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A RAND NOTE

Space Transportation Systems, Launch Systems, and Propulsion for the Space Exploration Initiative: Results from Project Outreach

T. Garber, J. Hiland, D. Orletsky,
B. Augenstein, M. Miller

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Prepared for the
United States Air Force
National Aeronautics and Space Administration

RAND

PREFACE

This Note describes the findings of the Space Transportation Systems, Launch Systems, and Propulsion panel, one of eight project panels established by RAND to evaluate submissions to the Space Exploration Initiative (SEI) Outreach Program, also called Project Outreach. Project Outreach is a NASA-sponsored program to elicit innovative ideas, concepts, and technologies for space exploration. The project was sponsored by Project AIR FORCE and by RAND's Domestic Research Division, with technical oversight provided by the Assistant Secretary of the Air Force (Space).

The findings of other RAND panels are reported in the publications listed below.

Space and Surface Power for the Space Exploration Initiative: Results from Project Outreach, by C. Shipbaugh, K. Solomon, and M. Juncosa, with D. Gonzales, T. Bauer, and R. Salter, N-3280-AF/NASA, 1991.

Automation and Robotics for the Space Exploration Initiative: Results from Project Outreach, by D. Gonzales, D. Criswell, and E. Heer, N-3284-AF/NASA, 1991.

Human Support Issues and Systems for the Space Exploration Initiative: Results from Project Outreach, by J. Aroesty, R. Zimmerman, and J. Logan, N-3287-AF/NASA, 1991.

SUMMARY

This Note analyzes a number of transportation and propulsion options for Mars exploration missions. As part of Project Outreach, RAND received and evaluated 350 submissions in the launch vehicle, space transportation, and propulsion areas. After screening submissions, aggregating those that proposed identical or nearly identical concepts, and eliminating from further consideration those that violated known physical principles, we had reduced the total number of viable submissions to 213.

In order to avoid comparing such disparate things as launch vehicles and electric propulsion systems, six broad technical areas were selected to categorize the submissions:

- Space transportation systems
- Earth-to-orbit (ETO) launch systems
- Chemical propulsion
- Nuclear propulsion
- Low-thrust propulsion
- Other

To provide an appropriate background for analyzing the submissions, an extensive survey was made of the various technologies relevant to the six broad areas listed above. We discuss these technologies with the intent of providing the reader with an indication of the current state of the art, as well as the advances that might be expected within the next 10 to 20 years.

Technology alone is not a sufficient basis for judging the merits of the submissions. It is also necessary to examine the submissions in the context of their usefulness to various space transportation and ETO launch system options.

For space transportation options, the operational figures of merit that are of interest are (1) initial mass in low Earth orbit (IMLEO) and (2) transit times to and from the destination planet. It is desirable for both of these parameters to have low values. In the case of ETO systems, the usual figures of merit are payload-per-launch, payload-per-unit-time, and cost-per-pound of payload delivered to LEO. In this Note we do not address costs.

A number of space transportation options are available, and submission concepts that could be used to either implement or support a particular option were examined. Both nonnuclear and nuclear options were considered. Of the space transportation options examined, two approaches appear particularly interesting—split missions and the use of in-situ propellants. A split mission uses cargo spacecraft that follow low-energy trajectories to pre-position the mass needed for Mars exploration and Earth return in Mars orbit. A small vehicle on a high-energy trajectory would be used for crew transportation. The use of in-situ propellants offers the potential for large reductions in IMLEO. Both the Moon and the Martian system offer materials that can be used to produce suitable rocket propellant; however, infrastructure is required.

To reduce IMLEO and trip time substantially, nuclear systems should be considered. Several nuclear systems are discussed in this Note, ranging from propulsion systems incorporating modest modifications over the ROVER/NERVA program to fusion/antimatter propulsion systems. It is clear that research priorities and specific areas of research should be considered in the light of policy with regard to the use of nuclear systems in space.

Earth-to-orbit launch system options were examined in detail. These systems range in payload from a few thousand pounds to over half a million pounds. Included in this group are electromagnetic launch systems, the Shuttle, Shuttle-derived vehicles, advanced launch systems, the national aerospace plane, Saturn V upgrades, and ultra-large lift vehicles. Given the magnitude of IMLEO requirements that appear to be necessary, an ETO transportation system with a large payload capability appears to be desirable.

Most notably, we found that almost all of the space transportation options we considered would benefit from the availability of orbital transfer systems that can economically transfer large masses from LEO to high Earth orbits and cis-Lunar space. In addition, all of these space transportation options could benefit greatly from the development of propellant sources either on the Moon, on Martian systems, or on both. Although engineering feasibility has not yet been demonstrated for the advanced nuclear concepts we considered, the performance potential warrants a research program to identify those concepts best suited for development, assuming the use of nuclear propulsion in space is permissible.

In summary, we have received many interesting submissions through the NASA Outreach Program. We recommend that the following submissions be considered further by the Synthesis Group:

- **Lunar/Mars Return Propulsion System (#100767)**
- **High-Energy Chemical Propulsion for Space Transfer (#101212)**
- **The Pony Express to Mars (#100714)**
- **Lunar-Derived Propellants (#100932)**
- **In-Situ Propellants for Mars Lander—Chemical Engines (#101178)**
- **Solar Electric Orbital Transfer Vehicle (SEOTV) (#101157)**
- **Pulsed MPD Electric Propulsion (#100170)**

- **Earth-Based Microwave Power Beaming to Interorbital (LEO to and from HEO) Electrically Propelled Transport Vehicles (#101536)**
- **The "Enabler," A Nuclear Thermal Propulsion (NTP) System (#100933)**
- **Low Pressure Nuclear Thermal Rockets (LPNTRs) (#100157)**
- **NIMF Concept to Enable Global Mobility on Mars (#100103)**
- **Heavy-Lifting Launch Vehicle Concept (#100192)**
- **A Fall-Back-to-Spring-Forward Strategy to a Heavy-Lift Launch Vehicle: Reviving Saturn V Technology (#100185)**
- **Ultra Large Launch Vehicle (ULLV) for Moon and Mars Missions (#100110)**

ACKNOWLEDGMENTS

We are grateful to many people for their help during Project Outreach. We would like to thank all of the individuals who provided us with information during our visits to the Air Force Astronautics Laboratory and Marshall Space Flight Center. We wish to especially thank Dave Perkins and Bob Durrett for organizing and coordinating those visits. We also appreciate the technical expertise in the field of nuclear physics provided by our RAND colleague Calvin Shipbaugh. We are grateful to Jack L. Kerrebrock for his careful technical review of this document. We also wish to thank Mark Nelsen and Jerry Sollinger for improving the structure of this document, and John Barrymore and Elaine Wagner for preparing it for publication.

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ACRONYMS AND ABBREVIATIONS

AIAA	American Institute of Aeronautics and Astronautics
ALDP	Advanced Launch Development program
ALS	advanced launch system
AMEROC	American Rocket Company
AMLS	advanced manned launch system
ASRM	advanced solid rocket motor
atm	atmosphere
AU	astronomical unit
B	magnetic field
β	ratio of plasma particle pressure to magnetic field pressure
C	Celsius
DC	direct current
deg	degree
DT	deuterium tritium
ECCV	earth capture crew vehicle
ECR	electron cyclotron resonance
EML	electromagnetic launcher
EOS	equation of state
ET	external tank
ETO	Earth-to-orbit
ETR	Eastern Test Range
EVA	extravehicular activity
FFQ	full flight qualification
FLOX	fluoride and liquid oxygen
FRP	flight-rated prototype
FSE	full-scale engineering
ft	foot
g	gram
g,g's	gravity (Earth)
GCR	gas core rocket
	galactic cosmic radiation
GEO	geosynchronous Earth orbit
GeV	gigaelectronvolt
GW	gigawatt
HEO	high Earth orbit
HLLV	heavy-lift launch vehicle
hr	hour
ICF	inertial confinement fusion
IFR	internal fusion rocket
IMLEO	initial mass in low Earth orbit
in.	inch
I_{sp}	specific impulse
IUS	inertial upper stage
J	joule
JPL	Jet Propulsion Laboratory
K	Kelvin
keV	kiloelectronvolt
kg	kilogram
km	kilometer
kN	kiloNewton
ksec	kilosecond
kW	kilowatt

kWe	kilowatt electric
lbf	pound force
LCO	liquid carbon monoxide
LEO	low Earth orbit
LEP	laser electric propulsion
LLO	low Lunar orbit
LMO	low Martian orbit
LOX	liquid oxygen
LPNTP	low-pressure nuclear thermal propulsion
LPNTR	low-pressure nuclear thermal reactor or rocket
LTP	laser thermal propulsion
m	meter
MCF	magnetic confinement fusion
MEP	microwave electric propulsion
MET	microwave electrothermal
MEV	Mars excursion vehicle
MeV	megaelectronvolt
min	minute
Mj	megajoule
MPD	magnetoplasma dynamic
M_T	thrust efficiency
MTP	microwave thermal propulsion
MTV	Mars transfer vehicle
MW	megawatt
NASA	National Aeronautics and Space Administration
NASP	national aerospace plane
NDR	NERVA-derived technology
NDV	NASP-derived vehicle
NEP	nuclear electric propulsion
NERVA	nuclear energy for rocket vehicle applications
NF	Nuclear Furnace
NIMF	nuclear rocket using indigenous Martian fuel
nmi	nautical mile
NTP	nuclear thermal propulsion
NTR	nuclear thermal rocket
OTA	Office of Technology Assessment
OTV	orbital transfer vehicle
P/L	payload
PBR	particle bed reactor
PLS	personnel launch systems
ppm	parts per million
psi	pound per square inch
psia	pound per square inch absolute
SDV	Shuttle-derived vehicle
sec	second
SEI	Space Exploration Initiative
SEP	solar electric propulsion
SRM	solid rocket motor
SSF	space station freedom
SSME	Space Shuttle main engine
ST	spherical torus
STME	space transportation main engine
STP	solar thermal propulsion
STS	Shuttle Transportation System
T/W	thrust-to-weight ratio
T_B	boiling temperature
TEI	trans-Earth injection

TMI	trans-Mars injection
TOGW	takeoff gross weight
ULLV	ultra-large launch vehicle
USAF	United States Air Force

I. INTRODUCTION

This Note documents the analyses and evaluations of the Space Transportation Systems, Launch Systems, and Propulsion panel (hereinafter called simply the Transportation panel), one of eight panels created by RAND to screen and analyze submissions to the Space Exploration Initiative (SEI) Outreach Program. In addition to managing and evaluating the responses, or submissions, to this public outreach program, RAND conducted its own analysis and evaluation relevant to SEI mission concepts, systems, and technologies. The screening and analysis of Project Outreach submissions were conducted on an accelerated schedule between July and October 1990, and involved staff and consultants throughout RAND's departments and research divisions.

The eight panels created to screen and analyze the submissions encompassed

- Space and Surface Power
- Space Transportation Systems, Launch Systems, and Propulsion
- Structures, Materials, Mechanical Systems, and Extraterrestrial Resource Utilization
- Automation and Robotics
- Communications
- Human Support
- Information Systems
- Architectures/Missions

This Introduction describes the background of the SEI, the overall methodology used in submission handling, the analysis procedures, and some general results and observations.

BACKGROUND

President Bush has called for a Space Exploration Initiative that includes establishing a permanent base on the Moon and sending a manned mission to Mars. The national space policy goals developed by the National Space Council and approved by President Bush on November 2, 1989, were the following:

- Strengthen the security of the United States.
- Obtain scientific, technological, and economic benefits.
- Encourage private sector investment.
- Promote international cooperative activities.
- Maintain freedom of space for all activities.

- Expand human presence and activity beyond Earth orbit into the solar system.

To support these goals, Vice President Quayle, Chairman of the National Space Council, asked NASA to take the lead in identifying new and innovative approaches that will be required to travel to the Moon and Mars and to live and work productively on both. In response to the President's announcement, NASA conducted a 90-day study (commonly referred to as "the 90-Day Study" [NASA,1989]) that presented a variety of strategies for accomplishing the objectives. It also solicited new ideas and concepts for space exploration through the SEI Outreach Program, which consists of three principal efforts:

1. Direct solicitation of ideas from academia, nonprofit organizations, for-profit firms, and the general public.
2. Review of federally sponsored research.
3. A study by the American Institute of Aeronautics and Astronautics (AIAA).

The results of the three efforts listed above will be presented to a Synthesis Group chaired by Thomas P. Stafford, Lieutenant General (ret.), USAF. The recommendations of the Synthesis Group will, in turn, be reviewed by NASA. From this process, a number of alternative mission paths will emerge, from which NASA may select several for detailed study over the next few years. In addition, the process is expected to yield innovative technologies and system concepts for possible development.

GENERAL OBSERVATIONS ON THE SUBMISSIONS

Our first observation was that the submissions did not contain any new scientific laws or principles, or wholly new areas of technology. For example, some submissions suggested applications of high-temperature superconductivity, which five years ago could have been considered a new technology. However, superconductivity was first discovered in the early 1900s, and the possibility of high-temperature superconductors was discussed soon afterward, so it should be understood that "new" technology areas are a matter of perspective.

The submissions did contain, however, a number of old ideas that have new implications in the context of the SEI. For example, several submissions included the concept of a spacecraft orbiting at a libration point, a concept that has been proven by NASA's International Sun-Earth Explorer-3, which was put into orbit around the sun-Earth libration point, L-1, in 1978. Libration concepts take on considerable new meaning in the context of potential use as transportation nodes for a Mars mission.

The submissions also contained ideas that had not been heretofore supported by the submitter's organization, which may have been an industrial firm, university, or NASA itself.

This is a natural consequence of the priority planning process and resource allocation decisions of each individual organization. Thus, many of the submitted ideas were not completely new, but simply had not received much support.

Lastly, we observed that the submissions were sufficiently diverse to support a wide range of SEI mission concepts and architectures.

THE SUBMISSION PROCESS

Figure 1.1 presents a flow diagram of the Outreach evaluation process. RAND mailed out 10,783 submission packets in addition to the 34,500 that were mailed out by NASA. A total of 1697 submissions were received and were initially processed by a subcontractor firm, KPMG Peat Marwick. Of the 1697 submissions received, 1548 were judged by Peat Marwick to contain sufficient information for screening by RAND. The screening process selected approximately 215 submissions for more formal analysis. The output of that analysis process was the set of priority submissions and recommendations reported in this and several companion Notes.

For further discussion of the sources of submissions and their management by RAND, please see App. A.

THE SCREENING PROCESS

The screening process objectives were to

- Assure relative insensitivity to the quantity of submissions.
- Select submissions to be analyzed at length.
- Have each submission reviewed by at least two technical experts working independently.
- Examine robustness by providing more than one ranking method.
- Maintain analytic rigor.

The first objective of the screening process was to assure a good capability to deal with the quantity of submissions, whatever their numbers. Therefore, we established a submission-processing "production line" that was insensitive to the quantity of submissions.

The next task of the screening process was to decide which submissions would be analyzed. We decided that the range and depth of our analysis would have to be a function of (1) the resources available, (2) the perceived quality of submissions across panels, and (3) the relative importance of topics to the overall SEI program. One obvious pair of very important panels (because of the tradeoffs between them) consisted of the Human Support and Transportation panels.

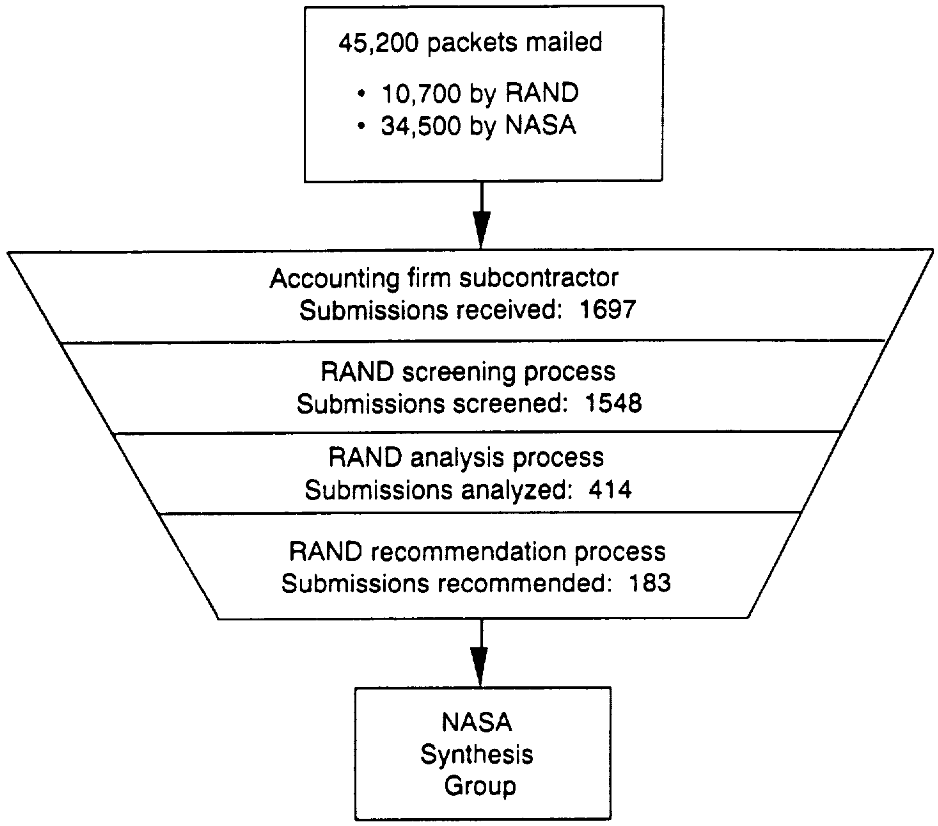


Fig. 1.1—RAND's Outreach Process

In the screening process, each submission was reviewed by at least two technical experts working independently. We allowed for robustness by providing more than one ranking method. A related goal was to maintain analytic rigor through the maintenance of tracking systems to enable later analysis of our methodology.

Multi-attribute decision theory was used in the screening process; i.e., a group of attributes was used to evaluate each submission. The panels chose to score their various submissions using the same five principal attributes:

- Utility
- Feasibility
- Safety
- Innovativeness
- Relative cost

Each panel tailored its own criteria for scoring an attribute according to the panel's specific needs. For example, *safety* meant a very different thing to the Transportation panel than it did to the Information Systems panel.

Attributes were independently scored by two or more reviewers on a scale of one to five, with five being the best. Written justification for the scoring was input into the text field in the database. We used a widely accepted Macintosh relational database, Fourth Dimension by ACIUS, Inc., for storing and using the various information components of each submission.

For each submission, pertinent background information was logged into the database, including the unique ID number of the submission, the reviewer, the date, the name of the panel performing the review, and the title or subject of the review. To remove any bias from the process, the panels did not have information concerning the submitter's name or organization. Reviews of the submissions were entered in a text field. Each reviewer was required to briefly explain the reasons for scoring a submission as he or she did.

If any attribute score varied by more than one among different reviews of the same submission, the submission was reviewed again, this time with the panel chairman participating with each of the original reviewers. However, there was no pressure to reach consensus.

A complete discussion of the quantitative means by which panels used their attribute criteria to rank and evaluate submissions is provided in App. A. The specific criteria used by the Transportation panel in assigning attribute scores are also discussed in App. A.

THE ANALYSIS PROCESS

The object of the analysis process was to select the submissions to be recommended for further consideration by the Synthesis Group. Where possible, we analyzed the submissions quantitatively within the context of the important performance tradeoffs in their respective technical areas.

Each panel prepared a working draft reporting on the results of its analysis in its area of technical responsibility. Each working draft was organized into technical discussions of the important technical subareas identified by that panel. Where possible, important performance tradeoffs in each subarea were examined quantitatively.

Submissions that arrived with no backup paper, i.e., no detailed substantiating information or documentation, were analyzed in the context of the technical discussions of the appropriate subareas, thus providing necessary background. The majority of submissions did not, in fact, include backup papers, making an extended analytical discussion almost mandatory in most cases.

SCOPE OF THE NOTE

This Note presents analyses of various transportation and propulsion options for the Mars exploration mission. As part of Project Outreach, RAND received and evaluated 350 submissions in the launch vehicle, space transportation, and propulsion areas. A list of all evaluated submissions appears in App. B.

Of these submissions, approximately 30 percent were judged infeasible because they either violated known physical laws or the performance claimed for a concept would be impossible to achieve. Although the remaining submissions covered a wide range of ideas applicable to SEI missions, nothing was presented that is truly new and revolutionary. There were, however, a number of concepts proposed that could offer substantial improvement in space transportation capabilities *if* various technical issues are resolved through vigorous R&D programs. Another group of submissions proposed concepts that could be very useful once the required infrastructure is in place. Most of the submissions, however, proposed concepts or ideas that are currently being considered or have been examined in the past.

In order to avoid comparing such disparate things as heavy-lift launch vehicles and ion-electric propulsion systems, six broad technical areas were selected to categorize the submissions:

- Earth-to-orbit (ETO) launch systems
- Space transportation systems
- Chemical propulsion
- Nuclear propulsion
- Low-thrust propulsion
- Other

The Other category was used for concepts that did not fit into the first five areas, such as Mars exploration vehicles. We also used aggregation as a simplifying procedure. During the screening of the submissions, it was noted that a number of them proposed an identical, or nearly identical, concept. Thus, rather than analyze each separately, an aggregate of submissions was formed to represent the group of individual submissions. Through this process, the total number of viable submissions was reduced to 213.

STRUCTURE OF THE NOTE

Section II presents a background discussion of the relevant technologies associated with each of the six categorizing areas mentioned above. This discussion is intended to present the reader with an understanding of the technologies involved with ETO launch systems and space transportation systems. The degree to which a topic is discussed is not indicative of the likelihood that it will be used for SEI missions. Rather, the depth of the discussion is based on establishing clarity of the subject matter. As a result, nuclear propulsion is treated much more thoroughly than ETO launch systems, although it is more likely for ETO launch systems to be developed than any of the nuclear propulsion systems. Section III examines the space transportation/propulsion options that are available to fulfill mission performance requirements. Section IV considers the ETO launch system options that are available to support the deployment of the various space transportation systems discussed in Sec. III. Selected submissions are analyzed within the framework provided by Secs. II through IV. Our conclusions and observations appear in Sec. V.

In App. A, as mentioned earlier, we discuss the submission handling and evaluation processes, as well as the specific criteria used by the Transportation panel in evaluating submissions. Appendix B lists all submissions that the Transportation panel screened. In App. C, we discuss velocity requirements for round-trip missions to Mars. Appendix D provides a discussion of vehicle mass determination. Appendices E through R present extended discussions of individual or aggregated submissions.

II. BACKGROUND: MAJOR TECHNOLOGIES AND SYSTEMS

This section discusses the many technologies that are of importance to the SEI in the areas of ETO launch systems, space transportation systems, and their associated propulsion systems. The specific technology/system areas we considered are as follows:

Chemical Propulsion Technologies

- Liquid systems
- Solid systems
- Hybrid systems
- High-energy propellants
- In-situ propellants
- Propellant storage

Nuclear Propulsion Technologies

- Nuclear thermal reactors
- Liquid and gas core reactors
- Fusion reactors
- Antimatter propulsion

Low-Thrust Propulsion Technologies

- Electric
  - Electrothermal
  - Electrostatic (ion)
  - Magnetoplasma dynamic (MPD)
  - Microwave electrothermal (MET)
  - Electron cyclotron resonance (ECR)
- Solar thermal
- Beamed energy
- Solar sails
- Magnetic sails

Earth-to-Orbit Launch Systems

- Ultra-heavy-lift launch vehicles
- Improved Saturn V

Advanced launch system (ALS)

Shuttle-derived vehicles (SDVs)

National aerospace plane (NASP) and NASP-type vehicles

Air-launched vehicles

Electromagnetic launchers (EMLs)

Light gas guns

Space Transportation Systems

Chemically propelled vehicles

Nuclear thermal transfer vehicles

Solar electric propulsion (SEP) vehicles

Nuclear electric propulsion (NEP) vehicles

Cycling vehicles

These areas are obviously broad, and frequently a submission logically fell into more than one area. Under such circumstances, the dominant concept of the submission was used as the basis for selecting the appropriate category. Specific submissions that fell into this category are examined in Apps. Q and R. Below, we examine the technology/systems areas, and their subareas, in more detail.

CHEMICAL PROPULSION TECHNOLOGIES

Chemical propulsion is a mature technology. In many cases, rocket engine performance, both liquid and solid, is near the theoretical limit for conventional propellants. Thus, for these systems, future performance improvements can be expected to be marginal. New engine developments emphasize low cost and reliability rather than performance.

There are propellant combinations that offer significant increases in specific impulse (I_{sp}); however, the results of past investigations by NASA and the USAF have been discouraging. Nevertheless, the 100-sec or more improvement in I_{sp} offered by tripropellants such as Be-O_2-H_2, relative to LOX/LH_2, warrants continued research for SEI missions.

Another class of propellants that has a very high I_{sp} potential is specially prepared metastable variants of elements such as He or N. At the moment, however, the problems associated with the production and storage of metastable propellants appear to be insurmountable.

Thus, unless unexpected progress is made over the next 20 years in the areas of tripropellants, metastable, or free-radical propellants, chemical propulsion systems for SEI applications will be limited to I_{sp}s of less than 500 sec. Under these circumstances, the possible utilization of propellants obtained from the Moon or the Martian system should be vigorously pursued.

Liquid Systems

Liquid propellants can theoretically provide I_{sp}s in excess of 700 sec. However, to date, delivered I_{sp}s have been limited to less than 500 sec in operational systems. Thrust levels from a fraction of a pound to 1.5 million pounds have been attained in single engines. Liquid propellants can be employed as monopropellants, bipropellants, or tripropellants, and also can be formulated as high-density slurries or gels.

Some of the attractive characteristics of liquid propellant rocket engines include

- Start/stop capability
- Throttling capability
- High reliability
- High combustion efficiency
- Hypergolic ignition
- Component cooling (regenerative, film, or transpiration) by the propellant
- Possible use of Lunar or planetary in-situ propellants

Alternatively, design, performance, and operational limitations can arise due to

- Relatively low propellant bulk densities
- Turbopump life and reliability
- Cryogenic propellant storage and leakage
- Toxicity of some propellants
- Ullage (residual propellants)
- Complexity and weight of plumbing, valving, and controls

The Space Shuttle main engines (SSMEs) typify the current operational state of the art for liquid rocket engine booster and space applications. These nontoxic, cryogenic LOX/H_2, high-pressure turbopump engines operate with high-combustion efficiency and reliability in a reusable configuration. Single-engine thrust is about 470 klb at an I_{sp} of 460 sec, and the thrust-to-engine weight ratio is approximately 75:1. As part of the ongoing ALS activities, R&D programs are under way to design lower cost, less complex, more reliable LOX/H_2 engines with higher thrusts and with I_{sp}s comparable to those of the SSME.

While liquid systems are considered one of the more mature propulsion technologies, there are several areas where further development would result in liquid propellant rocket engines that could provide interesting options for SEI applications. Several liquid propellant combinations exist that offer the potential of substantial performance improvement. Some examples are F_2/H_2, monopropellants, tripropellants, space storables such as compound A/hydrazine, slurries, gels, etc.

Table 2.1 presents the ideal and theoretical I_{sp}s of a number of liquid monopropellants, bipropellants, and tripropellants that have either been used or considered for use during the past 40 years.

Table 2.1

Ideal and Theoretical Specific Impulse of Various Liquid Propellants

Propellant	Ideal	Calculated I_{sp} ODE[a] (1000 psia→0.2 psia)
Monopropellants		
H_2O_2	245	192
$N_2H_2H_4$	269	264
Biopropellants		
ClF_5/N_2H_4	386	372
N_2O_4/N_2H_4	404	354
O_2RP-1	461	380
F_2/N_2H_4	–	436
F_2/H_2	528	489
O_2/H_2	528	470
O_3/H_2	607	501
Tripropellants		
F_2/Li-H_2	703	
O_2/Be-H_2	705	

[a]One-dimensional equilibrium.

Solid Systems

Solid propellant theoretical I_{sp}s can approach 400 sec, as indicated in Table 2.2.

Table 2.2

Theoretical Specific Impulse of Typical Solid Propellants

Propellant Combination	Calculated I_{sp} @ 1000/0.2 psia (sec)
$10CH_2/72\ NH_4\ Cl\ O_4/18Al$	340
$10CH_2/52\ NH_4\ Cl\ O_4/20Al$	347
$14CH_2/72\ NH_4\ Cl\ O_4/14Be$	370

The current operational state of the art for solid motors is exemplified by the Space Shuttle solid rocket motor (SRM) strap-on boosters and the IUS Orbus motors. The IUS motors achieve a delivered I_{sp} of 306 sec and incorporate an extendable nozzle exit cone that increases the nozzle expansion area ratio from 50:1 to 180:1 in a volume efficient manner for space operation. Thrust levels ranging from about a pound to greater than 7 million pounds have been demonstrated in single motors having diameters of a few inches up to 260 in. Solid propellants can generally be classified as composites or doublebase.

Some attractive features of solid propellant rocket motors are

- High density
- Long-term storability
- High reliability
- Reproducible performance
- Instant readiness/reduced launch preparation
- Relative low cost
- High thrust-to-weight ratio
- Thrust/time profile flexibility through grain design

Limitations that can affect some applications are

- Short burn times compared to those of liquid engines
- Generally lower performance than that of liquid engines
- Difficult start-stop operation
- Limited throttling with penalties in complexity, weight, and cost
- Exhaust products that can be abrasive and environmentally hazardous

Although there is some sacrifice in I_{sp} (relative to liquids), the benefits that can accrue with solid propellants make them valuable options for the future. For example, solids are, and will continue to be, considered in the design of any advanced launch vehicle, either as strap-ons or as main-stage boosters where low cost, operational simplicity, high reliability, and payload capability are emphasized. If ultra-heavy-lift launchers are required, Table 2.3 reminds us of the solid rocket capability that was *demonstrated* in the 1960s when such vehicles were being seriously considered. The 260-in. solids were proposed as strap-ons to the Saturn V, as well as to serve as the main stages of heavy-lift concepts.

For long-duration space flights that involve destination planet orbit insertion, deorbit, planetary surface boost, and planetary escape, considerations such as long-term propellant storage, propulsion system volume, and reliability take on added importance from an overall mission viewpoint. Solid systems offer such benefits; moreover, I_{sp}s comparable to those for space-storable liquids (350 to 400 sec) are possible through substitution of beryllium or beryllium hydride for the more commonly used aluminum metal additives. Such a

Table 2.3

Sample Demonstrated Solid Rocket Performance

	Contractor/sponsor		
Capability	UTC/NASA	Lockheed/USAF	Aerojet/NASA
Diameter (in.)	120	156	260
Avg. thrust (lb)	1.0M	2.84M	5.12M
Burn time (sec)	107	55	70
I_{sp} del. (sec)	266	236	227
Motor weight (lb)	500K	784K	1843K
Propellant fraction	.853	.877	.893

substitution should be acceptable for space operations where the net performance benefits would far outweigh any potential toxicity issue.

For near-Earth operations, it is becoming increasingly more important to eliminate hydrogen chloride from solid motor exhaust products because of the potential effect of free chlorine on the ozone layer. A potential solution is to develop ammonium nitrate propellants. Intensified R&D programs are needed to resolve concerns regarding combustion efficiency, processing, handling, and phase stabilization.

Another area of improvement is toward lighter, higher-strength solid rocket motor cases. Current programs are attempting to transition from aluminum, maraging steel, titanium alloys, and composite filament-wound structures, such as Kevlar-epoxy and S-glass epoxy, to cases made from filament-wound graphite-epoxy. Aside from reducing motor/vehicle structural weight, such cases can permit operation at higher chamber pressures, which can, in turn, enhance I_{sp} and permit use of smaller exhaust nozzles.

Hybrid Systems

Hybrid rocket systems combine many of the advantages of both liquids and solids by using an inert solid fuel, or solid fuel-rich propellant, and a liquid oxidizer. They generally preserve the simplicity, storability, and quick reaction of solid rockets and add the higher performance, start-stop, and throttling features of all-liquid systems. Overall, operational performance, complexity, density impulse, and cost tend to be intermediate to the two pure approaches.

The solid fuel can be almost anything, from cured rubber, plexiglass, and aluminized HTPB (hydroxyl-terminated polybutadiene) up to the more exotic high-energy fuels. Since the fuel charge is essentially inert, it is safe to produce and handle. Moreover, it can be configured and loaded as unbonded wafers, which could conceivably safely facilitate in-space assembly by astronauts. There are a number of liquid oxidizers that can be used, ranging from storables such as nitrogen tetroxide, to cryogenics such as liquid oxygen (LOX) or a mixture of fluorine and oxygen (FLOX). Specific impulses as high as 400 sec have been attained in hybrid systems, but they generally fall in the 300- to 350-sec range.

The hybrid combustion process is markedly different from an all-solid propellant rocket. As the hybrid fuel charge surface is heated by combustion, it vaporizes; the vapor products are then mixed with the oxidizer as it is injected into the thrust chamber. This forces combustion to take place above the fuel charge rather than on its surface; hence, voids and cracks in the fuel grain, which can otherwise have disastrous effects in a conventional solid rocket, have no impact on the chamber pressure or the regression rate of the hybrid fuel.

Hybrid rocket propulsion has been explored at various levels of intensity since the 1960s, when multiple start-stop and throttling capabilities were demonstrated. A hybrid motor currently powers the United Technologies Corporation "Firebolt" target drone, and such motors are being developed in a commercial venture by the American Rocket Company (AMEROC) to power a family of low-cost, robust space launch vehicles. The inherent, attractive characteristics of hybrid systems, along with the somewhat unique requirements of various aspects of the SEI, make them an option worth serious consideration. They could potentially enhance safety, ease environmental concerns, reduce payload propulsion volume requirements, and provide important operational flexibility as boosters or space propulsion devices.

High-Energy Propellants

Up to this point, chemical propellants have been considered that consist of compounds or elements that, when combined in the proper manner, release energy. Prior to combustion, both the fuel and the oxidizer are stable or at least stable under reasonable temperature and pressure conditions.

There are two types of potential propellants that, under normal conditions, quickly revert to a more stable state in a few microseconds to a few hours. In the process, a great deal of energy—on the order of 20 times that released by burning LOX/LH_2—is released.

Atomic hydrogen, produced by the dissociation of H_2, has a recombination energy that would yield an I_{sp} of over 2000 sec if the conversion to rocket exhaust energy is 100 percent efficient. There are, however, major problems in producing atomic hydrogen in concentrations that would be of use for propulsion. Experimental techniques include the use of strong magnetic fields and very low temperatures to trap atomic hydrogen in a matrix of H_2. Long-term storage of atomic hydrogen will require the continued application of high magnetic fields and low temperatures—30 kg and 0.2K, respectively. If a 25 percent concentration of H in a matrix of H_2 can be achieved, an ideal I_{sp} of about 800 sec could be achieved.

Metastable helium (helium that has been raised to an excited state) is relatively easy to produce, but there appears to be a fundamental limit to the length of time it can be stored because of its inherent radiative lifetime. Unless there is some fundamental breakthrough, the storage of metastable He or other metastable elements for a period of time that would make them useful for propulsion purposes is not possible.

In-Situ Propellants

The use of in-situ propellants could provide a substantial reduction in IMLEO, trip time, and ultimately cost for many SEI missions. In some cases, the use of in-situ propellants may be the driving factor that makes a mission feasible. The Moon, Mars, and the Martian moons (Phobos or Deimos) all have the potential to provide useful propellants for SEI missions. We discuss this issue below in terms of those sites.

The Moon. The Lunar regolith is fairly abundant in metal oxides, which could be broken down to provide liquid oxygen (referred to as Lunar LOX). Since the Lunar gravity force is much smaller than that of the Earth, LOX placed in orbit from the Moon would be substantially cheaper than LOX from the Earth (once the infrastructure is in place). The Moon has no atmosphere and a relatively shallow gravity well; thus, mass drivers could be used in the near term to place the LOX in low Lunar orbit (LLO). This is a very attractive possibility, since many SEI missions are expected to require a large quantity of LOX, and this approach could provide the LOX in orbit at a very low cost. However, mass drivers require a great deal of electrical power and will thus require a major investment. It should be noted, however, that in-situ processing of propellants is also a very electrical power-intensive undertaking.

In addition to oxygen, the Lunar regolith contains a small quantity of hydrogen (expected to be 20 to 200 ppm). The possibility of efficiently processing the regolith to obtain hydrogen in sufficient quantities to justify the initial investment in required infrastructure is being considered. In addition to the obvious benefits obtained by the Lunar surface providing both LOX/LH_2 (rocket fuel, fuel cells, and surface transportation), the hydrogen could be used as the reducing agent of the regolith to extract the oxygen. Hydrogen from the Moon could also be used as fuel for fusion reactors. Another possible source of hydrogen (and oxygen) would be the existence of Lunar polar ice. Due to the complexity of the regolith processing to obtain hydrogen and the uncertain existence of polar ice, it is expected, at least initially, that hydrogen will be transported from the Earth's surface.

The Lunar surface is also abundant in aluminum and magnesium. A rocket engine using these metals, oxygen, and possibly hydrogen could produce an I_{sp} in the range of 300 to 450 sec. However, many technical problems exist with the design of aluminum/magnesium-fueled rocket engines: combustion, plumbing, injection, reliability, etc.

The most likely near-term Lunar-derived material that could offer substantial benefit for SEI missions is Lunar LOX. The Lunar regolith contains about 10 percent ilmenite ($FeO*TiO_2$). Three methods are currently being considered to extract O_2:

- Chemical reduction
- Magma electrolysis
- Vapor-phase pyrolysis

The method that will ultimately be used to perform this function will be chosen on the basis of technological feasibility, quantity and quality of O_2 produced, material considerations, required Lunar infrastructure, and reliability. All methods require a great deal of electrical power, which should be a major consideration in the analysis, since placing a power source on the Moon is a very costly endeavor. All of these methods are energy intensive. We discuss each of these methods in more detail.

Chemical Reduction. Chemical reduction is considered the most feasible method to obtain O_2 from the Lunar regolith. An example of this process uses hydrogen (H_2) to reduce the iron oxide in the ilmenite to produce iron, titanium oxide, and water. A temperature of 700 to 1000^{o} C is required for this process. The water is then electrolyzed to produce hydrogen and oxygen. The hydrogen is recycled and used to reduce more ilmenite.

Magma Electrolysis. A more advanced method to extract O_2 is through magma electrolysis. In this process, a molten ilmenite would be electrolyzed to break the iron oxide into its component elements. Two major problems exist with this process: (1) consumption of electrodes, and (2) materials to contain the molten material.

Vapor-Phase Pyrolysis. Another advanced method to obtain oxygen from the Lunar regolith is vapor-phase pyrolysis. This process uses high temperature and very low pressure (the Moon's "free" perfect vacuum) to vaporize the regolith. The oxygen is then separated from the metal by condensation or electromagnetics. Several technical issues must be resolved before this process can be used in a production facility. First, the oxygen must be condensed with a high degree of purity. Second, materials must be developed to withstand the high temperatures required to sustain this process.

Mars and the Martian Moons. The main component of the Martian atmosphere is carbon dioxide (nearly 96 percent). The carbon dioxide could be used for propulsion in two ways: directly and chemically (after processing).

Direct Use of Carbon Dioxide. This scheme simply uses the gas as the working fluid in a nuclear thermal rocket (NTR). Current NERVA, or slightly advanced, reactor technology could provide an I_{sp} in the range of 250 to 350 sec with carbon dioxide propellant. This provides a great potential benefit, which could be realized with no infrastructure required on the planet prior to first mission. The use of carbon dioxide fueled NTRs could provide a vehicle with propulsion capability for surface-to-surface transportation, surface-to-LMO (low Martian orbit) transportation, and possibly even trans-Earth injection capability. This concept is discussed further in App. N.

Chemical Carbon Dioxide Propulsion. This alternative would require some infrastructure on Mars prior to the first mission. Several alternatives are being considered to provide suitable rocket propellants:

- LOX/LCO (liquid carbon monoxide)

Although this propellant combination has very low efficiency, it may be the best choice for the overall system. A carbon monoxide and oxygen rocket engine would not require the transportation of nonindigenous chemicals to the Martian surface and would most likely require the least infrastructure of all alternatives. Because of the low I_{sp}, it is unclear whether these propellants would be suitable for rocket fuel; however, this combination seems well suited for planetary transportation and power.

- LOX/LH_2 or LOX/CH_4

This alternative would require either a source of hydrogen on Mars or its moons or that hydrogen be transported from the Earth-Moon system. If a suitable quantity of hydrogen exists in the Martian systems, it will most likely be found in the form of ice in the polar regions. Currently, planetary geologists believe that the Martian moons, Phobos and Deimos, have a fairly good chance of providing hydrogen. If ice is found in the Martian system, it would simply be a matter of melting the ice, electrolyzing the water, and storing the hydrogen (cryogenically, in the form of a hydride, or as methane or methanol) and oxygen, all of which are current technology. Long-term storage of hydrogen and oxygen also could be in the form of water. Using this alternative, high-performance rocket engines could be built.

In addition to rocket fuel, these chemicals could be used for planetary transportation (airborne and surface) and surface power.

The first step to obtain either of the above in-situ rocket propellants is splitting the carbon dioxide into oxygen and carbon monoxide. Several processes are currently being considered to accomplish this objective. As an example, two of these processes are dissociation through a temperature and pressure combination, and emulation of the photosynthesis procedure. Further study is necessary to determine the best method to accomplish this necessary task. Without a permanent habitat on Mars, this task would be difficult.

- Dissociation

This process uses heat to establish an equilibrium mixture of the CO_2, CO, and O_2 (about 1000^oC should be sufficient). The constituent parts are then extracted. A zirconia membrane could be used to separate the O_2 from the mixture, and cooling the remaining CO and CO_2 would liquify the CO first.

- Photosynthesis Process

This process emulates the natural photosynthesis process to break CO_2 into CO and O_2. The process uses a rhenium catalyst and visible light, in the range of 385 to 392 nmi. Although this process has a very low efficiency, the simplicity of the approach could make it the preferred approach.

Propellant Storage

If large quantities of hydrogen and oxygen are required to be stored or transported to perform SEI missions, methods other than cryogenic liquefaction should be considered. The most obvious easily stored compound that can provide both hydrogen and oxygen is water. Solar- or nuclear-powered electrolysis can then be used to split water into its component parts just prior to use. The hydrogen and oxygen could then be used as rocket fuel or in fuel cells.

It is possible that oxygen will be available from other sources (i.e., Lunar LOX). In this case it will be far more economical to transport only hydrogen from the Earth's surface. To transport and store hydrogen without using cryogenic techniques, hydrides may be used. For example, lithium hydride (LiH) is 12.6 percent hydrogen by weight. This storage technique could use materials that are available from the Lunar surface (Al, Mg, etc.). Research should be done on the following attributes to determine the best material for hydrogen storage:

- Hydrogen weight density
- Ease and efficiency of hydride productivity and hydrogen recovery
- Availability from Lunar surface

- Storage conditions (temperature, pressure, etc.)
- Energy requirements

NUCLEAR PROPULSION TECHNOLOGIES

This subsection discusses propulsion systems that exploit a relatively high converted mass fraction reaction as the basic energy source ($\sim 9\text{X}10^{-4}$ for fission, up to $\sim 4\text{X}10^{-3}$ for fusion, and 1.0 for antimatter reactions). Such systems can develop high I_{sp} values, and can also enjoy a relatively high thrust-to-weight ratio (T/W) for the propulsive system (relatively high in this context means T/W from about 10^{-2} to 10^{-1} and up).

The class of systems considered often requires considerable hardware mass to provide the propulsive power; general relations or couplings between I_{sp} and T/W result. If P = total power and W = hardware mass, the power per unit mass, a_p, or specific power, is $a_p = P/W$, and $P = g\,T\,I_{sp}/2$, or $T/P = 2/g\,I_{sp} = 2/V_e$, where V_e = exhaust velocity. Thus, a high I_{sp} reduces propellant mass flow rates, but also reduces the thrust for a given power.

Further couplings arise if one now also specifies the propulsive nature of the mission. For example, suppose one wants to effect propulsive burn continuously in going from A to B (accelerating first and then decelerating, from, say, a parking orbit around planetary body A to a parking orbit around planetary body B, with the acceleration/deceleration occurring during the transit between A and B). Then additional relations arise, noted briefly later.

The propulsion systems include

a. Fission—nuclear thermal solid-core reactors
b. Fission—liquid core reactors
c. Fission—gas core reactors
d. Fission or fusion—several explosion-driven systems
e. Fusion—magnetic confinement fusion reactors
f. Fusion—inertial confinement reactors
g. Antimatter—direct use of annihilation
h. Antimatter—annihilation-driven fission/fusion systems

Of these eight systems, system h, using annihilation-driven fission/fusion systems, is apparently the most recent promising proposal. It shows very significant promise of being a realizable set of concepts that retain much of the promise of antimatter systems, without immediately posing the formidable scaleup problems for antiproton production which direct use of antimatter systems implies. (To put one metric ton of payload into LEO, using annihilation directly, requires about one million times the current annual antiproton production capability of Fermilab. Using fission/fusion intermediary concepts might reduce this by a factor of about 10^5 and more.)

Two comments on nuclear propulsion need to be made:

1. Many fine studies by NASA, contractors, and academic personnel have been documented on various nuclear propulsion options. Recently, further reviews by groups such as the AIAA and the National Research Council (NRC) have been undertaken. Parts of this subsection borrow and adapt such documentation freely.

   A recent review, "Nuclear Thermal Propulsion," a Joint NASA/DoE/DoD Workshop, July 10–12, 1990, included discussions of upwards of about 100 papers on various aspects of nuclear propulsion, including solid core (and low pressure), liquid core, and gas core technology issues, together with reviews of missions, safety, development plans, and related items.

   Unfortunately, few archival-class records of this work are available. Secondary histories of very good quality are available, in several forms. A few sources for various aspects include Bond (1971), Thom (1972), Hilton (1963), Reinman (1971), Nance (1965), Balcomb et al. (1970), Boyer and Balcomb (1971), Borowski (1987 et seq.), Haloulakos and Bourque (1989), and others. The summaries by Borowski are reasonably complete and are significantly drawn on here. A definitive history to date remains to be written.

2. There still remain some societal concerns on use of nuclear power for spacecraft propulsion. Indeed, constraints are in effect. Development and operational uses of nuclear power for propulsion on any useful scale would require deliberate setting of new public policy permitting such use.

As a side note, antimatter does not legally fall into the definition of "special nuclear materials," and in principle could be freely used. Some of the antimatter applications noted in this subsection, however, involve nuclear materials.

Changing public policy to allow unconstrained use of nuclear systems with large fissile inventories or other sources of radioactive contamination, such as neutrons produced in fusion burning, will need very careful planning and preparation. The difficulties of public policy changes—to allow both testing and operations—are not usually appreciated by nuclear propulsion enthusiasts, with the consequence that some of the asserted RDT&E "time lines" can be highly misleading.

Below we discuss these nuclear and nuclear-related propulsion concepts in more detail. Note that we generally do not explicitly discuss hybrid or dual use systems that can provide propulsive thrust and electrical power for spacecraft use.

Solid Core Concepts

Studies of nuclear rockets at a substantial level of engineering and physics detail were under way in 1946; such a major study was undertaken by North American Aviation. For the next decade further studies were made, and an extensive experimental program was then initiated by Los Alamos.

Various fission thermal rocket designs based on solid core reactor concepts were studied during the Los Alamos ROVER/NERVA program. The solid core systems were considered to be the logical first step toward achieving a working nuclear rocket engine, and indeed progress by Los Alamos was very rapid. The more advanced gaseous core engines, capable of operating in the multi-kilosecond I_{sp} regime, were also considered by Los Alamos and others, and RDT&E was planned to give insights on them. In the solid core reactor systems proposed in the late 1950s and early 1960s, the fissioning uranium was contained in a variety of fuel element forms, ranging from prismatic graphite assemblies, to packed beds of particulate fuel spheres, to the thin ribbed tungsten plates used in the Dumbo reactor concept. More recently, particle bed fuel element assemblies have again been emphasized. In principle, particle bed reactors could provide some gains in performance (via improvements in heat transfer), but the test base for older NERVA-type fuel elements is much larger and more suited for early engineering. Thermal energy generated in the fuel elements by the fission process would be transmitted via heat conduction to a working fluid flowing through or over these fuel elements. The reactor coolant, heated to high temperatures, is then exhausted through a convergent-divergent nozzle at high velocities.

The basic research and technology development required to build a flight-rated solid core nuclear rocket engine was "essentially in hand at the completion of the ROVER/NERVA program in 1973."[1] This view of program participants is repeated now, and submissions to RAND's Outreach Program reemphasize this position. During the years 1959 through 1972, 19 reactors were built and tested at various power levels, with Phoebus-2A the most powerful nuclear rocket reactor ever constructed. Designed for 5000 MW and using hydrogen as

[1]Personal communication from Richard Bohl, Los Alamos National Laboratory, and D. R. Koenig, "Experience Gained from the Space Nuclear Rocket Program," LA-10062-H, Los Alamos, 1986.

propellant, the reactor had a nominal thrust of 1110 kN (250,000 lbf) and an I_{sp} of 840 sec. Phoebus-2A was intended to be the prototype for NERVA-2, a propulsion system for manned missions to Mars. Replication of Phoebus-2A would, it has been stated, give a running start for nuclear thermal propulsion (NTP) systems.

