ability of such crews to perform efficiently in the isolated environment of space. The Soviet space program has long recognized this as a potential problem--especially since they have kept several individuals in space for up to 240 days. But even the Soviets have little experience with crews of more than two or three individuals. However, from the beginning of their training, Soviet cosmonaut candidates are subjected to the most grueling psychological tests and the candidates are grouped according to compatibility (19).

Almost all of the group studies done to date have been done on all-male crews, and thus far, the potential consequences of adding women to long-duration missions have not been dealt with. The issue of sexuality--especially on long-duration missions such as that to Mars--is not a trivial one. No matter how well trained the crew is, there is reason to expect that such issues as sexual arousal, tension, and competition are just as likely to occur in space as they are in any Earth-bound endeavor, and possible solutions may be to balance the crew with individuals of both sexes; or perhaps to have married couples. This is another area which may require careful thinking in advance.

One method of approaching the complexity of handling group problems on the way to Mars would be to train the crewmembers in simple group dynamics. In this way, an ongoing group process would help identify and resolve potential trouble areas and help crews develop problem-solving techniques before they arise in the more dangerous space environment. As a practical method, this would also give the crew a model for resolving in-flight hostilities and tensions that can lead to group-threatening behaviors and decrease crew performance (21,22). After all, on the journey to Mars, there will be nowhere to go to get away from it all. Such a method would offer useful ways of discharging tension and anger, and possibly help alleviate the symptoms of depression and somatic complaints reported in other isolated groups.

One final issue should be mentioned in this section and that is the question of how leadership should be structured on a Mars mission. Leadership style can have a significant impact on group morale and performance. There has been considerable research on different kinds of leadership on group function--obviously, some types work better than others. What will be the best type (or types?) of leadership for a Mars mission? (23)

HUMAN PERFORMANCE IN STRESSFUL ENVIRONMENTS

Research in human performance and productivity has grown to very large proportions in industry and the military. Primarily this has come about because of the desire to increase human productivity in these areas, each of which has its own peculiar environment which often impedes attainment of maximum efficiency by its workers. The relevance of this type of research for an extended mission to Mars cannot be over emphasized. Such factors as the habitability of the space ship and other man/machine interfaces must be carefully planned in advance taking into account the specific complexities of a mission to Mars.

For example, what will the scientists on the mission do during the time it takes to get to the planet? How will we prevent boredom and restlessness from occurring--or at least minimize them? A mission to Mars will provide many environmental challenges to crew performance, efficiency, and productivity. Such factors as temperature, isolation, work/rest cycles, exposure to unfamiliar and possibly dangerous contingencies (which no amount of training beforehand can possibly cover completely), exposure to various physical/physiological alterations which may alter the body's ability to cope with other types of stress--all of these factors will have their effect on individual and crew performance capabilities. Can we devise a simple way for individuals to keep track of their own performance status, and thus give them some kind of feedback which they can use to enhance their efficiency? This is done in other environments where it is much less dangerous to fall below a certain level of functioning. Why not on a trip to Mars? This would enable an individual (or the entire crew) to put into effect pre-planned strategies to increase their own effectiveness. (24)

In paying attention to these factors, we can maximize crew performance for the entire mission and decrease the incidence of any psychological sequelae.

PSYCHOPHYSIOLOGIC RESPONSE TO STRESS

Coping responses, or how individuals avoid being stressed when exposed to threatening environmental situations, will be of particular interest on a long-term mission to Mars or other planets. Researchers have found that individual psychological defenses, such as isolation and denial, resulted in subjects' low cortisol secretion before a stressful event, compared with subjects who were overtly distressed before the event (25). Thus, a defense may be effective physiologically, but maladaptive psychologically, and it becomes an even greater threat to the organism. For example, women who had breast tumor biopsies and who denied the situation had low cortisol levels; however, because these women had waited much longer before coming to medical attention than women who feared the situation and immediately sought help, they put their lives at greater risk (26). In the same way, defensive structures in Mars voyagers may be vitally important; they may be even more important than the voyagers' actual physiologic responses to stress.

PSYCHOSOCIAL SUPPORT FOR INDIVIDUALS ON A MARS MISSION

NASA has long recognized the importance of habitability in determining astronaut morale. Food items, for example, are selected to meet the tastes of individual astronauts, and favorite music has been permitted on longer missions such as Skylab. The Soviets make use of their Group for Psychological Support to help their cosmonauts get through record-breaking stays on the Salyut Space Station.

The literature in this area has many suggestions as to what factors might be particularly useful to focus on for the purpose of maintaining mental health or improving the quality of life on long-term space missions. The suggestions come from studies of other isolated groups such as those on submarines or in the Antarctic, and have been extrapolated to the space environment. It remains to be seen, however, if these "psychological support" measures are really going to be supportive or not in the rather extreme isolation of the voyagers to Mars. For example, how does one deal with the death of a loved one on Earth when you are millions of miles away--or handle a family crisis? Should Mars astronauts be told about such things when they may not be able to return to Earth for months or years? If they are told, then what can be done to help them deal with their feelings (especially the feeling of helplessness)? Will constant communication with families be supportive psychologically--or might it also be disruptive for the individual physically and mentally coping with the stresses of space? It seems likely that some important family problems or situations will develop over the two years of a Mars mission. How will we help astronauts and their families cope with this prolonged separation? Perhaps part of the solution would be to select individuals who are not married, or have no children, or have children who are grown. However, no astronaut that might be selected could possibly come from a complete social vacuum, and the problem of psychosocial support still remains.

There is probably no one simple answer for all individuals, but this area needs to be carefully thought about prior to a mission to Mars.

SUMMARY

The research on isolated environments over the last thirty years suggests that psychological factors associated with such environments will lead to negative changes in individual and group performance. A mission to Mars will be the greatest undertaking ever devised by the human species. The members of such a mission will be in an environment whose potential dangers are not even completely known at this time. The psychological factors generated by such an environment, and which might adversely affect accomplishment of mission goals, can be minimized or planned for in advance. This paper was written in the hope that these issues will not be ignored in planning for this great adventure.

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N87-17775

NEED FOR ARTIFICIAL GRAVITY ON A MANNED MARS MISSION ?

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ABSTRACT

Drawing upon the extensive Soviet and our own Skylab medical observations, the need for artificial gravity (g) on a manned Mars mission is discussed. Little hard data derived from well done experiments exist. This dearth of information is primarily due to two factors. We cannot collect tissues from astronauts for ethical or operational reasons. Second, there has not been opportunities to fly animals in space to systematically evaluate the extent of the problem, and to develop and then to prove the effectiveness of countermeasures. The Skylab and Space Station will provide the opportunity to study these questions and validate suggested solutions.

The need for some form of "artificial gravity" aboard a spacecraft may be necessary during Earth-Mars-Earth transit. The most conservative approach would be to artificially provide one g during a round trip to Mars. Economic and engineering prudence will demand a validation of the assertion one g need be provided. Fortunately, the need can be determined onboard the Space Station, given proper and early study of people and mammals. If those studies should prove the need for one g and the consequent extensive engineering measures, the determination of "how much" or "what kind" of accelerations would provide the necessary minima will also require rather extensive testing. By its very nature, this testing will require years to conduct. One important question that would then follow is if 0.17, 0.38, or near 1.0 g is sufficient? There are no data to guide us (although it is presumed continuous 1.0 g would be adequate).

Observation of people who have spent extended periods of time in free fall suggest at least two, and possibly three, reasons to suspect that some form of artificial gravity might be needed on very long duration missions; especially on missions requiring on-surface EVA. These observations are: (1) Rather extensive orthostatic hypotension following long exposure to free fall, in spite of considerable hours of exercise designed to counter this cardiovascular sequella; (2) There is consistent, measureable, and progressive skeletal and muscle atrophy of the anti-gravity bones and muscles of the body; and finally (3) there seems to be a continuous loss of calcium from the skeleton, and possibility, from other calcium deposits within the body.

On the Soviet four to seven month missions, Cosmonauts have needed considerable assistance to egress from their spacecraft. Their ability to do on-surface extensive EVA is not known, but is suspected to be minimal. Furthermore, their return to full pre-flight cardiovascular competence has reportedly taken weeks to months. This, even with special on-orbit exercises and devices designed to tax their hearts and their large muscle groups. In spite of various exercise regimes, there have been measurable muscle mass losses in both cosmonauts and astronauts. Skylab input/output studies actually measured a negative nitrogen balance (indicative of muscle loss), in spite of exercises designed to prevent such losses. It should be remembered that once muscle fibers are lost they are never replaced. The residual fibers, can through exercise, increase their bulk, and thereby their strength, but if too many fibers are lost that muscle's function can never be recovered.

Soviet and U.S. studies of rat muscle tissue suggest a rapid and massive change from "slow twitch" to "fast twitch" fibers in muscles that normally have both types of fibers. In the predominantly "slow twitch" or anti-gravity muscle groups, the changes are even more remarkable following only seven days of weightlessness. These morphological changes and other histochemical changes observed in the animal studies, and on relatively short duration missions, give physiologists considerable cause for concern for the integrity of muscles exposed to weightlessness for years. The longest a rat has flown in space is 21 days, so long term effects are not known in even this simple mammalian species. No direct data exist on the extent of comparable changes in humans. To make obtaining valid answers even more difficult, astronauts and cosmonauts rightfully resist muscle (or any tissue) biopsies.

It is not known if some partial g loading, or if aperiodic g loading, would prevent these "normal" adaptations to the microgravity environment. There may be exercise devices that could attenuate or prevent these changes. The Soviets have developed special flight garments designed to exercise the heart by putting the lower half of the cosmonaut's body at a negative pressure. This to expand the volume of the legs and lower abdomen into which the body fluids shift, causing the heart to work harder. They have required their cosmonauts to wear other devices designed to exercise the large anti-gravity muscles. This suit consisted of strong elastic bands attached in such a way as to force these various muscles to work against the elastic bands during flight. The Soviets have reported no systematic data on the effectiveness of these devices. It is known that some of the cosmonauts do not like wearing them. There may even be dietary means to help alleviate the muscle wastage seen on long duration missions....but again, no data exist.

It should be emphasized that the changes here described are considered to be the normal adaptations of the body to a new environment, weightlessness. These adaptations seem to be well suited to the micro-gravity environment and present complications only when the body is then placed into an environment requiring adaptation to increased g loadings, or when asked to "work" in physically demanding situations. Since Mars' gravity is approximately 0.40 that of Earth's, there would be significantly less stress placed upon the body than when returning to Earth. However, EVA and a desire to accomplish as much as possible while on the surface of Mars, will be physically demanding. The procedures to best assist astronauts to re-adapt to the rigors of 1.0 g here on Earth, after 2 or 3 years of weightlessness are but dimly understood.

From the physiological point of view there are, then, several necessary sensitivity analyses that should be undertaken to determine the relative importance of each element of a research program. For example, it is not known if the loss of calcium in weightlessness will stabilize at some acceptable high level...or, if some partial level of g is necessary to prevent excessive losses of this important mineral from the body. If any form of acceleration is found to be necessary, then a similar analysis must be done to determine the "best" means of providing it, e.g., if the habitat modules must be designed to rotate, the maximal acceptable angular velocity will need to be determined (relative to the radius of the habitat and the position of the body within it).

Similar analysis must be done to determine the priority of research programs in preventing skeletal muscle atrophy or cardiovascular deconditioning, and developing proper exercise physiology methods (germane to EVA on the surface, if such be the case), etc.

Clearly the Spacelab, followed by the Space Station, will provide the United States for the first time the means to objectively assess these problems (rather than via experts' opinions and pronouncements). Most importantly, these machines will provide the laboratories where the best answers to these vexing questions can be obtained. It must be kept in mind that these problems are complex in-and-of themselves. They are even more complex when the physiological requirements interact importantly with engineering design considerations. The opportunity to start the processes of identifying the solutions must be seized now, and the search for answers begun. Since the fundamental issues deal with long duration exposures to zero gravity and with biology, there is no way to "speed up" some of the experiments! For example, if there is a question of how long it takes the loss of calcium to reach some asymptotic level during weightlessness, (and we strongly suspect it's longer than 238 days based on Soviet experience) then it behooves us to start the experiments early. This is especially true if we then must devise and test inexpensive countermeasures rather than artificially providing one g. The time it will take to do these experiments is further compounded if one is either curious about or ethically compelled to understand how (or if) the bones re-adapt once the crew returns to a one g environment.

It is reasonably well understood that animal models will have to be used to conduct many of these experiments. At some point however, it will be programatically prudent, and possibly ethically acceptable to validate conclusions from the animal studies by experiments upon humans. These experiments would include biopsies from people exposed to weightlessness for long periods, biopsies from others who have been "protected" by proposed countermeasures, and then later more tissue biopsies after varying periods of recovery. The alternative is to either take the risk of using unproven techniques and countermeasures or to provide a suitable one g environment for most of the round trip voyage to Mars.

MANNED MARS MISSION RADIATION ENVIRONMENT AND RADIOBIOLOGY

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ABSTRACT

This paper discusses potential radiation hazards to crew members on manned Mars missions. It deals briefly with radiation sources and environments likely to be encountered during various phases of such missions, providing quantitative estimates of these environments.

This paper also provides quantitative data and discussion on the implications of such radiation on the human body. Various sorts of protective measures are suggested. Recent re-evaluation of allowable dose limits by the National Council of Radiation Protection is discussed, and potential implications from such activity are assessed.

DISCUSSION

The crewmembers of a manned mission to Mars (MMM) will be unavoidably exposed to ionizing radiation as they pass through the inner trapped proton belt, the outer trapped electron belt, and through the galactic cosmic ray (GCR) flux of interplanetary space. Moreover, outside of the Earth's magnetosphere, there is the possibility for exposure to proton radiation from solar particle events (SPE). On the surface of Mars, the GCR and SPE fluxes will be less than half that of free space because of the 2-pi shielding by the planetary mass and the shielding provided by the thin Martian atmosphere. Some representative dose equivalents in these regions are shown in Table 1.

It should be emphasized that the listed dose equivalents are approximate. In the future, as planning for MMMs matures, the depth-dose-equivalent projections must be refined. These dose projections are complex functions of the particle fluence, the charge and energy (velocity) of the particles, the interaction of the primary particulate radiation with the spacecraft material, the production of secondary particles, body self-shielding, the ionization density or linear energy transfer (LET) of the particle in tissue, relative biological effectiveness (RBE) of different particles, and other factors. For many of these factors, the uncertainties are large. The factor which is, perhaps, the most uncertain is the RBE upon which is based the quality factor (Q) to be applied for radiological health risk assessment. Recent experimental data indicate that high LET radiation such as in GCR may be 50 or so times as effective as low LET radiation such as the gamma and X-rays to which the Japanese A-bomb survivors were exposed. Moreover, the application of conventional radiological health practices to GCR is likely not warranted. Before a Manned Mars Mission is attempted, the radiological health risks must be refined and uncertainties reduced.

The implications of the approximate dose equivalents listed in Table 1 can still be considered in relationship to general radiological health impacts. In Table 2, note that the doses to achieve a certain biological end point must be given in a short time (hours) to be effective in eliciting the response. If the dose is protracted over several days, 2.5 times the dose is required to elicit the response. If the exposure is protracted over a very long time, the dose-response relationships shown in the table are replaced by entirely different types of dose responses resulting from hematological depression. With this in mind, a comparison of the doses in Table 2 with those in Table 1 indicates that only in the case of an anomalously large SPE (ALSPE) need we be concerned with the potential for an immediate mission impact. Although such ALSPE are rare events, having occurred only once or twice per 11-year solar cycle during the past 3 solar cycles for which measurements are available, their potentially serious effects dictate that they be protected against. Moreover, it has been estimated that the dose rate for the August of '72 event could have been 10 times higher if it had occurred 4 days later when the Sun's rotation would have placed the flare zone in a more damaging location relative to the near-Earth vicinity.

Various possible means for the management of ALSPE risks during travel in free space are as follows: (1) Schedule mission for period around solar minimum--there is about a 6-year period during which SPE's are not expected to occur; (2) Shield spacecraft with nonfunctional mass against the known worst-case event (August 1972) times a safety factor to reflect the facts that (a) the August 1972 event would have been worse if it had originated in the optimum region of the Sun, and (b) it is not known how large an ALSPE can be; (3) Arrange stowage, water tanks, and waste tanks to provide shielding as above using parasitic shield mass only to fill the gaps; (4) Provide a storm cellar--a

TABLE 1

APPROXIMATE BLOOD FORMING ORGAN DOSE EQUIVALENTS AND
RADIATION HEALTH RISK FOR A MANNED MARS MISSION DURING SOLAR MINIMUM

Radiation Source	Skin Dose Eq.	Deep Organ (5 cm) Dose Eq.	Excess Lifetime Cancer Incidence in a 30-Year Old Male*
CHRONIC EXPOSURE TO GCR			
FREE SPACE			
behind 4 g/cm^2 Al	36 rem/yr	27 rem/yr	1~%/yr of exposure
ON MARS			
behind $4/cm^2$ habitat shielding and 10 g/cm^2 CO_2 atmosphere	12 rem/yr	10 rem/yr	<0.5%/yr of exposure
ACUTE EXPOSURE TO ALSPE (a' la August '72)			
FREE SPACE			
behind 2 g/cm^2 Al	1106 rem	105 rem	~6%
behind 15 g/cm^2 Al	27 rem	7 rem	<0.5%
ON MARS			
behind 10 g/cm^2 CO_2	83 rem	17 rem	<1%
behind 10 g/cm^2 CO_2 +	~0 rem	~0 rem	Negligible

*The % increased cancer incidence for a 30-year old female is roughly twice that for a 30-year old male.

Note: Low Earth Orbit Phase and Van Allen Belt Passage Phase together contribute <4 rem. The % increase in cancer for a typical Mars Mission is obtained by multiplying the yearly % by the number of years of exposure.

TABLE 2

EARLY EFFECTS OF ACUTE (LESS THAN 1 DAY) RADIATION (IN RAD AT >5 CM)

	1ST DAY				20-60 DAYS
	ANOREXIA	NAUSEA	VOMITING	DIARRHEA	LETHALITY
ED_{10}*	40	50	60	90	220
ED_{50}	100	170	215	240	285
ED_{90}	240	320	380	390	350

EARLY EFFECTS OF RADIATION GIVEN AT LOW RATE (4-6 DAYS)

AS ABOVE X 2.5	ABOVE X2

*EFFECTIVE DOSE FOR 10, 50, OR 90 % OF A POPULATION OF NORMAL PEOPLE.

TABLE 3

RADIATION EXPOSURE LIMITS

CONSTRAINTS IN REM	SKIN (0.1 MM)		EYE (3 MM)		BONE (5 CM)	
	NAS[A]	NCRP[B]	NAS	NCRP	NAS	NCRP
1 YR AVERAGE DAILY RATE	0.5	-	0.3	-	0.2	-
30-DAY MAXIMUM	75	150	37	100	25	25
QUARTERLY MAXIMUM	105	-	52	-	35	-
YEARLY MAXIMUM	225	300	112	200	75	50
CAREER LIMIT	1200	600	600	400	400	100-400[C]

[A]NAS = NATIONAL ACADEMY OF SCIENCES, 1970, CURRENT OFFICIAL LIMITS.

[B]NCRP = NATIONAL COUNCIL ON RADIATION PROTECTION AND MEASUREMENTS, 1986, RECOMMENDED BY SCIENTIFIC COMMITTEE 75. NOT YET OFFICIAL.

[C]VARIABLE DEPENDING ON AGE AT START OF EXPOSURE AND ON SEX. THE CAREER LIMITS CAN BE APPROXIMATED BY 200 + 7.5 (AGE-30) FOR MALES AND 200 + 7.5 (AGE-38) FOR FEMALES.

smaller region of the spacecraft which utilizes shielding from stowage, tankage, and parasitic mass; and (5) Provide a group partial body shield consisting of a cylinder inflatable up to a wall thickness of about 20 cm with stored water. The cylinder in operation would surround the torsos of the crewmen huddled back-to-back to improve shielding of the blood forming organs (BFO) in the spine. [During the August 1972 event, most of the dose (60%) was received in a 6-hour period. Conceivably a 12-hour stay in the "water bed" shield would be tolerable.] This crew shield concept could take different forms with a variety of tradeoffs.

On the surface of Mars, one could shield against an ALSPE by using only 10 cm (4 inches) of Martian soil, which, with a density of 3.5 g/cm^3, would provide excellent shielding and reduce the skin dose from an August 1972 event to below 1 rad. Conceivably an astronaut could cover himself with soil as one does with sand at the beach or an astronaut could insert an inflatable storm cellar into a crater on Mars and cover it with soil by means of explosive charges.

In the case of an ALSPE occuring either in flight or on the Martian surface, adequate warning will be required. The Earth-based optical network currently used to warn STS astronauts of potential SPE will not be able to view the region of the Sun which poses the greatest threat to a Mars-bound spacecraft. A system comparable to NOAA's proposed Solar X-ray Imager (SXI) will be required. Also, active, alarmed dosimeters will be required to alert the crew of the arrival of the first particles.

Adequate protection against ALSPE must be provided to preclude exceeding the official space radiation exposure limits: currently 25 rem to the blood forming organs, 37 rem to the lens of the eye, and 75 rem to the skin (Table 3). The 30-day limits are set to avoid immediate radiological health impacts on a mission involving nausea, vomiting, etc. After protection against immediate impacts, the remaining radiological health issue concerns radiogenic stochastic effects, primarily cancer induction.

Radiocarcinogenesis results from a combination of physical, chemical, and biological events occurring over the years and with low probability. The severity of cancer is independent of the dose received, but the probability that cancer will occur increases with dose.

Moreover, any radiation dose increases the risk. Therefore, limits are set based on an acceptable level of risk, not precluding any risk.

The current astronaut career radiation limits, which were published in 1970, were based primarily upon radio-epidemiological data from Hiroshima-Nagasaki A-bomb survivors. These data indicated that 400 rem doubled the natural cancer risk for males between 35 and 55, a group comparable to astronauts. The risk was deemed acceptable considering the other risks of space flight.

These limits are currently being reevaluated by Scientific Committee 75 of the National Council on Radiation Protection (NCARP) and measurements in light of the following considerations: (1) The appreciation of radiation-induced cancer risks has changed markedly since the earlier guidelines were developed prior to 1970; (2) HZE particle effects were not well known at that time and while they were deemed, in the early 1970's, to be unlikely to be limiting, the question needed reexamination as soon as real experimental evidence became available; (3) Philosophies relating to occupational risks, for example, comparisons with relative risks in chemical industries and with risks of fatal accidents in"safe" and "less than safe" industries; and (4) The numbers and the nature of the people, including sex, and the roles they are to perform and the time they are to spend in space have also appreciably changed.

Sinclair (1984), President of NCRP, has discussed these points in some detail. The basic thrust of the reevaluation is embodied in the following extended quote:

> "Among the considerations which the committee will no doubt discuss are the following. On Earth, we tend to compare the risks from occupational exposures of radiation workers to the accidental fatality rates of "safe" industries, which we consider to be 10^{-4}/year or less. . . Fatality rates for travel to and from work are in the same range. . . However, many industries described as 'less safe', but quite normal industries, are in the range up to 10^{-3}/year. . . and it may be justified to compare with them. Thus, it may be appropriate to consider a lifetime risk of say 50 years x 10^{-3} or 5%. This could be a limit which can be

received in a space worker's lifetime, or after a defined number of missions, if the dose or risk permission is known. At low doses, which applied to most space circumstance, 2×10^{-4}/rad might be used as the risk."

Sinclair's considerations imply a career dose limit of 205 rem to the organs susceptible to radiocarcinogenesis, which are essentially encompassed within the blood forming organs or 5 cm dose. Sinclair's risk factor of 2×10^{-4} cancer death/rad is admittedly rough. Susceptibility varies with age at time of irradiation and sex.

Since Sinclair's statement, NCRP Scientific Committee 75 has refined its' risk assessments and philosophy and is recommending to the Council as a whole the limits shown in Table 3. The tentative career limits for the deep organs are predicated on a 3% lifetime risk of cancer mortality. Because the risk per rem depends upon age at exposure and on sex, these factors are considered.

The 3% lifetime mortality is comparable to the accidental death risk incurred in careers in quite normal industries such as mining, transportation, and agriculture, and is therefore deemed an acceptable risk.

However, cancer incidence, in contrast to mortality, may be a more important endpoint in that quality of life is impacted by contracting cancer, even if cured. In short, risk factor estimates and considerations of acceptable risk can be refined further. However, if we accept a career dose of 200 rem, then the total estimated dose from Table 1 for a reasonable 3-year MMM scenario does not exceed the career limit for a 35-year old even with allowance for a number of previous low Earth orbit missions in, for example, Space Station, where up to about 10 rem/90-day tour could be accumulated.

In conclusion, radiation concerns will not prohibit MMMs but must be considered in the operation and the design of the spacecraft and the Mars base. Moreover, NASA is committed to the radiation protection principle of ALARA, that is, keeping doses As Low As Reasonably Achievable; therefore, every reasonable effort should be made to reduce the total dose-equivalent the crew will receive. Substantial effort will be required to reduce the dose uncertaintites and thus reduce unnecessary shielding mass to achieve optimum radiological health protection consistent with MMM goals.

N87-17777

NATURAL RADIATION HAZARDS ON THE MANNED MARS MISSION

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ABSTRACT

We consider the hazards of the natural radiation environment--cosmic rays and solar energetic particles--on a manned mission to Mars. These hazards are addressed in three different settings: (1) the flight to Mars where astronauts are shielded only by the spacecraft, (2) on the surface of Mars under an atmosphere of about 10 g/cm^2 carbon dioxide, and (3) under the surface of Mars where additional shielding would result.

INTRODUCTION

The manned mission to Mars is confronted with a high energy nuclear radiation exposure two orders of magnitude greater than that encountered on previous space missions. The dose rate is comparable to what Apollo astronauts received on Moon missions; however, the flight duration is expected to be about 3 years, or 100 times longer than the average 10 day Moon mission. Longer space flights, such as Skylab, are not comparable to the Mars mission because they were not exposed to the full force of the radiation environment.

A baseline dose equivalent rate for the Mars spaceflight is 43 rem/year. This is based on a computation (Silberberg et al., 1984) of the free space cosmic ray flux just under the surface (0.1 cm) of a 30 cm diameter sphere of water. The natural radiation environment of Adams et al. (1981) was used as a model of the cosmic ray flux ($Z < 29$) at solar minimum. The model does not accurately predict free space cosmic ray fluxes at energies < 10 MeV/nucleon, but these particles are removed by very thin shielding. Particles surviving 0.1 cm of water originate at energies above this limit.

The baseline dose as decribed here maintains a fairly continuous intensity. The solar cycle introduces downside variations of about a factor of 2 in integral fluxes above 150 MeV/nucleon, and up to a factor of 10 in low energy fluxes. Aluminum shielding (4g/cm^2) reduces the dose to about 36 rem/year. Self-shielding of the spherical phantom reduces the dose to about 24 rem/year at its center. The baseline dose is essentially inevitable. Energetic particles associated with solar flares are the primary risk of higher dose rates. This risk is not presently quantified and is strongly dependent on shielding.

The expected dose equivalent rate on the surface of Mars is reduced from the baseline a factor of 2 by shielding with the planet's mass. Further attenuation results from atmospheric shielding. For an assumed vertical atmospheric depth of 10 g/cm^2 the dose equivalent rate due to cosmic ray primaries is estimated to be 10 rem/year. Neutrons should not be an appreciable fraction of the dose at this depth - we guess neutrons would increase the surface dose by no more than 25%. We suggest the surface dose equivalent is 12 rem/year.

Under Martian soil, the dose continues to fall, perhaps by a factor of 2 from the surface to 20 g/cm^2 (~ 10 cm) below the surface. Another reduction by a factor of 2 can be expected down to 60 g/cm^2 (~ 30 cm) below the surface. At this depth, neutrons dominate the dose equivalent and further reductions are not so rapid. We have not estimated the neutron dose under the surface.

CONCEPTS: THE NATURAL RADIATION ENVIRONMENT

The natural radiation environment encountered on a mission to Mars consists primarily of galactic cosmic rays and solar energetic particles.

Galactic Cosmic Rays: (a) Mostly protons, 10% He, 1% heavier ions; (b) Hard spectrum (E-2.2 for protons); (c) Relatively constant intensity (factor of 2-3 variation with solar cycle); and (d) High energy (mean about 2 GeV/nucleon).

Solar Energetic Particles: (a) Mostly protons, variable heavy ion composition usually not as rich as cosmic rays; (b) Soft spectrum (E-5 or so for protons); (c) Widely varying intensity (many orders of magnitude); (d) Low energy (mean < 100 MeV/nucleon); and (e) Unpredictable.

Figure 1 shows the differential proton energy spectrum for cosmic rays at solar minimum and solar maximum, and for a large solar event (4-7 Aug 1972). The cosmic ray spectra are integrated over a week, while the solar protons are integrated over the flare duration. Above a few GeV/nucleon there is no solar cycle variation. Low energy fluxes vary by up to a factor of 10. Integral fluxes above 100 MeV/ nucleon vary by factors of 2 or 3.

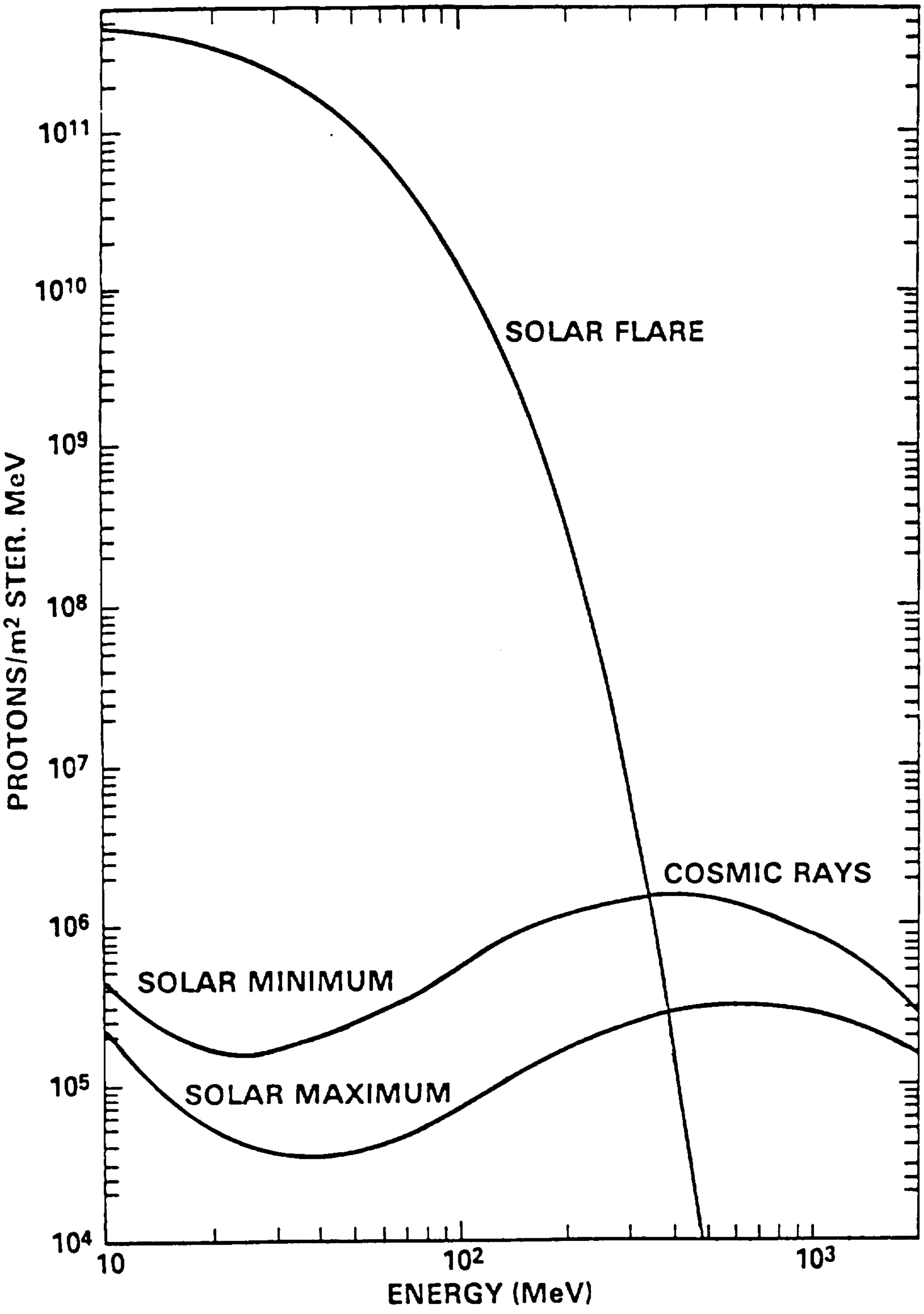

PROTONS/m² STER. MeV
10¹¹
10¹⁰
10⁹
10⁸
10⁷
10⁶
10⁵
10⁴
SOLAR FLARE
COSMIC RAYS
SOLAR MINIMUM
SOLAR MAXIMUM
10
10²
10³
ENERGY (MeV)

FIGURE 1

How well do we understand the environment? We can predict the galactic cosmic ray fluxes to within a factor of 2, well ahead of time. After the fact, much better estimates of the accumulated dose should be possible by examination of data from satellite-borne particle monitors. There is no complete engineering model of the risks associated with solar energetic particles. Important factors in such a model would be peak intensity, duration, energy spectrum, heavy ion enrichment, and time-intensity profile. All of these factors are critical for estimates of the biological risks of solar energetic particles.

It is worth noting that we are interested in the natural particle environment in the vicinity of Mars. This differs in several ways from the environment around Earth. The cosmic ray flux at 1.5 AU is somewhat greater than at Earth. Measurements from Pioneer 10 and 11 (McKibben et al. 1983) show radial gradients of 3-4%/AU at solar minimum at energies > 67 MeV. Below this energy, variations of up to 15%/AU have been observed. Mars also has a negligible magnetic field. The associated magnetic rigidity cutoff, which protects astronauts in low inclination orbits around Earth from most cosmic rays, is missing. In addition, there is no trapped radiation presenting a risk of high dose in Mars orbit.

CONCEPTS: PARTICLE TRANSPORT AND SHIELDING

Astronauts are never exposed to free space radiation intensities. In addition to the shielding provided by space vehicles and suits, self-shielding provides some protection. An example of self-shielding is shown in Figure 2. This is the pathlength distribution 0.1 cm below the surface of a 30 cm spherical phantom as used in computing the baseline dose in free space. Figure 2 shows an exposure of 3.16 steradians through less than 0.2 cm shielding. On the other hand, cosmic rays are shielded by between 6.0 and 30 cm of water (uniformly distributed) over 40% of the solid angle.

We would like to understand the properties of shielding to guide us in defining structures and procedures for protecting astronauts from space radiation. To understand the effects of shielding, we must understand the transport of high energy nuclei in materials. Much work has been done in this field (see, for example, Letaw et al. 1984, Letaw et al. 1983, Silberberg and Tsao 1973). We briefly explore several concepts below.

3631–86

SPHERICAL PHANTOM PATHLENGTH DISTRIBUTION

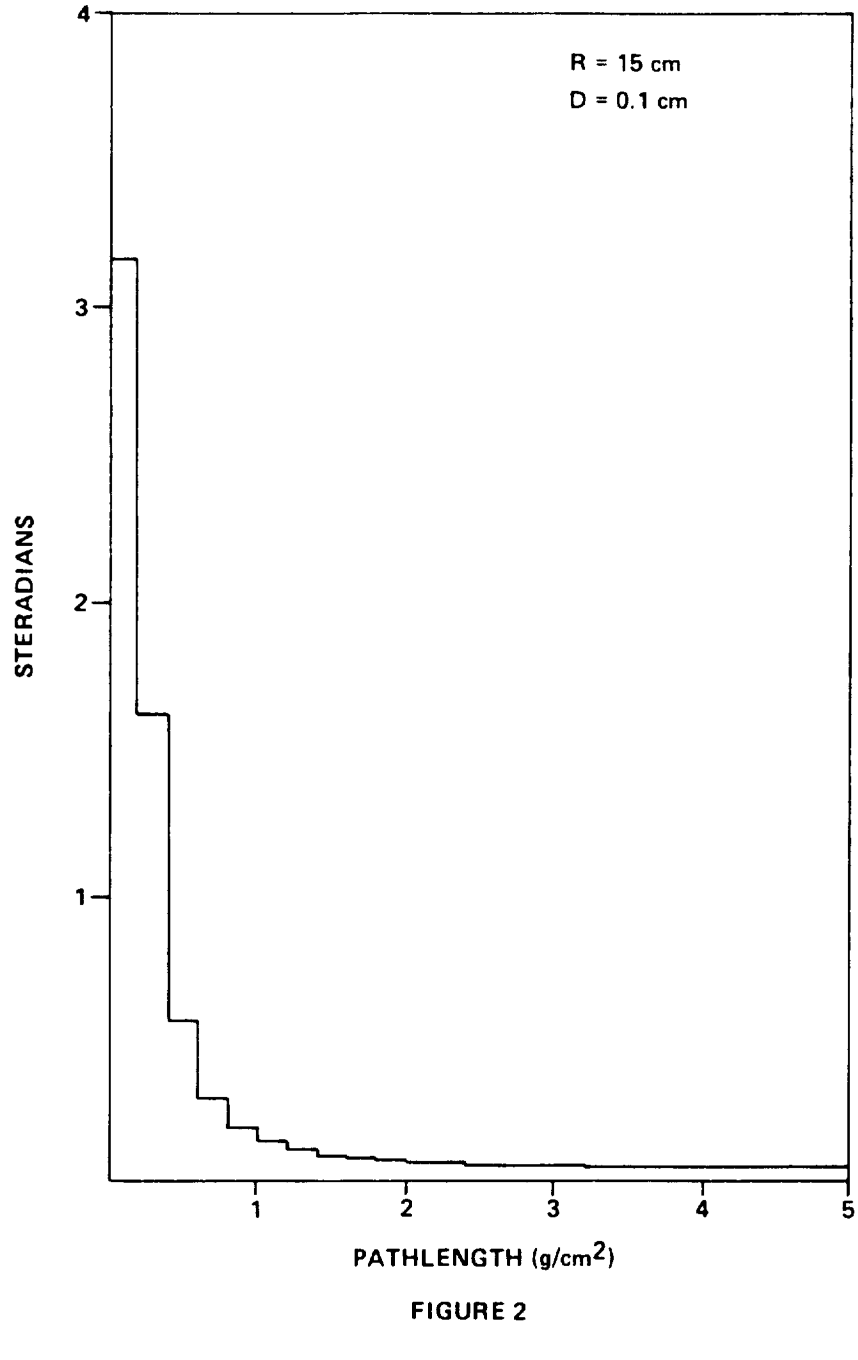


FIGURE 2

There are two important mechanisms for degrading high energy particle fluxes in matter: (1) ionization loss, and (2) nuclear fragmentation. Ionization loss is a continuous slowing down of charged particles introduced by their collisions with atoms. It effectively gives the charged particles a finite and well-defined range in materials. Table 1 shows the ranges of several ions at several energies in water, aluminum, and carbon dioxide. The table shows that: (a) Shielding materials consisting of lighter atoms are more effective at stopping fast ions, and (b) A few g/cm^2 of shielding has essentially no effect on most cosmic rays (> 1 GeV/N), but stops the heavy ions (and much of the proton flux) from solar energetic particle events.

Table 2 shows approximate interaction mean free paths for several ions in several materials. Unlike ionization loss rates, the interaction mean free paths are roughly independent of energy. Table 2 shows that: (a) Shielding materials consisting of lighter atoms are effective at degrading heavy ions by fragmentation, and (b) At some energy below 1 GeV/nucleon, nuclear fragmentation is a <u>more</u> efficient degradation mechanism than ionization loss.

One additional factor not comprehended in the Tables is the buildup of neutrons. Especially in materials of high molecular weight, neutrons are released from the target nuclei in ion interactions. The majority of the neutrons are released in proton nucleus interactions. Neutron build-up is best treated with an intranuclear cascade code (for example, Armstrong and Chandler, 1972).

<u>CONCEPTS: DOSE ESTIMATION</u>

Particle transport codes give high energy particle fluxes at any point within a structure or a body. The biological effects of this radiation are estimated by computing the rate of energy deposition by each particle type at each energy. A quality factor compensates for the increased damage associated with higher density of energy deposition. We use the following integral to compute dose equivalent:

$$D(S) = \quad J(S)\,Q(S)\,S\,dS$$

where J(S) is the flux of particles having LET of S and Q(S) is the quality factor associated with LET of S.

$$Q(S) = \begin{cases} 1 & S < 35\ MeV/(g/cm2) \\ 0.072S0.74 & 35 < S < 2000 \\ 20 & S > 2000 \end{cases}$$

TABLE 1

RANGES OF IONS IN MATERIALS (G/CM^2)

		H_2O	CO_2	Al
H :	30 MeV/N	0.9	1.0	1.2
	100 MeV/N	7.7	8.9	10.0
	1 GeV/N	330.0	370.0	410.0
	10 GeV/N	4700.0	5100.0	5800.0
C :	30 MeV/N	0.3	0.35	0.4
	100 MeV/N	2.6	3.0	3.3
	1 GeV/N	110.0	120.0	140.0
	10 GeV/N	1600.0	1700.0	1900.0
Mg :	30 MeV/N	0.16	0.18	0.21
	100 MeV/N	1.3	1.5	1.7
	1 GeV/N	54.0	61.0	68.0
	10 GeV/N	780.0	840.0	950.0
Fe :	30 MeV/N	0.09	0.11	0.12
	100 MeV/N	0.67	0.78	0.90
	1 GeV/N	27.0	30.0	33.0
	10 GeV/N	380.0	410.0	470.0

Note: This table is based on theoretical calculations and empirical fits known to be approximately correct. It has not been checked explicitly against measurements.

TABLE 2

INTERACTION MEAN FREE PATHS OF IONS IN MATERIALS (G/CM2)

	H_2O	CO_2	Al
H	74	84	99
He	36	40	51
C	19	25	34
Mg	13	18	25
Fe	8	11	16

All values are given at 1 GeV/nucleon. Variations of up to a factor of 2 occur at lower energies down to 10 MeV/nucleon. Little variation occurs above 1 Gev/N.

Note: This table is based on theoretical calculations and empirical fits known to be approximately correct. It has not been checked explicitly against measurements.

This is our parameterization. Note that relativistic protons have S = 2, relativistic C has S = 72, slow protons (a few MeV) have S = 100, relativistic Fe has S = 1400, and all cosmic rays of interest have S < 105.

It is important to note that relativistic Fe is thousands of times more damaging than relativistic protons (using our quality factor). Slow Fe, for example from a heavy ion rich solar flare, is tens of thousands of times more damaging than the minimum ionizing particles. We emphasize the most effective shielding is the (approximately) 5 g/cm^2 needed to eliminate heavy ions from solar flares and low energy cosmic rays.

RESULTS

We have previously (Silberberg et al. 1984) calculated the dose equivalent rate to a 30 cm spherical phantom at various depths. Results are shown in Figures 3 and 4. Figure 3 shows the free space exposure. A rate of 36 rem is taken from the 0.1 cm depth. To this is added an estimated neutron dose of 7 rem to give our baseline of 43 rem. Figure 4 is the same dose calculation except under 4 g/cm^2 aluminum shielding. This shielding thickness is thought to be typical of spacecraft. The maximum cosmic ray dose in Figure 4 is 26 rem, to which we add 10 rem for neutron buildup in the shielding. Little reduction in dose is associated with shielding.

Figure 5 shows the relative contributions of various charge groups to the dose equivalent. Note that heavy ions are the most important component of the dose at all depths.

During the writing of this report, we have recomputed the baseline dose in free space. This recalculation was suggested by the many improvements in our transport codes and particle environment models over the past few years. We quote a preliminary result of 47 rem for the cosmic ray primary dose, to which must be added 7 rem from neutrons. Thus the baseline dose may be as high as 54 rem. We emphasize the preliminary nature of this result which is given as a guide to the uncertainty of our calculations.

Figure 6 shows the dose equivalent rate (per solid angle) at slab depths of up to 60 g/cm^2 CO_2. Cosmic rays at solar minimum in the charge range Z < 29 were used as the incident flux. The "zero" depth point is actually under 0.1 g/cm^2 so very low energy fluxes have been removed.

3636–86

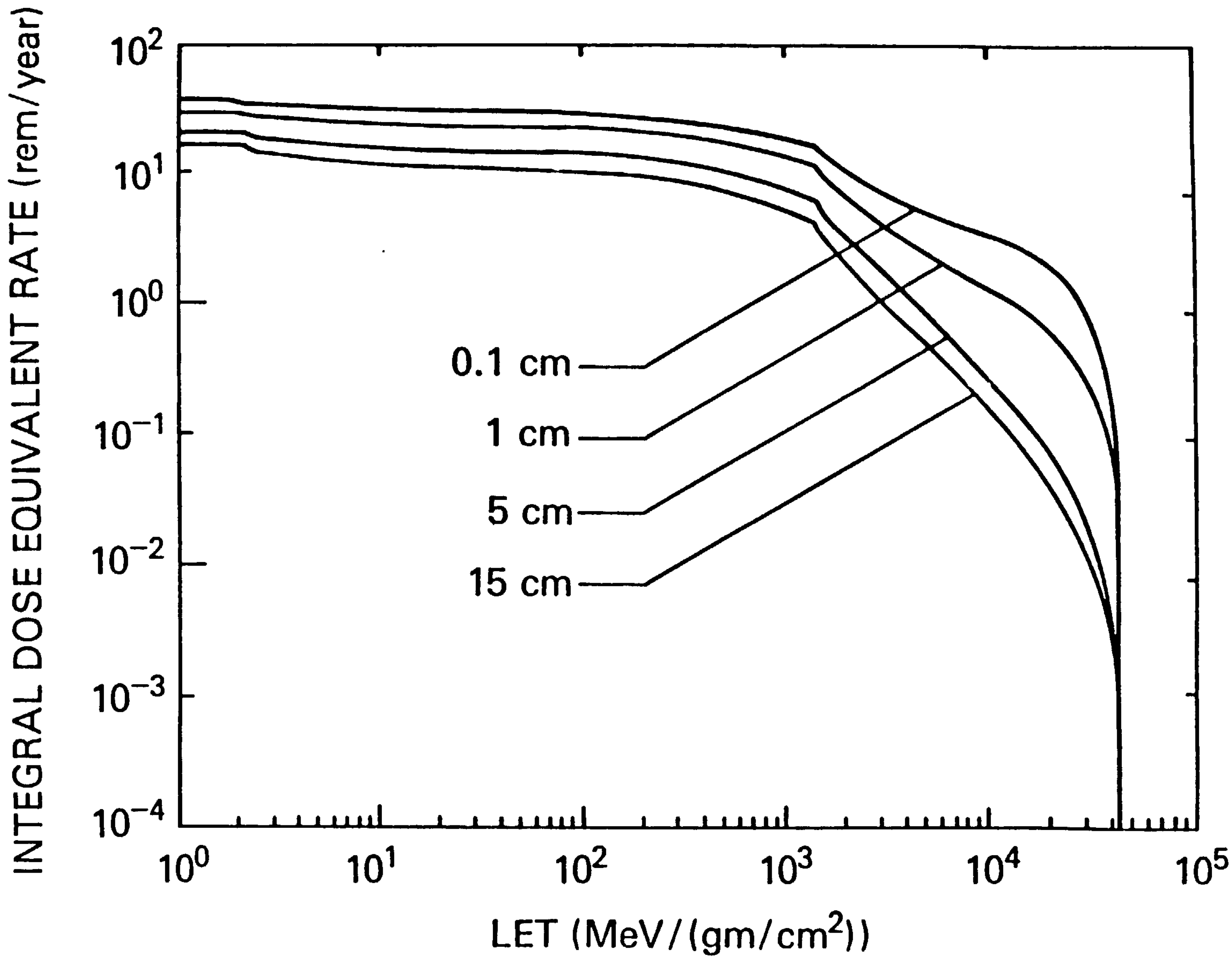
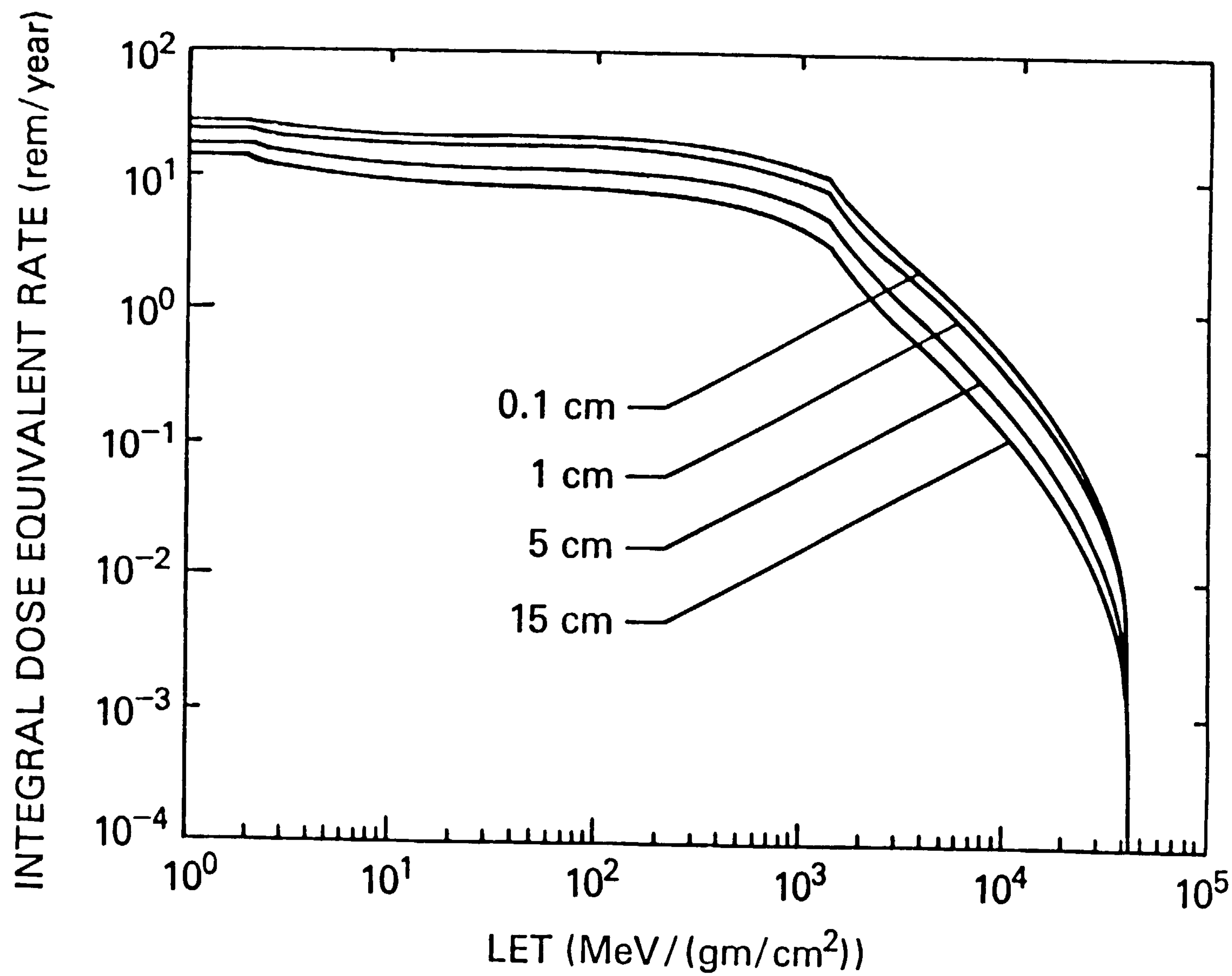


FIGURE 3

3635–86

FIGURE 4



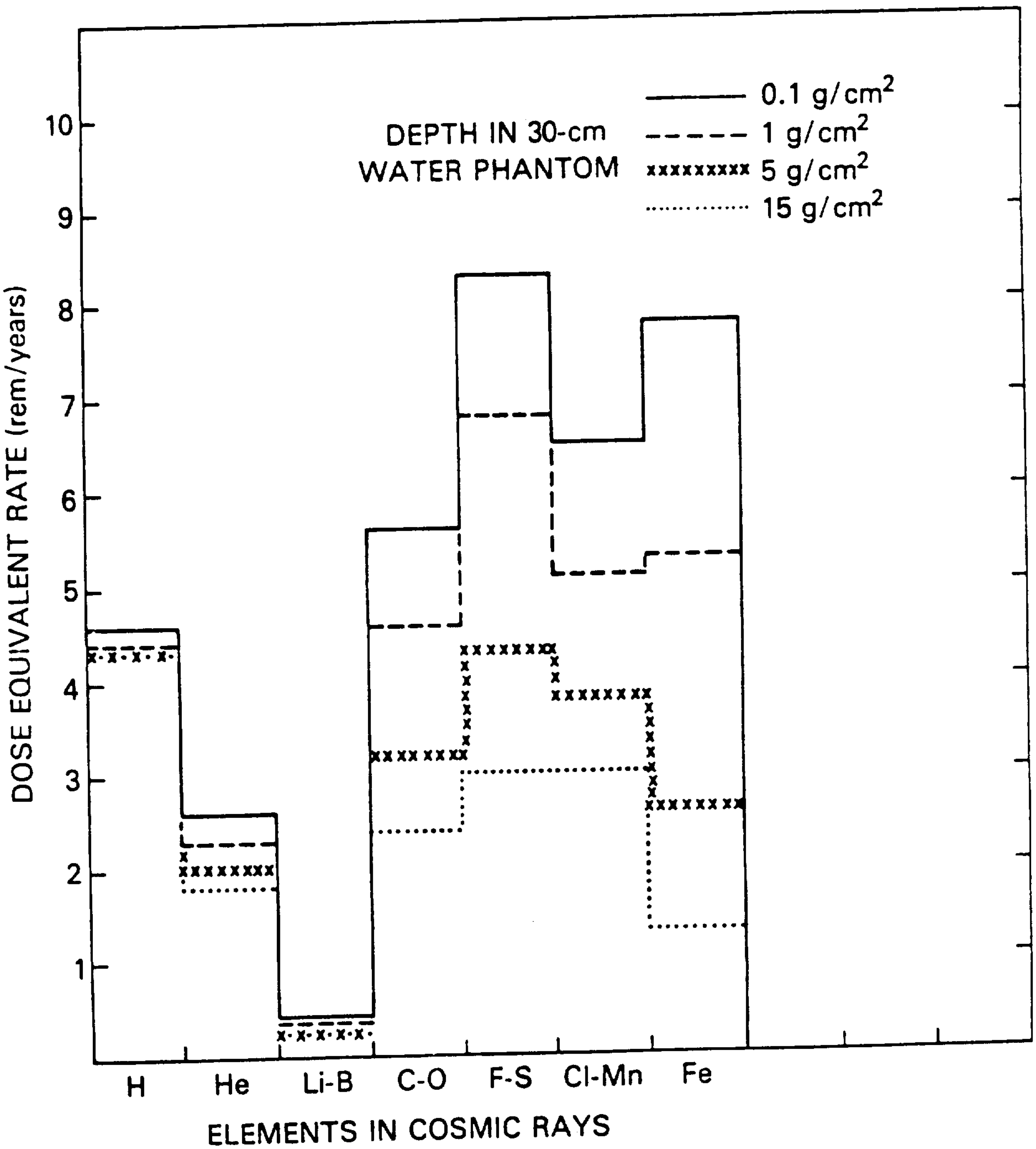

0.1 g/cm²
DEPTH IN 30-cm WATER PHANTOM
1 g/cm²
5 g/cm²
15 g/cm²
DOSE EQUIVALENT RATE (rem/years)
10
9
8
7
6
5
4
3
2
1
H
He
Li-B
C-O
F-S
Cl-Mn
Fe
ELEMENTS IN COSMIC RAYS

FIGURE 5

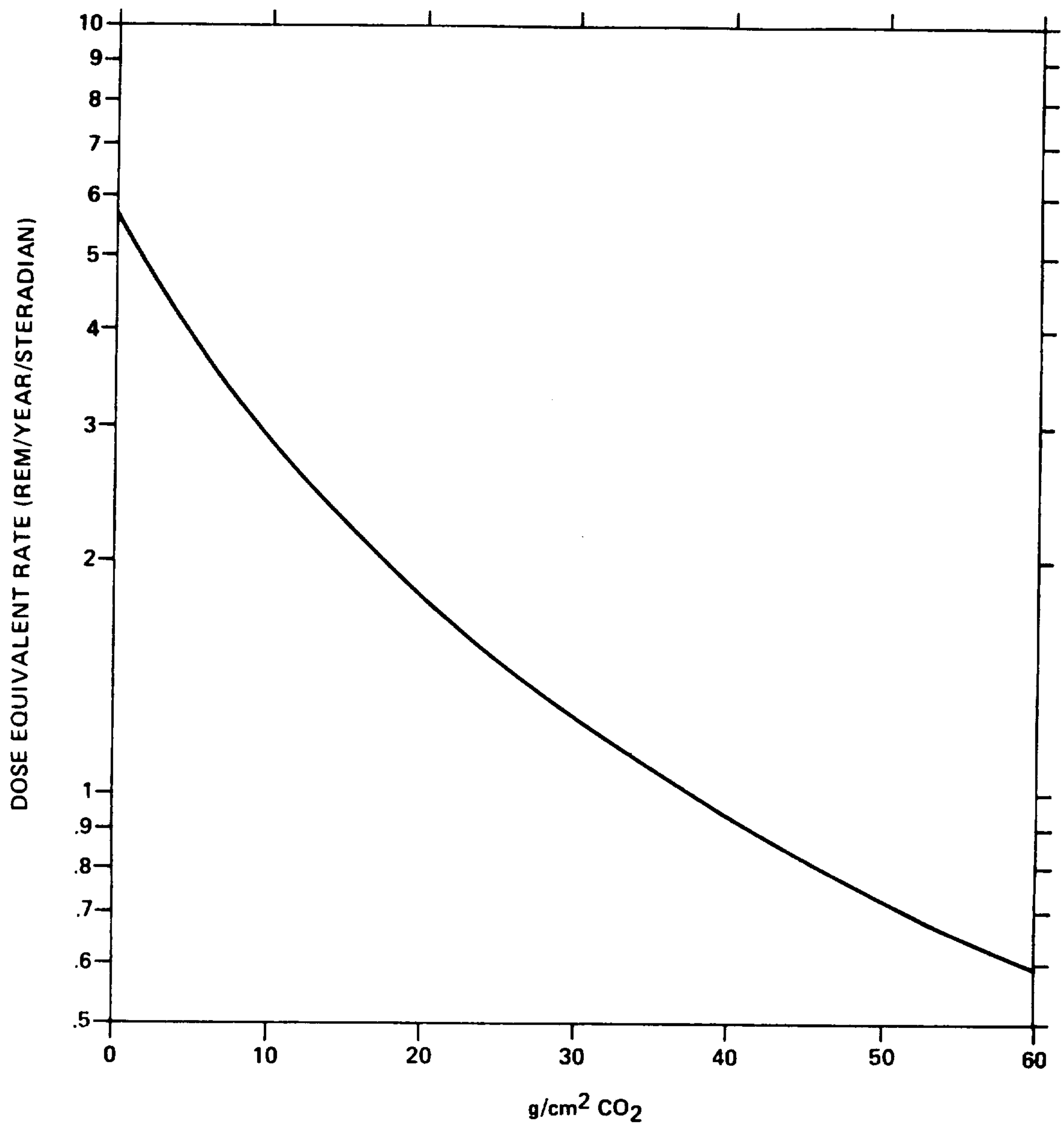

DOSE EQUIVALENT RATE (REM/YEAR/STERADIAN)
10
9
8
7
6
5
4
3
2
1
.9
.8
.7
.6
.5
0
10
20
30
40
50
60
g/cm^2 CO_2

FIGURE 6

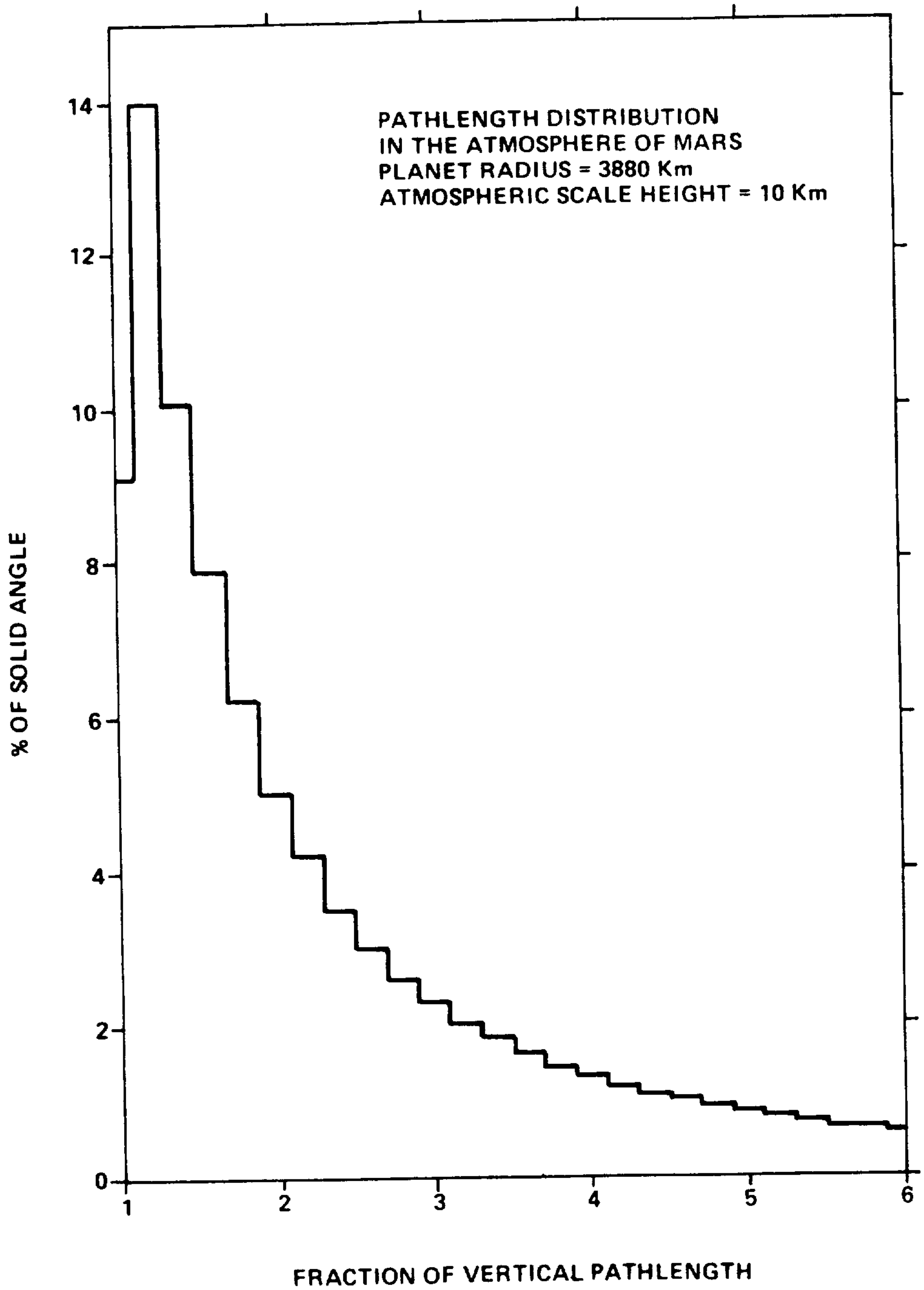

PATHLENGTH DISTRIBUTION
IN THE ATMOSPHERE OF MARS
PLANET RADIUS = 3880 Km
ATMOSPHERIC SCALE HEIGHT = 10 Km
% OF SOLID ANGLE
0
2
4
6
8
10
12
14
1
2
3
4
5
6
FRACTION OF VERTICAL PATHLENGTH

FIGURE 7

Figure 7 shows the pathlength distribution in the martian atmosphere. Since the vertical depth is variable and uncertain, the function is described in terms of fractions of this pathlength. An atmospheric scale height of 10 km was used to determine the distribution, though the results are insensitive to the scale height. Combining Figures 6 and 7 (with an overall factor of 2 pi steradians) gives a total cosmic ray dose at the planet's surface of 10 rem/year. We estimate a contribution of about 2 rem/year giving a surface dose of 12 rem/year.

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6. Silberberg, R., and Tsao, C.H. 1973, Ap. J. Suppl. 25, 315.
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N87-17778

SOCIO/PSYCHOLOGICAL ISSUES FOR A MARS MISSION

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ABSTRACT

This paper addresses some of the socio/psychological problems expected to accompany such a long-duration mission as a trip to Mars. The emphasis is on those issues which are expected to have a bearing on crew performance.

Results from research into aircraft accidents, particularly those related to pilot performance, are discussed briefly, as a limited analog to space flight. Significant comparisons are also made to some aspects of long-duration antarctic stays, submarine missions, and oceanographic vessel voyages. Appropriate lessons learned from U.S. and Russian space flight experiences are provided throughout the paper.

Design of space missions and systems to enhance crew performance is discussed at length, considering factors external and internal to the crew. The importance of incorporating such design factors early in the design process is stressed.

INTRODUCTION

A manned mission to Mars is expected to last some 600 days. Forty days would be spent on the planet and the round trip would take some 280 days each way. Such a mission would require a high level of investment and consequently would carry expectations of a good return. The crew would be the focus--the hand of man--in this first direct human touch on this distant planet.

The socio/psychological problems that need to be addressed for such a long mission to such a distant planet are the issues related to crew performance. The acceptable range of performance will vary from a non-negotiable criteria for survival to high productivity, both while at Mars and on the trip to and from the planet. The approach is to explore the areas susceptible to planning and design that will sustain the crews and optimize their performance.

PERFORMANCE

Crew performance, in its simplest definition, means that the crews will be able to carry out the mission objectives successfully and will complete the entire trip safely. Enhanced performance implies that the crew will not only do all that is expected well, but they will also be able to do things that were not planned for and which clearly enhance the mission outcome. The foundations for such crew performance are those which foster crew members who are alert, attentive, vigilant, motivated, flexible, skilled, knowledgeable, aware of and understand the mission goals and spacecraft systems, able to successfully operate and maintain the equipment during the entire mission, and are capable of functioning effectively as a small team.

Of course it would be unreasonable to expect a mission of this length and type to be without any problems. With this as a basic position, the high level of performance is set as an ideal goal, and planning and design should focus upon the optimization of that goal given the knowledge, funding, and resources available.

The job to be done in the planning phase relative to crew performance, then, is to determine the levels of performance that can be reasonably expected on the flight, to identify the design features and operational approaches that can be developed to achieve those standards, and to establish methods within the program structure for the design and development of the hardware, training, and operations that will effectively include these features.

ANALOGS

PILOT ERROR

Space flights have been enormously successful, and thus it is easy to assume that the procedures used in the past are more than sufficient to the task of preparing for a Mars mission. A Mars mission is different from our past experiences in a number of fundamental ways which makes this an untenable assumption. First, the trip to Mars is far longer than any mission ever undertaken in space. It includes a descent to the planet, operations on the planet, and return to the mother ship that will demand the maintenance of complex landing, ascent, and docking skills. It will require dependence on high levels of automation. It will require a high degree of interpersonal and group living skills.

Though flying a modern commercial jet airplane is not an exact replica of such a mission, there is much that can be learned from the intense work that has been done in looking at the human factors that are related to accidents.

The first thing we find is that the human factors are a major ingredient in accidents. We also find that though automation solves some of these problems, it can cause other problems. It is also extremely clear that the human factor is extremely complex, and is not accessible from simple common sense examinations. First we need to ask the right questions, then devise ways to seek answers, and finally we will be ready to ask how we can optimize performance.

In the early research into aircraft accidents, the question that was asked was "what went wrong, or what happened?" The next set of questions begin to explore "why?" These need to be followed by "what goes wrong?" and "why?" Some of the areas that have been found to be related to accidents have been identified in a recent book on pilot error, and include: (1) Human perception, information processing, attention, decision making, and action; (2) Visual illusions as related to refraction, textures, and autokinetics; (3) Assumptions when related to expectancy, anxiety, focus of attention, and as related to periods of high concentration; (4) Habits; (5) Motivation with its level and direction; (6) Stress and stressful environments; (7) Fatigue; (8) Workload; (9) Judgement; (10) Failures of automatic equipment; (11) Failures of automatic equipment compounded by crew error; (12) Failure to monitor; (13) Loss of proficiency; (14) Lack of proper vigilance; (15) Crew coordination; (16) Confusing documentation; (17) Workplace design; (18) Displays; (19) Software; (20) Cockpit discipline and professionalism; (21) Command as leadership or intimidation; and (22) Communication. (see Hurst, PILOT ERROR, NY: Aronson, 1982)

This research into aircraft accidents, and consequently pilot performance, shows a very complicated set of variables that occur in very dynamic contexts. It also demonstrates dramatically that the performance of the person inside of and running a complex machine needs to be examined with as much intensity, rigor, dispassion, detachment, and objectivity as any other system in that machine. Because a person <u>can</u> adapt, does not mean they always will, or that their capacity to do so is unlimited. What has been found in this field is that when a professional approach is taken to the understanding of the person as a legitimate subsystem interfacing with many other subsystems, much can be done to optimize the performance of both the machine and the person--total system optimization can be enhanced. Any serious attempt to develop the elements required for a Mars mission will need to include a thorough immersion and understanding of the work that has been done in this field.

PRECEDENTS

The most obvious precedents to a Mars mission are the very long Soviet missions on their Salyut Space Stations. The record to date is 237 days. Crews showed that they could perform successfully for that amount of time, though many problems were identified. As these missions have become longer and longer, the Russians have gradually enhanced the design of the station, the communications, the types of supplies, and the daily operational schedules. They still have much to do, but they have clearly shown progress. Any serious planning for a mission to Mars would need to evaluate carefully the lessons learned on these flights. It should be noted that the Russians have found that the provisions they have made for the socio/psychological factors have been extremely important in maintaining crew performance.

Other analogs, such as long stays at the Antarctic, nuclear submarine patrols that last for 90 days or more, oceanographic research vessel voyages, etc., also provide much valuable insight into the factors related to the performance of people in isolated and confined environments for long periods of time. One important outcome of the examination of these analogs, however, is the point that isolation and confinement, per se, do not usually cause dysfunctional performance on the part of crews. Rather, isolation and confinement exacerbates conditions that are stressful or error generating, acting as a catalyst which makes difficult conditions much worse than they would be in any other environment. This leads us to the need to consider the factors related to the generation of stress, error, and otherwise dysfunctional performance with the assumption that once identified, many of the factors can be attenuated through design, training, and planning. The range of experience and research to draw upon thus extends beyond that found in isolated and confined environments to the whole realm of performance, productivity, error, and stress.

THE CONTEXTUAL APPROACH

There are two levels of context to be taken into account in designing to enhance performance. The first is the context of the mission itself, meaning the spacecraft, the crew, the operations, etc. The second is the context of the design process that generates the spacecraft, the mission objectives, the operations, etc.

Any study of the work to be done relative to performance, productivity, error, and stress, requires a systematic approach which will ensure the results will be pertinent to a Mars mission. The space environment is significantly different because of microgravity, radiation, total provision for life support, and lack of accessibility for rescue, and thus the total context of the mission must constantly be kept in mind. Human performance is the result of a vast range of constantly changing events and influences which occur in a series and over time within this context.

Figure 1 shows some of the elements that need to be taken into account in evaluating the performance of crew in a space flight.

THE FLIGHT PERFORMANCE CONTEXT

People act in a context of perceived and unperceived factors which, in concert at any given time, influence the nature, content, and quality of their thoughts and actions. Furthermore, this context is constantly changing. There is a present, past, and future milieu, all of which impinge on the moment. None of these circumstances is static (the rose and the banana never remain the same, day after day).

The changing context can be looked at from two points of view: Internal and External. Actions are carried out by an individual within a space and time that includes many features of the physical environment as well as the presence and actions of other people--the External Context. At the same time, these people act relative to the scope of their own capacities, perceptions, physical state, and experiences--the Internal Context. Performance enhancement includes both of these contexts.

The broken lines with two arrowheads signify the idea that the external context impinges upon the individual, but that it is filtered by the internal context, resulting in a mental and physical tone which, coupled with motivation, processing and perception, results in a performance or behavior.

HUMAN PRODUCTIVITY CONTEXT

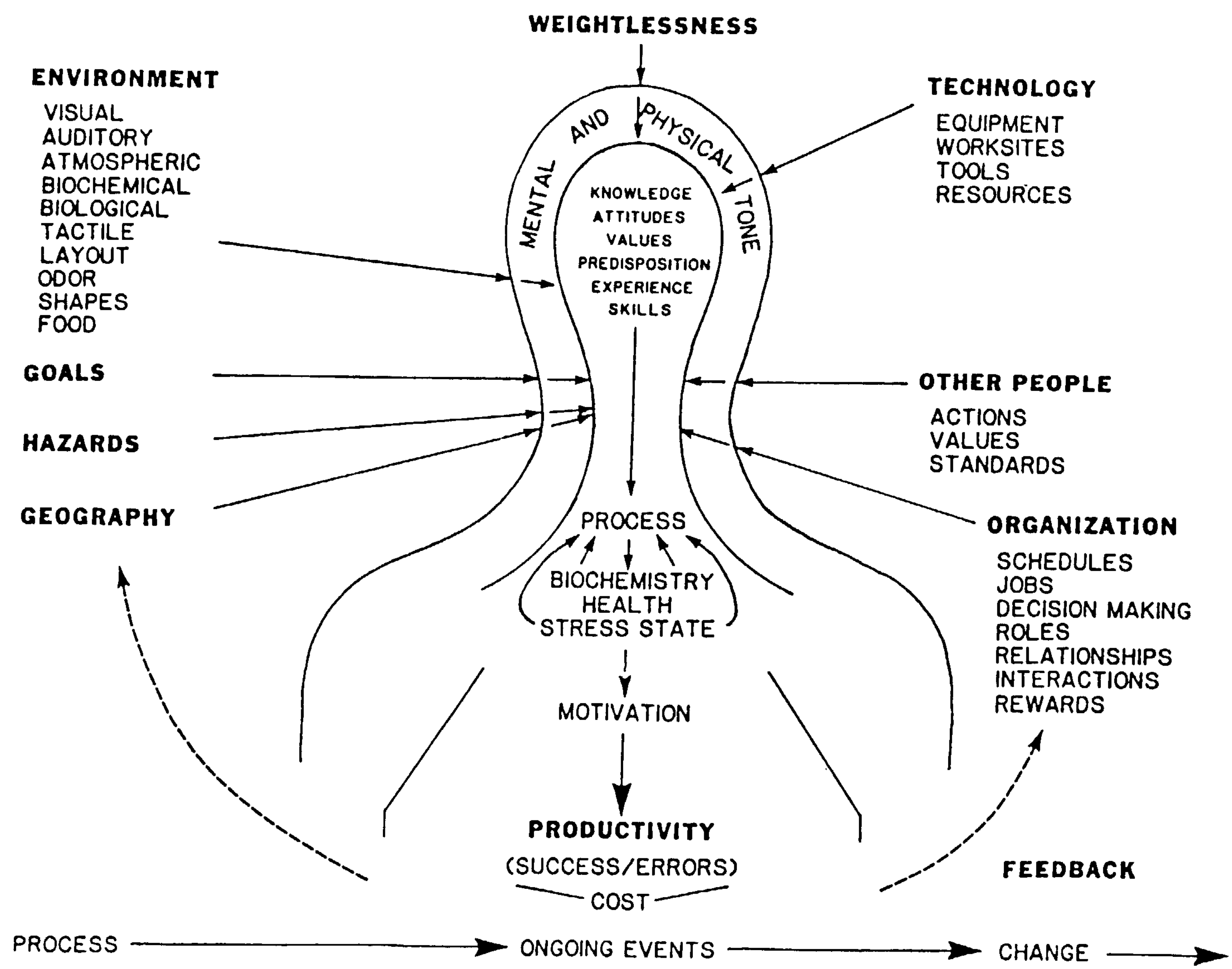

FIGURE 1

DSG-48

The External Context

There are eight factors that make up the External Context: (1) Weightlessness; (2) Environment; (3) Technology; (4) Goals; (5) Other People; (6) Hazards; (7) Organization; and (8) Geography, or more properly, Location.

The External Context has been found to be an important factor in crew performance. Poorly designed environments can cause undue frustration, stimulate error, create fatigue, and impact overall motivation and morale, consequently affecting the performance of the crews. The effects of these factors are related to errors, accidents, increases in the time needed to perform tasks, decreases in the amount of effective work time, poor planning, increases in the time it takes to detect errors or problems, and higher numbers of experiments or tasks that need to be repeated or redone because of original mistakes. Once the cycle of difficulty starts, it can snowball as the crews try to make up for the losses and attempt to maintain the original objectives. Stress in the External Context quickly increases stress in the Internal Context. In the confined and isolated environment of a spacecraft on the trip to or from Mars, there are also few diversions to permit a turnaround--re-creation--making it all the more important to minimize as many potential stress or error generators as possible.

The design and control of the External Context is therefore significantly related to the overall performance and success of the crews, especially on a mission as long as the one to Mars. The initial danger, however, is to focus on the engineering and technical tasks in the design and development phases, assuming that the human elements can be inserted at some later time, or to assume that the crew will be so well chosen that they could be expected to adapt to design difficulties. A certain degree of failure would be expected, so that success beyond the primary mission of going, landing, and returning safely, would be a bonus given such a dangerous and unknown mission. This does not have to be the case. So much is known about the design of External Contexts that facilitate human performance that the inclusion into the early phases of the engineering design would not be extremely difficult or expensive. The outcome would not only go a long way in providing for the success of the mission, but also would enhance the cost effectiveness of the return

c-3

relative to the investment. The real challenge here is the organizatonal one of providing an effective method for embedding the human performance factors into the design and development process itself.

Weightlessness

Crew performance is dramatically affected by the microgravity environment due to its effect on all types of material objects and mediums, the posture of the human body, the need for various kinds of restraints to obtain leverage or stability to perform a wide range of actions and to manage equipment or materials, and the freedom of movement and placement it provides that is not present in the one-gravity of Earth.

Skylab crews found that attempting to maintain an erect posture for long periods of time was painful on stomach muscles. Equipment needs to be designed to the neutral posture the body assumes in space. Shuttle experience shows that a significant amount of time can be spent in caring for personal hygiene and daily maintenance. Control of small articles of material or equipment could be a major problem if there were not adequate restraints. The Soviets discovered that they could solder in space, but that the residue could easily float into their eyes. Small pieces of paper, tools, water, food, or parts could get lost or float behind panels either to be gone, or possibly to damage the equipment. Because of the freedom of movement that weightlessness presents, lighting is also affected. Simple ceiling lighting would not be helpful if someone chooses to work in an anomalous position. Traffic patterns, layout, and handholds will be different because people float to translate from one place to another, and need some means of stopping their forward motion without danger to themselves or to the equipment.

Weightlessness provides significant opportunities for the modularization of interior components so that they can be detached and moved or replaced as the need arises regardless of the weight.

In weightlessness, odors and heat can collect in various nooks and crannies of the structures which can result in unpleasantness or contamination that could affect health.

Tools and equipment need to be designed for weightlessness. The Soviets have made special screwdrivers, hammers, wrenches, and cutters to be used in the weightless environment. Medical equipment, for instance, will provide special challenges because so much of current medical practice and equipment is implicitly dependent on one-gravity (for instance, a special IV system will need to be designed).

Design thus will need to be fitted to weightlessness if the crew is to avoid the frustrations, accidents, and errors that could be generated by designs that are not carefully thought through.

Environment

The Russians have made many changes to enhance the physical elements of the space station interior environment. In order to lower the noise from fans and other equipment, they have developed tools and systems that permit the crews to replace and move the equipment during the flights. Space Shuttle measurements of noise show an 80 DB level in the Forward Avionics Bay at the floor level (25 DB over the recommended 55 DB design standard), 68 DB in the center of the Mid Deck, 61-64 in the sleep areas, and spikes up to 87 DB when the Waste Collection System is used. Skylab also showed that in spite of the low 5 psia atmosphere, sleep could be interrupted by intermittent noise or the movement of another crew member around the cabin. Soviet cosmonauts and sailors have also commented on the comfort of a constant noise at a reasonable level, but that intermittent noise, extremely loud noise, or the starting or stopping of noise in unexpected ways could be quite stressful.

The Soviets have also made changes in the visual appearance of the interiors of the Salyut Space Stations with the addition of stronger colors on the walls and ceilings, contrasting accent colors, and provision for the display of posters, pictures, and other personal items brought by the crews. Soviet uniforms also are characterized by a variety of colors and designs to provide visual stimulation and variety. Color television has been installed to permit the crews to interact with family, friends, scientists, and engineers on the ground as well as permit the use of videocassettes which can provide a wide range of visual stimulation. All of these have been provided to alleviate boredom and the monotonous nature of life in such confined and unchanging surroundings.

Another environmental contextual factor that has been found important is food. Tastes seem to change over the total scope of a mission, and meals are important times of the day both to enjoy the food, but also to fill social needs. At one point, the Soviets ceased planning the meals for the flight crews and simply asked them to meet a given caloric intake each day in order to permit them to make the choices themselves. (However, nutritional requirements have been given priority again and some system of meal planning is to be reinstated). Crews are supplied with fresh fruits and vegetables by the Progress resupply ship which comes every three to six weeks, and a special hatch was installed to permit this loading to take place, but a few hours before the Progress launch. The Soviets have also worked at learning to grow lettuce and other vegetables onboard the ship (watching the growth of plants and flowers also seems to supply an important psychological boost).

Technology

The balance of automation and machines that require human manipulation will be an extremely important variable in a trip to Mars. There must be enough automation to permit the crew to be fairly small and to leave the crew sufficient time to carry out the experiments and daily operations of the long flight. At the same time, there can not be too much automation--leaving the crews with little to do--the seeds of boredom. Furthermore, the automated systems must have a level of reliability, as perceived by the crews, that inspires confidence for such a long trip. If a system breaks down, can the crews fix it? If it fails, can they carry out the operations manually and will their skill level be maintainable for a wide range of operationally related failures that can not be easily predicted? Will they know if the equipment has failed or perhaps shifted data or operations in some minimally detectable way? What is the back up? How transparent is it? How much skill is required to explore the system for malfunctions or to reset it to respond to unexpected events? To what degree can false alarms which will persuade the crew to take action that is not required affect safety or equipment, or to what degree can they become so frequent as to be generally ignored? To what degree can the crews induce error in setting up automated equipment and how significant can that be? Can the crew fail to monitor the equipment adequately either from boredom or from excessive confidence? How much knowledge or skill would be required to perform a major repair on any of the automated systems? How will the automation affect the basic attitudes the crew members have toward their roles and their importance in carrying out tasks? Can we design automated systems the way users want them designed when the user may not be identified for some ten years or so?

Automated systems are necessary for space flight, but they do misbehave. The Soviets installed a flight navigation computer called the Del'ta which at one point began to store data that was to be used, but it erased data that was still needed. The crew had to replace the memory, and the memory replacement took a week with the use of the telemetry from the ground. The Soviets are moving to higher and higher levels of automation, however. They have installed much more sophisticated systems on their Kosmos 1443 type logistics module to provide for precise navigation and pointing, and they use automatic systems for transferring fuel, gases, and liquids from the Progress resupply vehicle to the Salyut-7. They have found that large ground support teams are extremely expensive to maintain and to keep alert and say they hope to transfer a large number of these current ground operations to the space station itself by means of automation.

The Soviets have gone to considerable lengths to provide for onboard maintenance and repair of both small and large systems. They have carried out a major fuel line bypass, installed new solar arrays, disengaged a very large tangled and stuck antenna, and moved and replaced thermal pumps which were permanently installed with welded metal clamps. They are working on a cutting, welding, soldering, and spraying tool. They have used drills and power saws. The assumption here is that things can and will malfunction or break down and they need to be fixed by the flight crews. To do this, it is necessary to supply adequate tools and information to carry out the tasks whether they are anticipated or unanticipated, IVA or EVA.

Goals

The goals of the mission, and the way they are understood and perceived by the crews will be important drivers in guiding their actions and supporting their motivation. The first Mars mission will be sufficiently unique and outstanding in that it will carry a high leverage for the crews and thus this factor will not be as critical as goals will be later when missions in space and to places like Mars become routine.

Other People

The special chemistry of a given group of people at a given time, and as it changes over time, will be an extremely important factor in the capacity of the crews to perform well and to maintain their motivation, morale, attention, vigilance, and alertness over the entire length of time it will take to go and return from Mars. Since there is total isolation and total confinement, the crews will be forced to meet and manage all of the problems created in the dynamics of their small community. With events, they will probably change, and so it is not possible to predict all of the factors that will be present over time.

Much is known about the dynamics of small groups, and that can be of great value in preparing the station and the crews for the social and interpersonal relationships they will encounter.

As an External Context, other people are salient in terms of their values, the degree to which they meet and abide by perceived standards and rules, their capacity to support the team, and their general compatibility in terms of customs, culture, and the resulting manifest behavior. Submarine crews speak of the "testing" that is done to new members of a crew to see if they can be "depended on" in an emergency and the Soviets have spoken and written frequently on the need for compatibility if crews are to be successful. In spite of their rather extensive efforts to provide for compatibility, they still report on instances of interpersonal stress and conflict that can intrude on the mission goals.

To meet the need for manageable and smooth interpersonal relationships, methods for selection and training can be merged with the development of organizational systems that will enhance the day-to-day management of the small group dynamics of the crew.

Hazards

For socio/psychological reasons, the perception of hazards and methods used to combat those hazards may be more important than the actual reality. On a mission of this length, there are a wide range of potential hazards which include radiation, hits by space debris or meteorites, medical emergencies, or mechanical failures, to mention a few. The crew needs to be confident that they have a good chance of identifying and compensating for these kinds of hazards. In the Nuclear Submarine Service, for instance, crews do not see the escape or rescue measures as the most significant control for an accident at sea. Rather, they perceive their ongoing maintenance and repair capability combined with the skills and knowledge of their crewmates as the primary means of preventing such eventualities before they ever happen. On modern SSBN or Attack submarines, none of the life support or ship's control systems are fully automated. Rather, crew members carry out most of these monitoring and control activities and, where computers are used, they perform a backup function.

Organization

The organization of a small group needs to fit the job to be done, the specific conditions of the job, and the people who are the members of the group. An organization that works well in one situation may not be transferable to another situation, or to another group of people.

Both astronauts and cosmonauts who have flown on very long missions say there is a qualitative difference between short missions of a month or less and long missions of three or more months. It is much easier to make adjustments for a short mission than on the long ones.

Decision Making

The Soviets have told their crews not to stress the command structure in daily activities, and have been willing to let the crews themselves determine who will carry out various activities during the mission. The kinds of decisions that need to be made for scientific research require a high level of flexibility and group consensus where priorities are involved. The decision making in an emergency, however, leaves little room for discussion, requiring a clear line of authority and unambiguous instructions. In the Antarctic with the current system of using civilian support teams, one team leader made a clear distinction between the routine decisions to be made relative to support and science, and those to be made in an emergency. The scientists were to carry out their own activities with no interference from him or his support team, but they were to keep him informed of what they were doing. He would become involved only if there were some problem of safety, use of resources, or of conflict with other station activities. However, if an emergency occurred, he was to be in unquestioned control.

Schedules

As their experience with long missions increased, the Soviets created a schedule that permits many breaks during the day and two days off each week (see Figure 2). Crew members will work themselves very hard when a mission begins, and will soon become fatigued and stressed if they keep up a high pace of activity. The Soviets have required that the crew take time out for leisure activites and they have a Group for Psychological Support on the ground who provide activities and supply special foods, videotapes, cassettes, books, surprises, two-way TV interviews and conversations with friends or famous people on the ground, TV or radio broadcasts of sports events, music, news, etc. In the beginning, the crews complain about this "waste of time" but, as the mission becomes longer, they speak of looking forward to these simple pleasures. The experiment system is also varied. Crews will focus on one or two types of experiments for a few months, and then will shift to another regime. On a given day, they will also focus on one or two experiments or activities. Crew members are trained to do all of the experiments (they may train for four years for a mission) but tend to specialize in terms of their own degrees of interest during the mission itself.

American Skylab astronauts also have spoken frequently about the need for a degree of onboard control of schedules and activities by the crews themselves. The lack of control was especially stressful when the ground specified in extreme detail every action down to the minute with little room for error or change.

Relationships

The Soviets have experimented with mixed crews for short periods of time by inviting cosmonauts from different countries to fly for a week or so and to devise and carry out a wide range of experiments. They have also flown a woman twice on these short missions. V. Remek, the Czechoslovakian cosmonaut, commented on the need to do a very wide range of planning and training relative to language and cultural differences if such mixed crews were to fly together for very long periods of time due to the potential for misunderstandings and the consequences which could result.

SALYUT SCHEDULING

ARISE	8:00
TEST OF THE STATION	8:00 - 8:20
MORNING TOILET	8:20 - 9:00
BREAKFAST	9:00 - 9:40
WORK	9:40 - 12:00
PHYSICAL EXERCISE	12:00 - 13:00
FREE TIME	13:00 - 13:20
WORK	13:20 - 14:20
DINNER	14:20 - 15:00
FREE TIME	15:00 - 16:00
WORK	16:00 - 17:30
TEA	17:30 - 18:00
WORK	18:00 - 19:00
PHYSICAL EXERCISE	19:00 - 20:00
SUPPER	20:00 - 20:30
FREE TIME	20:30 - 23:00
SLEEP	23:00 - 8:00

DSG-1293

FIGURE 2

Experiences at the Antarctic strongly reinforce the need to prepare with diligence for small crews with mixed cultural backgrounds even within nationalities but from different subgroups (scientists and Navy support teams) where misunderstandings and conflicting values and views can easily come to impede the mission goals. During one mission, the team leader attempted to minimize the gap between his civilian support team and the scientists by having Saturday evening reports on the scientific experiments that were being done (which ended up as parties), and by encouraging the support team members to help the scientists out in collecting their data and maintaining the equipment. To reciprocate, the scientists helped in some of the station maintenance tasks. Crews who participated in this sharing gave it very high marks and this mission apparently experienced less of the interpersonal problems encountered on other missions.

Rewards

Crews need some means to measure and recognize accomplishments that they find significant. Soviet crew members can look forward to national recognition, medals, trips, and career advancement for the long run, as well as confirmation of scientific breakthroughs by talking to renowned scientists during the flight. A 600-day mission will need some sustaining method to reinforce the efforts of the crews during the long stretches of the trip.

The Location

A Mars crew will be beyond rescue for almost the entirety of their voyage. This will be of enormous significance in the socio/psychological aspects of the mission.

The Internal Context

Each crew member will bring to the mission a whole set of internal predispositions, and these will be constantly influenced by the External Context and the events of the mission as it progresses. It is this Internal Context which is the seat of the capacity of the crew member to perform well. The choice that will need to be made prior to the mission will be whether to stress selection or training in putting together the crew for the mission. Selection is an easy choice, but people change over time; thus selection is a useful but a limited option. It is necessary, but not sufficient. Training can facilitate skills and knowledge, but more than is currently believed, also general internal predispositions, thus making the pool of available people larger when focusing upon general skill and knowledge. People can successfully be taught how to change attitudes, habits, and perceptual orientations as well as how to understand and interact successfully in small groups.

THE PROGRAM DESIGN AND DEVELOPMENT CONTEXT

Oddly enough, this Program Context is probably the most difficult one to change, and yet absolutely essential to the actualization of performance enhancement measures. To date, space programs have been driven within limited budgets and primarily to engineering criteria. Vehicles were experimental or developmental, missions were all fairly short, and crews were expected to adjust to the compromises that had to be made throughout the whole design and development process. As long as missions were short, this was a reasonable expectation and crews have shown both ingenuity and creativity in their ability to make these systems work.

With the maturity of the space program, however, there is a danger that these systems which were seen as very successful will be brought into a Mars mission design complete with the engineering focus. (If it isn't broke, don't fix it.) In this perspective, many of the elements that are required to provide for crew productivity and socio/psychological stability are seen as either luxuries or superfluous and are the first things to be cut as the program proceeds. Old program systems do not automatically include these design issues, and change is generally resisted because of the alterations it requires in the design processes. It is very difficult to include the performance factors as equal to the power or life support factors in change boards and in budgeting criteria. With the engineering culture of the aerospace community in government and in industry, such an inclusion will represent a qualitative change in the way they do business, and hence will require major alterations in attitudes, values, and procedures--a change in the culture.

What is paradoxical about this issue is that early inputs relative to performance factors and crew support are not always that expensive. They become prohibitive when they are introduced later, once the design has been set, and thus involve significant and costly redesign. An early legitimacy for the performance and crew support factors in the design and budget system would thus provide for the inclusion of the contextual factors that are most conducive to the enhancement of the living and working conditions of the crew over the long 600 days of the flight. The success of the mission may depend on it.

N87-17779

SOLAR PARTICLE EVENT PREDICTIONS FOR MANNED MARS MISSIONS

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ABSTRACT

Manned space missions to Mars require consideration of the effects of high radiation doses produced by solar particle events (SPE). Without some provision for protection, the radiation doses from such events can exceed standards for maximum exposure and may be life threatening. Several alternative ways of providing protection require a capability for predicting SPE in time to take some protective actions. The SPE may occur at any time during the eleven year solar cycle so that two year missions cannot be scheduled to insure avoiding them although they are less likely to occur at solar minimum. Reliable predictions of SPE can be made 20-30 minutes before particle fluxes reach a standard threshold that indicate an event has begun. An additional 20-30 minutes are available before the SPE rises to a dangerous level. Forecasts for SPE one to ten days in advance are made in a probabilistic mode. The forecasts are sufficiently accurate to use for setting alert modes but are not accurate enough to make yes/no decisions that have major mission operational impacts. Forecasts made for one to two year periods can only be done as probabilistic forecasts where there is always some chance of a SPE occuring. These are current capabilities but are not likely to change significantly by the year 2000 with the exception of some improvement in the one to ten day forecasts. The effects of SPE are concentrated in solar longitudes near where their parent solar flares occur, which will require a manned Mars mission to carry its own small solar telescope to monitor the development of potentially dangerous solar activity. The preferred telescope complement includes a solar x-ray imager, a hydrogen-alpha scanner and a solar magnetograph.

RADIATION HAZARDS FROM THE SUN

Space missions to other planets, including Mars, where most of the mission is outside the protection of the Earth's magnetosphere, will subject the mission crew to radiation hazards from solar particle events (SPE) produced by solar flares. Radiation from this source may reach levels of several hundred rads in periods of a few hours. Alternatives for avoiding these exposures are to schedule missions when no events will occur, provide on-board shielding sufficient to reduce the radiation to acceptable levels during the entire duration of the mission, or to provide temporary protection where the crew can remain during the few hours when the solar particle event radiation is at a high level.

We do not at present have sufficient information to schedule a two year mission to avoid all solar particle events. Figure 1 (from reference 1) shows that events may occur at any time, even at the minimum of the well known eleven year solar sunspot cycle. The capability to predict, with lead times of months to years, a two year period when no SPE events will occur is highly unlikely by the year 2000. However, it is clear that the chance of encountering a major event is reduced if the mission is scheduled around the time of sunspot minimum. The required mass to provide continuous shielding throughout a space vehicle and while on the surface of Mars is probably too great to be feasible for missions in the 2000 time period. The apparent alternative is to explore ways of temporarily avoiding the high SPE radiation levels by using temporary protection. SPE predictions are required to provide time for the crew to seek shelter.

SIZE OF THE PARTICLE EVENTS LIKELY TO BE ENCOUNTERED

SPE occur at the rate of about 100 per eleven year solar cycle. Most of the events are small and only a few are sufficiently large to be of danger. The number of SPE observed in the present solar cycle, which began in June 1976, are shown as a function of size in Figure 2 (from reference 2). The present cycle, designated as Cycle 21, has not produced events as large as those of the preceding Cycles 20 and 19, where peak fluxes of protons with energies greater than 10 MeV exceeded 10,000 (protons per centimeter squared per second per steradian) on several occasions. It is important to both mission design and mission operations planning to be able to rapidly distinguish the few large events from the many smaller events.

SPE FORECASTING

Efforts to predict SPE for operational users such as NASA are made by the Space Environment Services Center (SESC), operated jointly by the National Ocean and Atmospheric Administration and the U.S. Air Force, in Boulder, Colorado. These forecasts have been made routinely for the past

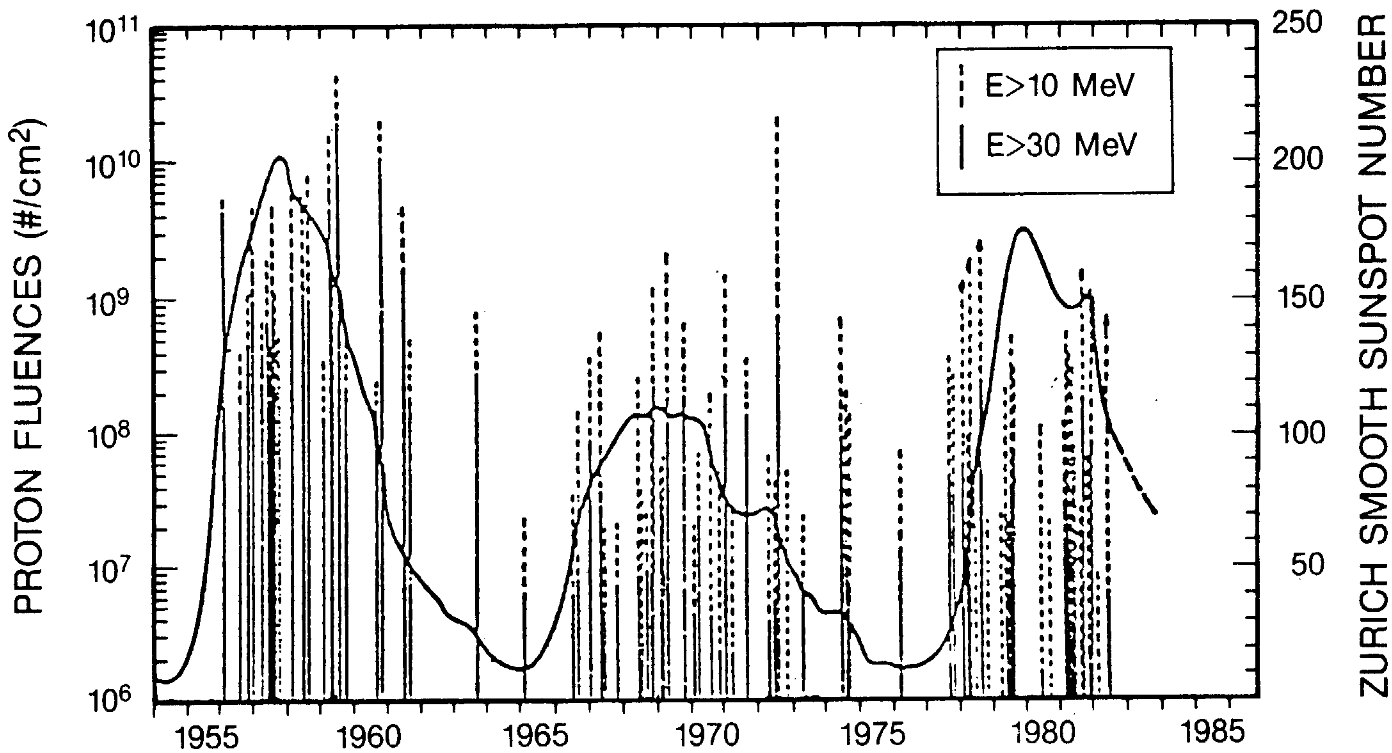

FIGURE 1: PROTON FLUENCES ABOVE 10 AND 30 MeV IN SOLAR FLARE EVENTS DURING SOLAR–CYCLES 19, 20, AND 21. THE SOLID CURVE REPRESENTS ZURICH SMOOTHED SUNSPOT NUMBERS.

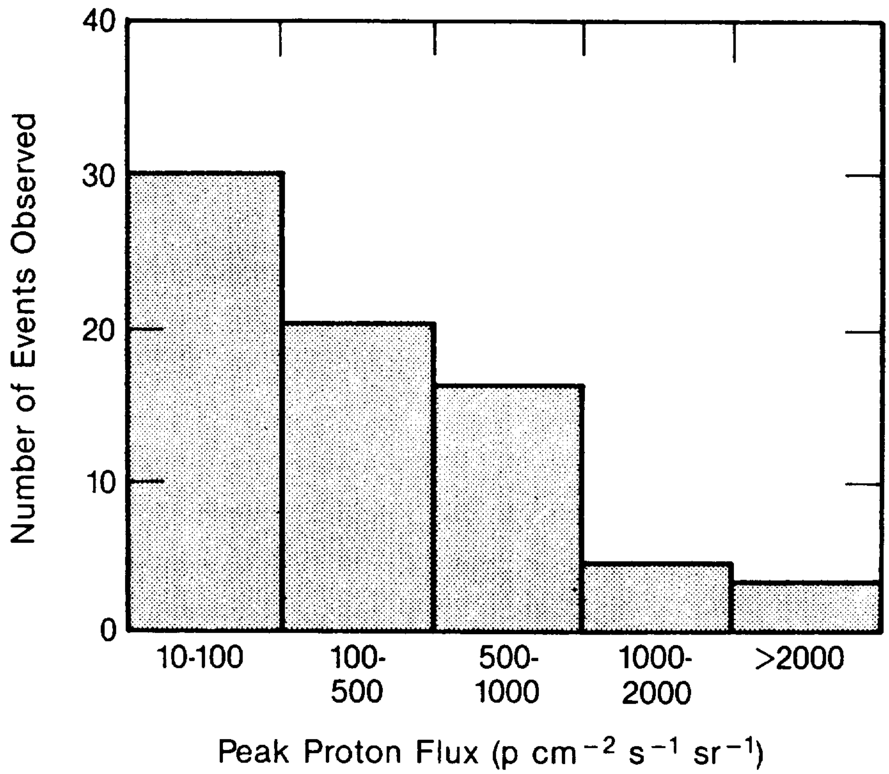

FIGURE 2. SOLAR CYCLE 21 EVENTS

20 years and have been supplied to NASA for mission operations support since the Apollo 8 mission. SESC makes predictions of SPE for two time periods. The first is the prediction, based on the electromagnetic radiation of a flare, of the occurrence or non-occurrence and the size of a particle event that may follow (less than 5 percent of moderate to large solar flares produce a particle event at the Earth (2). The flare is recorded by solar telescopes and subjected to computer and human analysis. The evaluation can be done in time for an alert to be issued and a prediction of the size of the expected event to be made 20 minutes before the protons reach a standard threshold. The threshold--a flux of 10 protons per centimeter squared per second per steradian--was chosen to meet several operational requirements, and corresponds to a radiation level of approximately one rad per hour. There are another 30 minutes available to seek shelter as the flux will continue to rise for several tens of minutes or more before it reaches a dangerous level. For Mars mission planning, it can be expected that the alert, if based on observations made at Mars, would have a 20 to 30 minute lead time.

The other type of SESC particle event forecast is made one, two and three days in advance. It requires the evaluation of the potential for a large solar flare to occur, and if it does, the potential for it to produce a particle event. These forecasts are probabilistic in form. They are useful as a "particle alert watch" signaling that a SPE may occur, but the forecasts are nonspecific and inaccurate enough that any action beyond a watch mode would result in many false alarms and missed events.

The present forecast system (for predicting SPE near Earth) operates with a combination of ground based optical and radio telescopes, supplied by NASA and the U.S. Air Force, and a whole Sun x-ray detector carried on the NOAA GOES weather satellites. NOAA is presently in the process of planning a solar x-ray imaging (SXI) telescope that will provide images of the Sun every few minutes. The instrument design is based on telescopes carried on the Apollo Telescope Mount during the Skylab missions. It is expected that SXI will raise the short-term SPE forecasts to a 95 percent accuracy rate -- that 95 percent of all events at the Earth will be predicted and no more than 5 percent of the predicted events will be false alarms.

SUMMARY OF FORECAST CABABILITY

A summary of the various forecast lead periods is shown in Table 1. The capabilities shown are those presently available. Barring a breakthrough in the research into the physics of solar flares, it is unlikely that these capabilities will improve substantially by the year 2000. The most probable improvement is in the probability forecasts done one to a few days in advance.

LOCATION OF SOLAR FLARES OF CONCERN TO MARS MISSION

Figure 3 illustrates the spiral configuration of the interplanetary magnetic field caused by the constant rotation of the Sun about its axis. Particles from solar flares do not travel freely in interplanetary space, but are generally guided by the existing magnetic field. Though substantial numbers of particles scatter across the field, the peak fluxes of particles are generally observed along field lines that extend radially outward from near the solar flares that produced the particles. As a result, there are dangerous longitudes where flares have a higher probability of producing large SPE at the Earth. Figure 3 shows those longitudes for near-Earth space. Travelers to Mars will be endangered by a different set of longitudes as they travel around to the opposite side of the Solar System. From Earth, it will be impossible to observe the solar flares that can produce SPE near Mars.

SOLAR TELESCOPES ON THE MARS MISSIONS SPACECRAFT

If short term SPE alerts are to be used on a Mars mission, solar telescopes will have to be carried and operated as part of the payload. When Mars is on the opposite side of the Sun from the Earth, it will be necessary for the mission to do its own solar observations because the solar flares that are of danger to the mission will occur on the side of the Sun opposite to the Earth. In addition, the time required to transmit those observations to the Earth and for an alert to be transmitted back to Mars exceeds 30 minutes when Mars is in opposition. It will be necessary for some on-board solar observing to be carried out by members of the crew and for solar image analysis, part of which can be automated, to be done in real time on the spacecraft. For reliability, and because Earth-based analysts can assume some of the watch tasks when proton events are not imminent, it is also desirable to transmit the solar images to Earth. The solar telescopes of choice, based on the

TABLE 1

SUMMARY OF CAPABILITY FOR SPE PREDICTIONS
FOR VARIOUS FORECAST PERIODS

<u>FORECAST PERIOD</u>	<u>CAPABILITY</u>
1-2 Years	- Probabilistic Forecast (Goddard SOL-PRO Program) - No current capability for reliable yes/no forecast
1-10 Days	- Probabilistic forecasts - SPE watch (region on Sun may produce a proton flare
23-30 Minutes	- Reliable yes/no forecasts 95% accuracy. Prediction of event size (accurate to one order of magnitude over a possible range of 5 orders of magnitude). Requires on-board x-ray solar imager.

