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Research on the Use of Space Resources

William F. Carroll
Editor

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The research described in this publication was carried out by the Jet Propulsion Laboratory, California Institute of Technology, under contract with the National Aeronautics and Space Administration.

ABSTRACT

This report covers the second year of a multiyear research program on the processing and use of extraterrestrial resources. The fiscal year 1981 results are reported in JPL Publication 82-41, dated April 15, 1982 and entitled Extraterrestrial Materials Processing, chief author Wolfgang H. Steurer. The report was subsequently identified as NASA CR 169268. A multiyear research plan was also developed in parallel with the research activity reported here. The plan was submitted to NASA under document JPL D-217, dated August, 1982.

The research tasks included: 1) silicate processing, 2) magma electrolysis, 3) vapor phase reduction, and 4) metals separation. Concomitant studies included: 1) energy systems, 2) transportation systems, 3) utilization analysis, and 4) resource exploration missions. Emphasis in fiscal year 1982 was placed on the magma electrolysis and vapor phase reduction processes (both analytical and experimental) for separation of oxygen and metals from lunar regolith. The early experimental work on magma electrolysis resulted in gram quantities of iron (mixed metals) and the identification of significant anode, cathode and container problems. In the vapor phase reduction task a detailed analysis of various process concepts led to the selection of two specific processes designated as "Vapor Separation" and "Selective Ionization." Experimental work was deferred to fiscal year 1983. In the Silicate Processing Task a thermophysical model of the casting process was developed and used to study the effect of variations in material properties on the cooling behavior of lunar basalt.

Preliminary indications are that the power source is likely to be the most massive subsystem of an extraterrestrial materials processing facility. Based on a simple mass pay-back ration figure-of-merit, chemical propulsion systems using near-term technology and oxygen produced at the Moon can transport lunar products to low earth orbit (LEO) for net gains in mass. In the Utilization Analysis Task, lunar oxygen was identified as the most important early product to be derived from space resources. In the Resource Exploration Missions Task, four important uncertainties (questions) were examined: 1) water at the lunar poles, 2) existance of Earth Trojan asteroids, 3) accessible "near-earth" asteroids, and 4) lunar transient phenomena as indicators of gas deposits.

PREFACE

This report documents the research accomplished during the second year of a multiyear program on the processing and use of extraterrestrial materials. The work was sponsored by the NASA-OSSA Materials Processing Office. During the first year of the program, research was carried out under RTOP 179-29-62, "Extraterrestrial Materials Processing." Results of the first year's efforts are reported in Extraterrestrial Materials Processing, JPL Publication 82-41, NASA CR 169268, dated April 15, 1982, chief author Wolfgang H. Steurer.

The work reported here was carried out under RTOP 176-46-20, "Research on the Use of Space Resources." This report, covering work completed during the second year, has been compiled by William F. Carroll and Wolfgang H. Steurer from texts prepared by the principal technical contributors to the program. Editorial changes have been made by the editors, but there has been no attempt to rewrite the text to produce uniform style throughout. Names of principal authors for individual sections appear in the Table of Contents and in the text in order to identify them with their work, for the readers.

In parallel with the research reported here, a multiyear research plan was developed. Details of the plan have been submitted to NASA under document JPL D-217, dated August, 1982.

Robert A. Boundy, Task Manager

EXECUTIVE SUMMARY

R. A. BOUNDY

This report documents the technical work completed during fiscal year 1982 under the program Research on the Use of Space Resources (RUSR). It continued the research on extraterrestrial materials processing with emphasis on two analytical and experimental methods for extracting oxygen and metals from lunar, and other, extraterrestrial surface oxides and on analytical studies of processes for mixed-metal separation. Work was also continued on methods for fabricating useful shapes from indigenous oxides (silicates) and a study was initiated on mixed-metal separation. Studies were completed which clarified the constraints and the roles of propulsion and energy technology on extraterrestrial materials processing. Additional studies addressed the limits of feasibility for extraterrestrial products, and the requirements and approaches for missions to extend our knowledge of space resources.

Silicate Processing

Prior analytical and experimental work on fabricating useful shapes from mixed lunar oxide powders established the attainable properties as a function of compaction, processing temperature, and cooling rate. During this past reporting period a thermophysical model of the casting process has been developed which incorporates temperature-dependent properties of materials, the liberation of heat on going from liquid to solid (heat of solidification), and a capability to vary the properties of the mold, liquid, and solid independently of one another. This model has been utilized to study the effect of variations in material properties on the cooling behavior of a lunar basalt.

The thermophysical model data demonstrate that for material with an emissivity greater than approximately 0.7 the rate-limiting step in the heat-transfer process is not the radiation of heat from the surface but rather the rate of heat transfer through the solid (free surface) and the mold. At present the thermal history of the cast basalt has been determined over the time period required for the last remaining liquid to reach the solidus temperature. The model, however, is quite general in nature and can be used to follow cooling behavior of any material for which sufficient property data are available.

The predictions of the thermophysical casting model are based on the results of the prior experimental cooling rate studies. Although the present work is discussed in the context of lunar materials and processes, the program scope also includes asteroids and Martian moons as sources of silicate materials for study.

Magma Electrolysis

Magma means "molten rock." Usually we understand, in the context of Magma Electrolysis, a melt composition similar to basalt, or to minerals separated from basalt. Magma Electrolysis, though similar in principle, differs considerably in practice from "electrowinning" of metals as carried out on Earth. Even if we compare only processes conducted at high temperature, there are basic differences. For example, Magma Electrolysis employs no fluorides or other fluxes to lower viscosity and alter reduction voltages. Also, it does not employ a consumable anode (e.g., graphite) to increase the driving force of the reaction or to avoid the expense of a permanent anode capable of withstanding the high temperature of the process.

For Magma Electrolysis to work, one must first melt the rock, using heat alone (no flux). This puts the process in the range 1500-2000 K, or more realistically, 1800-2000 K. The strain this puts on materials of construction is extreme. Fortunately, much of this strain can be relieved, on the works scale, by properly designing the thermal gradients within the cell. However, on the laboratory scale the problems can be severe.

Analysis has shown that semiconduction, as opposed to ionic conduction, must be avoided. Semiconduction is essentially a parasitic process detracting from the manufacture of useful products, namely oxygen and metals. During the reporting period several experimental studies were initiated. These included conductivity measurements of magma samples in molybdenum crucibles and direct-current magma electrolysis experiments. These conductivity experiments established that molten basalt is conductive enough to support electrolysis, even at temperatures just above the liquidus line. Since these and related studies have shown that the conductivity of magma increases with temperature, the feasibility of Magma Electrolysis was thus firmly established. Representative experimental data are given in the report.

The experimental work on Magma Electrolysis using basalt resulted in the production of gram quantities of iron (mixed metal) and the identification of significant anode, cathode, and container problems. These are discussed in detail in the body of this report.

Vapor Phase Reduction

An alternate process for the extraction of metals and oxygen from basaltic raw material, most abundant in near-space, is "Vapor Phase Reduction." This process employs the gaseous state, either in the form of dissociated vapor, or in the form of a thermal plasma. It is characterized by total reliance on environmental space resources, primarily vacuum and solar energy, thus eliminating the need for consumables from Earth.

Analytical studies of a number of specific processing concepts led to the adoption of two processes for in-depth study, designated as "Vapor Separation" and "Selective Ionization." In the reporting period, the evaluation of these processes was confined to analytical studies, while experimental verification was deferred to FY'83.

The "Vapor Separation" process is aimed at the production of oxygen, which has been identified as the most desirable product for near-term space operations. In this process, the granulated raw-material oxides are vaporized and the temperature raised to a point where dissociation takes place and a substantial amount of oxygen is set free. Rapid cooling of the dissociated vapor to a discrete temperature causes condensation of the oxides and suboxides while the free oxygen remains intact and can be collected downstream. The relatively low operational temperature of 2500-3000 K permits extensive use of direct (concentrated) solar energy, which reduces the total facility mass substantially, in contrast to the use of solar-electric energy.

The analytical studies comprised: (1) the definition of the vapor composition at temperatures from 500 to 3000 K in terms of partial pressures of oxygen and other dissociation species--data were generated for individual oxides, as well as for oxide mixtures representative of extraterrestrial basalt; (2) definition of oxygen yield in weight percent of the starting material which was found to be in the order of 20-25%; (3) assessment of energy requirements.

The "Selective Ionization" process takes advantage of the ionization gap between metals and oxygen in the 7000-10,000 K regime. Separation is achieved in an electrostatic or electromagnetic field, where the highly ionized metals are extracted while the essentially neutral oxygen continues to flow downstream for recovery.

In the analytical studies a single-element approach was employed for the generation of numerical data on the equilibrium ratio of ionized-to-neutral atoms over the temperature range from 4000 to 10,000 K. The results were then translated into product yield. The theoretically obtained yield at 8000 K is 37% metals and 25% oxygen, and at 10,000 K 51% and 38%, respectively. Tentative calculations place the net energy requirements at approximately 35,000 kWh/ton for oxygen (only) and 26,000 for metals, and at 15,000 kWh/ton when both metals and oxygen are recovered simultaneously.

The efforts on both processes further included preliminary facility design studies and mass estimates.

Metals Separation

Both Magma Electrolysis and Vapor Phase Reduction processes are likely to produce (in addition to oxygen) metal mixtures which will require further processing into useful metal products. A study was initiated to address this subject.

To understand why mixes are likely to be produced, it is necessary to grasp why processes that proved successful in metal winning on Earth are not considered for extraterrestrial materials processing. Neither the Moon nor the asteroids, as far as we know, contain "ores" in the terrestrial sense: that is, concentrations of metal sulfides or other compounds that have been deposited in limited areas and show concentration factors of 3 to 20 orders of magnitude. Any metallurgical process conducted on the Moon must rely on an absolute minimum of imported materials. Water, fluorides, chlorides, etc., which have proved so useful in terrestrial metal winning, have no part in lunar metallurgy.

The study identified vacuum distillation as the most effective way (given lunar conditions) to separate critical pairs of metals. Distillation may be used, for example, to purify aluminum for structural purposes; to purify silicon for electronic purposes; and to purify the alkali and alkaline-earth metals. Of these, magnesium and calcium may be used for structural purposes, and all may be used for chemical service.

Distillation may be potentially used to purify titanium for structural purposes. Modified by preliminary chemical steps, such as liquation or reaction with lunar-abundant elements, distillation may be potentially used to ultrapurify aluminum for electrical conductivity or semiconductor use, and to provide alternate routes to the ultrapurification of silicon.

Energy Systems

Energy systems may handle energy in various ways to perform four general functions: (1) collection, (2) conversion, (3) storage, and (4) transmission. Energy systems may deal with energy in any one of its several forms, i.e., mechanical, thermal, electrical, magnetic, electromagnetic, chemical, and nuclear. The objective of the RUSR energy studies is to provide energy in the most economical manner for all the major energy-consuming elements under the envisioned operational conditions.

For the future operating processing facility, one can identify three major categories of operation that will require the use of energy systems: (1) material conditioning and processing, (2) transportation, and (3) miscellaneous support operations (crew support, housekeeping, communications, etc.). These operations may be carried on at different locations depending on eventual overall design. For any one operation, the characteristics of these different locations will influence the choice of the appropriate energy system required to perform a certain function. The characteristics of these locations insofar as they influence the choice of energy systems are discussed.

The general operations (material processing, transportation, and miscellaneous support) will require energy in either electrical or thermal form. Some material-processing and material-conditioning operations will be able to utilize direct thermal power while others will need electrical power. The majority of the transportation and miscellaneous operations will require their energy to be provided in the form of electrical power. For realistic scenarios, only nuclear and solar energy sources can be considered. Waste thermal power from material-processing functions may also be recovered and recycled, but this cannot be considered an energy source. The preliminary energy requirements per ton of process output for the specific processes under each general operation were determined.

Preliminary indications are that the power subsystem is likely to be by far the most massive subsystem of an extraterrestrial materials processing facility. Therefore any steps that can be taken to reduce the imported mass of the power system may have large benefits. The largest fraction of the power system will be the power-source itself, most likely either a solar array or nuclear reactor power source. Both of these sources can conceivably supply power from 10-1000 kW_e at 100 to 1000 volts DC or AC. The study identified several power-source components which might be made from extraterrestrial materials. These included the cover glasses which shield solar cells from UV and some particulate radiation; the radiation shield for the nuclear reactor power source; and large current conductors.

Several useful Energy System tables are given in the report. These include Location Characteristics, Energy Requirements, Power Requirements, Energy Collection Systems, Energy Conversion Systems, Energy Transmission Systems, and Energy Storage Systems. The latter four indicate the status of technology in these areas.

Transportation

Space-based transportation systems for the transport of extraterrestrial materials from their point of origin (e.g., lunar surface) to final delivery point in Earth orbit and transport of equipment and supplies for extraterrestrial processing to the operational site were analyzed. The emphasis has been on the use of near-term (pre-year-2000) propulsion systems, such as advanced chemical (H_2/O_2) and nuclear-electric systems. More exotic systems (mass drivers, nuclear-thermal rockets, etc.) were also considered to evaluate the potential to be gained by use of post-year-2000 technologies. Finally, the effects of using extraterrestrial propellants (e.g., lunar-derived O_2 for chemical systems) were compared to those of propulsion systems deriving all or part of their propellants from Earth.

In this year's efforts, transportation of materials to and from Earth-orbit and the lunar surface has been emphasized. Transportation for asteroidal materials was also considered, but in less detail. Near-term propulsion systems were emphasized since the technology base being developed for space-based orbit transfer vehicles (OTVs) designed for Earth-orbit transfers can be applied directly to Earth-orbit-to-Moon or asteroid vehicles. This is especially important economically, since the Earth-orbit satellite market will drive and support development of space-based OTV technology independently of, and probably prior to, an extraterrestrial materials program.

Based on a simple mass pay-back ratio (MPBR) figure-of-merit, the following conclusions are drawn:

(1) Chemical propulsion systems using near-term technology (aerobraking, space-based H_2/O_2 OTVs, I_{sp} = 460 s, O/F = 6

engines) can transport lunar material to LEO or GEO for net gains in mass. The MPBRs found for the baseline chemical system are highly significant since they are large enough to be potentially attractive without the need to develop any new major technologies.

(2) There are several options available that can provide significant (but not dramatic) improvements in MPBR. These include the relatively near-term electric propulsion system using 50- to 100-cm Ar electric thrusters.

(3) For asteroid materials, only Earth-Trojan asteroids appear to offer significant promise on a mass basis, although the presence of valuable resources (H_2 from H_2O, etc.) might change this conclusion.

(4) Most advanced systems, like nuclear-thermal rockets, are less effective than chemical systems since all the nuclear rocket propellant (LH_2) would have to be supplied from Earth, assuming no lunar H_2.

(5) Specific technologies and vehicle components required for a lunar materials program would be directly adaptable from OTVs developed in the next few decades to satisfy LEO-to-GEO satellite traffic demands (independently of a lunar materials program).

(6) Finally, there is a nontechnology issue that vastly overshadows all others, i.e., the discovery of a source of lunar hydrogen. This could yield infinite MPBRs by eliminating the need for any H_2 from Earth.

Utilization Analysis

It requires 7 times more energy to go from Earth's surface to low Earth orbit (LEO) than it does from the Moon to LEO. LEO is particularly important since that is the orbit in which initial large-scale utilization of space is likely to occur--for example--in the form of manufacturing, micro-g science experiments, astronomical and astrophysical observations, and staging of spacecraft for geostationary or interplanetary orbits. At times, it requires even less energy to go from some asteroids to LEO than from the Moon to LEO. Launch energies to most higher Earth orbits or to interplanetary trajectories are even more favorable to payloads of lunar or asteroidal origin. Therefore, there may be advantages to be gained from the use of extraterrestrial materials in large-scale space operations.

The conditions under which such advantages are possible have been examined. The basis for the evaluation is an examination of the mass of Earth-origin supplies and equipment required to produce and transport extraterrestrial materials, compared to the mass of conventional Earth-origin space hardware or materials that might be displaced. The following is a list of some of the conclusions reached as a result of the analyses.

(1) Lunar products for use in LEO or GEO:

  (a) Lunar O_2 or some nonconventional form of propulsion (e.g., an electromagnetic launcher) is required for transportation in order for any product to be practical.

  (b) Products whose performance is directly proportional to mass (e.g., lunar O_2 for propulsion, lunar products for radiation shielding) appear to have a high probablilty of significant advantage over comparable terrestrial products.

  (c) Structural metal-alloy parts are not likely to be viable early candidates unless their performance closely approaches comparable terrestrial products.

  (d) Components made from unrefined or partially refined lunar surface material ("lunar bricks") which function primarily as structure are very unlikely to be competitive.

  (e) Products which require production facilities and processing aids or reagents on the Moon in excess of a few percent (1-3%) of the life cycle product mass are very unlikely to be competitive.

(2) Asteroidal products for use in Earth orbit require less transport energy than from the Moon. While this difference may improve the potential advantages of extraterrestrial materials, it is unlikely to result in conclusions which are significantly different from those regarding lunar products.

(3) In the case of lunar products for use on the Moon, the conditions under which an advantage can be shown over terrestrial competition are significantly less constrained.

  (a) Lunar products are competitive even with a performance disadvantage of 20 to 100 compared to terrestrial counterparts. This appears to allow the use of ceramic structural parts made from lunar surface materials and metal structural/mechanical products with relatively "crude" composition and shape.

  (b) For products where performance is equivalent to terrestrial competition, total imports, including production facility, energy supply, and expendable processing aids/reagents can approach the life-cycle mass of the product. (This condition applies only for products with a large potential market <u>on the Moon</u>.)

(4) Manned operations will be practical only for initial exploration and possibly for nonroutine maintenance.

(5) Extraterrestrial materials will be practical only for operations requiring large quantities (tons/year) of individual products.

Resource Exploration Missions

Early analysis indicates that the use of certain nonterrestrial resources may be commercially advantageous within two decades. Oxygen from the Moon presently appears to be the most likely candidate for an initial space-resource product: in this case, a product to be used for propellant for orbit transfer vehicles (OTVs) carrying high-value cargo from low Earth orbit (LEO) to geostationary Earth orbit (GEO). Such OTVs would utilize hydrogen as a fuel, to be burned with oxygen; the latter represents about 60% of the gross loaded mass of an OTV, and about 4 times the useful payload mass.

Strong advantages have been identified (by Frisbee, Carroll, and Staehle) if hydrogen could be found along with oxygen in useful, accessible quantities. At this time there is no known source of both hydrogen and oxygen that meets both the favorable concentration and the accessibility criteria (Mars is rather inaccessible, while known lunar hydrogen is at a very low concentration.) However, there are good reasons to suspect undiscovered hydrogen sources on the Moon and among the asteroids.

Other resources of potentially high value may also exist, particularly among the near-Earth asteroids. During fiscal year 1982, Resource Exploration Mission activities focused on identifying the potentially important early resources, identifying the key uncertainties regarding their presence in accessible locations, and proposing methods of resolving those uncertainties.

Also during 1982, 4 of the 10 most accessible known asteroids were discovered using ground-based photographic searches. Two of these, 1982 DB and 1982 XB, were discovered by Eleanor Helin of JPL, whose work was partially supported by this program. Of the "top ten" asteroids, these rank first and fourth, respectively, in terms of lowest energy to reach them from Earth using an optimal ballistic trajectory. They also rank first and second in lowest energy of ballistic return to Earth using aerocapture, a parameter of greatest importance when considering the accessibility of materials for use in Earth orbit.

The present extraterrestrial materials-processing research program, based on known natural availability, is planned with the flexibility to take advantage of any newly discovered resources. Four particular uncertainties regarding resource availability are key issues in focusing processing-technology development. The sooner these uncertainties can be resolved, the sooner a sound space-resource utilization scenario may be constructed, allowing all relevant technology areas to be focused much more effectively than is possible today. These four uncertainties, in descending order of priority, are:

(1) Is there water (or other volatile ices containing H or C) at the lunar poles?

(2) Do Earth-Trojan asteroids exist, and if so, what are their orbital parameters, what resources do they harbor, and how large are they?

(3) Are there easily accessible asteroids (i.e., low-inclination "near-Earth" asteroids) with water or with substantial quantities of free metals at or near their surfaces?

(4) Do transient lunar phenomena indicate the existence of substantial gas or volatile ice deposits on the Moon, and if so, what gases of resource importance are available?

A positive answer as early as possible to any of these questions would be of great value by providing a focus for early resource-utilization research.

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APPENDIXES

SECTION 1

INTRODUCTION

William F. Carroll and Wolfgang H. Steurer

1.1 BACKGROUND

This report covers the second year of a multiyear research program on the processing and use of extraterrestrial resources. The initial year's results are reported in JPL Publication 82-41, dated April 15, 1982 and entitled Extraterrestrial Materials Processing, chief author W.H. Steurer. The report was subsequently identified as NASA CR 169268.

This report will cover only work completed during the second year of the program and will not duplicate results included in the first, except where repetition is essential for clarity. The reader is referred to previously distributed copies of the JPL report or to copies available through NASA.

A multiyear research plan was also developed in parallel with the activity reported here. The details of that plan have been submitted to NASA under document JPL D-217, dated August, 1982.

The principal results of the work reported here are: research on the three key processes selected for detailed evaluation and covered in Sections 2, 3, and 4; analysis of metal refinement and other processes reported in Sections 5 and 6; assessment of supporting technology in Section 7; analysis of the potential utility of extraterrestrial materials in the near-term, e.g., 2-3 decades, and probable constraints on utilization in Section 8; and progress, needs, opportunities, and requirements for exploration in Section 9.

The primary motivation for interest in the use of extraterrestiral resources is the difficulty of launching materials from the Earth's surface. It takes 7 times more energy to get 1 kg from the Earth's surface to Low Earth Orbit (LEO) than it does to move 1 kg from the Moon to LEO. Thus, at least some part of the materials, equipment, and supplies for large-scale space operations might be supplied from extraterrestrial sources, reducing the requirements for costly, energy-intensive launches from Earth.

Extraterrestrial materials are not "free"--their production, transport, and use will require some equipment and supplies that must be launched from Earth. The magnitude of such equipment and supplies, compared to the amount and function of the extraterrestrial products, will determine whether there is a net advantage to be gained from the use of extraterrestrial materials.

Currently, candidate processes, process efficiencies, equipment requirements, and product characteristics are not well-enough understood to accurately determine the practicality and/or limits of the use of extraterrestrial materials.

As a result, the primary objective of the present research is to develop the information necessary to make meaningful comparisons. These comparisons are necessary to make objective decisions whether or not to proceed with the major engineering development programs necessary to develop processes and products for extraterrestrial use. A second objective is to provide the technical base for such development in the event of a positive decision.

1.2 APPROACH

1.2.1 Programmatic Definition

During fiscal year 1981, a broadly based program was pursued, including analyses of extraterrestrial resources (starting with raw materials), mining and material conditioning, and primary and secondary processing. The results of this broad examination formed the basis, in the second year, of the research on specific selected processes, and of the examination of critical supporting technologies. It also provided the background for more detailed analyses of the various processes, processing systems, and candidate products in order to examine and document perspectives of extraterrestrial material applications and limitations.

The general approach, and the basis for the research and for the long-range plan, is the maximum use of both the indigenous materials and the extant environment of space. The direct adaptation of terrestrial mining, conditioning, extraction, and manufacturing processes to the space environment is not considered a viable approach. Therefore, the research primarily emphasizes application of the basic chemistry and physics to the examination of candidate processes and products rather than emphasis on conventional processes (e.g., conventional extractive metallurgy).

There are large numbers of potential candidate starting materials, processes, products, and end-uses which can be considered for extraterrestrial operations. The various individual and selected combined options identified during the first year are summarized in Table 1-1. From Table 1-1, it becomes obvious that the very large number of combinations makes it essential that a systematic approach be used to establish priorities.

The methodology which has been used to examine classes of options and the results of the examination are described in detail in Section 8.0.

Table 1-1. Space Resource Utilization Options

Resources	Processes	Products	Uses
Mixed oxides Lunar Asteroids Mars Phobos & Deimos	Mining & material conditioning Acquisition Comminution Sizing Beneficiation	Mixed oxide shapes Glass/crystalline Porous/dense	Structure[a, b] Insulation (thermal/elect)[b] Radiation shielding[b] Optical/chem/other[c]
Metals Lunar fines Asteroids	Primary Processes Extraction Shaping Post-treatment	Oxygen	Propulsion[b] Life support[c] Energy Storage[c]
Water Lunar poles (?) Asteroids Mars Phobos & Deimos	Secondary Processes Fabrication Assembly Installation	Metal alloy shapes	Structures[b] Containers/tanks[b] Misc.
Hydrogen Solar wind Lunar soil		Hydrogen	Energy storage[b] Life support[c] Extraction process Chemistry[c, d] Propellant[d]
Carbon Asteroids Phobos & Deimos		Silicon	Solar photovoltaics (unlikely in near term)
		Carbon	Reducing agent Life support

[a] Use limited to near point of production (transportation).
[b] Potential large volume product--objective of research.
[c] Very limited quantity needed or available.
[d] High priority if available in large quantities.

1.2.2. Priorities

If we examine probable scenarios starting with "use", Low Earth Orbit becomes the most probable first site for major space operations. LEO is currently reachable with large payloads and for manned operations with the Space Shuttle. It is below the radiation belts and therefore habitable for extended periods. LEO is also the most probable transfer point for Shuttle-launched communication satellites for geosynchronous orbit (GEO) and is also the most probable starting point for manned or unmanned operations elsewhere in space.

Extraterrestrial materials technology is not expected to be ready for the currently planned Space Station, but could certainly become a factor in either an evolutionary or revolutionary second generation thereof. The first generation Space Station can be expected to be a "user" in the context of a storage and fueling facility for extraterrestrial propellants for use in orbit raising. In a broader sense, the Space Station could provide a capability for system assembly, including installation or assembly of extraterrestrial components.

If we compare the bodies which are considered to be part of this current study--the Moon, near-Earth asteroids, and the Mars system--the Moon becomes the obvious first choice for initial studies. The Moon is not only the closest, it is also the only one that we know the surface composition of from firsthand examination of samples. It has other critical advantages as described in Section 1.3., below. Mars has obvious advantages for the production of propellant, specifically for a Mars sample return mission. Propellant production on Mars has been studied as part of the NASA propulsion research program, totally independent of the processing program described here.

Based on the analyses described in Section 8.0, oxygen extracted from lunar rocks and delivered to Low Earth Orbit for orbit raising, becomes an obvious high priority product. In fact, if current projections of communications satellite traffic prove correct, there is a market for extraterrestrial oxygen which is completely independent of any or all NASA space plans, and a market of a size which is sufficient to warrant major extraterrestrial processing investment. Therefore, the extraction of oxygen from lunar soil is the subject of two of the three major experimental investigations during this fiscal year.

The third major area of experimental investigation, fabrication of products from unrefined or partially refined lunar material, is based on other considerations. These products are expected to have potential structural applications only on the Moon. However, since the performance efficiency of radiation shielding is approximately proportional only to mass, shielding made from refined or partially refined lunar surface material can find significant advantage for use in Earth orbits. Shielding (of terrestrial or extraterrestrial origin) will be required for manned operations above LEO, and may find advantage in space nuclear power applications. Furthermore, next to LEO, the lunar surface is the most likely site for near-term large space operations. Such operations may very well involve construction of laboratories and habitats using rather crude lunar origin structural materials.

Although not obviously competitive for use in LEO (see Section 8.0), fabricated metal parts made from naturally occurring metals or metals extracted from oxides have broad potential utility for many space operations and are considered important. The processes for extracting oxygen from mixed oxides (electrolysis, vapor phase separation) are likely to yield mixtures of metals which will need to be refined for many of their potential uses. Since refined products will be important and the nature of the mixtures and their refinement will impact the details of the oxygen-extraction processes, refinement of metal mixtures was examined (analytically only) in some detail (see Section 5.0).

With limitations on resources and higher priorities for other products, it was necessary to defer work on fabrication of metal shapes until later in the program. The deferral is not a serious detriment [illegible], when the work is initiated, there will be a better understanding of the [illegible] and forms of metals which are likely to be available either from naturally occurring resources or from oxygen extraction and refining processes.

Results of analyses like those described in Section 8.0 will need to be used to guide program priorities on a continuing basis. Moreover, re-examination of conclusions will continue as planned as part of the prioritization process. For example, early analyses indicated that fabrication of conventional silicon photovoltaics devices is very unlikely to be practical at least for the first generation of extraterrestrial products. This conclusion still appears to be valid for use in LEO. However, information from recently developed processes for terrestrial photovoltaics indicates that the conclusion warrants reexamination for silicon devices for use on the lunar surface (and possibly GEO) as well as for newly developed thin film photovoltaics in general. The production of extraterrestrial photovoltaics will be re-examined in the coming months and may result in modification of research priorities.

1.2.3 Exclusions

There are many areas in which research could be done and which are of obvious importance to extraterrestrial materials processing and use, but which are specifically omitted from current plans. These are in addition to areas like metal part fabrication, which are being deferred for later phases of the program.

Most required mining and material conditioning processes for lunar operations appear to be well enough understood to proceed with development during a future engineering phase without further research. Mining and material conditioning in the microgravity of asteroids cannot be based on existing technology and will require research, but will also require better understanding of asteroidal composition and structure than is currently available.

There are other processes which are uniquely extraterrestrial but do not appear to require research at this time and which can proceed with development when the time arises. For example, there is a general programmatic assumption that solar-wind hydrogen trapped in the lunar soil will be extracted if the soil is handled in large quantities for any reason.

However, in the case of the solar-wind hydrogen, enough is known about the thermal release from studies on Apollo lunar samples to move directly into process development without the necessity for any additional research.

Likewise, it is assumed that metallic iron-nickel fines in the lunar soil will be extracted magnetically and somehow physically separated from the attached oxide materials. Such processing appears to be reasonably straightforward engineering development without the need for research.

1.2.4 Future Development

An overall program leading to the large-scale production and use of extraterrestrial materials is illustrated schematically in Figure 1-1 (taken from JPL document D-217, August, 1982). The current research program, the second year of which is reported here, is represented by the three boxes to the extreme left Once the current phase of research is completed, and adequate information is available to make a decision (assuming the decision is positive), the program must move through a series of complex, lengthy, and expensive steps prior to commercial implementation. The details of such a development program depend strongly on the results of the research on supporting technology which is likely to be developed in the interim, and on the nature and magnitude of the space program plans at the time of decision. Further detailed discussion is therefore inappropriate at this time.

1.3 COMPARISON OF LUNAR AND ASTEROID RAW MATERIALS

Of all of the extraterrestrial bodies, it is easier to move material from the near-Earth asteroids to Earth orbit than from any other extraterrestrial body including the Moon. However, the Moon has certain advantages which appear to more than make up for the somewhat greater difficulty in moving material to LEO.

The distance from the Earth to the extraterrestrial body to be utilized will determine the degree of automation and process autonomy which will be required. Of the candidate resource bodies, only the Moon is consistently close enough (approximately 3×10^{-3} AU) for teleoperation control of equipment and processes. For the asteroids and for the Mars system, the Earth-body distances vary from a fraction of an AU to several AU. Operations during extended periods at large distances will require highly reliable autonomous facilities.

The Sun to body distance and the resulting intensity of solar radiation incident at the surface of the Moon remains nearly constant at approximately 1 AU. The same is true for a group of asteroids called Earth Trojans whose existence has been postulated but not verified. For these bodies, high efficiency solar energy utilization is practical. The solar intensity at Mars is 1/3 to 1/4 that at Earth, and solar energy is a marginal energy source. The orbits of most of the Earth-approaching asteroids are highly eccentric, so that solar intensity varies and high-efficiency solar-energy utilization is impractical.

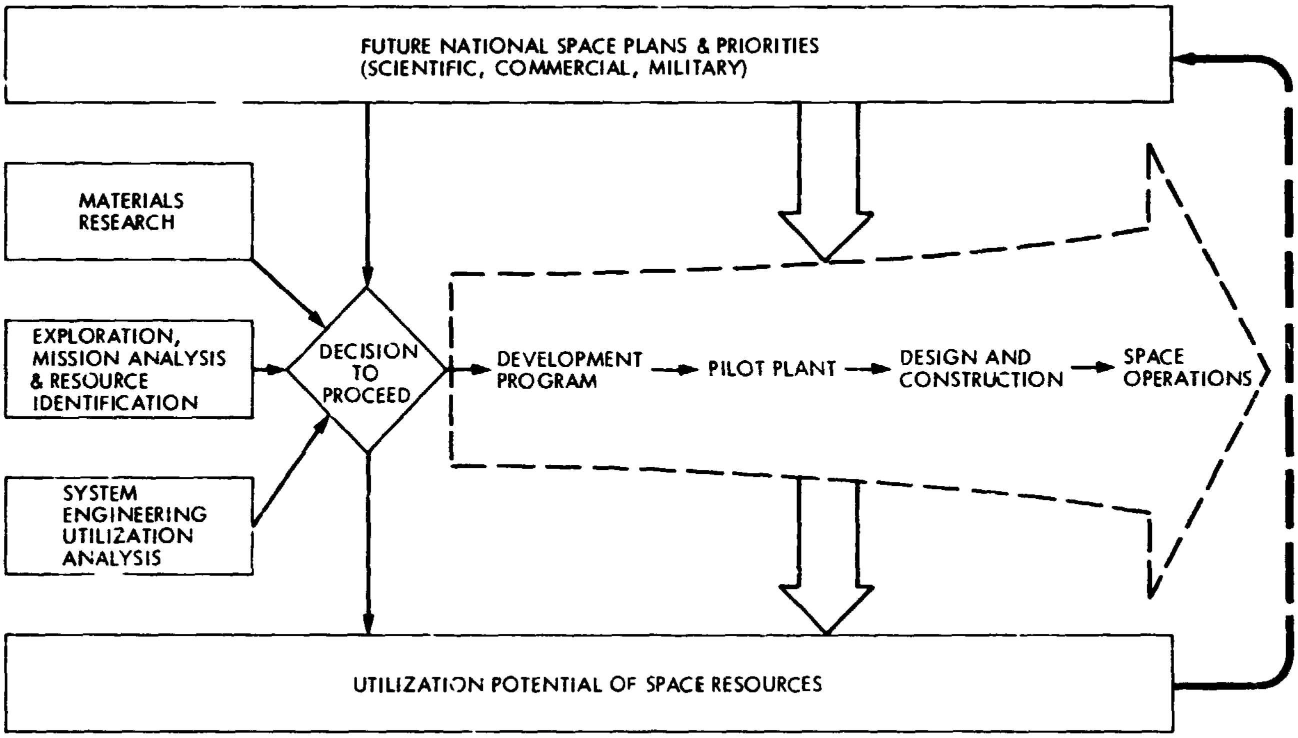

Figure 1-1. Extraterrestrial Material Processing Development--Program Flow Diagram

The final constraint on extraction and utilization of asteroidal materials is the requirement to operate in microgravity. While microgravity is an attraction for certain kinds of processes which are anticipated to be carried on in LEO (e.g., for production of certain types of crystals), it can present a serious constraint in the extraction and utilization of materials from extraterrestrial bodies for use in space.