Smaller research reactors, such as Pewee and the Nuclear Furnace (NF), were designed primarily as test-beds for evaluation of various fuel element designs. Pewee attained a peak power density of 5200 MW/m^3, an exit gas temperature of 2550K, and an I_{sp} of 845 sec. The smaller NF reactor operated for a total record time of 109 min with an exit gas temperature of ~2450K and with peak fuel power densities in the range of 4500 to 5000 MW/m^3.

Considerable insights were gained on the endurance of graphite matrix fuel, composite fuel, and carbide fuel systems. For example, ~10 hr endurance for a carbide fuel operating at ~3000K was projected. This test base could be exploited early on in a reinstituted NTP program.

Modern particle bed reactors appear to offer now, as earlier, rather slight increases in I_{sp} and possibly significant increases in T/W. If projected power densities for the fuel elements can be attained, the core of the very small fuel elements might operate in a molten state at the high end of the projected power densities. Some added safety issues also arise.

The particle bed reactor can have more advantages if we focus on missions where the higher T/W has greater potential, however. Such missions might include, for example, payload delivery into LEO or rapid linear transfer missions. For such missions, high T/W can have relatively higher utility than it would for missions involving interplanetary transfer orbits. Borowski (1991) has considered linear transfer missions using three classes of technology development for nuclear reactor systems: 1972 NERVA technology (I_{sp} = 870, T/W = 3.0); NERVA-derived technology (NDR), using carbide fuel forms (I_{sp} = 925, T/W = 3.9); and particle bed reactor (PBR) technology (I_{sp} = 915, T/W = 20). His results give the following values for the IMLEO for fully reusable Lunar nuclear rocket vehicles, which return the crew, the transfer module, and a Lunar excursion vehicle:

System	IMLEO (tons)
1972 NERVA	235.9
NDR	206.0
PBR	181.4

In this case, the benefits of reduced engine weight from PBR use give an IMLEO savings of ~12 percent over NDR use. Such a savings is potentially attractive to some mission specialists.

Any of the solid core concepts operated at low pressures could in principle achieve I_{sp} conservatively in the 1100 to 1300 range at relatively low T/W. Realization of the higher I_{sp} depends on a number of factors (including energy recovery in the nozzle, exploitation of shifting versus frozen states, details of propellant state changes, heat transfer augmentation, etc.). The mission benefits of this higher I_{sp} can in many cases outweigh the benefits of the much more marginal improvements provided by the modern concepts of particle bed reactors operated at high pressures. Accordingly, a combined mission study and experimental program on low pressure systems may seem substantially more promising than emphasis on, for example, high pressure particle bed systems. Tradeoff studies not yet done adequately would allow better evaluation of the low pressure NTP option.

Existing studies, many done some 30 years ago, differ considerably on the magnitude of the I_{sp} benefits from low pressure operation. At 100 psia chamber pressures, for example, I_{sp} estimates range from about 1000 to 1300 sec, and at 1 psia from about 1200 to 1700 sec, for systems operating at 3500K and with 100:1 nozzle area ratios. There are also many other potential ancillary benefits from low pressure operation: safety, reliability, elimination of turbopumps, etc.

Liquid Core Nuclear Reactor Propulsion Systems

These systems contemplate use of a dense, high temperature fluid containing fissionable material in the liquid state. The intent is to avoid the solid core constraint of having to keep the *maximum* system temperature below the melting point (more realistically, below the temperature at which the material gets structurally weak) of structural materials. Working fluid (e.g., hydrogen) is "bubbled" through the liquid metal to achieve temperatures intermediate between the melting and boiling temperatures of the selected material. Candidate materials include tungsten (density 19.3 g/cc, boiling temperature, T_B, = 6170K), and such other possibilities as osmium (density 22.5 g/cc, T_B = 5770K), rhenium (density 21.0 g/cc, T_B = 6170K), perhaps tantalum (density 16.6 g/cc, T_B = 6370K), etc. For engineering reasons, a large temperature spread between melting and boiling is useful.

The general arrangement contemplated uses a cylindrical shell of molten material held against a solid outer wall by centrifugation (which can be accomplished by proper injection of the working fluid or by mechanical drives). The bulk of the working fluid is injected radially inward, establishing a temperature gradient that serves to cool the outer containing wall while in principle permitting the gas exiting from the inner surface of the molten material and entering the core of the engine to attain nearly the maximum temperature of the melt. Questions of bubble size, flow conditions, material entrainment, sequence of startup operations, etc., are all clearly important. This design is more complex than the corresponding solid core nuclear rocket design, and many more constraints are operative. The material used in this concept is typically kept heated primarily by absorption of fission products from the fissile material it contains. For test purposes, one could experiment with materials made molten by external means (several schemes are directly applicable, including radiation introduced in one of several ways) and obtain data on working fluid temperature rise and the other questions just noted.

The effects occurring and necessary to take into account at molten metal temperatures include dissociation and recombination; radiation transport to the working fluid, particularly if additives increasing absorptivity are used; and, if we consider additives to hydrogenous propellants that ionize at relatively low temperatures, the increased rate of energy transport from ionization effects. If, for example, cesium is used, cesium atoms may be ionized by contacting a surface such as tungsten whose work function is greater than the cesium ionization potential.

Estimates of the net consequences (in, e.g., I_{sp}) of all these effects are best done by numerical calculations, when the effects of working fluid operating pressure, finite nozzle size (which affects recombination), etc., are also desired. However, it is possible to make relatively simple calculations showing that I_{sp} in the range 1500 to 2600 sec should be possible at working fluid operating pressures of ≥ 1 atm and at working fluid operating temperatures consistent with the boiling temperatures of the four materials mentioned earlier.

The liquid core system is perhaps simpler than the gaseous core systems to be discussed, but shares some of their problems (e.g., core material entrainment in the exhaust). It was never very clear, therefore, to what degree such liquid core systems should be emphasized, if at all. A modern study illuminating such issues would probably be worthwhile.

Gas Core Fission Thermal Rockets

Temperature limitations imposed on the solid core and liquid core thermal rocket designs by the need to avoid, or handle, material melting can be overcome, in principle, by allowing the nuclear fuel to exist in a high temperature (as high as 10,000 to 100,000K has been proposed), partially ionized plasma state. Studies of gas core systems were reasonably well accomplished in the 1950s, and interest in them has remained high.

In this "gaseous- or plasma-core" concept, an intensely radiating cylinder or sphere of fissioning uranium plasma functions as the fuel element. Nuclear heat released within the plasma is dissipated as thermal radiation from its surface and is absorbed by a surrounding envelope of seeded hydrogen propellant that is then expanded through a nozzle to provide thrust. Propellant seeding (with, e.g., small amounts of graphite or tungsten powder) ensures that the thermal radiation is absorbed predominantly by the hydrogen and not by the cavity walls that surround the plasma. With the gas core rocket (GCR) concept, I_{sp} values ranging from 1500 to 7000 sec have been suggested as feasible, at various parameter combinations. Two concepts have emerged that have again had considerable current interest: an open-cycle configuration emphasized by NASA, which has the uranium plasma in direct contact with the hydrogen propellant, and a closed-cycle approach emphasized by United Technologies Corporation, the "nuclear light bulb engine" concept, which isolates the plasma from the propellant by means of a transparent, cooled solid barrier.

Porous Wall Gas Core Engine. The open-cycle, or "porous wall," gas core rocket is typically conceived as basically spherical in shape and consisting of three solid regions: an outer pressure vessel, a neutron reflector/moderator region, and an inner porous liner. Beryllium oxide (BeO) can be selected for the moderator material because of its high operating temperature and its compatibility with hydrogen. The open-cycle GCR requires a relatively high pressure plasma (500 to 2000 atm) to achieve a critical mass. At these pressures, the gaseous fuel is dense enough for the fission fragment stopping distance to be comparable to or smaller than the dimensions of the fuel volume contained within the reactor cavity. Hydrogen propellant is injected through the porous wall with a flow distribution designed to generate an appropriate central fuel region. A small amount of fissionable fuel (1/4 to 1 percent by mass of the hydrogen flow rate) is generally exhausted, however, along with the heated propellant. Thus, issues of fissile investment loss are high on the list of questions to resolve.

The uranium plasma and hot hydrogen are essentially transparent to the high-energy gamma rays and neutrons produced during the fission process. The energy content of this radiation (~7 to 10 percent of the total reactor power may be in such radiative form) is deposited principally in the solid regions of the reactor shell. The ability to remove this energy, either with an external space radiator or by regeneratively using the hydrogen propellant, determines the maximum power output and achievable I_{sp} for the GCR engines.

Nuclear Light Bulb Engine. In this closed-cycle nuclear light bulb engine concept, thermal radiation is transferred from the gaseous fuel to the seeded hydrogen through an internally cooled transparent wall that physically isolates the uranium fuel and fission products from the propellant exhaust. The wall material might be constructed of a variety of materials, including silica or beryllium oxide. The uranium fuel is prevented from condensing on the cooled wall by a vortex flow field created by the tangential injection of a neon "buffer" gas near the inside surface of the transparent wall. Neon (or argon) discharged from the system exits through ports located on the centerline of the forward cavity wall and passes to a fuel recycle system. Here fission products are removed and the nuclear fuel entrained in the neon is condensed to liquid form, centrifugally separated from the neon, and pumped back into the fuel region of the vortex. The neon is also pumped back into the cavity to drive the vortex. The light bulb engine has as its most attractive feature complete containment of unburned fuel and fission products—in principle. At least that is the design goal.

Recent emphasis in the light bulb concept lowers the various technology gains aimed for, and results in I_{sp} in, roughly, the 1800- to 2400-sec range. Some of the implications for moderating interplanetary environmental conditions for manned flight, such as life support/radiation handling, become rather marginally tractable at such I_{sp} levels, compared with, say, I_{sp} in the 3500- to 6000-sec range. Such issues arise when the need for high shield weights and/or fast transit times is paramount.

Considerable research has been done in the past to establish critical features of gaseous core reactors (criticality, confinement, propellant heating). Hot-flow experiments were run in the light bulb program using induction heated plasmas for testing transparent wall models and simulating radiation heating of simulated seeded propellants. Other propellant heating tests were run. A common current consensus is that a great deal of technology maturation remains to be done for the combined system, and technology closure time is still far off.

The potentials for the gaseous core engine—I_{sp} up to perhaps $\sim 7X10^3$ and T/Ws in the 10^{-1} to 1.0 range (perhaps as high as the order of unity and more in certain advanced light bulb design concepts)—have generated renewed interest in such engines, especially in the light bulb concept. Very formidable RDT&E problems remain to realize these development potentials. But the promise likely justifies a vigorous RDT&E program. At this time, serious concerns exist within a major portion of the community about the hydrodynamic containment of the fuel element in the open-cycle GCR. It is not clear that the stable flow conditions required to contain the fuel core can be established. Therefore, the light bulb concept is regarded as the more serious GCR candidate for RDT&E. However, the validity of this view, prior to a major RDT&E effort, is not yet wholly convincing to some. There is a significant community that does not believe that an RDT&E effort in gas core reactors (both open-cycle and light bulb) has an appreciable likelihood of culminating in a successful operational engine. In addition, some believe that there are less technologically ambitious ways to achieve the same benefits realizable with GCRs. For example, proponents of ORION (discussed in the next section) believe that explosion-driven spacecraft offer comparable performance with lower technology demands.

A United Technologies Corporation comparison, which is predicated on the successful accomplishment of gas core RDT&E efforts and of open-cycle/light bulb concepts, summarizes some of the key performance parameters potentially achievable for gas core engines, and engine design issues. It is shown in Table 2.4.

Table 2.4

Potential Performance Parameters of Gas Core Rockets

	Open Cycle	Light Bulb
Engine mass (kg)	$40X10^3$–$110X10^3$	$30X10^3$–$300X10^3$
Operating pressure (atm)	400–2000	400–1600
I_{sp} (sec)	2500–7000	1100–3200
Engine T/W	0.05–0.10	0.4– 5.0

Explosion-Driven Spacecraft

Nuclear-explosive propulsion of several forms was studied extensively during the ORION program. Substantial RDT&E work was done in ORION about 30 years ago, ending in 1965, on spacecraft propelled by a series of nuclear (fission bomb) explosions external to a specially designed spacecraft. The concept originated at Los Alamos in the late 1940s, but was first explicitly studied there in the mid-1950s. The explosion products from the fission bomb, expanding essentially isotropically, impinge on a pusher plate at the rear of the spacecraft. The pusher plate moderates the explosion duration by an elaborate system of recoil absorbers and damping techniques to produce spacecraft accelerations tolerable to humans. Representative designs gave effective acceleration of about one Earth g and more. The energy dissipation in the damping system is a crucial design issue. The fission-bomb designs of the time dictated minimum bomb sizes, tailored repetition rates for the explosions, and the consequent generally large spacecraft sizes necessary to handle the explosions. Thus, vehicles in the 10^3 to 10^4 and greater metric ton size, and effective I_{sp} in the 2000- to 5000-sec range, were the general consequence.

The nuclear test ban agreements in the 1960s halted work on ORION. It should be noted that plans were well advanced to test critical ORION phenomenologies in underground nuclear tests. This could still be done today, if desired; but naturally operational use of ORION could not be implemented in the ways then desired, unless, again, public policy on use of nuclear devices is changed.

Proponents of ORION believe that the technology to achieve explosion-driven spacecraft is less demanding than that to produce, for example, gas core reactor systems while yielding all the benefits of such systems. Likewise, such explosion-driven spacecraft systems were then, and are still today, considered much nearer in time than magnetic confinement fusion systems for spacecraft propulsion. Thus, many attractions are inherent in the general class of explosion-driven propulsion systems, and there would seem to be considerable attraction to revival of ORION-like considerations.

Basic research on ORION was comprehensive and detailed. Several hundred basic reports were written; most of these remain classified because specific variants of nuclear device designs were considered (e.g., devices giving asymmetric effects) and for other, analogous reasons. The many reports produced covered such issues as pulse systems, propellant-pusher interactions, questions of opacity and equation of state, computational methods, ablation, experiment design, hydrodynamics, radiation, pusher design, shock absorber design, charge (i.e., nuclear device) delivery, stability and control, flight issues, design integration, missions, parameter studies, and systems analyses. Quite adventurous missions were considered, up to rudimentary interstellar flight capabilities. Many summary mission studies intersected the parameters useful for current SEI operations. For example, a payload of ~90 metric tons and a delta V (ΔV) of ~25 km/sec gave a takeoff gross weight (TOGW) of about 1500 metric tons for a one-stage ORION, compared with a TOGW of ~25,000 metric tons for a three-stage NTR and a TOGW of ~320,000 metric tons for a five-stage O_2/H_2 system.

Various ORION design alternatives were considered. These included features involving small subcritical bomblets specially triggered to produce explosion pulses providing, effectively, more nearly continuous thrust. Explosions within a cavity (HELIOS) were considered to use the bomb energy more efficiently. Shaped nuclear charges, producing nonisotropic external nuclear explosions, were also contemplated. Special methods of loading the systems to reduce ablation of the pusher plate were investigated. Use of thermonuclear fusion charges was proposed early on. Extensive systems studies were made of the interactions of various parameters (e.g., I_{sp}, bomb yield, pusher plate size). Shielding studies were extensive.

Early studies (SIRIUS) likewise proposed inertial confinement fusion (ICF) ideas to "smooth" the explosions even more (lower unit energy release at a higher explosion repetition rate). SIRIUS looked at both internally and externally driven designs.

Very generally, modern work on various ICF ideas suggests that the most attractive combinations of T/W and I_{sp} may derive from these systems. However, currently considered ICF devices have turned out to be a good deal larger and heavier than originally anticipated, and the notion of lifting into orbit propulsive devices of the scale of such ICF machines is surely daunting.

In any event, the lineage from ORION to modern ICF propulsion ideas is direct, even to the extent that early ICF ideas had proponents for both pusher plate technologies, a la the original ORION, and internal nozzle flow ideas, a la HELIOS. We return to these concepts later in this subsection.

Fusion: Initial Comments

As a preliminary to consideration of general fusion propulsion systems, some general comments can be made. *Confinement* issues for the fusion systems (giving enough time for thermonuclear reactions to take place) fall into the two general classes shown in Table 2.5, of which we review only the tokamak-derived and ICF ideas.

Table 2.5

Classes of Fusion Confinement Systems

Low Initial Particle Density ($<10^{18}/cm^3$)	High Initial Particle Density ($\gg 10^{18}/cm^3$)
• Long time for reaction	• Short time for reaction
• Magnetic systems	• Inertial
Tokamak	ICF
Stellarator	Impact
Mirror	Explosive compression
Plasmak	Magnetically insulated
Compact torus	• Quasi-steady state
Etc.	Several concepts
• Electrostatic	
• Wall confined	
• MIGMA/non-Maxwellian	

Correspondingly, of the 36 or so possible fusion fuels to consider, 11 have special features that make them initially attractive in terms of fusion power density, nature of the reaction products (seven produce no neutrons—only charged particles controllable, in principle, by electromagnetic systems), and so on. The eleven "survivors" of the initial screening are

D-$\underline{T}$	$\rightarrow \alpha$ (3.52*) + $\underline{n}$ (14.07)	
D-D	$\rightarrow He^3$ (0.82) + $\underline{n}$ (2.45)	
	$\rightarrow \underline{T}$ (1.01) + p (3.02)	
D-He^3	$\rightarrow$ p (14.68) + a (3.67) + Trace $\underline{T}$ + $\underline{n}$ from D-D	
p-B^{11}	$\rightarrow 3\alpha$ (2.89 each) + Trace $\underline{C}^{14}$ from B^{11} - α	
D-Be^7	$\rightarrow$ p (11.18) + 2α (2.8 each)	
$\underline{T}$-He^3	$\rightarrow$ D (9.5) + α (4.8)	41 percent
	$\rightarrow$ p (5.4) + α (1.3) + $\underline{n}$ (5.4)	55 percent
	$\rightarrow$ p (10.1) + α (0.4) + $\underline{n}$ (1.6)	4 percent
p-Be^9	$\rightarrow$ D (0.3 + 2α (0.16 each) + Trace $\underline{n}$ from Be^9 - α	
	$\rightarrow \alpha$ (1.3) + Li^6 (0.85)	
p-Li^6	$\rightarrow He^3$ (2.3) + α (1.7)	

He^3-He^3	→ 2p (5.72 each) + α (1.43)
p-Li^7	→ 2a (8.67 each) + n (endothermic)
He^3-Li^6	→ p (12.39) + 2α (2.25 each)

*Particle energy in MeV.

where the underlinings denote potential sources of radioactive concerns.

Further screening of these 11 fuels usually suggests that the interesting fuel candidates are typically DT, DD, DHe^3, PB^{11}, He^3He^3, and possibly THe^3. Three of these fuels have the salient characteristics listed in Table 2.6.

Table 2.6

Characteristics of Fuel Candidates

	DT	DHe^3	PB^{11}
Relative power density	100	10	1
Operating temperature (keV)	14	55	150
Relative neutron flux	100	0.25	0

The other fuels have some additional shortfalls, in charged particle power density, ignition temperature, and/or operating temperature. Thus, the presumed advantages of relatively neutron-free reactions must be weighed against the other characteristics. Also, DD reactions require only very common reactants. The net consequence is that one usually considers as the three primary fuel candidates the following most "viable" ones: DT, DHe^3, and DD. DD would be significantly more interesting if catalyzed reactions (Cat. DD) are possible, burning the T and He^3 produced also.

With these preliminaries we next review the most common fusion propulsion concepts.

Fusion—Magnetic Confinement Fusion (MCF) Reactors

Fusion reactors based on the magnetic confinement concept use, typically, superconducting coils to generate the strong magnetic fields needed to confine and isolate the ultrahot power-producing plasma from the reaction chamber walls. The fusion plasma, consisting of positively charged fuel ions and negatively charged free electrons, has a kinetic pressure that has to be sustained via the confining magnetic field pressure. For example, a 15 tesla field may be required, corresponding to a confining pressure of about 13,000 psi; the pressure varies with the square of the field.

Rocket propulsion driven by thermonuclear fusion reactions, whether in magnetic confinement devices or via inertial confinement, can be an attractive concept (see below): a large amount of energy can be released from a relatively small amount of fuel, and the charged reaction products can often be manipulated electromagnetically for thrust generation. Propulsion systems deriving their energy from high energy density fusion fuels have the potential to demonstrate simultaneously large exhaust velocities and the high jet powers and T/W values that could make solar-system-wide travel feasible.

Interest in fusion propulsion is not universal, and many are skeptical of claims in this field. Of course, fusion is a field that has seen a great deal of excessive hyperbole in the last nearly four decades. The fact of the matter is, however, that recent experiments and studies now suggest that some claims are within shouting distance of reality. Things are in fact going on at an accelerating pace in the field. A good, brief but detailed overall summary, for example, appears in Gierszewski, Harms, and Nickeson, 1990. The JET (Joint European Torus) machine has recently (1990) approached a Q-value of 0.8, thus almost achieving the minimum initial energy breakeven value of Q = 1.0. There are probably 100 mainline machine embodiments that are producing a broad base of knowledge currently. Many other experiments have in the last four to five years given new optimism for fusion power, to the extent that actual tritium burning is now scheduled for 1994–1995 in two large tokamak machines, the schedule being driven largely by environmental issues. These machines are the JET and the Tokamak Fusion Test Reactor (TFTR), and the tritium burning is to explore alpha particle heating and associated physics. For example, the machine DIII-D has recently achieved peak ß values of about 10 percent, contrasted to earlier values for any machine on the order of 1.0 percent or less. In terms of the old physics feasibility criterion, the ntT product, the current value of this is quite close to the necessary value for plasma ignition, which is why at least two net fusion power-producing systems are now under extremely active international consideration. These machines will be aimed at, for example, ~1000 MW of fusion power in a test reactor before a 2005 international thermonuclear engineering test.

The fact of the matter is that present large machines are now poised on the edge of achieving energy breakeven, an event that will surprise those jaded by the claims of the last two decades.

Does this recent progress suggest that fusion is a sure thing? Not at all; but it clearly demonstrates that the chances of something useful happening (e.g., substantial net fusion power production in an engineering device) are also now very far from vanishingly small. Note that we are here talking about energy production per se; it is quite possible that machine weight may make MCF machines relatively unsuitable for space propulsion, even if energy outputs are appropriate.

A number of MCF propulsion concepts exist, with the mainline concept still the tokamak. One possible candidate MCF system that could be developed for propulsion applications is based on an advanced tokamak concept, the spherical torus (ST). Other compact torus systems have been under active consideration. Steady progress is being made worldwide in tokamak plasma physics understanding and technology development. Energy breakeven-size tokamaks are currently operating in the United States (Princeton's TFTR), in England (the JET, sited at Culham Laboratories), in Japan (the Japanese Tokamak-60), and the Soviet Union (the superconducting T-15 tokamak). Expectations are now quite high that energy breakeven, and possibly thermonuclear ignition, can be achieved in the TFTR and JET devices in the next several years.

The results obtained to date in TFTR and JET are impressive. In TFTR, central ion temperatures of ~32 keV have been obtained using 15 MW of neutral beam heating power. The much larger JET device, with a toroidal plasma volume of ~150 m^3, has also made very significant progress, achieving most attractive combinations of plasma densities, temperatures (~20 keV), and energy confinement times. The outlook therefore appears promising that an operational power-producing system can be available within the first few decades of the twenty-first century. Use of a high-performance, steady-state tokamak reactor as a driver for a fusion rocket engine consequently has some reality, if the development work progresses as well as its advocates now suggest and as recent results make plausible.

In the ST concept, what is retained of the standard design includes a first wall/vacuum chamber arrangement and a center conductor that carries current to produce the tokamak's toroidal magnetic field. Other components, such as an inner solenoid and inboard neutron shield, are to be eliminated. The twenty-first century can likely see the development of fusion propulsion systems based on high power density magnetic confinement fusion concepts, using some of these technology embodiments. Magnetic fusion engines with specific powers in the range of 2.5 to 10 kW/kg and I_{sp}s of ~20,000 sec might result, giving us the ability to carry efficiently heavy cargoes on fast interplanetary trips.

Very recently (summer 1990), Culham Laboratory studied the use of "low technology" MCF systems, i.e., those not requiring extensive technology extrapolations (the STARLIGHT concept). They proposed a system with the following characteristics:

- Tokamak confinement concept
- Deuterium-tritium (DT) fuel
- Exhaust velocity 8 to 10 km/sec
- Specific impulse ~1000 sec
- Reactor indirectly heats the propellant
- Short operational lifetime (10s of hours)
- Compact design due to high fields

and suggested that such a DT-fueled tokamak fusion reactor heating hydrogen propellant to high temperature has potential for high delta V missions. In contrast to earlier fusion propulsion studies, STARLIGHT employs much more conventional physics and near-term technology. The main features of the specific 10 GW device examined were:

- Total mass of about 300 tons
- Exhaust velocity of 9.4 km/sec
- Cryogenic nonsuperconducting magnet coils
- No tritium breeding required

This study was brief and not all subsystems could be examined in detail. The poloidal field coils, reactor startup, auxiliary heating, plasma exhaust, and electrical power supplies and conditioning were not examined. The nature of the application and the space environment suggest that these tasks and an overall design may be considerably easier to realize than they would be in a ground-based installation.

There are a number of critical reasons for this; for instance,

- The necessary reactor life (hours instead of years) greatly diminishes the materials problems overall.
- The short life gives the ability to use stored tritium, instead of having to breed tritium. The design of blankets to breed tritium in Earth-based (very long life) fusion reactors is one of the enormously complicating issues of fusion reactors.
- For the space application, where the plasma can be exhausted for propulsion, substantially smaller plasma densities can be used than are required for *economical* operation of Earth-based fusion machines for electricity production. The power density varies as $\beta^2 B^4$, so there is much more freedom to relax values of β, B, or both β and B; this freedom translates into much easier physics.
- In a low β machine, the reduction in power density greatly affects advanced fuel potential. This is certainly tolerable, if perhaps not optimum, in a space application using DT burning, whereas for Earth operation there are great economic incentives for using advanced fuels. Just as a matter of interest, the

ability of TFTR and JET in 1988 and 1990, respectively, to routinely achieve temperatures of ~30 and ~20 keV was more than enough for DT operation, and gets us tantalizingly close to DHe^3 and DD operational levels.

- Short-term operation (i.e., hours instead of years) greatly reduces concerns about impurity control and complexity of fueling operations.
- Balance-of-plant issues diminish greatly because the machine only has to produce enough electricity to run itself, and that relatively small recycled power is extractable in ways not suitable for economical Earth-based electrical power producers.
- More relevant for ICF devices, we can use various microfission concepts to provide additional means of energy production, whereas such concepts would be very hard to justify on Earth, for environmental reasons.
- Relevant again to ICF devices, here the target, reaction chamber, and driver are largely decoupled. Normally the chamber must be pumped out before the next driver beam can propagate to the target, a process that can be aided by the space environment.

This STARLIGHT design is deliberately primitive, near term, and of modest performance. As a fusion concept, the design should be regarded as a proof-of-concept exercise and as a step toward more advanced designs, somewhat in the way in which solid core fission reactors are presumed to be along the path to gaseous core reactors.

Fusion—Inertial Confinement Fusion (ICF) Reactors

Early work on ORION spawned work at both Los Alamos (LANL) and Livermore (LLNL) on the "microexplosion" concepts. LANL work (by Balcomb et al., and by Boyer and Balcomb, 1970–1971) retained the concept of a pusher plate, accepting the problems of plate ablation and the consequent constraints on I_{sp}, a matter treated by NASA (Reynolds, 1972) also. LLNL work instead emphasized magnetic redirection of charged reaction products to expel thrust-producing mass out of a nozzle. Ideally, the charged particles never contact the chamber wall, and I_{sp} can be higher because structural limits are not as constraining. Thus, early LANL work had an I_{sp} of $\sim10^4$, while LLNL work talked about an I_{sp} of up to the order of $\sim10^6$.

Roughly at this time (early 1970s) much work on ICF was declassified, and at the same time a considerable body of open (never classified) work appeared that generated much public interest in ICF propulsion possibilities (Winterberg, 1971; Bogolyuskii, 1976; Fraley et al., 1973; Hyde et al., 1972). The result was that serious consideration was given to ICF propulsion, and that consideration has remained vigorous to this time.

For most magnetic confinement concepts, confinement times of a second or more are generally required in order to get a substantial burnup of the fuel. In the ICF approach, requirements for density and confinement times are reversed. Here, multimegajoule (typically) pulses, several nanoseconds in duration, of photons or ions from a "driver" are used to ablate the outer surface of a fuel pellet. Spherical rocketlike reaction forces implode the remaining fuel to stellar densities while simultaneously heating the central core of the pellet to thermonuclear ignition temperatures, 5 to 10 keV for DT mixtures. As the fuel burns, the energy generated is used to heat and ignite more fuel. A thermonuclear burn wave driven, for example, by a particle self-heating propagates radially outward through the compressed fuel. Compared to the disassembly time of the pellet ($t_r \sim R_c/C_s$, with R_c being the compressed pellet radius and C_s the ion sound speed), the fuel is understood to react so rapidly (in as little as a few picoseconds) that it is confined by its own inertia—thus the term "inertial confinement fusion." The process has been discussed publicly in some detail (e.g., Winterberg's early work [1971] is often referenced), and fairly good approximate analytic treatments are possible. Many issues of burn dynamics are treated in Fraley et al. (1973) for a rapid overview. Very comprehensive computer calculational codes are also available, to a smaller group of researchers, giving more verisimilitude to the analyses.

Considerable public literature exists on ICF concepts and issues. The preceding very brief discussion does not do justice to the complex physics involved and to the tools used to treat the detailed phenomenologies. A highly useful discussion is found, for example, in the book *Inertial Confinement Fusion,* by Duderstadt and Moses (1982). In addition, fresh ideas continue to be advanced for drivers for ICF devices; such ideas may significantly reduce the mass and complexity of current prime drivers.

Despite three decades of magnetic fusion research effort, the currently less-developed inertial confinement approach offers the possibility of more compact, lower weight propulsion systems. Heavy superconducting coils in the primary reactor are not needed. The pellet energy release and microexplosion repetition rate can be tailored to produce the desired power level for specific propulsion applications. The high repetition rates for fuel pellet burning (10 to 100 Hz) and energy gain possibilities of ICF would permit an inertial fusion rocket (IFR) to operate, in principle, at very high power levels (10 to 100 GW). Such powers would be extremely difficult if not impossible to achieve with continuous drive MCF. The energy release is in the form of a small and potentially manageable explosion. The initiation of a sustained series of these fusion microexplosions characterizes inertial fusion rocket propulsion and so reflects a limiting version of the ORION concept. The thrust of the spacecraft would be produced, in many concepts, by redirecting the charged plasma debris from the microexplosion via a nozzle out the rear of the vehicle. This debris can also be mixed with additional matter to provide higher T/W at the expense of I_{sp}.

Early on, T. Hyde (1972) performed a typical detailed analysis of an IFR using the assumption of two 2-MJ, 6 percent efficient, high-temperature (1000K) krypton-fluoride (KrF) lasers, each operating at 50 Hz, as the driver. With slightly tritium-enriched deuterium as fuel and a high gain target (G = 1000), the calculated fusion power output consisted of 1280 MW of charged plasma power. With a repetition rate n = 100 Hz, the exhaust velocity and jet power were estimated to be 2650 km/sec (I_{sp} = 270 ksec) and 53 GW. The corresponding thrust level was ~40 kN. The total weight of the engine system was estimated to be 486 metric tons, of which he attributed 54 percent to the drive system and 34 percent to the magnetic thrust chamber. Based on the above parameters, the specific power of the IFR is ~100 kW/kg, in this example.

There have been arguments that such estimates of ICF propulsion system weights and I_{sp}s have very little basis in fact. Of course, such estimates cannot now be verified, since no actual operating device has been built. But we can now make some educated estimates of what such ICF parameters could be. We have a very good idea of what representative current drivers (laser, ion beam) weigh for a given current delivery of effective compression energy onto a target, and we know reasonable scaling laws for how such drivers scale in mass for higher compression energy. For the targets themselves we have extensive numerical simulations, buttressed by many experiments at near breakeven. *Currently* NOVA is running at about a factor of (only) 4 to 6 (in combination of confinement time and fuel temperature) away from the value needed for fusion ignition and high gain. As recently as 1980 we were about a factor of 10^3 away from that value. In addition, one particular drive technique has demonstrated its feasibility in a series of classified nuclear explosion tests. Balance-of-plant designs for extracting enough electrical power to run the driver are extensive (and naturally, as in the propulsion magnetic fusion designs, we need only a small fraction of the electrical power that an Earth-based power plant would need to produce). Mixing propellant with the fusion fuel (to produce greater thrust at the expense of I_{sp}) has been very extensively investigated theoretically and experimentally, at least since ORION days. The associated calculations of I_{sp} starting from the basic energy releases in the targets are reasonably conventional. Our view is that this information base can provide reasonable estimates now.

Does this mean that we can guarantee the weights and I_{sp}s of inertial fusion propulsion systems? Of course not; that is why we need a proving and validating RDT&E program, if we elect to take this path. But recognizing that there are uncertainties that cannot be fully resolved without an RDT&E program should not be used as an occasion to say we know almost nothing about the relevant issues of weight and I_{sp}.

The development of such inertial fusion rockets could usher in the era of the true manned interplanetary-class spaceships. Possessing specific powers and impulses of ~100 kW/kg and as much as 200 to 300 ksec, respectively, IFRs could offer outstanding performance over a wide range of interplanetary destinations and round-trip times. The whole solar system would become accessible, expanding SEI possibilities. Thus, there are many attractions to developing ICF technology for propulsion of spacecraft.

Some of the motivations for developing propulsion devices that could very significantly outperform NTRs have been noted. (It has been suggested in a dissenting view that little point exists for considering advances beyond the NTR in the next few decades.) One can now argue that NTR systems, in the long run, might be almost as much of a dead end as would be, say, a focus on just solid rocket propulsion systems. To show how this argument could go, consider the possible shielding needs, some of which are noted elsewhere in this document, for protecting humans against galactic cosmic radiation (GCR). J. Aroesty, R. Zimmerman, and J. Logan (1991) also argue some of these human support needs in detail, and conclude (as have others) that shielding weights of about 10^2 times the basic spacecraft habitat weights can be required for roughly a minimum energy transit. Coupled with this is the possibility of solar flare radiation and means for handling that radiation exposure source. There are gross uncertainties in our current knowledge of GCR and its biological implications. A decrease in these uncertainties could take a very long time indeed; end results could be either decreases or increases in required shielding weights. One can accept these enormous increases in initial LEO weights, or look at alternatives, one of which is major decreases in transit time from, say, Earth to Mars. Then, for example, a transit ΔV of 50 km/sec reduces one-way flight time to about 30 days, thus reducing GCR exposure by about an order of magnitude.

The minimum one-way Earth-to-Mars flight time and the velocity increment ΔV (km/sec) to leave the Earth parking orbit and circularize in a low altitude Mars orbit are related in the following typical way:

ΔV (km/sec)	5.7	6.5	10.0	23	50
Time (days)	270	200	120	60	30

Therefore, to *reduce* the GCR exposure by an order of magnitude implies an order of magnitude *increase* in ΔV.

The GCR is effectively a 4π source of radiation, while the solar flare radiation is more complex to treat. For flares originating on the sun at solar locations that are roughly connected to the spacecraft (at roughly 1 astronomical unit [AU] from the sun) by interplanetary field lines, the events are about as follows. The very earliest part of the flare flux contains the highest energy particles (e.g., protons), arrives at the spacecraft at times corresponding to velocities that are a substantial fraction of light velocity, and has arrival directions covering a small solid angle. These highest energy protons therefore tend to constitute quite anisotropic events. Subsequent particles arriving at the spacecraft tend, over a period of hours, to have lower energy and to develop into a flux having more nearly isotropic arrival directions. This low-energy tail of the flare transitions after periods of many hours into the steady-state interplanetary flux. Protons from flares originating at solar locations substantially displaced in angle from this most effective flare location arrive at a spacecraft via scattering, and thus tend to arrive later, at lower energies, and more gradually in time, and again tend toward more nearly isotropic arrival directions. Also, as we go farther away from the sun, the events become more diffusive and more isotropic (thus proton distributions tend more towards isotropy as we get near Mars, compared with near Earth).

The precise interplay of these events as they affect spacecraft design warrants more attention than it seems so far to have received. This is because one tempting strategy to handle *both* GCR and solar flare radiation is to use speed to minimize GCR exposure, and asymmetry in the spacecraft structure (i.e., exploiting a natural asymmetry from the positioning of stores, propulsion equipment, or even deliberate use of a shadow shield) to protect from the intense portion of solar flare events, relying on the normal structure to shield against the longer tail of low-energy, more nearly isotropic particle arrival events. This strategy could very greatly reduce the amount of 4π water or other shielding otherwise necessary to handle GCR exposure. But the viability of this approach obviously rests on a very detailed time history of solar flare events, tracing the transition from highly anisotropic particle arrival directions to more nearly isotropic arrival directions, and translating this history directly into vehicle design implications.

In any case, achieving ΔVs on the order of as much as 50 km/sec is far beyond the practical reach of any NTR design. An advanced propulsion capability giving us the possibility of I_{sp} producing ΔVs in this range would have many additional potential uses: much bigger payload fractions; the ability to handle shield weights if desired; insensitivity to any Earth-Mars positional geometry; much greater freedom for choosing flexible combinations of departure, stay, and return times; means for extending large-scale activities beyond Mars if we so choose; and so on. Our feeling is that one could argue persuasively that the goal of routine "colonization" of Mars, or even large-scale exploration and base settlements, could rest on having these degrees of freedom. Such arguments would not be likely now to convince everyone, but our view is that it would also be imprudent now to deliberately exclude and foreclose RDT&E in these fields in favor of the narrow focus of the NTR.

Many other fusion propulsion studies have been performed. Two major ones certainly are the DAEDALUS project and the VISTA project. From the results of these and other studies, one generally concludes that ICF systems usually outperform MCF systems by a factor of about ten in terms of specific powers (as the examples in this section also suggest). This general finding about the attraction of ICF concepts will resurface in following subsections on antimatter systems.

The DAEDALUS project (1973–1978) was conducted by a small Journal of British Interplanetary Society team. Although aimed at an *interstellar flight* use of ICF, a flyby of Barnard's star, the engineering studies for ICF propulsion were comprehensive and thorough, using public information available at the time, and are directly useful for more modest SEI applications. For the interstellar mission, an enormous ship was designed—launch mass of 150,000 metric tons for a final payload of 500 metric tons, flying 40 years, using an engine exhaust velocity of 10^7 m/sec. Two aspects of the project are worth noting.

First, a complete systems study of the mission was documented, and a wealth of engineering detail was presented. No study since then has been pursued at a degree of comprehensiveness exceeding (or even approaching) DAEDALUS, in our opinion. Second, the fusion fuel of choice (DHe^3) brought to the fore the immediate issue of providing He^3 (the mission required *30,000* metric tons of He^3). Three methods of producing He^3 were considered in significant detail (noting that the terrestrial abundance of He^3 is too low—in conveniently accessible and economic form—for the intended amounts). This substudy suggested the level of detail pursued by the DAEDALUS team. The methods included (a) artificial breeding in fusion breeder reactors (a "possible" route, but involving energy needs appropriately one to ten times the entire world's current energy consumption); (b) collection of He^3 ions from the solar wind using enormously large electromagnetic structures (the DAEDALUS team did not consider the prospects for recovering He^3 from Lunar surface materials intercepting the solar wind, the "method of choice" today); and (c) mining the Jovian atmosphere via many large aerostat factories (the analysis of this method was fairly complete, and this method was the choice of the project team). Net economic comparisons of Jovian mining with Lunar soil extraction would be interesting, even though Jovian mining is not seen as an immediate SEI mission necessity.

All in all, the DAEDALUS study is still very worthwhile reading today, even if many details would be modified now.

The much more recent (1990) VISTA project at LLNL is of direct SEI relevance and considers, in this version, Mars missions. The VISTA study is a reasonably detailed system study, using DT fuel, aimed at 100-ton payloads flown on 100-day round-trip missions, with nozzle exhaust velocities giving maximum vehicle speeds of ~300 km/sec. Peak acceleration is $\sim 2 \times 10^{-2}$ Earth g. The LLNL design has a maximum dimension of nearly 200 m; the vehicle is correspondingly heavy. The study has a number of interesting aspects, including a detailed power flow analysis that shows how 150 GW of microexplosion power results in ~13.5 GW of jet power, and describes the various loss mechanisms. The latter include a "plasma drag" contribution that is not generally discussed. VISTA emphasizes DT vice DHe^3 reactions because the latter does not perform as well in debris energy (considering *both* lower expected pellet gain and the high percentage of pellet energy in the debris for He^3).

Antimatter—Direct Use of Annihilation

Antimatter is the ultimate compaction of energy and has long been considered for high-I_{sp} spacecraft propulsion. A primary difficulty is that the annihilation of antiprotons does not generally release energy in a very usable form. The shower of pions and the subsequent electromagnetic cascade result in a spatial spreading of energy and a large increase in entropy. The space-time compression is lost. A straightforward technique for retaining spatial concentration of energy is to deposit the antiprotons in a high-Z material. Both radiation and charged-particle transport are thereby curtailed. This technique is exploited in the simplest form of an antimatter rocket, one that uses a solid core (like the corresponding nuclear rocket) heated by antimatter annihilations. Many individuals and groups, including some at RAND, have considered the application of antiprotons to advanced space propulsion. A portion of the conclusions of RAND studies is summarized in Table 2.7.

Table 2.7

Antimatter Engine Concepts

	Solid core	Liquid core	Gas core	Plasma core
I_{sp} ¥ 10^{-3}	~1.2	~1.5–2.0	~2.5–4.0	~4–25
T/W	High	High	High	High
		Hybrid concepts		
	Beam Core	Pulsed	Electric	Fusion Assist
I_{sp} ¥ 10^{-3}	$\sim 10^4$	~20	$\sim 10–10^2$	~2, up to $3X10^2$
T/W	Low	High	Low	High

Note that the lower end of the I_{sp} range for antimatter engines overlaps the I_{sp} available for a spectrum of fission and fusion engine concepts. An antimatter engine would definitively come into its own at the high I_{sp} suitable for very deep space exploration and for interstellar probes. Even, however, in the overlap region between fission and fusion concepts, detailed analyses show that antimatter engines can frequently be lighter, potentially, than fission or fusion engines of comparable I_{sp} and T/W.

The engine concepts listed have been studied at various levels of detail. As is well known, significant problems emerge when these conceptual designs are scaled up to operational sizes. Primary problems are the facilities and technologies needed to produce, collect, and store antiprotons. A rough rule of thumb relates the major propulsion parameters possible using a variety of conceptual engine types, such as are listed above, to the numbers of antiprotons required. This rule assumes optimization, making the exhaust velocity (V_e) = 0.63 ΔV, with ΔV the desired velocity increment to be gained.

The rule suggests that, per gram of antiprotons, we can relate M_e, the final mass, and Δv, the desired mission velocity increment in km/sec, via the rough rule: $M_e(\Delta v)^2 \int 10^5$. Thus, ideally a gram of antiprotons could give a final mass of 100 metric tons a Δv of ~30 km/sec, or a milligram could give a final mass of 1 metric ton a Δv of ~10 km/sec (corresponding to insertion of one metric ton in LEO).

The difficulty is that even a milligram of antiprotons exceeds by a factor of $\sim 10^6$ the number of antiprotons that can be accumulated by current European Center for Nuclear Research (CERN) or Fermilab facilities per year. To store such numbers of antiprotons efficiently, ways of creating neutral forms of antimatter need development, e.g., in the form of antihydrogen. Creating and storing antihydrogen appears possible on the basis of current studies. While such scaleups in antiproton accumulation capability, by factors of 10^6 to 10^9, have been studied extensively at RAND and elsewhere, the problems are daunting, and the current information base needs upgrading.

Thus, we have a dilemma in using antiprotons directly to power rockets via annihilation energies. Conceptually, the engines are relatively simple, but the number of antiprotons needed is very large indeed, even for modest missions. The question naturally arises: Are there ways in which a few antiprotons can go a much greater way in producing energy? The answer to this query appears to be yes. The reasons for this affirmative answer follow.

Antimatter—Annihilation-Driven Fission/Fusion Systems

Stated in the simplest way, we are looking for possible mechanisms for amplifying the energy directly available from antiprotons. This is now immediately reminiscent of the classical ICF problems intensively investigated in many laboratories, where we try to amplify the energy in a driving beam of some sort to produce fusion reactions, with the fusion reactions yielding energies much greater than those inherent in the initial driving beam.

One would quickly conclude that consideration of antiprotons would make substantial sense only if some special antiproton annihilation phenomenologies were involved. Are there such possible phenomenologies, and how well are these understood currently? These issues are taken up next.

Thus, what we are interested in are *efficient* ways of using antiprotons. Experiments already done suggest such ways.

CERN experiments using the Low Energy Anti-Proton Ring (LEAR) facility have shown that antiproton annihilation in U^{238} produces striking phenomenology, including 100 percent probability of uranium fission and about 10 neutrons per annihilation. Fission deposits locally about 0.2 GeV per antiproton (additional effects can raise this deposition, in perhaps a somewhat more diffuse way, to about 0.8 GeV). The fission probability is effectively independent of antiproton kinetic energy. Thus, the deposition may be done via very low energy beams (e.g., a few megaelectron volts), giving very short longitudinal range, while the radial beam dimension can be made small. The consequence is the possibility of fission fragment deposition in very small volumes, giving very high energy densities, with appropriate beam space-time compression.

The annihilation-produced high energy densities in uranium and the release of about 10 neutrons per annihilation can individually and together plausibly produce effects of great interest in the ICF field. A pusher exploded by antiprotons could compress thermonuclear fuels to ignition; the sudden production of many neutrons in a critical micromass (made critical by using antiprotons or by other means) could initiate prompt and sustained fissile burning—an antimatter "spark initiator." These effects combined in a number of ways could come into play in microcapsules in which both fusion and fission phases can occur. The pusher may require as few as $\sim 10^{14}$ antiprotons to achieve ultimately thermonuclear energy releases in the 10^8 to 10^9 J range, while the spark could be done well within currently deliverable numbers of antiprotons (as low as about 10^7 antiprotons would be interesting). These conclusions are supported by current calculations that use approximate hydrodynamic codes and equations of state (EOSs).

The results so far obtained by approximate calculations can be very considerably improved by using known, more competent codes, with inclusion of hitherto neglected effects, and employing more realistic EOSs.

It should be emphasized that these properties of antiproton annihilation in U^{238} (or U^{235}) are based on two large-scale experiments, PS 177 and PS 183, run at CERN in the early 1980s. The phenomenology is therefore firmly based.

A further gain could be realized by using the energy compression inherent in an antiproton beam to drive a nuclear capsule to produce fusion, fission, or some combination of these. Tentative classes of designs (using publicly available data and codes) have been suggested by several groups, including researchers from Penn State, the University of Michigan, LANL, and RAND. These researchers have explored several basic concepts (practical implementation of these concepts would bring obvious challenges). The concepts include

1. An exploding-pusher fusion capsule using a U^{235} shell imploding a DT core.
2. A U-DT antiproton-driven shock tube that compresses a DT fuel by explosion of a U^{235} plug at each end of the tube.
3. A gold capsule using normal matter lithium beams for implosion and antiprotons for ignition of a central PuDT fuel.
4. A scheme again using light-ion compression, special geometries, antiproton ignition, and strong magnetic fields to suppress energy loss from alpha-particle escape and electronic thermal conductivity.

5. Schemes using several such concepts in combination (a few moments of thought will convince researchers that a potentially very rich class of these combined concepts arises).

One of the aims of some of these concepts is to compress microassemblies of fissile material toward criticality, so that the initial neutrons produced by a pulse of antiprotons—the "antimatter spark"—lead to multiplication.

The antiproton numbers implied range up from about 10^{14} for $\overline{p}$ compressed systems, and from about 10^{7} for $\overline{p}$ spark-ignited systems. These numbers are in a sense floating parameters related to the size of the systems under consideration. Gains of many orders of magnitude over the initial investment in the $\overline{p}$ rest energy might be realized, according to initial results of rough calculations for these concepts.

The numbers of antiprotons needed, according to these initial estimates, may be compared with present production levels. Fermilab's antiproton source currently can deliver about a nanogram ($6X10^{14}$) of $\overline{p}$s annually for use in the high-energy physics experiments run there. For purposes of testing microexplosions, we would need additional deceleration stages (costing about $10 million to construct) to reduce the kinetic energy of the antiprotons into the kiloelectron volt/magaelectron volt level. The Fermilab source actually produces over 10^{3} times these numbers of antiprotons but suffers known inefficiencies, correctable to a large extent in a new design, in capture and cooling of the antiprotons produced. Thus, a de novo Fermilab might deliver $\sim 10^{16}$ to 10^{17} low-energy antiprotons annually. A new high current accelerator, such as those already designed for Hadron/Kaon machines, would rather easily have the potential with known technology for getting us into the range of $\sim 10^{18}$ to 10^{19} delivered antiprotons per year. We conclude that competent experimentation with the antimatter microexplosion concepts can be clearly accessible early on, and that operationally useful numbers of antiprotons are also within reach, exploiting the near-term characteristics of Hadron/Kaon machines for antiproton production and accumulation. Compared with pure annihilation propulsion systems (i.e., systems that do not use fission/fusion to produce energy gains), the need to scale up antiproton production levels would be reduced by the very large fission/fusion energy gain factor to a good approximation. This major change could make a dramatic difference in the way we view the near-term reality of antimatter propulsion.