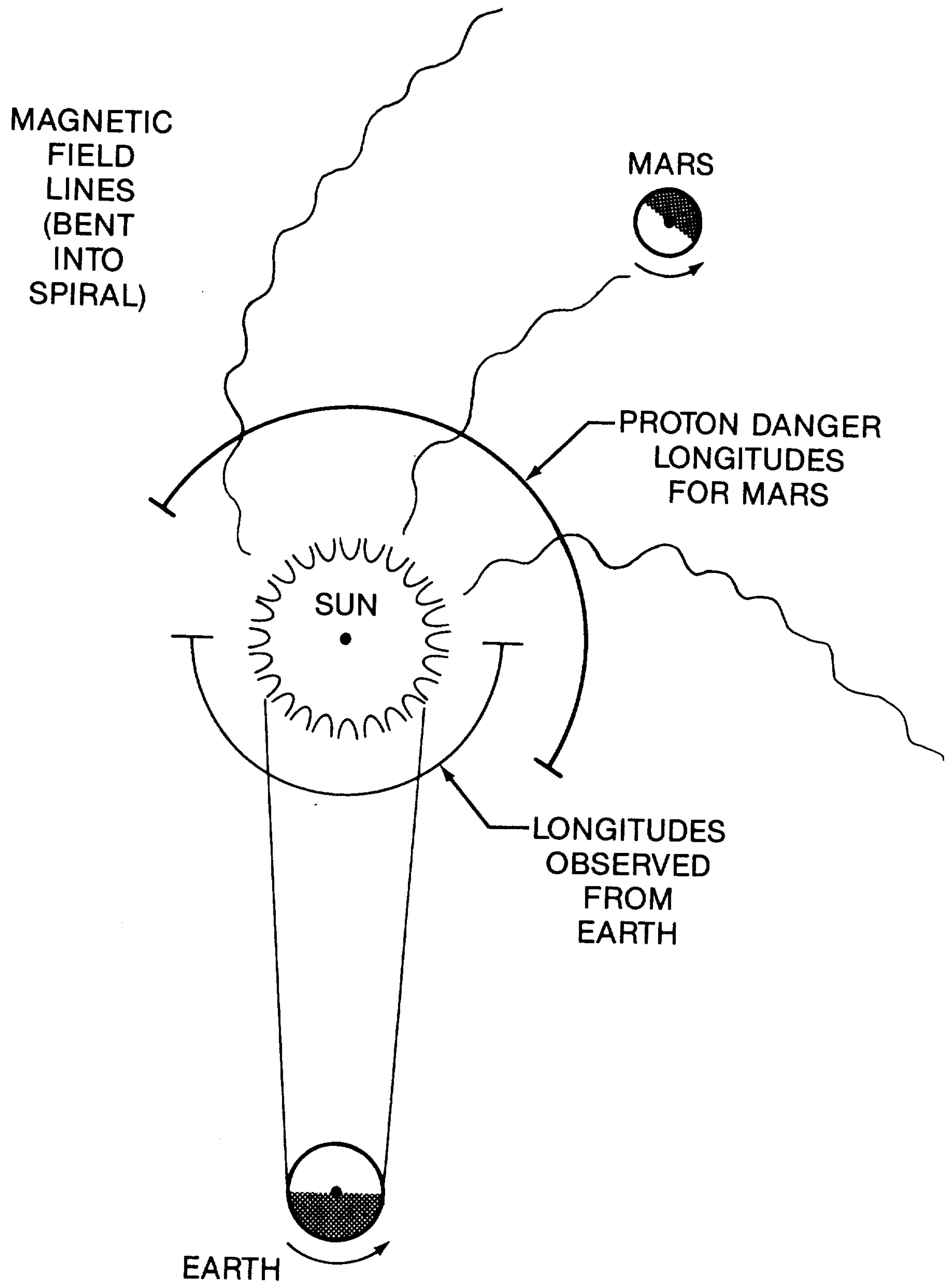

FIGURE 3. VIEW FROM ABOVE NORTH POLE

experience of the SESC, would be an x-ray imaging telescope, a hydrogen-alpha chromospheric scanner and a solar magnetograph, with a solar radio telescope as a useful addition. These instruments would also be of use in solar astronomy-science work that can be done by a Mars mission.

ISOTROPICS

The issue has been raised whether protection from solar particles can be accomplished by use of a large shield placed in the direction of the Sun during SPE. The technique will provide some radiation reduction during the very first phases of particle events when most of the particles are arriving from the direction of the Sun, but the particle flux rapidly becomes isotropic--the particles are scattered by the interplanetary field until they appear to come from all directions. The reduction in total dose from using such a shield, depending on the nature of the proton event, would usually be small.

HIGH ENERGY, HEAVY MASS SOLAR PARTICLES

High energy, heavy ions (HZE) produced by solar flares have been observed regularly for only a few years (3) and thus far, the energies of the particles are such that they can be shielded by normal shielding thickness. However, no direct observations of these particles were possible in the very large solar flares in solar cycles 19 and 20 and the knowledge of the energy spectra involved is incomplete at this time.

RECOMMENDATIONS

1. Scheduling Mars missions at solar minimum reduces, but does not eliminate, the chance of encountering SPE.
2. Reliance on forecasts should be based on nothing longer than 20-30 minute predictions plus a few tens of minutes during low exposure rates early in an event.
3. Mars missions should include on-board solar telescopes (preferably an x-ray imager, a hydrogen-alpha scanner, a magnetograph and if feasible, a radio telescope), the images used for radiation alerting (as well as science) and the images processed in an on-board system with parallel Earth-based analysis.
4. Solar high mass, high energy (HZE) events should be observed and additional data reduced to determine the energy and spectra of ions from very energetic solar flares.

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SOVIET EXPERIENCES WITH A MARS PROGRAM OR MISSION

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ABSTRACT

This paper provides a very brief summary of some of the Soviet Union's experiences related to manned missions to Mars. Most of the discussion is involved with the Russians' long duration space flight experiences and potential implications on the human body.

DISCUSSION

The Soviet Union has been trying to establish a Mars program for the past twenty five years. In 1960, under the leadership of a General Nedyelin, while Premier Nikita Krushchev was busy pounding his shoe on the rostrum at the United Nations, a large booster with an unmanned Mars probe went through the traditional countdown. When the command "Lift-off" was given, nothing happened. Men (somewhere between 20 and 200) rushed to the rocket to find out what was wrong. At that moment, the rocket exploded, killing an unconfirmed number of men, and definitely General Nedyelin. This first attempt of the Soviets to go to Mars has since become known as the "Nedyelin Disaster".

Since then, three or four attempts have been made, none of them successful, and none with casualties.

At the present time, The Soviet Space Program, along with the "Intercosmos" council, has marshalled available resources of the Warsaw Pact Countries, has invited the European Space Agency, France, Sweden and Finland to participate in the launch of an unmanned space probe to place a platform on Phobos about three years from now.

As a matter of fact, Professor V. L. Barsukov, Director of the Vernadsky Institute of the USSR Academy of Sciences, when he spoke in Houston on the occasion of the Sixteenth Lunar and Planetary Conference in March of 1985, indicated that there would in all probability be two probes. Should the first one fail its assigned task of landing on Phobos, the second would be directed toward Deimos, the second moon of the planet Mars.

The Soviet Union has until now stayed with the thought of keeping manned missions in near-Earth orbit. In doing this they have achieved an impressive record in duration of stay in orbit. The last completed mission aboard the "Salyut 7", in spite of several anxious moments and a record number of EVA hours, continued for 237 days. The last crew consisted of two cosmonauts and one physician-cosmonaut who made a number of changes in the daily regime of cosmonauts in order to facilitate their capacity for readaptation. After three weeks in a readaptation center at the Baykonur Cosmodrome, the three crewmembers were flown to Moscow, where they gave a large press conference. This gave the impression that the cosmonauts had recovered, at least sufficiently to physically withstand the rigors of such a press conference.

The disinformation worked--for a while. However, since late October, 1984, when the press conference took place, there has not been a word mentioned about the cosmonauts in any of the Soviet media. The three men seemed to have ceased to exist!

Having attained such record as 237 days in space, the Soviet medical authorities are well placed to study the difficulty in the process of readaptation. Blood volume (cardiovascular changes), erythrocyte formation, vestibular disturbances (i.e. motion sickness), muscle tone and volume, and psychological interaction among crewmembers are all known and well recognized possible dysfunctions caused by prolonged stay in weightlessness. All have proven to be reversible during readaptation. The problem not mentioned is calcium loss: Following every long-duration flight, a certain amount of bone loss was registered overall, and especially in the calcaneous region. The amount of loss appeared first to follow in a linear pattern the ever increasing duration of flights, but then it seemed to peak, recede and then stablize, but does not seem to reverse upon readaptation. Since a flight to Mars would last somewhere in the neighborhood of 237 days, the Soviets would be faced, as anyone else, with the necessity of finding prophylactic measures for these dysfunctions, especially the last.

(Information gathered from USSR News Agency TASS releases.)

N87-17780

RADIATION ENVIRONMENT AND SHIELDING FOR EARLY MANNED MARS MISSIONS

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ABSTRACT

The problem of shielding a crew during early manned Mars missions is discussed. Requirements for shielding are presented in the context of current astronaut exposure limits, natural ionizing radiation sources, and shielding inherent in a particular Mars vehicle configuration. An estimated range for shielding weight is presented based on the worst solar flare dose, mission duration, and inherent vehicle shielding.

RADIATION EXPOSURE LIMITS

Dose Limits

The most radiation critical organs are the bone marrow (blood forming tissue), the skin, the lenses of the eyes, and the reproductive organs. Irradiation of these areas can cause delayed effects such as leukemia, skin cancer, cataracts, and sterility/genetic defects respectively. It can also cause shortening of lifespan and an increase in general malignant tumors. High doses over a short time can also cause more immediate medical problems.[1]

Table 1 shows the current radiation exposure limits established for flight crewmen.[2] These limits were established by the Radiation Safety Panel for Manned Spaceflight and represent the total allowable radiation limits for the crew from all sources, including routine medical X-rays. The rationale for adopting these limits, instead of limits used in the nuclear industry, are as follows:

"1. Radiation is only one of many recognized and accepted potential risks that may jeopardize the success of any flight mission.

2. Individual astronauts are carefully selected for their special skills and motivation. The application of existing standards of radiation safety established for large, occupationally exposed groups would unduly limit the ability of this small group of specialists to achieve their objectives.

3. The parameters of some space-radiation risk cannot be precisely predicted; therefore, optimal protective measures will not always be available or even feasible. Since any radiation shielding will add to

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TABLE 1
RADIATION EXPOSURE LIMITS RECOMMENDED FOR SPACEFLIGHT CREWMEMBERS

CONSTRAINT	BONE MARROW (REM AT 5 CM)	SKIN (REM AT 0.1 MM)	OCULAR LENS (REM AT 3MM)
1–YEAR AVERAGE DAILY DOSE	0.2	0.6	0.3
30–DAY MAXIMUM	25	75	37
QUARTERLY MAXIMUM	34	105	52
YEARLY MAXIMUM	75	225	112
CAREER LIMIT	400	1200	600

the weight of a spacecraft, the reduction in risk to be achieved by the shielding must be balanced against the other uses to which this weight might have been put.

4. Since flight missions may vary in both duration and radiation exposure, the probability and importance of the radiation risk compared with those of other risks must be taken into account for each specific mission. A risk-versus-gain philosophy is most appropriate for this comparison, and the philosophy is particularly useful for evaluation of radiation risk. The latter is generally a cumulative one that should not require an urgent all-or-none type of decision."[3]

Since these limits were defined, many of the underlying tenets have changed. Consequently, the limits are being revised to be more stringent. Since they will not be officially redefined for several months, the limits previously cited are used in this paper.

Radiation Sources

There are three major sources of natural ionizing radiation which the Manned Mars Mission crew will encounter. They are the trapped particles in the Van Allen radiation belts, galactic cosmic rays, and solar flares. In the following sections, each source is discussed with respect to the hazard it poses to a crew.

Van Allen Belts

The Van Allen belts consist of electrons and protons trapped in the geomagnetic field. They are generally described as two somewhat overlapping radiation belts, an inner one comprised predominantly of protons and an outer one comprised primarily of electrons. A simplified diagram of the belts is shown in Figure 1. Doses from the protons are due mainly to primary particles however, the dose from electrons can be far more severe from secondary radiation than from the primary particles. As low energy electrons are absorbed by high-Z materials, they generate x-rays with penetrating power far greater than that of the electrons producing them.[4] Experience from the Apollo flights indicates that the dose from ascending directly through the belts to the moon and returning incurs average mean doses less than .16 to 1.14 rads.[2] Doses, from the terrestrial radiation belts, would probably be comparable for a manned Mars mission.

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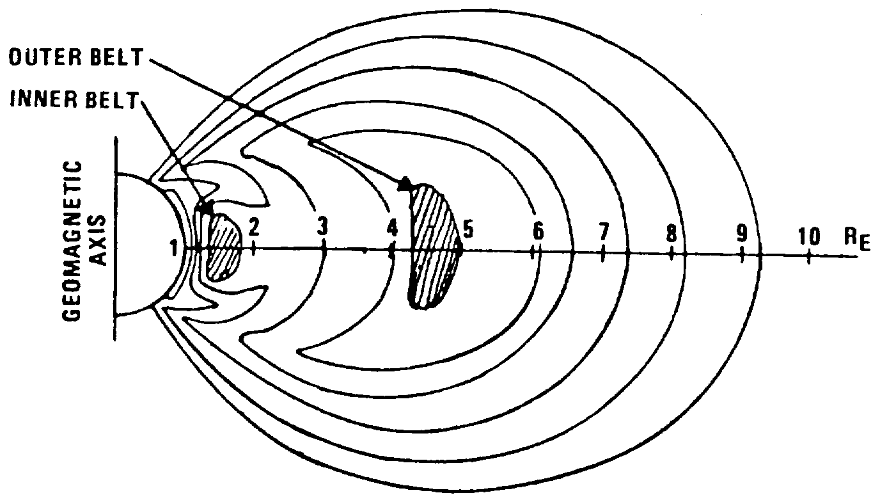
FIGURE 1



VAN ALLEN RADIATION BELTS

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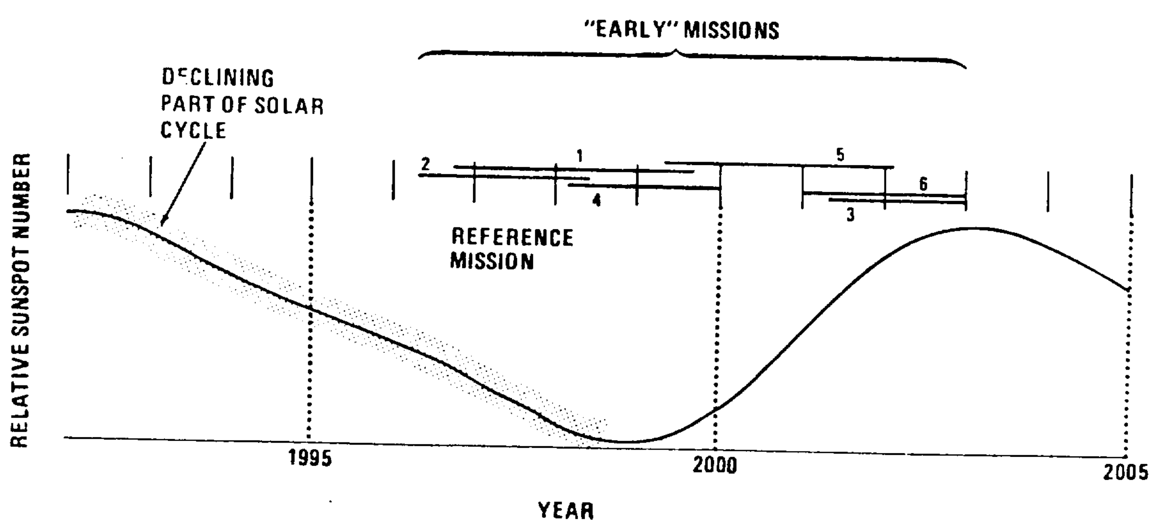
FIGURE 2.
IDEALIZED SUNSPOT NUMBER VS. TIME FRAME FOR SELECTED MISSIONS



Galactic Cosmic Rays

The second category of radiation encountered by an interplanetary spaceflight crew is galactic radiation. It consists of low intensity, extremely high-energy particles. These particles, 85% protons, 13% alpha particles, and 2% heavier nuclei, bombard the solar system from all directions. The flux levels beyond the influence of earth's magnetic field are relatively constant except during enhanced solar activity when galactic cosmic ray flux decreases. The decrease is caused by an increase in the strength of the interplanetary magnetic field which shields incoming particles.[6] The integrated dose for galactic radiation (without shielding) is 4 to 10 rads/year.[7] No trapped radiation belts, similar to those encircling Earth, have been found around Mars. Therefore, the radiation environment in Martian orbits resembles that in interplanetary space[8] although it is reduced somewhat due to blockage by Mars itself.

Solar Flares

A third significant source of ionizing radiation that may be encountered by the Mars crews is solar cosmic rays or solar flares. A flare is an area on the solar disk where surface temperatures are nearly a thousand times that of surrounding areas. Flares tend to occur more frequently during the declining portion of the eleven year sunspot cycle (Figure 2 shows relationship between sunspot cycle and several mission "windows" in a recent study). Flares always occur in so-called active regions or centers. An active region begins with structural abnormalities in the surface granulation called plages. These plages are followed after a few days by sunspots and those, in turn by flares. Strong active regions sometimes live through several 25-day solar rotations. An active region frequently produces several flares during one passage across the visible side of the Sun, and this is the most important clue for flare prediction. In at least one case during a solar maximum, the same active region produced a second major flare after its second appearance on the visible side.

Large flares are rare events, occurring only a few times during the 4 to 6 year period of high sunspot activity in the 11 year solar cycle. Flares require only a matter of minutes to develop. The optical phenomenon on the Sun usually lasts only 30 to 50 minutes. The emission of electromagnetic radiations from flares is limited to the time of visible activity, but solar protons continue to arrive in the vicinity of the Earth up to 36 or even 48 hours after a flare.

Classifying flare events according to radiation hazards is difficult because no distinct types of flares can be defined. Not only does the total dose for individual flares vary over an extremely wide range from fractions of a rad to doses approaching 1,000 rads, but so do the instantaneous dose rate and spectral configuration at different times during the same event. The time profile of flux buildup and decay and the slope of the energy spectrum for each solar particle changes as the flare event progresses. This greatly complicates calculation of depth-dose distribution and compensation for shielding effects.

Flares from the solar maximum in Cycle 19 have been studied in detail to determine depth-dose distributions behind simplified vehicle-shield systems. The largest events from one study are depicted in Table 2. Due to measuring limitations, the data in this table should be considered representative of the general exposure levels, rather than as exact individual doses. Furthermore, data in the table assumes uniform shield thickness. Actual space systems always show a complex distribution of shield thicknesses covering a wide range. For example, on the Apollo vehicle the range extends from about 1.75 g/cm^2 to 212 g/cm^2. In such systems, the dose distribution throughout the body becomes extremely complex.

Table 2 reveals that 92% of the total dose during solar cycle 19 was delivered in eight critical periods, each of which was 10 days or less, randomly spaced over six years. Additionally, 64% of the total was confined to periods around February 23, 1956; July 10-16, 1959; and November 12-15, 1960.

One method for assessing the flare hazard for humans is to determine the maximum and minimum doses encountered for various launch dates on a mission of a given duration. Table 3 shows just such data for Cycle 19. Notice that the worst dose expected for a week is the same as for two weeks, and a month, and almost the same dose as for several months.[9] This further illustrates the sporadic nature of flares.

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TABLE 2
TOTAL ESTIMATED SOLAR FLARE DOSES BY EVENT FOR 10 SHIELDING CONFIGURATIONS

DATE	SHIELDING CONFIGURATION									
	1/0[a]	2/0	5/0	10/0	20/0	1/5	2/5	5/5	10/5	20/5
2/23/56	280.00	181.00	91.80	50.20	24.80	64.78	58.00	43.75	30.40	17.90
8/3/56	8.50	5.00	2.20	1.00	0.40	1.39	1.21	0.85	0.53	0.27
1/20/57	122.00	43.50	8.30	1.80	0.30	3.42	2.57	1.23	0.46	0.11
8/29/57	77.00	25.10	4.20	0.80	0.10	1.63	1.20	0.54	0.19	0.04
10/20/57	18.50	10.30	4.10	1.80	0.70	2.53	2.17	1.46	0.88	0.41
3/23/58	148.00	53.60	10.90	2.50	0.40	4.67	3.55	1.75	0.69	0.17
7/7/58	150.00	53.70	10.50	2.30	0.40	4.38	3.30	1.60	0.61	0.15
8/16/58	23.70	8.60	1.80	0.40	0.10	0.75	0.57	0.28	0.11	0.03
8/22/58	45.00	14.90	2.50	0.50	0.10	0.96	0.71	0.32	0.11	0.02
8/26/58	75.00	23.10	3.40	0.50	0.10	1.19	0.85	0.36	0.11	0.02
5/10/59	470.00	211.10	59.30	18.30	4.40	30.18	24.28	13.60	6.70	2.10
7/10/59	420.00	214.00	73.20	27.40	8.40	41.56	34.65	21.76	11.84	4.80
7/14/59	650.00	284.50	75.90	22.30	5.00	37.56	30.00	16.75	7.80	2.50
7/16/59	382.00	194.80	67.20	25.30	7.80	38.30	31.98	20.16	11.03	4.50
9/3/60	13.00	7.20	2.90	1.20	0.50	1.77	1.52	0.10	0.06	0.03
11/12/60	484.00	269.60	105.50	44.90	16.20	64.53	55.12	36.87	21.83	10.05
11/15/60	288.00	151.90	55.90	22.40	7.50	30.04	7.91	18.14	10.33	4.49
11/20/60	17.30	9.50	3.60	1.50	0.05	2.14	1.82	1.20	0.69	0.31
7/12/61	25.70	8.40	1.40	0.30	0.03	0.54	0.40	0.18	0.06	0.01
7/18/61	128.00	64.20	21.60	8.00	2.40	12.16	10.11	6.30	3.39	1.35

a. SHIELDING CONFIGURATIONS ARE GIVEN AS X/Y WHERE X IS THE SHIELDING THICKNESS IN g/cm^2 OF ALUMINUM AND Y IS THE SHIELDING THICKNESS IN g/cm^2 OF TISSUE.

TABLE 3
MAXIMUM AND MINIMUM MISSION DOSES*
FOR BEST AND WORST LAUNCH DATES DURING ACTIVE PERIOD OF CYCLE 19

MISSION DURATION	MAXIMUM DOSE (RADS)	MINIMUM DOSE (RADS)
4 YEARS	3492	2439
3 YEARS	3229	974
2 YEARS	2781	526
1.5 YEARS	2415	176
1 YEAR	2110	15
9 MONTHS	1963	2
6 MONTHS	1963	0
3 MONTHS	1962	0
1.5 MONTHS	1492	0
1 MONTH	1452	0
2 WEEKS	1452	0
1 WEEK	1452	0

* SURFACE DOSE INSIDE 1 g/cm^2 UNIFORM ALUMINUM SHIELDING.
(LANGHAM, 1967)

SHIELDING REQUIREMENTS

Since any additional weight added to a spacecraft decreases its payload capacity, it is prudent to add as little shielding mass as possible without compromising crew safety. This entails using the vehicle mass as much as possible to provide shielding capability and adding supplementary shielding until it offers sufficient protection. The following sections outline a general approach to bracket the shielding mass requirements for a manned Mars mission.

Baseline Dose Limit

Any calculation of shielding requirements must begin with criteria for the maximum dose limit acceptable for personnel. Revised dose criteria are expected to reduce the permissible dose limits when approved. However, until the astronaut dose limits are revised, we will use the current official limits (see Table 1).

Several mission durations were considered based on various launch opportunities. The candidates for "early" missions are shown in Table 4.[10] To derive a reasonable maximum shielding mass estimate, it was decided to use the mission, from a recent study, which had the longest travel time. Several missions had longer total mission times. However, on those missions, it was felt that: (1) the entire crew would probably be on the surface; and (2) providing protection from solar flares and galactic cosmic rays would be easier on the surface than on orbit because of material available for shelter and more flexible operational strategies. Therefore, a long duration mission with a ~60 day stay time, where part of the crew remains in Mars orbit, imposes the most severe shielding mass penalty. For simplicity, it was assumed that there would be no appreciable additional dose from either a nuclear propulsion system or a nuclear power system. Similarly, doses from routine medical x-rays and doses from medical experiments were not included, although they would be relatively easy to incorporate.

The mission meeting all of the above criteria is the "1997 Double Swingby Mission". The duration of that mission is 738 days (2.02 years).

Residual Acceptable Dose

The portion of the dose absorbed during passage through the radiation belts and from galactic cosmic rays is comparatively easy to estimate. The radiation dose from the Van Allen belts is estimated at

TABLE 4. EARLY MANNED MARS MISSION FLIGHT OPPORTUNITIES (REF 10)

NO.	MISSION NAME	MARS STAY TIME (DA)	TOTAL MISSION TIME (DA/YRS)	EARTH DEPARTURE WINDOW
1.	1997 CONJUNCTION MISSION	385	1025/2.81	OCT 28—NOV 27, 1996
2.	1997 DOUBLE SWINGBY MISSION	60	738/2.02	MAR 24—APR 3, 1996
3.	2001 INBOUND SWINGBY MISSION	60	628/1.72	MAR 12—APR 11, 2001
4.	OUTBOUND VENUS SWINGBY FOR 1999 OPPOSITION OPPORTUNITY	60	668/1.83	JAN 11—FEB 10, 1998
5.	CONJUNCTION CLASS MISSION FOR 1999 OPPOSITION OPPORTUNITY	485	1005/2.75	MAR 12—APR 11, 1999
6.	INBOUND VENUS SWINGBY	60	708/1.94	DEC 22, 2000—JAN 21, 2001

less that 1.14 rem. The dose rate from galactic cosmic rays is estimated to be .165 to .265 rems/day, by considering the biological effectiveness of the galactic radiation.[11,12] The bone marrow dose is the limiting dose in these types of calculations. Since the bone marrow dose is generally less than the skin dose for this mission, the dose contribution over the 2.02 years from these sources is less than 122 to 196 rems. Referring to the acceptable dose limits in Table 1, we see that for a career bone marrow dose (400 rem) the galactic cosmic rays and radiation belt exposures would leave 204 to 278 rem available for solar flare doses. A shielding mass could be estimated from this data, but it would be inadequate because the human responses to radiation are dose rate, as well as total dose dependent. The correct level of shielding can only be estimated if the worst dose for each of the time periods and tissue depths cited in the exposure limits (Table 1) are checked.

Worst Likely Dose

Assuming that Cycle 19 is representative of an unusually active solar cycle, and that its most hazardous flares are typical of the most hazardous flares that would be encountered by a Mars mission crew, we can establish a basis to estimate maximum shielding requirements. The maximum doseage in the most hazardous two year period of Cycle 19 is 2781 rems.[11 & 12] However, according to Table 3, a dose of 1452 rems, over 50% of the total, is encountered during a single one week period! This dose is significantly more hazardous than the total two year dose because of the high dose rate.

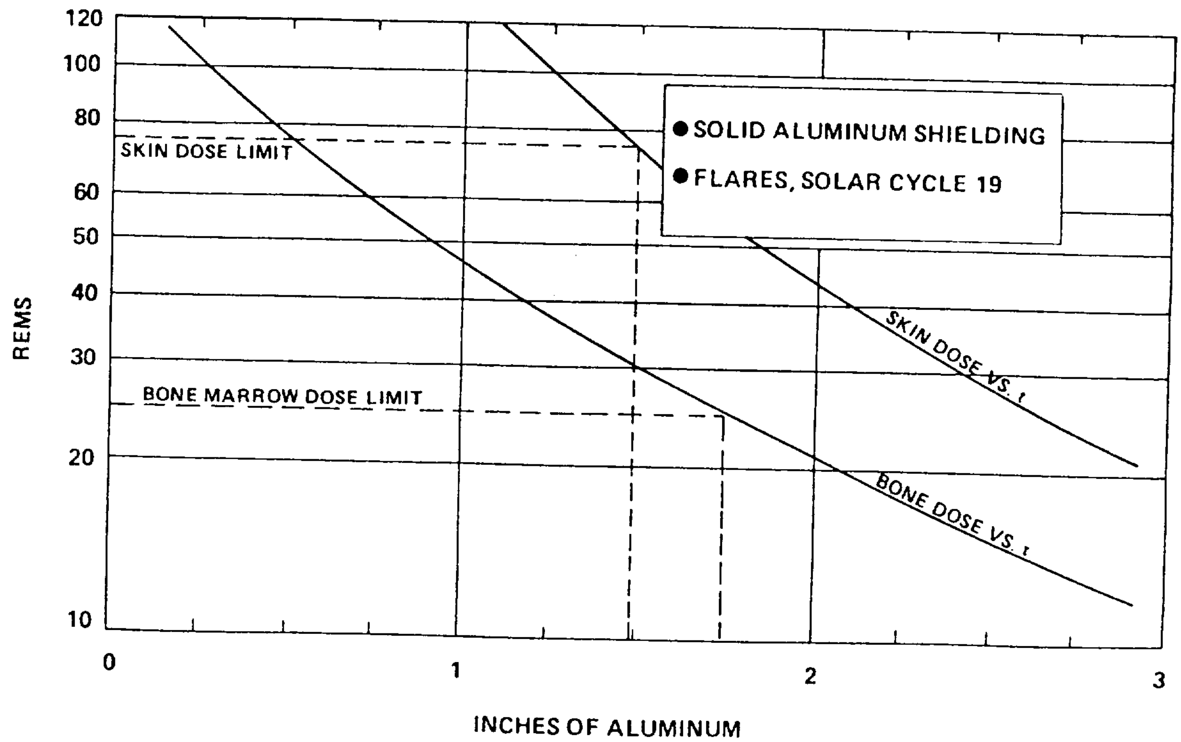
The dose limits in Table 1 show the maximum acceptable dose for a 30-day period (the closest corresponding period) is 25 rem for the bone marrow dose. Figure 3, based on Table 2, shows that for the period cited a uniform shield of aluminum approximately 4.44 cm (1.75 inches) thick would provide sufficient protection. The corresponding thickness based on skin dose limits is also shown to stress the importance of considering all of the dose limits when estimating shielding requirements.

Shielding Mass Required

Given the shielding thickness, the shielding mass can be readily determined if we ignore the shielding effects of vehicle mass and provide a dedicated mass for the shield. Referring to the Celentano criteria for minimum free volume, we allow 1.42 m^3 (50 ft^3/man) for a "storm shelter".

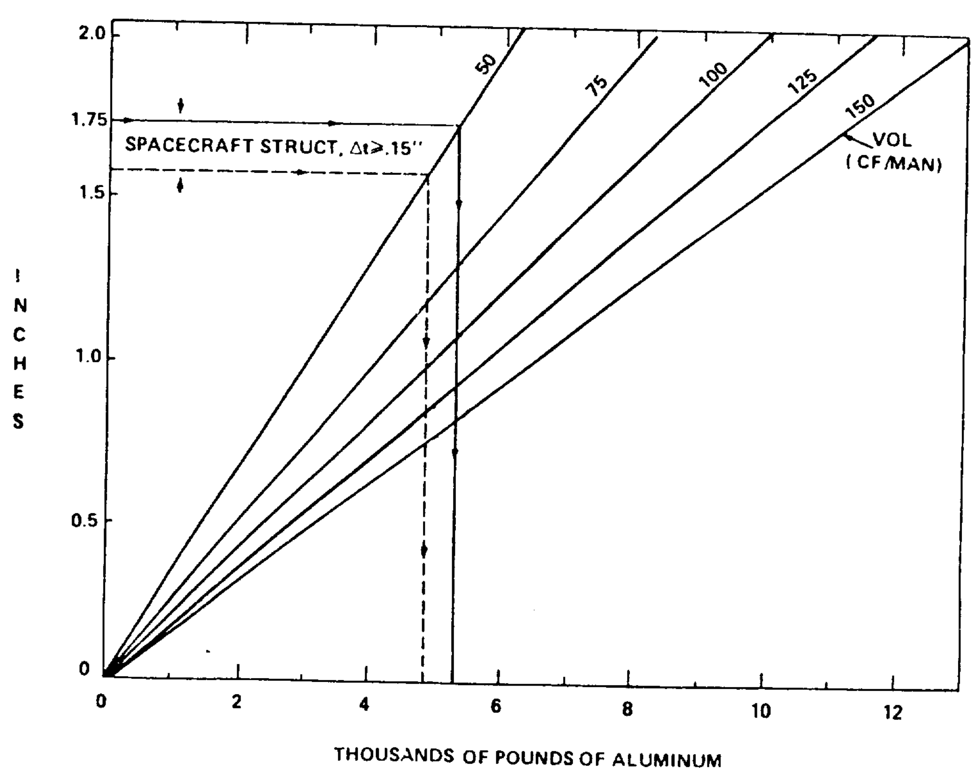
3168-85

FIGURE 3. MAXIMUM 30 DAY FLARE DOSE VS. SHIELD THICKNESS (t)



3489-85

FIGURE 4. RADIATION
SHIELDING THICKNESS VS. SHIELDING WEIGHT



This further assumes that the maximum time the shelter would be used is no more than a few days at a time. The crew of six would require 2.82 m^3 (300 ft^3). This corresponds to a spherical enclosure with an inside diameter of 2.52 m (8.3 feet). A sphere that size, fabricated from Al, would weigh about 2430 kilograms (5360 pounds), about 2 1/2 metric tons (see Figure 4).

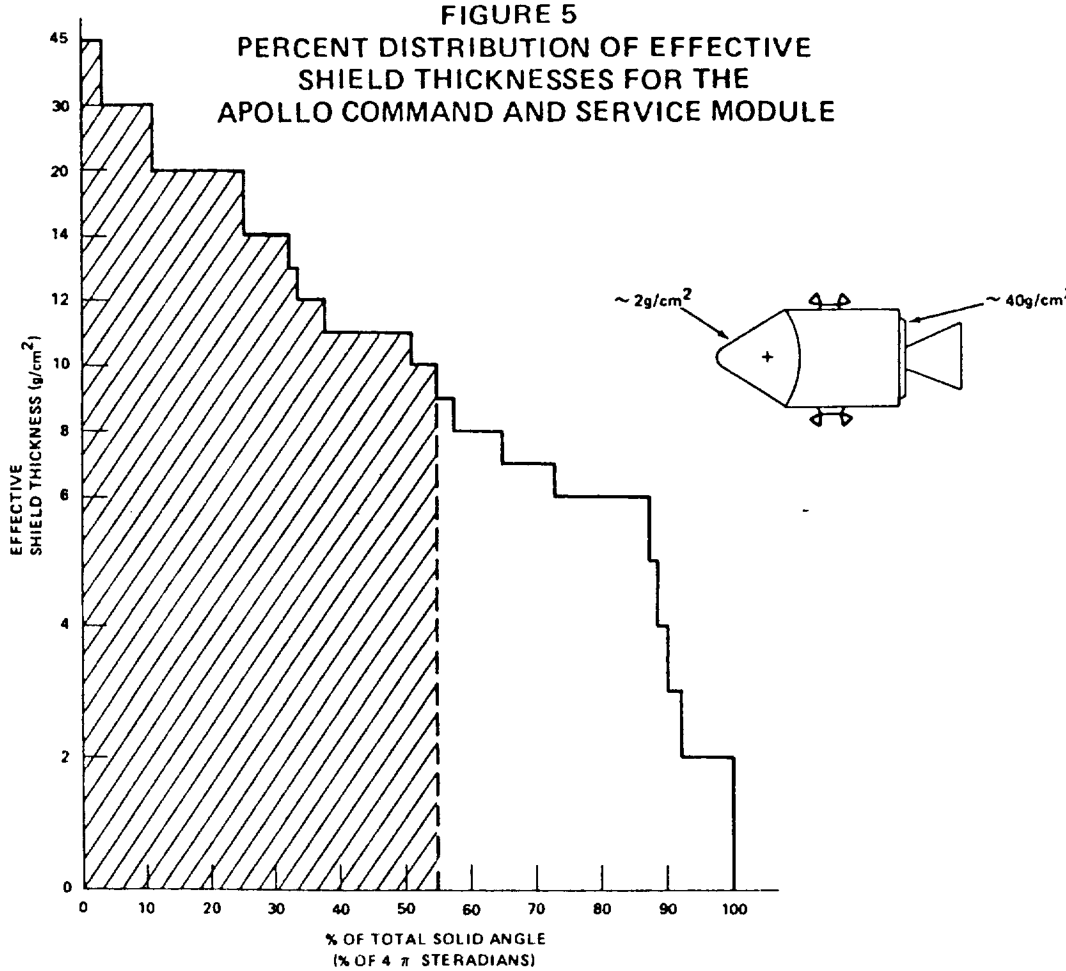
Shielding Available

Past experience indicates that the effective shielding for a typical point inside a spacecraft is considerably higher than would be expected from merely measuring the spacecraft wall thicknesses. For Skylab, the wall thickness was about 1.0 g/cm^2, whereas typical points in the Workshop had effective shielding of approximately 10 to 15 g/cm^2. (13) In the Spacelab module, effective shielding thicknesses ranged from about 1 to over 20 g/cm^2 equivalent aluminum. Figure 5 shows the distribution of equivalent thicknesses for a typical spacecraft. Obviously, the protection afforded by the structure and systems can be significant. It is equally obvious that the magnitude of such shielding cannot be accurately estimated without fairly detailed design concepts.

However, even in early design it is possible to begin estimating the minimum amount of protection that would be provided. This enables the shielding mass to be bracketed and also suggests optimum locations to locate a "storm shelter" for protection from solar flares. The results of such an analysis are depicted in Figures 6, 7, & 8. The first figure indicates the location in the laboratory module that was selected for analysis with an "x". The entire vehicle is depicted here. However, only the shaded portion accompanies the inhabited areas all the way to Mars and back. Consequently, shielding from the first stage, the braking stage, the MEM, and the Mars departure stage are not considered in the analysis. Figure 7 shows the cross section of the lab/logistics module with a detail of the equipment racks. In Figure 8 we see the equivalent thicknesses of material shielding the spot analyzed. The effective shielding thickness graph is drawn in spherical coordinates, with the origin at the point indicated in Figure 6. In Figure 8, we are looking aft along the vehicle centerline. The field of view extends 90 degrees left and right , and 90 degrees above and below the centerline. The shielding levels shown are categorized, the lowest level being .38 cm

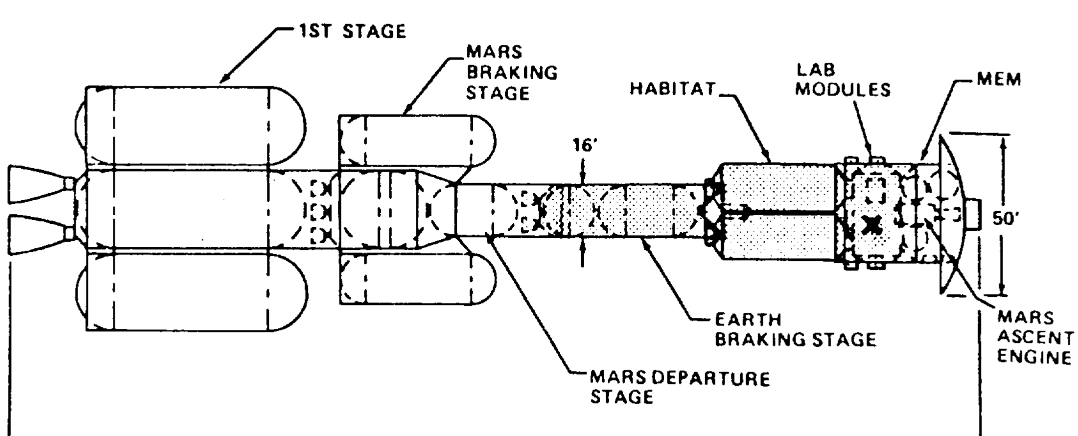
3602–86

FIGURE 5
PERCENT DISTRIBUTION OF EFFECTIVE SHIELD THICKNESSES FOR THE APOLLO COMMAND AND SERVICE MODULE



4186–86

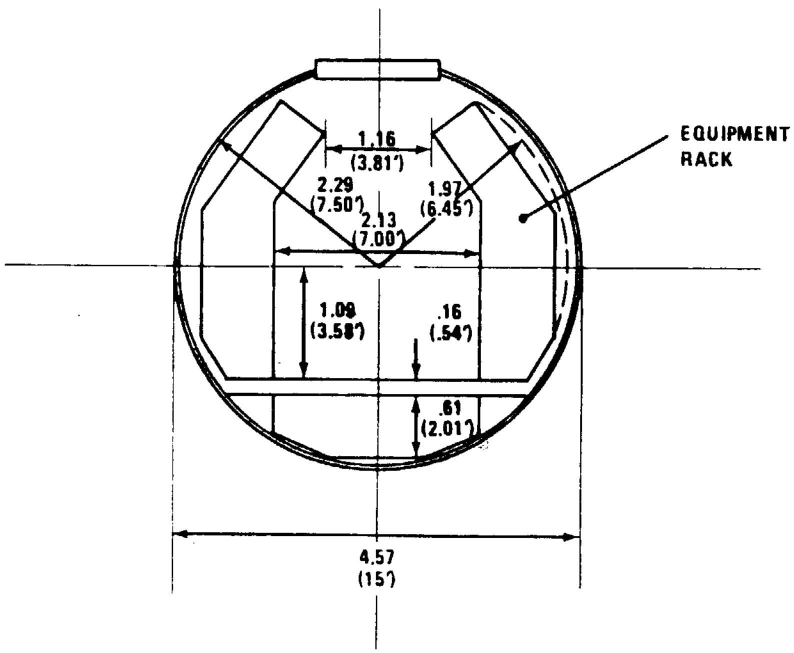
FIGURE 6
MARS MISSION
ALL PROPULSIVE OPTION



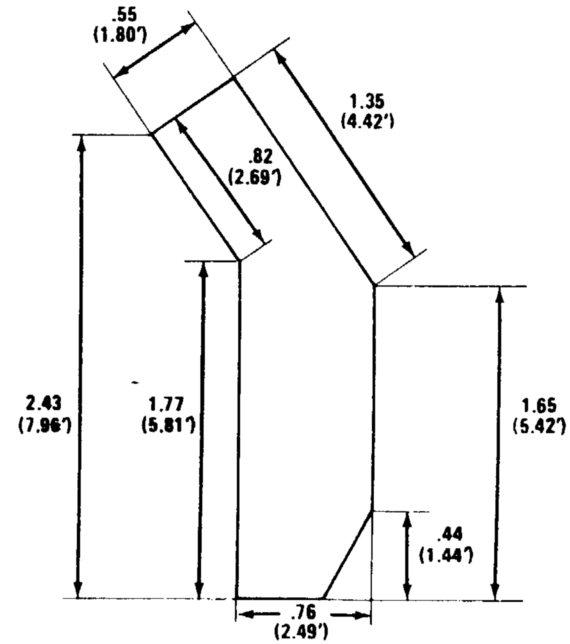
3242—85

FIGURE 7
LAB/LOGISTICS AREA

MODULE CROSS SECTION *



EQUIPMENT RACK DETAIL*

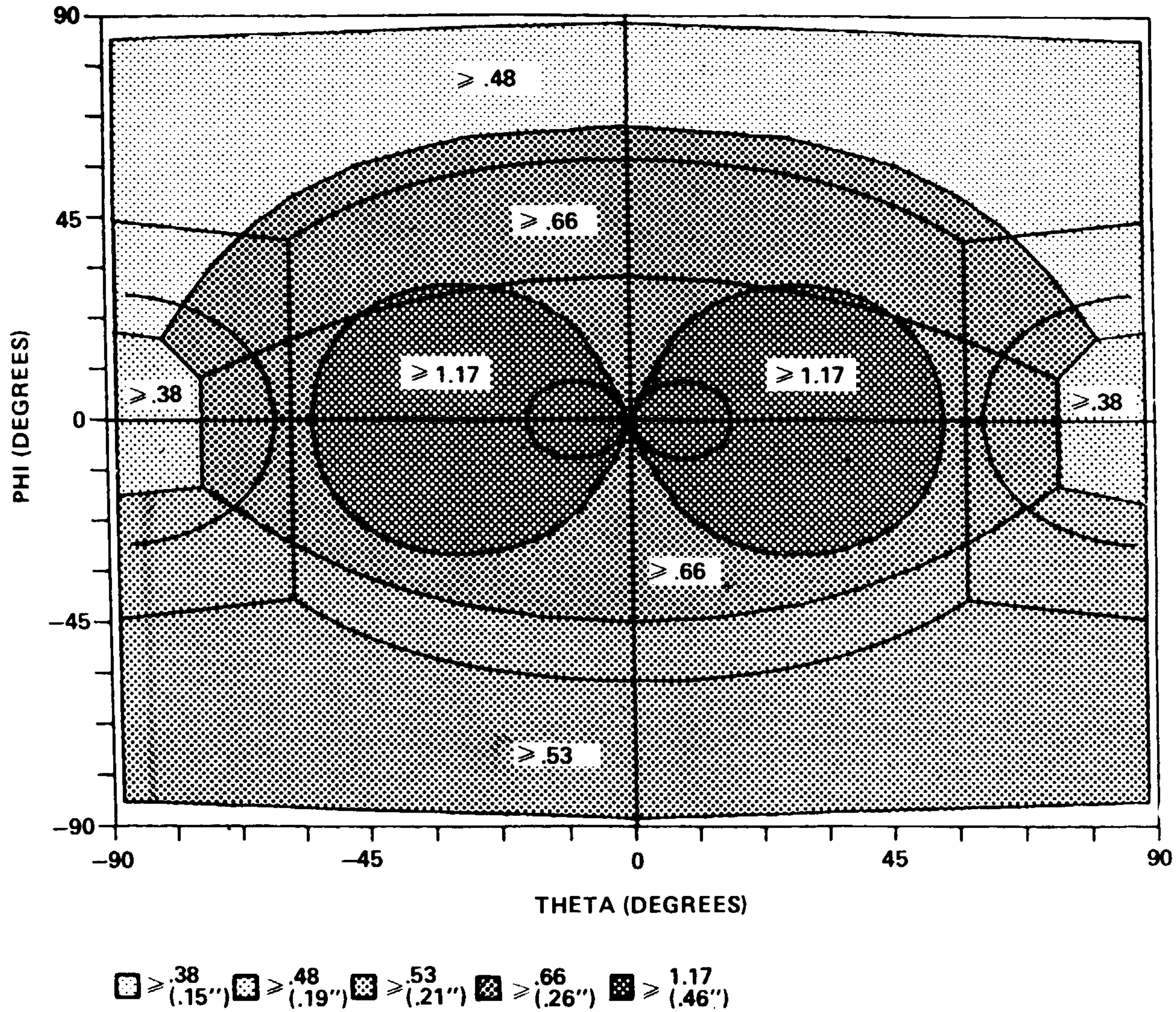


* ALL DIMENSIONS GIVEN IN METERS, WITH ENGLISH UNITS IN (   )

3219-85

FIGURE 8
EFFECTIVE SHIELDING THICKNESS*

LOOKING FROM LAB AREA TOWARD HAB AREA



* THICKNESS GIVEN IN CENTIMETERS, WITH ENGLISH UNITS IN (   ).

(.15 in) or more of aluminum, the next category being .48 cm (.19 in) or more, etc. The least protection is given at the ends of the module where only the .25 cm (.10 in) outer shell plus the .13 cm (.05 in) external support structure is available. The next level is identical except we add .10 cm (.04 in) of overhead locker structure. The .53 cm (.21 in) level is available where we have floor and subfloor structure .15 cm (.06 in) available, instead of the overhead lockers. The thickness is equivalent to .66 cm (.26 in) of aluminum where we have the equipment racks .28 cm (.11 in) in addition to the shell and external support structures. Finally, there is a region where we have the shell of adjacent modules providing shielding. This superimposed .51 cm (.20 in) on the .66 cm (.26 in) previously mentioned, for a total of 1.17 cm (.46 in).

In this simplified analysis, obliqueness of shielding was not taken into account. Also, the equipment racks were assumed to be empty, although they would in reality be nearly full of hardware. These factors would significantly increase the effective thickness of inherent shielding.

To aid conversion among various shielding terms, a nomagraph was prepared relating range (g/cm^2) to equivalent thicknesses in aluminum expressed in centimeters and inches (Figure 9).

Configuration Sensitivity

Vehicle configuration will significantly influence shielding mass and vice-versa. The selection of a site for locating a "storm shelter" for solar flare protection can appreciably reduce mass requirements for the shielding. The analysis done in the current activity suggests that with the configuration shown, the maximum inherent shielding might be available in one of the habitability modules at a site close to the centerline of the vehicle and as far aft as possible.

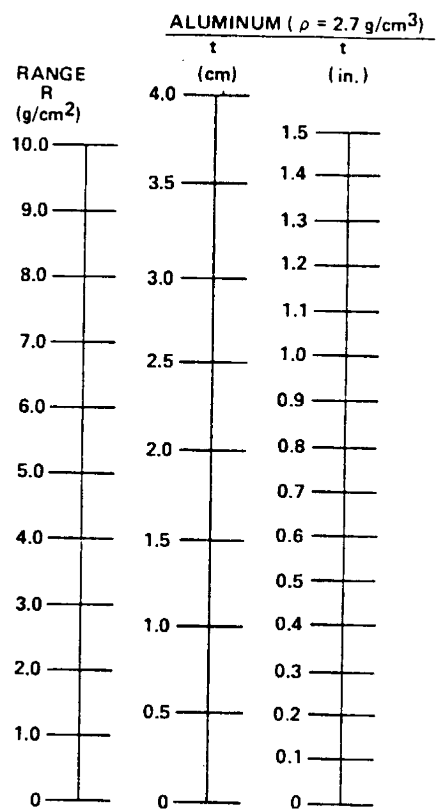
If artificial gravity is provided by spinning the vehicle, it would probably lead to a less compact configuration in which inherent shielding would be more difficult to exploit. In such an instance, it might be feasible to despin the vehicle in response to an impending flare, and reconfigure it to maximize the inherent shielding from vehicle.

Shielding Options

The previous remarks are predicated on using aluminum or an equivalent material for shielding. However, several other materials are at

2549–85

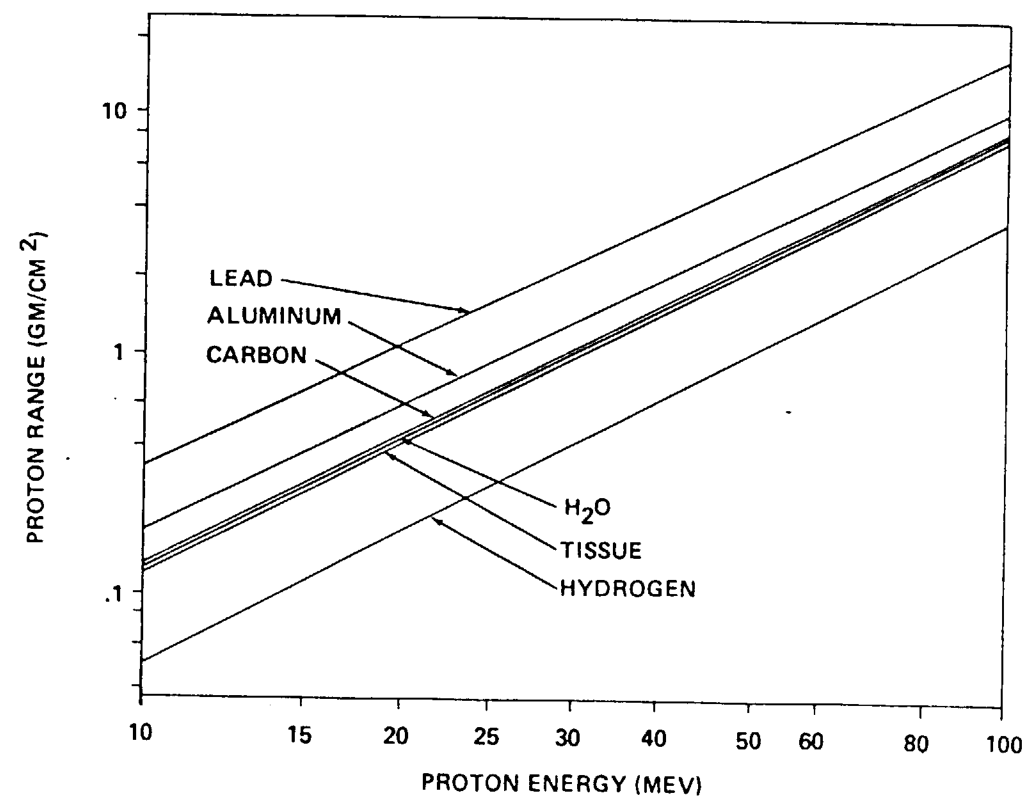
FIGURE 9
CONVERSION SCALES FOR SHIELDING THICKNESS



1.0 IN. = 2.54 cm

3169–85

FIGURE 10
RANGE–ENERGY CURVES FOR PROTONS



least as effective and some are, pound for pound, more effective. Figure 10 shows the relative effectiveness of several examples as well as a few materials that are less effective. Several polymeric hydrocarbons are also good shielding materials because they absorb particles without generating appreciable secondary radiation.

SUMMARY

An approach has been suggested to bracket the range of weight for radiation shielding. The sources of radiation have been described. Precise doseage estimates are difficult to make because of the sporadic nature of solar flares, and because the mechanisms of radiation damage are still under investigation. However, rough estimates of shielding mass can be made. The contribution of vehicle mass to shielding also can be estimated. In further studies of manned Mars missions, the effects of secondary radiation, neutron buildup, and high-Z particles should be fully accounted for as tools are developed to quantify them. Configuration options should be conceptualized that make optimum use of vehicle structure, system, and consumable masses for shielding. Finally, consolidated dose limit criteria and shielding performance data should be developed in consistent, easily interpreted terminology to support future trade studies and design efforts.

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N87-17781

TOXICOLOGICAL SAFEGUARDS IN THE MANNED MARS MISSIONS

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ABSTRACT

Safeguards against toxic chemical exposures during manned Mars missions (MMMs) will be important for the maintenance of crew health and the accomplishment of mission objectives. Potential sources of toxicity include offgassing, thermodegradation or combustion of materials, metabolic products of crew members and escape of chemicals from containment. Toxic exposures during MMM's are of especially great concern because of their long duration. This would give toxins a long time to accumulate and a long time to exert their adverse effects. Other concerns regarding toxic exposures during MMMs are the long time that would be required for crew members to return to Earth for medical aid and the great cost of crew impairment.

Safeguards against introduction of excessive environmental toxins and against excessive crew exposure include proper materials selection in regards to offgassing, heat stability and flammability; proper containment of bulk chemicals, alarms warning of chemical release into the atmosphere, the wearing of protective clothing, goggles and masks, the use of fume hoods while handling toxic chemicals, and the availability of safe havens for crew members to take refuge in event of high levels of toxic chemical contamination of their living environment.

Decontamination of breathing atmospheres will probably be performed by the use of regenerable absorbents, catalytic oxidizers, and condensors. Spilled chemicals may, in some instances, be recovered by the use of vacuum pumps.

New spacecraft maximum allowable concentration (SMAC) limits will have to be established for potential contaminants during the MMMs. The following factors will be used in establishing these new limits: Duration of missions, simultaneous exposure to other contaminants, deconditioning of crew members after long periods of reduced gravity, and simultaneous exposure to ionizing radiation.

Atmospheric contaminant levels in all compartments of the transit spacecraft and Manned Mars Station (MMS) will be monitored at frequent

intervals with a real time analyzer. This analyzer will be highly automated, requiring minimal crew time and expertise. The atmospheric analyzer will find other usages during the MMMs such as analyzing Martian atmospheres and soils, exhaled breath and body fluids of crew members, and reaction products in chemical processing facilities.

INTRODUCTION

Safeguards against toxicological exposures will be an important factor in the maintenance of crew safety, health and well-being during the manned Mars missions (MMMs). As in current spaceflight missions, the greatest emphasis will be on controlling toxic contaminants in the breathing atmospheres. Skin, eye and oral toxicities will also be important concerns. Possible sources of toxic chemicals in the transit spacecraft and the Manned Mars Station (MMS) environments include: (1) Offgassing of non-metallic materials--a few examples of chemicals frequently seen during offgassing tests of various spaceflight hardware materials include acetaldehyde, acetone, methylethyl ketone, isopropyl alcohol, xylene, trichloroethane, toluene, methane and Freon 113. These same chemicals are also often seen in air bottle samples taken from the crew quarters during Space Shuttle flights; (2) Metabolic products release by crew members including carbon dioxide, carbon monoxide, pyruvic acid, methane and skatole; (3) Escape from containment of stored chemicals--as from heat exchangers, fire extinguishers, tissue fixatives or reactants from chemical processing facilities; and (4) Thermomoderation or combustion of electric wire insulations or other materials.

Toxic contamination of the transit spacecraft or the MMS environment would probably have greater consequences than it would during missions of shorter duration for the following reasons:

Contaminant Accumulation

The transit spacecraft will require about nine months to travel from Earth to Mars and an equal time to return. The MMS will probably exist for many years. Due to the sparseness of the Martian atmosphere, and its lack of oxygen, the MMS will have a closed environment, similar to that of the transit spacecraft vehicle. Therefore, atmospheric contaminants that were not adequately removed by an inside decontamination system in the MMS could accumulate to dangerous levels over long periods of time.

Extended Crew Exposure

The individual MMMs will probably be of 20 months or longer duration, including the time in transit to and from Mars. Continuous exposure to some chemical contaminants over a long period of time would probably cause harmful effects to crew members at levels that would be harmless after shorter periods of exposure.

Long Time to Return to Earth for Medical Aid

If a crew member were made ill from a toxic chemical exposure and could not be adequately treated at the MMS he would have to wait a long time before receiving medical care on Earth since he would probably have to wait until the tour of duty on Mars ended and then would have to travel for nine months to reach Earth.

Cost Impact of Crew Impairment

The many scientific experiments and observations to be performed along with survival in the harsh Martian environment will require high levels of mental alertness and physical stamina. Therefore, considering the high cost of a MMM, the cost impact of improper or incomplete performance of planned tasks or early termination of a mission would be very great.

SAFEGUARDS AGAINST ENVIRONMENTAL TOXICITY

Considering the above consequences of environmental toxicity to the MMM, adequate safeguards against toxic exposures will be quite important. Most of these safeguards are similar to those currently used. They will include the following:

Proper Materials Selection

Candidate materials selected for use in the interior linings, insulations and flight hardware of the transit spacecraft and the MMS will have to undergo extensive preflight testing and be proven to be of low toxicity potential before usage. Selection will include:

Minimal Offgassing Characteristics

Most plastic and other nonmetallic materials slowly release a number of chemical vapors into the atmosphere. This is called offgassing. The rate of offgassing is increased by reducing the atmospheric pressure. Materials that offgas excessive amounts of trace gas contaminants (TGCs) are not considered to be acceptable for spaceflight. Due to the long duration of the MMMs and the probability of atmospheric pressures in the transit spacecraft and MMS being less than the 14.7 psi used today, minimal offgassing requirements for materials to be used in habitable areas will probably be more stringent than those required for materials used in today's spacecraft.

High Heat Stability and Low Flammability

Thermodegradation or combustion of many commonly used synthetic materials produces carbon monoxide, hydrogen cyanide, hydrogen fluoride and many other highly toxic gases. It is likely that in association with reduced atmospheric pressures, the breathing atmospheres of the transit spacecraft and the MMS will be 28% oxygen or higher. This will greatly increase the chance of combustion of many materials. Therefore, high heat stability and low flammability will be very important criteria for selection of materials to be used in the MMMs.

Containment of Bulk Chemicals

High levels of containment of bulk chemicals will be extremely important, since they could cause high levels of toxicity within a short time following escape. Bulk chemicals will probably include heat exchangers, disinfectants, fire extinguishants, tissue preservatives, electrolytes in storage batteries, and chemicals used in various types of scientific experiments and chemical processing facilities. Paints, adhesives and ingredients for insulation foams, all containing volatile chemicals, might be used in constructing the MMS. Even very small leaks in a large chemical reservoir might cause the release of toxic quantities of chemical vapors into habitable areas, causing physical or mental impairment or adverse health effects on crew members.

In cases where bulk chemicals do present a toxicological hazard, double or triple containment will probably be required. An airtight storage cabinet would count as one level of containment. It must also be definitely established that materials used in the containment vessels, including valves and O rings, are compatible with the contained chemicals so they will not be eroded over longer periods of time. Pressure alarms may be set up in certain types of containers, giving warning of overpressurization or a sudden loss of pressure (indicating that a chemical had escaped).

Alarms Near Potential Sources of Chemical Release

The atmospheres near stored chemicals, especially near actively functioning systems such as heat exchangers or processing facilities, must be monitored with alarms that would immediately warn the crew members of an increase in the concentration of vapors of these chemicals. Auditory alarms that warn of the sudden release of aldehydes and ketones, aromatic hydrocarbons, and a wide range of other chemicals are currently available and are widely used in industry.

Wearing of Protective Clothing, Goggles, and Masks

Because of the seriousness of illness or injury resulting from contact with toxic chemicals during a MMM (see above), extraordinary care must be taken by crew members to avoid contact during handling or working near these chemicals. It should be remembered that many gaseous or liquid chemicals are readily absorbed into the bloodstream through the skin. Some caustic chemicals may cause long term or even permanent visual impairment upon direct contact with the eyes. Therefore, persons working with or around toxic chemicals should be required to wear goggles along with gloves and other protective clothing. If there is a risk of the release of noxious vapors, crew members should also either wear an oxygen or charcoal filter mask or have one immediately available.

Fume Hoods

When one is handling volatile and toxic chemicals in certain processes such as infixing tissues with glutaraldehyde, the use of a fume hood, preferably a closed one with glove ports, will be important. A fume hood used within the closed environment of the MMS or transport vehicle would vent its outflow air through charcoal or another absorbent that could remove any contaminant vapors. It would be similar to the ones designed for the Spacelab.

A Safe Haven

In both the transit spacecraft and in the MMS, there must be a safe haven, that can be closed off from the principle living area. Here the crew members could take refuge in the event of a chemical spill or if toxic substances were released through thermodegradation. The safe haven should be able to support all of the crew members until the principal living area could be decontaminated.

MEANS OF DECONTAMINATION DURING THE MANNED MARS MISSION

As mentioned above, the toxic chemical vapors will be continuously released into the habitable areas of the transit spacecraft and the MMS through offgassing and crew respiration. In addition, large amounts of toxic vapors may enter the atmosphere unexpectedly through leakage of stored chemicals or thermodegradation of construction materials. Therefore, continuous removal of TGCs will be essential for maintaining a clean, safe environment. Equipment for the efficient removal of high concentrations of noxious vapors or quantities of spilled liquids or solids will also be necessary. Systems for accomplishing these tasks include:

Regenerable Absorbent Air Filters

The air filters to be used during the MMMs should effectively absorb a wide range of chemical vapors, including those of relatively low molecular weight. They should also be regenerable in order to minimize bulk and weight requirements. Most TGCs could be absorbed by charcoal, molecular sieves or other absorbents and then desorbed by heating at reduced atmospheric pressures. TGCs not removed by normal charcoal or molecular sieves may be removed by chemically treated charcoals.

Catalytic Oxidation

Catalytic oxidation, using platinum or other catalysts at ambient or elevated temperatures, may be useful in oxidizing carbon monoxide and other organic chemicals to carbon dioxide. Incomplete oxidation of many chemicals, however, may produce even more toxic products, so this should be avoided.

Condensation

Vapors of many alcohols and other water soluble substances will be condensed, along with water, by the duhumidifier system. These and other vapors may also be condensed in a cold trap, perhaps downstream from the duhumidifier. The cold outdoor climate of Mars at most times could provide very adequate cold for a cold trap.

Vacuum Pumps

Larger quantities of spilled liquids could, in many instances, be recovered with a vacuum pump. The Martian gravity would make this moot, however, since a liquid would fall to the floor and spread out, rather than float around in the air as a large sphere as would be the case in a microgravity environment.

CONSIDERATION FOR TOXIC CONTAMINANT EXPOSURE LIMITS

The design of toxic contaminant control systems for the MMM breathing atmospheres will be centered around keeping TGC's below designated SMAC* limits. Previously established SMAC limnits for space ventures of shorter duration will probably be revised for the MMMs. Factors to be considered in establishing these revised SMAC limits will include: (1) The anticipated maximum length of the MMM tours of duty; (2) The potential for simultaneous exposure to other toxic vapors; (3) The expected effect of physical deconditioning during the MMMs which could reduce one's resistance to the effects of certain toxic exposure; and (4) The expected effect of simultaneous radiation exposures which could enhance some types of toxicity.

A REAL TIME ONBOARD ANALYZER

Description

One essential item for toxicological control during MMMs will be a real time atmospheric analyzer, both in the transit spacecraft and in the MMS. This analyzer will probably be a more advanced design of the gas chromatograph/mass spectrometer (GC/MS) planned for the Space Station. It should be capable of monitoring atmospheric TGC's at fairly frequent intervals (at least several times daily) and should be capable of taking air samples through vent lines from each room or compartment that is used for human habitation. It will be highly automated, requiring minimal crew time and will not require a highly trained analytical chemist.

Other Usages Besides Atmospheric Analysis

The real time analyzer will probably find other applications besides analyzing atmospheres during the MMMs. It may be used in the analysis of the Martian atmospheres, soils and mineral deposits. It could also be used in health maintenance and in physiological experiments, analyzing body fluids and the exhaled breath of crew members for different metabolites. Still another usage might be the monitoring of reaction products produced in chemical processing facilities, seeing how well chemical reactions would precede in this new environment.

*Spacecraft Maximum Allowable Concentration

S E C T I O N   V I I

SUBSYSTEMS AND TECHNOLOGY DEVELOPMENT REQUIREMENTS

N87-17782

AMTEC: HIGH EFFICIENCY STATIC CONVERSION FOR SPACE POWER

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ABSTRACT

Future manned and unmanned space missions will require reliable, high efficiency energy conversion systems. For a manned Mars mission, power levels in the range 10kWe-100's kWe will be needed. The Alkali Metal Thermoelectric Converter (AMTEC) is a new direct energy conversion technology with the potential to meet these needs. It's characteristics include compactness, light weight, reliability and simplicity (no moving parts), and modularity, where efficiencies of 20-40% have been predicted. AMTEC is a thermally regenerative electrochemical device that derives its operation from the unique sodium ion conducting properties of beta-alumina solid electrolyte (BASE). It's high temperature operating range, 900 K-1300 K, makes it well suited for space power heat sources. To date, an efficiency of 19%, area power density of 1 W/cm^2, and a lifetime of 10,000 hrs at high temperature have been demonstrated in laboratory devices. Systems studies show that projected AMTEC systems equal or surpass the performance of other static or dynamic systems in applications of 1 kWe-1 MWe. Thus, the laboratory experiments and applications studies conducted to date have shown that the AMTEC possesses great potential. In order to bring this technology to the stage where prototype units can be built and operated, several technical issues must be addressed. These include the need for long life, high power electrodes, minimization of radiative parasitic losses, and high temperature seals. In summary, the evidence shows that if the AMTEC is developed, it can play a significant role in future space power applications.

INTRODUCTION

A manned Mars mission (MMM) will require a reliable high efficiency power conversion system that can provide electrical power in the 10 kWe-100's kWe range. The conversion system also has to be adaptable to a variety of energy sources (nuclear, solar and isotope) to be useful for various phases of the mission. The MMM will probably consist of mission module, lander and rover and can evolve to a permanent base on Mars. The electrical power requirements can vary from tens of kilowatts for the lander to 100's of kilowatts for a base. The lifetime of the power conversion system should be in the 7-10 year range to accommodate the various mission scenarios. Reliability of the conversion system is especially important for the manned missions, and redundancy in components or the total system has to be offered. Low mass is always an important factor in space power. It becomes even more important for MMM because of the large ratio of mass required in Low Earth Orbit (LEO) to that delivered to the surface of Mars. The conversion system also has to have a small area/volume, especially if aerocapture for Mars landing or the Earth return is employed.

It is desirable for a power conversion system to meet the above criteria as well as other mission peculiar requirements and considerations. Previous NASA missions, with the exception of short duration missions (e.g., Apollo, Space Shuttle) that used fuel cells, have usually taken advantage of static conversion systems (e.g., photovoltaic, thermoelectric). Static conversion systems are modular, reliable and have a moderate specific power. However, they offer a low energy conversion efficiency (7-10%) which in view of high fuel cost, makes them less attractive. Dynamic conversion systems, on the other hand, offer higher conversion efficiencies, but a mass penalty has to be paid to make them redundant and reliable. This is particularly true for power levels less than ~25 kWe. A system that promises to offer the advantages of both static and dynamic conversion systems is the Alkali Metal Thermoelectric Converter (AMTEC), a system with the modularity and reliability of static conversion systems.

The AMTEC is a thermally regenerative electrochemical device for high efficiency, direct thermal to electrical energy conversion.(1,2) Its operation derives from the unique sodium ion conducting properties of beta-alumina solid electrolyte (BASE).(3) The AMTEC accepts heat at 900-1300 K and rejects heat in the range 400-800 K, with predicted device efficiencies of 20-40%.(4) These projected efficiencies are much higher than any other direct thermoelectric device. To date, an efficiency of 19%(5) and an area power density of 1 W/cm^2 (4) have been demostrated in laboratory devices.

Among the AMTEC characteristics that make it potentially attractive for space applications are compactness, light weight, reliability and simplicity (no moving parts) and modularity. Since the high temperature sodium reservoir may be heated externally, the AMTEC may be coupled with chemical, nuclear or solar heat sources. The high temperature range is well suited for current and projected space power heat sources. Its high efficiency and light weight would provide high specific power (W/kg), and radiator area would be minimized. Modular units could be designed to produce efficient power generation over a range of power system sizes. Also, a particular AMTEC system could be operated at reduced power levels without large drops in efficiency during periods of low demand.

A comprehensive technical review of thermally regenerative electrochemical systems, including AMTEC, was carried out by the Solar Energy Research Institutes for DOE.[6] It found that the AMTEC (or sodium heat engine) has potentially the highest power density and efficiency of such systems. Thus, the characteristics and potential of the AMTEC make it a candidate for a variety of applications, including space and remote power.

PRINCIPLES OF OPERATION

The operating cycle of the AMTEC is illustrated in Figure 1. A closed vessel is divided into a high- temperature/pressure region in contact with a heat source and a low-temperature/pressure region in contact with a heat sink. These regions are separated by a barrier of BASE which has an ionic conductivity much larger than its electronic conductivity. The high-temperature/pressure region contains liquid sodium at temperature T_2, and the low-pressure region contains mostly sodium vapor and a small amount of liquid sodium at temperature T_1. Electrical leads make contact with a porous (positive) electrode which covers the low pressure surface of the BASE and with the high temperature liquid sodium (negative electrode). When the circuit is closed, sodium ions are conducted through the BASE due to the difference in vapor pressures (or chemical activity) across the BASE, while electrons flow to the porous electrode surface through the load, producing electrical work. The unique feature of the AMTEC cycle is that sodium vapor is expanded nearly isothermally through the BASE, causing the sodium atoms to separate into sodium ions and electrons. The AMTEC thus converts the work of isothermal expansion of sodium vapor directly to electric power.

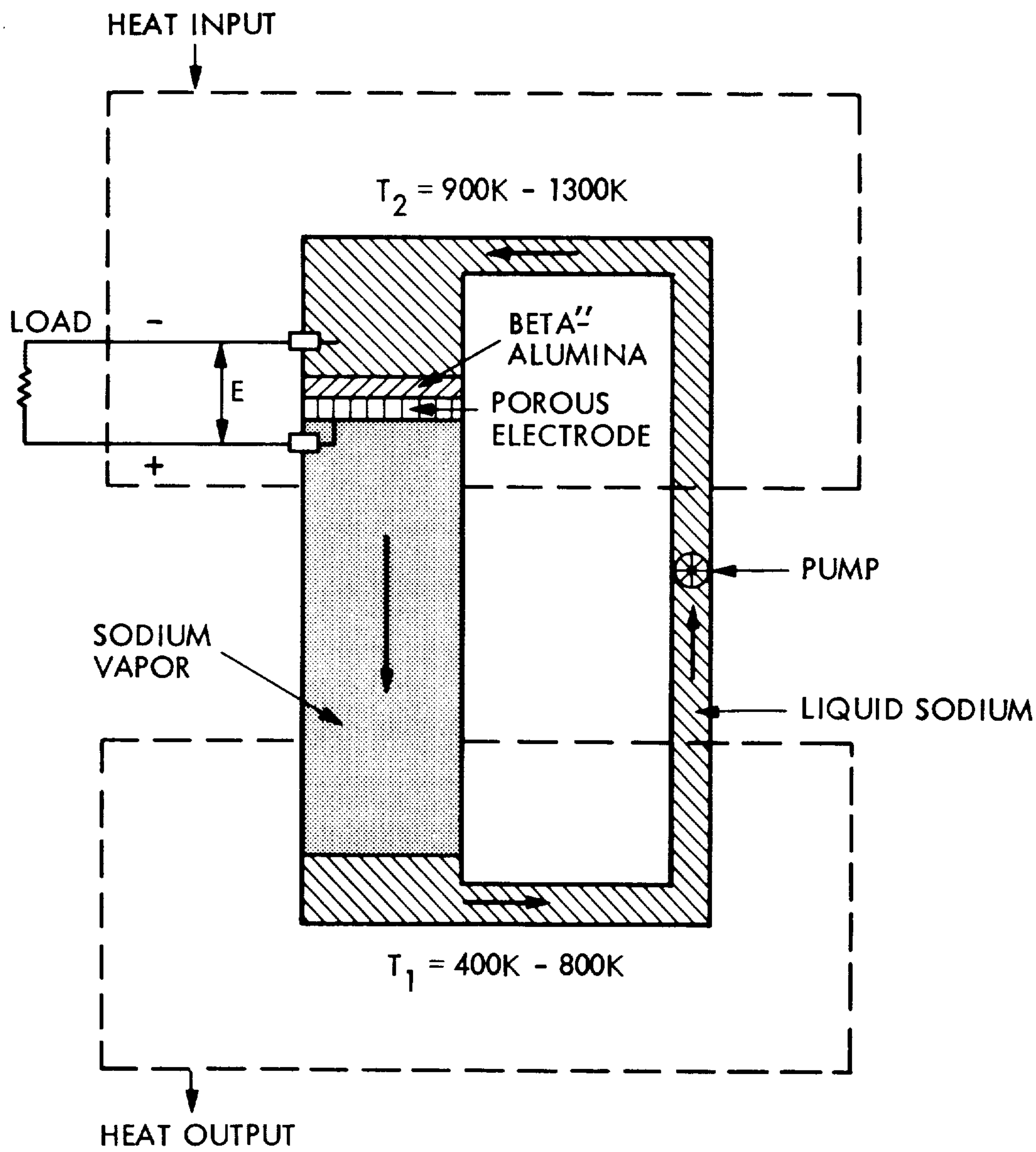


Figure 1. AMTEC Schematic

A return line and an electromagnetic pump (no moving parts) circulate the sodium working fluid through the AMTEC. Complete details on the AMTEC cycle may be found in References 1, 2, 4 and 7.

The operating characteristics and system performance may be determined from AMTEC voltage vs. current characteristics:

$$V = A - B\ln(I + \delta) - Ihr_o, \tag{1}$$

where A,B and δ depend on temperatures T_1 and T_2, and r_o is the resistivity of the BASE which is also dependent on T_2. I is current density in A/cm^2 and h is BASE thickness in cm. The relationship for A,B, δ and r_o are given in Ref.7. Equation (1) is derived from kinetic theory and an equation relating pressure and the rate of evaporation of sodium atoms from the low pressure hot surface.[2,4] Representative voltage vs. current curves for different temperatures are shown in Figure 2.

AMTEC efficiency is given by:

$$\eta = IV/[\ IV + I(L + C(T_2 - T_1)/F + Q_{loss}], \tag{2}$$

where L is the latent heat of vaporization for sodium, C is the average specific heat of liquid sodium, and F is the Faraday. Q_{loss} is the sum of parasitic conductive and radiative heat losses. This quantity depends on the specific AMTEC design and should be minimized in any practical device. Its effect on system performance is illustrated in Figure 3. Note that both power output and parasitic losses scale with electrode area. The AMTEC thus lends itself to modular design, since efficiency will be relatively independent of size (or output).

TECHNOLOGY STATUS

Work on the AMTEC concept has to date predominantly been directed toward the development of a long life, high power porous electrode. State-of-the-art electrodes are composed of either magnetron sputter deposited molybdenum (1-3 m thick) or of bi-conductor cermet composed of a mixture of molybdenum and BASE powders. For the cermet electrode, power densities are very low initially, but have been observed to be stable for up to approximately 10,000 hours.[8] In the case of the thin film molybdenum electrode, the data show that power densities are initially high (near theoretical), followed by a decay by a factor of 3-5 in the first 1000 hours of operation. Power output usually remained stable thereafter. However, more recent studies at JPL indicate that the high initial power densities for thin electrodes may be explained by certain

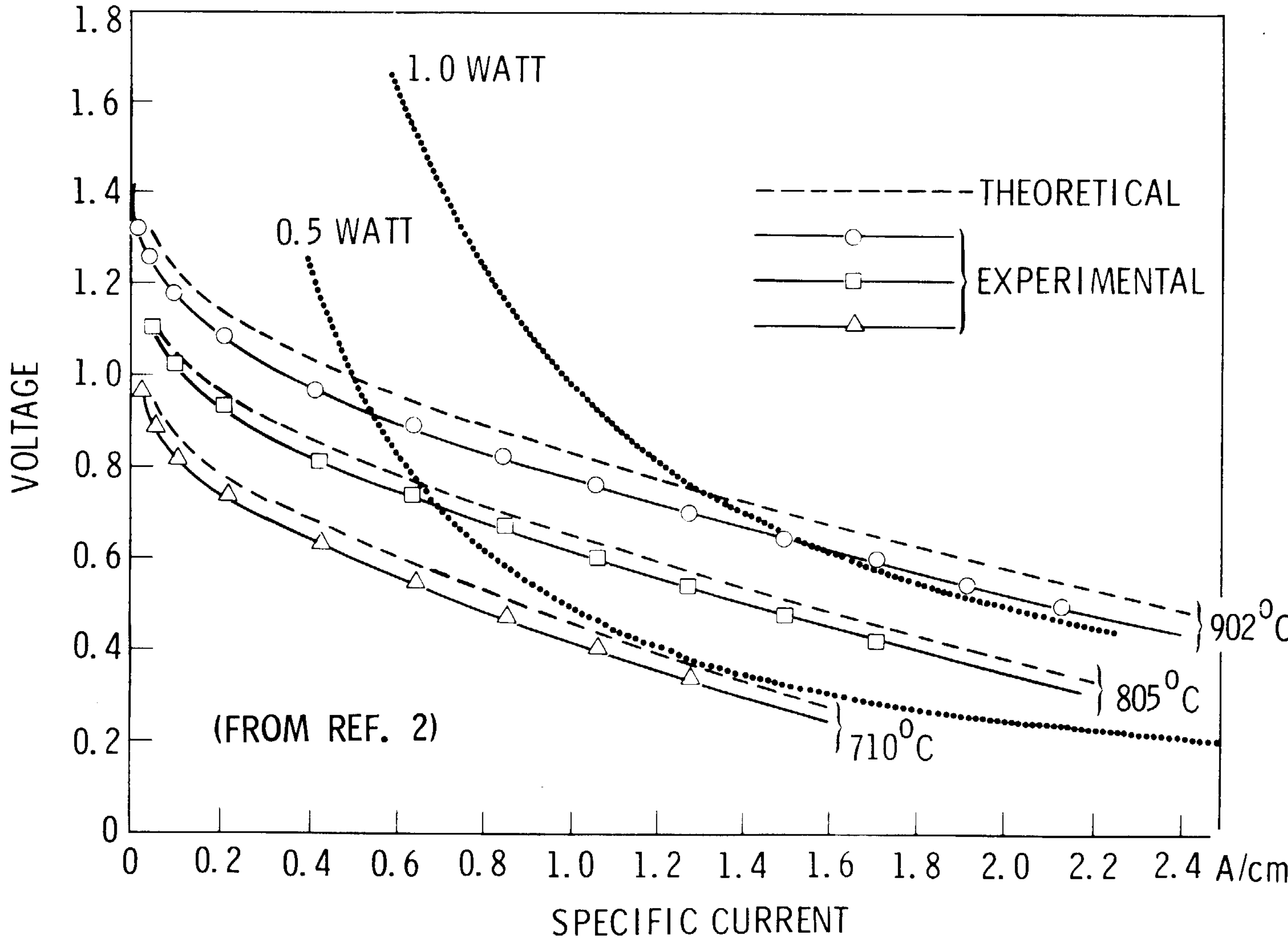


Figure 2. AMTEC: Voltage vs Current

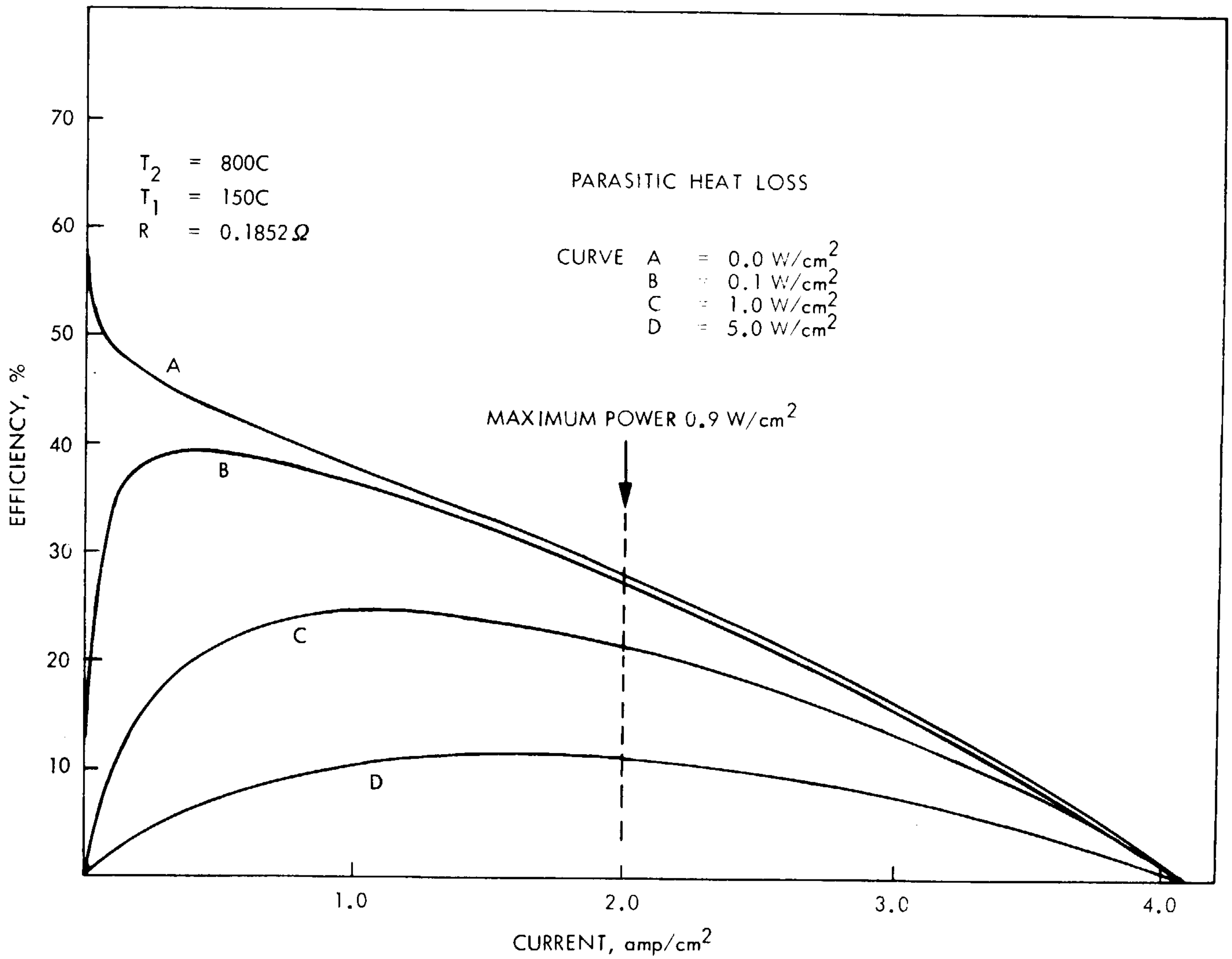


Figure 3. Efficiency vs Current

electrochemical mechanisms and that the high power lifetime may be restored and extended by oxygen treatment of deposited Mo electrodes.[9]

As mentioned previously, recirculating devices have been built which have demonstrated relatively high efficiency[5] and more recently long lifetime potential (10,000 hours).[8] A single cell recirculating device schematic is shown in Figure 4. Also, series connection of AMTEC cells has been demonstrated.[8]

SYSTEM STUDIES

System studies of the AMTEC have been carried out at JPL for space power applications (7,9,10) over a wide range of power levels (1-10,000 kW). In the concept designs studied by Bankston, et al.,[7,9] the results (Figure 5) show that small radioisotope space power systems (~1kW) would operate at an efficiency and specific power (W/kg) which is 3-4 times the currently utilized Seebeck effect generators. For larger nuclear reactor based systems (1000kW), AMTEC projections equal or surpass efficiency and mass characteristics of both the static (thermoelectric, thermionic) and dynamic (Brayton, Rankine, Stirling) system projections. The AMTEC would also allow the nuclear reactor to be operated at lower temperatures than most other conversion options. Ewell[10] extended these projections to the 1MW level and found the AMTEC to be very competitive with the static and dynamic systems. In the most recent study,[11] it was found that if the AMTEC is developed, it would be the preferred power converter for an unmanned Mars rover.

Preliminary solar energy system projections were carried out at Ford by Subramanian and Hunt.[12,13] They calculated that with a "near term" AMTEC technology, a point focused system would achieve a converter-receiver efficiency of 25%, and with advanced technology, 33%. In an AMTEC/Rankine cycle topping/bottoming system combination, estimated efficiencies would increase to 37% and 45 % for near term and advanced levels. These efficiencies are highly competitive with alternate technologies. Recently, in a study for the Department of Energy,[14] Sandia National Laboratories ranked the AMTEC first among heat engine technologies with the potential to meet DOE's long-term goals for dish electric solar systems.

In the transportation sector, the AMTEC has been proposed as a prime power source and as the battery charger in an AMTEC-battery hybrid elec-

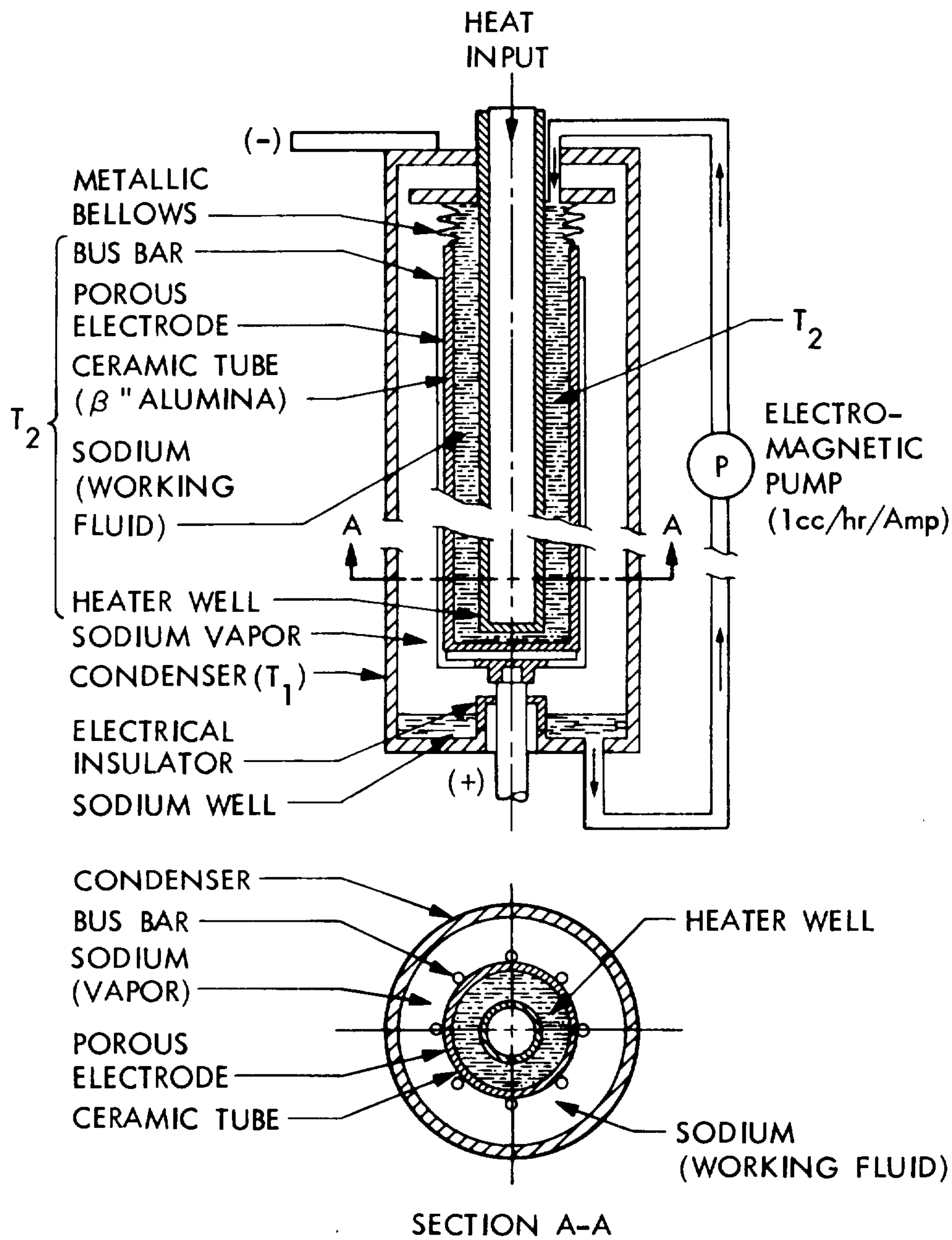
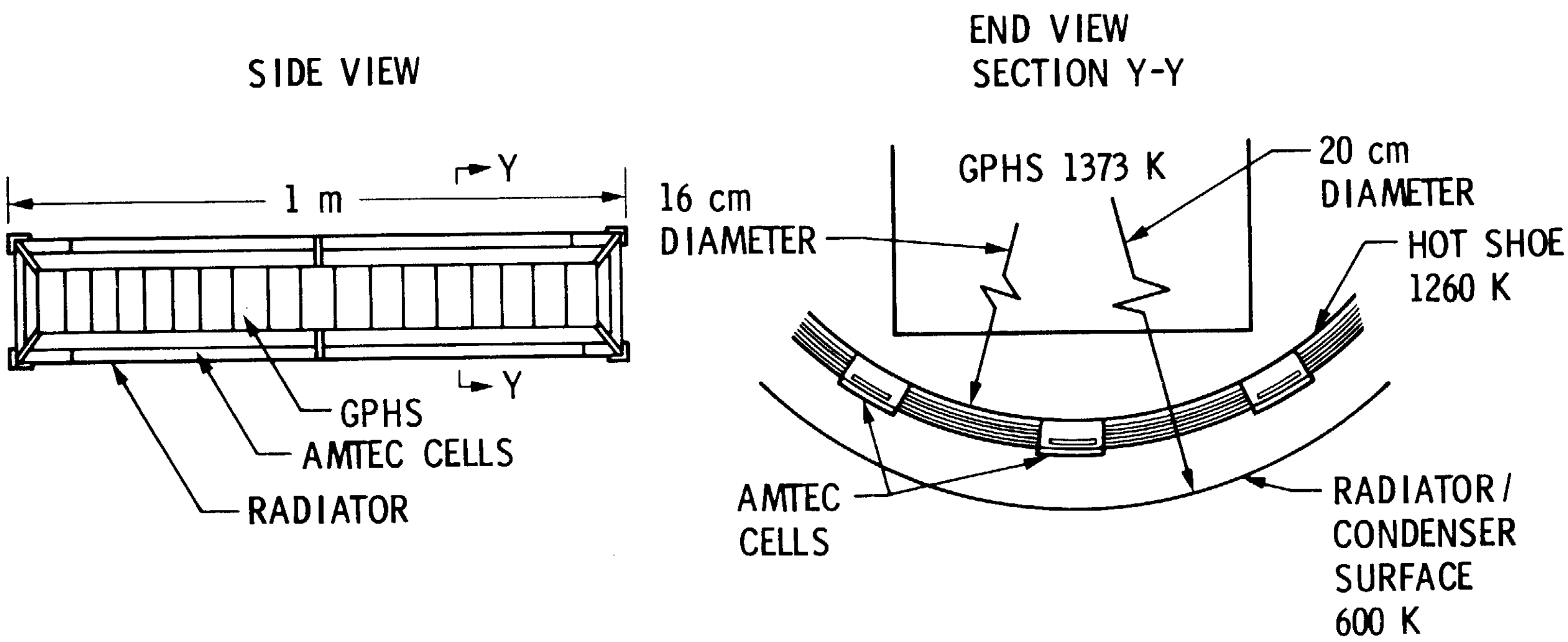


Figure 4. AMTEC Recirculating Test Cell



- AMTEC POWER SOURCE NEARLY IDENTICAL TO GPHS RTG CONFIGURATION, DIMENSIONS, AND MASS---PROJECTED OUTPUT POWER IS MORE THAN TRIPLED BECAUSE OF INCREASED EFFICIENCY
- BASELINE DESIGN HAS A MASS OF 55 kg AND PROJECTED OUTPUT POWER OF 890 W=> SPECIFIC POWER 16.2 W/kg; SPECIFIC POWER BASED ON DEMONSTRATED LABORATORY PERFORMANCE IS 14.7 W/kg (SOA GPHS RTG 5.2 W/kg)

Figure 5. AMTEC Radioisotope Power Source Concept

tric vehicle. As a primary power source, the compactness (~ 0.5 kW/liter) of the AMTEC makes it a potential competitor for >1MW electric drives for locomotives or other large land vehicles. In a hybrid system, a well engineered AMTEC should be able to generate direct current at high efficiency for battery charging.

COMPARISION OF AMTEC WITH OTHER POWER CONVERSION SYSTEMS

As mentioned earlier, AMTEC offers efficiencies and specific power (kW/kg) better than (in the 1-100 kWe range) or comparable (in the 100 kWe-1 MWe range) to dynamic conversion systems. This is illustrated in the following figures. Figure 6 shows the radiator area requirement for various power conversion systems. The radiator area requirement for AMTEC is lower because of its high temperature heat rejection. Figure 7 shows the efficiencies of various conversion systems. The figure shows the efficiencies for intermediate technology (dashed lines) and advanced technology. The basic difference between the two technologies is the hot side temperature. Figure 8 shows the total system mass for various conversion systems in the 1-100 kWe power range. The numbers in the 1 kWe range are the system studies that have been performed by various organizations (7,15,16,17), and the 100 kWe numbers are the result of SP-100 system studies. Figure 9 shows the same comparison for 100 kWe-10 MWe power ranges.

TECHNICAL ISSUES

The laboratory experiments and applications studies conducted to date have shown that the AMTEC concept possesses great promise. However, it is still at an early stage of development, relative to more mature energy conversion devices. Thus, a significant technology development effort will be required before prototype units can be tested. Work to date has led to the conclusion that to bring the AMTEC to a state of technical maturity, several key issues must be resolved.

(a) The porous electrode is crucial to the operation of the AMTEC. To date, high power density operating periods of 300-1000 h have been achieved. Recent experimental work has resulted in regeneration of degraded electrodes and led to an understanding of electrode degradation mechanisms. Continued work on stabilizing electrode performance is needed to extend lifetimes to 10,000 h or more.

(b) The efficiency equation for the AMTEC shows that parasitic heat loss by radiation from the porous electrode to the condenser must be tightly controlled to achieve 20-40% efficiency. A smooth film of liquid sodium on the condenser surface would meet this requirement. Thus, materials or surface treatments that will insure a smooth film on the condenser surface must be identified. Also, if operated in zero-g, this condenser must also be capable of collecting and transporting sodium for recirculation.

(c) Operation of the AMTEC for thousands of hours will require several high quality ceramic to metal and metal to metal seals in the high temperature zone. State-of-the-art techniques have been demonstrated for short term testing; however, durable, long life integrity must be demonstrated.

(d) Control circuitry and optimum power/voltage conversion components must be developed.

(e) What are the effects of thermal, electrical and mechanical transients on the durability of the device? Testing must be conducted.

(f) The initial design studies have shown that the AMTEC would provide significant performance advantages in space power applications. However, design optimizations have not been carried out, particularly with respect to coupling with appropriate heat sources.

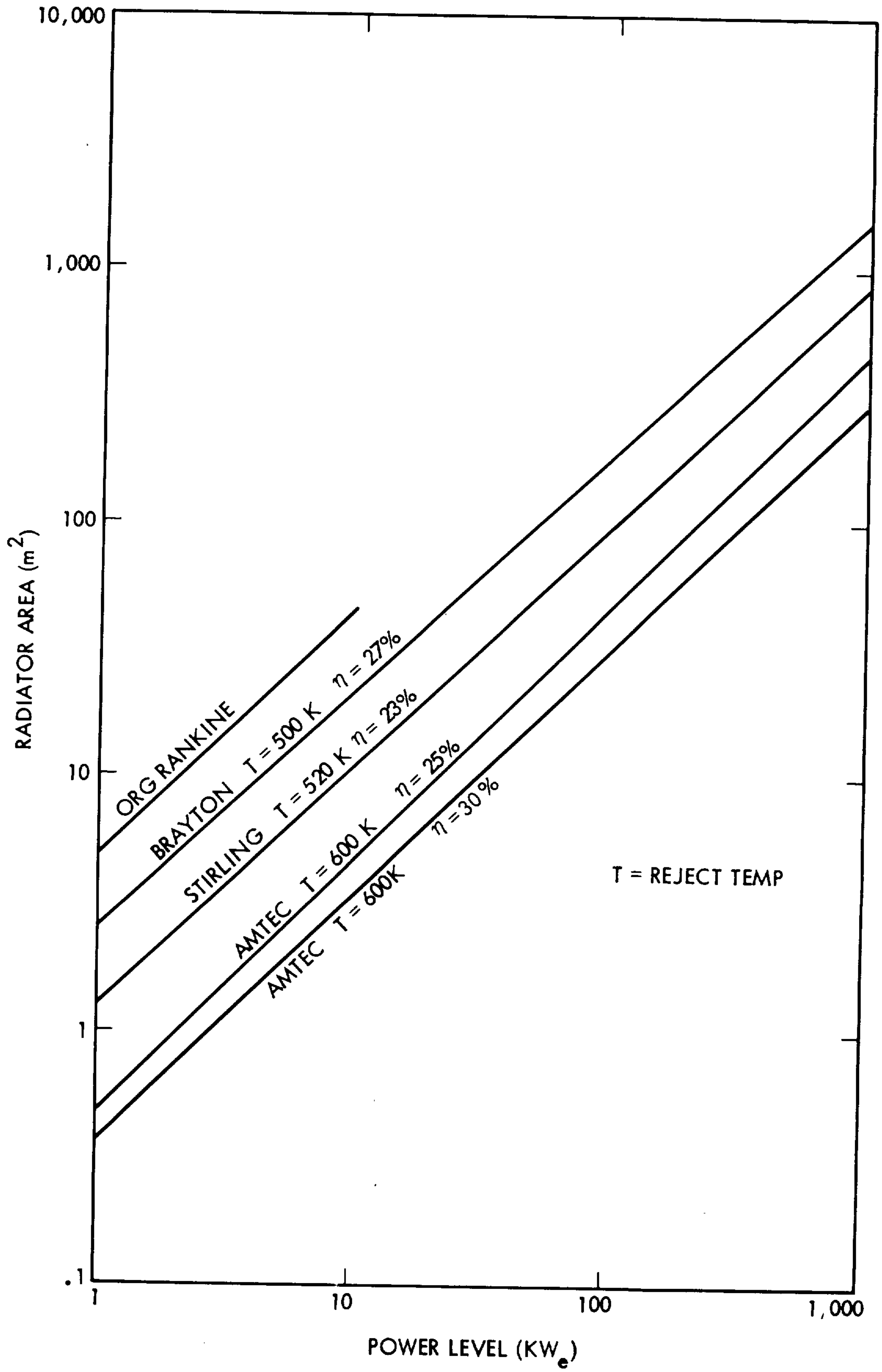


Figure 6. Radiator Area vs Power

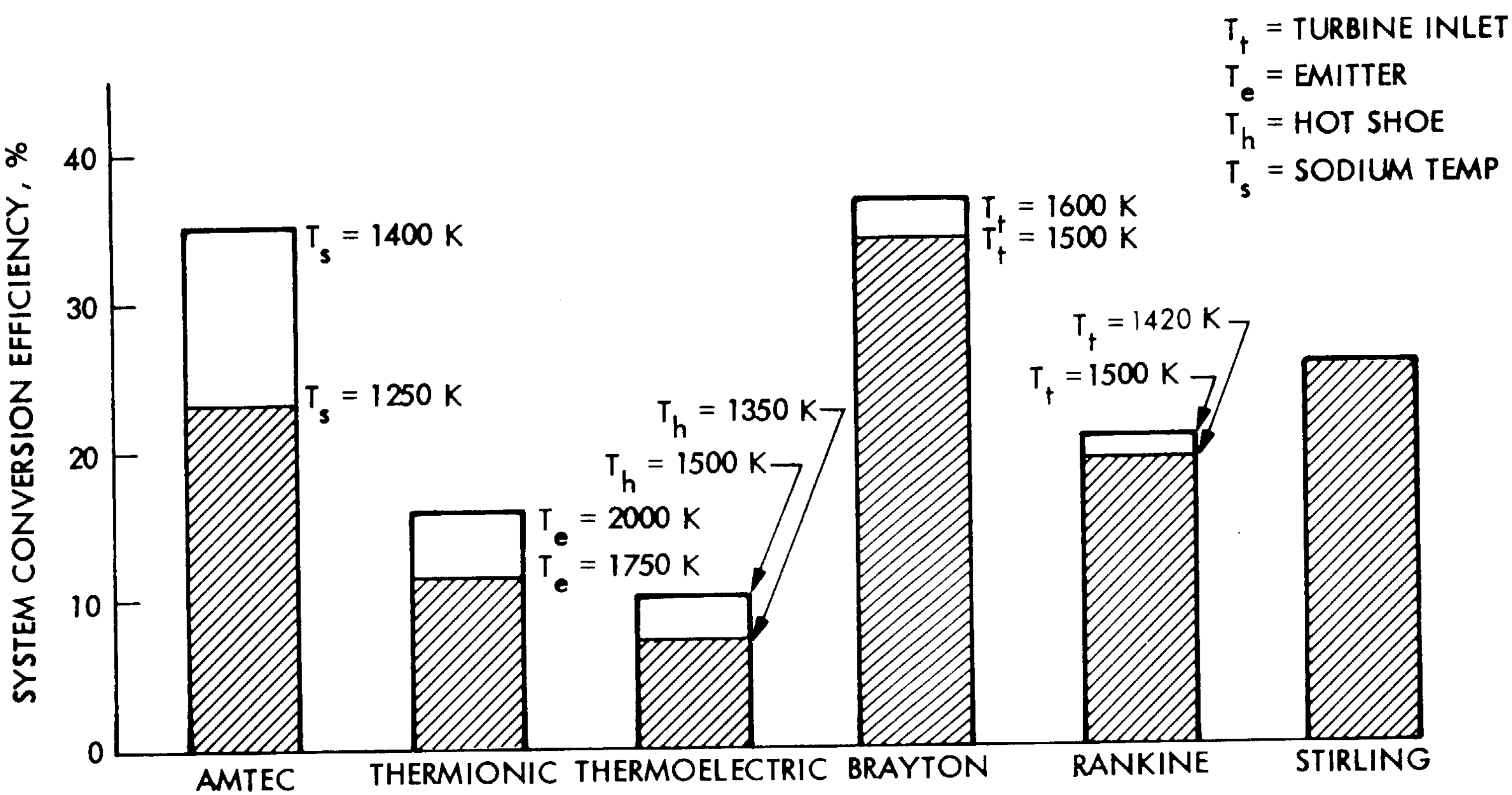


Figure 7. System Conversion Efficiencies

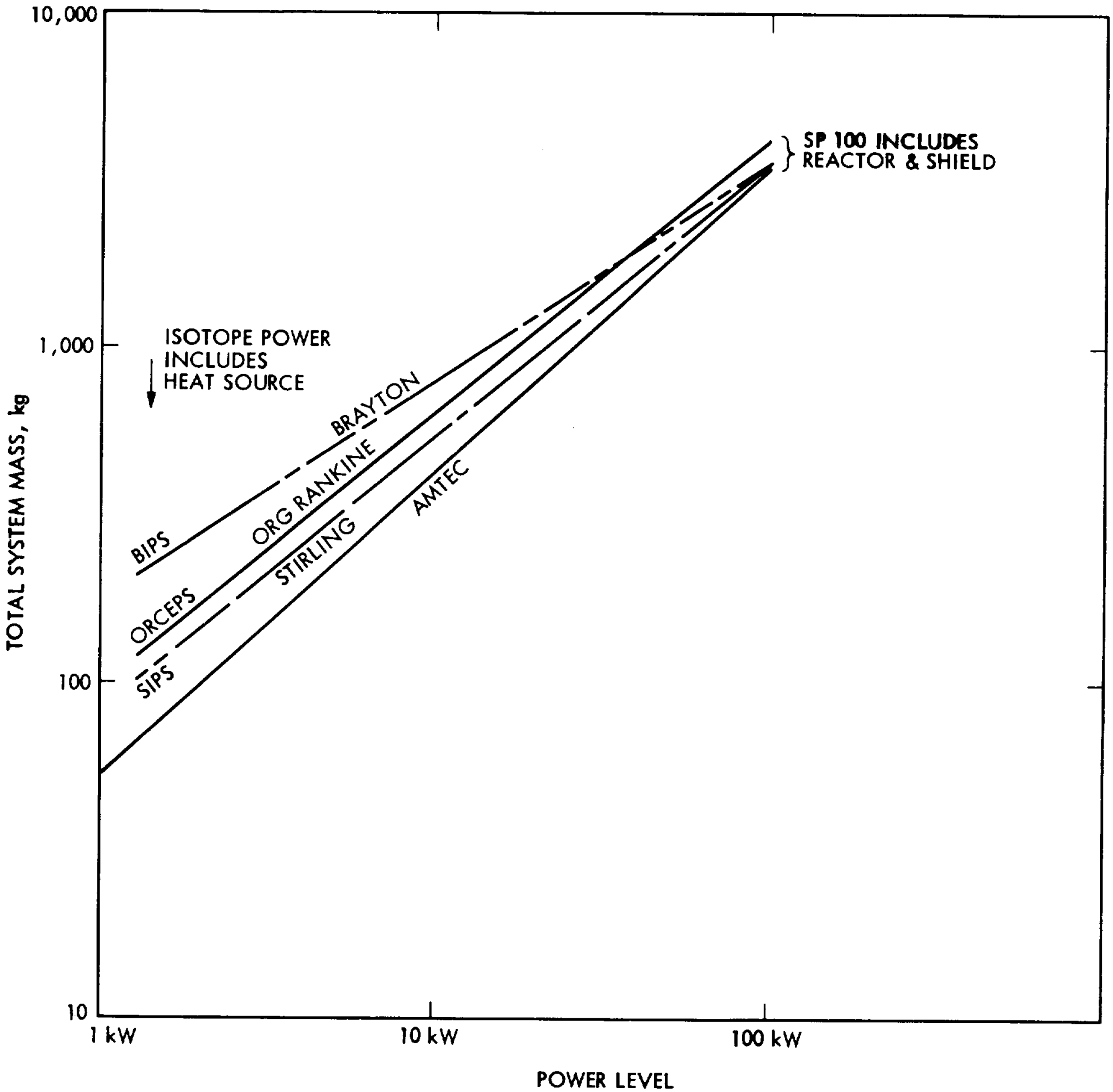


Figure 8. System Mass Comparisons 1-100 kWe

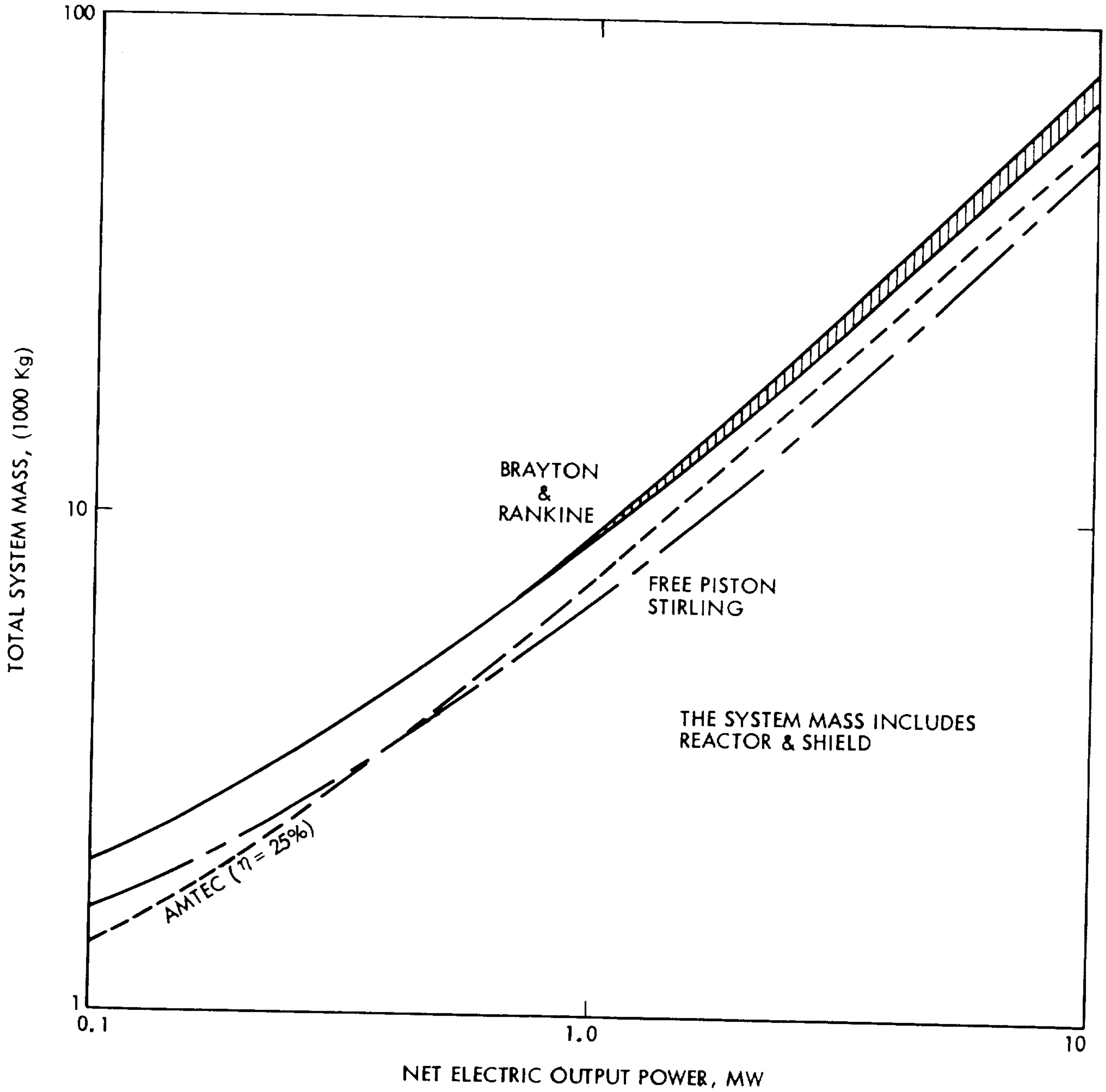


Figure 9. System Mass Comparisons 0.1-10 MW_e

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N87-17783

MARS LANDING MISSION: A STRUCTURAL APPROACH

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ABSTRACT

A Mars landing mission in the 2000 opportunity presents a structural challenge. Earlier studies have indicated that a Mars landing was then feasible using current structural techniques. Since these earlier studies, technology advances have been made to enhance the capability. Lighter and stronger materials, large structures programs, and super computers now exist and even greater advances are expected.