SECTION 2

SILICATE PROCESSING

Richard H. Smoak

2.1 INTRODUCTION

The development of large structural elements (e.g., columns, beams, blocks, braces, etc.) by casting molten silicates requires a knowledge of the rate of heat transfer during cooling. The rate of cooling will vary with position in the cast structure. The cooling rate will affect the morphology (size and shape) of the grain structure, the types of crystalline phases that develop, and the amount of amorphous/crystalline material in the structure. The variation of composition with position in the casting may lead to establishment of deleterious residual tensile stresses. Rapid rates of cooling will produce fine-grained crystal structures which may contain significant amounts of amorphous (glassy) material. Slow cooling rates, on the other hand, will permit crystallization of more nearly equilibrium phases. The added time for nucleation and growth of crystalline phases during slow cooling will reduce the amount of amorphous phase present.

The mechanical properties of these structural elements will be dependent on the size, shape, and composition of the crystalline phases and on the location and amount of noncrystalline/crystalline material in the structure. The composition and crystal structure control the thermal expansion/contraction behavior of each of the crystalline phases. A large mismatch in the contraction of two different phases during cooling will result in residual stress in the part which will often lead to crack initiation/propagation and failure. The deleterious effects of a mismatch in thermal contraction behavior can be minimized to some extent if a homogeneous, fine-grained mixture of crystalline phases can be produced.

Since the mechanical properties of a cast structure may be controlled by the interrelationship between the type and morphology of phases formed during cooling, it is of interest to predict the development of crystalline phases as a function of cooling rate and location. It is also of interest to relate mechanical properties to the size and shape of the various phases present and to the composition of these phases.

During the past reporting period a thermophysical model of the casting process has been developed which incorporates temperature-dependent properties of materials, the liberation of heat on going from liquid to solid (heat of solidification), and a capability to vary the properties of the mold, liquid, and solid independently of one another. This model has been utilized to study the effect of variations in material properties on the cooling behavior of a lunar basalt.

The results of experimental cooling rate studies, which were described in a prior report (Ref. 2-1), are described. The predictions of the thermophysical casting model are discussed in light of the results of the experimental cooling rate studies. The studies reported herein are part of an overall program to develop a greater knowledge of production methods for and properties of extraterrestrial silicate structures. Although the present work is discussed in the context of lunar materials and processes, the program scope also includes asteroids and Martian moons as sources of silicate materials for study.

2.2 SILICATE MELT SOLIDIFICATION STUDIES

The study of formation of silicate structures by casting from the melt had two objectives: (1) development of an understanding of the role of cooling rate on phase composition of typical extraterrestrial materials and (2) development of a model to predict the effect of material properties on the cooling rate during the casting process. The purpose of this phase of the silicate processing task was to develop a thermophysical model of the process of casting large structural elements. The rate of cooling of a casting was evaluated in terms of its thermal properties and of the properties of the mold into which it was cast.

The experimental studies of the effect of cooling rate on phase composition and grain morphology are discussed in Section 2.3. There are plans for additional work to evaluate mechanical properties as a function of phase composition and morphology.

2.2.1 Prior Work

The model which has been developed is similar in nature to those employed in studies of the solidification of magmatic rock in lava lakes and of solidification of molten rock formed, for example, by meteorite impact. Onorato, et al. (Ref. 2-2), have reviewed models which have been developed to predict cooling behavior of silicate melts. Both numerical and analytical methods were employed by the various investigators. These models suffer from lack of one or more of the following:

a) inclusion of latent heat of solidification on going from liquid to solid,

b) capability to vary the properties of the mold (basement) independently of the liquid and the solid being formed, and

c) use of temperature-dependent material properties.

Jaeger (Ref. 2-3) presented an excellent discussion of the cooling of igneous rocks. He provided information on models utilized up to the late 1960s and described case histories of cooling behavior of magmatic melts. The models discussed by Jaeger are analytical ones utilizing material properties which are independent of temperature. An analytical solution for the depth of crust formed at the free surface of a lava flow is presented and takes the form

$$X = C(kt)^{1/2} \tag{2-1}$$

where C is a dimensionless constant, k is the thermal diffusivity of the solid, and t is time.

It is noted (Ref. 2-3) that the role of convection in the cooling of magma is uncertain. Convective stirring may be caused by multiple or pulsating injection during flow as well as by differences in density within the body of magma. Jaeger states that it is believed that convection in the magma sets in after a short period in which the boundary layers solidify. Convection then becomes the primary heat transfer mechanism in the liquid lava.

The model of Provost and Bottinga (Ref. 2-4) is of particular interest since it contains many of the features of the current model. These authors evaluated the situation in which a sheet of lava was instantaneously emplaced on the surface of the Moon. The thickness of the lava was assumed to be small compared to its lateral dimensions so that a one-dimensional model could be formulated. The thermal conductivity of the molten lava was taken to be infinite; i.e., heat transfer through the liquid to the liquid/solid lava interface was assumed to be instantaneous. This is a valid assumption if convection occurs to such an extent that the overall liquid temperature is uniform, or nearly so.

The extent of crystallization during cooling below the liquidus temperature was accounted for by use of a relation observed to hold for solidification of Hawaiian lava lakes (Refs. 2-5 and 2-6), i.e.,

$$T = T_L - F(T_L - T_S) \qquad (2\text{-}2)$$

where T is the temperature of the liquid, T_L is the liquidus temperature, F is the fraction of flow which has crystallized, and T_S is the solidus temperature.

It is noted that the description of a unique liquid/solid boundary is simplistic in light of studies of Hawaiian lava lakes which demonstrate that the interface is diffuse, containing both liquid and solid, before giving way to a completely solid layer. This indicates that relatively strong convection does not take place in the diffuse interface and that, as a result, there will be a temperature gradient in the region where the liquid and crystal mix, once this region is established. The consequence of this is a slower rate of cooling than that predicted by assuming an isothermal liquid (due to convective stirring) and a discrete liquid/solid interface. However it has been demonstrated by earlier work that the predicted results from thermophysical solidification models based on the assumption of uniform liquid temperature are not significantly different than what is observed experimentally (Refs. 2-2, 2-3, and 2-4).

2.2.2 Casting Model

In the formulation of a model for the formation of a cast silicate slab, one may consider a layer of molten material of depth D and at a uniform temperature, T_0, that is poured into a mold initially at temperature T_{bo}. The surface of the material is allowed to radiate to space at 0 K. The surface emissivity is ε and the surface temperature, T_{su} varies with time. The molten silicate having a thermal conductivity of $K_\ell(T)$, a function of its temperature, heats the mold by conduction. The mold has a temperature-dependent thermal conductivity, $K_b(T)$. These relationships are illustrated in Figure 2-1.

The silicate loses energy at the upper or free surface by radiation and simultaneously by conduction through the lower interface with the mold. The conduction and convection processes within the molten silicate are assumed to ensure a uniform temperature within the melt. Provost and Bottinga (Ref. 2-4) showed that the temperature gradients set up by radiation at the free surface and by conduction to the mold are sufficient to generate free convection flows which in turn tend to mix the molten region and therefore ensure essentially a uniform-temperature melt. The molten silicate is cooled uniformly to the liquidus temperature at which point the first layers of solid form at the free surface and at the surface of the mold simultaneously. The rate of growth of the solidified layers is not the same, however.

The heat transfer mechanisms from the time of introduction of the molten silicate in the mold to the time when the first solidified layers appear are radiation at the free surface and unsteady conduction to the mold. In this case the temperature of the interface between the melt and mold is the same as the free surface temperature. Following the onset of solidification, conduction within the solid layers is included in the heat transfer processes. The solid/liquid interface within the cooling body is a moving source of heat generation as the heat of fusion is liberated upon solidification. The heat generated at the interface must travel through the solidified regions and either be dissipated at the upper surface by radiation or conducted into the mold from the lower solid silicate layer. Total solidification is achieved when the upper and lower layers meet.

With reference to Figure 2-2, where the temperature profile is schematically shown as a dashed line, the complete formulation of the problem will now be presented.

2.2.2.1 Before Solidification. Let z represent the distance in the downward direction. The free surface radiates to space according to the Stephan-Boltzmann law:

$$q_1 = -\varepsilon_\ell \sigma T_{su}^4 \qquad (2\text{-}3)$$

where ε_ℓ is the emissivity of the liquid, σ is the Stephan-Boltzmann constant, and T_{su} is the temperature at the free surface.

The heat conducted through the interface at the bottom of the melt into the mold is given by:

$$q_2 = K_\ell \left.\frac{\partial T}{\partial z}\right|_{z=D} \qquad (2\text{-}4)$$

The reduction of the melt temperature with time (before solidification) is obtained from the relation:

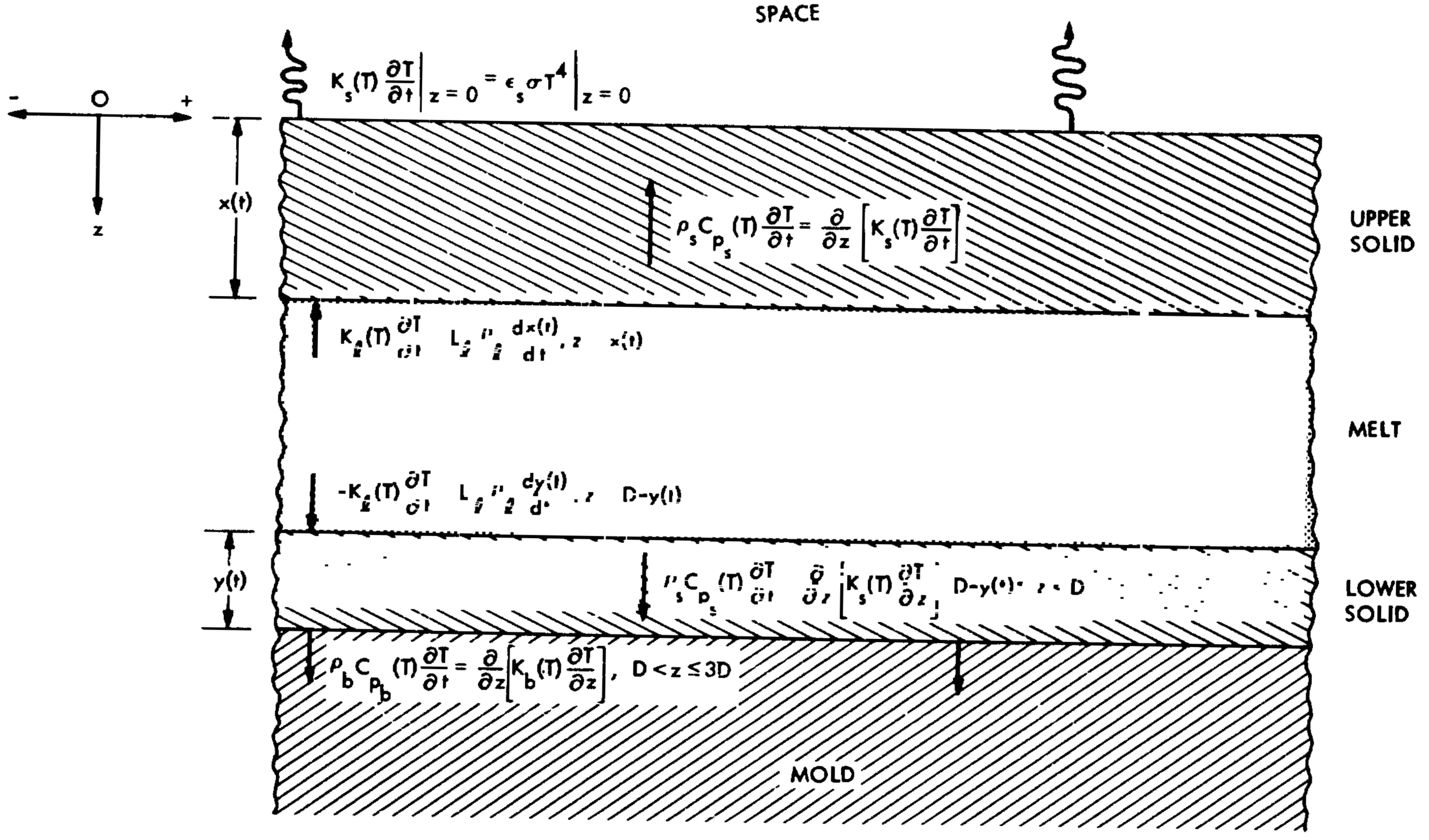

Figure 2-1. Thermodynamic Relationships in Casting Model

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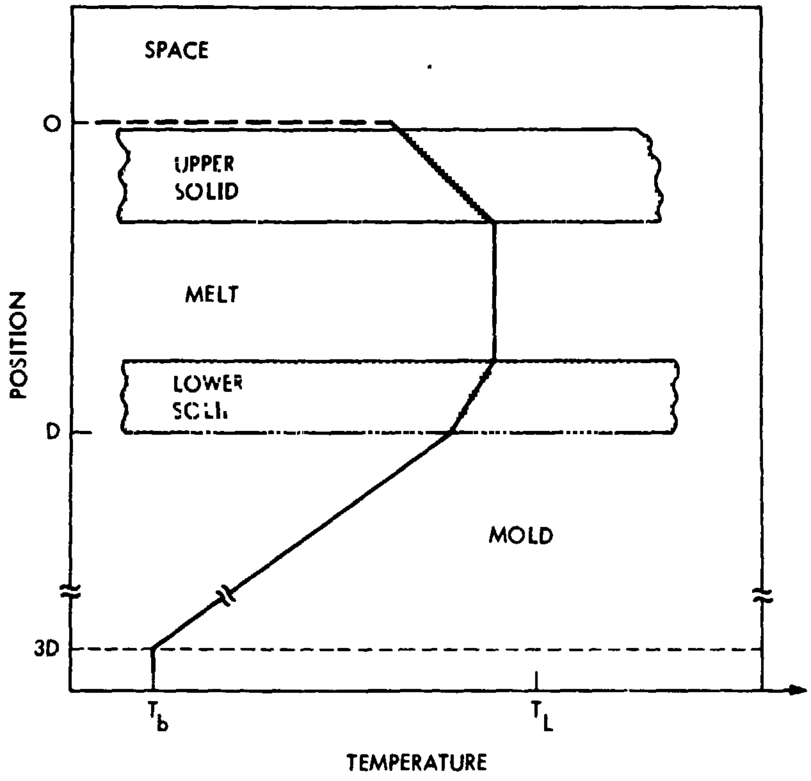

Figure 2-2. Schematic Representation of Temperature Profile in Silicate Casting as a Function of Position

$$D\rho_\ell C_{p(\ell)}\frac{\partial T}{\partial t} = q_1 + q_2 \qquad (2\text{-}5)$$

where D is the thickness of the molten material, ρ_ℓ is the liquid density, and $C_{p(\ell)}$ is the heat capacity of the liquid.

2.2.2.2 Upper Melt. Let x represent the position of the solid surface moving downward. For unit area of the slab, the heat transfer rate, q is:

$$q = \varepsilon_s \sigma T_{su}^4 \qquad (2\text{-}6)$$

Let the temperature distribution within the upper solid be represented by T = T(z,t). Then the heat conducted to the surface,

$$K_s \frac{\partial T}{\partial z}\bigg|_{z = 0} \qquad (2\text{-}7)$$

must be equal to the heat radiated to space; therefore

$$K_s \frac{\partial T}{\partial z}\bigg|_{z=0} = \varepsilon_s \sigma T_{su}^4 \qquad (2\text{-}8)$$

Within the solid the conduction equation is:

$$\rho_s C_{p(s)} \frac{\partial T}{\partial t} = \frac{\partial}{\partial z} K_s \frac{\partial T}{\partial z} \qquad (2\text{-}9)$$

where ρ_s is the density and $C_{p(s)}$ is the heat capacity of the solid.

At the upper melt surface the heat liberated during solidification will be conducted to the solid according to the relationship

$$K_s \frac{\partial T}{\partial z}\bigg|_{z=x} = L\rho_\ell \frac{dx}{dt} \qquad (2\text{-}10)$$

where ρ_ℓ is the density of the liquid and L is the latent heat of fusion. Furthermore the temperature boundary condition at

$$z = x \text{ is } T \approx T_\ell \qquad (2\text{-}11)$$

2.2.2.3 Lower Melt. Let y represent the thickness of the solid surface moving up. Equation (2-9) is valid for the region within the solid and the temperature boundary condition at z = y is $T \quad T_\ell$.

At the lower solid/liquid interface Equation (2-10) prevails but with y variable, or:

$$-K_s\frac{\partial T}{\partial z}\bigg|_{z=D-y} = L\rho_\ell\frac{dy}{dt} \qquad (2\text{-}12)$$

From the principle of conservation of energy the heat flux from the lower solid must equal the heat conducted to the mold; i.e.,

$$-K_b\frac{\partial T}{\partial z}\bigg|_{z=D} = K_s\frac{\partial T}{\partial z}\bigg|_{z=D} \qquad (2\text{-}13)$$

Within the mold

$$\rho_b C_{p(b)}\frac{\partial T}{\partial t} = \frac{\partial}{\partial z} K_b\frac{\partial T}{\partial z} \qquad (2\text{-}14)$$

where ρ_b, $C_{p(b)}$ and K_b are density, heat capacity, and thermal conductivity of the mold, respectively. And finally, continuity of temperature requires that

$$T_b\Big|_{z=D} = T_s\Big|_{z=D} \qquad (2\text{-}15)$$

A computer code was developed to evaluate the heat transfer during cooling of the molten silicate slab from above its liquidus temperature to below its solidus temperature. The code uses a standard finite difference analysis approach to numerically analyze the thermodynamic partial differential equations (shown above) that describe the temperature dynamics. A one-dimensional model is used, with the mold considered to be semi-infinite in the two remaining dimensions. Table 2-1 is a list of the system parameters which may be varied.

The numerical solution by finite difference analysis allows one to take into account the latent heat of solidification (fusion) which is liberated on going from liquid to solid. Temperature-dependent material properties can also be included in the model. In addition the numerical method used in this model permits one to vary the properties of the liquid, solid, and mold independently of one another. This model thus provides a means to assess the impact of a variation in the properties of the casting or mold material on the temperature profile and rate of cooling. For example, the significance of an uncertainty in a particular material property on the cooling behavior of a casting can be determined. Based upon this determination a decision can then be made as to whether a need exists to have a greater knowledge of that particular property.

Table 2-1: Computer Code System Parameters for Model of Casting of Molten Silicates.

---

1) Thermal properties of molten state

   a) Emplaced (initial) temperature
   b) Liquidus temperature
   c) Solidus temperature
   d) Density
   e) Thermal conductivity (as a function of temperature)
   f) Heat capacity (as a function of temperature)
   g) Latent heat of fusion
   h) Emissivity

2) Thermal properties of solid state

   a) Density
   b) Thermal conductivity (as a function of temperature)
   c) Heat capacity (as a function of temperature)
   d) Emissivity

3) Thermal properties of lunar basement

   a) Initial temperature before molten basalt emplaced
   b) Density
   c) Thermal conductivity (as a function of temperature)
   d) Heat capacity

4) Resolution of finite difference analysis

   a) Thickness of mold
   b) Number of differential spacial nodes
   c) Number of differential temporal nodes

5) Cooling time limit (absolute or completely solid)

6) Hardcopy format

2.2.3 Evaluation of the Model

After the model was established, it was debugged by checking its results against those obtained by Provost and Bottinga (Ref. 2-4). The work of Provost and Bottinga was particulary appropriate in this regard since they evaluated cooling behavior of lava flows in both lunar and terrestrial environments.

The materials property data utilized by Provost and Bottinga was input to the model and the thermal history of a 50-cm-thick slab was determined. The temperatures at points 0, 2, 4, 10, and 50 cm below the free surface of the slab are shown as a function of time in Figure 2-3. Temperature as a function of time taken from Figure 2 of Provost and Bottinga (Ref. 2-4) is shown by the ● for depths of 0 and 10 cm from the free surface. The correspondence of the current model with that of Provost and Bottinga is excellent.

The advantage of the current model over that of Provost and Bottinga is that the heat flow properties of the mold ("basement" in the Provost and Bottinga paper) can be varied independently of the properties of the solid and liquid basalt. The finite difference technique used in the determination of heat transfer into the mold in the current model is felt to be more precise than the error function utilized by Provost and Bottinga.

2.2.3.1 Material Properties. The material properties that affect the rate of solidification of a large casting of molten material are listed in Table 2-2. The properties of the mold used to form the casting must be considered as well as those of the liquid and solid casting.

Table 2-2. Material Properties That Affect the Rate of Solidification of Cast Molten Silicates.

Property	Solid	Liquid	Mold
bulk density	X	X	X
thermal conductivity	X	X	X
heat capacity	X	X	X
latent heat of solidification	-	X	-
emissivity	X	X	-

Data for lunar basalt was taken from the literature when available. In some cases there was a range of values for a particular property. In such instances a "most likely value" was selected. In other cases there was no data available and material property data on analogous terrestrial materials were utilized.

The bulk density of a medium-grained porphyritic basalt with no porosity (lunar rock sample 12002,85) was determined to be 3.31 gm/cm^3 (Ref. 2-7). The bulk densities of a fine-grained lunar basalt rock (70215) and a porphyritic lunar basalt rock (74275) were reported as 3.37 gm/cm^3 and 3.36 gm/cm^3, respectively (Ref. 2-8). For the purposes of this study the highest reported density (3.37 gm/cm^3) was selected for the model material. It was assumed that the density of the liquid and solid were the same and that density was independent of temperature.

The density of the lunar soil has been determined at several locations through return of core-tube samples. The Apollo 16 probe brought back a sample having a density of approximately 1.5 gm/cm^3 (Ref. 2-9). The average density of an Apollo 11 core-tube sample was approximately 1.64 gm/cm^3,

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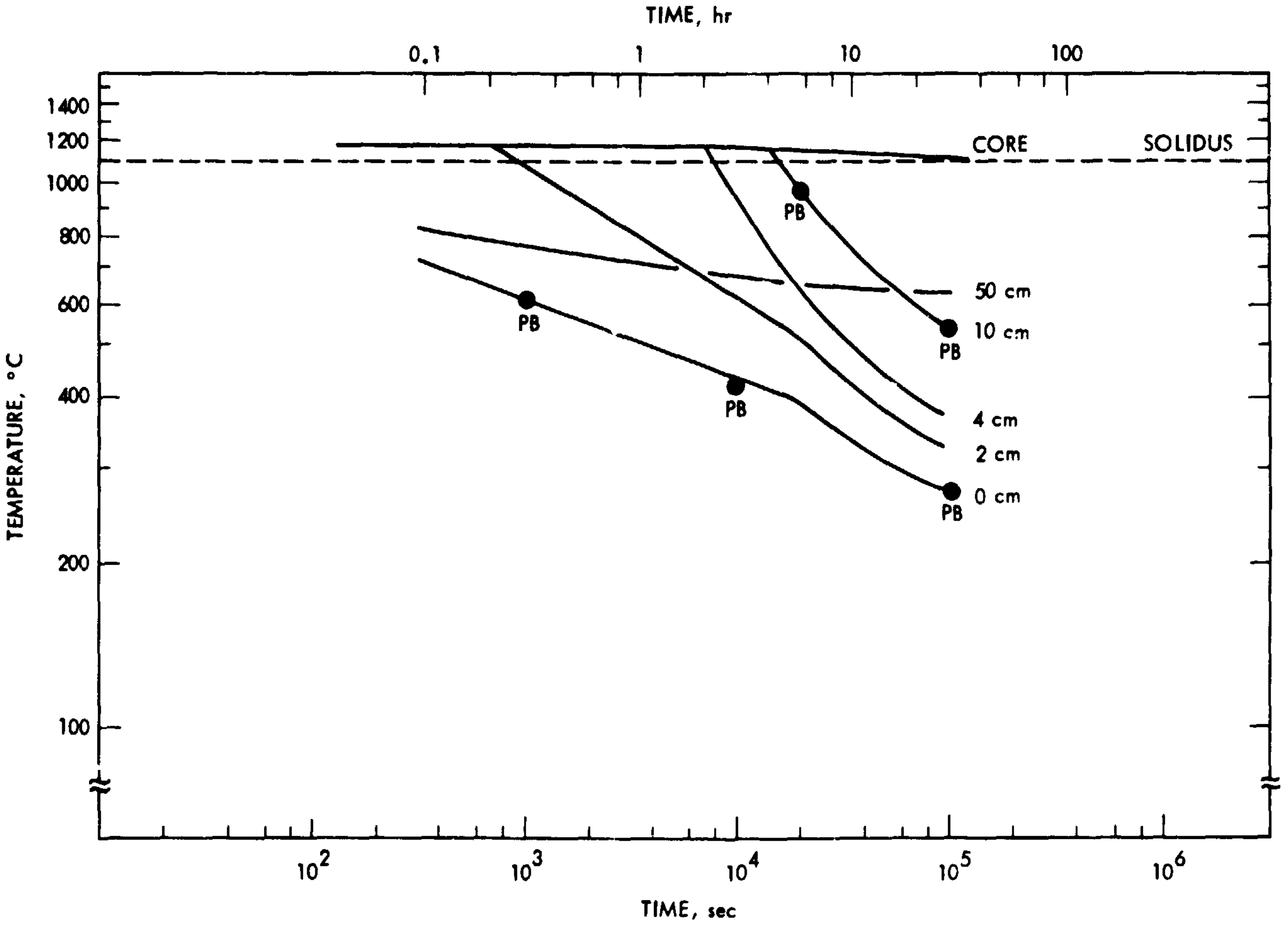

Figure 2-3. Comparison of Current Model with That of Provost and Bottinga (Ref. 2-4)

while the densities of the Apollo 12 core tube samples were reported to range between 1.74 and 1.98 gm/cm^3 (Ref. 2-10).

Thermal conductivity of lunar soil has been measured by a number of investigators (Refs. 2-11 to 2-14). The effect of soil density and composition on thermal conductivity of a number of lunar and terrestrial soil samples is discussed in the review by Cremers (Ref. 2-12). This thermal conductivity data is illustrated in Figure 2-4.

Thermal conductivity was measured on a synthetic lunar magmatic rock and Columbia River basalt over the temperature range 25° to 1500° C (Ref. 2-13). These data are shown in Figure 2-4 as SLB (synthetic lunar basalt) and CRB (Columbia River Basalt). The range of values for lunar soil reported by Cremers (Ref. 2-12) are illustrated by the hatched region in the upper left-hand corner of Figure 2-4. The thermal conductivity-temperature relationship used in the current investigation is indicated by the line marked "model". This relationship has the form thermal conductivity, $K = 4.619 \times 10^{-3} - 2.05 \times 10^{-6}T$ [cal.cm^{-1}.sec^{-1}.°C^{-1}]. The thermal conductivity of the molten basalt is taken to follow the same relationship as that of the solid.

Data on lunar soil are scattered over a wide range as illustrated in Figure 2-4. Data for thermal conductivity of lunar soil were not available above 150° C. Since the data for thermal conductivity of the lunar soil have been found to fit the relationship $K = A + BT^3$, this expression has been used to extrapolate the available data to 800° C as indicated by the dotted lines in Figure 2-4.

The relationship between thermal conductivity and temperature chosen for the model is illustrated in Figure 2-4 by the line marked "soil model". This curve has a shape different from that of the others because it has a linear relationship with temperature of the type, $K = A + BT$ rather than the cubic dependence on temperature followed by the lunar soil samples.

The linear temperature dependence was chosen so as to conform to the type of relationship used for the lunar rock. This simplifies the finite difference model used in calculation of heat flow across the interface between the case basalt and the lunar soil used as the mold. There is no loss of accuracy in utilizing such a relationship since the scatter in the available thermal conductivity data is so large.

The relationship for heat capacity used in this model is one calculated by Horai, et al. (Ref. 2-14), for Apollo 11 sample 10020, a fine-grained, vesicular rock. The latent heats of fusion for lunar materials are unknown. Data on terrestrial minerals (Ref. 2-15) indicate that values range from approximately 100 cal/gm to 150 cal/gm with the higher values attributed to the more refractory minerals. It is reasonable to expect that basalt, with a relatively low liquidus temperature, would also have a relatively low latent heat of fusion. Thus, a value of 100 cal/gm has been selected for use in the model. This value was also used by Provost and Bottinga (Ref 2-4) in their thermophysical model.

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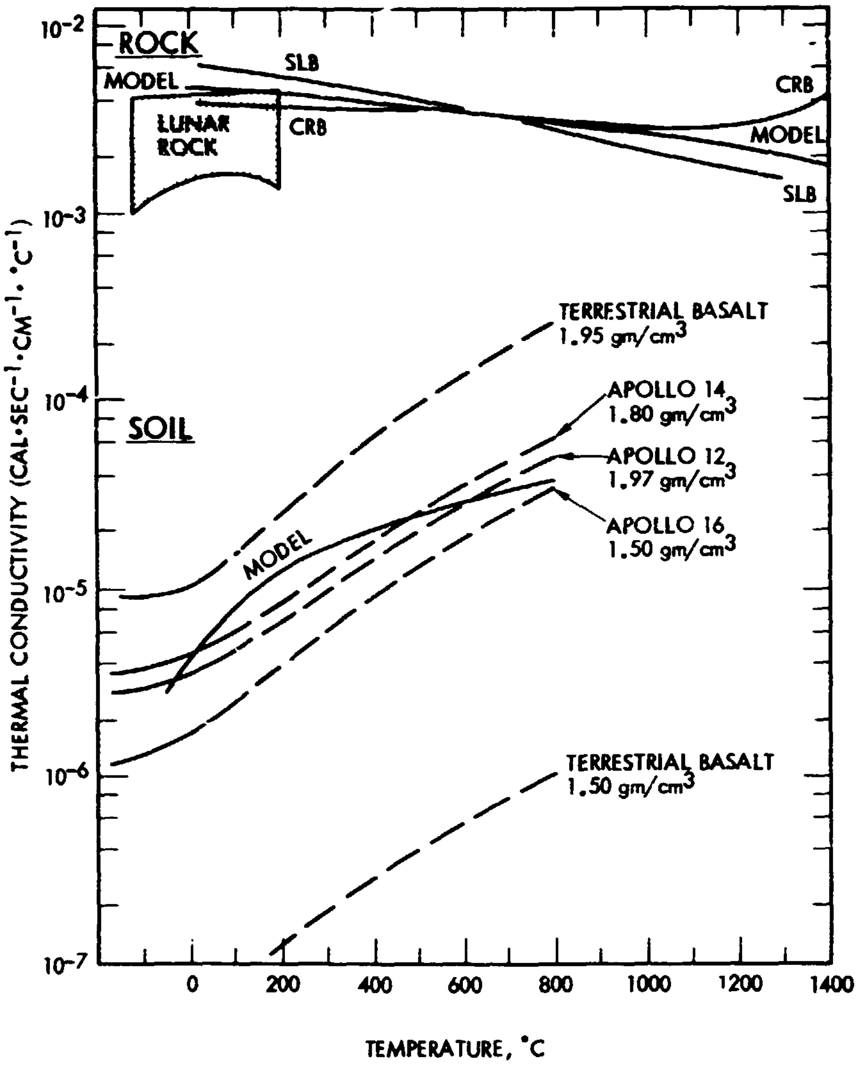

Figure 2-4. Temperature Dependence of the Thermal Conductivity of Lunar and Terrestrial Rock and Soil

The emissivity of some analogue lunar materials has been determined by Arndt, et al. (Ref. 2-16). Values for a lunar basalt ranged from approximately 0.72 to 0.75 over the temperature range 700° to 1250° C. Below 700° C the emissivity increased rapidly to a value of 0.94 at approximately 620° C. A value of the emissivity of 0.74 has been selected for use in this study and represents the lower limit expected for basalt.

2.2.3.2 Convection. There is some question as to the occurrence of convective stirring and its role on cooling rate in a slab of molten silicate. In the observing of lava flows, the differentiation is usually observed and this implies mass transport by convection. However, these flows generally have thicknesses of 100 m or more. Fluid instability due to thermal gradients between the top and bottom of a liquid layer is predicted to occur when the Rayleigh number for that liquid is greater than approximately 1700 (Ref. 2-17). The Rayleigh number, R, is given by the relation (Ref. 2-3, 2-17)

$$R = \rho g a^3 \alpha \Delta T / \eta k \qquad (2\text{-}16)$$

where ρ is the density of the liquid, g is the acceleration due to gravity, a is the layer thickness, α is the coefficient of thermal expansion, ΔT is the temperature difference across the layer, η is the liquid viscosity and k is the thermal diffusivity.

In the present model, using typical values of the material properties as listed in Table 2-3, the value for the Rayleigh number is calculated to be $8.7 \times 10^{-4}\Delta T$, $35\Delta T$, and $2365\Delta T$, at temperatures of 1110° C (just above the solidus temperature), 1150° C, and 1200° C (the liquidus temperature), respectively. The above values suggest that when the greater part of the casting has solidified and the temperature of the liquid remaining is near the solidus temperature there will not be convective stirring of the liquid. Large temperature gradients in the liquid can be tolerated and heat transfer through the liquid will be primarily due to conduction.

At temperatures near the liquidus even small temperature differences, ΔT (on the order of 1° C), between top and bottom of the layer are sufficient to promote convective stirring. In the beginning of the cooling process then, the rate of heat transfer through the liquid to the liquid/solid interface will be controlled by convective stirring. However, as the liquid cools to near 1150° C and below, theory predicts that the temperature difference required for convection to take place must be greater than approximately 50° C.

During the initial part of the cooling process the temperature is, to a good approximation, uniform over the entire thickness of the molten basalt. Thus the assumption used in the model, that the temperature through the liquid is everywhere the same, agrees with predictions based on the theory of hydrodynamics. During the initial build up of the solid at the free surface and the bottom surface the model conforms with the expected behavior of the liquid. As the temperature drops and material begins to solidify the viscosity increases exponentially and the thickness of the liquid layer

Table 2-3. Typical Properties of Lunar Basalt Casting at Temperatures Above the Liquidus

	Temperature		
	1110° C	1150° C	1200° C
Density, gm/cm^3	3.37	3.37	3.37
Thickness of liquid,[a] cm	8	27	50
Thermal Expansion Coefficient, $°C^{-1}$	4×10^{-6}	4×10^{-6}	4×10^{-6}
Viscosity, poise	500,000[b]	500[b]	50[d]
Thermal Diffusivity,[c] cm^2/sec	2.64×10^{-3}	2.52×10^{-3}	2.38×10^{-3}
Acceleration Due to Lunar Gravity,[d] cm/sec^2	167	167	167
Rayleigh Number	$8.7 \times 10^{-4}\Delta T$	$35\Delta T$	$2365\Delta T$

[a] Determined from computer model.

[b] Estimated based on data given by Jaeger, 1968 (Ref. 2-17).

[c] Calculated from thermal conductivity, heat capacity and density data.

[d] Ref. 2-4.

decreases. Both factors tend to stabilize the liquid. At some point, most likely around 1150° C in the current case, based on Rayleigh number calculations, the convective flow is largely suppressed and heat transfer in the liquid primarily occurs by conduction. At this point the current model begins to overestimate the rate of heat loss.

The assumption of uniform temperature in the liquid provides for a good approximation of the predicted situation, particularly at the beginning of the solidification process where the surface layers are formed, while it provides for simplification of the mathematical model of the heat transfer process.

2.2.3.3 Results. A baseline set of properties was obtained from the literature as discussed above and utilized in establishing a baseline cooling rate condition for lunar basalt. The values of the properties used for the baseline are listed in Table 2-4.