Finally, it should be noted that storage on the order of $\sim 10^{8}$ $\overline{p}$ can be conveniently done in quite small Penning traps. Such traps have been scaled up, in conceptual designs, to contain $\sim 10^{12}$ to 10^{14} $\overline{p}$, and thus allow us to avoid the added complexity, for even rather ambitious early experiments, of having first to create neutral antihydrogen to circumvent the space charge limitations of the Penning trap.

With any of the thermonuclear reactions (DT, DD, DHe^3), the questions of fuel cost could change substantially over those same questions using only antiprotons. D costs about \$500 to \$1000 per kilogram, T produced today in very limited production amounts is said to have an effective cited price of about \$30,000 per gram, and costs for delivered He^3 are problematic but are certainly bounded by the D and T costs.

Note that He^3 is produced terrestrially as a separable isotope of normal helium gas in natural gas wells, and via decay of tritium used in U.S. nuclear weapons and for other purposes. Monsanto can now sell about 1.3 kg of He^3 per year, while weapon-based tritium decay might give the order of 10 to 20 kg per year if fully exploitable. Thus, no dearth of *experimental* quantities arises. Lunar sources of He^3 have been suggested (possibly 10^6 metric tons may exist there). *If* He^3 becomes a fusion fuel of choice, there is every likelihood that amounts for major applications can be accessible. Some of these applications emphasize utility as a *concentrated* energy source (useful for space propulsion, say), and *net economies* to a lesser extent. Note, in this context, the earlier remarks on the utility of He^3 for fusion use.

If we assume an "average" fuel mixture price of ~\$15,000 per gram of DT fuel and remember that each of the three fuels mentioned can produce a converted mass fraction of nearly $4X10^{-3}$, then, for fractional fuel burnup of b percent an amount of DT fuel giving about the same release energy as 1 mg of antiprotons would cost ~$(15X10^3)/4b$ dollars. Burnup efficiencies as low as 1 percent would produce DT costs of ~\$$0.4X10^6$, corresponding roughly to the low end of the estimated antiproton costs of ~\$0.5 to 10.0 million per milligram at high production levels. DD costs would of course be much lower, if such fuels could be burned at virtually any appreciable efficiency.

A sense of the role of thermonuclear fuel costs can be gotten by going through a simple propulsion example. The example suggests the large amounts of fuel needed for relatively demanding missions. For fuels with an energy release of ~$3.4X10^{11}$ J/g (i.e., DT, DHe^3, Cat. DD) and with a burn efficiency of one-half, a microexplosion giving $1.5X10^9$ J requires burnup of ~9 mg of fuel. If half the burn products are directed into a propulsive jet, $7.5X10^8$ J are in the jet. If further we add propellant mass ~20 times the initial burn mass, the total propellant expelled, m_p, per microexplosion is ~180 mg. Assume now that the propellant is formed into a directed jet with an efficiency, ε, of ~1/2, so that the jet exhaust velocity $V_e \sim \varepsilon ([1.5X10^9 \text{ J}]/180 \text{ mg})^{1/2} \sim 1.5X10^8$ cm/sec = g I_{sp}, so that I_{sp} is ~$1.5X10^5$ sec. If the microexplosion repetition rate is n = 50 explosions/sec, the jet power $P = 1/2 \cdot m_p \cdot n \cdot V_e^2 \sim 9X10^9$ W. The thrust $T \sim m_p \cdot n \cdot V_e \sim 14X10^3$ N ~ 1.4 metric tons. If we use a specific power a_p of ~100 kW/kg, which may be achievable in an antiproton-driven fusion system (possibly a_p values in the 2 to $5X10^2$ kW/kg range might be achievable), the total engine system all-up weight would be ~90 metric tons.

For this system, the fuel burned per second, M_s, is M_s = 9n = 450 mg/sec = $4.5X10^{-1}$ g/sec. Assume now a hypothetical vehicle with a payload weight of 260 metric tons, an engine weight of 90 metric tons, and a structural weight of 100 metric tons, or a total weight empty (sans fuel) of W_e = 450 metric tons. For the continuous thrust case mentioned in the Introduction, and with straight line paths between A and B (these very simple extreme hyperbolic transfer paths are a useful approximation if the vehicle accelerations are significantly greater than the sun's gravitational pull at the Earth of $\sim 6X10^{-4}$ g) and a few other simple approximations, an approximate relation for the trip time, t_{EM}, in going from, say, Earth to Mars, an assumed distance of $\sim 7.8X10^{10}$ m, gives a t_{EM} of $\sim 3.3X10^6$ sec (about 38 days). The midpoint space vehicle velocity increment is $\sim 3X10^{-2}$ X $1.7X10^6$, or ~50 km/sec. This simple approximation assumes that, in addition to constant P, n, m_p, and I_{sp}, the value of the spacecraft acceleration and deceleration is also constant at the final value of $\sim 3X10^{-2}$ m/sec. Since the total fuel burned is ~0.18X50 X $3.3X10^6$ gr, which is $\sim 3X10^7$ gr, or ~30 metric tons, or 1/15 the vehicle empty weight, constant acceleration/deceleration is not a bad approximation.

The amount of thermonuclear fuel burned in this trip is then 1.5 metric tons. From this, it is seen that current T costs of $30,000 per gram would be impractical or unacceptable for spacecraft use for equimolar DT mixtures. Fuel compositions emphasizing the lower cost thermonuclear fuels, a high production breeding method drastically reducing T costs (and/or He^3 costs), and, especially in the case of antiproton-induced fusion, possible significant reliance on fissile burning, are possible ways out of the thermonuclear fuel cost issues, singly or in combination. Still, this very demanding mission (a payload of 260 metric tons one way to Mars in less than 40 days) would be at the limits of realism for any chemical propulsion system and for any reasonable NTR system, because of the enormous initial vehicle weights needed in LEO.

Achieving antiproton-initiated fusion with such characteristics would thus have many implications of very significant interest. This modern proposal for efficient and unique uses of antiprotons may give propulsion concepts of very great promise. The implications include but are not limited to

- Possibilities for antiprotons becoming a net energy source.
- For comparable energy releases, far fewer antiprotons would be needed, with consequent great reductions in antiproton production scaleup requirements.

- Propulsion, power, and energy sources capable of a very broad and controllable range of energy release rates, available in small engineering embodiments.

Summary

A great premium can be placed on propulsion concepts with high I_{sp} and appropriately high T/W for routine interplanetary travel. The class of nuclear propulsion systems discussed in this section appears to offer the promise of realizing such concepts.

The solid core nuclear rocket system has been extensively tested, and has already achieved in one version I_{sp} and T/W capable of giving robust interplanetary transportation systems.

The progression through liquid core and gaseous core nuclear rocket systems would offer factors of about two and five over the solid core I_{sp}. Awaiting development are important tests (component, subscale, full scale; nonnuclear and nuclear) to achieve actual engines.

ORION (nuclear explosive driven propulsion) systems continue to offer great promise; serious consideration should be given to reviving interest in such capable systems.

Fusion engines of several kinds have been studied. ICF-based concepts offer significant advantages over MCF possibilities. Work going on in ground-based ICF and MCF designs for power production can evolve directly into propulsion-related versions of such fusion devices. Fusion propulsion devices potentially combine I_{sp} that is substantially higher than that of gaseous core nuclear rocket systems while retaining useful, effective T/Ws.

Antimatter, the most compact form of energy routinely available to us, has great attractions as an ultimate form of energy for propulsive uses. Direct use of annihilation energy promises a large range of compact engine concepts, but also implies needs for very large scaleup of antiproton production capabilities. The efficient use of antiprotons in the near future can lie in ICF applications, that is, using antiprotons to induce fissile burning, fusion burning, or combinations of these. These modern uses of antimatter might revolutionize the ICF field and provide quicker paths to very advanced propulsion systems. Exploring such a use of antiprotons would appear to be a very high priority in advanced propulsion RDT&E.

We conclude that nuclear propulsion, in all these embodiments, warrants a carefully structured research program in the next years. The promise of nuclear propulsion of the kinds briefly discussed for SEI use supports such a conclusion. A very exciting and productive period for development of advanced propulsion may result; if so, mission applications once thought to be far beyond our reach may become accessible.

This view that we can usefully subscribe to a comprehensive nuclear propulsion RDT&E exploration program is not currently universally accepted. We earlier noted the dissenting view that successful achievement of routine NTR use is all that is necessary, and we suggested why, in our belief, this may not be so.

Other reasons for a focus on NTR development might be advanced. A possible cost argument would seem to us to be misdirected. Appendix M of this document, and many other studies, including estimates in the July 1990 Workshop noted earlier, suggest that bringing an NTR to full manned flight qualification status would, based on chemical precedents and on additional issues for nuclear systems, be quite expensive—easily on the order of $5 billion. For a modest fraction of this funding a very substantial advanced propulsion research exploration could be accomplished, so there can be no question of advanced propulsion research in effect foreclosing the next step goal of operational realization of the NTR system. Fusion research applied to propulsion has recognized difficulties; but very significant basic advances achieved in the last decade have moved us immeasurably beyond the status of a decade ago by progressively working through many of the difficulties we are all aware of. As for antimatter, there is a new conceptual basis, namely, use of antimatter for supporting microfission, microfusion, or both energy release techniques, and possibly making ICF in general more real and more immediate (a proof-of-principle experiment proposal, using an existing machine, is already being prepared). It is hard to push such ideas to definitive go/no-go levels in the absence of advanced propulsion research. Our view is that the utility of advanced propulsion, *if developable,* is such that research to guide us on *whether* it can, or should, be developed is merited in a well-constructed research program. This view is in part based on the belief that NTR work, while a significant advance if operationally used, still has its own set of dead ends and quickly reached technology asymptotes, if we have in mind some of the more challenging SEI objectives.

Thus, we feel that our notion of an advanced nuclear propulsion research program to put various advanced ideas to the test and to winnow out those concepts (if any) that could result in significant further operational advances beyond NTR capabilities is completely appropriate. That notion in no way impedes the next step of NTR operational implementation in any sensibly planned propulsion program. We later reiterate this view (see Nuclear Space Transportation Options and Technologies in Sec. V). The key operational statement there is worth noting here, however: "The potential increase in performance . . . is so great that we recommend that a research program be undertaken to *identify . . . options*

. . . *most promising for development*" (emphasis added). This theme does not automatically assume that any of these advanced concepts *will* in fact be developed; but it is asserted that finding out *whether* such concepts should be developed (if technically possible) is a productive endeavor.

LOW-THRUST PROPULSION TECHNOLOGIES

The characteristic common to all low-thrust propulsion systems is the low vehicle accelerations that are normally achievable. As a class, accelerations range from 10^{-5} to 10^{-2} g's, with the specific accelerations achievable being system dependent. Typically, solar sails and magnetic sails are at the lower end of the range. In the middle of the range are the various electric propulsion systems. At the upper end of the range are solar thermal, laser thermal, and electrothermal (e.g., arcjet) propulsion systems. As a consequence of the low thrusts and subsequent accelerations, these propulsion systems must operate continuously for periods of many weeks or even months.

Of the low-thrust propulsion systems, ion and magneto plasma dynamic propulsion systems currently offer the best potential in terms of both I_{sp}s and thrust levels. Solar sails and magnetic sails have infinite I_{sp}s but are limited to low-thrust levels. Other low-thrust propulsion systems, such as solar thermal and laser thermal systems, are limited to more modest I_{sp}s, in the range of 1200 to 1500 sec, respectively.

A special category of low-thrust propulsion is that of beamed energy systems. This category includes laser thermal propulsion (mentioned in the preceding paragraph), microwave thermal propulsion, laser electric propulsion,and microwave electric propulsion. Although these propulsion systems do not require onboard power sources, they all suffer an operational range limitation because of the divergence of the beamed energy.

Overall, the most likely application of low-thrust propulsion systems for SEI missions is for unmanned cargo carriers. This likelihood results from the fact that transit times for this class of systems are usually considerably longer than for either chemical or nuclear propulsion systems. Exceptions to this rule would be the multimegawatt electric systems, if they can be developed, where trip times comparable to those of either chemical or nuclear thermal systems can be achieved.

Electric Propulsion

Unlike chemical propulsion, which is energy limited (amount of energy limited by chemical bonds and the efficiency of converting this energy into gas velocity), electric propulsion is power limited. The source of electricity could either be solar or nuclear. The major technical problems with electric propulsion systems occur with developing an efficient power source and efficient power conversion and conditioning systems that are suited for space vehicle applications. We first discuss the various types of electric thrusters and then describe aspects of solar and nuclear power supplies for electric thrusters.

There are at least five types of electric thrusters that could be suitable for space propulsion:

- Electrothermal (resistojet and arcjet)
- Electrostatic (ion rocket)
- Electromagnetic (magnetoplasma dynamic [MPD] rocket)
- Microwave electrothermal (MET)
- Electron cyclotron resonance (ECR)

The I_{sp} for electric thrusters is much higher than that for chemical rockets, as shown in Table 2.8.

Table 2.8

Propulsion System Technology

Propulsion Technology	I_{sp} (lb_f-s/lb_m)	System Efficiency (%)
O_2/H_2	480	Not applicable
Arcjet	1500	49
Ion	2000–10000	60–85
MPD	2000–10000	50

The relationship between power, thrust, and I_{sp} is

$$T \times I_{sp} = \frac{2P_w \eta_T}{9.81},$$

where thrust, T, is in Newtons, I_{sp} is in seconds, and power, P_w, is in watts. The parameter η_T is the efficiency of the thruster in converting input power to thruster power. For a constant power, there is a direct trade between I_{sp} and thrust, with the optimum combination being mission dependent.

Electrothermal. Two basic types of thrusters are included in electrothermal propulsion: resistojet and arcjet. A resistojet simply heats a propellant with a resistor in the gas flow. This technique has been used to augment the propulsion systems on commercial and military satellites since 1965.

An arcjet produces thrust by heating a gas with an electric arc. The heated plasma then expands through a conventional nozzle. Several problems exist with arcjets, including electrode erosion and material problems due to hot gases. The efficiency of arcjets is low compared to that of other forms of electric propulsion, since much energy is lost in the resistance of the electrodes, electromagnetic radiation, and heat lost to chamber walls. Since the I_{sp} of electrothermal rockets is typically less than that of other types of electric thrusters, electrothermal rockets are usually considered only for orbital transfer and possibly Earth-Moon transportation and would not be considered for Mars missions.

Electrostatic. Electrostatic, or ion, rockets produce thrust by accelerating ions using electrostatic force. An ion rocket first ionizes a neutral propellant by stripping off electrons from the atoms or molecules. Then the ionized gas is accelerated by an electrostatic field. Finally, the gas is neutralized by recombining the ions and electrons to prevent the vehicle from acquiring a net negative charge, which would attract the expelled ions back to the vehicle and result in zero thrust. Ionization potentials, mission requirements, and handling and environmental characteristics should be considered in choosing a propellant. Cesium and mercury have been studied in the past; however, due to environmental concerns, xenon and argon are now the preferred candidates. Table 2.9 shows the current and projected operational thruster characteristics for these propellant choices.

Table 2.9

Current and Projected Ion Thruster Performance

	Xenon		Argon	
	Current	Projected	Current	Projected
I_{sp} (ksec)	3.3–5.0	2.5–5.5	5.7–7.7	4.4–9.4
Efficiency (%)	66–75	69–78	61–64	67–75
Thrust (N)	0.29–0.67	16–34	0.29–0.68	16–34
Power/unit (kWe)	7–22	290–1160	13–40	525–2105
Operating life (hr)	<5000	>5000	<5000	>5000
Effective diameter (cm) (equivalent area)	30	160	30	160

MPD. An MPD thruster uses electromagnetic forces to accelerate plasma. An advantage of MPD thrusters is that they can use a wide range of propellants (however, I_{sp} will vary). Hydrogen is currently considered to be the best propellant choice because of its low molecular weight and, therefore, high I_{sp}. Currently the primary problem with MPD thrusters is electrode erosion and the associated short lifetimes and low efficiency in comparison to those of ion thrusters. Current and projected operational characteristics for MPD thrusters using hydrogen and argon as propellants are presented in Table 2.10.

Table 2.10

Current and Projected MPD Thruster Performance

	Current[a]	Current[b]	Projected[c]
Propellant	Hydrogen	Argon	Hydrogen
I_{sp}	4900	1100	5000
Efficiency (%)	43		60–70
Thrust (N)	27	8.6	100
Power/unit (kWe)	1500	273	1500
Operating life (hr)	1	1	5000
Operating mode	Pulsed	cw	cw

[a]Highest observed performance at conditions below "onset" of high erosion. Ref. IEPC Paper 84-11, 1984 (ISAS, Japan).

[b]Highest steady-state power data. Ref. AIAA Paper 87-1019, 1987 (Stuttgart, Germany).

[c]Significant uncertainties exist in high power MPD thrust (efficiency) and life.

MET. An MET thruster uses microwave energy to heat a propellant gas, producing a plasma flame with temperatures as high as 4000 to 6000K. Because of material considerations, the temperature of the propellant in contact with the thruster surface must be kept in the range of 2000K. Experiments to date with MET thrusters have produced I_{sp}s in the range of 6000 sec with a power input of 1.5 kW. Research is also being conducted on magnetic nozzles, which would allow much greater propellant temperatures and are expected to allow I_{sp}s as high as 20,000 sec to be achieved using MET thrusters.

ECR. This propulsion technique is currently being researched. It is projected that very high I_{sp}s (5000+ sec) and high efficiencies (50 to 85 percent) may be obtainable. In contrast with MTP, where thrust is produced by thermally accelerating the propellant, ECR produces thrust by coupling the microwave energy to the propellant electromagnetically.

An ECR thruster uses circularly polarized microwave radiation to ionize the propellant. The electrons in the plasma then spiral around diverging magnetic field lines produced by a solenoid magnet. This process produces a net body force on the plasma, which is then accelerated out of the thruster. With the proper choice of microwave frequency and magnetic field strength, the energy may provide a forcing function at the resonant frequency of the electrons.

ECR thrusters have two very advantageous characteristics: (1) no electrodes (and therefore the possibility for very long operational lifetimes) and (2) the ability to operate on a variety of propellants (many are available in situ).

Experiments to date using ECR thrusters have produced I_{sp}s in the range of 1000 sec with a 20 kW microwave power source. JPL is currently researching the possibility of producing higher I_{sp}s from ECR thrusters.

Power Sources for Electric Propulsion. Both solar and nuclear power sources are being considered for electric propulsion. Depending on mission requirements, power levels of 1 to 100 MW are being studied for SEI missions.

Solar electric propulsion (SEP). The photovoltaic power supply (solar arrays) for near-term applications will most likely be made of gallium arsenide or amorphous silicon. Since the power from a solar array is proportional to the projected collector area (and therefore mass), the specific mass will stay approximately constant regardless of power level. The solar flux (and therefore the power output of the solar array) is inversely proportional to the square of the distance from the sun; however, since the efficiency of the solar array increases with decreasing temperature, the reduction in power produced by the array drops off slightly slower than $1/R^2$. Current technology can produce solar arrays with a specific mass of about 6.5 kg/kW and a 14 percent efficiency. It is projected (Palaszewski, 1988) that in the 2010–2020 time frame, the specific mass of an SEP power source could be reduced to 3 or 4 kg/kW with a 25 percent efficiency.

Nuclear electric propulsion (NEP). Near-term nuclear power sources will probably be liquid-metal-cooled fission reactors of the SP-100 type. The SP-100 is currently an R&D project to build a 100 kW reactor.

Economies of scale become dominant for high power nuclear reactors. Much of a nuclear reactor's mass is required, regardless of power level. Therefore, nuclear electric power sources become much more attractive as power output is increased (specific mass decreases as power level increases). According to a JPL study (Sercel, 1987), the specific mass of an NEP for high power reactors (100 to 500 MW) is about one third that of an SEP of the same power output. The major contributor of mass (approximately 60 percent of the total for a high power system) for a NEP power system is the radiator required to remove waste heat from the thermal-to-electric power conversion system. The reactor and shielding of an NEP power system are expected to be less than 5 percent of the total mass, with the remaining mass composed of the power conversion system (radiator, alternators, turbines, boiler, and plumbing). It is somewhat difficult to estimate specific mass over a wide range of power levels. In addition to economies of scale, technological advancements and type of power conversion cycle also affect the specific mass. At higher power levels, it is expected that more technologically advanced designs will be chosen and different power conversions will be used (i.e., the SP-100 is expected to use Sterling cycle, whereas the 100 MW class will probably use Rankine cycle). Table 2.11 provides very rough rule of thumb estimates of specific mass for various power levels.

Table 2.11

Nuclear Electric Power System Specific Mass

Power Level (MW)	Specific Mass (kg/kW)
.1–1	25
1–10	5
100–500	1

Solar Thermal Propulsion

Solar thermal propulsion (STP) concepts have been under study since the late 1950s. The Air Force Astronautics Laboratory (formerly the Air Force Rocket Propulsion Laboratory) funded much of the early work and is currently the focal point for STP research and development.

Solar thermal systems are simple in concept. A mirror or concentrator collects solar energy and focuses it onto a chamber. The chamber, a part of the thruster, contains the propellant, or working fluid, usually hydrogen, that is heated by the incident radiation. The heated propellant expands through a conventional nozzle to produce thrust. The major technology issues are the development of large, lightweight solar concentrators that can be packaged compactly and then easily erected in space, and the efficient coupling of the solar energy to the propellant.

Current solar concentrator designs are inflatable and thus easily packaged and self-deployable. After deployment, the concentrator can be stiffened by a number of methods, thus eliminating shape distortion resulting from deflation caused by micrometeoroid impacts. Other approaches to concentrator design include the use of holographic techniques and film creep-formed surfaces.

Thruster design concepts fall into two broad categories: blackbody cavity absorbers and volumetric absorbers. In the former design, a heat exchanger made of refractory material, such as rhenium, is heated by the focused sunlight. Hydrogen or another propellant passes through the coils (after having regeneratively cooled the thruster), where it is heated and then expanded though a nozzle. This type of thruster has achieved an I_{sp} of about 870 sec in tests. Such a performance level is about the upper limit for this type of heat transfer mechanism because of material temperature limitations. Another approach would replace the rhenium coil exchanger with a series of cylindrical discs constructed of graphite foam on which hafnium has been deposited. The material is porous, and the focused sunlight passes through the first disc and is absorbed deep within the cavity. Hydrogen gas flowing through the pores would be heated by the hot hafnium carbide. The theoretical I_{sp} of this type of heat exchanger is expected to be about 1000 sec, with materials again being the limiting factor.

Volumetric absorbers, where the incoming solar radiation is absorbed or trapped by particles suspended within the propellant, are projected to achieve I_{sp}s on the order of 1200 sec. Again there are a number of possible design approaches. One potential disadvantage of seeding the propellant with particles is that the average molecular weight of the propellant is increased, which increases thrust but decreases I_{sp}. Volumetric absorbers are still in the developmental stage but, with reasonable funding, could be available for application within ten years or so.

Table 2.12 presents the physical and performance characteristics of an STP system scheduled for testing in approximately ten years.

Table 2.12

Solar Thermal Propulsion System Characteristics

I_{sp} (sec)	870[a]–1200
Thrust chamber input power, Pc (kW)	1500 at 1 AU
Concentrator area (m^2)	~1200
Propellant flow rate (kg/sec)	$2P_c$ (Watts)/(9.81 I_{sp})^2
Chamber power conversion efficiency, η_T	63.3 percent[a]
Thrust (N)	$2\eta_T P_c$ (Watts)/(9.81 I_{sp})^2 = 222[a]
Propulsion system T/W	0.117[a]

[a]For the rhenium tube cavity heat exchanger system.
Note: These data were extracted from Laug (n.d.).

In summary, STP, although first proposed at least 34 years ago, still remains to be demonstrated in a space vehicle test. Thus, although the rhenium coil thruster technology has been demonstrated in ground tests, the successful integration of solar concentrator and thruster technologies into an operational solar thermal rocket still faces a number of technical hurdles. In particular, the ability of solar concentrators to maintain the quality and shape of their reflecting surfaces in a space environment remains to be demonstrated.

Beamed Energy Propulsion

Beamed energy propulsion systems can use either thermal or electric thrusters of the types discussed previously in this section. Although, in principle, beam systems can produce thrusts high enough for ETO applications, in practice, the power levels required are extremely high. For example, assuming an I_{sp} of 1200 sec for a thermal thruster, the radiated power required to produce the thrust of a single SSME, 470,000 lb, would be approximately 30 GW. Laser power levels of this magnitude are well beyond the current state of the art, and even if they were available, the operational hazards involved make the use of such ground-based laser systems doubtful.

Both laser and microwave power sources can be either ground based or orbital based. In the former case, the deployment and maintenance costs would be considerably less than in the latter case. For continuous illumination of a space vehicle, however, multiple ground stations would be required. Also, in the case of ground-based lasers, additional atmospheric transmission losses would occur. Assuming diffraction-limited optics, microwave beaming systems are limited to a transmission distance corresponding to geosynchronous orbital altitude, while for lasers operating at near-visible light wavelengths, transmission to Lunar distances may be possible. Given the range limitations of beamed power propulsion systems, the most likely applications for SEI would be for orbital transfer vehicles (OTVs).

With regard to the onboard propulsion equipment, a laser thermal propulsion (LTP) system would be very similar to the STP system discussed earlier. Laser light is focused by an inflatable concentrator into the thruster to heat the propellant, typically hydrogen. The laser beam spot intensity is higher than that of an STP system, resulting in an I_{sp} of about 1500 sec.

In the case of laser electric propulsion (LEP), the tuned laser light illuminates a solar photovoltaic array made up of gallium arsenide cells. Because the laser beam can have a much higher intensity than sunlight at 1 AU, the specific mass of the array can be less than that of an SEP system. Ion thrusters would most likely be used in an LEP system.

A microwave thermal propulsion (MTP) system uses microwave energy to heat the propellant. Two possible techniques might be used to heat the propellant. Both of these, MET propulsion and ECR, are discussed under Electric Propulsion.

The last beamed energy propulsion system is microwave electric propulsion (MEP). In this system, a rectifying antenna converts the microwave radiation to electrical energy, which then is used to power ion thrusters. Rectennas have conversion efficiencies of about 85 percent, so an overall efficiency of beam power to jet power of more than 50 percent can be expected.

Both the MTP and MEP systems would have very large collector antennas. To operate in geosynchronous orbits, the antenna would require a diameter of about 1 km.

Solar Sail Propulsion

A solar sail uses radiation pressure from the sun to produce a propulsive force. Because of the small force involved, solar sail vehicles must be deployed at high altitudes—greater than 2000 km—to minimize residual atmospheric drag.

Because no propellant is required, the I_{sp} of a solar sail is infinite. The radiation pressure force, 9 N/km^2 at 1 AU, results in low accelerations and long trip times. Typically, solar sails have areas greater than 1 km^2.

Solar sails were extensively studied at JPL for the Halley Comet mission. Analyses were made of sail fabrication techniques, sail deployment, and trajectory control. Although the Halley Comet mission was not undertaken by the United States, the research concluded that solar sails are technically feasible.

An important performance parameter of a solar sail is areal density. This parameter directly determines the acceleration of the sail. Included in the areal density calculation is the mass of any structure required to support the sail. The sail material itself is typically kapton with a silvered or aluminized reflecting surface.

Currently, there are two basic approaches to constructing solar sail vehicles. In the first, studied for the Halley mission, the vehicle is built on Earth and launched into orbit, where the sail is unfurled. Such a sail needs to be relatively thick in order to withstand the wear and tear of being folded and packed into a launch vehicle and then unfolded and erected in orbit. In this case, "relatively thick" means a sail thickness of 2.5 microns. Sails of this type would have an areal density of about 5×10^3 kg/km^2.

Another sail concept, proposed by Garvey (1987) and Drexler (1978) would have the sail fabricated in orbit. Without the need to be folded and unfolded, such sails could be much thinner, 0.015 to 0.2 microns thick. The areal density of these sails would range from 10^3 kg/km^2 (Garvey) to 300 kg/km^2 (Drexler). Thus the Garvey and Drexler sails would, for a given area, have both a higher acceleration and a smaller mass than Earth-launched sails. However, on-orbit fabrication would require a substantial infrastructure that would be costly to establish and maintain. In all cases, OTVs would be needed to transfer the solar sail vehicle from LEO to an orbital altitude of at least 2000 km.

Two specific sail designs have been studied extensively: (1) a square sail supported by a lightweight boom system and (2) a heliogyro sail, which is rotated like a propeller or helicopter rotor. The "blades" of sail material are unrolled and stabilized by centrifugal force. Because of its angular momentum, the heliogyro is more difficult to turn than is the square sail, but, at the same time, it is less sensitive to disturbances.

Technical issues that need to be resolved for both square and heliogyro sails are the deployment and control of large flexible structures. For the advanced space-fabricated sails, on-orbit assembly techniques must be developed along with the materials that would permit a reduction in areal density by factors of five to ten. Finally, the ability of sail materials to withstand long-term exposure to the space environment (micrometeorites, solar protons, etc.) without undue performance degradation *must* be established.

Magnetic Sail Propulsion

Magnetic sails, or *magsails*, are devices that interact with the solar wind to produce a drag force that can be used for propulsion. A cable, a few millimeters in diameter, is fabricated from superconducting materials. The cable is formed into a loop or hoop that is tens of kilometers in diameter. Passing current through the loop creates a magnetic dipole that interacts with solar protons, resulting in a drag force upon the loop. This drag force acts radially outward from the sun. By turning the dipole, a force perpendicular to the radial drag force can be generated. Like the solar sail, the magsail uses no propellant, so I_{sp} is infinite. The magsail thrust is predicted to be about 200 N for a 64-km loop diameter and a magnetic flux density of 10^{-5} tesla.

The magsail concept is new, and many technical and operational questions remain to be resolved. Among the technical issues are superconductor technology, temperature control, structural design, and the effects of the space environment upon the superconductive material. Operational issues include the deployment and erection of the loop and attitude control. In addition, the magnetic field of the loop could trap protons, which might pose a radiation hazard for crew or cargo.

Magsail designs by Zubrin and Andrews (1989) assume a significant advance in high-temperature superconductor technology. Current densities on the order of 10^{10} amps/m^2 must be achieved. Current materials are subject to flux creep in the presence of a magnetic field, which results in resistance in the superconductor and a reduction in current density and critical temperature. This would significantly degrade magsail performance.

In summary, it is evident that magnetic sails are at a very early stage of development, with their feasibility depending upon the successful resolution of a number of technical and operational issues. Even if proven feasible, it appears that magsail operation will be restricted to heliocentric space. It is not clear how close to a planet's magnetosphere magnetic sails can operate. Thus, OTV will be required to service magnetic sails operating in 1 AU orbits.

EARTH-TO-ORBIT LAUNCH SYSTEMS

The launch systems discussed in this subsection cover a variety of types and thus technologies. Payload capabilities to LEO range from hundreds of kilograms to hundreds of metric tons.

For rocket boosters, the technologies involved are relatively mature and thus performance improvements over the next 20 to 30 years will most likely be evolutionary and predictable. The current technology emphasis, as exemplified by the ALS, is to reduce manufacturing and maintenance costs and to increase system reliability. However, one development that could dramatically improve the payload fraction delivered to orbit would be the incorporation of nuclear thermal propulsion (NTP) systems in the upper stages of launch vehicles.

At the other end of the spectrum are vehicles like the national aerospace plane (NASP) and systems such as electromagnetic launchers (EMLs), where the technologies are still developing. The launch capabilities of both the NASP and EMLs remain to be determined, as do operational factors such as reliability and cost.

Ultra-Heavy-Lift Launch Vehicles

A rich background exists of ultra-heavy-lift concepts, including unmanned launch vehicle concepts and design/operation approaches. These could serve as a foundation for development of a new launcher to provide SEI payload capabilities ranging approximately from 500,000 to over 2,000,000 lb. Many of these concepts date back to the late 1950s and early 1960s, when aggressive space exploration endeavors beyond Apollo first appeared likely to materialize. The concepts generally involved development of extremely large propulsion devices and structures. Uncertainties regarding manufacture, achievable performance, and other scaling problems were addressed in many, mostly successful, incremental hardware feasibility demonstrations. Overall concept emphasis (aside from heavy-lift capability) was on achieving markedly lower cost-per-pound-of-payload-to-orbit through simple, modest technology designs that provided comfortable design and operational safety margins and through varying degrees of component or overall vehicle reusability.

Vehicle configurations ranged from single stage to as many as five or six stages and included all-solid propellant designs, all-liquid, or a combination, typically liquid core and/or upper stages with solid boosters. Launchers in this size class require new launch sites and infrastructure, as well as unique assembly and handling procedures. Many innovative approaches were proposed and evaluated.

The following few examples very briefly illustrate the kinds of characteristics these very large concepts encompassed. If a long-term commitment to SEI is ultimately made, this entire body of past work might be seriously reviewed with an eye toward retaining salient features and modernizing/modifying the concepts to incorporate today's technology, where appropriate, as a means of satisfying the massive initial mass to low earth orbit (IMLEO) requirements of an aggressive SEI program.

In the early 1960s, NASA sponsored a series of studies known as NOVA, Post-Saturn, and Post-NOVA. They included concepts such as NEXUS and RHOMBUS. In the NEXUS studies, blunt-shaped single, one-and-a-half, and two stage designs were considered, with single stage the desired goal. The approach stressed simplicity in design, manufacture, and operation, as well as total vehicle recovery, to achieve low operating cost. NEXUS was powered by a large, high-pressure, throttleable, LOX/H_2 truncated-plug nozzle engine using multiple thrust chamber modules. Hydrogen was contained in a central tank and LOX was carried in a toroidal tank made up of compartmented spheres. Payload-to-gross-weight ratios of .042 and thrust-to-gross-weight ratios of 1.3 were estimated. A 24-million-pound gross weight design provided 1 million pounds of payload to LEO and 2 million pounds was thought possible with a 48-million-pound gross weight. Vehicle base diameters were 164 and 202 ft, respectively, at a common height of 400 ft. Risks in achieving the required single stage mass fractions and I_{sp}s with then existing technology were recognized, and one-and-a-half and two stage designs were studied as backups.

In contrast, some NOVA designs used as many as five to six stages, incorporating various numbers of F-1, M-1, J-2, and RL-10/LR-115 engines.[2] One 360-ft-tall vehicle used LOX/RP-1 (eight F-1s @ 12 million pounds of thrust) in the first stage, LOX/H_2 in the intermediate stages, and LOX/F_2 in the upper stage.

Another innovative concept, called SEA DRAGON, was capable of 2 million pounds of payload. It was extensively investigated by Aerojet Corporation. It incorporated two liquid pressure-fed stages using LOX/kerosene and LOX/H_2 in a simple, low cost, reusable configuration. The first stage was recovered via a parachute-like drag device and the second stage via a heat shield and drag device. SEA DRAGON was to be built "ship fashion" in dry dock, fueled at sea, and water launched. Rudiments of this concept are currently being explored in a privately funded experimental program.

As mentioned in the subsection on chemical propulsion, very large solid rocket motors were successfully demonstrated in the 1960s, the largest being a full-length (160 ft), 260-in. diameter motor delivering 7 to 7.5 million pounds of thrust. It was believed that solid motors 30 to 50 ft in diameter might ultimately be possible (a 30-ft diameter motor would produce

[2]The M-1 engine was under exploratory development by Aerojet in the 1960s. Over $100 million was invested before the program was terminated. It was a 1.5-million-pound thrust LOX/H_2, gas generator cycle engine intended for primary use in the NOVA vehicle second stage. It was 26 ft high with a nozzle exit cone diameter of 18 ft, and had a vacuum I_{sp} of 428 sec. Follow-on versions were believed to be scalable to the 2-million-pound thrust level.

about 10 million pounds of thrust). Four of the full-length, 260-in. motors were proposed as strap-ons to the Saturn V to give an 820-klb payload capability to LEO.

Boeing (with NASA sponsorship) conducted extensive ultra-HLLV (heavy-lift launch vehicle) studies using large solids in all-solid stage designs as well as combined liquid/solid configurations. A proposed vehicle family (using the latter approach) appeared capable of tremendous lift capability, as shown in Table 2.13. The family used a single stage, LOX/H_2 liquid core that produced 36 million pounds of total thrust, augmented with various numbers of full-length, 260-in. solid strap-on boosters.

Table 2.13

Typical Lift Capability of Vehicle Family Using Solid Strap-On Boosters

Configuration	Payload-to-LEO (million pounds)
1) Liquid core + (2) 260-in. solids	1.2
2) Liquid core + (4) 260-in. solids	1.7
3) Liquid core + (8) 260-in. solids	2.4
4) Liquid core + (8) extended-length 260-in. solids	2.8

We have only touched lightly on the ideas and approaches that are available from the past (and in some cases fairly recent efforts) that could provide a basis for future ultra-HLLV development.

Improved Saturn V

Assuming that a payload capability ranging from 250 to over 300 klb is desired, an option some feel might save time and money compared to a complete new start would be to revive the Apollo Saturn V vehicle or create a modernized configuration. The Saturn V is a three stage vehicle, 33 ft in diameter and approximately 365 ft in length (when topped with the Apollo payload). Fully fueled, it weighs 6.1 to 6.4 million pounds and delivers 250 to 280 klb of payload to LEO; hence, payload-to-gross-weight ratios of 4 to 4.5 percent are achieved. Characteristics of the three stages are summarized in Table 2.14.

Table 2.14

Saturn V Stage Characteristics

	1(S-IC)	2(S-II)	3(S-IVB)	4(IU)
Weight				
Airframe (lb)	138,900	46,000	13,500	400
Engine section (lb)	127,200	24,700	6,100	
Astrionics (lb)	8,500	5,700	4,400	3,070
Propellant (lb)	4,351,900	986,100	236,800	260
Interstage (lb)		9,600	7,700	
Gross weight (lb)	4,626,500	1,062,500	260,800	3,730
Configuration				
Length (ft)	138.0	81.4	59.0	3.0
Diameter (ft)	33.0	33.0	21.7	21.7
Propulsion				
Engine type	F-1	J-2	J-2	
Manufacturer	Rocketdyne	Rocketdyne	Rocketdyne	
Number of engines	5	5	1	
Engine thrust/engine (lb)	1.5×10^6	205,000	205,000	
Propellant type	RP-1/LO_2	LH_2/LO_2	LH_2/LO_2	
I_{sp} (sec)	264	423	426	
Restarts (number)	0	0	1	

One approach to recapturing the past capability is to retain the original design configuration, through refurbishment of existing surplus hardware and/or the reopening of all Saturn V production lines. An alternative approach is to upgrade the vehicle through upper-stage enhancements. While it would still involve reopening production of the 1.5-million-pound thrust LOX/RP-1 F-1 engine, the S-IC first stage may not require major modification for an upgraded vehicle. Most benefits of a Saturn V derivative vehicle might accrue through improvements in the second (S-II) and third (S-IVB) stages, where the previous 205 klb thrust LOX/H_2 J-2 engines could conceivably be replaced with current or upgraded SSMEs or, potentially, the space transportation main engines (STMEs) being explored in the Advanced Launch Development program (ALDP). For example, three SSMEs or STMEs might replace the five original J-2s in the second stage for less total weight and higher performance.

Regardless of the approach taken, some modifications and new construction would undoubtedly be required at Kennedy (or elsewhere) to facilitate assembly, checkout, and launch of such vehicles. Moreover, current environmental concerns may weigh heavily against use of large hydrocarbon-fueled boosters.

Detailed analyses would be required to ascertain the ultimate performance and overall cost benefits attainable with Saturn V derivative vehicles for providing heavy-lift launch capabilities for SEI.

Advanced Launch System

The ALS is a totally new, partially reusable, unpiloted launch system concept under study by NASA and the USAF. In July 1987, seven contractors were awarded $5 million each for one year to conduct concept definition studies of a family of vehicles that would dramatically reduce the cost-per-pound of payload to LEO (by a factor of 10, it was hoped) and markedly improve reliability, produceability, and operability. The general approach is to trade launch vehicle performance efficiency for low cost and high reliability by incorporating design and operating margins (such as engine-out capability) and using redundant subsystems that are highly fault tolerant. Current launch infrastructure would be reduced, simplified, and standardized. A typical family of ALS configurations is shown in Fig. 2.1.

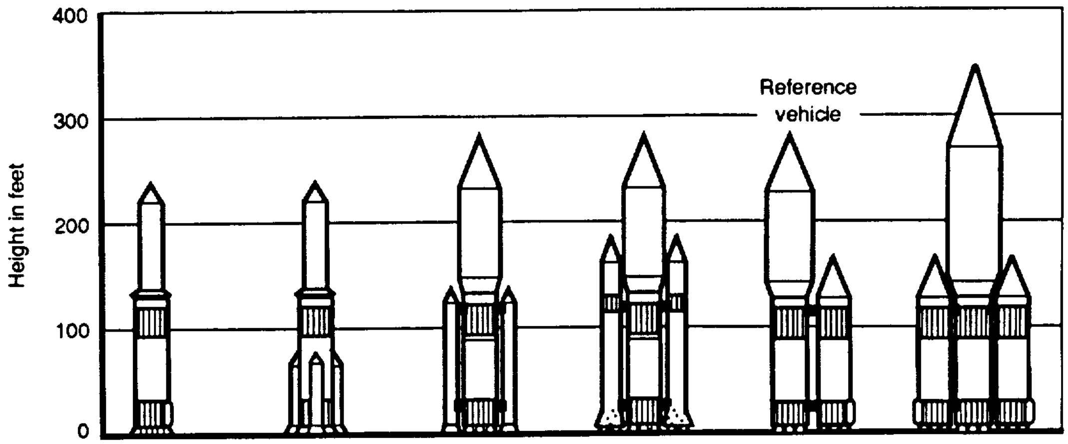
Payload capabilities from 40 to 300 klb can be provided by a modular approach involving two new propulsion developments: a 580-klb thrust LOX/H_2 pump-fed engine (the STME) that can provide an I_{sp} of about 430 sec at roughly 60 percent of the current SSME cost, and solid rocket booster propellants that provide environmentally clean exhausts at lower cost and perhaps improved performance compared to current Shuttle SRMs. It is estimated that a moderate-payload version of ALS might be operationally available by the year 2000, given adequate funding and early go-ahead, and that a launch rate of 20 to 30 per year may be possible post-2000.

In late 1989, DoD decided not to proceed with procurement of an ALS at that time and restructured the program as a technology development effort, the Advanced Launch Development Program (ALDP). The new program emphasis is in three areas:

1. Develop technology to improve operability and cost of current launch vehicles.
2. Develop technology for future launch systems.
3. Develop prototype hardware—in particular for the low-cost LOX/H_2 STME.

NASA's Marshall Space Flight Center now has the lead role for the STME effort and is continuing to evaluate various ALS-type launch vehicle options for SEI, with particular emphasis on extending the payload lift capacity, as indicated in the vehicle family shown in Fig. 2.2.

The L3 and L4 configurations have LEO payload capacities of 330 and 450 klb respectively; however, new launch sites may be required at these sizes. Earlier ALS launch site selection studies suggest that Kennedy Space Center may be limited to launches of no more than 300 klb payload because of safety considerations (overpressure, etc.). Cost (in year-2007 dollars per pound) and reliability estimates for this ALS family are also shown in the figure.



Payload (thousands of pounds)	40 to 80	40 to 80	80 to 110	100 to 251	80 to 120	120 to 300
Booster:						
Type	Stage & half	ALS-SRM	SRMU	STS LRB	ALS LRB	ALS LRB
Propellant	LO_2/LH_2	Clean solid	Clean solid	LO_2/LH_2	LO_2/LH_2	LO_2/LH_2
Number	N/A	3 to 8	2 to 4	2 to 6	1	2 to 4

KEY: ALS = advanced launch system; LRB = liquid rocket booster; SRM = solid rocket motor; SRMU = solid rocket motor unit; STS = space transportation system.

SOURCE: U.S. Congress, Office of Technology Assessment (1990).

Fig. 2.1—Typical Family of ALS Configurations

Shuttle-Derived Vehicles

For the past several years, NASA has been studying a wide variety of Shuttle follow-on vehicles, both manned and unmanned. These shuttle-derived vehicles (SDVs) would incorporate, in various ways, both current Shuttle hardware and modified or new components such as

- Modified/improved external tank (ET)
- Upgraded SSMEs or STMEs
- Advanced solid rocket booster motors (ASRMs)
- Pump- or pressure-fed liquid rocket boosters (using LOX/RP-1 or LOX/H_2 propellants)
- Hybrid (solid fuel/liquid oxidizer) boosters
- Recoverable propulsion/avionics modules
- Cargo carrier and payload shroud

Several versions of manned follow-on Shuttles have been considered which, in varying configurations, would include an improved ET, upgraded orbiter engines, and advanced solid or liquid rocket boosters.

Among the many unmanned, heavy-lift, cargo launch derivatives under study, the Shuttle-C is closest to providing an interim capability until heavier lifters can be developed. It is estimated that this vehicle could be operational within 4 to 5 years from program go-ahead to provide 85 to 150 klb of payload to LEO, and would use most of the Shuttle-proven subsystems and launch infrastructure. The current manned orbiter would be replaced with a cargo carrier and payload shroud, powered by two or three existing SSMEs, while retaining the expendable ET and reusable solid rocket boosters (a two-SSME version is shown in Fig. 2.3). While providing an early improved-lift capability, this vehicle would have modest effect on substantially reduced launch costs relative to other, longer-term HLLV options.

Beyond Shuttle-C, NASA is considering other Shuttle-derived HLLVs with LEO payload capabilities as high as 400 klb; however, a 300-klb configuration is currently favored to avoid construction of a new launch complex. The basic rationale for a SDV approach is to save the up-front development costs that would be required for a completely new launch system such as the ALS.

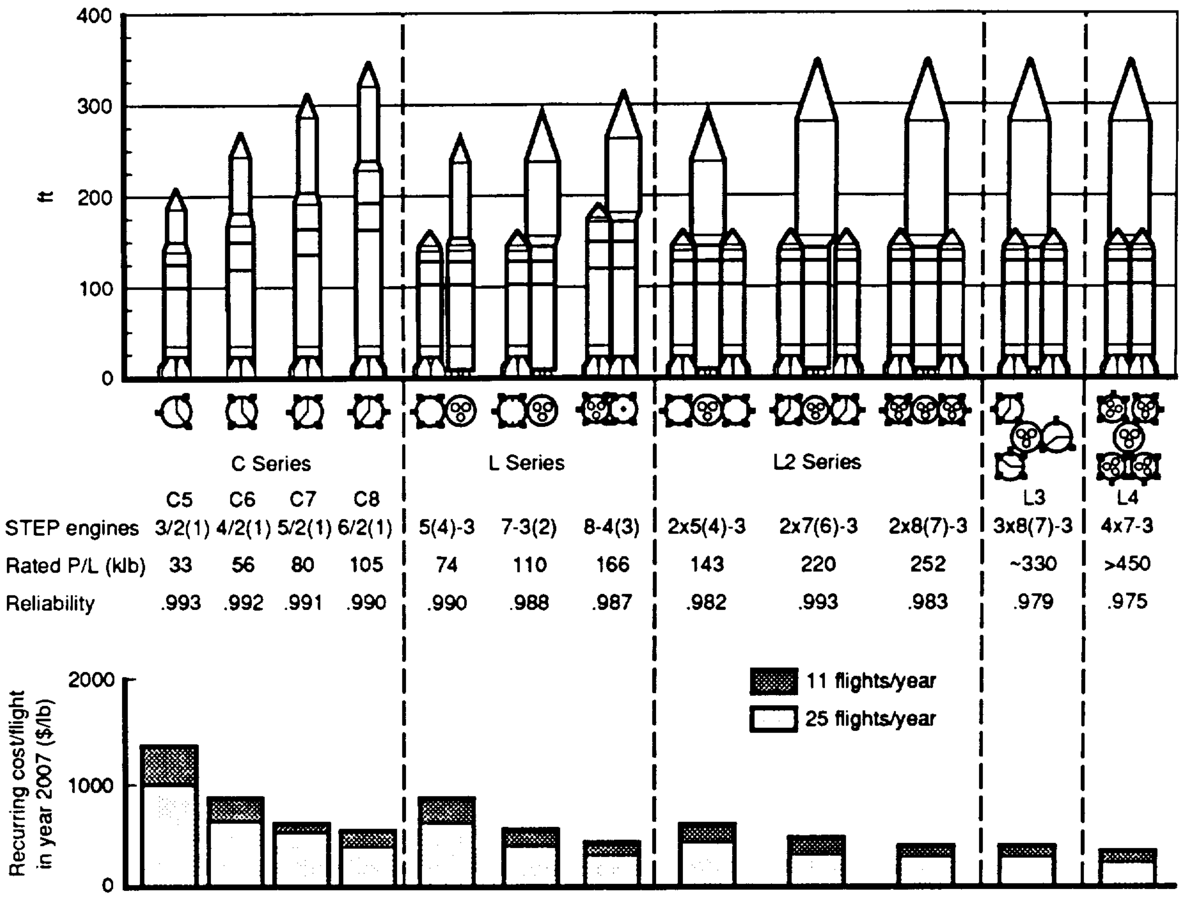
A sampling of the Shuttle-derived HLLV configurations currently under evaluation by NASA and their characteristics is shown in Fig. 2.4.