The feasibility of a Mars landing does not depend on the structure. If the space travelers can withstand the trip, the necessary structures can be provided to deliver them. If artificial gravity is required the structure can also provide for it.

The structural challenge is to provide structural designs that are lightweight with high reliability. In order to do this advanced technology must be utilized to the fullest on all structural elements.

LOADS ENVIRONMENT

Shown in Figure 1 are the load conditions imposed on the structure during the course of a manned Mars landing mission. It is obvious the Earth launch condition is the most severe load condition the structure will encounter for the entire Mars mission.

PRIMARY MISSION OPTIONS

The classical Mars landing mission of past studies has considered propulsive stages for braking into Mars orbit and for braking into Earth orbit. Development in the understanding of aero-braking technology has led to the concept of placing aerodynamic brakes and heat shields on the spacecraft to provide the delta-V necessary to brake the spacecraft into Mars and Earth orbit. The propulsive stages are replaced by these structural/thermal shields. Shown on Figures 1a and 1b are representative configurations for the propulsive and aero-braking concepts respectively.

FIGURE 1

STRUCTURAL LOAD ENVIRONMENT FOR MARS LANDING MISSION

o EARTH LAUNCH

(Most severe structural loads)

-Aero, Thrust, bending, max q Liftoff

o EARTH ORBIT ASSEMBLY

-Docking Loads

-Manuvering Loads

o EARTH DEPARTURE

-Thrust Loads (well defined)

o BRAKING

-Thrust Loads

-Aero Braking

o MARS LANDING

-Aero

-Landing

o ASCENT

-Thrust Loads

-Mars Launch Loads

o MARS DEPARTURE

-Thrust Loads

o EARTH BRAKING

-Propulsive Loads

-Aero Braking

FIGURE 1a.
MANNED MARS MISSION
1999 OPPOSITION
ALL PROPULSIVE OPTION

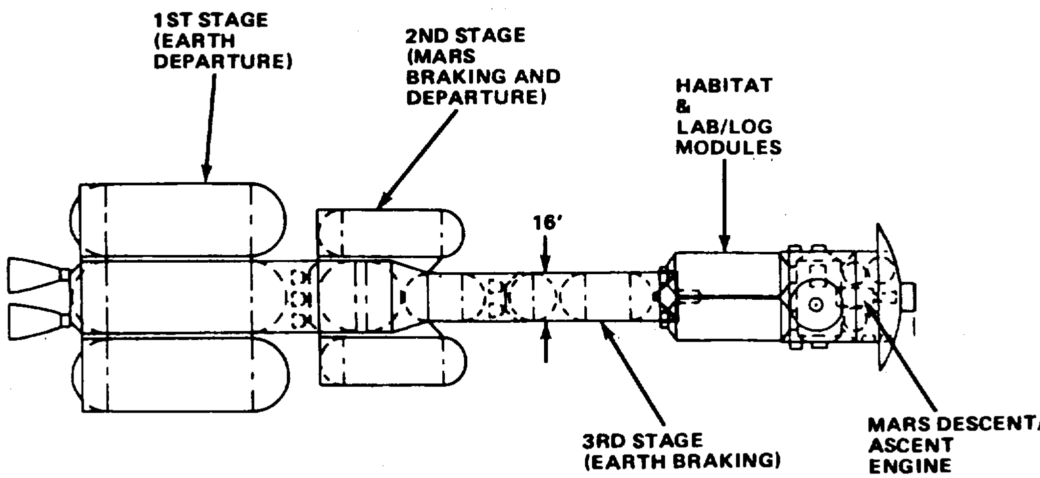
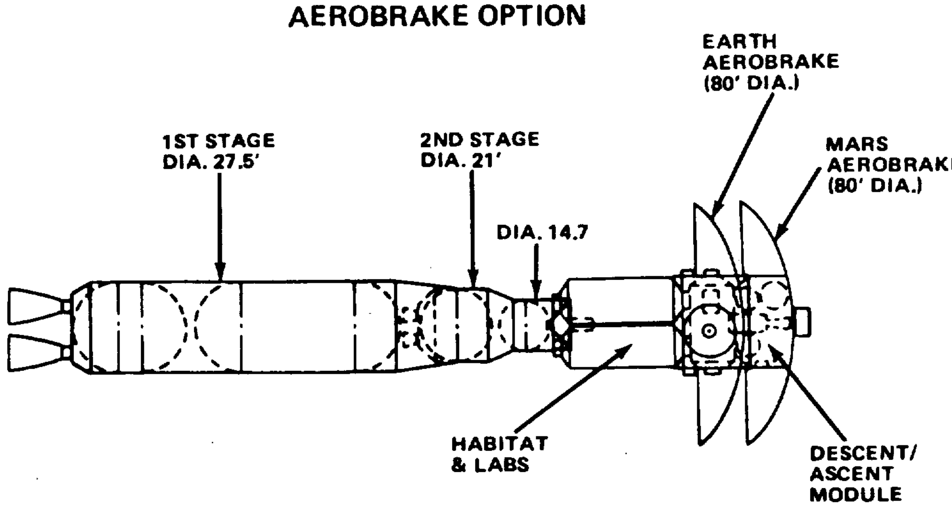


FIGURE 1b.
2001 OPPOSITION
MANNED MARS MISSION
AEROBRAKE OPTION



EARTH LAUNCH PHASE

All elements of any trans-Mars injection configuration are required to endure the Earth launch condition. The launch condition includes vehicle bending and acceleration loads, liftoff loads, and max q conditions. Shown in Figure 2b is an assumed launch vehicle. All structural elements are required to reflect the launch environment. It is important the Earth launch vehicle relieve the Mars stages from as much load as possible. Since the launch vehicle is in the early phases of selection this phase of the Mars study should define requirements imposed on the launch vehicle by the Mars landing mission. Shown in Figure 2a are the Shuttle Derived Vehicle (SDV) and the Heavy Lift Launch Vehicle (HLLV) compared to the current Shuttle configuration. A larger vehicle will allow for more efficient structural elements.

STRUCTURAL ELEMENTS

Shown on Figure 3a and 3b are representative configurations of all the propulsive and aero-braking configurations respectively.

The following major structural elements have been identified:

(1) Trans-Mars Injection Stage - LO_2/LH_2 stage that required multiple launches (smaller stage required for aerobraking)

(2) Mars Braking (*) / Departure Stage

(3) Earth Braking Stage (*)

(4) Mars Excursion Module (MEM) - Landing Stage; Pressurized Habitat and Lab; Departure Stage

(5) Interstages - The interstages are light weight; no launch loads are carried thru them.

(*) Indicates Aero-braking option

TRANS-MARS INJECTION STAGE

The trans-Mars injection stage puts more requirements on the selection of the Earth launch vehicle than any of the Mars stages because of the massive size of the stage and the amount of propellant it must hold. Since the stage is separated after the trans-Mars burn, the trans-Mars injection stage is less technology critical than other elements. As seen from Figures 3a and 3b the injection stage for the aerobraking concept is smaller than the stage for the all-propulsive option. The structural loads are primarily due to launch and to LEO environments.

FIGURE 2a. SHUTTLE–DERIVED VEHICLE

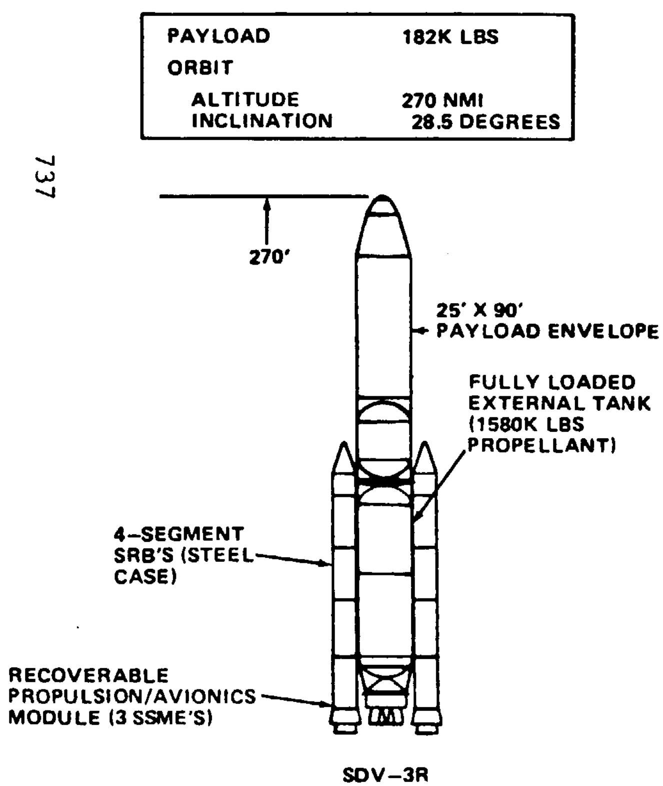


FIGURE 2b. TYPICAL EARTH–TO–ORBIT VEHICLES

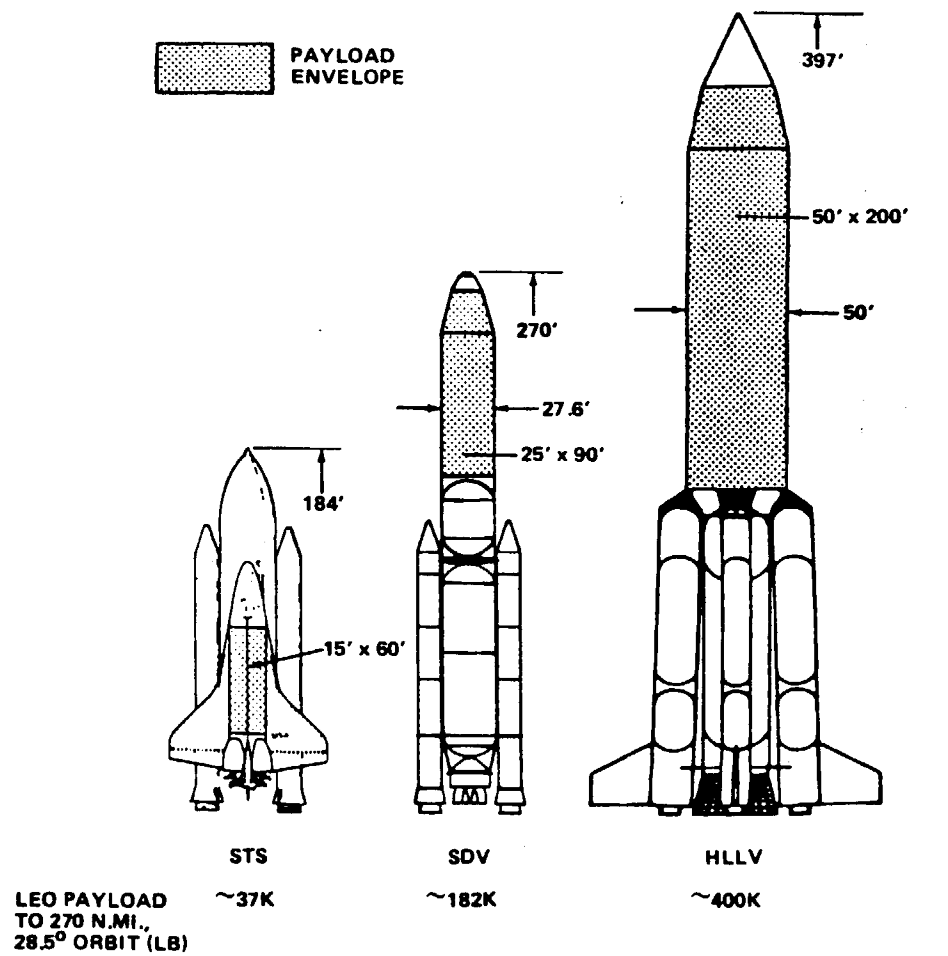


FIGURE 3a. ALL–PROPULSIVE CONFIGURATION

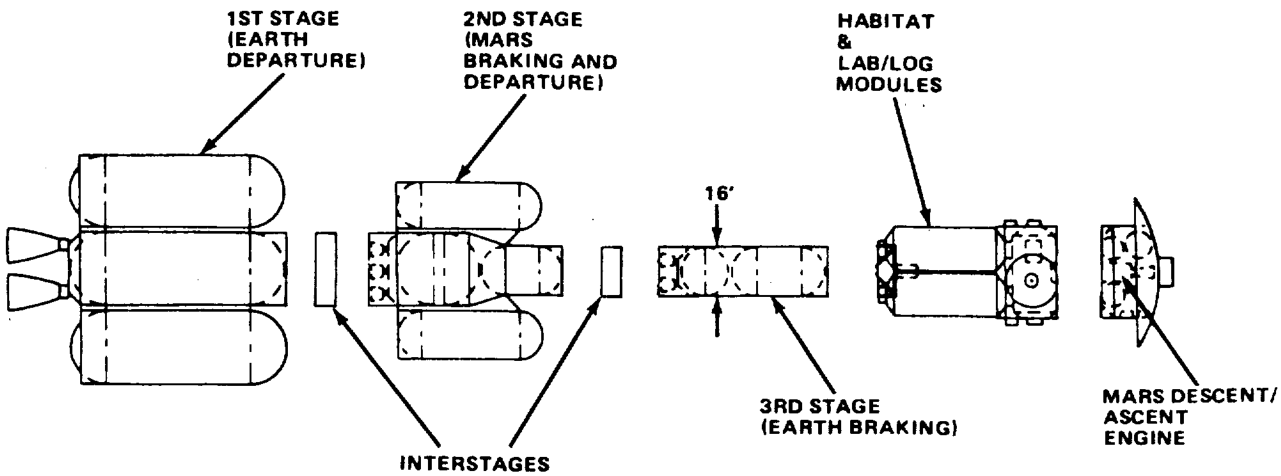
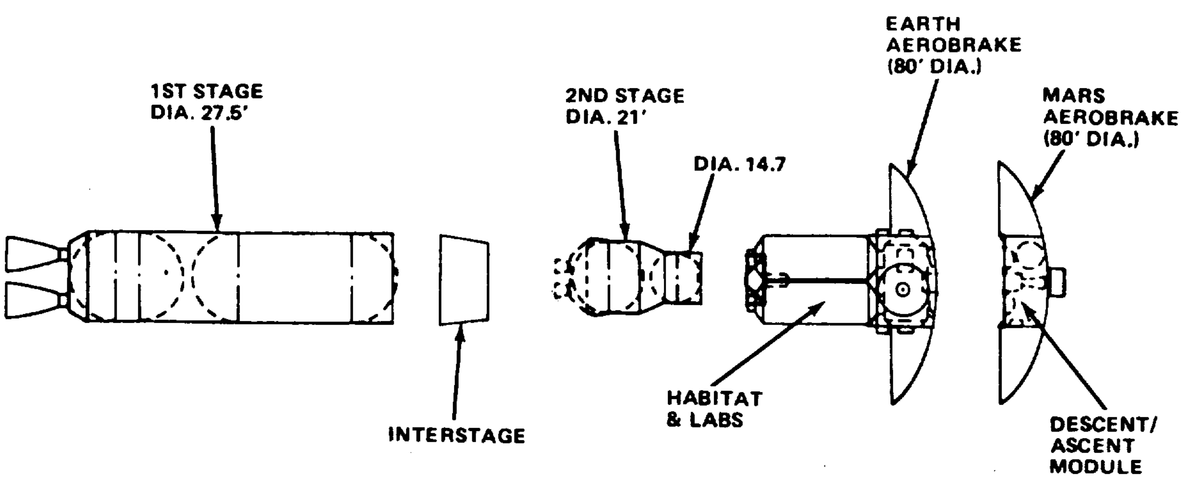


FIGURE 3b. AEROBRAKING CONFIGURATION



The propulsive stage is a LO_2/LH_2 stage launched in three separate launches. The core stage contains the engines and LOX tanks. Much of the propellant will be launched after the stage is launched.

A definite advantage of having an all aero-braking configuration is the reduced size of the trans-Mars stage. Even with the reduced size the stage will have to be launched partially full with the balance of the propellant to be launched later; therefore, a propellant "tanker" concept is required to support the Mars landing mission.

MARS BRAKING STAGE

Mars braking can be provided by propulsion or by an aero-braking shield. The all-propulsive option utilizes the braking stage combined with the Earth return stage. The structural approach for the braking/return stage is to employ maximum technology to reduce the stage weight. The aerobraking concept utilizes an aero-shield to provide the braking. The diameter of the Mars braking shield will require assembly in Earth orbit unless it is a slender shape which can be launched intact. Though the braking concept is estimated to be much lighter than a propulsive stage the loading conditions and temperature considerations will require the maximum use of high technology materials and analysis.

Earth Braking Stage

Braking at Earth is provided by a propulsive stage or an aero-braking stage. In either case the technology requirements are high. This stage weight impacts the structural weights of the trans-Mars stage, Mars braking and departure stage. The aero-braking shield may require assembly in LEO whereas the propulsive braking stage and propellant can be carried to LEO on one launch without requiring LEO assembly. The aerobraking concept is estimated to be much lighter than a propulsive stage; however, the loading conditions and temperature considerations will require the maximum use of high technology materials and analysis.

MARS MISSION MODULE (MMM)

The Mars Mission Module goes through every phase except landing. It must protect the space travelers throughout the entire mission. There are two concepts considered; the single MM and the multiple concept which utilizes the Space Station type modules to build up to the MM. Shown in Figure 4 are weight comparisons between two concepts. Since the single large module is much lighter the structural preference is the single

FIGURE 4a.

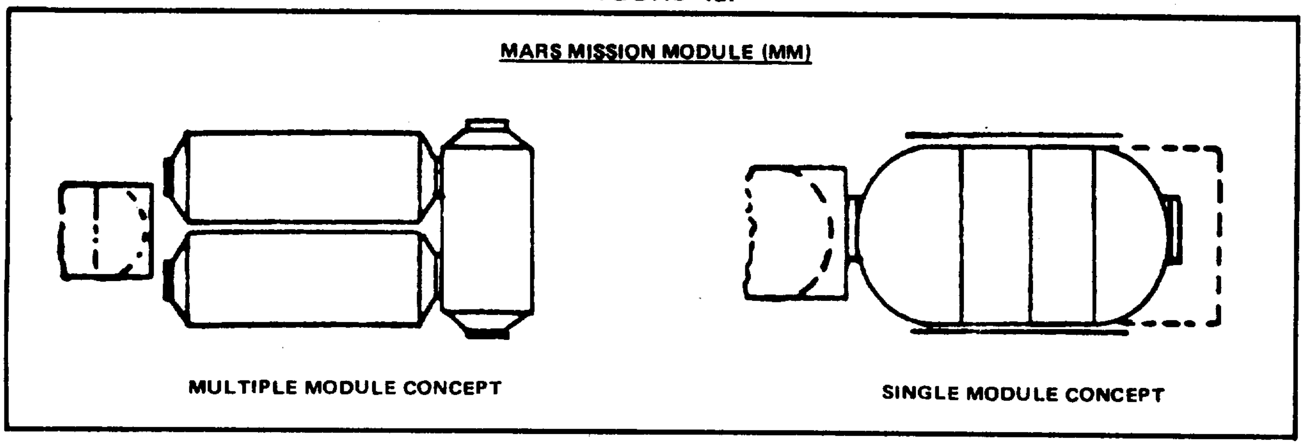
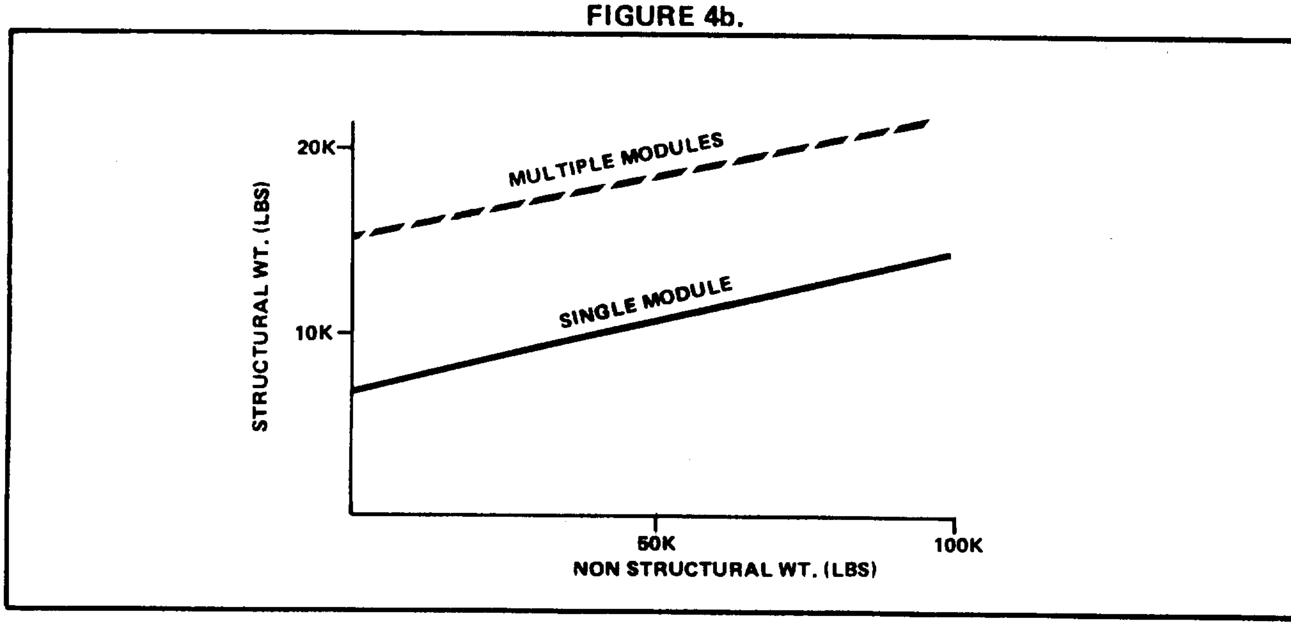


FIGURE 4b.



module. Every stage weight is impacted by the Mission Module, therefore, the MM is constructed of the highest technology material available.

MARS EXCURSION MODULE (MEM)

The MEM goes through every loads environment except the Earth return and braking, and the ascent stage must experience those environments in some mission scenarios. The MEM structure faces more unknown conditions than any of the other elements. The MEM consist of four primary elements: (1) Aero-brake; (2) Propulsive landing stage; (3) Ascent stage; and (4) Pressurized Habitat. Also, the MEM must deliver several independent sets of equipment such as Mars rover vehicle, and Mars surface test equipment along with providing the capability to return with samples. The MEM has to be as light as possible and still meet the Mars mission requirements. Also, the MEM impacts the Mars braking and the trans-Mars injection stages. Since the aero-braking shield will not be tested after assembly the structural approach and materials must provide for a light-weight structure with high reliability. It is therefore necessary that a high technology approach for the entire MEM structure be taken.

Because of the complexity of the design requirements and loading conditions proper structural analyses have not been made to determine the MEM structural weights.

INTERSTAGES

The interstages will be supported for Earth launch such that the only loads they see are self-induced loads. The design loads for the interstages then are the trans-Mars propulsion or braking loads. They will be constructed of lightweight material.

TECHNOLOGY REQUIREMENTS

Figure 5 shows the relative technology ranking for the various structural elements. The ranking is from 1 through 10 where 10 is the highest technology requirement. The various structural elements do not have the same sensitivity to improvements over current technology. The purpose of Figure 5 is to determine where the structural technology emphasis needs to be.

FIGURE 5

TECHNOLOGY EMPHASIS

Item	all propulsive	aero-braking
trans-mars injection stage	3	2
Mars braking stage	5	7
Mars departure stage	6	6
Earth braking stage	9	10
Mission Module(MM)	10	10
Mars Excursion Module(MEM)	10	10

STRUCTURAL TECHNOLOGY PROJECTION

The primary focus on structural technology is to reduce the structural weight. Much advancement with composite materials has been made that allow for lighter and stronger structures. Shown on Figure 6a is a projection of weight reductions that can be expected through the year 2000. Also, shown on Figures 6b and 6c are specific characteristics for some composite materials. These type material advances when applied to the MEM and MM have much potential to enhance the Mars landing mission.

FIGURE 6a.

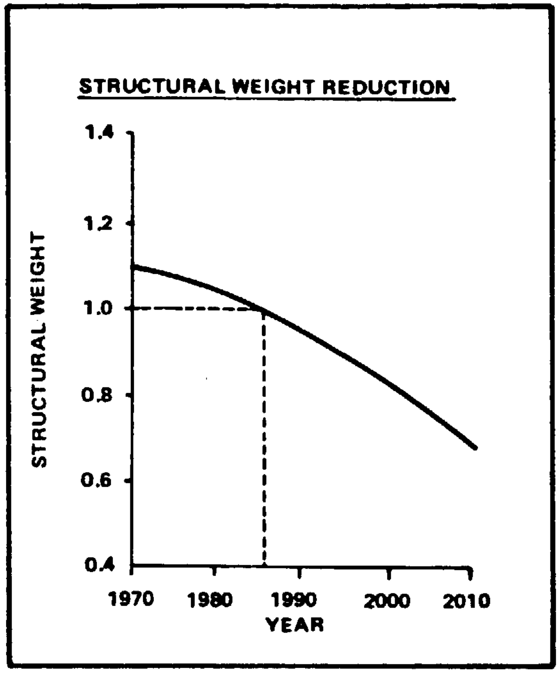


FIGURE 6b.

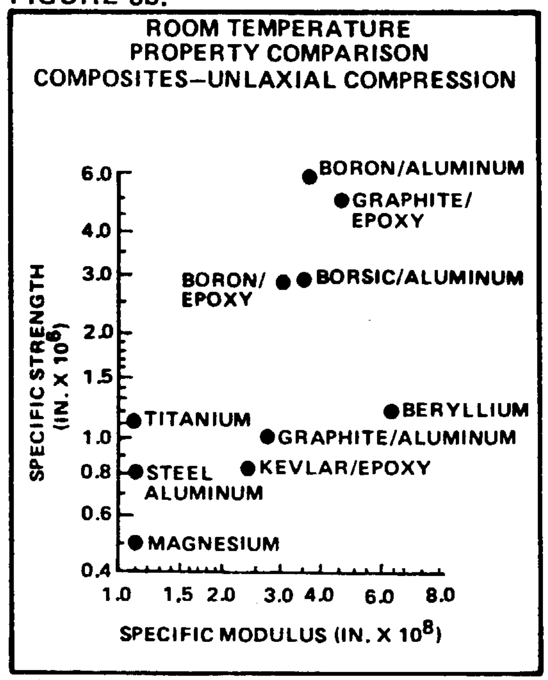
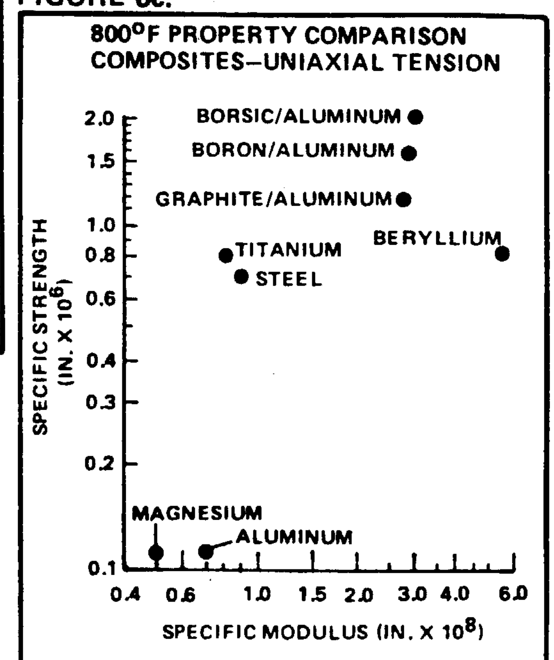


FIGURE 6c.



MICROMETEOROID PROTECTION

Meteoroid protection must be provided during all phases of the Mars landing mission. The following table shows the considerations for the various structural elements.

Item	Exposed Area	Mission Time	Probability of no Penetrations (Po)
Trans-Mars Injection Stage			.99
Mars Braking Stage			.995 (*)1.000
Mars Departure Stage			.995
Earth Braking			.995 (*)1.000
MM			.999
MEM			
Aero-brake			1.000
Landing stage			.99
Ascent stage			.9999
Habitat			.995

(*) Aero-braking Shield

The overall mission requirement for micrometeoroid protection has not been established. Flux models are required for near Earth, the trans-Mars orbit, and near Mars. The above probabilities are estimates. The total probability of penetration of all structural elements should equal the overall mission requirement. An estimate for the overall requirement is .995 for the mission duration.

SUMMARY

A manned Mars landing presents a number of challenges in the area of structural design; however, it appears that the structure is not the critical element for a Mars mission. The structural approach is to utilize advanced technology to make the mission more reliable and cost effective. The purpose of this paper is to present a survey of structural considerations in order to focus thinking on future work.

N87-17784

MARS TRANSIT VEHICLE THERMAL PROTECTION SYSTEM: ISSUES, OPTIONS, AND TRADES

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ABSTRACT

A Mars mission is characterized by different mission phases. The thermal control of cyrogenic propellant in a propulsive vehicle must withstand the different mission environments. Long term cryogenic storage may be achieved by passive or active systems. Passive cryo boiloff management features will include multilayer insulation (up to 4 in.), vapor cooled shields, and low conductance structural supports and penetrations. Active boiloff management incorporates the use of a refrigeration system. Key system trade areas include active verses passive system boiloff management (with respect to safety, reliability, and cost) and propellant tank insulation optimizations. Technology requirements include refrigeration technology advancements, insulation performance during long exposure, and cryogenic fluid transfer systems for mission vehicle propellant tanking during vehicle buildup in LEO.

INTRODUCTION

The manned Mars missions are characterized by different mission phases. They are low Earth orbit (LEO) buildup, vehicle transit, and Mars orbit and surface. The thermal environment in each phase presents varied challenges to maintain thermal control of stored cryogenic propellants. For example, during LEO buildup, the vehicle is in a severe environment in which propellants must be stored for periods of months. This situation presents unique problems requiring a high performance and reliable thermal control system (TCS). Likewise, in the mission transit phase and Mars orbit, degraded insulation performance and system reliability become major factors when analyzing thermal environments. The thermal control options include active and passive type systems. The TCS and fluid acquisitions system must withstand each environment. Mission trades, issues, and technology requirements have been identified in order to determine the feasibility of the TCS options.

PASSIVE THERMAL PROTECTION

Advanced passive thermal control systems for cryogenic propellant tanks include the following: multilayer insulation (MLI), vapor cooled shields (VCS), thermodynamic vent system (TVS), low heat leak support struts, and a para-ortho hydrogen converter.

Multilayer insulation consists of thin radiation shields separated by low-conducting material. Coatings have been developed which provide environmental protection for the radiation shields with minimal performance loss. Cryogenic storage will require multiple layers of MLI with fabrication/assembly techniques that minimize seam heat leaks. Currently, there is no significant experience in installing or testing MLI in thicknesses over 4 in.

Another subsystem of passive thermal control is the vapor cooled shield (VCS). A VCS is similar in design to a conventional cross flow heat exchanger. Cryogenic fluid is routed across a metallic shield to intercept part of the heat leak which would otherwise reach stored cryogens. Assuming a LH_2/LO_2, propellant combination, the VCS system is most efficient when utilizing a coupled tank configuration (See Figure 1). Liquid or vapor may be drawn from the LH_2 tank for VCS use; however, studies have shown that saturated vapor (boiloff) may be the more feasible choice. After the vented fluid leaves the LH_2 heat exchanger, it is routed to the LO_2 tanks, in the coupled tank configuration, to perform the same function. Study results have shown approximately 50% reduction in heat leaks when using an optimally located VCS. Optimal location of a VCS is estimated to be 40%-50% of distance through the MLI measured from the tank wall.

The thermodynamic vent system performs a dual function. The primary function is to regulate tank pressure through controlled escape of saturated vapor (boiloff). The TVS is an effective pressure control device for any system (active or passive). Additionally, when used in conjunction with a VCS, the TVS provides the saturated vapor from the tank. Utilizing both the TVS and VCS and an effective insulation system, relatively long term storage of cryogens is possible.

Tank support struts are a key element in thermal design of cryogenic tanks. Heat leaks due to structural supports have been estimated at approx. 30% of the total environmental heat leak. Various composite materials with high strength and low conductivity properties are under investigation for use as support structures. Other options include

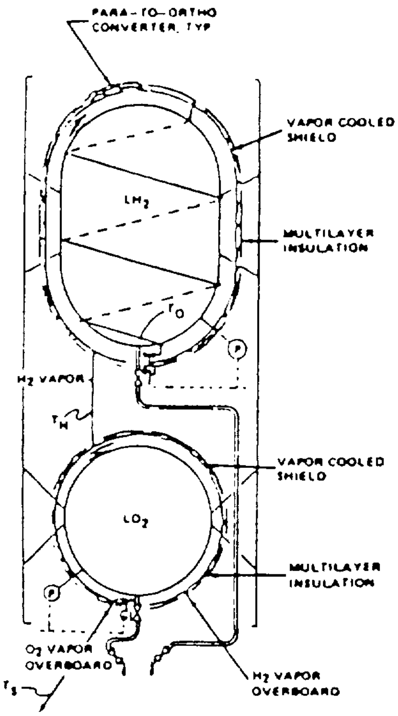

FIGURE 1. TYPICAL PASSIVE TPS (COUPLED TANK DESIGN)

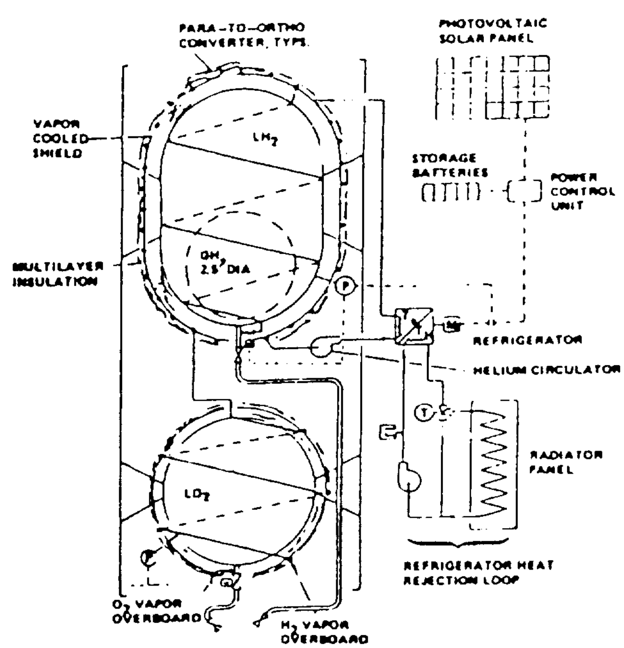

FIGURE 2. TYPICAL ACTIVE TPS

configurations of orbital disconnect struts where highly efficient struts are complemented by larger struts for support during launch loads.

A para-ortho hydrogen converter exploits the endothermic reaction of transforming para hydrogen into a para-ortho mixture. Feeding the para hydrogen saturated vapor (boiloff) over a catalyst bed allows additional heat adsorption. Development is continuing on improved catalysts and the converter efficiency.

ACTIVE THERMAL CONTROL

Primarily, there are two methods of active thermal control; refrigeration and reliquefaction. A refrigeration system design would intercept a large portion of the total heat load into the tanks. A reliquefaction system would be designed to reliquefy the boiloff of propellants. Elements in a spacecraft refrigeration system include the refrigerator, a power supply, power conditioning equipment, and a heat rejection system. (See Figure 2) A number of different refrigerator cycles have experienced various degrees of development. Current technology refrigerator cycles (such as Brayton, Claude, Stirling) have the potential to provide cooling in the range of 10K to 100K in capacities ranging up to a few hundred watts.

Reliquefaction would eliminate the need for any tank venting by reliquefying the boiloff vapor. The efficiences associated with reliquefaction are lower than those using refrigeration systems. Reliability, along with power, weight, and volume requirements are areas of development for space based reliquefaction systems.

FLUID TRANSFER

An important part of the propellant tankage design is fluid transfer. All transfers must be accomplished with minimum waste and ullage venting. The primary fluid transfer subsystems of concern are zero-g liquid acquisition devices, pressure control systems, and mass gauging systems. Currently, significant progress in liquid acquisition devices for storable propellants has been achieved. However, progress has been very slow in developing similar devices for cryogenics. Cryogenic tank pressures must be maintained within a specified range. Control of the pressure during engine burn may be maintained by inert gas or autogenous pressurization, that is, pumping liquid to a higher pressure, vaporizing and heating it, and then returning the heated vapor back into the tank. Thermodynamic venting (required with VCS) can be utilized to control pressure during storage periods. Mass gauging of cryogenic liquid in a tank under zero-g conditions has yet to be demonstrated. Currently, work is being conducted in this area. The potential use of subcooled or slush LH_2/LO_2 appears to be feasible to enhance propellant storage during vehicle buildup. The subcooled liquids will provide a limited thermal sink to absorb heat leaks. It is estimated that propellant with up to 60% solids content can still be transferred using state-of-the-art fluid lines.

ISSUES

Support structure requirements form the main issues in the LEO buildup phase. Assembly time should be minimized and propellant loading scheduled to optimize cryogenic control. With the relatively high boiloff rates in LEO, long term refrigeration of propellant tanks may be required. The main issue associated with active thermal protection systems is the power to performance ratio. A normalized range of required input power for refrigeration systems operating at LH_2/LO_2 temperatures are shown in Figures 3 and 4, respectively. The estimated refrigeration input range due to mission vehicle propellant boiloff is well above any current state-of-the-art system. Current systems are generally providing refrigeration capability of less than 50 watts, whereas this system will be required to have a capability in excess of 400 Watts. Additional issues when utilizing long term active systems include weight and volume requirements and reliability concerns. The passive TCS will experience thermal cycling, long term exposure, degraded insulation, and potential micro-meteoriod impact. The propellant transfer/resupply issues include transfer methods, advanced technology systems, and minimum leak joints for safety considerations.

The vehicle transit mission phase is characterized by relatively low boilff due to an efficient passive TCS and reduced environmental heating. An important point to make is the issue of preferred vehicle orientation. The preferred orientation of the space vehicle can be represented as the longitudinal axis of the vehicle "pointing" toward the sun. This orientation results in minimum tank area being exposed to direct solar flux. Any deviation of the longitudinal axis of the vehicle will expose greater surface area to solar flux and result in increased environmental heating.

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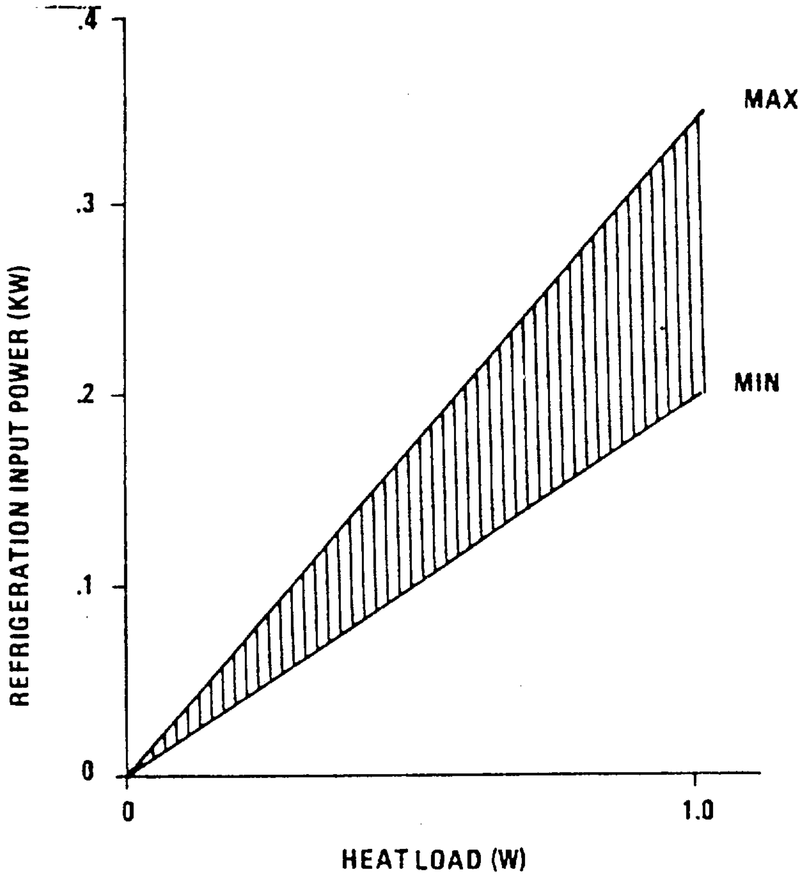

FIGURE 3. REFRIGERATION REQUIREMENTS @ LH_2 TEMPERATURE (38°R)

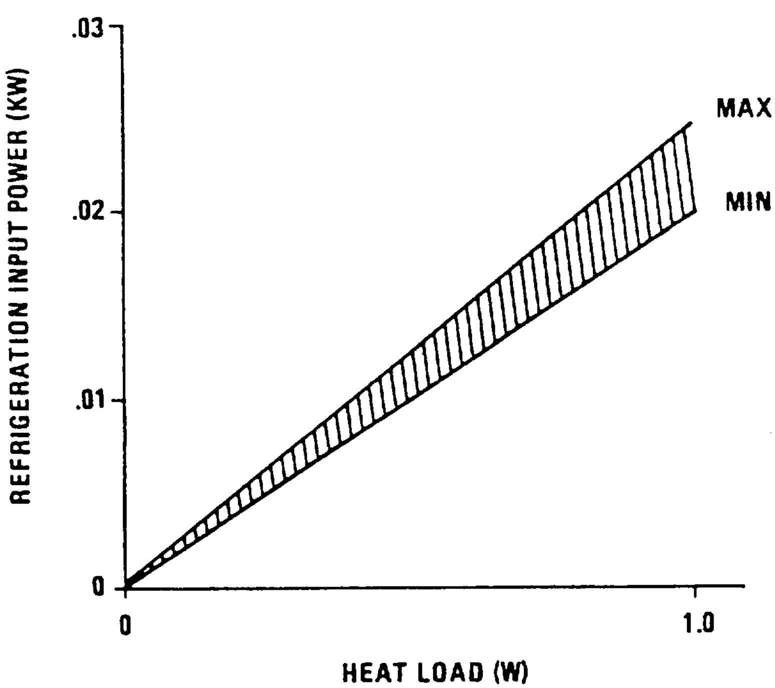

FIGURE 4. REFRIGERATION REQUIREMENTS @ LO_2 TEMPERATURE (162°R)

Additional insulation or shields may be placed around tank bulkheads (in the all cryogenic stage configuration) to reduce environmental heating effects. Long-term space exposure effects, e.g. degraded performance of MLI/coatings, are areas of concern when trying to maintain very low heat leak rates.

The Mars orbit and surface mission phase is concerned with variable environmental conditions. The vehicle in orbit around Mars could experience medium environmental heating rates (compared to LEO) due to orientations dictated by mission requirements. If a preferred orbital orientation could be achieved (resulting in minimal surface area exposed to direct solar flux), environmental heating would be reduced. The TCS must maintain a high level of performance throughout the Mars mission duration. The insulation performance is common to other mission phases. The lander propellant tanks (LO_2) must have additional thermal control due to degraded insulation performance on the Martian surface resulting from increases in pressure (i.e. gas conduction in insulation). A Dewar type tank design may be used in order to minimize cryogenic boiloff during the surface stay time. Dewar tank sizing, along with weight are issues that impact the MEM vehicle.

KEY TRADES

In LEO, the cryogenic boiloff rate will be relatively high, potentially requiring a form of active thermal control. Refrigeration, reliquefaction, or a combination could be used. The support structure system would have to be sized for the time period required during LEO buildup. Key trades identified for active thermal control systems include: 1) performance versus power requirements, 2) performance versus weight/volume requirements, and 3) reliability issues. Additional analysis should evaluate the impact of providing an active system during the entire mission due to environments, driven by vehicle orientation, dictated by mission requirements.

Insulation optimization is another key trade in the long term control of cryogenic propellants. Preliminary estimates show the optimum range of tank insulation to be 2-4 in. (See Figures 5 and 6). Trades identified for insulation optimization include insulation performance, weight, and degraded characteristics.

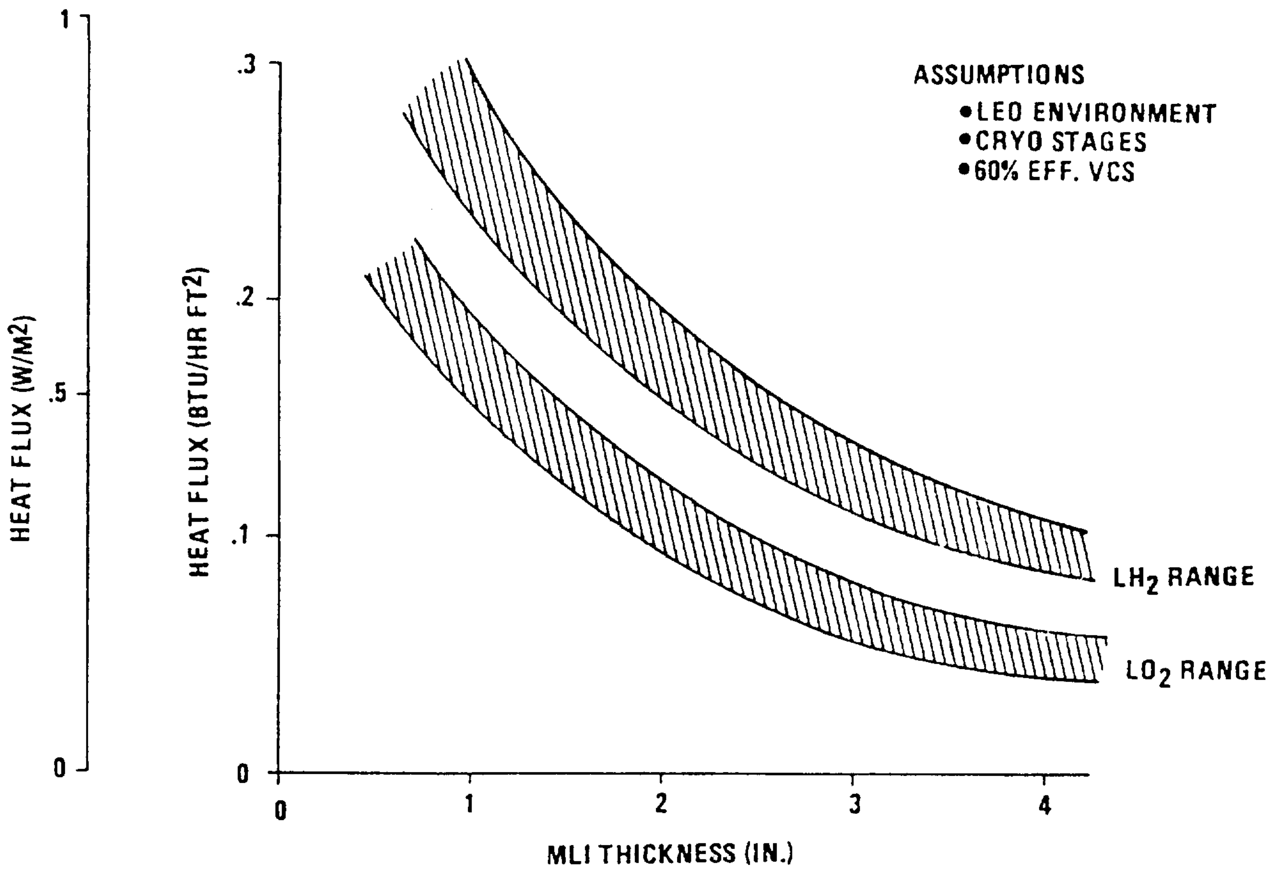

FIGURE 5. MARS TRANSIT VEHICLE HEAT FLUX VS. MLI THICKNESS

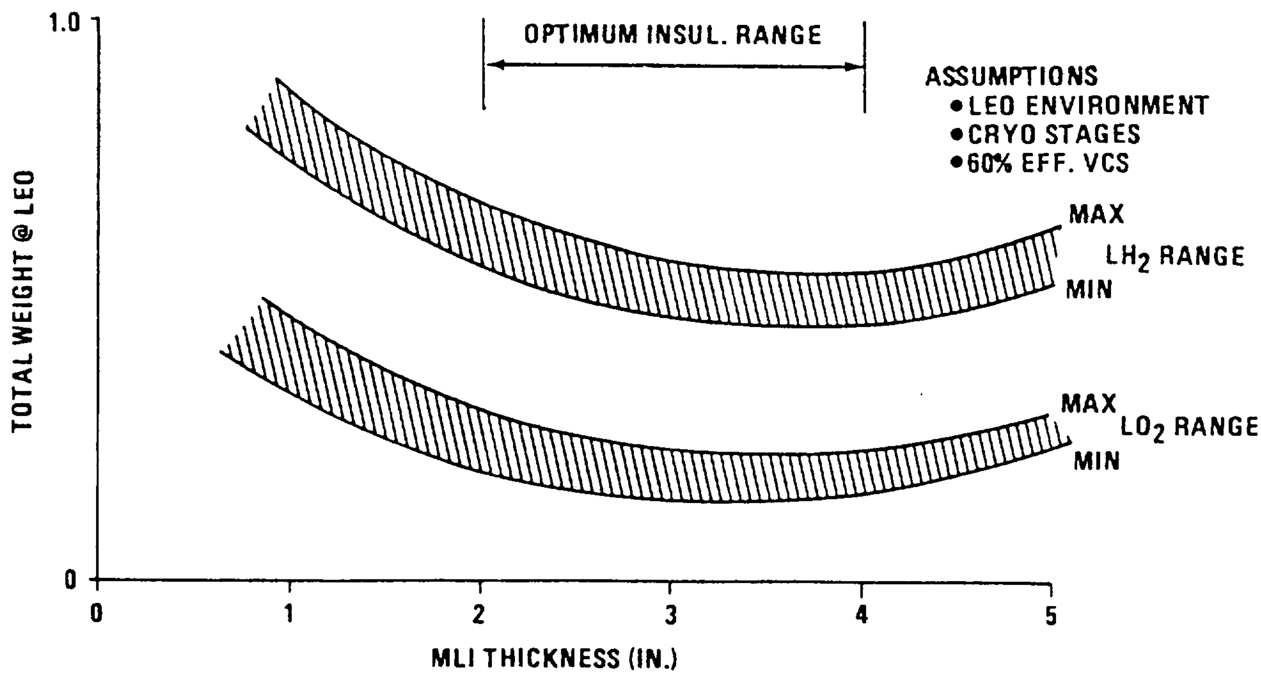

FIGURE 6. MARS TRANSIT VEHICLE TYPICAL INSULATION OPTIMIZATION

A third trade area is propellant transfer methods (mainly concerned with LEO operations). The option of zero-g versus induced artificial-g transfer should be evaluated. Artificial-g fluid transfer may be achieved by thruster firing, rotation of tethered vehicles, etc. Liquid acquisition devices for zero-g cryogenic fluids must be developed and tested. Systems trades should include various acquisition systems and fluid transfer systems.

TECHNOLOGY

Refrigeration system technology advancement could be required for utilization on Mars missions. Due to the vehicle tank sizes and duration in LEO, the anticipated refrigeration power input range could approach a megawatt (MW), using current efficiencies. The current capacities of refrigeration systems are well below the MW range. Additionally, support structure weight and volume restraints will place restrictions on any form of active thermal protection system. Another technical concern is the system's reliability during long term exposure. Factors affecting reliability and performance of active systems include the retention of the working fluid, the contamination of internal components, and the need for advanced electronic controls.

High insulation performance requirements are common to all mission phases. Current state-of-the-art multilayer insulation properties have been shown to degrade during long term exposure. Improvement in thermal protective coatings will be required for the Mars mission. Support structure heat leaks must be minimized using advanced composite materials or deployable struts.

Technology drivers in the fluids transfer area include; 1) minimal waste transfer, 2)advanced liquid acquisition devices, and 3) mass gauging techniques. Current estimated fluid transfer losses are less than 5 percent. Liquid acquisition systems must be developed and tested for use in a zero-g environment. Mass gauging of cryogenic propellants must be successfully demonstrated. In order to utilize a no vent transfer system, leak proof joints/fluid interfaces must be developed.

SUMMARY

The Mars mission transit vehicle TCS options include active and passive systems. Insulation performance, active TCS support structure requirements, and propellant transfer highlight the vehicle issues while in LEO. During vehicle transit and Mars orbit, reduction of environmental heating through preferred vehicle orientation is a key issue. Vehicle orientation may be affected by mission requirements, thus varying the environment the vehicle experiences. The transit vehicle trades are identified as active TCS utilization, insulation optimization, and propellant transfer methods. Advanced active TCS, advanced MLI and other passive TCS options, and fluid transfer systems for zero-g resupply are recommended for technology development.

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N87-17785

MARS VEHICLE TCS AND AEROBRAKE TPS

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ABSTRACT

General design approach, some unique problems and new technology needs for a Mars vehicle manned module Thermal Control System (TCS) and aerobrake Thermal Protection System (TPS) are discussed. The design approach of the TCS will be similar to that of the Space Station. Mars atmospheric dust storms are identified as an unknown that will impact the design of the Mars landing vehicle and base facility. New technology may be needed for thermal control surfaces to functionally survive the dust storms. The TPS for the Mars aerocapture vehicle will be subject to marginal stagnation heating rates for conjunction class missions and very high heating rates for opposition class missions. New technology TPS materials or an ablative heat shield will be required for the high stagnation heating rate trajectories. No significantly new technology is needed for the manned modules that do not descend to the Mars surface.

INTRODUCTION

The Mars vehicle Thermal Control System (TCS) will provide temperature control for the manned modules. This includes temperature and humidity control for the cabin atmosphere, temperature control for the support sub-systems (power and avionics), and temperature control for experiment equipment. This accomodation will be provided during the travel to and from Mars and during the time on the Mars surface.

Before launch, during those periods when the manned modules require cooling, ground support equipment will provide a thermal sink. Similarly, Earth orbit assembly and launch support facilities will provide this accommodation for the orbiting modules until the manned Mars vehicle departs from low Earth orbit.

The aerobrake Thermal Protection System (TPS) will provide protection for the vehicle structure against high aerodynamic heating rates during aerocapture maneuvers at Mars and at Earth.

During interplanetary travel and while at Mars, orbit TCS heat dissipation will be very efficient due to the cold space environment. This condition will set the TCS heat dissipation capability of the orbiting Mission Modules. The Mars Excursion Module (MEM) TCS will be driven by the warmer environment on the Mars surface. The Martian surface environment here will be variable due to the atmospheric dust storms. On clear days heat can be dissipated to both the Mars surface and deep space. Dust storms, however, will block the view to space interfering with radiative heat rejection. Dust particles encountered at high velocity may cause erosion of the shield as well. This will be the worst case design condition for the MEM.

An extended duration Mars base facility would of course be subjected to the same atmospheric condition as the MEM and would have similar heat rejection problems.

The design approach for the Mars vehicle manned modules would be practically the same as that for the Space Station. Both must have extended lifetime capability. The general approach is to provide dual heat transport loops with inflight maintenance capability for replacement of critical components. One probable difference would be design of the radiators. Whereas, on the Space Station, failed heat pipe panels are inflight replaceable, the Mars vehicle radiators would most likely be oversized so that needed heat rejection could be maintained with a number of failed heat pipe panels. This would reduce the inflight maintainance task.

Parametric studies conducted for the Space Station TCS radiators are shown on Figure 1. Radiator size is shown as a function of heat rejection for a deployed radiator, with radiator effective temperature as a parameter. Radiator size as a function of a 14 feet diameter module lenth for a body mounted radiator is also shown, again with effective radiator temperature as a parameter. Radiator sizes will be much smaller than this for the Mars vehicle Mission Modules because of the much colder thermal sink during interplanetary travel and in Mars orbit.

Another subject quite separate from the manned module thermal control is the aerobrake design for aerocapture maneuvers at Mars and Earth. Two categories of missions types are being considered for manned Mars missions: conjunction class and oppositon class. Entry interface speeds at Mars for conjunction class missions range from 17,700 ft/sec to 20,500 ft/sec and result in convective stagnation heating rates of 50 to 150 BTU/ft^2/sec. This results in a maximum convective stagnation heating

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SPACE STATION TCS
RADIATOR PERFORMANCE

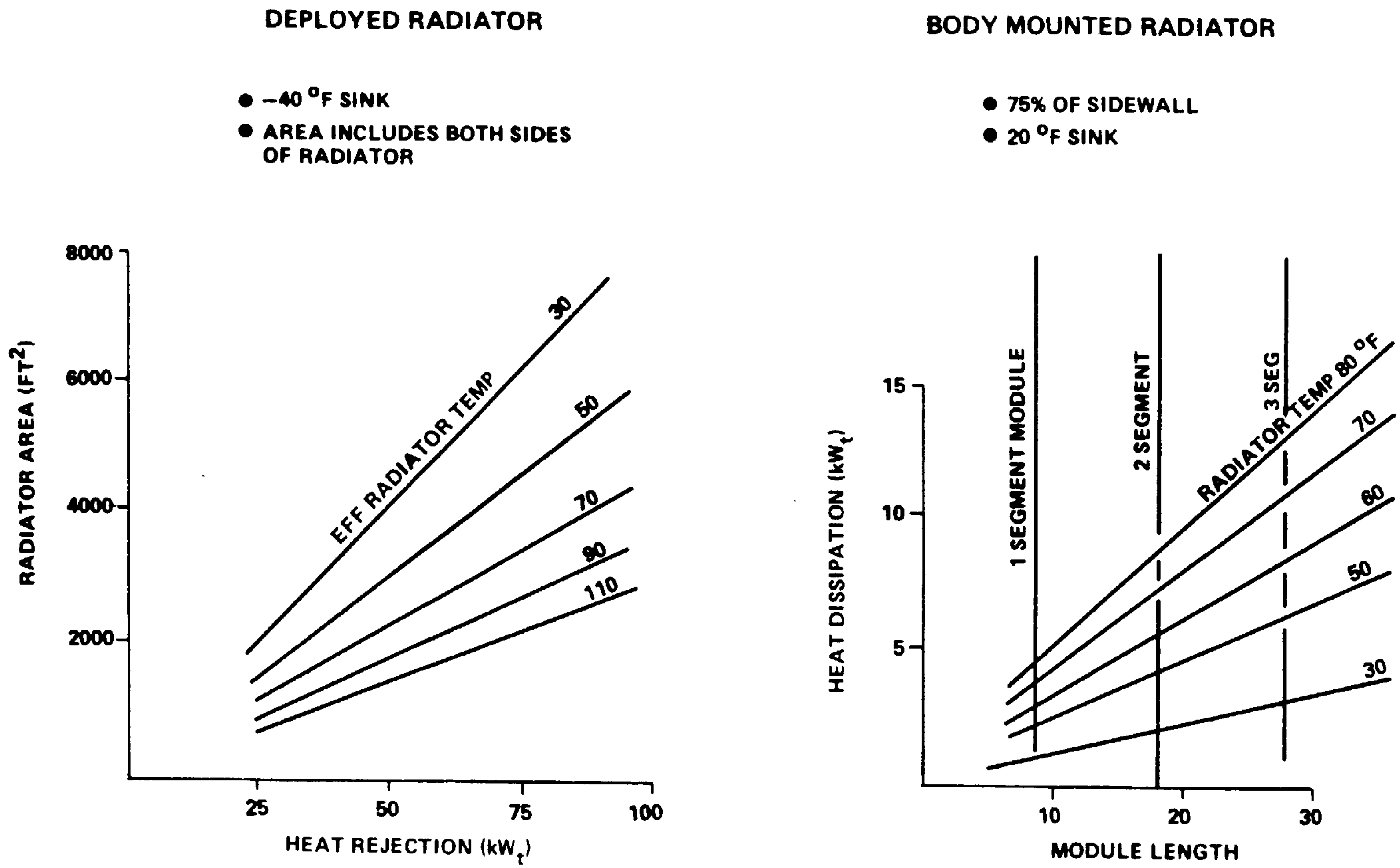

FIGURE 1

rate of 450 BTU/ft^2-sec. These rates are based upon a 1-foot radius sphere.* The TPS requirements for aerocapture maneuvers at Mars are highly mission dependent and should be designed accordingly. Shuttle type reusable TPS may be usable in the lower range but new technology TPS will be required for reusability at the higher end. Ablative heat shields are usable but impose a weight penalty.

The entry interface velocity at Earth for a free-return trajectory from Mars is 38,000 ft/sec and results in a maximum convective stagnation heating rate of 600 BTU/ft^2-sec. A propulsive maneuver near Earth can reduce the entry interface velocity to 33,7800 ft/sec resulting in a maximum convective stagnation heating rate of 390 BTU/ft^2-sec. Heating rates of this magnitude require ablative heat shields, which are heavy, or a new technology TPS.

The physical description or definition of the Mars atmosphere during dust storms is one the greatest unknowns we must deal with. This lack of definition requires that we keep open design options for heat dissipation by the MEM TCS. The most likely candidate for heat dissipation is the conventional fin tube type radiator (either pumped fluid or heat pipe). In the event this is not adequate due to a warm environmental sink condition, a flash evaporator might be used for heat dissipation. Design options for a Mars base facility would not likely include the flash evaporator because of the amount of water required for an extended duration facility. Options for the Mars base facility might include other concepts such as utilization of the soil as a heat sink.

Another concern on the Mars surface is the effect of the dust storms on thermal control sufaces. The abrasive effect will most likely degrade the surface coatings, affecting thermal performance of the radiator and affecting passive temperature control of the MEM structure. Options to deal with surface degradation include: (1) avoidance of solar heating loads by selective orientation of thermal control surfaces, and (2) development of surface maintenance techniques.

Selection of the appropriate option for MEM TCS heat dissipation will depend on the severity of the atmospheric dust storms. Trades could

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*An approximate rate for a shield of curvature Rm can be obtained by dividing by Rm.

be conducted with two objectives in mind. One study could be for the purpose of determining mission constraints (MEM landing and duration) based on severity of the dust storms. A second study could be for the pupose of defining a TCS that would be acceptable regardless of the dust storm severity.

TECHNOLOGY

Most of the on-going TCS technology is oriented toward efficient accommodation of long duration missions and is appropriate for the Mars mission. On-going TPS technology for atmospheric re-entry is appropriate for the needs of a Mars vehicle aerobrake and design goals are probably adequate for conjunction class missions. However, present design goals are not likely to be adequate for opposition class missions. Heat rates for opposition class Mars missions are expected to be considerably higher than those experienced by vehicles up to this time and new materials will most likely need to be developed.

The convective stagnation heating rates resulting from an aerocapture maneuver at Earth will require an ablative heat shield or new technology TPS. In addition to convective heating, radiative heating must also be considered. More detailed study is required to determine the total heating rates and the requirements for aerocapture maneuvers at Mars and at Earth.

Past studies on the subject of thermal coating contamination during space flight have indicated need for maintenance and have proposed at least two approaches to be pursued: (1) cleaning procedures, and (2) surface coating refurbishment or replacement. A new concern involving thermal coatings on Mars missions is the dust environment on the Mars surface. New technology efforts may be needed to deal with the subject of erosion and contamination of thermal coatings by Martian dust storms.

SUMMARY

The discussion presented here is intended to give some idea of the unique problems, general design approaches and technology needs for a Mars landing vehicle. Mars atmospheric dust storms are identified as an unknown that will probably impact design of the MEM and the Mars base facility. The design approach of the TCS will be similar to that of the

Space Station. The most outstanding new technology requirement may be new materials for the aerobrake TPS. No significantly new technology is needed for the manned module TCS.

N87-17786

NUCLEAR POWER SUPPLIES: THEIR POTENTIAL AND THE PRACTICAL PROBLEMS TO THEIR ACHIEVEMENT FOR SPACE MISSIONS

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ABSTRACT

This paper discusses various issues associated with getting technology development of nuclear power systems moving at a pace which will support the anticipated need for such systems in later years. The projected power needs of such advanced space elements as growth space stations and lunar and planetary vehicles and bases are addressed briefly, and the relevance of nuclear power systems is discussed. A brief history and status of U.S. nuclear reactor development is provided, and some of the problems (real and/or perceived) are dealt with briefly. Key areas on which development attention should be focused in the near future are identified, and a suggested approach is recommended to help accelerate the process.

BACKGROUND FOR A MANNED MARS MISSION: THE RELATIONSHIP OF POWER REQUIREMENTS TO OTHER SPACE POWER ACTIVITIES

The next major US program for the development of space infrastructure will be a manned space station scheduled to be operational in the early 1990's.[1,2] The space station program will provide for an expandable manned facility to conduct scientific and commercial activities. The facility will be serviced from the Earth by the shuttle.

Envisioned is an evolving facility, increasing in capacity, features, and capability to support research and development activities, materials processing, and other tasks which are, or may prove to be, cost beneficially performed in the unique environment of space microgravity and near vacuum. Too, the space station is anticipated to serve as a key element to the expansion of space exploration and utilization, including the possibilities of establishing a manned base on the Moon and missions to Mars. As this evolution of the space station use occurs, the power requirements of the station, including the accommodation of future scientific and commercial activities, can reasonably be expected to grow. The basic question is simply how best to meet these future requirements in a timely, cost effective fashion.

The initial space station power requirements are projected to amount to a connected load of about 200 kWe, with peak loads in the 100-200 kWe range, average loads in the 90-100 kWe range, and with station keeping requirements amounting to 25-35 kWe. These ranges provide for 6-8 person crews and include the best estimates for the experimental and/or commercial tasks.[3] The power source will be photovoltaics, the only available system capable of supplying this magnitude of power in the time frame desired. The magnitude of these power loads strains the applicability of photovoltaic systems--the systems simply become very large, constitute a real design concern, and are a serious constraint to future space activity expansion possibilities.

The second generation space station, although ill-defined at this time, is projected to require power levels in the range of 200-300 kWe. If space activities expand as might reasonably be expected, this modest growth in power requirements may prove to be seriously underestimated, particularly if commercial processing of materials proves viable. It would be unfortunate indeed if the horizons of space station activities were to become constrained by the lack of a suitable power source.

Establishing a manned base on the Moon or visiting Mars makes the power supply question even more serious. Both the Moon and Mars have day/night cycles to contend with and the solar source strength decreases with distance from the Sun. For the long-duration Mars mission, the choice of feasible power sources is very restricted. The power requirements for either of these two missions are unknown because the planning is in such an early state, but life-support and in-transit experimentation can be expected to be comparable to that of the space station, i.e., up to a few hundred kilowatts. If, in addition, nuclear electric propulsion is used, we might reasonably expect levels up to a few thousands of kilowatts.

The anticipated growth of the space station power requirements noted above provides a good example of the problem the space nuclear power supply developers have to contend with: should a reactor power supply be developed that attempts to be all things to all missions, i.e., is highly flexible in its ability to meet a wide variety of missions, or should the development of a reactor system await a specific mission definition and be customized to this mission? This leads, of course, to a chicken-and-egg situation which will be addressed subsequently. Suffice it to say that, for power requirements of several hundreds of kilowatts or more, no nuclear power source exists or is even far enough along in the definition stage (much less the development stage) for NASA to reasonably assume probable availability within the next 10 years.

STATUS OF SPACE NUCLEAR REACTOR DEVELOPMENT

The history and status of space nuclear reactor development can be summarized quickly. In 1965 the US launched the SNAP 10A reactor system, a nominal 0.5 kWe power plant consisting of a zirconium hydride moderated, fully enriched uranium core, a beryllium reflector which contained control drums for startup in orbit, and a thermoelectric power conversion system mounted on a waste heat rejection conical radiator. The reactor was liquid metal cooled with pumping of the liquid metal by a electromagnetic pump. The power system was successfully launched into Earth orbit, started up, and operated for over 40 days with a performance level as designed and in accordance with an identical and simultaneously operated ground based system. Operation ceased upon the failure (suspected) of a voltage regulator.

Further launchings were postponed because of the lack of missions requiring the particular capabilities of SNAP 10A and because of the successful development of solar electric systems. The fact of the matter is that for the NASA or DoD applications of that era, SNAP 10A was unable to compete in a cost effective manner.

The development of the SNAP 10A and the associated engineering phases conducted for the 3 kWe SNAP 2, the 30 kWe SNAP 8, and the 50-100 kWe thermionic reactor systems resulted in a considerable amount of space-specific reactor system technology. These activities took place during the period from the late 50s to 1972, and the technologies included dynamic (turbine) power conversion systems, thermionic and thermoelectric conversion, shadow shielding, development of system and sub-system analysis techniques, compact reactor design and analysis, related launch and orbital safety aspects. In addition, an active program to develop a nuclear rocket for space propulsion proceeded through a relatively extensive experimental phase. These programs were all terminated at year's end in 1972.

The US renewed its' interest in space reactor power in 1981 with a modest study activity that has become known as the SP-100 program. The first objective of this program has been to reevaluate the competing technologies which might be capable of providing 100 kWe for periods up to 7 years in duration. To date the old technologies have all been revisited, evaluated in light of advances during the intervening years, and reduced to four candidate power conversion systems: in-core thermionics, Brayton cycle (expansion of a hot gas through a turbine), Stirling cycle (a reciprocating piston engine), and thermoelectrics. Selection of one of these systems took place in July 1985 and a Request for Proposal for the selected system development is to be issued.

A second space reactor program has only recently been initiated to develop reactor power systems for the Strategic Defense Initiatives program. This so-called multimegawatt program encompasses systems beyond the capability of the SP-100 project and includes nuclear propulsion as well. The program is classified and consequently its status cannot be presented here.

RESEARCH AND DEVELOPMENT REQUIREMENTS

The achievement of a successful space reactor program will not occur overnight. Considerable applied research and engineering development is required to advance beyond the achievements of the 1960's. This R&D encompasses many technology areas, will require the services of our "best and brightest," and will be expensive and time-consuming. The key areas and potential solutions are already identified, so the problem is not that of fundamental technical feasibility, but rather of conducting the systematic studies, experiments, breadboard testing, component prototype development and demonstration, and finally of building flight-qualified hardware.

Here, only a few key R&D areas are touched on, with a few examples to provide an indication of the types of issues being addressed (or projected to be within the immediate future).

It should be emphasized from the outset that the overriding consideration for the development of a space reactor power source is that the system must be safe. Safe means safe in every aspect: from construction and checkout testing, through delivery to the launch site, during launch and startup, throughout its useful life, and after shutdown. This applies to the crew, support personnel, and the general public. Every aspect of the R&D is guided by this consideration. A safety record at least comparable to that of commercial power reactor program, which by any measure is unparalleled in the technologies of power generation, can and must be achieved.

For the reactor power source, the key areas requiring attention are fuels which: (1) can accommodate requirements in the event of a launch failure under worst-possible scenarios (e.g., immersion), and (2) will provide long life through the accommodation of fission products and the use of burnable poisons; mechanical/neutronic control systems which provide acceptable margins of safety and redundancy under all situations and which will operate reliably and accurately for long periods; cooling systems which will not introduce additional safety complexity, constitute a single point of failure, vulnerability, or require excessive power for achieving their function of maintaining system temperatures at design levels; neutronic shields which are light-weight and stable with regard to radiation damage; and overall reactor systems that are reliable in their operation and flexible with regard to their operational power level and longevity or which can accommodate changes in these parameters with a minimum of difficulty.

The general objectives for power conversion systems (PCS) follow the same line of thought: they must not constitute a compromise to safety at any phase of the mission, they must operate reliably for long periods under essentially maintenance-free conditions, they must be adaptable to a wide variety of mission-specific conditions and requirements, and they must be capable of power level growth with a minimum of system modification or flight requalification. These will be the governing principles, whether the PCS is a static (thermoelectric, thermionic), or dynamic (Brayton, Rankine or Stirling cycle) system.

As adjuncts to the PCS, electrical transmission to the payloads, power conditioning, and system control also require considerable attention. These areas are equally crucial to achieving attractive reactor-based power systems but are not considered to constitute limitations to feasibility. Power transmission includes laser, tether, or microwave options (allowing separation between the reactor power source and radiation-sensitive areas), all of which have appealing features on paper but which also have significant problems to their practical realization. Highly efficient power conditioning, capable of accommodating a wide variety of operational conditions and high (for space) combinations of currents and voltages, will be required for the power levels of interest. System controls will likely be required to incorporate artificial intelligence in the form of a real-time "expert system", combining the desirable features of sophisticated process control with systems which will automatically detect, analyze, and correct disturbances using on-line fault-free analysis, faster-than-real-time predictions, and automated decision making.

In space, waste heat rejection can be accommodated by thermal radiation or very specific applications possibly by a mass loss means. Development of light-weight materials for radiators, geometrical configurations which change according to the amount of heat being rejected, and mass ejection devices are all under consideration, but the practical realization of any of the options at higher power levels is far from being clearly defined. Complicating the development is the strong interdependence between the amount of heat to be rejected and the design features of the rest of the system, including the payload requirements.

To repeat, these development requirements cannot be resolved overnight and will not be resolved within a time frame useful for a lunar base or Mars mission unless a broad and concerted technology development effort is soon undertaken. If the US is serious about establishing a meaningful space infrastructure beyond the space station and in making significant technological advances which unquestionably will have consequences far beyond just the space mission applications, then this commitment to development must be pursued. NASA must take initiative for this to happen.

THE REAL PROBLEMS OF DEVELOPING SPACE NUCLEAR REACTOR SYSTEMS

As NASA well knows and has experienced, the real problems of developing and delivering technologically advanced systems do not tend to be only technical in nature, but rather political and budgetary. The Manhattan and Apollo projects are good examples of the importance of these factors. Nuclear programs, in addition to these "normal" factors, must also contend with a further complexity: the perceived "safety" of anything having to do with nuclear-derived energy. As the civilian power program in this and other countries have amply demonstrated, public participation in matters nuclear have become an accepted way of life. The nuclear community is well aware of this fact of life, and hence the previously stressed emphasis on ensuring safety through all phases of a nuclear system's lifetime.

Fortunately, the use of nuclear power plants in space for NASA missions will likely involve little general public concern, provided the launch safety can be shown not to entail a radiation risk (avoided by system startup only after a safe orbit has been achieved) and the systems are not used in low Earth orbit, where a repetition of the Cosmos 954 reentry could be perceived. Technically, these safety issues can be or are resolved. More likely, the misconceptions and safety concerns will come from the higher echelons of NASA who are in politically sensitive or vunerable positions and who will be on the firing line with the public, with Congress, and the the Executive Branch. It will be important for these persons to be knowledgeable about the nuclear aspects and the detailed attention accorded to the safety issue. Those within the nuclear community are confident that, properly handled, the safety issue can be a non-issue, but it will require careful communication between DOE and NASA to ensure that this happens, and that public sensitivity receives proper attention.

The real problem of space nuclear power is what was previously referred to as the "chicken-and-egg" syndrome: DOE will not develop a space reactor system for NASA without a firm mission, and NASA will not specify a firm mission requiring a space reactor because such a system doesn't exist and development is perceived not to be feasible within the time frame of the mission. The problem is how to break this cycle.

The SP-100 program has taken an important first step to breaking this cycle, but this program is too design-specific to achieve a broad technology base necessary to provide latitude in achievable power level.

In contrast to the SP-100 approach, a wider perspective is needed to facilitate the development of the technologies required to provide the latitude in power levels, the desired power levels can be broken into ranges, say, from 100 kWe to 1000 kWe, and from 1000 kWe to 10,000 kWe or greater. The various technologies will require careful evaluation through meaningful developmental tests, in and during which evolutionary improvements are incorporated in order to determine their true potential with regard to operational longevity. It is vitally important that this recommended approach be recognized for what it is: a broad technology program, not a mission-specific one! The technology development goals cannot be dictated by a mission schedule, but must instead be based on desired levels of performance.

There are several recognized difficulties with this approach, of course. It doesn't have any public appeal or the pizazz of an Apollo or Mars mission. It will probably not be perceived as a bold, imaginative program (even though, in reality, it will be if pursued appropriately), so that strong Congressional support may be difficult to obtain. A lot of blind alleys will inevitably be pursued. It will be a politically vulnerable program since it may not be identified with a specific large-scale programmatic goal (the same argument advanced in the congressional evaluation of the space station mission versus consideration of a large number of smaller activities).[1]

If NASA foresees the eventual desirability of being unconstrained by power availability in their planning of future missions, they are going to have to take the initiative to break this cycle. Needless to say, the entire energy-related community would welcome such an initiative, since these developments would inevitably benefit terrestrial power generation as well.

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N87-17787

PEGASUS: A MULTI-MEGAWATT NUCLEAR ELECTRIC PROPULSION SYSTEM

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ABSTRACT

A propulsion system (PEGASUS) consisting of an electric thruster driven by a multimegawatt nuclear power system is proposed for a manned Mars mission. Magnetoplasmadynamic and mercury-ion thrusters are considered, based on a mission profile containing a 510-day burn time (for a mission time of approximately 1000 days). Both thrusters are capable of meeting the mission parameters. Electric propulsion systems have significant advantages over chemical systems, because of high specific impulse, lower propellant requirements, and lower system mass.

The power for the PEGASUS system is supplied by a boiling liquid-metal fast reactor. The power system consists of the reactor, reactor shielding, power conditioning subsystems, and heat rejection subsystems. It is capable of providing a maximum of 8.5 megawatts of electrical power of which 6 megawatts is needed for the thruster system, leaving 1.5 megawatts available for inflight mission applications.

INTRODUCTION

With the Space Transportation System (STS), the advent of space station Columbus and the development of expertise at working in space that this will entail, the gateway is open for missions to Mars. The missions are possible with state-of-the-art hydrogen/oxygen propulsion, but would be greatly enhanced by the higher specific impulse of electric propulsion. This paper presents a concept that uses a multi-megawatt nuclear power plant to drive an electric propulsion system. The concept has been named PEGASUS, PowEr GenerAting System for Use in Space, and is intended as a "work horse" for general space transportation needs, both long- and short-haul missions.

The advantages of electric propulsion are well-known in the aerospace community. But this high specific impulse, propellant-saving, potentially cost-saving, and mission-enabling technology has not received serious consideration for spacecraft propulsion, primarily because of the lack of a suitable, light-weight electric power source. The recent efforts of the SP-100 program indicate that a power system capable of producing upwards of 1 megawatt of electric power should be available in the next decade. Of greater interest are efforts in other areas which indicate that a power system with a constant power capability an order of magnitude greater could be available near the turn of the century. With the advances expected in megawatt-class space power systems, the high specific impulse propulsion systems, such as a magnetoplasmadynamic (MPD) or ion propulsion system, powered by a nuclear electric power plant, must be reconsidered as potential propulsion systems. A conceptual drawing of a manned Mars spacecraft powered by the PEGASUS drive is shown in Figure 1. The electrical power for the propulsion system is provided by a nuclear multi-megawatt fast reactor space power system. This power system is capable of meeting both the propulsion system and spacecraft power requirements. The size and mass limitations of the STS are a prime consideration in the design, and the collapsed system will be capable of lower earth orbit insertion by two shuttle missions. Development of this power system could be completed by the mid 1990's and the system available near the turn of the century. Since this is an advanced system concept, some development efforts are still needed in the fuels, heat rejection, and turbo-alternator areas.

MISSION ANALYSIS

Any number of mission profiles may be devised for a manned Mars expedition, depending on the assumptions made concerning desired objectives and available technology. This paper uses a previously developed set of mission parameters (Ref. 1) as a basis for design. However, the propulsion system concept is adaptable to a wide range of mission profiles. The parameters defining this Mars mission consist of a 344 metric ton initial mass vehicle in Low Earth Orbit (LEO) which autonomously raises itself through the radiation belt to Geosynchronous Earth Orbit (GEO), whereupon a crew of three boards. The trip to Mars takes 601

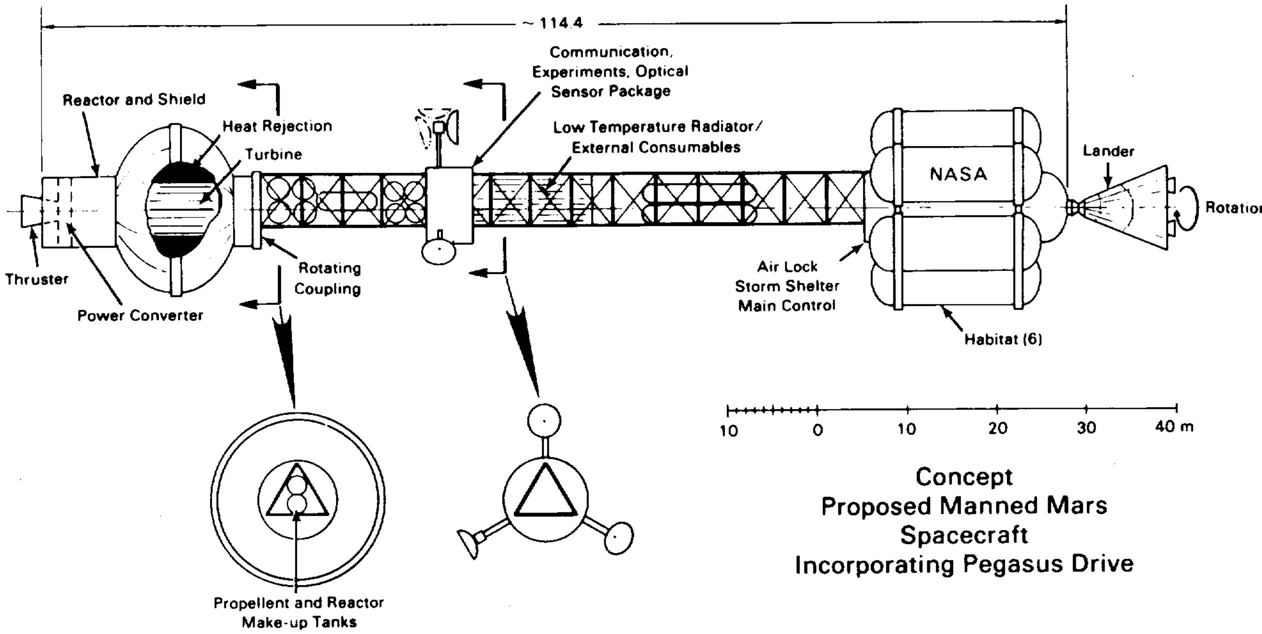


FIGURE 1. CONCEPT OF PROPOSED MANNED MARS SPACECRAFT WITH PEGASUS DRIVE

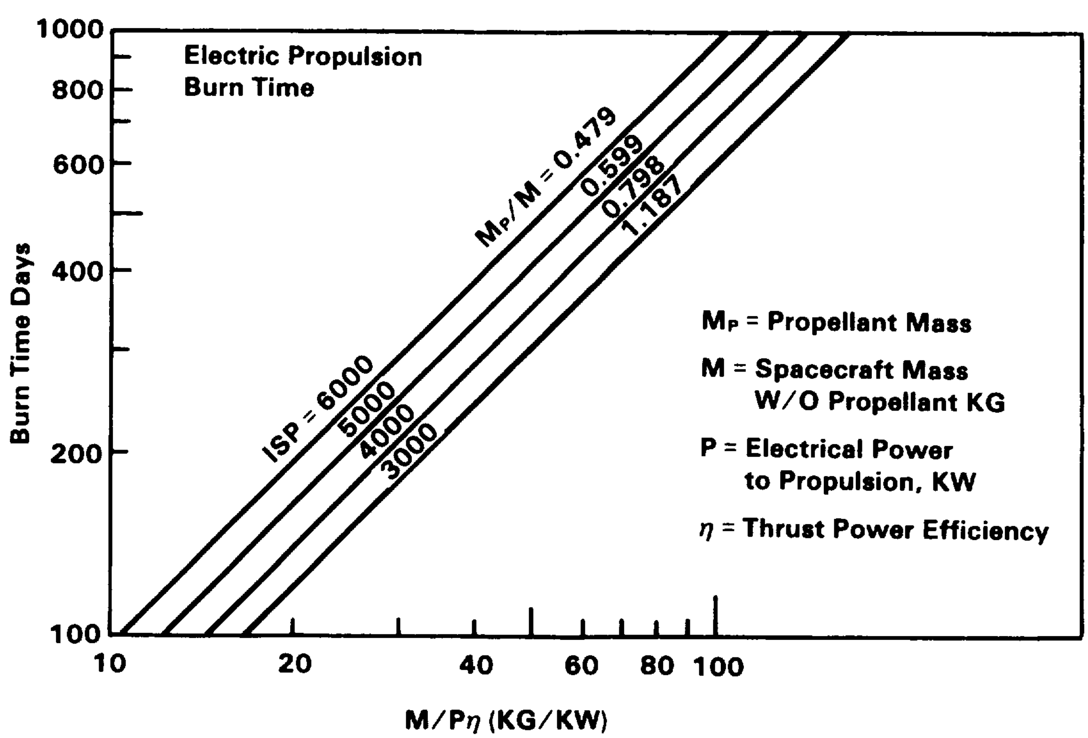


FIGURE 2. ELECTRIC PROPULSION BURN TIME

days, exploration 100 days, and the return trip 268 days. Propellant consumed is 111 metric tons with a 5% reserve. The return mass to GEO is 103 tons, 76 tons having been left on Mars. The allowance for power and propulsion system was 34 tons, including 11 tons for tankage.

Figure 2 shows how vehicle mass M (excluding only propellant) can be traded against power P, overall efficiency, eta, and burn time, t, to provide a desired specific impulse. Improvements to the propulsion system would be reflected in higher values of eta, and have the same effect as reductions in vehicle mass M in reducing the required burn time. Actual trip time for a mission is a function of many variables, but it is clear from Figure 2 that the more power developed by the propulsion system (all other things being approximately equal), the shorter the trip time. The time saving can be accomplished either by the shorter burn times required to produce the same velocities as projected for the chemical-fueled system, or by longer burn times that produce higher velocities in the coast phase. Such trade-offs in power, burn time, and coast time would have to be studied extensively to determine the optimum combination, but it is clear that electric propulsion systems are greatly mission-enhancing in this regard.

Electric propulsion also shows an advantage for this mission in the modest propellant requirements as compared to chemical systems. While the scope of this paper excludes a detailed comparison, the propellant required is characterized by the specific impulse of the propulsion system. Since electric engines have a specific impulse that is 10-20 times that of hydrogen-oxygen chemical systems, the propellant required for electric systems can be from 1/3 to 1/10 that of chemical propellant.

PROPULSION SYSTEM

Electric propulsion systems have not been seriously considered for use with large spacecraft due to the lack of a suitable electric power source to drive them. However, recent efforts to develop megawatt-class space power sources show such systems to be technologically feasible, and a multi-megawatt lightweight nuclear electric propulsion system would enable missions of almost any conceivable duration and scope. The manned Mars mission would be well within the capabilities of such a system.

The electric propulsion could be provided by a magnetoplasmadynamic (MPD) or ion propulsion system. Section 3.1 discusses MPD thrusters as part of such a system. Section 3.2 discusses ion thrusters for this concept. The design of the nuclear power plant that provides the required power is outlined in Section 3.3.

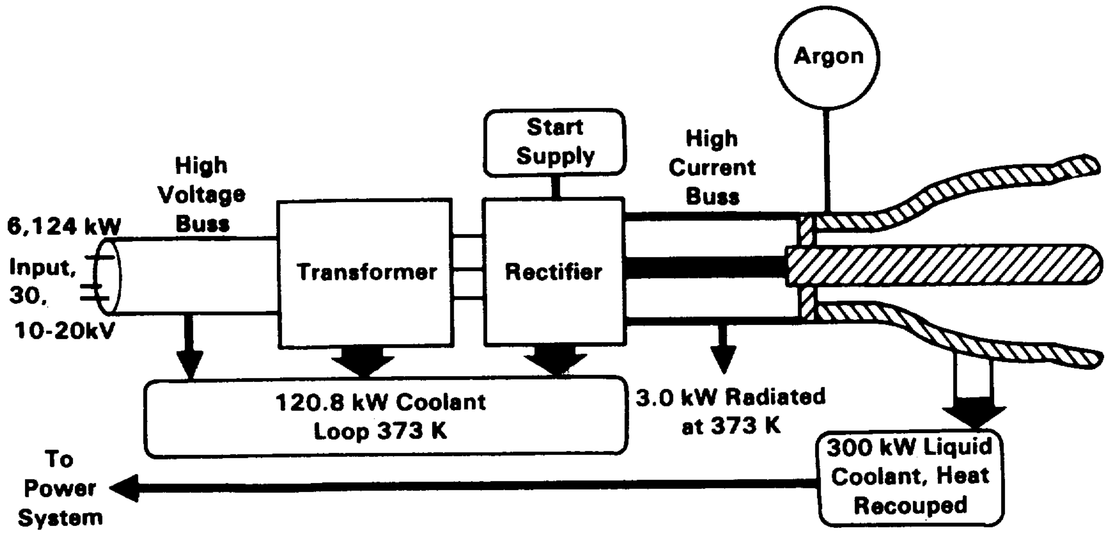
Magnetoplasmadynamic Thrust System

The MPD system is composed of thrusters, propellant tanks, and thermal control subsystems. A schematic of the major components of the MPD thruster system is shown in Figure 3. Three phase ac at 10-20 kV and 1500 Hz is fed through a transformer and rectifier to the MPD thruster. Power control and regulation is confined to adjustments of the ac output from the alternator. The high tension buss runs from near the turbo-alternator past the reactor shield to the transformer and thruster. This 15 meter length of cable is insulated and electrically shielded to reduce EMI, and masses amount to 69 kg.

The transformer mass has been estimated from design equations and weighs only 1911 kg (Ref. 2). (Efficiency of 99% is a reasonable design goal for this delta-delta transformer.) The rectifier is designed to use 3,000-ampere diodes derated to 1,500 amperes. Eight diodes are connected in parallel to handle 12,000 amperes. Six units are required for full wave rectification of three phase AC. The 48 diodes and cooling assembly are estimated to have a total mass of 200 kg, and will dissipate 59.5 kW when rectifying 25,000 ADC. The transformer, rectifier, and high tension buss require 120.8 kW of heat to be removed by a cooling plant associated with the cryogenic superconducting alternator.

The current buss to the MPD thruster is designed to be self-radiating at 373 K; it has a mass of 817 kg, which includes 0.5 cm of insulation between the coaxial conductors and on the outside. An alternative that may reduce overall mass at the cost of complexity is to reduce the cross-section of aluminum, which increases the dissipation and requires active cooling. Also, 3 meters were allowed to electrically connect the MPD thruster, which operates at about 1,600 K, and the rectifier assembly which is designed for 373 K, thus providing thermal isolation.

The MPD thruster assembly consists of seven engines which are used one at a time, for a 510-day burn. Based upon ongoing work at JPL to evaluate thruster lifetime with subscale devices, and the present understanding of cathode physics, thruster lifetime (which is limited by the



Subsystem Specific Mass 1.066 kW/kG

Overall Efficiency 0.4984 = (0.9811) (0.508) (Power) (MPD)

FIGURE 3. MPD THRUSTER SYSTEM COMPONENTS

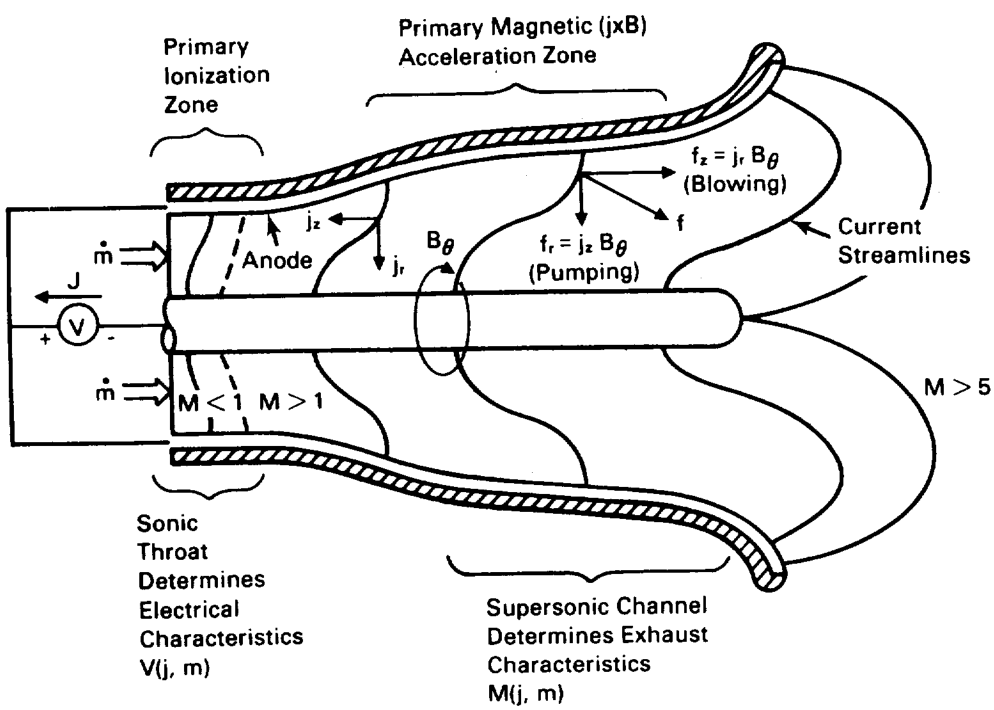


FIGURE 4. MPD THRUSTER

cathode) is estimated to be 2,000 hours. But life test of a multimegawatt MPD thruster can only follow a vigorous development program, neither of which are presently planned in NASA's research program.

Figure 4 shows a schematic of the MPD thruster which accelerates propellant by means of a magnetic body force. Propellant is injected through the rear insulator and is ionized by a high-current, diffuse discharge. The large multi-kiloampere current creates an azimuthal magnetic field which interacts with the current to produce a magnetic body force directly on the ionized gas. Exhaust speeds of 15,000 to 80,000 m/s have been measured on a thrust stand using a variety of propellants (Ref. 3). Thruster conversion efficiency of electrical power to directed kinetic energy has been measured at 35% on a thrust stand and analysis suggests that 50-60% is ultimately possible (Refs. 3,4).

All seven engines are connected in parallel to the high-current buss, but each has a separate propellant valve and a contactor on the cathode current feed. Though performance development may change the electrode shape somewhat, the overall dimensions of a multimegawatt thruster will remain about the same (Ref. 5). Each engine is assumed to have a center-body cathode 3 cm in diameter and 10 cm long, with an anode about 12 cm inside diameter, and of a comparable length. The most massive part of the MPD engine is the anode heat removal system, which must remove approximately 300 kW. An allowance of 700 kg is made for the assembly of seven engines, contactors, and thermal control.

The MPD thruster is started by application of 500 V and a low propellant flow which results in a glow discharge. As the power in this glow discharge is increased to 2-5 kW, the cathode is heated to incandescent temperatures. High temperature results in prodigious thermionic cathode emission and the arc jumps to a low impedance arc mode. The alternator output can then be ramped up, along with the propellant flow rate, to full power. The MPD thruster exhaust velocity and thrust may be throttled down to half of nominal by adjusting the mass flow and power.

A summary of the mass breakdown and power consumption is shown in Table 1. A contingency of 10% of the system mass is included, and 20% on top of that is added for structure. The net propulsion system mass, excluding power and the cooling plant, is 5,144 kg. The electrical system is 98% efficient, while the MPD thruster is assumed to have a 50%

TABLE 1: MANNED MISSION TO MARS PROPULSION SYSTEM SUMMARY

ITEM	MASS	POWER DISSIPATED
MPD THRUSTER ASSEMBLY - 7 engines required for 510 day burn - 2000 hr life per engine assumed - 240 VDC, 25,000 ampere input - 5,000 S specific impulse - 50% thrust efficiency - 2.5 G/S propellant flow rate	700 kg	6,000.0 kW
ENGINE STARTUP AND CONTROL SUBSYSTEM	50 kg	
HIGH CURRENT BUSS - 25,000 amp insulated coaxial conductor - 0.36 M dia by 3 M length - self-radiating at 373 K	817 kg	3.0 kW
RECTIFIER - 48 diodes, 3,000 amp rating each - heat sink requires active cooling at 373 K	200 kg	59.5 kW
TRANSFORMER - 6,123 kW, 1,500 HZ, 3 phase, 10-20 KV input - 99% efficiency - 0.5 M^3 volume	1,911 kg	60.0 kW
HIGH VOLTAGE BUSS - 10-20 KV, 3 phase, 500 A rating - insulated, shielded - self-radiating at 373 K - 10 cm dia. by 50 M long	69 kg	1.3 kW
THERMAL CONTROL		
- liquid coolant to MPD anode, recoups 300 kW at up to 1,600 K	100 kg	
- coolant to diodes and transformer and high voltage buss	50 kg	
Subsystem Total	3,897 kg	6,123.8 kW
Contingency Mass 10%	390 kg	
Structural Mass 20%	857 kg	
Propulsion System Total Mass	5,144 kg	
Specific Mass	0.840 kg/kW	
Power Processing Electrical Efficiency	98.0 %	

conversion efficiency. An interesting point on the waste heat of the MPD thruster is that since the engine operates at a high 1,600 K temperature, the 300 kW of heat rejected can be recouped to the advantage of the power conversion system.

The MPD thruster has been studied extensively with argon propellant, and the range of specific impulse available with this propellant is well suited to this mission. Cryogenic storage of argon has been considered in detail elsewhere, and presents no unusual constraints or problems. A 16,301 kg storage tank has been designed, with a mass of 721 kg (Ref. 6). This tank is designed as Shuttle-compatible, and operates in an environment of 293 K, while allowing only 0.14 of heat energy to leak into the fluid. Additional heat is needed to drive off gas at 0.0025 g/s at 2 atm for the MPD thruster. Seven of these tanks are required to store 111,000 kg of propellant for the mission for a total tankage mass of 5,047 kg.

Potassium, which is the working fluid in the dynamic conversion equipment, has nearly the same atomic mass as, and one more proton than, argon. Since it's ionization potential is about one-fourth of argon, potassium should provide a higher thrust efficiency than argon. Potassium offers a simpler storage problem in that it can be stored in a solid form and then liquified at 62 C. Since the density of solid potassium is comparable to liquid argon, and mercury tanks for ion propulsion have a tank fraction comparable to 4.4% for liquid argon, this study will assume that the tank mass for potassium will be the same as for argon.

There is a concern that the small fraction of the exhaust that flows back toward the spacecraft may plate out on many surfaces after 111 metric tons of potassium are exhausted. Though potassium may be a better propellant from a performance and storage standpoint, contamination of the spacecraft surfaces may require argon propellant.

Ion Thrust System

Thrust system characteristics for the Mars mission trajectory are presented in Table 2 for 100 cm, 50 cm, and 30 cm ion thrusters for both mercury and xenon propellant. The J-series 30 cm ion thruster served as the baseline for this study, since it has undergone extensive design evolution over the past fifteen years (Ref. 7) and has been operated over a full range of high performance levels necessary to support future

TABLE 2: THRUST SYSTEM CHARACTERISTICS

	100 CM ION THRUSTER						50 CM ION THRUSTER						30 CM ION THRUSTER					
	CASE I			CASE II			CASE III			CASE IV			CASE V			CASE VI		
MISSION PARAMETERS:																		
Velocity increment (n/s)	22,600																	
Thrust time (days)	510																	
Final mass (kg)	103,000																	
Specific impulse (S)	4665	5390	6030	5770	6600	7400	4665	5390	6030	5770	6600	7400	4665	5390	6030	5770	6600	7400
Thrust Module:																		
propellant	4g	4g	4g	Xe	Xe	Xe	4g	4g	4g	Xe	Xe	Xe	4g	4g	4g	Xe	Xe	Xe
total accel. voltage (V)	3000	4000	5000	3000	4000	5000	3000	4000	5000	3000	4000	5000	3000	4000	5000	3000	4000	5000
number of thrustors *	6	3	2	5	3	2	21	6	5	20	9	5	57	25	14	54	24	13
total mass (kg)	9977	10250	11018	11105	11960	10236	11104	10581	10820	12595	12298	12701	13295	11857	11762	14752	13493	13478
input power (kw)	2181	2396	2594	2563	2816	3067	2183	2397	2594	2565	2817	3068	2188	2399	2595	2569	2819	3069
Interface Module:																		
propellant mass (kg)	65801	54950	47942	50564	43042	57632	65801	54950	47942	50564	43042	37632	65801	54950	47942	50564	43042	37632
tank mass (kg)**	398	327	282	298	250	217	398	327	282	298	250	217	399	327	282	298	250	217
total mass (kg)	70013	58799	51709	54182	46579	11223	10019	58692	51448	54253	46445	40902	70122	58754	51496	54349	46486	40506
input power (kw)	2192	2408	2607	2576	2830	3082	2194	2409	2607	2578	2831	3083	2200	2411	2608	2582	2833	3084
Transmission Line:																		
mass (kg)	3830	4207	4554	4500	4945	5385	3833	4209	4554	4504	4946	5386	3841	4212	4556	4511	4949	5388
dissipated power (kw)	191	210	228	225	247	269	192	210	228	225	247	269	192	211	228	226	247	269
OVERALL Thrust System:																		
dry mass (kg)	18018	18307	19338	19224	20442	21931	19155	18532	18880	20787	20640	21357	21456	19874	19872	23048	21886	22140
total mass (kg)	23820	73256	67281	69788	63484	59563	84957	73482	66822	71351	63690	50989	87257	74823	67815	73612	64528	59772
input power (kw)	2373	2607	2822	2788	3064	3337	2375	2608	2822	2790	3064	3337	2380	2610	2823	2795	3066	3338
radiator area (n^2)	31	34	37	37	40	44	32	35	37	37	41	44	33	35	38	39	41	44
specific impuse (S)	4665	5390	6030	5770	6600	7400	4665	5390	6030	5770	6600	7400	4665	5390	6030	5770	6600	7400
thrust (N)	68.5	66.1	64.5	65.1	63.4	62.1	68.5	66.1	64.5	65.1	63.4	62.1	68.5	66.1	64.5	65.1	63.4	62.1

* No thruster redundancy included

** Propellant storage was assumed to be noncryogenic pressurized spherical tanks planetary missions. The design-level performance of the 30 cm J-series thruster is shown in Table 3. It has a lifetime design goal of 15,000 hours (demonstrated 10,000 hr and projected in excess of 25,000 hours (Ref. 8)) at the design-level operating conditions.

Figure 5 shows a cross-section of the 30 cm J-series mercury ion thruster. The thruster consists of a cathode, an anode, a cylindrical discharge chamber, an arrangement of magnets in the discharge chamber, and a set of closely-spaced grids downstream of the cathode and propellant injection. Thrust is developed when electrons passing through the grid system form an ion beam accelerated to high velocities by a large electric field between the screen and acceleration grid. Electrons injected into the ion beyond the extraction system neutralize the current to prevent charge buildup on the spacecraft and beam divergence. The thrust levels are not comparable to those of chemical rockets, but the extremely high exhaust velocities permit large characteristic-velocity missions to be accomplished.

Some component development will be required to produce advanced thrusters for planetary missions, particularly for the 50 and 100 cm thrusters. Present magnetic field and chamber designs may require modification to reduce losses. Operation of the accelerator system at voltages above 3,000 V will require the development of advanced grid materials and fabrication procedures (Ref. 9). The present J-series cathodes are designed for operation up to 20 amperes emission current, but the systems presented in Table 2 would require currents ranging from a low of 80 amperes for a 30 cm ion thruster (ISP=4665) to a high of 2800 amperes for a 100 cm ion thruster (ISP=7400).

NUCLEAR ELECTRIC POWER SOURCE

The proposed power source for the electric propulsion system is PEGASUS, an 8.5 MWe boiling liquid-metal, space-based nuclear power system. The system employs a direct Rankine power cycle and is designed to meet the power requirements of 6 MWe for the electric propulsion system with an additional 1.0 to 1.5 MWe available for mission-specific tasks and experiments.