The cooling history of a 50 cm thick slab of lunar basalt was evaluated over the temperature range of 1250° C (50° C above the liquidus) to 1100° C, the solidus temperature. The change in temperature with time and position in the cast piece and mold is illustrated in Figure 2-5. Cooling from 1250° C to the liquidus temperature, 1200° C, takes place in 193 sec (0.054 hr). The liquid basalt then cools to 1150° C in approximately 14 hr and is completely solid 49.8 hr after cooling commences. Note that while the first 10 cm of material below the free surface of the cast piece are below 500° C at the time of complete solidification, the bottom surface (50 cm) is at a temperature slightly above 1000° C. These results demonstrate that

Table 2-4. Baseline Material Properties Used in Establishment of Fusion Casting Behavior of Lunar Basalt.

Liquid	
Liquidus Temperature	1200° C
Solidus Temperature	1100° C
Density	3.37 gm.cm^{-3}
Thermal Conductivity	$4.619 \times 10^{-3} - 2.05 \times 10^{-6}T$ $\text{cal.cm}^{-1}.\text{sec}^{-1}.°\text{C}^{-1}$
Heat Capacity	$0.213 + 4.92 \times 10^{-5}T - 5.00 \times 10^{3}T^{-2}$ $\text{cal.gm}^{-1}.°\text{C}^{-1}$
Heat of Fusion	100 cal.gm^{-1}
Emissivity	0.74
Solid	
Density	3.37 gm.cm^{-3}
Thermal Conductivity	$4.619 \times 10^{-3} - 2.05 \times 10^{-6}T$ $\text{cal.cm}^{-1}.\text{sec}^{-1}.°\text{C}^{-1}$
Heat Capacity	$0.213 + 4.92 \times 10^{-5}T - 5.00 \times 10^{3}\ T^{-2}$ $\text{cal.gm}^{-1}°\text{C}^{-1}$
Emissivity	0.74
Mold	
Density	1.95 gm.cm^{-3}
Thermal Conductivity	$4.25 \times 10^{-6} + 3.96 \times 10^{-8}T$ $\text{cal.cm}^{-1}.\text{sec}^{-1}.°\text{C}^{-1}$
Heat Capacity	$0.213 + 4.92 \times 10^{-5}T - 5.00 \times 10^{-3}T^{-2}$ $\text{cal.gm}^{-1}.°\text{C}^{-1}$

equipment having high temperature capability will be required to handle cast pieces, even after solidification is complete. The alternative, of course, is to allow the cast piece sufficient time to cool before removing it from the mold. The model provides the information on temperature profiles required to perform "tradeoff" studies as to the rate of formation of fusion-cast slabs and the time available for production.

Due to the rapid rate of heat transfer by radiation at the free surface a large thermal gradient is established between the free surface and the center of the cast piece. For example, the material 2 cm below the free surface is below 800° C after 1/2 hour of cooling while the center of the body is above 1190° C. While the material is above 1100° C it will deform viscously as the outer solid layers contract during cooling. However, on reaching 1100° C the material in the center becomes a rigid solid and can no longer deform. At this point the material in the center must cool over a greater temperature range than the material at the surface. Thus, the greater amount of contraction in the center relative to the surface will cause strain to build up in the structure. This strain, if not relieved, will give rise to stresses which will will be compressive in nature at the surface and tensile in nature in the interior.

A compressive stress at the surface will actually strengthen the cast body. However, if the strain is too large for the material to accommodate, cracking will occur. The formation of cracks will relieve the strain (and thus the stresses) in the material. It will also weaken the structure as a whole. More importantly, the cracks may propagate through the body and cause mechanical failure.

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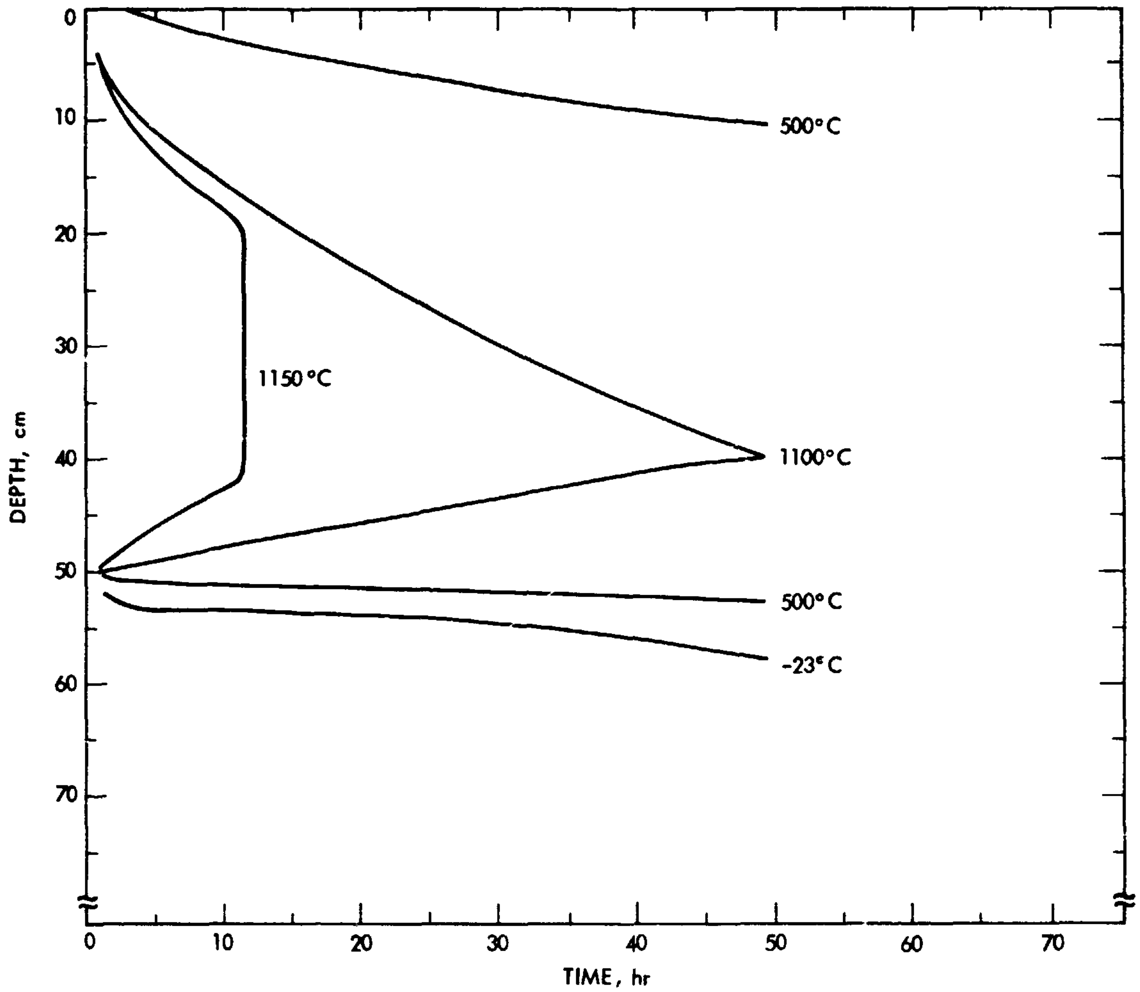

Figure 2-5. Baseline Case: Isotherms in Casting During Cooling from 1250° to 1100° C

There is another mechanism for development of strain which results in stress in the body on a microscopic scale. In this case there is a mismatch in thermal expansion coefficient between two or more adjacent phases in the same region in the structure. The difference in contraction of the different phases during cooling will cause strain across the phase boundaries. These localized strains result in stresses which can cause microcracking to occur. The microcracks become defects which may lower the overall strength of the cast body. In order to produce useful shapes, it will be necessary to control the microstrain in the body by, for example, control of phase composition and grain size.

2.2.3.4 Lower Thermal Conductivity. The data selected for the baseline model is a compilation of the best information available from the literature. However, it is of interest to determine the effects of variations in material properties on the thermal history and temperature distribution of a fusion cast lunar basalt. Accordingly, the thermal conductivity of the liquid and solid basalt in the computer model was decreased by an order of magnitude. This level is expected to represent the lower limit for the thermal conductivity of lunar basalt.

The results of cooling the basalt having lower conductivity from 1250° C to 1100° C are illustrated in Figure 2-6. There is no difference in time required for cooling the baseline basalt and the lower thermal conductivity basalt from 1250° C to 1200° C. The heat loss in both cases is due to radiation in space and is not influenced by variations in conductivity of the liquid.

The interior of the cast piece reaches 1150° C in approximately 68 hr and reaches 1100° C in approximately 326 hr. The time required to reach 1150° C and 1100° C is greater than that for the baseline by a factor of 4.9 and 6.5, respectively. While the time required for complete solidification to occur is sensitive to the thermal conductivity of the liquid and solid, the increase in time has less than a 1:1 correspondence with the change in magnitude of the thermal conductivity.

The temperatures of the upper (free) and lower surfaces of the lower-thermal-conductivity (LTC) cast basalt after complete solidification are lower than those of the baseline casting as shown in Table 2-5. Even so, the temperature at the cast basalt/mold interface is high enough (948° C) that special equipment will be necessary to handle the cast piece unless additional time is allowed for cooling to take place.

2.2.3.5 Effect of Conductivity of the Mold. The material property having the greatest degree of uncertainty is that of the thermal conductivity of the mold. Thermal conductivity is a function of material density as well as chemistry. The density can be effected by mechanical operations (packing to higher density or by turning the mold material to achieve lower density).

Because of this uncertainty it is instructive to observe the role of thermal conductivity of the mold on the time required to cool molten lunar basalt to its solidus temperature. Two cases were examined. In one case the

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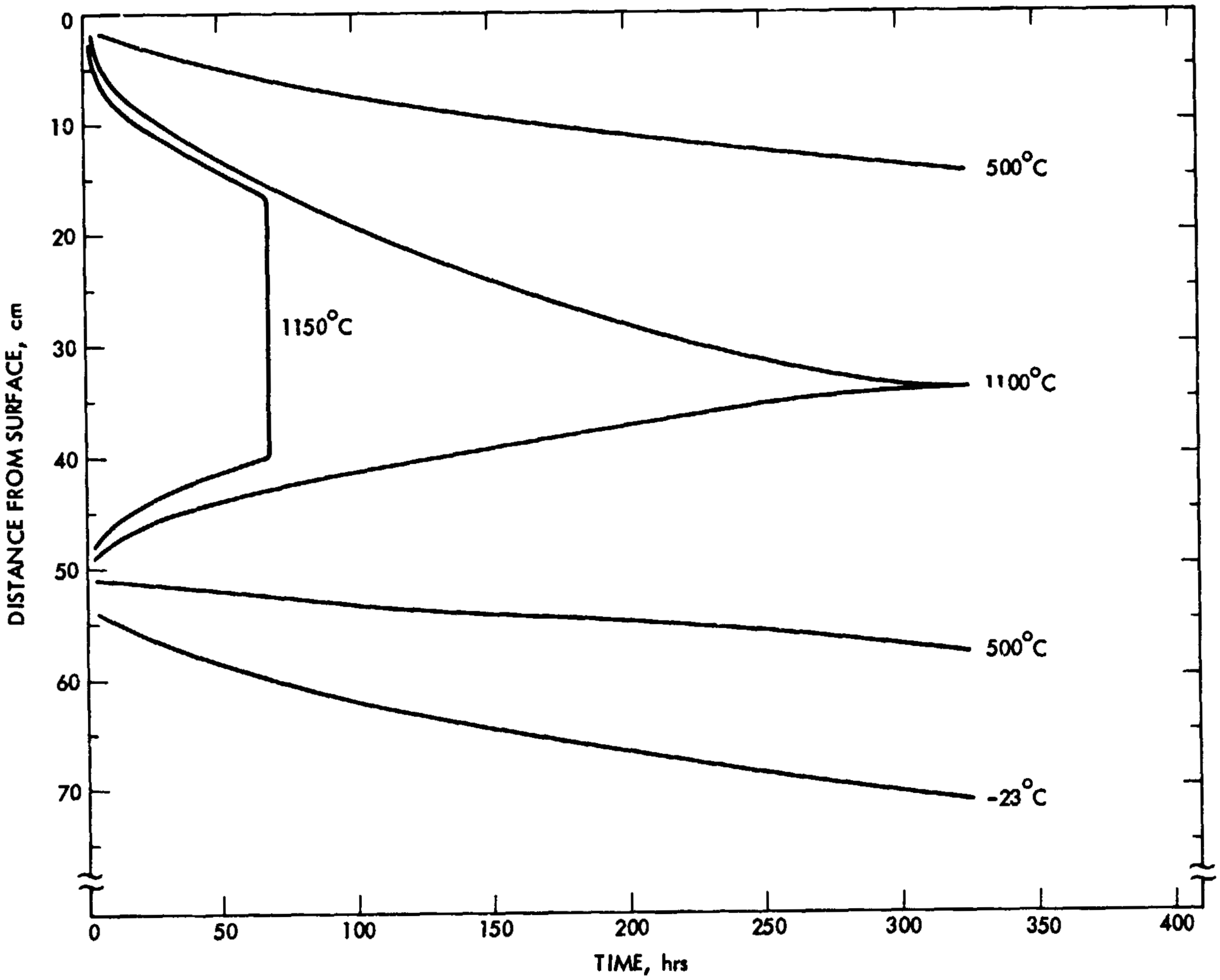

Figure 2-6. Low Thermal Conductivity Basalt Case: Isotherms in Casting During Cooling from 1250° to 1100° C

Table 2-5. Surface and Near-Surface Temperatures, °C, of Baseline Basalt and the Low Thermal Conductivity Basalt Castings at the Time of Complete Solidification.

	Depth from Free Surface, cm				
	0	2	48	50	52
Baseline Basalt	269	312	1067	1061	673
LTC Basalt	62	119	965	948	829

thermal conductivity of the mold was increased over baseline by an order of magnitude and in the second case the thermal conductivity of the mold was decreased by an order of magnitude from baseline.

Figure 2-7 illustrates the effect of decreasing the thermal conductivity of the mold by a factor of 10 on the isotherms in the 50-cm-thick casting. The isotherms are plotted as a function of position below the free surface and of time. Above the center of the cast basalt the location of the 500°, 1000°, and 1100° C isotherms of the low thermal conductivity case are similar in shape and close in position to those obtained in the baseline and high thermal conductivity cases. Below the center of the cast piece, however, a change in thermal conductivity of the mold markedly affects the isotherms. Comparatively little heat is conducted into the mold in the low conductivity case, as indicated by the -23° C isotherm remaining constant at a location approximately 4 cm beneath the surface.

The "envelope" of probable locations of the solidus (1100° C) isotherm in time and depth in the system is illustrated in Figure 2-8. The 1100° C isotherms for the two extremes in thermal conductivity of the mold (10^{-1} times baseline and 10 times baseline) are shown as the outer and inner bounds on the envelope. The dotted line within the envelope is the solidus isotherm for the baseline case.

The time required for cooling the central core of liquid in each of the cases is listed in Table 2-6, with the results of the baseline run included for comparison. Cooling begins on emplacement of the molten basalt at 1250° C into the mold at -23° C. The cooling times reflect the changes in thermal conductivity of the mold. As with the role of thermal conductivity of the liquid and solid basalt, however, there is less than a 1:1 correspondence of cooling time with conductivity change. This is to be expected of course, since a decrease in the thermal conductivity of the mold will result in a greater percentage of heat transfer via radiation into space. Likewise, an increase in thermal conductivity of the mold will increase the amount of heat being carried away by the mold. But since the heat being dissipated by the mold is only a fraction of the total, the fractional amount of change in time will be small compared to the change in thermal conductivity.

The effect of thermal conductivity of the mold on surface and near-surface temperatures at the moment of complete solidification of the cast basalt can be seen in Table 2-7. The temperature of the free surface ranges

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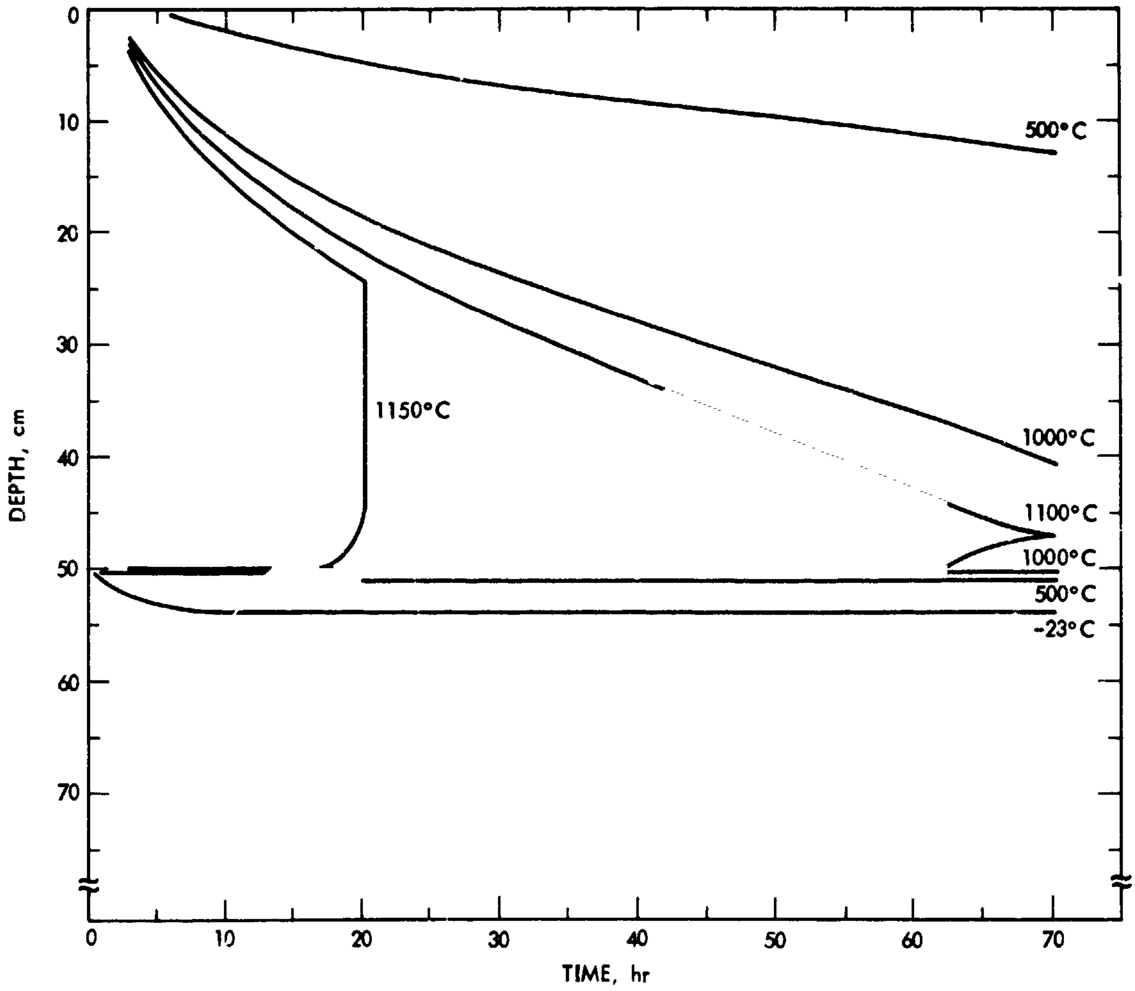

Figure 2-7. Low Thermal Conductivity Mold Case: Isotherms in Casting During Cooling from 1250° to 1100° C

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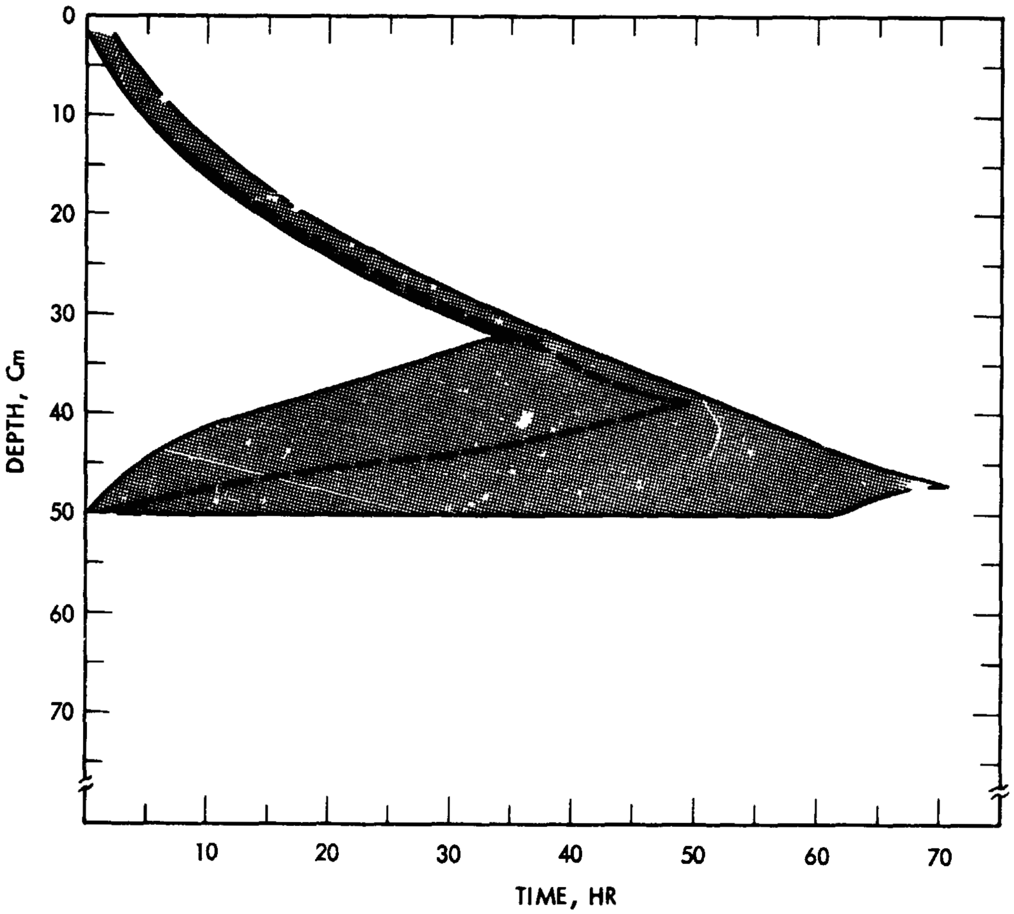

Figure 2-8. Solidus Isotherms for Basalt During Cooling in a Mold Which Has: (a) Baseline Thermal Conductivity (Dashed Line), (b) Thermal Conductivity 10^{-1} x Baseline (Outer Bound), and (c) Thermal Conductivity 10 x Baseline (Inner Bound)

Table 2-6. Time Required for Cooling the Liquid Core to Selected Temperatures.

Temperature, °C	Cooling Time		
	10^{-1} x Baseline	Baseline	10 x Baseline
1200	0.49 hr	0.05 hr	0.006 hr
1150	20.1 hr	11.7 hr	7.6 hr
1100	70.5 hr	49.8 hr	37.6 hr

from 251° C to 289° C. At a point 4 cm below the free surface the temperatures are still moderate, ranging from 326° to 383° C. The surface in contact with the mold (at 50 cm), however, is above 900° C in all cases. Again, special equipment will be required to handle material with temperatures of this magnitude.

Table 2-7. Temperature, °C, of the Outer 4 cm of the Basalt Casting at the Moment of Complete Solidification.

Distance from Free Surface, cm	10^{-1} x Baseline	Baseline	10 x Baseline
0	251	269	289
2	288	212	344
4	326	355	383
46	1080	1072	973
48	1085	1067	958
50	1082	1061	942

An interesting feature of the low thermal conductivity mold material is that the solidus isotherm (1100° C) is located slightly below the surface of the mold for times less than approximately 62 hr. In all other situations evaluated in this study a solid skin formed rapidly at each surface and this skin grew in thickness as solidification proceeded. The presence of the skin will permit the cast piece to be handled (using special equipment) before solidification is complete. The casting could be removed to a storage area, thereby freeing up the mold for additional casting operations. Such a skin will not be present if the thermal conductivity of the mold is approximately 10 times less than the thermal conductivity used for the baseline cooling study. Thus the cast piece could not be removed early, thereby tying up the mold for 60-plus hours.

Another consequence of the solidus temperature being below the casting/mold interface is that lower melting components of the mold will be taken into the melt. The composition of the lower surface layer of the casting will thus be affected by differences in composition between the casting and the material from which the mold is made. If there are differences in composition between the mold and the casting, the layer of material at the casting/mold interface will take on a composition different from the bulk casting. This difference will affect the phase composition and may affect the mechanical properties of the casting.

The thermal conductivity of the mold markedly affects the rate of cooling of the liquid basalt. Low conductivity molds lead to lower rates of cooling. The texture of the basalt microstructure will be affected by the change in cooling rate. Since the rate of cooling at the free suface is much less than that of the mold/basalt interface, the surface layer of the casting would be expected to have finer texture. The phase composition at the surface may also be different than that at the casting/mold interface. Equilibrium phases would appear at the slow cooling casting/mold interface whereas nonequilibrium phases are likely to occur at the free surface.

2.2.3.6 Effect of Emissivity. The value of emissivity of the cast silicate may be expected to vary with composition and surface roughness. The amount of heat radiated to free space from the surface of the casting is proportional to emissivity times temperature of the surface raised to the fourth power. Since the heat radiated from the surface is a function of emissivity it is of interest to determine the effect which variations in emissivity would have on the overall rate of temperature change.

The practical upper limit of emissivity of basalt is expected to be approximately 0.94 (Ref. 2-16). Use of $\varepsilon = 0.94$ in the model instead of the most likely value of 0.74 results in very little change in overall cooling time for the casting. The solidus isotherms for both the standard basalt and the high emissivity basalt are shown in Figure 2-9. The solidus isotherm for the high emissivity basalt is approximately 1/2 cm closer to the center of the casting than that for the standard basalt for the same cooling time. The times required for cooling to the liquidus and solidus temperatures for the standard basalt ($\varepsilon = 0.74$) are 193 sec and 179,200 sec (49.78 hr), respectively. Similar values for basalt with an emissivity of 0.94 are 192 sec and 172,250 sec (47.85 hr), respectively.

The data demonstrate that for material with an emissivity greater than approximately 0.74 the rate-limiting step in the heat-transfer process is not the radiation of heat from the surface. Rather, the rate of heat transfer through the solid and the mold are far slower modes of heat removal and as such control the overall cooling rate.

At present the thermal history of the cast basalt has been determined over the time period required for the last remaining liquid to reach the solidus temperature. The model, however, is quite general in nature and can be used to follow cooling behavior of any material for which sufficient property data are available. The model is thus applicable to analysis of a wide range of both extraterrestrial and terrestrial situations.

2.3 EXPERIMENTAL STUDIES

2.3.1 Introduction

The objectives of the liquid-phase-production experimental studies are (a) to develop an understanding of the kinetics of crystallization of extraterrestrial silicates, (b) to develop an understanding of the effect of cooling rate on texture and phase-composition of extraterrestrial silicates

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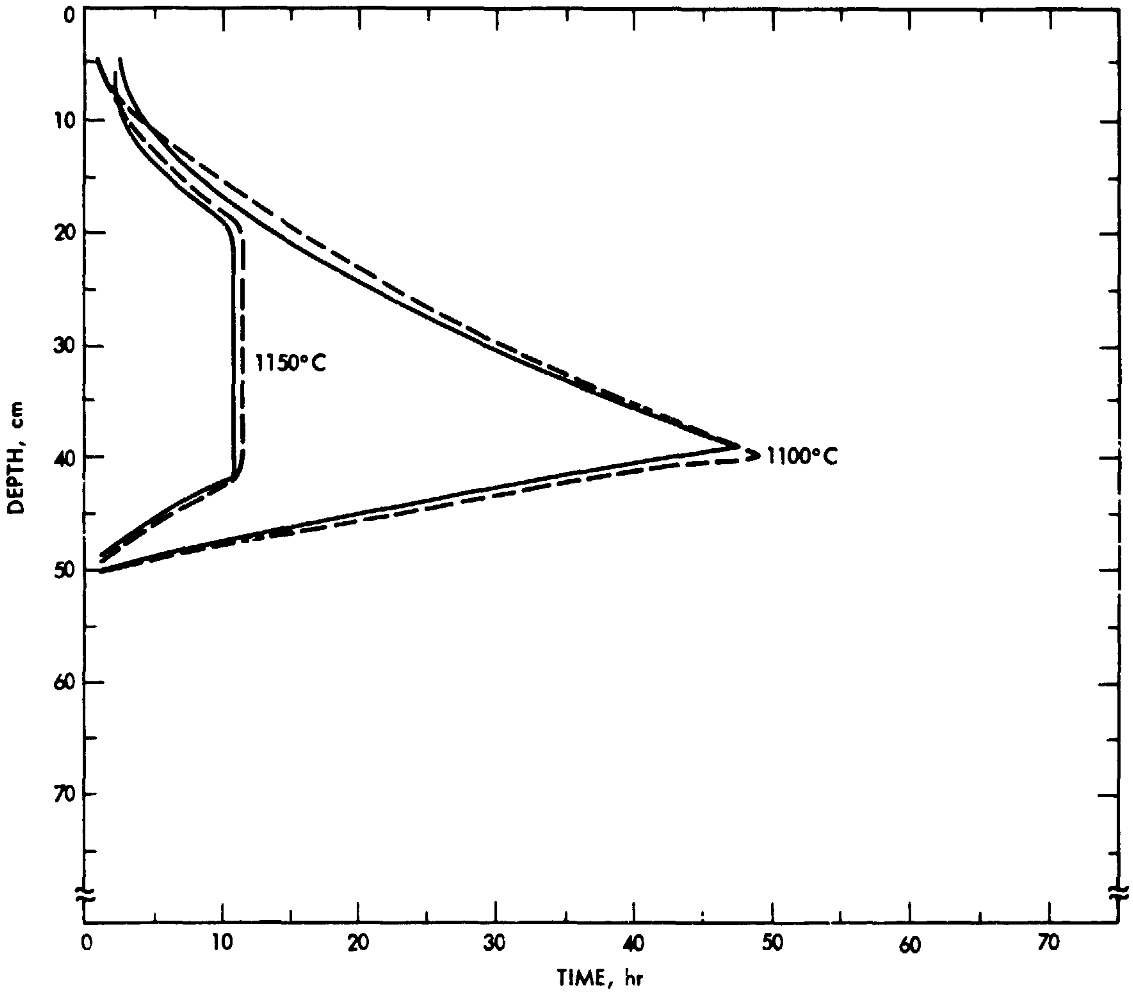

Figure 2-9. 1150° C and 1100° C (Solidus) Isotherms for Baseline Basalt (Dashed Line) and High Emissivity Basalt (Solid Line)

cooled from the molten state, and (c) to evaluate the properties of extraterrestrial materials produced from the molten state.

The experimental procedure for the production of mock extraterrestrial materials and their heat treatment was described in Reference 2-1. Briefly, extraterrestrial materials were simulated with a mixture of terrestrial minerals. The mock extraterrestrial materials formed were (1) a mock lunar maria basalt, (2) a mock lunar highlands anorthosite, and (3) a mock ordinary chondrite (meteorite composition).

Each of the three mock materials was placed in a closed molybdenum crucible and held for 1 hr (a) at 1400° C and cooled at 500° C/min or (b) at 1350° C and cooled at 7° C/min, 4° C/min and 0.8° C/min to below its solidus temperature. Both thin sections and polished specimens were made of each sample formed under each of the heat-treatment conditions and were evaluated via reflected and transmitted light microscopy, scanning microscopy and electron beam microprobe.

2.3.2 Results

The phases observed by optical and scanning electron microscopy in specimens evaluted in this study are listed in Table 2-8. Also presented are the heat-treatment conditions and the phase identification made from X-ray diffraction analyses (Ref. 2-1).

2.3.2.1 Mock Basalt. The samples of basalt were found to be contaminated with molybdenum. The molybdenum was present in the solidified structure along with iron and titanium. Figure 2-10 shows both the blocky and dendritic habit of the Fe-Ti-Mo phase in mock lunar basalt cooled at 0.8° C/min. The basalt/crucible interface is at the left of the figure. The white areas are the Fe-Ti-Mo phase, the light gray is Ca-rich pyroxene, while the dark gray is feldspathic material. The dendritic crystals of Fe-Ti-Mo appear to be associated with the feldspathic material and in most cases to have precipitated from solution at the pyroxene/feldspar interface.

The mock lunar basalt cooled at 7° C/min had Fe-Ti-Mo phase present as both blocky crystals and dendrites in a feldspar matrix. The blocky crystals form at the basalt/crucible interface while the dendritic crystals grow in the feldspar matrix as shown in Figure 2-11a. Figure 2-11b shows in more detail the pockets or veins of a very fine-grained material which pervade the entire specimen and which appear to consist of at least two components. The first is feldspathic in composition and is probably a glass, while the other component is a ferromagnesium mineral, possibly a Ca-pyroxene. These materials probably represent the crystallization products of a residual liquid.

These results indicate that reaction with the Mo crucible has occurred before solidification of the sample. We have tentatively identified two coexisting pyroxenes (low Ca and high Ca) which form a compact, coarse-grained mass. All ferromagnesium minerals appear to have a high Fe/Mg ratio and some silicates contain a substantial amount of Mo.

Figure 2-10. Mock Lunar Basalt Cooled at 0.8° C/min (Reflected Light)

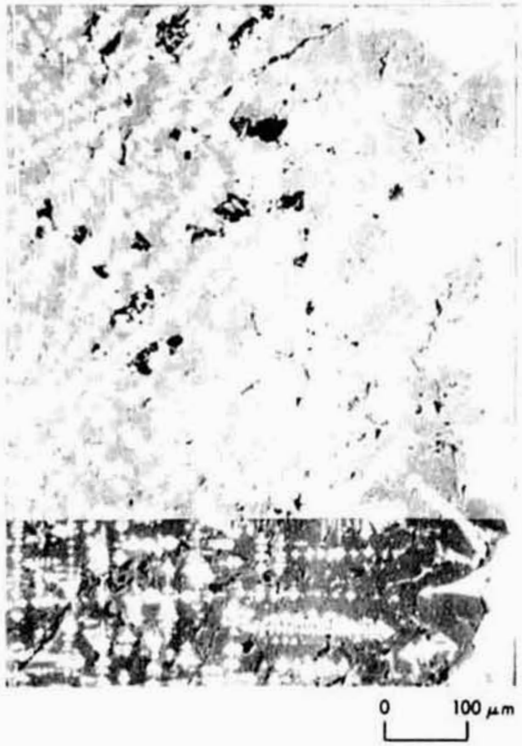

Figure 2-11a. Mock Lunar Basalt Cooled at 7° C/min (SEM)

Figure 2-11b. Detail of Microstructure in Figure 2-11a Showing Fe-Ti-Mo (White), Feldspathic Material (Dark Grey), and Ca-Rich Pyroxene (Light Grey) (SEM)

Table 2-8. Crystalline Phase Content of Specimens Evaluated by Optical and Scanning Electron Microscopy

Material	Heat Treatment	Phase Identification	
		X-ray Diffraction	Microscopy
Basalt	held 1350° C, 1 hr; cooled 0.8° C/min	augite, unknowns	Fe-Ti-Mo dendrites, feldspathic material, Ca-pyroxene
	held 1350° C, 1 hr; cooled 7° C/min	augite, unknowns	Fe-Ti-Mo, glass, Ca-pyroxene
	held 1400° C, 1 hr; cooled 500° C/min	augite, unknowns	Pyroxene, feldspathic glass, Fe-Ti-Mo
Anorthosite	held 1350° C, 1 hr; cooled 7° C/min	analbite, unknowns	plagioclase, olivine, pyroxene, feldspathic glass, augite
Chondrite	held 1350° C, 1 hr; cooled 0.8° C/min	olivine, unknowns	olivine, pyroxene, feldspathic glass, Ni-sulfides
	held 1350° C, 1 hr; cooled 7° C/min	olivine, trace bronzite, unknowns	olivine, Ca-poor pyroxene, feldspathic glass, Ni-sulfide

The mock lunar basalt cooled at 500° C/min had the Fe-Ti-Mo phase distributed throughout the sample. Blocky crystals of this phase were not present at the basalt/crucible interface as was observed in the basalt specimens cooled at the lower rates. However, small dendritic crystals formed in a random arrangement within the matrix as shown in Figure 2-12. The rapid cooling rate of this sample has produced a matrix with a fine-grained, sugary texture consisting of small pyroxene crystals (medium gray), an interstitial glass of predominantly feldspathic composition (dark gray), and the Fe-Ti-Mo phase (white). The two silicate phases shown in Figure 2-12 contain traces of Mo. A substantial amount of Fe, Ti, and Mg and a small amount of Mn is dissolved in the feldspathic glass, indicating that the fast cooling has led to incomplete crystallization of the mafic components from the melt.