NASP/NASP-Type Vehicles

NASP is projected to be a manned vehicle designed to take off horizontally from conventional runways and accelerate to high hypersonic speeds (Mach 20 to Mach 22). At this point a rocket motor provides the remaining delta -V to place the vehicle into a circular orbit. Upon completion of the mission, the vehicle deorbits and lands at an airport. The terminal portion of the flight is powered so that NASP has a cross-range capability. Unless special runways are constructed, the gross takeoff weight of NASP-derived vehicles (NDVs) would be limited to about 1 million pounds.

With projected payloads on the order of 13,000 kg, the most likely application of NDVs for SEI would be the transport of personnel and high priority cargo. Because of aerodynamic considerations, the payload bay of NASP will have a limited volume. Thus, it will not be suited for the delivery of low-density cargo such as liquid hydrogen.

Currently, NASP, or the X-30, is being designed by a consortium of airframe and engine contractors. Research work has been under way since 1984.



SOURCE: Hueter (1990).

Fig. 2.2—Advanced Launch System: the ALS Family

The primary technology drivers in the development of a NASP-type vehicle are the propulsion system and the high-temperature, high-strength materials needed for the engine and certain parts of the airframe. In addition, vehicle flight path and attitude control must be carefully integrated with the control of the propulsion system to maintain stable operation over the entire flight envelope.

Technical issues include the following:

- Supersonic combustion ramjet operation at speeds in excess of Mach 8, including the issues of
    1. fuel-air mixing

2. skin friction with reacting flows
3. finite rate chemistry
4. maintaining inlet/nozzle efficiency

- Integration of low speed ramjet and scramjet propulsion systems with airframe, including the issues of
  1. boundary layer transition
  2. shock wave interactions
- Vehicle stability under engine-out conditions
- Aerothermoelastic effects

Current ground facilities that provide the proper enthalpy and Reynolds number conditions are limited to Mach numbers less than 10. Computational fluid dynamics can be used to aid vehicle design, but actual flight testing will be required to resolve most of the issues listed above.

Air-Launched Vehicles

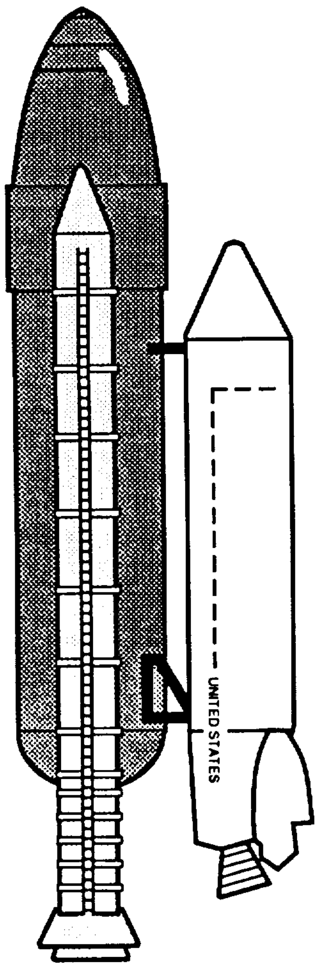
Another approach to flexible access to space is to use an aircraft as the first stage of a launch system. Usually the second stage is a rocket-propelled vehicle, although designs have been proposed in which the second stage employs scramjet propulsion. Again, because of runway limitations, total gross takeoff weights much in excess of 1 million pounds are not feasible.

With relatively small payloads, the most likely application of this type of launch system for SEI would be the transport of personnel and light cargo.[3]

A major design issue for this type of system is the Mach number at which the second stage is launched. If subsonic staging is employed, the delta-V required by the upper stage to reach orbit is still substantial, resulting in a high propellant fraction. On the other hand, if supersonic staging is used, then the size of the first stage grows, automatically limiting the upper-stage weight because of the runway constraint.

A number of studies were made in the recent past using Boeing 747-size aircraft as the first stage. These studies indicate that for a launch at about Mach 0.85 at an altitude of 35,000 to 40,000 ft a payload of approximately 5 klb can be placed into a 100 nmi polar orbit. The rocket stage that carries the payload has an initial weight of about 275 klb. At the other extreme, the German Sanger concept would use turboramjets to achieve a Mach number on

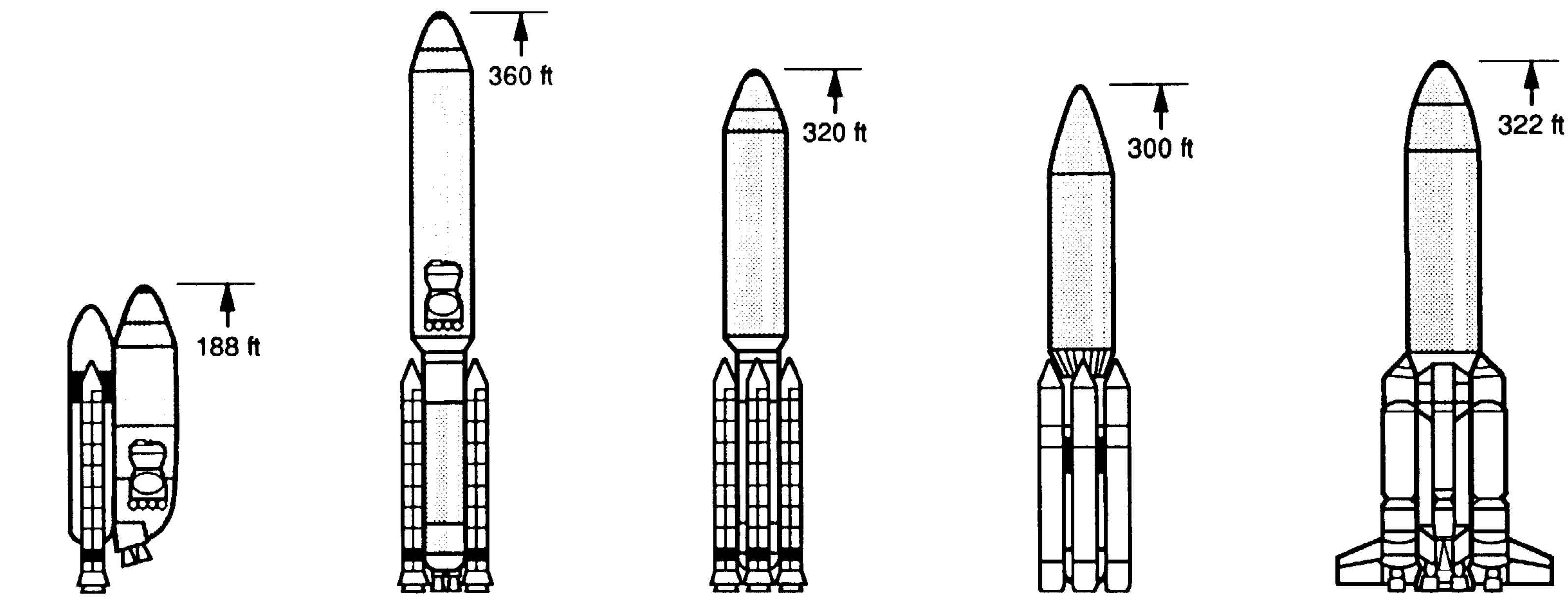
[3]The "Pegasus" air-launched space booster, developed in a privately funded joint venture, is currently operational for placing small payloads (600 to 900 lb) into LEO. The launch vehicle weighs about 42 klb and is launched from a B-52 aircraft at Mach 0.82 and a 40,000-ft altitude.



- Standard 4-segment SRBs (reusable)
- Standard ET (expendable)
- Orbiter boattail (expendable)
  - 2 SSMEs (remove SSME #1)
  - Remove verticle stabilizer
  - Remove body flap
  - Cap SSME #1 feedlines
  - OMS pods (do not install OMEs, RCS tanks, and 4 RCS thrusters/pod)
  - RCS performs circularization and deorbit
  - Cover and thermally protect SSME #1 opening
- Payload carrier (expendable)
  - New shroud/strongback
  - Skin/stringer/ringframe construction of Al 2219
  - 15 x 82 ft usable payload space
  - 15 x 60 ft changeout on pad capability
- Avionics
  - Uses mature design components from STS and other applications
  - Requires some new integration and software
- Eastern test range payload:
  - 114 klb (160 nmi/28.5° inclination)
  - 109 klb (22 nmi/28.5° inclination)

SOURCE: Hueter (1990).

Fig. 2.3—Shuttle-C Configuration



STS evolution (SM)

- 2 ASRMs
- Modified ET
- 3 SSMEs @ 104%
- Shuttle-C boattail
- 3rd stage—6 ASEs
- 40 ft D x 50 ft L payload envelope
- Performance to 220 nmi/28.5° = 193 klb

STS evolution (IL)

- 2 ASRMs
- Modified ET
- 3 SSMEs @ 104%
- New boattail
- 3rd stage—6 ASEs
- 40 ft D x 50 ft L payload envelope
- Performance to 220 nmi/28.5° = 265 klb

STS derived HLLV

- 4 ASRMs
- 4 x SSMEs on 33 ft D core
- Recoverable P/A module
- 40 ft D x 98 ft L payload envelope
- Performance to 220 nmi/28.5° = 300 klb

STS/ALS evolution

- 4 LOX/LH_2 boosters, 4 STMEs per booster
- 4 STMEs on 33 ft D LOX/LH_2 core stage
- 45 ft D x 125 ft L payload envelope
- Performance to 220 nmi/28.5° = 310 klb

New concept HLLV

- 4 LOX/HC boosters
- 6 STBEs total
- LOX/LH_2 core (33 ft D)
- 5-engine recoverable P/A module
- 50 ft D x 125 ft L payload envelope
- Performance to 220 nmi/28.5° = 400 klb

SOURCE: Hueter (1990).

Fig. 2.4—Shuttle-Derived Heavy-Lift Launch Vehicles

the order of 5.5 to 6, at an altitude of about 10^5 ft, where an unmanned payload of approximately 30 klb can be placed into orbit.

The most likely SEI application of air-launched vehicles would be for the transport of personnel and priority cargo.

Electromagnetic Launchers

The EMLs can be divided into three types: coil guns (or mass drivers), maglev (magnetic levitation) launchers, and rail launchers. Research on EMLs began in the World War I era, initially in weapon applications.

All EMLs depend upon the Lorentz force to accelerate the payload along some sort of guideway. The differences in mechanization among the three types of devices, however, influence their suitability for launching payloads into orbit.

Coil guns are induction devices that employ a series of coils that are energized sequentially at carefully timed intervals. An armature, within the tube formed by the coils, has a current induced within it by the rising magnetic field of an adjacent coil. The interaction between the armature current and the coil magnetic field pushes the armature down the tube. Sensors along the tube detect the position of the armature as a function of time so that each coil receives a current pulse at the correct time to exert a push. As described, no physical contact is required between the armature and the tube, although there are coil guns that use sliding contacts to produce a current in the armature.

Maglev launchers are basically linear electric motors, usually with the armature levitated and accelerated along a guideway. Again, the phasing or timing of current pulses is critical for successful operation. This type of technology was developed in Germany and Japan for high-speed ground transportation.

Rail launchers use a pair of rails to guide the armature. In this type of device's simplest form, current enters one rail, passes through the armature, and then returns through the second rail. The current in the armature interacts with the magnetic field formed by the current passing through the rails to push the armature, along the rails. Because the armature is in contact with the rails, a plasma is generated behind the armature that can erode the rails. Also, ohmic losses are high, so rail launcher efficiency is low relative to the induction-type devices.

To date, coil guns and rail launchers have been limited to launch masses on the order of 5 kg, with launch velocities of about 0.33 and 2 km/sec, respectively. Maglev systems have demonstrated very large payload capabilities (tens of tons) but at low velocities of approximately 0.13 km/sec.

Light Gas Guns

The velocity that can be achieved with guns or launchers that accelerate payloads by means of an expanding gas is limited in by the speed of sound in the gas. Thus, in order to maximize the limit, heated hydrogen is usually used as the working fluid. These devices have been used for more than 30 years for various types of hypervelocity tests. A velocity of over 11 km/sec has been achieved using a light gas gun.

Most light gas guns use an explosive to drive a piston that, in turn, compresses and adiabatically heats the hydrogen. When the peak temperature is reached, a diaphragm that separates the compressed hydrogen from an evacuated tube containing the payload is ruptured. Typically, the launch acceleration is on the order of 100,000 g's. In practice, barrel erosion can be a problem.

To date, only small payloads (usually less than a kilogram) have been launched from light gas guns. Even if a gun can be constructed to launch payloads in the metric ton range, launch g's would limit the types of payloads. Nose-tip erosion could cause severe aerodynamic problems if high-speed launches near sea level are contemplated. Based on past history, maintenance of a light gas gun must occur frequently, and thus the rate of launch achievable in practice might be relatively low—perhaps one per day.

SPACE TRANSPORTATION SYSTEMS

Design criteria for space vehicles are quite different from those for launch vehicles. Thrust accelerations can be considerably lower, and there are no aerodynamic loads. As a consequence, spacecraft structures, in general, can be substantially lighter than launch vehicle structures.

A number of design issues are common to all spacecraft:

- Shielding against micrometeorites and debris
- Shielding against radiation (solar protons, galactic cosmic rays, solar flares)
- Reusability
- Presence or absence of an aerobrake

If the vehicle is manned, these additional issues are important:

- Presence or absence of artificial gravity
- Crew size
- Type of life support system
- Mission duration

The type of propulsion system employed usually determines the general configuration of the vehicle.

Chemically propelled transfer vehicles can be built in a compact fashion, as illustrated by the 90-Day Study baseline design. The length-to-diameter ratio of this design, which is an important factor when aerobraking is used, is about 1.6, including the aerobrake and the trans-Mars injection (TMI) stage. At Earth departure, the payload fraction is about 32 percent, while the propellant fraction is about 61 percent.

If artificial gravity or shielding against GCR is required, the baseline design would be altered markedly. In the former case, after TMI, it would be necessary to separate portions of the vehicle, using tethers, in order to have a large enough radius of revolution that the crew does not suffer adverse vestibular effects. The penalty paid is a much more complicated design and a 15 percent increase in IMLEO. Cosmic ray shielding could have an even greater impact. Studies suggest that shielding masses greater than 10^5 kg might be required to protect the crew from the long-term effects of cosmic rays (see J. Aroesty, R. Zimmerman, and J. Logan, 1991). Such a requirement would obviously dominate the design of the transfer vehicle.

Nuclear thermal transfer vehicles require a higher length-to-diameter ratio than chemical vehicles because of reactor shielding considerations. Thus, for a Boeing-designed NTP Mars transfer vehicle (MTV), the length-to-diameter ratio is 3.7. The vehicle is basically a truss structure with the propulsion system at one end and the crew module and Mars excursion vehicle (MEV) at the other. Propellant tanks are attached along the truss. The payload fraction is about 52 percent at Earth departure, with a corresponding propellant fraction of about 39 percent.

The configuration of the nuclear transfer vehicle is advantageous from the viewpoint of incorporating artificial gravity into the design. The truss has to be lengthened to provide an adequate rotation arm gravity, but no tethers are required. As a consequence, the IMLEO penalty for antigravity is only about 7 percent. As in the case of the chemical transfer vehicle, shielding against cosmic rays would dominate the NTP vehicle design if masses on the order of 10^5 to 10^6 kg are required. This would be true for *any* manned vehicle unless very short transit times are possible.

Both nuclear electric propulsion (NEP) and solar electric propulsion (SEP) transfer vehicles tend to be large elongated structures. In the former case, the propulsion system and the payload module must be separated because of shielding considerations. In addition, a large-area radiator is usually required to reject the low-temperature heat that is a by-product of the conversion of thermal energy to electrical energy. The radiator poses a design problem in that it should be as light as possible but, at the same time, rugged enough so that micrometeorites will not damage it.

SEP transfer vehicles require solar panels with relatively large areas. These panels must be capable of tracking the sun and, at the same time, be lightweight and rugged.

Although electrically propelled vehicles are usually thought of as cargo carriers, Boeing and others have studied multimegawatt designs suitable for manned Mars missions. Multimegawatt NEP and SEP transfer vehicles have the potential for reducing both IMLEO and transit times as compared to the baseline LOX/LH_2 vehicle. On the other hand, if artificial gravity is required, NEP and SEP vehicles face a major design problem. Since both types of vehicles must thrust for a large fraction of the trip time, the thrusters have to be despun so that they can be pointed in the desired direction. In addition, for the SEP vehicle, the large solar arrays must also be despun so they can point at the sun. The resulting designs are complex, with rotating joints required between different parts of the structure.

Finally, another type of spacecraft that has been studied for the Mars mission is the cycling vehicle. Basically, a cycler spacecraft is placed into a heliocentric orbit with a period such that it passes "close" to both Mars and the Earth periodically. It is not necessary that the transit time from Earth to Mars be the same as the transit time from Mars to Earth. At the destination planet, smaller transfer vehicles rendezvous with, or depart from, the cycler, carrying personnel and cargo.

Cycler spacecraft are envisioned as very large vehicles, with the capability of transporting large payloads and providing life support for a relatively large number of personnel. In some of the more advanced versions, the cycler would employ low-thrust NEP, or possibly SEP, to continuously correct the trajectory.

Given the fact that the cycler is going to be a very large vehicle, shielding against cosmic rays and solar flares might be less of a design problem than it would be with more traditional spacecraft. However, if artificial gravity is a cycler requirement, the problems discussed above with regard to despinning the thrusters of an NEP or SEP system must be resolved.

OBSERVATIONS

Some of the submissions received propose concepts that would, in total, draw upon almost every one of the specific technologies discussed above. To properly analyze submissions as to their potential value or utility, however, it was necessary to relate technological capability to the relative effectiveness of various launch and space transportation system options in meeting Mars mission requirements. This was done as described in the next two sections.

Finally, even though a technological capability may exist, certain options might be closed because of U.S. policy or international treaties. For example, the use of nuclear energy for both launch and space vehicle propulsion could have a substantial effect upon SEI mission cost. At the moment, however, U.S. policy with regard to the use of this type of technology in either launch or space vehicles is not clear.

III. SPACE TRANSPORTATION SYSTEM OPTIONS

This section identifies a range of space transportation options for the Mars mission. Tradeoffs between mission duration and IMLEO are examined, and options that significantly reduce either IMLEO or two-way transit times, as compared to the 90-Day Study baseline, are pinpointed. Submissions that proposed either these transportation options or the technologies that support them are then analyzed in more detail. Both nonnuclear and nuclear systems are examined.

The 90-Day Study on Human Exploration of the Moon and Mars selected, as a baseline, a space transportation system that would use liquid oxygen/hydrogen propulsion systems and aerobraking at Mars and on Earth return. Since the 90-Day Study, work has continued to refine this initial concept. The physical characteristics of a current (May 1990) vehicle design appear in Table 3.1.

Table 3.1

Current NASA LOX/LH_2 Design

Element	Mass (kg)
Trans-Mars injection (TMI) stage	546,010
Inert stage mass	54,560
Propellant load	490,950
Interstage mass	500
Mars transfer vehicle (MTV)	163,732
Mars aerobrake	23,758
Crew habitation module	28,531
Consumables and resupply	7,096
Science	1,000
Inert propulsion stage	18,206
Propellant load	85,141
Mars excursion vehicle (MEV)	84,349
Mars aerobrake	15,138
Ascent stage	22,754
Descent stage	21,457
Surface cargo	25,000
Earth capture crew vehicle (ECCV)	7,000
Initial mass in LEO	801,091

SOURCE: Data provided in briefing to RAND at Marshall Space Flight Center, July 19, 1990.

The vehicle of Table 3.1 was designed for a 2016 opposition-class mission, with a duration of 565 days, including a 30-day stay on Mars. Upon approaching Mars, the MEV separates from the MTV. The MEV aerobrakes and lands on Mars, while the MTV uses aerobraking to dissipate sufficient energy to go into orbit about Mars. Upon completion of the 30-day stay, the MEV ascent vehicle rendezvouses with the orbiting MTV, which then departs for Earth. Approximately one-half day before Earth arrival, the crew moves from the MTV into the ECCV, which then separates from the MTV. The ECCV enters the Earth's atmosphere where, after sufficient braking, parachutes deploy for landing. The reference or baseline LOX/LH_2 vehicle adopted for this Note is based on the vehicle of Table 3.1 (see App. D).

When comparing different space transportation options, the most frequently used measures of merit are two-way transit time and IMLEO. Short transit times are important from the viewpoint of crew safety and comfort. The second measure, IMLEO, is a surrogate for mission cost and thus should be minimized. In the case of the 90-Day Study mission, the Mars flight trajectory employed passes close to the planet Venus. The net effect of this maneuver is to provide the spacecraft with a gravitational assist on its way to Mars. This permits a reduction in IMLEO but, at the same time, results in an Earth-to-Mars transit time somewhat longer than when a Venus swingby is not used.

Appendices C and D describe a program that has been used to establish the relationship between mission duration and IMLEO for various types of transportation systems. The program does not have the capability to include the effect of planetary swingby upon vehicle motion, so the calculated values of IMLEO are not necessarily minimum for a given set of launch conditions.

Let us first consider nonnuclear options that are variants of the 90-Day Study.

NONNUCLEAR SPACE TRANSPORTATION OPTIONS

This sub-section discusses several space transportation options considering a nonnuclear space policy:

- Option 1: Beryllium-loaded space storables
- Option 2: Planetary swingby
- Option 3: Advanced tripropellant rocket engines
- Option 4: Lunar-derived propellants
- Option 5: In-situ propellants from Mars
- Option 6: Split missions

Option 1: Beryllium-Loaded Space Storables

The general characteristics of the vehicle in LEO are the same as those shown in Table 3.1, except that the Earth-return propulsion system of the MTV is replaced by a space-storable solid system that employs metal additives such as Be or Be hydride (see Solid Systems, Sec. II). The elimination of LH_2 and LOX propellants eliminates both the problems of leakage and boil-off and the weight of propellant tanks with their heavy insulation and/or cooling systems. This results in a much more compact propulsion system, and because of this, the size of the aerobrake can be reduced, resulting in further mass savings. The penalty, of course, is a reduction in I_{sp} from 480 sec to about 350 to 400 sec.

The use of space-storable solid propellant rockets for the Mars departure propulsion proposed in submission **#100767**, entitled **Lunar/Mars Return Propulsion System**, entails a small penalty in IMLEO as compared to the LOX/LH_2 baseline system. Nevertheless, if space-storable propellants (either solid or liquid) can be formulated with I_{sp}s near 400 sec, then their use still might be attractive for long-duration missions where hydrogen leaks and boil-off could pose problems.

In Figure 3.1, for every value of mission duration, which includes a stay time of between 25 and 65 days on the Martian surface, the value of IMLEO has been minimized (see Appendices C and D). Thus, a "rubber vehicle" is being considered with only the payload delivered to Martian orbit being held constant. This payload is the MEV, the dry mass of the MTV, and the ECCV (see Table 3.1). Thus, the purpose of Fig. 3.1 and subsequent plots in this section is to enable the reader to compare, for this fixed payload, the IMLEO requirements of different transportation system/propulsion options for the Mars mission.

Option 2: Planetary Swingby

Although not a propulsion option in the conventional sense, planetary swingbys are frequently used to reduce mission propulsion requirements and thus IMLEO. The gravity field of the planet or moon involved is used to either accelerate or decelerate the spacecraft as it makes a close encounter.

The mission selected as the baseline for the NASA 90-Day Study would have a duration of 565 days, assuming a Venus swingby and a 2016 launch date. With direct transits to and from Mars, the mission duration would be decreased by about 20 percent, or 115 days. From Fig. 3.1, it can be seen that without a Venus swingby, a Mars mission duration of 450 days would require an IMLEO of about 900 metric tons to deliver the same payload to Mars orbit as the 800-metric-ton vehicle of the 90-Day Study. Thus, a Venus

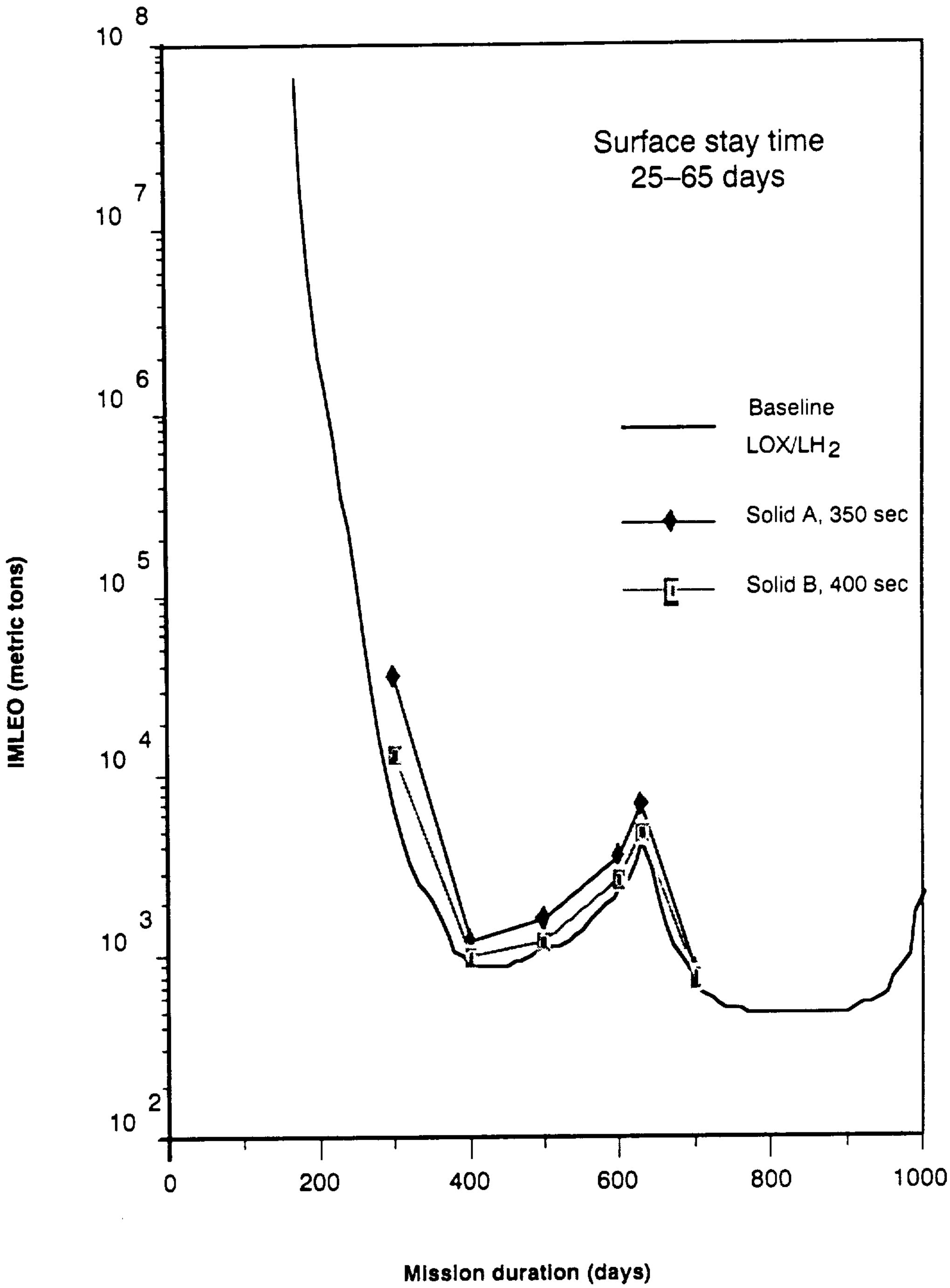


Fig. 3.1—Storable Solids for Earth Return

swingby maneuver gains savings in IMLEO of about 100 metric tons at the expense of about 115 days of additional mission time.

This comparison of swingby and nonswingby missions is only approximate. The RAND model used to determine velocity requirements for Mars missions neglects the eccentricity of the Martian orbit (see Appendix C). Because of the orbital eccentricity of Mars, both the mission duration and the corresponding minimum value of IMLEO vary markedly from one opposition launch date to another.

Submission **#100121**, entitled **Achieving Mars Transfers via Multiple Lunar Swingbys**, indicates that a gain in payload of as much as 40 percent can be achieved by using three Lunar swingbys in combination with a Venus swingby. After the last Lunar swingby, the spacecraft is on a trajectory that passes close to Venus for the final gravity assist.

A major disadvantage of using multiple swingbys of the type described above is that the total one-way transit time to Mars is about two years. This would not be suitable for manned flight, but a chemical cargo rocket could use such a concept. Another possible disadvantage is the frequency with which the various bodies involved—the Earth, Moon, Venus, and Mars—would have the appropriate relative positions to make the maneuver feasible.

Option 3: Advanced Tripropellant Rocket Engines

The general characteristics of the vehicle in LEO are the same as those shown in Table 3.1, except that advanced high-energy tripropellants with an I_{sp} of 600 sec are used in place of LOX/LH_2 (see Liquid Systems, Sec. II). The potential advantage of the use of tripropellants is a possible increase in I_{sp}, relative to LOX/LH_2, of as much as 100 to 200 sec.

Figure 3.2 compares IMLEO over a range of mission durations for both the baseline LOX/LH_2 system and one using advanced propellants. From Fig. 3.2, it is evident that the use of tripropellants, as proposed in submission **#101212**, entitled **High Energy Chemical Propulsion for Space Transfer**, and submission **#100133**, entitled **Metallized Propellants for the Space Exploration Initiative**, has the potential of substantial reductions in IMLEO as compared to LOX/LH_2. As pointed out in the discussion of chemical propulsion techniques in Sec. II, however, tripropellants have not come close to demonstrating I_{sp}s near their ideal limits. In past tests there have been problems in achieving mixing and good combustion efficiency. Metal additives, such as Be, also result in toxic combustion products that might pose problems in testing. Although past experience has been discouraging, if nuclear propulsion systems are banned in space, renewed research in various tripropellant combinations is definitely warranted.

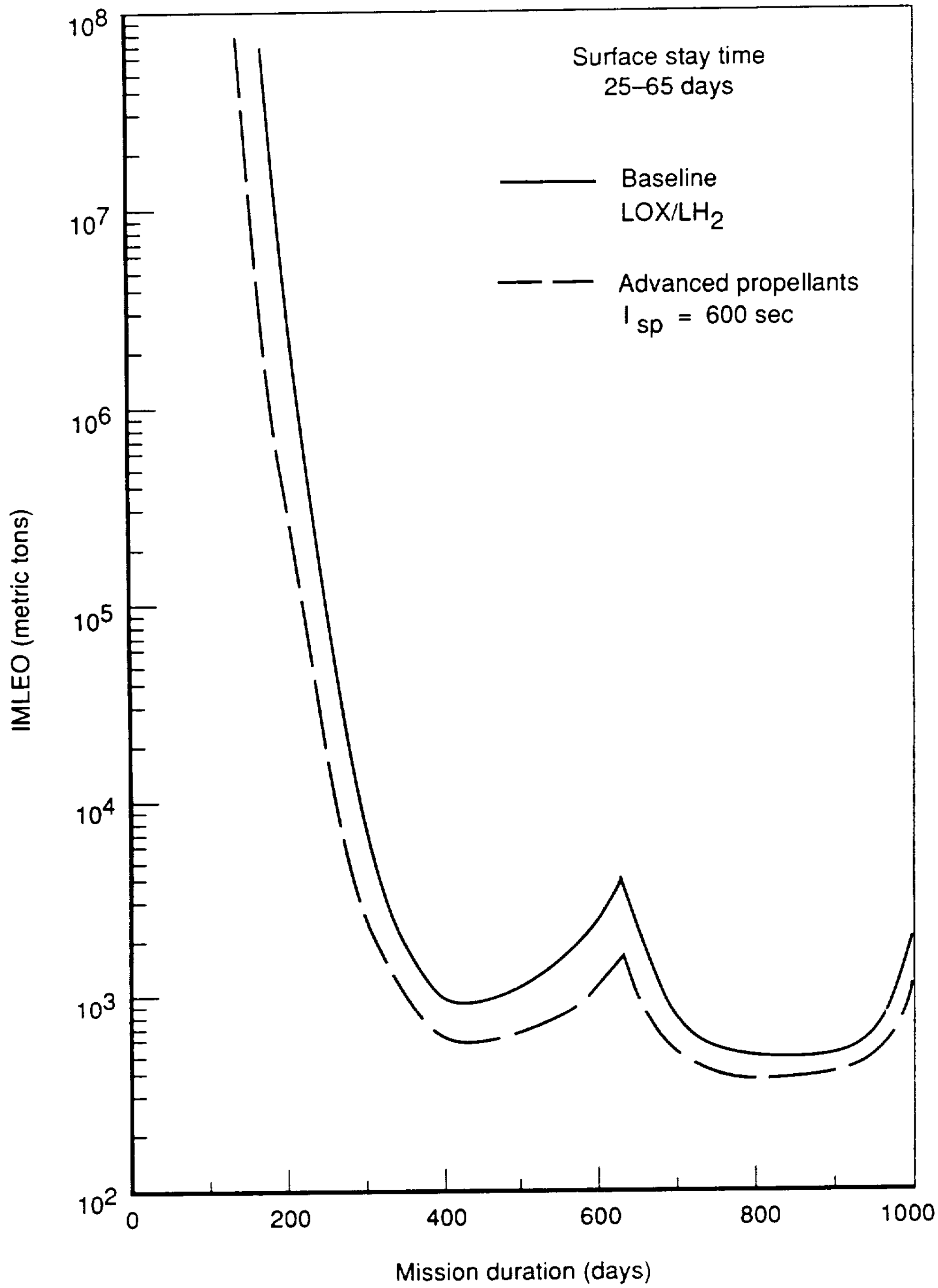


Fig. 3.2—A Comparison of IMLEO Requirements for LOX/LH_2 and Advanced Chemical Propellants

Option 4: Lunar-Derived Propellants

In this variant, the baseline system is unchanged, but it is assumed that a Moon base has been established and that large quantities of Lunar oxygen are available (see Liquid Systems, Sec. II). Using Lunar oxygen can reduce IMLEO by anywhere from about 80 percent (departure from LLO) to 60 percent (Lunar oxygen delivered to LEO). The reduction in ETO launch costs would be substantial, but Lunar launch systems would be needed to deliver O_2. Because of the differences in the gravity wells of the Earth and the Moon, operating from the Moon should require a transportation system with about 1/15 to 2/15 the capability of an Earth-based system. Cost savings would depend on the mission model assumed and the continued use of chemical propulsion systems that use O_2 as the oxidizer.

There are possible transportation nodes other than Earth and Moon orbits. One in particular, the Earth-Moon L_2 libration point, offers advantages if Lunar LOX is available. For a Mars transfer from L_2, the delta V is about 3000 m/sec less than from LEO. Spacecraft departing from the L_2 point can use both the Moon and the Earth for gravity assist.

In one operational scenario, the MTV would operate between L_2 and Mars orbit. Oxygen would be transported to L_2 from the Moon. A low-thrust OTV would carry hydrogen and other supplies from LEO to L_2. Prior to departure, the crew would leave the Moon and join the MTV.

Other libration points, either Earth-Moon or sun-Earth, could also serve as transportation nodes. Careful consideration should be given to these alternatives to LEO or LLO staging.

Although it is certain that oxygen can be found on the Moon, the presence of water and thus easily obtained hydrogen is not a certainty. A number of submissions that propose the development and use of in-situ propellants have been aggregated under submission **#100932**, entitled **Lunar-Derived Propellants**. These submissions are discussed in some detail in App. E.

The first step in delivering Lunar LOX to any transportation node is the development of a surface-to-space transportation system. If chemical rockets are used, then it would be desirable to develop engines that can use Moon-derived fuels such as aluminum rather than depend upon fuels delivered from Earth. An alternative to rockets would be the use of EMLs to deliver Lunar LOX to LLO.

A number of submissions propose the use of various types of EMLs to place payloads in orbit, either about the Earth or Moon. Those that are very similar were aggregated under submission **#101029**, entitled **Earth to LEO Electromagnetic Launch**. A different type of launcher is proposed in submission **#100575**, entitled **Lunatron—Lunar Surface-Based**

Electromagnetic Launcher (see Electromagnetic Launchers in the Earth-to-orbit launch system portion of Sec. II for a brief discussion of the technology). The Lunatron is a linear electric motor accelerator that, when developed, will have a greater payload capability per launch than would a coil gun. However, an on-orbit system of the general type described below would still probably be needed. The Lunatron approach is examined in more detail in App. F.

Even when launch is from the Moon, coil guns or mass drivers would have relatively small payloads—probably a few tons or so, at most. Thus, an orbiting platform, perhaps with SEP, would be needed to collect payloads, transfer the LOX to onboard storage tanks, and then return the empty LOX tanks to a recovery area on the Moon. The SEP LOX tanker would then rendezvouses with the MTV at the appropriate transportation node.

Submission **#101157**, entitled **Solar Electric Orbital Transfer Vehicle (SEOTV)**, proposes an SEP vehicle that employs an inflatable solar array structure. Amorphous silicon solar cells, fabricated on a flexible substrate, are attached to a plastic sheet supported by the inflatable structure. After inflation, the structure could be rigidized. Although amorphous silicon cells are, at best, only 5 to 10 percent efficient, they are many times less costly than, for example, germanium cells. The amorphous silicon array would also be much lighter than conventional arrays. This appears to be a promising approach to an inexpensive, modest performance SEP for Earth-Moon cargo applications.

It is obvious that the development of a Lunar LOX capability to support Lunar and Mars SEI activities would require an extensive infrastructure. A careful analysis of projected SEI missions is needed to establish the cost effectiveness of such an approach.

Option 5: In-Situ Propellants from Mars

The potential benefits of in-situ propellant production are even greater if the destination planet can provide both fuel and oxidizer. For this option, it is assumed that both H_2 and O_2 can be obtained from either Phobos or Deimos. The obvious infrastructure problems will be ignored for the moment. At Earth departure, the MTV consists of the habitation module, an aerobrake, the 25-metric-ton surface cargo, the ECCV, consumables, and a propulsion system for Earth return and, if necessary, to aid in braking at Mars. The propellants for Mars departure and the MEV are not included.[1]

[1]It is assumed that an MEV from a prior mission is left in Mars orbit. Its propellants are replenished from the Phobos/Deimos facility, as needed.

C-2

Figure 3.3 shows how IMLEO varies over a range of mission durations for both the baseline case and transportation option 5. From Fig. 3.3, it is evident that the availability of propellants at the destination planet markedly reduces the required IMLEO over the entire

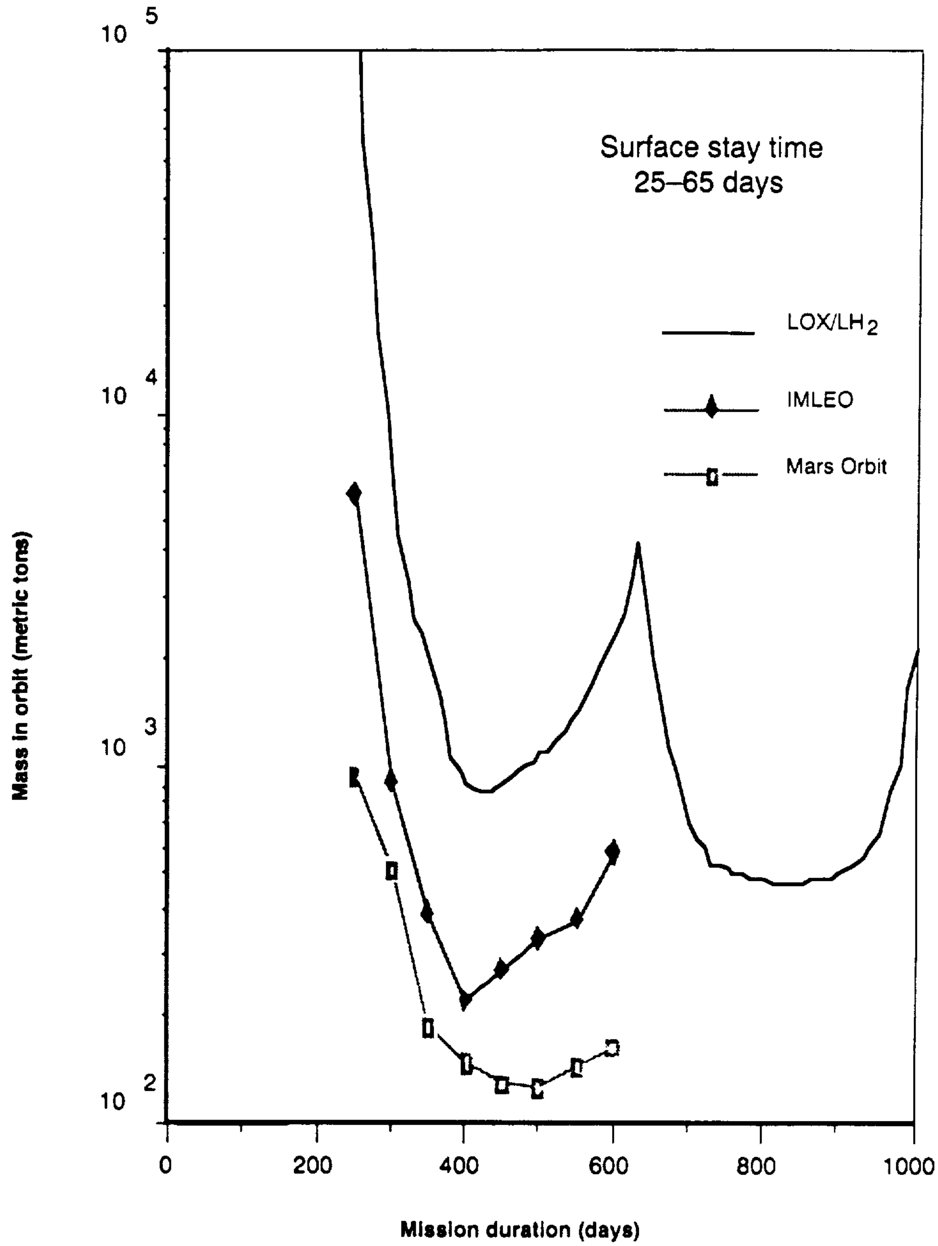


Fig. 3.3—Mass-in-Orbit Requirements When LOX/LH_2 Is Available on Either Phobos or Deimos

range of mission durations. (See App. G, which discusses submission **#101178**; entitled **In-Situ Propellants for Mars Lander—Chemical Engines**, for an analysis of in-situ propellant production in the Martian system.)

Figure 3.3 shows both the initial MTV mass in LEO and the mass of the MTV in Mars orbit prior to Earth return. Since it is assumed that the MTV directly obtains propellants from a storage facility on either Phobos or Deimos, the MTV mass in Mars orbit is not a good surrogate for operational costs.

Option 6: Split Missions

A reduction in IMLEO can result from splitting the space transportation system into two parts. A cargo vehicle would be launched first, following a low-energy trajectory to Mars. On board would be consumables, the MEV, and the propulsion stage for Earth return. Upon arrival at Mars, the cargo vehicle would go into orbit to await the arrival of the manned vehicle. The manned vehicle would consist of the MTV (minus the Earth-return propulsion system and some consumables) plus the TMI stage. This vehicle would be launched after an extensive systems check of the orbiting cargo vehicle was completed.

There are a number of possible variants of option 6. In the case of the cargo carrier, chemical, solar electric, solar sail, or possibly solar thermal propulsion might be used (see Low-Thrust Propulsion Technologies, Sec. II). The trans-Earth injection (TEI) system could use LOX/LH_2, high-energy chemicals such as tripropellants, or high-performance space-storable liquids or solids.

Previous studies by JPL and others indicate that solar sail propulsion would result in the smallest value of IMLEO, but that the one-way trip time is very long—in excess of 500 days. With nuclear systems excluded, SEP using ion propulsion would probably be the best overall selection for the cargo vehicle.

Figure 3.4 presents IMLEO as a function of mission duration for three cases: (1) the baseline, (2) split mission with manned vehicles using LOX/LH_2, and (3) split mission with manned vehicles using tripropellants. In all cases, the cargo carrier uses SEP; its contribution to total IMLEO has been calculated using data from Frisbee et al. (1989).

The SEP spacecraft that has been used as the split mission cargo carrier has a loaded mass of 185 metric tons with a payload of 100 metric tons.[2] In the examples presented in

[2] A LOX/LH_2 cargo vehicle, with a 100-metric-ton payload, would, assuming a Hohmann transfer trajectory, have an IMLEO of about 325 metric tons. Although not as attractive as an SEP vehicle when operating from LEO, a LOX/LH_2 cargo vehicle could be very attractive when operating from Martian orbit, with propellants supplied by the Phobos/Diemos facility. When used in conjunction with a manned vehicle, the total IMLEO required for a 400-day mission would be about 160 metric tons, averaged over ten manned missions.

Fig. 3.4, no attempt has been made to minimize the combined cargo and manned vehicle IMLEO for each mission duration time. Instead, the manned vehicle IMLEO has been minimized for each mission duration time (excluding mass reductions that can be realized only by using Venus swingbys), but the same SEP vehicle is always used. Once the mass in orbit required for Mars exploration and Earth return is determined, together with the number of SEP vehicles needed to transport that mass, the total IMLEO is then calculated for the particular mission duration being considered.

Submission **#100714**, entitled **The Pony Express to Mars**, proposes a variant of the above split mission option using SEPs (see App. H).

The use of a solar sail cargo vehicle proposed in submission **#101392**, entitled **Solar Sail Cargo Vessels to Reduce Mars Expedition Costs**, is undoubtedly attractive from the viewpoint of IMLEO. In the submission, a sail with an area of 4 km^2 and a mass of 19 metric tons is assumed. This appears to be a Staehle-type sail with an areal density of about $5X10^3$ kg/km^2 (see Low-Thrust Propulsion Technologies, Sec. II). The payload of the sail is stated to be 32 metric tons.[3] Recent work at JPL indicates that 26 metric tons is a more realistic estimate of the payload for this type and size of sail. Thus, four solar sail vessels of the type described above would be needed to transport 100 metric tons to Mars. A 50-kWe SEP OTV would be needed to transfer the sails from LEO to the 2000-km departure orbit. The Earth-to-Mars transit time would be about 1300 days.

Submission **#101016**, entitled **A Solar Sail Design for Space Transportation and Power Beaming**, presents another sail design that is analyzed in App. I.

A possible alternative to the 50 kWe SEP OTV is proposed in submission **#101536**, entitled **Earth-Based Microwave Power Beaming to Interorbital (LEO to and from HEO) Electrically Propelled Transport Vehicles**, which is discussed in App. J.[4]

Finally , solar thermal propulsion (STP) could be used for either the Mars cargo vehicle or an OTV to support solar sail cargo vehicle operations. The latter application is proposed in submission **#101188**, entitled **Solar Thermal Orbital Transfer Vehicle (STOTV)**. The submission does not propose a specific design, but refers to the work on STP being done at the Air Force Astronautics Laboratory (see Low-Thrust Propulsion Technologies, Sec. II).

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[3]The submission is based on work done at JPL during the late 1970s.

[4]Another possible approach to orbit transfer, tether systems, is discussed in App. Q.

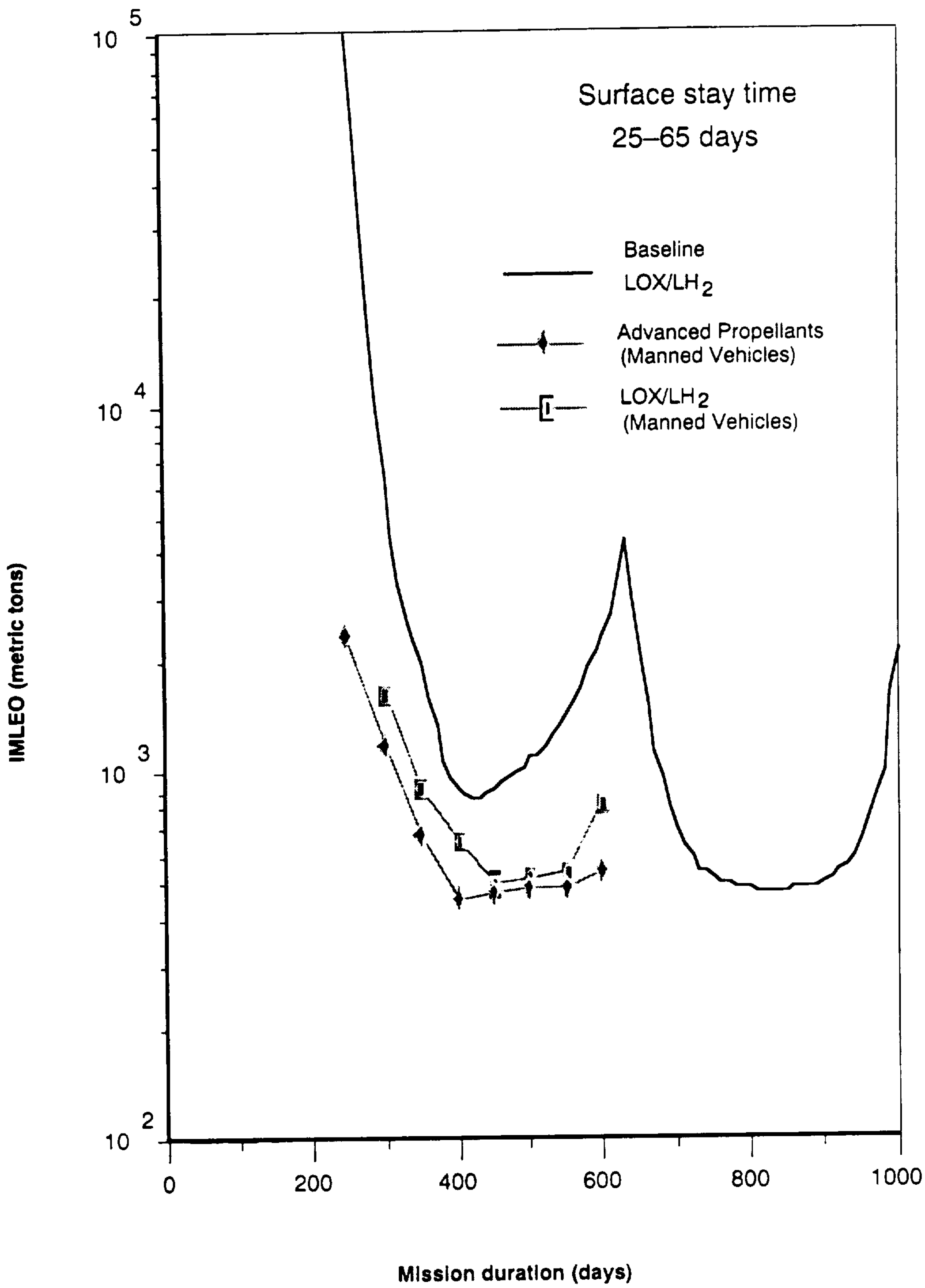


Fig. 3.4—IMLEO Requirements for Split Missions

An OTV design under study at that lab would be able to transport 13 tons from LEO to GEO in approximately 30 days. The IMLEO of the vehicle would be about 25 tons. An OTV similar to the Astronautics Lab design would be capable of supporting solar sail operations with a reduction in transfer time at the expense of an increase in IMLEO as compared to SEP.