PEGASUS comprises five major subsystems and components. These are a cermet fueled, boiling liquid metal fast reactor; a four-pi contoured

TABLE 3: 30 CM J-SERIES THRUSTER PARAMETERS

PERFORMANCE	DESIGN-LEVEL DEMONSTRATED PERFORMANCE	DEMONSTRATED EXTENDED PERFORMANCE	MARS MISSION CASE V PERFORMANCE REQUIREMENTS
Beam Current, J_B (A)	2.0	7.9	13.3
Beam Ion Production Cost, E_i (W/A)	192	220	150
Specific Impuse, ISP (S)	3000	4880	4665
Thrust, T (N)	0.13	0.51	1.20
Thruster Efficiency, N_t	72.3%	71.2%	69.5%
Thruster Input Power, (W)	2650	17,210	38,300
Propellant	MERCURY	XENON	MERCURY
Lifetime (hours)	25,000	-	12,500

TABLE 4: MANNED MISSION TO MARS OVERALL MASS SUMMARY

		MPD		ION
1. SPACECRAFT, HABITAT, LANDER, ETC.		143,310 kg		129,640 kg
2. POWER PLANT SUMMARY		36,500 kg		36,500 kg
Reactor	3,730		3,730	
Shield	28,350		28,350	
Turbine	1,290		1,290	
Generator	840		840	
Pumps	590		590	
MBR	540		540	
ACS	1,160		1,160	
3. PROPULSION SYSTEM		121,190 kg		125,310 kg
Engines, etc.	5,144		19,400	
Propellant	111,000		101,450	
Tankage	5,047		4,460	
TOTAL MASS IN GEO		301,000 kg		291,450 kg

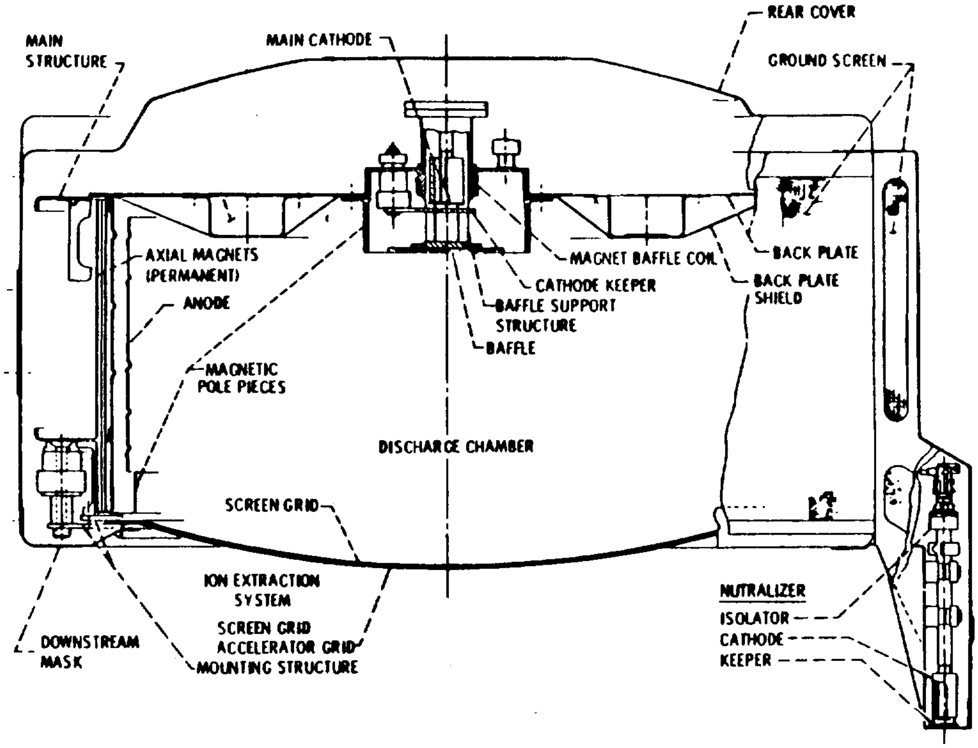


FIGURE 5. MERCURY ION THRUSTER

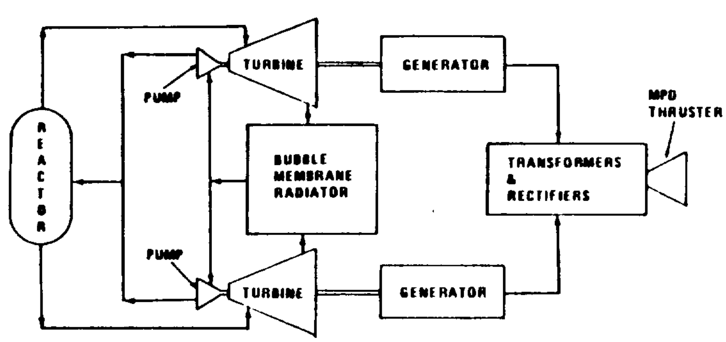


FIGURE 6. PEGASUS SCHEMATIC DIAGRAM

man-rated reactor shield; an axial flow turbine and superconducting alternator for power conversion; a power conditioning subsystem; and a heat rejection and thermal control subsystem. A schematic of the system is shown in Figure 6, and the basic design of these subsystems is discussed in the following sections.

Reactor Subsystem

The reactor selected for the PEGASUS system uses a boiling liquid metal coolant and cermet fuel. The reactor is a right cylinder approximately 53 cm long and 50 cm in diameter. It has a 15 cm diameter exit plenum at each end and a 2.5 cm inlet plenum in the center, forming a double wafer. The reactor is controlled by means of control drums located in the reflector region and is fueled with cermet blocks. Each block is 7.5 cm high and 7.5 cm on a side, with a hexagonal cross-section honeycombed by coolant channels. The coolant channels are 5 mm across the flats with a fueled web 3 mm thick. A preliminary thermal analysis of the cermet fuel, assuming full power operation and stable nucleate boiling, indicates that a peak centerline fuel temperature of 1300 C can be expected for a coolant channel bulk temperature of 1100 C. It should be noted, however, that detailed thermal analysis of this system (or any system involving two-phase flow) will require additional experimental and theoretical work.

Reactor Shielding Subsystem

The PEGASUS system incorporates a four-pi contoured shield to reduce radiation from the reactor to safe levels during full-power operation. The shield is a composite material of LiO, W, and LiH. To maintain the physical integrity of the shield, it is cooled by the reactor inlet coolant. The mass of the shield requires that the major portion of it be delivered to orbit on a separate Shuttle load, to be installed after the system is deployed.

Power Conversion Subsystem

Power conversion is accomplished by means of a pair of turboalternators. The turbine designs incorporate a multi-stage axial flow design which is based on previous development work performed at NASA's Lewis Research Center. This is a saturated potassium vapor turbine which is operating with an inlet pressure of 1.5 MPa and an exit quality of 80% or less. Optimum turbine efficiency is obtained with blade speeds of 300 to

600 m/sec. System shaft speeds will be determined through trade-off studies involving stress analysis of the turbine components at desired operating temperatures and size optimization of the turboalternator system.

Superconducting alternators are chosen to develop the electrical power because of their high power-to-weight ratio. Each superconducting alternator is expected to operate at 1500 Hz and 10 to 20 kva, and be capable of providing a continuous electric power output of 5.0 megawatts. The overall power conversion subsystem is expected to have a specific weight on the order of 0.05 kg/kW and an efficiency of 85%.

Power Conditioning Subsystem

The requirements for the power conditioning system will be set by the specific propulsion unit with which the power system is to be integrated. However, primary power conditioning is expected to be accounted for in the generator portion of the power conversion system.

Heat Rejection Subsystem

Heat rejection for the PEGASUS is accomplished by means of both a high and a low temperature heat rejection subsystem. The high temperature subsystem handles waste heat rejection from the turbines. The low temperature subsystem takes care of waste heat from the alternator and other components requiring cooling.

The high-temperature heat rejection system consists of a bubble membrane radiator, associated pumps, plumbing, and structure. The radiator is sized to reject 21.5 MW of waste heat from the system during full-power operation. This system consists of a 19.9 m diameter thin film spherical envelope, coolant stowage tanks, structural piping, thermal insulating material, and associated pumps. Turbine exhaust vapor enters the radiator and is condensed on the inner surface of the bubble membrane. System rotation causes the condensed fluid to travel along the inner surface of the membrane to the gravity trough at its equator. Once collected, low pressure EM pumps are used to return the coolant fluid to the turbine driven high-pressure reactor feed pumps.

The low-temperature heat rejection system consists of an auxiliary cooling system designed to reject waste heat produced within the alternator and other equipment operating at much lower temperatures. The auxiliary cooling system has its own working fluid, pumps, and radiator.

This system utilizes helium as its working fluid and is composed of a Stirling cycle cryogenic cooler, an auxiliary chiller, a low-temperature radiator, and associated pumps and piping. The cryogenic cooler removes heat from the liquid helium and transfers this heat to the auxiliary coolant in the chiller. A closed loop system is used to pump this coolant through the auxiliary cooling radiator located around the perimeter of the main radiator. Before returning to the chiller, this coolant is used to cool pumps and other components.

PEGASUS SYSTEM PARAMETERS

Table 4 contains the major parameters that have been examined thus far. All calculations are of a scoping nature and large tolerances exist in all areas. The specific impulse and the thrust of the ion system were matched to that of the MPD system, and the burn times held the same for both systems so as to match the trajectory calculations. The trajectory used may not be an optimum for either the MPD or the ion thruster system.

ARTIST'S CONCEPT

Figure 7 is an artist's sketch of the Pegasus concept.

FIGURE 7

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N87-17788

A PHYSICOCHEMICAL ENVIRONMENTAL CONTROL/LIFE SUPPORT SYSTEM FOR THE MARS TRANSIT VEHICLE

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ABSTRACT

The environmental control/life support system (ECLSS) of the Mars transit vehicle (infrastructure) must be as small and maintenance free as possible to allow maximum mission flexibility. This paper describes a "new technology" physicochemical ECLSS concept similar in many ways to several of the partially closed ECLSS concepts proposed for the Space Station (SS). However, this new concept eliminates several of the SS ECLSS subsystems and potentially eliminates the use of cryogenics and high-pressure gaseous storage.

The unique technology in the new concept is supercritical water oxidation (SCWO). The properties of supercritical water allow it to act as a medium in which organics and oxygen can mix freely. The extreme conditions that form supercritical water (630 K, 250 atm) also induce complete combustion of the organics. Virtually all organics break down and reform into carbon dioxide, water, and nitrogen. Inorganics form salts which are less soluble in supercritical water than in water in its natural state. Invariably, the inorganics precipitate out. This paper will explain how technology based on these phenomena can be used in an ECLSS for carbon dioxide removal, partial humidity control, trace contaminant control, water reclamation, nitrogen generation, and untimately trash and garbage reduction. Then a qualitative comparison between an ECLSS using SCWO technology and an ECLSS using SS era technology will be given. Mass balances are included to enhance the comparison.

INTRODUCTION

The environmental control/life support system (ECLSS) of the Mars transit vehicle (infrastructure) must be as small and maintenance free as possible to allow maximum mission flexibility. This paper describes a physicochemical ECLSS concept similar in many ways to several of the partially closed ECLSS concepts proposed for the Space Station (SS). However, this new concept eliminates several of the SS ECLSS subsystems by performing more than one ECLSS function in one "new technology subsystem. Furthermore, inherent in the simplified concept is the potential for eliminating the use of problematic cryogenics and high-pressure gaseous storage (the forms of nitrogen supply considered for the SS). Other advantages of the new concept are discussed and additional quantitative studies are recommended to increase confidence in supporting development of the unique technology in the new concept.

To summarize, the SCWOS technology is based on the physics and chemistry of water molecules (H_2O) at conditions above their super-critical pressure and temperature (at 25.3 MN/m^2 (250 atm) and 627.59 K (670^o F) (Anon., 1982; Josephson, 1982; Temberlake, 1982; Modell, 1983; Swallow, 1984-85). Under these conditions, the dielectric constant of H_2O weakens which causes two important phenomena to occur: hydrocarbons and other normally immiscible organics become miscible in the water medium, and normally-dissolved inorganic salts precipitated out of solution. Solid salts can be separated from the process stream in the same solids separator that removes any metal particles found in solution. At the high temperature, complete combustion of the organics result if sufficient oxygen (O_2) is presesnt. Complete combustion yields H_2O, carbon dioxiode (CO_2) and N_2.

To achieve and sustain the high temperature for the supercritical combustion, O_2 and hydrogen (H_2) can be introduced to the feed mixture for their "heat of reaction" value (O_2 + $2H_2$ > $2H_2O$ + heat) (Modell, 1984). An alternative, which is to preheat the feed electrically, would consume about the same amount of energy; however, the effect of the extreme conditions on the heat exchanger would make corrosion and structural problems difficult to control (Modell, 1984). These problems could be avoided by using the O_2/H_2 feed method. Maintaining the temperature is a matter of "superinsulating" the system. The vacuum of space could be utilized for this purpose. The heat of combustion of the reactants ensures that the temperature during reaction would not fall below the lower limit for rendering complete combustion. Reaching and maintaining the desired pressure is also achievable using current technology.

BASIC LIVING REQUIREMENTS FOR A MARS TRANSIT VEHICLE

The basic living requirements (i.e., maximum partial pressure of CO_2, minimum partial pressure of O_2) (Lin, 1983) for a Mars transit vehicle are the same as those for the SS. However, the ground rules and methods for meeting those requirements may be different for the Mars transit vehicle, especially since the mission objectives are so different.

The problem of inaccessibility to civilization for resupplies is much more profound for Mars missions than the SS missions. The astronauts will have to take everything they need (for themselves and for the vehicle) to survive over two years without resupply. Therefore, the less dependent on terrestrial resupplies the mission is, the more flexible the mission can be, and the less the time wasted on housekeeping.

CANDIDATE ECLSS CONCEPTS

Many elements are common to the proposed Mars transit vehicle ECLSS (Fig. 1) and a SS-type ECLSS (Fig. 2). Both have the same atmospheric pressure/composition control subsystem, O_2 generation subsystem, CO_2 reduction subsystem, and hygiene facilities. In fact, most of the hardware in the two ECLSS's is the same, but the few differences make the two concepts quite dissimilar.

SCWOS-ECLSS

Food, water posttreatment supplies, and a nitrogen-containing solid, which is discussed later, are the required resupplies for the ECLSS designed around an SCWOS. The byproducts would be salts, minerals, dense carbon, and excess water and hydrogen, all of which could be used elsewhere on the Mars transit vehicle or Mars base. A mass balance and a functional schematic for the SCWOS-ECLSS is shown in Figure 1.

Five subsystems would make up the air management group of the SCWOS-ECLSS: the atmospheric pressure/composition control subsystem, the O_2 generation subsystem, the SCWOS (for CO_2 removal, trace contaminant control, N_2 makeup, and partial humidity control), the CO_2 reduction subsystem, and the humidity/temperature control subsystem. The multifunctional SCWOS also would be part of the waste management group and the water management group, which are discussed later in more detail.

The atmospheric pressure/composition subsystem would be similar to that of the Space Shuttle; however, the sources of the gases (O_2 and N_2) would be different. Oxygen would be generated by water electrolysis

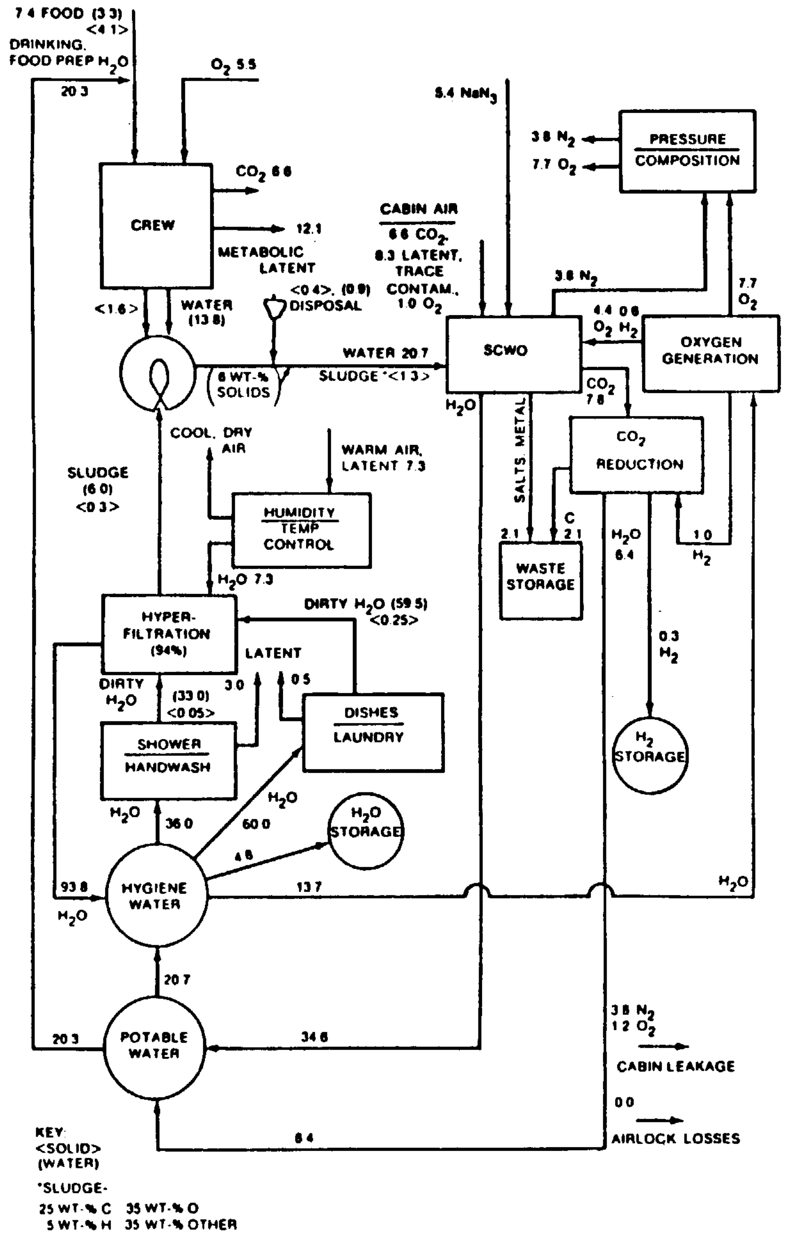

Fig. 1 - SCWOS-ECLSS; daily mass balance for a three-member crew.

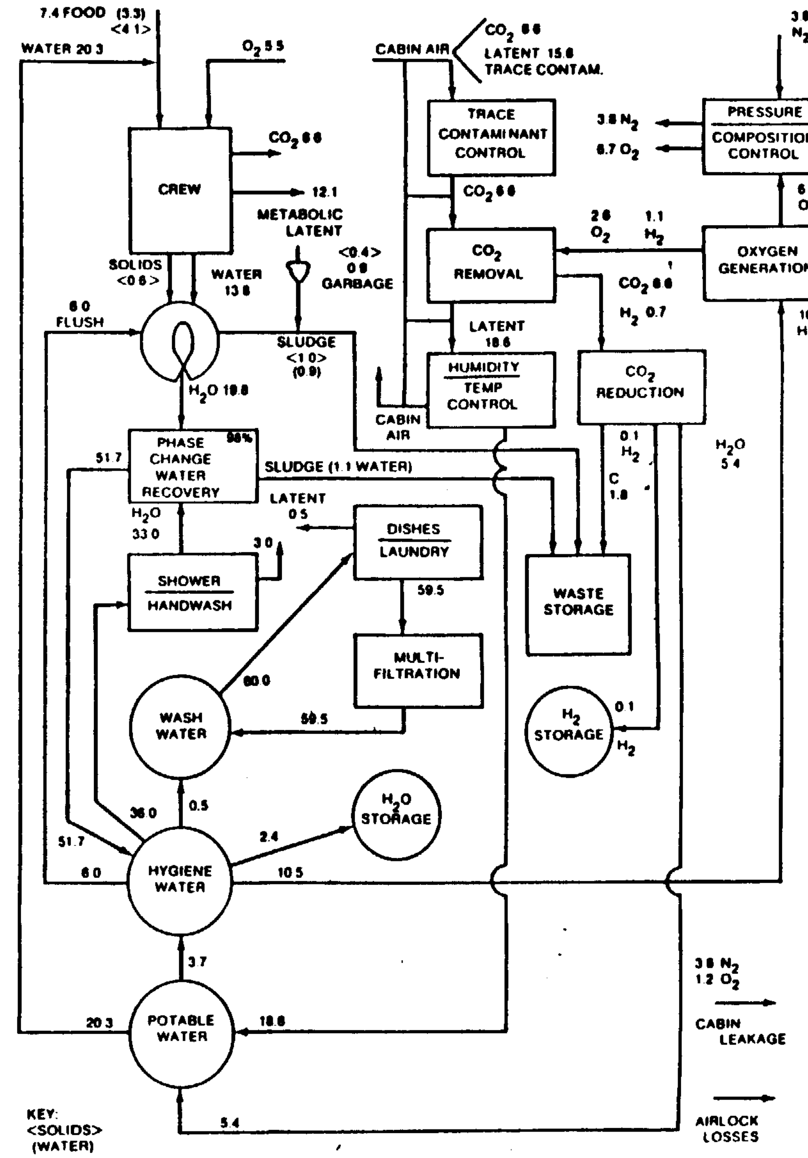

Fig. 2 - Space Station-type ECLSS daily mass balance for a three-member crew.

($2H_2O$ + electrical power > $O_2 + 2H_2$). Nitrogen would be derived from the SCWOS.

Since not enough N_2 for ECLSS needs could be generated by the normal ECLSS wastes fed to the SCWOS (urine, feces, garbage, dirty water, and trace contaminants) (Marrero, 1983), the SCWOS feed could be supplemented with a nitrogen-containing solid or liquid compound supplied from Earth. There are several compounds to choose from that would benefit the Mars transit vehicle in ways beyond N_2 generation. Information on several nitrogen-containing compounds is contained in Table 1 (Sax, 1965; Weast, 1977). As discussed later in the comparison between ECLSSs, the most important consequence of the use of this N_2 generation concept is that the compound could be resupplied as a solid or liquid. If this N_2 generation scheme is rendered undesirable, the SCWOS-ECLSS could revert to the same N_2 supply subsystem to be used for the SS (probably cryogenic storage). The atmospheric pressure/composition control subsystem would regulate the release of O_2 and N_2 into the cabin to maintain the cabin O_2 content and total pressure.

The CO_2 reduction subsystem in the air management group would receive all the H_2 from the O_2 generation subsystem except that used in the SCWOS. All the CO_2 entering the SCWOS with the process air and that formed by combustion inside the reactor would leave the SCWOS in a concentrated stream. The CO_2 reduction subsystem would receive this CO_2 stream and convert the CO_2 and H_2 into water and dense carbon. The excess H_2 would be stored for other Mars transit vehicle or Mars base needs.

The SCWOS-ECLSS water management group would consist of two water loops: the potable water loop and the hygiene water loop. Normally, to save energy and expendables, the hygiene water would not be made potable (in the palatable sense) but nonetheless free from contamination.

"Prepotable" water would come from urine, water vapor in the air (e.g., metabolic latent), SCWOS combustion product water, and CO_2 reduction product water. Potable water would be derived from two intense sterilizing processes which operate at temperatures above 533 K (500^{o} F): the CO_2 reduction subsystem and the SCWOS. The sterile water would be chemically enhanced for flavor and for bacterial growth prevention to yield potable water. Once the potable water tanks were full, the

TABLE 1

NITROGEN-CONTAINING COMPOUNDS WHICH ARE CANDIDATES REACTANTS FOR NITROGEN GENERATION

[From Sax, 1965, and Weast, 1977]

Compound and description	Molecular formula	Molecular weight g/g-mole	Density or specific gravity (a)	Melting point	Boiling point	Heat of formation, kcal/g-mole	Comments (b)
Ammonium hydroxide Colorless liquid	NH_4OH	35.5	---	-77° C	---	-87.64 (acqueous)	---
Hydrazine Colorless fuming liquid, white crystals	N_2H_2 (NH_2-NH_2)	32.05	1.011^{15} (liquid)	1.4° C	113.5° C	+12.05	Flash point is 126° F (open container) Autoignition occurs at 518° F
Hydrazine axide White powder	N_2H4-HN_3	75.07	---	75.4° C	---	---	---
Hydrazoic acid (Azoimide) Colorless liquid	HN_3	43.03	1.094^{25}	80° C	37° C	70.3	May be used to sustain SCWOS reaction temperature
Sodium Amide White crystalline powder	$NaNH_2$	39.02	---	210° C	400° C	-28.04	Yields heat with moisture
Sodium axide Colorless hexogonal crystals	NaN_3	65.01	1.846^{20}	---	---	---	Decomposes in a vacuum
Sodium nitride Dark grey crystals	Na_3N	82.98	---	$300^{\circ}C^{c}$	---	---	To be considered only if sodium ions are highly desirable
Sodium nitrite Slightly yellowish or white crystals	$NaNO_2$	69.00	2.168^{0}	271° C	320° C^{c}	-85.9	The oxygen elements may fuel the SCWOS combustion reaction
Sodium Nitrate Colorless crystals	$NaNO_3$	84.99	2.261	308.8° C	380° C^{c}	-101.54 (crystalline) 106.65 (aqueous)	The oxygen elements may fuel the SCWOS combustion reaction (aqueoous)

[a] Superscripts and subscripts are temperatures in deg C.

[b] Many of the compounds are dangerous, some explosive; however, if there are ways to minimize the danger, they have been retained for comparison.

[c] Decomposition point.

processed water would be redirected to the hygiene water supply. In fact, the mass balance (Fig. 1) shows that there would be enough of this redirected water to be used for taking showers or for rinsing in the dishwasher and laundry machine.

Ordinarily, hygiene water would be used for laundering, dishwashing, showering, and handwashing. Surplus hygiene water could be stored for other Mars transit vehicle or Mars base operations. Dirty hygiene water and whatever humidity condensate was not processed by the SCWOS would be cleaned by reverse osmosis, a selective regenerable filtering process. After posttreatment, the clean water would be returned to hygiene water storage.

SPACE-STATION-TYPE ECLSS

One SS ECLSS (SS-ECLSS) concept (Anon., 1983; Lin, 1984) is depicted in Figure 2. This ECLSS concept is closed and resupply-free except for water filters, posttreatment chemicals, N_2 makeup, and food. Feces, garbage, hygiene sludge, and carbon (C) would be the byproducts requiring extensive waste management facilities. Excess clean water, however, would have many other uses outside the ECLSS. The system would need little scheduled maintenance except for frequent water filter changes, but even this chore represents a waste of time and precious storage space.

Seven subsystems would make up the air management group: the atmospheric pressure/composition control subsystem, the N_2 supply subsystem, the O_2 generation subsystem, the CO_2 removal subsystem, the CO_2 reduction subsystem, the trace contaminant control subsystem, and the humidity/temperature control subsystem. The water management group would have three reclamation subsystems: one for producing drinking water, one for hygiene water, and one for wash water (laundry and dishwashing). Having these three water groups would minimize energy and expendables.

COMPARISON OF THE ECLSS CONCEPTS

The differences between the two ECLSS concepts go beyond what appears on the schematics. Several ways in which the concept differences impact the Mars mission are disclosed in the following discussion.

As mentioned earlier, resupply weight and volume requirements are extremely crucial design considerations. The handling of wastes (trace contaminants, feces, trash, and garbage) by the SCWOS-ECLSS saves significant resupply weight and volume in terms of filters, bactericides, and waste containers. The wastes (solid, liquid, and gaseous) would actually be broken down into harmless combustion products. Bacteria would be destroyed, so concern about masking or filtering odors, resupplying bactericides, or venting and dumping wastes would be greatly reduced. In fact, the materials derived from the SCWOS-ECLSS waste reduction could be incorporated back into the ECLSS to help further close the system: CO_2 would go the CO_2 reduction, H_2O would go to potable water storage, and N_2 would go to the atmospheric pressure/composition control subsystem.

The N_2 supply concept of the SCWOS-ECLSS may be preferable to cryogenic storage. Cryogenic N_2 requires insulation and isolation to reduce boiloff. If boiloff rates exceed use rates, a high-pressure tank and pump may be required to eliminate loss of N_2. High-pressure storage is costly in terms of volume and weight. These problems may be eliminated in the SCWOS-ECLSS. The alternative offered by the SCWOS is to carry a powder, a grindable solid, or a liquid that is rich in elemental nitrogen (N) which can be reduced to N_2. The compound, being solid or liquid, would assume any desired shape for storage. In addition, several of the candidate compounds would break down into wastes that would reduce resupply weight elsewhere. Carrying nitrogen in solid or liquid form would greatly simplify logistics. The potential for this simplification exists with the SCWOS-ECLSS.

The air management group of the SCWOS-ECLSS is simpler than that of the SS-ECLSS. In one package, the SCWOS would remove the CO_2, the trace contaminants, and more than half of the water vapor from the air. Essentially two and one-half SS-ECLSS air management subsystems would be replaced by the SCWOS. Having fewer unique subsystems would reduce the crew's training load and cut down on the spare parts inventory, not to mention increasing the reliability and decreasing the maintenance of the ECLSS.

The water management group of the SCWOS-ECLSS is also simpler than that of the SS-ECLSS. The former has two water loops; the latter, three. The mass balances mentioned earlier indicate that the potable water supply level of the SCWOS-ECLSS, as compared to that of the SS-ECLSS, is less critically dependent on subsystem production and consumption rates.

In the SCWOS-ECLSS, twice as much potable water is produced daily as is used for drinking and food preparation. In the SS-ECLSS, the ratio of "produced" to "humanly ingested" potable water of nearly one to one signifies greater dependence on timing between the ECLSS entities. The relative abundance of potable water in the SCWOS-ECLSS opens up new integration possibilities, such as potable water showers and potable water rinse cycles for the laundry machine and dishwasher. These luxuries cannot be afforded as easily in the SS-ECLSS.

STATUS OF SCWOS DEVELOPMENT

Substantial work has been done to understand the chemistry of supercritical water oxidation and ways to develop the technology. Many sludges and solutions have been successfully converted to the products of complete combustion (CO_2, H_2O, and N_2) in a breadboard reactor.

However, there are still chemical and mechanical difficulties associated with processing some ECLSS wastes. For instance, although a recent discovery led to the complete combustion of urea (a major component of urine) at a lower than expected temperature (Swallow, 1984-85), the processing of urine has not been successful to date. (The as yet uncontrolled precipitation of urinary salts has clogged the reactor.) Furthermore, the preparation of trash and garbage for processing has not been successful. (Very little work has been done in this area.) Much development work lies ahead in reaction optimization, design optimization, and automation to use this technology for the Mars transit vehicle ECLSS.

For comparison with other candidate ECLSS waste management subsystems, the estimated SCWOS power level for processing the wash water, urine, and feces of an eight-person crew is 300-400 watts, continuous (Thomason, 1985). This power level excludes the energy for producing supplementary oxygen, but does not take credit for the carbon dioxide removal, the trace contaminant control, and the partial humidity control that results from the waste processing. (The process uses cabin air for the combustion oxygen supply.) Since the SCWOS process can be compared favorably with other candidate waste management subsystems, an ECLSS designed around an SCWOS should certainly do well in comparison with the more conventional partially closed ECLSS designs being considered for SS use.

CONCLUSIONS AND RECOMMENDATIONS

The following are some of the many reasons given in this paper for supporting the candidacy (and development) of the SCWOS-ECLSS for operation in the Mars transit vehicle; (1) Trace contaminants would be controlled without the consumption of expendables; (2) Waste management would be simplified and would require less storage room and maintenance; (3) The SCWOS would make useful byproducts out of trace contaminants and wastes; (4) Air management would be simplified; (5) Nitrogen logistics would be more manageable; (6) Logistics would be reduced and facilitated in many ways; (7) Water management would be simplified; (8) Luxuries such as bathing in potable water or having potable water rinse cycles for the laundry machine and dishwasher could be afforded; and (9) The SCWOS-ECLSS would allow more mission flexibility.

Hopefully, these qualitative advantages have stimulated interest in learning more about the quantitative differences (power consumption, heat rejection, weight, and volume differences) between the SCWOS-ECLSS and other candidate ECLSSs. These quantitative analyses are needed to fully appreciate the advantages or disadvantages of the ECLSSs.

The discovery of supercritical water oxidation could be fortunate for space exploration. This paper reveals the manner in which this technology would enhance long-duration Mars missions. Next, quantitative analyses are needed to gain a better appreciation for the advantages and disadvantages of the ECLSS concepts. Preliminary calculations encourage optimism toward the use of the SCWOS-ECLSS. This paper also presents the challenge of determining the nitrogen generation compound that would be most beneficial to the Mars mission.

N87-17789

POWER SYSTEM TECHNOLOGIES FOR THE MANNED MARS MISSION

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ABSTRACT

The high impulse of electric propulsion makes it an attractive option for manned interplanetary missions such as a manned mission to Mars. This option is, however, dependent on the availability of high energy sources for propulsive power in addition to that required for the manned interplanetary transit vehicle. Two power system technologies are presented here. They are the nuclear and solar technology options. The ion thruster technology for the interplanetary transit vehicle is described for a typical mission.

The power management and distribution system components required for such a mission must be further developed beyond today's technology status. High voltage-high current technology advancements must be achieved. They are described in this paper. In addition, large amounts of waste heat must be rejected to the space environment by the thermal management system. Advanced concepts such as the liquid droplet radiator are discussed as possible candidates for the manned Mars mission. These thermal management technologies have great potential for significant weight reductions over the more conventional systems.

POWER/ENERGY SYSTEMS

Nuclear Power System (megawatt class)

Reference 1 describes a nuclear power system for a manned interplanetary spacecraft utilizing electric propulsion. This power system consists of a lithium cooled-uranium nitride fueled reactor. The characteristics for this power system are given in Table 1. This system represents technology which should be operational in the year 2000. A typical spacecraft configuration utilizing this power system is shown in Figure 1. The 300 foot separation distance between the power plant and the manned modules is required for crew radiation isolation.

A schematic of the nuclear power/energy conversion system is given in Figure 2. On this Figure are shown the various states throughout the system. Table 2 shows the weight breakdown of this system.

TABLE 1

Nuclear Turboelectric Indirect Rankine System
Lithium Cooled Reactor
(Reference 1)

Reactor Power	34.3 MW_{THER} @ 1510 K Outlet Temperature
Net Electric Power	$5MW_e$
Turbine Inlet Temperature	1450^o K
Turbine Efficiency	80%
Number of Stages	5
Inlet Pressure	209 PSI
Exit Pressure	27.5 PSE
Condenser Temperature	1100^o (Exit Quality 88%)
Cycle Efficiency	16.8%
Overall Efficiency	14.6%

TABLE 2

Nuclear Power System Weights (KG)

Reactor	3900 KG
Shield	30000 (Man Rated)*
Reactor Pump	1750
Boiler	2400
Turbogenerator	3300
Feed Pump	1250
Condenser	3000
Radiator	8700 (535 M^2)
Power Conditioner	2000
Miscellaneous (Structure, Etc.)	2900
Total Weight in LEO	59200 KG

*Man Rated: 13 rem total integrated mission dose

This nuclear power system concept results in a specific weight of 12 KG/KW_e.

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FIGURE 1: SPACECRAFT CONFIGURATION

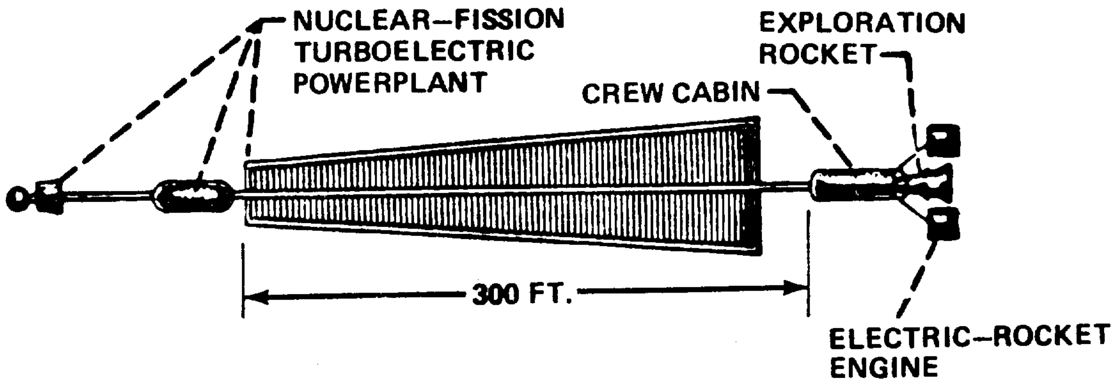
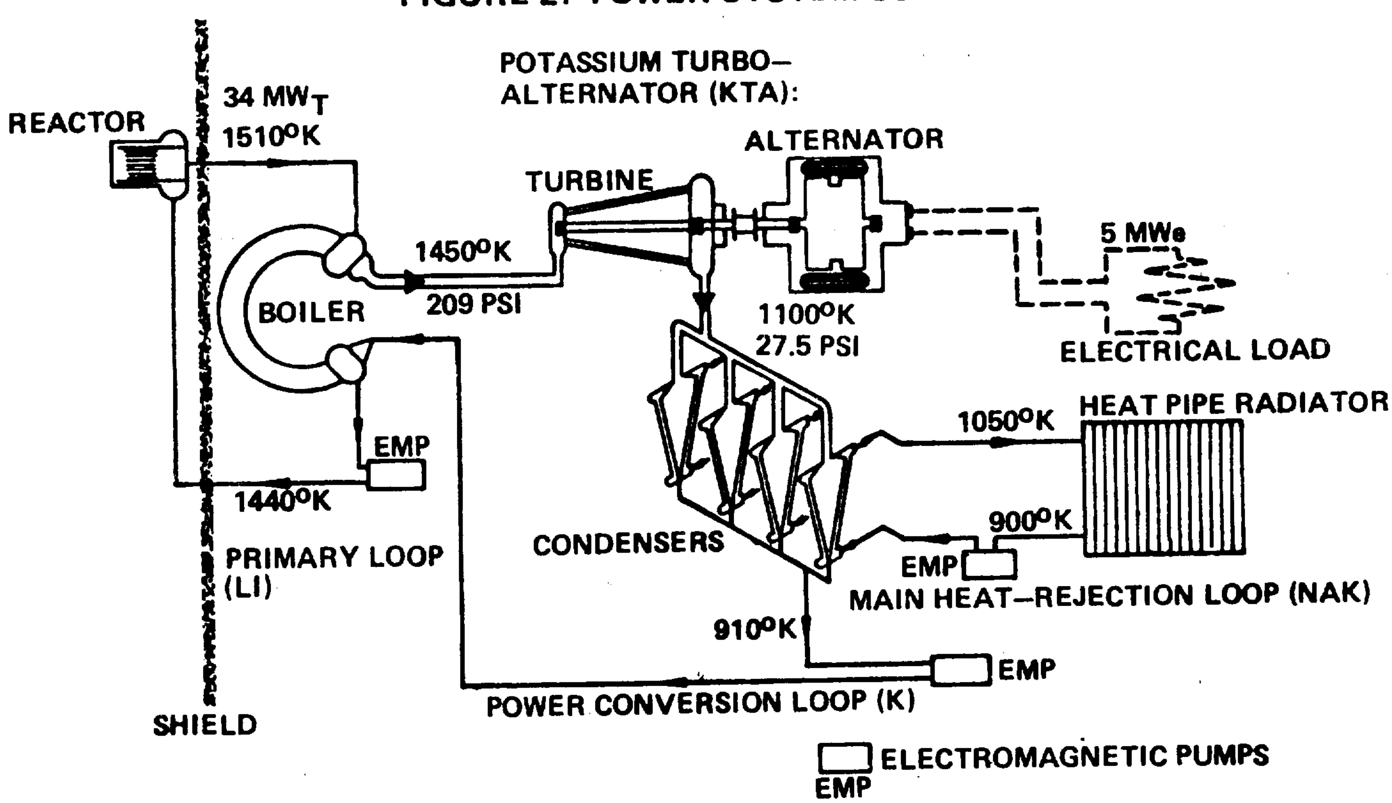


FIGURE 2: POWER SYSTEM SCHEMATIC



Technology advances projected for the nuclear power systems for the early part of the 21st century indicate a possibility of reducing this weight by a significant amount. Reactor designs employing cermet fuels, boiling liquid metal cooled reactors, improved turbo-alternator machinery, and advanced heat rejection concepts offer potential for substantial weight savings over the technologies which make up the design shown here. Projections of achieving specific weights of 7 KG/KW_e for manned nuclear systems by the first part of the century have been made, and appear attainable.

Other Power Options

The possibility of using other non-nuclear power sources for the manned Mars mission was also addressed.

Solar Photovoltaic Power Systems

Photovoltaic system energy densities of 300 Watts/KG, at 1 AU, are projected to become available at the early part of the 21st century. Such systems, when sized for low Mars orbit (1.52 AU) and with regenerative fuel cell storage for shadow period operation, appear to indicate specific weights on the order of 25-30 KG/KW, depending on the system hardware parameters assumed. However, special mission designs, wherein partial or no electric propulsion thrusting would take place during the Earth or Mars shadow phases could significantly reduce this figure by eliminating the weight penalty associated with providing shadow period operation. Technology advances beyond the 300 Watts/KG and special trajectory design could possibly make solar photovoltaic systems weight competitive with nuclear systems. This would come about, however, at a penalty in transit trip time. Also, the enormously large areas of megawatt-size photovoltaic systems would pose an additional problem.

Solar Thermal Dynamic Systems

These systems were also investigated for application to the manned Mars mission. However, even with optimistic projections for techonolgy advances, these systems do not seem weight competitive for those missions/orbits where extended shadow period operation is required. If the weight penalty associated with shadow period operation (storage) can be eliminated (or reduced), these systems may be competitive for the Mars mission.

ION THRUSTER PROPULSION SYSTEM

The interplanetary phase (trajectory) of the manned Mars mission has not yet been designed. However, in order to carry out a preliminary evaluation of possible thruster technologies, a representative mission scenario has been selected (Ref. 2). One possible thruster system which shows considerable potential for application to this mission is an outgrowth of the 30-cm ion thruster technology (J-series mercury ion thrusters), developed by the Lewis Research Center. This thruster has been investigated for application to that performance range required for future interplanetary missions, such as the manned Mars mission. In addition, the technical maturity of this technology makes it a prime candidate for this mission. The interplanetary trajectory analysis for the manned Mars mission carried out in Ref.2 used this technology to derive thruster performance data. A cross section of the 30-cm mercury ion thruster is shown in Figure 3.

The design level performance of the 30-cm J-series thruster is shown in Table 3. This shows the extended performance (demonstrated) and the required performance if this thruster were to be used for the manned Mars mission.

TABLE 3

30-cm J-Series Thruster Parameters

	Design Performance*	Extended Performance*	Manned Mars Mission Requirements
Beam Current	2.0 A	7.9	13.3
Beam Ion Prod. Cost	192 W/A	220	150
Specific Impulse	3000 sec	4880	4665
Thrust, N	0.13 N	0.15	1.20
Thruster Effic.	72.3%	71.2	69.5
Thruster Input Pow.	2650 W	17210	38300
Propellant	Mercury	Xenon	Mercury
Lifetime	25000 hrs	-	12.500

*(Demonstrated by Test)

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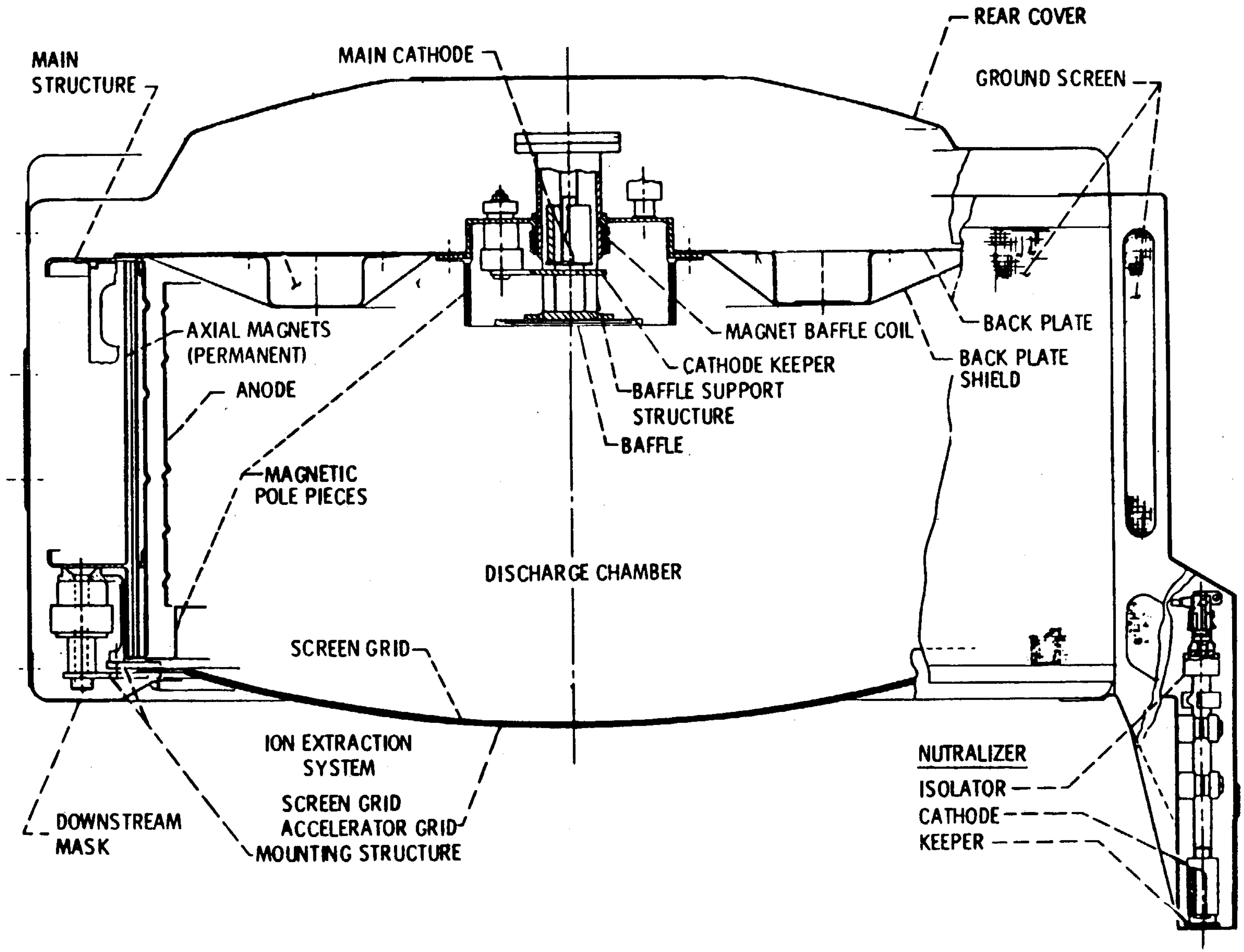
FIGURE 3. J–SERIES 30 cm MERCURY ION THRUSTER



In carrying out the analyses for the manned Mars mission, thruster system characteristics were also developed for 50-cm and 100-cm ion thrusters. Both mercury and xenon propellants were considered at specific impulses associated with 3000, 4000, and 5000 total volts across the accelerator grids. Operation of the J-series thruster with xenon and other inert gases has been demonstrated (Ref.3). The performance is comparable to that documented with mercury propellant.

There are critical technology advances which must be made before this technology can be incorporated into a flight system. The major developments required are:

Discharge Chamber: The present magnetic field and chamber design of the J-series thruster may require minor modifications to reduce losses to the level used in the analyses. However, recent research at Lewis (Ref.4) has demonstrated a 30-cm xenon ion thruster operating at losses below this level.

Accelerator System: Operation of the J-series accelerator system at 3000 v total has been demonstrated. However, higher voltages will require technology levels beyond those of present J-series accelerator grid materials and fabrication procedures (Ref.5). Further, although thrusters of up to 150-cm dia have been successfully designed and operated (Ref.6), dished-grid optics of the J-series type have not been fabricated for thrusters larger than 30 cm.

Hollow Cathode: The present J-series cathodes are designed for operation up to 20 A emission current. However, recent tests at Lewis demonstrating an extended performance of the J-series thruster (Table 3) have operated a modified cathode design at up to 50 A emission current. The systems analyzed would require cathode operation ranging from a low of 80 A for the 30-cm ion thruster scenario (Isp = 4665) to a high of 2800 A for the 100-cm ion thuster (Isp = 7400). These levels may be achieved by operating a larger cathode, or by operating a multiple cathode arrangement. This is a technology that also requires demonstration.

The thruster technology required for the Mars mission scenario at 4665 seconds specific impulse is considered near-term.

Power Conditioning Subsystem

The AC Power Management and Control approach used in this study is that presented in Ref.5. This includes the methodology employed and assumptions made to define the thrust system characteristics in this analysis.

System Parameters

Figures 4 to 9 present thrust system parameters as a function of specific impulse. Figure 4 presents total number of thrusters in the propulsion system as a function of specific impulse for the 30-, 50-, and 100-cm thrusters, for both mercury and xenon propellants. Because the ratio of net to total accelerator voltage in fixed at .9 to maintain a high specific impulse, the mercury propellant systems require less total thrusters to attain the mission specified thrust, at a specific impulse. Because the mission parameters are fixed, increasing the thruster diameter decreases the total number of thrusters required to accomplish the mission. Aside from the near-term technology capability of the 30-cm thruster, the 100-cm thruster technology would represent a significant reduction in system complexity through a reduction in component count.

The thrust system characteristics are relatively insensitive to thruster size (for the specified mission profile): figures 5-8 present thrust system parameters for the 100-cm thruster technology. Figure 5 presents overall propulsion system total mass (including propellant) as a function of specific impulse. The system mass decreases strongly with increasing specific impulse due to the reduction in propellant mass. The propulsion system dry mass is relatively insensitive to changes in specific impulse, as indicated in Figure 6. This is due to the competing facts that the power supply and thermal system masses increase with specific impulse, while the thruster, tankage, and support structure masses decrease with increasing specific impulse. Figure 7 presents the total payload mass (power and nonpower) as a function of specific impulse. The payload mass (the final return mass less the propulsion system dry mass), is relatively insensitive to changes in specific impulse for the region investigated.

Propulsion system total efficiency vs. specific impulse is presented in Figure 8. The system efficiency, about 67%, is relatively insensitive to specific impulse and propellant type. This reflects the near-constant

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FIGURE 4. TOTAL NUMBER OF THRUSTERS VS. SPECIFIC IMPULSE CASE I–VI

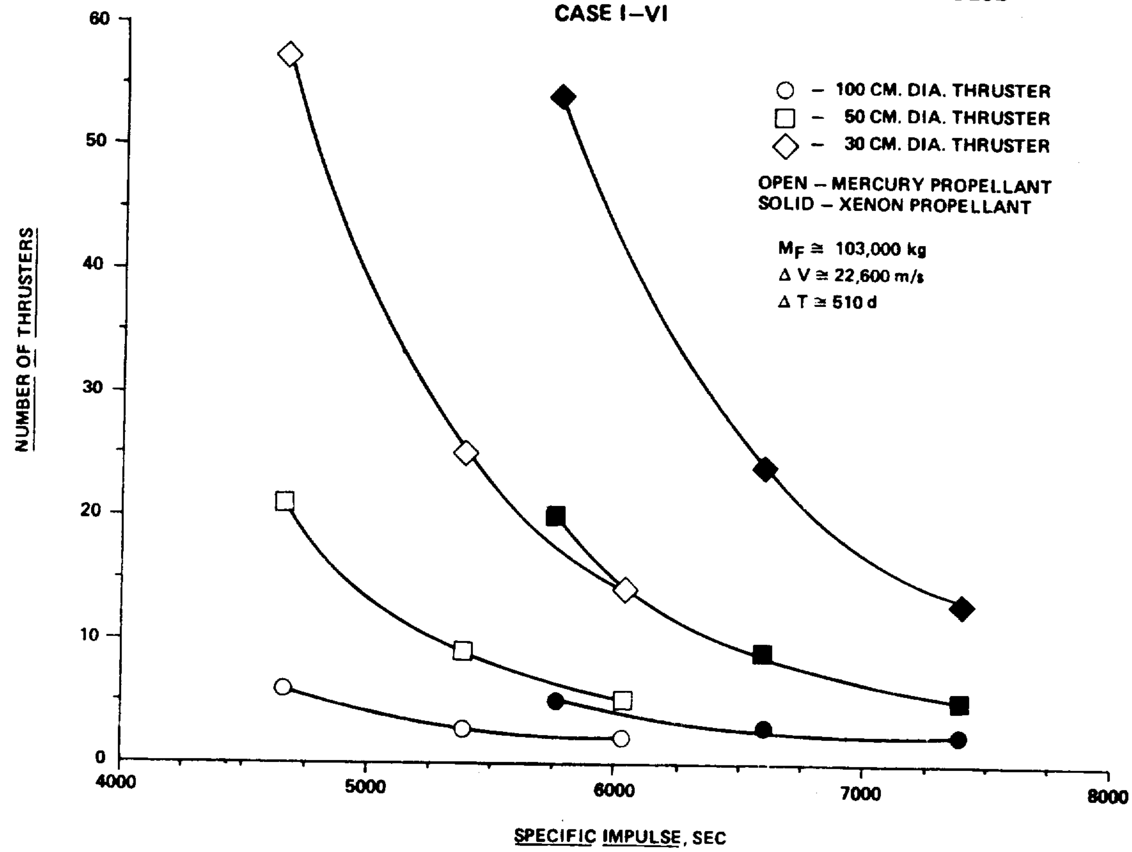

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FIGURE 5. TOTAL PROPULSION SYSTEM MASS VS. SPECIFIC IMPULSE CASE I–II

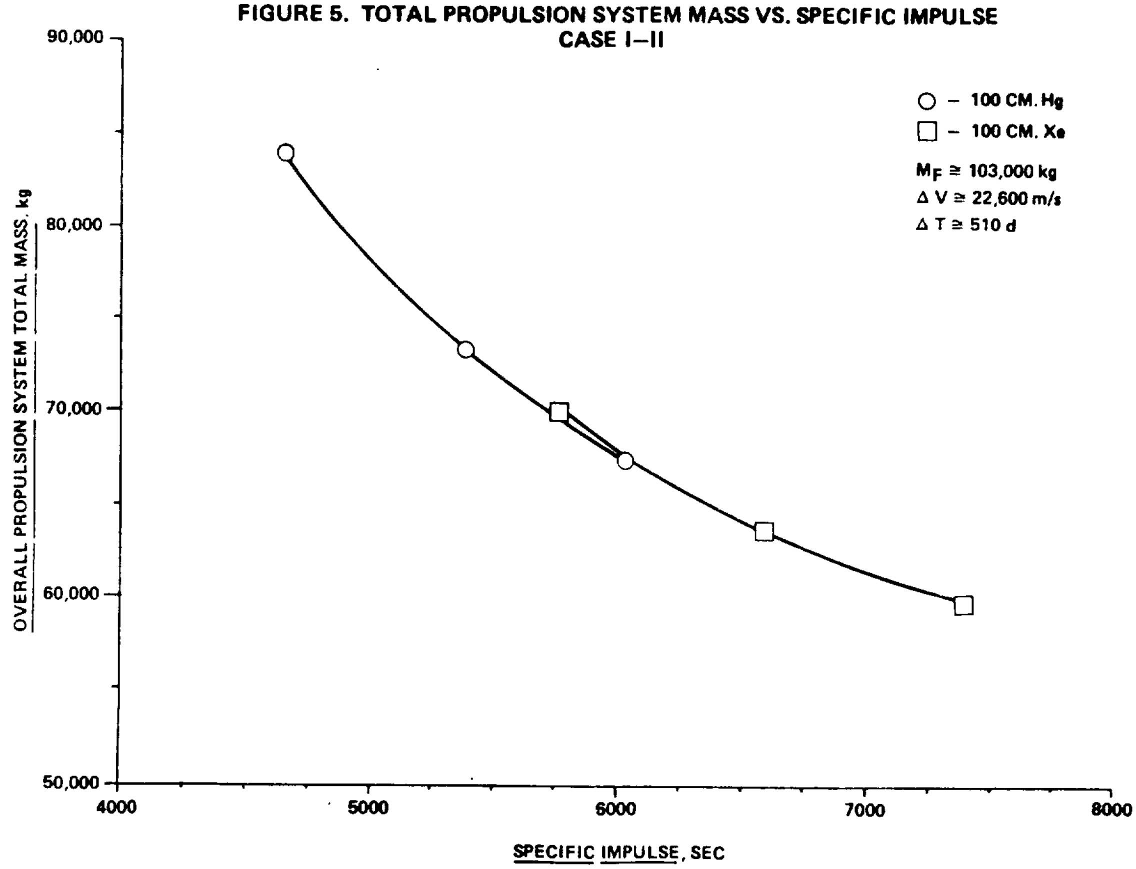

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FIGURE 6. PROPULSION SYSTEM DRY MASS VS. SPECIFIC IMPULSE
CASE I–II

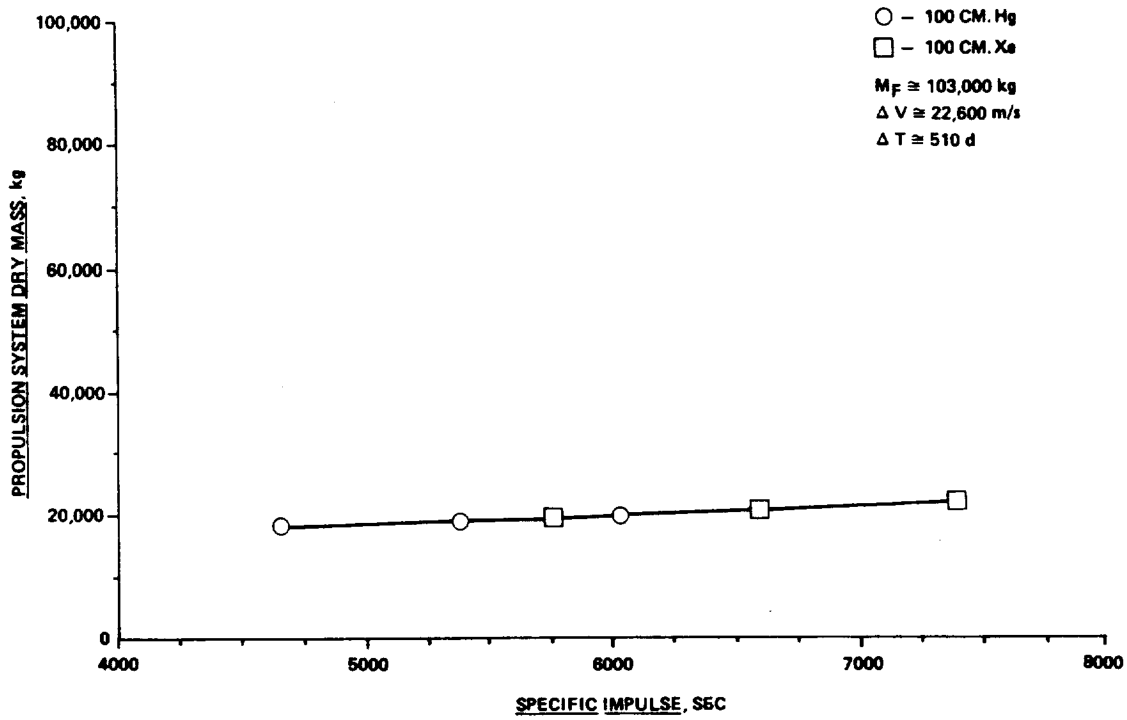

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FIGURE 7. TOTAL PAYLOAD MASS VS. SPECIFIC IMPULSE
CASE I–II

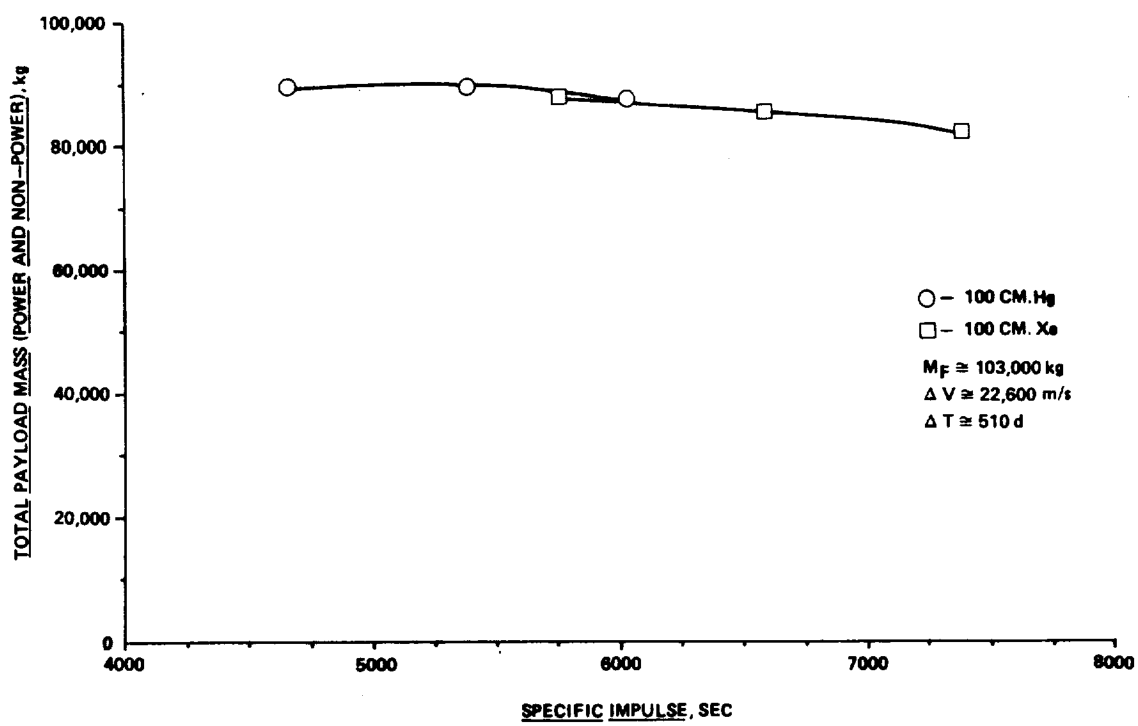


FIGURE 8. PROPULSION SYSTEM TOTAL EFFICIENCY VS. SPECIFIC IMPULSE
CASE I–II

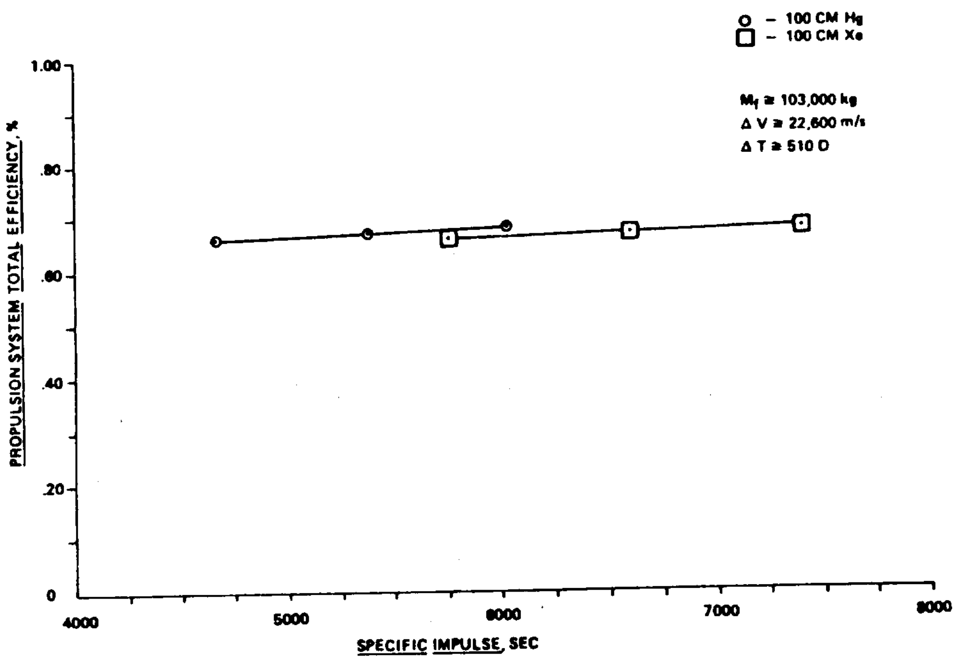
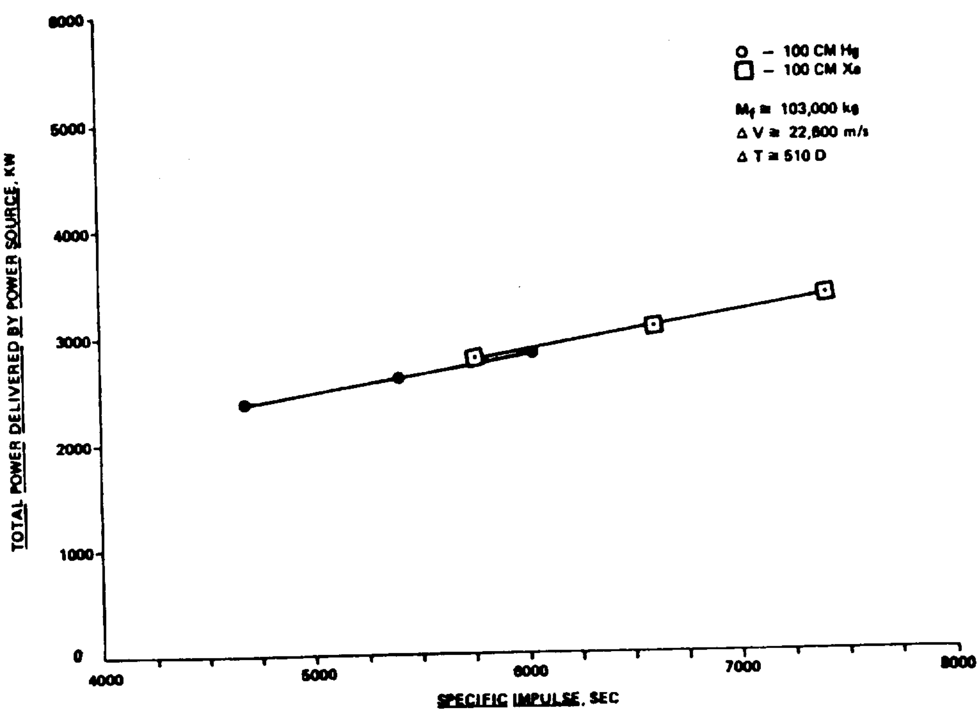


FIGURE 9. TOTAL POWER VS. SPECIFIC IMPULSE
CASE I–II



thrust-to-power ratio for a wide range of specific impulse. Figure 9 presents the system power as a function of specific impulse. Because the mission parameters were fixed, the system power increased by approximately 30% from 2370 KW to 3340 KW as the specific impulse was varied from 4665-7400 seconds.

POWER MANAGEMENT AND DISTRIBUTION TECHNOLOGY

The multimegawatt power requirements of a manned Mars mission will require the development of high power-high voltage-high current power conditioning, transmission and distribution systems. Power requirements have been increasing at a rapid pace since the early 70's when space power requirements could be fulfilled by systems of 1 KW or less. Figure 10 shows the increase which space power systems have experienced over the years. The manned Mars mission will require power electronic components in the multimegawatt range to be ready by the turn of the century.

The basic power requirement for the manned Mars mission is to the ion thruster with the beam supply being the predominant user at 3000, 4000, or 5000 volts, depending upon the thruster concept used. Since the mission and spacecraft design have not as yet been determined, the exact power requirements are not known. However, Table 4 was constructed to give an estimate of the mission power requirements.

TABLE 4

MARS MISSION POWER REQUIREMENTS

Housekeeping Power	100 KW
Low Voltage Thruster Power	200 KW
High Voltage Thruster Power	3000 KW

The mass of the power management and distribution system will depend upon the actual spacecraft and mission design and thruster concept used. For the concentrated loads, such as the ion thruster beam supply, a projection of present high frequency technology to the early 2000's results in a specific weight of approximately 1 KG/KW. The same projection for the low voltage power supplies for the ion thruster may be in the 5 KG/KW range. Due to the smaller power range and higher complexity, the housekeeping power, with provision for emergency operation and some storage capacity, will be closer to a specific weight of 50 KG/KW. Transmission line weights for a 150 meter line at 3000 volts are expected to be on the order of 3500 KG. It is estimated that the losses incurred in the power management and distribution system are on the order of 500 KW.

TABLE 5

POWER MANAGEMENT AND DISTRIBUTION SYSTEM WEIGHT

Housekeeping Power	5000 KG
Low Voltage Ion Thruster Power	1000
High Voltage Ion Thruster Power	3000
Transmission Line	3500
TOTAL WEIGHT	12500 KG

As indicated in Figure 10, about 4 years are required to advance the state of the art in power management and distribution system components by one decade of power. At the present time - 1985, the power technology level for these components is less than 100 KW. To arrive at the required level of 3000 KW or greater will require approximately 6-8, years provided that these power levels will not require "relatively" more development than the lower power levels. Circuit, systems development, and testing will require approximately 3-4 years additional. Thus, the feasibility of the power management and distribution systems technologies could be established by the turn of the century to support the manned Mars mission.

ADVANCED THERMAL CONTROL

Multimegawatt space power/energy systems will require the control of multimegawatt quantities of waste heat which must be rejected (radiated) to the space environment. Even the most optimistic estimates of the more conventional space radiator concepts such as those proposed for the nuclear power system described in a preceding section of this paper, represent a sizable fraction of the total power system weight (see Table 2).

FIGURE 10. POWER ELECTRONIC COMPONENT DEVELOPMENT AT LEWIS RESEARCH CENTER

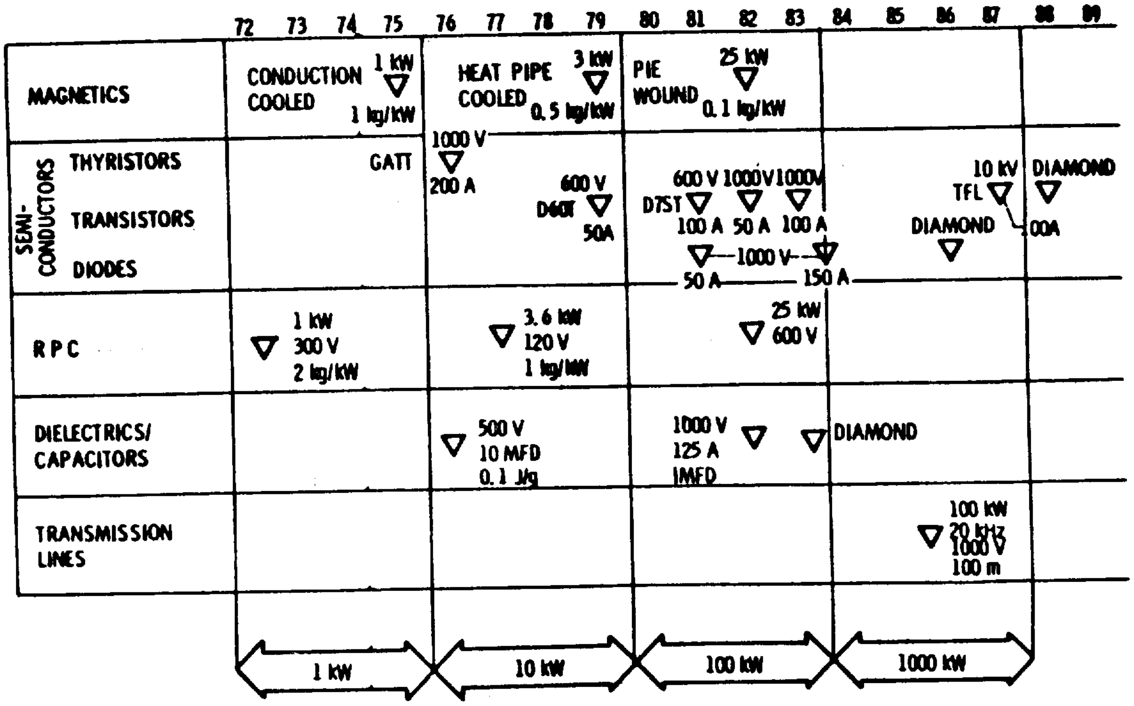
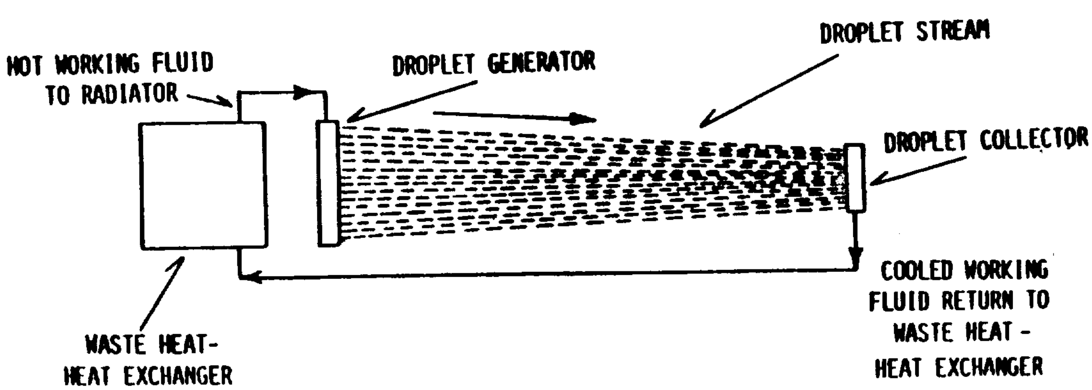


FIGURE 11. LIQUID DROPLET RADIATOR CONCEPT

RADIATIVE "FINS" AND "HEAT PIPES" OF CONVENTIONAL RADIATORS REPLACED BY MULTIPLE STREAMS OF UNIFORM LIQUID DROPLETS



There are a number of advanced heat rejection concepts, under investigation at the Lewis Research Center, which show potential for considerable reductions in radiator system weights in addition to providing greater operational flexibility. These devices are, at present, in the concept feasibility assessment stage and still require extensive development before they can be incorporated into prototype or flight systems.

One such device is the liquid droplet radiator being jointly studied by the Lewis Research Center (NASA) and the Rocket Propulsion Laboratory (Air Force) through an interdependency agreement. A schematic of such a device is shown in Figure 11.

Design approaches to this concept are being investigated at this time. The design of the droplet generator, selection of a fluid compatible with its application and the space environment (operating temperature), lossless collection of the drops, and coalescence of the drops into a pumpable fluid stream for recirculation through the return loop, are areas which will require significant development.

As a result of studies presently underway, it was possible to arrive at preliminary estimates of the parameters which characterize a liquid droplet radiator system. A liquid droplet radiator which would meet the requirements of the nuclear power system described in the first section of this paper would be:

TABLE 6

LIQUID DROPLET RADIATOR PARAMETERS

Fluid	Tin (SN)
Density	6600 KG/M^3
Droplet Sheet Emissivity	0.5
Droplet Sheet Flow Rate	765 KG/SEC
Droplet Sheet Area (One Side)	289 M^2
Droplet Temperature (Generator)	1050° K
Droplet Temperature (Collector)	900°

Preliminary design estimates, (Phase I, Report, Ref.7), indicate that such a radiator could be constructed for 1.72 KG/M^2 of droplet sheet area, (both sides). This estimate includes the hardware items shown in Table 7 below.

TABLE 7

Liquid Droplet Radiator Components
Droplet Sheet
Fluid Inventory
Droplet Generator and Plenum Chamber
Droplet Collector
Return Loop and Pumps
Astro Mast, Boom, Miscellaneous Structure

One design concept of such a radiator is shown in Figure 12. This concept, if sized to meet the heat rejection loads of the nuclear power system of Section I of this paper, would result in a radiator mass of approximately 1000 KG which represents a considerable weight savings over that given in Table 2 (8700 KG). It is felt that at this point in its development stage, no problems have surfaced which indicate that this concept is unfeasible.

Another concept, somewhat similar in principle to the liquid droplet radiator, is the liquid belt radiator which is also being investigated for space power systems applications. One concept of such a device is shown in Figure 13. The primary difference between this concept and the liquid droplet radiator is that the free flowing droplet stream is replaced by a fluid belt upon which the radiating fluid is retained by surface tension forces. The circular shape of the belt is maintained by centrifugal forces as the belt is driven through the plenum chamber where it "picks up " the hot fluid.

The development status of this concept is not as advanced as that of the liquid droplet radiator. Areas which require further investigation are those related to the belt properties, fluid properties, seals, and the operational aspects of this device. The liquid belt radiator may show some operational advantages over the liquid droplet radiator from the

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FIGURE 12. LIQUID DROPLET RADIATOR

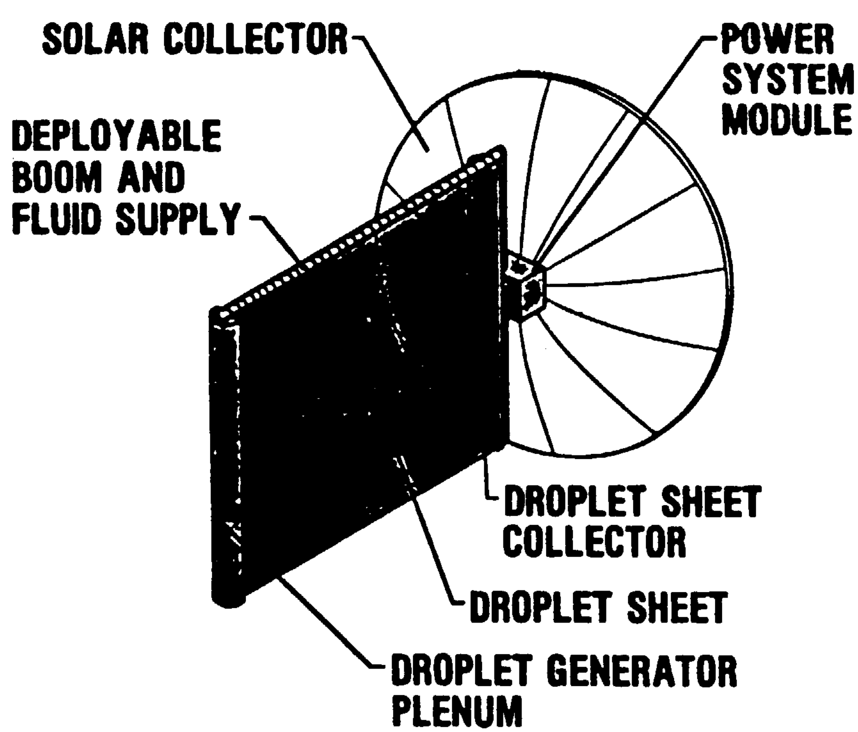
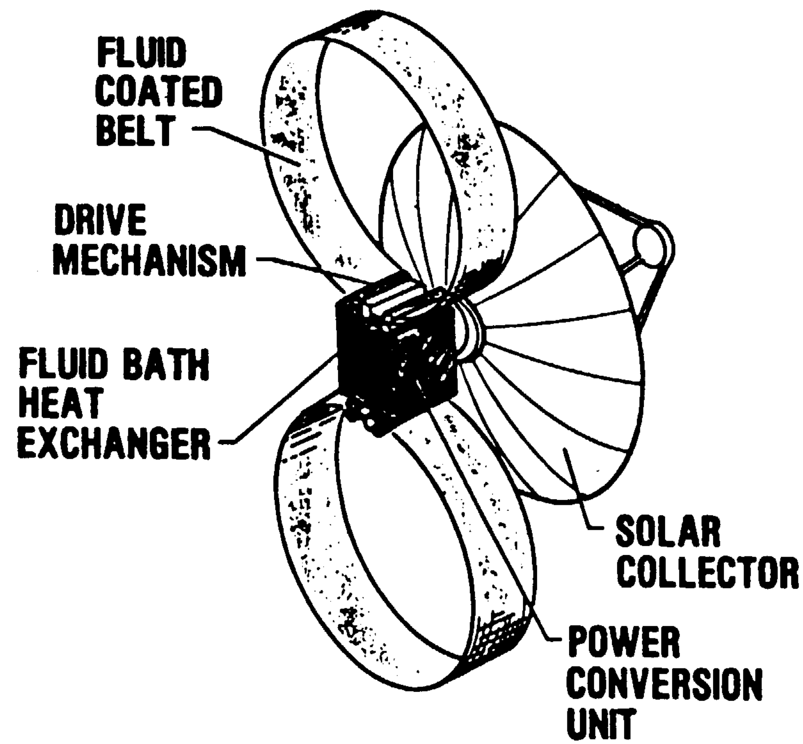


FIGURE 13. LIQUID BELT RADIATOR



viewpoint of management of the radiating fluid. An early projection indicates that such a device can be built at possibly 1.10-1.25% of the weight of a liquid droplet radiator.

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PROPULSION ISSUES, OPTIONS, & TRADES

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ABSTRACT

This paper briefly discusses several different types of propulsion concepts: (1) pulsed fission, (2) continuous nuclear fission, (3) chemical, and (4) chemical boost with advanced upper stage concept. Some of the key characteristics of each type are provided, and typical concepts of each are shown.

COMPARSION OF ADVANCED PROPULSION CONCEPTS

Considerable confusion exists concerning the relative attributes of various advanced propulsion concepts. Figure 1 shows a relative performance comparison of propulsion concepts with respect to important vehicle design parameters.

In general, propulsion concepts to the left of the dashed line result in unsatisfactory trip times for a manned MARS mission because of insufficient vehicle acceleration. However, these advanced propulsion concepts could become feasible if combined with a nuclear or chemical boost from LEO, or if the vehicle starts from a Lunar libration point or GEO, thus reducing Earth escape spiral time. For Mars missions there is little advantage for low thrust if it is necessary to boost to escape from LEO. The libration points or GEO options are mission design options beyond the scope of this paper. The discussion herein is therefore restricted to: (1) pulsed fission, (2) continuous nuclear fission, (3) chemical, and (4) chemical boost with advanced upper stage concept.

NUCLEAR FISSION PULSE PROPULSION

Nuclear fission pulse propulsion was studied extensively as a space transportation device from 1958 until 1965 under project Orion. An illustration of the NASA Orion vehicle, sized for compatibility with the Saturn V launch vehicle, is shown in Figure 2. This vehicle, according to reference 1, would be capable of completing a manned Mars surface-excursion mission from a single Earth launch, using a Saturn first stage. For this mission, the nuclear pulse propulsion would begin at suborbital velocity, starting at an altitude greater than 100 km (50 n mi). The vehicle shown has an estimated specific impulse of 2500 sec, a dry mass

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Figure 1. Advanced Propulsion System Performance

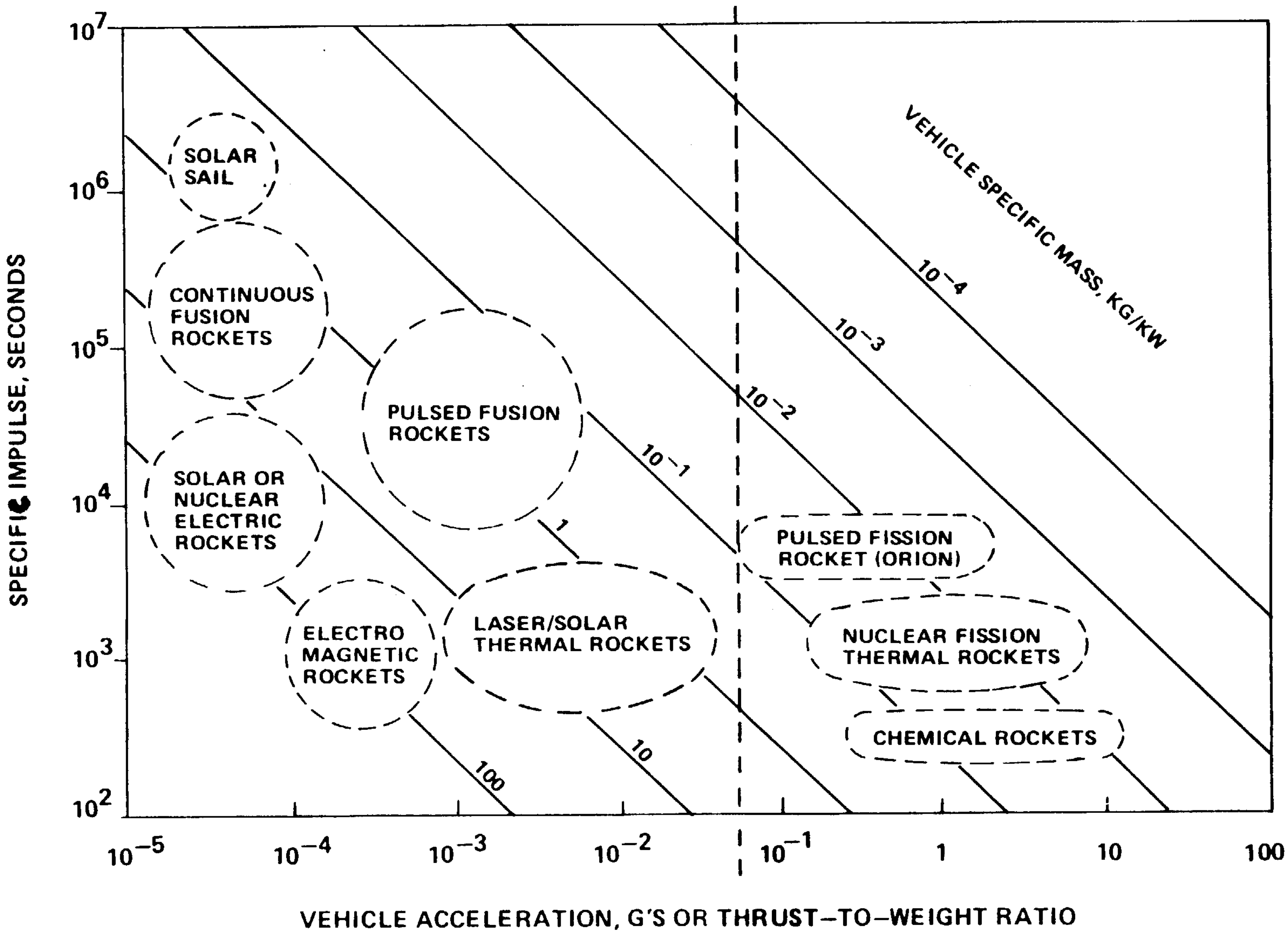

Figure 2. Summary of Nuclear Fission Pulsed Rocket (Orion) Characteristics

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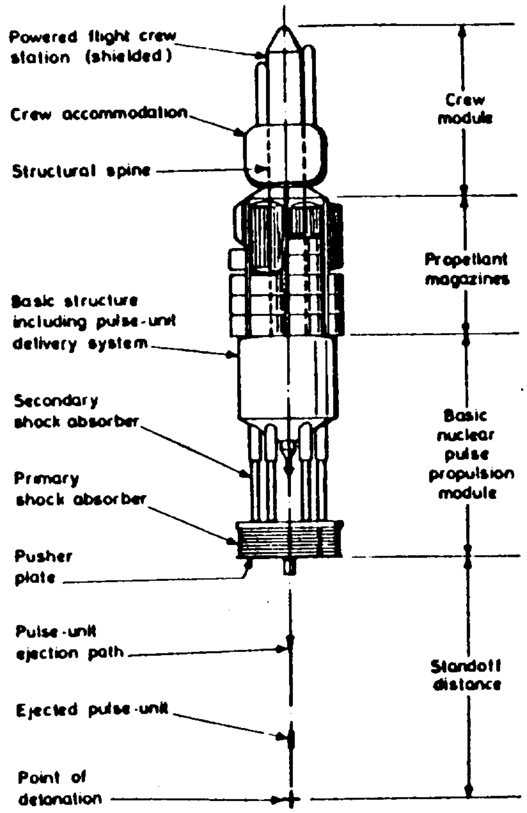

PERFORMANCE

- I_{sp} = 2000-6000 SEC
- ENGINE T/W~4

} DEPENDS ON DIAMETER OF PUSHER PLATE

OPERATIONAL CAPABILITY

- VERY LONG LIFETIME
- CAN BE MAINTAINED
- OPERATION BELOW GEO QUESTIONABLE (PROJECT STARFISH)

RISK/FEASIBILITY ISSUES

- DATA BASE BELIEVED GOOD (CLASSIFIED)
- NUCLEAR TEST BAN TREATY PROHIBITS TESTING
- HIGH DEVELOPMENT COST MODERATE RADIATION HAZARD

RECOMMENDATION

THIS CONCEPT HAD THE BEST ALL-AROUND PERFORMANCE OF ANY CONCEPT EXAMINED, BUT FEASIBILITY FOR NEAR EARTH MISSIONS NEEDS FURTHER INVESTIGATION. THIS CONCEPT RECOMMENDED FOR FUTURE STUDY CONSIDERATION

Figure 3. Summary of Nuclear Fission Thermodynamic Rocket Characteristics

REACTOR TYPE	PERFORMANCE	OPERATIONAL CAPABILITY	RISK/FEASIBILITY ISSUES
SOLID CORE REACTOR	800 - 1000 SECS ENGINE T/W ≃ 3	LIMITED LIFETIME POOR MAINTAINABILITY	RADIATION HAZARD FROM USED ENGINE: MUCH DESIGN DATA AVAILABLE.
ROTATING BED REACTOR	1000 - 1200 SECS ENGINE T/W ≃ 6	LIMITED LIFETIME CAN BE SERVICED	RADIATION HAZARD MODERATED BY CORE REMOVAL; DESIGN LEVEL TECHNOLOGY AVAILABLE.
LIQUID CORE REACTOR	1400-1600 SECS ENGINE T/W ≃ 1	VERY SHORT LIFETIME ONE SHOT MISSIONS	NO CONTAINMENT OF FISSION PRODUCTS: VERY LIMITED DATA BASE.
OPEN-CYCLE GAS-CORE REACTOR	1500 - 2000 SECS ENGINE T/W ≃ 1	LONG LIFETIME, BUT MUST BE REFUELED EVERY BURN	NO CONTAINMENT OF FISSION PRODUCTS: GOOD DATA BASE BUT FEASIBILITY NOT PROVEN.
CLOSED-CYCLE GAS CORE REACTOR	1500 - 2000 SECS ENGINE T/W ≃ 1	LIFETIME UNKNOWN CAN BE SERVICED	"LIGHTBULB" EXTREMELY HIGH RISK GOOD DATA BASE BUT FEASIBILITY NOT PROVEN.

RECOMMENDATION

SOLID CORE AND ROTATING BED REACTORS SHOULD BE CARRIED INTO TASK 2.

of 90,000 kg (200,000 lb), and an effective thrust level of 3,470,000 N (780,000 lbf).

Unfortunately, the same grounds used in 1965 to terminate the original Orion project are still valid today. For instance: (a) The large size and power of the vehicle made full-scale tests difficult and very expensive (final testing in space required); and (2) The 1963 nuclear-test-ban treaty specifically excluded nuclear explosions in the atmosphere or in space.

NUCLEAR FISSION THERMODYNAMIC ROCKET

The characteristics of five types of nuclear fission thermodynamic rockets are summarized in Figure 3. Much work was expended on these concepts prior to abandonment of the U.S. nuclear rocket program in 1973 and, for most of these concepts, the data base is quite good. Of the five concepts, the solid-core and rotating-bed rockets are recommended for vehicle-level assessment. The liquid-core reactor was dropped for not being reusable, the open-cycle gas-core reactor was dropped for being too large and too expensive to operate in near-term applications, and the closed-cycle gas-core or "light bulb" reactor was dropped because of feasibility issues concerning the light bulb.

CHEMICAL PROPULSION

Space vehicle design work at MSFC in 1985 has centered primarily on the cryogenic system, utilizing liquid oxygen/liquid hydrogen as propellants. Advanced engine candidates include the STME 625 (SSME derivative) for Stage 1 engines and the advanced expander cycle engine (RL-10 derivative) for Stage 2 and Stage 3 engines.

The storable propellant option utilizing nitrogen tetroxide/monomethyl hydrazine as propellants has been pursued to alleviate the cryogenic propellant boil-off problem; however, the storable propellant option has a significant vehicle weight penalty compared to the cryogenic. Figure 4 depicts typical chemical propulsion engine concepts.

MULTIPLE ENGINES SIMPLIFY ATTITUDE CONTROL

Consider the space vehicle of reference 2 as depicted in Figure 5. Note that Stage 2 and Stage 3 have single main engines. If these engines were replaced with two or more smaller engines with gimballing capability, the outbound midcourse correction system, the inbound midcourse

1274-85

SYSTEM	CHEMICAL	
	CRYOGENIC	STORABLE
SPECIFIC IMPULSE	461–482	327–346
ENGINE THRUST–TO–WEIGHT	38–78	29–41
PROPELLANT	LOX/LH_2	N_2O_4/MMH

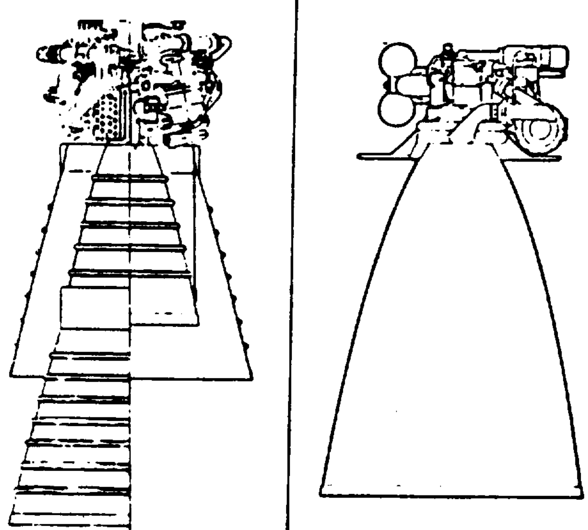

Figure 4. Summary of Chemical Propulsion Characteristics

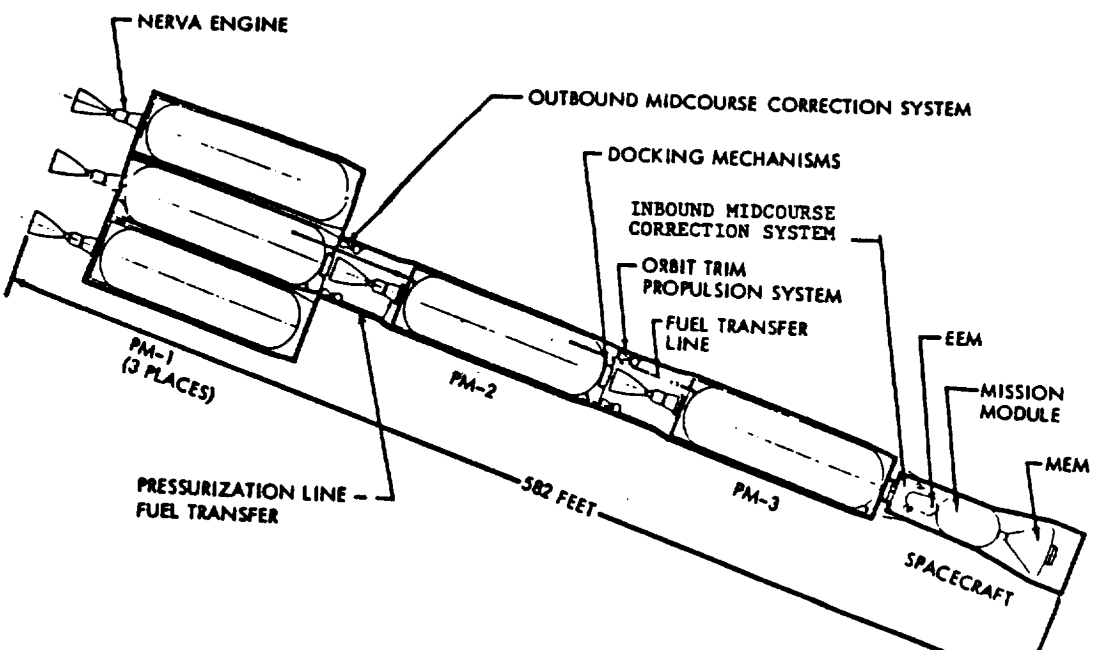

Figure 5. Space Vehicle Design of 1968.

correction system, and the orbit trim propulsion system could be eliminated.

The resulting vehicle design would be much simpler, with three fewer propulsion systems as well as an "engine-out" capability. A small chemical attitude control system could be incorporated to handle small correction maneuvers, rather than restarting the reactor.

MARS EXCURSION MODULE (MEM) DESCENT/ASCENT ENGINE OPTIONS

Early studies (ref.3) investigated the trade-offs between plug nozzle engines and bell nozzle engines. The envelopes of these engine types is shown in Figure 6. Propellant combinations evaluated were OF_2/MMH, FLOX/CH_4, and LO_2/LH_2.* Plug nozzle engines were baselined at that time in order to fit the MEM envelope.

MSFC studies in 1985 have centered around engine types and propellant combinations which are closer to state-of-the-art. Two engine designs were evaluated, both utilizing two-position nozzles. A summary of the performance characteristics of these engines is shown in Figure 7.

*These formulae and acronyms denote:

OF_2	oxygen difluoride
MMH	monomethyl hydrazine
FLOX	a mixture of liquid fluorine and liquid oxygen
CH_4	methane
LO_2	liquid oxygen
LH_2	liquid hydrogen

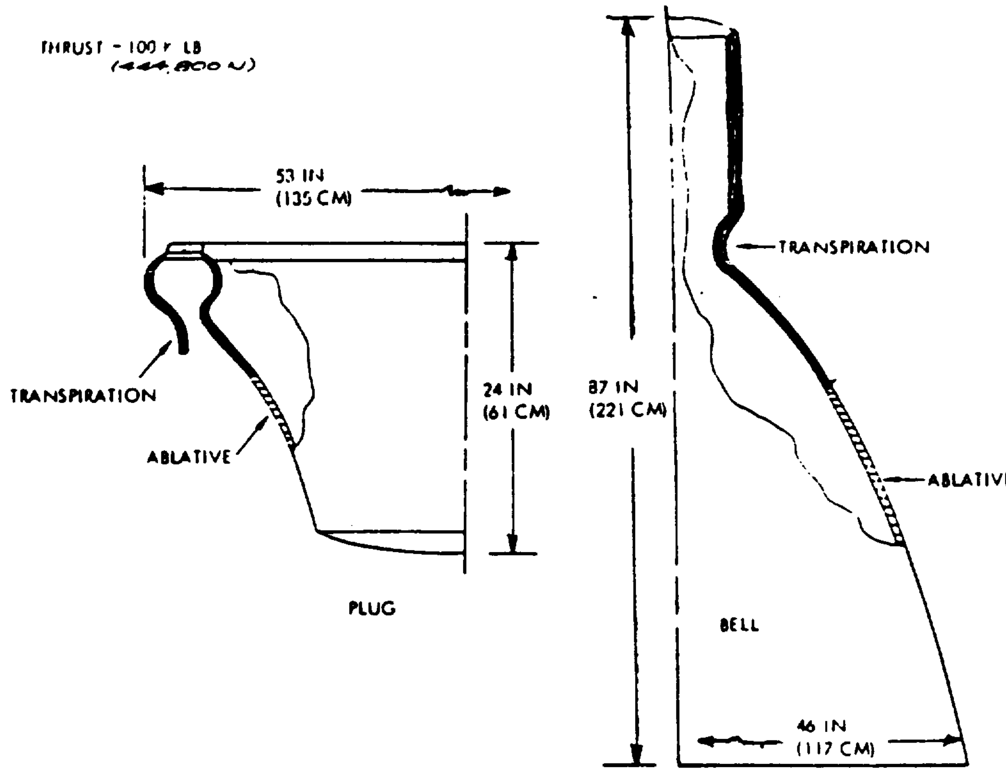

Figure 6. Plug Nozzle and Bell Nozzle Envelope

MEM DESCENT/ASCENT ENGINES

	BASELINE	OPTION
• PROPELLANTS	LOX/MMH	N_2O_4/MMH
• NOZZLE AREA RATIO (FIXED/EXTENDED)	30/75	30/75
• VACUUM THRUST (LBF)	40K	40K
• CHAMBER PRESSURE (PSIA)	1430	1430
• MIXTURE RATIO (O/F)	1.7	2.0
• DEL ISP VAC (SEC)	360.5	328.6
• LENGTH (IN)	52.6/76.8	53.7/78.4
• DIAMETER (IN)	37.6	38.5
• DRY WEIGHT (LBM)	555	573

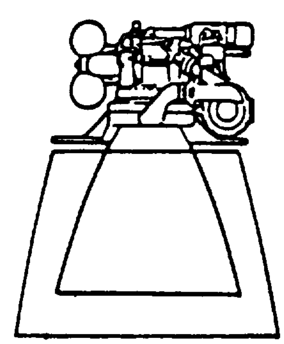

Figure 7. Two-Position Nozzle Designs

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N87-17791

ROTATING BUBBLE MEMBRANE RADIATOR FOR SPACE APPLICATIONS

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ABSTRACT

An advanced radiator concept for heat rejection in space is described which uses a two-phase working fluid to radiate waste heat. The development of new advanced materials and the large surface area per mass makes the Bubble Membrane Radiator an attractive alternative to both conventional heat pipes and liquid droplet radiators for mid to high temperature applications. A system description, a discussion of design requirements, and a mass comparison with heat pipes and liquid droplet radiators is provided.

INTRODUCTION

By the turn of the century, electrical power requirements for space activities will increase significantly as additional power is needed for orbiting space stations and platforms, electric propulsion systems, communication facilities, space based radars, and other proposed commercial and military applications. To meet this increased demand for power, solar dynamic and nuclear power systems, which operate on a closed heat engine cycle or use direct conversion of thermal to electric power, are being investigated for their significant reduction in size and mass over comparable photovoltaic systems. This reduction in power system mass and size may translate into reduced initial and life cycle costs as well as improved orbital operations in the areas of stability, control, and maintenance.

For any space-based activity, waste heat must ultimately be radiated to space. Spacecraft system studies by NASA and industry have shown that heat rejection radiator systems are a major weight and volume contributor to any power or thermal management system. The optimumal design and development of future power or thermal management systems will require advanced heat rejection concepts utilizing new and innovative approaches to reduce overall system mass and size, while increasing system efficiency and thermodynamic performance. These advanced heat rejection systems will be required to withstand the detrimental effects of meteoroid and space debris impact, radiation, and ionizing atoms, in addition to addressing such pertinent mission requirements as: reliability and maintainability, operation and control, system integration and life-cycle cost. Current research and development efforts are being focused on heat pipe and liquid droplet radiator technologies.

ROTATING BUBBLE MEMBRANE RADIATOR SYSTEM DESCRIPTION

An alternative approach to the problem of space-based heat rejection is the Rotating Bubble Membrane Radiator (RBMR). The RBMR is a hybrid radiator design which incorporates the high surface heat fluxes and isothermal operating characteristics of conventional heat pipes with the low system masses associated with liquid droplet radiators. The RBMR is designed to take full advantage of the microgravity environment of space through the integration and selection of components, and the elimination of mechanical phase separators.

The Rotating Bubble Membrane Radiator is an enclosed two-phase direct contact heat exchanger consisting of nine major components as shown in Figure 1. These components are: the attachment boom, rotation platform, central rotating shaft, central spray nozzle, thin film radiating surface, fluid collection troughs, return pumps, return piping and structure, and main feed/return lines.

Though the RBMR is capable of working with single-phase fluids by incorporation of a heat exchanger within its rotation platform, the operations discussion will be limited to a radiator system conceptualized for a two-phase working fluid. In this system, the two-phase working fluid enters the radiator via feed lines incorporated within the central axis of rotation. The working fluid is then ejected from the central spray nozzle as a combination of liquid droplets and vapor into the radiator envelope. Within this envelope, both convection and radiation heat transfer occurs between the droplets, the vapor and their cooler surroundings, with the dominant mode of heat transfer being radiation. As the droplets move radially outward, they grow in size by the condensation of vapor upon their surface and by collision with other droplets before striking the thin liquid film on the inner surface of the radiator. Once assimilated into the thin surface film, the working fluid begins to flow from the poles of the sphere toward the equator due to the rotationally induced artificial gravity. Heat transfer between the fluid and bubble radiator then becomes a combination of conduction and convec-

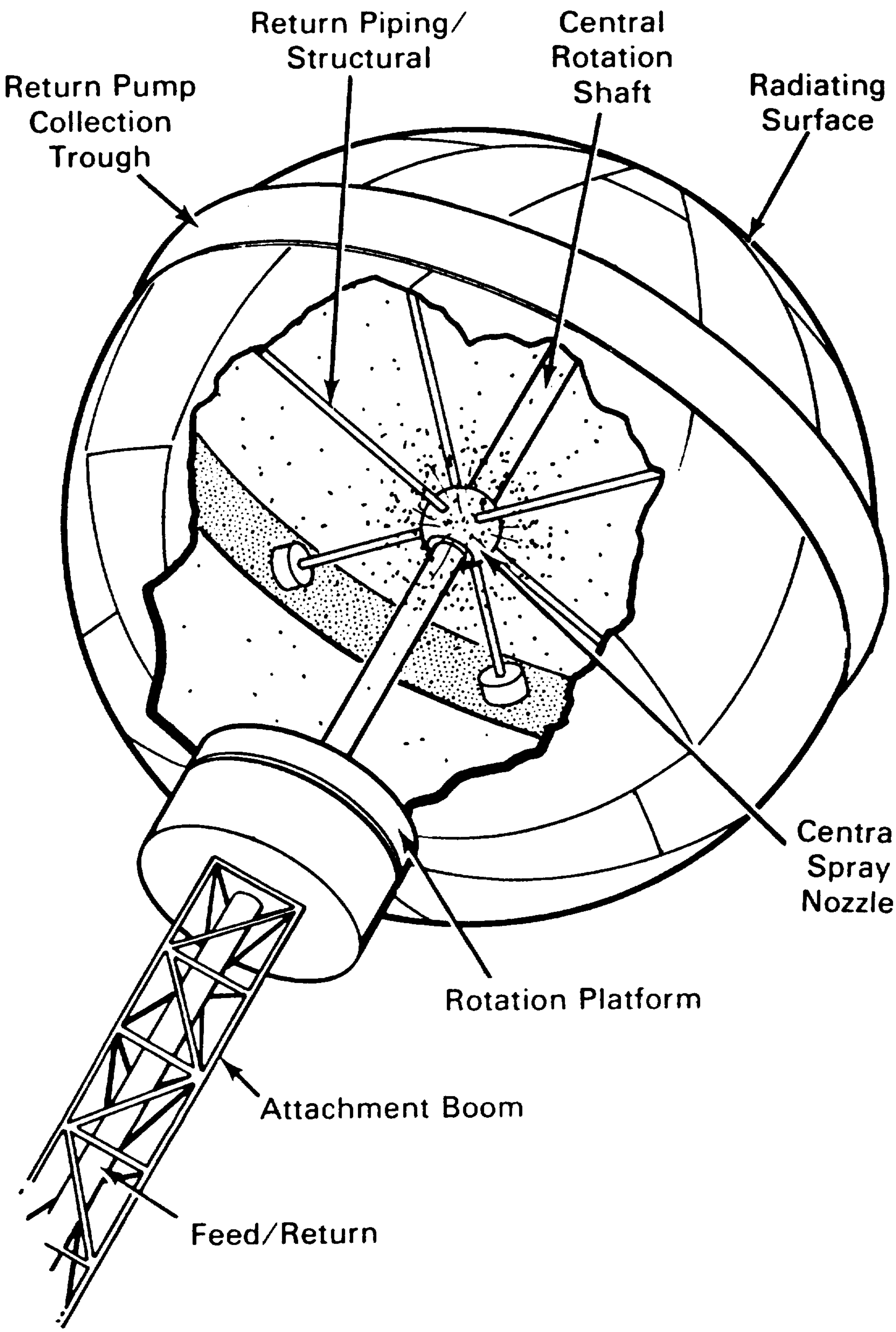


FIGURE 1

BOOM MOUNTED ROTATING BUBBLE MEMBRANE RADIATOR tion. As the fluid reaches the equator of the sphere, it is collected in gravity wells (throughs) and pumped back to repeat the process.

To operate reliably in space, the RBMR will include design features to minimize tears and mitigate coolant losses that may result from meteoroid and space debris impacts. To minimize tearing of the radiator surface, structural support filaments will be incorporated into the radiator surface material. Coolant losses will be mitigated by incorporating self-sealing features which utilize system rotation. By developing self-sealing options substantial mass savings may be realized over other designs which provide protection against meteoroid impact by armoring the heat transfer surface.

Prime candidate materials for the thin film envelope include epoxy-carbon, zirconium and titanium alloys, and niobium-tungsten composites with final selection of the envelope material depended upon the radiator fluid and its intended operating temperature. Pump selection will also be determined by radiator fluid with EM pumps as possible candidates for liquid metal coolants, and mechanical or electric pumps favored for other applications.

RBMR SYSTEM ANALYSIS AND DESIGN CONSIDERATIONS

Primary design considerations for the RBMR were centered on the minimization of overall mass and payload volume and meteoroid survivability. Since all of these criteria are interrelated, a tradeoff study was performed based on current heat pipe and liquid droplet designs to determine the limitations and features of each type of radiator.

Heat pipe radiators, in general, consist of a circular pipe with a layer of wicking material covering the inner surface and leaving a central void region. The working fluid is added to this configuration and by capillary action per- meates the wicking material. When heat is added to the evaporator end of the pipe, the working fluid is vaporized and driven to the central void region. At the condenser end of the pipe, the vapor condenses back to liquid in the wicking material as heat is removed from the pipe structure via radiation to space. The liquid is then returned to the evaporator by capillary action through the wick.