2.3.2.2 Mock Ordinary Chondrite. The chondrite specimen cooled at 7° C/min from 1350° C to below the solidus contained large angular or sub-rounded areas of polycrystalline materials surrounded by a ground-mass of feldspathic glass. The polycrystalline material consists of an equigranular mixture of coarse-grained olivine and Ca-poor pyroxene and appears as the medium gray phase in Figure 2-13a. The polycrystalline material is shown in more detail in Figure 2-13b. The grain in the lower left of the figure is olivine and feldspathic glass is located in the center. Note the microcracking of the olivine grain. These microcracks have the potential for degrading the

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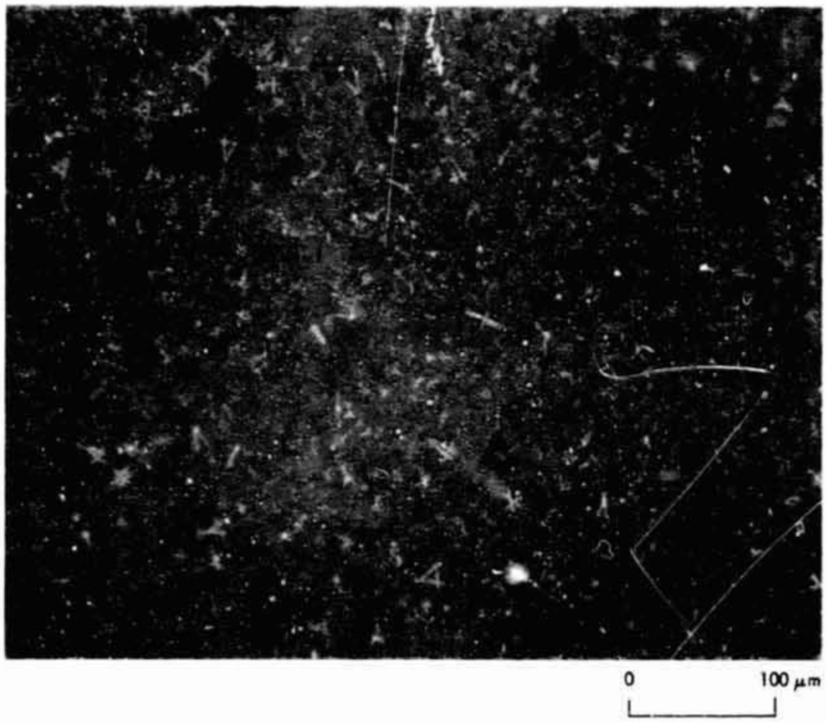

Figure 2-12. Mock Lunar Basalt Cooled at 500° C/min
(Reflected Light)

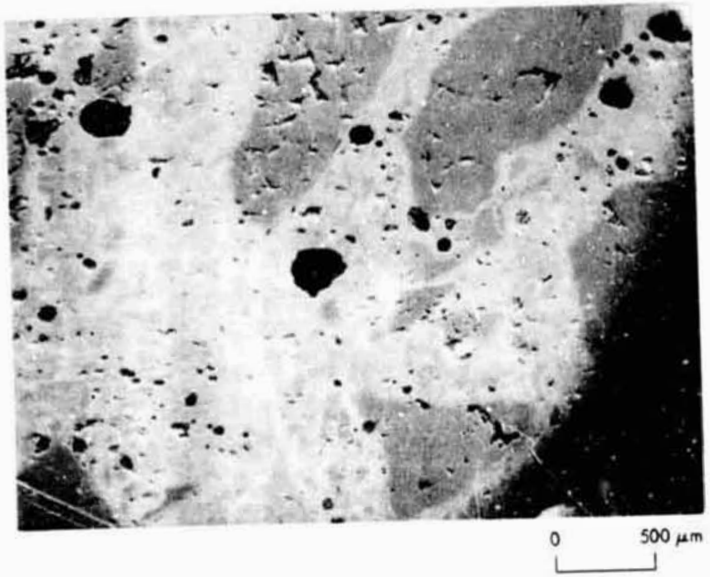

Figure 2-13a. Mock Ordinary Chondrite Cooled at 7° C/min (SEM)

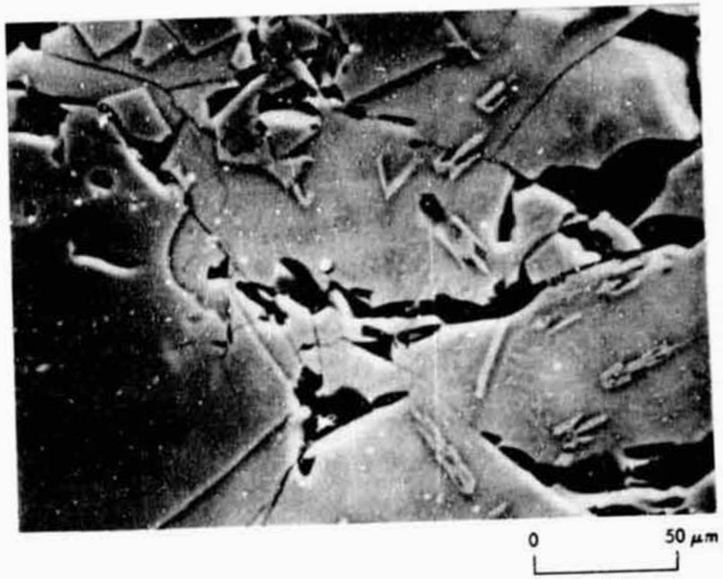

Figure 2-13b. Detail of Microstructure in Figure 2-13a Showing Olivine and Feldspathic Glass Phases (SEM)

strength of the material. The feldspathic glass matrix (light gray in Figure 2-13a) also contains a metallic phase shown as the white region in Figure 2-14a. This metallic component consists of an Fe-Ni phase which occurs in association with a Ni-bearing Fe sulfide displaying an eutectic texture as shown in Figure 2-14b. The regions of high reflectivity are the Fe-Ni phase while the light gray regions are FeS. The Fe-Ni phase also occurs as very small droplets scattered throughout the ferromagnesium silicates as shown in Figure 2-15.

The chondrite specimen cooled at 0.8° C/min appeared similar in texture to that specimen cooled at 7° C/min. As shown in Figure 2-16a the specimen cooled at the slowest rate consists of angular fragments (medium gray) of coarse-grained olivine and pyroxene immersed in a ground-mass of feldspathic glass and crystallites of ferro-magnesium minerals (light gray). The Fe-Ni metal present in the mock chondrite appears as spherules (bright areas in Figure 2-16a) within the feldspathic glass matrix. Sulfide inclusions occur both within the Fe-Ni metal spheres and around their perimeters as shown in Figure 2-16b. Energy dispersive X-ray analyses indicate that the sulfides are nickel-bearing.

2.3.2.3 Mock Anorthosite. This specimen consists of areas containing coarse-grained plagioclase with minor olivine and pyroxene as shown in Figure 2-17. Within and among these areas are pockets or veins of feldspathic material (probably glass) which show enrichments in Na and K and include acicular crystals of augite.

2.3.3 Conclusions

There is substantial admixture of Mo at a level of 5-15% in the Fe-Ti phase found in the mock lunar basalt samples. This was a result of reaction of the basalt with the molybdenum crucible itself. All three mock lunar basalt samples exhibited this feature in the form of dendritic Fe-Ti-Mo phases which appear metallic. However, the phases studied in the mock lunar anorthosite and mock meteoritic ordinary chondrite do not show evidence of Mo.

There is strong evidence from literature studies that Mo exhibits strong chalcophile and siderophile as well as minor lithophile behavior. The exact partitioning between the different behaviors of Mo mentioned above depends on the partial pressures of the various gas phases (O_2, CO_2, H_2O, SO_2) which are in equilibrium with the sample. The introduction of the Mo from the crucible into the melting system will strongly affect the bulk composition of the mineral phases crystallizing from the melt. Consequently, the partition of other elements among coexisting mineral phases will also be severely affected. The unexpected difference between lunar basalt and lunar anorthosite is puzzling and suggests that the physico-chemical conditions during the melting process were substantially different. Another possibility is the addition of substantial amounts of ilmenite in the preparation of mock lunar basalt samples, leading to large amounts of Ti in the melt, which may have played a major role. Further studies are clearly needed to delineate between these possibilities.

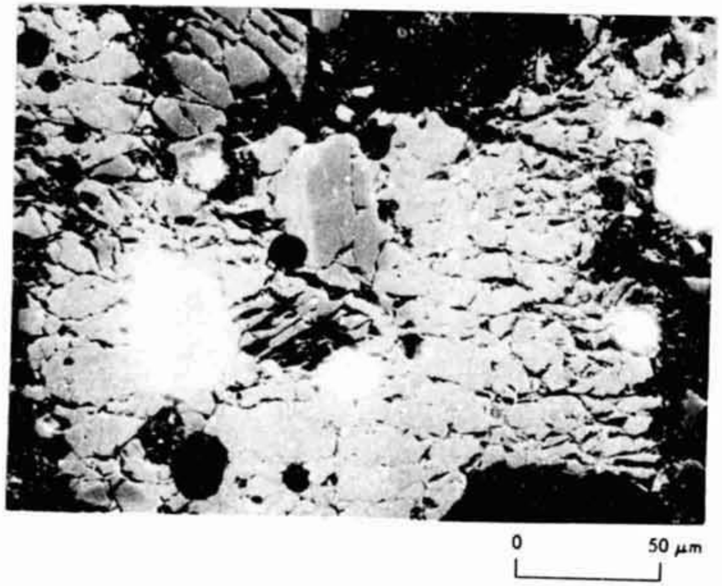

Figure 2-14a. Mock Ordinary Chondrite Cooled at 7° C/min Showing Metallic Phase as White Regions (SEM)

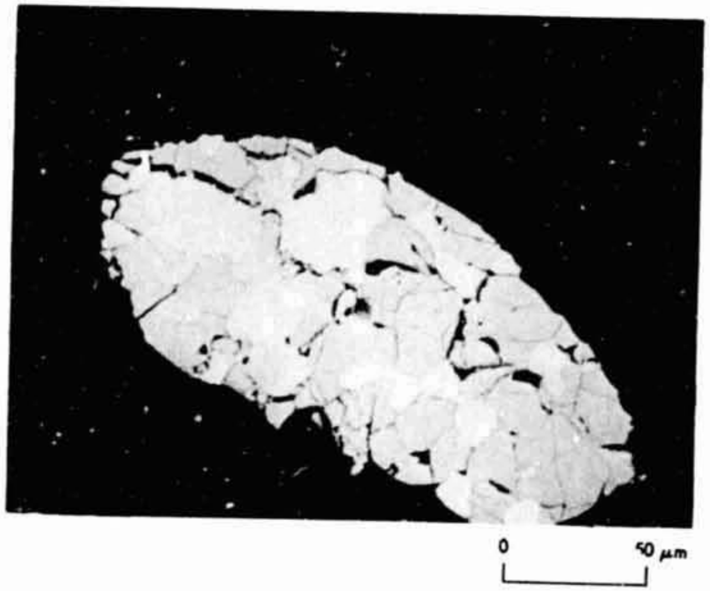

Figure 2-14b. Detail of Metallic Phase in Figure 2-14a Showing Eutectic Structure of Fe-Ni and FeS (SEM)

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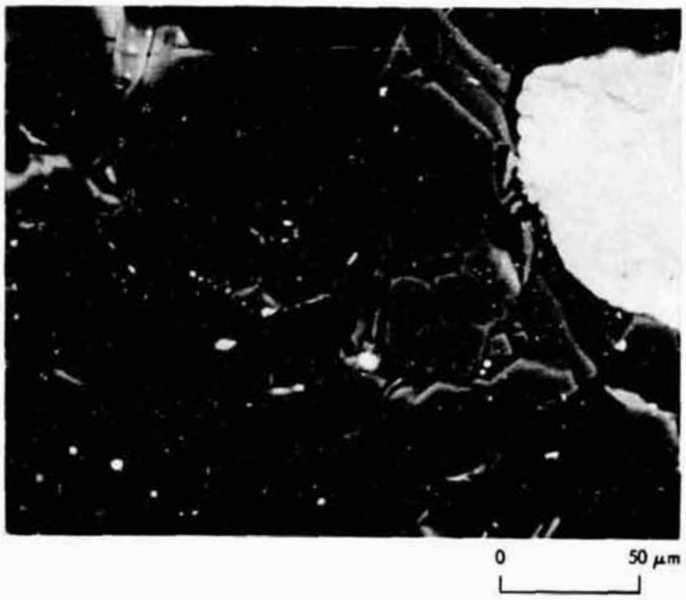

Figure 2-15. Mock Ordinary Chondrite Cooled at 7° C/min Showing Fe-Ni Phase as Small Droplets Dispersed in the Ferromagnesium Silicate (SEM)

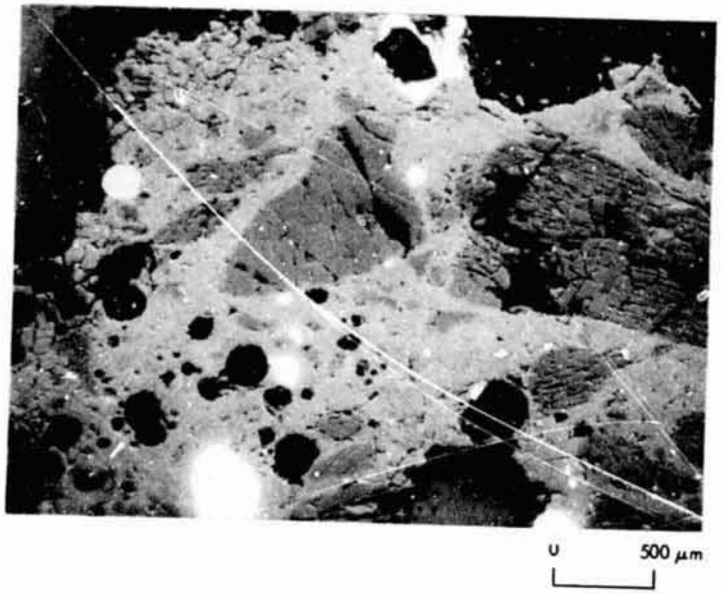

Figure 2-16a. Mock Ordinary Chondrite Cooled at 0.8° C/min Showing Angular Fragments of Coarse-Grained Olivine and Pyroxene (Medium Grey) in a Ground-Mass of Feldspathic Glass and Ferromagnesium Silicates (Light Grey) (SEM)

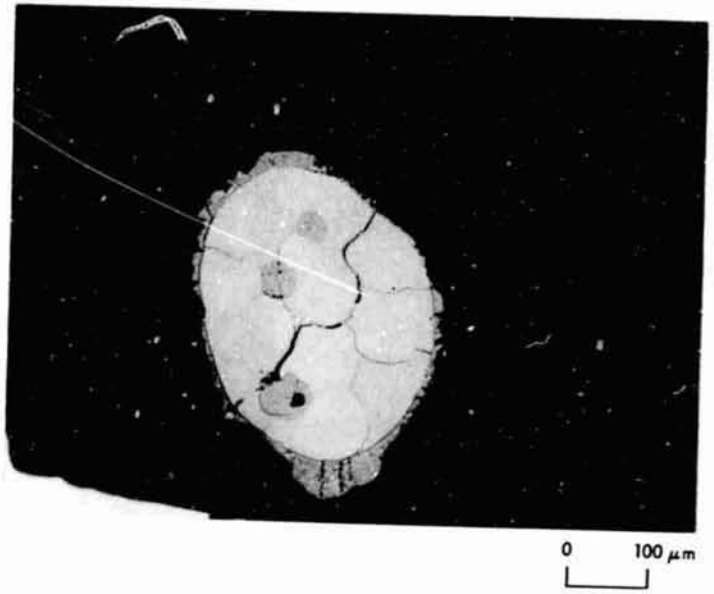

Figure 2-16b. Detail of Fe-Ni Metal "Sphere" in Lower-Left-Hand Corner of Figure 2-16a. FeS Inclusions Are Darker Grey (SEM)

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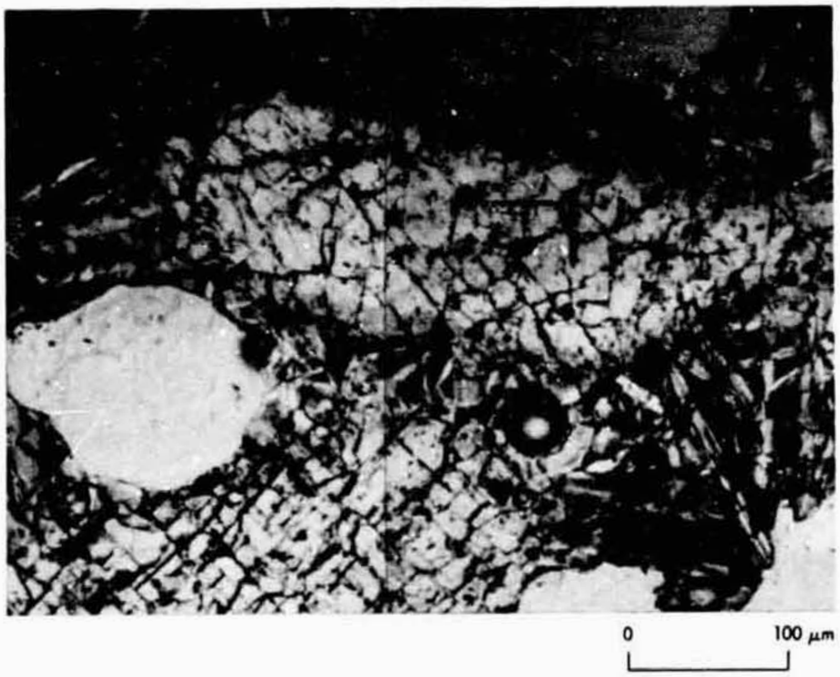

Figure 2-17. Mock Anorthosite Cooled at 7° C/min
(Transmitted Light)

In summary, the chemical composition of the various resulting phases in the basalt specimens may have been altered and the results compromised, due to the introduction of substantial amounts of Mo from the crucible. The textural studies are probably more valid with respect to the silicate materials, which have crystallized from the melts. In future studies in this area, we would propose the use of crucibles made up of less reactive materials such as, perhaps, aluminum oxide or zirconium oxide.

2.4 COOLING RATE CALCULATIONS

The rate of cooling of a liquid silicate through its solidus temperature will determine the texture and phase composition of the solidified structure. Since the thermal gradient is greatest at the free surface the heat transfer rate is greatest there. The cooling rate of a cast silicate is thus a function of position within the silicate body.

The change in temperature with time at 2 centimeter intervals within the baseline basalt is illustrated in Figure 2-18 over the first 6.5 hr of cooling. The line marked "core" shows the change in temperature of the central liquid. The free surface (0 cm) and the interface between casting and mold (50 cm) deviate immediately from the core temperature. There is an "initiation" period at locations away from the surfaces which takes place prior to the onset of rapid cooling. The length of this "initiation" period is dependent on location. After this period is complete the rate of cooling at 2-cm intervals between 0 and 12 cm ranges between 140° C/min at the free surface to 0.9° C/min at 12 cm below the free surface as listed in Figure 2-18.

The most rapid rate of cooling to below the solidus (1100° C) is 140° C/min which is approximately 2/7 the maximum rate in the experimental melt studies (500° C/min). There is good correspondence between the lowest rate of cooling in the experimental melt studies (0.8° C/min) and that predicted to occur at 12 cm (0.9° C/min) below the free surface. This means that we can predict with some degree of accuracy the texture and phase content which will occur in the surface layer of the casting based on the experimental studies.

The rapid rate of cooling of the first 4 cm occurs before the temperature of the central portion of the casting drops more than approximately 10° C. There may be some segregation of the highest melting components which will change the overall chemistry of the remaining liquid. However, it is not expected that this change will be great enough to negate application of the data obtained in the experimental studies in prediction of texture and phase composition of the immediate free surface layers (0-6 cm) in larger castings. At a depth of 12 cm, the temperature drops approximately 25° C below the liquidus over a period of 5.5 hr before rapid cooling begins. Segregation by crystallization of the most refractory minerals may occur during this period of slow cooling (average rate 0.08° C/min). Such crystallization of the most refractory components would cause a change in the overall chemistry of the liquid silicate remaining. Additional experimental melt studies are required to obtain data applicable in such situations.

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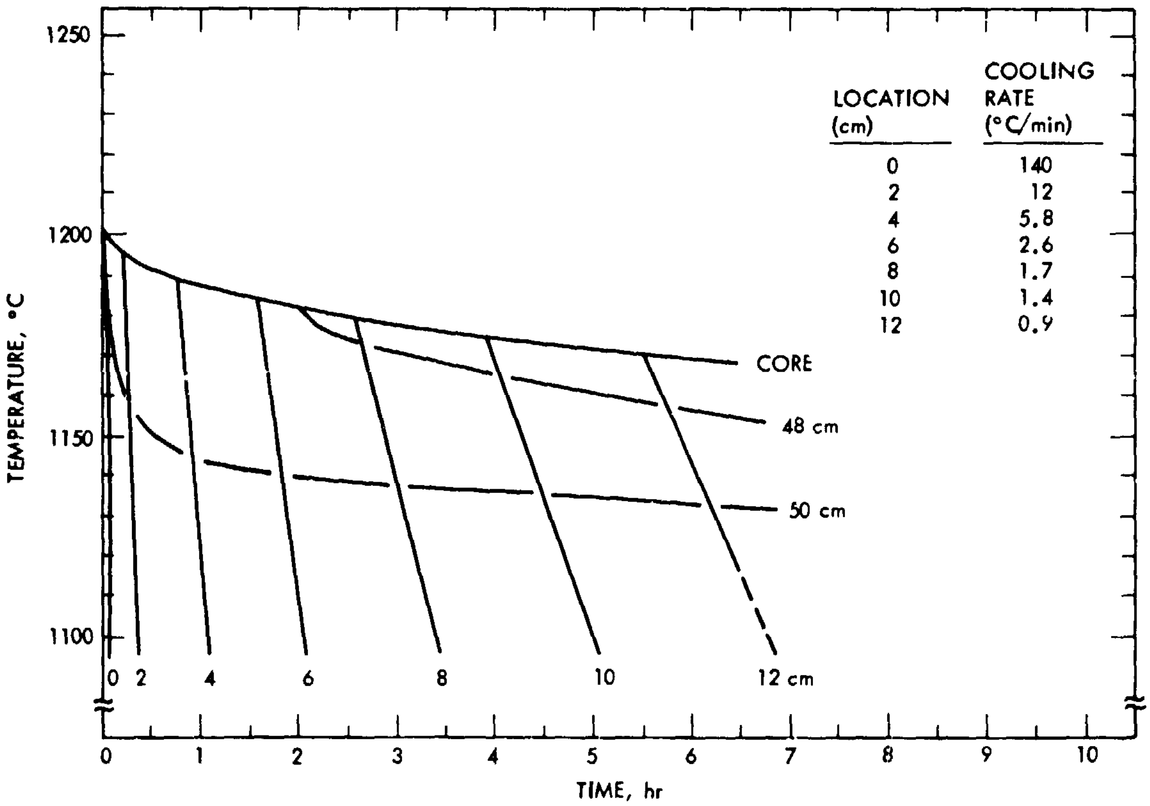

Figure 2-18. Change in Temperature as a Function of Time at Locations Between the Free Surface (0 cm) and the Casting/Mold Interface (50 cm) During Cooling of the Baseline Basalt

Based on the experimental studies we can predict that the free surface layer of a large basalt casting of the composition studied in this program will have a fine-grained, sugary texture. Pyroxene will form in a feldspathic glass matrix and the glass will contain Fe, Ti and Mg in solution.

At locations 2 to 4 cm below the free surface the cooling rate will be between 12 and 6° C/min and the texture and phase composition is expected to be similar to the material cooled at 7° C/min in the experimental studies. It is predicted that at this location there will be a feldspar matrix which will contain pockets or veins of a feldspathic glass and a Ca-pyroxene. The matrix will also contain low Ca and high Ca pyroxenes as compact, coarse-grained masses.

The task which lies before us now is to equate the effect of the changing texture and phase composition with mechanical properties of the structure.

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SECTION 3

MAGMA ELECTROLYSIS

E. du Fresne and J. E. Schroeder

3.1 DEFINITIONS

Magma means simply "molten rock." Usually we understand, in the context of Magma Electrolysis, a melt composition similar to basalt, or to minerals separated from basalt, although there are exceptions.

Magma Electrolysis, though similar in principle, differs considerably in practice from "electrowinning" of metals as carried out commercially on Earth. Even if we compare only processes conducted at high temperature, there are these basic differences:

(1) Magma Electrolysis employs no fluorides or other fluxes to lower viscosity and alter reduction voltages, and

(2) It does not employ a consumable anode (e.g., graphite) to increase the driving force of the reaction or to avoid the expense of a permanent anode capable of withstanding the high temperature of the process.

The reason for both of these is simple: we cannot afford the transport costs.

In the case of flux, we must not only import an initial supply, but must make up process losses as well. In addition, the flux must be recovered, insofar as it can be, and this calls for an additional plant, which must also be imported.

Consumable anodes not only must be imported, but they frustrate a main object of Magma Electrolysis, namely to obtain oxygen. Consider a graphite anode: this will be attacked by oxygen to form CO or CO_2 (usually a mixture of both), and this gaseous effluent must be processed further to obtain oxygen and recover carbon.[1]

Direct oxygen production is important for reasons that go far beyond the cost of importing consumable anodes. W. Carroll has shown that bulk oxygen for propulsion is likely to be the most important immediately useful product to be expected from our efforts (Ref. 3-1). The logic is impressive: beside the importance of propulsive oxygen, even the oxygen dedicated to life-support dwindles to insignificance, being less important by several orders of magnitude. This being taken as given, it is natural to concentrate efforts on the manufacture of oxygen for propulsion, and of propellants in general. This must appear disconcerting, a decided shift in

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[1] While a solid-state electrolysis has been proposed for CO_2 recovery, it has not yet been demonstrated. Routes to carbon and oxygen from CO or CO_2 are likely to be circuitous and costly.

emphasis from terrestrial processes. Oxygen is virtually "free" on Earth, so there is no anxiety about producing it from an electrolysis scheme: the more usual goal is a structural metal. Hence conventional electrowinning processes ignore oxygen production and employ consumable anodes.

3.2 CONTEXT: WORST-CASE ANALYSIS

The context in which Magma Electrolysis is being developed is a worst-case analysis, in which Nature is everywhere unkind. In this scenario, every object that comes close to the Earth, that is, the Moon, the Apollo asteroids, and the (as yet to be discovered) Earth-Trojan asteroids, are mere masses of rock, totally devoid of water ice, other ices, or forms of carbon. In fairness, we should discuss what could be done if this worst-case analysis is wrong.

3.2.1 The Alternative Context: Water is Available

Of the infinity of possible cases, three seem more probable than others:

(1) Water-ice persists at sheltered locations, such as the poles of the Moon (Refs. 3-2 to 3-4), or under the silicate surfaces of asteroids (Ref. 3-5).

(2) Gas hydrates and water-ice persist under the dusty crusts of burnt-out comets which masquerade as Apollo asteroids.

(3) Some of the Apollo asteroids (or Earth-Trojans, if they exist) are similar to carbonaceous chondrites in composition.

These three cases can be resolved, from the standpoint of processing propellants, into one major opportunity, the availability of water, with several minor beneficial side issues, such as the recoverability of ammonia or carbonaceous compounds.

Water can be electrolyzed directly, at room temperature, into hydrogen and oxygen in stoichiometric proportions. There is no need for a solid-state electrolysis process, as recommended by some enthusiasts. The liquid process is well-established on the commercial scale, as well as being a time-honored demonstration in high-school chemistry. Hence a virtually perfect fuel-oxidant combination is available at little cost, provided only that water is to be found.

3.2.2 Gas Hydrates and Frozen Free Radicals

Gas hydrates add little to this: they may contain CO_2, ammonia, argon, etc., but their main product of interest is <u>water</u>. The rest contribute little or nothing to propulsion.

It has been argued that comet ices also contain frozen free radicals, which would be a boon to propulsion. On examination, this hypothesis, based on the irregular flare-ups of comet Schwassmann-Wachmann II, turns out to be an argument against the existence of ices within such burnt-out comets as may circulate close to the Earth. For if Schwassmann-Wachmann II can occasionally show a coma of gas, and blow off its dust covering, at a distance of 5 A.U. or more from the Sun, then surely a similar object at 1 A.U., where the light is at least 25 times more intense -- and the heating effect likewise -- will be even more prone to flare up, display a coma, and blow off its dust cover! Which is to say that a comet nucleus cannot become a dust-covered object, mistakable for an asteroid, at a distance of 1 A.U. from the Sun, if its composition resembles that of Schwassmann-Wachmann II.

One can eliminate gas hydrates for much the same reasons. These clathrate compounds can be formed in the region of 0-20°C in the presence of gas pressures of some hundreds of atmospheres. To preserve them at 1 atmosphere or less, it is necessary to drop the temperature considerably: CO_2 hydrate can be kept for weeks to months at -40°C (Ref. 3-6). If one allows them to warm gradually to 0°C, they decompose, often by explosive decrepitation. Again, it is difficult to see how a comet nucleus can acquire a dusty mantle at 1 A.U. from the Sun, where the black-body equilibrium temperature is close to 0°C, if it contains any substantial amount of gas hydrates.

The chain of logic forces us to the conclusion that, if any burnt-out comet nuclei are masquerading as Apollo asteroids, they either consist of residual dust alone, or of such dust plus water ice, but without any substantial content of frozen free radicals or of gas hydrates.

3.2.3 Hydrocarbons

The case of carbonaceous chondrite-like asteroids is more interesting, though perhaps worse understood. Among the paradoxes presented by these objects is that they contain no chondrules, though described as chondrites. Some of them, such as Orgueil, appear to have been extensively reprocessed, while others, superficially similar, have been classed as primitive solar nebula material (e.g. Renazzo).

In the threefold classification enunciated by Wiik (Ref. 3-7), we find a gradation in carbon content from I-III, and Group III, the extreme end of the range, is so poor in resources (other than those found in silicates generally) as to be scarcely worth special attention. These carbon-poor meteorites are hardly distinguishable from the ordinary type known colloquially as "stones." But, progressing toward the other end of the gamut, Group I, we find increasing amounts of water and carbon.

The water content has often been challenged, in spite of the definitive work of Giovanni Boato (Ref. 3-8). It has been argued that the carbonaceous chondrites have absorbed water from Earth's atmosphere in museum storage. This is true, to a degree -- but Boato's isotopic studies showed that some, at least, of the hydrogen and oxygen of the exchangeable water was of extraterrestrial origin. Moreover, du Fresne and Anders (Ref. 3-5) were able to prove that much of the primordial water was held in layer-lattice silicates, as "water of constitution," that is, as fixed hydroxide ions, capable of being liberated (or exchanged) only at elevated temperatures, approaching 600C. In the Orgueil-type meteorites, even more of such primitive water was held in vacuum-stable compounds such as the glassy form of magnesium sulfate monohydrate. (This last does indeed rehydrate under some storage conditions to Epsom salts -- the heptahydrate -- causing poorly kept specimens of Orgueil and other Class I carbonaceous chondrites to fall apart, while becoming contaminated with water of terrestrial isotopic composition.)

The question of carbon content is even more interesting. The carbon is not in the form of graphite, or of metallic compounds, such as the cohenite (Fe_3C) of the great iron meteorites. Rather, it is distributed as a thin, polymerized hydrocarbon film, called "retigen," over the surfaces of the mineral grains that have been compacted to form the bodies of these meteorites. Solvent extraction has proved difficult (not that it would be practical to import solvent). Recovery of carbon (or better yet, of hydrocarbon) values thus resembles the oil-shale problem. Essentially, it is a matter of retorting (pyrolysis) to obtain tarry matter and combustible gases. Since the same process of destructive distillation would be used to recover water from such material, it is worthwhile to consider what might happen in a joint operation.

(1) Low temperature (up to 200°C): water released from hydrated salts; retigen essentially unaltered. The process can stop here, at discretion, with considerable savings in energy, if only the water is desired.

(2) Intermediate (200-500°C): pyrolysis of retigen, releasing tars and hydrocarbon gases, and leaving a carbon-rich residue. Again, the process can be interrupted at this point.

(3) Hot (500-700°C): water is released from phyllosilicates, etc., in which it is stored as OH of constitution. Some of this may react with carbon left over from retigen pyrolysis to produce "town gas," that is, H_2 + CO. This reaction is relatively unimportant. Again, the process may be concluded here.

(4) Ultra-Hot (700°C to melting, say, 1200°C): the carbon will reduce iron in the silicates, liberating CO.

This "oil-shale" model has the advantage that it is immediately testable with samples of known carbonaceous chondrites. Naturally, there will be variations between one specimen and another. But enough work has been done already (some of it going back to the 1860's, in the case of Orgueil) to indicate that the "oil-shale" model is reasonably good. This has certain consequences:

(1) No special solid-state electrolysis need be provided for CO_2.[2] There won't be enough CO_2 to worry about.

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[2] Some carbonate exists in Orgueil, but it is too rare to be worth extracting. (Refs. 3-9 and 3-10.)

(2) A series of options exists, depending on how much energy one is willing to expend to obtain a given product. Hydrogen and oxygen from water are at once the most valuable products (from the point of view of propulsion) and the most easily extracted. Then come the hydrocarbons. To obtain more than the second installment of water ("Hot" option), or phyllosilicate water, is difficult using solar energy, through dish collectors alone, at 1 A.U. from the Sun, as 700°C is the practical limit attainable by dishes at that distance; and rather pointless, since the product of the "Ultra-hot" option is CO, which has a very low specific impulse. With hydrogen and hydrocarbons available, who would want to go to great expense to make CO?

To sum up the argument thus far:

The best we can hope for is a source of abundant water-ice or of water-rich salts, supplemented by pyrolyzable hydrocarbons. Otherwise, we must make do with rock.

The possibility of using electrowon aluminum as a fuel in place of hydrogen has been discussed by R. Frisbee et al. (Ref. 3-11) Here, we confine ourselves to the technical problems of getting Magma Electrolysis to work reliably.