The use of STP for the Mars cargo mission is proposed in submission **#101399**, entitled **Solar Thermal Rocket System for Orbital/Injection Transfer Vehicle**. This submission is considered in some detail in App. K.

SEP vehicles can use electric thrusters other than the ion type that have been assumed in the above examples (see Low-Thrust Propulsion Technologies, Section II). Currently, ion thrusters are further along in development than other types, but the potential performance of other electric thrusters, such as MPD thrusters, makes them promising candidates for SEI applications. Submission **#100170**, entitled **Pulsed MPD Electric Propulsion**, proposes an approach for improving MPD efficiency. This submission is examined in App. L.

It is apparent from Fig. 3.4 that split missions are advantageous in terms of IMLEO required, particularly when the two-way transit times are short. In terms of a Mars exploration program that extends over years, transportation option 6 would probably evolve into transportation option 5. Split missions could be used until an in-situ propellant manufacturing capability is established, either on the Martian moons or on Mars itself. After propellants are available in the Martian system, transportation option 5, or a combination of options 5 and 6, would be used.

Transportation option 4 could be used in conjunction with any chemical transportation system that requires oxygen as an oxidizer. Thus, a large number of other transportation options should eventually be examined to determine the least costly approach for Mars exploration missions.

Up to this point, the transportation options considered have assumed a mass of 28.5 metric tons for the MTV crew habitation module (see Table 3.1). This mass does not provide an explicit allowance for shielding against GCR. The effectiveness of various materials in shielding against GCR is very uncertain. For the volume of the 90-Day Study crew module, GCR shielding mass estimates range from 10^4 to 10^6 kg, where the material is water (see Aroesty, Zimmerman, and Logan, 1991).

In examining the effect of shielding mass upon IMLEO, a minimum energy trajectory to and from Mars has been selected. Table 3.2 shows how vehicle mass varies with shielding mass at Earth departure, Mars arrival, Mars departure, and Earth arrival. Two separate cases are considered. In the first, upon Earth arrival, the crew transfers to an ECCV for Earth return, leaving the MTV and its shielding to be destroyed upon reentry. In the second case, the MTV plus shielding is propulsion braked into LEO. For both cases, the I_{sp} is 480 sec, and the MTV is propulsion braked into Mars orbit after having separated from the MEV, which aerobrakes and lands on Mars.

Table 3.2

Galactic Cosmic Ray Shielding (chemical propulsion)
(all masses in metric tons)

Case	Shielding Mass	Earth Departure[a]	Mars Arrival	Mars Departure	Earth Arrival
1	0	526	127	69	7
2		894	276	153	84
1	10	621	165	91	7
2		1163	385	215	119
1	100	1473	511	286	7
2		3586	1369	770	432
1	1000	9990	3970	2237	7
2		27,820	11,212	6323	3561

[a]Figures indicate IMLEO required

It is evident from Table 3.2 that once the shielding mass exceeds 10 metric tons, the IMLEO requirements rapidly become excessive. Also, it appears that, in all cases, propulsion braking at Earth in order to save the MTV plus shielding for further use is not cost effective in terms of IMLEO.

The availability of high-energy propellants would help reduce IMLEO (e.g., option 3), but for shielding requirements in the range of 10 to 100 metric tons, a split mission transportation option should be considered, with the manned vehicle using the highest energy propellants available.

An alternative to shielding the crew module is to reduce the exposure time to GCR to an acceptable value. Unfortunately, the tradeoff between shielding mass required and exposure time is not well understood.

A reduction in exposure time means a shorter mission. As can be seen from Fig. 3.2, even with high-energy chemical propellants, IMLEO becomes large for mission durations on the order of 250 to 300 days. Figure 3.4 shows the reduction in IMLEO, relative to standard missions, that split missions make possible. However, there should be some combination of shielding mass and mission duration that would result in an acceptable crew risk and, at the same time, yield a minimum value of IMLEO.

NUCLEAR SPACE TRANSPORTATION OPTIONS

The use of nuclear energy for space propulsion would provide a combination of high I_{sp} and thrust that cannot be duplicated by chemical propulsion systems. Although little development work has been done since the 1970s, solid core NTP technology is currently at the stage where a flight-test article could be developed within ten years, assuming adequate funding together with a regulatory environment that is not overly restrictive (see Nuclear Propulsion Technologies, Sec. II).

Table 3.3 presents the physical characteristics of a vehicle powered by an NTP system designed by Boeing as a possible alternative to the 90-Day Study LOX/LH_2 system.[5]

Table 3.3

Boeing NTP Design

Element	Mass (kg)
Trans-Mars injection	<u>329,238</u>
Propellant load	286,146
Propellant tanks	43,092
Mars transfer vehicle	<u>54,716</u>
Crew habitation module	28,531
Consumables and resupply	5,408
Science	1,000
Propulsion, frame, and shield	19,777
Mars orbit capture	<u>177,252</u>
Propellant	151,680
Propellant tanks	25,572
Mars excursion vehicle	<u>73,123</u>
Descent aerobrake	7,000
Ascent stage	22,464
Descent stage	18,659
Surface cargo	25,000
Trans-Earth injection, Earth orbit capture	<u>100,846</u>
Trans-Earth injection propellant	59,245
Earth orbit capture propellant	27,756
Common propellant tank	13,845
Initial Mass in LEO	735,175

SOURCE: Data provided in briefing to RAND at Marshall Space Flight Center, July 19, 1990.

This NTP vehicle has been designed for a 2016 opposition-class mission. The mission duration is 434 days, including 30 days on the Martian surface.

[5]Nuclear-propelled vehicles will require an orbital transportation system to transfer them from LEO to a nuclear "safe" orbit (~1000 km).

It can be seen from Table 3.3 that only the crew habitation module is common with the LOX/LH_2 vehicle (see Table 3.1). The MEV of Table 3.3 is smaller that that of the LOX/LH_2 vehicle because, on approaching Mars, propulsion braking is used to slow the MTV/MEV combination before the MEV is released to land on Mars. Perhaps the major difference between the NTP and LOX/LH_2 vehicles is that, in the former case, a single nuclear propulsion system is used to provide the delta V required for Earth departure, Mars arrival, Mars departure, and Earth arrival.

As in the case of the LOX/LH_2 vehicle, a reference NTP vehicle, based on the Boeing design of Table 3.3, has been used to determine IMLEO requirements as a function of mission duration. The mission duration includes stay times on the Martian surface that are constrained to fall within a 25- to 65-day span. As before, only the MEV and the habitation portion of the MTV are fixed, with the propulsion system and propellant masses varying with mission duration (see Apps. C and D). Unlike the Boeing studies, this Note considers aerobraking of nuclear vehicles. The same vehicle model is used for other types of nuclear propulsion systems where the only input parameters that are varied are the I_{sp}s, the reference propulsion thrusts and T/Ws, and the average thrust accelerations.

Figure 3.5 presents IMLEO as a function of mission duration for the baseline LOX/LH_2 vehicle, a LOX/LH_2 vehicle that aerobrakes the MTV into orbit on Earth return, and two NTP vehicles. It can be seen from Fig. 3.5 that the high I_{sp}s of the nuclear systems provide an advantage in IMLEO relative to both of the LOX/LH_2 systems. Both nuclear systems use propulsion braking, combined with aerobraking, to go into orbit at both Mars and Earth. The first NTP vehicle uses a solid core, NERVA-type reactor and is based on the Boeing design of Table 3.3. The second NTP vehicle uses a low-pressure nuclear thermal reactor (LPNTR). The I_{sp} assumed for the solid core engine is 925 sec with a propulsion system T/W of 3.5. The corresponding values for the LPNTR are 1200 sec and 3.5.

We received a number of submissions that advocated the use of NERVA technology, which was developed in the 60s and 70s. These submissions have been aggregated under submission **#100566**, entitled **Nuclear Rocket Power and Propulsion System for Mars**. The development of this technology and its current status is discussed in some detail in Sec. II under Nuclear Propulsion Technologies.

Another submission that proposed employing NERVA technology is **#100158**, entitled **Clustered Low Thrust Nuclear Thermal Rocket Engines**. The suggested approach is

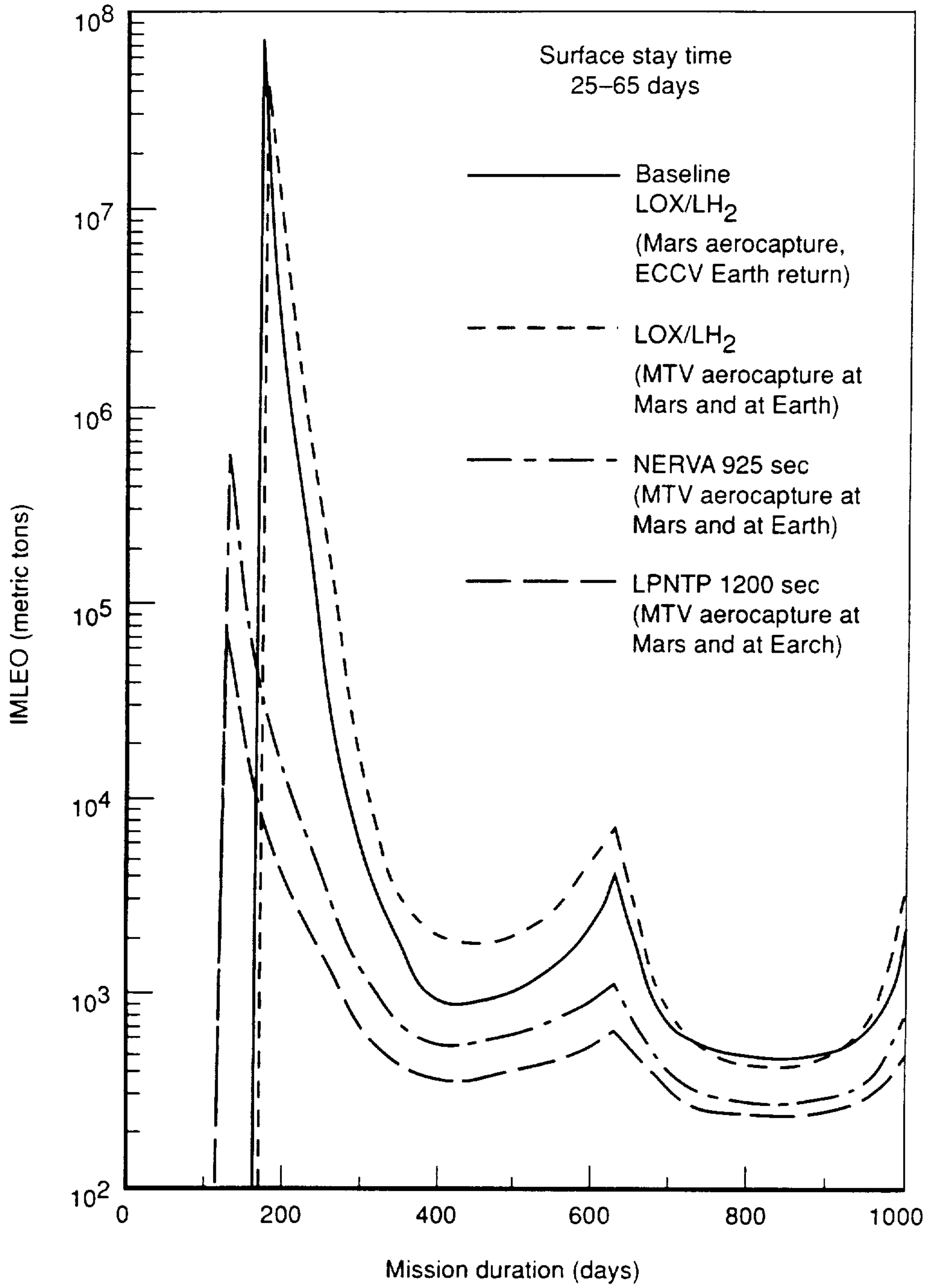


Fig. 3.5—A Comparison of Two NTP Systems with Two LOX/LH_2 Systems

not new, but the author claims that use of clustered, lower thrust NTP units in place of one unit with a higher thrust has not been given much consideration in NASA studies.

A major benefit of using clustered engines is the improvement in crew safety and the probability of mission success in case of engine failure. Other benefits include the development of reactors with thermal power levels lower by a factor of three to ten than the 1500 MW of a 75-klb thrust engine. This could significantly reduce the time to construct ground-test facilities and also reduce the time and cost of testing.

The negative aspect of clustered NTP units is a lower T/W ratio than that of a single, higher thrust unit. Also, with multiple reactors, the propulsion system will be more complex, with additional pumps, propellant lines, etc., but this appears to be a small price to pay for an "engine-out" capability.

By the end of the ROVER/NERVA NTP programs of the 1950s, 1960s, and 1970s, advanced developments in the area of fuel element materials made possible the design of solid core reactors that could operate at higher temperatures than had been achieved to date. Submission **#100933**, entitled **The "Enabler," A Nuclear Thermal Propulsion (NTP) System**, which proposes advanced NTP systems, is discussed in App. M.

The second NTP system of Fig. 3.5 offers an improvement in I_{sp} of about 300 sec relative to NERVA by operating a solid core reactor at low chamber pressures. Again, a number of submissions have proposed this concept and have been aggregated under submission **#100157**, entitled **Low Pressure Nuclear Thermal Rockets (LPNTRs).**

An increase in I_{sp} is achieved in low pressure reactors, without exceeding material temperature limits, by providing operating conditions where the dissociation of the hydrogen propellant occurs. The dissociation process raises propellant energy for a given temperature (i.e., via energy of dissociation). As the propellant expands out of the core into a nozzle, if the dissociated hydrogen recombines, this energy is released to the exhaust jet. Some gain via lower molecular weight is possible if no or only partial recombination occurs. Depending on chamber pressure, I_{sp}s on the order of 1200 sec are anticipated. See Nuclear Propulsion Technologies, Sec. II, for more details concerning low pressure reactor technology.

When the NTP vehicle uses propulsion braking only, the IMLEO advantage of the NERVA vehicle with respect to the baseline LOX/LH_2 system that uses ECCV for Earth return disappears (see Fig. 3.6). However, if the LOX/LH_2 system aerobrakes the MTV into Earth orbit, then, even with propulsion braking, the NTP systems have an IMLEO advantage. Comparing Figs. 3.5 and 3.6, it is obvious that the ability to use aerobraking/aerocapture is a very important factor in reducing IMLEO. Given the uncertainties in the Martian atmosphere, however, aerobraking is a very risky procedure and

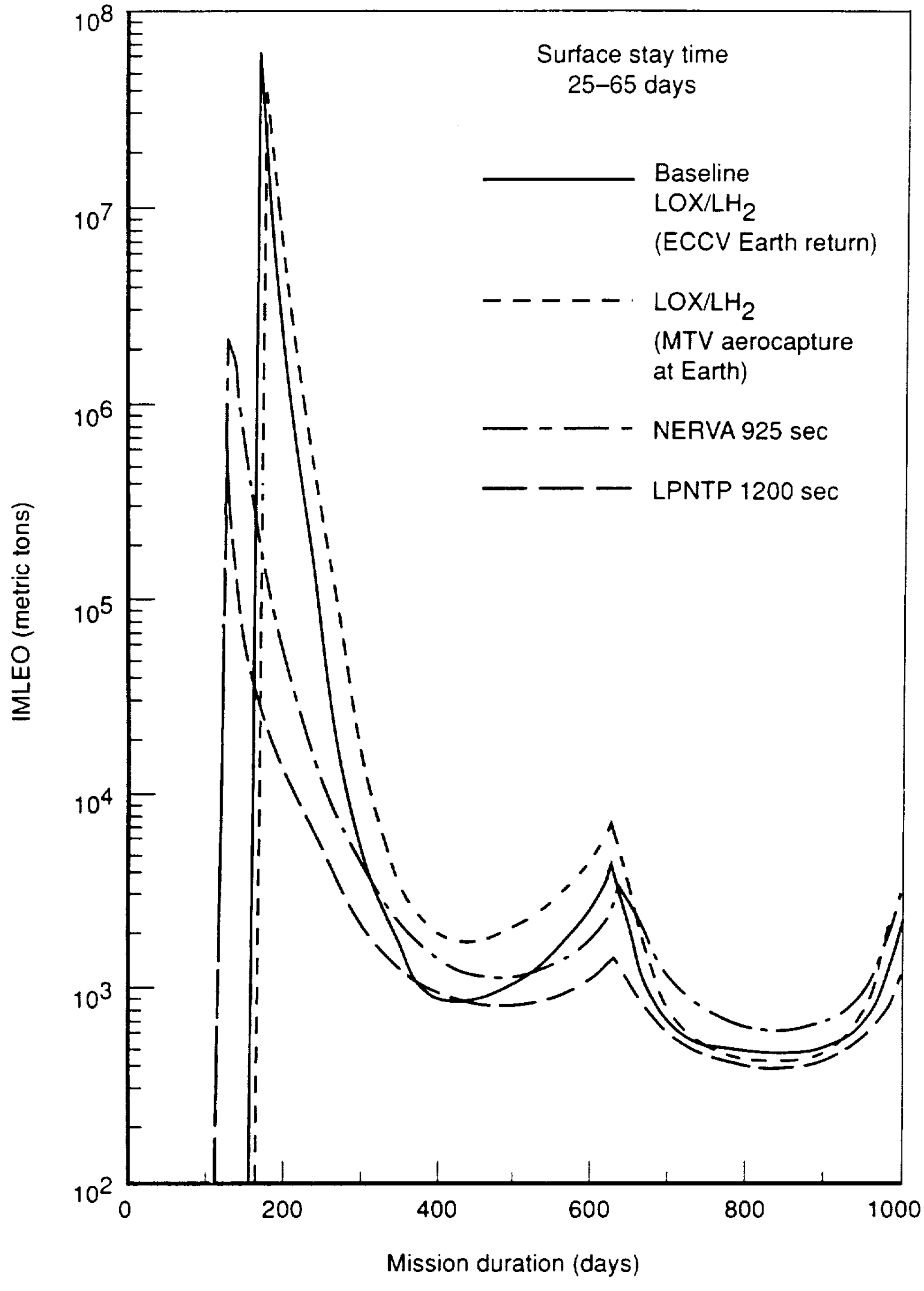


Fig. 3.6—IMLEO Requirements for Two NTP Systems, Propulsion Braking Only

the development of advanced NERVA-type NTPs would eliminate the need for aerobraking (see the LPNTP [low pressure nuclear thermal propulsion] curve, Fig. 3.6).

Another nuclear option that would both avoid aerobraking and reduce the required IMLEO relative to propulsion braking the MTV into HEO is the use of an ECCV to return the crew to earth. After ECCV release, the nuclear propulsion unit would be placed in a heliocentric orbit. This option has been examined only for two mission durations—300 and 400 days. At 300 days, the NERVA MTV has an IMLEO of about 2100 metric tons, while that of the LPNTP MTV is about 960 metric tons. At 400 days, the values of IMLEO are 820 and 510 metric tons, respectively.

With ECCV return, both nuclear options have considerably lower values of IMLEO for a mission duration of 300 days than does the LOX/LH_2 baseline system. At 400 days, NERVA has an IMLEO slightly lower than that of the LOX/LH_2 baseline, while the LPNTP system has an IMLEO 400 metric tons less than that of the baseline.

Figure 3.7 shows how IMLEO varies with mission duration for three advanced nuclear propulsion concepts. The first of these is the nuclear light bulb, which was studied by United Technologies Corporation during the 1960s and 1970s. This is a closed-cycle gas core reactor that uses hydrodynamic forces to stabilize a plasma within a transparent enclosure (see Nuclear Propulsion Technologies, Sec. II). The second concept represents an open-cycle gas core reactor propulsion system that has been studied by Lewis Laboratories and other facilities since the 1960s. There are fundamental questions concerning the feasibility of this concept (see Nuclear Propulsion Technologies, Sec. II). Finally, the third concept is meant to represent the level of performance that might be achieved by fusion propulsion or perhaps a combination of fusion and antimatter propulsion. The I_{sp}s assumed for the three conceptual vehicles are 1800, 5000, and 10,000 sec, respectively. Feasibility, in an engineering sense, has not been demonstrated for any of these concepts.

It should be stressed that only the NERVA-type solid core reactor has been ground tested as a complete propulsion unit. Specific impulses as high as 845 sec have been demonstrated in conjunction with thrusts of 200 klb.

From Fig. 3.7 it can be seen that, even for short mission durations, the IMLEO requirements for these advanced transfer vehicles are quite low compared to those of the LOX/LH_2 baseline. Given the performance potential of these advanced concepts, a vigorous RTD&E nuclear propulsion program is warranted (see Nuclear Propulsion Technologies, Sec. II, for a detailed discussion of a number of advanced nuclear propulsion concepts).

As in the case of chemically propelled transfer vehicles, NTP vehicles would benefit if propellants were available at the destination planet. We received a number of submissions

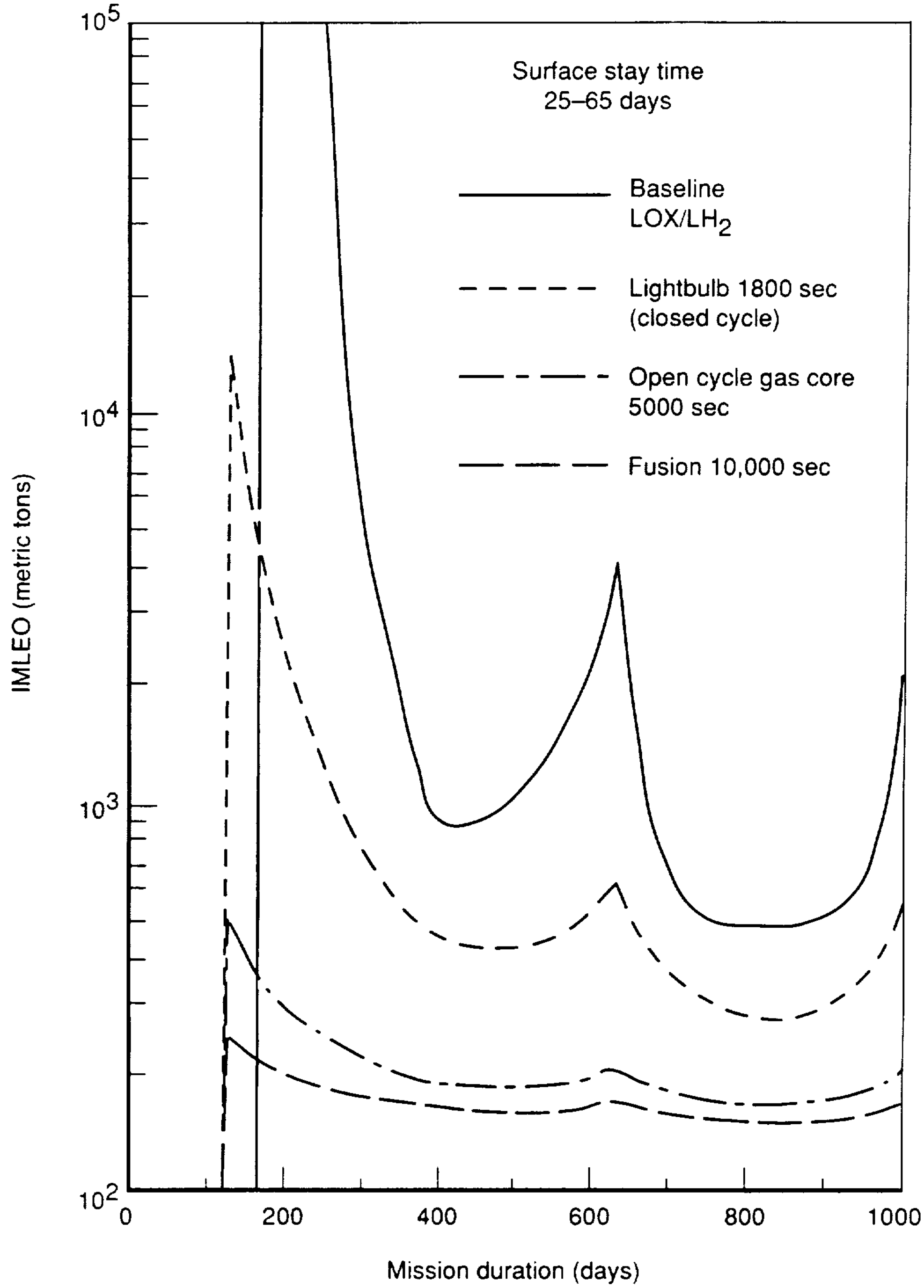


Fig. 3.7—IMLEO Requirements for Three Conceptual Advanced Nuclear Systems

that advocated the NIMF (nuclear rocket using indigenous Martian fuel) concept of R. Zubrin (1990). These have been aggregated under submission **#100103**, entitled **NIMF Concept to Enable Global Mobility on Mars**. This concept is examined in App. N.

Assuming, as before, that water, and thus hydrogen, is available from the Martian system, the IMLEO requirements for nuclear-powered MTVs can be substantially reduced. Table 3.4 presents mass-in-orbit requirements, for both Earth and Mars departures, as a function of mission duration. Both NERVA and low pressure NTP systems, using propulsion braking, are considered.

As in the case of chemical systems, the availability of propellants at the destination planet reduces the IMLEO required. The price that must be paid, of course, is the establishment of propellant production and storage facilities and, possibly, a transportation system for placing propellants into LMO. If propellant production occurs on either Phobos or Deimos, then the MTV could effectively "dock" with the storage facility because of the weak gravity fields of the Martian moons. Thus, the energy expended in "placing" the propellant into LMO is minimal and a separate transportation system is not needed. This would not be the case if propellant production takes place on the Martian surface. One possible approach to transporting propellant from the surface into Martian orbit is discussed in App. N.

Table 3.4

Required Mass in Orbit: Martian In-Situ Propellants (all masses in metric tons)

Mission	Mass in Orbit (NERVA)		Mass in Orbit (LPNTP)	
Duration (days)	Earth	Mars	Earth	Mars
200	—	—	2233	820
250	1705	940	1062	393
300	1315	415	576	230
350	790	305	408	186
400	665	210	368	142
450	645	180	364	128
500	585	205	337	139
550	590	225	337	147
600	790	310	401	179

Comparing the data presented in Table 3.4 with those shown in Fig. 3.3, it is apparent that LOX/LH_2 MTVs, using Martian propellants, have lower IMLEO requirements than do NERVA and LPNTP vehicles for mission durations in excess of about 300 days. Again, the advantage of the LOX/LH_2 system can be eliminated if the nuclear vehicles use ECCVs for Earth return, rather than propulsion braking the MTV into HEO. For a 300-day mission, a NERVA vehicle using Martian propellants and ECCV Earth return has an IMLEO of 500 metric tons as compared to about 620 metric tons for its LOX/LH_2 counterpart (see Fig. 3.3). An LPNTP vehicle has an IMLEO of about 300 metric tons for the same mission duration. At 400 days, the corresponding IMLEOs are NERVA, 300 metric tons; LOX/LH_2, 225 metric tons; and LPNTP, 210 metric tons.

Before in-situ propellants could be produced on Mars, a split mission, similar to the one described for chemical systems, might be used. Again, a low thrust cargo vehicle would be used to transport to Mars all of the mass that is needed for the exploration, plus the propellant required for Earth return. The transport vehicle could use SEP or NEP.

Submission **#101144**, entitled **Nuclear Electric Powered (NEP) Interplanetary Cargo Vehicle**, proposes such a vehicle for a Mars split mission. The submission points out that by using a high I_{sp}, low thrust vehicle to carry a large percentage of the overall system mass to Mars, the manned vehicle can be lighter and faster than when a single vehicle is used. In addition, because of the efficiency of NEPs, a substantial savings in IMLEO is possible. The submission does not provide any technical details concerning the NEP system other than to indicate that a Rankine cycle would be used with liquid potassium as the working fluid.

The major technical issue associated with NEP systems is the development of a lightweight, reliable power conversion system together with a reactor that would yield electric power in the range of 500 to 1000 kW or more. At these low power levels, however, SEP systems have lower specific masses and are less complex than NEP systems.

Figure 3.8 presents IMLEO as a function of mission duration for a split mission in which the manned vehicle uses either a NERVA or a low pressure NTP system. The LOX/LH_2 baseline mission is also shown for reference. The cargo carrier that is used is the same SEP vehicle used for the chemically propelled split mission cases. The nuclear vehicles use only propulsion braking.

Comparing Figs. 3.4 and 3.8 reveals that, with propulsion braking, both NTP vehicles require higher values of IMLEO than do LOX/LH_2 vehicles for mission durations of 450 days or longer. However, the NTP vehicles, using propulsion to brake into orbit, retain the MTV and nuclear propulsion system for reuse. As in the case of the nuclear thermal systems that fly single vehicle missions (see Fig. 3.6), the use of ECCVs for Earth return, as opposed to propulsion braking the vehicle into HEO, can significantly reduce the IMLEO of split mission vehicles. For a mission duration of 400 days, the IMLEO of the NERVA system of Fig. 3.8 would be reduced from 900 to 550 metric tons. The corresponding reduction for the LPNTP system would be from 570 to 500 metric tons. The improvement in IMLEO is even more

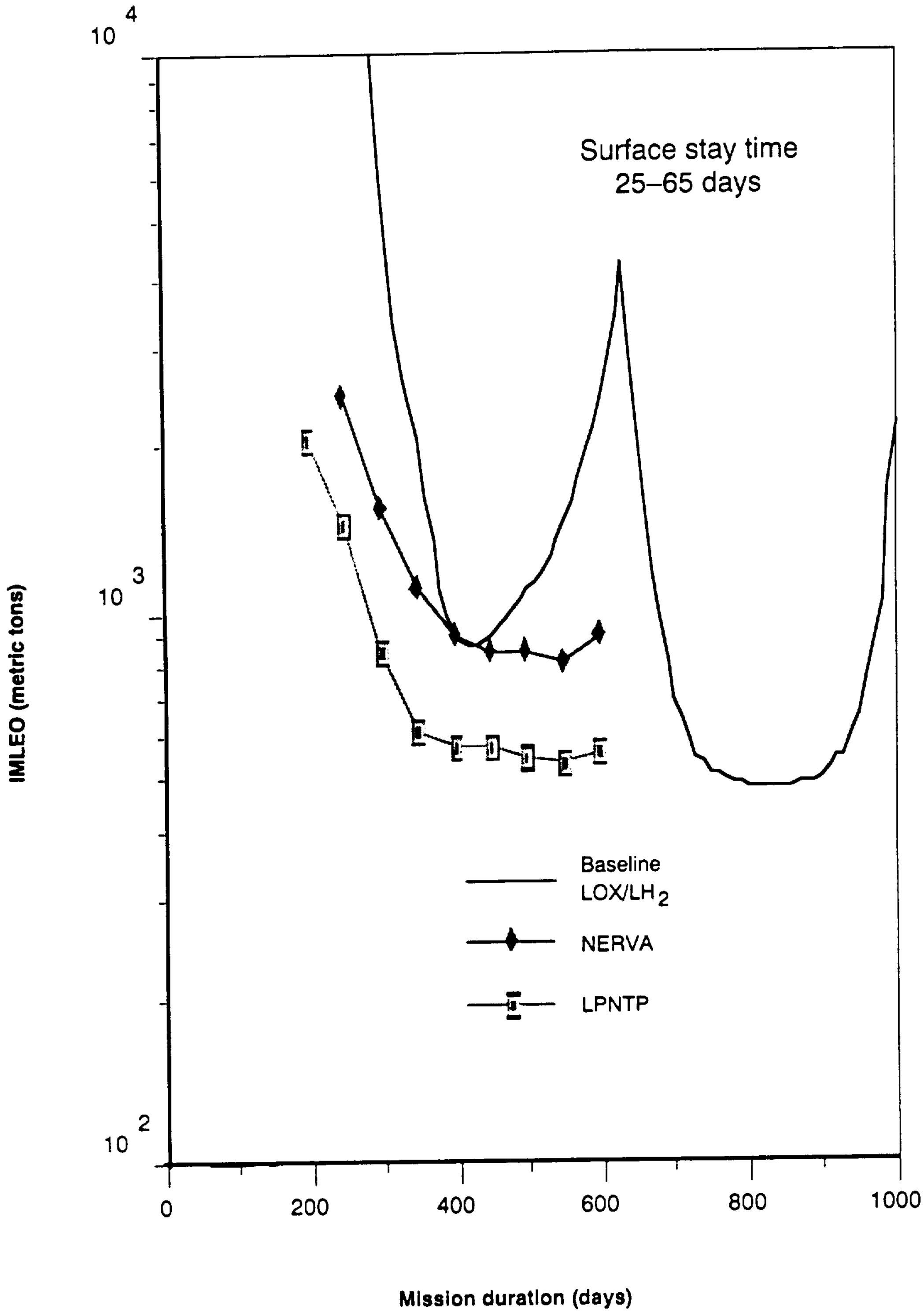


Fig. 3.8—IMLEO Requirements for Two NTP Systems—Split Mission

impressive for a mission duration of 300 days, where the reductions for NERVA and LPNTP systems are 1600 to 835 metric tons and 850 to 545 metric tons, respectively.

Although recovering the NTP vehicle in HEO for reuse might appear attractive, there are a number of factors that make recovery of questionable value. First, the reactor will be highly radioactive, which will impose severe shielding requirements to protect personnel during on-orbit maintenance and propellant loading. It should be kept in mind that all of these operations must take place in a nuclear "safe" orbit (~1000 km). Second, the number of round-trip missions that can be performed by a nuclear thermal vehicle will probably be less than five. Thus, it is questionable that recovery will permit a net reduction in IMLEO as compared to five individual missions using ECCVs, particularly for missions with short durations. Finally, as the reactor becomes more radioactive, there is the question of crew safety in carrying out operations in Mars orbit, particularly for split mission vehicles or vehicles using in-situ propellants.

A final question is: If large masses are required for GCR shielding, how would the use of nuclear propulsion systems affect IMLEO requirements? Table 3.5 presents IMLEO requirements for NERVA, LPNTP systems, and the three advanced concepts considered previously (see Fig. 3.7), assuming shielding masses of 10, 100, and 1000 metric tons. On Earth return, the shielding mass is discarded prior to propulsion braking. A minimum energy trajectory (Hohmann) is assumed. It is evident from Table 3.5 that the three advanced propulsion concepts offer substantial savings in IMLEO compared to chemical systems (Table 3.1) or to NERVA and low pressure NTP systems.

Table 3.5

Galactic Cosmic Ray Shielding (nuclear propulsion)
(all masses in metric tons)

	IMLEO				
Shield Mass	NERVA	LPNTP	Light Bulb	Gas Core	Fusion
0	405	300	227	150	138
10	435	323	244	162	149
100	701	529	399	271	249
1000	3360	2588	1954	1365	1240

Unfortunately, at this time, the feasibility of various advanced concepts—gas core reactors, fusion reactors, antimatter propulsion—that can theoretically provide the I_{sp}s and T/Ws assumed for Table 3.2 has not been demonstrated. But, as has been stated before in this section, the performance potential for SEI missions is so great that an early research effort should be mounted to identify those concepts that appear to be most promising for eventual development. Thus, Table 3.5 should be regarded as a "what if" table, which assumes the five propulsion concepts are realized; we do not imply that these five concepts are equally likely to be realized.

OBSERVATIONS

This subsection presents some observations that have been drawn with regard to the IMLEO requirement of the various space transportation/propulsion options examined in this section. We have selected two mission durations to compare the relative merits of these options.

As indicated earlier, the NASA 90-Day Study chose a mission duration of 565 days, including 30 days on the surface of Mars. From a human support perspective, a mission of this duration might be undesirable. Therefore, mission durations of 300 and 400 days were selected as possible alternatives. A 400-day mission was chosen because it appears feasible using current LOX/LH_2 technology. A 300-day mission was chosen to identify the most promising transportation/propulsion options that could be pursued if future research indicates it is unadvisable for humans to spend as much as 400 days in space.

Table 3.6 presents the IMLEO requirements for various space transportation/ propulsion options for these two mission durations. The data presented have been taken from Figs. 3.1 through 3.8. The performance shown, along with the supporting technology, becomes more speculative as one reads down the table. It should be kept in mind that only LOX/LH_2 -powered vehicles have flown in space.

It can be seen from Table 3.6 that the various space transportation/propulsion options examined in this section fall into four broad categories: single vehicle missions, split missions, missions employing in-situ propellants, and single vehicle missions that employ advanced but unproved propulsion concepts. It is almost certain that the first Mars missions will use one of the options in the first category. If policy permits the use of nuclear reactors in space, the prime contenders for 400-day missions are the LOX/LH_2 baseline system and the LPNTP system. For the LOX/LH_2 system to achieve an IMLEO of 900 metric tons, it is necessary that lightweight aerobrakes be developed so that the amount of propulsion necessary to go into orbit about Mars is minimized. For the LPNTP system, the primary technology driver is the development of low-pressure nuclear thermal reactors with adequate performance and operating lifetime. In both cases, the development of lightweight tanks that can store liquid hydrogen for extended periods of time without excessive loss is required.

The most attractive options in the first category, in terms of relatively low values of IMLEO, are NERVA and LPNTP systems using aerocapture to go into Mars and Earth orbits. However, it is doubtful that aerobraking a radioactive nuclear reactor on Earth return would be acceptable from a policy perspective. As discussed earlier in this section, the use of propulsion braking in conjunction with ECCVs for Earth return yields IMLEOs that that fall between the pure propulsion braking and aerocapture cases shown in the first category of Table 3.6. (After ECCV release, the nuclear reactor would be placed in a heliocentric orbit.) With ECCV Earth return, the propulsion unit and the MTV would not be available for reuse, but the reductions in IMLEO relative to nuclear options of NERVA with propulsion braking and LPNTP with propulsion braking would be substantial. The IMLEO for the nuclear options with ECCV Earth return appear in the parenthetical entries in Table 3.6. Thus, for early missions to Mars, the development of LPNTP transfer vehicles appears desirable in that it would provide the capability to perform either 300- or 400-day missions.

Finally, for 400-day missions, advanced chemical propellants appear attractive, but they would not be suitable for 300-day missions. In view of past difficulties in attempts to develop tripropellants and other high-energy chemical propellants, they should only be considered if policy considerations preclude nuclear systems in space.

The next two categories of Table 3.6, split missions and the use of in-situ propellants, would most likely provide the space transportation options that would support the development of a Mars base following the initial exploratory missions. In the case of split missions, the major technology driver is the development of electric propulsion systems that would propel efficient (in terms of low IMLEO per ton of payload) unmanned cargo carriers from the Earth to orbits about Mars. With the payload needed for Mars exploration carried by the cargo vehicle, the manned vehicle can, for a given initial mass, achieve shorter transit time than the transfer vehicles in the first category listed in Table 3.6. (The values of IMLEO listed for split missions include both the cargo carriers and the manned vehicle.)

Before in-situ propellants can be used, the infrastructure required for the production, storage, and distribution of those propellants must be established. Thus, comparing in-situ transportation options with those in the other categories of Table 3.6 on the basis of IMLEO is invalid because of the cost associated with the establishment and maintenance of the required infrastructure.

Table 3.6 indicates that for split missions, the two nuclear thermal options using ECCV Earth return are most promising in terms of the IMLEO requirements. Without a nuclear option, chemical systems with advanced propellants have an IMLEO advantage over

Table 3.6

IMLEO Requirements for Various Space Transportation/Propulsion Options (in metric tons)

Transfer Vehicle/Propulsion Option	Mission Duration in Days 400	300
SINGLE VEHICLE MISSIONS		
Chemical systems:		
Baseline LOX/LH_2 (ECCV return)	900	6400
Advanced propellants (ECCV return)	600	2700
Nuclear thermal systems:		
NERVA (propulsion braking)	1800 (820)[a]	5000 (2125)
LPNTP (propulsion braking)	959 (515)	2350 (960)
NERVA (aerocapture in LEO)	550	1550
LPNTP (aerocapture in LEO)	400	750
SPLIT MISSIONS		
Chemical systems:		
LOX/LH_2 (ECCV return)	660	1650
Advanced propellants (ECCV return)	450	1200
Nuclear thermal systems:		
NERVA (propulsion braking)	900 (550)	1600 (835)
LPNTP (propulsion braking)	570 (500)	850 (545)
MISSIONS EMPLOYING IN-SITU PROPELLANTS		
Chemical systems:		
LOX/LH_2 (ECCV return)	225	910
Nuclear thermal systems:		
NERVA (propulsion braking)	665 (300)	1315 (500)
LPNTP (propulsion braking)	368 (210)	576 (300)
SINGLE VEHICLE MISSIONS		
Advanced nuclear concepts:		
Closed-cycle gas core (propulsion braking)	470	800
Open-cycle gas core (propulsion braking)	200	220
Fusion/antimatter (propulsion braking)	160	175

[a] The values of IMLEO in brackets are nuclear thermal systems that use propulsion braking plus ECCVs for Earth return.

LOX/LH_2 systems. For the category using in-situ propellants, LPNTP systems are most attractive in terms of IMLEO with or without the use of ECCVs for Earth return.

The last category in Table 3.6, missions employing advanced nuclear concepts, indicates that if a nuclear propulsion system with an I_{sp} of about 1800 sec becomes available, then a truly reusable space vehicle could be built with IMLEO values of 470 or 800 metric tons. (It should be noted that these values of IMLEO are not that much smaller than those for LPNTP systems using ECCV Earth return.) All of the options of the last category are completely speculative insofar as being feasible in an engineering sense and should not be considered in the same light as the nuclear thermal systems in the first three categories.

Finally, the IMLEO results presented in Table 3.6 and throughout this section are not necessarily optimum (i.e., minimum) for a given mission duration. Furthermore, the RAND model used to calculate IMLEO values is relatively simple (see Apps. C and D). The results should be good enough to support the relative comparisons discussed, although the values of IMLEO listed are undoubtedly approximate.

IV. EARTH-TO-ORBIT LAUNCH SYSTEM OPTIONS

The success and, indeed, even the political/economic feasibility of any future space endeavor such as SEI may be determined by the availability of low-cost, reliable launch systems for placing large payloads into LEO. This section briefly discusses available launch system options and some of the tradeoffs involved in the context of submissions received.

BACKGROUND

The NASA 90-Day Study indicated IMLEO requirements of 110 to 200 metric tons for Lunar trips and 550 to 850 metric tons for Mars trips based on LOX/H_2 propulsion. A high percentage of this mass consists of propellant—roughly 75 percent for Lunar missions and about 85 percent for Mars missions.

If the long-term effects of GCR dictate either massive shielding or very short mission duration times, then, for LOX/LH_2 systems, large values of IMLEO will be required. From Fig. 3.1 it can be seen that for mission durations of 200 to 300 days with no GCR shielding, IMLEO requirements are extremely high. Even for split mission transportation systems using LOX/LH_2 for the manned vehicle, total IMLEO requirements are very high (see Fig. 3.4). On the other hand, if vehicle shielding is used, even minimum energy trajectories to and from Mars can still lead to large values of IMLEO (see Table 3.2). Estimates of the mass required are very uncertain, but even without GCR shielding, IMLEO requirements for each Mars mission, using LOX/LH_2 propulsion, should range from 500 to 1000 metric tons.

The overall observation to be made is that IMLEO requirements for Mars missions are apt to remain high, somewhat irrespective of overall trip time. For long but propulsively efficient missions, the IMLEO required due to delta V may be low, but GCR shielding provisions will be high. Alternatively, very short-duration missions that require little shielding have high propulsive delta Vs and thus high IMLEO requirements.

As mentioned above, for the Mars mission, most of the required IMLEO consists of large quantities of propellant, along with vehicle tankage, structure, and systems mass. Only a small percentage of IMLEO involves personnel or small critical payloads (where reliability may take precedence over launch cost); thus, two distinctly different types of launch vehicles are suggested. The personnel and critical payloads would be carried aboard the Shuttle, personnel launch system (PLS), advanced manned launch system (AMLS), or, eventually, a NASP derivation.

There is a wide spectrum of possible approaches for launching the large mass associated with the MTV. One way of characterizing this spectrum of options is in terms of launch systems payload capability and the corresponding number and frequency of launches required to transport a fixed quantity of payload to LEO. Obviously, important tradeoffs require consideration as payload capability increases and the number/frequency of required launches decreases.

With the relatively small payload/high frequency approach, many of our current launchers and launch sites could be used or modified to save the major development costs associated with an all-new, heavy-lift launch system. But other implications (and costs) ensue:

- Opportunities for substantial reductions in launch cost (dollars per pound in orbit) may be limited.
- Intensive launch schedules may be difficult to maintain for long periods.
- Extensive on-orbit assembly of large structures with considerable amounts of EVA will be required.
- Means for in-space collection, storage, and utilization of many small packages of propellant, supplies, etc., will have to be established and maintained.
- Additional launches will be required to provide the support/supplies during the extended program time that will be needed to conduct extensive on-orbit assembly operations.
- Opportunities for ground assembly, test, and checkout of large components and structures will be limited and, in some cases, eliminated.

Alternatively, opting for a very large payload capability/low number of launches approach has a different mix of advantages and disadvantages. Development of a new heavy-lift launcher (payload capability in the range of 300K to 1M lb) can possibly facilitate

- Opportunity for reduction in cost to orbit through economies of scale, minimization of number of launches, and technological approaches that permit high margins of safety in design and operation.
- Minimization or elimination of extensive on-orbit assembly and thus extravehicular activity (EVA), as well as the need to develop such capabilities.
- Ground assembly, test, and checkout of large components and structures of the MTV.

Conversely:

- A large up-front development cost and long lead time will be required.
- New launch sites and infrastructure will be necessary.

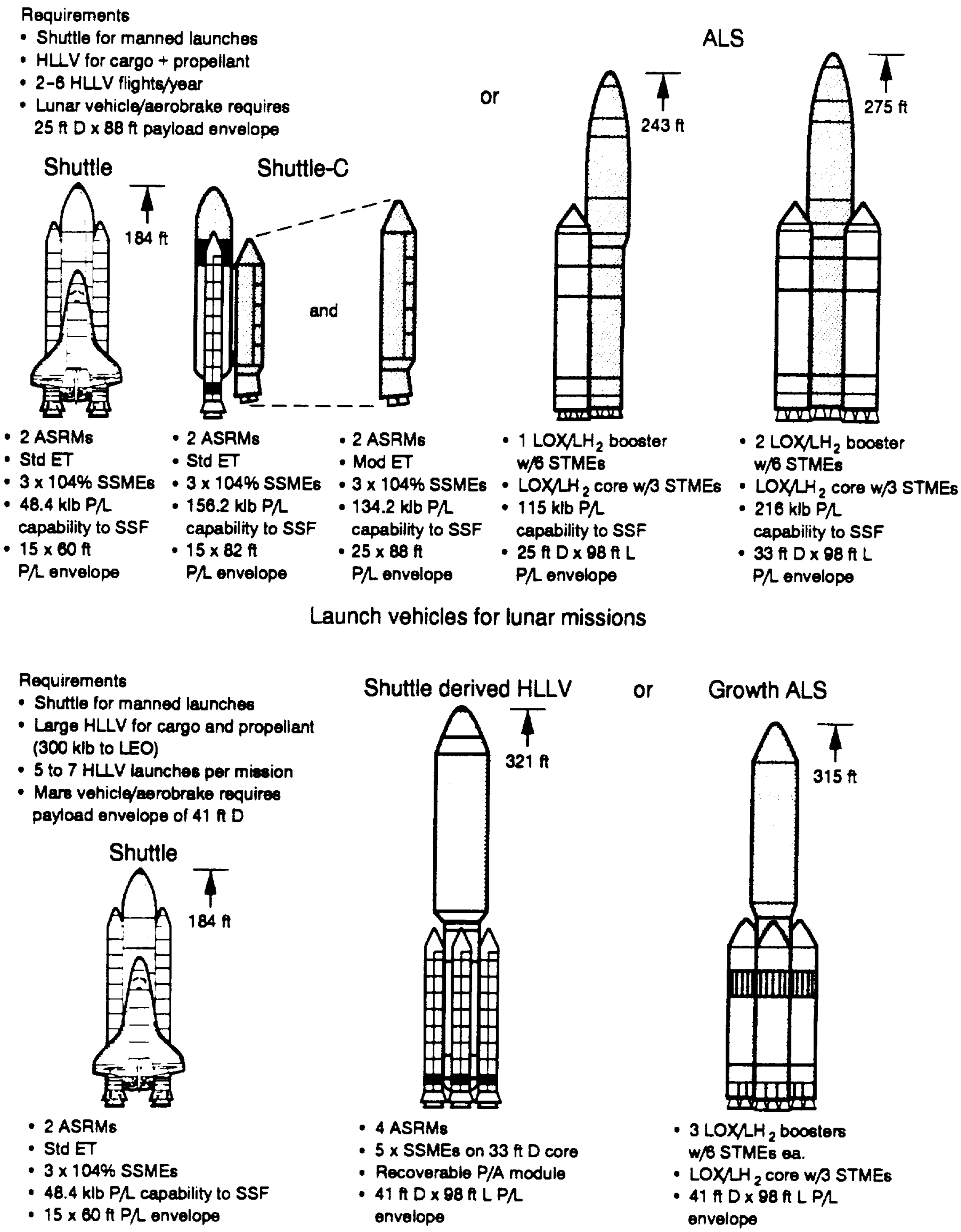
- There is concern regarding the relative program effect of losing a very large single payload versus a few small ones to accidents.
- Launch costs (on a dollar-per-pound basis) to orbit can rise for partially loaded payload launches.
- The overall utility and potential cost benefits of such vehicles are heavily dependent on a long-term, robust mission model.