The design review of the heat pipe radiator found that its design often incorporated armor plating to minimize the damage from meteoroid impact and that the orientation of the heat pipe during acceleration or deacceleration may directly affected its performance. The addition of this armor plating increases the overall system mass and the required payload volume. To compensate for the increase of mass due to the armor plating, efforts are underway to increase the operating temperature to minimize the mass to heat rejection ratio of the system. This in turn restricts the efficient usage of heat pipes from low temperature operations.

The liquid droplet radiator uses a stream of heated liquid droplets to radiate waste heat directly to space. A liquid droplet radiator consists of a droplet generating device, a collection device, makeup working fluid storage tanks, and return piping. The radiator fluid is typically heated by means of a heat exchanger and an additional working fluid. The radiator fluid is passed through a pressurized plenum, where it is ejected as jet streams into space towards a collector. Because of surface tension effects the fluid streams quickly break down into droplet form. The collector captures the droplets, pressurizes them, and returns them back to the heat exchanger.

The liquid droplet radiator was able to minimize its mass by spraying the radiator fluid directly to space. The design was found to require little payload volume and high meteoroid survivability, but it required extensive radiator fluid resupply for long-term continuous operations because of fluid losses to space. To reduce these losses, present research is focused on retention of radiator fluid by proper aiming of the droplet streams, minimizing evaporation, and efficient collection.

The ideal system, it was reasoned, would have as little structural mass as possible, be enclosed to reduce radiator fluid losses to space, have high surface heat fluxes, be self sealing, and self deployable. To achieve the high surface heat fluxes, a two-phase system was selected. Two-phase direct radiators offer higher surface heat fluxes then single phase indirect radiators, the overall system benefits from mass and volume savings. Initial estimates are that the RBMR is capable of heat rejection equivalent to a single-phase system, but with one-fourth the fluid mass and one-twentieth the mass flow requirements for the same operating temperatures.[1]

To reduce radiator fluid losses and structural mass and be self-deploying, a sphere or bubble type structure with support and return flow spokes was selected. To aid in the collection of the radiator fluid, artificial gravity was introduced into the system through rotation. Surface punctures could then be sealed by thin interior sheets that rely on pressure and gravity gradients to hold them in place.

The assumptions for the problem were:

Reactor Power (P)	1.11 MW
Waste Heat to Reject (Qr)	1.01 MW
Radiator Temperature (Tr)	775 K (935 F)
Temperature of Space (Ts)	0 K (-453 F)
Emissivity of Radiator	.9
Stefan-Boltzman Constant	5.67E-8 W/m^2-K^4
Percent Error Introduced By Ignoring Solar and Earth Radiation (2)	5 %

The equation for finding the radiating surface required for the heat load is then:

$$\text{Area} = 1.05 * Qr/0.9 * 5.67E-8 * (Tr^4 - Ts^4) \text{ sq. meters}$$

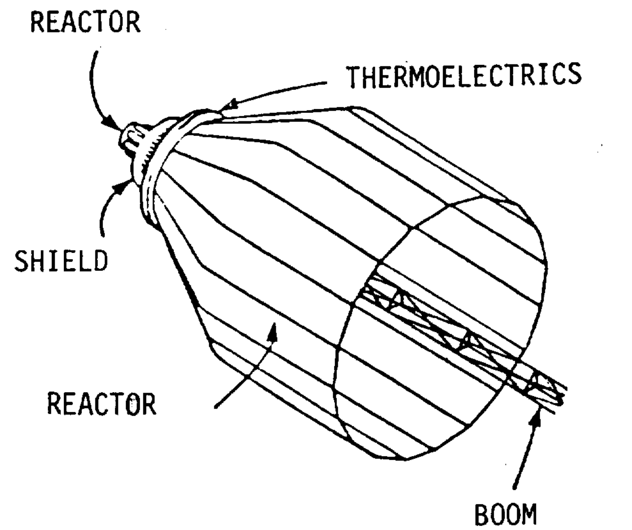
$$\text{Area} = 57.18 \text{ sq. meters}$$

This surface area then corresponds to a sphere with a radius of 2.13 meters (6.98 feet) to reject 1.01 MW of waste heat. The RBMR system mass, based on an 0.15 mm (0.006 in)-thick thin film envelope is then estimated to be approximately 91 to 137 kg (200 - 300 lbs) and is envisioned as being a small self-deployable unit.

MASS COMPARISON WITH OTHER HEAT REJECTION SYSTEMS

As a comparison of radiator system masses, the RBMR and the liquid droplet radiator have been directly substituted for the heat pipe radiator used for the 100 KWe Thermoelectric/SPAR Reactor (see Figure 2). A similar comparison previously made between the liquid droplet and heat pipe radiators was used as the basis of comparison of the RBMR.[2] A table of total system mass, estimated radiator masses, specific powers, and percent mass savings is given in Table 1.

The results of this comparison indicate that the RBMR is approximately four times lighter then today's heat pipe radiators while being four times heavier than a liquid droplet radiator with no operational coolant losses to space.



Source Temperature	1800 K
Hot Shoe Temperature	1400 K
Cold Shoe Temperature	775 K
Radiator Temperature	775 K
SPAR Reactor Power Rating	100 KWe
Total System Mass	1465 kg
Radiator Mass	455 kg
Specific Power with Heat Pipe Radiator	31 W(e)/lb

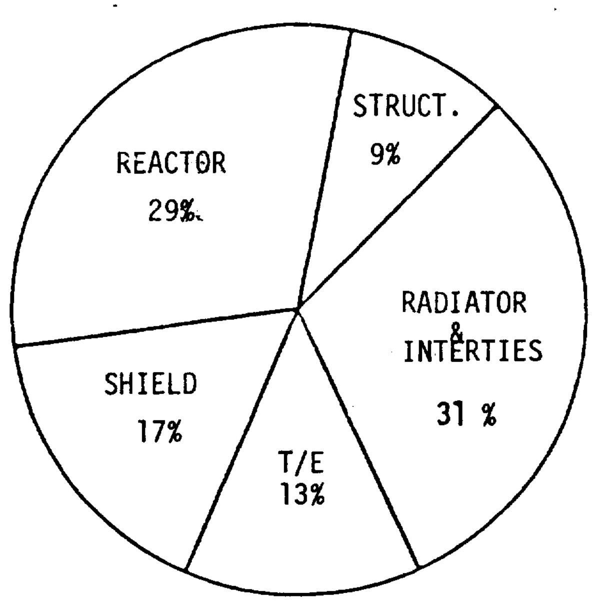


FIGURE 2. **Mass Distribution and Schematic for 100 kWe Thermoelectric/SPAR Reactor**

TABLE 1

SPAR REACTOR SYSTEM COMPARISON

Radiator Type:	Heat Pipes	RMBR	Liquid Droplet
Total System Mass	1465 kg	1103-1148 kg	1033 kg *
	(3226 lb)	(2426-2526 lb)	(2273 lb)
Radiator Mass	455 kg	91-136 kg	21 kg
	(1000 lb)	(200-300 lb)	(47 lb)
Specific Power (W(e)/lb)	31	39- 41	44
% Mass Savings	-	23	31

* Note: This value does not include system coolant mass estimates to compensate for coolant losses to space, since these are dependent upon the operating temperature, history and alignment of the system.

<u>RBMR APPLICATIONS</u>

As currently envisioned, the Rotating Bubble Membrane Radiator is capable of a broad range of space applications.

* As a thermal management device, the RBMR is suited to both high and low temperature applications through the proper selection of working fluid and radiator component materials.

* For power generation applications, the design can be modified by selection of materials to accommodate solar dynamic or nuclear two-phase systems.

* Using a boom-mounted configuration, the RBMR can be substituted for the type of heat pipe radiator panels currently envisioned for the NASA space station and orbital platforms. Because of its design, the boom-mounted RBMR need not require a gimbal attachment nor sensors for continuous alignment of the radiator to minimize solar heat input.

*The RBMR can be used as an integral part of advanced space apropulsion units (See Figure 3).

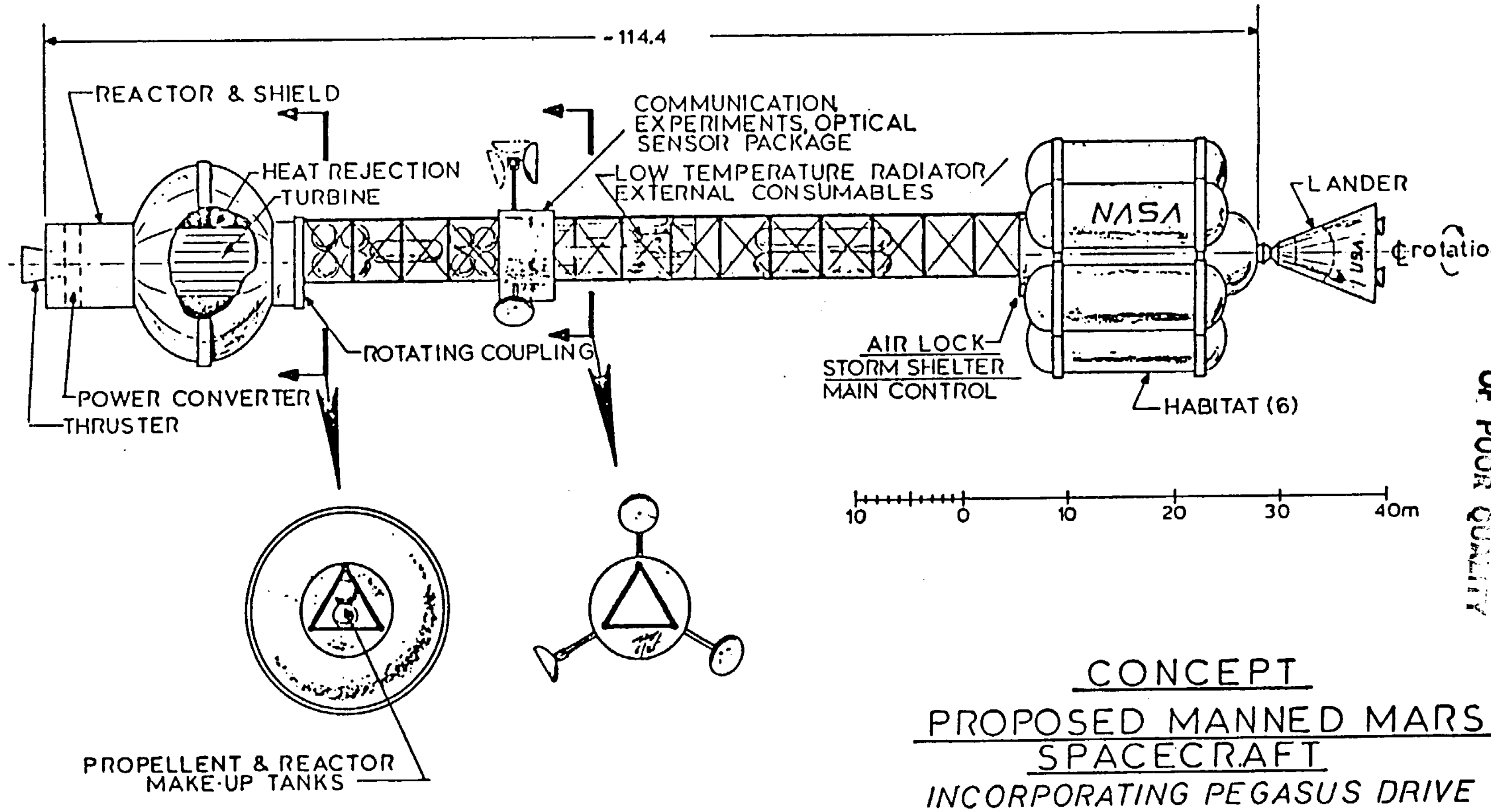


FIGURE 3. Rotating Bubble Membrane Radiator Incorporated in Spacecraft Design

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N87-17792

TELECOMMUNICATIONS AND RADIO-METRIC SUPPORT FOR A MANNED MISSION TO MARS[1]

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ABSTRACT

This paper describes some general characteristics of the Deep Space Network and relates them to services needed by a manned mission to Mars. Specific details of the Network's current capabilities and those planned for the near future may be found in the reference.

DISCUSSION

NASA's Deep Space Network is a multimission telecommunications and radio-metric facility used to support space science and exploration. The Network is physically located on three continents, i.e. in southern California, U.S.A.; near Madrid, Spain; and near Canberra, Australia, and provides nearly complete coverage of deep space. It is designed and managed overall by California Institute of Technology's Jet Propulsion Laboratory (JPL), and its California facilities are installed and operated by contractors to JPL. The Network's overseas facilities are managed locally and operated by agencies of the Spanish and Australian governments via international agreements.

The Network's basic services are telemetry reception, command transmission, and radio-metric data (position and velocity) acquisition. The ability to provide these services enables the Network also to perform flight radio science, radio astronomy, very long baseline interferometry (VLBI), geodynamics measurements, and searches for extraterrestrial intelligence (SETI). It's very stable radio-metric instruments have been used to attempt gravitational wave detection, and its long-term radio-metric measurements of the Viking Landers contributed to tests of general relativity.

The Network transmits at frequencies of 2025 to 2120 MHz and receives at frequencies of 1659 to 1675 MHz, 2200 to 2300 MHz, and 8400 to 8500 MHz. Transmission at 7145 to 7190 MHz will be provided for

[1]This work was performed by Jet Propulsion Laboratory, California Institute of Technology; Pasadena, California; under contract to the National Aeronautics and Space Administration.

Project Galileo in 1987, and spectrum space has been allocated for deep space communications in the vicinity of 32 GHz downlink and 34 GHz uplink. Space-based antennas, both microwave and optical, have been considered from time to time and may become a reality in the 21st century.

There are three or four antennas at each site, i.e. California, Madrid, and Canberra. The most sensitive antenna system at each site can receive data from Voyager-class spacecraft (20 Watts transmitter power and 48.2 dB antenna gain at 8400 to 8500 MH_z) at rates of 115.2 kbps from Jupiter or 44.8 kbps from Saturn. The Network's other antenna systems are also very sensitive. If, for example, a Mars mission has a 1.5-meter directional antenna and a 100-W transmitter operating within 8400 to 8500 MHz, the Network's regular 34-meter antenna system can simultaneously receive digital voice communications (9.6 kbps), 16 kbps telemetry, and 16.4 kbps slow scan TV (8 seconds per black and white picture of 128 x 128 pixels). Three to four times this basic total amount of data can be received using the Network's most sensitive antennas, and about five times the basic amount can be received if antennas are arrayed.

Network transmitters operate at 20 kW to 100 kW, depending upon mission need. These transmitted power levels are up to 10^{28} times greater than the received telemetry power level cited in the previous paragraph and provide ample uplink communications capability to an omnidirectional antenna at Mars and substantial signal-to-noise ratios at Mars, thereby enabling satisfactory radio-metric measurements. A transmitter capable of 400 kW is also available for emergency situations.

With present or currently planned equipment, the Network can process downlink data at rates of 6 bps to 5 Mbps with an undetected bit error rate of 1 part in 10^6 or less. The Network achieves a frequency stability of 7 parts in 10^{14}, which enables radio-metric data precisions in angle and angle rate of 50 nanoradians and 5 picoradians per second, respectively, and in range and range rate of 5 meters and 2 millimeters per second, repectively. Together, these capabilities allow important position and velocity measurements at Mars.

Beginning with its first antenna in 1958, the Network has been managed as an evolving capability. New technologies have been planned as potential needs appeared and then implemented as these needs were confirmed. Equally important, Network research and development has often identified opportunities for new telecommunications or radio-metric capabilities, which were implemented when user requirements arose. The Network's objective has always been to offer the most capable and cost-effective telecommunications and radio-metric service possible, integrated over all Network users.

Current Network research and development and planning studies are considering the following capablities: reception in selected frequency bands over the entire range of 1 to 50 GHz, 500 MHz instantaneous band-widths, communications coding techniques that will double link efficiencies, 1 part in 10^{17} frequency stability, and 5 nanoradian radio navigation by very long baseline interferometry (VLBI), as well as the orbiting antennas mentioned above. These capabilities will be developed whenever users require.

As indicated above, the Network's current technology can amply support both the communications needs and the interplanetary and local navigation needs of a manned mission to Mars. New capabilities installed to meet the requirements of future missions will extend that ability before the turn of the century.

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N87-17793

ANTIMATTER PROPULSION: STATUS AND PROSPECTS

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ABSTRACT

The use of advanced propulsion techniques must be considered if the currently envisioned launch date of the manned Mars mission were delayed until 2020 or later. Within the next 30 years, technological advances may allow such methods as beaming power to the ship, inertial-confinement fusion, or mass-conversion of antiprotons to become feasible. A propulsion system with an ISP of around 5000 s would allow the currently envisioned mission module to fly to Mars in 3 months and would require about one million pounds to be assembled in Earth orbit. Of the possible methods to achieve this, the antiproton ($\bar{p}$) mass-conversion reaction offers the highest potential, the greatest problems, and the most fascination. Antiprotons are currently being produced in the world at the rate of about 10^{14} particles per year. Based on the past 30 years of production experience, antiproton production rates have increased by an order of magnitude every 2.5 years. If this trend continues, almost a mg/yr (6×10^{20} particles) could be produced by the early 2000's. To accomplish this level of production, significant progress needs to be made in accelerator technology. Increasing the production rates of antiprotons is a high priority task at facilities around the world. Rapid progress can be expected in the shorter term. Antiprotons are currently stored in large synchrotron rings. By lowering the particle energy, storage can be achieved in compact structures known as ion traps. Current experiments plan to decelerate and capture up to 10^{10} antiprotons in such as trap. The storage capability of ion traps is limited. However, these traps will provide a source of sub-thermal $\bar{p}$'s for development of better storage mechanisms suitable for propulsion. The application of antiprotons to propulsion requires the coupling of the energy released in the mass-conversion reaction to thrust-producing mechanisms. In addition, there are recent proposals which would enhance the average energy released per $\bar{p}$ used. These proposals entail using the $\bar{p}$'s to produce inertial confinement fusion or to produce negative muons which

can catalyze fusion. By increasing the energy released per $\bar{p}$, the effective specific cost, (dollars/joule) can be reduced. These proposals and other areas of research can be investigated now. These short term results will be important in assessing the long range feasibility of an antiproton powered engine.

INTRODUCTION

The type of propulsion system used on Mars missions may depend on when a particular ship is launched. If the mission is launched later than around 2010, several currently envisioned advanced propulsion concepts may be feasible and could be utilized for improved mission performance.

An advanced propulsion system would offer the potential for reducing (1) the required total ship mass to be assembled in Earth orbit for a given payload mass; (2) the total amount of material and the costs of launching the material from Earth's surface to orbit; and (3) the round trip transit time from years to a few months.

Within the next 30 years, technological advances may allow systems with a specific impulse (I_{sp}) of 2000-5000 s and with thrusts of around a meganewton to be developed. The effects that such a system could have on a Mars mission are shown in Table 1. To duplicate the baseline mission profile of 360 days outbound-260 days return for a 100 ton payload, about 220 metric tons of mass would be required in Low Earth Orbit (LEO). By comparison, the chemical propelled system (LO_2/LH_2) would require about 1800 metric tons. If a shuttle based delivery system is used, i.e. 65,000 lbs/launch, the LEO mass requirements imply 61 launches for the chemical system compared to 8 launches for an advanced propulsion system.

In addition to the tremendous reduction of the required LEO mass, high I_{sp} systems also offer the possibility of faster transit times. The LEO mass requirements for a 1 yr round trip mission and a 6 month round trip time are also shown in Table 1 and are about 308 metric tons and 422 metric tons, respectively. Thus, a round trip time of 6 months could be accomplished for less total mass than is currently estimated for the chemically propelled 680-day mission.

The reduced trip time may be necessary in view of the physiological and psychological responses of the Russian cosmonauts after 239 days of weightlessness. If less than 100 days of weightlessness were endured, a

TABLE 1

MARS MISSION

Mass Comparison (Klbs)

	Chemical Propulsion	Case 1[a]	Case 2[b]	Case 3[c]
EOI				
Payload	112.69	112.69	112.69	112.69
Engine	.78	100	100	100
Structure	16.88	3.84	8.05	11.74
Propellant	198.92	19.21	40.30	58.70
Δvelocity (km/s)	3.72	3.72	7.40	10.0
TEI				
Structure	28.46	1.81	8.57	15.63
Propellant	183.13	9.00	42.86	78.14
Δvelocity (km/s)	1.62	1.62	6.50	10.0
MOI				
MEM	128.20	128.2	128.2	128.2
Structure	26.50	4.94	14.47	27.88
Propellant	694.71	24.70	72.36	139.41
Δvelocity (km/s)	2.76	2.76	6.50	10.0
TMI				
Probes	24.48	24.48	24.48	24.48
Structure	148.68	9.30	20.92	38.47
Propellant	3105.70	46.54	104.58	192.34
Δvelocity (km/s)	4.43	4.43	7.40	10.0
TOTAL MASS	4667.00	484.67	677.44	927.7

[a]360 day outbound/200 day return/60 day stay.

[b]1-yr round trip - 20-day stay.

[c]3-month each way transit.

duration about equal to the U.S. Skylab experience, the requirements for closed environment life support systems (CELSS) and for artificial gravity might be reduced. As a result, the overall complexity of the ship design might be reduced.

Several possible types of advanced propulsive systems have been proposed over the last few decades. Low-thrust electric or variations of the nuclear-thermal rocket are not considered here because they are either under development or are already developed and are not advanced concepts. The truly conceptual designs can be grouped into beamed-power propulsion and improved specific-energy density concepts.

The beamed-power concept is one in which the power generation is performed at a fixed location and the energy to drive the spaceship is beamed to the ship's receptor in the form of lasers (optical or x-ray), microwaves, nuclear particles, or material pellets. These systems are usually low-thrust, high I_{sp} designs and operate over the duration of the trip. Consequently, the demands on beam divergence, pointing accuracy, and efficient power reception/conversion are very stringent. Although such systems should be considered, especially for transport of bulk material, greater potential is offered by the second group of engines within the next few decades.

The second group of systems relies on developing a propellant or propellant heating method with a high specific-energy (joule/kg). Consequently, these concepts depend on fission, fusion, or the mass-conversion of antiprotons ($\bar{p}$) as power sources to heat a working fluid. The development of these concepts must inherently deal with radiation of some type and thus must use massive engines. Furthermore, in some cases these engines will require the production of intense magnetic fields and stronger radiation resistant structural materials.

One of the earliest studies of using fission/fusion energy for space propulsion was the ORION concept utilized thermonuclear bombs detonated behind a massive pusher plate which ablated and drove the ship forward. Although the ORION concept used a simple propulsive method, copious amounts of neutrons and fission products were produced which made the concept unattractive.

Since the ORION study, the concept of using small, contained fusion microexplosions was developed. These systems employed an intense magnetic field to channel the charged reaction products and to contain the expanding plasma by flux compression. Usually, these explosions were assumed to be driven by photons, electron beams, or heavy ions. A recent study [1] estimated that the mass of laser driven or heavy ion driven ICF engines would be almost 500 tons.

The concept of using antimatter as a power source for propulsion has existed for decades.[2] Because antimatter annihilation has the highest specific energy of any reaction now known, the potential advantages of an antimatter propulsive system are very great. The obvious problems, however, are whether: (1) sufficient quantities of antimatter can be produced; (2) sufficient quantities can be conveniently stored for long periods; and (3) the products of the annihilation reaction can be converted efficiently into usable thrust.

INTRODUCTION TO ANTIPARTICLES

The concept of antiparticles began with the work of P. A. M. Dirac in the early 1930's on the dynamics of electrons.[3] This work for the first time needed the then-new, quantum mechanics with Einstein's relativistic kinematics. The need for this advance arose from atomic physics where it had recently been estimated that the electrons in an atom are moving in their orbits with velocities near the velocity of light. Dirac's new relativistic theory of electrons was an enormous breakthrough and explained a host of observed phenomena in an elegant and fundamental way. However, the new theory predicted the existence of a new particle in nature that was in every way the mirror image or antiparticle counterpart of the electron. In the mid-1930's, the anti-electron, that is the positron, was discovered.[4]

The tremendous success of the Dirac theory and its experimentally confirmed prediction of the existence of an antiparticle for the electron, touched off widespread speculation that the existence of antiparticles was a fundamental symmetry of nature. All particles have an opposite, an antiparticle. for protons there are antiprotons. For neutrons there are antineutrons, you have all the ingredients needed to make anti-atoms. Thus, it was speculated that there could exist a whole periodic table of anti-elements identical in every way to the familiar elements except that they are constructed of antiparticles. Soon the term antimatter was coined.

Although the existence of the antiproton was predicted in the 1930's, it was not until 1955 that its existence was experimentally observed. Chamberlain and coworkers[5] at the Lawrence Berkeley Laboratory had labored since the late 1940's to build a proton particle accelerator with enough energy to produce antiprotons. They knew exactly what they were after and tailored the accelerator design for the production of antiprotons. Their discovery of this new antiparticle rocked the world of physics and Chamberlain and Segre were awarded the Nobel Prize in physics for this observation. The award cited specifically the experimental confirmation of the particle-antiparticle symmetry in nature. This work opened the door for cosmologists and astronomers to ask in earnest if there were antimatter in our universe and stimulated a host of other investigations.

ANTIPROTON PRODUCTION

Since their discovery, the rate of antiproton production has increased by an order of magnitude every 2.5 years (on the average). This trend line is shown in Figure 1 where the relevant physics and detector technology are indicated as well.[5-22] The slope of this trend line is limited by funding and the available accelerator and magnet technology. The LEAR facility,[15-18] which recently came on-line at CERN, fits clearly on the trajectory, as does a proposed facility at Los Alamos.[20,21] The early part of this trend line was driven by the advent of the zero gradient synchrotron (AGS)[9] at the Brookhaven National Laboratory. In fact, most antiproton production in this era actually exceeds the trend line which is drawn on a conservative trajectory. The present and future production rates will be driven by a new technology, stochastic and electron cooling.[23] The facility at the Fermi National Accelerator Laboratory (FNAL)[19] is already considerably above the trend line. In addition, a practical antiproton factory, using existing magnet and accelerator technology, could be built by the 1990's and would produce 100 to 1000 times more antiprotons than the conservative Los Alamos proposal. This possible factory is still further above the trend line, which shows that the projected limits of the new cooling technology are not properly indicated. Actual limits could be considerably higher. Nevertheless, if the conservative trend line is followed, the annual production of antiprotons could exceed a gram by the year 2010.

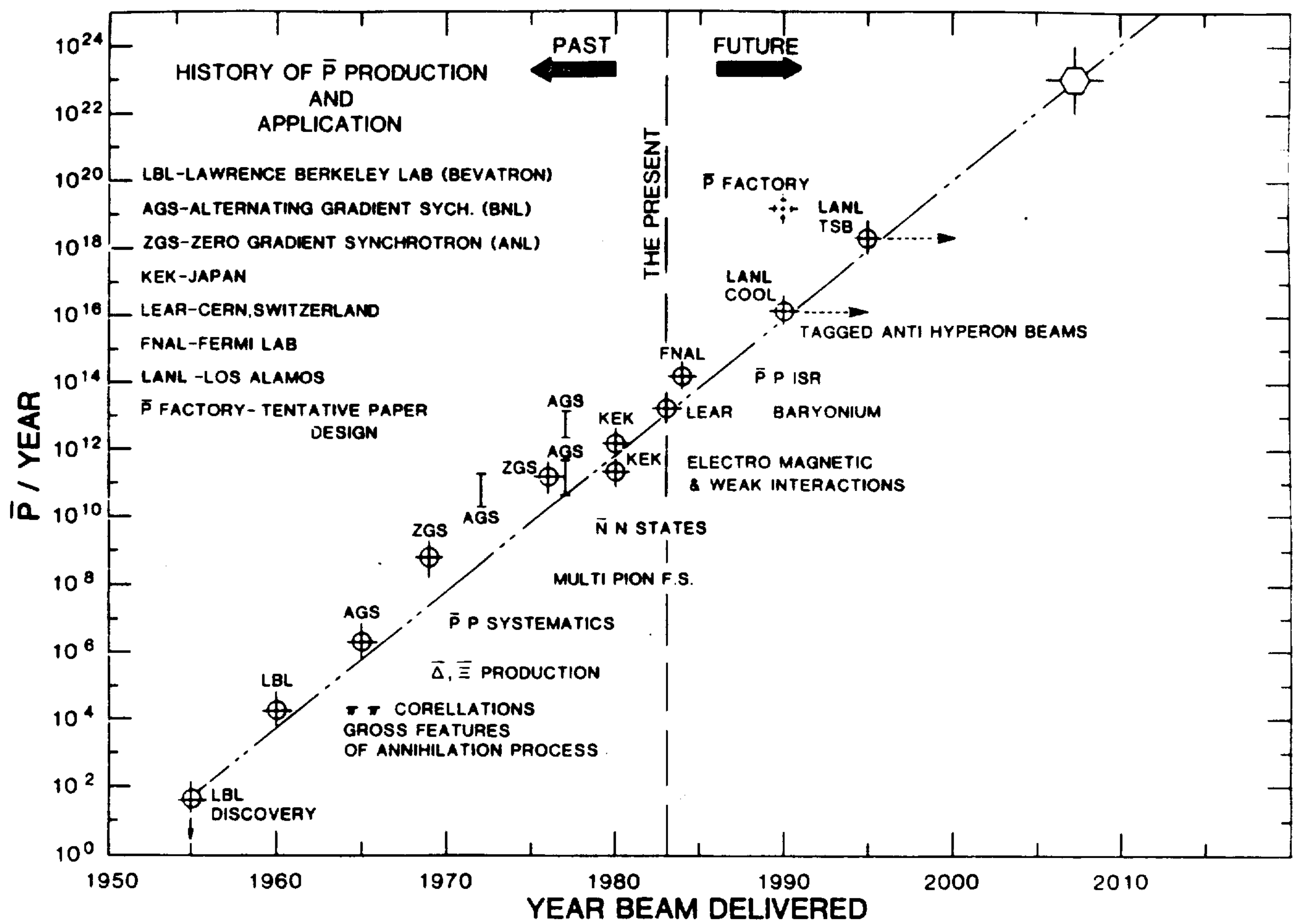

Fig. 1. Annual antiproton production versus year for most high-energy physics facilities around the world. The circled points represent the published flux value; the vertical bar indicates the range of fluxes cited in the literature (Refs. 5-22). The point labeled $\bar{p}$ factory represents a practical design using existing magnet and accelerator technology. The physics of interest for each era is also noted.

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The advent of the new cooling technology has already made possible major advances in high energy physics. These same techniques offer uniquely exciting possibilities for ultralow energy physics as well. Through a combination of deceleration stages, antiprotons produced at several GeV (where the production is at a maximum) can be made available for experiments at thermal velocities. This availability opens many new avenues of basic and applied research in atomic, condensed matter and nuclear physics.

Aside from the success of the new cooling technology in antiproton production, there is little understanding of the fundamental production mechanism. A simple view of the production of antiprotons has a high energy proton incident on a nucleon at rest in, for instance, a liquid hydrogen target. Such an initial state can reach a multitude of possible final states ranging from simple elastic scattering to multiple pion and kaon production, depending upon the incident beam monentum. However, let us consider only those final states which produce antiprotons. To conserve baryon number and charge, antiprotons are produced as part of a proton-antiproton pair. The minimum beam momentum required for this reaction is 6.5-GeV/c, whereas the likelihood for production increases rapidly with increasing momentum. Typical antiproton production facilities for basic research use incident beam energies in excess of 20 GeV. Usually, these facilities use targets of beryllium, carbon or tungsten instead of liquid hydrogen. This simplifies the production system structure and leads to slightly different kinematic properties of the distribution of antiprotons emerging from the target. There have been a great number of measurements of antiproton production from nuclear targets, although only over a limited range of antiproton momentum and production angle.

Despite the lack of fundamental understanding of the production process, several empirically derived production cross section formulations describe the limited data available. These empirical formulations have been used to design the collection facilities at CERN and FNAL. Neither of these facilities were designed originally with antiproton production or collection in mind. Their collection facilities were added onto the existing accelerator systems. Nevertheless, the antiproton production capabilities of these facilities is impressive. At

CERN or FNAL, 10^{13} -10^{14} antiprotons can be produced, with 10^{15} per year in the near future.

Two other facilities are currently being planned in the free world for producing, among other particles, antiprotons: TRIUMF in Canada and a facility at the Los Alamos National Laboratory. The antiproton production rates at these facilities could far exceed those currently available at CERN and FNAL. However, even these facilities are not optimized solely for antiproton production and do not exploit fully the available magnet and accelerator technology. These and all previous antiproton facilities represent the very best that could be done with a fiscally constrained basic research budget. The current Los Alamos plan, for example, is a $300M project, not including an antiproton collector and cooler. If the fiscal constraint were lifted for the design of an antiproton factory, several orders of magnitude more antiprotons per year could be produced using existing technology. However, before this increase in production can be cooled and accumulated, very significant progress needs to be made in accumulator/cooling technology. In addition, before the milligram-to-gram size quantities, projected for the next decade and beyond, can be produced, very significant progress in accelerator technology needs to be made as well. Increasing the production/cooling rates is a high priority task at antiproton facilities around the world. Rapid progress in these areas can be expected in the short term. Thus, technological research and development here in the US should proceed on the assumption that such quantities of antiprotons will be available in the coming decades.

The only facility in the world today that is capable of producing low energy antiprotons is at CERN. The facility at FNAL accumulates antiprotons at high energy, and at present has no low energy capability. The possibility of developing a low energy capability at FNAL is probably the best option for a low energy antiproton facility in the United States before 1990. After 1990, a true antiproton factory is needed. Without such a facility, by the next decade, the United States will be a third world country in antiproton technology, behind the Soviet Union, Switzerland, and Canada.

STORAGE OF ANTIPROTONS

At the present time the particle physics community stores a significant number of antiprotons for several tens of hours for basic research on particle dynamics at very high energies. The storage technique used is electromagnetic confinement in very large rings inside which the antiprotons are circulated or accelerated to the desired energy. Although well-suited to the requirements of many applications in basic research, this type of storage is not readily adapted to the applications we envision. We have considered two general types of storage: Bulk storage, in which antimatter at low temperature is stored in a high vacuum, and dispersed storage, in which the antimatter is stored in a uniform mix with normal matter. Whether in bulk or dispersed storage, the antimatter can be charged, as in the case of antiprotons or it can be neutral, as in the case of antihydrogen atoms.

The discovery of the positron in 1932[4] started the theoretical and experimental work on the fundamental interaction between matter and antimatter. The discovery of the antiproton in 1955[5] triggered a series of cosmological studies investigating the signatures and consequences of antimatter in our universe.[24-27] These studies addressed the basic symmetry between the existence of both matter and antimatter on a cosmological scale. A model for the separation of matter and antimatter was presented to explain the apparent absence of antimatter in our local space.[24] This early work marked the beginning of the quantification of the matter-antimatter interaction problem. Later work by Morgan and Hughes[28] pointed out, for the first time, the importance of atomic scale processes in antihydrogen-hydrogen collisions. Morgan and Hughes calculated the cross section for annihilation as a function of temperature. This cross section together with the number density of particles, determines the average lifetime of the plasma. For very long lifetimes, very low densities must be used (10^{-4} to 10^{-10} per cm^3).

The principal operating feature in these calculations is the long-range van der Waals force, which is attractive for normal matter-matter mixtures and is still attractive for matter-antimatter mixtures. As the matter-antimatter atoms or molecules draw more closely together, the interaction potential grows increasingly more attractive, until finally the protons and antiprotons annihilate along with the electrons and positrons. With normal matter-matter interactions, as the two atoms or molecules draw more closely together, the potential also becomes more strongly attractive until the two objects are close enough to start exchanging electrons. At this point, a repulsive exchange force overwhelms the attractive force and the two objects can get no closer.

Let us consider what is required to store antimatter. Stated simply, the antiprotons (and any positrons) must be kept away from their normal matter counterparts to prevent annihilation for timescales of a year or longer. For the bulk storage of antimatter, contact with the confining walls must be eliminated, whereas for dispersed storage, a metastable state for the antimatter within the normal matter matrix must be found. Consider the assumptions that led to the result that the van der Waals force is attractive. Firstly, it is assumed that the anti-atom and the atom are interacting as free particles, as in a dilute gas, uninfluenced by nearby neighbors. Also, it is assumed that the atoms are in a ground state which is assumed spherically symmetric, without any electromagnetic moments higher than the monopole charge. Finally, it is assumed that there are not external electric or magnetic fields.[29] Changing any of these basic assumptions can lead, in principle, to a repulsive barrier.

The scale of the barrier needed to confine the antiprotons can be estimated by treating the confinement as a one-dimensional barrier penetration problem.[30] The transmission coefficient for such a barrier should be in the range 10^{-30} - 10^{-35} in order to realize long-term storage of gram-like quantities. The calculation reveals that transmission coefficients in this range can be obtained with barrier heights of about 0.5 eV and widths of 2 to 4 angstroms for thermal antiprotons (10 - 100K). The scale set by these results are atomic in size. Thus, much of our effort in searching for a storage medium for antimatter will necessarily be concentrated in atomic and condensed matter systems.

A simple and obvious way to prevent antiprotons from impinging upon the walls of a storage vessel is to electrically charge the walls so as to repel them. Storage devices of exactly this sort have been intensively studied both theoretically and experimentally for the confinement of normal matter ions.[31] All of this "ion trap" work is directly applicable to the storage of antiprotons. Briefly, the charged particles are stored in a volume defined by a combination of electric and magnetic fields or in an inhomogeneous RF field. In addition, techniques for cooling the confined ions to very low temperatures have been developed.[32]

To explore any of the atomic or condensed matter storage approaches, a thermal source of antiprotons is required. Because of the cooling capability of ion traps, these devices can serve as an intermediate technology allowing for the study of more advanced concepts. More importantly, however, ion traps could allow for the storage of significant quantities of antiprotons today. The practical limit on storage of this type in sensibly dimensioned equipment is of order 10^{15} - 10^{17} antiprotons. This not only represents more antiprotons than is currently being produced yearly at existing facilities, but it also represents an engineeringly significant amount of energy (0.3 - 30.0 megajoules).

APPLICATIONS

The capability to store large numbers of antiprotons at thermal velocities will open many avenues of basic and applied research. The potential applications that we envision utilize the very high specific energy characteristic of antimatter annihilation. The specific energy in joules per kilogram for a variety of exoergic reactions is shown in Table 2. The fact that antiproton annihilation has specific energy 10^8 times chemical values and about 10^3 times fission/fusion reactions, indicates the enormous potential of antiprotons as an energy source for space based prime power and propulsion applications where mass is a principle consideration.

Because the energy release modes of antiproton annihilation are vastly different than any other energy source, the questions confronting designers of antiproton propulsion or power sources must be approached from fundamental viewpoint. Although in their infancy, several propulsion system concepts have been discussed.[33-37]

One concept which has not been discussed but which may offer a near term potential is the Solid Core Thermal Rocket (SCTR). The SCTR would utilize the antiprotons by stopping all of the annihilation products in a solid core of high-melting-temperature material such as tungsten. The core is honeycombed to allow the heat transfer to the propellant. Such a

TABLE 2

SPECIFIC ENERGY COMPARISON

Source	Specific Energy (joule/Kg)
Chemical	
gasoline + air	9.1 e06
hydrogen + flourine	1.3 e07
hydrogen recombination	2.2 e08
metastable helium	4.8 e08
Fission	
U-235	8.2 e13
Fusion	
D(t,n)4He	3.4 e14
D(d,n)3He	7.9 e13
D(3He,p)4He	3.5 e14
Antiproton Annihilation	
$\bar{p} + p$	9.0 e06

concept is similar to the nuclear rockets developed during the NERVA program and could possibly utilize many of the non-nuclear components, such as liquid hydrogen (LH_2) turbo pumps, already tested. A schematic diagram of the small nuclear rocket engine (SNRE) designed in 1971 is shown in Figure 2. This engine would have produced about 16000 lb of thrust and would have weighed about 5887 lb. The figure shows the layout of the liquid hydrogen transport lines, valves, and pumps which were tested in the NERVA program. Preliminary calculations indicate that a tungsten cylinder which has been sized to stop most of the $\bar{p}$ annihilation products would be slightly smaller than the nuclear reactor core designed for the SNRE. These calculations included the 36% void fraction for the hydrogen flow channels used in the SNRE. A $\bar{p}$-NERVA engine based on the most thoroughly tested nuclear rocket, designated NRX, would have a thrust of 4.4×10^5 N (100,000 lb), a power level of around 2700 MW, a mass of near 11000 kg, an I_{sp} of 1100 s, and a mass flow of antiprotons of around 13 μg/s. Such an engine would require about 400 metric tons of material in LEO to accomplish the baseline manned Mars mission--a factor of 4.5 times less than a chemically propelled system.

Another engine concept utilizes a reaction chamber filled with high pressure gas into which the antiprotons are deposited. The charged annihilation products are trapped by an intense magnetic field, slow down, and heat the gas for expulsion. This engine concept has the advantage of adjusting the ratio of antimatter to produce a wide range of I_{sp} depending upon the mission. The possible effects of pion and muon thermalization times, wall losses, reaction chamber structural requirements, and losses of pions or muons due to nuclear reactions or decay, need to be evaluated after more fundamental data have been collected, and will require complex computational studies.

The amount of antimatter required by either concept will depend upon the mission delta V requirements. Typical missions such as launch from Earth's surface, orbital transfer to GEO, or a mission to Mars will probably require between tens to hundreds of milligrams. The ship's mass ratios for these missions would be about 3 to 10.

Other results presented during this workshop indicate that (1) artificial gravity may be required on the Mars-mission ship to alleviate bone and muscle mass loss, (2) radiation dose rates of about 50 rem/yr

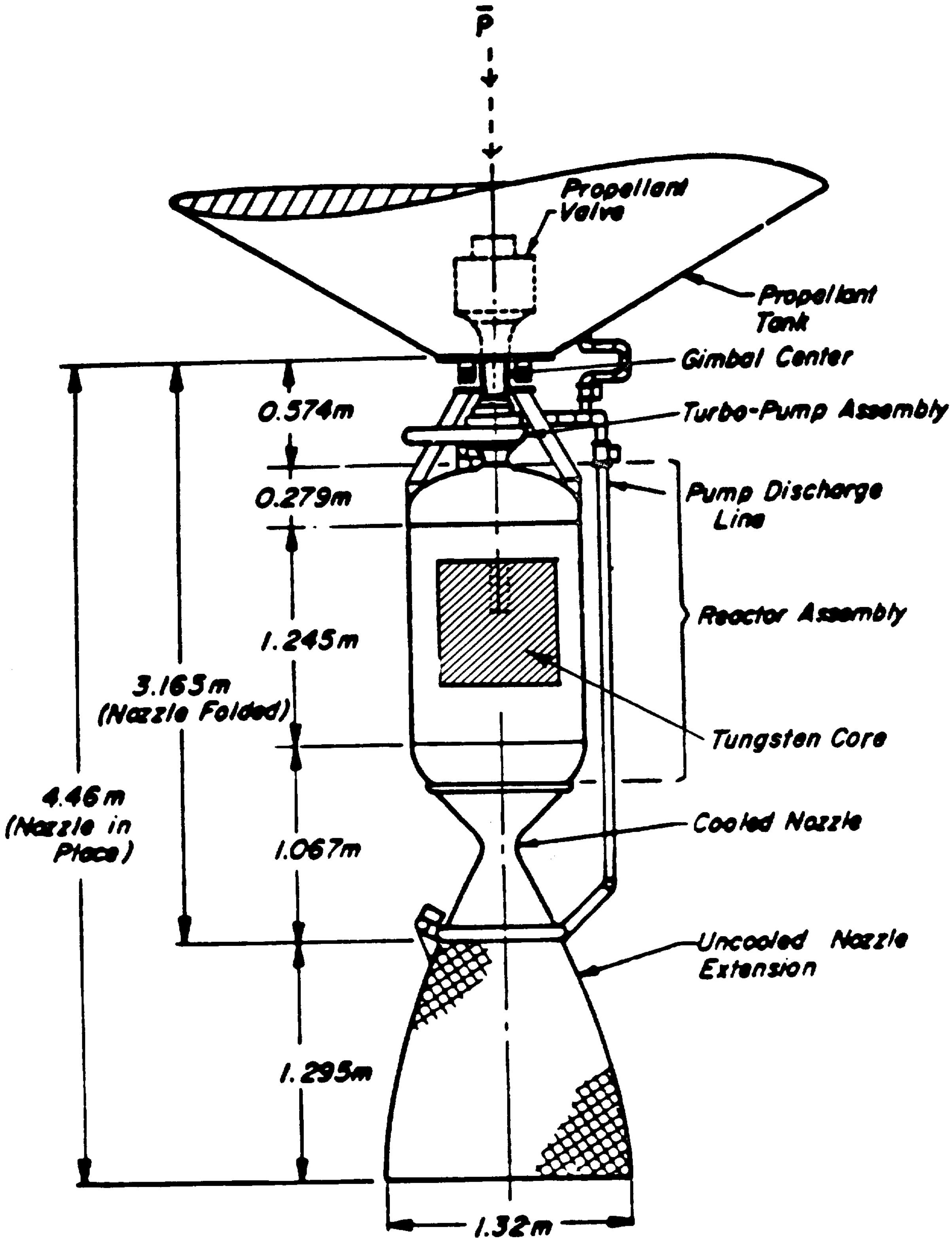

Fig. 2. Schematic diagram of the Small Nuclear Rocket Engine designed during the NERVA program. The nuclear reactor core has been replaced with a possible configuration of the metal-honeycomb used to convert the antimatter annihilation energy into heat.

in interplanetary space may limit missions to 3 yrs or less, i.e. Mars and Venus only for chemically propelled manned systems, and (3) confinement times of over a year in a spacecraft may induce psychological difficulties. Although these problems may be tenable by more complicated and massive ship design, the use of an antimatter engine which could reduce trip times to under a year could also alleviate most of the problems.

In general, the antiproton powered engine may allow low mass-ratio ships and/or fast transit-time missions to become possible. These two characteristics may not be simply enhancing but actually enabling to certain space missions such as planetary exploration.

ENHANCEMENT

The specific cost of production of antimatter (dollars per unit mass) is a convenient but misleading quantity. A more significant quantity is the dollars per unit energy. Reduction of these ratios has always been assumed to depend on improving the production and collection efficiency of the antimatter factory accelerator. Use of the latter ratio, however, shows that improvements can be made if the energy output for each incident antiparticle is increased or amplified.

One possibility is to consider the antiproton as a stable repository of negative muons. An average $p\bar{p}$ annihilation will produce about 1.45 negative pions with an average energy of 250 MeV. If the pions can be either trapped in magnetic field or quickly thermalized by collisional losses, then the negative muons (μ^-) resulting from the pion decay may be generated in a small volume. By thermalizing these muons in a volume containing a mixture of gaseous deuterium and tritium, fusion of the DT atoms can be catalyzed.[38-42] Recent measurements of $D\mu T$ molecular formation rates[43] and of other factors inherent in $\bar{\mu}$ catalyzed DT fusion have observed up to 180 fusions per muon. The resonant molecular-formation theory which accounts for the observations predicts that up to 300 fusions per muon could be induced in DT mixtures at appropriate density and temperature. Thus, an upper limit of about 7.8 GeV in fusion energy could be released per antiproton in addition to the 1.8 GeV of annihilation energy--more than a factor of 5 enhancement. Clearly, losses due to pion capture and inter-actions, muon decay during thermalization, and muon-wall interactions, as examples, will reduce this upper limit in an operating system. Efforts to estimate the magnitude of different loss factors and of a possible reactor geometry are currently underway.

Another method of producing fusion energy using antiprotons is inertial confinement fusion (ICF). This technique relies on stopping the antiprotons in a thin, uranium shelled capsule containing DT gas. The stopped antiprotons annihilate on the uranium nuclei and induce fission. The localized deposition of the fission energy ablates part of the shell and implodes the capsule. Early calculations show that more than 10 GeV[9] could be released, with much higher gains possible. Experiments characterizing the $U(\bar{p},f)$ reactions are underway at CERN with the ultimate goal of investigating antiproton- produced implosions.[44] The major attraction of the ICF technique is that the incident antiproton energies could be a few keV or less so that the required accelerators would be small. Thus, depending upon the mass of the antiproton storage device, low mass ICF reactors might be possible. Evaluations of pulse structure, implosion symmetry, and optimum capsule design are required, and significant work in those areas can be performed with currently existing codes.

SUMMARY AND CONCLUSIONS

Since their discovery in 1955, antiproton production rates have increased by an order of magnitude every 2.5 years. The advent of the new cooling technology could make the production rates rise even faster. Nevertheless, if the conservative trend is followed, a gram of antiprotons could be produced yearly by the year 2010. Many of the applications we envision for antiprotons require only milligram-size quantities. These applications are in the area of energy sources for prime power and propulsion for space-based systems where high-energy density is of principal importance. Storage of antiprotons can be accomplished in sensibly dimensioned equipment using ion traps for quantities up to 0.1 micrograms. Higher density storage techniques have been investigated theoretically and require experimental work to make progress. For this work, the ion trap storage device will serve as an intermediate technology, supplying a thermal source of antiprotons. Antiproton technology will be upon us in the coming decades. Now is the time to consider what technical steps are required to enable the concept of antiproton power sources to be put on a more scientific basis.

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N87-17794

ASSESSMENT OF THE ADVANTAGES AND FEASIBILITY OF A NUCLEAR ROCKET FOR A MANNED MARS MISSION

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ABSTRACT

The feasibility of rebuilding and testing a Nuclear Thermal Rocket (NTR) for the Mars mission has been investigated. Calculations indicate that an NTR would substantially reduce the Earth-orbit assembled mass compared to LO_2/LH_2 systems. The mass savings were 36% and 65% for the cases of total aerobraking and of total propulsive braking, respectively. Consequently, the cost savings for a single mission using an NTR, if aerobraking is feasible, are probably insufficient to warrant the NTR development. If multiple missions are planned or if propulsive braking is desired at Mars and/or at Earth, then the savings of up to $7 billion will easily pay for the NTR development.

Estimates of the cost of rebuilding a NTR were based on the previous NERVA program's budget plus additional costs to develop a flight ready engine. The total cost to build the engine would be between $4-5 billion. The concept of developing a full-power test stand at Johnston Atoll in the Pacific appears very feasible. The added expense of building facilities on the island should be less than $1.4 billion.

INTRODUCTION

The concept of using a Nuclear-Thermal Rocket (NTR) for a manned mission to Mars has been considered for over 30 years.[1,2] The obvious advantage of producing about 2 times the Isp of chemical rockets allows (1) a lower total mass to be assembled for a given payload mass; (2) the possibility of much faster, high-energy orbit to be used; or (3) more relaxed launch windows to be used. One other distinct advantage of the NTR is that the development and use of NTR engines will bring the possibility of future missions to more distant planets into the realm of possibility.

The major tasks of this study are to: (a) compare the use of a NTR system to a chemical (LO_2/LH_2) system for the proposed 1999 launch scenario; (b) assess the economic feasibility of redeveloping the NERVA class NTR; (c) determine the possibilities of testing the NTR; and (d) assess the concept of using the NTR as an electrical power source during the mission.

NUCLEAR ROCKET PRINCIPLE

The fundamental principle of a NTR is that a nuclear reactor operating at high power levels can heat and expel injected coolant at very high temperatures.[3] Thus, the reactor simply is an energy source which replaces the chemical energy released in LO_2/LH_2 reaction engines, for example.

A schematic diagram[4] of a "standard" NTR is shown in Figure 1. The reactor core is composed of highly enriched uranium-carbide fuel in a graphite matrix. Control drums composed of borated cages around beryllium cylinders, to either absorb or reflect neutrons, surround the cylindrical core. Liquid hydrogen is injected into the core, heated to temperatures as high as $4500^{o}R$, and ejected through the nozzle. A small amount of liquid hydrogen is also heated and diverted to run the LH_2 turbopumps. The pumps are located at the top of the engine and are protected from the intense radiation fields of the reactor by a ZrH shield.

The intense neutron and gamma-ray radiation fields produced by the operating reactor are clearly the main difficulty in using a NTR on a manned mission. The ejected propellant poses a relatively minor radiation hazard since the LH_2 does not become radioactive and the fuel element particulate which abrades into the LH_2 from the core will rapidly disperse into the interplanetary environment.

Shielding the crew from the reactor during the "propulsive burn" can be accomplished by the combination of a tungsten and LiH shield. Further, reduction in the neutron dose to the crew can be accomplished by incorporating a few meters of LH_2 in a tank between the crew and engine. This tank, for example, might contain the 15% contingency LH_2 and would be the last tank to be used.

After the full power burn of the engine, the radiation from the reactor will be only gamma-rays and within a few days the intensity will have dropped by over three orders of magnitude. The thickness of tungsten required to shield the reactor in transit will be substantailly less than for propulsive maneuvers. Thus, the tungsten shield may be designed to "unfold" around the reactor for post burn shielding which

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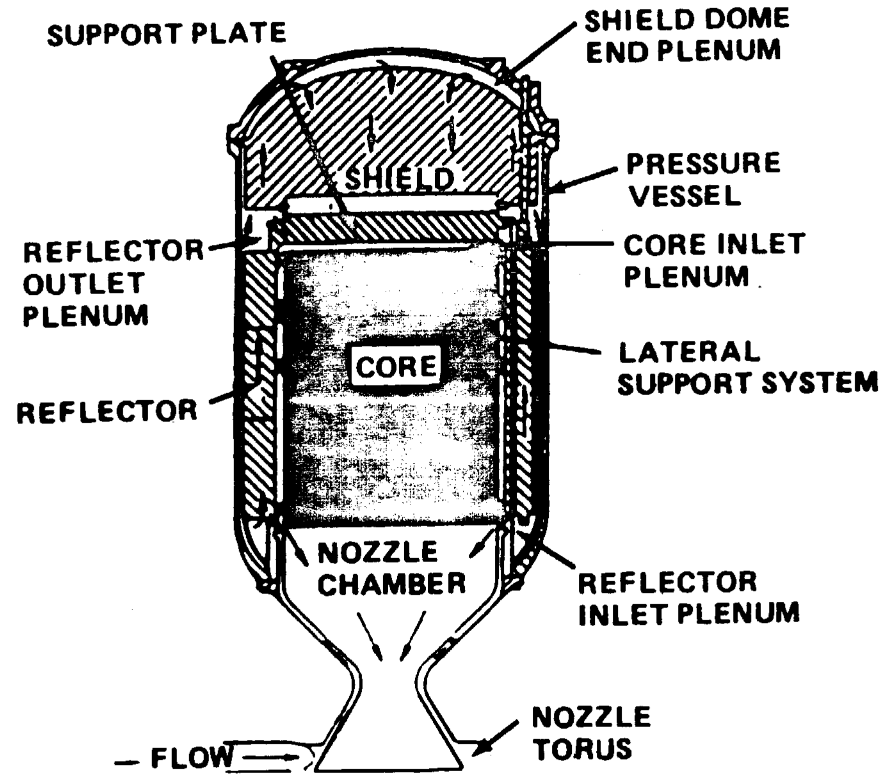


FIGURE 1. SOLID CORE NUCLEAR ROCKET DIAGRAM

will provide a 2 π or greater shield around the reactor and allow docking or EVA activity. Another possibility is to use mercury as the gamma-ray shield. Change of configuration is then accomplished by pumping the Hg into preformed reservoirs as shown in Figure 2.

After the full power "burn" of the engine, delayed neutrons in the sub-critical reactor core continue to produce fission heating. This delayed heat output causes a penalty in propellant mass, since LH_2 must be fed to the reactor for a few days at a reduced flow rate. If the ejected hydrogen is not used to provide thrust, an extra mass of propellant must be carried. The amount of extra LH_2 which must be carried along to cool the core is around 24% of the mass of LH_2 used during the burn of the engine[5].

Another approach is to utilize the delayed heat to produce low Isp thrust. Since the Isp of the NTR scales as the square root of the propellant temperature, the cooldown flow can be used to provide thrust with an Isp of around 400 s. This application reduces the average Isp of the engine by between 6-10%, and will necessitate carrying extra fuel. For most missions with delta V requirements of a few km/s, the extra LH_2 required will be less than the 24% penalty previously described.

NTR VS. CHEMICAL

A comparison between NTR and chemical propulsion systems is shown in Table 1. In the comparison, an Isp of 450 s and 825 s was used for the chemical and NTR respectively. The NTR value was chosen as a reasonable compromise between cooldown losses which would lower the effective Isp and studies in the NERVA program which concluded that a flight ready version of the NRX reactor, which would include a topping or bleed cycle to power the turbines, would have an Isp of about 900 s.

The tankage mass for the NTR was determined as being 0.15 of the propellant mass. This factor derived from the LH_2 tankage used in the chemical system study.

The dry-weight masses of the ship were also taken from the chemical study and totaled to 128,208 lb for the MEM, 112,690 lb for the mission modules, and 24,480 lb of probes.

The flight scenario assumes that the entire ship is launched from Earth orbit, the probes are jettisoned before Mars Orbital Insertion (MOI), the MEM is detached and remains in Mars orbit, and the remaining ship

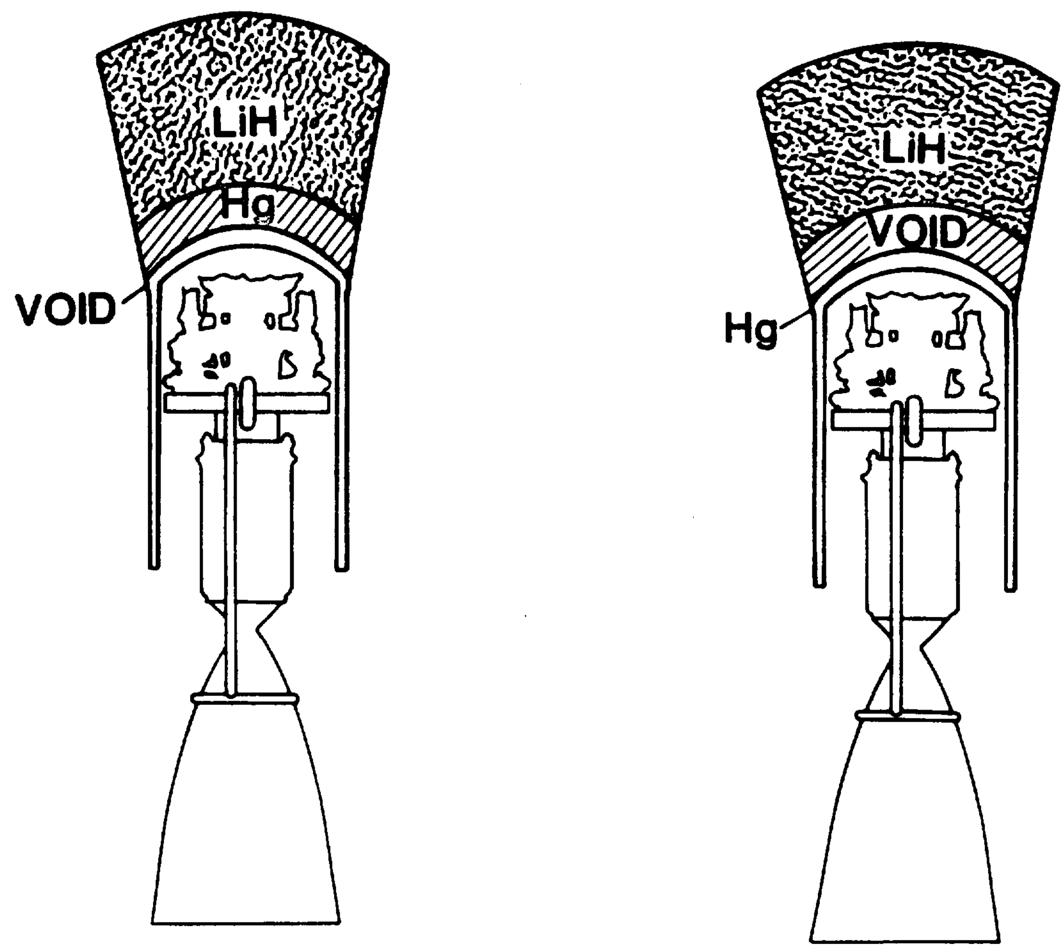


Fig. 2. Possible configuration of a moveable shield using mercury. The figure is to scale except for the plenum around the engine which was expanded for clarity.

TABLE 1

NTR TO CHEMICAL PROPULSION

	Aerobraking (Mars and Earth)		Propulsive Braking		Hybrid (Earth Aerobrake)
	Chem	NTR	Chem	NTR	NTR
EOI					
Payload (K lbs)	112.69	112.69	112.69	112.69	112.69
Engine/Aeroshell	19.83	19.83	.78	26.00	19.83
Tankage			16.88	13.34	
Propellant			198.92	102.31	
	132.52	132.52	329.26	254.34	132.52
TEI					
Engine	1.96	26.00	1.96		26.00
Tankage	20.37	6.29	26.50	10.09	6.29
Propellant	77.18	41.95	183.13	67.30	41.95
	232.03	206.76	540.85	331.73	206.76
MOI					
MEM	128.20	128.20	128.20	128.20	128.20
Tankage/Aeroshell	63.40	58.95	26.50	34.33	25.00
Propellant	423.63		694.71	228.87	166.68
	423.63	393.92	1390.26	723.13	526.64
TMI					
Probes	24.48	24.48	24.48	24.48	24.48
Engine	16.02	26.00	16.02	52.00	26.00
Tankage	75.02	62.79	130.66	112.97	81.54
Propellant	1035.33	418.58	3105.70	753.16	543.59
	1574.	926.	4667.	1666.	1202.

including all waste products are returned to Earth. Both an aerobraking maneuver (ABM) at Earth and Mars and a propulsive braking maneuver (PBM) are considered. The mass of the aeroshell was assumed to be 0.176 of the mass required to brake. The delta V's of the flight plan were 4.4289 (TMI), 2.7569 (MOI), 1.6238 (TEI), and 3.7246 (EOI) km/s. All propellant masses include an extra 15% for contingency and boiloff following the example of a previous Mars study.[6]. No mass penalties were made for post burn cooldown of the NTR since the average Isp which was used included the penalty.

The propulsive NTR scenario assumes that 3 engines of 75,000 lb thrust are used in Earth-orbit departure. After MOI, 2 engines are detached and are left in Mars orbit and a single engine is used on the return trip. The aerobraking-NTR scenario is similar except that only 2 engines are used for Earth departure. The number of engines was chosen to produce thrust-to-weight ratios of near 0.20 in Earth Orbit. This value was chosen following the results of a study[7] which optimized thrust to weight ratios for maximum payload fraction for orbital launch. The mass of engines includes an 11,000 lb shield for each engine which will allow approximately a 10 Rem dose to the crew from the engine burns.

The final calculation shown in the last column of Table 1 is for a combination of propulsive braking at Mars, where the ship is bulky and difficult to cover in an aeroshell, and of aerobraking at Earth where the mission modules should be easy to cover in a shell. Before EOI, the NTR is assumed to detach and boost itself into an appropriate helio-centric orbit, possibly the stable Lagrange point, L_2, lying between the Earth and the Sun.

The masses in Table 1 show that the NTR has significant advantage over chemical propulsion. The ratios of NTR mass to chemical for the entire ship in Earth orbit are 0.64 and 0.35 for the aerobraking and propulsive braking scenarios respectively. The hybrid scenario mass is 26% of the mass for the chemical PBM scenario. The number of shuttle launches to put the difference of the respective masses into orbit for assembly are 8 to 46 for the ABM and PBM respectively, assuming 65000 lb per launch. At $0.15 billion per launch this equates to $1.2 billion and $6.8 billion in savings due just to launch the mass in orbit. Further savings will be incurred from reduced handling of the mass both on Earth and in orbit.

NERVA PROGRAM

In 1960, the Space Nuclear Propulsion Office (SNPO) was established by joint AEC/NASA agreement. The Nuclear Engine for Rocket Vehicle Application or NERVA program began in 1961 with selection of an industrial-contractor (I-C) team of Aerojet General Corporation and the Astronuclear Laboratory of the Westinghouse Electric Corporation (see Figure 3). The I-C team was to "pursue the development of nuclear-rocket engine technology with reactor designs based on the KIWI concepts"[8] of the Los Alamos National Laboratory. The KIWI reactor was the product of project ROVER which began at Los Alamos National Laboratory in 1955.

The NERVA program existed for 11 years and succeeded in developing and testing the NRX reactor series and the Phoebus reactor series as shown in Figure 4. The NRX reactor operated between 1100 to 1500 MW and produced 75000 lb of thrust while the Phoebus reactor operated at 4500 MW and developed 250,000 lb of thrust. Characteristics of both engines are shown in Table 2. Both engines were tested at the Nuclear Rocket Development Station (NRDS) in Nevada with the NRX series being much more thoroughly developed. The NRX-EST and NRX-XE tests actually incorporated the non-nuclear system components such as LH_2 turbo pumps, valves, and regenerative LH_2 cooled nozzles in the tests at NRDS. By the end of the program in 1971, a fully integrated engine had been tested under simulated altitude, and efforts were shifting to define and develop a flight ready engine.

The budget for the NERVA program and the total costs of the entire rocket development effort are shown in Figure 5. Clearly, a major portion of the effort was the development of the reactor/engine with the next largest category being material and non-nuclear component development. In the event that this nation would decide to build a nuclear engine for a Mars mission, much of the costs of the NERVA program would not be duplicated by the new effort. The KIWI, much of the technology, and some of the NERVA categories would be removed. The magnitude of effort to build the engine will depend on whether only a redesign of the NRX or Phoebus engines is requested from existing data bases or if a redevelopment and improvement is desired. Reestablishing

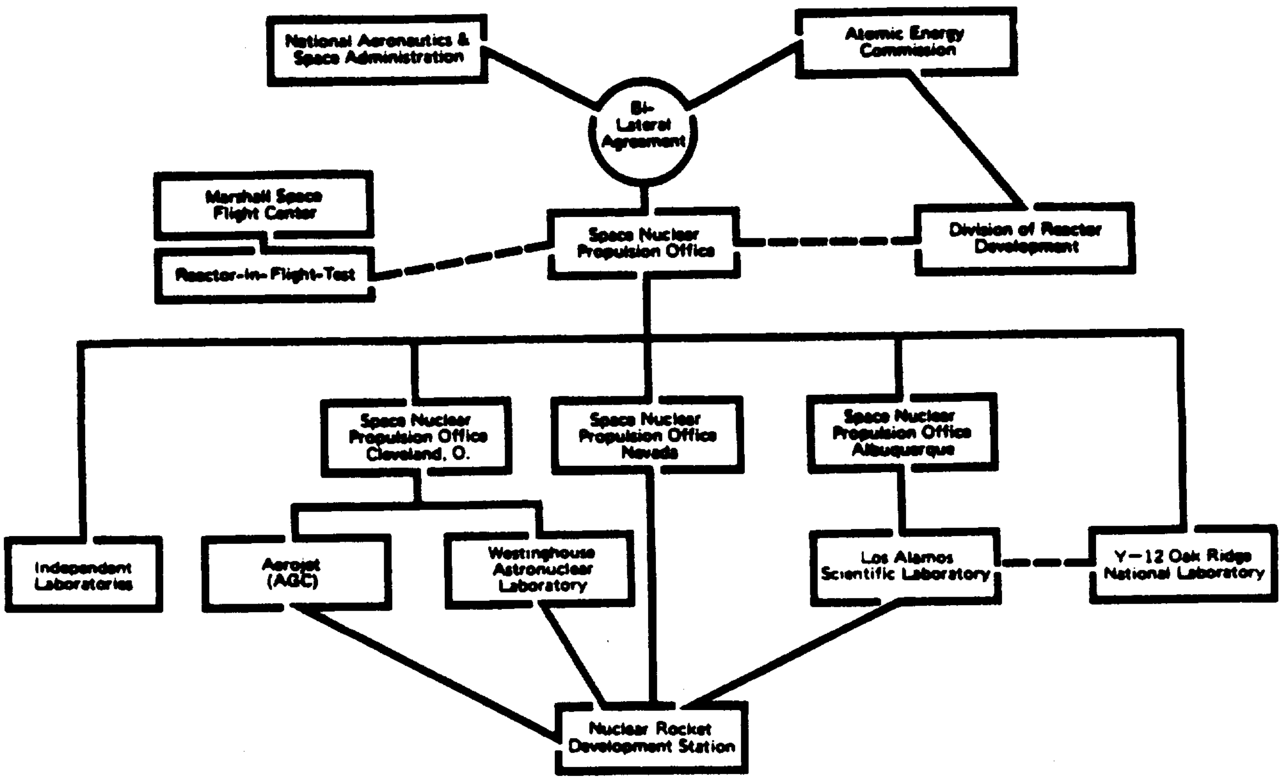


Fig. 3. Organization chart for the NERVA program.

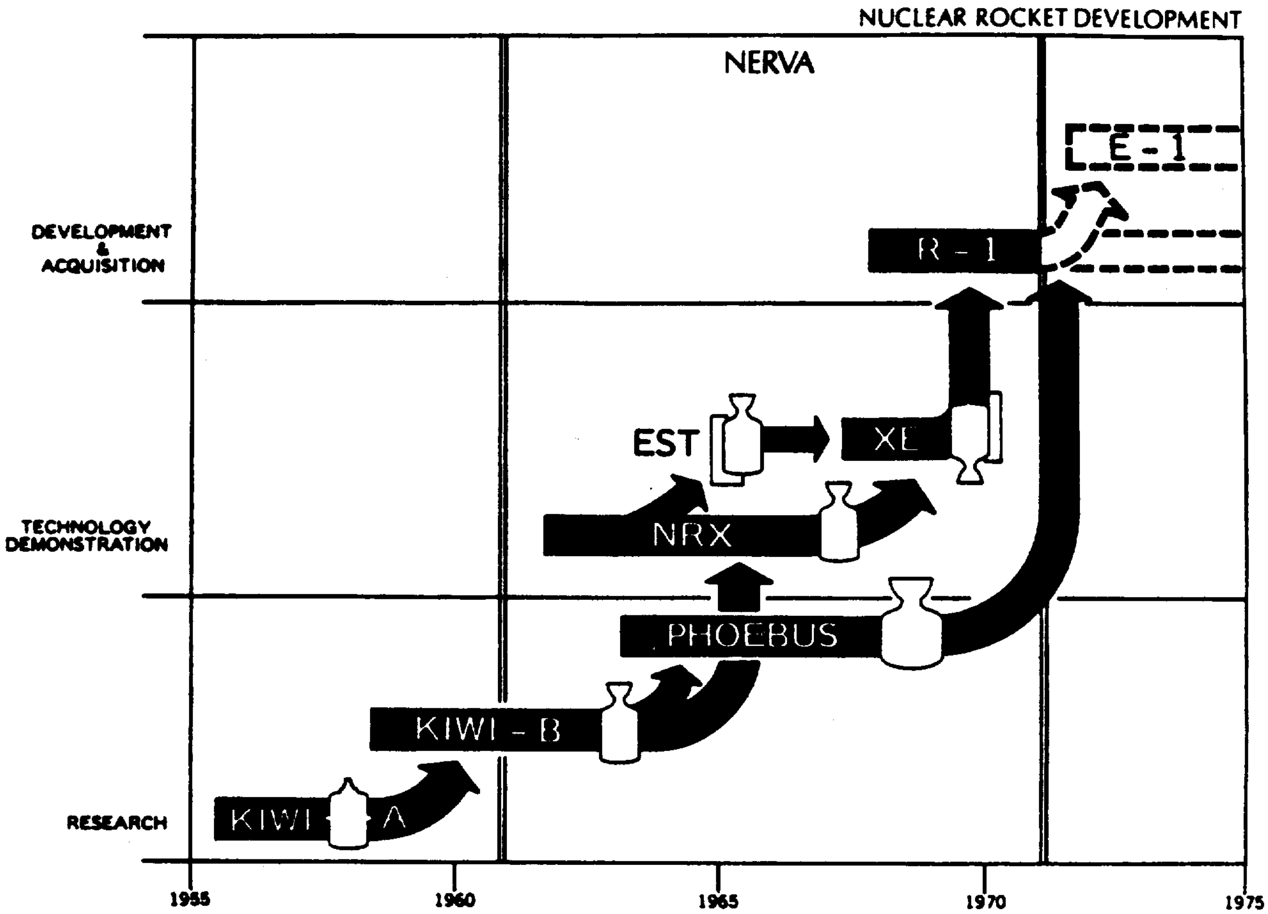


Fig. 4. Nuclear rocket development time line.

NERVA BUDGET (M$)

1962	63	64	65	66	67	68	69	70	71	72
20	58	84	80	70	72	65	53	53	50	25

TOTAL = 662

PROGRAM TOTALS (M$)

KIWI	177	Los Alamos
NERVA	662	Westing House (342/Aerojet)
Techology	328	Materials Development
NRDS	90	Operating
	153	Capital/Test Facilities
TOTAL	1410	AEC (866)/NASA (566)
Proposed NRDS Ugrades	112	1972 Estimate
	1522	
	(3501 in 1985 $)	

Fig. 5. NERVA program budget.

TABLE 2

ENGINE CHARACTERISTICS

	NRX	Phoebus
Power (MW)	1,500	4,500
$\dot{m}$ (lb/s)	90	285
Thrust (lb)	75,000	250,000
Tested Isp (s)	825	820
Mass (lb)	15,000	40,000

TABLE 3

ESTIMATED COSTS TO REBUILD A NERVA ENGINE

Engine Design and Construction		1218	M$	(80% of NERVA)
Technology		377	"	(50% of Previous)
NRDS:	Capital	460	"	
	Operating	210	"	
		2,265	"	

the capabilities existent at the termination of the NERVA Program can probably be accomplished for under $2.5 billion (1985 $) as shown in Table 3.

The capital investment of NRDS is estimated by subtracting the value of the major facilities currently at NRDS which could be refitted and adding the cost of improvements estimated in a 1972 Los Alamos Study.

In addition to the costs of rebuilding the engine, significant costs will be incurred to make the engine flight ready. Determining this expense is more difficult, since only estimated requirements exist from the previous program. Furthermore, many of the costs previously estimated will already be incorporated in the new rebuilding effort. A reasonable estimate, according to researchers who were involved in the previous program, is $2-3 billion. Thus, a reasonable estimate for the cost of rebuilding a flight-ready, nuclear-thermal rocket is between $4-5 billion dollars.

TESTING FEASIBILITY

The estimates of the costs of a new NERVA type program are somewhat dependent on the ability to test the new engines when built. The NRDS at Nevada still retains some major facilities such as the EMAD building (for post-test reactor analysis), the tank farm for pressurized gases, and several large (up to 500,000 gal of LH_2) dewars.[9] The possibility of refurbishing some of these facilities, the accessibility of the Nevada Test Site (NTS), and the existence of experienced operations and security personnel currently at the site make the testing of the engines at NTS appear quite feasible.

The major obstacle to testing at NTS will be the reduced levels of radioactive debris which are allowed to transport into the public domain. The levels are more stringent than those present during the NERVA program. The current exposure limits of 150 m rem to civilian personnel may restrict the tests of the NTR to low power levels and mass flows in the reactor.

While low power tests may be sufficient for early tests and rebuilding, eventually a full power test will be necessary for flight readiness. A simple solution to this problem may be to utilize one of the Pacific Ocean islands owned by the United States--namely Johnston Island (JI). Johnston Island is part of a large atoll lying about 700 miles southwest of Hawaii at 15°N latitude. The island currently supports an active military base, an airstrip, and an active shipping port as seen in Figure 6. The advantages of using JI for NTR testing are: (1) that several hundred acres of slightly submerged coral atoll can be dredged to make test stands or can be used to anchor test platforms; (2) constant easterly trade winds 10 months out of the year; (3) ecological desert of ocean surround the area due to the stagnation of the return of the Japanese current; (4) exposure limits for badged, base personnel are 500 m rem; (5) several hundred miles to the nearest human settlement and a 100 mile warning radius can be extended into international; and (6) a 100 channel telephone cable exists to the Hawaiian islands.

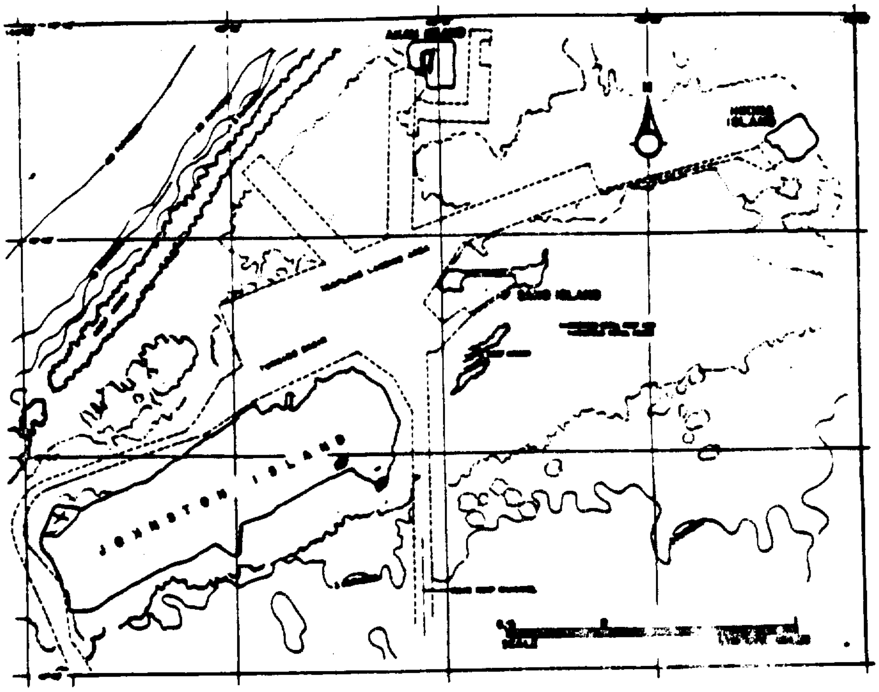
The Defense Nuclear Agency and U.S. Army currently have activities on the island so that personnel with security experience already exist on the island.

Clearly, construction of facilities will be more expensive at JI, primarily due to transportation costs. The general rule of thumb, however, is that facilities cost a factor of 3 more. Applying this factor to the capital costs in Table 3 results in only a $1.4 billion increase if the entire NRDS facility were reproduced on JI. Since only the full power test stands may be needed, the use of JI may not entail a major cost increase at all!

The other option instead of testing at JI is to explore zero power tests at NTS, the still "cold" NTR could then be launched on the Shuttle, docked with previously filled LH_2 tanks in orbit, and full-power tested. Once completed, the NTR could launch itself into small helio-centric orbit for disposal. The increased difficulty, however, of post-burn analysis of engine components may preclude orbital testing.

POSSIBLE DEVELOPMENT

Several improvements to the NRX engines are possible if time and budget allow for some development. Studies performed at the end of the NERVA program indicated that an Isp of 900 is achievable and that the use of UC-ZrC fuels might allow an operating Isp of 975 which would substantially reduce the required mass of the Mars ship in Earth orbit. Even without a change in the nuclear fuel structure, improvements in the engine Isp may be possible by reducing the operating lifetime require-



VICINITY MAP – JOHNSTON ATOLL

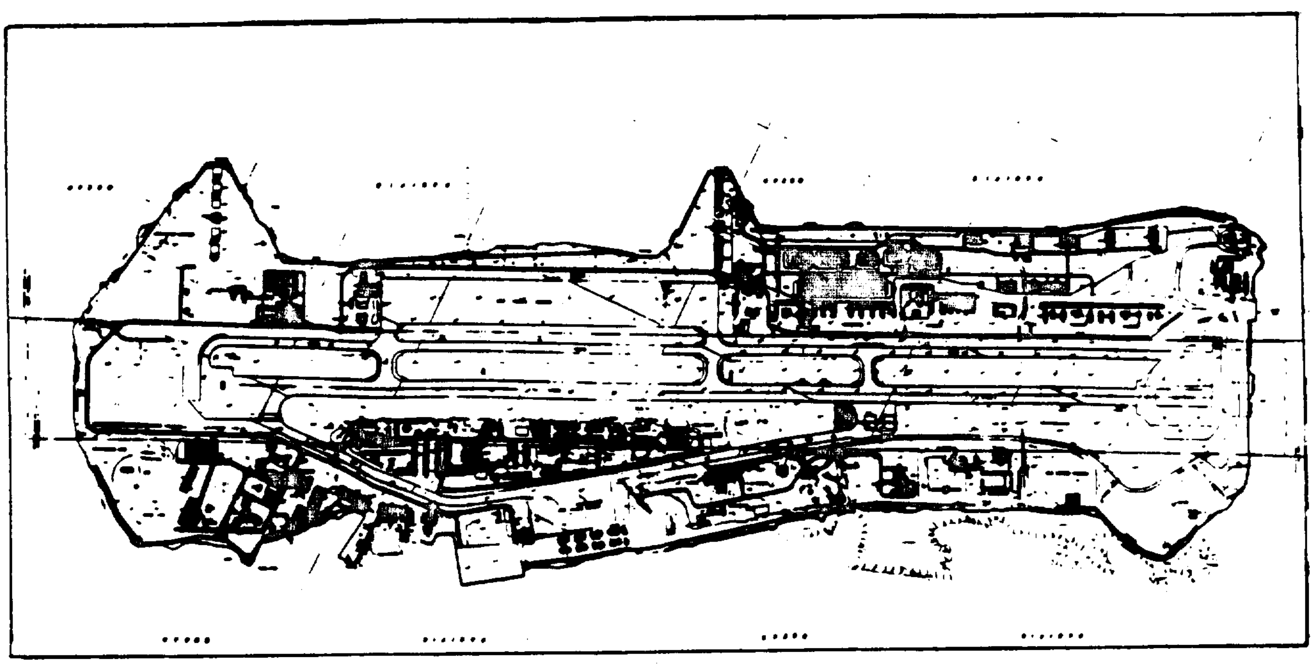


Fig. 6. Schematic views of the Johnston Atoll and Johnston Island.

ments from 10 h down to 3-5 h, thus allowing the reactor to run at higher temperatures. Furthermore, several new materials with improved qualities such as Mo-Re alloy and high strength ceramics have been developed in recent years and may significantly improve device performance.

An attractive concept which was developed in the early 1970s[10] was to operate the NTR in a lower power mode after the impulse burn to produce eletrical power for the ship. Calculations at the time indicated that a closed loop rankine cycle using an organic working fluid could provide electricity for an additional mass of about 70 kg/KW(e).

The development of the SP-100 program and the associated technology has provided another avenue for dual mode NTRs. After the high power burn of the NTR, high temperature heat pipes may be inserted into the core. The heat conducted out of the core would then be used to operate thermoelectric converters to provide a fluctuating power level as required. The electrical power produced could even be of sufficient magnitude to power an electric propulsion system. Such a dual mode of propulsion system employing a single set of reactors may provide the ideal symbiosis between impulse and continuous thrust systems and allow the shortest, feasible transit time to Mars of any near term propulsion systems.

SUMMARY

An operating nuclear thermal rocket engine has been thoroughly tested during the NERVA program which ended in 1971. Estimates made at the end of the program concluded that the ground tested Isp of the engine of 825 s would equate to about 900 s in a flight-qualified engine. If NTR's were used for a manned Mars mission, the required mass in LEO would be reduced by almost a factor of 3. For the all propulsive braking scenario, this translates into about 1.6 million pounds instead of about 4.5 million pounds for the NTR and chemical systems, respectively. The launch costs which would be saved would be greater than \$5 billion. Preliminary cost estimates to rebuild the NRX engine tested in the NERVA program are between \$3-5 billion. These estimates include the expense of building a full power test stand at Johnston Island in the Pacific Ocean. If an all propulsive-braking mission is planned or if multiple Mars missions are planned, the cost of rebuilding a nuclear rocket appears to be justified.

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N87-17795

ELECTRICAL POWER SYSTEMS FOR MARS

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ABSTRACT

Electrical power system options for Mars Mission Modules and Mars surface Bases were evaluated for both near-term and advanced performance potential. The power system options investigated for Mission Modules include photovoltaics, solar thermal, nuclear reactor, and isotope power systems. Options discussed for Mars Bases include the above options with the addition of a brief discussion of open loop energy conversion of Mars resources, including utilization of wind, subsurface thermal gradients, and super oxides.

Electrical power requirements for Mission Modules were estimated for three basic approaches: (1) as a function of crew size; (2) as a function of electric propulsion; and (3) as a function of transmission of power from an orbiter to the surface of Mars via laser or r-f.

Mars Base power requirements were assumed to be determined by production facilities that make resources available for follow-on missions leading to the establishment of a permanently manned Base. Requirements include production of buffer gas and propellant production plants.

INTRODUCTION

The discussion of electrical power systems for the Mission Module considers a range of power requirements and power source options. Photovoltaics (PV) were selected for more detailed discussion. General performance characteristics of solar arrays and regenerative fuel cells are presented and weights for a 25 kW system are tabulated. Solar thermal (ST), nuclear reactor (RX), and isotope power system (IPS) are also considered.

The discussion of electrical power systems for a Mars/Phobos/Deimos base includes the above options in addition to methods of converting surface and atmospheric resources to electrical energy. Hybrid voltaic-isotope power systems and a nuclear reactor are selected for more detailed discussion.

MISSION MODULE

Power Requirements

Electrical power requirements are driven by crew size, electric propulsion options, and options to supply Mars surface power from a Mars Orbiter via laser or r-f transmission.

The crew size function is estimated to be 1.9 kW per person for life support systems. This leads to a 25kW Mission Module for a 6-person crew: 12 kW for life support and another 13 kW for subsystems and science. A 12-person crew requires 24 kW for life support and requirements for subsystems and science result in a 40 to 50 kW power system. Electric propulsion requiring multi-MW's dwarfs these requirements.

Laser transmission of power from a synchronous Mars orbit to the surface requires substantial technology development but is attractive because it may offer surface mobility and because of synergism with electric propulsion, i.e., surplus power is available from the electric propulsion power source after arrival at Mars. However, neither caveat is essential if an end-to-end efficiency of 6.3 percent can be achieved. For example, a 160 kW orbital power source could supply 10 kW of usable power to the surface if the transmitting antenna can be pointed with a 0.01 arc second accuracy. The diameter of the surface receiver would be 16 meters.

The viability of r-f transmission is dependent on the ratio of the square of transmission length to the area of the transmitter. Therefore, practical systems have either large size or modest transmission length. For example, the Space Power Satellite (Ref. 2) supplied 6.8 GW to an 85 km^2 ground rectenna from a 7.1 GW, 0.78 km^2 space antenna which orbited the Earth at 36,800 km. Attempts to scale this system into the MW range resulted in negligible power to the ground rectenna. Similar results may be expected for a Mars power satellite because synchronous orbit, 17,034 km, is of the same order as Earth synchronous orbit. However, synchronous orbit for Phobos was estimated to be 14 km. If such an orbit is possible, r-f transmission in the 10's of kW range may be practical.

Power Source Options

The following discussion places emphasis on, but is not limited to, a 25 kW manned Mission Module.

Four power system technologies shown in Figure 1 are considered to be good candidates for the Mission Module. PV is selected for detailed discussion as the most probable choice, although we caution that trade studies were not performed and that power source selection is strongly influence by mission requirements. Possible merits and limitations of the alternate technologies are explained briefly.

ST systems relative to PV systems have approximately the same weight require one-half to one-third as much planform area and provide 80 to 90 percent thermal shielding. Shielding may be an advantage for certain Mission Module configurations if LOX/LH_2 propulsion is used. However, ST systems require a relatively large concentrator mirror that may require retraction or added structure to accommodate burn periods and possible artificial-g requirements (also added pointing complexity to achieve typical pointing accuracy of 0.1 degrees).

RX systems that do not require man rated shielding should be weight competitive with solar power systems at power levels greater than 25 kW. System weight of the SP-100 reactor (Ref.6) is 2800 kg (35 W/kg) at 100 kW. A 25 kW system, weighing 1700 kg is presented in the discussion of Mars Bases. Performance of very large reactors (1-10 MW) for nuclear electric propulsion is estimated to be 125 w/kg elsewhere in this paper.

RX systems requiring man rated shielding are not competitive with solar power systems unless power levels of 400 kW to 1 MW are considered. System weight is estimated to increase from 15 MT (1 MT = 1000 kg) to 30 MT over a 10 kW to 400 kW power range. On this basis, performance is 13 W/kg at 400 kW and 35 W/kg at 1 MW. Redundancy requirements could further penalize RX performance even though typical designs utilize multiple fuel rods, heat pipes and converters.

IPS, using either dynamic or Alkali Metal Thermoelectric Converters (AMTC's), are presented in the discussion of Mars Bases. Performance does not exceed solar power systems for Mission Module application and cost, safety and availability considerations suggest that isotope power systems may be reserved for the more stringent environment on the surface of Mars.

Photovoltaic Design Values

Photovoltaics design values are tabulated in Table 1. The table lists seasonal solar intensity variation for Earth and Mars orbits.

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FIGURE 1. MARS MISSION MODULE ELECTRICAL POWER SYSTEM OPTIONS

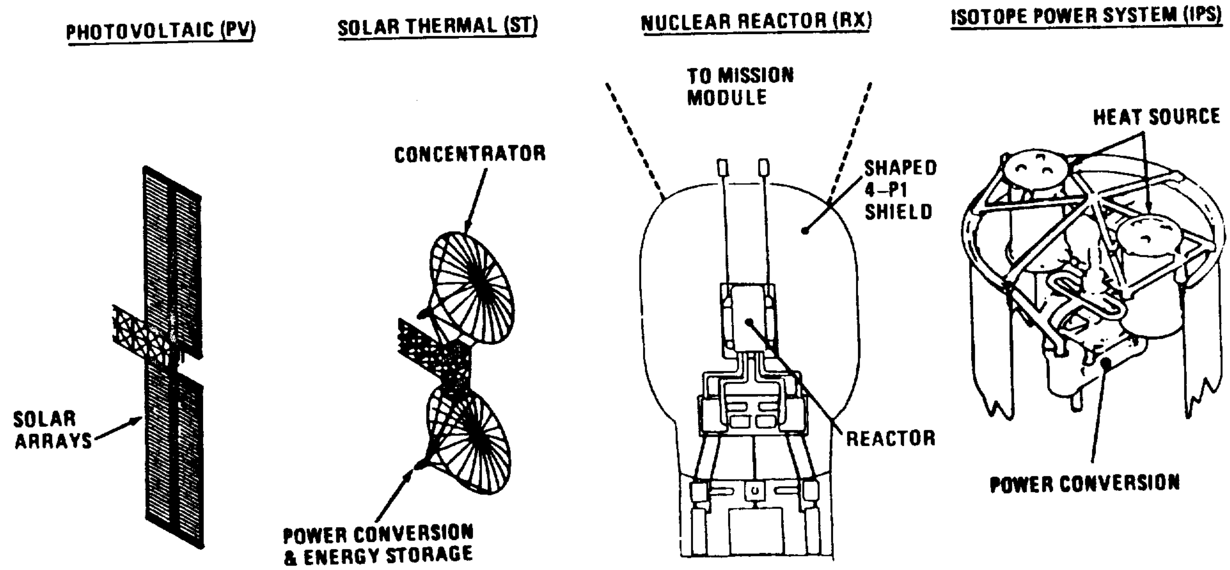

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TABLE 1. PHOTOVOLTAIC DESIGN VALUES FOR MISSION MODULE

	EARTH	MARS
SOLAR INTENSITY		
PERIHELION	1399 W/M^2	708.8 W/M^2
AVERAGE	√ 1353	√ 582.8 (.43 RELATIVE)
APHELION	1309	487.0
SOLAR CELL DEGRADATION		
LEO ASSEMBLY	.95	
MARS ARRIVAL		.9
LEO RETURN	.87	
SOLAR ARRAY PERFORMANCE		
Si PLANAR		
W/M^2	120–155** (150) √	X.43 RELATIVE INTENSITY
W/KG	75–144 (100) √	X1.09 LOWER TEMP
KW/M*	37	X COS 20° OFF POINTING
$/W	46–100	
GaAs CONCENTRATOR		
W/M^2	150–133	1° POINTING
W/KG	25–41	
KW/M*	54–38	
$/W	30–166	

* LAUNCH PACKAGING EFFICIENCY IN UNITS OF KW PER METER OF STS ORBITER PAYLOAD BAY

** INCREASE FROM 155 W/M^2 TO 200 W/M^2 IF GaAs

√ REFERENCE SELECTION

Selected values are 1353 W/m^2 in Earth orbit and 582.8 W/m^2 in Mars orbit. Degradation from solar radiation is estimated to be 0.95 to 0.87 for two-year round trips. Table 1 also tabulates a range of performance values for state-of-the-art Si-planar and GaAs concentrator (1 degree pointing required) solar array designs. Earth performance values of 150 W/m^2 and 100 W/kg for a Si-planar array are selected for the analysis assuming simple retraction for burn periods and 20 degree pointing capability to accommodate artificial-g or other orientation constraints. Accounting for reduced solar intensity, degradation and temperature performance at Mars is 63 W/m^2 and 42 W/kg.

Energy Storage

Energy storage options required by PV systems, if full Sun operation is not possible, include regenerative fuel cells (RFC), fused salt batteries, and flywheels. RFC's are selected for analysis assuming energy storage must accomodate both a highly elliptical orbit at Mars such that the Mission Module is in the Sun for 20 hours and in shadow for 4 hours (worst case) and a circular orbit at Earth with 1 hr sun and 0.6 hr shadow periods. RFC's have a particular advantage in handling either 4 hours Mars shadowing or 0.6 hour LEO shadowing because fuel cell size remains the same for either shadow period. However, the electrolyzer unit must be sized for LEO operation and H_2O and H_2/O_2 tanks must be sized for longer occultation periods at Mars. RFC performance for LEO application is typically 15 to 45 Wh/kg depending on whether the system is optimized for weight or efficiency. Compromises are necessary for dual use at both LEO and Mars: electrolyzer sized by LEO and tanks sized by Mars. Resulting performance, for an efficiency optimized system, is 19 Wh/kg at LEO and 125 Wh/kg at Mars (19 x 4 hr/0.6 hr).

Energy storage is probably not required for large PV or ST propulsion systems because the penalty for not thrusting during occultation periods at Earth or Mars orbit is minimal.

Photovoltaics--Regen Fuel Cell Performance

The weight of a state-of-the-art 25 kW PV system, with and without energy storage requirements, is tabulated in Table 2. Note that the solar array is oversized in the RFC option to recharge the electrolyzer and in the PV only option to overcome distribution losses. The performance range is 13 to 35 W/kg.

3479–86

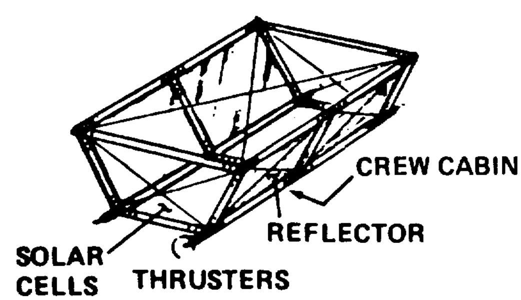
TABLE 2. 25 KW, PV SYSTEM – MARS ORBIT

COMPONENT	PV–RFC	PV ONLY
SOLAR ARRAY		
POWER	40 KW	30 KW
AREA	650 M^2	475 M^2
WEIGHT	980 KG	710 KG
RFC AND TANKS	930 KG	0
	1910 KG (13 W/KG)	710 KG (35 W/KG)

3480–86

FIGURE 2: PHOTOVOLTAIC ELECTRIC PROPULSION (PEP)

	WEIGHT (MT)	
SOLAR ARRAY		333
SOLAR PANELS	230	
CONCENTRATOR	33	
STRUCTURE	51	
DISTRIBUTORS	11	
OTHER	8	
THRUSTER ASSEMBLY		189
THRUSTERS	24	
POWER PROC.	2	
BATTERIES	154	
OTHER	9	
TANKS (DRY)		85
		607



An advanced Photovoltaic Electric Propulsion (PEP) system is shown in Figure 2 along with a weight tabulation (Ref. 1). This conceptual design, part of the Space Power Satellite (SPS) studies (Ref. 2), transferred a 6860 MT cargo from 486 km Earth orbit to GEO in 120 days. The PEP concept is considered to be applicable to Mars missions as an alternative to Nuclear Electric Propulsion. Power, 309 MW, for the ion thrusters was provided by a 2 to 1 GaAs concentrator solar array weighing 333 MT (930 W/kg). Overall dimensions of the PEP was 1500 M x 1400 M planform area x 606 M concentrator height and the total dry weight was 607 MT. NaS batteries, 155 MT, where utilized to accommodate maximum LEO gravity gradient torques during occultation periods but may not be required for Mars missions. A specific weight of 930 W/kg, based on LEO to GEO intensity, would be reduced to approximately 400 W//kg at Mars.

The analyses of Mission Module power systems reaches the following conclusions: (1) Requirements for crew delivery modules are 25 to 50 kW; (2) Electric propulsion requires multi-MW's; (3) Laser transmission is practical in the 100's of kW range but r-f transmission requires GW's; (4) Several power system technologies are available to accomodate a variety of mission objectives and power levels; (5) State-of-the-art performance referenced to Mars orbit is 13 to 35 W/kg for 10-100 kW systems; and (6) Performance of large, MW electric propulsion systems is 100 to 400 W/kg.

MARS BASE

The production of electrical power on the Mars surface is considered to be difficult, and existing terrestrial and space power system technology development programs do not address environmental and operational constraints unique to Mars. Because the potential to produce energy from surface or atmosphere constituents of Mars is not well understood, a basic approach to power generation cannot be clearly established. Compounding the difficulty are considerations of fixed or mobile bases at Mars, Phobos, or Deimos, each having a variety of growth scenarios.

The possibility of supplying electrical power from either the conversion of Mars surface and atmospheric constituents or the collection of solar energy is briefly explored. Two conventional power systems, a

10 kw PV-IPS system and a 25 kW RX system are presented for the purpose of investigating first-order design constraints.

Open loop energy conversion, similar to Earth power plants, would be the preferred method of power generation at Mars if a readily available source of fuel, such as superoxides, can be extracted from the surface. However, the present understanding of in-situ fuel production of e.g., CH_4 and O_2 from the atmosphere and surface, as described by Ash (Ref. 4), while useful to store energy for intermittent power generation, is not practical for steady state operation; 6 kWh/kg input required energy to the process plant and only 1 kWh/kg is converted to electrical energy by a fuel cell or generator.

LOX and LH_2 can be brought from Earth and converted to electrical energy by a fuel cell with a conversion efficiency of 2.5 kg/kWh. Reactant and tank weight make this approach unacceptable for operation beyond a few weeks.

Cursory investigation of wind or subsurface thermal energy sources initiated that these did not appear to be a readily available solution for either Mars, Phobos or Deimos. Mars windmills do not appear to be practical because the atmosphere is very thin (0.008 atm) and nominal wind speed is moderate (2 to 7 m/s). Subsurface thermal energy production (geothermal equivalent) may be possible at certain sites at Mars.

ST systems were not considered in detail but may be a competitive option if they can be manufactured from natural resources at Mars.

Photovoltaic Design Values for Mars Base

Photovoltaic systems for a Mars Base are considered with reference to Table 3. A nominal solar intensity value, $582.8 W/m^2$, is modified by several factors at the Mars surface: (1) increased performance over Earth orbit because the solar array operates at about -20° C instead of typical LEO, temperatures of 70° C; (2) radiation degradation, 5 percent after initial deployment and 10 to 20 percent after 15 or 20 years; (3) a 20 percent allowance for dust obscuration and abrasion; and (4) a 50 percent weight improvement over typical self-deployed solar arrays designed for LEO assuming manned or simple spring deployment. Simple deployment also implies that the solar array be deployed flat along the surface. The performance penalty is 0.53 at 30° latitude and 0.614 if the array is located at the equator or is tilted to compensate for higher latitudes. Performance of a flat array at 30^{o} latitude is 28 W/m^2 and 28 W/kg as compared with LEO performance of 150 W/m^2 and 100 W/kg.

Two additional considerations are included in Table 3. First, the time available to collect solar energy on Mars is approximately 50 percent of the 24.62 hr. rotation period. The percentage was modified slightly because solar arrays are not useful for morning and evening sun angles below 15^{o}. Secondly, dust storms were assumed to totally obscure the Sun for 60 days of the 687-day Martian year. These factors reduce performance to 12 W/m^2 or 12 W/kg on an annual basis. Detailed solar radiation data can be found in reference 7.

Solar energy collection at Phobos and Deimos is only slightly better than solar collection at Mars. Both moons orbit with the same side facing Mars. Also, Phobos is obscured by Mars for 0.9 hr. of a 7.65-hr orbit while Deimos is obscured by Mars for 1.5 hr. of a 30.6-hr orbit.

Point Design Power Requirements

Early outposts, or even an initial outpost, may place processing plants into operation in preparation for future missions. Power estimates assumed for this analysis are tabulated in Table 4. The processing plant produces Ar/N_2 buffer gas (carbon leakage make-up for next crew); water and compressed CO_2 for a greenhouse; and CH_4/O_2 propellants for surface and/or return vehicles in preparation for an 8-person crew planning 45 or 350 day stay times. Power requirements, 5 to 25 kW, matched 4 to 6-person crew requirements, 10 to 25 kW, and are considered to be reasonable for early missions. Power levels are modest because the power is furnished over the 400-day interval between missions (low power but high energy). The implication to power design is that long life, unattended, systems will be required.

Requirements for production of Ar/N_2 buffer gas are 0.5625 kWh/man-day yielding requirements of 200W to 1.5 kW. Compression of CO_2 from 0.008 to 0.1 atm for the greenhouse requires 0.027 kWh/kg. If the greenhouse requires 10 kg/day, 112 W of compressor power is required (Ref. 3). Water is extracted from the soil at 9.3 kWh/kg (Ref. 4) or from the atmosphere at 42 kWh/kg (Ref. 3). The power requirement range is 2.7 to 12 kW to supply 7 kg/day. Production of CH_4 and O_2 in quantities of 330 to 12,000 kg requires 2.1 to 7.5 kW of electrical power (Ref. 5).

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TABLE 3. PHOTOVOLTAIC DESIGN VALUES AT MARS

SOLAR INTENSITY	582.8 W/M^2 (.43 RELATIVE TO EARTH ORBIT)
TEMP. CORRECTION	1.09 RELATIVE TO EARTH ORBIT
DEGRADATION	.95
OBSCURATION	.80 DUST/SPECTRAL RESPONSE
GEOMETRY	.53 FLAT ARRAY/NO ORIENTATION 30° LATITUDE/15° MIN SUN SINGLE
STRUCTURE	1.5 SIMPLE DEPLOYMENT
REF. LEO PERFORMANCE	150 W/M^2 AND 100 W/KG
MARS PERFORMANCE	28 W/M^2 AND 28 W/KG
DAY/NIGHT TIMES	
ROTATIONAL PERIOD	24 HR, 37 MIN, 23 SEC
SUN TIME, EQUATOR	50%
USABLE SUN TIME	11.89 HRS, 15° TO 165° SUN ANGLE
ENERGY STORAGE TIME	12.73 HRS
ANNUAL POWER AVAILABILITY	
ORBITAL PERIOD	687 DAYS
DUST STORM OBSCURATION	60 DAYS

3086–85

TABLE 4. EARLY MARS BASE POWER REQUIREMENTS

	POWER (KW)
• Ar/N_2 BUFFER GAS	
– STORED FOR 8–MEN CREW	
– 45 DAY OR 350 DAY STAY TIMES	.2 TO 1.5
– 202 TO 1575 KG STOCKPILE	
– CONVERSION ENERGY 9.4 KWH/KG OF AR/N_2	
• H_2O FOR GREENHOUSE	
– 7 KG/DAY REQUIREMENT	
– CONVERSION ENERGY	2.7 TO 12
– 9.3 KWH/KG FROM SOIL	
– 42 KWH/KG FROM ATMOSPHERE	
• CO_2 COMPRESSED FROM 0.008 ATM TO .1 ATM FOR GREENHOUSE	
– 10 KG/DAY REQUIREMENT	.1
– CONVERSION ENERGY OF .027 KWH/KG	
• CH_4 AND O_2 IN SITU PROPELLANT PRODUCTION	
– 3330 KG TO 12,000 BY REQUIREMENT	2 TO 8
– CONVERSION ENERGY 6 KWH/KG	
	5 TO 25 KW

A 10 kW Photovoltaic/Isotope Power System

A reference design is defined for the purpose of exploring first order system considerations. Three combinations of PV and IPS power systems to supply 10 kW to an initial Mars Outpost are shown in Figure 3. Liberty is taken in interperting the 10 kW requirement. The requirement is assumed to be satisfied if day/night loads are unbalanced to provide 12 kW of day power and 7 kW of night power. Unbalanced loading favors PV designs by minimizing energy storage requirements. Although a regenerative fuel cell was selected for this analysis, inertial energy storage and high performance batteries, such as NaS and Li, should be investigated.

The use of IPS was minimized based on the assumption that isotope inventory should be minimized for cost, availability and safety reasons. These considerations tend to limit IPS to a few Kw output. Performance numbers are based on typical 20 to 30 percent dynamic converter efficiencies of existing Organic Rankine or Brayton cycle converters. Static Alkali Metal Thermoelectric Converter (AMTEC) or dynamic Stirling cycle engines may achieve 30 to 40 percent efficiency.

The first design supplies 10 kW from a solar array and 5 kW from a RFC energy storage system. IPS supplies 2 kW of continous power to satisfy the 12 kW day and 7 kW night power requirement. A series of concerns emerge from this design.

First, an 860 sq. meter solar array, 20 percent of a football field, must be deployed. Crew capability to deploy the solar array is unknown and because existing technology deals only with free-flyer deployment, there is little basis for evaluating spring loaded or other unmanned deployment alternatives.

Secondly, existing RFC technology emphasizes LEO Space Stations utilizing low pressure (150 psi) and high temperature (100^{o} C) H_2 and O_2 gas storage. Tank size, for these conditions is prohibitive for 12-hour energy storage requirements at the Mars surface. Cool-down to 20 degrees C and operation of the electrolyzer unit at 400 psi is assumed in projecting a packaging volume of 5 cu. meters for the total PV-IPS system. Package volume for the solar array is estimated to be 550 W/cu. meter.

Heat rejection requirements for the IPS and fuel cell portion of the RFC, 11 kWt, may be a major system level factor. Thermal control will be presented in other papers, however, in-house discussion suggests that requirements will be modest under normal conditions but possibly severe during large dust storms.

A final design option utilizes a 5 kW solar array without a RFC, and a 7 kW IPS to supply 12 kW day loads and 7 kW night loads. Performance is reduced slightly from 6.3 w/kg to 6.1 W/kg but solar array area is reduced from 860 to 210 sq. meters and packaging volume is reduced from 5 to 2.5 cu. meters. Heat rejection increases from 11 to 25 kWt.

Power requirements for the Initial Outpost were estimated previously to be in the 10 to 25 kW range. Results from the 10 kW PV-IPS system study were indication that a more optimistic approach should be investigated for 25 kW systems. Accordingly, a RX concept is selected for analyses.

A 25 kW Nuclear Reactor Power System

The RX concept depicted in Figure 4 requires a hole to be excavated, at say 50 meters from the Outpost; that a "reactor sled" be towed into the hole by an existing surface vehicle; and that the power system be connected electrically to the Outpost by a tether.

A weight tabulation, based on the on-going SP-100 program (Ref. 6), is included in Figure 4. The weight is 1700 kg or 15 W/kg. Performance more than doubles that of the hybrid PV-IPS system and should therefore be given consideration for the initial outpost. The largest weight contribution to the system is a 700 kg instrument rated shield. If the shield can be eliminated altogether, e.g., by excavating a right angle hole or burying the reactor, performance increases to 25 W/kg. Although this performance is inviting, consideration must be given to potential nuclear contamination of Mars, the lack of redundancy, heat rejection during dust storms, and requirements for power while the reactor is being placed in operation. Also, the SP-100 reactor is expected to use refractory metals which would be unacceptable in the CO_2 atmosphere of Mars. Refractory metals would not be a problem on Phobos or Deimos.