3.3 PRACTICAL CONSTRAINTS

3.3.1 Heat Balance, Temperatures, and Containment Materials

For Magma Electrolysis to work at all, one must first melt the rock, using heat alone (no flux). This puts the process in the range 1200-1700°C, or more realistically, 1500-1700°C. The strain this puts on materials of construction is extreme. (Fortunately, much of this strain can be relieved, on the works scale, by properly designing the thermal gradients within the cell -- it is on the laboratory scale that the problem is most severe.)

Where is the necessary heat to come from?

Earlier attempts at the problem suggested a need for continuous solar heating of the cell, as through a transparent window. More rigorous treatment by J. Schroeder, given elsewhere in this report (Ref. 3-12), shows that the ordinary resistance losses involved in electrolysis are sufficient to keep a works-scale cell at the proper temperature. Hence there is no need for a window, which would prove an embarrassment in the practical case, as there is no easy way to keep it clean and transparent.

In effect, once a molten pool has been established, it is possible to insert temporary electrodes and to heat it by purely electrical means until it has grown (by melting its surroundings) to a size practical for full-scale electrolysis. At this point, the permanent electrodes can be activated, oxygen-collecting apparatus installed, etc., and a serious campaign begun.

The process of establishing a molten pool, through such methods as striking an arc, etc., is widely known in terrestrial industrial practice. So is the process of enlarging the pool through resistance losses -- this is the standard start-up procedure for an electric furnace. ("Skull" melting is a name often given to the procedure.) Again, the procedure of enclosing a molten pool with solids that themselves ultimately dissolve in the pool has been standard practice in the glass industry for at least 2000 years.

3.3.2 Process Chemistry and Semiconduction

Questions left unanswered by previous analyses, and which remain to be settled by experiment, are:

(1) Are there kinetic blocks to the reduction of particular metals?

(2) What role does electronic conduction (n-type semiconduction) play in the dissipation of power by the melt?

On question (1), it has been rather glibly stated that silicon and aluminum cannot be reduced directly from magma. Actually, there is no direct evidence for this. The reasoning follows an analogy with aqueous solutions of silicates and aluminates, in which these anions are repelled from the cathode. In a magma rich in silica or alumina, however, such repulsion as may occur cannot entirely deplete the layer next to the cathode of these elements, so that their reduction remains possible, even likely. The subject is thus more complex than has been stated previously.[3] Such matters can only be settled by experiment, and not by a debate in the style of the ancient Greeks, even though the debaters might be Socrates, Plato, and Aristotle reincarnated. Hence experimental tests of the reduction of these elements, silicon, aluminum, and several others, are early on our lists of the cathode efficiency studies that must be performed.

Question (2) is more easily answered on theoretical grounds. Semiconduction in oxides has been well-studied in the past few decades. Consider mill-scale, Fe_3O_4. This is a remarkably good conductor, and, being cathodic to iron, is thus a pronounced promoter of corrosion in the basis metal. Semiconduction in Fe_3O_4 is therefore not a matter so arcane as to be considered an academic inanity, but has profound economic consequences. We need no further apology for using magnetite/mill-scale as a model of oxide semiconduction.

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[3] We may remind those that hold to the aqueous solution analogy that chromium is plated from solutions of chromic acid, i.e. those rich in dichromate anions, which are repelled from the cathode, while chromium cations, e.g. the Cr (III) aquo complex, are not usually reduced to metal at the cathode, owing to kinetic (stereochemical) factors!

3.3.2.1 A Model for Semiconduction. The framework of the Fe_3O_4 crystal is a close-packed array of oxide ions (cubic close-packing). In making a model of this crystal, one might represent the oxide ions by tennis balls -- on this scale, the iron atoms would be about the size of marbles or of large bearing balls, which can fit into the "holes" in the array of tennis balls. (There are two types and sizes of holes: "octahedral" and "tetrahedral," but this distinction need not concern us here.) On this model, two "marbles" can never touch each other: there is always "tennis ball stuff" in between them.

Imagine now two marbles, one colored blue, the other red, that are as close together as permitted by the intervening tennis balls. Let the blue marble be a ferrous ion, the red one, ferric. The elementary act of electronic conduction is the switch of the colors of these two marbles, by the passage of one electron from the ferrous to the ferric ion, converting the ferric to ferrous, and vice versa. This elementary act would be absolutely trivial in a metal, where all valence electrons are held in common as a "conduction band," since all the metal ions are cheek-by-jowl. But in magnetite, this is a non-trivial process. The electron must "tunnel" through a negative, repelling potential (the negative charge on the oxide ion, represented by the tennis ball) to get to its destination. The process does take place, but it is of comparatively low probability, as reflected by the fact that the conductivity of magnetite is of several orders of magnitude less than that of a metal, when these are compared in the solid state, as at room temperature.

Let's play with this magnetite structure. We can replace some of the red marbles, or ferric ions, with aluminum ions, which can't be reduced to a lower valence state without a large expenditure of energy. This decreases the chances of electron-transfer in proportion to the decrease in concentration of ferric ions.

Alternatively, we can replace some of the blue marbles, ferrous ions, with magnesium ions, which are hard to oxidize. Again the average distance from red to blue is increased, and conductivity goes down.

On this reasoning, we might expect that decreasing both the ferrous and ferric concentrations simultaneously and in equal proportions would result in some such double effect as the electronic conduction falling off as the inverse square of concentration (i.e., being proportional to the product of the individual concentrations). But this is not so. Referring again to our tennis ball and marble analogy, we can see that reducing both concentrations increases the average distance between marbles of different colors. Now, the probability of electron tunnelling through a barrier falls off exponentially with the distance through the barrier, everything else being equal), and hence electronic conduction falls off far more rapidly than as the inverse square.

This argument has laid no stress on the actual crystal structure, and is in fact largely independent of any structure other than the alternation of "tennis balls" with "marbles". Such an alternation would be maintained even in an amorphous substance, by the electrostatic attraction between anions and cations. Hence we can melt the crystal without substantially changing the argument. The only real change is that the volume increases by roughly one-seventh, and hence the average ferrous-ferric distance by roughly 5%.

This is enough to drop the electronic conductivity by a substantial factor, but not so much as to change the qualitative picture: the change in semiconduction upon melting will be less than an order of magnitude.

If we then play the dilution game in the liquid as we did in the solid, we should obtain the same qualitative results, most notably a drastic decrease in electronic conduction if both valence states are cut back simultaneously.

Note that ionic conduction remains essentially unaffected throughout all these exercises. Well below the melting-point (m.p.), within the rigid crystal, it is essentially zero. As the m.p. is approached, the rate of solid-state diffusion increases, and ionic conduction often (but not always) becomes detectable and measurable. With melting, the situation changes drastically, with the ionic component of conduction jumping by three to ten orders of magnitude. Finally, ionic conduction will increase in the liquid with increasing temperature, following an Arrhenius Law (log X vs. 1/T makes a straight line) corresponding to the decrease in viscosity with increasing temperature. (In effect, ionic conduction is proportional to the reciprocal of viscosity, so both, when taken on a logarithmic basis, are linear with 1/T). In contrast, electronic conduction remains steady, or actually decreases with the dilation of the melt.

We may draw several practical conclusions from this model:

First, there is no possibility of semiconduction in the melt unless a metal ion is present which can exist in more than one valence state.

Second, the potentiality of existing in several valence states, though necessary, is not sufficient -- there must be more than one valence state actually present. (If we eliminate ferrous ion by substituting magnesium, or ferric with aluminum, the electronic conductivity collapses, as noted.)

Third, practically any metal with more than one valence state available under the conditions prevailing in the magma will foster semiconduction. Since the net effect of the elementary act of conduction is merely to "swap" two ions, without a net reduction or oxidation, there is no net energy change to be taken into account.

Apart from the two states of iron, we may note the many valence states available to titanium (Fig. 3-1). These two elements are quite lunar-abundant, and one might expect them to be especially troublesome.

Other elements which have been used in experimental work, or proposed for use, are likely to foster semiconductivity in magmas. Among these are lead (two states) and molybdenum (several valence states), as will be discussed later.

A jump of several orders of magnitude in conductivity upon melting thus signalizes the ionic process, while electronic conduction will break downwards as abruptly. Of course both processes can coexist, but, in case of doubt, the dilution test can be applied: ionic conduction should be unaffected, while electronic conduction will drop off exponentially.

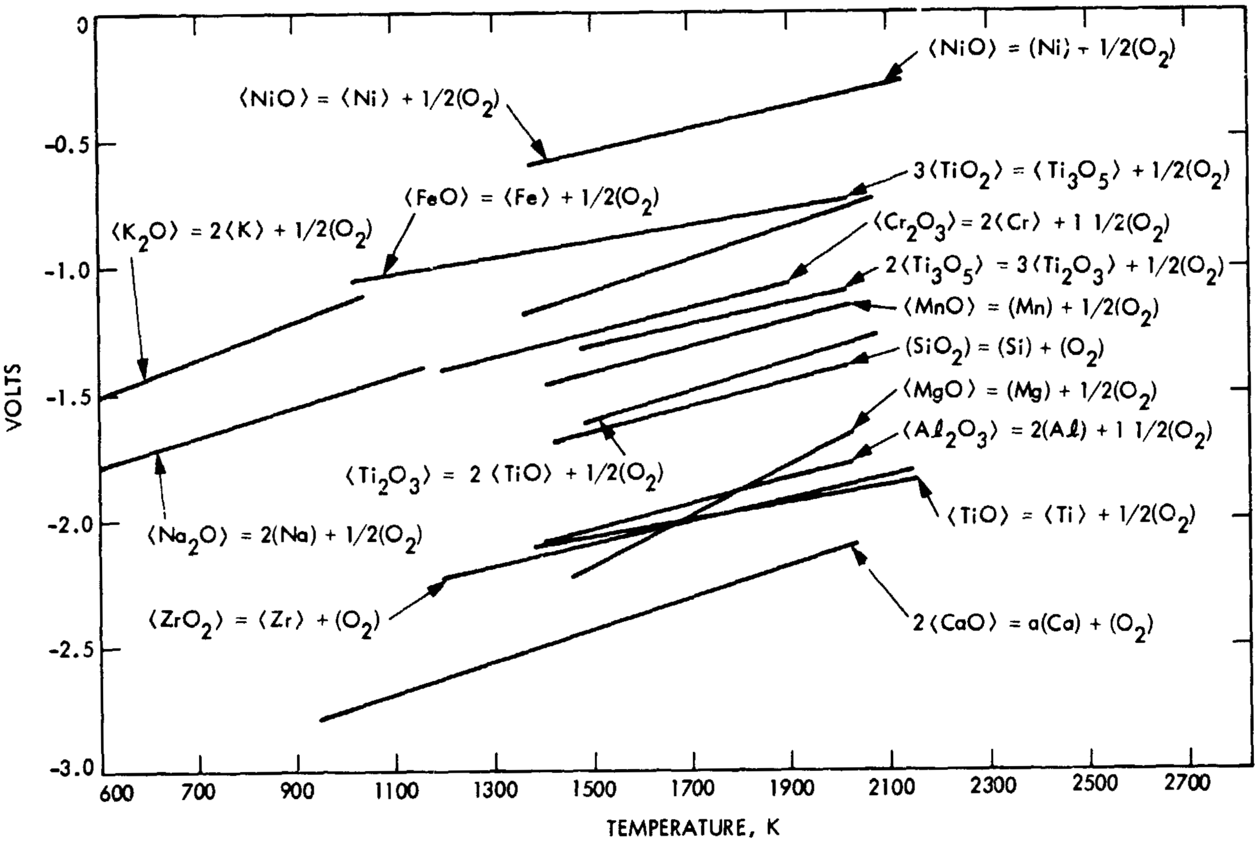

Figure 3-1. Reduction Voltages -- Lunar-Abundant Metals

Fifth, DC measurement, if prolonged, can falsify the results (independently of electrode corrosion). Consider a melt in which ferrous ion is a majority species, and in which ferric ion is initially absent. Impose a substantial DC voltage. Initially, all conduction (in the absence of other semiconductive systems) will be ionic. But, with the passage of time, some ferrous ions will transfer electrons to the anode. Hence a ferrous-ferric mixture will be established near the anode, and spread outward from it. Within the region of this mixture, and at its boundary, electronic conduction will compete with ionic, to the confusion of the experimenter.

In the context of the lunar-abundant rocks and minerals available for magma formation, these practical conclusions have the following applications:

(1) Semiconduction is not to be expected from feldspar-rich melts (e.g. anorthosite, maskelynite) unless erosion of the cell material (e.g. molybdenum) contributes to it in the course of a protracted experiment. There simply are not enough polyvalent species present in feldspars to support electronic conduction on an appreciable scale (i.e., to reach within two orders of magnitude of ionic conduction in such melts).

(2) Semiconduction is not to be expected in a lunar iron-rich melt under start-up conditions. Initially, virtually all iron will be in the ferrous state (and titanium in the tetravalent state) so that the mixed-valence condition does not exist.

(3) The passage of current over a prolonged period can drastically alter the conduction properties of such an iron-rich melt. Consider a magma containing substantial amounts of ilmenite, or of other minerals that contribute ferrous ion and titanate. Even if ferrous ion is not oxidized to ferric at the anode, the liberation of oxygen there will indirectly produce ferric, by the interaction of the pure oxygen, as it bubbles out, with the adjacent melt. Electronic conduction may thus rise to an appreciable level with the passage of time. Similarly, titanate is unlikely to be reduced directly to the metal at the cathode, even if the voltage is very high. What is more likely is reduction to one or more intermediate valence states. With either element, iron or titanium, the mixed valence state condition is likely to be established, and on a scale sufficient to be compared to ionic conduction.

3.3.2.2 Potential Effect of Semiconduction. It is clear that the subject is worthy of experimental study. This year's work (see summary below) has been carried out in such a way as to exclude semiconduction. The implicit assumption has been that Magma Electrolysis required a separate source of heat to maintain the molten state. (Early versions of an electrolytic cell showed a window for admission of solar energy from a concentrator.) Under this assumption, electrical losses due to semiconduction were economically harmless, as they would be sources of that heat supplement which needed to be superadded in any case. J. Schroeder's more rigorous analysis, (Ref. 3-12) proved that assumption to be incorrect.

In the new picture, semiconduction is shown as an essentially parasitic process, stealing hard-earned watt-hours from the manufacture of useful products: oxygen and metals. It now becomes needful to incorporate this revised thinking into our experimental program, for we must find out whether this parasitic process assumes proportions comparable to that of ionic conduction itself. If so (that is, if it amounts to 10% or more of the direct production process), we must explore methods of circumventing it.

This is, of course, a refinement rather than a fundamental problem. When one contrasts Magma Electrolysis with any other process proposed for the purpose, it emerges as the most energy-efficient of them all, and would still do so even if its efficiency were cut arbitrarily by 50%. That such refinements can now be seriously considered is a measure of the progress made, on both the experimental and theoretical levels, in the past year.

3.4 EXPERIMENTAL STUDIES

3.4.1 First Phase: Conductivity Measurements

Conductivity of magma samples was first determined in a crucible formed of bored-out molybdenum bar stock. The crucible served as one electrode, the other being a molybdenum rod supported in the magma pool, and insulated from the crucible by a boron nitride lid. The crucible also served as a susceptor in an induction furnace. An AC potential (6 volts RMS) was imposed between the electrodes at 60 Hz for a fraction of a second, thus avoiding complications from chemical reaction at the electrodes. Hence the observed conductivity is wholly ionic.

These crucial experiments established the essential fact: molten basalt is conductive enough to support electrolysis, even at temperatures just above the liquidus line (at which all constituents are wholly liquid).

Since these and related studies have shown that the conductivity of magma increases with temperature, the practicability of Magma Electrolysis was thus firmly established.

Representative experimental data are given below. Conductivities are in reciprocal ohms per centimeter (that is, $[\text{ohm-cm}]^{-1}$).

T, °C	Basalt	Anorthosite
1200	0.08	0.013
1250	0.115	0.02
1300	0.15	0.033
1350	0.22	0.045
1400	0.30	0.06
1450	0.43	0.09

These data accord well with the more reliable of the few to be found in the literature. (The data are distributed vertically -- along the conductivity axis -- about the regression line as Student's t with four degrees of freedom, and a standard deviation of very nearly 5% in both cases. The accompanying Arrhenius plot -- Figure 3-2 -- of the log of conductivity vs. $1/T_K$ shows the excellent correlation. In this figure, the error ellipses show approximate "two-sigma" limits, i.e., there is roughly one chance in a hundred that a datum will fall outside such an ellipse.) It is clear that this method can be used in evaluating any simulated sample of lunar rock.

Note that the simulated lunar basalt contains very considerable percentages of iron and titanium, but in single valence states (ferrous ion and titanate ion respectively). Hence semiconduction is neither expected nor observed. Had extra ferric ion been added to the melt (or any other appropriate ion or ions in large amount), to bring semiconductivity up to values comparable with ionic conductivity, no straight line would exist on the Arrhenius plot for basalt: it would be concave downwards.

3.4.2. Second Phase: Direct Current Experiments

Work on actual electrolysis was begun with a highly restricted budget, which permitted the use of few materials beyond those already at hand. Our watchword was that of Saint Bartimaeus: "Patience in Poverty." It was natural, therefore, to re-use the molybdenum apparatus (shown in Figure 3-3) for DC electrolysis.

Using the central rod as anode, and a mock basalt magma at 1550°C, current was passed at a formal density of 1.25 A/cm^2 (on the original anode). This virtually destroyed the anode by oxidation in less than an hour and a half (1.4 hr, estimated). Small deposits of molybdenum trioxide (MoO_3), which is volatile at this temperature, were found outside the crucible, where it had been carried to a cooler area by the argon used as sweep gas.

Had we been sure that the entire anode had been converted to MoO_3, it would have been easy to compute the current efficiency. The greater likelihood was, however, that MoO_2 and lower or intermediate oxides had also been formed, and had dissolved in the magma -- not to mention several minor side-reactions that might have taken place. Thus it is useful to calculate electrolysis efficiency on the basis of the simple conversion of the anode to dioxide, instead of the trioxide, as a first cut.[4] This gives an

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[4] The dioxide and trioxide are the more frequently produced of the oxides of molybdenum. Now, if conditions had been favorable for conversion of most of the anode to the trioxide, the gas exit would have been entirely clogged with it. The other logical choice is dioxide.

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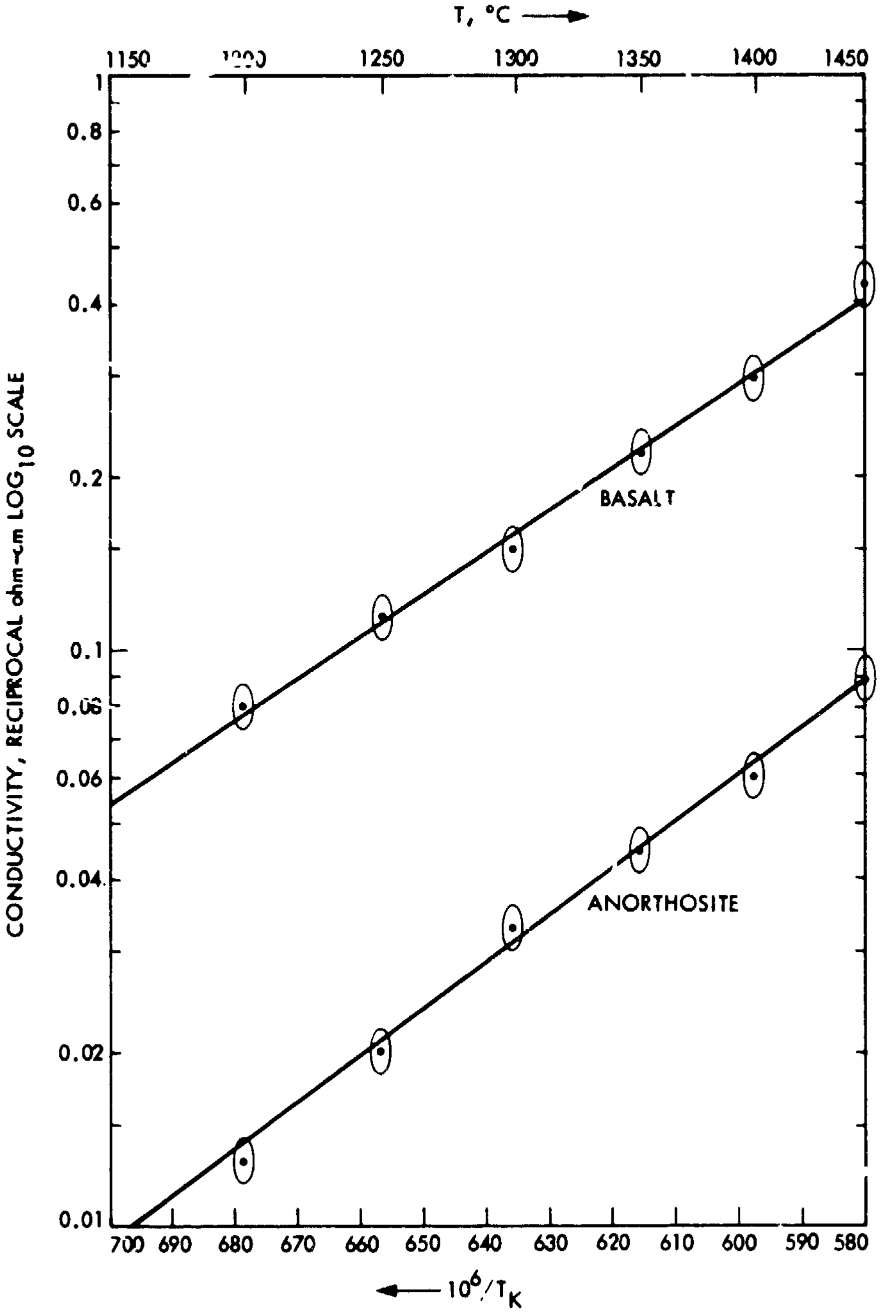

Figure 3-2. Arrhenius Plot of the Log of Conductivity vs $1/T_K$

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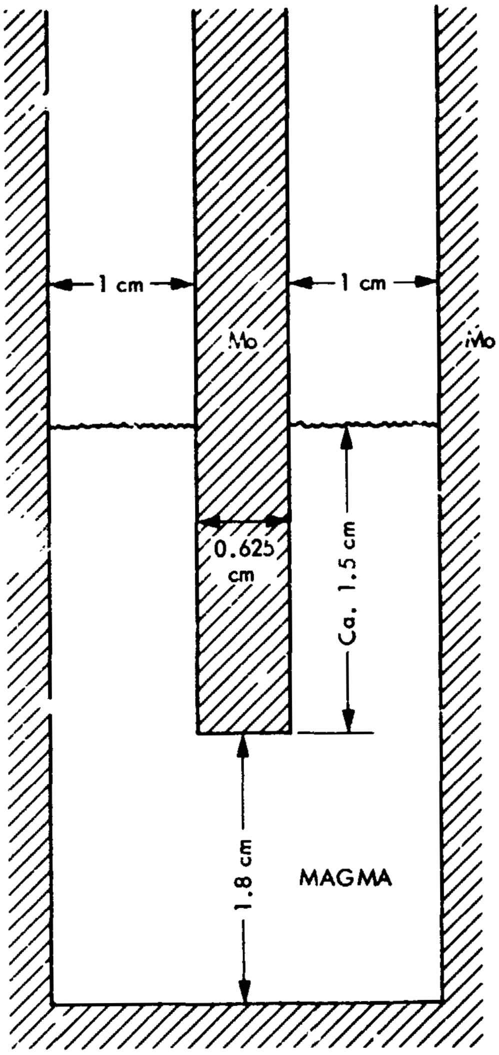

Figure 3-3. Conductivity cell, with scale magnified by 2.54, i.e., 1 inch on the figure = 1 cm on the actual object.

electrolysis efficiency of 101.76%, a remarkable agreement, which must be partly due to happy coincidence, considering the errors involved. (Note, however, that with our cell geometry, as shown in Figure 3-3, the greatest errors, errors due to misestimation of the depth of immersion of the anode, cancel one another, as the mass of anode immersed, and the anode surface exposed to effective electrolytic oxidation, both are incremented in proportion to the depth of immersion.[5] Several other major sources of error cancel almost as neatly.) The observed production of MoO_3, which would raise the putative electrolytic "efficiency" above the calculated value, if included in the calculation, must be compensated for in some way. These are the possibilities: partial formation of a lower oxide of molybdenum (probably Mo_2O_3), and direct chemical attack on the anode by metal ions contained in the magma (for which there are several candidates).

When all of these effects are taken into account, it still seems clear that the actual electrolytic efficiency approaches 100%, and is almost certainly higher than 95%.

But this result, though satisfying as far as it goes, is not enough. More directly relevant DC experiments would quantify:

(1) Oxygen as the element, produced at the anode.
(2) Iron or other metals produced at the cathode, and
(3) The corresponding number of coulombs passed across the cell.

But both anode and cathode conditions are difficult to study on the small scale of laboratory models, and under the artificial conditions thereby imposed.

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[5] Here is the reasoning in greater detail: the resistance of the magma is high, and current is proportional to the reciprocal of resistance. Hence most of the current will follow the shortest route, straight across the annular space between the anode and the wall of the crucible. End effects are thus negligible, a few percent at most. The initial current density at the anode is therefore found by dividing the total current by the product of the circumference of the anode times its depth of immersion. The mass of the anode that will be dissolved is the cross-section of the anode, times the depth, times the density. Increasing the depth therefore increases both the mass and the total charge passed during the experiment in "exact" proportion, "exact" meaning: "with a few percent error."

3.4.3 The Anode Problem

Most of the potential anode materials that survive temperatures of 1550°C (or higher) are subject to oxidation. (The notable exception is platinum, of which more below.) We can rule out all but the rare or precious metals on the grounds of melting-point, susceptibility to oxidation, or both. As for the conductive non-metals, carbon is rapidly attacked by iron-rich magmas, yielding metallic iron without the passage of current. Carbon therefore defeats the purpose (namely, measuring an output of elemental oxygen). It is difficult to see how a carbon anode experiment would improve on the molybdenum anode experiment described above, as it shares all of its errors. Likewise, silicon carbide, which is ordinarily protected by a film of silica (produced by an initial oxidation) will not be thus immunized when bathed in basaltic magma, for this rapidly dissolves the silica layer, exposing the basis to further oxidation. Again the result is likely to be the release of CO or CO_2 in the place of elemental oxygen, and this without the passage of current being necessary for the desired reaction.

Platinum is therefore the only readily available anode material capable of giving a clean result. But the melting point of pure platinum is 1772°C, compared to that of iron, 1535°C. Thus, rapid temperature excursions of the furnace could melt the platinum components of the cell, particularly those exposed more directly to radiation from the heating elements or from low thermal mass parts of the system. This is particularly true of experiments done in resistance-heated furnaces, which are difficult to control if the temperature is rapidly raised. A number of failures have occurred, with the characteristic aspect that all platinum components outside the magma are damaged, while those immersed in (or touching) the magma survive. An alternate hypothesis is that basaltic magma produces a gaseous "emanation" capable of causing damage to the platinum below 1600°C.

Resolution of the alternative hypotheses has not been accomplished because of higher priorities for other experiments. However, some facts are appropriate to this discussion:

(1) While some compounds (notably metal halides and carbonyls) could conceivably fill the role of the "emanation," they are likely to be present in parts-per-billion quantities (or less) in the gas phase, amounts inadequate to produce the results observed, and,

(2) Finally, "emanations" from the magma will have lower fugacities in the gas phase than in the magma itself. Hence, the platinum submerged in the magma should be damaged first, not the platinum in the gas phase -- which is contrary to the observations reported so far.

Unfortunately, as will be seen below, in Section 3.4.5, the best experimental work is to be done in ceramic crucibles, which must be heated in a resistance furnace, while the easiest resolution of the problem would be done in a metal crucible/susceptor, in an induction furnace. To keep within fiscal constraints, we have therefore deferred this test, while proceeding with matters more directly related to our charter. But the time is fast approaching when the resolution of this issue becomes indispensable to the charter, for, as indicated above, platinum is the most practical anode readily available.

3.4.4 The Cathode Problem

This is a problem of experimental design, which is expected to disappear upon scale-up. The cathode product is iron (at least in this present generation of experiments) and iron is either soluble in, or dissolves, the practical materials of which a cathode might be made. Haskins and Lindstrom (Ref. 3-13) encountered this problem in their micro studies, all of the iron they produced being found in solid solution in their platinum cathode, and detectable only by spectrographic analysis.

On a truly large scale, the problem ceases to exist: the cathode can itself be made of iron, molten at the reaction surface, but with the distal region cooled to below the melting-point of whatever metal is used for the electrical connection. But in an experiment in which the entire apparatus, including the cathode and its connector, must be immersed in a furnace, this practical solution does not apply.

Benefiting from Haskins' experience, we refrained from using platinum as cathode material; instead molybdenum and tungsten were used on various occasions. Predictably, the cathode products were iron alloys (or intermetallic compounds) with the cathode elements, rather than pure iron or spiegeleisen.

Nevertheless, these products represented a considerable advance over the previous efforts reported in the literature, for several reasons.

First, iron was produced in quantities up to a gram or more.

Second, the iron product, even though alloyed, was isolable from the bulk of the cathode. (Usually the product was clearly visible upon taking apart the electrolysis cell.)

Such compounds as Fe_3Mo_2 were readily identified as cathode product by X-ray diffraction and EDAX. Little, if any, of the iron that was made diffused into the cathode bulk. Hence:

Third, quantification of cathode efficiencies became possible with this type of experiment, and this promising avenue of work will be followed to its practical conclusion.

Note that estimates of cathode efficiency are subject to ambiguity, depending on

(1) magma composition, and
(2) the voltage drop at the cathode.

The ambiguity can be resolved by careful work, as we shall see.

For an example, take a typical experimental magma composition, which was (wt. %) 42 silica, 40 magnetite, and 18 magnesia. Given a sufficiently low voltage drop at the cathode, only magnetite would be reduced, giving iron as the sole product.[6] But there are other resistances in the circuit, and not all of these have been evaluated. Hence, to produce a practical result in a reasonable time, the overall voltage may be increased to a point where the local voltage at the cathode may exceed the value needed to produce iron alone. Under these conditions, silica or magnesia may be reduced also. In some experiments, there seems to be evidence for the production of magnesium or silicon monoxide at the cathode, in that some volatile has left a trace in the form of bubbles in the magma extending from the cathode. Both magnesium and silicon monoxide are gases at the magma temperature, 1450°. The question of which species is responsible remains to be resolved by more extensive work.

This will include:

(1) Coulometry. Exact comparisons between weighable cathode product and the current integral will be made.

(2) Search for alternate products:

  (a) Volatile, such as Mg and SiO, and

  (b) Reduced states of metals in the melt, exemplifying notably the reduction of ferric to ferrous, and transformations between states of titanium, and

  (c) Light-weight cathode products, notably silicon. Clearly, one need not proceed beyond step (1) if the iron produced at the cathode exactly equals the integral of current (over time) determined in the course of the experiment. In some cases this will perhaps be so. Yet each such experiment will suggest a further one in which current densities and voltages are raised by an arbitrary factor. At some point, the cathode process will result in an additional product or products. Then 2a, b, and c apply.

3.4.5 The Containment Problem.

Again, this problem is an artifact of the small scale of the experiments.

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[6] Production of metallic iron may be delayed during preliminary reduction of the +3 state to the +2 state. Obviously this melt, containing both ferrous and ferric ions, is capable of semiconduction.

For two thousand years, the glass industry has lived with the fact that most of its tools, including melting-pots (and the furnace-bricks themselves) ultimately become part of the glass. The point is that the container has a higher melting-point than the melt contained, and dissolves only slowly into the melt, rather than too rapidly. Hence the components of the container remain minority components of the melt, given a fast throughput of more fusible ingredients. The basis of this trick is that the reaction products of the container and the melt are much more viscous than the melt itself, which slows the rate of erosion.

An alternative way of reaching the same result is to impose a thermal gradient, rather than a compositional one. For "works-scale" experiments, or any other practical use in the lunar context, thermal gradients are seen as the basic answer. (This is not to exclude mixed strategies.)

But where the furnace imposes its own thermal gradient, which is frequently the opposite of that desired,[7] the viscosity shield must be obtained by means of a compositional gradient. Barriers of aluminum oxide are particularly effective, as this refractory reacts with basaltic magmas to form highly viscous aluminosilicates. Representative experimental setups will be found in Figures 3-4 and 3-5.

In sum, the year's endeavor has produced notable advances:

(1) Production of gram quantities of iron

(2) Reliable measurements of conductivity

and posed the next generation of problems, which are now in the process of being overcome.

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[7] That is, the furnace creates a condition where the reaction zone is hotter on the outside than on the inside. (This is typical of a rapidly heated resistance furnace.) But J. Schroeder has shown (Ref. 3-12) that large-scale electrolysis produces the desired condition, i.e., a hotter interior, due to resistance losses along the path of the electrolysis currents.