The general considerations addressed above, along with the question of balance among the cargo launch vehicle payload sizing tradeoffs, were addressed in the NASA 90-Day Study in the following way. The existing Space Shuttle and expendable launch vehicle fleet (Delta II, Atlas II, and Titan IV families) would be used to support all currently planned robotic missions, with the Shuttle continuing throughout the SEI program as the ETO transport means for personnel and limited cargo. Due to its potential for early availability, the Shuttle-C was suggested to provide the primary cargo-lift support for the Lunar mission to at least the year 2000. At that time or later, the primary cargo-carrying function might be transitioned to a heavier-lift vehicle such as an ALS configuration. In the Mars mission timeframe, it was assumed that a heavy-lift cargo vehicle of about 300 klb payload capacity could be made available from the families of Shuttle-derived or ALS concepts currently under evaluation (as discussed in Sec. II). A pictorial layout of these candidates is shown in Fig. 4.1.

ETO APPROACHES SIMILAR TO NASA'S

A significant number of submissions that dealt with ETO launch vehicles discussed ideas or made proposals that were very close to those covered in the 90-Day Study or known to be under active consideration by NASA. Others deviated in varying degrees but generally fell in the categories of options previously discussed in Sec. II.

Submission **#101150**, entitled **Low Cost Launch Vehicle for Fluid Transport**, recognizes the potential benefits of separate, unmanned cargo-launch vehicles, especially for propellants such as LH_2. The author stresses the need for a low-cost design and proposes an expendable "big dumb booster" approach that might sacrifice some reliability to ensure meeting cost goals. A 100 klb payload class is mentioned. Since no backup material is provided, it is unclear what was envisioned by the term *big dumb booster*. The reference may have been to a large body of work, so labeled, that began in the 1960s and was recently reviewed in an Office of Technology Assessment (OTA) workshop on 1 December 1987. The workshop debated the pros and cons of such an approach and outlined various options available to Congress should they decide further R&D is warranted.



SOURCE: Hueter (1990).

Fig. 4.1—Launch Vehicles for Lunar and Mars Missions

Submission **#100638**, entitled **Low Cost Earth-to-Orbit Launch System**, reiterates the rationale for separate manned and unmanned launch systems, emphasizing a simple, designing-to-low-cost approach for the cargo vehicle(s) (payload size is not given). Most of the design and operational aspects discussed are being explored in the ongoing ALS/Advanced Launch Development Program (ALDP) activities. The author further argues for construction of a new launch site near the equator in order to realize performance gains in added orbit plane velocity due to the Earth's rotation. These benefits can be significant but would need to be weighed against additional logistics problems and costs. The net gains are unclear, but the issue may warrant serious study.

ETO APPROACHES DIFFERING FROM NASA'S

The following submissions markedly depart from current NASA planning in basic approach and/or payload capacity.

Submission **#100192**, entitled **Saturn V Heavy-Lifting Launch Vehicle Concept**, and submission **#100185**, entitled **A Fall-Back-to-Spring-Forward Strategy to a Heavy-Lift Launch Vehicle: Reviving Saturn V Technology**, both argue for development of an updated version of the Apollo Saturn V launch vehicle as an effective means (considering time, cost, and proven performance/reliability) of providing a 250- to 300-klb cargo lift capability for SEI. These two submissions are discussed jointly in App. O.

Another submission, **#101166**, entitled **Advanced Heavy-Lift Launch Vehicles (HLLVs)**, advocates development of a HLLV with payload capability up to 660 klb to LEO. If propellant transportation costs to LEO could be reduced by a factor of two to ten, ostensibly other systems could be made more economical at the expense of using more propellant. While specific design details are omitted (and no backup material provided), the author feels that the technologies and design philosophies being developed in ALS/ALDP could facilitate the proposed development.

An ultra-HLLV concept with payload capability to LEO in the 1 million to 1.5 million pound class is proposed in submission **#100110**, entitled **Ultra Large Launch Vehicle (ULLV) for Moon and Mars Missions**. The concept, described in App. P, promises transportation costs on the order of $100/lb, utilizing design characteristics largely derived from past NOVA/SEA DRAGON studies.

Submission **#100662**, entitled **Needed, a Site to Launch Nuclear OTVs and Other Potentially Unsafe Equipment**, argues for a near-equatorial launch site that would permit launch of otherwise environmentally undesirable and/or hazardous vehicles and payloads, reduce launch delays due to bad weather, and take advantage of the Earth rotation velocity increment and 15-mi-high equatorial bulge. The author's preference is Christmas Island, followed by Kwajalein or Eniwetok atolls. As earlier mentioned, careful analyses are required to adequately assess the countereffect of logistics and overhead costs of operating and maintaining a remote launch facility. Of course, for very hazardous payloads or operations, there may not be much choice if such systems are operationally required.

Potential launch systems with payload capabilities in the 10,000 to 20,000 kg class include current rocket launchers such as Titan IV and future vehicles such as the NASP. The limitations of current rocket launchers with regard to the cost of placing payloads into orbit have already been discussed, so let us now examine the use of NASP-derived vehicles (NDVs) for launching SEI payloads into orbit.

The primary goal of the NASP program is to develop an air-breathing vehicle that can fly into LEO, using a minimum of rocket propulsion for final insertion. The vehicle is intended to be operated like an airplane, with simple payload integration procedures and short turnaround times. The net result would be a launch system with a high launch rate and a low cost per pound to orbit.

Currently, it is not certain that these objectives can be met. There are still technical hurdles to overcome (see discussion of NASP/NASP-type vehicles, Sec. II). Even if single stage to orbit with a reusable payload is demonstrated, the ability to operate NDVs like current aircraft may not be possible, given the complexity of the vehicle, the severity of its flight environment, and the use of liquid or slush hydrogen as fuel.

As discussed previously, a major portion of mass delivered to LEO for SEI missions is propellant. Because of aerodynamic considerations, NDVs are not well suited for transporting bulky, low-density payloads. They certainly could not transport loaded propellant tanks to orbit like the Shuttle-C is designed to do for the Lunar missions. With an NDV propellant transport, it would be necessary to perform on-orbit transfer of liquid oxygen and hydrogen to orbiting storage tanks. These, in turn, would later supply Lunar or Mars spacecraft with the required propellants. In any case, a substantial on-orbit infrastructure would be required for propellant transfer and long-term storage.

If NDVs were used to transport spacecraft structural components to LEO for on-orbit assembly, the time required for vehicle construction and the subsequent amount of EVA required would both be substantially greater than if large, preassembled spacecraft structures are launched by heavy-lift rockets. Again, the greater the amount of on-orbit assembly required, the greater the amount of on-orbit infrastructure required.

The logical role for NDVs in supporting SEI missions would be the transport of personnel and priority cargo. With the increase in on-orbit activity associated with Lunar or

Mars missions, the operational flexibility of NDVs or similar aerodynamic vehicles, as compared to Shuttle-type vehicles, makes them prime contenders for this role.

Finally, at the low end of the launch system spectrum are devices such as EMLs and light gas guns (see Sec. II, Earth-to-Orbit Launch Systems). Both of these devices accelerate payloads through a tubular or barrel-like structure. By the time the payload exits the launcher, it has achieved a substantial fraction of the velocity needed to orbit the Earth. For a constant launch acceleration of 1000 g's, the tube or barrel must be approximately 2 km long.

Both EMLs and light gas guns are limited to very small payloads, a few thousand kilograms at most. Because of the long launch tube, an installation must be fixed and could only launch in one direction, thus limiting access to one orbital plane. Finally, launchers of this type have a unique range-safety problem. For ballistic placement into LEO, the flyout trajectory must be relatively flat—20 to 30 deg above horizontal at launch. If a malfunction occurs so that the payload does not achieve sufficient velocity to go into orbit, the payload will impact somewhere downrange, conceivably at intercontinental distances. Even if the launch vehicle is fragmented, there still will be some area downrange from the launcher that is at risk because of the fragments.

As indicated in Sec. III, we received a number of proposals that advocated the use of EMLs or a light gas gun to put payloads consisting of either propellants or water into LEO. In the latter case, the water would be collected at an on-orbit facility that would, by electrolysis, produce hydrogen and oxygen, which would then be liquified and stored for future use as propellants.

All of the points that were raised with regard to NDVs being unsuited as propellant tankers also hold for EMLs and light gas guns. Indeed, because of the small size of the payloads relative to those of NDVs, the problems associated with the on-orbit collection and transfer of packages to an orbiting storage facility would be exacerbated.

In the case of the on-orbit electrolysis of water, the facility would very likely use solar energy, and thus low-inclination orbits would be preferred (plus or minus 23.4 deg). For such orbits, the EML/light gas gun installation should be located outside the continental United States to minimize orbital plane changes. Because of Earth rotation, there is a brief period every 24 hours during which payloads can be launched into an orbit that is coplanar with that of the facility. Operationally, EMLs/light gas gun systems are very limited in both payload mass and, because of launch g's, payload types. Although suitable for certain types of small scientific payloads, they appear to have very limited utility for SEI ETO launch applications.

V. CONCLUSIONS AND RECOMMENDATIONS

A large number of potential space transportation options could support a Mars exploration mission similar to that of the 90-Day Study baseline. In general, from the viewpoint of reducing IMLEO, spacecraft trajectories that make use of Venus for gravity assist—either on the way to Mars or on the way back from Mars—are desirable. The disadvantages of such an approach are the restrictions on launch opportunities, the increased hazard to the crew from solar flares, and the increase in travel time as compared to direct flights to Mars.

Almost all of the space transportation options examined in Sec. III could benefit from the availability of orbital transfer systems that can economically transfer large masses from LEO to high Earth orbit (HEO) or cis-Lunar space. This suggests that the development of electric propulsion systems would be highly desirable to minimize the orbital support costs of such a transfer system (the required propellant mass in orbit).

All of the space transportation options considered could benefit greatly from the development of propellant sources on the Moon or in the Martian system, or both. It would then be necessary to develop not only the propellant manufacturing and storage facilities, but also a transportation system to transfer the propellant directly to the spacecraft or to a storage facility located at a selected transportation node. A careful examination should be made of transportation nodes other than those in LEO and LMO. In particular, if propellants are available from the Moon and the prerequisite transportation system is in place, the Earth-Moon L_2 point offers a number of advantages as a departure point for Mars.

Finally, a major concern in designing a manned vehicle for the Mars mission is protecting the crew from the harmful effects of GCR. Currently, only three alternatives exist for reducing crew exposure to GCR. All three alternatives—mass shielding, magnetic shielding, and short trip times—require very large values of IMLEO. Unfortunately, the mass required for shielding is very uncertain. Figure 3.2 indicates that with chemical propellants, a shielding mass greater than about 10 metric tons is probably unacceptable. Reducing exposure to GCR by reducing trip time also can lead to very large values of IMLEO for chemical systems. For a trip time of 200 days, the value of IMLEO can vary from a low of about 7000 metric tons to a high of over 200,000 metric tons, depending on the type of mission and propellants assumed. If either short trip times (150 to 200 days) or large shielding masses (100 to 1000 metric tons) are required, then only high-I_{sp}, high-thrust nuclear systems appear to be viable candidates for the Mars mission.

NONNUCLEAR SPACE TRANSPORTATION OPTIONS AND TECHNOLOGIES

The nonnuclear space transportation options and technologies we recommend for further study are summarized below.

(1) If, for the Mars mission, it is decided to pursue the use of LOX/LH_2 propulsion systems as the baseline, then we recommend that space-storable propellants be considered for all vehicle propulsion requirements except for TMI, where LOX/LH_2 would be used (see Figure 3.1). By eliminating LOX/LH_2 propellants, the penalty that must be paid for heavy, insulated tanks or refrigeration systems and the waste due to boiloff can be avoided, along with potential problems due to leaks. In addition, space storables can have a much higher density than liquid hydrogen, leading to a more compact and lighter design, including a smaller aerobrake. Based on the above considerations, we recommend the submission entitled **Lunar/Mars Return Propulsion System (#111767)** for further consideration.

(2) To reduce crew exposure to the hazards of the space environment, it would be desirable to shorten flight times from Earth to Mars and return. Thus, the development of chemical propellants and propulsion systems that can perform better than current LOX/LH_2 systems should be a high priority research item, under the assumption, of course, that nuclear propulsion is not admissible. Tripropellants, such as BE-O_2-H_2 and Li-O_2-H_2, have the potential for delivering I_{sp}s in the range of 550 to 650 sec. Experimental work in the past with these propellants has produced disappointing results. In a nonnuclear world, however, the potential of tripropellants should not be overlooked (see Fig. 3.2). We recommend the submission entitled **High-Energy Chemical Propulsion for Space Transfer (#101212)** for further consideration.

(3) Another approach to reducing manned vehicle trip time to and from Mars is to use a split mission in which cargo spacecraft, following low-energy trajectories, pre-position the mass needed for Mars exploration and Earth return in Mars orbit. A small manned vehicle would follow high-energy trajectories to and from Mars. Because the cargo vehicles can use a propellant-efficient, low-energy trajectory, there is the potential for reducing total IMLEO requirements as compared to the baseline LOX/LH_2 approach.

Substantial additional mass savings can be realized by using solar electric cargo vehicles (see Fig 3.4). Such an approach is proposed in the submission entitled **The Pony Express to Mars (#100714)**. We recommend this submission for further consideration even though the mission example that is provided has much less payload capability than that of the 90-Day Study baseline, a payload capacity that has been adopted for the transportation options discussed in this Note.

(4) An established base on the Moon offers the potential for producing large quantities of oxygen, aluminum, and magnesium from the Lunar regolith. Liquid oxygen, transported to Lunar orbit, could be used to support Mars missions. The use of aluminum and oxygen as rocket propellant would provide a Lunar rocket transportation system with an excellent performance capability.

If hydrogen can be found on the Moon in usable quantities, a tripropellant, Al-O_2-H_2, could be formulated that would have an I_{sp} of about 475 sec with 30 percent of the fuel hydrogen by weight. The development of a tripropellant engine that could operate on materials wholly available from the Lunar surface is proposed in the submission entitled **Lunar-Derived Propellants (#100932).**

We recommend that the use of the Earth-Moon L_2 point for a Mars departure point be examined for the case where the MTV uses Lunar-derived propellants exclusively. Although Lunar-derived propellants do not offer I_{sp}s as high as those of Be-loaded tripropellants, the fact that they can be delivered to L_2 from the Moon rather than the Earth would provide a substantial reduction in IMLEO requirements.

(5) As with Lunar-derived propellants, the utilization of materials available from the Martian system could substantially reduce IMLEO requirements. The Martian atmosphere is nearly 96 percent carbon dioxide. It is also likely that water exists in the Martian system (polar ice caps on Mars or ice on the moons). The use of Martian in-situ propellants is proposed in the submission entitled **In-Situ Propellants for Mars Lander—Chemical Engines (#101178)**. We recommend that an early effort be undertaken to determine whether or not water is available in usable quantities in the Martian system, particularly on either Deimos or Phobos.

(6) If transportation options are developed that make use of Lunar-derived propellants, then an orbital transfer system will be required to transfer propellants from LLO to orbital storage facilities located at a transportation node. A prime candidate for this mission is a vehicle using SEP. The submission entitled **Solar Electric Orbital Transfer Vehicle (SEOTV) (#101157)** proposes such a vehicle. The innovative feature of this submission is the use of inflatable structures to achieve a large, lightweight, low-cost solar array that uses amorphous silicon cells on a Kapton film. Although amorphous silicon cells are currently only 6 percent efficient, the power density of such an array is five to ten times that of conventional photovoltaics. We recommend that the concept receive further consideration.

(7) OTVs using electric propulsion systems will almost certainly be required to support any long-term SEI program. Of the various types of electric thrusters that are

suitable for OTV operations in cis-Lunar space or for Mars missions, ion thrusters are currently the most advanced. The submission entitled **Pulsed MPD Electric Propulsion (#100170)** proposes a magnetoplasma dynamic electric thruster concept that is projected to achieve an efficiency of about 60 percent with an I_{sp} of 5000 sec. This performance is comparable to that of ion thrusters, and the combined specific mass of the MPD thruster and power conditioning unit is considerably lower than that of ion thrusters. A major advantage of MPD thrusters, as compared to ion thrusters, is that they can use almost any kind of material for propellant. We recommend that an experimental program be initiated with the goal of demonstrating both high efficiency and low electrode erosion rates for this MPD concept.

(8) SEI missions will require large amounts of mass to be transferred from LEO to higher orbits. An approach for performing this mission is proposed in the submission **Earth-Based Microwave Power Beaming to Interorbital (LEO to and from HEO) Electrically Propelled Transport Vehicles (#101536)**. The major advantage of this concept is that the massive power-generating infrastructure is kept on the Earth's surface and this infrastructure could support many OTVs over its lifetime. A variation of this concept would be a power beaming station on the Moon for transportation from LLO to L_2.

For LEO-to-HEO operations, the optimal configuration would have four Earth stations to provide continuous power to the OTV. Beamed-energy OTVs would be strong contenders for transferring nuclear powered vehicles from LEO to higher (nuclear safe) orbits. We recommend that this concept be considered for orbital transfer missions.

NUCLEAR SPACE TRANSPORTATION OPTIONS AND TECHNOLOGIES

The nuclear space transportation options and technologies we recommend for further study are summarized below.

(1) The submission entitled **The "Enabler," A Nuclear Thermal Propulsion (NTP) System (#100933)** proposes an NTP system that builds on technology developed and tested in the NERVA/ROVER program. The submission advocates updating NERVA by incorporating advances that were initiated in the latter part of that program while including features that address safety and environmental concerns. The major advantage of this concept is that the technology is relatively mature (and tested). A prototype engine with an I_{sp} in the range of 925 to 1000 sec could be available within ten years without unduly restrictive regulatory controls. We recommend that an aggressive RDT&E program be undertaken to develop this capability.

(2) An NTP system that offers an improvement in I_{sp} of about 300 sec relative to NERVA by operating a solid-core reactor at low chamber pressures is proposed in the submission entitled **Low Pressure Nuclear Thermal Rockets (LPNTRs) (#100157)**. The increase in I_{sp} is achieved in low-pressure reactors without exceeding material temperature limits by providing operating conditions that enhance the dissociation of the hydrogen propellant. Depending on chamber pressure, an I_{sp} of about 1200 to 1300 sec is projected. We recommend an RDT&E program to develop this technology.

(3) A nuclear propulsion concept that could be very useful for Mars exploration is proposed in the submission entitled **NIMF Concept to Enable Global Mobility on Mars (#100103)**. This submission proposes the use of CO_2 from the Martian atmosphere as the reactor working fluid in an NTR. The NIMF (while on the Martian surface) would collect and liquify CO_2, which would then be used as propellant. With an unlimited propellant supply available, NIMF would have a global capability, and the I_{sp} of 280 sec is even sufficient to go into Mars orbit. We recommend that this submission be given further consideration.

(4) A substantial reduction in IMLEO and trip time for the Mars mission could be achieved if an advanced nuclear option was available. The point here (also made earlier) is that an advanced propulsion RDT&E program—to establish **whether** any of these advanced concepts **could**, or **should**, be pursued—makes sense if potentially available increases of performance give us extra freedom in manned SEI missions. Three representative advanced nuclear propulsion concepts, which provide I_{sp} at the average levels indicated, were examined:

- Nuclear light bulb (I_{sp} = 1800 sec)
- An open-cycle GCR propulsion system (I_{sp} = 5000 sec)
- Fusion propulsion or a combination of fusion and antimatter propulsion (I_{sp} = 10,000 to 100,000 sec)

The nuclear light bulb concept offers a substantial reduction in IMLEO over the baseline LOX/LH_2 system. The latter two concepts appear to make mission times in the range of 130 days feasible. All three of these concepts are in the early phases of research, and engineering feasibility has not yet been demonstrated. Although doubt exists within the community concerning the containment of the fuel element of the open-cycle GCR, the potential increase in performance, especially for the latter two, is so great that we recommend that a research program be undertaken to identify, as quickly as possible, those advanced nuclear options that are most promising for development.

Antimatter offers the potential for a great increase in performance over the other advanced propulsion options discussed. Many difficult technical problems exist with the large-scale production and storage of antimatter and the development of antimatter engines. However, we believe the potential benefits of antimatter propulsion are such that a long-term antimatter research program should be initiated.

ETO LAUNCH VEHICLES

In the area of ETO launch vehicles, it seems clear that separate manned and unmanned systems will be required for SEI. The options for personnel and small critical payloads appear to be relatively straightforward, i.e., Shuttle, PLS, AMLS, and NASP.

The difficult decision involves the unmanned, heavy-lift vehicle(s) to be used for transporting the extremely large propellant and cargo mass to orbit. Clearly, low cost, sizable payload capacity, operational simplicity, and robustness must be emphasized. Pivotal issues are: (1) whether these attributes are best achieved with a completely new design/development or whether they can be adequately achieved through modification and utilization of existing hardware and launch infrastructure, and (2) the balance to be struck between single-launch payload size and number of launches required for a given IMLEO. The tradeoffs involved in setting this balance were discussed in Sec. IV.

To date, most of the initial planning has evolved toward a payload size on the order of 300,000 lb. This sizing would probably facilitate continued use of Kennedy Space Center as the primary launch site. Resolution of issue 1 is less clear. An SDV in this payload class would avoid a complete new development but would have limited ability to reduce operational costs substantially. An ALS-type approach could go further toward reducing costs but requires new vehicle development, launch infrastructure, and longer lead time. If this general payload class is selected for SEI missions, then we recommend that an upgraded, modernized version of the Saturn V vehicle be added as a candidate for consideration, as is proposed in the submissions entitled **Heavy-Lifting Launch Vehicle Concept (#100192)**, and **A Fall-Back-to-Spring-Forward Strategy to a Heavy-Lift Launch Vehicle: Reviving Saturn V Technology (#100185)**.

Alternatively, some have urged development of a much larger cargo vehicle. Our brief assessment during this Project Outreach activity would lead us to favor a payload capacity of perhaps 500,000 lb or larger in view of the very large IMLEO required and assuming SEI is to be a long-term program. We have discussed (see Secs. II and IV) a large menu of options for providing such capability. From the submissions received, **Ultra Large Launch Vehicle (ULLV) for Moon/Mars Mission (#100110)**, outlines a concept that encompasses many interesting features (discussed in detail in App. P). We recommend that this submission be given serious consideration in the pursuit of very heavy payloads. This, or any other such development, would involve large up-front costs, new launch site(s) and infrastructure, and long lead times, but it may well provide the most net benefits (in terms of the aforementioned desired attributes) over the long haul. Moreover, it may provide operational flexibility to facilitate follow-on space ventures beyond the Mars exploration missions.

Appendix A

SUBMISSION HANDLING, EVALUATION METHODOLOGY, AND TRANSPORTATION PANEL CRITERIA FOR EVALUATING SUBMISSIONS

Submitters were asked to select the appropriate category for their ideas from among those listed in Table A.1. The table shows that all categories received a fair number of submissions. Of the 1697 submissions received, 149 (less than 9 percent) were judged to be incapable of being screened. Another 105 submissions were received after the cutoff date of August 31, 1990.

Table A.1

Submissions Distributed by Category

Category	Screened	Not Analyzed
Architecture	290	1
Systems	52	0
Transportation	350	0
Power	138	1
Life support	156	2
Processing	75	3
Structures	119	1
Communications	45	1
Automation	52	1
Information	21	1
Ground support	28	0
Others	194	4
Undetermined	28	134
Total	1548	149
Received after 8/31/90	105	

A submission was ruled incapable of being screened if it (1) was marked as classified or proprietary or (2) contained no supporting information of any kind. A submission marked as either proprietary or classified was automatically destroyed by the subcontractor. In such cases, the subcontractor noted who destroyed it, the date, and any particulars, then informed the submitter of the destruction of the submission and the reason for it.

As shown in Table A.2, the majority of submissions (63 percent) came from individuals, with 22 percent coming from for-profit firms and 5 percent from educational institutions. The relatively few submissions from educational institutions may have been a problem of timing, because Project Outreach's publicity and submission process began in the summertime, when most lower-level schools are closed and most universities have reduced staffs and enrollments.

Table A.2

Sources of Submission

Source	Submissions Number	Submissions % of Total
Individuals	1061	63
For-profit firms	381	22
Educational institutions	89	5
Nonprofit organizations	72	4
Other	46	3
Groups of individuals	48	3
Total	1697	100

Nevertheless, Project Outreach generated broad national interest. All of the states except Alaska, Arkansas, and Wyoming were represented, as were five foreign countries—Argentina, Australia, Canada, Israel, and Scotland. Interestingly, 40 percent of the submissions came from three states—California with 26 percent, Texas with 9 percent, and Florida with 5 percent.

NASA personnel also contributed to Project Outreach: submissions were received from the Johnson Space Center, Goddard Space Flight Center, Marshall Space Flight Center, Lewis Research Center, Ames Research Center, Jet Propulsion Laboratory, Langley Research Center, the Reston Space Station Program Office, and the Stennis Space Center. A total of 121 submissions were received from NASA locations.

SUBMISSION FORMAT

Submitters were asked for a two-page summary and simple outline of their idea. Submitters were also given the option of submitting an additional ten-page backup explanation of their idea. Only 22 percent of the total submissions included backups. This had implications for the analysis process, which we discuss below.

SUBMISSION HANDLING

Because of time constraints, RAND was obliged to follow an abbreviated six-month schedule. Figure A.1 shows the flow of the process we developed and implemented for handling the submissions. Our task involved simultaneously processing the submissions, developing a methodology, training the panels, and building the software. This time frame allowed no margin for error.

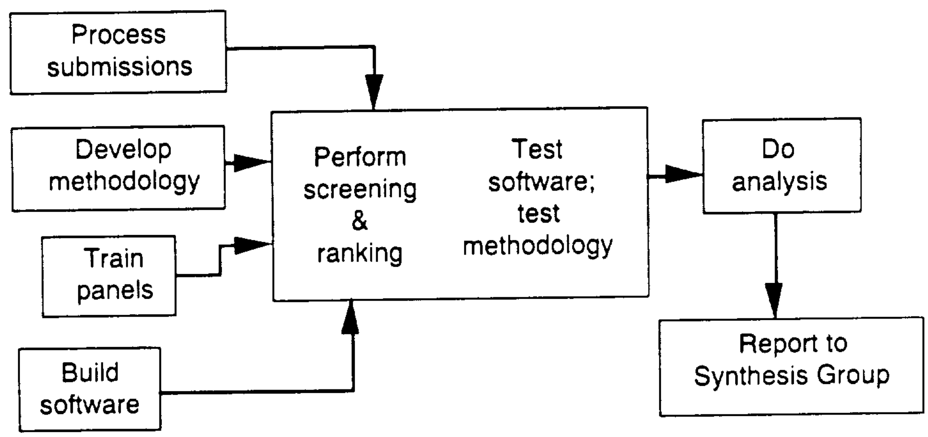


Fig. A.1—Flow of Submission Handling

During our screening and ranking process, we were, in effect, testing the software and the methodology, a highly risky process. We are happy to report they both performed well.

SUBMISSION DATABASE

For each submission, pertinent background information was logged into the database, including the unique ID number of the submission, the reviewer, the date, the name of the panel performing the review, and the title or subject of the review. To remove any bias from the process, the panels did not have information concerning the submitter's name or organization. Reviews of the submissions were entered in a text field. Each reviewer was required to briefly explain the reasons for scoring a submission as he or she did.

PANEL RANKING OF SUBMISSIONS

Primary Ranking Method

Submissions were ranked initially using a method based on weighted sums of five attribute scores. In this case, the attribute weightings were numbers between zero and one that summed to one over the five attributes. These weightings represented the consensus of each panel concerning the relative importance of the attribute for the panel's particular technology/mission area.

Table A.3 presents the screening process weights determined by each panel for each of five common attributes. Each submission received a composite score, computed by summing over all attributes the product of the attribute score (1–5) and its weight. Thus, cardinal rankings represent the overall score of a submission relative to all the submissions within its panel. Rankings by composite score can be sorted within the Fourth Dimension database and recomputed using different attribute weights to perform sensitivity analysis.

Table A.3

Screening Process Weights Determined for Each Panel

Panel	Utility	Feasibility	Safety	Innovativeness	Cost
Architecture	0.30	0.30	0.15	0.20	0.05
Transportation	0.30	0.25	0.25	0.05	0.15
Power	0.25	0.25	0.25	0.10	0.15
Human support	0.40	0.25	0.08	0.25	0.02
Structures	0.30	0.25	0.20	0.10	0.15
Robotics	0.30	0.25	0.01	0.04	0.20
Communications	0.50	0.25	0.01	0.04	0.20
Information	0.29	0.23	0.11	0.20	0.17

Prioritized Ranking Method

To test the robustness of the screening process, each panel also ranked submissions using prioritized attribute ranking methods. In ordinal ranking, the *most important* (primary) attribute is selected, and submissions are ranked according to their scores for that attribute alone. Submissions with equal scores on the primary attribute are then ranked by their score on the next most important, or secondary attribute. The panels found that it was rarely necessary to use a third attribute to rank all the submissions by this process. The prioritized ranking of a submission can then be compared with its general ranking results to determine if there are significant differences. The lack of significant differences in the two ranking systems would indicate that the results are somewhat robust.

In addition, a secondary prioritized ranking was created by reversing the order of the first two attributes in the primary ordinal ranking. Thus, if safety was the most important and utility the second most important attribute for a given panel, the order was reversed. This provided a further check on robustness.

Comparison of Methods

Figure A.2 compares the results of the rankings from the Structures panel submissions. The vertical axis represents the primary rank of a submission, and the horizontal axis measures its prioritized rank. The intersection points of these rankings are shown by small black boxes or squares. The figure contains a 45-degree line from the origin out through the total number of submissions. Submissions that had the same primary rank and the same prioritized rank would fall directly on the 45-degree line. The "best" submission for this panel would be the one closest to the origin, because it would be the one that ranked first in the primary rankings or first in the prioritized rankings, or first on both. Thus, the closer that each of the small black boxes falls to the 45-degree line, the better the congruence of the two ranking methods. Figure A.2 shows that the dark blocks representing the top 20 or 25 submissions are in the lower left-hand corner, indicating good agreement. The agreements of the two ranking methods become less congruent as one moves out into the lower-ranked submissions, which is to be expected.

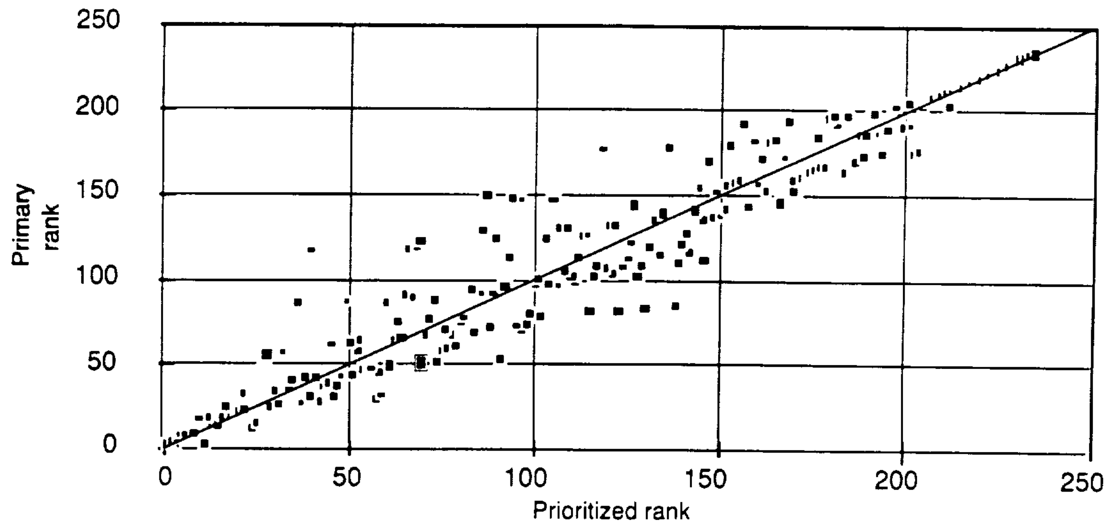


Fig. A.2—Example of Primary Versus Prioritized Ranking

Table A.4 compares the percentage of common submissions found in the lists of the top 20 submissions as created by the three ranking methods just discussed. The left-hand column shows the percentage of submissions that appeared on both the primary and "primary prioritized" lists; it indicates that the percentage of overlap of the top 20 submissions on both lists ranged from 75 to 85 percent. The right-hand column shows the commonalties among three lists: the primary rankings, the "primary prioritized" rankings, and the "secondary prioritized" rankings discussed above. This comparison was made as a more stringent test of robustness; it also reveals a fairly high correlation among the three ranking methods.

This correlation gives confidence in the consistency of the evaluation method used to screen submissions. It shows that whether we extracted the top 20 submissions using the primary or the prioritized methods, they would still be nearly the same.

Table A.4

Comparison of Ranking of Top 20 Submissions for Each Panel

Panel	Percentage of Submissions Appearing on Two Lists[a]	Three Lists[b]
Architecture	75	40
Transportation	75	35
Power	85	75
Life support	80	55
Structures	85	80
Communications	85	55
Robotics	85	55
Information	80	80

[a]Primary and prioritized.

[b]Primary, prioritized, and reverse prioritized.

Appendix B

LIST OF ALL TRANSPORTATION PANEL SUBMISSIONS

Submission ID	Title/Subject
100042	Back to the Future
100055	Project Adonis
100100	Shuttle to the Moon
100101	Electromagnetic Coilgun Launcher
100102	A New Rocket Propulsion Engine Utilizing a Full-Flow Topping Cycle and Gas-Gas
100103	* NIMF Concept to Enable Global Mobility on Mars
100105	Whirley-Go
100106	A Method of Translation
100107	Mars Mission with Present Technology
100108	Aerobie Shuttle
100109	A System for Economical Transport to E. Orbit
100110	Ultra Large Launch Vehicle (ULLV) for Moon/Mars Mission
100111	Mass Transportation Between Earth and Moon
100112	Alternate Propulsion Systems/Basic Research
100113	Untitled
100114	American Energia and Buran
100115	The Cyclone Motor
100116	Orbiting Tug
100117	Mass Transfer Device (MTD)
100118	Conceptual Design and Analysis of the Lunar Operations Vehicle
100119	Space Propulsion Using a D-3He Field Reversed
100120	Space Station
100121	Achieving Mars Transfers via Multiple Lunar Swingbys
100122	Hydride Solid Rocket Fuel
100123	Methods of Controlling the Effects of Gravity and a Cold Fusion Apparatus
100124	Advanced Aerospace Propulsion System
100125	Atomic Considerations Regards Propulsion
100128	A Consumable Lunar Supply Craft
100129	Advanced Launch Vehicle Concept

Submission ID	Title/Subject
100130	High Thrust Ion Propulsion
100131	High Density Fuels
100132	Direct Nuclear Thrust (DNT)
100133	Metallized Propellants for the Space Exploration Initiative
100134	Boron Reaction Drive
100135	The Development of NERVA and Other Advanced Propulsion Systems
100136	Inertial Drive Unit
100138	Space Induction Magnetic System
100139	Cheap Launch from Earth to Orbit
100140	Nuclear Rocketry
100144	The Use of Solar Energy in Space Flight
100145	Cesium and Rubidium in Ion-Propulsion Systems
100146	Antimatter Inducer
100147	Simulating Tornado to Make Fuel Last Eighteen Times Longer
100148	Ray Propellor
100149	Levitation of Payloads into Space
100150	Lift Generating Descent and Ascent Mechanism for a Directionally Controlled Mart
100151	Alternate Propulsion System for Shuttle Vehicle
100153	Beyond Electric Propulsion
100154	Use of the Space Shuttle to Return to the Moon
100155	Gravitationally Boosted Impact Propulsion and Power
100156	A Novel Fusion Propulsion Scheme
100157	* Low Pressure Nuclear Thermal Rockets (LPNTRs)
100158	Clustered Low Thrust Nuclear Thermal Rocket Engines
100159	Electrogravities: An Energy-Efficient Means of Spacecraft Propulsion
100160	Refuelling from Ion-Engine Propelled Tankers for a Fast (200-day) Round Trip
100161	Vortex-Based Propulsion System
100162	Concerned American (Pro-World)
100163	Integrated Power and Propulsion Systems Based on Hydrogen and Oxygen
100164	Space Bicycle
100165	Hybrid Boosters

Submission ID	Title/Subject
100166	Future Reuse Options for OV-102
100168	Bump Propulsion
100169	Electronic Balloon
100170	Pulsed MPD Electric Propulsion
100171	Plug Cluster Nozzle Options for Space Transfer and Excursion Vehicles
100172	Trash Hybrid Attitude Control/Propulsion Systems
100173	Gas Core Nuclear Light Bulb (NLB) Rocket
100174	Gyro Propulsion
100175	Scramjet Accelerator for Orbital Applications
100176	Rail Launchers for Ground-to-Orbit Payloads
100177	Low-Cost Surplus Hardware Moon/Mars Proposal
100178	Defining Antigravity
100179	Earth-Moon Transport Vessel
100182	Interplanetary Shuttle
100183	Small Launch Vehicles for Mars Network Missions
100184	Integral Bipropellant Propulsion for Orbit Transfers
100185	A Fall-Back-to-Spring-Forward Strategy to a Heavy-Lift Launch Vehicle
100186	A Solar Wind-Jammer
100187	A Solar Space Mission to Mars from Moon
100188	Reuse Apollo Design for Lunar Excursion Module
100189	Developing Heavy Lift Pilotless Launch Vehicles with Existing Space Shuttle Tech
100190	Isothermal Expansion Nuclear Thermal Rockets
100191	A Joint Space Powers Exploration of Mars
100192	* Saturn V Heavy Lift Concept
100392	Plan for Astronaut Mobility Device
100400	Long-Endurance Aircraft as a Mars Exploration Vehicle
100414	Solar Incineration
100434	Space Travel—Is There a Practical Way?
100443	Flex Wing for Martian Transportation
100450	Energy Impulse Engine
100455	Tether Tower, Lunar
100462	Antimatter Driver Fusion Propulsion System

Submission ID	Title/Subject
100470	Docking Plan, Space Service Vehicle (SPV), Diving Bell
100481A	Laser Thermal Propulsion
100481B	Tether Transport from LEO to Lunar Surface
100481C	Solar Thermal Propulsion
100481D	Non-orbiting Spacecraft
100481E	Solar Photon Thruster
100481F	Cable Catapult
100481G	Tether Variant
100481H	Metallic Hydrogen
100481I	Magnetic Engines and Nozzles
100481J	Antiproton Engines
100481K	Magnetic Sails
100481L	Solar Sails
100551	Space Transportation, Launch Vehicles and Propulsion
100561	Space Transportation Reliability Simulator
100562	Duo Shuttles to Mars
100563	Neutrino Accumulator
100564	Orbital Placement Vehicle
100565	Solar Power Reaction Jet Engine
100566	* Nuclear Rocket Power and Propulsion System for Mars
100567	Speed Unlimited (In Space)
100568	Multi Stage Space Transportation System
100569	The Reaction Propulsion Unit
100570	Plasma Engine for Mars Transfer Vehicle
100571	Modular Autonomous Design of a Mars Transfer Vehicle
100572	Transfer and On-Orbit Storage of Hydrogen and Oxygen Rocket Propellants as Water
100574	Hydrogen/Oxygen Powered Turbofan for Mars Aircraft
100575	Lunatron—Lunar Surface-Based Electromagnetic Launcher
100576	Trimarket Aerospace Transport
100577	Self-Returning Rocket Engine
100638	Low Cost Earth-to-Orbit Launch System
100639	Air-Launch Personnel Transportation System

Submission ID	Title/Subject
100654	Solar-Powered, Coil-Gun Star Gate Network
100662	Needed, a Site to Launch Nuclear OTVs and Other Potentially Unsafe Equipment
100663	Solar Sails
100664	Space Carrier
100665	Big Smart Ramjet Booster
100666	Maximizing Usage of Surplus Hardware
100667	Summary of Single-Stage Rocket (SSR)
100668	Multiconfiguration Space Transportation System
100669	* Lunar-Mars Propellants
100671	Nuclear Aerospace Plane
100672	Advanced EM Propulsion Systems
100700	Ozone Replenishment
100703	A Heavy Payload Walking Vehicle
100714	The Pony Express to Mars
100721	Sailing the Solar Wind
100722	Solar Ionic Propulsion
100723	Future Space Propulsion
100740	Materials and Mechanisms
100749	An Infrastructure for Solar System Exploration
100759	Space/Supply Rescue Vehicle
100760	Yield Propulsion
100761	New Shuttle Design
100762	Electric Propulsion for Aerospace Application
100763	Probable Immediate Alternate Orbital Launch Capability
100764	Untitled
100765	Untitled
100766	Solid Separation and Retro Rockets for Assured Crew Rescue Vehicle (ACRV)
100767	Lunar/Mars Return Propulsion System
100768	High Expansion Uncooled Nozzles
100769	Strap-on Solid Rocket Boosters
100802	Tethered Propulsion System

Submission ID	Title/Subject
100803	Plastic-Fuel Rocket Motors
100804	Applications of Orbital Electromagnetic Rail Gun
100805	Lunar-Earth Dual-Energy Skyhook
100820	Satellite Propulsion—Magneto, Hydrodynamics, Ion Thruster
100832	Mars Spacecraft Using Thermionic Nuclear Power and Plasmoid Thrusters
100833	Waverider Shuttles
100834	Ballooning into Space
100835	Electrostatically Powered Aerospace Vehicles
100836	Piggyback Fuel Station
100837	Slinging Payloads into Space from Aircraft
100838	Pulsed Plasmoid Electric Propulsion
100863	Advanced Propulsion for Mars Mission
100868	American/Soviet Low-Cost ICBM Launch Vehicles
100869	Untitled
100871	Untitled
100921	An Alternative Mission Concept
100928	Earth-Moon (Mars) Vicinity Transport System
100929	A Reusable Exploration Vehicle
100930	Laser Launch
100931	Controlled Thermonuclear Fission/Fusion
100932	Lunar-Derived Propellants
100933	The ENABLER, A Nuclear Thermal Propulsion System
100934	A Roller Coaster Launch (RCL)
100935	Contracting Out for Freight Delivery
100936	STS—Space Transportation System
100937	Mini Service Station
100938	LEO Tether Transportation Node
100939	Antimatter Propulsion
100940	Lunar Orbit Tether Transportation Node
100941	Phobos Tether Transportation Station
100942	Fast Orbit-to-Orbit Propulsion System
100943	Half-Sized Shuttle Orbiter and Booster
100944	Microwave Propulsion

Submission ID	Title/Subject
100945	Enhancements to the SASSTO Design
100946	Impulse Drive
101012	A Practical System of Optimal Rocket Staging
101013	Micronuclear Physics—A New Presentation of Quantum Wave Mechanics
101014	Air Scoop
101015	Emergency Orbital/Positioning
101016	A Solar Sail Design for Space Transportation and Power Beaming
101017	Direct Solar Powered Space Propulsion System
101018	Extended Range Orbiter
101019	Modular, Reusable Spacecraft Utilizing Existing Space Hardware
101020	Fission Fragment Rocket
101021	Orbital and Positioning
101022	* Shuttle Heavy Lift Vehicles
101023	Flight and Cruise
101024	Backup Orbital/Positioning
101025	Project Charon
101026	* Nuclear Pulse Propulsion
101027	Space Transport Vehicle
101028	Proposal for Water-Vapor Based Life System
101029	* Earth-to-LEO Electromagnetic Launch
101030	Beaming Across the Universe
101031	Concentrate Effort on Space Plane
101032	High Payload SSTO Vehicle
101033	Two-Launch Vehicle Architecture
101144	Nuclear Electric Powered (NEP) Interplanetary Cargo Vehicle
101145	Methane (Hydrocarbon) Rocket Engines for Martian Transfer and Excursion Vehicles
101146	Modularized Launch Vehicles
101147	Rotor Supported by Fluid Film Elements Solely
101148	Carbide-Fueled Nuclear Thermal Rocket
101149	Common Propulsion Module for Space Transfer
101150	Low-Cost Launch Vehicle for Fluid Transport
101151	Balloon Launch of Small Rockets

Submission ID	Title/Subject
101152	Magsail Asteroid Survey Missions
101153	Magsail Stabilization of Lagrange Point Structures
101154	Another Use for a Space Elevator
101155	Automated Propulsion System Checkout, Installation and Removal
101156	Transportation: Martian Orbit-to-Surface
101157	Solar Electrical Orbital Transfer Vehicle (SEOTV)
101158	Storable Engine for Mars and Lunar Lander
101159	Magsail Mars Missions
101160	Nuclear Electric Propulsion for Mars Transfer
101161	Nuclear Thermal Propulsion for Mars Transfer
101162	Liquid Strap-on Boosters
101164	In-Orbit Modification of a Shuttle External Tank to Transport Lunar-Produced He^3
101165	Transportation of Payloads into Low Earth Orbit Using Railguns
101166	Advanced Heavy Lift Launch Vehicles (HLLVs)
101167	* Storage of Hydrogen Using Metal or Silicon Hydrides
101168	Integrated Rotating (Artificial Gravity) Habitat/Nuclear Power System
101169	CO_2 Cracking to CO and O_2 for Energy Needs on Mars
101170	Wire Core Reactor for Nuclear Thermal Propulsion
101171	To GEO
101172	Anode Plasma Engine
101173	A Cable Car to Space Made of Existing Materials
101174	Pop Up/Burn-to-Orbit Launcher
101175	Integrated Reaction Control/Main Chemical Propulsion
101176	Transkinetic Nozzle
101177	Propulsion for Space Exploration Using Electron-Beam Storage Rings on Space Sate
101178	In-Situ Propellants for Mars Lander—Chemical Engines
101179	Application of Mirror Fusion Technology to Propulsion for Interplanetary Satellite Vehicle
101180	Light-Weight, High-Temperature Materials Development for Advanced Combustion Development
101181	Carbon Dioxide Breathing Propulsion for Mars Spaceplane

Submission ID	Title/Subject
101182	Magneto Plasma Dynamic (MPD) Propulsion
101183	Indigenous Propellant (CO_2) Nuclear Thermal Rocket (NTR) for a Mars Exploration
101184	Advanced O_2/H_2 Transfer and Lander Vehicle Propulsion System
101185	Regolith as Propellant for Mars Mission
101186	CIS Lunar Ferry
101187	Upgraded Shuttle with One Propulsion Pod
101188	Solar Thermal Orbital Transfer Vehicle (STOTV)
101189	Combined Cycle Nuclear Propulsion
101190	Multimegawatt Nuclear Electric Propulsion
101191	Flying Wing/Rocket Launch Vehicle
101192	Use of Launch Vehicle Components for Space Exploration Vehicles
101193	Common Storable Engine for Mars and Lunar Landers
101194	Nozzle Extension with Ablative Insert
101195	Damping Bearings for Cryogenic Turbomachinery
101196	Upgraded Shuttle with Two Propulsion Pods (US2)
101197	External Fuel Tanks as Expandable Spacecraft
101199	Upgraded Shuttle with Orbiters (USO)
101200	Hot Gas Balloon Mars Hopper
101201	Potassium Rankine Nuclear Electric Propulsion
101202	Replenishing an Orbital Propellant Depot
101203	Methanol-Fueled Rover
101204	Reusable, All Propulsive Lunar Transportation System
101205	Propulsion System Integrated Control and Health Monitoring
101206	Multipropellant Chemical Engine
101208	Rail Gun/Ramjet/Rocket Three-Stage Launch Vehicle
101209	Efficient, Long-Term Storage of Hydrogen
101210	Transfer and Lander Vehicle Engine Commonality
101211	Redundant Component Engine Cluster
101212	High-Energy Chemical Propulsion for Space Transfer
101213	The Low Pressure Nuclear Thermal Rocket
101214	Alternate Shuttle-C Heavy Life Launch Vehicle
101215	Advanced O_2/H_2 Deep Throttling Lander Vehicle Engine

Submission ID	Title/Subject
101216	Nozzle Isolation System for Dual Mode NTR/NEP
101347	H/O Propellant as Life Support Resource
101355	Augmented Thrust Devices for Rocket Vehicles
101356	Laser-Heat-Transfer Rocket
101357	Cold Fusion for Rocket Engine Propulsion
101358	Modified Interplanetary Shuttle
101359	Untitled
101360	The Space Ferry
101392	Solar Sail Cargo Vessels to Reduce Mars Expedition Costs
101393	The "MALT" Launch System
101394	Heavy Payload Launching Vehicle
101396	Redesign of Extant Motor
101397	Propulsion Systems for Mars Missions
101398	Shock-Absorbing Landing Pads
101399	Solar Thermal Rocket System for Orbital/Injection Transfer Vehicle
101401	Modified Launch Stack for Larger, Safer Space Station Modules
101442	Laser Simulation of Nuclear Rocket Propulsion
101450	Closing a Thermodynamic System with Gravity
101455	Solar-Wind Exploration
101456	The Searl Levity Disc
101466	Kests Technology and Forms
101473	A Free-Flying Martian Explorer
101478	Cure and Propulsion Thru Attraction
101518	Induced Electron Capture and Supercold Fusion
101519	Mobile Space Station Delivery System (MSS)
101520	Shuttle-C
101521	Hydrogen Ice for On-Orbit Refueling
101522	Lunar Orbital Electro-Magnetic Ring Catcher Satellites
101552	Orbiter Space Platform
101555	Electromagnetic Coilgun Launcher
101563	Earth-Based Microwave Power Beaming to Interorbital Electrically Propelled Trans
101564	An Efficient Launch Vehicle CM1

Submission ID	Title/Subject
101565	A Self-Sustaining Rocket Engine CM_2
101568	Piggyback Ramjet
101570	Inertial Engine
101571	Antiproton and Fusion Powered Aerospace Plane
101572	Space Ferry SVF
101573	Space Crawler Vehicle
101574	Mars Shuttle
101605	Rocket Propellant Production from the Martian Atmosphere
101619	Combustion Studies for Large Liquid Propellant Rocket Engines
101620	Revive Project Orion
101621	Surface Vehicle Engine
101622	Closed Reactive Propulsion System
101630	Gravitational Propulsion
101638	Ultra-Lightweight and High Efficiency Electric Motor Design
101640	Sensorless DC Brushless Electronic Motor Drive
101645	Unmanned Heavy Lift Launcher
101651	Planetary Space Transportation
101653	Thruster
101654	Space Drive
101679	Mars Exploration with Lighter Air Vehicles
101688	Walking Robot for Mars Rover
101694	Liquid Rocket Booster Elements for STS
101695	Real-time Health Monitoring, Diagnostics and Control System for Aerospace
101696	The Sharp Mars Concept—Revised Copy
101697	The Keys to Mars, Titan and Beyond?
200453	How to Build a Flying Saucer
200456	Space Plane
200457	Untitled
200459	The Combined Launch and Construction System (COMLACS)
200461	The Future of the World and its Economy
201119	United Space Fleet (USF)
201129	Reusable, All Propulsive Transportation Vehicle Architecture

Submission ID	Title/Subject
201342	A Justification for a Policy Favoring Reusable Manned Spacecraft
201374	A Citizen's Overlook
201476	Space Exploration Vehicle
201641	Power Factor Corrected Electronic Motor Drive
301127	Combined Lunar Transfer Vehicle Electric Propulsion and Lunar Surface

NOTES: (1) There were a total of 350 Transportation submissions.
(2) The * denotes submissions that aggregated a number of similar submissions under a single title.