SUMMARY

Several power source technologies were evaluated for Mission Module application: (1) Photovoltaics (PV), (2) Solar thermal (ST), (3)

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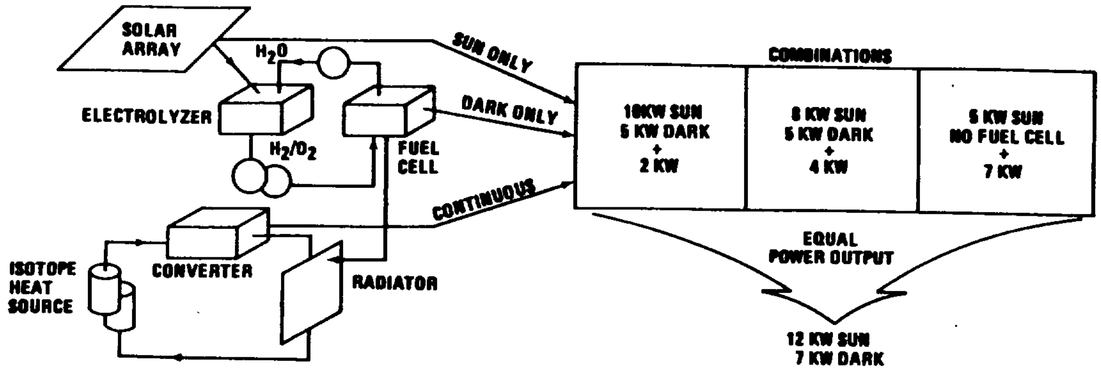
FIGURE 3. ELECTRICAL POWER FOR MARS LANDER PHOTOVOLTAIC + REGEN FUEL CELL + ISOTOPE



SOLAR ARRAY AREA	$860M^2$	$600M^2$	$210M^2$
SYSTEM MASS	1580 KG	1610 KG	1630 KG
SYSTEM VOLUME	$4.8M^3$	$3.9\ M^3$	$2.5M^3$
HEAT REJECTION	11 KWt	16 KWt	25KWt

1467—86

FIGURE 4. ELECTRICAL POWER FOR MARS LANDER 25 KW NUCLEAR REACTOR

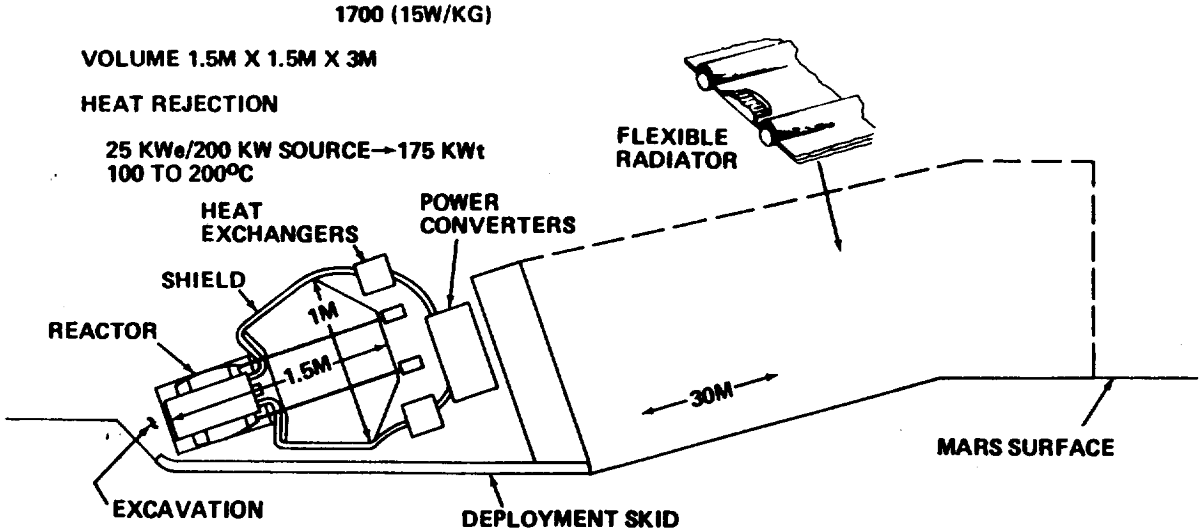
WEIGHT (KG)

REACTOR	350
SHIELD	700
CONVERTER	300
RADIATOR	100
SKID	250
	1700 (15W/KG)

VOLUME 1.5M X 1.5M X 3M

HEAT REJECTION

25 KWe/200 KW SOURCE→175 KWt
100 TO 200°C



Nuclear reactors (RX), and (4) Istope power systems (IPS) with either dynamic cycle or AMTEC engine energy conversion. Performance of a 25 kW state-of-the-art PV-regenerative fuel cell system was estimated to be 13 W/kg and performance for an advanced 309 MW photovoltaic electric propulsion system was estimated to be 400 W/kg. The state-of-the-art systems PV, ST, RX and IPS, require modest development to satisfy early missions, whereas large, advanced high performance systems, necessary for electric propulsion, require significant development.

The same power source options were evaluated for a Mars/Phobos/Deimos Base. Performance ranged from 6 W/kg for a 10 kW PV-IPS hybrid system to 25 W/kg for a 25 kW RX system. Requirements of long life, unattended systems were hypothesized assuming early, or even initial, mission objectives to operate in-situ production plants (5 to 25 kW power level). Reduced solar intensity, self-deployment over rough terrain, 0.3 gravity, CO_2 atmosphere, wind, dust storms, long night periods, etc., were considered to pose severe design challenges which are not being addressed by existing power technology programs. Energy systems to convert Mars resources to electrical power super oxides, CH_4/O_2, and subsurface thermal gradients should also be investigated.

The author wishes to thank Mr. W. Brantley and others from the Preliminary Design Office at MSFC, Dr. K. Faymon from LeRC, Mr. M. Shirbacheh from JPL and Mr. S. Howe from LANL for providing valuable assistance in preparing this paper.

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N87-17796

MANNED MARS MISSION

COMMUNICATION AND DATA MANAGEMENT SYSTEMS

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ABSTRACT

A manned Mars mission will involve a small crew and many complex tasks. The productivity of the crew and the entire mission will depend significantly on effective automation of these tasks and the ease with which the crew can interface with them. The technology to support a manned Mars mission is available today; however, evolving software and electronic technology are enabling many interesting possibilities for increasing productivity and safety while reducing life cycle cost. Some of these advanced technologies are identified.

1.0 INTRODUCTION

The Communications and Data Management Systems for the manned Mars mission are part of a number of end-to-end information systems. The function of each of these systems is to transfer information between a user and a domain of interest. The user may be on the Earth or in space. The domain of interest may be a user investigation, an engineering subsystem, or a human being. The information transferred may be for control, monitoring or mutual interaction. This paper is aimed primarily toward the space data systems and the communication links involved with the transfer of information beyond the Earth's atmosphere.

Specific requirements for the communication and data systems to support a manned Mars mission depend largely on the mission objectives, mission duration, and the number of vehicles involved; however, the following general characteristics are required to support any manned mission to Mars:

- o Transparency: Users should not be forced to deal with complex embedded systems.
- o Reliability: The systems should operate in space for years with little or no maintenance.
- o User Responsive: The systems should provide for rapid and adaptive turnaround of pertinent easily understood information.

- o Cost Effective: Small gains in performance should not drive costs when adequate alternatives exist.

Key issues regarding flight Communication and Data Management Systems (CDMS) for a Mars mission are:

Communications

- o Extravehicular Space Links
  - Data relay versus direct links between a Mars base and the Earth
  - Frequency (Hz)
  - Communication coverage
  - Data rates
  - Communication security
- o On-Board Communications
- o Data Systems
  - Degree of Autonomy/automation
  - Data system architecture

2.0 COMMUNICATIONS

The number of communication links depends on the number of vehicles involved in the mission and the amount of communication coverage required. Figure 1 shows some potential communication links in support of a Mars mission. The engineering options identified include:

- o Data relay verus direct links between a Mars base and the Earth
- o Frequency of the communication links

2.1 DATA RELAY VERSUS DIRECT LINKS

Some concepts for a manned mission to Mars involve a habitable Mars orbiter (MO) that will remain in orbit around Mars while crewmen are on the surface of Mars. If a relay system is used for communications between a Mars base and the Earth, either the Mars orbiter(s) or a set of dedicated communication satellites in orbit around Mars can be used to provide the relay capability. If the Mars orbiter(s) (assumed to be in a highly elliptical orbit) is used as a communications relay, its orbital period must be the same as the rotation period of Mars and the phasing must be such as to permit its position to oscillate about the zenith of the Mars base. Figure 3 shows typical coverage provided for a single base by a single Mars orbiter. For the orbit and mission shown, communication coverage for the Mars base is approximately 40 percent of the time. Additional coverage could be provided by a second Mars orbiter or by a system of dedicated relay satellites.

Use of a direct link between the Mars base and the Earth will reduce the orbital parameter restrictions on the Mars orbiter and/or eliminate the cost of dedicated communication relay satellites. A direct link can provide communications for approximately 50 percent of the time, but the duration of a single blockage is approximately 76 percent greater than for a link using a single Mars orbiter as a relay. The direct link will also require more weight on the surface and consume more power from the surface elements than a relay link via an orbiting vehicle.

A relay system appears attractive for the primary communication link between a Mars base and Earth. Regardless of whether the relay or direct approach is chosen for the primary link, the alternate approach will probably be used as a backup.

2.2 FREQUENCY OPTIONS

Frequency options for communication links between Earth and Mars include S/X band, K_a or MM-wave communications and optical. The data rate requirement will be a major factor in the ultimate choice of transmission frequency. A comparison of these frequency options is shown in Figure 2.

The existing Deep Space Network (DSN) uses S-band for transmission to the spacecraft and S&X band for reception. An X-band uplink capability is being developed with planned evolution toward a unified X-band two-way system in the 1990's. An X-band system together with the planned 70-meter antenna subnet and reasonable spacecraft antenna size and power levels can support a Mars mission with data rates on the order of 10-30 Mbps or less provided the bandwidth of the DSN 70-meter subnet electronics is increased to support these rates.

In 1979, the World Administrative Radio Conference (WARC) allocated a 34 GHz/32 GHz (K_a) band for deep space use. NASA is currently pursuing K_a-band technology and the DSN expects to provide a K_a-band receive capability on 70-meter antennas in the early 1990's. A K_a-band system can be expected to support data rates of five to ten times (depending on system noise temperatures) the rates supported at X-band for the same antennas and power levels.

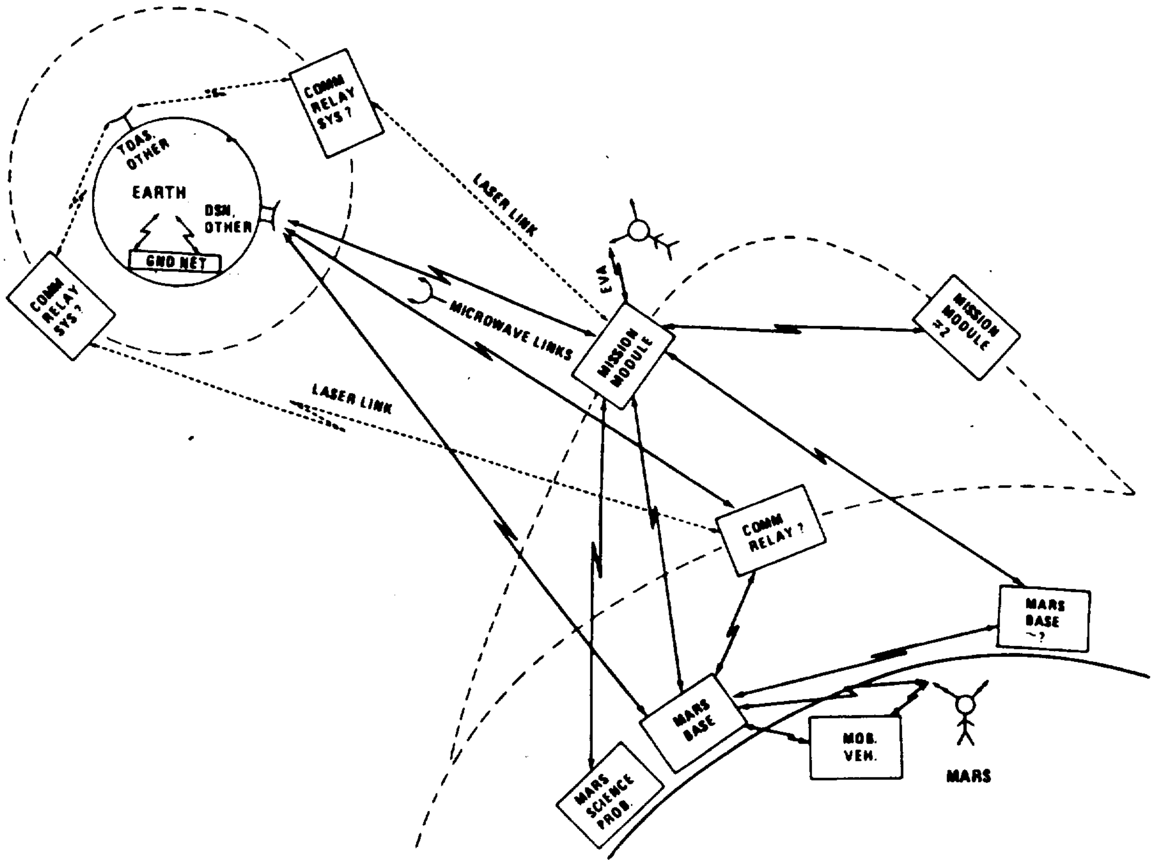

FIGURE 1. MANNED MARS MISSION
WHAT ARE THE COMMUNICATION LINKS?

	S/X BAND	KA/MM-WAVE	OPTICAL
BANDWIDTH	FEW MBPS	INCREASED	MUCH INCREASED
ANTENNA GAIN	REFERENCE	INCREASE OF 12 dB OR MORE OVER X-BAND	INCREASE OF 60-80 dB OVER KA
IMMUNITY TO INTERCEPTION & JAMMING	POOR	BETTER	EXCELLENT
SIGNAL ACQ.	EASY	SATISFACTORY	DIFFICULT
PTG. ACCURACY	FEW ARC MIN	ARC SEC RANGE	ARC SEC-TO-SUB ARC SEC RANGE
LIFE TIME	OK	OK	SHORT LASER LIFETIME
COMPATIBILITY WITH EXISTING SYSTEMS	YES	NO	NO
TECHNOLOGY STATUS	MATURE	IMMATURE (TECH. DEV. PLANNED)	IMMATURE (SOME RISK)

FIGURE 2. FREQUENCY OPTIONS

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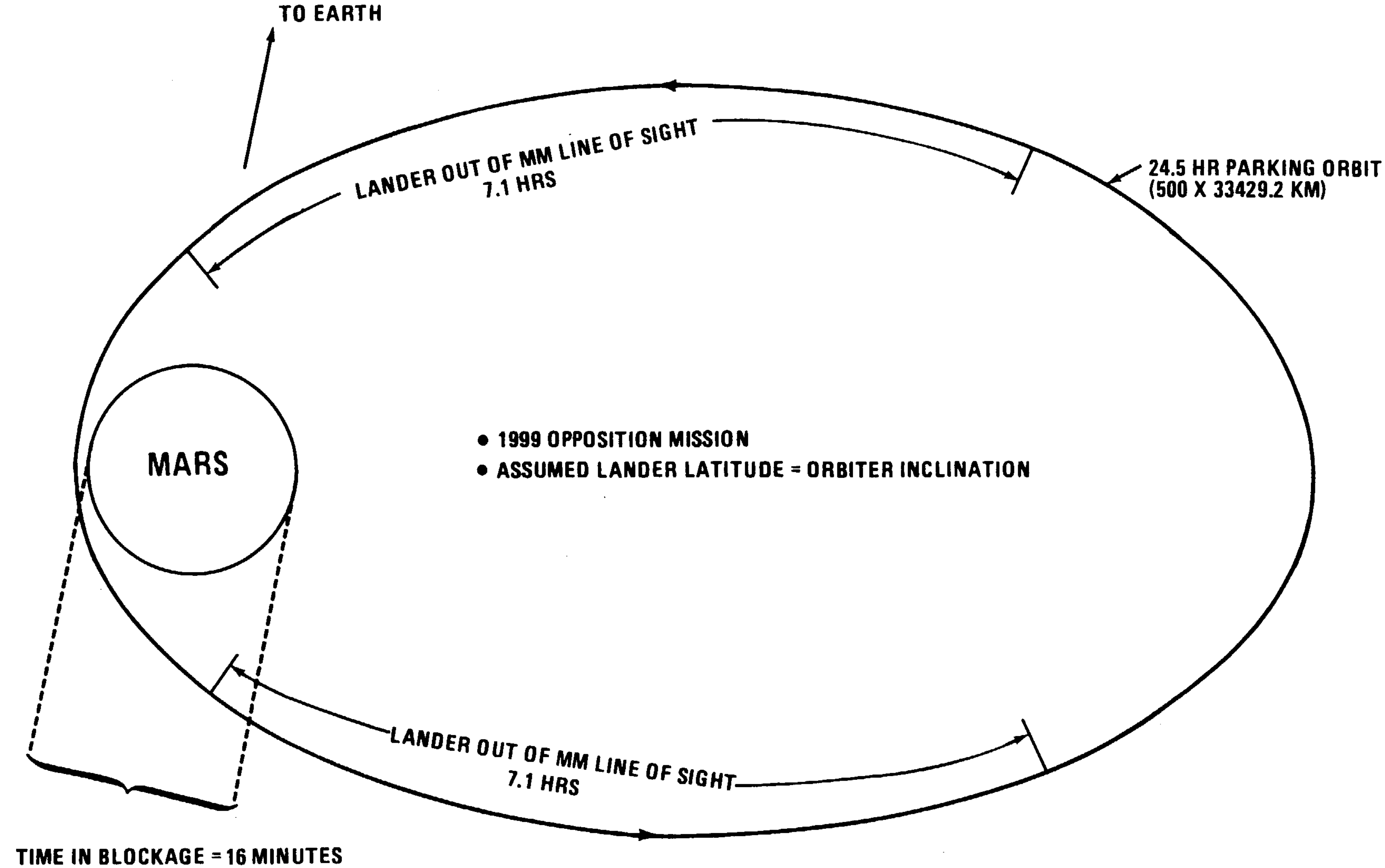

FIGURE 3. EARTH COMMUNICATIONS BLOCKAGE FOR A MISSION MODULE (MM) IN ORBIT AROUND MARS

For data rates greater than 100 Mbps, lasers should be considered as an alternative to the more conventional microwave links. Lasers can support large data rates with small transmitting/receiving apertures. However, some of the characteristics which make laser communication links attractive also make them difficult to use. The high gain which makes possible small apertures also requires very accurate pointing systems. Atmospheric attentuation, which together with narrow beamwidth makes unauthorized access to the communication links more difficult, requires a relay system of satellites orbiting the Earth in order to provide dependable communications with Earth based stations. A similar situation exists at Mars where a dust storm could block laser communication links. Therefore, it is envisioned that a laser link would be used only for communications between vehicles in Earth and Mars orbit with lower frequency communications to the surfaces.

For present near-term planning, a K_a-band system appears attractive for communications between the Earth and Mars. A K_a-band system can support moderate data rates, providing dependable communications with the Earth without a data relay system in orbit around the Earth.

2.3 COMMUNICATION COVERAGE

The amount of communication coverage required is a key factor that will influence the overall communication system architecture and the design of individual communication links. Figure 3 shows the communication coverage for a 1999 opposition mission that could be provided for a Mars base via a Mars orbiter in a highly elliptical 24.5 hour orbit of Mars. Because of orbit geometry and relative motion, the orbiter will drop below the horizon of the Mars base twice daily for approximately 7 hours each time. In addition, direct communications between the orbiter and the Earth will be blocked for approximately sixteen minutes per day as the orbiter swings behind Mars with respect to the Earth. If the communication coverage provided by a Mars orbiter is unacceptable to mission planners, direct links between the Earth and the Mars base together with relay via the Mission Module will reduce blockage to approximately 7.3 hours per day. If this amount of blockage is unacceptable, an additional relay system in orbit about Mars must be provided.

2.4 DATA RATES

The data rate requirement for each of the various communication links used to support a manned Mars mission is needed not only to size the system in terms of antenna size and power levels, but also to determine applicable technologies such as microwave versus optical transmissions. For a manned Mars mission, the data rate requirements will probably be driven by the science requirements and the video requirements. Data compression should be used on data from both sources to avoid the transmission of redundant or unneeded data. The degree to which data compression is used will be a major factor in determining the data rate for transmission. Figure 4 shows some communication links and the equivalent of the data rate in terms of color TV transmission assuming a data compression equivalent to 1 bit per pixel.

2.5 COMMUNICATION SECURITY (COMSE)

Communications links may require protection from unauthorized access, electronic deception and intelligent jamming. The extent of protection will impact the communiction system design. Early COMSE planning should include:

- o Identification of the treat environment
- o Degree of protection required
- o Assessment of required crypotography techniques, authentication methods, and anti-jamming features.

2.6 ON-BOARD COMMUNICATIONS

On-board communication systems include closed circuit TV, internal audio, the routing of scientific data, and interfaces with external communication links. For mobility, it is assumed that each crew member will be provided with a wireless communication set. Some of the issues/options that must be addressed for each manned vehicle are analog versus digital distribution of audio and video, dedicated versus multiplexed channels, electrical versus optical distribution, and centralized versus distributed control of these systems. The architecture and sizing of the on-board communication systems depend heavily on the data rates that they must support.

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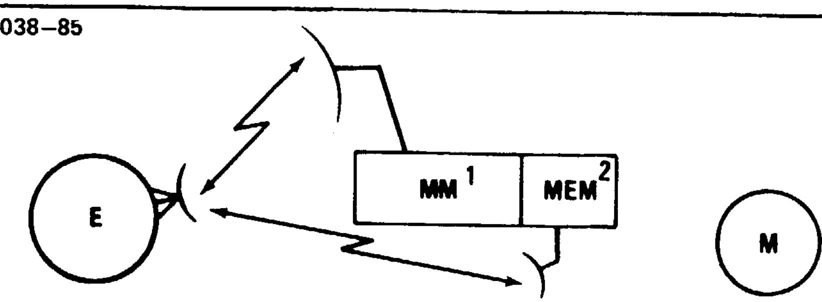
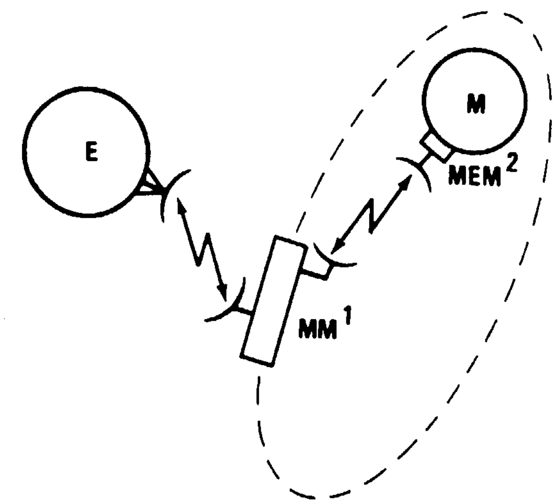


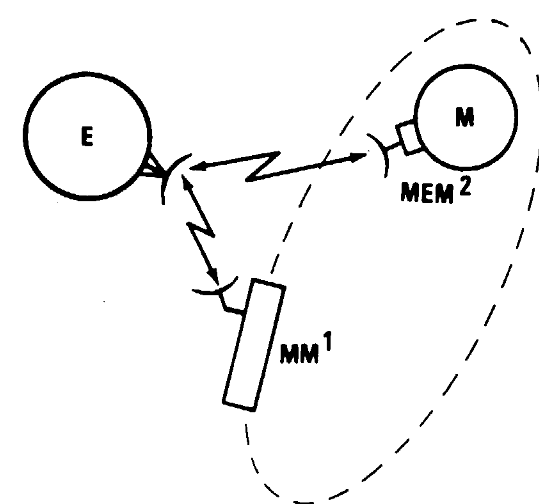
TRANSIT PHASE

MARS ORBIT PHASE

OPTION 1 – ORBITER RELAY



OPTION 2 – DIRECT COMMUNICATIONS



RETURN LINK REFERENCE CAPABILITY* @ X–BAND

	RANGE	DATA RATE (MBPS)	EQUIV COLOR TV TRANSMISSION**		ANT DIA (FT)	RF POWER (WATTS)
			FRAMES/ SEC.	SPATIAL RESOLUTION (PIXEL)		
RELAY OPTION						
●TRANSIT PHASE						
– MM	1 AU	9.2	30	640x480	16	230
– MEM	1 AU	.0528	~.2	640x480	4.3	20
●MARS ORBIT PHASE						
– MM	1 AU	← SAME AS TRANSIT PHASE →				
– MEM	33,000KM	10.0	32.5	640x480	4.3	20
DIRECT OPTION						
– MM	1 AU	← SAME AS FOR RELAY OPTION →				
– MEM	1 AU	8.0	26	640x480	16	200

* ASSUMES 2.3dB ADVERSE TOLERANCE FOR CLOUD COVER AND 1.9 dB IMPROVEMENT IN DSN
** ASSUMES DATA COMPRESSION EQUIV TO 1 BIT PER PIXEL
1 MISSION MODULE
2 MARS EXCURSION MODULE

FIGURE 4. RF LINK PERFORMANCE

3.0 DATA SYSTEMS

3.1 DEGREE OF AUTONOMY/AUTOMATION

For this paper, the word "autonomy" refers to the independence of the flight systems (those systems in space including the human systems) from Earth based support. The term "automation" refers to operations performed by machines, computers, etc., that otherwise would have to be done by humans. Automation is required implement autonomy for systems in space because of the limited number of flight crew personnel.

Except for the resupply of a long-term Mars base, the only support of the flight systems from Earth must be support that can be provided via the communication links. Because of the distances involved in a Mars Mission (up to 2 A.U.) and corresponding signal propagation delays, the onboard systems must at a minimum be capable of operating acceptably independent of Earth based support for periods of approximately 30 minutes plus some turnaround time on Earth. Beyond this basic requirement, the degree of autonomy and automation is a major issue that must be considered.

Because of the long mission duration and the distances involved, a high degree of automation including "expert" and other knowledge based systems is highly desirable. Automation of functions should result in the following benefits: (1) Relief of the flight crew or Earth based personnel from time-consuming tasks, thus improving productivity, (2) Enhancement of reliability and safety via continuous monitoring of system health, (3) Enhancement of system performance via faster or more consistent response, and (4) Reduction of operating cost.

Potential functions for automation (including system, subsystem and application functions) must be identified and the benefit of automating them must be assessed against the potential risks and implementation costs.

Assuming a high degree of automation, the next question is: Where should the automation be performed? Should the automation be performed by the flight data systems (autonomously) or by systems on Earth? The most critical factor in determining whether automation should be performed by the flight systems versus based systems is the response time requirement. Many functions will not tolerate the delays involved in an Earth communications loop. For these functions, the question is a non-issue. For other functions, trades must be performed to determine the desirability of implementing them with Earth based processing nodes versus the flight systems. Factors that must be considered include the cost of flight versus Earth based processing systems, differences in performance and safety levels, and cost differences for communication systems required to support different degrees of autonomy.

For missions using vehicles that orbit Mars in addition to those that land on the surface, a similar issue exists regarding the location of processing support for Lander functions; i.e. what support capability is left in the orbiter versus being carried in the lander.

3.2 DATA SYSTEM ARCHITECTURE

A manned mission to Mars may involve several vehicles, each having its own data system. The vehicles may vary from a module transferring men and materials to Mars to a Mars surface rover to intelligent robots. The architecture of each data system is influenced by vehicle configuration, data throughput requirements, and criteria for system operational reliability. An example of a hierarchical data system architecture potentially applicable to a manned vehicles for a Mars mission is shown in Figure 5.

Before selecting a data system architecture, the design engineer must consider a large number of options and trades.

- Processing Architecture: The data systems for the manned vehicles used for a Mars mission will utilize distributed processing to provide high througput, lower integration costs, and operational flexibility. However, the degree and type of distribution must be studied. Trade studies include hierarchical versus non-hierarchical, the number of levels in a hierarchical system, module-oriented versus subsystem oriented architectures, etc.

- Number of Physical Buses: Are all data routing functions performed by the same physical network or are separate buses used for the data management, audio, video, and science functions? Are timing and mass memory distributed on the same bus with realtime computer data, etc.?

- Bus Topologies: Major factors that influence the selection of network path structure include transmission medium, data rate, number and type (active versus passive) of bus interface units, growth, bus

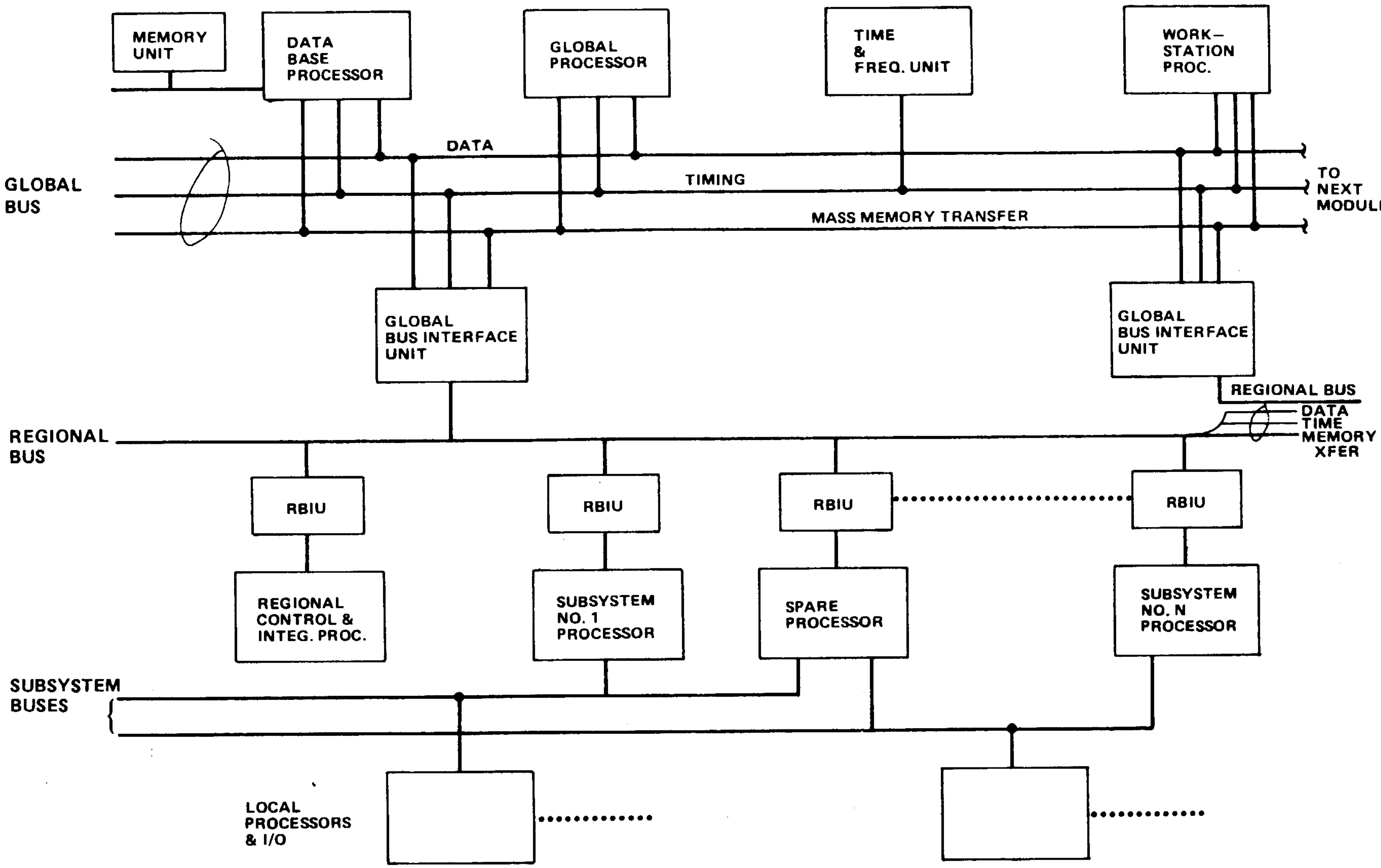


FIGURE 5: HIERARCHIAL DATA SYSTEM ARCHITECTURE

protocol, bus length, and system reliability. Options include rings, stars, serial buses, graphs, and various variations and combination of these.

- Data Bus Medium: One must trade the advantages and technology and cost uncertainties of optical fibers versus wire.

- On-Board Data Base: The data base management system becomes larger and more complex with increasing autonomy. Issues include management, degree of distribution, on-orbit versus Earth based storage, replication, and type of storage device.

Two studies are currently being performed by TRW and McDonnel Douglas Aircraft Corporation to recommend a data system architecture for the Space Station. Since the Space Station will utilize distributed data systems with a high degree of autonomy to support manned space missions for a long duration, the results of these studies should have application to the manned Mars mission.

4.0 CDMS TECHNOLOGY

No new or advanced communication and data management technologies have been identified as enabling for a manned Mars mission. However, the use of a number of advanced technologies to improve productivity and safety and to reduce mission cost is highly desirable. The following is a list of advanced technologies that are applicable or potentially applicable to a manned Mars mission.

- o Fault and Damage Tolerant Distributed Data Systems
  - Processors
  - On-Board Communication Network (e.g., Fiber Optic Networks)
  - Large Mass Memories (e.g., Bubble, Optical Memories)
  - Software
- o On-Board Data Reduction and Processing Techniques
  - Artificial Intelligence/Expert Systems
  - Video and Science Data Compression
- o Man-Machine Interfaces
  - Solid State Multifunction Color Displays
  - Voice Recognition and Natural Language Understanding

- o Laser Communications
  - High Power Solid State Lasers
  - Pointing Systems
- o Reliable High Power RF Amplifiers
  - Approximately 200 Watts at X-Band
  - Approximately 100 Watts at K-Band

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N87-17797

MANNED MARS MISSION
ENVIRONMENTAL CONTROL & LIFE SUPPORT SUBSYSTEM
PRELIMINARY REPORT

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ABSTRACT

The purpose of this paper is not to present a specific design but to discuss the general philosophy regarding potential ECLSS requirements, concepts, issues and technology needs. The focus is on a manned Mars mission occurring in the late 1990's. Discussions on the Trans-Mars Vehicle, the MEM, and a Martian base facility are covered. The paper lists the functions, performance requirements, and design loads of a typical ECLSS, and briefly discusses issues and technology. It identifies several ECLSS concepts and options, and provides comparative weights and volumes for these. It contrasts several aspects of the Space Station ECLSS with Mars element ECLSSs.

INTRODUCTION

A proposed manned Mars mission to be flown around the turn of the century presents some unique requirements for the Environmental Control & Life Support Subsystem (ECLSS). The mission will require not only extremely long life verified equipment, but also several types of systems. The vehicle transporting the men to Mars will most likely utilize a different type ECLSS design from that required by the Mars Excursion Module (MEM) and that to be utilized by facilities on the Mars surface.

The mission to Mars from low-Earth-orbit (LEO) may take up to one year each way. Additionally, stay times on Mars may be from several days to months. The overall mission times from LEO to Mars and return could be on the order of three years. Since resupply will not be possible and weight/volume will be at a premium, redundancy, spares, and expendables will have to be minimized. Therefore, the equipment will have to be highly reliable for operating times in excess of three years. These types of requirements will dictate additional design/development efforts on the Space Station (S/S) ECLSS hardware in order to make it applicable to the Mars mission. The MEM will probably be used primarily for short durations and the ECLSS systems could range anywhere from open to closed loop. A permanent Mars base facility would most likely have ECLSS equipment different from that being utilized by the S/S due to the gravity environment (1/3 that on Earth) and the availability of consumables (oxygen, nitrogen, and water) in the Martian atmosphere. Additionally, the possibility of growing food in greenhouses on the Martian surface would have an effect on the total system architecture. The ECLSS design for a manned Mars mission will also be very dependent on factors other than those mentioned above. Factors such as crew size, propellants utilized, type of power system, vehicle architecture, extra-vehicular activity (EVA) requirements, safe haven philosophy, and artificial gravity requirements will also heavily influence the ECLSS design.

REQUIREMENTS

The primary system functions performed by the ECLSS are habitat temperature and humidity control, atmospheric pressure and composition control, water processing and management, waste management, and EVA support. The primary functions and major equipment associated with each is shown in Table 1.

The mission success, health, and safety of the crew will require as a minimum a fail operational/fail safe design criteria for the ECLSS hardware. The data presented in Table 2 provides a summary of currently utilized design criteria for the ECLSS equipment. The degraded level signifies a "fail operational" condition. The average design loads imposed by the crew are specified in Table 3. These loads will size the ECLSS hardware for all mission phases.

CONCEPTS

The ECLSS concepts for the Trans-Mars Vehicle will be strongly influenced by the designs chosen for the S/S. As mentioned earlier, additional requirements to that imposed by the S/S will be imposed on the equipment. The primary difference will be that of equipment lifetime, redundancy, and reliability. The potential options possible for consumables regeneration and carbon dioxide / carbon monoxide control are shown in Figure 1. The issue of resupply, which is an acceptable design approach for S/S but not for the Trans-Mars Vehicle, will drive the design to systems that are further closed than that required by S/S.

TABLE 1. EC/LSS SYSTEM FUNCTIONS

ECLSS FUNCTION	MAJOR EQUIPMENT
• ATMOSPHERE PRESSURE & COMPOSITION CONTROL – TOTAL & PARTIAL PRESSURE CONTROL & MONITORING – FIRE DETECTION & SUPPRESSION	PRESSURE REGULATION PORTABLE OXYGEN SYSTEM SMOKE/FIRE DETECTORS FIRE SUPPRESSION SYSTEM
• MODULE TEMPERATURE & HUMIDITY CONTROL	DEHUMIDIFICATION VENTILATION FANS AIR COOLING HEAT EXCHANGERS
• ATMOSPHERE REVITALIZATION – CO_2 CONTROL/REMOVAL/REDUCTION – O_2 & N_2 MAKEUP – TRACE GAS MONITORING & CONTROL	CARBON DIOXIDE REMOVAL AND COLLECTION CARBON DIOXIDE REDUCTION CONTAMINATION CONTROL ODOR CONTROL ATMOSPHERE MONITORING OXYGEN GENERATION EMERGENCY OXYGEN AND NITROGEN STORAGE
• WATER MANAGEMENT – WASTE WATER COLLECTION/PROCESSING – WATER QUALITY MONITORING – STORATE & DISTRIBUTION OF RECOVERED WATER	EVAPORATION PURIFICATION WATER QUALITY MONITORING WATER STORAGE
• WASTE MANAGEMENT – COLLECT/PROCESS URINE – COLLECT/STORE FECAL MATTER	WASTE COLLECTION AND STORAGE EMERGENCY WASTE COLLECTION HOT/COLD WATER SUPPLY
• EVA SUPPORT – PROVIDE EXPENDABLES/SUPPORT TO EMU & MMU – PROVIDE LIFE SUPPORT SERVICES TO AIRLOCK/HYPERBARIC FACILITY	SUITS AND BACKPACKS RECHARGE STATIONS AIR LOCK SUPPORT

TABLE 2. EC/LSS PERFORMANCE REQUIREMENTS

PARAMETER	UNITS	OPERATIONAL	DEGRADED (1)
CO_2 PARTIAL PRESS	MMHG	3.0 MAX	7.6 MAX
TEMPERATURE	DEG F	65–75	60–85
DEW POINT (2)	DEG F	40–60	35–70
POTABLE WATER	LB/MAN–DAY	6.8–8.1	6.8 (3)
HYGIENE WATER	LB/MAN–DAY	12 (3)	6 (3)
WASH WATER	LB/MAN–DAY	28 (3)	14 (3)
VENTILATION	FT/MIN	15–40	10–100
O_2 PARTIAL PRESSURE (4)	PSIA	2.7–3.2	2.4–3.8
TOTAL PRESSURE (5)	PSIA	10.2 OR 14.7	10.2 OR 14.7
DILUTE GAS	––––	N_2	N_2
TRACE CONTAMINANTS (8)	MG/M^3	TBD	TBD
MICRO–ORGANISMS	CFU/M^3 (6)	500 (7)	750 (7)

NOTES:

(1) DEGRADED LEVELS MEET "FAIL OPERATIONAL" CRITERIA.
(2) RELATIVE HUMIDITY SHALL BE WITHIN THE RANGE OF 25–75 PERCENT.
(3) MINIMUM.
(4) IN NO CASE SHALL THE O_2 PARTIAL PRESSURE BE BELOW 2.3 PSIA, OR THE O_2 CONCENTRATION EXCEED 25.9 PERCENT OF THE TOTAL PRESSURE AT 14.7 PSIA OR 30 PERCENT OF THE TOTAL PRESSURE AT 10.2.
(5) ALL SYSTEMS SHALL BE COMPATIBLE WITH BOTH 10.2 AND 14.7 PSIA TOTAL PRESSURE.
(6) CFU – COLONY FORMING UNITS.
(7) THESE VALUES REFLECT A LIMITED BASE. NO WIDELY SANCTIONED STANDARDS ARE AVAILABLE.
(8) BASED ON NHB 8060.1B, (J8400003).

TABLE 3. EC/LSS AVERAGE DESIGN LOADS

– METABOLIC O_2	1.84 LB/MAN DAY
– LEAKAGE AIR	5.00 LB/DAY TOTAL
– EVA O_2	1.22 LB/8 HR EVA
– EVA CO_2	1.48 LB/8 HR EVA
– METABOLIC CO_2	2.20 LB/MAN DAY
– DRINK H_2O	4.09 LB/MAN DAY
– FOOD PREPARATION H_2O	1.58 LB/MAN DAY
– METABOLIC H_2O PRODUCTION	0.76 LB/MAN DAY
– CLOTHS WASH H_2O	27.50 LB/MAN DAY
– HAND WASH H_2O	4.00 LB/MAN DAY
– SHOWER H_2O	8.00 LB/MAN DAY
– EVA H_2O	9.68 LB/8 HR EVA
– PERSPIRATION AND RESPIRATION H_2O	4.02 LB/MAN DAY
– URINAL FLUSH H_2O	1.09 LB/MAN DAY
– URINE H_2O	3.31 LB/MAN DAY
– FOOD SOLIDS	1.60 LB/MAN DAY
– FOOD H_2O	1.00 LB/MAN DAY
– FOOD PACKAGING	1.00 LB/MAN DAY
– URINE SOLIDS	0.13 LB/MAN DAY
– FECAL SOLIDS	0.07 LB/MAN DAY
– SWEAT SOLIDS	0.04 LB/MAN DAY
– EVA WASTEWATER	2.00 LB/8 HR EVA
– CHARCOAL REQUIRED	0.13 LB/MAN DAY
– METABOLIC SENSIBLE HEAT	7000 BTU/MAN DAY
– HYGIENE LATENT H_2O	0.96 LB/MAN DAY
– FOOD PREPARATION LATENT H_2O	0.06 LB/MAN DAY
– LAUNDRY LATENT H_2O	0.13 LB/MAN DAY
– WASH H_2O SOLIDS	0.44%
– SHOWER/HAND WASH H_2O SOLIDS	0.12%
– AIR LOCK GAS LOSS	1.33 LBS/USE
– TRASH	1.80 LB/MAN DAY
– TRASH VOLUME	0.10 FT^3/MAN DAY

TABLE 4. EC/LSS TECHNOLOGY REQUIREMENTS

- FECAL WASTE MANAGEMENT
- TRASH/FOOD MANAGEMENT
- SENSOR DEVELOPMENT
  - MASS GAUGING
  - TRACE GAS
  - AIR/WATER QUALITY
- WATER RECLAMATION/PROCESSING SYSTEMS
- REGENERATIVE CO_2 REMOVAL/REDUCTION SYSTEM
- MARS ATMOSPHERE PROCESSING SYSTEM FOR OXYGEN, NITROGEN & WATER

Figures 2 and 3 provide weight and volume penalties for consumables requirements as a function of system closure.

The MEM ECLSS could consist of either an open or closed loop system, depending on the design approach that will be implemented. If the MEM will be used as either a working volume or safe haven during trans-Mars travel, the MEM ECLSS would probably be a closed system similar to the Trans-Mars Vehicle system. Additionally, if contamination of the Mars atmosphere by overboard venting is not permitted, a closed system will be required. If, however, it is decided that the MEM is to be used only for short durations (less than approximately ten days), the ECLSS could be envisioned as an open system. An open system has the advantage of design simplicity and low cost. Figure 4 shows the weight trend of open versus closed systems.

The Mars base facility is envisioned to be a self-contained ecological system utilizing the Martian atmosphere as a source of consumables. The primary consumables available from the atmosphere are oxygen, nitrogen and water. Oxygen is obtainable through the reduction of carbon dioxide which constitutes approximately 95% of the Martian atmosphere. Nitrogen, 2.5% of the atmosphere, can be directly extracted from the atmosphere. Water is found in very small quantities in the atmosphere (approximately 0.03%). However, additional sources are potentially available in the Martian permafrost and polar caps. The supply of food is envisioned to be brought from Earth in the early stages of a Mars base facility. Future development could lead to a greenhouse-type facility that would permit growth of the food supply. The greenhouse could also be a potential source of oxygen for the habitat.

ISSUES

The major issues pertaining to the ECLSS are associated with the degree of closure of the subsystems and the reliable operation of the hardware for many years. Due to the length of the mission and the weight sensitivity of a Mars vehicle, the amount of consumables will have to be minimized. Also, the weight and volume allocation for spares and/or redundant systems will have to be critically evaluated to assure a viable vehicle design. The whole issue of how much redundancy one builds into the basic design versus onboard spares and repair philosophy needs addressing in future studies.

FIGURE 1. EC/LSS SYSTEM OPTIONS

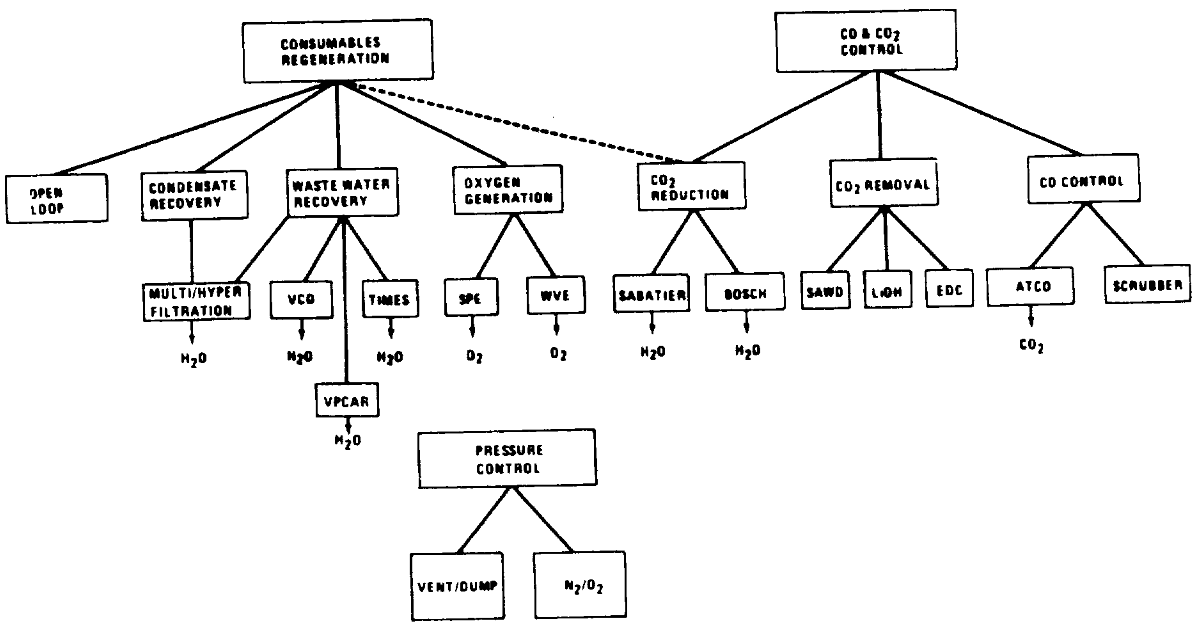


FIGURE 2. WEIGHT VS. EC/LSS SYSTEM CLOSURE OPTION

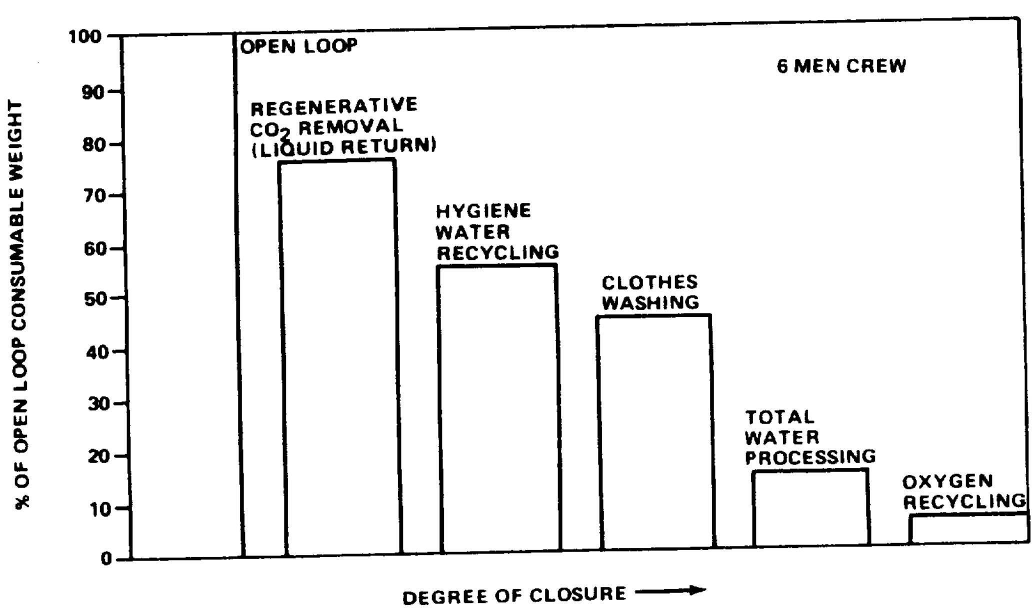


FIGURE 3. VOLUME VS. EC/LSS SYSTEM CLOSURE OPTION

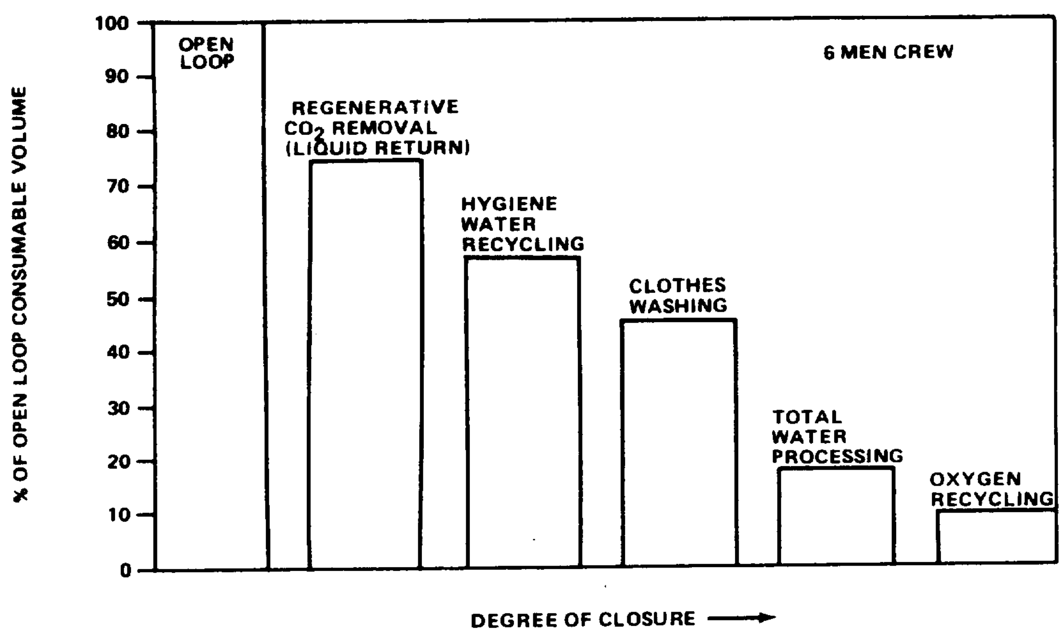
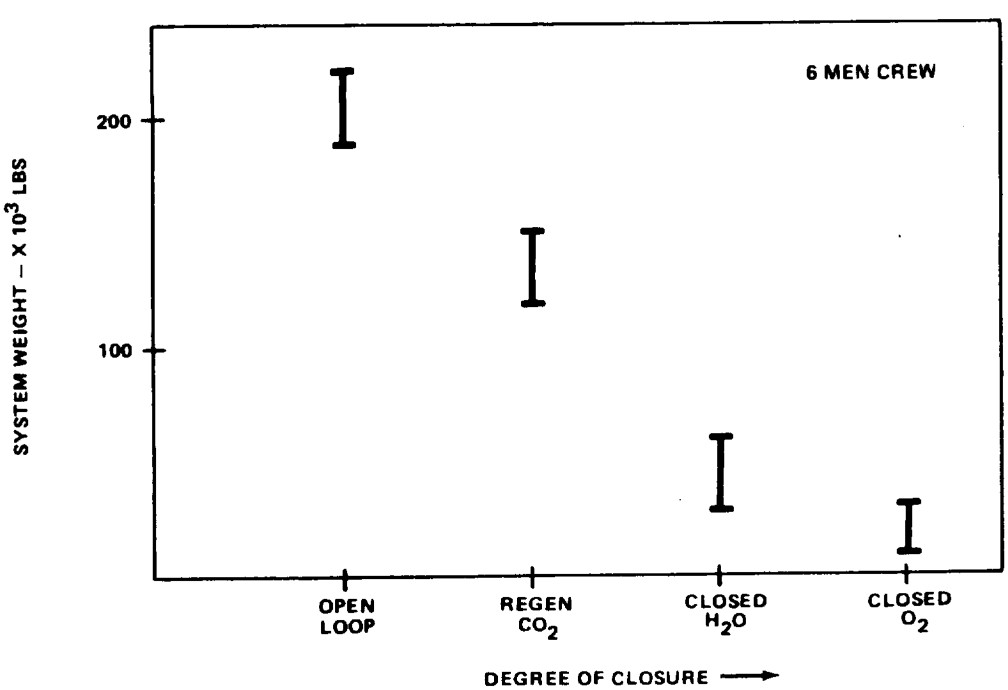


FIGURE 4. TOTAL WEIGHT VS EC/LSS SYSTEM CLOSURE



The issue of system closure for the ECLSS will become a major design driver for the overall vehicle. The Trans-Mars Vehicle will be more heavily affected than Mars surface equipment since some of the consumables will be available on the Martian surface. Ideally, it would be desirable to close the system as much as possible. However, system inefficiences, structural leakage of gases, consumed gases/liquids, and emergency supply of consumables will always dictate that a storage facility for consumables be required. The actual size of such a facility will require further study. Other factors influencing the design will be the EVA requirements and the design of non-venting/closed EVA systems. The largest part, both by weight and volume, of such a facility will be driven by the water requirements.

One of the key parameters concerning the redundancy issue is the crew safe haven philosophy. Also, the overall architecture of the vehicle will have direct bearing on the type of ECLSS subsystems that will eventually be implemented. For example, the use of several separate modules to form the habitat area would in essence allow nonredundant loops in each module with repair capability of the failed loop. The crew would temporarily be restricted to the active module(s) while repair was being performed. The use of a singular module, although structurally more weight efficient, will place very stringent safety requirements on the design. The resultant repair/maintenance philosophy would be different from that of a multiple module design.

One other issue that will effect the design of the ECLSS is the potential requirement of artificial "g" during the flight from LEO to Mars and return. The level of gravity proposed will dictate the type of system design that may be implemented. If the gravity is comparable to that expected on the Martian surface (1/3 that on Earth), the system used might be similar to that which will be used on the Martian surface. The primary effect a higher gravity environment has on the ECLSS is in the area of fluid acquisition/feed systems and vapor/liquid separation devices.

TECHNOLOGY

The major areas of technology in the ECLSS that will have to be advanced for a manned Mars mission are: regenerative systems, consumable storage/generation, and waste/trash management. Table 4 provides a list of some of the key ECLSS technology items that will need to be addressed prior to commitment of a design for a manned Mars mission. Some of the issues will be worked by the Space Station technology program. However, the aspects of operational life requirements of approximately three years without resupply/refurbishment will place additional requirements on the applicable Space Station hardware. Also, the potential of artificial "g" during transit flight to Mars and the gravity on the Martian surface will provide the opportunity to use systems that take advantage of the gravitational force. The areas that will require major emphasis are waste management and systems for extracting the consumables from the Martian atmosphere.

SUMMARY

The major issues that need further emphasis are the degree of closure that can be effected by the ECLSS and the operational reliability of the hardware. The minimization of redundancy and spares will be an important factor due to the vehicle weight sensitivity for a Mars mission.

A manned Mars mission will be a very challenging undertaking for the design/development of the ECLSS. Current and advanced technology will be required to meet the mission objectives. A lot of the development work being conducted and planned by S/S will be directly applicable to this mission. The Trans-Mars Vehicle's ECLSS design could be very similar in design to that of the S/S. The MEM and Mars base facility ECLSS designs remain as open issues because of the many undefined variables mentioned earlier.

N87-17798

PHYSIOLOGICAL AND TECHNOLOGICAL CONSIDERATIONS FOR MARS MISSION EXTRAVEHICULAR ACTIVITY

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ABSTRACT

The nature of the suit is a function of the needs of human physiology, the ambient environment outside the suit and the type of activity to be accomplished while in the suit. In the following paragraphs the physiologic requirements that must be provided for in the martian EVA suit will be reviewed. We will elaborate on how the martian environment may influence the EVA suit, EVA capabilities, and will compare the martian environment with the lunar environment and point out differences that may influence EVA design. The type, nature, and duration of activities to be done in transit to Mars on the Mars surface will be evaluated and the impact of these activities on the requirements for EVA systems will be discussed. Furthermore, the interaction between martian surface transportation systems and EVA systems will be covered. Finally, options other than EVA will be considered such as robotics, non-anthropometric suits, and vehicles with anthropometric extremities or robotic end effectors.

DISCUSSION

Extravehicular activity (EVA) refers to excursions outside the spacecraft cabin environment in a suit that provides its own protective environment. The experience of Skylab has demonstrated the value and versatility of micro-G EVA in terms of planned resupply and maintenance of spacecraft components as well as in repair of disabled spacecraft. The Apollo experience on the lunar surface has shown that in a self-contained space suit, man can move about freely on the lunar surface. He can perform useful work, deploy equipment, drill soil samples, make measurements, and select and collect geological samples. The crewman can also explore on foot and using motorized transportation. The Shuttle program has provided even greater experience and definition of what can be accomplished in micro-g EVA with improved suits and support systems. EVA is planned to be a very important component of Space Station. To meet the needs of Space Station, EVA suits will have to be durable and easy to repair; they will have to operate for long periods of time; and all cleaning, refurbishment, and repair will have to be done on board the vehicle. Also Space Station EVA will have to be done with a minimum utilization of expendables.

In meeting the objectives of the Mars missions all the EVA experience of earlier missions and all the evolution of EVA equipment will be required. During the transit between Earth and Mars, EVA outside the vehicle may be required for vehicle repair and suited IVA will surely be to maintain training of crewmen for martian surface EVA. On the martian surface crewmen will be involved in exploring, mapping, surveying and detailing the martian surface. Scientific equipment will be set up and measurements and obervations will be made by crewmen. Finally, crewmen wil be involved in fabricating and extending a martian habitation base.

Certainly a desirable way to perform these activities would be to walk about outside the spacecraft and on the martian surface in shirtsleeves, to pick up samples with bare hands, to use these hands to work with scientific equipment and make fine adjustments, and to ride in open vehicles and to mount and dismount at will. This simple approach is not possible because the martian environment, like the free space environment and the lunar environment, does not provide the physiologic requirements of the crewmen. To modify this environment man will need to be enclosed in a controlled habitable environment. An extravehicular activity suit will provide a minimum enclosure and interdiction between the man and the external environment to most closely approach shirtsleeve activity capability.

The EVA suit will have to provide adequate control of the following environmental factors: pressure, oxygen pressure, temperature, humidity, and radiation. At the same time, the suit will have to accommodate other physiologic needs. The suit will have to remove CO_2 produced by the crewmen. Food will have to be provided in the suit for the crewman if the duration in the suit is long enough to require it. There must be provision for waste management certainly of urine and possibly of feces again if the duration of suit wear is such that this would be required. EVA suits have sometimes been referred to as pressure suits; and although an EVA suit must control much more than pressure, pressure is one of the most critical environmental factors. A minimal atmospheric pressure (about 0.9 psi) is required to keep body fluids in the liquid state. For all practical purposes, however, acceptable pressures for an EVA suit are determined by the required partial pressure of oxygen and by an acceptable change in pressure from the ship cabin pressure to the suit pressure without causing altitude decompression sickness.

The ambient pressure on Mars is about 7 torr(1), a level well below that required to sustain human life. So pressure control will be required during EVA. If 100% oxygen is used in the suit, minimum operational suit pressure would be 3.7 psi. This pressure would provide a normal O_2 pressure in the alveoli of the lungs for transmission to the body for use in metabolism(2). A 100% O_2 environment was used in the Apollo Program for both the cabin at 5.0 ps and the pressure suit at 3.7 psi. The Apollo Program included exposures up to 2 weeks in length. For longer exposures some diluent gas is needed to avoid atelectasis in the lung and other potential problems with 100% O_2(3). In the Skylab Program, a 5.0 psi cabin pressure was used with 70% O_2 and 30% N_2 as the diluent gas. There was no indication of physiological problems with this atmosphere for periods of up to 84 days(4).

If different pressures are used in the cabin and in the pressure suit, care must be taken to avoid decompression sickness. Decompression sickness occurs when the pressure of dissolved gases in the tissues exceeds the ambient pressure. Under these conditions, bubbles may form in tissues and be carried by the blood-stream throughout the body. Decompression sickness is not normally a problem when the pressure of the diluent gas in the atmosphere does not exceed the final decompression pressure by more than a ratio of 1.25 to 1. If this ratio is to be exceeded, the crewmen must breathe O_2 prior to decompression to reduce the N_2 pressure in the body. It can be seen, therefore, that the pressure in the pressure suit depends on the cabin pressure as well as the minimum O_2 pressure required in the suit.

Options for different combinations of cabin and suit pressure are now being considered for Space Station (table 1). The main trade consideration for Space Station are the reduction in flammability associated

TABLE 1

CABIN AND SUIT PRESSURE AT THE THRESHOLD OF BUBBLE FORMATION

Cabin Presssure	Nominal % O_2 In cabin	Suit Pressure	Constraints
14.7 psi	21	9.5 psi	None
10.2 psi	28	6.0 psi	Equilibration at 10.2 psi for 72 hours prior to EVA or 1-hour pre-breathe prior to 10.2 psi plus 24 hours at 10.2 psi.
11.0 psi	25	6.7 psi	Equilibration at 11.0 psi for 72 hours prior to EVA or 1-hour pre-breathe prior to 11.0 psi plus 24 hours at 11.0 psi.
12.75	24	8.00 psi	Equilibration at 12.75 psi for 24 hours prior to EVA.
10.0 psi	50	4.3 psi	Equilibration at 10.0 psi for 72 hours prior to EVA or 1-hour pre-breathe prior to 10.0 psi plus 24 hours at 10.0 psi.

with higher levels of diluent gas versus the decreased pressure suit mobility associated with higher suit pressures. For the Mars mission, the tradeoffs may not be the same. High mobility pressure suits are now being worked on, and if they are operationally developed for Space Station, suit mobility may no longer be a tradeoff consideration. On the other hand, increased loss of consumables at higher cabin and suit pressures may become an overriding consideration.

The EVA suit must allow the crewman to maintain thermal balance. That is, a balance between heat production and heat loss. The man's internal heat production can vary from rest to work over a range of 10 to 1 or more on occasion and commonly varies over a range of 4 or 5 to 1. At the same time the outside of the suit may be exposed to a wide range of radiant thermal environment. This makes thermal balance difficult and requires a variable controlled rate of heat loss. The successful approach in EVA systems to date has been to isolate the suit from the external environment and to match heat loss to heat production using a liquid cooled garment bringing body heat to a heat exchanger cooled by sublimating H_2O to the space vacuum. For Space Station other approaches are being looked at to avoid the loss of the water involved in the sublimation and to avoid contamination of the near station space environment with water vapor(5). Typically, options now being looked at rely on change of state of water from solid to liquid as a heat sink. Because of lower quantity of heat involved in the change of state from solid to liquid compared to the heat involved in the change of state from liquid to gas, these systems will tend to be bulkier, heavier, and support shorter EVA's than systems involving sublimation. There may be other alternatives on the martian surface. The temperature environment on the martian surface will depend on the landing site, the martian season and the time of day; however, the Mars environment relative to the Earth environment will typically be cold(1). It may be possible to devise a controlled variable heat loss system from the suit that would use the martian environment as the heat sink. Such a system might involve radiators mounted on the surface of the suit with control of heat loss inplemented by flow of a coolant from the liquid cooled garment. A system of this type would be most effective in the really cold martian environments.

A second alternative for thermal management would be to utilize martian resources as a substitute for the water currently used. Although water will probably not be easily accessible on the martian surface for use as a chage of state heat sink, solid CO_2 is available at the poles and may be absorbed in surface soils in extra-polar regions(1). Solid CO_2 could be used as a heat sink in an EVA system. The combined heat of fusion and of vaporization of CO_2 is about 130 cal/gram compared to the 80 cal/gram heat of fusion of ice. A more reliable source of CO_2 would be the atmosphere. The martian atmosphere is 95% CO_2 to about 7 torr(1), so it would be conceivable to compress, cool, and solidify the martian atmosphere and use the CO_2 as an EVA heat sink.

While working in the EVA suit, the crewmen will generate CO_2 that must be removed from the suit atmosphere or maintained at acceptable levels (about 7 torr)(2). In all of our portable life support systems to date, we have used Lithium Hydroxide (LiOH) to absorb the CO_2 and react with it to form various Lithium carbonates. This reaction is not easily reversible and expended LiOH cartridges are discarded. In Skylab, molecular sieve ion resins were used to absorb CO_2. CO_2 could later be removed from the beds with the application of low pressure(6). For Space Station, recoverable systems are being planned in which not only can the CO_2 absorbent be recovered and reused but the CO_2 itself can be recovered and converted back to O_2(7). Systems of this type will be essential for Mars missions where conservation of consumables will be critical. The CO_2 systems will consist of beds or liquid containers in the EVA back pack that will absorb CO_2. These beds or liquids would be regenerated in the spacecraft, the Mars lander vehicle or the Mars base facility to convert the CO_2 back to O_2. The regenerable CO_2 absorbers tend to be larger than current LiOH system so this will impact EVA capability.

The EVA system will also have to provide protection from environmental radiation. Mars does not have a strong magnetic field(7) and therefore, the martian surface is not protected against space radiation as is the Earth. Galactic radiation will be about one-half of that in open space due to the shielding provided by the planet itself. With pressure suits similar to those that will be developed for Space Station, which will probably provide more radiation protection than our Shuttle suits, galactic radiation on the Mars surface will not limit EVA for martian stays of several months and frequent EVA. Galactic radiation during EVA may be limiting for martian stays of a year or more involving EVA or for Mars colonization. However, galactic radiation is not the only radiation threat. Crewmen on the Mars surface would also be at risk from episodic radiation from solar flares. These potential high radiation flux episodes would require retreat to a radiation "Safe Haven" and may be an important consideration in planning mobile explorations across the Mars surface. The problem of radiation is treated in depth in a separate chapter.

In addition to the environmental considerations already mentioned, the EVA system will have to provide food, water, and waste management to the crewmen. These requirements become more critical and difficult as EVA duration is extended (table 2). Water requirements are 8 oz/hour for EVA durations in excess of 3 hours. Food requirements are: a snack of about 200 kcal for EVAs of less than 6 hours and 750 kcal/8-hour duration for longer exposures. Some urine collection capability should be provided for even short EVAs and a 1000 cc capability should be provided for 8 hour EVAs. Some level of containment of an uncontrollable bout of diarrhea or any other unscheduled defecation must be provided for EVA's up to 8 hours. Stays in the suit in excess of hours would require more serious containment capability. The longest EVAs to date have been about 7 hours in length, and such EVA'S have been done in each of the Apollo, Skylab and Shuttle Programs. The Apollo Program included a contingency capability to return from the Moon over an up to 115 hour period in a pressurized suit(8). To achieve this capability the suit helmet had a feeding port that could be utilized with a 3.7 psi differential pressure to take food and liquids into the suit and into the mouth. The suit also had a urine transfer system to transfer urine out of the suit. The gaseous environment in this situation was supplied by umbilical so the CO_2 system was part of the cabin ECS. The suit system also included a fecal containment system that could be described as a large diaper. This system was designed only as a get-back system aimed at survival. Although there is little doubt that the system would have resulted in crew survival if it had been required the use of an anthropometric form fitting EVA suit for EVA durations in excess of 8 hours is

TABLE 2

CONSIDERATIONS FOR MARS EVA LIFE SUPPORT

<u>THERMAL EQULIBRIUM</u>:

Thermal balance in crewmen at range of EVA metabolic rates.

Need: Active temperature, control system & distribution system.

Design Consideration:

Variable insulation or sublimation of H_2O or sublimation of CO_2. Liquid cooled garment or other distribution system.

Supply of O_2 for metabolism, control of CO_2 produced.

Need: 0.2 lb/hr O_2 @ 1000 BTU/hr

0.24 lb/hr CO_2 @ 100 BTU/hr

Design Consideration:

1) O_2 supply
2) Regenerable CO_2 absorption system.

<u>WATER MANAGEMENT</u>:

Avoid dehydration - Allow urination

Need: Provide for collection - 1000 ML

Provide in-suit water at 8 oz/hr after 3 hours

Design Consideration:

1) In-suit water supply and drinking system
2) In-suit urine bag

<u>NUTRITION</u>:

Need: 750 cal/8 hours

Design Consideration:

1) In-suit food system, or
2) Limited duration EVA

<u>MONITORING</u>:

Provide measure of stress and consumables usage to crew and or others.

Need: Physiological monitoring as needed

Design Consideration:

1) O_2 usage
2) CO_2 level
3) Heart rate

TABLE 2 (CONTINUED)

<u>MAINTENANCE</u>:

Suit must be kept operational.

Need: 1) Durability

2) Simplicity

3) Cleanability

4) Repairability

Design Consideration:

1) Largely hard components - high cycle life

2) Anticipated repair time minimal

3) Smooth surfaces,easy cleaning procedures & systems.

4) Component replacability

5) Repair facility

<u>MECHANICAL MOBILITY</u>:

Man must be able to perform useful work in suit without injury or abrasion.

Need: Good low-effort Joint systems - Comfortable fit

Design Considerations:

Improvement over current systems

<u>GRAVITY EFFECTS</u>:

Need: Center of gravity of man in suit must be compatable with walking.

Design Consideration:

Suit design must consider gravity.

not recommended because of limitations related to personal hygiene, waste management, and general crew health and well-being.

EVA has shown the real potential for vehicle repair in space, so designers of the Mars mission may want to assure this capability during the period of travel to and from Mars. However, EVA is not free. The design of equipment that may have to be repaired, the translation aids on the surface of the vehicle, the airlocks and the EVA support systems must be carefully planned to accommodate EVA. One important factor in the decision as to whether or not to provide a trans-martian EVA system may be whether or not artificial gravity is provided in the crew module during this mission phase. If artificial gravity is generated by rotating all or part of the vehicle, then EVA may be much more difficult because it would then be possible to "fall off" the vehicle and certainly moving around and about the vehicle would be much more difficult.

During the long duration of the trans-martian mission phase, IVA will be desirable to maintain training for Mars surface EVA. This training period would be particularly useful if the spacecraft is maintained at Mars' normal g level.

For surface EVA in the area of the landing site, the gravity force field on the martian surface is a consideration that will impact the nature of Mars surface EVA and the systems that support it. Prior to Apollo 11, there was considerable speculation on how well man could walk and move about in the 1/6-g lunar environment. The best simulations of 1/6-g indicated it would be easier to work in 1/6-g(9) and the lunar surface EVA proved the point(10). The Apollo EMU weighed about 200 pounds and had to be supported during the 1-g training exercises to allow crewmen to move. The martian gravity force will be less than .4 times that on the Earth(7). As in the Apollo EVA system, careful attention will have to be paid to the center of gravity of the man/suit complex. Because of the relatively higher weight of the regenerative life support systems and the greater apparent weight of the martian backpack relative to lunar backpacks, it is likely that self-contained EVA systems will be limied to 2 to 4 hours of support capability.

An alternative to self-contained EVA systems is an umbilical system. In such a system, some of the life support components could be mounted in the martian base or on a mobile vehicle or platform. The crewmen would be tied to the support system with umbilicals. Such a system can extend EVA time but umbilical tending is a constant concern with this type of arrangement and the length of the umbilicals is limited. Umbilical systems would seem to be particularly attractive to allow some EVA in a limited perimeter around a mobile exploration vehicle.

EVA will be a component part of any exploration plan for the martian surface. For short distances some exploration will be done on foot. But to cover greater areas, a motorized vehicle will be needed. Martian rovers are discussed in other papers; however the following paragraphs discuss some of the options of how a vehicle might interact with an EVA system.

One system might be similar to the lunar rover system (figure 1). In this system, the transportation system was completely separate from the EVA systems. The crewmen rode on the vehicle with their own self-contained EVA system. Such a system provides maximum freedom for the crewmen and is limited by the duration of life support provided by the backpacks on the crewmen. Because of considerations mentioned in earlier paragraphs, the duration of life support systems that could be carried on a regenerative backpack system on the martian surface would be relatively short (2 to 4 hours). An alternative would be a similar system with the pressure-suited crewmen tied to the transportation system with umbilicals (figure 2). Such a system would be limited in time and range to the duration that crewmen could stay in the pressure suit (8 to 10 hours). Longer range exploration vehicles would have to provide a pressurized volume for crewmen. In its simplest form, such a vehicle might be a motorized-non-anthropometric pressure suit with arms and hands extending from the pressurized volume (figure 3). With the capability to withdraw from the arms, the crewmen could tend to food, drink, and waste management in a larger volume. Such a system would be range limited by power and consumables. Finally, given sufficent size and volume, a transportation system with a pressurized volume could in addition carry a pressure suit to be used as needed. This would provide the most versatile and far ranging, but not the most complex system of all.

A rover vehicle that could have flexible pressurized arms might instead have mechanical end effectors or robot arms. Robotics is a fast developing field, and it is likely that robotic systems will be developed

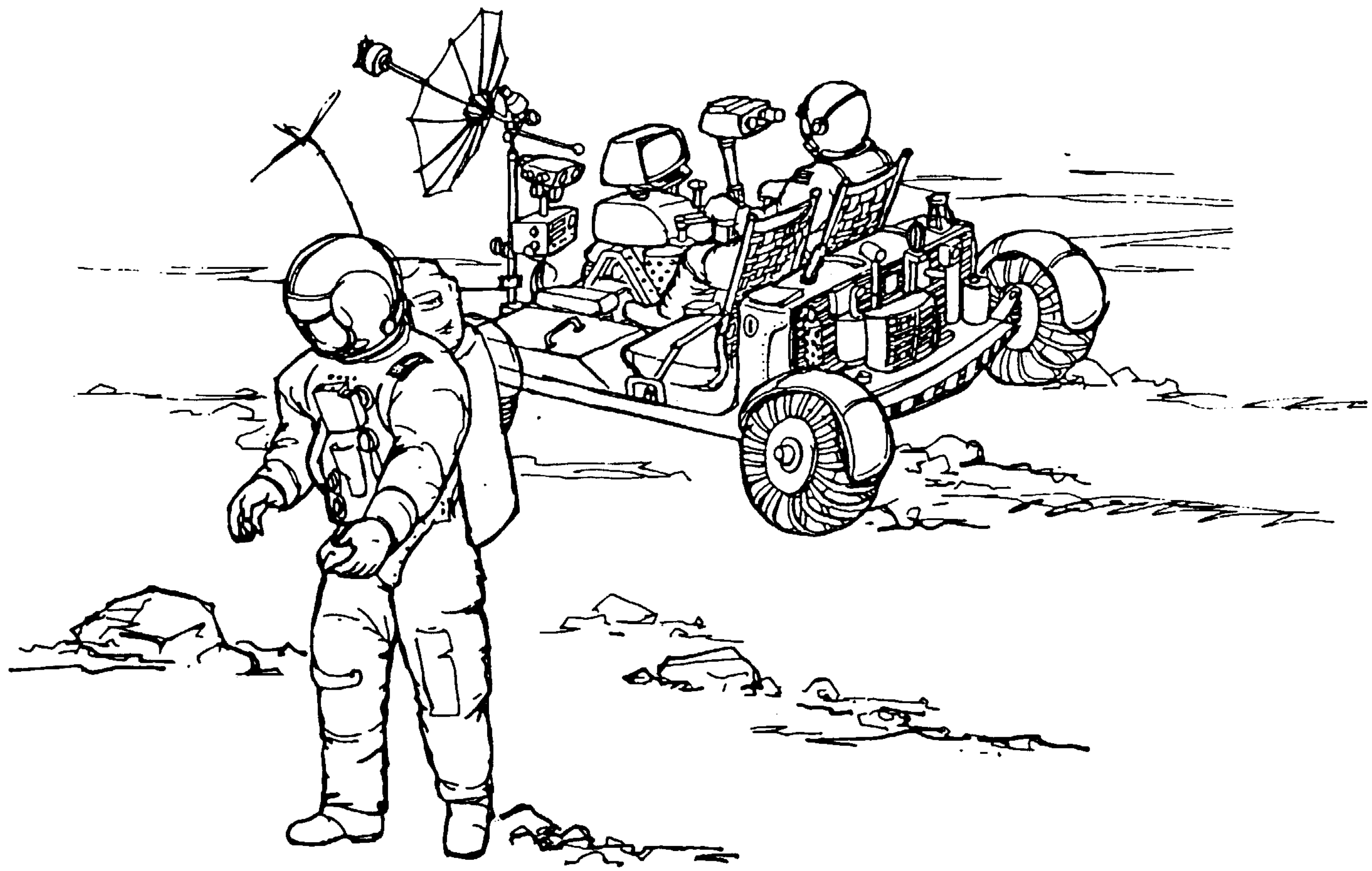

FIGURE 1
Transportation System with Suitmounted ECS Systems

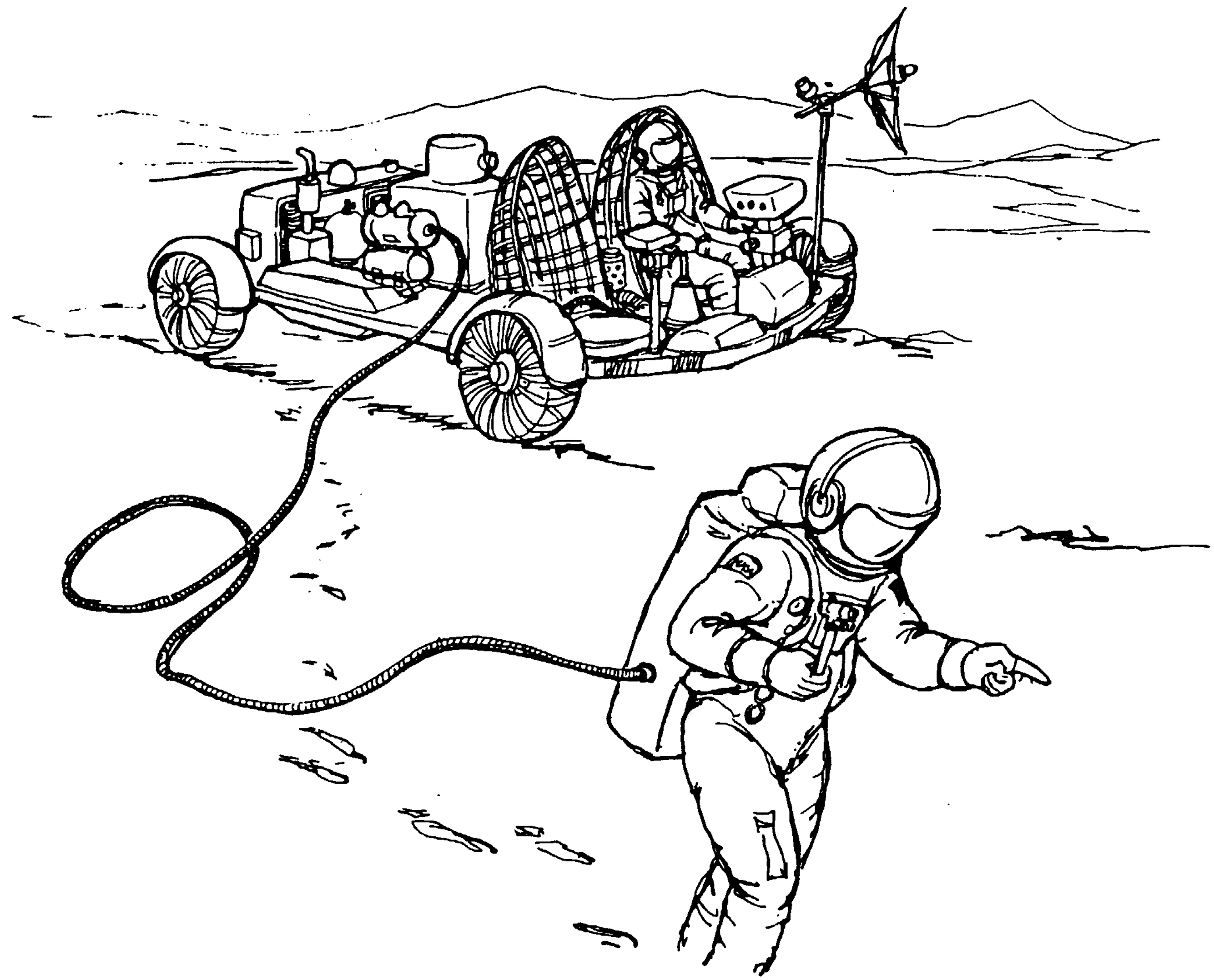

FIGURE 2

Transportation System with Umbilical ECS System

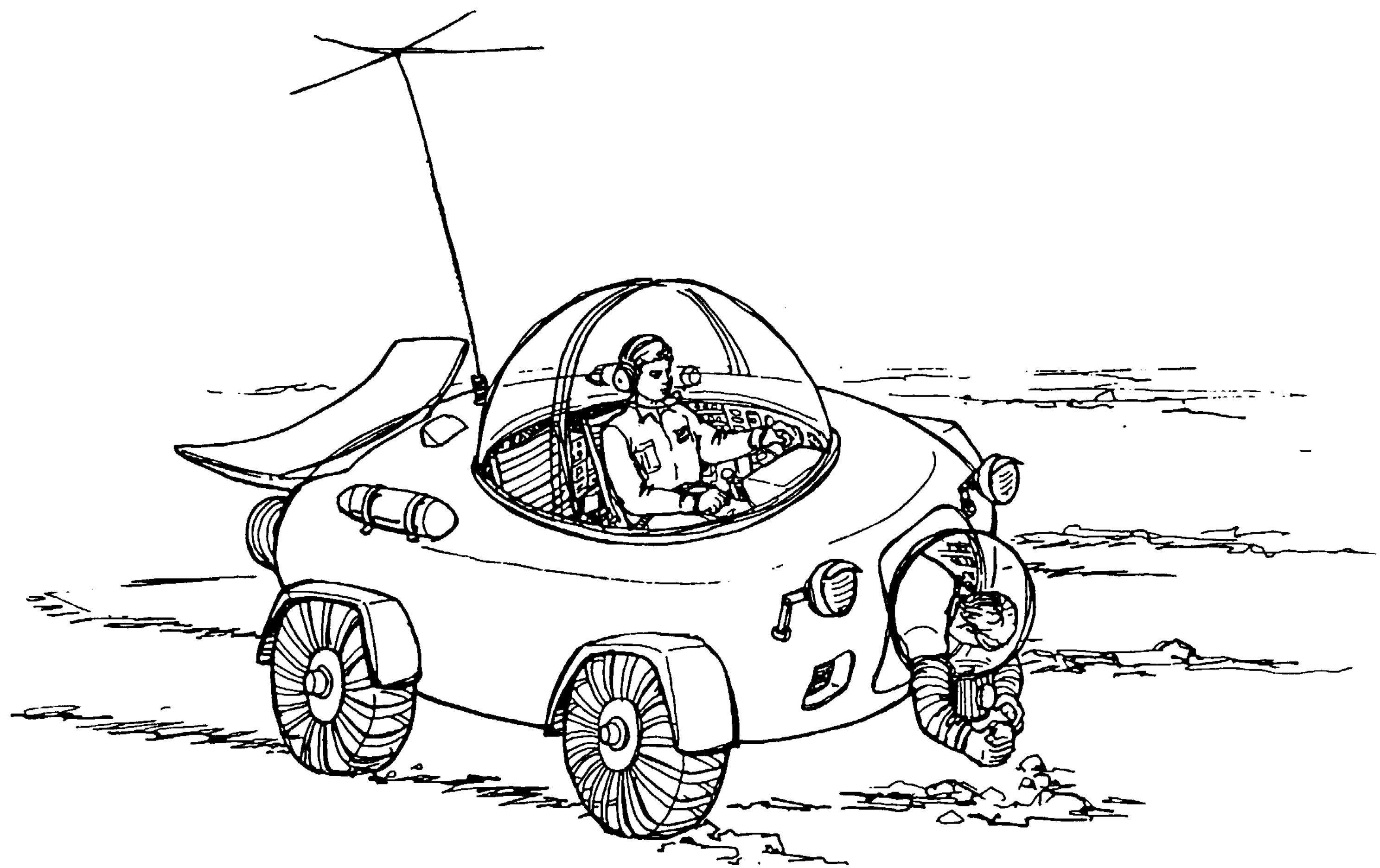

FIGURE 3
Transportation with Pressurized Volume and Anthropomotric Suit Extension

to aid in Space Station construction. It is very likely that some tasks that might be done with an EVA crewman could be done with a robotic systems or with hybrid systems using mechanical end effectors to aid the EVA crewmen. Robotic systems in EVA will probably evolve in a process of using such systems to aid EVA crewmen and considerable use of robotics will be made in developing the Space Station. Depending on the direction and scope of this evolution prior to the Mars mission, robotics will have a greater or lesser impact on EVA on Mars, but we can expect that robotics will supplement rather than replace manned EVA in a pressure suit.

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S E C T I O N   V I I I

POLITICAL AND ECONOMIC ISSUES

N87-17799

BUDGET AVAILABILITY

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ABSTRACT

The report describes a forecast of the total NASA budget required to achieve a manned mission to Mars at around the end of this century. A methodology is presented for projecting the major components of the NASA budget, including the NASA base, Space Flight, Space Station, Shuttle Derived Launch Vehicle, and the Manned Mars Mission. The NASA base, including administrative expenses, construction of facilities and research and development other than manned space flight, is assumed to level off at the present (1985) level and remain constant at approximately $3.5 billion (constant fiscal year 1985 dollars). The budget for Space Flight, which consists of Shuttle research and development, operations, and tracking and data acquisition costs, is projected to decrease from approximately $4 billion in 1985 to just under $2.5 billion by 1989 and then level off. Planning profiles for three new major programs are constructed: (1) a permanently manned Space Station; (2) a Shuttle Derived Vehicle; and (3) a Manned Mars Mission. It is concluded that all of the new programs can be conducted by the year 2002 with a 3 percent real growth rate in the total NASA budget.

INTRODUCTION

This report contains an estimate of the total NASA budget required to support a manned Mars mission at around the turn of the century. The purpose of this report is to document the methodology, groundrules and assumptions used in preparing the forecast. The following sections will describe the scope of the analysis and the method used to project each major element of the budget.

SCOPE

The budget projection includes all NASA outlays for direct program expenses during the period from fiscal year 1987 thru fiscal year 2010. Specifically excluded are any expenses which will eventually by paid for by someone other than NASA, such as the launch of commercial satellites.

GROUNDRULES

All costs presented in this report are in constant fiscal year 1985 dollars. Historical data thru 1984 are actual direct program costs, inflated to 1985 dollars using the NASA R&D index for advanced programs. Budget data for 1985 and 1986 are estimates from the Budget of the United States Government. Other assumptions for major categories of the budget are listed in Table 1.

RESULTS

Forecast results for major categories and subcategories of the present budget are shown in Figure 1 and Figure 2. Planning profiles for the Space Station, the Shuttle Derived Vehicle, and the Manned Mars Mission are depicted in Figures 3-5. A list of abbreviations to assist in deciphering the graphs is included in Table 2.

CONCLUSION

A summation of all the major budget categories is shown in Figure 6. Also shown is total NASA budget if a 3 percent real growth rate beginning in 1986 is assumed. The illustration demonstrates that the Manned Mars Mission as well as a Space Station and a Shuttle Derived Vehicle could be developed in the time period shown within the 3 percent growth line. The 3 percent growth rate is based on the assumption that NASA budget remains a constant percent of gross national product (GNP) and that GNP continues to grow at an average rate of approximately 3-4 percent per year as it has for more than 100 years.

TABLE 1

FORECAST METHODOLOGY

NASA Base:

- FY85-86 estimates from the President's FY1986 budget
- FY87-2010 3 year moving average

Manned Space Flight:

TDRSS:

- FY85-86 estimates from the President's FY1986 budget
- FY87-2010 3 year moving average

Other:

- FY85-86 estimates from the President's FY1986 budget
- FY87-90 OSF 1986 congressional budget
- FY87-2010 3 year moving average

Space Station:

- $8 billion IOC cost
- $300 million annual operations cost
- 60/40 spread for 5 year development program

Shuttle Derived Vehicle (SDV):

- $5 billion development cost (2 flight units)
- 60/40 spread for 6 year development program
- $100 million per flight
- 8 flights per year

Mars:

- $27 billion IOC cost estimate
- $1 billion for LEO assembly facility
- operations cost included in IOC cost estimate
- 50/50 spread for 12 year development program

Total NASA budget:

- NASA budget equals fixed percent of GNP
- GNP average annual growth at 3% in constant year dollars

LIST OF ABBREVIATIONS

APPL	Science and Applications
CF	Construction of Facilities
DDT&E	Design, Development, Test and Evaluation
ETB	Engineering and Technical Base
FY	Fiscal Year
HDWR	Hardware
IOC	Initial Operating Capability
MISC	Miscellaneous
OPS	Operations
OSF	Office of Space Flight
PROD	Production
REQTS	Requirements
RPM	Research and Program Management
R&D	Research and Development
R&T	Research and Technology
SCI	Space Science
SDV	Shuttle Derived Vehicle
SFCDC	Space Flight Control & Data Communications
SS	Space Station
STS	Space Transportation System
TDRSS	Tracking and Data Relay Satellite System

FIGURE 1

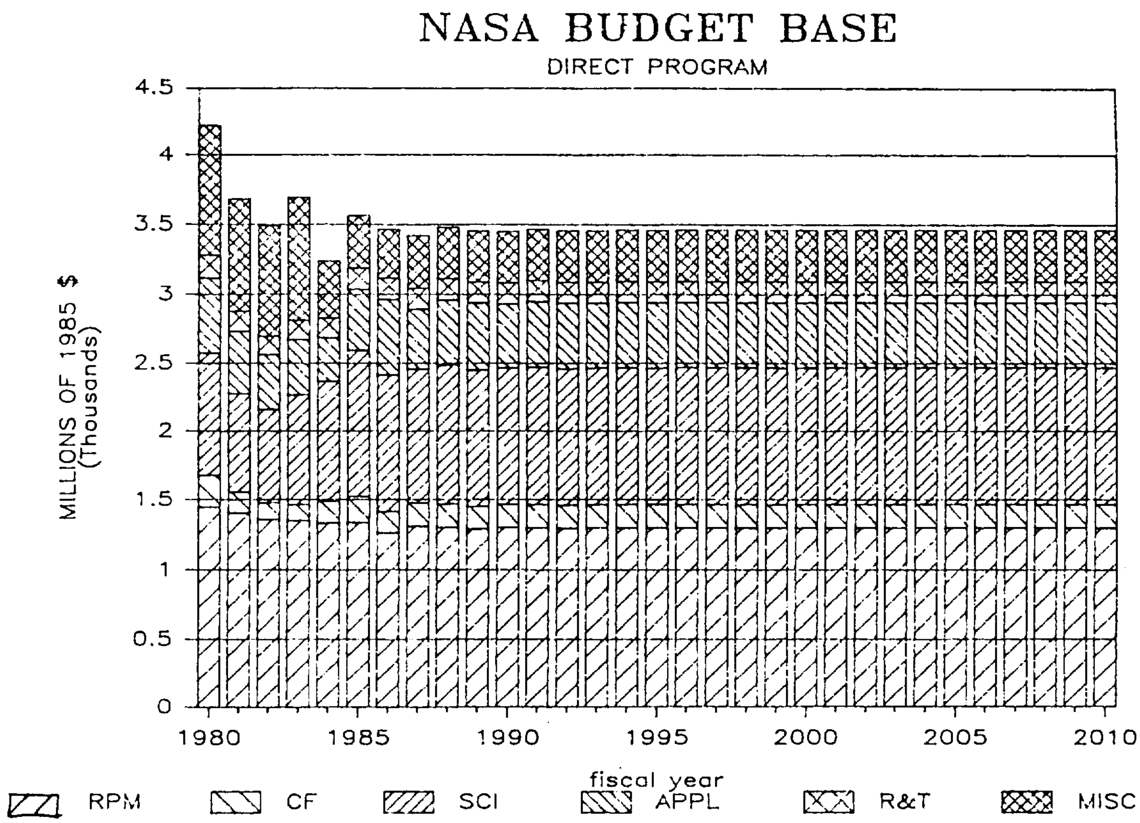

FIGURE 2

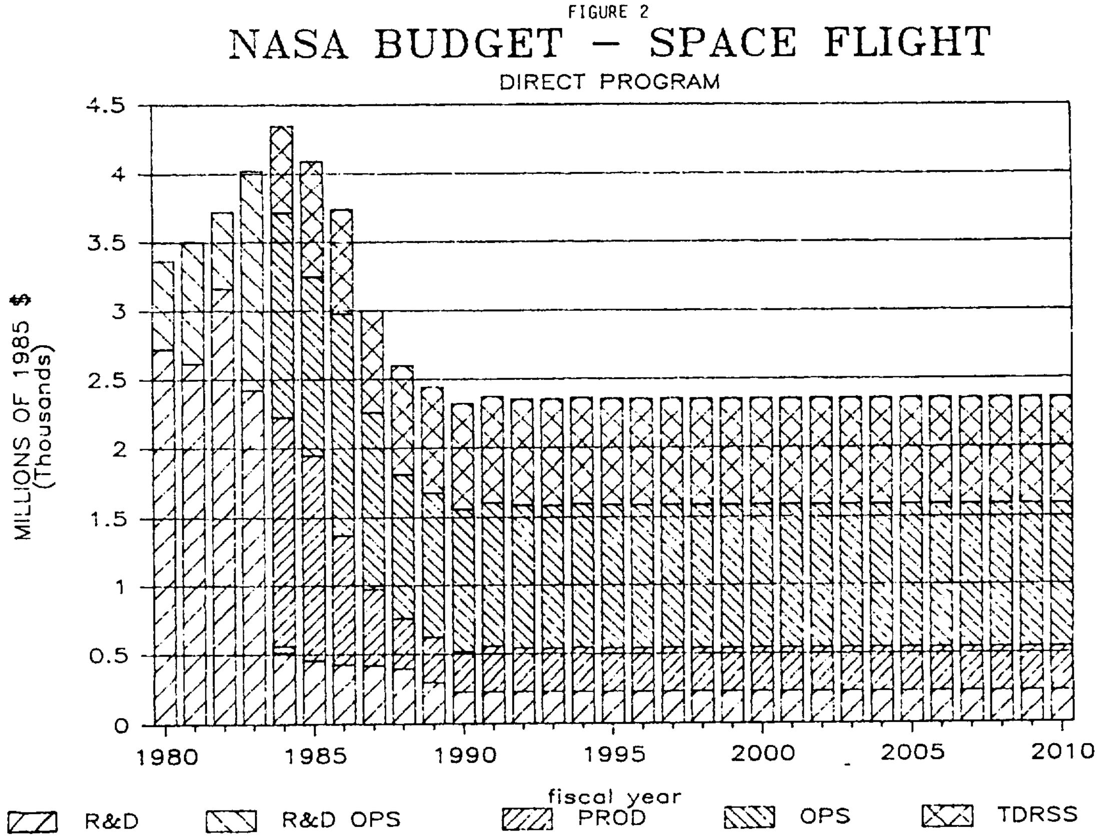

FIGURE 3

PLANNING PROFILE

SPACE STATION (SS)

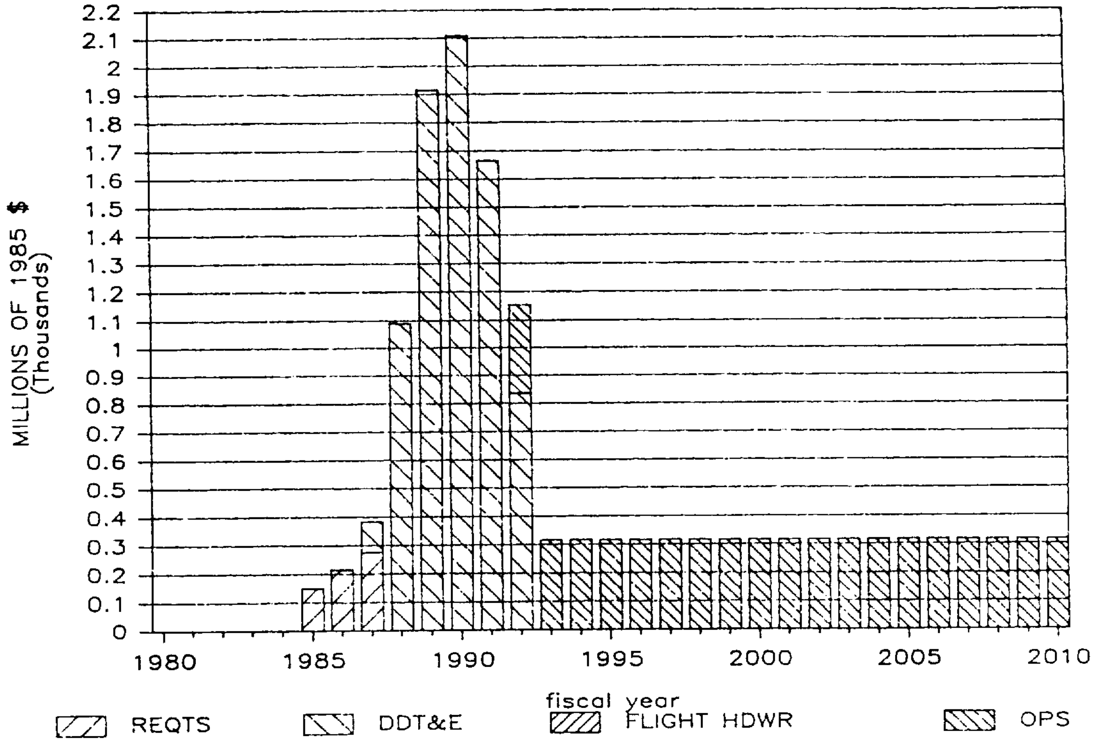

FIGURE 4

PLANNING PROFILE

Shuttle Derived Vehicle (SDV)

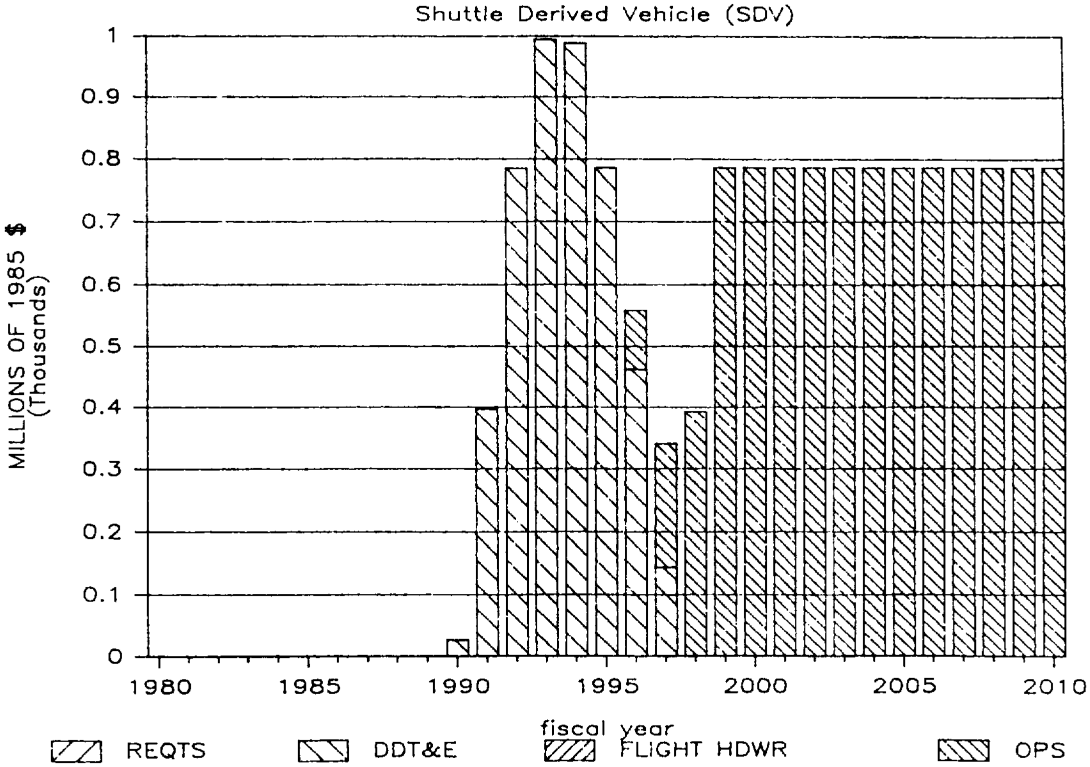

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FIGURE 5

PLANNING PROFILE

Manned Mars Mission (MARS)

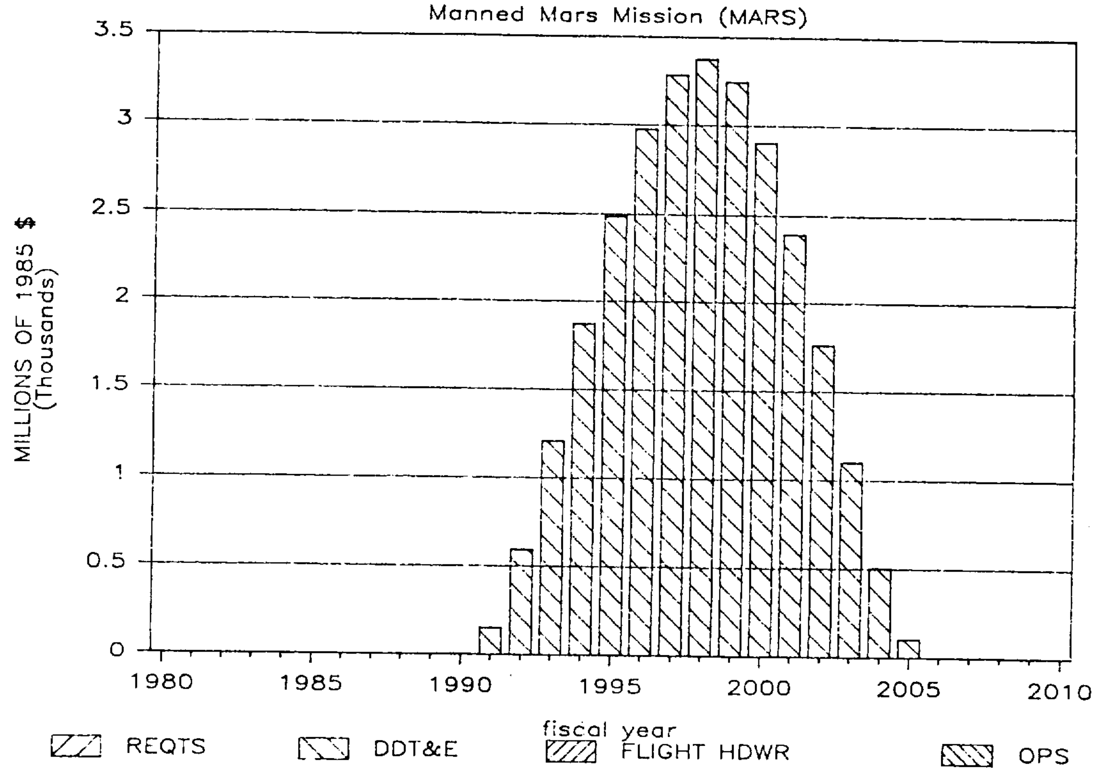
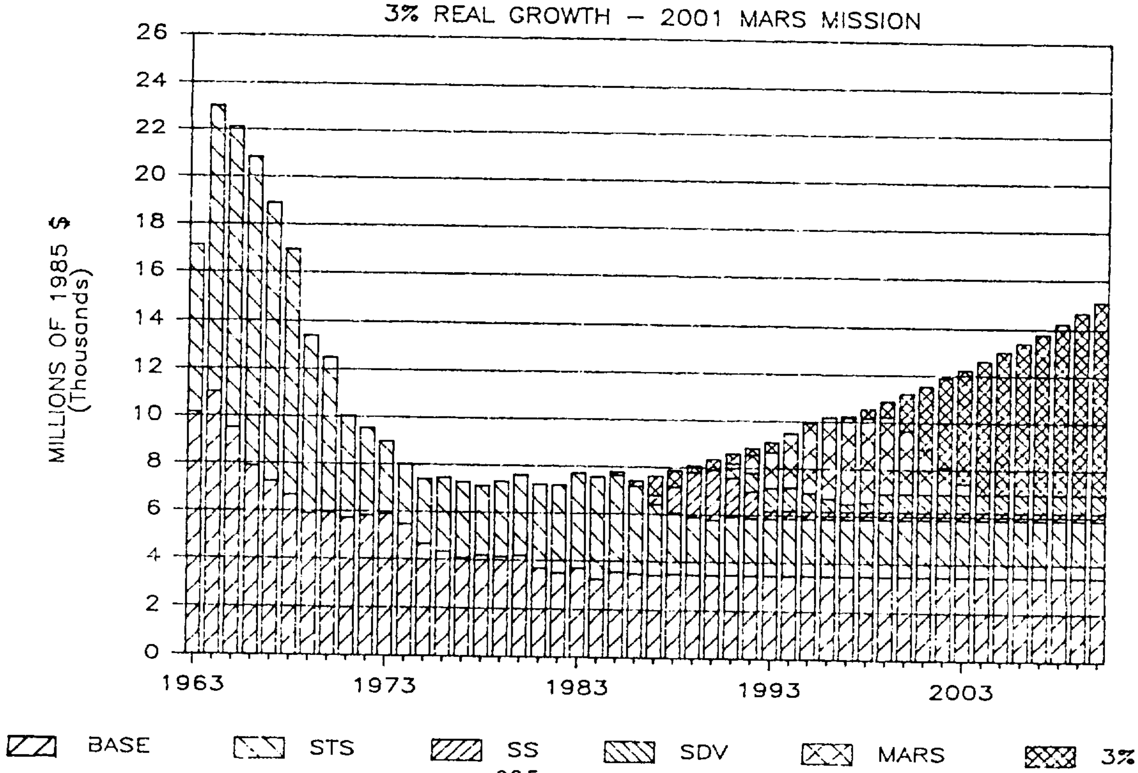


FIGURE 6

TOTAL NASA BUDGET

3% REAL GROWTH – 2001 MARS MISSION



C-6

N87-17800

MANNED MARS MISSION COST ESTIMATE

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ABSTRACT

The potential costs of several options of a manned Mars mission are examined. A cost estimating methodology based primarily on existing Marshall Space Flight Center (MSFC) parametric cost models is summarized. These models include the MSFC Space Station Cost Model and the MSFC Launch Vehicle Cost Model as well as other models and techniques. The groundrules and assumptions of the cost estimating methodology are discussed and cost estimates presented for six potential mission options which have been studied. The estimated manned Mars mission costs are compared to the cost of the somewhat analogous Apollo Program cost after normalizing the Apollo cost to the environment and groundrules of the manned Mars missions. It is concluded that a manned Mars mission, as currently defined, could be accomplished for under $30 billion in 1985 dollars excluding launch vehicle development and mission operations.