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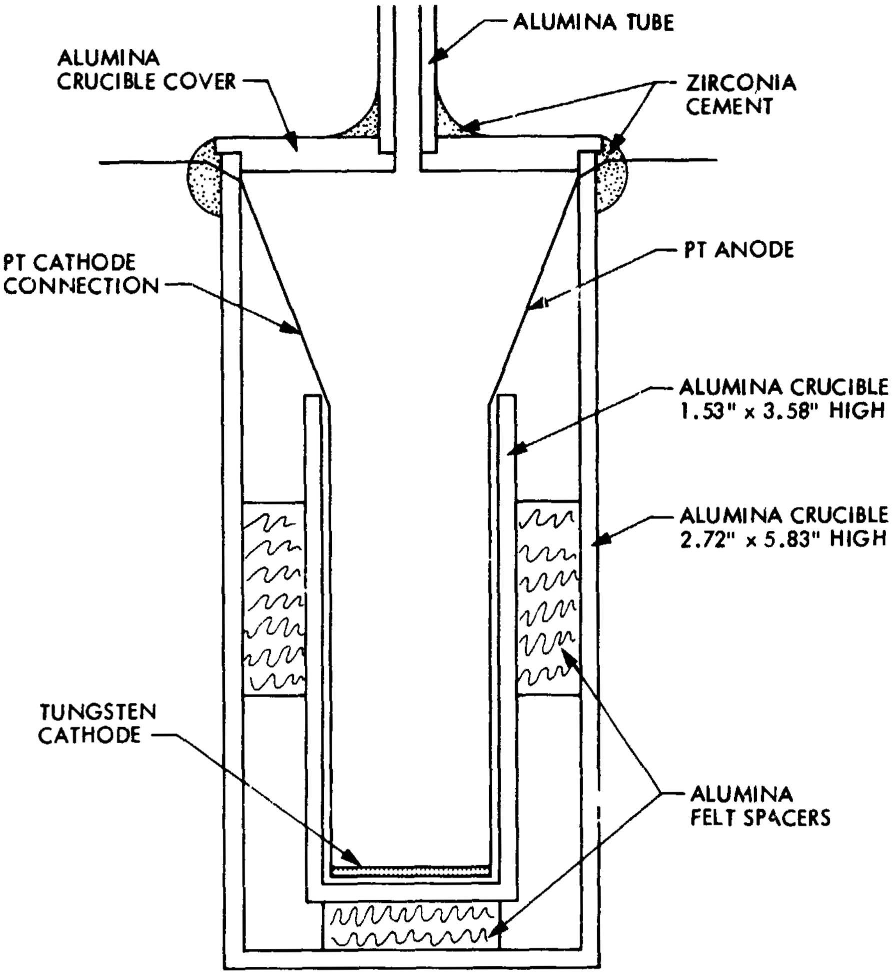

Figure 3-4. Electrolysis Cell

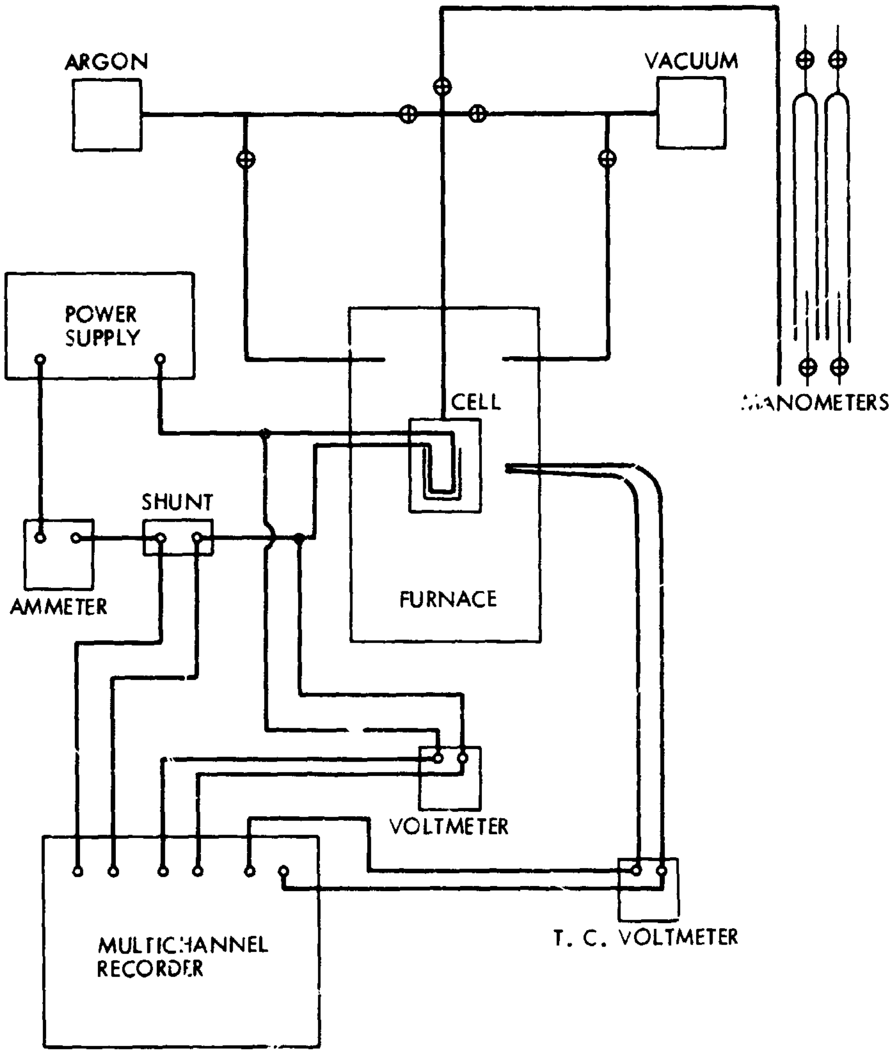

Figure 3-5. Schematic of Apparatus for Furnace Experiment

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SECTION 4

VAPOR PHASE REDUCTION

Wolfgang H. Steurer and Bruce A. Nerad

4.1 INTRODUCTION

The term "vapor phase reduction" comprises a family of processes in which metals and/or oxygen are extracted from the vaporized, dissociated or ionized raw material by various separation and recovery techniques. As pointed out in the preceding annual report (Ref. 4-1), these processes rely entirely on space environment and have therefore no terrestrial counterpart. By the same token, they are highly adaptable to space operations. Some of the attractive features of these processes are:

(1) Complete absence of consumables whose import from Earth is extremely expensive.

(2) Instant and complete consumption of the raw material.

(3) The process can be turned on or off at any time, without any preheating or shut-down time.

(4) Extensive utilization of space environments, such as vacuum and solar energy.

(5) High predictability of process performance by theoretical analysis.

The sole disadvantage is the high energy consumption, particularly in the case of operation in the ionized state. For this reason, extensive emphasis has been placed in FY'82 on the recovery of products from the dissociated vapor, which requires only moderately high temperatures and, therefore, lends itself to direct solar heating.

4.2 SELECTION OF SPECIFIC PROCESSES

A detailed analysis of various process concepts, identified in principle in the foregoing annual report, led to the selection of two specific process types which serve as the basis of further theoretical and experimental investigations. These two processes have been designated as "Vapor Separation" and "Selective Ionization".

The vapor-separation process is aimed at the production of oxygen, which has been defined as the most desirable product for use as propellant in near-term space operations. In this process, the granulated raw-material oxides are vaporized and the temperature raised to a point where dissociation takes place and a substantial amount of oxygen is set free. Rapid cooling of the dissociated vapor to a discrete temperature causes condensation of the oxides and suboxides while the free oxygen remains intact and can be collected downstream. The process concept is illustrated in Figure 4-1.

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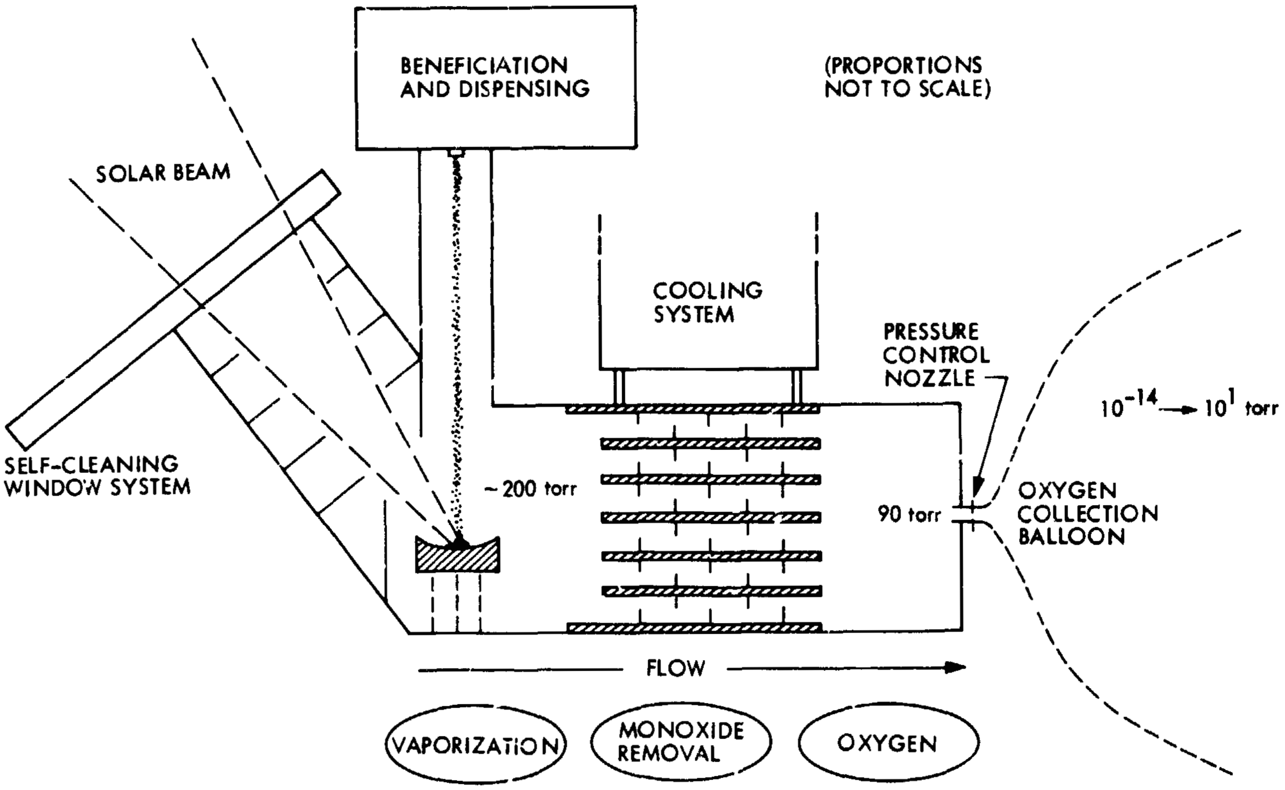

Figure 4-1. Vapor Separation--Process and Facility Concept

Theoretical studies indicate that the optimum operational temperature for this process is on the order of 3000 K. This moderate temperature permits extensive use of direct (concentrated) solar energy, which reduces the total facility mass substantially as opposed to solar-electric energy. A disadvantage of the process is the low operational pressure which implies either a low throughput or a high equipment volume, and consequently, equipment mass.

It could be argued that a further temperature increase by some 1000 K would produce a higher degree of dissociation and a higher oxygen yield; condensation of the free metal atoms would further produce a metal-rich residue. However, this potential was not pursued, at least for the time being, because it no longer lends itself to direct solar heating and entails other technological difficulties. Once solar-electric heating is introduced, it will probably be more effective to go one step further and to increase the temperature to a magnitude where a thermal plasma is obtained. The ionized species can then be separated, according to their positive or negative charge, in an electrostatic or an electromagnetic field.

Studies and calculations on the degree of ionization of various species over temperature led to the realization that, between 4000 and 8000 K, metals are increasingly ionized (+), while the oxygen remains essentially neutral up to 9000 K. This effect is illustrated in Figure 4-2, in which the ionization potential of various elements is plotted over temperature. It can be seen that at 8000 K (vertical line) over 90% of the metals are ionized, while oxygen, nitrogen or argon are below 1%. Consequently, in passing this plasma through an electrostatic field, the positively charged metals are captured at the cathode(s), while the neutral oxygen continues to flow downstream into an appropriate collection system (Figure 4-3).

Designated as "Selective Ionization" this concept was chosen as the second model process for in-depth study in view of the following unique features:

(1) It takes advantage of the ionization gap occurring at a discrete temperature.

(2) It provides a clean separation of metals and oxygen.

(3) It requires no earth-imported consumables, as it is based exclusively on space environments (solar-electric energy, vacuum).

(4) It can be turned on and off without leaving any residue or needing extended heat-up or cool-down periods.

4.3 VAPOR SEPARATION

The investigation of this process, described in principle above, was confined to theoretical studies with the objective to quantify the process effectiveness in terms of recoverable products, and to identify those problems which can only be solved by experiments, scheduled for FY'83. The analytical studies consisted of:

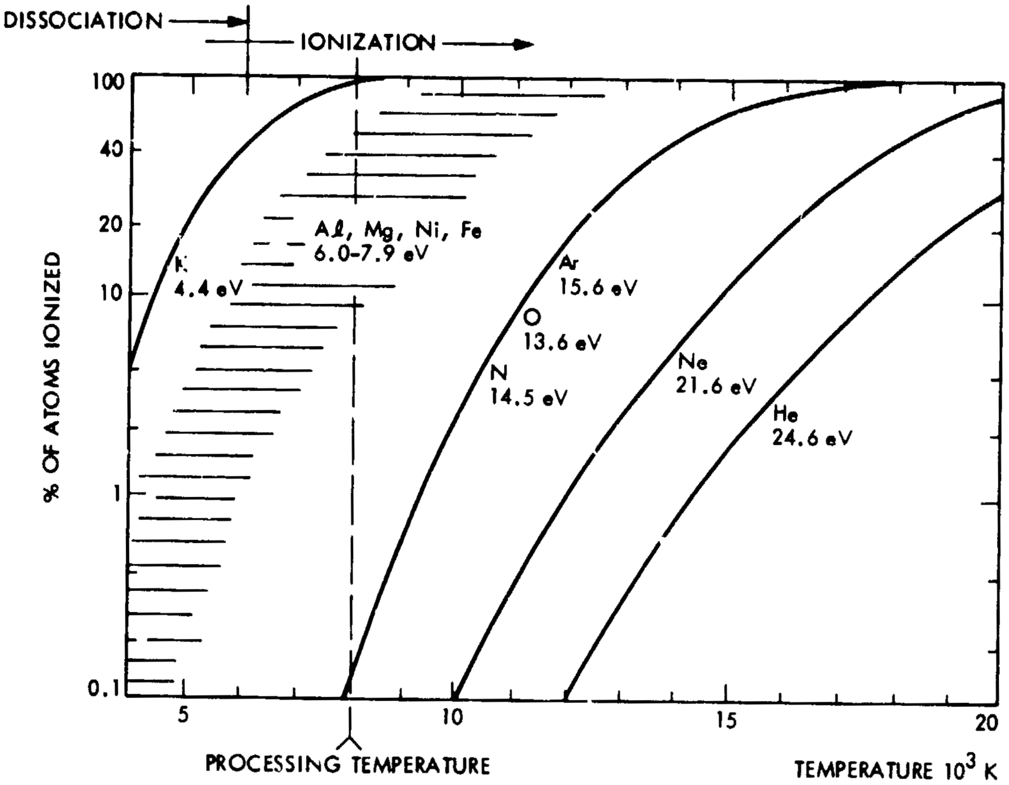

Figure 4-2. Ionization Potential of Elements vs Temperature

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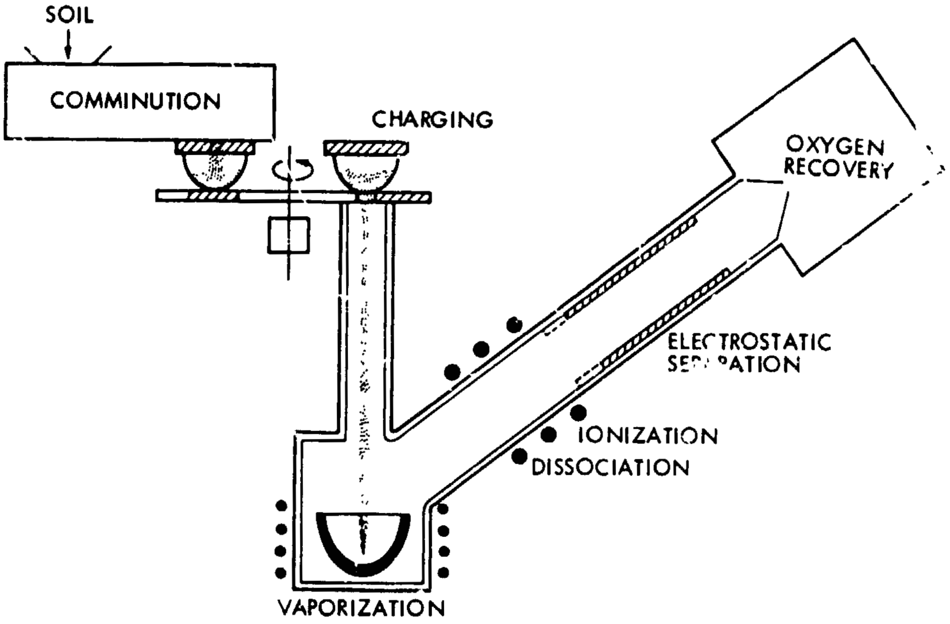

Figure 4-3. Selective Ionization--Process and Facility Concept

(1) Determination of the degree of dissociation of single oxides, as related to temperature.

(2) Assessment of the product yield with emphasis on oxygen.

(3) Composition of the vapor of an oxide mixture representative of the lunar soil (basalt).

(4) Determination of the energy requirements.

Conceptual studies were also carried out on process equipment and laboratory experiment designs.

4.3.1 Dissociation of Single Oxides

The increasing degree of vaporization and dissociation as related to temperature was measured by the vapor pressure of individual species vs temperature. Four oxides, predominant in the lunar basalt (soil) were selected for this analysis:

$$TiO_2, \; SiO_2, \; FeO, \; Al_2O_3$$

The approach taken is best illustrated by an example (TiO_2). In examining the JANAF tables (Ref. 4-2), the possible vaporization products of TiO_2 are $TiO_{2(g)}$ and $O_{2(g)}$.[1] To determine the equilibrium vapor pressures at a given temperature, the following equations were introduced:

$$TiO_{2(s)} \rightarrow TiO_{2(g)} \tag{4-1}$$

$$TiO_{2(s)} \rightarrow TiO_{(g)} + 1/2 \; O_{2(g)} \tag{4-2}$$

The equilibrium constants (K) for these reactions are:

$$K_1 = \frac{\gamma_{TiO_{2(g)}}}{\gamma_{TiO_{2(s)}}} \tag{4-3}$$

---

[1] The subscripts (s), (l) and (g) denote the solid, liquid and gaseous state, respectively.

$$K_2 = \frac{\gamma_{TiO_2(g)} \cdot \gamma_{O_2}{}^{1/2}{}_{(g)}}{\gamma_{TiO_2(s)}} \qquad (4\text{-}4)$$

where γ is the "activity". These equilibrium constants can be simplified by noting that the activity of a gas, if ideality is assumed, is equal to the partial pressure. Furthermore, the activity of a solid is defined to be 1 (unity). Therefore:

$$K_1 = P_{TiO_2(g)} \qquad (4\text{-}5)$$

$$K_2 = P_{TiO(g)} \cdot P_{O_2}{}^{1/2}{}_{(g)} \qquad (4\text{-}6)$$

Equation 4-2 further indicates

$$P_{TiO(g)} = 2P_{O_2(g)} \qquad (4\text{-}7)$$

We now have three independent equations (4-5, 4-6, and 4-7) with three unknowns, $P_{TiO_2(g)}$, $P_{TiO(g)}$ and $P_{O_2(g)}$.

Using the free energy (ΔG) data from the JANAF tables and the thermodynamic relationship

$$\Delta G = -RT \ln K_{eq} \qquad (4\text{-}8)$$

the temperature dependence of the vapor pressures can be determined. For example, at 500 K, the free energies of the individual species are:

$TiO_{2(s)}$: ΔG = -203.692 kcal/mole

$TiO_{2(g)}$: ΔG = -61.801 kcal/mole

$TiO_{(g)}$: ΔG = 8.230 kcal/mole

$O_{2(g)}$: ΔG = 0.000 kcal/mole

Thus, the equilibrium constants at 500 K are

$$K_1 = \exp\left[-\left(-\Delta G_{TiO_2}(s) + \Delta G_{TiO_2}(g)\right)/RT\right] = 9.43 \times 10^{-63} \qquad (4\text{-}9)$$

$$K_2 = \exp\left[-\left(-\Delta G_{TiO_2}(s) + \Delta G_{TiO}(g)\right)/RT\right] = 2.30 \times 10^{-93} \qquad (4\text{-}10)$$

From Equations 4-5 and 4-9 we find

$$P_{TiO_2(g)} = 9.43 \times 10^{-63} \text{ atm} = 7.16 \times 10^{-60} \text{torr}$$

To calculate $P_{TiO(g)}$ we use equations 4-6, 4-7, and 4-10. Substituting Equation 4-7 in Equation 4-6 we find:

$$K_2 = P_{TiO(g)}\left(1/2\ P_{TiO(g)}\right)^{1/2} \tag{4-11}$$

$$= P_{TiO}{}^{3/2}{}_{(g)}\ \sqrt{2}$$

Thus

$$P_{TiO(g)} = \left(\sqrt{2}\ K_2\right)^{2/3} \tag{4-12}$$

$$(500\ K) = 2.20 \times 10^{-62}\ atm = 1.67 \times 10^{-59}\ torr$$

and

$$P_{O_2(g)} = 1/2\ P_{TiO(g)} \tag{4-13}$$

$$(500\ K) = 1.10 \times 10^{-62}\ atm = 8.35 \times 10^{-60}\ torr$$

For SiO_2, the calculation of the partial pressures of the individual vapor species (SiO_2, SiO, O_2) followed the same procedure as outlined above for TiO_2. The vaporization products of Al_2O_3 consist of Al_2O_2, Al_2O, AlO_2, AlO and O_2; thus, the calculations involve 5 independent equations. For $FeO_{(s)}$, the work is trivial, as the sole vapor species present is $FeO_{(g)}$.

The results of the calculations for the four selected oxides (SiO_2, TiO_2, Al_2O_3, FeO) are presented in Table 4-1. The temperature dependence of the partial pressures of the individual dissociation species is further illustrated in Figure 4-4 in the form of the customary ln P (atm) vs 1/T plot. While the Clausius-Clapeyron equation postulates straight lines for ideal gases and constant enthalpy changes vs temperature, the calculated data produce slight bowed lines due to the now-constant ΔH over the entire temperature range.

An examination of these data indicates relatively high partial pressures at temperatures above 2,500 K which permit high throughput and production rates. This applies particularly to the free oxygen, considered the prime near-term product of interest. A quantitative assessment of the oxygen yield is presented in the following section.

4.3.2 Oxygen Yield

The oxygen yield was defined as the oxygen weight fraction of the total gas weight or

$$\%\ O_2 = \frac{\text{Weight of Gaseous Oxygen}}{\text{Weight of Total Gas}} \times 100 \tag{4-14}$$

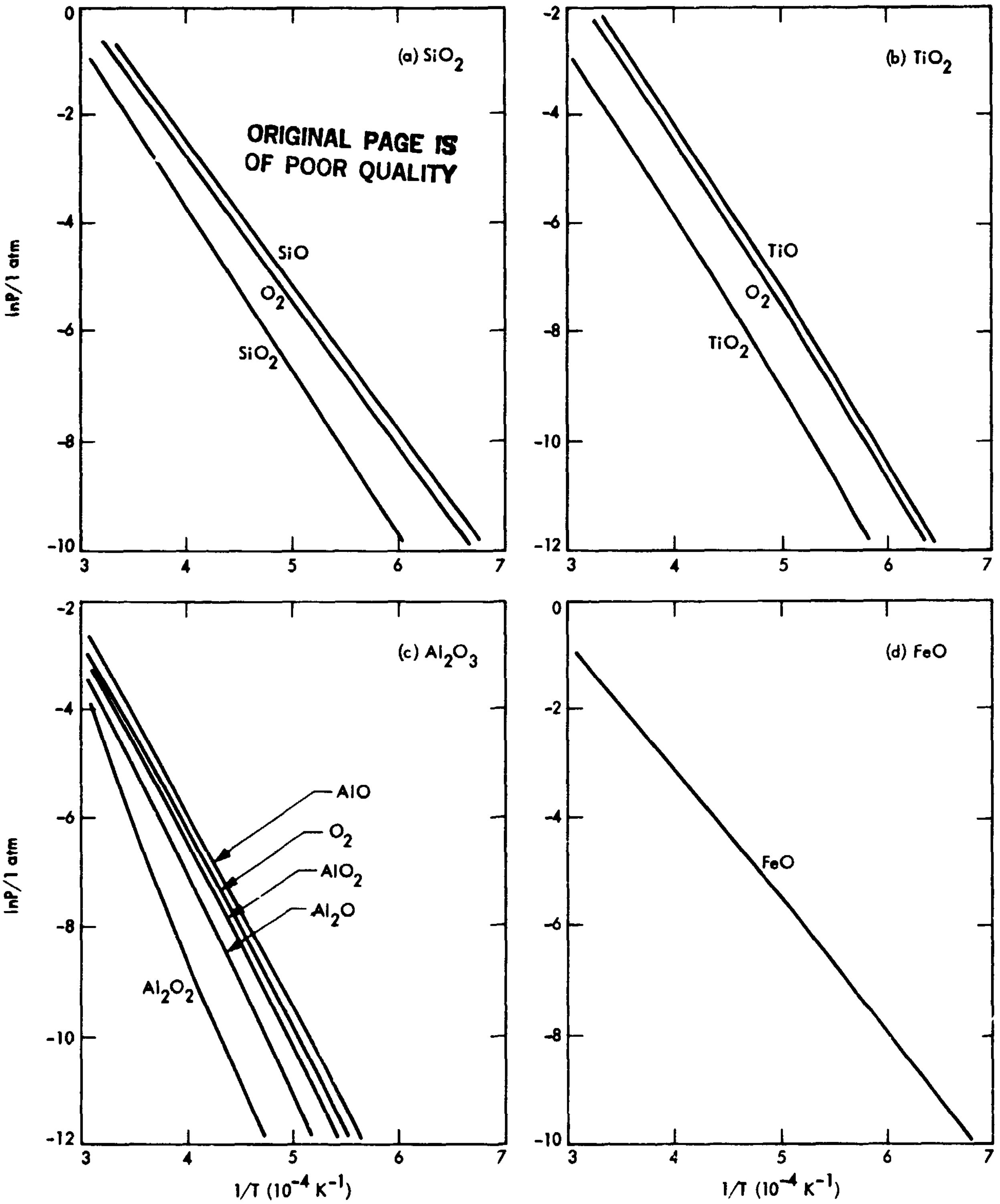

Figure 4-4. Partial Pressures of the Dissociation Species of Selected Oxides vs Temperature: a) SiO_2, b) TiO_2, c) Al_2O_3, d) FeO.

Table 4-1. Vapor Pressure of Oxide Dissociation Species at Increasing Temperature.

		Vapor Pressure of Species (torr) at K				
Oxide	Vapor Species	1000	1500	2000	2500	3000
SiO_2	SiO_2	6.96×10^{-20}	1.60×10^{-9}	1.92×10^{-4}	1.77×10^{-1}	1.53×10^{1}
	SiO	1.18×10^{-16}	2.07×10^{-7}	7.13×10^{-3}	3.26	1.80×10^{2}
	O_2	5.90×10^{-17}	1.04×10^{-7}	3.57×10^{-3}	1.63	9.00×10^{1}
TiO_2	TiO_2	2.57×10^{-24}	1.36×10^{-12}	8.22×10^{-7}	1.23×10^{-3}	1.20×10^{-1}
	TiO	1.91×10^{-21}	1.81×10^{-10}	4.78×10^{-5}	5.36×10^{-2}	4.71
	O_2	9.55×10^{-22}	9.05×10^{-11}	2.39×10^{-5}	2.68×10^{-2}	2.34
Al_2O_3	Al_2O_2	---	2.52×10^{-18}	8.13×10^{-11}	1.50×10^{-6}	5.46×10^{-3}
	AlO_2	---	4.15×10^{-16}	5.72×10^{-9}	7.30×10^{-5}	2.84×10^{-2}
	Al_2O	---	9.80×10^{-15}	4.63×10^{-8}	2.99×10^{-4}	6.43×10^{-2}
	AlO	---	1.31×10^{-13}	2.24×10^{-7}	9.42×10^{-4}	2.11×10^{-1}
	O_2	---	4.26×10^{-14}	1.01×10^{-7}	5.17×10^{-4}	1.09×10^{-1}
FeO	FeO	3.18×10^{-16}	1.88×10^{-7}	2.56×10^{-3}	5.81×10^{-1}	1.92×10^{1}

For the example of TiO_2 the oxygen yield is

$$\%\ O_2 = \frac{M_{O_2} P_{O_2}}{M_{O_2}P_{O_2} + M_{TiO}P_{TiO} + M_{TiO_2}P_{TiO_2}} \times 100 \qquad (4\text{-}15)$$

where M is the mole weight and P the partial pressure of each species. Introducing the numerical values for TiO_2 at 2000 K:

Species	M(g/mole)	P(torr)
$TiO_2(g)$	79.9	8.22×10^{-7}
$TiO(g)$	63.9	4.78×10^{-5}
$O_2(g)$	32	2.39×10^{-5}

results in an oxygen yield of 19.7%. The calculated oxygen yields (in % by weight) for the four selected oxides and various temperatures are listed in Table 4-2.

Table 4-2. Oxygen Yield (% by Weight) of Oxides vs Temperature

	Temperature				
Oxide	1000 K	1500 K	2000 K	2500 K	3000 K
TiO_2	20.0	19.9	19.7	19.6	19.5
SiO_2	26.6	26.4	25.9	25.3	24.5
FeO	0	0	0	0	0
Al_2O_3	-	17.7	19.6	20.1	18.2

To place these data in perspective, the approximate oxygen yield in the 2000-3000 K range is compared with the total oxygen (in % by weight) contained in each oxide:

Oxide	O_2 Yield (% of Oxide) %	Total O_2 (% of Oxide) %	O_2 Yield (% of Total O_2)
TiO_2	19.6	40.1	49
SiO_2	25	53.2	47
FeO	0	22.3	0
Al_2O_3	19.3	47.1	40

Thus, in every case except FeO, approximately half of all oxygen contained in each oxide is extractable.

Two additional observations can be made from the data of Table 4-2:

(1) The oxygen yield is insensitive to temperature. It should be noted, however, that only the higher temperature regime (above 2000 K) is of practical interest, since the higher overall pressure permits a useful throughput and, consequently, output rate.

(2) The oxygen yield of FeO is zero. It will, therefore, be beneficial to remove Fe-rich particles prior to vapor-phase processing by magnetic separation. The extracted FeO/Fe_3O_4 could be used to advantage in electrolytic processing.

4.3.3 Oxygen and Metal Recovery

As stated initially, the envisioned technique of suboxide removal is rapid cooling of the gas to a temperature in the order of 500 K and at a rate which prevents reaction with the free oxygen and recombination to the original oxide. The produced oxygen flowing downstream under a controlled pressure differential can then be collected in a balloon (see Figure 4-1), whose pumping capability is provided by the environmental vacuum of 10^{-14} torr. (Liquefaction may not be necesary, since in the drag-free and volume-insensitive environment of space operations the use of oxygen in gaseous form is entirely feasible.)

In the rapid cooling process the condensing vapors, such as TiO, SiO, TiO_2, SiO_2, AlO, etc., will in some instances disproportionate to form the original oxide and free metal, according to the following equations.

$$TiO_{(g)} \rightarrow TiO_{(s)} \qquad (4\text{-}16)$$

$$FeO_{(g)} \rightarrow FeO_{(s)} \qquad (4\text{-}17)$$

$$2\ SiO_{(g)} \rightarrow SiO_{2(s)} + Si_{(s)} \qquad (4\text{-}18)$$

$$AlO,\ AlO_2,\ Al_2O \rightarrow Al_2O_{3(s)} + AL_{(s)} \qquad (4\text{-}19)$$

As can be seen from Equations 4-18 and 4-19 the condensation of the suboxides of Al and Si may be associated with the formation of free metals. Tentative calculations of the metal yield indicate the following maxima (in % by weight of the original oxide):

Oxide	Metal	1000 K	1500 K	2000 K	2500 K	3000 K
SiO_2	Si	23.4	23.2	22.7	22.2	21.5
Al_2O_3	Al	-	19.9	21.1	22.6	20.5

The data represent close to half of all metal contained in the oxide (SiO_2 46.8% and Al_2O_3 52.9%).

4.3.4 Vapor Composition of Oxide Mixtures

To approach the conditions of the processing of lunar soil by vapor phase techniques more accurately, the theoretical analysis was extended to the prediction of the vapor composition of a mixture of oxides. The model mixture consisted (as in the treatment of individual oxides in 4.2.1) of SiO_2, TiO_2, FeO and Al_2O_3, representing major constituents of the lunar soil. The vapor composition was again expressed in terms of partial pressures.

Assuming an ideal mixture of these oxides, i.e., that the activity of all solids is equivalent to the mole fraction, and the activity of the gases is equal to their respective partial pressures, the thermodynamic data of the JANAF tables (Ref. 4-2) can be used to calculate compositional vapor pressures. The equilibrium constants for the various species of dissociation were defined as follows:

Solid	→	Gas	Equilibrium Constants	
Al_2O_3	→	$2AlO + \frac{1}{2}O_2$	$A = [AlO]^2[O_2]^{\frac{1}{2}} / x_{Al_2O_3}$	(4-20)
Al_2O_3	→	$Al_2O + O_2$	$B = [Al_2O][O_2] / x_{Al_2O_3}$	(4-21)
$Al_2O_3 + \frac{1}{2}O_2$	→	$2Al_2O_2$	$C = [AlO_2]^2 / [O_2]^{\frac{1}{2}} x_{Al_2O_3}$	(4-22)
Al_2O_3	→	$Al_2O_2 + \frac{1}{2}O_2$	$D = [Al_2O_2][O_2]^{\frac{1}{2}} / x_{Al_2O_3}$	(4-23)
SiO_2	→	$SiO + \frac{1}{2}O_2$	$E = [SiO][O_2]^{\frac{1}{2}} / x_{SiO_2}$	(4-24)
SiO_2	→	SiO_2	$F = [SiO_2] / x_{SiO_2}$	(4-25)
TiO_2	→	$TiO + \frac{1}{2}O_2$	$G = [TiO][O_2]^{\frac{1}{2}} / x_{TiO_2}$	(4-26)
TiO_2	→	TiO_2	$H = [TiO_2] / x_{TiO_2}$	(4-27)
FeO	→	FeO	$I = [FeO] / x_{FeO}$	(4-28)

The equilibrium vapor species are coupled through the common evolution of oxygen. From stoichiometry the total oxygen vapor pressure is:

$$[O_2] = 1/4[AlO] + [Al_2O] - 1/4\,[AlO_2] + 1/2[Al_2O_2] + 1/2[SiO] + 1/2\,[TiO] \qquad (4\text{-}29)$$

Substituting in terms of the various equilibrium constants, above, the oxygen vapor pressure can be expressed as:

$$[O_2] = \frac{A^{1/2}X_{Al_2O_3}}{4\,[O_2]^{1/4}} + \frac{BX_{Al_2O_3}}{[O_2]} + \frac{C^{1/2}[O_2]^{1/4}X_{Al_2O_3}}{4}$$

$$+ \frac{DX_{Al_2O_3}}{2[O_2]^{1/2}} + \frac{EX_{SiO_2}}{2[O_2]^{1/2}} + \frac{GX_{TiO_2}}{2[O_2]^{1/2}} \qquad (4\text{-}30)$$

Multiplying by $4[O_2]$ and letting $y = [O_2]^{1/4}$ leads to the following eighth power equation:

$$0 = 4y^8 + X_{Al_2O_3}C^{1/2}y^5 - X_{Al_2O_3}A^{1/2}y^3$$

$$- \left(2X_{Al_2O_3}D + 2X_{SiO_2}E + 2X_{TiO_2}\right)y^2 - 4X_{Al_2O_3}B \qquad (4\text{-}31)$$

By solving this eighth power equation and the nine equilibrium equations, the compositional vapor pressure can be determined. This was carried out by a specially developed computer program for the temperature range from 1000 to 3000 K and for the following model composition of the starting material, representative of major constituents of lunar basalt:

Oxides	% (Weight)
Al_2O_3	15
SiO_2	50
TiO_2	10
FeO	25

The results in terms of compositional vapor pressures are presented in Table 4-3. The table shows that the pressures rise sharply with temperature, as expected. At 3000 K, for example, the pressure of the free oxygen is on the order of 0.1 atm. While the oxygen yield per unit of raw material is almost independent of temperature (as shown 4.3.2/Table 4-2) and on the order of 20%, higher pressures permit higher throughput and, consequently, output rates.

The output can be further increased by magnetic removal of the iron-oxide particles from the raw material prior to processing, since FeO does not contribute to the generation of free oxygen. The enhancement of the free oxygen content by such beneficiation--in terms of vapor pressure--is identified in Table 4-4. This enhancement increases the oxygen yield from the earlier defined 20% to 24% of the feedstock (=100%).

Table 4-3. Compositional Vapor Pressure (atm) of a Mixture of Al_2O_3, SiO_2, TiO_2 and FeO.