Appendix C

VELOCITY REQUIREMENTS FOR ROUND-TRIP MISSIONS TO MARS

INTRODUCTION

This appendix describes the mathematical model used to determine the velocity-added requirements for a round-trip mission to Mars. The model employs the *patched conic* approach, in which the spacecraft trajectory is analyzed under the successive gravitational influences of the departure planet, the sun, and the arrival planet, respectively. Because the intent is to obtain only first-order approximations to the velocity requirements, several simplifying assumptions are invoked to facilitate the analysis, but with relatively minor loss in fidelity.

We focus attention exclusively on the so-called *direct mission*, in which the spacecraft receives no gravity assistance from swingbys of Venus or other bodies. Moreover, we impose no restrictions on the time (e.g., year, month, etc.) at which a given mission must occur. Rather, we compute the relative geometry of Earth and Mars necessary to satisfy a given set of mission requirements, and assume that the mission could be scheduled at a time for which the geometry is appropriate.[1]

METHODOLOGY

For our calculations, we assume that Earth and Mars revolve about the sun in coplanar, concentric, circular orbits. Relevant orbit and planetary data are given in Table C.1.[2]

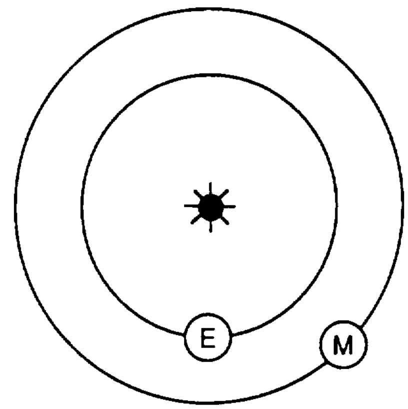
[1]Under our simplifying assumptions, the Earth-Mars geometry relative to the sun repeats every $780 = \dfrac{1}{\dfrac{1}{365.25} - \dfrac{1}{687}}$ days, the so-called *synodic period* of Mars. This is therefore an upper bound, in our analysis, on the waiting time to achieve a desired Earth-Mars configuration.

[2]For the remainder of this appendix, unless otherwise indicated, all distances are given in kilometers, all times in seconds, and all speeds in kilometers per second.

Table C.1

Orbit and Planetary Data

Characteristic	Value
Earth heliocentric orbit radius (R_e)	149,500,000 *km*
Mars heliocentric orbit radius (R_m)	227,800,000 *km*
Earth heliocentric orbit speed (V_e)	29.78 *km/sec*
Mars heliocentric orbit speed (V_m)	24.12 *km/sec*
Earth heliocentric angular orbit speed (ω_e)	1.991×10^{-7} *rad/sec*
Mars heliocentric angular orbit speed (ω_m)	1.059×10^{-7} *rad/sec*
Earth heliocentric orbit period	365.25 *days*
Mars heliocentric orbit period	687 *days*
Mars synodic period (syn)	780 *days* (2.13 *yr*)
Earth radius (r_e)	6376 km
Mars radius (r_m)	3380 km
Earth gravitational constant (μ_e)	398,603 km^3/cm^2
Mars gravitational constant (μ_m)	43,050 km^3/cm^2
Sun gravitational constant (μ_s)	1.327×10^{11} km^3/cm^2

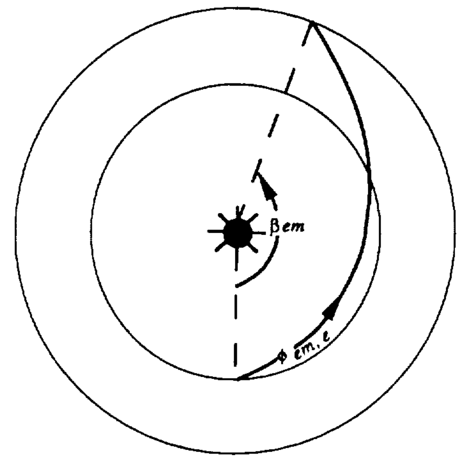


We assume that the spacecraft is initially in a circular parking orbit about Earth at altitude a_e, with corresponding orbit speed $v_{circ,e} = (\frac{\mu_e}{r_e + a_e})^{1/2}$. At an appropriate place and time in this orbit, which we call *time zero*, the spacecraft is given a tangential velocity increment to inject it into a hyperbolic (relative to Earth) orbit that will escape the Earth's effective gravitational field. When the craft is sufficiently far from Earth (say 1,000,000 km), its motion becomes dominated by the gravitational influence of the sun. It remains under this influence until it arrives sufficiently close to Mars, at which time its motion is referenced to a Mars-centered system. The hyperbolic (relative to Mars) approach trajectory carries the craft to a perigee altitude a_m above the Martian surface, at which time the craft is given a retro tangential velocity increment to slow it down and circularize it into a Martian parking orbit, in which its speed is $v_{circ,m} = (\frac{\mu_m}{r_m + a_m})^{1/2}$. After some specified interval in Martian orbit, the above process is repeated: the spacecraft injects tangentially into a hyperbolic (relative to Mars) escape trajectory and returns to Earth, where it circularizes into a parking orbit at altitude a_e.[3] Our goal is to determine four velocity-added increments: the injection increments leaving Earth and leaving Mars, and the circularizing increments needed upon arrival at Mars and arrival back at Earth.

[3]The possibility of employing aerobraking (in combination perhaps with propulsion braking) to slow the vehicle as it approaches Mars and (on its return) Earth is discussed in App. D.

EARTH-TO-MARS TRAJECTORY ANALYSIS

We begin our trajectory analysis by examining the heliocentric Earth-Mars transfer orbit, which is perturbed at its planetary endpoints by the gravitational influences of Earth and Mars. Because the spacecraft spends such a small fraction of its total travel time under these influences, there is little loss in generality in treating this orbit as an unperturbed conic passing through the center of both Earth and Mars. We shall assume, as fundamental inputs to our model, two variables that totally characterize this conic: (a) the angle $\phi_{em,e}$, between $-\frac{\pi}{2}$ and $\frac{\pi}{2}$, at which the conic intersects the Earth's heliocentric orbit path; and (b) the sun-central angle β_{em}, between 0 and 2π, measured in the direction of the Earth's heliocentric orbital motion, between the Earth at time zero and Mars at the time of spacecraft arrival there. From these inputs, the orbital elements and other parameters of the transfer orbit can be computed:



Spacecraft true anomaly at Earth departure point (time zero)[4]

$$\theta_e = arg\,(R_m - R_e - R_m \tan\phi_{em,e} \sin\beta_{em}\ ,\ R_m \tan\phi_{em,e}\,(1 - \cos\beta_{em}\,))$$

Eccentricity

$$e_{em} = \frac{R_m - R_e}{R_e \cos\theta_e - R_m \cos(\beta_{em} + \theta_e)}$$

Parameter of orbit

$$p_{em} = R_e\,(1 + e_{em}\cos\theta_e)$$

[4]Here $arg(x,y) = \tan^{-1}(y/x)$ if $x > 0$; $= \tan^{-1}(y/x) + \pi$ if $x < 0$; $= \pi/2$ if $x = 0, y > 0$; $= -\pi/2$ if $x = 0, y < 0$; $=$ undefined if $x = 0, y = 0$. The true anomaly measures the central angle between the perihelion of the transfer orbit and the spacecraft position at Earth-vicinity departure.

Semi-major axis

$$a_{em} = \frac{p_{em}}{1 - {e_{em}}^2}$$

Perihelion (of Earth-to-Mars conic segment)

$$R_{per,em} = \begin{cases} a_{em}(1 - e_{em}) & \text{if } \phi_{em,e} \le 0 \\ R_e & \text{if } \phi_{em,e} > 0 \end{cases}$$

Energy of orbit

$$EN_{em} = \frac{-\mu_s}{2\, a_{em}}$$

Specific angular momentum

$$h_{em} = \left[\mu_s p_{em}\right]^{1/2}$$

Sun-referenced spacecraft speed at Earth-vicinity departure point

$$v_{em,e} = \left[2\,(EN_{em} + \frac{\mu_s}{R_e})\right]^{1/2}$$

Earth-referenced hyperbolic excess speed as spacecraft leaves Earth's sphere of influence

This latter value, $v_{em,e}$, approximates the spacecraft speed when it leaves the Earth's sphere of influence. The magnitude of the vector difference between the Earth's heliocentric orbital velocity and the spacecraft velocity vector (of magnitude $v_{em,e}$) near the Earth-vicinity departure point is the so-called (Earth-referenced) *hyperbolic excess* speed of the craft as it leaves the Earth's sphere of influence. Its value is given by:

$$v_{hyp,eme} = \left[{v_{em,e}}^2 + {V_e}^2 - 2v_{em,e}V_e \cos\phi_{em,e}\right]^{1/2}$$

Earth-referenced spacecraft speed following injection from Earth parking orbit

$$v_{i,eme} = \left[{v_{hyp,eme}}^2 + \frac{2\mu_e}{r_e + a_e}\right]^{1/2}$$

The difference $v_{LE} = v_{i,eme} - v_{circ,e}$ thus represents the required velocity-added to leave Earth parking orbit.

Earth-to-Mars flight time

The computation of Earth-to-Mars flight time, T_{em}, varies according to whether the transfer orbit is elliptical or hyperbolic:[5]

$$\textit{Case 1: } e_{em} < 1 \qquad T_{em} = \frac{a_{em}^{3/2}}{\mu_s^{1/2}} \left[E_{2e} - E_{1e} - e_{em}\,(\sin E_{2e} - \sin E_{1e})\right]$$

$$\text{where} \quad E_{1e} = arg\,(e_{em} + \cos\theta_e,\ (1 - e_{em}^2)^{1/2} \sin\theta_e)$$

$$E_{em} = arg\,(e_{em} + cos\,(\beta_{em} + \theta_e),\ (1 - e_{em}^2)^{1/2}\, sin\,(\beta_{em} + \theta_e))$$

$$E_{2e} = \begin{cases} E_{em} & \text{if } E_{em} \geq E_{1e} \\ E_{em} + 2\pi & \text{if } E_{em} < E_{1e} \end{cases}$$

$$\textit{Case 2: } e_{em} > 1 \qquad T_{em} = \frac{(-a_{em})^{3/2}}{\mu_s^{1/2}} (e_{em}\,(sinh\ E_{2e} - \sinh E_{1e}) - E_{2e} + E_{1e})$$

$$\text{where} \quad E_{1e} = \frac{1}{2} \log\left[\frac{1 + c_{1e}}{1 - c_{1e}}\right]$$

$$c_{1e} = \frac{(e_{em}^2 - 1)^{1/2} \sin\theta_e}{e_{em} + \cos\theta_e}$$

$$E_{2e} = \frac{1}{2} \log\left[\frac{1 + c_{2e}}{1 - c_{2e}}\right]$$

$$c_{2e} = \frac{(e_{em}^2 - 1)^{1/2} \sin\,(\beta_{em} + \theta_e)}{e_{em} + \cos\,(\beta_{em} + \theta_e)}$$

Figure C.1 shows how Earth-to-Mars flight time varies with the input parameters $\phi_{em,e}$ and β_{em}. The curve segments bounded by asterisks represent transfer trajectories whose closest approach to the sun is less than 70 million km. As a point of reference, note the point on the middle (ninth) of the 17 curves, at a sun-central angle of 180 deg. This represents the geometry of the Hohmann transfer trajectory, whose flight time (as is well known) is about 260 days. Keep in mind that the plot ignores velocity-added considerations—the short flight times corresponding to points on the lower left portion of the curve swarm obviously require unreasonably high velocity increments v_{LE}, as indicated on Fig. C.2, which follows. Here we see, again as a function of the two basic inputs, how the required injection velocity from

[5]The parabolic case occurs with zero probability and is thus not treated.

Earth parking orbit to Earth-Mars transfer orbit varies. The lowest point on the bottom envelope (at 180-deg central angle) corresponds to the Hohmann transfer, requiring an injection velocity increment from 400-km Earth parking orbit of 3.57 km/sec. Figure C.3 combines the data of the first two figures and depicts, as a function of Earth-to-Mars flight time, the velocity increment v_{LE} required to inject from Earth parking orbit to Earth-Mars transfer orbit. Velocities shown are minimal among all choices for the input parameters $\phi_{em,e}$ and β_{em}, given the indicated constraint on transfer orbit perihelion.

Sun-referenced spacecraft speed as it enters Mars' sphere of influence

We next compute the speed $v_{em,m}$ of the spacecraft as it approaches Mars' heliocentric orbit path. By energy conservation,

$$v_{em,m} = (2\,(EN_{em} + \mu_s / R_m))^{1/2}$$

Angle at which Earth-to-Mars transfer orbit crosses Mars heliocentric orbit

The planetary transfer orbit crosses Mars' orbit path at an angle $\phi_{em,m}$, where (since angular momentum on the transfer orbit is constant)

$$\phi_{em,m} = \cos^{-1}\left[\frac{h_{em}}{v_{em,m}\, R_m}\right]$$

Mars-referenced hyperbolic excess speed as spacecraft enters Mars' sphere of influence

$$v_{hyp,emm} = \left[v_{em,m}^{\;2} + V_m^{\;2} - 2v_{em,m} V_m \cos\phi_{em,m}\right]^{1/2}$$

Mars-referenced spacecraft speed at perigee altitude above Mars

$$v_{i,emm} = \left[v_{hyp,emm}^{\;2} + \frac{2\mu_m}{r_m + a_m}\right]^{1/2}$$

Velocity-added in circularizing into Mars parking orbit

$$v_{AM} = v_{i,emm} - v_{circ,m}$$

Figure C.4, similar to Fig. C.3, shows how the minimum required velocity-added to circularize into Mars parking orbit varies with Earth-to-Mars flight time. As expected, the curve bottoms out at a v_{AM} value of 2.066 km/sec, corresponding to the 260-day-duration Hohmann transfer trajectory.

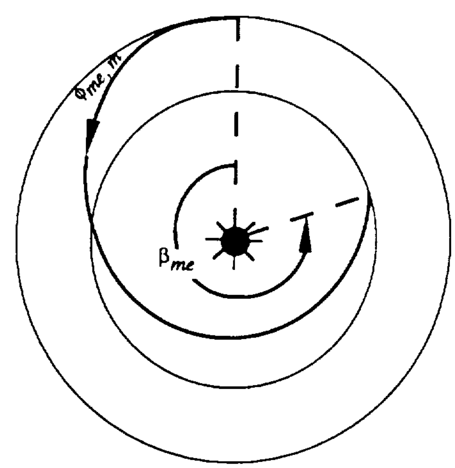
Sun-central angle by which Mars leads Earth at time spacecraft departs Earth

Finally we can determine, for a given set of input angles $\phi_{em,e}$ and β_{em}, the required relative phasing of Earth and Mars at zero hour; that is, the sun-central angle by which Mars leads Earth:[6]

$$phase_{em} = mod(\beta_{em} - \omega_m T_{em}, 2\pi)$$

MARS-TO-EARTH TRAJECTORY ANALYSIS

The geometry, assumptions, and requirements for the return flight from Mars to Earth are symmetrical in nature to those just presented. We introduce two new input variables: (a) the angle $\phi_{me,m}$, between $-\frac{\pi}{2}$ and $\frac{\pi}{2}$, at which the Mars-to-Earth (heliocentric) conic trajectory intersects the Mars orbit path about the Sun; and (b) the sun-central angle β_{me}, with $0 \le \beta_{me} < 2\pi$, measured in the direction of Mars' heliocentric orbital motion, between Mars at the time of spacecraft departure and Earth at the time of spacecraft arrival. As before, we compute:



Spacecraft true anomaly at Mars departure point

$$\theta_m = arg(R_e - R_m - R_e \tan\phi_{me,m} \sin\beta_{me}, R_e \tan\phi_{me,m}(1 - \cos\beta_{me}))$$

Eccentricity

$$e_{me} = \frac{R_e - R_m}{R_m \cos\theta_m - R_e cos(\beta_{me} + \theta_m)}$$

Parameter of orbit

$$p_{me} = R_m(1 + e_{me}\cos\theta_m)$$

[6]Here, $mod(x, y) = x - y\,[x/y]$, where $[x/y]$ denotes the greatest integer $\le x/y$.

Semi-major axis

$$a_{me} = \frac{p_{me}}{1 - e_{me}^{2}}$$

Perihelion (of Mars-to-Earth conic segment)

$$R_{per,me} = \begin{cases} R_e & \text{if } mod(-\theta_m, 2\pi) > \beta_{me} \\ a_{me}(1 - e_{me}) & otherwise \end{cases}$$

Energy of orbit

$$EN_{me} = \frac{-\mu_s}{2\, a_{me}}$$

Specific angular momentum

$$h_{me} = \left[\mu_s p_{me}\right]^{1/2}$$

Sun-referenced spacecraft speed at Mars-vicinity departure point

$$v_{me,m} = \left[2\,(EN_{me} + \frac{\mu_s}{R_m})\right]^{1/2}$$

Mars-referenced hyperbolic excess speed as spacecraft leaves Mars' sphere of influence

$$v_{hyp,mem} = \left[v_{me,m}^{\ 2} + V_m^{\ 2} - 2v_{me,m} V_m \cos \phi_{me,m}\right]^{1/2}$$

Mars-referenced spacecraft speed following injection from Mars parking orbit

$$v_{i,mem} = \left[v_{hyp,mem}^{\ 2} + \frac{2\mu_m}{r_m + a_m}\right]^{1/2}$$

Velocity-added in leaving Mars parking orbit

$$v_{LM} = v_{i,mem} - v_{circ,m}$$

Mars-to-Earth flight time

$$Case\ 1:\ e_{me} < 1 \qquad T_{me} = \frac{a_{me}^{3/2}}{\mu_s^{1/2}} \left[E_{2m} - E_{1m} - e_{me}(\sin E_{2m} - \sin E_{1m})\right]$$

$$\text{where} \quad E_{1m} = arg(e_{me} + \cos\theta_m, (1 - e_{me}^2)^{1/2} \sin\theta_m)$$

$$E_{me} = arg(e_{me} + cos(\beta_{me} + \theta_m), (1 - e_{me}^2)^{1/2} sin(\beta_{me} + \theta_m))$$

$$E_{2m} = \begin{cases} E_{me} & \text{if } E_{me} \geq E_{1m} \\ E_{me} + 2\pi & \text{if } E_{me} < E_{1m} \end{cases}$$

$$Case\ 2:\ e_{me} > 1 \qquad T_{me} = \frac{(-a_{me})^{3/2}}{\mu_s^{1/2}} (e_{me}(sinh\ E_{2m} - \sinh E_{1m}) - E_{2m} + E_{1m})$$

$$\text{where} \quad E_{1m} = \frac{1}{2} \log\left[\frac{1 + c_{1m}}{1 - c_{1m}}\right]$$

$$c_{1m} = \frac{(e_{me}^2 - 1)^{1/2} \sin\theta_m}{e_{me} + \cos\theta_m}$$

$$E_{2m} = \frac{1}{2} \log\left[\frac{1 + c_{2m}}{1 - c_{2m}}\right]$$

$$c_{2m} = \frac{(e_{me}^2 - 1)^{1/2} \sin(\beta_{me} + \theta_m)}{e_{me} + \cos(\beta_{me} + \theta_m)}$$

Sun-referenced spacecraft speed as it enters Earth's sphere of influence

$$v_{me,e} = (2(EN_{me} + \mu_s / R_e))^{1/2}$$

Angle at which Mars-to-Earth transfer orbit crosses Earth heliocentric orbit

$$\phi_{me,e} = \cos^{-1}\left[\frac{h_{me}}{v_{me,e} R_e}\right]$$

Earth-referenced hyperbolic excess speed as spacecraft enters Earth's sphere of influence

$$v_{hyp,mee} = \left[v_{me,e}^{2} + V_{e}^{2} - 2v_{me,e} V_{e} \cos \phi_{me,e} \right]^{1/2}$$

Earth-referenced spacecraft speed at perigee altitude above Earth

$$v_{i,mee} = \left[v_{hyp,mee}^{2} + \frac{2\mu_{e}}{r_{e} + a_{e}} \right]^{1/2}$$

Velocity-added in circularizing into Earth parking orbit

$$v_{AE} = v_{i,mee} - v_{circ,e}$$

Sun-central angle by which Earth leads Mars at time spacecraft departs Mars

$$phase_{me} = mod(\beta_{me} - \omega_{e} T_{me}, 2\pi)$$

Figures C.5 through C.8 are identical in structure to Figs. C.1 through C.4, but deal with the return trajectory from Mars to Earth. As discussed in conjunction with the earlier set of figures, the Hohmann transfer return route, of about 260-days duration, corresponds to the lowest point on each of Figs. C.6, C.7, and C.8.

WAIT TIME ON MARS

Given the four basic input variables, $\phi_{em,e}$, β_{em}, $\phi_{me,m}$, and β_{me}, we have computed the four required velocity-added increments:

Leaving Earth	v_{LE}
Arriving Mars	v_{AM}
Leaving Mars	v_{LM}
Arriving Earth	v_{AE}

Furthermore, given these inputs, we have determined the required relative orbital phasings of Earth and Mars at the times the spacecraft leaves Earth and leaves Mars. Thus far, we have treated the Earth-to-Mars and Mars-to-Earth legs of the mission as independent. The connecting link and the final step in our brief mission analysis is to compute the required wait time in Mars parking orbit to ensure that the Earth-Mars phasing is correct for the Mars-to-Earth return flight. This wait time is given by

$$wait = mod\left(\frac{syn}{2\pi}\left(phase_{em} + phase_{me}\right) - \frac{T_{em}}{86400}, syn\right) \ days$$

with total mission time, in days, being $\frac{T_{em} + T_{me}}{86400} + wait$. The four velocity increments above, modified as necessary to account for aerobraking opportunities, serve as inputs to a model (see App. D) that calculates, for various mission scenarios and propulsion options, the required initial mass in Earth orbit.

To gain some feel for the dynamics of a typical round-trip mission, we present in Fig. C.9 a schematic of the geometry for that particular 616-day mission requiring the least total velocity-added, given that the wait time on Mars is between 25 and 65 days. Shown are the positions of Earth and Mars at the times of Earth departure, Mars arrival, Mars departure, and Earth arrival.

Figure C.10 addresses the relationship between mission duration, wait time at Mars, and velocity-added requirements; it shows how total required velocity-added[7] (namely, $v_{LE} + v_{AM} + v_{LM} + v_{AE}$) varies with total mission time for eight different wait-time windows. Not surprisingly, short mission durations require huge velocity commitments. The behavior of the curves, in particular their non-monotonicities, is governed by the stringent Earth-Mars phasing requirements necessary to achieve given wait times on Mars and total mission times. In light of the complex Earth-Mars- spacecraft geometry, as determined by the degrees of freedom provided by the four basic angular inputs, there is no reason to expect the curves to be monotone.

The two plots of Fig. C.11, drawn for a 45-day Mars wait time, shed light on the trade-offs between the velocity-added required for the outbound and inbound legs of the round trip. Both plots include 12 curves representing various relative positions of Earth and Mars at the time of Mars departure. Although the sun-central angle by which Earth leads Mars at the time of Mars departure can vary from -180 to 180 deg, we restrict attention here to angles between -60 and 50 deg, which correspond roughly to the so-called *opposition class* missions that are typically most velocity efficient for short wait times on Mars.[8]

Note, in particular, the reverse ordering of the curves on the two plots. Suppose, for example, that we are interested in a 500-day total mission duration. By combining

[7] In Figs. C.10 and C.11 and Tables C.2 and C.3, indicated velocity requirements are minimal over the full range of input parameters ($\phi_{em,e}$, β_{em}, $\phi_{me,m}$, β_{me}), subject to the 70-million-km perihelion constraint.

[8] Curves for 60 deg, 70 deg, . . . , continue the downward progression on the left plot and thus represent velocity-efficient Earth-to-Mars transits. When combined, however, with their counterparts on the right plot, they represent highly velocity-inefficient options for the round-trip.

appropriate data on the plots, we can derive minimum velocity-added requirements as a function of the indicated sun-central angle, as shown in Table C.2.[9]

Table C.2

Velocity Requirements for 500-Day Total Mission Duration

Sun-Central Angle (deg)	Minimum Required Total Velocity-Added (km/sec)	Outbound Flight Time (days)[a]	Inbound Flight Time (days)
-60	23.9	260	240
-50	23.0	270	230
-40	22.5	280	220
-30	22.3	280	220
-20	22.4	280	220
-10	22.5	280	220
0	22.7	280	220
10	22.7	270	230
20	22.6	270	230
30	22.5	260	240
40	22.4	270	230
50	22.3	270	230

[a]Includes 45-day wait time on Mars.

The minimum velocity requirement, about 22.3 km/sec, is precisely the number plotted on Fig. C.10 above 500 days on the curve for 25 to 65 days. Note that the Earth-Mars phasings (i.e., central angles) that correspond to low velocity-added requirements on the outbound leg correspond to high requirements on the inbound leg, and vice versa. It is not possible, given the geometry constraints imposed by the 45-day wait period and 500-day mission time, to select inbound and outbound flight profiles that provide near-minimal velocity requirements for each separate leg of the mission.

Table C.3 presents similar data for a 640-day total mission duration, which corresponds to a local maximum point on the curve for 25 to 65 days in Fig. C.10. The total velocity requirements are substantially greater than those for the 500-day wait time. This increase can be traced on Fig. C.11 to the rapid increase (in both plots) in velocity requirements for one-way trip times between 300 and 400 days.[10] As total mission time is increased from 640 days, the total velocity-added requirement begins to decrease, as the opportunity then exists to exploit the decreasing one-way velocity requirements shown in the upper right corner of each plot.

[9]Small fluctuations in the velocity and flight time columns of Table C.2 and the next table are due primarily to "grain effects" of the computer simulation and are not significant.

[10]Note that between 400 and 450 days on either plot, one-way velocity requirements exceed 24 km/sec for the range of central angles presented. The curves all "peak" somewhere above 24 and work their way back down, as shown in the upper right corner of each plot.

Table C.3

Velocity Requirements for 640-Day Total Mission Duration

Sun-Central Angle (deg)	Minimum Required Total Velocity-Added (km/sec)	Outbound Flight Time (days)[a]	Inbound Flight Time (days)
-60	26.5	500	140
-50	26.4	510	130
-40	26.9	520	120
-30	27.0	310	330
-20	26.8	310	330
-10	26.7	320	320
0	26.1	330	310
10	26.4	330	310
20	26.2	350	290
30	26.3	360	280
40	26.4	360	280
50	26.9	370	270

[a]Includes 45-day wait time on Mars.

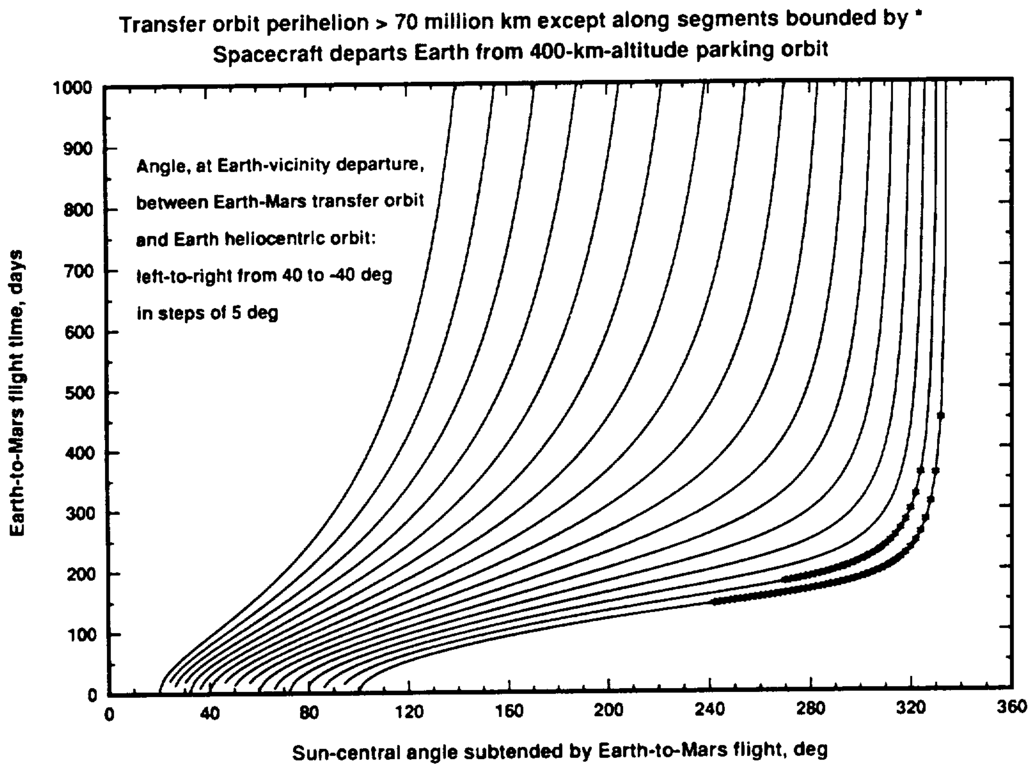


Fig. C.1—Earth-to-Mars Flight Time

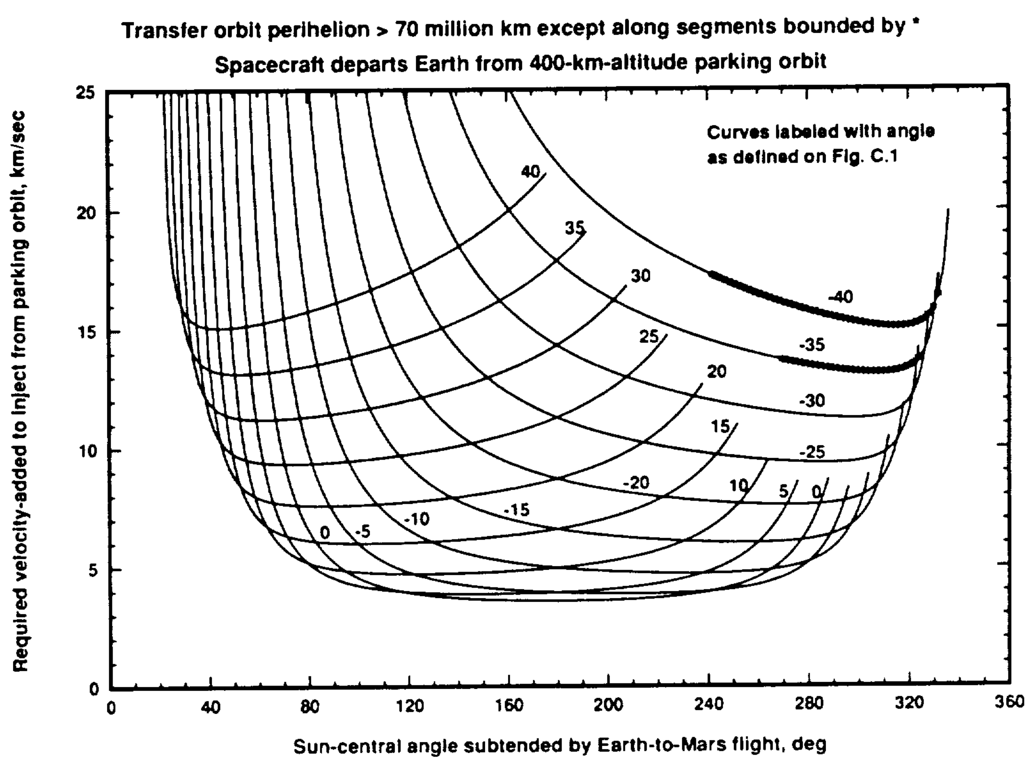


Fig. C.2—Required Velocity-Added: Earth Orbit to Earth-Mars Transfer Orbit

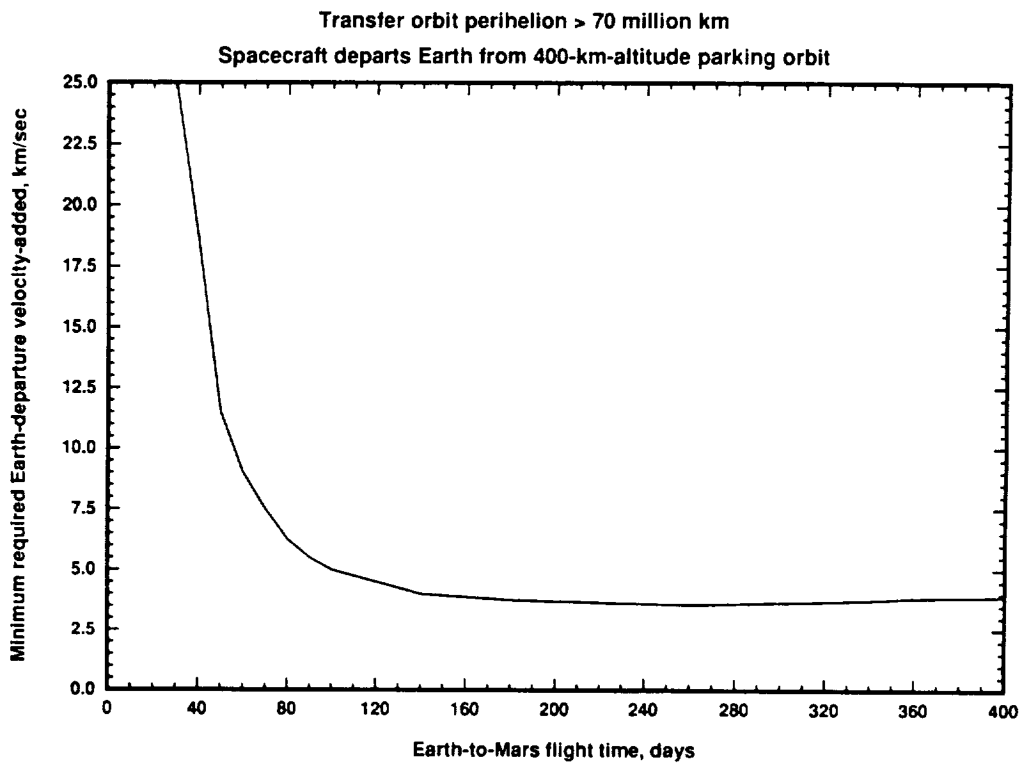


Fig. C.3—Earth-to-Mars Flight Time Versus Required Earth-Departure Velocity-Added