COST ESTIMATING METHODOLOGY

The costs for the manned Mars missions were primarily estimated using adaptations of existing parametric cost models which relate the cost of historical NASA programs to certain technical characteristics of those programs (e.g. weight, power, etc.). Figure 1 is a typical example of such a relationship. The majority of the hardware items required for the mission were estimated at the subsystem level using such cost estimating relationships (CER's). Specifically, the models utilized and their application were:

(1) COST MODEL: MSFC Launch Vehicle Cost Model

WHERE APPLIED:
- LEO Departure Stage and Engines
- Mars Arrival and Departure Stage and Engines
- Descent Stage and Engines
- Ascent Stage and Engines
- Earth Braking Stage
- KSC Launch Facilities

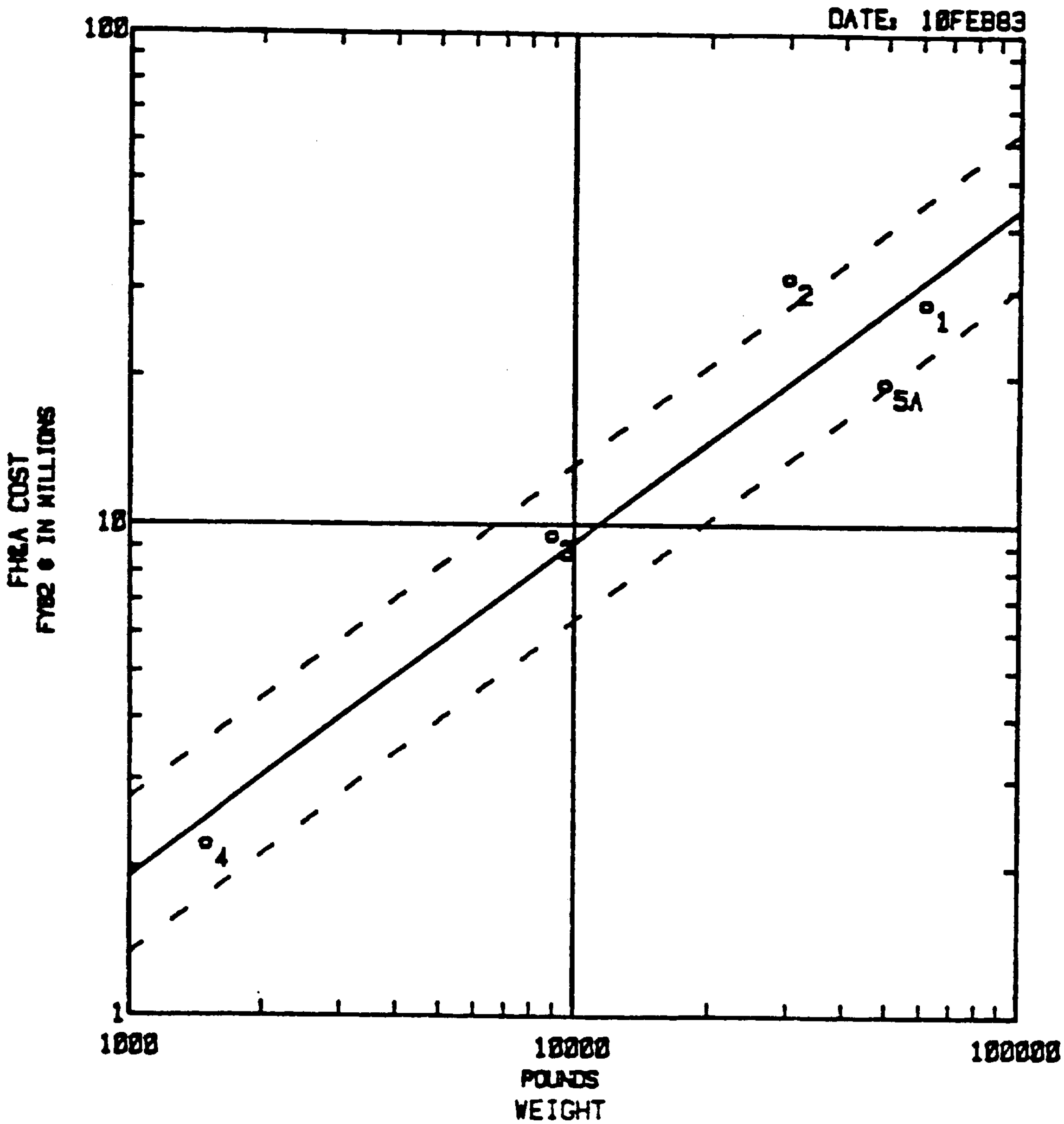
STRUCTURES/TPS-TANKS

FH&A COST

FY82 $ IN MILLIONS=(.0172) POUNDS$^{(.6815)}$

CC=.9578

SEA=.3027



EACH DATA POINT REPRESENTS COMBINED LOX AND FUEL TANKS

FIGURE 1

(2) COST MODEL: MSFC Space Station Cost Model
WHERE APPLIED: Mission Modules
Mars Excursion Module

(3) COST MODEL: GSFC Spacecraft Instruments Cost Model
WHERE APPLIED: Venus and Mars Moon Probes
Mission Module Experiments
Mars Surface Experiments

Certain elements of costs which were not estimated parametrically included STS and SDV-3R Launch Vehicle operations costs (i.e. costs per flight) which were taken from other currently ongoing MSFC studies. Also, insufficient definition existed to parametrically estimate Mission Control and Training Facilities and these were estimated strictly by engineering judgement. Some costs were not estimated, because of the lack of definition; these included low Earth orbit (LEO) assembly/logistics facilities, Orbital Transfer Vehicle (OTV) operations costs, Orbital Maneuvering Vehicle (OMV) operations costs, Space Station operating costs, Space Station facility cost impacts and Mission Operations costs.

OTHER GROUNDRULES AND ASSUMPTIONS

All costs presented are in constant 1985 dollars and include the prime contractor cost with fee.

A 14% allowance has been included for Program Support costs (which is generally consistent with other large NASA programs including Apollo and Shuttle and is also consistent with the allowance being carried in the current estimates of the Space Station Program). Program Support includes such activities as crew training and simulation, mission planning, computer support, software and data reduction, ground transportation, parallel development programs, propellants and consumables, tests using existing facilities (e.g. wind tunnel tests, KC 135 Zero G tests, etc.) and other costs which cannot be explicitly identified in a conceptual definition.

A 5% allowance has been included at the program level (i.e. 5% of total program cost) as an allowance for a major prime contractor integration contract. This is, again, generally representative of the experience of other major space programs.

The Mars spacecraft Mission Module assumes some inheritance from Space Station habitation modules and associated subsystems. This was reflected in the cost estimates of the appropriate Mission Module subsystems (e.g. pressurized structure, solar arrays, fuel cells, environmental control and life support, crew accommodations, and other selected subsystems) through the use of complexity factors in the range of 0.7 to 0.9. (Meaning that the items costed would be expected to cost from 70% to 90% of historical trends).

Some discussion has been given to utilizing inherited hardware for two stages of the Mars mission spacecraft. The Low Earth Orbit Departure Stage might be a derivative of External Tank (ET) hardware or possibly SDV-3R hardware (which itself might utilize ET hardware). The Mars Departure Stage in some configurations studied could be an Orbital Transfer Vehicle which will likely be in the NASA inventory of vehicles by the late 1990's. Because these hardware inheritance concepts are preliminary ideas and because they are dependent upon the configuration of the Mars mission stages, no cost savings have been assumed at the present time. The potential exists, however, for some reductions in stage cost as these options are further explored.

The Mars mission Earth-to-LEO transportation costs assumes use of the Shuttle at $100 million per flight and the SDV-3R at $80 million per flight. Because OTV and OMV traffic requirements have not been identified, no costs are included for these systems. It is expected that any such costs, when identified, will be relatively minor.

A 15% weight contingency has been included in all weights which were used as CER independent variables. In addition, a 35% cost contingency has been included at the module/stage level in the cost estimate. These contingencies are meant to reflect uncertainties in the weight estimating, cost estimating, and program definition processes and are considered adequate for the level of definition of the project.

In general, the cost estimates reflect the prototype approach, with one system test hardware article and one flight article. An exception to this is the Mars Excursion Module ascent and descent engine development program which assumed 15 test articles (consistent with historical engine development programs).

The manned Mars mission cost estimate assumes the existence, availability and use of the SDV-3R launch vehicle, Space Station, Orbital Transfer Vehicle, Orbital Manuervering Vehicle, STME Engine, RL100 Engine, TDRS and a deep space communications capability. This groundrule is based upon the assumption that each of the above systems will be developed prior to the time-frame of the manned Mars mission for use in other programs. This premise is perhaps most debatable in the case of the SDV-3R launch vehicle. However, if the Mars mission is the first user of the SDV-3R, there are undoubtedly other space programs which would greatly benefit from the existence of a heavy lift capability. Due to these uncertainties, the development cost of the SDV-3R is carried in the manned Mars mission as a "below the line" cost and is not charged to the Mars mission in this analysis.

COST ESTIMATES

Major cost estimating emphasis was placed upon estimating the costs of six potential manned Mars mission options. These cases were: (A) 1999 Opposition, LOX/LH_2 Propulsive-Braked; (B) 1999 Conjunction, LOX/LH_2 Propulsive-Braked; (C) 1999 Opposition, Aerobraked; (D) 1999 Conjunction, Aerobraked; (E) 2001 Opposition LOX/LH_2 Propulsive Braked; and (F) 2001 Opposition, Aerobraked.

While numerous other missions are possible, it is felt that these six represent a viable sampling of cases which should be representative of the costs to be expected for a manned Mars mission. Table 1 and Figure 2 present summary cost data for these configurations. These costs, which exclude the development cost of the SDV-3R launch vehicle, range from a total of about $23 to $24 billion for the aerobraked cases, to about $26 to $27 billion for the propulsive-braked options. Due to the flight mechanics of the 1999 mission opportunities, the propulsive energy required for the conjunction class missions is less than the energy required for the opposition class missions (see "Mission Concepts and Opportunities" by Young). This is reflected in the cost shown in Table 1 in the Stages and Transportation cost line items which are the costs of the stage hardware and the Shuttle/SDV-3R transportation operations, respectively. The Spacecraft and Science costs however, are higher for the conjunction class missions, reflecting the impact of the longer stay times of these missions.

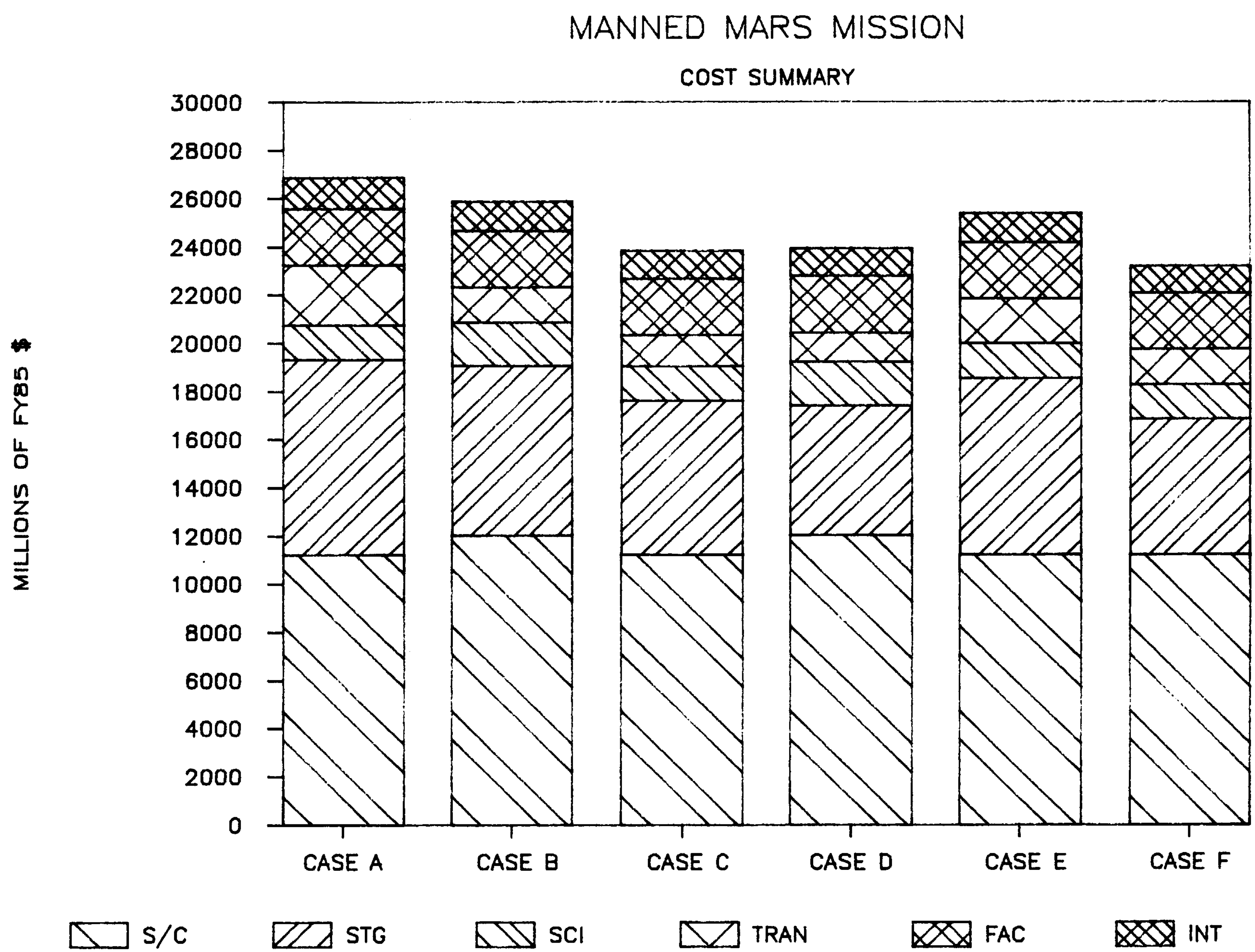
TABLE 1

MANNED MARS MISSION COST SUMMARY

(ALL COSTS FY85 $ IN MILLIONS)

	Case A 1999 Opposition Propulsive Braked	Case B 1999 Opposition Propulsive Braked	Case C 1999 Opposition Aerobraked
Spacecraft	$11,201	$12,009	$11,201
Stages	$8,095	$7,028	$6,389
Science	$1,439	$1,818	$1,439
Transport	$2,500	$1,460	$1,300
Facilities	$2,330	$2,330	$2,330
Integration	$1,278	$1,232	$1,133
TOTAL	$26,843	$25,877	$23,792

	Case D 1999 Conjunction Aerobraked	Case E 2001 Opposition Propulsive Braked	Case F 2001 Opposition Aerobraked
Spacecraft	$12,009	$11,201	$11,201
Stages	$5,387	$7,327	$5,629
Science	$1,818	$1,439	$1,439
Transport	$1,220	$1,860	$1,460
Facilities	$2,330	$2,330	$2,330
Integration	$1,138	$1,208	$1,103
TOTAL	$23,902	$26,365	$23,162



MANNED MARS MISSION
COST SUMMARY
MILLIONS OF FY85 $
30000
28000
26000
24000
22000
20000
18000
16000
14000
12000
10000
8000
6000
4000
2000
0
CASE A
CASE B
CASE C
CASE D
CASE E
CASE F
S/C
STG
SCI
TRAN
FAC
INT

FIGURE 2

From the cost and technical standpoint, Case F appears to be the most attractive option of the six cases investigated. For simplicity, the remainder of this paper will refer to Case F, the 2001 Opposition Mission with aerobraking.

Table 2 and the corresponding pie chart of Figure 3 detail the $23.1 billion dollar cost estimate for this mission. Nearly 50% of the total cost is attributable to the Spacecraft. As can be seen from Table 2, this cost further subdivides into the Habitation Module, the Laboratory/Logistics Modules and the Mars Excursion Module (MEM). Nearly one fourth of the total cost is for interplanetary stages which include the LEO Departure Stage, the Mars Arrival and Departure Stage, the MEM Ascent and Descent Stages and the Earth Braking Stage. The remaining fourth of total program costs is accounted for by the Experiments and Probes (6%), Transportation Operations (6%), ground based facilities (10%), and an allowance of about 5% for project level integration. The SDV-3R development cost is cited on Table 2 as a "below the line" cost.

As can be seen from Table 2, a little over three fourths of total cost would be expended in the development phase of the project and the remaining cost in the production phase. Development includes all system hardware DDT&E, and system test articles and facilities; production includes flight article procurement and transportation operations.

Figure 4 displays the anticipated annual funding requirements of the manned Mars mission. The distribution of funding assumes a nine-year development and production span and a distribution of funds corresponding to a Beta distribution with 60% of costs incurred in 50% of the time for development costs and uniform funding for production costs (a typical distribution for NASA projects). Peak year funding would occur in year three, with a requirement of about $5.6 billion. The inflection point in year seven is due to the buildup of flight hardware production activities.[1]

[1] It should be noted that this prediction process estimates actual costs as they would appear looking back in time from the end of the program. If the values are to be used for budgeting, reserves for program growth should be identified at the peak years and beyond, to reflect actual historical program trends.

TABLE 2

MANNED MARS MISSION PRELIMINARY COST ESTIMATE

(FY85 DOLLARS IN MILLIONS)

2001 OPPOSITION

ALL AEROBRAKE

CASE F

	DDT&E	PRODUC-TION	TOTAL
Mission Module-Habitation Module	$3,774	$1,135	$ 4,909
Mission Module-Lab/Log Module	$1,704	$ 283	$ 1,987
Mars Excusion Module	$3,778	$ 527	$ 4,305
Spacecraft Subtotal	$9,256	$1,945	$11,201
LEO Departure Stage	$2,533	$ 426	$ 2,959
Mars Arrival & Departure Stage	$1,131	$ 225	$ 1,356
MEM Ascent & Descent Stages	$1,212	$ 102	$ 1,314
Earth Braking Stages	$ 0	$ 0	$ 0
Stages Subtotal	$4,876	$ 753	$ 5,629
Experiments & Probes	$ 763	$ 676	$ 1,439
SDV Transportation	-----	$ 960	$ 960
STS Transportation	-----	$ 500	$ 500
Transportation Subtotal	-----	$1,460	$ 1,460
Launch Facilities	$2,130	-----	$ 2,130
Mission Control	$ 100	-----	$ 100
Training Facilities	$ 100	-----	$ 100
Facilities Subtotal	$2,330	-----	$ 2,330
Space Station Impacts	TBD	TBD	TBD
Program Level Integration	$ 861	$ 242	$ 1,103
Total	$18,086	$5,076	$23,162
Launch Vehicle Development	$ 3,000	$1,000	$ 4,000

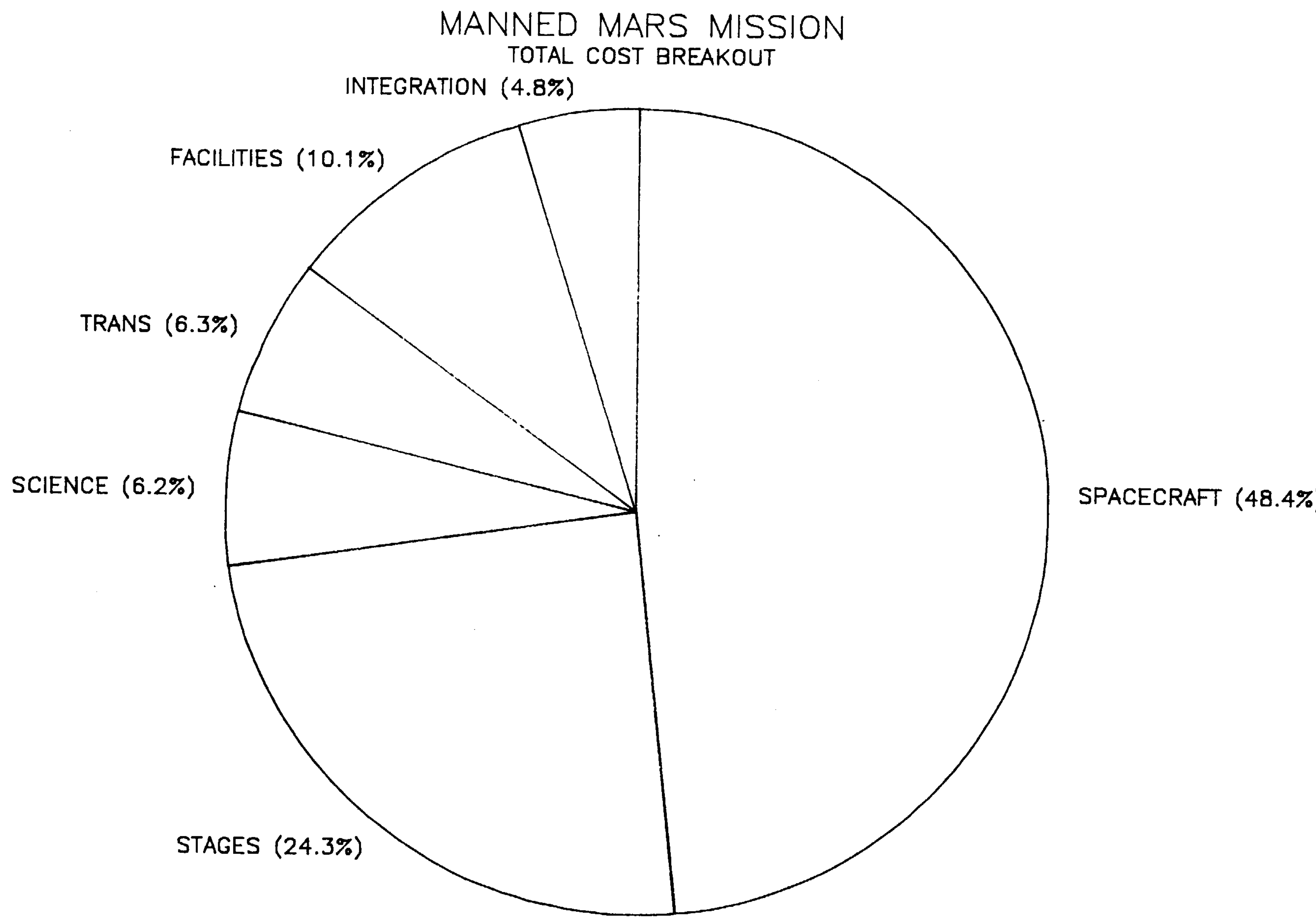
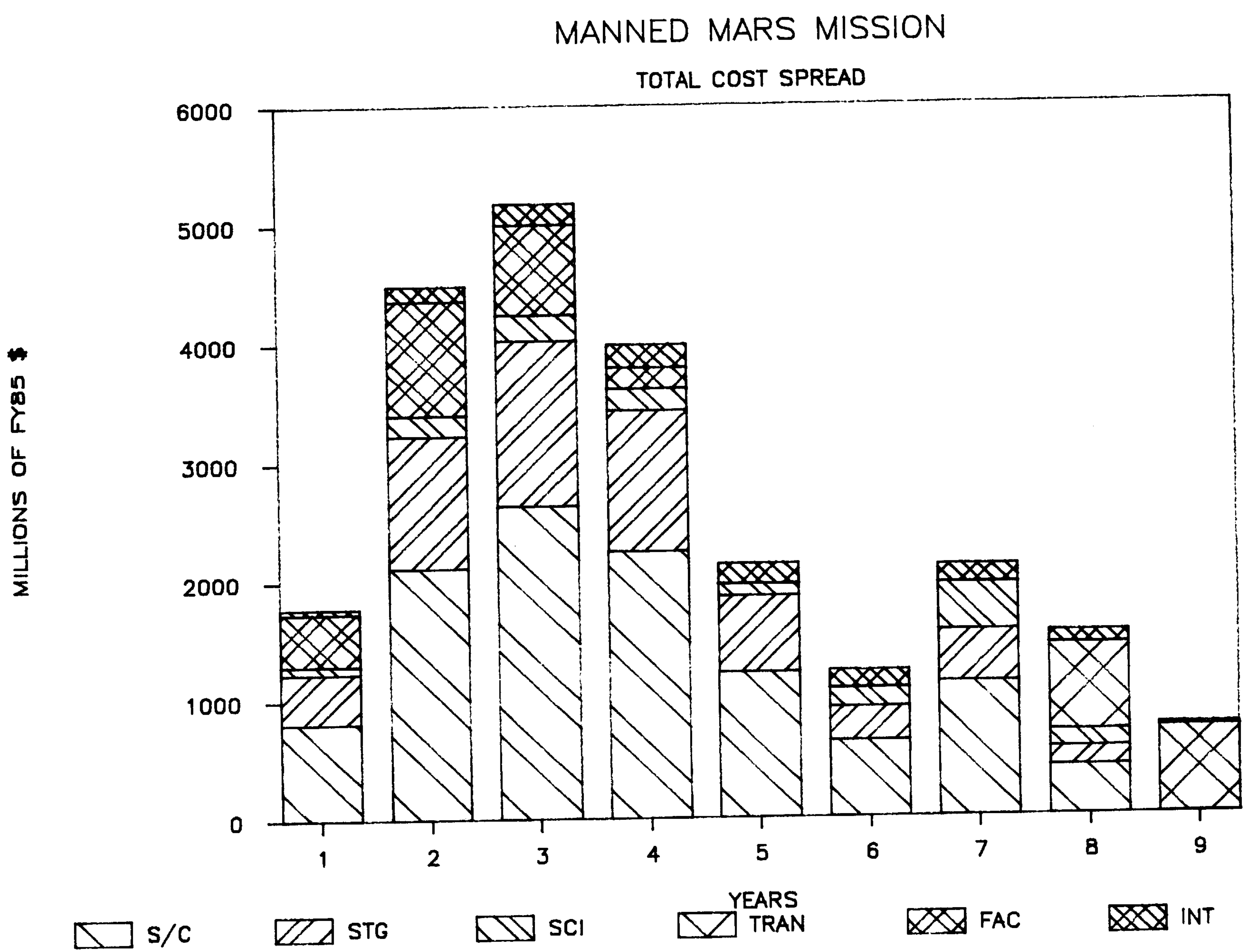

MANNED MARS MISSION
TOTAL COST BREAKOUT
INTEGRATION (4.8%)
FACILITIES (10.1%)
TRANS (6.3%)
SCIENCE (6.2%)
STAGES (24.3%)
SPACECRAFT (48.4%)

FIGURE 3



MANNED MARS MISSION
TOTAL COST SPREAD
MILLIONS OF FY85 $
6000
5000
4000
3000
2000
1000
0
1
2
3
4
5
6
7
8
9
YEARS
S/C
STG
SCI
TRAN
FAC
INT

FIGURE 4

COMPARISON OF MANNED MARS MISSION AND APOLLO COST

The historical space project most comparable to the manned Mars mission is the Apollo Program. In fact, there is a tendency to assume that the cost of a mission to Mars will cost as much as the Apollo Program (in the same year dollars). There are a number of reasons why this need not be true. The comparison of the cost of Apollo to the cost of a manned Mars mission requires cognizance of several fundamental differences in the cost drivers of these two programs.

First, the Apollo Program was mission and schedule constrained. The goal was to put man on the Moon by the end of the (1960's) decade. Thus cost, the third variable in any program, became the unconstrained variable. Cost was allowed to grow in order to meet the mission and schedule goals which were deemed to be unchangeable. Presumably, the Mars mission will be accomplished in an environment that allows flexibility in all three program variables such that something near the optimum mission, schedule, and cost can be achieved.

Secondly, at the beginning of the Apollo Program, the space infrastructure was still in its infancy. The Mars mission, on the other hand, will benefit from a space program with a forty year experience base.

Thirdly, because space was such a new and largely unknown environment in which to operate (and also probably because the funding was available), the Apollo Program had extremely intensive redundancy and test philosophies. The typical flight system was preceeded by dozens of test articles. From this beginning, the space program has matured to the point where the typical manned system today (e.g. Shuttle or Space Station) has, at most, one test article.

Finally, the cost of the Apollo Program which is widely quoted (about \$20 billion in "then-year" dollars or about \$80 billion in 1985 dollars) purchased the entire series of Saturn Vehicles and Apollo Moon landing missions. The manned Mars mission cost presented in this paper is for the initial mission only.

Therefore, before comparisons are valid, the \$80 billion price tag for Apollo must be analytically normalized to a basis consistent with the environment of the Mars mission as estimated in this paper.

Figure 5 shows such a normalization. The $20 billion ($80 billion in 1985 dollars) cost of Apollo is shown broken into its major components.

The first normalization excludes the Saturn I and Saturn IB launch vehicles which were precursors to the Saturn V development program and have no analogous requirements in the manned Mars mission. This reduces the $80 billion Apollo cost to around $73 billion.

The second adjustment reduces the large number of test articles in each of the Apollo Program line items to the equivalent of one prototype test article and one flight article for the non-engine program line items and to the equivalent of 15 test articles for the engine development line item. This adjustment to today's test philosophy reduces the cost down to around $61 billion.

In order to be consistent with manned Mars mission cost, the final adjustment deletes the cost of all Apollo missions beyond the first mission. This reduces the Apollo cost to about $37 billion.

This cost still includes some artifacts of the Apollo era way of doing business which were difficult to quantify. These include parallel development programs and heavy Supporting Research and Technology activities. Also note that about $16 billion ($14 billion plus a pro rata share of mission support) is relatable to the launch vehicle. Therefore, the basic Apollo cost which is comparable to the manned Mars mission cost estimate is around $21 billion, which is actually slightly lower than the range of costs estimated in this paper. Considering the increased challenges due to the greater interplanetary distances involved in the Mars mission, cost in the mid-to-upper twenty billion dollar range for the Mars mission seems appropriate.

1539-85

COST COMPARISON OF APOLLO AND MANNED MARS MISSIONS

APOLLO PROGRAM "THEN–YEAR" COSTS ($ M)	
SPACECRAFT	$ 8482
SATURN I	767
SATURN IB	1128
SATURN V	7172
ENGINE DEV.	901
MISSION SUPT.	2003
TOTAL	$20453

$80 BILLION
1985 $

ADJUSTING APOLLO COST TO EXCLUDE PRECURSOR LAUNCH VEHICLE DEVELOPMENT AND TO 1985 $

APOLLO EXCLUDING SATURN I & IB 1985 $ M	
SPACECRAFT	$33512
SATURN V	28337
ENGINE DEV.	3560
MISSION SUPT.	7101
TOTAL	$72510

ADJUSTING APOLLO COST TO TODAY'S TEST PHILOSOPHY

APOLLO WITH REDUCED TEST ARTICLES 1985 $ M	
SPACECRAFT	$25792
SATURN V	26962
ENGINE DEV.	2017
MISSION SUPT.	5946
TOTAL	$60717

ADJUSTING APOLLO COST TO 1 MISSION FOR COMPARISON TO MANNED MARS

APOLLO WITH REDUCED FLIGHT ARTICLES 1985 $ M	
SPACECRAFT	$17248
SATURN V	14072
ENGINE DEV.	2017
MISSION SUPT.	3619
TOTAL	$36956

- TO COMPARE COST OF APOLLO TO COST OF A MANNED MARS MISSION REQUIRES COGNIZANCE OF SEVERAL FUNDAMENTAL DIFFERENCES IN THE COST DRIVERS OF THESE TWO PROGRAMS
  - APOLLO WAS MISSION AND SCHEDULE CONSTRAINED (COST WAS THE UNCONSTRAINED VARIABLE)
  - MODEST INFRASTRUCTURE AT START OF APOLLO
  - APOLLO HAD EXTREMELY INTENSIVE TEST PHILOSOPHY
  - THE $80 BILLION PURCHASED MULTI–MISSIONS

- ANALYTICALLY NORMALIZING HISTORICAL APOLLO COST TO A BASIS MORE CONSISTENT WITH MANNED MARS DECREASES THE $80 BILLION APOLLO PROGRAM TO ABOUT $37 BILLION
  - THIS COST STILL INCLUDES PARALLEL DEVELOPMENTS, HEAVY SR&T AND OTHER ARTIFACTS OF APOLLO WAY OF DOING BUSINESS
  - ABOUT $16 OF THE $37 BILLION IS RELATABLE TO LAUNCH VEHICLE
- CONCLUSION: THE APOLLO ANALOGY SUGGESTS THAT THE COST FOR AN INITIAL MANNED MARS MISSION SHOULD BE IN THE $20 BILLION RANGE (1985 $) EXCLUDING LAUNCH VEHICLE/ENGINE DEVELOPMENT

FIGURE 5

CONCLUSIONS

The preliminary engineering cost analysis of the manned Mars mission indicates that the cost, excluding launch vehicle development and mission operations, should be less than $30 billion in 1985 dollars for the initial mission. This cost estimate independently compares well with the cost of the somewhat analogous Apollo Program when the cost of that program is normalized to the environment and groundrules of the estimate for the manned Mars mission.[2]

[2] A separate paper (by K. Cyr) describes the budgetary requirements and timing which might reasonably be anticipated for the program, illustrating that missions in this time period are feasible within reasonable budget levels.

N87-17801

INTERMARS: USER-CONTROLLED INTERNATIONAL MANAGEMENT SYSTEM FOR MISSIONS TO MARS

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ABSTRACT

Existing international space law as well as the best interest of all nations are consistent with the establishment of a user - based international organization, herein called INTERMARS. INTERMARS would provide access to facilities and services at a Martian base which would be of high functional potential, quality, safety and reliability. These opportunities would be available on an open and non - discriminatory basis to all peaceful users and investors.

INTERMARS is a model organization concept tailored to provide cooperative international management of a Martian base for the benefit of its members, users and investors. Most importantly, INTERMARS would provide such management through a sharing of both sovereignty and opportunity rather than unilateral control by any one nation or set of competing nations.

Through an Assembly of Parties, a Board of Governors, a Board of Users and Investors and a Director General, INTERMARS would meet its primary goal as it would be in the self-interest of all members, users and investors to do so. The internal structure and philosophy of INTERMARS would provide not only for all participants to have representation in decisions affecting its activities, but also would insure effective and responsive management. Surely this is the precedent we wish to establish for mankind at the now not-so-distant shores of the new ocean of space.

INTRODUCTION

People throughout the world want space to be a frontier for human cooperation as well as a frontier of freedom and achievement. Unfortunately, the narrow political designs as well as the legitimate national interests of the nations of the world make broad cooperation in any area very difficult.

Such cooperation, however, is not impossible. The success of the Apollo-Soyuz mission in 1974 and various Soviet-French efforts shows that joint efforts involving international adversaries are possible, at least if objectives are relatively limited and well-defined. The very successful Spacelab, developed by the European Space Agency and flown by the United States on the Space Shuttle demonstrates the potential for very close cooperation between free nations. Most importantly, in the INTELSAT and INMARSAT telecommunication organizations, we have examples where nations of all levels of economic development and all varieties of political persuasion have found it in their self-interest to cooperate in space-related projects.

If we are to see broadly based international cooperation in space, we must first see a commitment to the rule of a body of space law. Although not perfect, and certainly not complete, currently recognized and internationally sanctioned tenets of space law provide a workable base for future cooperation. The free world, however, must be very cautious about agreeing to any partial or total legal framework for space that either limits rational free enterprise activities or allows the one-nation, one-vote control of cooperative organizations. The United Nations developed "Moon Treaty" and "Law of the Sea Convention" are examples of the dangers of ill-conceived and badly negotiated new international law based on extreme applications of the otherwise acceptable notion that the Moon and the sea are the "common heritage of mankind". Fortunately, few nations have ratified these documents, nor should they be ratified without major amendment.

Currently, recognized principles in space law, as established by international treaties, are reasonably general and straightforward: (1) Space, including celestial bodies, is the province of mankind and should be developed for its benefit; (2) Space, including celestial bodies, should be free for access, exploration, scientific investigation and use by all countries; (3) Space, including celestial bodies, is not subject to national appropriation by claims of sovereignty, by means of use or occupation, or by any other means; (4) Space, including celestial bodies, shall be used exclusively for peaceful purposes; and (5) International law as formulated on Earth extends to space and celestial bodies.

These five principles, which have developed slowly over the last 25 years, are embodied in several multilateral treaties now in force, but most particularly in the Outer Space Treaty of 1967. They provide the currently recognized international legal framework for initiating, planning and implementing international cooperation in space.

The principal new notion currently being explored as a possible basic tenet of space law is that which states that space and all celestial bodies are the "common heritage of mankind". Not only is this notion somewhat inconsistent with the ongoing search for extraterrestrial intelligence, but it is seriously flawed in its more extreme application as currently embodied in the proposed 1979 "Agreement Governing the Activities of States on the Moon and Other Celestial Bodies", or "Moon Treaty", and 1982 "Law of the Sea Convention". Under the "common heritage of mankind" notion, several new principles of space law, as they apply to the Moon, would be added to the list given above.

First, a celestial body, such as Mars, or any part of it would not be subject to appropriation by any entity, including private, corporate, national, or international interests. It would be owned or possessed by no one.

Second, all nations would share equally in the management of activities in space. National and limited international interests would be subordinate to so-called universal interests.

Third, any benefits from the exploitation of natural resources in space would be shared by all nations, not just those who developed the capability to exploit such reaources.

The principal theoretical difficulty many see in the implementation of a "common heritage" regime is that unless there is international consensus on significant issues of administration, the resources of space would go unused indefinitely. Issues put to a vote would be divided on a one-nation, one-vote basis which would ultimately politicize decision-making. The practical difficulties in operating under comparable regimes of consensus and one-nation, one-vote principles are becoming increasingly evident in the politicization of international agencies such as UNESCO, the International Telecommunications Union and the World Intellectual Property Organization, not to mention the United Nations, itself. The stagnation of such agencies and their increasing antagonism toward the principles of the free flow of information and basic human rights is forcing nations back to bilateral and multilateral agreements in order to get anything done.

As it has been for 25 years and, as is international law in general, space law must continue to evolve subject to the realities of national and international interests and activities in space. It must adapt to changing political and technical conditions.

For example, the free world is already moving toward free enterprise commercialization of near-Earth orbit facilities. Does this violate any of the principles enumerated above? In the eyes of communist and many developing nations, it probably does. What about the hard resources of space, those in the Moon, planets and asteroids? Are they forever off-limits to free enterprise? probably not. Therefore, the question becomes "Under what national or international regime will such activities be conducted?"

Further, at least some permanent space stations will be the "sovereign" territory of single nations or limited groups of nations. Does this violate the letter or intent of the current principles? Obviously, it would seem to. Moreover, can we assume that territorial sovereignty will not be claimed for the first lunar or Martian bases if established by national entities as current trends would indicate they will be? Thus, it would seem that in space the concept of "functional" sovereignty is already clearly established. Functional sovereignty is considered the right of states or cooperating groups of states to exercise jurisdiction and control over assets and activities they have in space. Whether or not functional sovereignty will replace, or be replaced by, territorial sovereignty, only time and circumstances will tell.

Finally, it is clear to all, space is not being used exclusively for "peaceful" purposes in spite of treaty agreements to the contrary. It is probably unreasonable to assume that any new geographical frontier can be immune from either plans for aggression or the need to defend against aggression. There are no clear historical precedents to indicate this is possible, particularly when presence at that geographical frontier has significant implications for the balance of power between nations.

BACKGROUND

The next major crucible of legal experimentation and development in space law will probably come when men and nations return to the Moon to stay. The existing regimes of law for space discussed above create significant legal constraints on nations interested in the establishment of permanent Martian bases. The obvious practical difficulties the world is experiencing with one-nation, one-vote international organizations provides significant pragmatic constraints on nations interested in international participation in a space base or settlement. Further, to realize the many recognized psychological, political and technical benefits of international participation in Martian base activities, the management regime of such a base must offer clear self-interest incentives to participation by major powers rather than the alternative of "going it alone" and "damn the legal torpedoes".

Fortunately, we have international experience with a successful model of a high-technology management system. This system conforms to the legal, operational and self-interest constraints that exists on international operations in space. This model system is INTELSAT, a user-based management organization for the operation of international telecommunications satellites.(1)

The political and technical management of a global communication satellite system, as manifested by the INTELSAT organization, is a unique new entry into the international scene. It is an organization that developed because of a coincidence of new technology and obvious international need. To the everlasting credit of the United States, we perceived this coincidence and guided the gradual trial-and-error development of INTELSAT. To the everlasting credit of the INTELSAT organization, it has become an example of international coperation that is not only remarkably successful, but also utilitarian and profitable.

The INTELSAT model has already spawned one successful imitator, INMARSAT, which manages international maritime communication satellites and includes the Soviet Union as a member. Modified versions of this model have been proposed for the management of international waterways(2), space-based antenna farms and lunar bases(3). Here we suggest consideration of another modified version of the INTELSAT management model which is appropriate to the international management of a Martian base.

We believe that "INTERMARS"(4), as we have termed this suggested organization, would satisfy all the previously discussed constraints of space law as well as be consistent with the principles of free enterprise which are held so dear in democracies of the world. Most importantly, INTERMARS would bring into the management of a Martian base those nations and other interests with the greatest motivations for insuring the successful implementation of that management.

The concept of INTERMARS is a concept of the space age and of the recognition that space resources are common resources of the spaceship Earth. INTERMARS does not require that territorial sovereignty be given up in space; it does not require that free-enterprise opportunities be abandoned in space; it merely requires that sovereignty and opportunity be shared.

BASIS FOR INTERMARS CONCEPT

Technological advancements have produced a trend towards realization of a "common heritage of mankind" in certain international resources. This trend is most apparent in negotiations regarding the resources of the seas and outer space. It indicates a general realization that nations have common interests in sharing benefits from the exploitation and environmentally sound use of these resources.

It must be recognized that Mars can become a common heritage resource for mankind. It also must be recognized that Mars will not be available to mankind without a workable management system and a peaceful management environment. An institutional arrangement should be possible which would vest operation and control of Martian bases in an organization composed of nations who will actively participate in creating such bases with association of those other entities who are solely users of the bases or investors in the technologies required to establish them. Such nations and entities would be united by a common bond of policy and purpose which would be focused on both the technical and financial success of the enterprise.

The advantages of sharing sovereignty and opportunity under this concept should be clear. First, the potentially disastrous discontent over which nation should exert control over Martian operations would be largely alleviated.

Second, the concept can provide institutionalized access and influence to all participants. Nations, users and investors with any degree of participation in INTERMARS would have to be consulted, eliminating the possibility that small or temporarily small participants could be frozen out entirely.

Third, the operational objectives of a base or settlement would be best met by this concept. The most important of these objectives are (1) assuring access by all members to the base and its services; (2) assuring access to proprietary technologies and available material resources in proportion to investment; (3) assuring access to Martian scientific resources; (4) maintaining reasonable and uniform rate structures bearing a realistic relationship to the value derived from the use of the base (and of spacecraft moving to and from it) while also considering operating expenses and return on investment; (5) assuring administrative stability over long periods of time; (6) assuring effective maintenance and operation of facilities and services; and, (7) assuring continued and environmentally sound expansion, improvement and development of spacecraft, facilities, and services.

Finally, creation of an international organization of all nations, users and investors who wish to actively participate in the excitement of space pioneering cannot help but improve the friendship and unity of purpose of nations and peoples on Earth.

MANAGEMENT STRUCTURE

The conceptual advantages of a user-based international organization will only be realized if the actual institutional structure is designed to provide an equitable system for the various interests to exert influence and control, as well as provide for efficient and proper management of the base.

There are two distinct mechanisms for nations, users and investors to be involved in INTERMARS. The first mechanism relates to the creation and operation of a Martian base. It draws to it those nations that contribute directly and substantively to the activities required to establish the base and stabilize its initial operation. The second mechanism relates to the use and the terms and conditions for use of the base, its accessible resources and the proprietary technologies required to establish it. This second mechanism draws to it those nations, users and investors who contract with or invest in INTERMARS in order to benefit from its activities.

The main functioning bodies within INTERMARS would be the Assembly of Parties, the Board of Governors, the Board of Users and Investors and the Director General's office. The member nations of the Assembly of Parties would collectively exert policy authority over the major contributing nations comprising the Board of Governors which, in turn, would exert operational authority over the Director General, the operating entity of INTERMARS. The Board of Users and Investors, working within the policy framework set down by the Assembly of Parties, would develop recommendations on operational issues affecting their interests. These recommendations would be presented to the Board of Governors through the Board of Users and Investors formal representatives on that Board.[(5)]

PROVISION FOR SELF-DETERMINATION

Inherent in the concept of establishing a permanent Martian base is the high probability that such a base would ultimately become a human settlement of permanent residents. If our history on Earth is any indication, such permanent residents will eventually desire a controlling voice in the governing of their activities. We should take this possibility into account in the initial structure of INTERMARS so as to avoid the conflicts that plagued colonial establishments in the past.

The best way to do this is to create from the beginning of INTERMARS a clear mechanism by which the settlers can be represented in its organizational entities and by which the settlers can have majority control of INTERMARS at an appropriate level of population. Thus, the INTERMARS charter should contain concepts such as the following: (1) The provision for a seat for INTERMARS settlers on the Assembly of Parties, the Board of Governors and the Judicial Tribunal; (2) The provision for the systematic accumulation of voting shares for INTERMARS settlers based on the number of settlers who qualify as permanent residents; and (3) Clear recognition that the success of INTERMARS will guarantee that its settlers will ultimately gain voting control of the organization if they then desire such control. The net result of these concepts would be the transition of INTERMARS from an international exploration, management and investment organization to a true Martian government.

RIGHTS AND OBLIGATIONS OF MEMBERS

The INTERMARS charter must spell out the rights and obligations of its member nations, users and investors. Although this would be the subject of much negotiation, a few points appear to be critical to the success of the enterprise.

First, the member nations must agree to refrain from the establishment, or cooperation in the establishment, of any other facilities and services related to Martian bases outside those of INTERMARS unless it is done jointly with INTERMARS.

Second, the member nations must agree that INTERMARS facilities and services, including those national facilities and services committed to INTERMARS by contract, shall be neutral so that in time of hostilities or threatened hostilities, INTERMARS facilities, services and personnel would remain secure to peaceful use by all nations without discrimination. Thus, INTERMARS should not be a target of hostile forces in any armed conflict.

IMPLEMENTATION

It is never simple to initiate and then implement a new international concept or organization. INTERMARS will be no exception. However, the establishment of INTERMARS is clearly possible so long as the commitment of the United States to the establishment of a Martian base is unequivocal and there is a sincere willingness to search for a fair means of international participation in such an endeavor. On the other hand, if the United States is, or appears to be, hesitant and uncommitted to either the base or international participation, then it is highly probable that the Soviet Union and possibly other nations or groups of nations will "go it alone". If this should happen, a great opportunity for increased cooperation and trust among otherwise competing nations will be lost.

With commitment to a Martian base by the United States, the next logical step would be the convening of an international conference to consider a draft of an INTERMARS charter. This draft charter should be the product of extensive bilateral and multilateral discussion between nations critical to the ultimate political viability of the organization. The United States clearly would have to take the lead in this early drafting period, but there is no reason why the final drafting conference should not be by joint invitation of all interested nations. All nations should be invited to send official delegates or observers as they are so inclined. Potential user or investor entities should be invited as observers or allowed to participate as members of official delegations.

An obvious question is, "How can the Soviet Union, the Soviet Bloc nations and the Developing Nations be brought into the development and implementation of INTERMARS ?" The answer lies in making participation "an offer they cannot refuse " as has been largely the case with INTELSAT and INMARSAT. Such an offer is inherent in, first, an unequivocal commitment by the United States, Europe and Japan; second, a clear willingness to share sovereignty, opportunity and technology; and third, a clear articulation of direct human, scientific and economic benefits to all participating nations. Once a reality and once it is clear it will be successful, INTERMARS will attract many of those nations that may at first be reluctant to participate. Although conceived as an international self - regulating monopoly, INTERMARS should always be open to new members and investors if it is to achieve its broad humanistic goals as well as its technical and economic purposes.

REFERENCES

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(2) Schmitt, H.H.,1978, Congressional Record, pp. S.3385-S.3388, S.3197, and S.3626-S.3634.

(3) Joyner, C. and Schmitt, H.H., 1984, "Lunar Bases and Extraterrestrial Law: General Legal Principles and a Particular Regime Proposal", Proceedings of "Lunar Bases and Space Activities of the 21st Century," LPI, (in press).

(4) We acknowledge, gratefully the early and critical contributions of Dr. Delbert Smith to the extension of INTELSAT concepts to other applications.

(5) For a detailed discussion of the comparable INTERLUNE organization, see Joyner and Schmitt, 1984, " Lunar Bases and Extraterrestrial Law: General Legal Principles and a Particular Regime Proposal".

MANNED MARS MISSION SCHEDULE REPORT

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ABSTRACT

This section contains the schedules for hardware for the initial manned Mars mission. The mission for the purpose of this report is determined to be a 1999 opposition mission and the vehicle hardware configuration for the mission is as depicted in Figure 1.

BACKGROUND

NASA has experienced phenomenal success in its brief history with the major programs of Saturn (I, IB, and V), Skylab, Shuttle, and Spacelab. Prior to the flight of a manned Mars mission, it is anticipated that the Space Station, Shuttle Derived Vehicle (SDV), and Orbital Transfer Vehicle (OTV) will have become operational. It is also anticipated that DOD will be well into the Strategic Defense Initiative (SDI) development and operation by the year 1999, and that NASA will have a large role in the development and operation of hardware for that system. Accepting the challenge for manned Mars missions will be one that NASA is equally qualified for and willing to assume.

During the late 1960's, NASA performed in-depth studies of various Mars missions. A large portion of the data generated as a result of those studies can be utilized for a future definition study for manned Mars missions. As mentioned earlier, the development of technology/hardware through these other programs will be of tremendous benefit through the reduction of development time and cost to the manned Mars program.

This paper attempts to cover all facets of the initial manned Mars mission for this particular launch vehicle configuration. The program begins with definition studies and continues with schedules for hardware necessary for reaching Mars and returning to Earth. Since hardware/software is the most tangible criteria from a scheduling standpoint, seven categories of hardware were selected for analysis. These categories are as follows: (1) Rocket Vehicles; (2) Spacecraft; (3) SDV-3R Payload Adapters; (4) Experiments; (5) LEO Assembly

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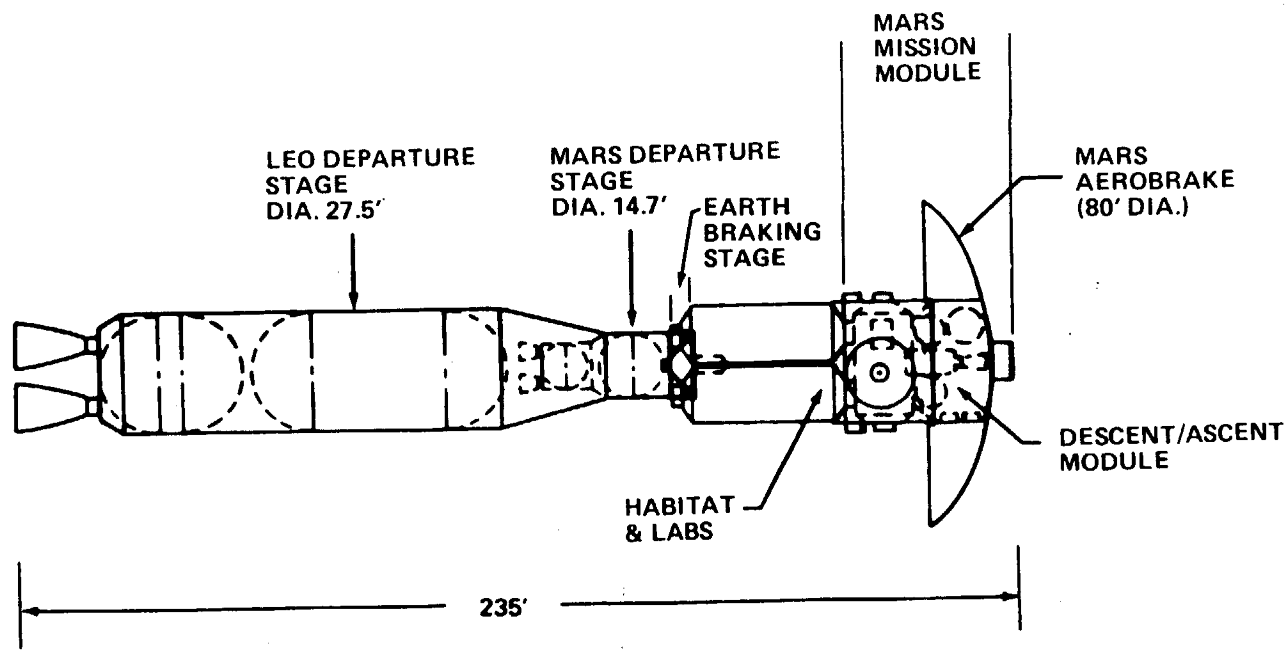
1999 OPPOSITION MANNED MARS MISSION AEROBRAKE OPTION



EARTH DEPARTURE VEHICLE

FIGURE 1

Equipment; (6) Training Hardware and Facilities; (7) Mission Control and Communications Network.

Launch site facilities were not covered, since it is assumed that these facilities will be in existence from other national space booster programs such as SDV, SDI, etc. In the time allocated for this study, schedule information for a large number of vehicle configurations was not developed nor would it have been desirable to do so at this time. Instead, one vehicle was selected that seemed most feasible to be developed by the 1997/1988 (start LEO operation 1997 and start Earth departure 1998) time period.

SCHEDULES

(Refer to Figures 2 through 8)

METHODOLOGY

Nine flights of the SDV-3R vehicle are required to place all hardware/equipment, including propellant into Low Earth Orbit (LEO). These flights will be scheduled as required to optimize the assembly at LEO. The SDV hardware to be recovered from these flights are the Propulsion/Avionics (PA) modules and the Solid Rocket Boosters (SRBs), however, there is insufficient time for these hardware items to be refurbished and reused for the remaining SDV-3R flights on the same manned Mars mission. An airplane will be leased to return the PA modules from the recovery site to the refurbishment site.

Five STS missions are planned for the crew during the manned Mars mission. Included are two assembly crew rotations required in LEO plus the placing of the flight crew into orbit and return. An OTV flight will be made available for rendezvousing with the manned Mars vehicle upon Earth return and transferring the crew to LEO to rendezvous with the Space Station or the Orbiter.

The power subsystem for the mission has not be selected. If a nuclear or isotope system is selected it will be imperative that early go-ahead be given for definition studies as these systems require very long lead times. (Refer to Table 1 for estimate of development time for various power systems.)

Orbital assembly of the space vehicle could require up to a year in duration, however, LEO assembly has not been totally assessed, therefore, this ample schedule cushion has been included. If this assembly is

MANNED MARS MISSION
ROCKET VEHICLES

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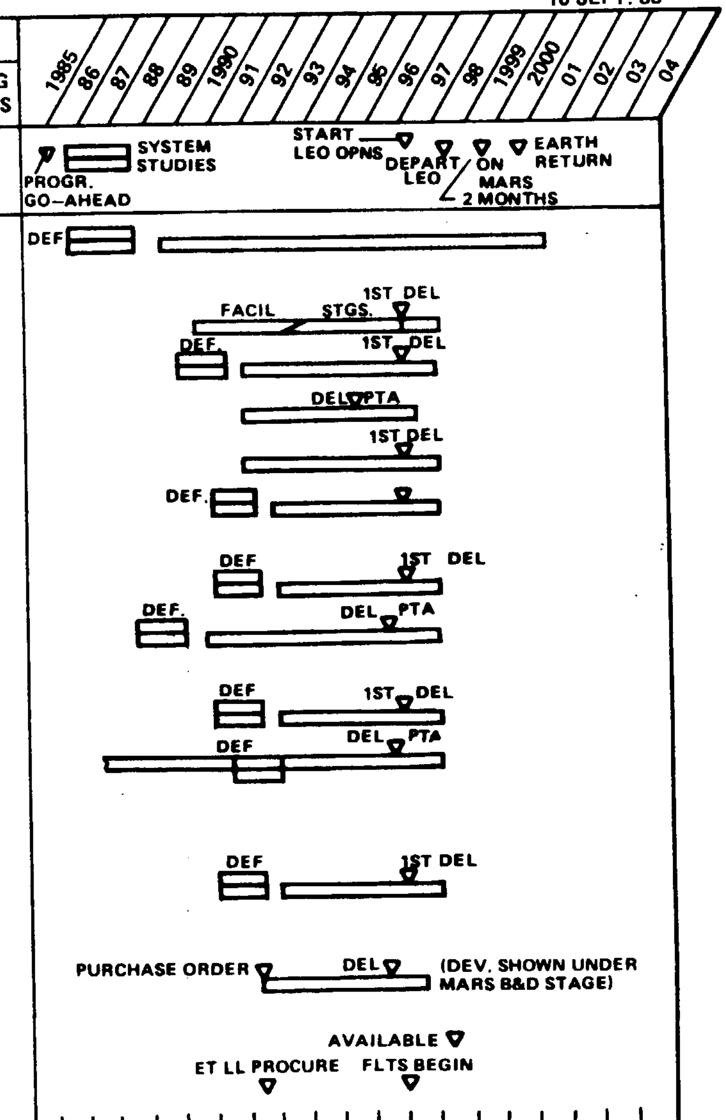
MAJOR HARDWARE ELEMENTS	HWD. QUANTITY: FLIGHT UNITS	GSE SETS	SOFT-WARE SETS	TESTS UNITS	TRNG UNITS
MAJOR MILESTONES					
● SYS ENGR & INTEGR					
● EARTH–TO–ORBIT VEH SDV–3R					
– ET	9	0	0	0	0
– P/A MODULE	9	3	3	2	0
● 3 STME'S	27	6	6	0	0
– SRB'S (DEVEL. EXIST)	18	6	0	0	0
– SHROUD	9	3	0	2	0
● LEO DEPARTURE STAGE					
– VEHICLE	1	2	1	2	0
– CHEM PROP (STME)	2	2	2	1	0
● MARS DEPARTURE STG					
– VEHICLE	1	1	1	2	0
– CHEM PROP (RL100)	2	1	1	2	0
● EARTH BRAKING STAGE					
– VEHICLE	1	1	1	1	0
– CHEM PROP (RL100)	2	1	1	2	0
● OTV (ASSUME EXISTING IN 1998)	~				
● STS FLIGHTS	(RESERVE		5 FLIGHTS)		



TEST AND TRAINING UNIT DELIVERIES PRECEDE THE FLIGHT UNIT DELIVERIES SHOWN ABOVE.

FIGURE 2

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MANNED MARS MISSION SPACECRAFT

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MAJOR HARDWARE ELEMENTS	HWD. QUANTITY: FLIGHT UNITS	GSE SETS	SOFT-WARE SETS	TESTS UNITS	TRNG UNITS	1985 86 87 88 89 1990 91 92 93 94 95 96 97 98 1999 2000 01 02 03 04
MAJOR MILESTONES						PROGR. GO-AHEAD; SYSTEM STUDIES; START LEO OPNS; DEPART LEO; ON MARS 2 MONTHS; EARTH RETURN
● MARS MISSION MODULE	1	1	2	1	2	DEF; 1ST DEL
● MARS EXCURSION MODULE (MEM)						
– LANDER	1	1	2	2	2	DEF; 1ST DEL
● DEORBIT PROPUL. (EXIST STAR 30B)	7	2	0	3	0	1ST DEL
● DESCENT PROPUL.	2	2	3	2	0	DEF; DEL PTA
● MODULE AND AEROBRAKE SHIELD	1	1	0	0	0	P.O.; DEL
– ASCENT						
● MODULE	1	1	2	2	2	COMBINE WITH LANDER PROCUREMENT
● ASCENT PROPUL.	1	~	~	~	~	THIS IS THE MIDDLE DESCENT ENGINE – ALSO USED FOR DESCENT
● INTEGR. SPACECRAFT VEH.	1	1	1	1	1	COMBINE WITH MISSION MODULE PROCUREMENT
– POWER SUBSYSTEM						
● MISSION MODULE	1	1	2	1	0	DEL *
● MARS EXCURSION MODULE	1	1	2	1	0	DEL *

* EXTENSIVE TIME WOULD BE REQ'D FOR NUCLEAR REACTOR, ISOTOPE DYNAMIC, SOLAR THERMAL AND OTHER NEW TECHNOLOGY SYSTEMS.

TEST AND TRAINING UNIT DELIVERIES PRECEDE THE FLIGHT UNIT DELIVERIES SHOWN ABOVE.

FIGURE 3

PP02/ROBINSON
7 JUNE 85

MANNED MARS MISSION
SDV–3R PAYLOAD ADAPTERS

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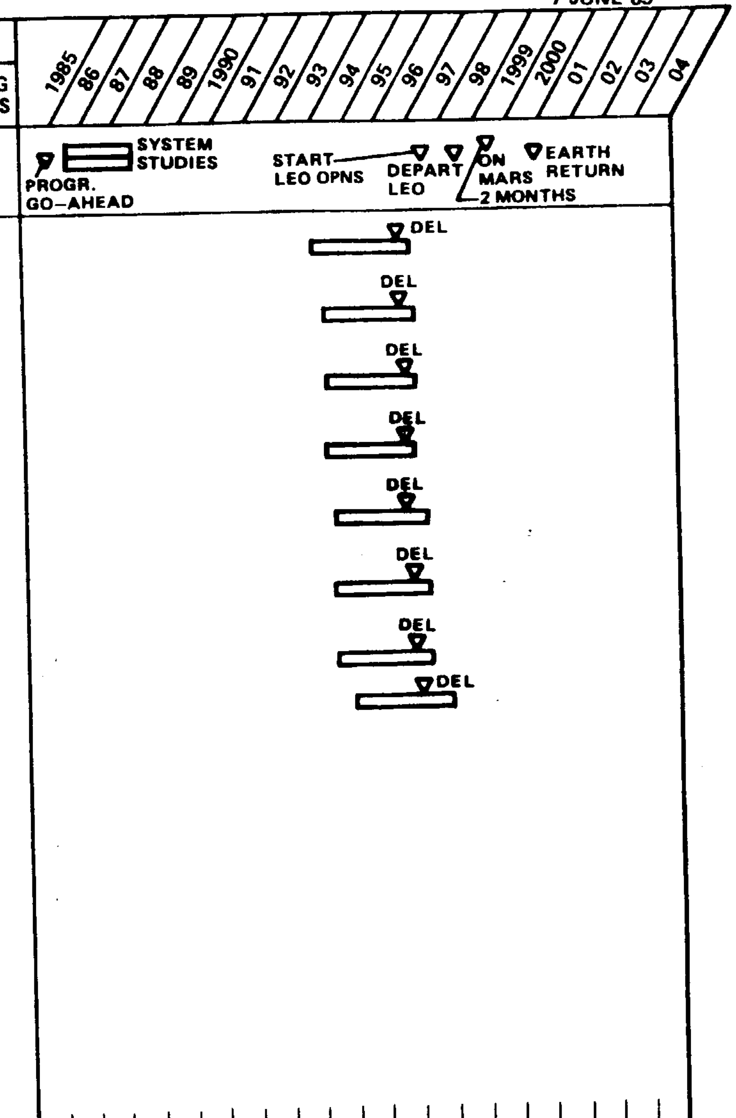
MAJOR HARDWARE ELEMENTS	QUANTITY: FLIGHT UNITS	GSE SETS	SOFT-WARE SETS	TESTS UNITS	TRNG UNITS
MAJOR MILESTONES					
● LEO ASSY STATION EQUIP ADAPTER	4	2	0	2	0
● PROPELLANT FARM EQUIP ADAPT	6	3	0	3	0
● EARTH DEPARTURE STG ADAPT	1	1	0	1	0
● SPACECRAFT STRUCTURE ADAPT	1	1	0	1	0
● VARIOUS MODULES WITHIN S/C ADAPTER	6	3	0	3	0
● MARS DEPARTURE STG. ADAPTER	1	1	0	1	0
● EARTH BRAKING STG. ADAPT	1	1	0	1	0
● PROPELLANT TRANSFER TANKS ADAPTERS	4	1	0	2	0



TEST AND TRAINING UNIT DELIVERIES PRECEDE THE FLIGHT UNIT DELIVERIES SHOWN ABOVE.

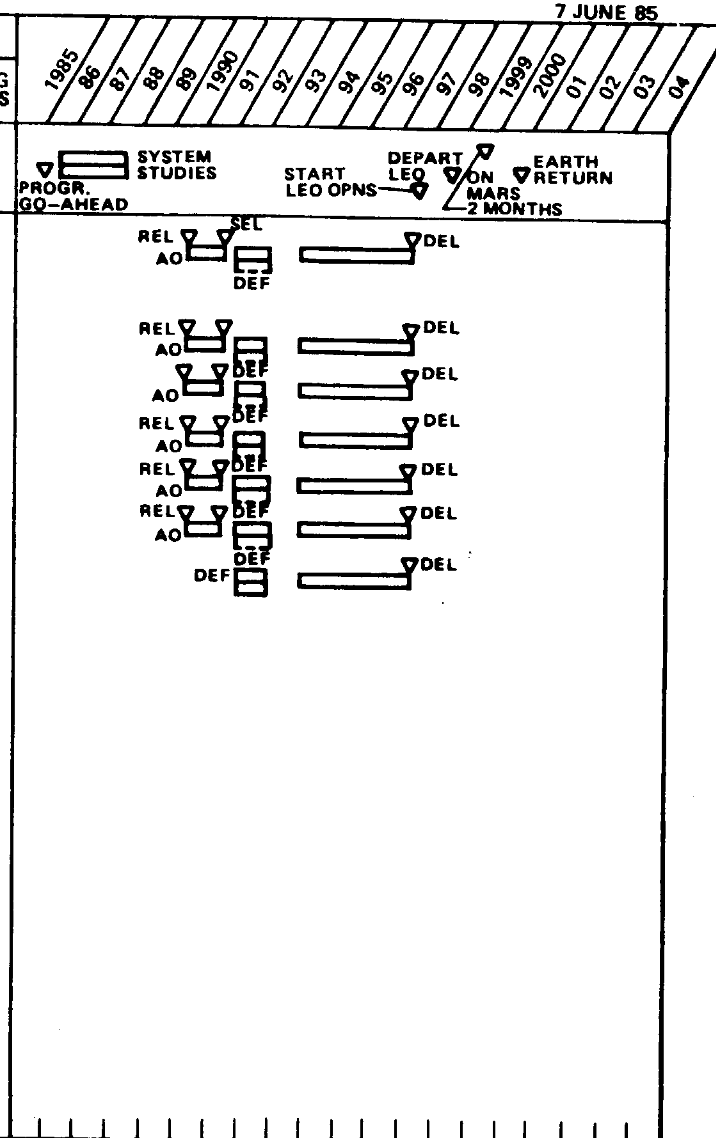
FIGURE 4

MANNED MARS MISSION EXPERIMENTS

641–85

PP02/ROBINSON
7 JUNE 85

MAJOR HARDWARE ELEMENTS	HWD. QUANTITY				
	FLIGHT UNITS	GSE SETS	SOFT-WARE SETS	TESTS UNITS	TRNG UNITS
MAJOR MILESTONES					
● VENUS PROBES	4	4	8	4	0
● MARS MOON PROBES:					
– PHOBOS	2	2	4	2	0
– DIEMOS	2	2	4	2	0
● MARS PROBES	1	1	2	3	0
● MISSION MODULE EXPERI.	100	5	10	50 SOME →	50
● MARS SURFACE EXPERI.	100	5	10	50 SOME →	50
● ROVING VEHICLE	1	1	2	1 →	1



TEST AND TRAINING UNIT DELIVERIES PRECEDE THE FLIGHT UNIT DELIVERIES SHOWN ABOVE.

FIGURE 5

MANNED MARS MISSION
LEO ASSEMBLY EQUIPMENT

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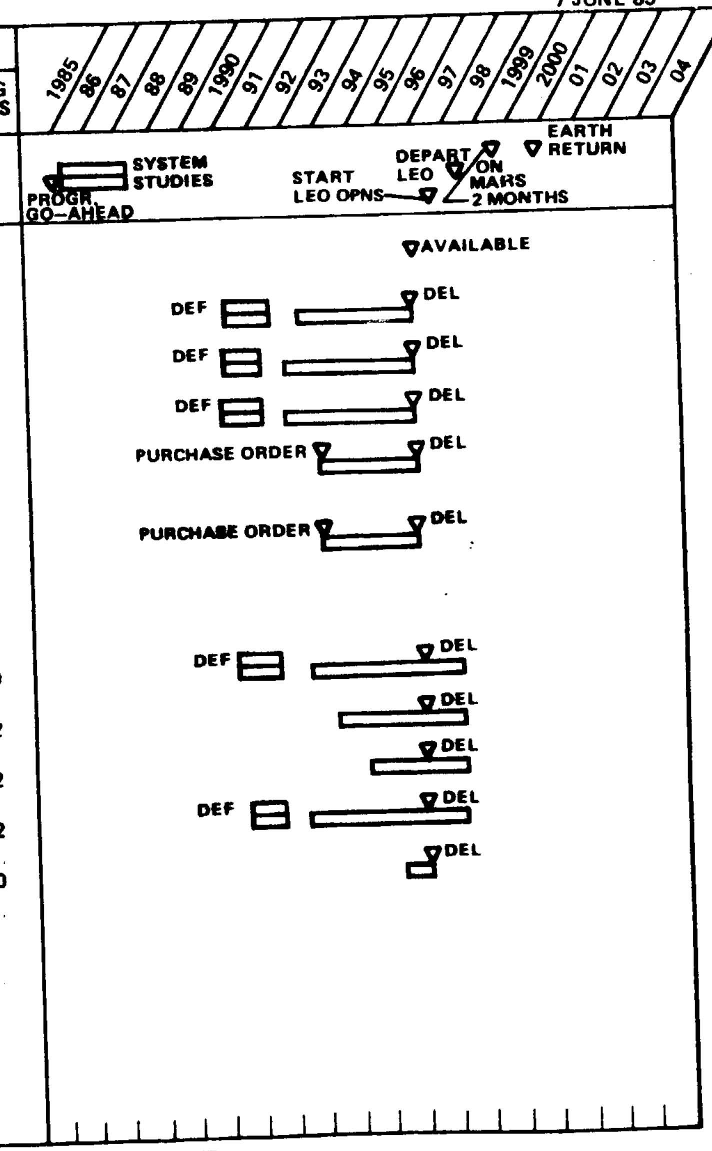
640–85

MAJOR HARDWARE ELEMENTS	HWD. QUANTITY: FLIGHT UNITS	GSE SETS	SOFT–WARE SETS	TESTS UNITS	TRNG UNITS
MAJOR MILESTONES					
● OMV	~	~	~	~	~
● ASSEMBLY FIXTURE	1	1	0	1 →	2
● CHERRY PICKER	1	1	1	1 →	2
● CONTROL MODULE	1	1	1	1 →	2
● HABITATION MODULE (SS DEVEL)	2	2	1	0	2
● LOGISTICS MODULE (SS DEVEL)	1	1	1	0	1
● PROPELLANT FARM:					
– TANKS	2	2	0	1	0
– PUMPS	2	2	0	1 →	2
– PLUMBING	1	2	0	1 →	2
– REFRIG & CONTROL S/S	2	2	1	1 →	2
– PROPELLANTS	1.4 M LB.	INCL. IN CRYO PLANTS	0	1	0



TEST AND TRAINING UNIT DELIVERIES PRECEDE THE FLIGHT UNIT DELIVERIES SHOWN ABOVE.

FIGURE 6

646–85

MANNED MARS MISSION
TRAINING HARDWARE AND FACILITIES

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7 JUNE 85

MAJOR HARDWARE ELEMENTS	HDW. QUANTITY: HDW UNITS	GSE SETS	SOFT-WARE SETS
MAJOR MILESTONES			
● LEO ASSY STATION TRNG			
– ENGR MODEL	1	1	1
– NEUT BOUY MOCKUP	1	0	0
● SPACECRAFT TRNG			
– OVERALL SPACECRAFT			
● 1–G MOCKUP	1	1	1
● N/B MOCKUP	1	0	0
– MISSION MODULE			
● 1–G MOCKUP	1	0	0
● N/B MOCKUP	1	0	0
– MEM			
● 1–G MOCKUP	1	0	0
● N/B MOCKUP	1	0	0
● ROVING VEH ENGR MODEL	1	1	1
● EXPERIMENTS			
– N/B MOCKUPS	40	0	0
– ENGR. MODELS	40	10	20

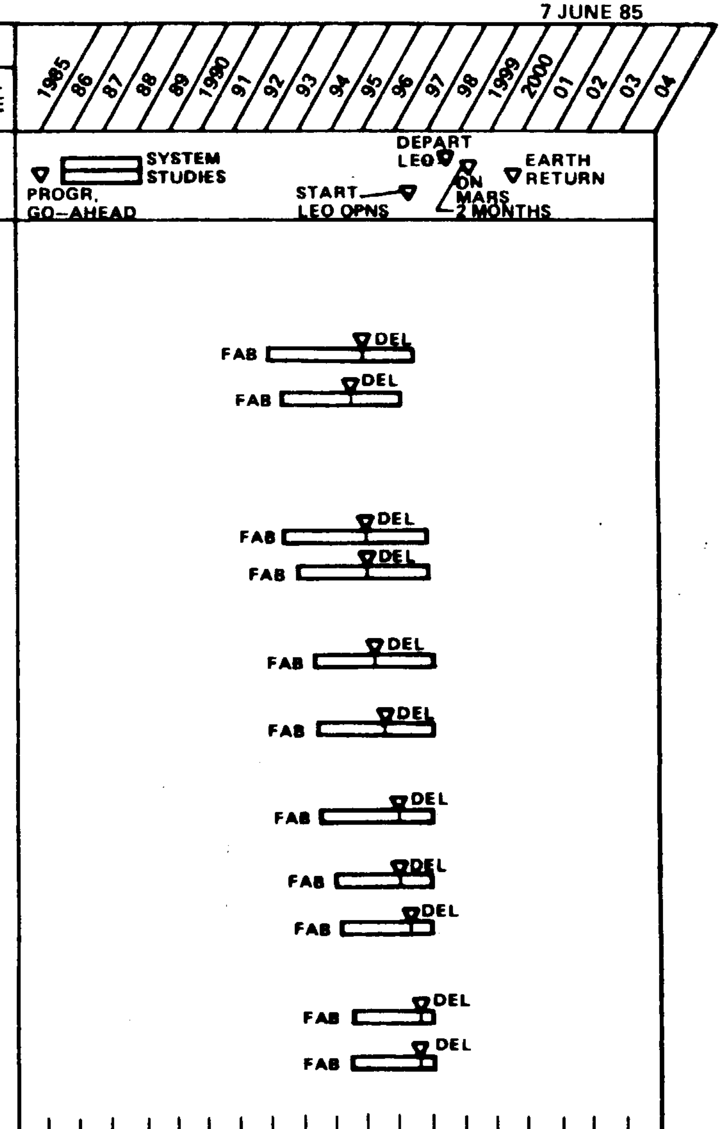

FIGURE 7

644–85

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7 JUNE 85

MANNED MARS MISSION
MISSION CONTROL & COMMUNICATIONS NETWORK

MAJOR HARDWARE ELEMENTS	HDW. QUANTITY: FACIL UNITS	GSE SETS	SOFT-WARE SETS	TESTS UNITS	TRNG UNITS
MAJOR MILESTONES					
● MISSION CONTROL CENTER	1	1	3	0	0
● COMMUNICATIONS NET					
– LEO OPERATIONS; 2ND GENERATION TDRS (ASSUME EXISTS BY 1996)	2	1	1	0	0
– DEEP SPACE; MODIFY JPL NETWORK	1	1	1	0	0

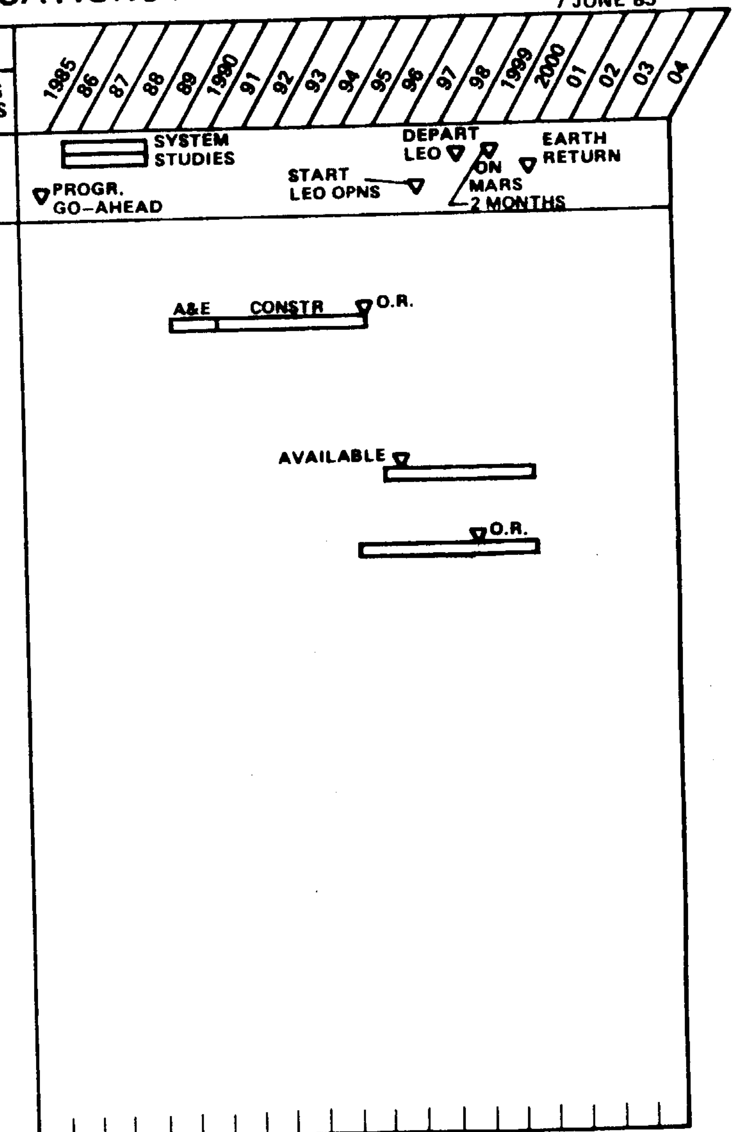

FIGURE 8

accomplished away from the Space Station, a control module will serve as a work station for the astronauts. It is anticipated that a multitude of operations will be required. These operations may utilize mechanized arms, robotics, and the OMV, and will require extensive EVA activities. The mission modules could serve as the habitation module for the astronauts during the assembly period.

Experiment operations could begin in LEO during assembly operations and continue until Earth return.

Extensive training hardware will be required for the manned Mars mission. In each applicable category of hardware, test and training unit deliveries will precede the flight unit deliveries.

ASSUMPTIONS

- No test flights are planned prior to a manned landing.
- Optimum Mars launch windows occur on approximately 2 year intervals.
- SDV-3R vehicle, manufacturing/test facilities and launch facilities development are not planned under this program (assume previous development).
- All flights planned in support of the manned Mars mission are in addition to the STS Program of 24 flights per year.
- Test and training unit deliveries precede the flight unit deliveries shown above.
- OTV is assumed to be in existence and available by 1997 (includes aerobraking shield).
- OMV is assumed to be in existence and available for assembly operations by 1997-98.
- Habitation and logistics modules used for LEO assembly will be copies of then existing SS modules.
- LEO assembly equipment is independent of existing Space Station equipment.
- The launch vehicle SE&I contractor will also be responsible for the payload adaptors, vehicle GSE, vehicle software and vehicle integration hardware.
- Existing neutral buoyancy facilities are adequate with judicious scheduling.

RISK ASSESSMENT

With the assumptions previously listed, the SDV-3R vehicle schedule should represent only a minimum risk. Early go-ahead is required for definition studies for the Space Transportation Main Engine (STME) for the Earth LEO departure stage and the RL100 engine for the Mars departure stage. Per previously mentioned assumptions, the OTV with its aerobraking shield and the OMV will have been developed by the time of the manned Mars mission. Early go-ahead is required for power systems for the spacecraft, particularly if nuclear or isotope systems are to be utilized.

CONCLUSION

In conclusion, it appears realistic from a schedule standpoint that a pre-2000 manned Mars mission is possible. However, it will be imperative that early go-ahead with adequate funding authorization be given so that the necessary planning and definition studies can be initiated for the long lead hardware.

TABLE 1

POWER SUBSYSTEMS

POWER SUBSYSTEM OPTIONS	DEVELOPMENT TIME
Mission Module (MM)	
o Photovoltaic	5-7 years
o Solar Thermal	10 years
o Nuclear Reactor	10 years
o Isotope Dynamic (DIPS)	10 years
o Regenerative Fuel Cell	5-7 years
Mars Excursion Module	
o Photovoltaic	5-7 years
o Solar Thermal	10 years
o Isotope Dynamic (DIPS)	10-12 years
o Open Loop Fuel Cell	5-7 years
o Nuclear Reactor	
- Multi-Hundred Watt	10 years
- General Purpose HS	10 years
- Hydride Reactors	10-12 years
- SP-100	10 years
o Laser/R-F Transmission	12 years
o Photovoltaic + Regen. Fuel Cell + Isotope	10 years
o Photovoltaic + DIPS	7-10 years
o Multi Megawatt	15 years

SECTION IX

IMPACTS ON OTHER PROGRAMS

N87-17803

AUTOMATION, ROBOTICS, AND INFLIGHT TRAINING FOR MANNED MARS MISSIONS

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ABSTRACT

The automation, robotics, and inflight training requirements of manned Mars missions will be supported by similar capabilities developed for the Space Station program. Evolutionary Space Station onboard training facilities will allow the crewmembers to minimize the amount of training received on the ground by providing extensive onboard access to system and experiment malfunction procedures, maintenance procedures, repair procedures and associated video sequences. Considerable on-the-job training will also be conducted for Space Station systems management, mobile remote manipulator operations, proximity operations with the Orbital Maneuvering Vehicle (and later the Orbital Transfer Vehicle), and telerobotics and mobile robots. A similar approach could be used for manned Mars mission training with significant additions such as high fidelity image generation and simulation systems such as holographic projection systems) for Mars landing, ascent , and rendezvous training. In addition, a substantial increase in the use of automation and artificial intelligence for systems management and in the use of automation and robotics for hazardous and tedious tasks would be expected for Mars missions. Mobile robots may be used to assist in the assembly, test and checkout of the Mars spacecraft, in the handling of nuclear components and hazardous chemical propellent transfer operations, in major spacecraft repair tasks which might be needed (repair of a micrometeroid penetration, for example), in the construction of a Mars base, and for routine maintenance of the base when unmanned.

INTRODUCTION

The manned Mars missions will require a substantial implementation of automation, robotics, and inflight training capabilities. The Space Station program is planning to incorporate a considerable amount of automation and robotics(A&R) into the Station and platform systems. In addition, extensive onboard training capability is being planned. Space Station program development and later operations should provide a good "test bed" for assessing Mars mission requirements for A&R and inflight training. The need for advanced development activities can be based on these assessments.

SPACE STATION AUTOMATION AND ROBOTICS PLANS

The Space Station automation and robotics (A&R) implementation is being emphasized in part because of the special mandate Congress has given to NASA. Congress has asked NASA to use the Space Station program to advance the state-of-the-art in application of A&R. In addition, NASA will give emphasis to A&R that will also support the needs of U.S. industry. NASA will brief industry and user representatives on the preliminary plans for A&R technology implementation and solicit comments and suggestions on these plans.

A draft of A&R priorities based on the needs of Space Station users has been developed (Ref 3). Later, a preliminary list of A&R technologies being considered for implementation in the Space Station program will be generated for user and U.S. industry review. Congress has provided additional funding (augmented funding program) to enable the development of a telerobotic system or robotic servicer. As part of this A&R development activity, NASA is evaluating plans for ground and flight demonstrations aimed at the development of the Initial Operating Capability(IOC) Station and the growth phase.

SPACE STATION FLIGHT TRAINING AND SIMULATION PLANS

The Space Station program has as a goal an evolution from extensive ground training to a minimal of ground training and extensive use of onboard training facilities and on-the-job training. The onboard training facilities should be capable of using the same computer disks used in ground training, with the exact same crew interfaces. Crewmembers with extensive ground training or previous Space Station experience will conduct on-the-job training, as required, for other crewmembers.

Only a select number of system and experiment malfunctions will be trained on in ground training facilities. The onboard training system will be capable of storing malfunction diagnostic procedures, repair procedures and video demonstrations of repair tasks for a very large group of potential malfunctions and contingencies. The training system may utilize voice actuated commands, optical disks, expert systems and artificial intelligence for interactive training.

The evolutionary Space Station onboard training system may be capable of placing any CRT terminal into an off-line training mode. The actual flight software and current systems parameters statuses can then be used in conjunction with selected malfunctions for training on malfunction procedures. Integrated contingency simulations (involving one or two crew members and ground controllers) for fire, micrometeroid penetration, release of toxic gases in the Station atmosphere, major system failures and collisions of vehicles would be conducted infrequently. The integrated simulations will keep the crew proficient on quick reaction procedures and will assist in verifying procedure changes uplinked from the ground. An occassional full-up "fire drill" would be conducted for the entire crew.

Training on the Mobile Remote Manipulator System, Orbital Maneuvering System, and EVA operations would be conducted primarily on the ground. On-the-job training would be used to supplement the ground training.

MARS SPACECRAFT ASSEMBLY AND CHECK OUT

Automation and Robotics

The Orbital Maneuvering Vehicle with a smart front end and the Orbiter or Servicing Platform Remote Manipulator System will be the primary means for assembling major components of the manned Mars Spacecraft. The connection of power, fluid, and data lines and the connection and deployment of smaller structures would require the use of Space Station EVA crewmembers. A combination of special EVA crewmembers dedicated to Mars spacecraft assembly and mobile robots (with telerobotics capability) will likely be used for these tasks. (See Figure 1).

The handling and checkout of nuclear power and propulsion systems before and after the first mission and other hazardous operations may necessitate the use of a mobile robotics capability which exceeds the capability of the OMV smart front end. A multi-arm robot which can grasp a structure and maintain an arbitrary position may be needed. The robot should have multiple sensors including vision in the visual and infrared range, vibration sensitivity (from 1 to 100 Hz, for example), and a laser ranging and attitude system.

Assembly and repair operations will require a versatility in manipulative tools for a mobile robot including power rotating

SERVICING PLATFORM AND MOBILE ROBOTS FOR MANNED MARS MISSION SPACECRAFT ASSEMBLY

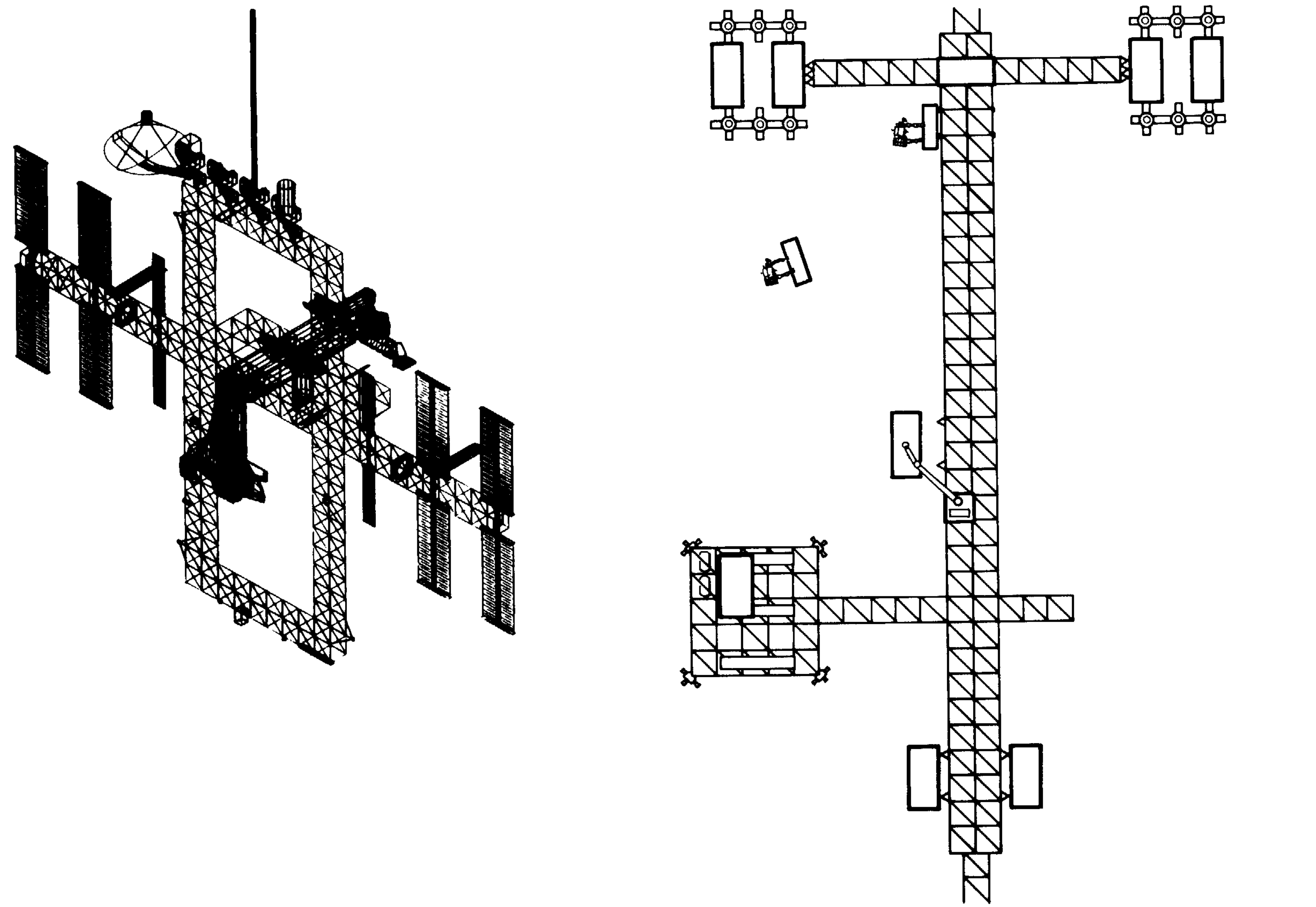

FIGURE 1

tools (angular momentum compensated tools), grasping end effectors, push and pull end effectors, and possibly a laser holographic diagnostic system for the detection of structural and surface faults. Extensive attitude control and translation capability for free flight maneuvers and small payload ferrying activities will be required.

Inflight Training and Simulation

During the assembly and checkout phases, the Space Station and later the Mars Mission Modules should have onboard training capability which supports the review of malfunction and repair procedures which may be required to overcome a problem which develops. Video presentations should be available for those problems which have a reasonably high probability of occurrence.

Malfunctions which occur during a checkout phase with a critical system such as a nuclear power system could require a crewmember to stop and review malfunction procedures associated with the new system configuration. This capability to review malfunction procedures should be available at the terminal being used for that portion of the checkout procedures.

As a goal, the Space Station will be designed such that failures in non-critical systems could be diagnosed by expert system software and, in some cases, a switch to a backup system could be executed without a crewmember's involvement. Depending on the type and frequency of the fault, the crewmembers could be immediately notified by an alarm or computer voice generation system. Criteria will have to be developed which help crewmembers determine when an expert system or artificial intelligence software can be relied on to handle a new system configuration or new system performance.

INTERPLANETARY OUTBOUND AND INBOUND ACTIVITY

Mars Spacecraft Automation and Robotics

Nominal spacecraft systems management should be conducted by expert systems and/or artificial intelligence software. Malfunctions and degraded performance trends should be identified and tracked by automated systems software. As a minimum, recommendations for corrective actions should be made by the software.

Criteria will have to be developed for determining the conditions and systems' types for which the expert or artificial intelligence systems would be allowed to take independent action. For certain emergency situations, it may be very helpful for the system to take immediate action, for example to minimize the impact of a fire.

For checks of external structures and external repairs, some form of mobile robotics may be required. Consideration should be given to providing a portable laser system for use in surface inspection and minor welding tasks. For external checks of nuclear components, remote viewing by an EVA crewmember with a Manned Maneuvering Unit may not be effective. A mobile robot which is stored outside could be used for a variety of purposes including the investigation of micrometeroid hits (detected by the robot's sensors or other sensors) and the repair of damaged areas, if required.

Mars Spacecraft Inflight Training and Simulation

The onboard training facility must be capable of supporting a great range of malfunction and repair training. Video presentations recorded during ground construction and checkout activities should be available for playback to support training using interactive expert system or artificial intelligence software. Special video and digital training materials should be available to provide training in crewmember fields-of-expertise, which may be related to experiments being conducted during the interplanetary phases or during Mars surface activities.

The training system should support procedures and malfunction training for inflight scientific experiments and technology demonstrations. Training for repairs of artificial intelligence and robotics systems would also be provided.

MARS ORBIT INSERTION, LANDING, LIFTOFF, AND RENDEZVOUS

Mars Spacecraft Automation and Robotics

Automation (and artificial intelligence) will be used substantially for systems' management and flight control for the Mars landing, ascent, and spacecraft rendezvous phases. Contingencies associated with these phases can be addressed by artificial intelligence software but they must also be integrated with pilot assessments as well.

The Mars Excursion Module or landing module may use techniques such as aerodynamic braking and other active flight control processes which will require the use of expert systems. The expert system may use laser ranging system navigational data as part of its logic processes.

Inflight Training and Simulation

Crew training on Mars landing, ascent, and rendezvous phases will be a primary focus of inflight training and simulation capabilities. Training on these phases could be accomplished in the Mars Excursion Module (MEM) if EVA access is not required to enter the Module.

High fidelity simulations could be accomplished by the use of holographic image projection systems which would project out-the-window views onto to the windows of the MEM. The same system could be used to provide video images for closed circuit TV screens to simulate external viewing of TV cameras. Off-line training modes could be used with MEM CRT terminals, MEM software and other controls and displays to provide training on various malfunctions and contingencies.

To use the MEM in a training mode will require redundancy in the safeguarding of the flight software to insure that simulated faults do not affect the actual flight software. Backup computers and software might be used for the training mode to provide the additional safeguarding.

Lower fidelity training for these phases could be accomplished using flight type controls and displays and software at a location in a Mission Module. Even for a largely automatic system it will be necessary to conduct extensive inflight training for landing and liftoff phases. Periodic training should be conducted throughout the outbound leg for these phases with more extensive training conducted during the 3 weeks prior to Mars orbital insertion.

MARS SURFACE ACTIVITIES

Automation and Robotics

A mobile robot would be very useful to have on the surface and could be used to assist in Mars base construction tasks, maintance and repair of nuclear power systems, external tasks required during periods of high solar flare radiation, experiment setup and deployments, area reconnaisance and Mars base management during the unmanned period. (See Figure 2). Other robotics would be useful on a Mars rover for sample collection and remote exploration using telerobotics.

Telerobotics modes for the mobile robot might enable many Mars base management tasks to be conducted during the unmanned phases including some repairs. Automation would be expected for the base systems

HUMANOID ROBOT ON MARS CONTROLS LANDING SEQUENCE OF UNMANNED LOGISTICS TRANSPORT

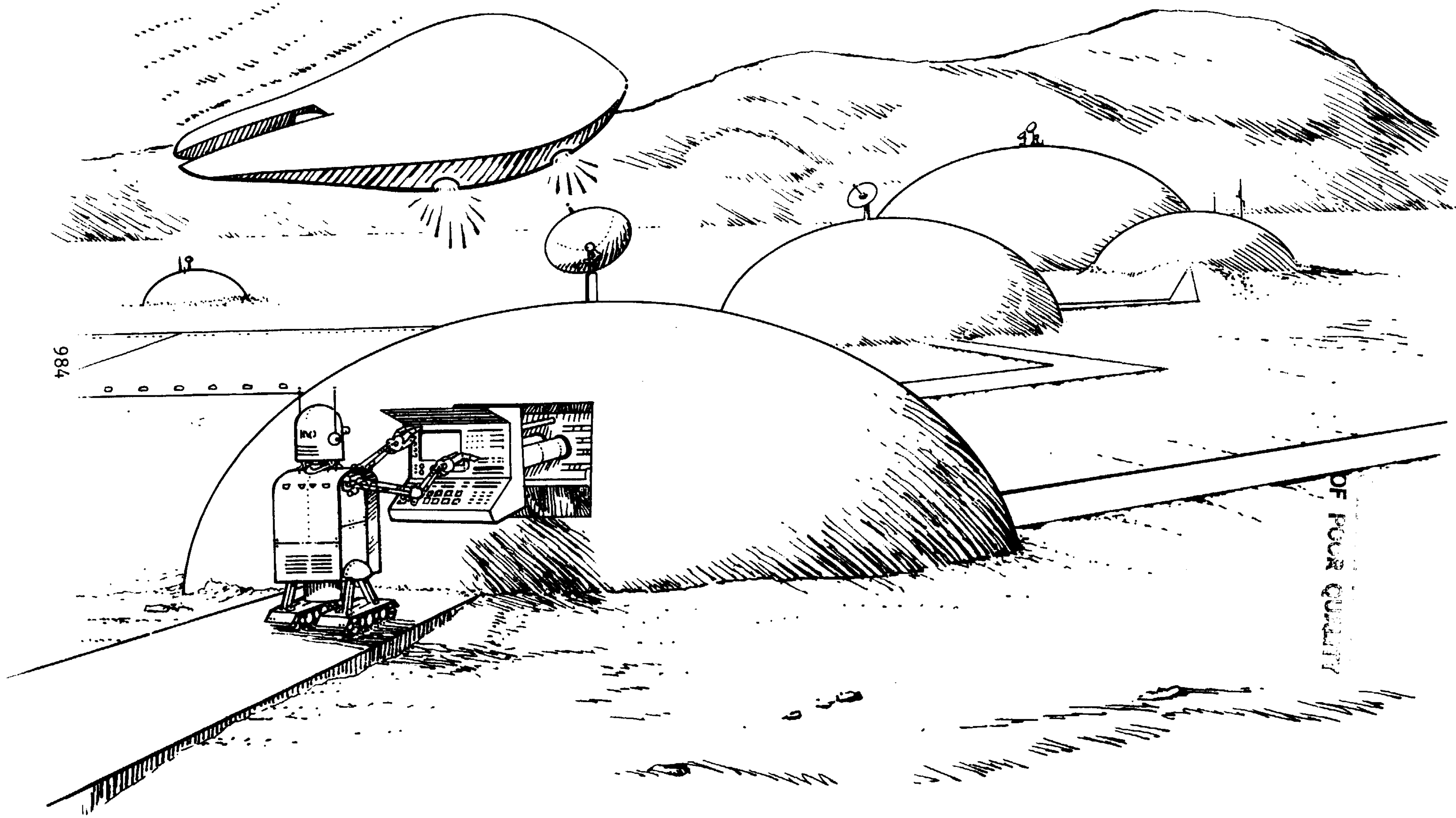

FIGURE 2

management similar to that used on the Mars spacecraft. Considerable attention should be given to the possible use of automation and robotics in support of surface activities. Performance and cost trade studies for Mars landing craft and base systems are required to insure that robotics can be integrated where beneficial. These studies should consider a 10 to 30 year Mars base development plan.

<u>Inflight Training and Simulation</u>

Training on the surface would include training for ascent and rendezvous. Surface stay times would be long enough in almost all cases to warrant additional crew training. If a MEM, built in holographic image generation and simulation system is available, then the same system used prior to landing can be used for ascent and rendezvous training.

If the training system can not be integrated in this manner or if the stay time is relatively short, then special off-line CRT displays could be used to review key contingencies and the nominal ascent activities. A surface training system should be available for the MEM and/or Mars base which could be used to call up system and experiment malfunction and repair procedures for review. Some video presentations of repairs should be provided also.

EARTH ORBIT INSERTION AND SPACE STATION RENDEZVOUS

<u>Mars Spacecraft Automation and Robotics</u>

The Mars spacecraft should utilize a laser ranging and positioning system for rendezvous with the MEM, and rendezvous with the Space Station upon return, and subsequent station-keeping. This data should be used by onboard expert systems to verify and maintain the relative position of Mars spacecraft and the MEM and the Space Station and the Mars spacecraft.

<u>Mars Spacecraft Inflight Training and Simulation</u>

Some nominal and contingency training will be required to support Earth orbital insertion and Space Station rendezvous. Visual imagery would be helpful for final approach phases but is not mandatory.

Malfunction or contingency training should emphasize any area where systems failures or performance degradations have been substantial during the flight. Simulations in the off-line training mode will help assure that there are no hidden problems in rendezvous and transfer procedures.

CONTINUING STUDY RECOMMENDATIONS

- Identify and prioritize potential Manned Mars Mission A&R requirements.
- Identify Space Station A&R activities which should be monitored and influenced to support potential Manned Mars Missions A&R requirements.
- Identify inflight training requirements and technology which exceeds that expected to be used on the Space Station.
- Conduct a feasibility study of using the MEM flight hardware and software in a training mode for inflight landing and ascent training, and identify key technical questions.

REFERENCES

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2. Space Station Automation and Robotics Implementation Plan, Space Station Systems, Engineering and Integration Office, PB, NASA JSC, June, 1985.
3. Customer Integration Automation and Robotic Priorities, Space Station Customer Integration Office, PD4, NASA JSC, 1985.

N87-17804

LUNAR AND MARTIAN HARDWARE COMMONALITY

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ABSTRACT

A number of different hardware elements were examined for possible Moon/Mars program commonality. These include manned landers, cargo landers, a trans-Mars injection (TMI) stage, traverse vehicles, unmanned surface rovers, habitation modules, and power supplies. Preliminary analysis indicates that it is possible to build a common two-stage manned lander. A single-stage, reusable lander may be practical for the lunar case, but much less so for the Martian case, and commonality may therefore exist only at the subsystem level. A modified orbit transfer vehicle was examined as a potential cargo lander. Potential cargos to various destinations were calculated for a Shuttle external tank sized TMI stage. A nuclear powered, long range traverse vehicle was conceptually designed and commonality is considered feasible. Short range, unmanned rovers can be made common without great effort. A surface habitation module may be difficult to make common due to difficulties in landing certain shapes on the Martian surface with aerobraking landers. Common nuclear power sources appear feasible. High temperature radiators appear easy to make common. Low temperature radiators may be difficult to make common. In most of these cases, Martian requirements determine the design.

INTRODUCTION

NASA's post Space Station options may include a return to the Moon and/or a manned Mars program. It may be easier to do all or part of both of these programs at the same time if some hardware can be made common. Cost savings through commonality require both lunar and Martian programs underway within five years or so of each other. Programs separated in time by more than this are much less likely to benefit from commonality because of advance of the technological state of the art.

USE OF A TRANS-MARS INJECTION STAGE FOR OTHER MISSIONS

Recent studies of a manned Mars mission identify the need for a very large chemical propulsion stage which provides the first maneuver of the trans-planetary space vehicle (Ref. 1). A "conjunction class" mission carrying about 340 metric tons of mission module and Mars landing vehicles (two) required a "Trans-Mars Injection" stage propellant load near the capacity of the Shuttle external tank (ET)--about 640 metric tons of hudrogen/oxygen propellant--and needed the engine thrust provided by a single high expansion ratio variant of the Shuttle main engine.

This led to the conceptual synthesis of a stage which was assembled and checked out on Earth, launched into the Space Station orbit as the Shuttle ET (i.e., using its propellant to power the STS), placed into LEO by a direct insertion ascent profile, then reconfigured and refueled at the Space Station. This concept has the advantage of eliminating the effort otherwise needed in LEO to assemble and test a modular tankage vehicle of the same class.

The possible utility of this large space propulsion vehicle for missions to lunar orbit, the lunar surface, and the several candidate future missions between the (500 km) Space Station orbit and GEO-stationary orbit is explored here in a tentative way. Figure 1 shows the original manned Mars concept and the modified TMI stage in a lunar lander configuration.

<u>TMI Stage Mass Properties and Engine Performance</u>

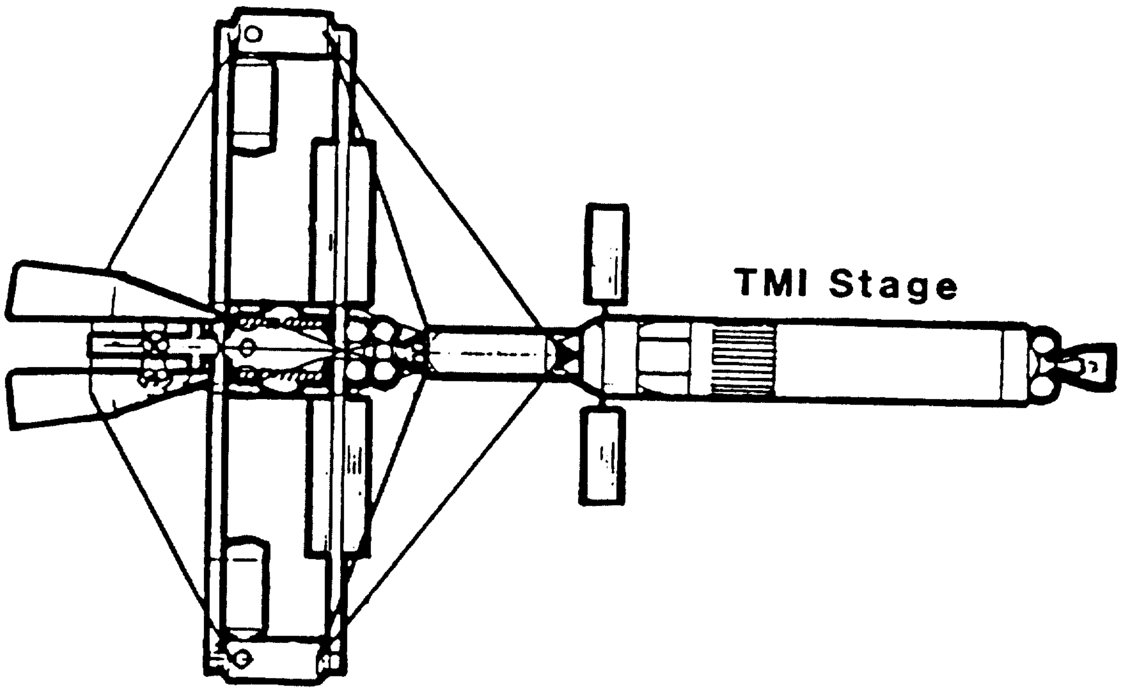
The elements and their estimated masses (Ref. 2) and the main propulsion system (MPS) engine performance are shown in Table 1.

<u>TMI Stage Mission Performance</u>

Velocity increments and mission performance for several all-propulsive missions are shown in Tables 2 and 3, respectively. The performance for each of these missions was calculated only for the case where the initial payload was the same as the payload for all subsequent mission phases. Obviously, other mission/payload combinations and partial propellant loading are possible. Additionally, full or partial re-use of the stage can be accomplished, rather than expending the stage. It may prove preferable to recover only the propulsion/avionics components and replace the propellant container for each mission with a once-used ET.