Species		1000 K	1500 K	2000 K	2500 K	3000 K
$O_{2(g)}$	atm	5.4×10^{-20}	9.2×10^{-11}	3.2×10^{-6}	1.5×10^{-3}	8.1×10^{-2}
$AlO_{(g)}$	atm	4.6×10^{-32}	1.6×10^{-18}	7.3×10^{-12}	5.6×10^{-8}	1.8×10^{-5}
$Al_2O_{(g)}$	atm	4.1×10^{-42}	9.9×10^{-25}	2.5×10^{-16}	1.6×10^{-11}	1.6×10^{-8}
$AlO_{2(g)}$	atm	1.7×10^{-31}	6.7×10^{-18}	2.9×10^{-11}	2.0×10^{-7}	5.8×10^{-5}
$Al_2O_{2(g)}$	atm	1.9×10^{-42}	3.3×10^{-25}	6.8×10^{-17}	3.7×10^{-12}	3.2×10^{-9}
$SiO_{(g)}$	atm	1.1×10^{-19}	1.8×10^{-10}	6.5×10^{-6}	3.0×10^{-3}	1.6×10^{-1}
$SiO_{2(g)}$	atm	5.3×10^{-23}	1.2×10^{-12}	1.4×10^{-7}	1.4×10^{-4}	1.1×10^{-2}
$TiO_{(g)}$	atm	9.6×10^{-28}	8.4×10^{-16}	4.8×10^{-10}	9.3×10^{-7}	1.0×10^{-4}
$TiO_{2(g)}$	atm	2.7×10^{-28}	1.8×10^{-16}	8.4×10^{-11}	1.4×10^{-7}	1.3×10^{-5}
$FeO_{(g)}$	atm	1.1×10^{-19}	5.3×10^{-11}	8.1×10^{-7}	2.0×10^{-4}	5.9×10^{-3}

Table 4-4. Oxygen Vapor Pressure of the Model Mixture With and Without Prior Beneficiation.

Temperature (K)	Without (atm)	With (atm)	Enhancement (%)
1000	5.43 10^{-20}	6.50 10^{-20}	19.7
1500	9.25 10^{-11}	1.11 10^{-10}	20.0
2000	3.23 10^{-6}	3.87 10^{-6}	19.8
2500	1.51 10^{-3}	1.81 10^{-3}	19.9
3000	8.12 10^{-2}	9.73 10^{-2}	19.8

4.3.5 Energy Requirements

The assessment of the energy required for the vapor separation process involves the two major consumers:

(1) The vaporization and dissociation of the raw material.

(2) The quenching of the vapors for the condensation of the sub-oxides.

For (1) reasonably accurate calculations were carried out using thermodynamic data from Refs. 4-2 and 4-3. Energy data for the quenching phase (2) were merely estimated since they depend extensively on the not yet defined design of the cooling system.

Since thermodynamic data for oxide mixtures are not available, the energy requirements for the vaporization phase (1) were calculated for each individual oxide from data on the heats of formation, vaporization and dissociation. The resulting figures were then adjusted commensurate with the raw-material fraction of each oxide. The energy requirement of the starting mixture was determined as the sum of the individual adjusted data. It amounted to 4.8 kcal/g or 5,100 kWh/ton. Interactions in the vapor, which may alter this figure slightly, were disregarded, since at this time only a first approximation was required.

For the cooling phase (2) a conservative estimate of 2000 kWh/ton was introduced. Thus, the total energy required for processing is on the order of 7,1000 kWh/ton of raw-material.

As mentioned earlier, at this time the emphasis is on oxygen production (even though the condensed metal-rich oxides and pure metals may eventually be also of interest). Since the maximum oxygen output is on the order of 24% (with enhancement), the total energy required per ton of oxygen is

$$E_{O_2} = \frac{7,100}{0.24} = 29.6 \text{ kWh/ton}_{O_2}$$

or roughly 30,000 kWh/ton of oxygen.

It should be remembered, however, that the major part of the total energy is provided by direct solar heating (solar concentrator). Electric energy applies primarily to the cooling system. Assuming 65% direct solar heating, the energy forms and amounts for the production of oxygen are:

Direct Solar	19.2	kWh/ton_{O_2}
Electric	10.4	kWh/ton_{O_2}

4.3.6 Experiments

In the overall plan of the RUSR program, experiments on vapor phase reduction were deferred to FY'83. The objectives of experimental investigations planned for FY'83 are as follows:

(1) Qualitative and quantitative determination of the vaporization/dissociation species of single oxides and oxide mixtures for verification or updating of the theoretical data.

(2) Determination of the oxygen yield of mixed oxides and of synthetic lunar raw material.

(3) Effectiveness of various cooling-system designs with regard to sub-oxide condensation and the cleanliness of the recovered oxygen.

Experiments (1) will be primarily carried out by Purdue University (JPL Contract 956322) at temperatures up to 2000 K, using mass-spectrometry techniques for the quantitative measurement of dissociation species. Starting materials will comprise single oxides, oxide mixtures and meteorite samples.

Experiments (2) and (3) will be performed at JPL using induction heating. Temperatures will range from 2000 K to 3000 K. Samples will comprise single oxides, mixed oxides and synthetic lunar minerals. Initial experiments will use argon as carrier gas, to be gradually eliminated.

4.4 SELECTIVE IONIZATION

As described in principle in Section 4.1, this processing concept utilizes the ionization gap in a thermal plasma at a discrete temperature range for the separation of metals and oxygen. The evaluation of this process was confined to theoretical studies in the reporting period, while experiments were deferred to FY'83. The studies were aimed at the definition of the encountered species at various temperature levels of the thermal plasma, the degree of ionization, the expected metal and oxygen yield and the required processing energy.

It is emphasized that the primary purpose of the generated data was to substantiate the feasibility and effectiveness of the process. They should, therefore, be regarded as tentative, to be refined in subsequent studies supported by experimental data.

4.4.1 Ionization Effectiveness and Ionization Species

The previous illustration of the ionization gap between metals and oxygen in the 7000 to 10,000 K temperature regime (Figure 4-2) was based solely on the ionization potentials of selected elements. In order to obtain more accurate numerical data on the equilibrium ratio of ionized to neutral atoms vs. temperature, the Saha equation (Ref. 4-4) was applied to the individual elements of interest. For the purpose of this phase of investigation, it was deemed adequate to rewrite the Saha equation for first-level ionization, which reads as follows:

$$\frac{n_1 n_e}{n_0} = \frac{2u_1(T)}{u_0} \frac{(2\Pi m_0 k)^{3/2}}{h^3} T^{3/2} \exp\left[\frac{\Delta E_0 - E_0}{kT}\right] \qquad (4\text{-}32)$$

where:

n_0 = number density of all neutral species
n_1 = number density of all positive ions
n_e = number density of electrons
$u_0(T)$ = partition function of the neutral atom
$u_1(T)$ = partition function of the charged ion
m_0 = electron rest mass
h = Planck's constant
k = Boltzmann constant
E_0 = ionization energy
T = absolute temperature
ΔE_0 = lowering of ionization energy

To evaluate this equation, a system containing a single element at 1 atmosphere was assumed. The equilibrium between the neutral species and singly charged ions was calculated, in the temperature range from 4000 to 15,000 K, for elements present in significant concentrations in the lunar soil, namely Al, Fe, O, Si and Ti. The results are depicted graphically in Figure 4-5 and in tabular form in Table 4-5. A substantial gap exists between the temperature at which the metal atoms and oxygen first show ionization. It is this gap which is exploited in selective ionization to separate metals and oxygen.

Assuming 100% collection efficiency of the charged species, a theoretical yield can be estimated for this simplified single element approach. As can be seen from Figure 4-5, the theoretical degree of metal ionization ranges from 27% for Si to 91% for Ti at 8000 K and from 72% to 98%

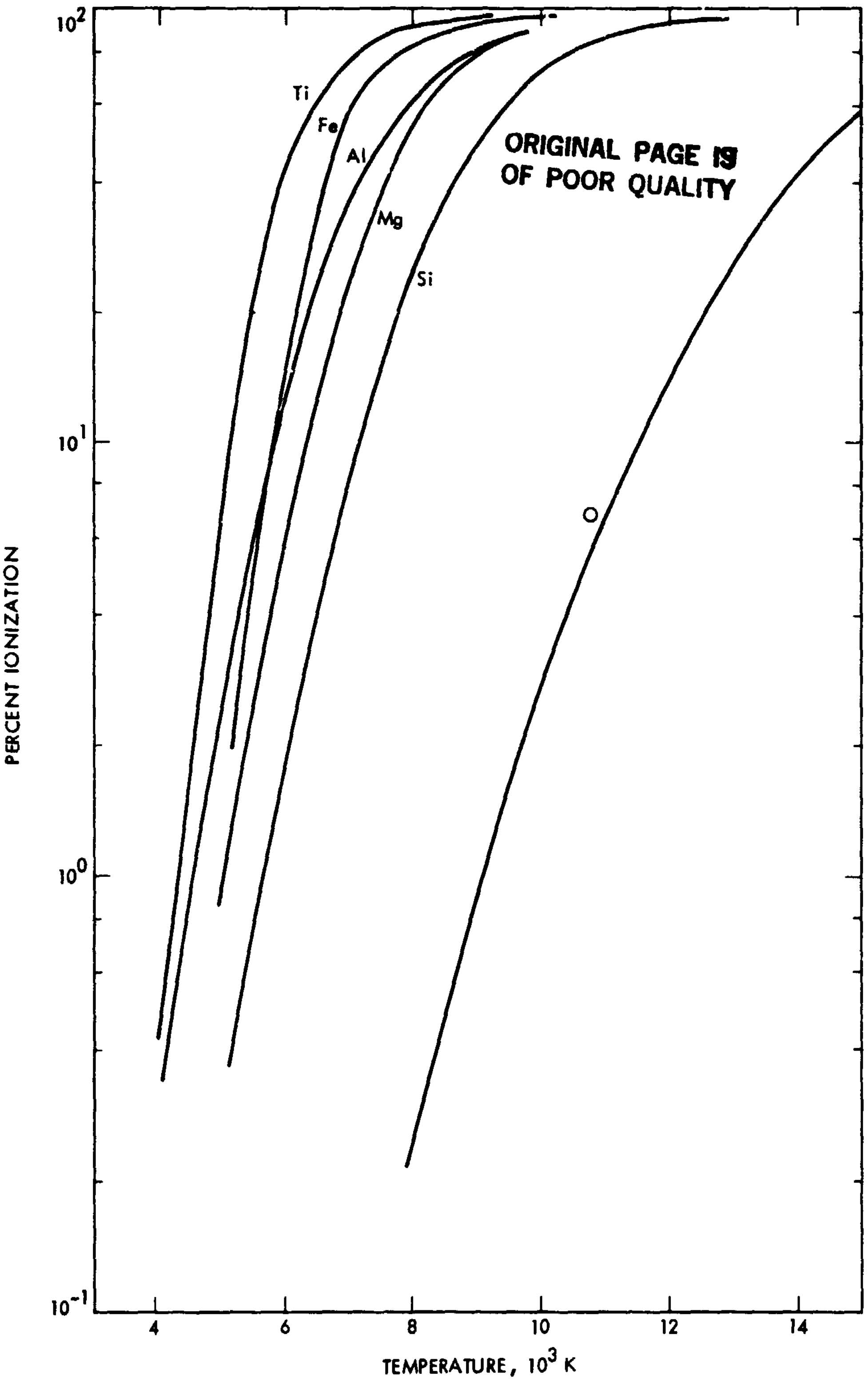

Figure 4-5. Percent ionization of Metal and Oxygen vs Temperature

Table 4-5. Degree of Ionization of Metals and Oxygen Between 7000 K and 10,000 K

Ratio M^+/M_{Total}	Temperature, K			
	7000	8000	9000	10,000
Ti^+/Ti_T	0.76	0.91	0.96	0.98
Fe^+/Fe_T	0.59	0.84	0.92	0.96
$A\ell^+/A\ell_T$	0.37	0.64	0.81	0.90
Mg^+/Mg_T	0.24	0.57	0.76	0.90
Si^+/Si_T	0.08	0.25	0.49	0.72
O^+/O_T	0.0007	0.0025	0.0095	0.026

at 10,000 K, respectively[2]. In contrast the oxygen remains essentially neutral with only 0.25% ionization at 8000 K and 2.6% at 10,000 K. However, these theoretical ion/atom ratios may be lowered to a yet to be determined degree by several effects, such as:

(1) The presence of stable metal-oxygen equilibrium products at high temperatures could reduce the predicted product yield. A thermodynamic approach was adopted to examine the equilibrium product distribution over the temperature range of 4000 to 10,000 K. The equilibrium distribution species of the oxides Al_2O_3, Fe_2O_3, SiO_2 and TiO_2 at 1 atmosphere were individually calculated. Free energy data were obtained from the JANAF tables (Ref. 4-2) and extrapolated quadratically to 10,000 K.

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[2] Other work (T. Yoshida et al, Ref. 4-5) suggests that better than a 90% yield of 95% pure iron is achieved at 10,000 in an iron-oxygen system.

The quadratic extrapolation avoided excessive computer time and yielded data with an uncertainty of a few percent at 6000 K and approximately 20% at 10,000 K. The data for the individual oxides are presented in Figures 4-6 to 4-9. In all cases, the only stable metal-oxygen compound is the mono-oxide. At 10,000 K, including an uncertainty of 20%, the mono-oxide is present in concentrations three orders of magnitude less than the free or charged metal. Therefore, metal-oxygen equilibrium compounds will not affect the possibility of obtaining theoretical yields of 70-100%.

(2) In the selective ionization process, free electrons are formed from the ionization of several species, and thus the ionization of one element in a mixture will affect the equilibrium loss of an electron from another. This "common-ion" effect is similar to the reduced solubility of mixed salts in water when an ion is common to each salt. The solubility of the least soluble salt will be suppressed by the presence of the common ion. In selective ionization, the tendency of oxygen to lose an electron will be lessened by the formation of metal-derived free electrons. Therefore, the oxygen ionization levels should be less than predicted by the simple single element approach of Figure 4-5.

(3) The theoretical calculation of yields is dependent on the existence of thermodynamic equilibrium during the selective ionization process. In the conceived thermal plasma method, near-equilibrium conditions exist for heavy particles such as molecules, atoms and ions. Electrons, however, violate Saha equilibrium, due to their rapid diffusion and the high temperature gradients near the confinement tube (Ref. 4-6). The result is two-fold: First, the Saha-derived temperature is too high near the walls. This fact is important only in plasma diagnostics and will not affect the predicted theoretical yield. Second, the total amount of ionization is limited by the diffusion of electrons to the confinement tube, where electron-wall collisions dissipate electron kinetic energy and result in ion-electron recombination rates higher than assumed by the Saha equation. The limitation on ionization may affect the possibility of obtaining the equilibrium electron density assumed in Figure 4-5, and thus would result in a decrease in the product yield. The impact of this electron non-equilibrium on ionization levels and yields will be examined when experimental data become available in FY'83.

4.4.2 Product Yield

Considering the factors outlined above, it is to be expected that the effective metal and oxygen yield of a multiple oxide system is substantially less than indicated by the single-oxide evaluation. To arrive

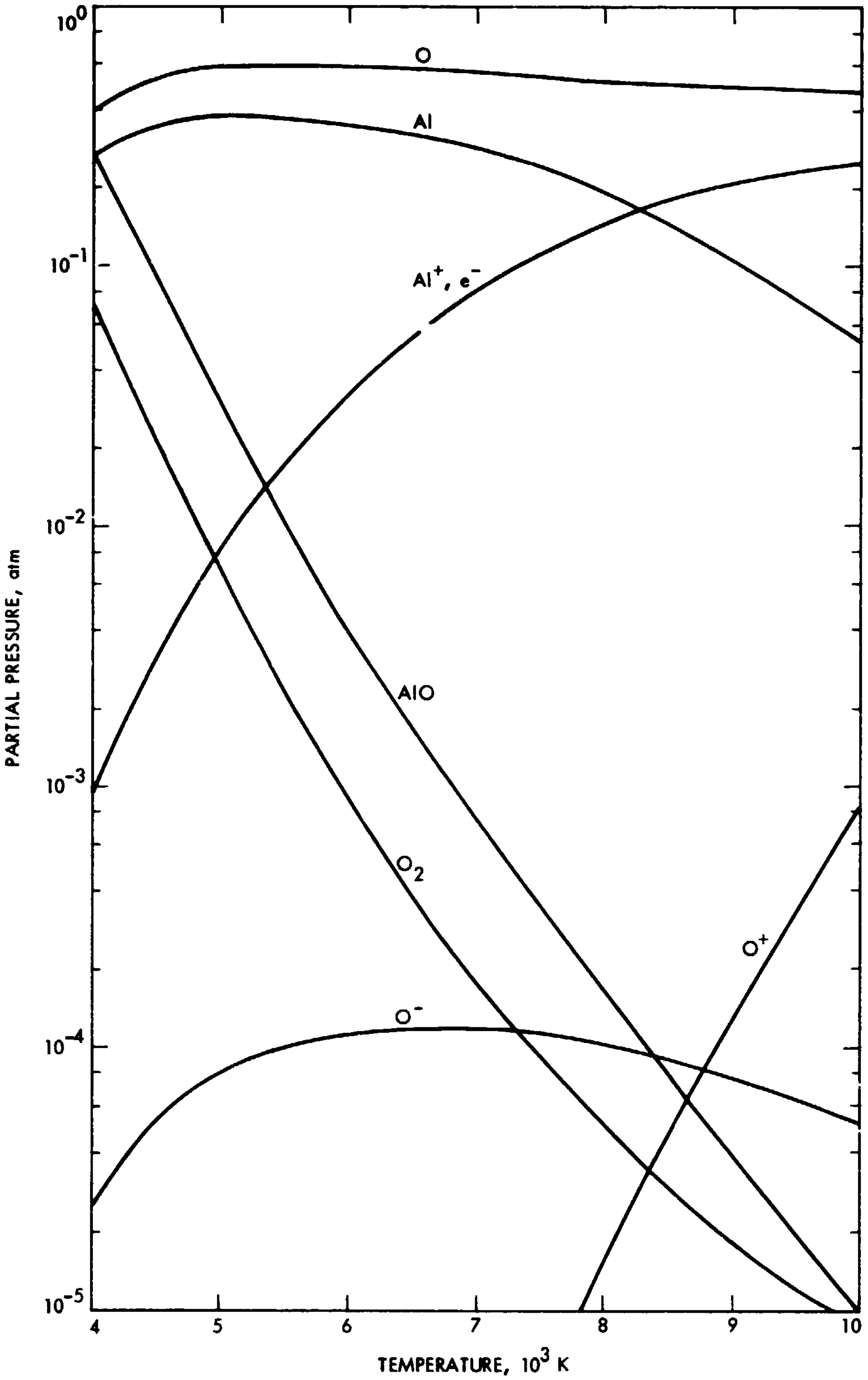

Figure 4-6. Equilibrium Species Distribution of Al_2O_3 vs Temperature

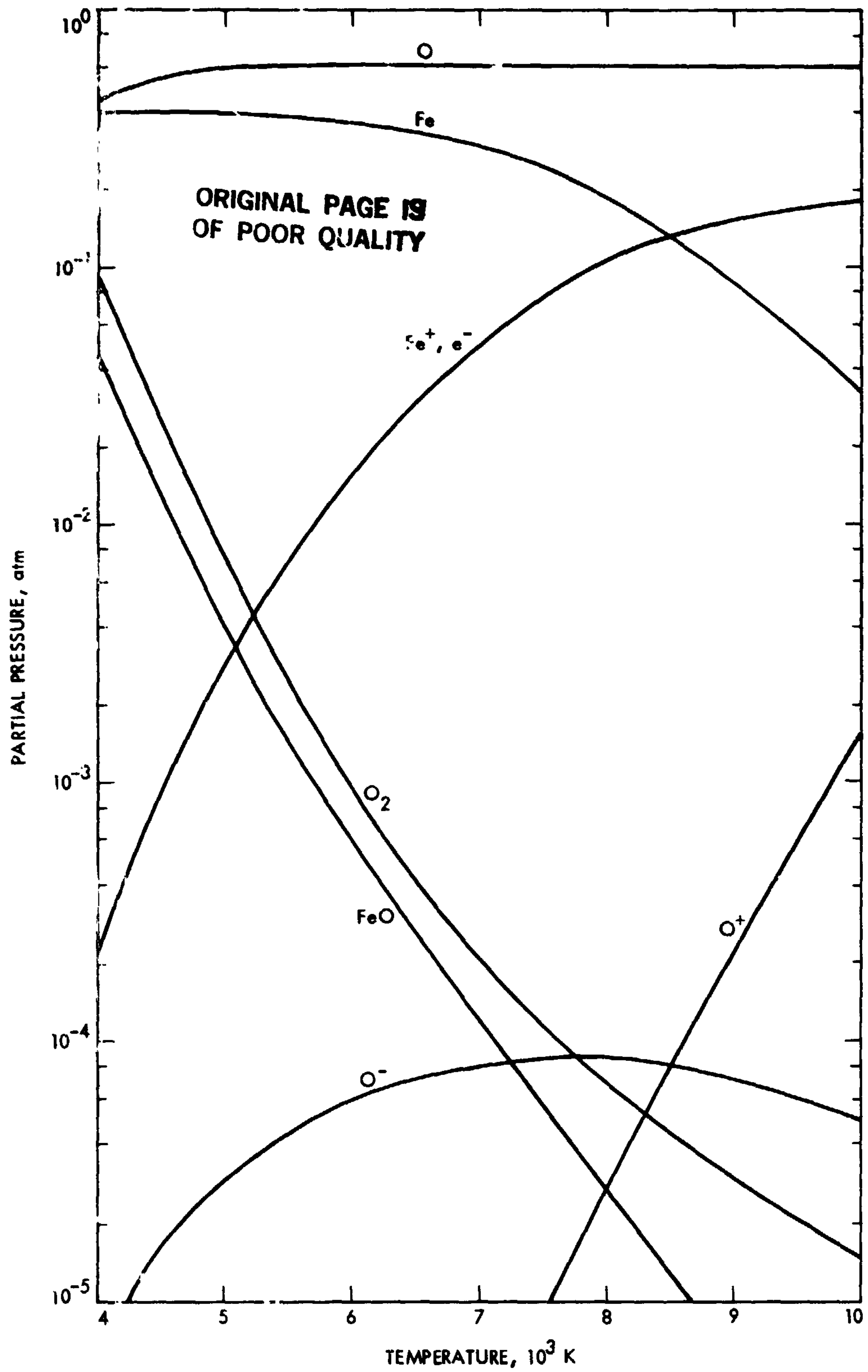

Figure 4-7. Equilibrium Species Distribution of Fe_2O_3 vs Temperature

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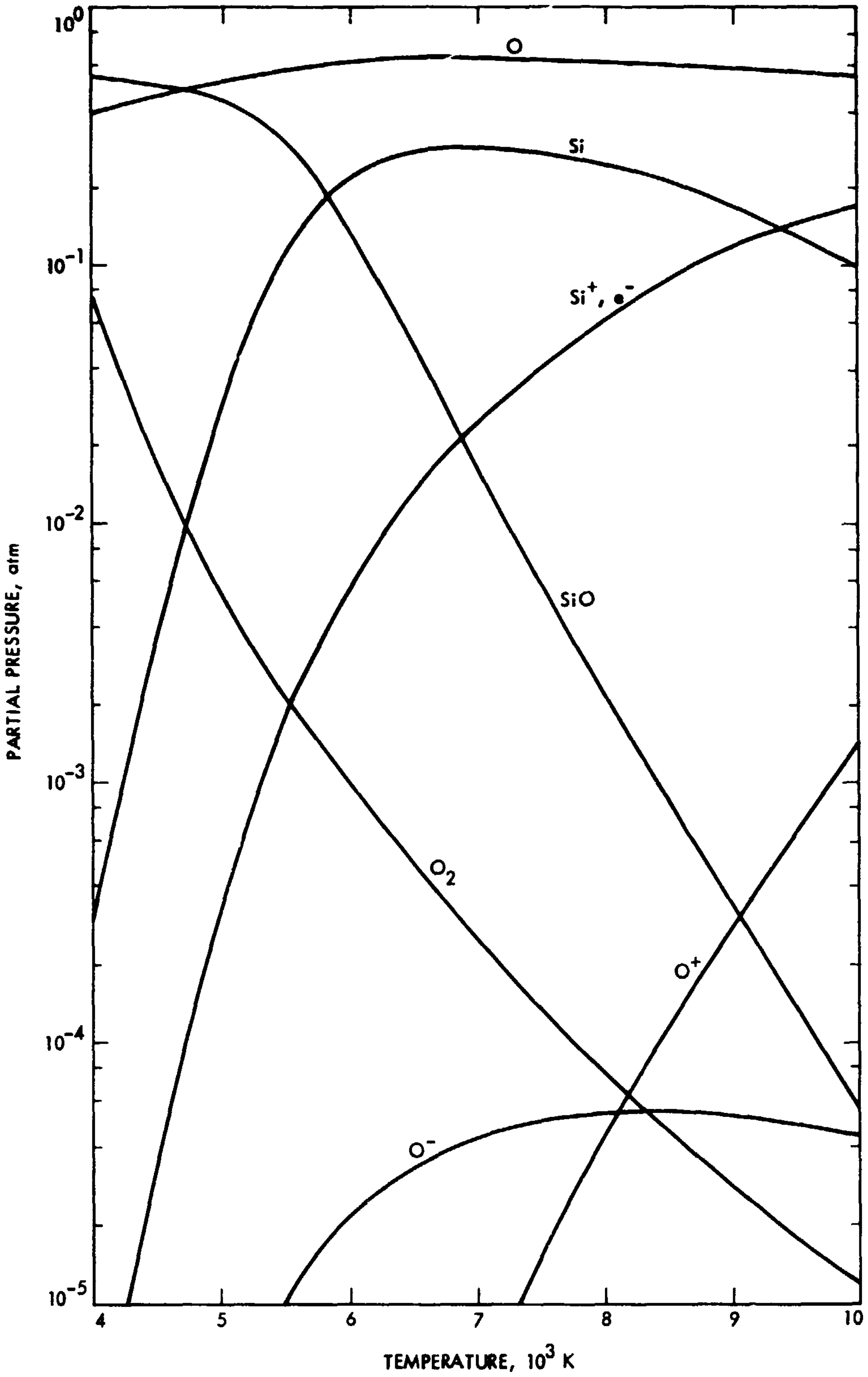

Figure 4-8. Equilibrium Species Distribution of SiO_2 vs Temperature

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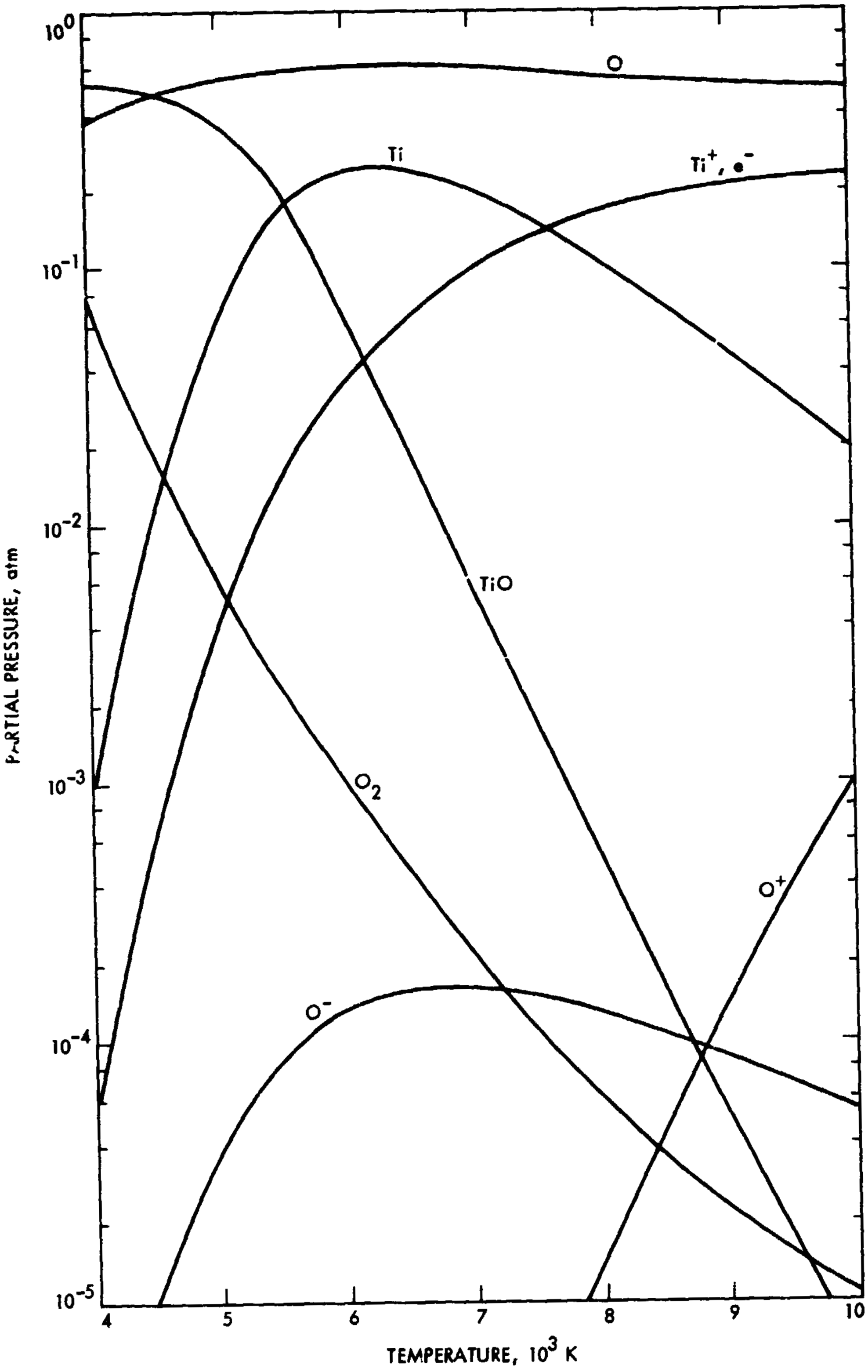

Figure 4-9. Equilibrium Species Distribution of TiO_2 vs Temperature

at a first-order definition of product yields and to establish a basis for the assessment of the all-important energy requirements, an effective ion/atom ratio of 0.65 to 8000 K and 0.90 at 10,000 K was tentatively adopted. Since the raw material consists of approximately 57% metals and 43% oxygen, a first approximation of the product yields is as follows:

Temp. K	Yield Factor	Product Yield/Throughput	
		Metals	Oxygen
8,000	0.65	0.37	0.28
10,000	0.90	0.51	0.39

The product yield values are related to the raw material input and may be considered as ton/ton.

4.4.3 Energy Requirements

For the purpose of a first-order numerical assessment, the energy requirements were broken down into three steps or consumption phases: (1) vaporization and atomization at 298 K, (2) temperature rise to equilibrium conditions at 10,000 K, and (3) electrostatic separation of the charged metal ions from the neutral oxygen. Energy requirements for Steps (1) and (2) were derived from thermodynamic analysis, based on the following transformation:

Step 1

$$M_xO_y(s)(298) \rightarrow xM(g)(298) + yO(g)(298) \tag{4-33}$$

Step 2

$$\begin{aligned} &xM(g)(298) + yO(g)(298) \rightarrow \\ &\rightarrow (x-a)M(g)(10{,}000) + aM^+(g)(10{,}000) + \\ &\quad (y-b)O(g)(10{,}000) + bO^+(g)(10{,}000) + \\ &\quad (a+b)e^-(g)(10{,}000) \end{aligned} \tag{4-34}$$

To quantify the first energy need, the values of model compounds of simple oxides were averaged to obtain an estimate for the complex metal silicates of the lunar soil. The thermodynamic values for some simple oxides are listed in Table 4-6 with the average about 6000 kWhr/ton. Heating of the atoms to their equilibrium levels of ionization at 10,000 K uses an additional 5500 kWhr/ton. Therefore, to form a 10,000 K plasma composed solely of vaporized lunar silicates would thermodynamically require 11,500 kWhr/ton of metal and oxygen produced based on an average yield of 90% (see Table 4-5).

For electrostatic separation (Step 3), which may be considered as a form of "vapor-phase electrolysis," additional energy is required to maintain the voltage between the charged plates due to the current drawn by the depositing electrons and metal cations. This deposition process is similar to the electroplating of metal with the exception that the fluid medium is gaseous rather than liquid. Therefore, as a first estimation of the collection plate voltage difference, a value in the range of those used in liquid electroplating was chosen. For convenience, the voltage selected was 1 volt. (Theoretical studies involving the mobility of ions in an electric field are underway to determine if 1 volt is a sufficient collection voltage.) This leads to an energy figure of 450 kWhr/ton of metal and oxygen produced.

The total energy required for the process, from raw material to product, is therefore:

Step 1 6,000/0.9 = 6,700 kWh/ton
2 5,500/0.9 = 6,100
3 450/0.9 = 500

Total 13,300 kWh/ton

Presently, oxygen is favored as the first near-term product. If only the oxygen is used, the simultaneously recovered metal mixture may be put aside for later refinement. Depending on the desired product and introducing a loss factor of 10% the energy requirements per ton are as follows:

Oxygen	34,500 kWh/ton
Metals only	26,000 kWh/ton
Metals and oxygen	14,800 kWh/ton

Studies are presently underway to translate these values into electrical energy requirements and to assess the feasibility of the partial use of direct (concentrated) solar energy.

Table 4-6. Energy Required to Atomize Some Oxides at 298 K and 1 atm.

Compound	Energy, kWhr/Ton
Al_2O_3	7500
SiO_2	7200
Fe_2O_3	3700
Fe_3O_4	3600
Al_2SiO_3	7500
TiO_2	5900
$FeSiO_3$, Fe_2SiO_4	4800
$FeTiO_3$	4700
$MgSiO_3$, Mg_2SiO_4	6800
$CaSiO_3$, Ca_2SiO_4	6000
Average	6000

4.4.4 Experimental Techniques

A literature survey of the state-of-the-art of plasma techniques for the decomposition of oxides was completed. Emphasis was placed on the processing of those metal oxides present in the lunar soil. Most articles accomplished the decomposition of oxides by the addition of reducing agents to chemically bind the oxygen, a method of limited application to extraterrestrial processing. However, Rains et al (Ref. 4-7) and Borgianni et al (Ref. 4-8), achieved yields of up to 33% free metal by the rapid cooling of the plasma effluent. They found that the yield was dependent upon the amount of heat transferred to the oxide particles. Therefore, higher power densities, longer residence times, and smaller particles would contribute to greater yields.

No literature is available on the magnetic or electrostatic separation of metals and oxygen from a plasma. Mitin and Pryadkin (Ref. 4-9), however, have shown the manipulation of an argon plasma with a magnetic field. Therefore, due to the limited body of reference material and experimental data, a comprehensive experimental program is planned to start in FY'83.

4.4.5 Experimental Program

The goals of the experimental plan are: 1) to evaluate the effectiveness of the selective ionization and the vapor separation techniques, 2) to isolate the parameters determining yield, and 3) to judge the applicability of the "vapor phase reduction" techniques to extraterrestrial materials processing. The experimental plan is designed as a stepwise build-up of capabilities from a pure gas plasma to the exclusive use of metal oxides (without a carrier gas). These steps are as follows:

Material	Objectives
1. Argon (only)	Techniques to start ionization. Gas flow characteristics for plasma containment, sustainment.
2. Ar + metal particles	Plasma diagnostics/cold plate recovery of metals.
3. Ar + oxide particles	Electrostatic recovery of metals. Degree of gas (Ar, O) ionization. Measurement of oxygen and metal yield.
4. Oxide particles (only)	Plasma initiation (e.g., sodium). Pressure/temperature optimization. Refinement of metal recovery systems. Recovery of oxygen.
5. Process demonstration	Implementation of results of (4).