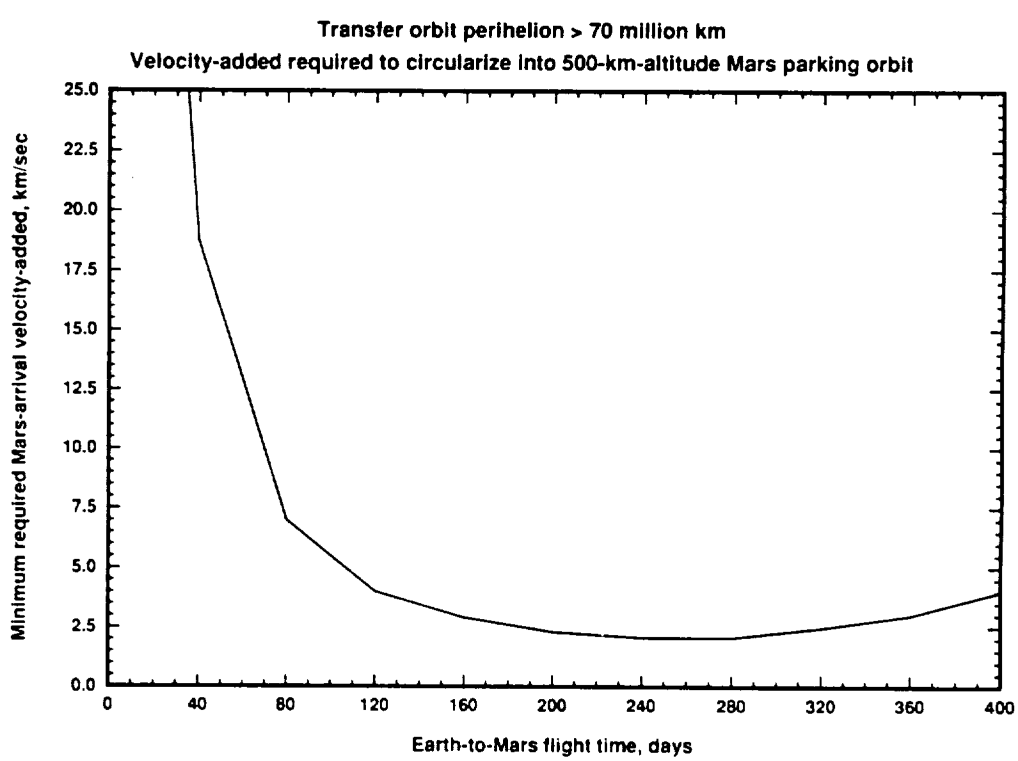


Fig. C.4—Earth-to-Mars Flight Time Versus Mars-Arrival Velocity-Added

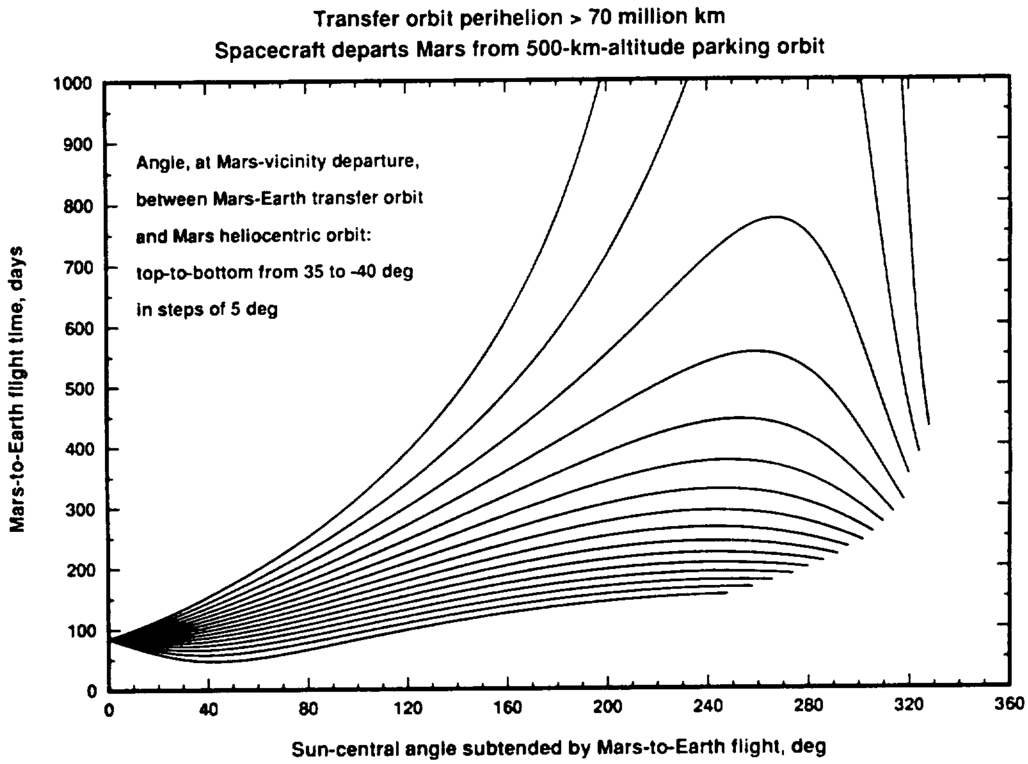


Fig. C.5—Mars-to-Earth Flight Time

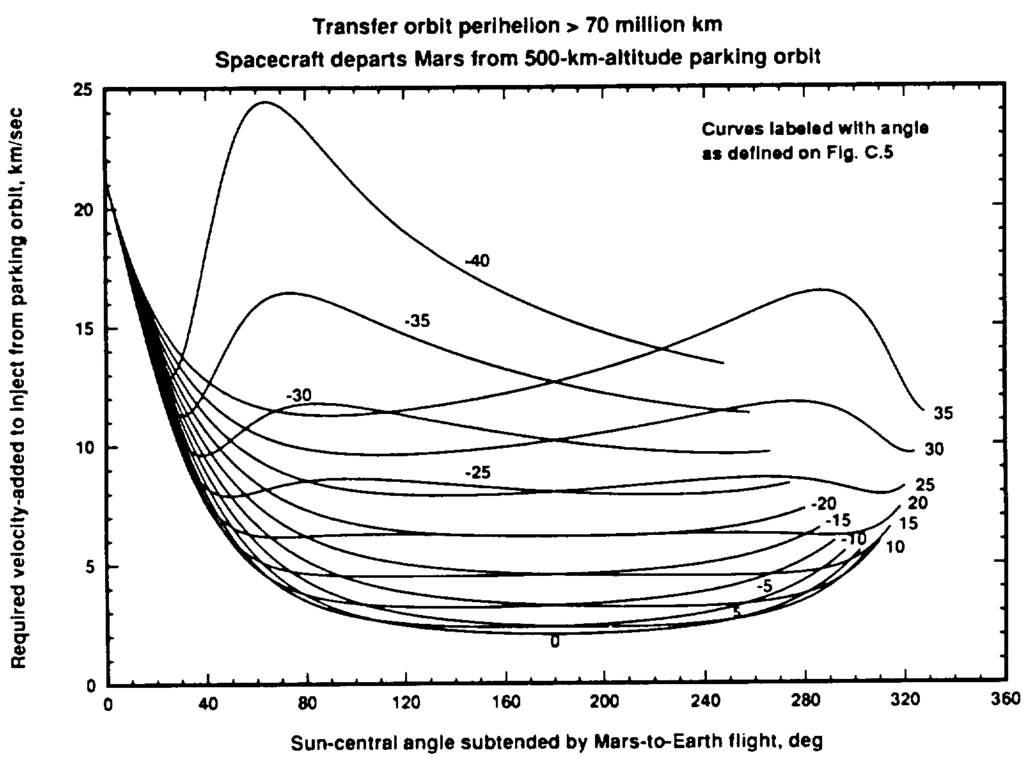


Fig. C.6—Required Velocity-Added: Mars Orbit to Mars-Earth Transfer Orbit

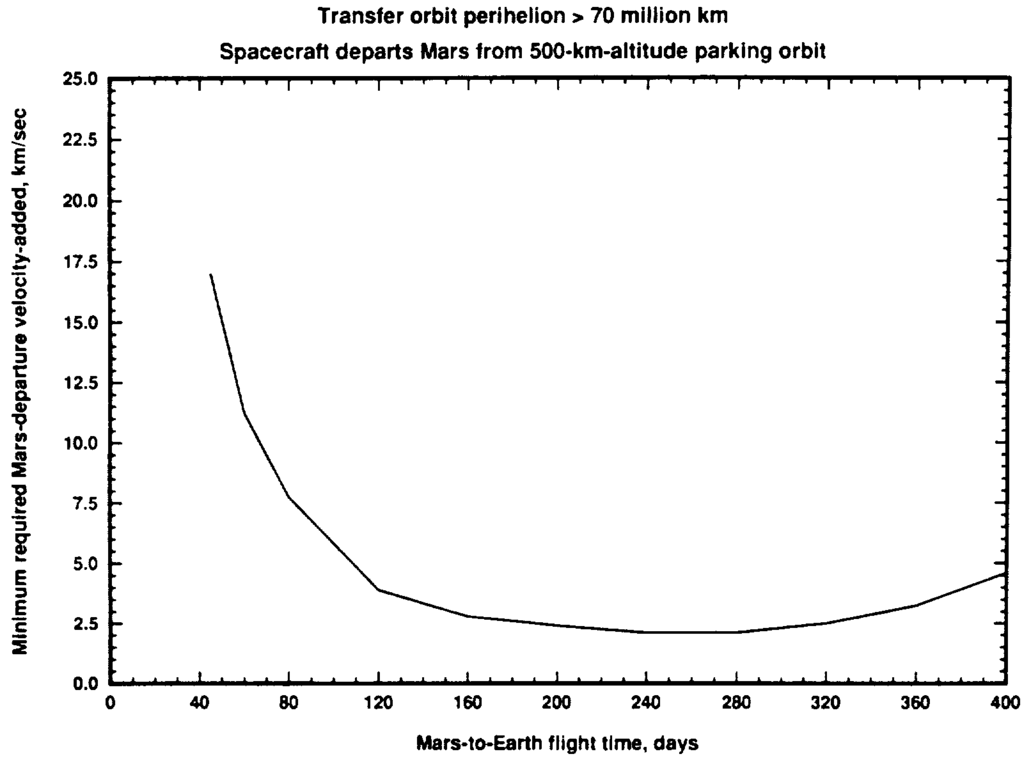


Fig. C.7—Mars-to-Earth Flight Time Versus Required Mars-Departure Velocity-Added

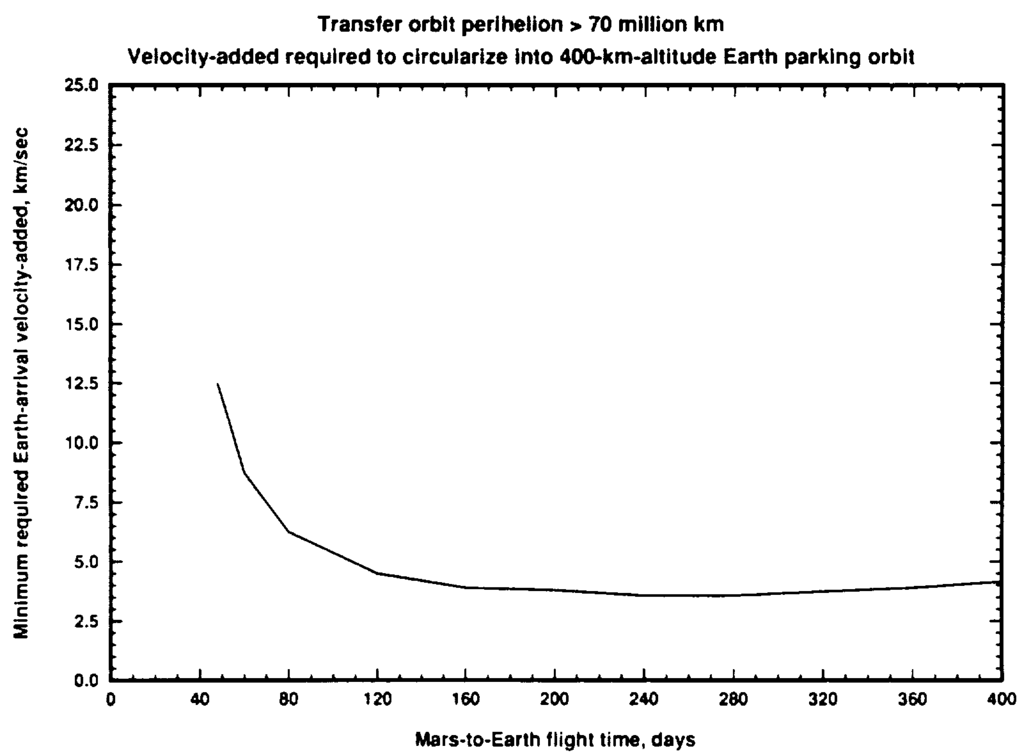


Fig. C.8—Mars-to-Earth Flight Time Versus Earth-Arrival Velocity-Added

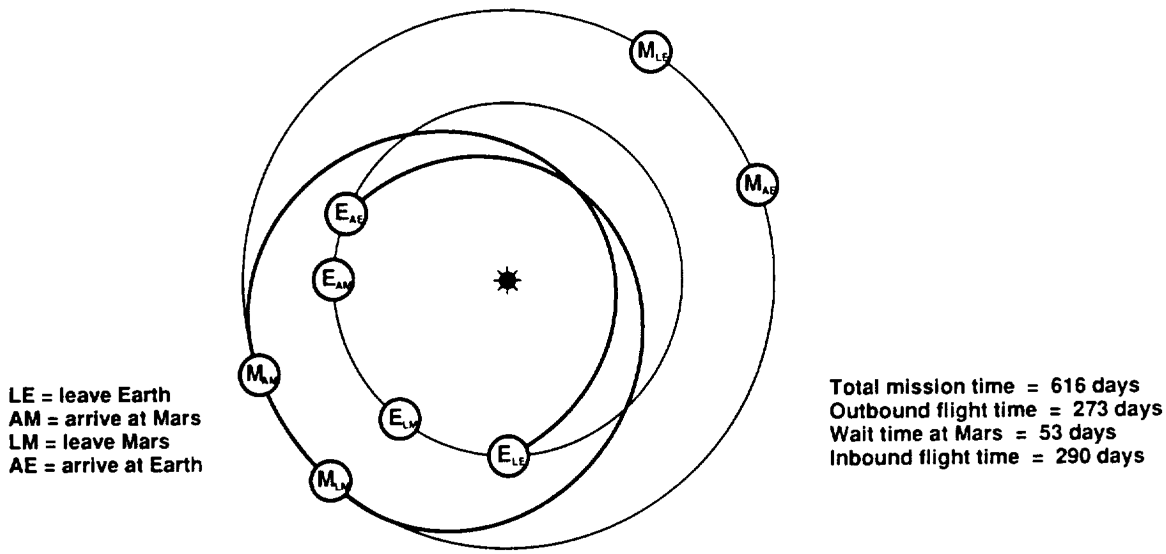


Fig. C.9—Geometry for Sample Round-Trip Mars Mission

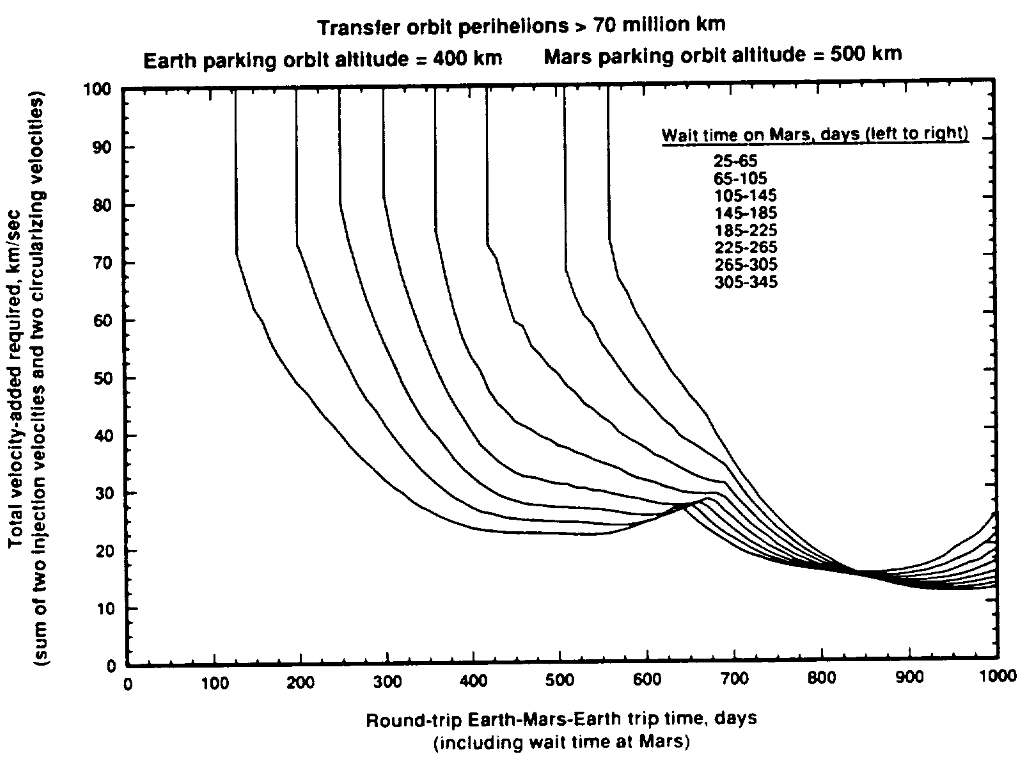


Fig. C.10—Velocity-Added Requirements for Round-Trip Mars Mission

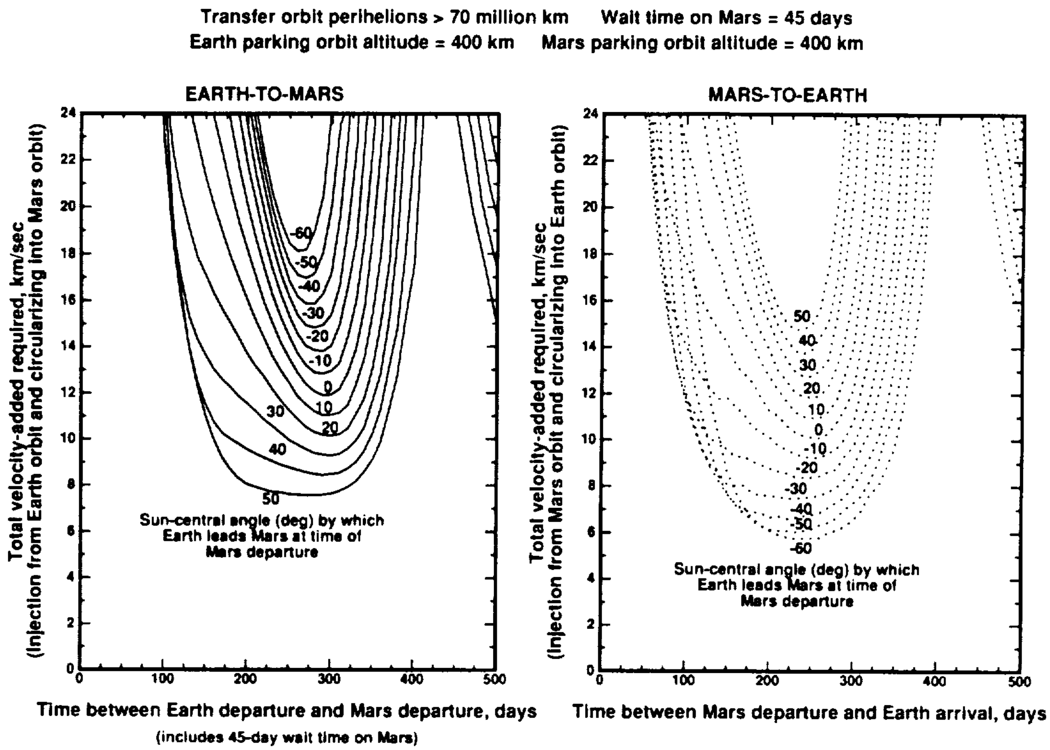


Fig. C.11—Velocity-Added Requirements Versus Earth-Mars Phasing

Appendix D
VEHICLE MASS DETERMINATION

Once the Earth departure velocity, the Mars approach velocity, the Mars departure velocity, and the Earth approach velocity are available from the patched conic program discussed in App. C, it is possible to calculate the propulsion requirements for each of the four flight phases. Working backward from Earth arrival to Earth departure, we can calculate the vehicle mass at the beginning of each flight phase.

On approaching Earth, three options should be considered. For the first option, the propulsion system brakes the vehicle speed until the desired orbital velocity about the Earth is achieved. The propulsion delta V required is thus the vehicle approach velocity minus the desired orbital velocity. For the second and third options, the vehicle is propulsion braked down to either 14 or 12.5 km/sec. The former value represents the maximum velocity with which the ECCV can reenter the Earth's atmosphere. The second value, 12.5 km/sec, is taken to be the maximum velocity with which a vehicle can use an aerobrake to be aerocaptured into Earth orbit. The propulsion delta Vs for these two options are Earth approach velocity minus 14 km/sec and Earth approach velocity minus 12.5 km/sec.

The mass that must be braked upon Earth arrival is a function of the braking option selected. With propulsion braking, the braked mass consists of the crew habitation module plus the science payload. For the ECCV Earth-return option, the braked mass is just that of the ECCV. In the case of the last option, the braked mass is the crew habitation module, the science payload, and the aerobrake.

In general, let the mass that is to be braked be designated by m_{BE}. Then the total vehicle mass immediately prior to braking is

$$M_{EB} = m_{BE} + m_{PS} + m_P + m_{PT} \tag{1}$$

where m_{PS} is the mass of the propulsion system, m_P is the mass of the propellant, and m_{PT} is the mass of the propellant tank and any necessary refrigeration equipment. The masses m_{PS}, m_P, and m_{PT} can be expressed as functions of M_{EB} by defining the following parameters: propellant fraction, system T/W, propulsion system T/W, propellant tank-structural fraction, and refrigeration system fraction.

For a chemically propelled spacecraft, Eq. (1) can be written as follows:

$$M_{EB} = G_{EB} m_{BE} \tag{2}$$

where

$$G_{EB} = \left[\frac{1}{1 - f_{EB} - \dfrac{\left(k_{EB} f_{EB} \lambda_{EB} + K_{EB}\right)}{\lambda_{EB}\left(1 - c_{EB}\right)}} \right]^{N_{EB}} \tag{3}$$

In Eq. (3), N_{EB} is the number of propulsion stages used in Earth braking, f_{EB} is the stage propellant fraction, k_{EB} is the propellant tank fraction, λ_{EB} is the propulsion system T/W, K_{EB} is the initial stage T/W, and c_{EB} is the percentage of the dry stage mass that represents the refrigeration system for the cryogenic propellants. It is assumed that these parameters are the same for all N stages.

Assuming impulsive burning, the delta V required for braking is

$$V_{EA} - V_{OP} = g_0 \left(I_{sp}\right)_{EB} N_{EB} \log_e \left(\frac{1}{1 - f_{EB}} \right) \tag{4}$$

where V_{EA} is the Earth approach velocity provided by the patched conic program, V_{OP} is the velocity associated with one of the three braking options discussed above, g_0 is the acceleration of gravity (9.81 m/sec), and $\left(I_{sp}\right)_{EB}$ is the specific impulse in seconds.[1] It is assumed that the I_{sp} is the same for all N stages.

With all of the parameters assigned values except the propellant fraction, Eq. (4) can be used to determine f_{EB}. If f_{EB} exceeds some specified limit, then N_{EB}, the input number of stages, is increased by one and f_{EB} is recalculated. Once f_{EB} is known, G_{EB} can be calculated from Eq. (3). If G_{EB} is negative, then N_{EB} is increased by one and f_{EB} is determined once more from Eq. (4). Finally, M_{EB} is determined from Eq. (2).

Working backward to Mars departure, the same procedure is followed as in the case of Earth braking. The patched conic program provides the value of the velocity

[1]If V_{EA} is less than V_{OP}, propulsion braking is not required.

needed to depart from the Martian system. Assuming no gravity losses, the propulsion delta V required is V_{MD} minus V_{OM}, where V_{MD} is the Mars departure velocity and V_{OM} is the velocity of the vehicle in orbit about Mars. Thus,

$$V_{MD} - V_{OM} = g_0 \left(I_{sp} \right)_{MD} N_{MD} \log_e \left(\frac{1}{1 - f_{MD}} \right) \tag{5}$$

With f_{MD} determined, the total mass just prior to Mars departure can be found.

$$M_{MD} = G_{MD} \left(M_{EB} + m_{MD} \right) \tag{6}$$

The parameter m_{MD} is the mass that leaves Mars orbit but is either consumed or discarded prior to Earth braking (excluding the Mars departure propulsion system and propellant). The parameter G_{MD} is

$$G_{MD} = \left[\frac{1}{1 - f_{MD} - \dfrac{\left(k_{MD} f_{MD} \lambda_{MD} + K_{MD} \right)}{\lambda_{MD} \left(1 - c_{MD} \right)}} \right]^{N_{MD}} \tag{7}$$

where, as before, the parameters f_{MD}, k_{MD}, λ_{MD}, K_{MD}, and c_{MD} have the same value for all stages. With f_{MD} determined from Eq. (5), G_{MD} and then M_{MD} can be found.

The same procedures described for Earth braking and Mars departure are followed for Mars arrival and Earth departure. Thus,

$$V_{MA} - V_{OP} = g_0 \left(I_{sp} \right)_{MA} N_{MA} \log_e \left(\frac{1}{1 - f_{MA}} \right) \tag{8}$$

where, as in the case of Earth arrival, the value of the velocity V_{OP} depends upon the type of vehicle braking employed. If the vehicle is propulsion braked into Mars orbit, V_{OP} is the orbital velocity. On the other hand, if an aerobrake is used to aerocapture the vehicle into orbit, V_{OP} is equal to 9.5 km/sec, the maximum value that is assumed feasible for Mars capture. As in the case of Earth arrival, if V_{MA} is less than V_{OP}, propulsion braking is not required. The mass to be braked at Mars arrival depends upon the braking option. If only propulsion is used, then the mass is M_{MD}, plus any mass that is consumed between Mars arrival and Mars departure, plus any mass that is left on Mars or in orbit about Mars, such as the MEV. With propulsion or propulsion plus aerobraking, the mass is M_{MD}, plus the consumables, plus the MEV and its aerobrake. Thus,

$$M_{MB} = G_{MB}(M_{MD} + m_{BM}) \tag{9a}$$

or

$$M_{MB} = G_{MB}(M_{MD} + m_{BM} + m_{AB}) \tag{9b}$$

It is assumed that the MTV aerobrake mass is 15 percent of the mass to be braked:

$$m_{AB} = 0.15(M_{MD} + m_{BM})$$

where m_{BM} is the mass consumed or left behind before Mars departure.[2]

Finally, at Earth departure

$$V_{ED} - V_{OE} = g_0(I_{sp})_{ED} N_{ED} \log_e\left(\frac{1}{1 - f_{ED}}\right) - V_L \tag{10}$$

where V_{OE} is the vehicle orbital velocity and V_L is the gravity loss

$$V_L = \frac{\mu}{24 r_0^3} \Delta V t_P^2 \left[1 - \left(\frac{\mu}{r_0\left(V_{OE} + \Delta V^2\right)}\right)\right] \tag{11}$$

In Eq. (11), μ is the Earth's gravitational constant, ΔV is V_{ED} minus V_{OE}, r_0 is the orbital radius distance, and t_P is the propulsion time. This gravity loss approximation is from Robins (1966).

At Earth departure the mass that leaves for Mars is M_{MB}, plus mass that is consumed prior to Mars arrival, plus any mass that is discarded en route. Thus,

$$M_{ED} = G_{ED}(M_{MB} + m_{ED}) \tag{12}$$

where m_{ED} is the mass consumed or discarded en route to Mars.

A nuclear propelled vehicle will most likely use a single propulsion system for all flight phases, although empty propellant tanks will be discarded. (It is assumed that a nuclear vehicle is normally recovered in Earth orbit.) This requires that the equations for chemical systems be slightly modified.

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[2]With aerobraking, the mass of the MEV and its aerobrake are included in m_{BM}.

Working backward, as before, from Earth arrival, the nuclear vehicle mass at Earth braking can be expressed as

$$M_{EB} = \frac{G_{EB}\left(m_H + m_{REF}\right)}{1 - \left(1 + \alpha\right)\frac{k_{EB}}{\lambda} G_{EB}} \tag{13}$$

where m_H is the habitation mass, or the habitation plus aerobrake mass, and m_{REF} is a mass associated with the reference propulsion system. The constant α determines how shielding and support structure mass vary as the propulsion reactor mass m_{RD} increases or decreases. Thus, m_{PS}, the propulsion mass system, is

$$m_{PS} = m_{ref} + (1+\alpha)m_R \tag{14}$$

The constant G_{EB} is

$$G_{EB} = \left[\frac{1}{1 - f_{EB} - \frac{\left(k_{EB}f_{EB}\right)}{\left(1 - c_{EB}\right)}}\right]^{N_{EB}} \tag{15}$$

where the various parameters are defined as before. For M_{EB} to be nonnegative, it is necessary that

$$f_{EB} < \left(\frac{1 - c_{EB}}{1 - c_{EB} + k_{EB}}\right)\left\{1 - \left[\left(1 + \alpha\right)\frac{K_{EB}}{\lambda}\right]^{\frac{1}{N_{EB}}}\right\} \tag{16}$$

As before, the propellant fraction, f_{EB}, is determined once the delta V required for Earth braking is defined (Eq. (4)).

The expression for M_{MD}, the mass just prior to Mars departure, is given by Eq. (6) where G_{MD} is now

$$G_{MD} = \left[\frac{1}{\frac{1}{1 - f_{MD}} - \frac{k_{MD}f_{MD}}{\left(1 - c_{MD}\right)}}\right] \tag{17}$$

The remaining steps for finding the mass at Earth departure, M_{ED}, are the same as described for the chemically propelled vehicle, with the G at each phase having the same algebraic form as G_{MD}.

Appendix E

LUNAR-DERIVED PROPELLANTS
(Submission #100932)

The submissions aggregated under the Lunar-derived propellant category discuss the applicability of Lunar materials for chemical rocket engines. The existence of oxygen, magnesium, aluminum, and possibly very small quantities of hydrogen (more if polar ice exists) in the Lunar soil is well documented. These submissions advocate investigating the feasibility of developing a chemical rocket engine that could operate exclusively on materials available from the Lunar surface. The potential payoff of such an engine is great because of the relatively small gravity well of the Moon compared to that of the Earth. The I_{sp} of a magnesium/aluminum/hydrogen/oxygen engine is expected to be in the range of 300 to 450 sec depending on the fuel ratio. An I_{sp} of 314 sec is expected using an engine fueled by aluminum and oxygen only. Although this is a relatively low I_{sp}, a great benefit arises from the fact that all of the propellant is available from the Lunar surface. The problems of finding and processing a large quantity of hydrogen on the Moon, or transporting it from the Earth, are eliminated. In addition, an aluminum/oxygen engine is less complex than a tripropellant engine. This performance is acceptable for Earth-Moon transportation, Mars cargo transportation, and possibly even crew transportation. A possible scenario using this concept would be an EML on the Lunar surface to put the LOX and Al in LLO, and the use of SEPs to transport the assembled space vehicle from LEO to L_2 and the propellant from LLO to L_2. The crew would then be transported to L_2 and the vehicle would depart for Mars.

With the addition of liquid hydrogen, I_{sp} may go as high as 475 sec. However, many technical problems exist with this engine design:

- Combustion stability and ignition are very uncertain
- Plumbing and injection of the fuel into the combustion chamber
- Performance
- Reliability

A substantial research effort must be undertaken to solve these problems. A more in-depth discussion of tripropellants is presented in Sec. II of this Note.

In addition, many technical challenges are associated with developing the capability of mining and processing materials on the Moon. The infrastructure required to produce suitable propellants would be extensive.

This concept does not appear to be useful for the 2019 missions; however, mission analyses must be performed to determine the applicability of this concept. A very desirable characteristic of this approach is that permanent structures are developed that make many space missions possible and more feasible.

Appendix F

LUNATRON—LUNAR SURFACE-BASED ELECTROMAGNETIC LAUNCHER (SUBMISSION #100575)

This submission proposes an EML that accelerates payloads in a guideway along the surface of the Moon. This is not a new concept. Electromagnetic accelerators, in general, were proposed before the turn of the century, while the specific application to a Lunar launch system was proposed by Arthur C. Clarke in a paper published by the Journal of the British Interplanetary Society in 1950.

The submission is based on a paper written by the author in the mid-1960s and does not include any material indicating the progress that has been made in the last 25 years. Nevertheless, the proposal is sound and should receive consideration as a potential Lunar launch system.

The proposed launcher would employ a linear electric motor in which the stator is fixed to the Lunar surface in a guideway. A carriage, to which the payload is attached, contains the polyphase stator coils that are placed closely adjacent to the vertical rail that constitutes the rotor. This nonferrous rail bears the forces imposed by the carriage and its payload. Bus-bar sets carry the primary polyphase power, with the rail being electrically grounded.

It is in the area of power generation and control that the submission is very nebulous and it is also in this area that great strides have been made during the past 25 years. Specifically, the development of high-power/high-current homopolar generators, compensated alternators (compulsators), and high-energy, low-weight/volume capacitors, in combination with high-speed sensing and computer control, makes possible the implementation of practical maglev devices.[1] Currently maglev transportation systems are close to commercial operation in both Germany and Japan.

As envisioned in the submission, the Lunar EML system would launch full-size spacecraft, both unmanned and manned. For manned launches, centripetal acceleration would impose an upper limit to the launch velocity. For a 9–10 g upper limit, the launch velocity relative to the Moon would be about 12 km/sec. In

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[1] The submission proposed the use of gas bearings to support the carriage, but magnetic suspension would be a desirable alternative.

heliocentric coordinates, the velocity would be about 42 km/sec, which corresponds to the solar system escape speed.

Initially, a much more modest launch system would be useful as a means of placing oxygen in Lunar orbit, as discussed in Sec. III. Such a system would also serve as a test-bed, providing both validation of the design and operational experience that could lead to EMLs with much greater performance.

Appendix G

IN-SITU PROPELLANTS FOR MARS LANDER—CHEMICAL ENGINES (SUBMISSION #101178)

The submissions aggregated under this category discuss the possibility of using materials available on Mars as propellant for chemical rocket engines. The Martian atmosphere is composed almost entirely of carbon dioxide (nearly 96 percent). The existence of polar water ice on Mars is also possible. Further, it is likely that the Martian moons, Phobos and Deimos, contain water.

In addition to the great amount of research and development that will be required to develop the expertise required to process material on extraterrestrial bodies, a great deal of infrastructure will be required before in-situ propellants can be produced.

If hydrogen is not available in the Martian system, propellant for a LOX/LCO engine (which would have an I_{sp} of around 270 sec) could be produced entirely from the Martian atmosphere. In addition, carbon monoxide is relatively easy to liquify and store. This propellant combination could be used for surface transportation and possibly to achieve orbit/deorbit. However, it is unlikely that a LOX/LCO engine would be suitable for TEI.

If hydrogen is available in the Martian system (or transported from Earth), a rocket engine could be used to run on LOX/LH_2 or LOX/CH_4. The LOX/LH_2 engine offers an I_{sp} of ■ 480 sec; however, the cryogenics storage requirement of liquid hydrogen is a substantial burden for long space missions. Methane is far more suitable as a propellant for space missions because of its relative ease of storage. In addition, a LOX/CH_4 engine has very respectable performance (I_{sp} = 340 sec). A LOX/CH_4 engine may even be considered for TEI.

The various methods that are available to split carbon dioxide to produce oxygen and carbon monoxide are discussed in Sec. II along with various storage alternatives.

Appendix H

THE PONY EXPRESS TO MARS (SUBMISSION #100714)

This submission proposes a split-mission transportation option that involves three SEP vehicles plus a LOX/LH_2-powered manned MTV. One of the SEP vehicles transports the Mars descent and ascent vehicles to Martian orbit while the second places the TEI stage into orbit. The third SEP vehicle goes into a heliocentric orbit, carrying a propulsion stage for Mars orbit insertion. The MTV rendezvouses with this SEP en route to Mars and docks with the orbital insertion propulsion stage. On arrival in Martian orbit, the MTV rendezvous and docks with the ascent/descent vehicles. On leaving the Martian surface, the three-man crew docks with the trans-Earth stage and departs. Earth return is by ECCV.

A total IMLEO of 690 metric tons is required for this mission. The mass breakdown and assumed performance parameters presented in the submission appear to support this value. The various system masses, however, are *much* lower than those assumed in the 90-Day Study. As described, it is essentially a "flags and footprints" mission. Other than a low value of IMLEO, the virtue of this approach is a total mission duration of 330 days—100 days to Mars, 30 days on Mars, and 200 days back to Earth. Although using rendezvous and docking en route to Mars helps reduce IMLEO, it is doubtful that the operational risk entailed makes it worthwhile. It is not clear how wide the launch window must be to ensure that the manned vehicle can rendezvous with the SEP vehicle in a heliocentric orbit.

Appendix I

A SOLAR SAIL DESIGN FOR SPACE TRANSPORTATION AND POWER BEAMING (SUBMISSION #101016)

This submission proposes a new type of solar sail design that would greatly simplify the construction, packaging, and deployment of the spacecraft. The design was selected as one of the winners for the Columbus 500 Space Sail Cup.

The novel aspect of the design is the manner in which the sail is folded, packaged, and unfolded. In this approach, the initial folds wrap around others but are not folded or creased again. The structural supports for the sail would be provided by ribs extending along the folds. Thus, the sail and supporting structure are deployed in a single unaided operation. The sail can also be refurled, which permits docking with other spacecraft.

The sail as currently designed is quite small, with an area of 0.06 km^2. The basic concept, however, should lend itself to being scaled to a size that could carry 20 to 30 metric tons.

One very important parameter, the areal density, is not provided in the submission. Based on the fact that multiple folds are not required for packaging, an areal density of $5X10^{-3}$ kg/km or smaller should be attainable.

Appendix J

EARTH-BASED MICROWAVE POWER BEAMING TO INTERORBITAL (LEO TO AND FROM HEO) ELECTRICALLY PROPELLED TRANSPORT VEHICLES (SUBMISSION #101536)

This submission proposes the development of an orbital transportation system that employs microwave-powered OTVs (see Beamed Energy, Sec. II). A baseline system with one high-power (60 MW) ground transmitter would be capable of transferring a 60,000-kg payload from LEO to GEO in about 100 days. An expanded system, using four transmitters, could place the same payload into GEO in 25 to 30 days.

The four large transmitting arrays would, ideally, be equally spaced along the equator so that, as the OTV spirals out from its initial 400 km orbit, it is irradiated by the four transmitters in succession. Each of the transmitting arrays would have an area of 2 km^2 and would use $2X10^6$ identical modules.

The OTV would have a large Rectenna that would provide 500 50-cm ion thrusters with 20 MW of DC power. The thrust of the vehicle would be 750 N.

The technology associated with the microwave components is mature. Lightweight radiation-hardened Rectennas have been developed that have a specific mass of 1 kg/kW. This value is lower by a factor of four to six than the specific mass of solar arrays.

This concept is not new, but SEI Lunar and Mars missions will almost certainly require an efficient orbital transportation system, and microwave beamed-energy OTVs should be prime contenders.

Appendix K

SOLAR THERMAL ROCKET SYSTEM FOR ORBITAL/INJECTION TRANSFER VEHICLE (SUBMISSION #101399)

This submission proposes an STP rocket that would use a volumetric absorber rather than a heat exchanger to convert focused solar energy into reaction jet energy. Fine carbon particles would be injected into the propellant stream (hydrogen), where they absorb energy from the solar radiation and, in turn, heat the hydrogen. An I_{sp} of 1500 sec is claimed, but AF Astronautics Lab studies indicate that 1200 sec is a more likely value. The fact that adding carbon particles to the hydrogen increases the average molecular weight and thus decreases I_{sp} is ignored.

The novel aspect of this submission, as compared to current work at the Astronautics Lab, is the proposal to use solar sail technology to construct very large, lightweight solar concentrators. This approach would yield very high specific powers—in the range of 50 to 100 kW/kg. The problems associated with maintaining the proper curvature of such a large, flexible surface are not addressed.

An STP cargo vehicle, based on the Astronautics Lab design discussed earlier, requires an IMLEO of 288 metric tons to deliver about 36 metric tons to Mars orbit in 370 days. An SEP cargo carrier with an IMLEO of 85.5 metric tons can deliver a payload of about 40.5 metric tons to Mars orbit in 398 days.[1] The inferior performance of the STP is due to its relatively low I_{sp} in combination with a mass penalty stemming from the need to refrigerate large quantities of H_2 propellant. The propellant tanks and cooling system of the STP cargo vehicle have a total mass of about 40 metric tons. Thus, it appears that STPs are more suited for orbit transfer applications in the Earth-Moon system than they are for Mars cargo missions.

[1] Low-thrust trajectory data from Frisbee et al. (1989) were used to perform these calculations.

Appendix L
PULSED MPD ELECTRIC PROPULSION (SUBMISSION #100170)

The efficiencies and thruster lifetimes of current experimental MPD devices are such that without substantial improvements in both, MPD propulsion will have very limited application (see Low-Thrust Propulsion Technologies, Sec. II). The concept proposed in this submission has the potential of increasing the efficiency of MPD thrusters by operating in a pulsed mode. The difference between the current proposal and what has been done in the past is the width of the current pulse used—tens of microseconds—and the frequency of pulsing, which is in the kilohertz range.

The mean power of an electric thruster is established by the power source. By operating in a pulsed mode, the peak power of the thruster is raised. Experimental research data indicate that MPD performance increases with increasing power.

The following example from the submission illustrates the concept. For a manned Mars mission, a 10 MW power source would provide 2 MW mean power to each of five thrusters. The peak thruster power would be 20 MW. For a typical pulse rate of 10 kHz, the pulse width would be 10 μsec with an energy of 20 J/pulse. Short pulse widths are essential to prevent electrode melting.

The performance goal of this concept is to achieve an MPD efficiency of 60 percent in combination with an I_{sp} of 5000 sec. This is equivalent to a thrust-to-power ratio of 24 N/MW.

MPD thrusters can operate with a wide range of propellants and, as compared to ion thrusters, the combined power conditioning thruster specific mass is low—less than 2 kg/kW.

Even if the performance goals can be met, it is necessary to demonstrate that low electrode erosion rates can be achieved at high powers. Thruster lifetimes in excess of 5000 hr would be desirable for SEI missions.

From the viewpoint of actual use (assuming the concept proves successful), the development of either solar or nuclear space power sources in the 10 MW range is probably the pacing item.[1]

[1] Studies by JPL and others have shown that multimegawatt electric propulsion systems are very attractive in terms of IMLEO and can achieve trip times comparable to, or less than, those of LOX/LH_2 systems.

Appendix M

THE "ENABLER," A NUCLEAR THERMAL PROPULSION (NTP) SYSTEM (SUBMISSION #100933)

This submission proposes to build on the technology developed in the NTP ROVER/NERVA program, with updating to include technology advances initiated in the latter part of that program and incorporation of more modern safety (and environmental) concerns. There is no discussion of the issue of changing public policy to support full-scale development and operational use of nuclear engines with a high investment of fissile material.

Overall, this is rated as a highly promising submission based on the technical and engineering merits of the proposal. Other aspects of the proposal are also discussed.

TECHNICAL PLAN

The ENABLER proposal for NTP development reflects the state of the art attained in the ROVER/NERVA program embellished by additional data on radiation damage phenomenology and thrust chamber design insights in modern chemical rocket programs such as SSME. In this sense, the technology choices are generally conservative. Changes, such as the improvement of nonnuclear components and increases in nozzle area ratio to 500:1, are incremental and can be fully demonstrated in the engine development program.

The proposal wisely (in our view) steers clear of such additional embellishments as emphasis on particle bed reactors for its main initial thrust. Such embellishments could introduce added material and safety problems not warranted by the modest further improvements in T/W and very slight I_{sp} increases.

The proposal notes dual-mode possibilities (thrust and electric power provision) derivable from the basic engine design. The proposal should be expanded to investigate low pressure (and low T/W) operation, where I_{sp} increases of ~20 to 35 percent might be achieved.

SCHEDULE PLAN

The proposal suggests an eight-year program to reach the goal of a full-scale engineering (FSE) test. It is not clear that the proposed FSE test is equivalent to a flight-rated prototype (FRP) test. If that equivalence is the case, an eight-year schedule is challenging. An approximate ten-year schedule for an FRP test is still possible but would require a highly dedicated team and very careful attention to anticipating and planning for the highly focused scrutiny on environment, safety, and public policy issues an NTP program would be guaranteed to elicit.

Even a ten-year schedule would be very tight for an FRP test. Additionally, there remain interfaces to consider carefully between FRP testing and full flight qualification (FFQ). The latter would need to include definitive calibrated analysis of reliability, durability, availability, and operational envelope achievement. To achieve this level will require a great deal of testing. Because of all these questions, and definitional ambiguities, the eight-year schedule proposed should be very carefully reevaluated with the FFQ objective in mind.

COST PLAN

The proposal suggests the development costs shown in Table M.1. The source of these estimates is not defined, except to say that these costs are consistent with a similar estimate provided to the NASA NTP Workshop in July 1990, and that a large demonstrated technology base is drawn on. Our view is that these costs are low by a factor of, minimally, three to five, with the high end of the cost range likely. This view is based on achieving FRP/FFQ status.

Table M.1

Development Costs of the ENABLER Nuclear Thermal Rocket ($ in millions)

Reactor development and design	$350
Engine development and design	150
Procure, assemble for full-scale test	100
Facility preparation	125
Test costs	30
Total	$755

A comparison can be made with the proposed ALS main engine development. This LOX/hydrogen engine also has a very large, more conventional-technology, prior-development base to build upon. The cost of an eight-year program to prototype/demonstrate the ALS engine is nevertheless estimated as ~$1.07 billion. This cost does not account for main portions of the facility and test costs, which result, when included in total development costs for the ALS main engine, in a total bill of ~\$1.67 billion. These ALS costs still do not include the equivalent of the reactor development and design costs and added facility and test costs to handle nuclear devices in a manner that might be acceptable to contemporary society. To bring a nuclear rocket to the level of FFQ demonstration will require order-of-magnitude increases in test costs alone.

The estimated development costs for the ENABLER are therefore very likely grossly understated. The engineering promise of the ENABLER warrants a substantially more realistic cost estimate. The costs of new chemical counterparts are such that citing much higher realistic ENABLER costs should add to the credibility of an ENABLER development program.

Appendix N

NIMF CONCEPT TO ENABLE GLOBAL MOBILITY ON MARS (SUBMISSION #100103)

SUMMARY

This submission discusses use of locally available volatiles (from in-situ sources directly or manufactured using in-situ components) in a nuclear propulsion system for exploration trips. The proposal exploits the fact that NTP rockets can use essentially any propellant (at the expense of reduced I_{sp} if hydrogen is not used), with special attention to NTP fuel element protection. The concept is very interesting and worth substantial RDT&E effort (for both the rocket and in-situ propellant production).

However, additional investigations on other possibilities particularly suited to SEI needs are indicated to perform some global tradeoff and priority studies. One such tradeoff, which gives a product far more widely usable in many applications, is noted.

TECHNICAL DESCRIPTION

Zubrin considers as candidate propellants those shown in Table N.1, with the I_{sp} at 2800K noted (arguments are given that operating temperatures as high as 3500K for NTP might be obtained).

Table N.1

Specific Impulses of Various Propellants

	Propellant				
	CO_2	Water	Methane	CO/N_2	Argon
I_{sp}	283	370	606	253	165

All but the methane case are well within the range of chemical propulsion systems; however, the NTP can use any of these candidates.

The paper next discusses energy cost issues. Producing CO_2 from the Mars atmosphere is presumably the cheapest option—CO_2 is 95 percent of the atmosphere and can be obtained in liquid form by compression at Martian environment temperatures. Compression would use a very small fraction of the NTP reactor power in electrical form, so the NTP system could essentially fuel itself on the Mars surface.

Other fuel candidates are discussed also. Water might be harvested from permafrost, but the operation would be more complex than CO_2 compression. Methane and oxygen could be produced, in the presence of water, by additional CO and CO_2 reactions, requiring still more complex production processes. The other cited propellants are energy intensive to produce, but are relatively inert as far as the fuel elements are concerned.

OTHER ALTERNATIVES

In-situ processing possibilities, especially for Mars in SEI missions, are much broader than the range discussed. A very interesting option, for example, is the production of hydrogen peroxide (H_2O_2), from the indigenous elements H and O, and/or direct use of the water molecule. This option is especially interesting as a "standard" product, the equivalent of petroleum on Earth.

Hydrogen peroxide has an enormous range of uses (a source of oxygen, water, heat, mechanical energy, transportation energy, electrical energy, explosive energy for industrial processes, chemical reactions, etc.), all of which could find immediate, ubiquitous applications on Mars. For transportation it can be used as a chemical rocket monopropellant, or bipropellant oxidizer, a use that intersects this submission's applications.

Numerous manufacturing methods are potentially available under Mars ambient conditions or on Phobos or Deimos. **Submission #101275** gives an excellent overview of H_2O_2 possibilities.

CONCLUSION

The possibilities set forth in this submission are worth significant RDT&E focus. However, a still broader range of in-situ potentials is apparent. The example of hydrogen peroxide as a product with a much wider scope of applications is a very important case in point. In-situ product possibilities demand a very extensive tradeoff RDT&E program.

Appendix O

SATURN V HEAVY LIFTING LAUNCH VEHICLE CONCEPT (SUBMISSION #100192)

A FALL-BACK-TO-SPRING-FORWARD STRATEGY TO A HEAVY-LIFT LAUNCH VEHICLE: REVIVING SATURN V TECHNOLOGY (SUBMISSION #100185)

The concepts discussed in these submissions have essentially been described earlier in Sec. II, Earth-to-Orbit Launch Systems. They involve creation of an updated Saturn V unmanned HLLV by utilizing a basically unmodified first stage and a higher-performing, lighter-weight second stage, wherein the original five J-2 engines are replaced by three SSMEs, along with other modernizations. In our evaluation of these submissions, the following points provided by the authors seemed worth noting.

BACKGROUND

- Basic Saturn V is a proven design. It flew 13 missions without a launch vehicle failure.
- Although much of the tooling was scrapped and launch teams were disbanded after Saturn V's last flight in 1973, blueprints have been preserved in NASA archives and two flight vehicles exist, one at Johnson Space Center and one at Marshall Space Flight Center, which can serve as further "specification banks."
- Twelve flight-ready F-1 engines are in mothballs at Rocketdyne.
- Rocketdyne has recently initiated a study concerning reopening of the F-1 engine production and assembly lines.

SALIENT ASPECTS OF THE CONCEPT

- Development of a modernized and uprated version of the Saturn V vehicle would provide 250 to 350 klb of payload capability in a cost-efficient, low-risk, and timely manner compared to other alternatives. It is estimated that a first firing could take place in four to six years (with additional time to assure high reliability). Shuttle-C will require about six years, and an ALS vehicle is not expected to be available until the year 2000 or beyond.

- Saturn V can be upgraded with flight-proven SSMEs in the second stage in a recoverable mode and would add an upper-stage throttling capability, Shuttle Transportation System (STS) or other state-of-the-art avionics, improved stage materials and fabrication techniques, and a composite materials payload nose fairing.
- Much of the Saturn V launch infrastructure still exists and is generally compatible with current Shuttle launch and assembly operations; hence, parallel Saturn V/STS flight operations may be uniquely possible. Some modifications and new construction would be required.
- An upgraded Saturn V would be within the currently estimated launch size limit of Kennedy Space Center (about 300 klb), considering safety, overpressure, and environmental factors.
- In comparison to other alternatives, ALS is a paper concept with no extant hardware or launch infrastructure in place. Shuttle-C has never flown and has much less payload capacity (85 to 150 klb to LEO).

Reviving Saturn V represents a major engineering effort, but so do other alternatives. The authors of these submissions provide extensive, soundly based detail in their comprehensive assessment of what would be required and how to accomplish the creation of an updated and uprated Saturn V HLLV. The basic arguments and approach seem sufficiently convincing to warrant consideration for SEI. Any further evaluation should probably also include the relative merits of using STMEs in place of the herein proposed SSMEs, should they become available. The potential compatibility of Saturn V with parallel Shuttle assembly/flight operations from Kennedy Space Center seems particularly important to consider.

Appendix P

ULTRA LARGE LAUNCH VEHICLE (ULLV) FOR MOON AND MARS MISSIONS (SUBMISSION #100110)

The concept proposed in this submission combines several design aspects that are somewhat akin to those of the past NEXUS and SEA DRAGON concepts, earlier described in Sec. II, to provide payload to LEO capabilities in the 1.0 to 1.5 million pound range. This concept, called EUCLID, has been under recent study for several years by the author and, hence, is envisioned to incorporate current technologies such as those being explored in the ALS/ALDP program.

Principal elements of the total system are the launch vehicle (LV), a multibay LV erection facility where vehicles are assembled on launch barges in dry dock, a metals fabrication factory to support the erection facility, launch barges equipped with propellant loading facilities below deck, canal access to the ocean, a way-station where LOX and LH_2 are generated for loading into the barge propellant tanks, and an oil platform structure at sea constructed so that the barge (from which the vehicle is launched) can be floated over pedestals and secured by ballasting.

The LV is a smooth, conical, single-stage design, 300 ft tall with a 138-ft maximum diameter. Hydrogen is contained in a 120-ft-diameter sphere mounted atop a toroidal LOX tank. General Dynamics has produced 55 120-ft-diameter aluminum alloy tanks in an LNG ship program in serial production at a cost of about $8 million each. The vehicle gross liftoff weight is 28.4 million pounds, providing 1.5 million pounds of payload to LEO for a payload-to-gross-weight ratio of 0.053. Liftoff thrust is 36 million pounds, using 18 2-million-pound thrust engines based on M-1 engine technology (see Sec. II). A single-stage propellant mass fraction of about 0.94 is required and believed achievable. The author estimates that the LVs could be produced for $150 million each, the total system for under $10 billion in less than ten years, and that payload could be delivered to LEO for about $100/lb. The concept stresses simple design using ALS-type cost-reduction principles in conjunction with the demonstrated H_2 tank fabrication techniques.

The benefits (and tradeoffs) associated with ULLVs were enumerated in Sec. II and could be of particular importance to SEI, given a sufficiently long-term commitment. While we cannot vouch for the author's performance and cost estimates, the overall concept is fundamentally sound. The general approach should be applicable for payload capacities in the range from perhaps 500 klb to 1.5+ million pounds. Of course, the required single-stage propellant mass fraction becomes more difficult to achieve as vehicle size is reduced, but as in the NEXUS studies, one and one-half or two stage configurations can hedge against these uncertainties. Several aspects of the concept have been demonstrated in principle, and the barge launch feature is particularly interesting. We believe the submission warrants further consideration if payload capabilities beyond current planning are envisioned.

Appendix Q
TETHERS

Tethers are long cables that can exchange energy or momentum between two objects. They are sometimes classified as propulsion systems, but they can have other applications as well, such as the production of power.

Many types of tethers or other momentum exchange devices have been investigated over the past ten years or so. Consider a tether in its simplest form—i.e., with objects of equal mass at each end, in orbit about the Earth. The center-of-mass of the system is at its midpoint, and it is this point that has the proper velocity for the tether system to maintain a circular orbit (the tether is initially aligned with the local vertical). Because of the gradient of the Earth's gravitational field, the gravitational force acting on the upper mass is less than the centrifugal force due to orbital rotation. The opposite is true of the lower mass, and thus the tether is in tension.

If a part of the upper mass (the payload) is released, it will have a velocity greater than circular orbital velocity for that altitude. As a consequence, the payload will go into an elliptical orbit with release point being perigee. Thus, the payload has, in effect, been given a delta V. The tether, with the two unequal masses, enters an elliptical orbit with apogee at the release point.

The tether would then be reeled back into the lower mass, which is the on-orbit station. In practice, the lower mass or station is much more massive than the payload that is released. This minimizes the perturbation to the original orbit due to payload release. In any case, propulsion must be used to bring the station back to its original circular orbit.

In an operational system, a station in circular Earth orbit would initially deploy a tether to an altitude lower than that of its orbit. A vehicle, on a path that, at apogee, has an altitude and velocity matching that of the tether end, is captured by the tether. The station then reels the vehicle in. After the vehicle reaches the station, the station rotates 180 degrees and reels the tether out until the vehicle reaches the release altitude. The process can be reversed by capturing vehicles returning from space and transferring them to a lower orbit.

Submission **#100938**, **Leo Tether Transportation Node**, proposes a tether system of the type described above. Assuming a material such as kevlar, the tether length would be limited to about 600 km. For an upwards release with a tether of this length, the vehicle would require a lunar transfer delta V 1.1 km/sec less than that needed from LEO. The station mass would be at least 400 metric tons with a tether mass of 20 metric tons. A low-thrust electric propulsion system could be used to maintain the station in a circular orbit.

It is obvious that a tether system requires a substantial infrastructure and its cost effectiveness would depend upon the volume of traffic between LEO and high-altitude Earth orbits or LEO and the Moon. Even for a 400-metric-ton station, the maximum mass of the vehicle to be transferred would be limited to about 50 to 60 metric tons. Also, there are technical issues regarding the dynamics and control of tethers.

Another submission, **#100941**, entitled **Phobos Tether Transportation Station**, proposes that the Martian moon, Phobos, be used as a tether system. Phobos is sufficiently massive that propulsion would not be needed to correct its orbit after releasing a vehicle. The submission proposes using a 1400-km tether to transfer vehicles from Phobos orbit to the upper atmosphere of Mars. No propulsion would be required. Deploying a vehicle "down" to Mars could generate power at the station because of the tension in the cable. Kevlar would be a suitable material for this tether.

Again, a substantial infrastructure would be required to operate such a tether syestem, which can be justified only if the traffic volume is high. Also, the problems involved in reeling in or out a 1400-km cable are not trivial. Controlling the flexure and torsional motion of such a long tether could also pose problems.

We received other submissions proposing various applications of tethers. In general, it is our judgment that tether systems can be justified economically only when large traffic volumes are anticipated. It would seem that the most likely application of tether systems would be on bodies that have relatively weak gravitational fields, such as the Moon and the Martian moons.

Because of required infrastructure and support, it is not clear that tether systems are more cost effective than alternative approaches. A recent study done at JPL (1989) examined a tether system proposed by Penzo (1984) for transporting cargo to Mars via Deimos and Phobos. The conclusion of the study was that "the tether-assisted propulsion system option is not sufficiently better than the baseline chemical system to warrant its use" (p. 9-12).

Appendix R

LONG-ENDURANCE AIRCRAFT AS A MARS EXPLORATION VEHICLE (SUBMISSION #100400)

This submission proposes the use of long-endurance, low-altitude, remotely piloted aircraft that would be capable of nonstop flight for periods up to a year or so for Mars planetary exploration. Possible missions might include

- High-resolution mapping or reconnaissance of a given region by circling or flying a grid pattern over the area.
- Mapping of magnetic and gravity fields of various regions near the planet's surface.
- Searching for subsurface water, geothermal sources, or volcanoes.
- Performing atmospheric soundings, composition measurements, and meteorological surveys.
- Deploying navigation beacons or other equipment at selected surface locations.
- Complementing other types of exploration vehicles, such as land rovers.

Basic concept feasibility and preliminary design requirements of such aircraft have recently been studied under contract to NASA Lewis Research Center. Both radioisotope heat engines and photovoltaic solar array power production systems have been considered. The results show, to a first approximation, that long-endurance aircraft flight within the Martian atmosphere may indeed be feasible. Aircraft size, weight, and performance appear comparable using either power source. All cases assumed a payload of 100 kg. For solar power, two solar cell efficiencies were considered, 14 and 25 percent. For currently available cells (14 percent efficient), the aircraft would weigh about 1200 kg with a wing span of over 100 m. With more advanced solar cells, the weight and wing span may reduce to approximately 500 kg and 50 m, respectively. In either case, the cruise velocity is about 30 m/sec.

If radioisotope power is used, the gross weight and wing span are comparable to advanced solar cell designs, but the cruise velocity can increase about 10 m/sec and year-round operational flexibility is enhanced. The radioisotope configurations would likely be substantially higher in cost.

Considerably more work is needed to realistically assess the performance, size/weight, cost, and utility of such aircraft, as well as to gain a better understanding of the problems of deployment, stowage, and stabilization of such large wing span vehicles. However, we believe this concept deserves further consideration, along with other alternatives (satellites, balloons, etc.) as a means for conducting the aforementioned types of Mars exploration missions.

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