The use of this means of transporting mass from the Space Station would require a large-scale space program to provide enough mission demand to justify the initial investment and to develop the logistics capability to modify the stage in orbit and reload it with propellant. A lunar surface base is one type of program that might require its large

Figure 1 - TMI Stage
Original Manned Mars Spacecraft Concept



Modified TMI Stage Landing 175 MT on Lunar Surface

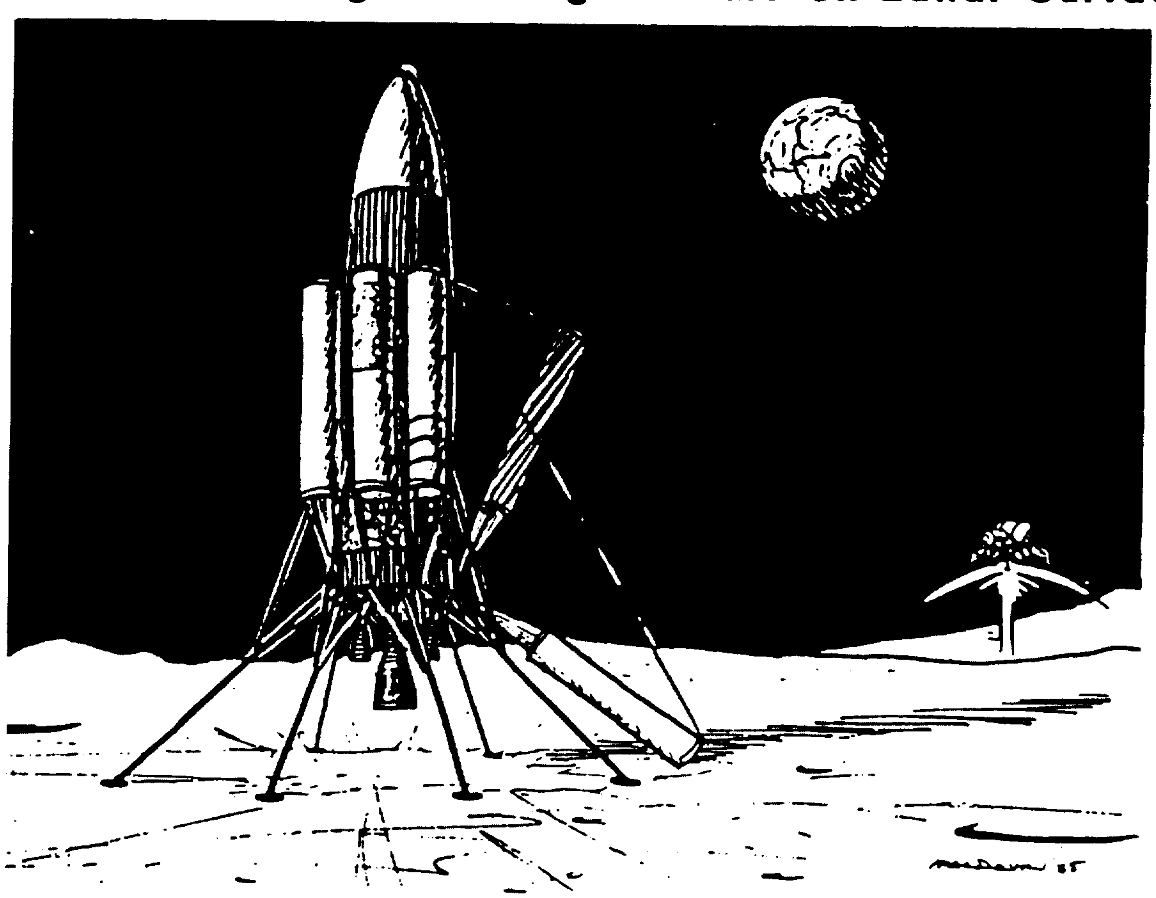


TABLE 1 - TMI STAGE OTV MASS PROPERTIES

Subsystem	Basic Mass launched	OTV Added in space	Total
"Light weight" ET	30,840		30,840
OTV MPS--1 SSME, high e	3,400		3,400
2 RL10's for roll control	340		340
Basic avionics suite	540		540
Propellant lines & valves	680		680
Helium pressurant system	910		910
Thrust truss	1,000		1,000
Attitude control sys. (dry)	410		410
Pyrotechnics & separation	230	50	280
Passive thermal control	540	1,810	2,350
25% Reserve--new items	1,080	460	1,540
Subtotal--dry mass	39,980	2,320	42,300
Unusable fluids			
Helium pressurant		680	680
Attitude control residuals		110	110
MPS residuals @ 1/4 %	1,770		1,770
Flight performance reserves	1,770		1,770
Subtotal	3,540	790	4,330
STAGE END--BURN MASS			46,630
USABLE FLUIDS			
MPS propellant		707,600	707,600
Attitude control propellants		2,270	2,270
Fuel cell reactants		540	540
Purge helium		1,130	1,130
Subtotal--consumables		711,540	711,540
START BURN MASS			758,170
"Mass Fraction"			0.933

Main propulsion system steady-state I_{sp} 470 seconds
Mission effective specific impulse 467.4 seconds

TABLE 2 - TMI STAGE OTV PROPULSIVE MANEUVERS, M/SEC

MISSION	#1	#2	#3	#4	#5	Total
Translunar Insert.	3,200					3,200
LEO - Lunar Orbit	3,200	30	1,070			4,300
LEO - L. Surface	3,200	30	1,070	90	2,040	6,430
LEO - LO - LEO	3,200	30	1,070	1,070	3,230	8,600
LEO - GTO	2,530					2,530
LEO - GEO	2,530	30	1,710			4,270
LEO - GEO - LEO	2,530	30	1,710	1,710	2,560	8,540
Planetary	3,200	3,200				6,400

LEO--Low Earth Orbit, GEO--Geosynchronous orbit, GTO--Geo-transfer orbit

TABLE 3 - TMI STAGE OTV MISSION PERFORMANCE

MISSION	Total dv	Mass Ratio	MR-1	Final Mass	Payload	Notes
TL Insert	3,200	2.011	1.011	700,090	653,460	
LEO - LO	4,300	2.555	1.555	455,100	408,460	
LEO - LS	6,430	4.070	3.070	230,480	183,850	Less lndg. gear
LEO - LORT	8,600	6.528	5.528	114,410	81,380	Round trip
LEO - GTO	2,530	1.737	0.737	960,150	913,520	Requires AKM
LEO - GEO	4,270	2.538	1.538	460,110	413,480	Less boil-off
LEO-GEORT	8,540	6.442	5.442	130,050	83,410	Round trip/boil-off
Planetary	7,770	5.455	4.455	158,860	112,230	High C3

payload capability. This stage could deliver a payload of about 175 metric tons from low Earth orbit directly to the Lunar Surface Base--perhaps necessary for economic placement of a large-scale, self-contained, highly automated lunar surface oxygen facility.

Another example would be the placement in GEO of a large, highly-shielded space station for GEO service crew habitation. The payload of 408 metric tons would permit several space station "common modules," radiation barriers, a large nuclear-electric power supply, and significant operational capability to be emplaced in one flight.

If large, relatively near-term space projects are contemplated, this system is a candidate. The Centaur 'G', now under development for the Galileo and other missions, may evolve into the workhorse OTV of the 22,700 to 68,200 Kg propellant class by use of drop tanks mounted at the Space Station. The first new OTV could possible be one of very large payload class such as this TMI stage, if a demand for such large payload delivery capability develops.

MANNED LANDERS--DESIGN CONSIDERATIONS

The use of a common or nearly-common vehicle for performing manned landings on Mars and the Moon would require that the vehicle meet several disparate performance and environmental requirement sets. The difference in gravitational attraction will dictate that different engine thrust levels be available in order to perform hovering flight and near-surface translational maneuvering. For Mars, an engine that can throttle over a wide range or some engines that are not used on the Moon may be required.

The Mars landing vehicle must accommodate entry heating and will almost certainly employ aerobraking to reduce the descent propulsion system size and mass; the lunar vehicle descends through a near vacuum and is untroubled by descent heating, but cannot make use of aerobraking. The presence of an atmosphere on Mars will cause the dust cloud raised by the terminal descent to persist and envelop the vehicle, whereas on the Moon, disturbed surface particles follow a ballistic path and do not dwell about the vehicle.

The different gravity fields of Mars and the Moon must be considered for every aspect of the lander crew interface--from physical support to egress and ingress. The Mars surface suit may have to be umbilical-supplied for makeup gas, coolant, and power, as otherwise, the pressure suit and backpack may be too heavy on Mars for a human to stand unsupported or gain the necessary mobility on foot. A long duration suit may weigh several hundred Earth pounds.

The texture of the surfaces may be sufficiently different that different landing gear design become necessary. Heat rejection on Mars cannot use the Apollo-era ice sublimator. A compression cycle "heat pump" coupled with external convectors or space radiators are needed for Mars; however, the dust problem of Mars must be carefully considered in assessing heat refection devices.

General arrangement may take several forms for either a Mars or a Moon landing vehicle. The requirements for ascent from the surface of Mars, however, are much more severe than from the Moon, such that a dedicated ascent stage with drop tanks for Mars ascent appears necessary. A lunar vehicle which uses hydrogen/oxygen propellants can descend and ascend with the stage. Thus, a common design for Mars and lunar landers does not look reasonable, if optimum performance for each is desired.

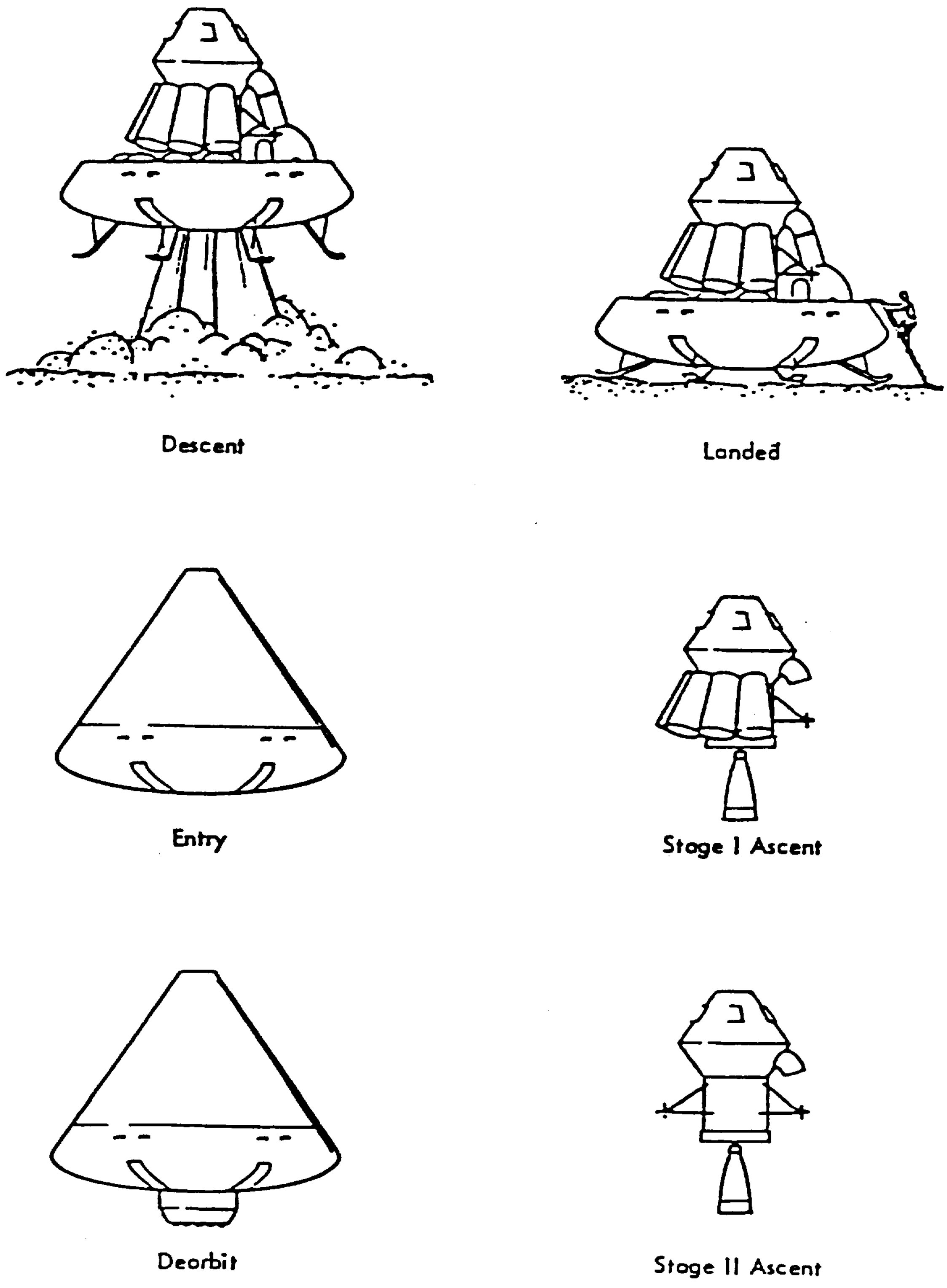
In spite of these problems, a common lander might be designed that would be primarily a Mars lander, capable of lunar landings. Table 4 shows the basic characteristics of a two and one-half stage lander, as it might be configured for Mars only and the Moon only, and the common configuration configured for both Mars and lunar missions. The Mars lander or Mars Excursion Module (MEM) design drives the lunar lander. The common lander in the lunar configuration does not carry the ascent drop tanks and the afterbody shroud, but is still assumed to carry the heat shield and the large tanks required for Mars first-stage ascent. Surprisingly, the descent propellant required for both cases is about the same, so there is no penalty associated with a common descent stage. All four of these landers carry a crew of 4 with 60 days life support, have an aerodynamic L/d of .5, a length of 8 meters, and a diameter of 9 meters. They will all look something like Figure 2, but the lunar version can be stripped of outer shell. All versions carry a 3.3 metric ton storm shelter. All versions use liquid oxygen/monomethyl hydrazine propulsion for both ascent and descent. This fundamental design is described in more detail in references 3 and 4.

TABLE 4 - LUNAR/MARS MEM CHARACTERISTICS

	MARTIAN ONLY MEM	LUNAR/MARS MEM (MARS CONFIG.)	LUNAR/MARS MEM (LUNAR CONFIG.)*	LUNAR ONLY MEM
DEORBIT MASS, MT	70.90	71.24	51.38	46.21
ASCENT 2ND STAGE PROPELLANT, MT	4.20	4.20	0.00	0.00
ASCENT 1ST STAGE PROPELLANT, MT	18.70	18.70	3.54	2.47
TOTAL DESCENT PROPELLANT, MT	22.10	22.20	24.95	22.44
ASCENT STG. TANKS & SYSTEM, MT	1.31	1.31	1.31	0.17
ASCENT CAPSULE (LESS PROPUL.),MT	2.42	2.42	2.42	2.42
DESCENT STAGE (LESS PROPULSION)	18.96	18.97	16.05	15.90
LANDED CARGO, MT	1.91	1.91	1.91	1.91
ASCENT CARGO, MT	0.14	0.14	0.14	0.14
CONSTANT ASCENT THRUST, KLBF	40	40	40	40
ASCENT THRUST/WEIGHT, KLBF/KLBM (MARS OR LUN. WEIGHT, STRT BURN)	1.75 (MARS)	1.75 (MARS)	14.35 (MOON)	20.56 (MOON)
CONSTANT DESCENT THRUST, KLBF	120	120	120	120
DESCENT THRUST/WEIGHT, KLBF/KLBM (MARS OR LUN. WEIGHT, END BURN)	2.87 (MARS)	2.85 (MARS)	11.87 (MOON)	13.20 (MOON)
ASCEND TO, KM (AND DESCEND FROM)	500x32,963 (24 hour)	500x32,963 (24 hour)	200x200 (circular)	200x200 (circular)
PROPULSION ISP, SEC	360.5	360.5	360.5	360.5
ASCENT STAGE 2 DELTA V, KM/SEC	2.66	2.66	0.00	0.00
ASCENT 1ST STAGE DELTA V, KM/SEC	3.43	3.43	1.92	1.92
DESCENT DELTA V, KM/SEC	1.23	1.23	2.17	2.17

* DROP TANKS AND AFTERBODY REMOVED

Figure 2 - Rockwell Lander with MSFC Updates (taken from REF. 2)



MEM - MISSION PHASE CONFIGURATIONS

MANNED LANDERS--DEVELOPMENT AND TESTING

Reference 4 (the last major MEM study, done in 1967) considered testing of a Mars-only lander on the lunar surface as an option in a development and test program that would also include an unmanned Earth entry test and several manned tests in Earth orbit. The MEM heat shield was to be built to withstand Earth entry, such that an unmanned Mars test could be avoided. A lunar landing of a Mars-only MEM is not a requirement for a test program and would, in reality, be a small additional lunar program with some testing benefit. Given the advance in technology since 1967, it may now be possible to test a Mars lander with perhaps one- or two-manned Earth orbital and entry flights.

OTV DERIVED CARGO LANDER

The Johnson Space Center has performed a conceptual design of an aerobraking orbit transfer vehicle (OTV) which uses the heat shield structure to support the propellant tanks and other vehicle systems. This 1990's vehicle concept is described in NASA TM 58264, March 1985. An attempt was made to adapt this space-based LEO to GEO and return vehicle to the task of landing on the surface of either Mars or the Moon while carrying an unmanned, or "cargo," payload. Figure 3 illustrates the original vehicle and its modified version landing on Mars.

Although no analysis has yet been performed, early indications are that a common heat shield may be designed for the LEO-GEO-LEO and Mars aerobraked landing missions. Placement of cargo and landing gear on the JSC OTV concept is complicated by the fact that the flight path during engine thrusting is approximately normal to the aerobraking path. This may be manageable by careful placement and attachment of both cargo and landing gear. Gimbal travel and engine throttling may be needed to an unprecedented extent to maintain stability. The space-based OTV may be reduced in mass for any mission not encountering an atmosphere, including lunar landing, but the basic structural arrangement must remain for the Mars and lunar vehicles to be considered "derivatives."

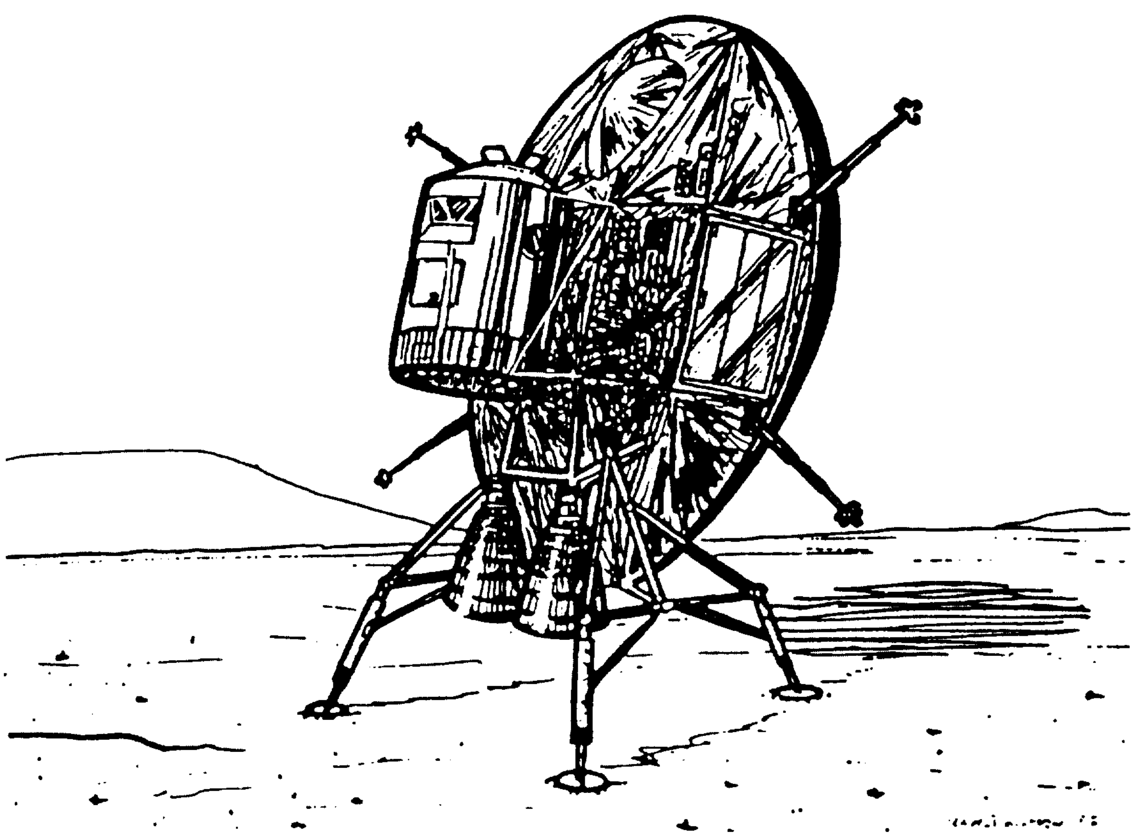
OTV DERIVED LANDER--MASS PROPERTIES AND ENGINE PERFORMANCE

The JSC OTV has an inert mass cited as 5,032 kilograms to house 38,000 kg of O_2/H_2 propellant which can transport a 7,120 kg manned crew module round trip from LEO to GEO and return. As both the lunar landing and Mars landing missions are much more sensitive to inert mass than are

Figure 3 - JSC OTV

Aerobraking into Low Earth Orbit

Modified Cargo Lander on the Martian Surface



GEO transfers, the "first pass" mass property estimates given in Table 5 for the lunar and Mars derivatives presume a composite structure OTV with a refined 25% lighter thermal protection system (TPS). In addition, a second generation main propulsion system (MPS) engine rated at 48,900 N thrust is assumed to be available, permitting the use of three engines, rather than two, and producing a higher delivered specific impulse than the RL 10 11B. Both of these assumptions for the "derivative" vehicles are consistent with their somewhat later need date than the space-based LEO-GEO vehicle. For the lunar case, a 973 m/sec delta V was used for lunar orbit insertion and 2,100 m/sec for descent. For the Mars case, 1,230 m/sec was used for de-orbit and descent (aerobraked).

OTV DERIVED CARGO LANDER PERFORMANCE--LUNAR MISSION

With the relatively low energy requirements of this mission, which originates in low lunar orbit with topped-off propellant tanks, the OTV-derived lander appears to have the performance capability to land from low lunar orbit with an unmanned payload of over 55 metric tons. There is considerable doubt, however, that a side-mounted payload of this magnitude can be balanced by engine gimbaling. Also, about 40% more engine thrust would be necessary in order to hover. A more appropriate mission mode may be to have the cargo lander perform its own lunar orbit burn, leaving the cis-lunar OTV only the task of establishing the trans-Moon trajectory and returning to the Space Station. When the vehicle is required to accomplish the insertion into lunar orbit, the payload is reduced to 31.9 metric tons, which is probably more than adequate for base buildup and resupply.

OTV DERIVED CARGO LANDER PERFORMANCE--MARS MISSION

The same comments apply here, only with a great deal more force than for the Moon due to the higher gravitational field and indicated payload for the Mars landing mission. Over 84 metric tons is neither required nor feasible, so again we must look for additional propulsion tasks for the Mars landing vehicle. If cryogenic propellant insulation can truly accommodate nine- to ten-month missions, an interesting possibility may be for the Mars cargo lander to proceed independently of the manned planetary vehicle and pre-position its cargo at the desired landing site, conducting its entire mission independent of the principal Mars space vehicle.

TABLE 5

ESTIMATED MASS PROPERTIES--OTV DERIVED CARGO LANDER
Estimated Mass in kilograms

SUBSYSTEM MASS	BASIC OTV	LUNAR DELTA	LUNAR LANDER	MARS DELTA	MARS LANDER	NOTE #
TPS	351	-263	88	11,592	11,943	#1
Structure--TPS	281	-140	140	9,216	9,496	
Structure--Other	510	127	637	170	680	
Payload Accom.	265	40	305	133	398	
Internal insulat.	239	60	298	79	318	
Power & Distrib.	218	54	272	73	290	#2
Reaction Control	204	153	357	204	407	
Avionics	210	63	274	63	274	
Tankage	1,301	0	1,301	0	1,301	
Main Engines	841	-136	705	-136	705	#3
Landing Gear	0	1,270	1,270	5,511	5,511	#4
Other	0	0	0	0	0	
Subtotal--Dry Mass	4,420	1,228	5,647	26,904	31,324	
Residual & Reserve Propellants	610	0	610	0	610	
Contingency Inerts	880	250	1,130	5,380	6,260	
EOM Inert Mass	5,910	1,478	7,387	32,284	38,194	
Payload (manned)	6,800		31,900	unmanned	84,400	#5
Total EOM Mass	12,710		39,280		122,600	
Re-entry Mass	13,020		39,280		122,600	#6
Usable Propellants	38,100		38,100		38,100	
Boil-off	190	-130	50	-100	80	#7
RCS Propellants	340	1,020	1,360	1,360	1,700	
Fuel Cell Reactant	120	120	240	150	280	#2
Start-burn Mass	51,770	1,000	79,040	1,400	162,750	
Mass Ratio	4.072				1.328	#5
"Mass Fraction"	0.847		0.808		0.486	
I_{sp} (Mission Effec.)	470.1		458.3		453.7	#8

NOTES:

1. 75% TPS removed for lunar, 15% of extra payload added for Mars
2. Assumes "power on" from internal power for OTV transfer/coast
3. Assumes new technology engines reduce dry mass/unit thrust
4. Assumes 3.5% of landed mass for lunar, 5% for Mars
5. Lunar & Mars landed cargo is found by iteration
6. No apogee raise maneuver for lunar on Mars landings
7. Assumes passive thermal control, top-off before deployment
8. Estimated by multiplying the steady state I_{sp} (483 sec) by the ratio of useful propellants to total fluids consumed, then deducting 1% for stop-start losses

SURFACE HABITATS

Significant design criteria such as crew size, stay time, pressurized volume, maximum weight, or dimensional limits for a Mars/Lunar Surface abitat Module (SHM) have not been defined as yet; therefore, the discussion that follows is based on assumptions which may or may not be applicable to later, more refined studies. As an example, the selected SHM shape (i.e., cylindrical vs hemispherical) obviously affects module weights considerably, but the shape will be dictated by other considerations such as SHM function, launch vehicle characteristics, orbit-to-surface delivery mode, etc.

Also, meeting program goals of minimizing development and testing costs can dramatically influence the design and manufacturing approach of the SHM. Interest in the evaluation of Space Station Common Module (CM's) for the SHM role stems primarily from this consideration. The Space Station CM used to consider Mars vs. lunar Surface SHM's is taken from reference 4.

MARS SURFACE HABITATION MODULE

The requirements for a Mars SHM are generally more severe than those for a lunar SHM in that the natural environment imposes more design complications. To illustrate, a comparison of the environmental factors affecting structural design is given in Table 6.

TABLE 6

ENVIRONMENTAL FACTORS AFFECTING SHM (TYPICAL)

FACTOR	EARTH	MARS	MOON
Atmospheric Pressure (mb)	1000	7-9	-0-
Temperature (^{o}K)	300	215-280	220
Soil Density (g/cc)	-	3.9	1.0-1.6
Gravity	1.0 g	0.38 g	0.165 g

In addition, the composition of the Mars surface material, combined with occasional storms with winds up to 100 mph, can create a significant erosion hazard. Although radiation protection from the SHM structure will be significant, for long times, the statistical probability of solar flares will likely require additional shielding or safe areas for crew protection. It is believed that the micrometeoroid hazard is somewhat less on Mars than on the Moon, but again, it is likely that additional protection would be needed if the SHM structure is designed to handle pressure, landing, and intertial loads only and deployed on the surface of Mars.

These concerns have led other studies to examine ways of burying or covering the SHM on the Mars surface. Such an approach appears feasible with considerable benefits. However, this will require considerations of budkling and local instability if the current family of Space Station CM's is considered. Certainly, there are ways of encapsulating the SHM without loading the skin structure with the overload from the Mars surface material. Such approaches as boring a self-supporting tunnel or trenching and erecting a roof structure could mitigate the penalty associated with CM re-design.

The long Earth-to-Mars transit time and the requirement for entry thermal protection would appear to cause significant differences in the Mars and the Moon SHM's. If needs for cxommonality in design are significant, these problems can be solved by deployable, single-use shields or panels. Trades of weight, cost and complexity of this approach versus separate SHM designs will be required to determine which approach is better.

Structurally, except for the possible load from burying the module, the Space Station Common Module would require modifications for attach points for Earth-to-Mars transfer and for deployment loads. The structure should be slightly over-designed for pressure loads for the Mars SHM application.

LUNAR SURFACE HABITATION MODULE

The requirements for a lunar SHM appear to be less rigorous than for a Mars SHM; notably, lower "g" and no "entry" heating will result in more design flixibility and, very likely, a design similar to the current Space Station CM design could be used, provided transportation and functional needs are satisfied.

Thermal control, radiation hazards, and micrometeoroid protection consideration may lead to a desire to place the SHM below the lunar surface as was discussed for the Mars SHM. If so, it is recommended that

that free-standing covers support the surface material rather than penalize the SHM design for this additional long-term load.

As an example, an early NASA-JSC design for a CM called for an 0.06 inch (.15 cm) thick wall of 2219.T87 aluminum alloy. Wall thickness was set by micrometeoroid criteria rather than pressure loads or flight loads. If it is determined that the micrometeroid hazard for the lunar SHM is less than for the Space Station, some structdural weight reduction is possible; otherwise, the additional weight of the lunar soi8l if the SHM is placed below the lunar surface could likely be accommodated by most of the current CM designs. Checks for local buckling and stability around cut-outs would be needed as well as Earth-to-Moon transportation load conditions. Design modifications to accomodate hard attachment points and support pads for surface deployment will be needed.

While there are many details yet undefined, it is feasible to consider a pressure module such as the space Station CM for both Mars and a lunar SHM. Perhaps the most significant impact will be the method of delivery of the Mars SHM to the surface; the cylindrical shape may not be feasible for the aero-entry system design. As far as a manned, pressure module, the development and certification of the Space Station CM will go a considerable way to the development of a manned surface habitat.

LONG-RANGE MANNED TRAVERSE VEHICLES

Before examining the feasibility of a long-range traverse vehicle capable of operating on both the lunar and Martian surfaces, it is necessary to establish the feasibility of such a vehicle for operation on either surface. A proposed requirement for a traverse vehicle is the capability to travel from the equator to the pole and back. This allows the manned exploration of half of the planetary surface from one base, which in some opinions, must be possible befroe committing to a surface base. On Mars, this trip is 11,300 km (7,000 miles), while on the Moon it is only 5,600 km (3,500 miles). Assuming travel at an average speed of 24 kilometers per hour (15 mph) for 12 hours per day and one day of scientific activity for each travel day, the trip on Mars will last 80 days. Weight of a Mars vehicle will be twice that of a lunar vehicle. The Mars requirements drive the design.

Figure 4 illustrates a Common/Mars Traverse Vehicle design concept. The nuclear power plant is in the trailing segment and cabin systems are located in the leading segment. This configuration is an adaptation of a vehicle currently under study (Ref. 6) Table 7 is an approximate mass summary. Subsystem weights are based on reference 6 information.

The first order feasibility of this vehicle depends on the power supply. A brief examination of fuel cells indicated that this trip is too long for their use as the primary power supply. Fuel cells should be adequate for short range missions, however, and they are included in the design as a secondary power system. A nuclear power supply is used as the primary power supply. The reactor segment provided about 15 square meters of radiator surface operating at 1300^{0} K on the upper portions of the body. The reactor is sized at 100 KW electric, assuming 50 Kg per KW typical of SP-100 type reactors.

The locomotion system consists of independently suspended 72-inch by 30-inch wheels. Power is provided to each wheel by electric motors similar to the Lunar Rover Vehicles. Track and elastic loop wheel systems were considered for greater obstacle clearing capability. Track systems, however, are notoriously unreliable and heavy, and are usually better suited to solid with high cohesion (Ref. 7). Fatigue in the elastic loop material for loop wheels appears to limit the reliability to an unacceptable level (Ref. 8).

The pressurized cabin system layout is similar to the 4 x 4 MOLAB configuration described in reference 9. The airlock is contained in the aft portion of the cabin and is provided with shielding for solar flare protection. The environmental control and life support is a closed cycle system. The Martian vehicle has the option of obtaining some consumables from the atmosphere. Secondary power is provided using fuel cells. This secondary system allows limited use of the vehicle in the event of reactor failure or prior to the reactor delivery.

The structural design of the vehicle is determined by the higher Martian weight. Examination of dynamic interactdion with the respective environment is necessary. Surface separation on the Moon during obstacle clearing may be higher than on the Martian surface. Braking capabilities will also vary with the lunar and Martian weights. Since the vehicle

Figure 4 - Long Range Traverse Vehicle

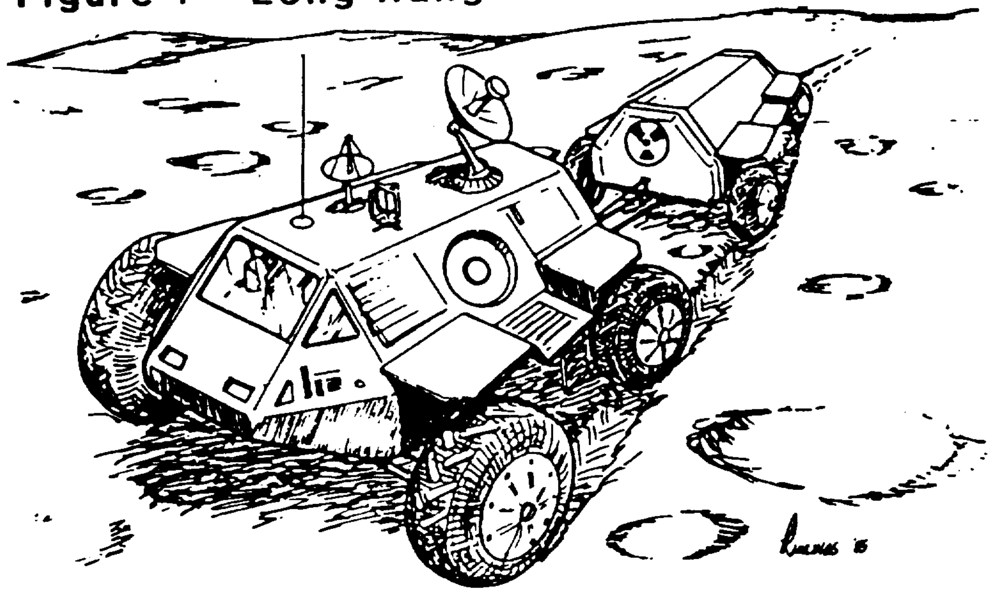


Figure 5 - Unmanned Rover

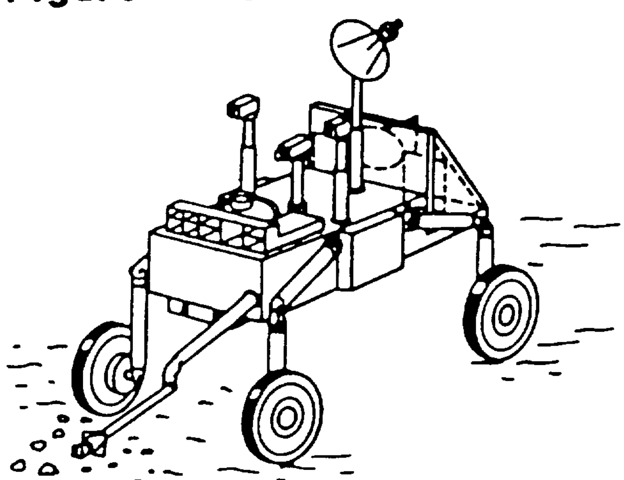
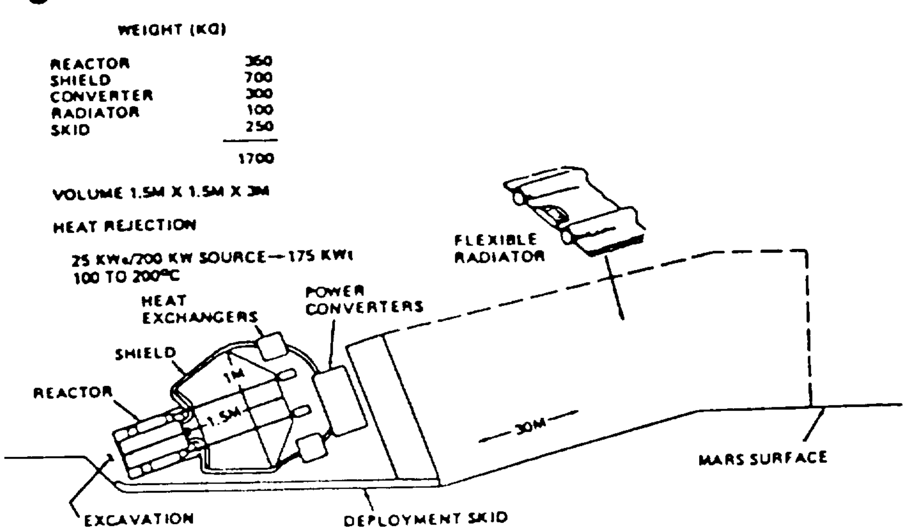


Figure 6 - Power Supply (from REF. 13)



weight is lower on the lunar surface and momentum is the same, lunar braking will be more difficult.

Differences in the soil characteristics may result in different operational capabilities. The range of angles of internal friction for each soil are approximately the same, but the Martian soil may be generally less cohesive than lunar soil. As a result, the vehicle will have greater slope climbing capability on the lunar surface.

The reactor radiators do not presesnt a commonality problem since they operate at over 600^{o} K. Study shows that radiators operating at temperatures higher than 600^{o} K do not have problems operating in either environment. Heat dissipation from the cabin segment is likely to occur at significantly less than 400^{o} K, thoughm, which may presesnt a commonality problem.

While space reactors are currently in development, the reactor used in this application will be unique. This reactor must be controllable while current applications do not appear to have wide variation in load characteristics. The high amplitude and random motion effects on the reactor that may be anticipated during Martian and lunar traverse must be considered also.

A vehicle designed for lunar and Martian long-range traverse appears feasible, assuming a nuclear reactor can be developed for this application. The vehicle will be over-designed for lunar applications, but a significant weight penalty is not anticipated.

UNMANNED ROVER VEHICLES

Unlike manned traverse vehicles, unmanned rover vehicles have received considerable recent attention. Reference 11 describes a 1984 Mars Rover initially planned for a mid-1980's follow-up to the Viking program. In addition, a vehicle very similar to this design has been baselined for the Mars Sample Return mission described in Reference 12. For the purposes of this comparison, this vehicle configuraiton will be used.

This configuration has a roughly rectangular body mounted on four articulated supports. One design uses elastic loop wheels for locomotion (Ref. 8) while the other uses wheels (Ref. 9). Figure 5 shows the wheeled vehicle.

The interaction between the terrain and the vehicle will not present capability conflicts for the same reasons described for traverse vehicles. Braking problems should be of little consequence since the vehicle will travel at slow speed.

Thermal considerations may cause problems in the use of a common vehicle since it appears that the thermal control system will operate at considerably less than 400^{o} K.

The computing system requirements may be different for the lunar application since ther period of radia contact with the vehicle will be considerably longer than for the Martian application. The Mars Rover must be capable of automated travel for periods of one day, while it is likely that the Lunar Rover can be controlled directly from Earth. A Martian vehicle baseline, however, should have exceptional operational capabilitiets on the lunar surface.

It appear that the major development requirement for this vehicle will be the image processing capabilities of the computing system. Some testing for the Mars application could be accomplished on the Lunar surface, but in reality, this would be an additional program.

Overall, the use of a common rover vehicle for the Moon and Mars appears feasible and desirable. Slightly increased design costs should far outweigh a complete repetition of the total design.

BASE POWER SUPPLY

Discussion of common electrical power systems for a mars base considers a power requirement of 25 Kwe. A Photo-PV system, including a regenerative fuel cell (RFC) and reactor (Rx) were selected for more detailed analysis. A major factor affecting the design of a common PV system is the relative solar intensity on each body. The lunar surface receives approximately 1,353 W/sq m during the day. On Mars, daytime solar intensity varies from 708.8 to 487 W/sq m, depending on the season. A nominal value of 582.8 W/ sq m, 0.43 relative to the lunar surface is specified in Table 8. This difference in solar intensity would relquire the Mars solar array to be 2.13 times larger than a solar array positioned on a lunar base.

In addition to overcoming the difference in solar flux between the Martian and lunar surfaces, the longer lunar night (18 Earth days), poses problems for a common design. Compared to a Mars RFC system, 35 times as

TABLE 7 - MANNED TRAVERSE VEHICLE ESTIMATED MASS, KGMS

TOTAL MASS	22,650
CABIN SEGMENT	12,150
Environmental Control & Life Support	1,140
Other Crew Systems	390
Food	130
Experimental Equipment	170
Communication	50
Navigation	40
Data	270
Displays	90
Shielding	2,000
Secondary Power	1,900
Pressure Hull	1,920
Structure and Drive System	4,050
REACTOR SEGMENT	10,500
100 KW Reactor	5,000
Shielding	2,000
Structure and Drive System	3,500

TABLE 8 - POWER SYSTEM COMPARISON

System type	Output (KWe)	System (Kg/Kwe)	Solar array or radiator area (sq m)	Solar array mass (Kg)	RFC mass (Kg)	Total mass (Kg)
LUNAR						
Photovoltaic	25 day only	48	396	1,188	-	1,188
Regen Fuel cell	25 constant	1070	1,117	3,351	23,402	26,753
Nuclear Reactor SF-100 program	25 constant	68 to 136	3 to 26	-	-	1,700 to 3,400
MARS						
Photovoltaic	25 day only	102	846	2,538	-	2,538
Regen Fuel cell	25 constant	337	2,383	7,149	1,268	8,417
Nuclear Reactor SP-100 program	25 constant	68 to 136	3 to 26	-	-	1,700 to 3,400

many storage tanks would be required on the Moon to deliver continuous power through the lunar night. From these two factors, a common PV-RFC design is not likely.

A nuclear reactor will deliver continuous power throughout the local night. At temperatures above 600^{o} K, waste heat radiator performance on the Moon and Mars is the same, making a common radiator design possible. Based on the SP-100 program, a 25 Kwe nuclear reactor was choses for consideration (Figure 6, taken from reference 13). The weight is 1700 Kg or 15 W/Kg. In this design, the reactor is towed into an existing hole or a hole made from an explosive charge. Once in place, the generation system is electrically connected to the manned base by a tether. The recessed position of the reactor and the added distance produced by the tether allow the system to use a lighter reactor shield.

The power plant consists of a nuclear reactor as a heat source, a radiation attenuation shield to protect the payload, the electric power conversion equipment, and a heat rejection system to eleminate waste heat. Power conversion is by the direct thermoelectric conversion of heat to electricity.

This design uses refractory metals in the power system's construction. Due to the reactive nature of refractory metals with the carbon dioxide atmosphere of Mars, different materials may have to be used. The metal would not be a problem on the Moon since there is no atmosphere.

Development and testing of this type of nuclear reactor is on-going in the tri-agency (NASA, DOE, and DARPA) SP-100 program. This type of system is expected to provide poer for mid-1990's missions.

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N87-17805

OTV IMPACTS AND INTERACTIONS

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ABSTRACT

The purpose of this brief paper is to identify the possible impacts to, and interactions with, the agency's planning activities for the Orbit Transfer Vehicle (OTV) that is tentatively scheduled for initial operational capability in the late 1990's. In general, the various Mars missions proposed elsewhere in this report require vehicles of significant size and performance far greater than that provided by any OTV configuration currently being seriously considered. Therefore, interactions and impacts on these current concepts are minimal. These impacts and interactions fall into categories of technologies, systems, and operations. Each category is addressed in the text.

INTRODUCTION

The civilian space agency is currently in the planning stages for the next step in the Space Transportation System (STS), a reusable OTV that can expand the range and capability of the Space Shuttle and Space Station. To accomplish preliminary definition of the OTV, Phase A development studies are in progress by three aerospace contractors. In addition, technology developments have been initiated in propulsion systems, aerobraking, and space servicing and operations. It is of value, if not the principal function of advanced program activities, to identify the impacts to, and interations with these current developments. It is in the best interests of this nation's future in space for the agency to define infrastructure and technology development programs that grow and evolve such that they enable, support, and promote these advanced programs. It is even more critical that the agency insure, through knowledge of these potential missions, that no developmental steps are made to support near-term needs at the sacrifice of the possible evolution into future missions. A particular point to be made here is that details of these future programs are much less important than how they interact with or impact current or near term space infrastructure element definition and technology developments. For the OTV, a useful breakdown of these impacts and interactions appears to fall into categories of technologies, systems, and operations. Each of these categories will be discussed in the following paragraphs.

OTV TECHNOLOGIES

Many times it is difficult to obtain public support for development of particular technologies when they are directly justified on the development of a single element. Advanced mission planning provides opportunities for broadening the base of support for technology development by enhancing the value of these technologies through a demonstration of how they can promote and support paths towards desirable future goals for the agency. For the manned Mars mission, supporting technologies are at least, but probably not exclusively: (1) Propulsion; (2) Life sciences; (3) Life support; (4) Robotics/automation/artificial intelligence; (5) Aerobraking; (6) Thermal protection; (7) Cryogenic fluids storage and handling; and (8) Power.

Propulsion

A manned Mars mission is a very demanding mission from the standpoint of propulsion. Concepts presented elsewhere in this report show departure masses of two million pounds or greater in low Earth orbit. Propulsion technologies in work that are primarily, but not totally, targeted for OTV development are the advanced space engine and propellant management. The advanced space engine, per se, will be of too low a thrust level for the Earth departure burn for a manned Mars mission but may be adequate for other portions of the mission. In addition, the technology developments that permit the increase in engine Isp may be applicable to development of a much larger Earth departure engine system. Other concepts presented in this volume rely on long term cryogenic storage (up to three years) and propellant transfer, not only between stages, but between fueling stations in space and the Mars vehicles. The most promising technology for enabling and enhancing manned Mars missions appears to be in-situ propellant production on the Mars surface, Mars moons, or even Earth's Moon. Some of these in-situ propellant production schemes will involve the transfer of cryogenic propellants in the "zero-g" environment of space.

Aerobraking

Analyses and conceptual designs for the agency's OTV that show aerobraking is a significant enhancing technology that can reduce the cost of transportation to Geosynchronous Earth Orbit (GEO) by almost doubling the delivery capability of an all-propulsive vehicle (reference 1). Analyses for manned Mars missions accomplished elsewhere in this report (reference 2), demonstrate a significant advantage of aerobraking in the performance of the Mars vehicle as well. However, there are significant safety problems involved with the concept of aerobraking either at Mars or at Earth on return. If these problems can be resolved, then the performance improvements offered by aerobraking are large enough for this technology to fall into the category of enabling. Reference 2, however, does make the point that current technologies for reusable Thermal Protection Systems (TPS) for the aerobrakes cannot tolerate the thermal environment for some masses and trajectories. Ablative TPS systems are more than adequate. Since ablative systems are non-reusable, the technology developments for the OTV must be pushed harder if they are to support reusable vehicle systems for manned Mars missions. Table 1 summarizes the impacts and interactions of manned Mars missions with OTV technologies.

Aerobraking technology developments are not limited to aero and aerothermal issues. Guidance through the atmosphere is another technology development receiving a significant amount of study at this time. To enter the atmosphere at a precise point, fly through a highly variable environment, and exit the atmosphere to fly to a precise point in orbit about a planet has proven to be a real challenge for Earth, where the atmosphere is known to a much higher degree than at Mars. Current efforts being devoted to development of guidance algorithms for the agency's OTV should include the requirement to enable aerobraking at Mars with little or no alterations. To address these technologies, the agency is planning a Shuttle launched Aeroassist Flight Experiment (AFE). The AFE will exit the Shuttle and fire a rocket motor to simulate an entry from GEO. The TPS concepts planned for the AFE will be reusable systems designed for the entry velocities encountered by a vehicle returning to Earth at or near the Earth relative parabolic trajectory. Conversely, vehicles returning from Mars will always be hyperbolic, with some trajectories having velocities as high as 55,000 ft/sec. As noted earlier, current state-of-the-art TPSs are not adequate for such a Mars return, hence this would require the AFE to consider the inclusion of

TABLE 1

INTERACTIONS AND IMPACTS WITH THE OTV

INTERACTIONS

I. TECHNOLOGIES

PROPULSION

- O High Performance Cryogenic Engines
- O "Zero-g" Propellant Transfer

AEROBRAKING

- O Utilize OTV Developed Aerobraking Technology for a Significant Performance Enhancement
- O Application of Guidance Algorithms to Mars Aeroflight

II. SYSTEMS

PROPULSION

- O Cryogenic Advanced Space Engine
- O Pump-Fed Storable Engine
- O Cryogenic Propellant Storable and Transfer Systems
- O Aerobraking Systems

III. OPERATIONS

- O Propulsive Vehicle for a Manned Mars Flyby
- O Utilization as a Ferry Vehicle To-From the LEO Space Station and the Returning Mars Spacecraft
- O Utilization as a Ferry Vehicle in the Mars Vicinity

IMPACTS

I. TECHNOLOGIES

AEROBRAKING

- O Reusable Vehicles will Require Significantly More Advanced Technology Development than are Currently Needed for the OTV
- O Aeroassist Flight Experiment:
  - \+ Alternative Advanced Reusable or Ablative TPS Samples
  - \+ Higher Entry Velocities or Dedicated Flight

II. SYSTEMS

- O None Identified

III. OPERATIONS

- O May Significantly Impact the Design of the TPS to Allow an Option of Either Performing a Block Change to the OTV TPS for Mars Missions or Allowing the Application of Ablative TPS some test TPS samples of ablative or advanced technology reusable concepts. The thermal environment to the TPS during entry from GEO-type orbits will be dominated by convective heating, whereas the thermal environment for the upper limit of the Mars returning vehicles will be dominated by non-equilibrium radiation. To fully understand this thermal environent, it may be necessary to provide a much larger booster for the AFE to properly simulate the entry velocities encountered in a Mars return. It may be more appropriate to make a separate dedicated flight of the AFE for assessment of a Mars-Earth return. These impacts and interactions are also noted in the table.

OTV SYSTEMS

Propulsion

The Agency is planning the development of an advanced high performance cryogenic engine that is space maintainable, man-rated, and capable of multiple starts. As mentioned in the previous section, this engine is probably suitable for all propulsive burns except the Earth departure. Pump-fed, high performance, storable propellant engines are also under consideration. The concept of pump-fed storable engines not only results in improved specific impulse for the storable propellants, but permits much greater structural efficiencies in the design of the propellant tankage due to the greater densities of the storables over the cryogenics. Additionally, the use of pumps to directly feed the engines rather than to pressurize the propellant tanks allows lighter tankage designs. These two concepts compete with each other in performance vs. boil-off of cryogens vs. lighter and more efficient structures. The winner will have to await more sophisticated analyses which are in process.

OTV OPERATIONS

Manned Mars Flyby

As discussed in reference 3, it is conceivable to utilize at least one concept of the agency's OTV for the main propulsive stage for a manned Mars flyby mission. The proposal in the reference suggests using two of these OTVs mated to a Space Station module to provide the required velocity increment for the mission. In addition, the proposed command module for the OTV is also required for this mission. On the other hand, however, the proposal in the reference not only discusses the mentioned interaction, but also notes a significant impact to the OTV design that must be addressed early in the definition stages of the OTV. That is, the TPS design proposed for this OTV will not tolerate the entry environment at Earth on return, for some masses and trajectories, and may well have difficulty at Mars also. This short-comming of the TPS requires either more advanced TPS technology development than is required for the current planned near-Earth support, or we must incorporate into the design of the vehicle an option to replace the reusable TPS with an ablator. Alternatively, the final design could allow for a block change to an advanced technology TPS to support Mars missions, or it might be possible to spray an ablative coating directly onto the reusable TPS without physically removing it.

Mars Landing Programs

This category actually covers all cases of Mars missions that, due to significant mass or other reasons, choose to only carry enough propellant to place the vehicle into an elliptical orbit with a very high apogee at arrival back at Earth. This leaves the crew and scientific equipment and Mars samples in an orbit that is not accessible by the Space Station. The plan, as described in reference 4, utilizes the OTV to rendezvous with the Mars Spacecraft and return the crew and equipment to the Space Station in LEO. At first glance, this requirement does not seem to be any more demanding than the current requirement for the OTV to ferry payloads to and from GEO. Another concept for utilization of the OTV (not addressed by any of the papers in this report) is as a sortie vehicle in the vicinity of Mars. There are probably more efficient means to execute sortie missions from the main spacecraft, but this OTV should be given attention until that point is proven.

SUMMARY AND CONCLUSIONS

This brief paper has taken a casual look at the possible impacts and interactions with the agency's proposed OTV, its technology developments, and its major systems. Several impacts and interactions have been found and have been tabulated in Table 1. It is hoped that such a tabulation is useful to the development process for the OTV to make it a more useful vehicle. Since this paper is very cursory, it is recommended that the studies of the Mars missions, as they mature, continue to define possible impacts and interactions to the OTV.

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N87-17806

SPACE STATION UTILIZATION AND COMMONALITY

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ABSTRACT

This paper identifies and discusses several potential ways of utilizing the Space Station (SS), including utilization of learning experiences (such as operations) utilization of specific elements of hardware which can be largely common between the SS and Mars programs, and utilization of the on-orbit SS for transportation node functions. The probablility of using the SS in all of these areas seems very good.

Three different ways are discussed of utilizing the then existing Low Earth Orbit (LEO) SS for operational support during assembly and check-out of the Mars Space Vehicle (SV) : (1) attaching the SV to the SS, (2) allowing the SV to co-orbit near the SS, and (3) a hybrid of the first 2 ways. Discussion of each of these approaches is provided, and the conclusion is reached that either the co-orbiting or hybrid approach might be preferable. Artists' concepts of the modes are provided, and sketches of an assembly system concept (truss structure and subsystems derivable from the SS) which could be used for co-orbiting on-orbit assembly support are provided.

SS CONCEPT

The initial Space Station (SS) is currently planned to be operational in the early 1990's. The timing for a growth version of the SS has not been established, but it certainly can occur in the time frame appropriate for support to Mars missions. The nature and capabilities of the growth SS will partially determine the ability of the manned Mars program to benefit from the SS program. This definition of the growth SS is in progress at this time.

There are several possible scenarios for the evolution of the SS, including phased growth; one growth mode might be replication. Exchange of new-technology equipment for old-technology equipment is a form of evolution, but this will occur as a part of any of the scenarios mentioned. If replication is the path chosen for growth, there would be in existence two or more smaller stations of somewhat limited size and capability. These might have a high degree of basic commonality among them, and yet might be dedicated to different functional purposes, e.g., one might be a more science-oriented SS and another might have a more operations-oriented capability--or, the stations might have identical capabilities and have all types of work evenly divided among them. If there are multiple stations, these might all be at the same orbit, or they could be at different orbits. If the growth path taken by the SS is an increase in the size of the IOC SS, this one would have responsibility for supporting a wide variety of science and operations activities. Such a SS would have larger dimensions, greater resources, and more functions than the initial SS. Each of these considerations would have some bearing on the potential usability of the SS for the manned Mars program.

An early concept of the growth SS was defined in reference 1 and is shown on Figure 1. Dimensions are shown on the figure; weights will be between 500K and 1M lbs. Solar dynamic and photovoltaic power systems are candidates for both the IOC and growth SS. A solar dynamic concept is shown in Figure 1, for reference. The Orbiter (not shown) would berth to one of the Habitability Modules during resupply missions. Some of the user accommodation equipment (experiment, servicing equipment, etc.) has been omitted from Figure 1 for simplification of the drawing. Figure 2 shows such equipment as it is envisioned for the IOC SS; the growth SS would have an increased complement of such equipment. The IOC SS weight is estimated to be slightly less than 500K lbs.

The flight orientation of the SS, as shown in Figure 3, is with the keel along the nadir - zenith line and with modules earthward; the transverse boom is kept perpendicular to the orbit plane.

SS UTILIZATION/COMMONALITY WITH MARS PROGRAM

There are several ways in which the manned Mars program can benefit greatly from the SS program. Some of the key benefits and impacts are listed in Table 1. The two general categories into which these applications fall are: (1) use of SS heritage including experience and use of SS technology, concepts, and/or specific hardware/software designs, and (2) use of the existing on-orbit SS.

As shown in Table 1, there are many areas in the first category where the manned Mars program could benefit greatly from the SS program. It is not apparent at the level of investigation done thus far that there

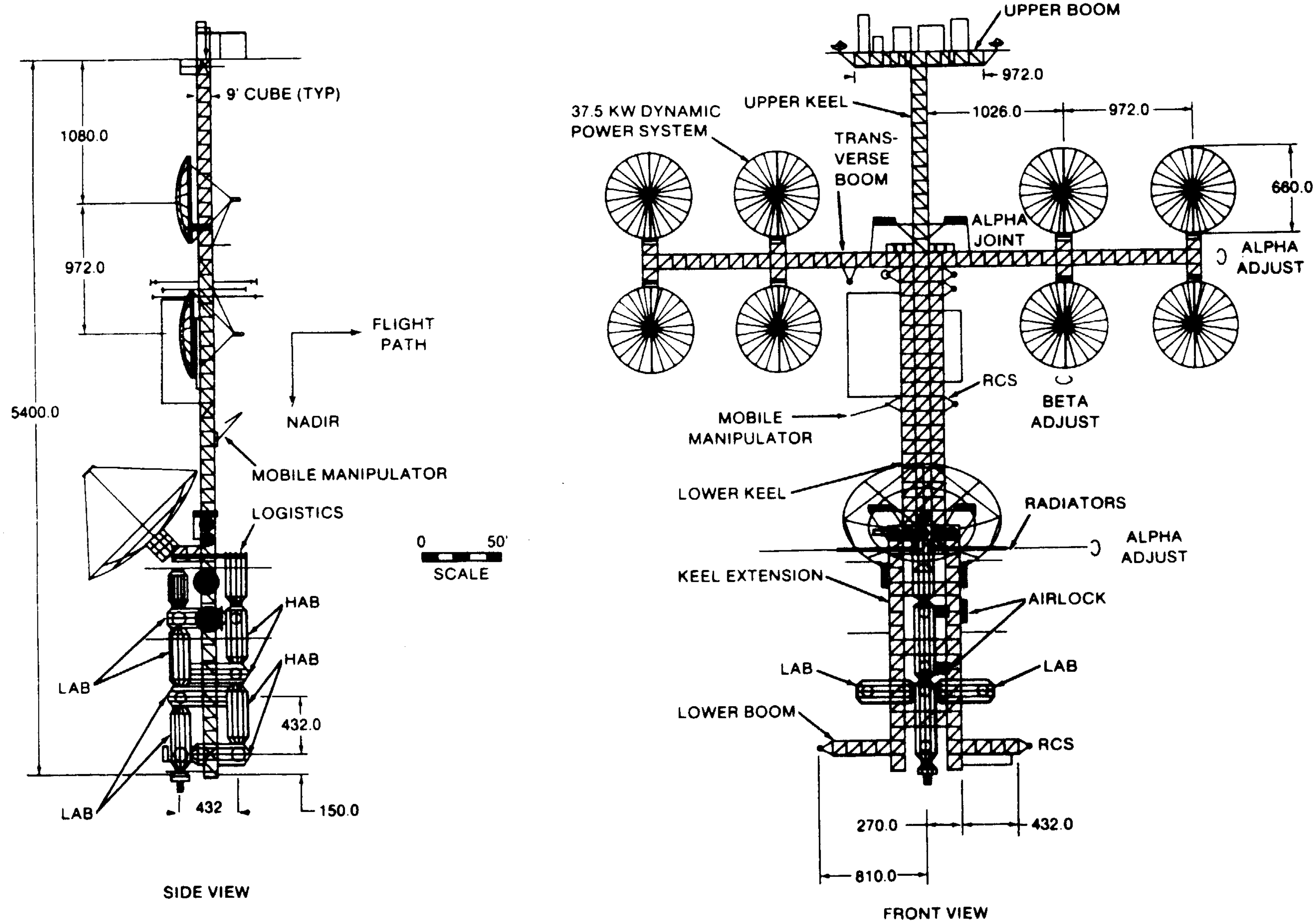

FIGURE 1. – SPACE STATION REFERENCE GROWTH CONFIGURATION

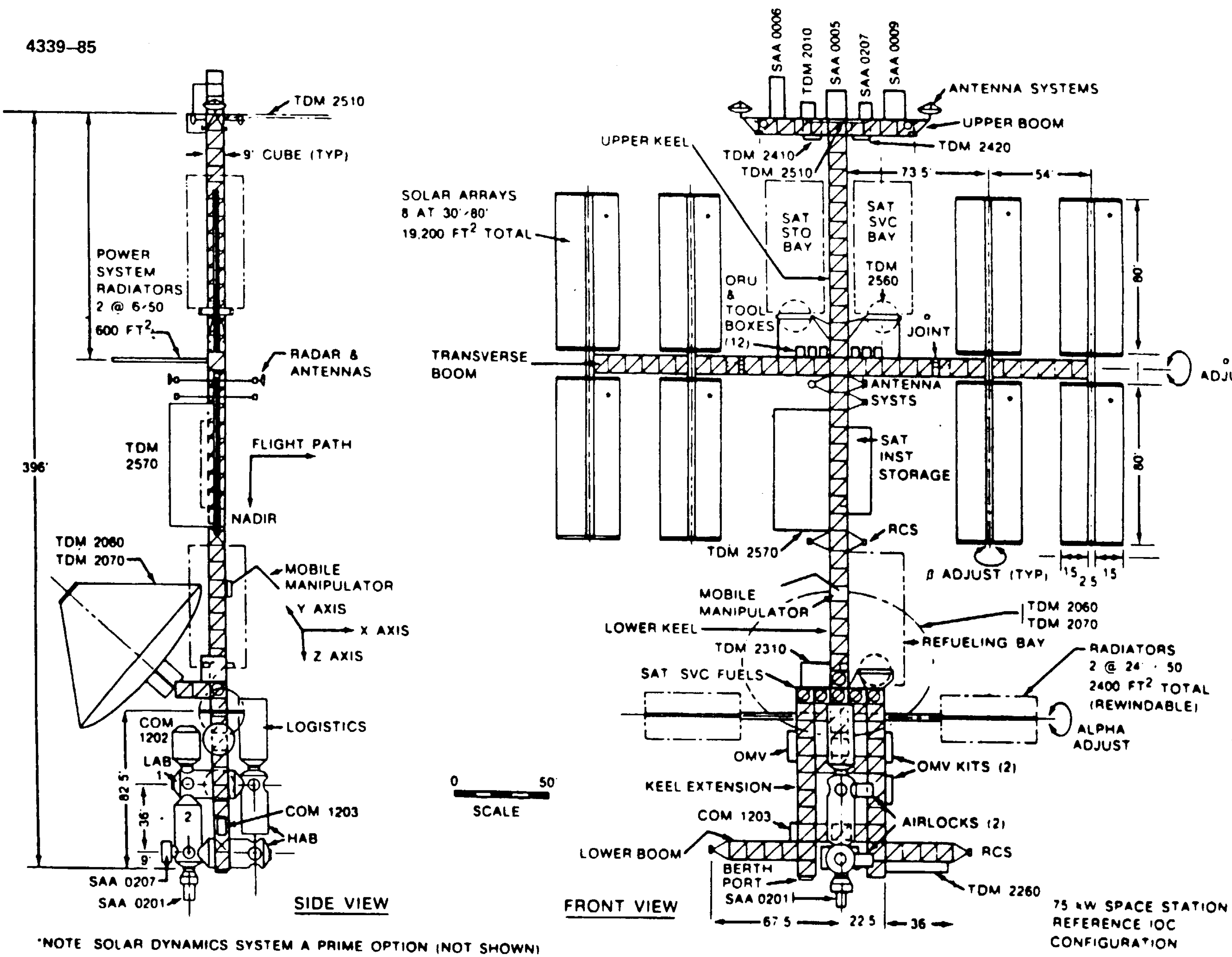

FIGURE 2. – USER ACCOMMODATIONS (IOC SPACE STATION)

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FIGURE 3. SPACE STATION ORIENTATION

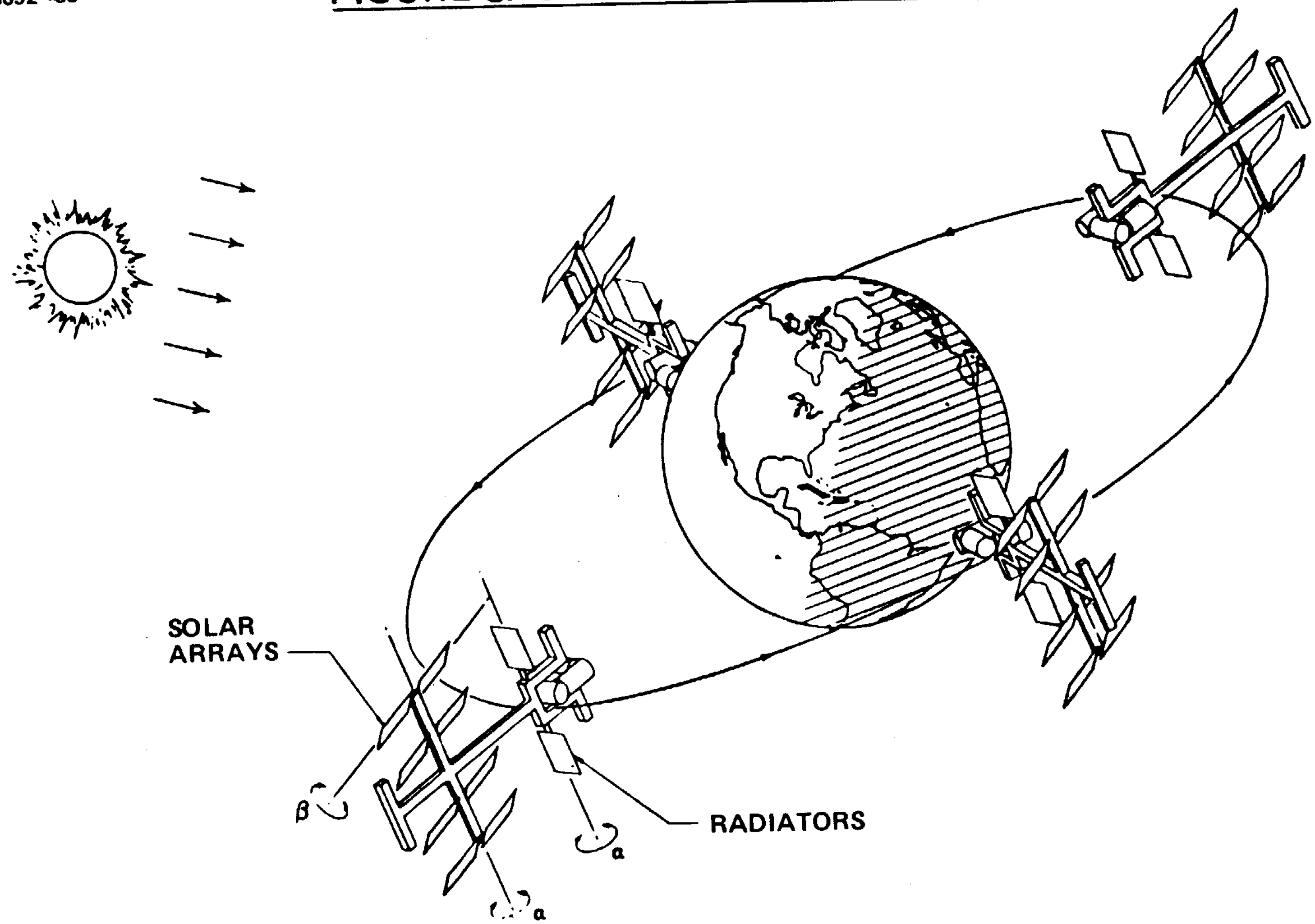

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TABLE 1.
SPACE STATION IMPLICATIONS

SS TECHNOLOGY	SS CONCEPTS	SS DESIGNS	SS EXPERIENCE	EXISTING ON–ORBIT SS

- MODULES & SUPPORT SYSTEMS
- ROBOTICS/AUTOMATION SYSTEMS
- SERVICING, ASSY., & LOGISTICS SYSTEMS
- CRYO FLUIDS SYSTEMS
- GRAVITY–FIELD SYSTEMS
- SOFTWARE
- CLOSED–LOOP ECLSS
- SPACE–BASED OMV & OTV SYSTEMS

- ASSEMBLY, OPERATIONS, & TRAINING
- LOGISTICS & SERVICING
- PAYLOAD INTEGRATION
- LONG–DURATION O–g EFFECTS
- SYSTEM TEST BEDS

CONSIDERATION	MARS VEHICLE ASSEMBLY OPS: ATTACHED	MARS VEHICLE ASSEMBLY OPS: CO–ORBITING	MARS SPACECRAFT RETURN OPS
• MINIMUM DISRUPTION OF SS		√	
• MINIMUM REAL ESTATE REQ'D. ON SS		√	
• MINIMUM QUARANTINE (FWD OR REV.) IMPACTS ON SS	√	√	√
• USE OF SS OTV & OMV	√	√	√ (RETRIEVAL)
• SHARED STS FLIGHTS	√	√	
• MINIMUM OVERALL ORBITKEEPING PENALTY	√		
• MINIMUM OVERALL COST	√		

SOME HYBRID MODE OPTIMUM?

would be any impact to the SS for the Mars program to benefit from use of items shown in this category. Some modification of the designs might be necessary as part of the Mars program due to the requirements for longer mission duration, higher reliability, differences in environments, weight and volume criticality, etc., but costs of incorporating such changes should be far less than those which would be incurred for development of a totally new system. Of course, the greater the similarity between the SS and Mars designs, the more usable will be the "experience" (logistics, servicing, etc.) listed in the first category in Table 1.

As shown by the items listed in the second category of Table 1, the existing SS should be highly useful as a development and qualification test bed for the Mars program systems, elements, operations activity and crew. "Qualification" of the crew will include verfication of methods of reducing or eliminating deleterious physiological effects of long-duration exposure to zero-gravity environments.

Utilization of the SS as a transportation node for the Mars program will potentially require support in the areas listed under that heading in Table 1. There are basically two modes of operation: (1) attaching the SV to the SS, and (2) allowing the SV to co-orbit with the SS. A modified version ofthe second mode would be to allow the SV to free-fly, but would not constrain it to co-orbit with the SS. However, this would essentially amount to not utilizing the SS as a transportation node. Implications of using the attached and co-orbiting modes are discussed in succeeding paragraphs.

<u>ASSEMBLY OF SV WHILE ATTACHED TO SS</u>

Figure 4 shows an artist's concept of the SS with a manned Mars SV attached; on-orbit assembly of the Mars vehicle is being completed here. The SS and SV appear at roughly their relative sizes here, so it can be seen that the SV is a very sizable vehicle in comparison to the SS. This concept of the growth SS is greater than 450 ft. long, and would weigh between 500K and 1M lbs. This concept of the SV is about 246 ft. long and would weigh about 1.6M lbs. fully loaded, of which about 1.2M is propellant. The large aeroshells shown near one end of the SV are about 80 ft. in diameter.

Attaching the SV to the SS could result in significant impacts to the SS, due to the large size, weight, and types of activities associated

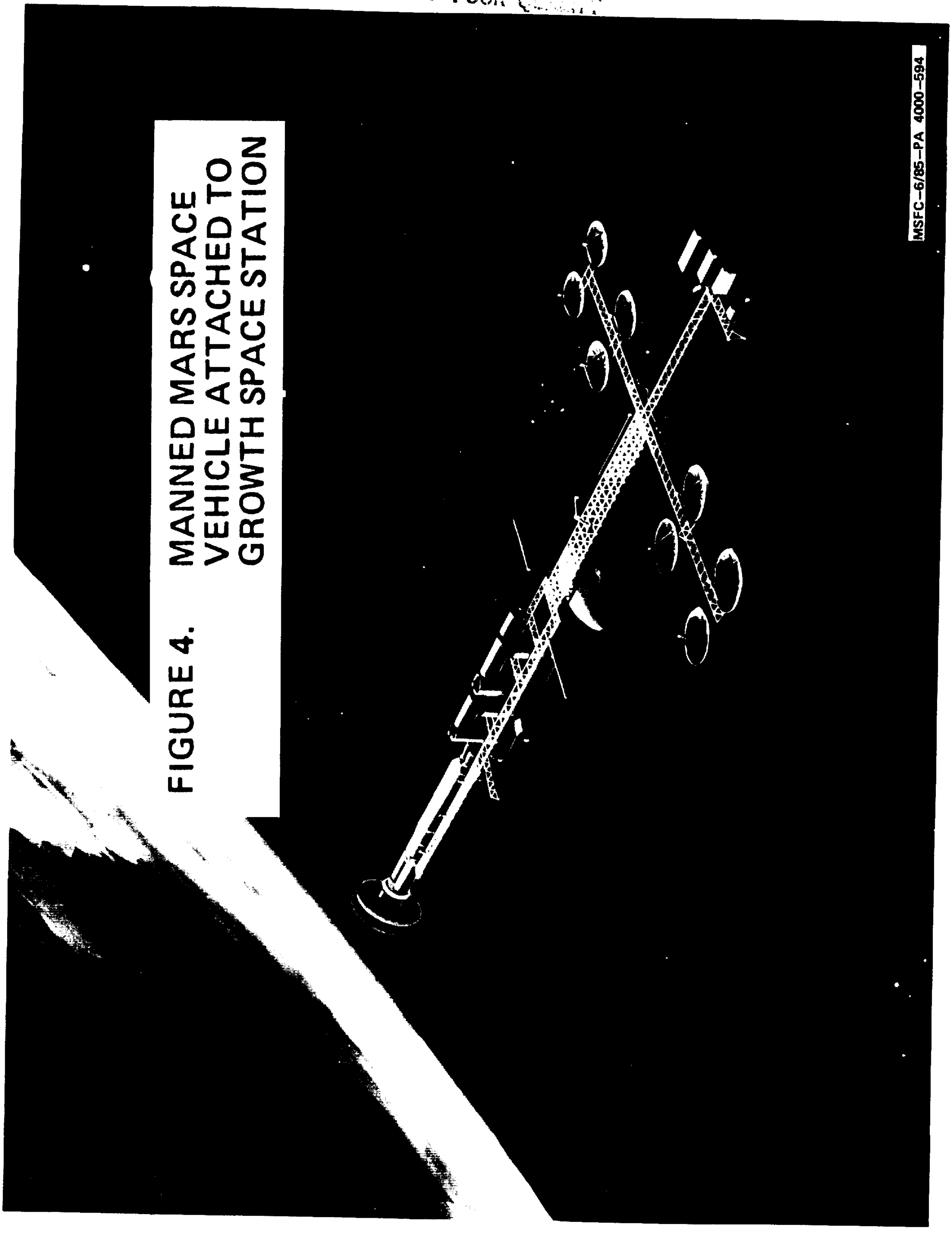

FIGURE 4. MANNED MARS SPACE VEHICLE ATTACHED TO GROWTH SPACE STATION

with the SV. Some of the potential difficulties associated with this mode of operation are discussed below. There will be a number of elements and activities associated with the SS which are not shown in Figure 4. For example, an OTV and spacecraft propellant storage and refueling area will probably be located in the lower central part of the keel. Also in this vicinity will be the berthing and servicing locations for two or more OMV's. In the SS reference concept, at least one face of the keel must be kept free of attached elements to allow for traverse of the MRMS up and down the SS. Payload servicing stations are located along both sides of the upper keel, and Earth-viewing payloads (not shown) will be attached to the lower boom. The payloads mounted on the upper boom need an entire hemisphere of unobstructed viewing in the zenith direction, and those mounted on the lower boom need an entire hemisphere of unobstructed viewing in the nadir direction. For these reasons, it is very difficult to find a location on the SS large enough to attach the SV without incurring some physical or field-of-view obstruction.

Any change in SS mass which would shift the center of gravity (c.g.) out of the orbit plane could quickly become a problem for the SS momentum exchange system, since controllability is fairly sensitive to such shifts. Consequently, the SV should not be attached to the side of the keel. Any SS c.g. shift within the orbit plane is much easier to handle from a momentum-exchange standpoint, and hence, if the SV were attached to the front or back surface of the keel, or to the bottom of the keel, controllability might be acceptable.

As previously mentioned, however, the central part of the keel will be congested, so the lesser of the evils might be to provide a keel extension on the lower end for attachment of the SV (see Figure 4). This would probably interfere with some of the Earth-viewing experiments, so some of them might have to be inoperable during this period. Design of such a keel extension would have to be done so as to ensure that STS berthing to the SS modules would not be impacted. Care would also have to be taken to ensure that the longitudinal c.g. shift did not exceed the bounds allowed by the RCS thruster arrangement, and that the center of pressure (c.p.)-to-c.g. shift did not overburden the momentum exchange system. On-orbit loading of SV propellants and other fluids in the vicinity of the Earth-viewing payloads might be very undesirable from a contamination point of view (depending on the types of propellants used). The SV assembly operations might cause disturbances to any materials science or other payloads desiring a low-g environment. Orbit decay and reboost of the SS may be affected; this might be improved or worsened, depending on the change in ballistic coefficient.

Quarantine constraints on the SV on either the outbound or inbound trip could cause impacts to the SS. In fact, this consideration alone might restrict the SV to a location isolated from the SS.

In spite of the potential difficulties mentioned above, attachment of the SV to the SS could no doubt be made to work if further study indicates that this mode is preferred. Some mitigating factors and steps which could be taken to minimize impacts are listed below:

- Since the reference SS is Earth-oriented, addition of a large payload to either end would minimize controllability impacts.
- In the early buildup phases of the SV, its physical dimensions and mass are smaller, hence impacts to the SS would be less than in later phases; the addition of the dry SV transportation elements represent the largest incremental increase in physical size, and propellant loading of the SV represents the largest incremental increase in mass (75% of total SV weight is propellant). Propellant loading and/or mating of the Mars habitable elements with the transportation elements could be done after separation from the SS.
- Propellant loading of the Mars SV should be accomplished by loading directly from Earth-to-orbit (ETO) tankers to the SV or from an on-orbit propellant depot to the SV, rather than requiring propellants to be stored on or pumped through the SS.
- The existence of a heavy-lift ETO system would allow delivery of larger pieces of the SV than if the STS must be utilized alone, thus reducing the on-orbit assembly and integration effort, skills, and time required at the SS.
- If the SS evolution has proceeded to the point where "branching" has occured (i.e., The SS has been replicated and functions have been re-aligned to provide a science SS and an operations

SS), the disturbances to pointing and low-g payloads due to SV assembly operations would be eliminated.

The SS Orbital Transfer Vehicle (OTV) can be used to circularize the orbit of the SV after return to Earth at end of the mission. This will eliminate having to round-trip a propulsive element for circularization, and will allow significant weight savings. The SS OMV can be used to ferry equipment back and forth and provide other assistance during the assembly period. A duplicate or derivative OTV will be usable as part of the SV propulsion system and possibly as an OTV in the Mars vincinity.

If the SS is in great demand for on-orbit operations or science activities as part of its normal course of business (particularly if commercial or international payloads are involved, any requirement to support the SV, particularly if it extended out to a several-month activity, would be a disruptive occurrence and would interfere with other potential activities. On the other hand, if the Mars activity is a national or international priority item, other workarounds (platforms, etc.) might be provided for the normal SS customers during the occasional periods of Mars mission involvement or a replicated SS could be devoted to support of Mars activities during the time needed.

There are several modes in which the attached SV modules and systems could be supported by the SS (see Table 2): (1) the SV provides most of the crew time for assembly, but the SV modules and systems remain dormant, with the SS providing all habitability (housing, food, etc.) for the SV crew, all resources (power, communications, heat rejection, etc.) and some of the crew time for assembly of the SV, (2) the SV provides habitability and most of the crew time, but the SS provides resources to operate the SV modules and systems and some of the crew time, and (3) the SV provides habitability, resources, and most of the crew time, and the SS provides some crew time. If the SS must provide all the resources to the SV, this could pose a significant problem to the SS, especially if all the other attached payloads continued to be operated using SS resources. Also, providing housing and food for the SV assembly crew would be a problem for the SS, since the SS would not normally be able to accommodate that many additional people. If the SV has to provide its own resources, that could necessitate the deployment of SV solar-energy-

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TABLE 2. POTENTIAL SS SUPPORT OF SV IN ATTACHED MODE

OPTION	PROVISION	FURNISHED BY		REMARKS ***
		SV	SS	
1	• CREW TIME	√ (MOST)	√ (SOME)	MAJOR IMPACTS TO THE SS HAB & RESOURCES CAPABILITY
	• HABITABILITY *		√	
	• RESOURCES **		√	
2	• CREW TIME	√ (MOST)	√ (SOME)	SIGNIFICANT IMPACTS TO THE SS RESOURCES
	• HABITABILITY *	√		
	• RESOURCES **		√	
3	• CREW TIME	√ (MOST)	√ (SOME)	POTENTIAL IMPACTS TO PHYSICAL CLEARANCES & FIELDS OF VIEW
	• HABITABILITY *	√		
	• RESOURCES **	√		

* INCLUDES HABITABILITY VOLUME, ECLSS, FOOD, ETC.
** INCLUDES POWER, COMMUNICATIONS, HEAT REJECTION, ETC.
*** CONTROLLABILITY NOT ASSESSED HERE – PROBABLY A MAJOR IMPACT

collecting devices, radiators, antennas, etc., which could quickly exacerbate the "real estate" and field-of-view situation.

ASSEMBLY OF SV WHILE CO-ORBITING WITH SS

Allowing the SV to co-orbit in the vicinity of the SS (see Figure 5), appears to offer significant potential advantages. Here, there is sufficient isolation and independence between the SV and SS to minimize interference with the SS, yet the SV could benefit from using some of the SS resources or equipment as part of the normal mode of operation (see Table 1). In an emergency, the proximity of the SS to the SV would allow use of the SS (or vice versa) for backup in a number of areas, such as those listed in Table 1.

The only significant impact to the SV to operate in the vicinity of the SS would be the propellant required to maintain proper orbit phasing. The quantity of propellant required for this activity has not been assessed, but would be a function of the degree of tolerance allowed in the orbit separations. For close tolerances, this might get to be a sizable quantity. The SS might provide part of the delta velocity required to maintain phasing, if this is cost-effective.

If the SV is assembled while in an orbit in the proximity of the SS (but not attached to the SS), an assembly system (structure and other subsystems) may be required. If it is required to have such a system, the central portion of the SS upper transverse boom (the portion between the two rotating alpha joints) and part of the keel, if necessary,could serve as the basis for such an element (see Figure 6). This structure is an open truss framework, expandable or erectable on orbit. The transverse boom contains the attitude control sensors, control moment gyros (CMG's), communications equipment, power conversion and conditioning equipment, and deployable radiators for heat rejection. This piece of equipment is an integrated free-flying element capable of providing its own stabilization, control, and resources, and provides resources to the user. If needed, the two gimbal joints and the solar dynamic (or solar array) energy collection elements can be included as part of the assembly. This total complement of equipment is used (together with the experiment accommodation portion of the SS truss) as the basis for some concepts of the SS unmanned platforms in the SS program. The SS Mobile Remote Manipulator System (MRMS) is designed to


FIGURE 5
Assembly of Manned Mars Space Vehicle
MSFC-5/85 PA 4000-591

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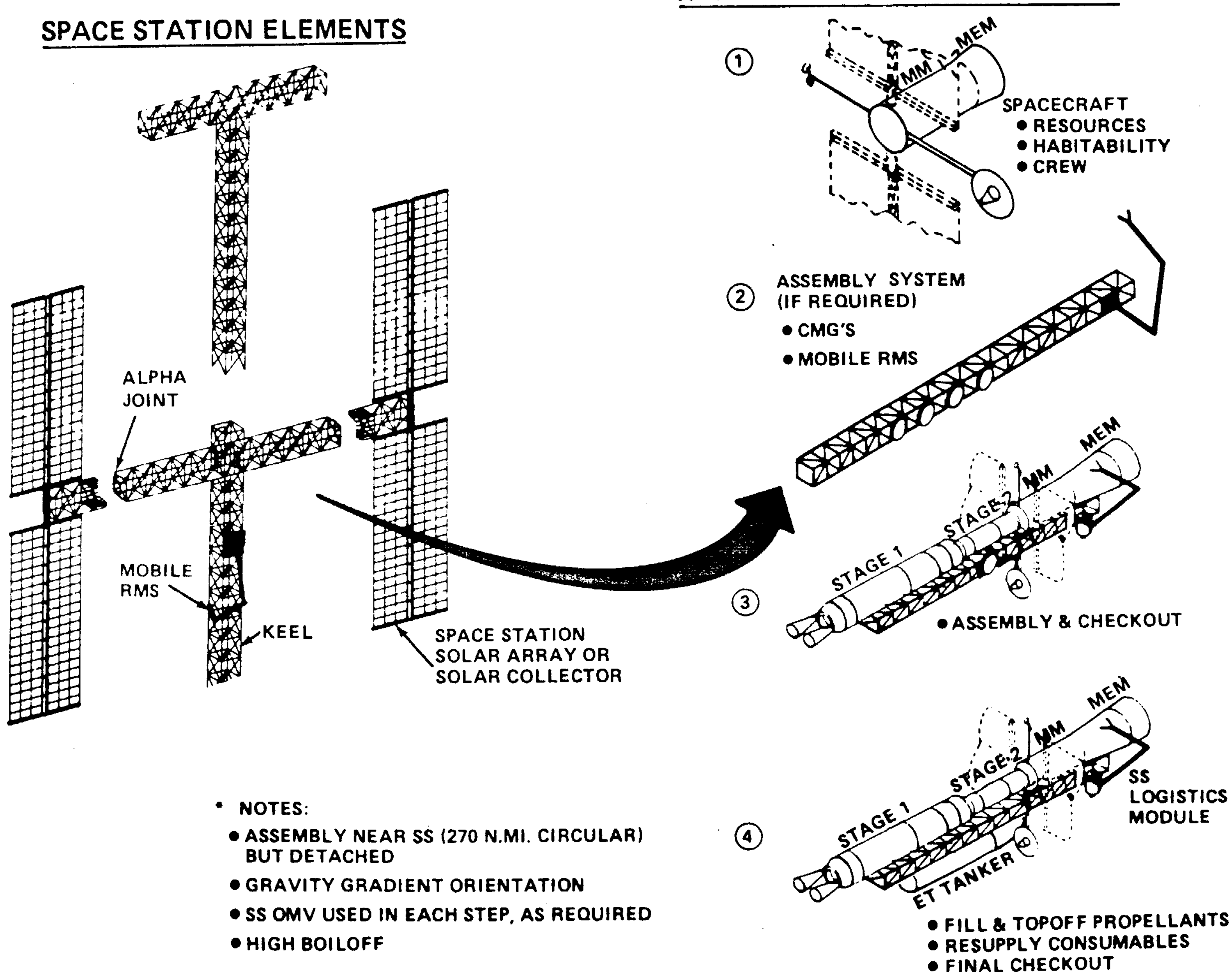
FIGURE 6. LOW EARTH ORBIT ASSEMBLY CONCEPT



traverse along the truss members, and can be used to aid in assembly of the SV.

If the on-orbit assembly time of the SV can be kept fairly short (a few weeks or months), the SV modules and systems might be used for habitability, power, communications, etc. during assembly. This would allow some burn-in time on these systems, and would allow the assembly system (if used) to be less complex and costly. If the assembly time is long, however, too much of the SV systems' lifetime might be expended, and the assembly system instead would need to provide the necessary habitability and resources during the assembly phase. It was assumed in this limited study that the assembly system would be kept simple and that the SV systems would be used during the assembly phase. The assembly system would be left on LEO after departure of the SV. An augmented co-orbiting platform might even serve as an assembly system.

The OTV and OMV would be useful in the co-orbiting mode of SV assembly for the same functions identified in the discussion on the attached assembly mode.

HYBRID MODE

Each of the other modes has advantages and disadvantages. The attached mode is more convenient, but disruptive. The co-orbit mode is less disruptive, but adds the expense of a separate assembly system and the mass of station-keeping propellant.

In the early years of a Mars program, with flight rates of about one per 2 years, a separate assembly system might not be very cost-effective, since it would be dormant for long periods. In later years this should change somewhat (although a Mars program will always tend to have greater functuations in activity levels from year to year than most other programs, due to the scarcity of flight opportunities). Program maturity thus might be a factor in determining the mode of assembly.

A hydrib mode in which the SV would be attached to the SS during early phases of SV assembly, then would be separated and co-orbit with the SS during later phases, might be an optimum mode and should be investigated further.

Futher study must be done to determine the most effective mode of utilizing the SS, but it appears that a high degree of usability should be possible.

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N87-17807

SPACE STATION SUPPORT OF MANNED MARS MISSIONS

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ABSTRACT

The assembly of a manned Mars interplanetary spacecraft in low Earth orbit can be best accomplished with the support of the Space Station. Station payload requirements for microgravity environments of $10^{-3}g$ and pointing stability requirements of less than 1 arc second could mean that the spacecraft may have to be assembled at a station-keeping position about 100 meters or more away from the Station. In addition to the assembly of large modules and connective structures, the manned Mars mission assembly tasks may include the connection of power, fluid, and data lines and the handling and activation of components for chemical or nuclear power and propulsion systems. These assembly tasks wilL require the use of advanced automation and robotics in addition to Orbital Maneuvering Vehicle and EVA crew support. Advanced development programs for the Space Station, including on-orbit demonstrations, could also be used to support manned Mars mission technology objectives. Follow-on studies should be conducted to identify Space Station activities which could be enhanced or expanded in scope (without significant cost and schedule impact) to help resolve key technical and scientific questions relating to manned Mars missions.

INTRODUCTION

The assembly of a manned Mars spacecraft will occur where long duration crew and robotics support will be available to insure that the spacecraft can be safely assembled and that the spacecraft systems are functioning as expected. The Space Station is the only planned facility capable of supporting these assembly and checkout operations in a cost-effective manner.

While the manned Mars mission (and Lunar missions) are being included in the Space Station mission data base, they can not be considered as drivers for the initial orbital capability (IOC) of the Station. However, they are encompassed by the Station's growth phase and therefore will be considered in the establishment of IOC interfaces (scar) which support growth requirements. Thus it is important that Phase B Space

Station definition studies consider the potential impacts of manned Mars missions on growth accommodation provisions. It is equally important that the planners of manned Mars spacecraft and missions be aware of the capabilities and limitations of Space Station support for assembly, departure, return, and refurbishment phases of manned Mars missions.

MARS SPACECRAFT ASSEMBLY SUPPORT

Space Station Motion and Disturbance Dynamics

The IOC Space Station will be designed to accommodate payloads which will require a 10^{-5}g microgravity environment and which will attempt to achieve sub-arcsecond pointing stability through the use of station (coarse) and user-provided fine pointing and stability systems. Meeting these payload requirements will be a substantial challenge for the Space Station.

Gravity gradient disturbances and drag effects on the Space Station will practically limit the microgravity environment achievable to 10^{-6}g. Many normal Station activities including crew movements may have to be restricted during critical payload operations' phases to achieve the Station's microgravity goal. Activities such as Orbiter berthing (10^{-2}g), crew wall pushoff (10^{-3}g), and crew console operations (10^{-4}g) will have to be timelined around or restricted during certain materials processing payload operations and solar and stellar observations if adequate isolation techniques are not developed by IOC.

The growth Space Station operations may not be as restrictive if most of the materials processing payloads and experiments move onto co-orbiting platforms where much lower microgravity environments may be achievable. Alternatively, the Space Station may be replicated, providing a Station for microgravity and sub-arcsecond payloads and a Station which is dedicated to servicing and transportation node functions.

However, it is possible that combination of budget constraints and mission times would dictate that only the IOC Space Station will be available for initial support of a Mars Mission. As one planning option, we should assume that the IOC Station must be able to support a manned Mars mission and determine how it can best be used to support Mars spacecraft assembly and checkout.

Attached and Station-Keeping Assembly and Checkout

The initial Mars spacecraft element to be brought up would likely be a Mission Module with life support systems, data management systems, communications and telemetry systems, habitability systems, and a power distribution and storage system. This module could be temporarily berthed at the Station, where the systems could be activated and checked out by the Station crew. The Mission Module would likely be common with Space Station modules. If the common module has a length of 49.5 feet (one of the Station options under study), it would fill a Shuttle cargo bay.

During this checkout phase of one day or so, micro-g and some solar and stellar operations could be suspended for a short time, if necessary. Once the systems have been activated and checked out, the Mission Module could be attached to the Station in a quiescent mode until additional spacecraft elements are brought up. Alternatively, if the Mission Module substantially alters the location of the Station's center of gravity to the detriment of materials processing payloads, the Mission Module could be moved to a position 100 meters or more from the Station for station-keeping operations. (See Figure 1) A temporary (or built-in) attitude control system package could be attached and activated. A contingency power system, probably of the same type used for the Space Station, could be erected until the primary power system is brought up and attached in the next Shuttle flight. If the next Shuttle can be scheduled within a week, then the suspension of certain Space Station materials processing activities for that period of time could probably be negotiated successfully.

Four to six Shuttle flights might be needed to bring up and assemble the power and propulsion elements. For a chemical propulsion system, additional flights would be needed to attach the required propellent tankage. An additional three or four flights could be used to bring up the second Mission Module, the Mars Excursion Module (landing vehicle), the module outfitting equipment (if required) and other scientific equipment. The second Mission Module would be checked out at the Space Station in a manner similar to the first module. The Mars Excursion Module could also undergo preliminary checks while attached to the

SERVICING PLATFORM AND MOBILE ROBOTS FOR MANNED MARS MISSION SPACECRAFT ASSEMBLY

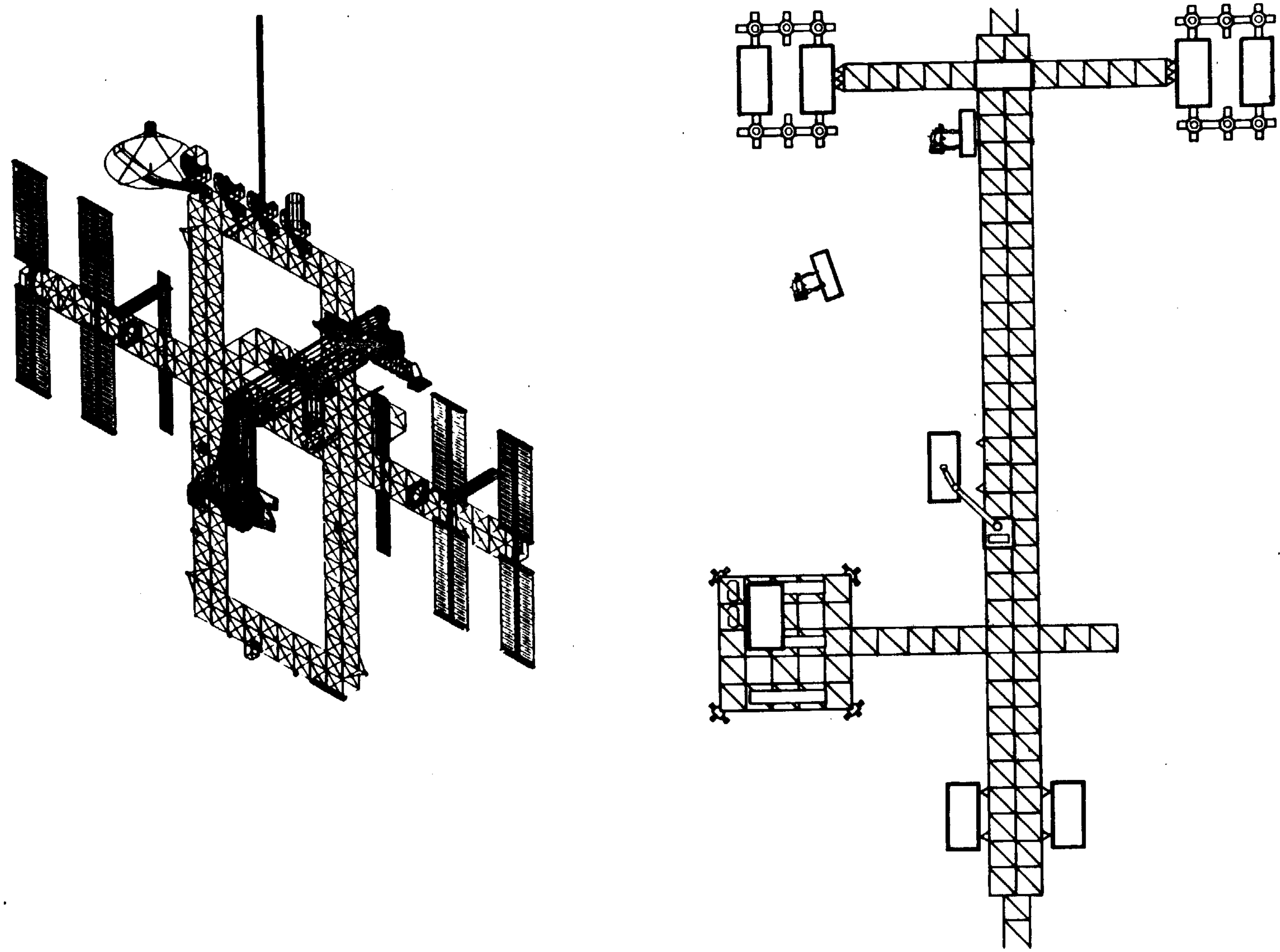

FIGURE 1

Station or could be checked out from a Mission Module after attachment to the Mars spacecraft.

The Space Station modules may be launched without internal outfitting, to allow utilization of larger modules and/or achieve higher orbit deployments. If the same procedure is required for the Mars Mission Modules, outfitting could be accomplished while the modules are attached to the Station for the initial systems activation and checkout. Tradeoffs between potential vibration disturbances of Space Station payloads and the advances of outfitting activities which could take several days would have to be made. With good planning and adherence to mutual schedules, this should not be any more of a problem for the Space Station than a normal Orbiter logistics visit.

Activation and check out of the power system and propulsion and attitude control systems would be conducted, in part, remotely from the Station. Activation and checkout would be completed by crewman in the Mission Module. For a nuclear propulsion system, remote activation and extensive checkout of the power and propulsion systems would be required to insure a safe environment prior to sending a crewperson to the Mission Module for final checks. Once the manned Mars spacecraft has been assembled, integrated systems tests would be conducted. These tests would include the exercise of contingency procedures and possible active tests of propulsion systems.

AUTOMATION AND ROBOTICS

Automation and robotics will play an important role in supporting many Space Station and platform activities. For servicing of the platforms and freeflyers at the Station, an Orbital Maneuvering System with a "smart front end" and the Module Remote System will almost certainly be required. In addition, the servicing area at the Station may require additional automation and possibly some robotics to assist with the connection of fuel lines and refueling operations, to replace modular elements, to resupply cryogen systems, and to assist with repairs. These activities will probably be conducted with substantial EVA crew and Mobile Remote Manipulator support. The availability of automated and robotic elements will reduce the amount of EVA required and the risks to the EVA crew associated with fluid transfers and possible spills.

In addition, some advanced robotic systems, such as a mobile robot, will be studied and may be available for use in the assembly of large space structures, for the handling of nuclear components and to assist in certain laboratory module operations. Congress has mandated that the Space Station program be used to support the development of automation and robotics which extends state-of-the-art applications and benefits the U.S. industry. Dependent upon the amount of Space Station funding that is available for IOC and Growth phase advanced development, a multi-purpose mobile robot could emerge from the program and would be available to support the assembly of a manned Mars spacecraft, inflight Mars spacecraft maintenance and repairs, and Mars surface activities.

A mobile robot coupled with an OMV with a "smart front end" could be used to handle active nuclear elements and conduct operations in areas with relatively high radiation levels which would help enable a more extensive use of nuclear power systems in space. These operations have their parallel in potential industrial co-orbiting platforms, which may require the use of nuclear power and nuclear electric propulsion.

SPACE STATION RENDEZVOUS AND SPACECRAFT MAINTENANCE

Parking Location for Mars Spacecraft

For a spacecraft which departs from and returns to the Space Station the natural parking location would be in an orbit co-planar with the Space Station. If a nuclear power system is used, nuclear electric propulsion would be the best choice to keep the spacecraft relatively close to the Space Station. The nuclear powered spacecraft could be brought into a station-keeping position on its own power or with the assistance of the OMV for maintenance and additions such as the replacement of an engine, a replacement for the Mars Excursion Module left on the surface and on Mars orbit, and outfitting of the Mission Modules and Logistics Modules for Mars base construction activities.

A non-nuclear powered spacecraft may have to be allowed to drift away from an optimum orbital location to reduce the amount of attitude control propellent required to maintain station-keeping. For maintenance activities, the attitude control and propulsion systems (partly re-supplied after the initial return to the Station) could be used to bring the spacecraft to the vicinity of the Station where the OMV can be used to assist in the final positioning.

If a manned Orbital Transfer Vehicle is available, other options could be considered, including a parking location of the Mars spacecraft in a much higher orbit to which a crew could be ferried. However, unless a GEO Space Station or an advanced robotics mobile servicing facility is available, maintenance and reassembly of a Mars spacecraft may be costly and time consuming in a higher orbit.

Space Station Isolation Quarters or Clean Room

The IOC Space Station reference configuration does not currently have isolation quarters or a clean room. Requirements for Mars and comet sample return missions using the Space Station as a transportation mode may result in the development of a clean room capability or the addition of a special module for this purpose. However, it is possible that provisions for the prevention of crew exposure to toxic gases or materials could result in the capability to close off or isolate a Laboratory Module. This module or a combination Habitation/Laboratory Module (a possible Space Station study option) could be used for temporary crew isolation, if necessary.

Spacecraft Maintenance and Reassembly

Spacecraft maintenance and reassembly activities in preparation for a return to Mars would be expected to utilize the same capabilities used in the initial spacecraft assembly activities. Resupply of propellent could take place by accessing Station propellent storage facilities, by use of a co-orbiting, mobile propellent storage facility (potential Space Station growth phase option), or by the use of Shuttle fuel tankers. Spacecraft propellent resupply operations near active or used nuclear systems would require more extensive use of mobile robotics and OMV activities to avoid crew exposure to radiation substantially higher than background radiation.

SPACE STATION/MARS SPACECRAFT SYSTEMS COMMONALITY

Space Station Common Module and Logistics Module use for Mars Spacecraft

A Space Station Common Module is being developed for the Habitability and Logistics Modules. These modules will have certain common infrastructure such as primary structure, power, data and thermal subsystems and interfaces. The module control and display and working area layouts are expected to be oriented along the long axis of the module, which would optimize the amount of usable space for artificial gravity orientations of Mars spacecraft Mission Modules.

While no spare modules are being planned for the IOC Station, growth phase requirements will mean that the capability to produce additional Common Modules will have to be maintained. For a manned Mars mission in the post 2000 period, it should be possible to place orders for Common Modules from the Space Station contractors and take advantage of IOC module experience in the outfitting of the Mars spacecraft modules.

The use of the Space Station Logistics Module should be considered for storage of certain consumables and for later mission use in the transfer of Mars base construction materials to the surface with an advanced Mars Excursion Module. Alternatively, unmanned missions could be used to land Mars base construction materials on the surface prior to first and/or subsequent missions.

COMMON DEVELOPMENT OF ADVANCED POWER, ATTITUDE CONTROL, AND PROPULSION SYSTEMS

The Space Station IOC power system is expected to be designed such that the transition to solar dynamic or nuclear power systems will cause a minimum of system reconfiguration. As a goal, the power distribution subsystem would have a standard interface with the power generation subsystem which would not change. In the process of developing power systems for the growth Space Station, design features could be considered which would enhance the ease of integrating these systems into a manned Mars spacecraft and a Mars base.

Resistojets and nuclear electric propulsion will be investigated for use on the IOC and growth Stations and platforms. Their performance parameters could be influenced (within certain cost/schedule constraints) by the requirements of manned Mars spacecraft activities. Advanced studies or development funding associated with manned Mars missions could be used to actively support the consideration of Mars mission requirements in Space Station definition and advanced development activities.

SPACE STATION TECHNOLOGY DEVELOPMENT AND EXPERIMENT SUPPORT

There are a number of advanced development activities and technology and scientific experiments being planned for the Space Station and Shuttle which will support some of the key technical and scientific areas of concern for manned Mars missions. A close interaction with these activities and their results by manned Mars mission interests is needed. For some activities, a more direct infusion of manned Mars mission concerns should be attempted.

If the National Commission on Space and/or NASA Headquarters chooses to outline some broad, long-range goals which include lunar and manned Mars missions, it should be possible to begin influencing ongoing Station definition studies, advanced development studies and technology in support of future programs. In Space Station and other government studies and research, the following areas, among others, should be investigated:

1. Long Term Weightlessness Effects and Countermeasures.
2. Progress Towards a Closed Environmental Control and Life Support System.
3. Development of Cryogenic Propellent Storage, Handling, Gauging, and Transfer Capabilities.
4. Large Space Structure Assembly and Construction Techniques
5. Plant Growth Techniques for Zero-G
6. Nuclear power and Propulsion Studies
7. Laser Communications and Positioning Systems
8. Automation and Robotics Studies
9. Inflight Training Techniques and Capabilities
10. Advanced Shielding Techniques

CONTINUING STUDY ACITIVITIES

In the area of Space Station program support of manned Mars mission studies and planning, each of the following activities should be pursued:

1. Identify specific Space Station design decisions which should be influenced by manned Mars mission planning - determine approach for inserting relevant information into the decision process.
2. Identify the advanced development activities which should be enhanced to support or otherwise consider manned Mars mission activities.
3. Determine the design studies and operations issues which must be addressed to support the assembly and maintenance of nuclear or other large power systems in space.
4. Identify specific technology demonstration activities which can support both the growth phase of the Space Station program

and manned Mars missions (such as mobile robots, advanced, active or passive shielding, closed environmental control systems, holographic imaging systems for onboard training, advanced propulsion systems, and artificial intelligence systems) and determine an approach for establishing joint study activities.

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N87-17808

STRATEGIC DEFENSE INITIATIVE IMPACTS ON MANNED MARS MISSIONS

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ABSTRACT

Research Conducted on a strategic defense system with space based elements may provide key components of systems necessary for Manned Mars Missions. Three areas of impact are Space Logistics, Space Power, and Supporting Systems. These are discussed briefly in this paper.

SPACE LOGISTICS

Emplacement, operation and maintenance of a reliable strategic defense system will necessitate enhancements in space logistics. These include the need to emplace payloads in excess of 100 MT (220,000 lbs.) in LEO, as well as sensor and weapons platforms in a variety of orbits (LEO, 1000 km, 10,000 km, GEO, 3xGEO). For example, boost phase surveillance and interception may require sensor platforms in GEO of 10 MT. The space logistics system must provide a capability to enhance and maintain these payloads. The SDI research effort has the stated goal of cost reduction to $300/lb for payloads into LEO (as a necessary prerequisite for practical defense systems). This will necessitate the development of a new generation of heavy lift launch vehicles. Such technologies will impact the manned Mars mission by providing the means to lift the large (500 klb -> 3 Mlb) LEO systems described with fewer launches and lower costs for the overall mission. In addition, on orbit logistical service may provide a relevant experience base for system buildup in LEO. The advanced power systems developed will also contribute to advanced electric propulsion for interplanetary orbital transfer.

SPACE POWER

The SDI research effort in power is driven by the need to provide standby, operational, and sprint power to weapons systems and operational power to sensor platforms. Mid-course discrimination sensor concepts involve power levels of 1-5 MW electric. Weapons concepts currently under investigation necessitate sprint power (1000's of seconds) operation at 1 - 100 MW. Systems being examined to provide these power levels include advanced chemical and nuclear options. In addition to increased power levels, advances must be made in power conditioning, precisely delivering and distributing the power to its eventual point of use at the required rate.

Manned Mars mission concepts described will be greatly influenced by the development of 1-5 MW (electric) systems for use in propulsion and surface activity. The existence of the SDI technology base will enhance the attractiveness of a decision to proceed with a Mars program focused on repeated visits and building of a permenent base.

SUPPORTING SYSTEMS

Advanced laser pointing and attitude determination systems, multi-purpose laser systems for laser communications, welding repair tasks, and varied automation and robotics systems required for on-orbit inspection and maintenance may find application on manned Mars missions.

APPENDIX A

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Battelle	Battelle Pacific Northwest Laboratories
CSC	Computer Sciences Corporation
EEI	Eagle Engineering, Inc.
GSFC	Goddard Space Flight Center
HQ	NASA Headquarters
JPL	Jet Propulsion Laboratory
JSC	Johnson Space Center
KSC	Kennedy Space Center
LANL	Los Alamos National Laboratory
LeRC	Lewis Research Center
LSPI	The Large Scale Programs Institute
MSFC	Marshall Space Flight Center
NCAR	National Center for Atmospheric Research
NOAA	National Oceanic and Atmospheric Administration
NRL	Naval Research Laboratory
ODU	Old Dominion University
U of HI	University of Hawaii
UAH	University of Alabama in Huntsville
UNM	University of New Mexico
USAF	U. S. Air Force
USGS	U. S. Geological Survey

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