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SECTION 5

INVESTIGATION INTO METALS SEPARATION

E. du Fresne

5.1 INTRODUCTION

Anticipating that one or another of the processes now under test for producing metals from lunar materials will ultimately be developed to a practical level, we have sought methods to render those metals useful. This implies achieving specification purities relevant to the uses intended. Both of the proposed processes, Plasma Dissociation and Magma Electrolysis, are likely to produce metal mixtures at some stage of operation, mixtures having undesirable properties unless refined to higher purity.

(In some modes of operation, the plasma process is likely to produce mixtures. Potentially, there are methods of overcoming this, but these methods have not yet been investigated, and in any event are related to the distillation process to be discussed below. For simplicity, and without prejudice to the plasma system, the discussion to follow will refer to the electrolytic process, but the fundamentals will be understood to apply to any process generating mixtures, be it the plasma process or some other.)

5.1.1 Axiomatics

To understand why mixes are likely to be produced, it is necessary to grasp why processes that proved successful in metal winning on Earth have been dismissed out of hand, leaving only Plasma Dissociation and Magma Electrolysis. The reasoning goes thus:

Axiom I: Neither the Moon nor the asteroids, as far as we know, contain "ores" in the terrestrial sense: that is, concentrations of metal sulfides or other compounds that have been deposited in limited areas and show concentration factors over the country rock of 3 to 20 orders of magnitude. Such ores are the result of aqueous extraction (usually hydrothermal) and subsequent precipitation. Terrestrial experience has shown that anhydrous differentiation of magmas simply does not produce such concentration factors. On the other hand, the operation of aqueous solutions on the Moon and the parent bodies of the meteorites (except for the carbonaceous chondrites) has been minimal. Do not, therefore, expect "ores" on the Moon.

Axiom II: Any metallurgical process conducted on the Moon must be done with the absolute minimum of imported materials. In particular, major construction materials and all process chemicals must be of lunar origin (see Section 8.8.3).

Corollary: Water, fluorides, chlorides, etc., which have proved so useful in terrestrial metal winning, have no part in lunar metallurgy. The same is true of such conveniences as consumable carbon anodes. (On the other hand, the outgassing of the Moon has not proceeded to the complete loss of sulfur, and it is theoretically possible to use lunar sulfur in some metal separations.)

The fluxes used in terrestrial metallurgy have selective chemical or electrochemical effects, sometimes by changing reduction free energies and voltages through the formation of metal-flux complexes, and sometimes by reducing only viscosity, which nonetheless has the important effect of improving selectivity. In the lunar milieu, the only flux cheap enough for us to afford in the near future is heat.[1] This is a highly effective flux for magmas low in network-forming elements, such as (notably) silicon and aluminum. When these elements are abundant in the magma, however, selectivity drops off in rough proportion to the increase in viscosity (which can increase by a factor of a million or more for magmas high in both aluminum and silicon, compared with those poor in these elements), so that the cathode product can be expected to consist of a crude mixture of the metals in roughly the proportions existing in the magma itself.[2]

A further constraint is that the most important initial prod[illegible] be made on the Moon is not a metal, but oxygen. In spite of our interes[illegible] the metals, they are at first only a byproduct of the production of oxyg[illegible] life-support, propulsion, and export. Eventually, of course, the metals will assume their own importance: initially in the same context as oxygen, i.e., for containment of life-support systems, and as vessels for propulsive oxygen (including that exported), and, in due course, as the structural elements of Moon-fabricated spacecraft.

5.1.2 Practical Consequences for Electrolysis in the Lunar Context

Once this constraint is understood, one looks for the most efficient electrolytic process for producing oxygen. In the lunar context, one searches for materials that produce low-viscosity melts at relatively low temperatures. These are iron-rich silicates and titanates, which also have the advantage of simple magnetic separation from the lunar regolith. Oxygen production from more troublesome lunar (mock) materials has already been demonstrated, as has been the production of small amounts of iron alloy. (As cell configuration problems are solved, permitting the collection of reduced iron in quantity -- a task now in progress -- we expect to demonstrate iron/oxygen production under even more favorable conditions than those now in use.) The anticipated byproduct of oxygen manufacture from iron-rich melts is iron containing varying amounts of silicon, manganese, chromium, and nickel (and perhaps sodium and potassium as well). See Figure 5-1, "Reduction Voltages."

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[1] Extraction of lunar alkalies for use as fluxes is also possible, but obviously belongs to a second-generation effort.

[2] Paradoxically, an excess of heat leads to the same result, as, in the gas or plasma phase, all ionic species of the same sign or charge, which in this case means the metals, are capable of mixing rapidly. The best-known remedy is to lower the density of the gas or plasma to accommodate the selection process. The result of low density, unfortunately, is low yield or low efficiency.

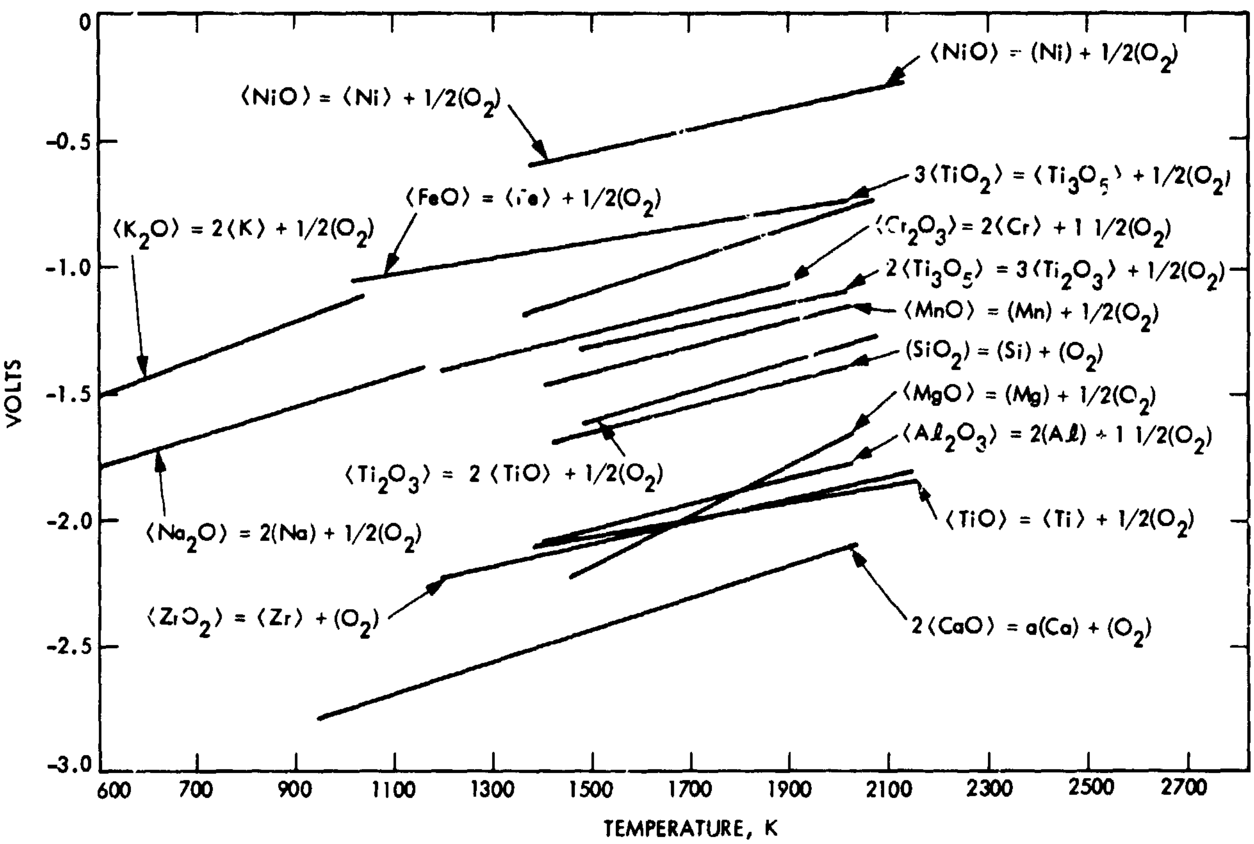


Figure 5-1. Reduction Voltages

Since oxygen is the primary goal, it is desirable to keep oxygen production efficient. This implies low temperatures, low voltages, and hence low viscosities, while (in the interest of production volume and heat efficiency) keeping current high. Under these conditions, electrolysis of an iron-rich silicate/titanate melt will yield, as voltage progressively increases, nickel in small quantity (due to low abundance), iron in large quantity, then chromium and manganese (both of low abundance), the alkalies, and finally silicon in quantity. Titanium, aluminum, and magnesium are unlikely to be reduced while silicon is still abundant in the melt. Given reasonable convection or turbulence in the oxide phase, iron, silicon, etc., will be delivered to the cathode in sufficient quantity-per-unit-time to permit the electrode reaction to proceed without further voltage increase at the cathode surface.

5.1.2.1 Uses for Metal Byproducts of Oxygen Manufacture. The issue is academic as long as the cathode deposit is a byproduct meant only for disposal. But when the iron is required for structural purposes, the silicon content becomes of great interest. Such alloying elements as nickel, chromium, and manganese toughen iron, but silicon, beyond a critical limit, embrittles it. Four to six percent of silicon can be tolerated, but much more is damaging. Terrestrial metallurgy knows a 50-50 silicon/iron product as spiegeleisen -- "mirror-iron", because of its proclivity to shatter, producing mirror-like fracture surfaces. We have used, and will use here, the term "spiegeleisen" to describe any iron alloy embrittled with silicon.

Any spiegeleisen by-product of oxygen manufacture can be used for further work. Here are the ways:

(1) Silicon can be removed, and iron (plus Ni, Cr, and Mn) added, by pouring the metal melt through an iron-rich oxide melt. This provides the needed agitation and contact surface for additional iron to be reduced at the expense of the oxidation of elemental silicon. The result is a structural-grade metal: impure, perhaps, but quite serviceable for many applications. (Any alkalies will be lost by evaporation, or by reoxidation in contact with the oxide melt.)

(2) Spiegeleisen can be used as a consumable anode in the reduction of further metals.

In the latter case, we are considering a minority process, wholly subordinate to the production of oxygen, which must be the major product of lunar chemistry. A portion of the by-product of oxygen manufacture, the iron or spiegeleisen, can be dedicated to use as consumable anodes, rather than converted to structural iron or simply dumped. It is a fact of nature that use of a consumable anode will not yield oxygen, and hence diverts electrical power from the main purpose of the lunar chemical industry. Yet, at a certain stage of development, this diversion of power will become necessary. Iron may be useful for structural products employed on the Moon, but is not efficient for structures that must be lifted off the Moon, whether vessels for oxygen transshipped elsewhere, or for other spacecraft components. This implies a need for one or more elements in the list of lunar-abundant light metals: Al, Mg, Ti, and possibly Ca. In addition, if the Moon is to be truly self-sufficient in energy, imported power sources must give way to photovoltaic units made from lunar silicon, or other power systems based on lunar materials.

5.1.2.2 Common Reduction of the Light Metals and Silicon. Clearly, the refinement of iron by the oxidation of silicon, as sketched above, does not afford a route to pure silicon. This element must therefore be won in a different campaign from that which yields iron and the siderophiles (Ni, Cr, Mn); in view of the "voltage clusters" shown in Figure 5-1, and the higher viscosities of iron-poor oxide melts, the first candidate for a process to win silicon must be that which also wins the light metals: Al, Mg, Ti, etc.

This exceptional electrolysis uses a spiegeleisen consumable anode because no known material is suitable as a permanent anode: between the heat and the oxidation potential, nothing that is electrically conductive enough to yield a product at a reasonable rate will stand up. At these high temperatures, the spiegeleisen anode is molten. This leads to an odd-looking geometry, with a central cathode pool, in which light metals accumulate upon reduction, surrounded by pools of molten anode spiegeleisen in the process of dissolution. (See Figure 5-2.) (Doubtless refinements of this scheme are possible: a dripping anode, for example, in which selective oxidation takes place. For the moment, we will go with the cruder design, because it is easier to visualize.)

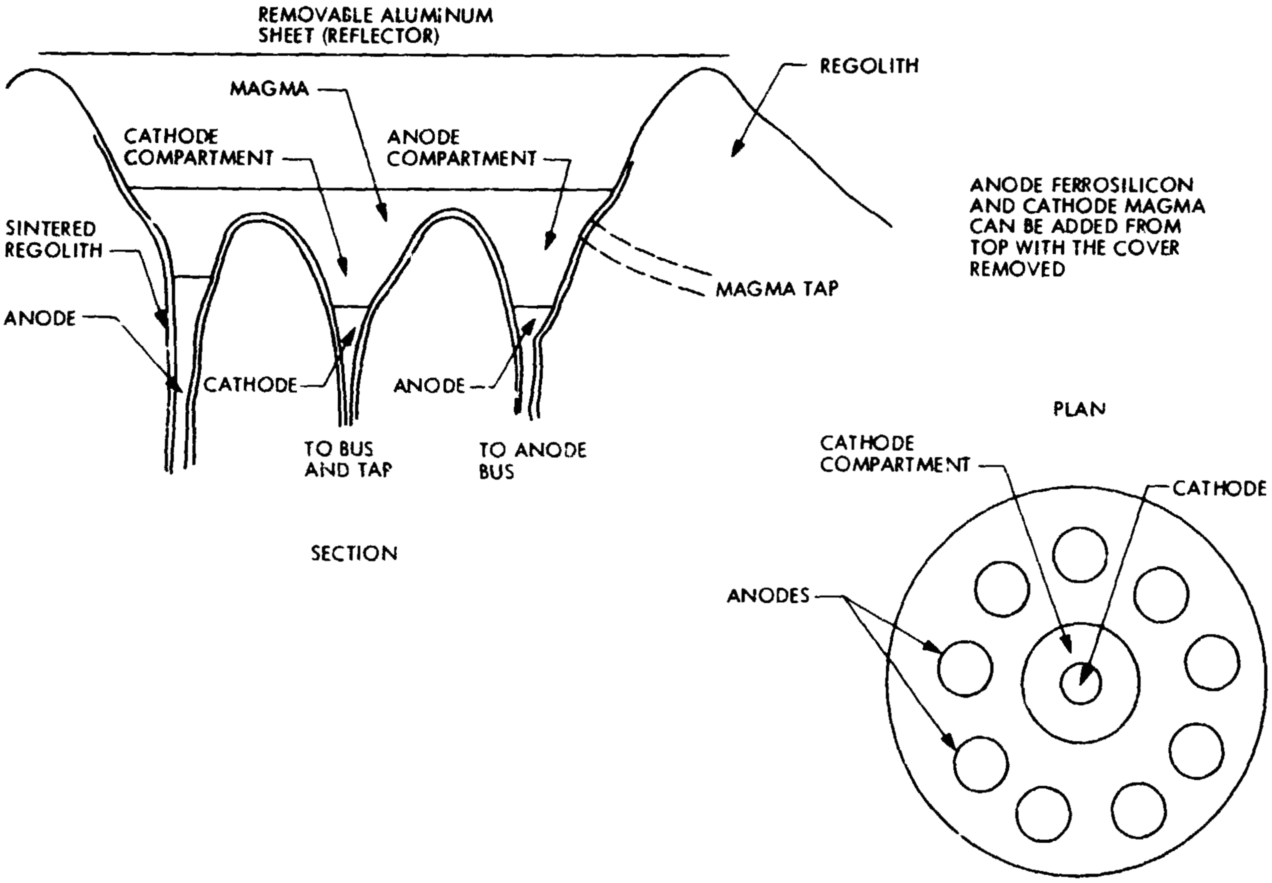
The net effect of high voltage, high viscosity, etc., is to reduce all available metals into a common metallic liquid. The problem is then to separate them, at least to a degree that will render them ultimately useful. In many cases, as with iron where the use is structural, this does not require high purity, with silicon as the notable exception.

5.2 VACUUM DISTILLATION

Vacuum distillation is an easy solution to some of the separations. A glance at Figure 5-3 ("Vapor Pressures") shows that the alkali and alkaline-earth metals can be easily separated from the cathode mixture, and from each other, by simple one-stage evaporation processes. Moreover, the final separations can be conducted in iron vessels of lunar manufacture. The other separations are less obvious.

With the removal of magnesium and calcium, the major constituents of the remaining metal are silicon and aluminum, since these are highly abundant in lunar rock and regolith. (Iron and titanium will occur in variable quantities, which are to some degree controllable, as these elements go together in magnetic minerals such as ilmenite. We can thus increase the amount of iron and titanium in the magma by magnetic beneficiation of the charge, or reduce these elements by taking the nonmagnetic -- or least magnetic -- portion of the feedstock. Later on we will need to come to grips with titanium production and consider the titanium/iron separation. For the moment let's consider a charge of relatively nonmagnetic material, depleted in iron and titanium. The separation of aluminum from silicon and most of the remaining elements (manganese being the only known exception) proves to be easy when done by distillation.

In effect, aluminum is so much more volatile than silicon at low temperature (just above the melting point) and low pressure (a thousandth atm), that, if Raoult's Law is followed, a one-stage vacuum evaporation should yield aluminum of high purity.



ANODE FERROSILICON AND CATHODE MAGMA CAN BE ADDED FROM TOP WITH THE COVER REMOVED

Figure 5-2. Molten Anode Cell

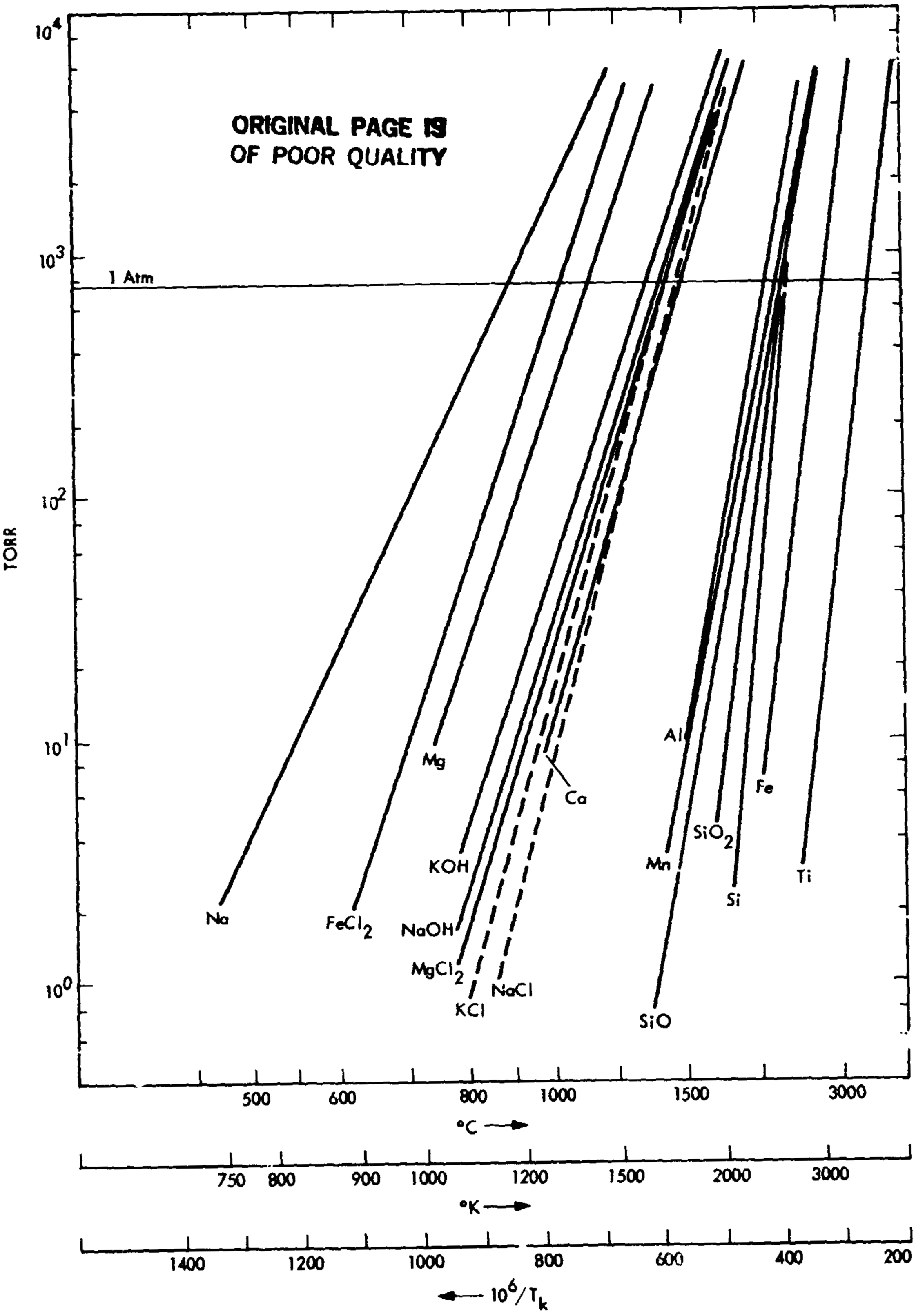


Figure 5-3. Vapor Pressures

5.2.1 Aluminum-Silicon Separation: An Example

It would thus be possible to pour an aluminum/silicon melt down through a shot tower, the metal solidifying into pellets on the way because of the heat loss by evaporation and radiation, and subsequently recover refined aluminum by stripping the solid from the walls of the tower. (This "prill-tower evaporator" is discussed in greater detail later.) Moreover:

(1) Literature search has uncovered data indicating the deviations from Raoult's Law that may be expected. The excess free energies of solution for aluminum and silicon are negative, indicating strong association in the melt. Paradoxically, this condition is known to result in greater separation upon distillation. Picture an atom of silicon in an aluminum-rich melt. The atom is tied to one or more aluminum atoms, effectively raising its molecular weight. Thus silicon does not escape into the vapor phase as readily as predicted by Raoult's Law. The excess, or unassociated, aluminum however, is just as volatile as that in an "ideal" (Raoult's Law) solution. The result is that the vapor phase is enriched in aluminum over the Raoult's Law prediction, throughout the whole range of liquid composition.

(2) We have developed computer programs to design metal distillation plants under a broad range of pressure and temperature conditions. Even at one atmosphere, and again assuming Raoult's Law for the purposes of simplicity and conservatism, only 3 to 5 theoretical plates (in the standard chemical engineering sense) are required to convert a 50-50 aluminum/silicon melt to aluminum rated "highly pure" by present-day commercial standards. The exact number of plates needed is a function of the "reflux ratio" assumed. The higher this ratio, the fewer plates needed, but energy consumption goes up with the reflux ratio. (Ref. 5-1).

5.2.2 Review of the Preconditions for Vacuum Distillation in the Lunar Context: Easy Separations

Let's review developments up to this stage.

(1) The main product of the lunar chemical industry is to be oxygen.

(2) To assure the efficiency of oxygen production, it is best to work with a low-viscosity magma containing an easily-reduced cation.

(3) Given the composition of lunar rocks and regolith, this will be a magma rich in iron, with a low content of silicate and aluminate, but containing some titanate. The feedstock for this magma is easily selected by conventional magnetic beneficiation of regolith.

(4) The by-product of the electrolysis of this magma is an impure form of iron. If the iron is not immediately fit for structural purposes ("spiegeleisen") it can be rendered so by intimate contact with more iron-rich magma.

(Note that any light-metal admixture in the crude iron will be removed by one process or another: silicon, aluminum, and the alkaline earth elements by oxidation, the alkali metals either by oxidation or evaporation from the molten iron.)

Matters are considerably different, however, when we consider the process for obtaining light metals, using crude iron as a consumable anode:

(5) Oxygen is not produced in this process, only raw metal.

(6) Magmas rich in aluminum and poor in iron are of high viscosity (assuming the suite of minerals observed in lunar regolith and rock, and calculating abundances of elements in the magma on that basis).

(7) Owing to the high viscosity, electrochemical reduction lacks selectivity, and a mixture of metals constitutes the cathode product, if our present model is accurate.

(8) Assuming that a regolith fraction of low magnetic susceptibility is taken as feedstock for this process, the principal constituents of the crude metal will be aluminum, silicon, the alkaline earths (magnesium and calcium), with minor amounts of titanium, the ferrous metals, and the alkali metals.

(9) Most, if not all, of this crude metal will remain molten at temperatures substantially below the melting point of iron.

It will be noted from Figure 5-3 that the vapor-pressure curves of the alkali metals are quite low: much of the sodium and potassium content of the crude metal could be lost by merely pouring it from one vessel to another!

Indeed, calcium and magnesium can also be separated from the crude metal by vaporization, and that quite simply. It is easy to show that the principal obstacle to the loss of these metals is the slowness of diffusion in the melt, not the passage of the liquid-to-vapor interface: it follows that agitation is the needed factor in the process of separation, and will be successful even at quite low temperatures, as long as the liquid phase remains.

Since the distillation of metals such as magnesium is a well-established metallurgical practice today, here at the Earth's surface, there is no need to dwell on it. We therefore confine ourselves to a few remarks, before getting on to more challenging problems:

(1) Throughout a rather long (practical) temperature range, the initial "distillate" from the crude metal will consist of the alkaline earth metals (Ca and Mg), the alkali metals (Na and K) in essentially the proportions present in the feedstock, and a variable content of aluminum.

(2) The content of aluminum is principally dependent on the temperature of the initial distillation.

(3) The initial distillation can be conducted in equipment made of lunar iron.

(4) The subsequent separations (of Na, K, Ca, and Mg from Al and from each other) can thus also be conducted in a simple apparatus made from lunar iron, and, with only one possible exception (Na from K), will involve one "theoretical plate" (or less) to attain practical degrees of purity.

5.2.3 Aluminum Purification

These are the "easy" distillation problems. The first distillation process that can be described as presenting any difficulties is the purification of aluminum, which we have already mentioned at the beginning of this summary. The "difficulties" depend upon the end use proposed.

(1) For structural purposes, small amounts of other metals such as Fe, Mn, and Si are tolerable, or even desirable.

(2) For electrical conductivity, as in wiring, purity above 99% or 99.5%, or as high as possible, consistent with cost, is preferred.

(3) For semiconductor use, as in the preparation of AlP, a III-V semiconductor, only parts-per-billion amounts of impurity are admissible.

Case 3 can be dismissed immediately. Only the substitution of AlP for silicon or germanium can justify production of AlP on the Moon in the foreseeable future. Now the primary need for semiconductor materials in bulk quantity on the Moon is likely to be for photovoltaic power sources. The adaptation of AlP to this purpose is problematical, in view of the large difference in band gap between AlP and silicon.[3]

Given the difficulty of ultrapurifying metals, it is more likely that high-purity silicon will be attainable rather than high-purity aluminum, which would then be adapted to an essentially foreign purpose. This, of course, is a prediction from the Cloudy Crystal Ball, which may be set aside by some future innovation.

In case 1, it is possible to continue processing in iron equipment, although molten aluminum is capable of dissolving small amounts of iron.

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[3] The room-temperature energy gap for silicon is 1.107 volts, while that for AlP is about 2.5, or more than double.

Minor amounts of iron are objectionable by terrestrial standards because of an adverse effect on corrosion resistance, which is scarcely a problem in a vacuum environment. For most structural purposes, the iron is a strengthening element. (So, for that matter, are most of the minor impurities expected: silicon, manganese, magnesium, etc.) It follows that two rather simple operations, which can be conducted in vessels made of lunar iron, will be sufficient to produce aluminum to structural specifications. These operations are:

(1) Liquation.

(2) Distillation.

5.2.3.1 Liquation as an Auxiliary Process. Liquation is the process of partial freezing of the metal melt with pouring-off of the remaining liquid. In this process, much of the hardening elements is lost, since these elements harden by forming high-meltingpoint intermetallic compounds with aluminum. Crystals of these compounds sink in the melt or adhere to the crucible, leaving a melt of nearly eutectic composition. This sets a natural upper limit to most of the hardening elements in aluminum, as long as liquation is practiced. This applies to the iron impurity also, even if iron vessels are used to handle the aluminum melt.

By the same token, there is a limit to what liquation can do (unless one is prepared to classify various complicated procedures of fractional crystallization and zone-refinement as "liquation", which is not a good use of the word). Liquation, as defined above, will not move a composition from the aluminum-poor to the aluminum-rich side of the eutectic, unless there is a stronger affinity between the impurities than between any of the impurities and aluminum.

Note that this introduces the concept of adding a foreign substance to scavenge impurities. A likely candidate is sulfur, a relatively abundant element on the moon, one that forms stable compounds with manganese and iron and volatile compounds with silicon; the aluminum compound is relatively unstable. Since this possibility is not yet fully investigated, it is unfair to include it in the principal discussion. We therefore confine ourselves to the following remarks:

(1) Metallic impurities that are not renewed from some other source -- manganese as an example -- can be removed to a level depending on:

  (a) The equilibria, as determined by free energies of formation.

  (b) The number of purification stages and the loss of aluminum that one is prepared to accept.

  (c) The natural abundance of the impurity in the feedstock.

(2) Metallic impurities that are renewed, as by dissolution of iron from iron vessels, cannot be reduced beyond a minimum that is set by the secular equilibria between the competing processes.

Hence manganese, which appears as an impurity in lunar iron. can be reduced to a very low level in <u>one</u> stage of purification of aluminum conducted in equipment made of lunar iron, upon addition of sulfur, but cannot be further reduced by sulfur-aided liquation in such equipment. The equilibrium limit will be reached in one stage, after which manganese will be supplied from the equipment as rapidly as it is removed by sulfur.

Without becoming quantitative, it is still possible to predict that the degree of removal of manganese will be inadequate to produce semiconductor-grade aluminum, but still sufficient to make good conductor-grade aluminum for electrical buses and wiring.

5.2.3.2 Quantitative Aspects. After this digression, we can return to the main subject, namely the removal of elements other than manganese from aluminum, and the general utility of vacuum distillation in lunar metallurgy.

The aluminum/silicon separation, already alluded to, gives a good example of the power of vacuum distillation as applied to lunar metals. In what follows, we will use Raoult's Law to be conservative, and ignore separations such as aluminum/titanium.

For aluminum/silicon, then, vapor pressures over the range of 1 to 760 torr are available from the Handbook of Chemistry and Physics. These data correlate well on an Arrhenius plot, (See Figure 5-4) i.e., the vapor pressure can be expressed as

$$p = \exp(m/T + b)$$

For Al, $m = -31205.89396$
$b = 20.03844917$

and for Si, $m = -60161.57516$
$b = 30.13548273$

If Raoult's Law can be assumed for the liquid phase, then the total pressure above a liquid mixture of aluminum and silicon will be equal to the sum of the vapor pressures of each element times its respective mol fraction, or

$$Xp + (1 - X)p' = P$$

This equation can be solved for $1/T$, and this in turn can be used to calculate p, $Y = p/P$, and $Y' = p'/P$, so that a phase diagram can be constructed, while a McCabe Thiele distillation plot can be laid out from a graph of Y vs X.

The traditional method of calculating Y is by a graphical construction, as the algebra is hairy. A Newton-Raphson solution is more convenient for our purposes, and has been used here.

For 760 torr, we obtain the following table:

Table 1. Newton-Raphson-Generated Data for Al/Si at 760 torr (1 atm)

X(Al)	t(°C)	Y(Al)
0	2286.68	0
	(compare Ref. 5-2)	
0.05	2273.58	0.15820
0.10	2260.13	0.29647
0.15	2246.45	0.41594
0.20	2232.66	0.51802
0.25	2218.86	0.60438
0.30	2205.18	0.67682
0.35	2191.72	0.73716
0.40	2178.57	0.78716
0.45	2165.79	0.82843
0.50	2153.44	0.86243
0.55	2141.54	0.89043
0.60	2130.11	0.91348
0.65	2119.16	0.93249
0.70	2108.68	0.94823
0.75	2098.66	0.96123
0.80	2089.07	0.97201
0.85	2079.91	0.98098
0.90	2071.16	0.98848
0.95	2062.78	0.99474
0.975	2058.72	0.99751
0.99	2056.33	0.99902
0.999	2054.91	0.9999038
0.9999	2054.77	0.99999039
1.0	2054.76	1
	(compare H'bk 2056)	

When this is plotted, there is not much difference from a chemical engineer's classical distillation chart. Several theoretical plates will be required to obtain reasonable purity.

Since we are intending to conduct the process on the surface of the Moon, with free vacuum available, it is of interest to find out what the Y-X chart would look like at, say, 10^3 atm = 0.76 torr. (This brings us close to the melting-point of pure Si = 1420.)

Table II. Newton-Raphson-Generated Data for Al/Si at 0.76 torr (1×10^{-3} atm)

X_A	t	Y_A
0	1705.20	0
0.05	1524.16	0.95560
0.10	1458.38	0.98821
0.15	1420.89	0.99484
0.20	1395.06	0.997198
0.25	1375.49	0.998288
0.30	1359.81	0.998897
0.35	1346.77	0.999228
0.40	1335.64	0.9994459
0.45	1325.94	0.9995952
0.50	1317.36	0.9996991
0.55	1309.68	0.9997826
0.60	1302.73	0.9998307
0.65	1296.39	0.9998729
0.70	1285.18	0.9999311
0.80	1280.17	0.99995398
0.85	1275.50	0.99998865
0.90	1271.12	0.99998960
0.95	1267.00	0.99999133
0.975	1265.03	0.99999585
0.99	1263.87	0.99999839
0.999	1263.19	1.0 (to accuracy of method)
0.9999	1263.12	1.0 (to accuracy of method)
1.0	1263.11	1.0 (to accuracy of method)

In the Y column, the sixth, seventh, and eighth decimals are probably not significant, in view of the precision of the original data.

Notice that, if Table II bears any resemblance to reality, we can take a mixture of 85 mol% Si, 15 mol% Al (this is practically weight percent, in this case) and obtain commercially pure Al in one pass. Again, a structural or casting alloy, 95% Al, will yield a purity grade seldom achieved in the laboratory. It is much better than conductivity grade.

5.2.3.3 Experimental Verification. The predictions made by Table II are so radical that they readily lend themselves to experimental test. Silicon-rich slugs can be brought to just above the melting point with a tungsten heater in a standard bell jar. The vapor emanating from these just-melted slugs will condense on any cold surface, and can be analyzed.

In a practical scheme, one would make a limited selection of high-silicon, low-aluminum slugs and bring them to appropriate temperatures just above the liquidus line.[4] A reasonable correspondence with theory between the analysis of the slug, the temperature, and the analysis of the deposit would justify the conditional acceptance of Raoult's Law, and, more importantly, verify engineering feasibility.

5.2.3.4 Engineering Embodiment. Molten aluminum welds to glass, reduces silica and many other oxides, and dissolves significant amounts of iron. Building a conventional rectification column, for aluminum, of Moon-won silicates or ferrous alloys is hence likely to be absurd. Importation of special materials for such a column is uneconomic. A one-pass system changes all that.

I have elsewhere disclosed the concept of the prill-tower evaporator.[5] It applies here as follows: Let molten alloy be discharged downward as droplets of precise size; let these droplets fall through a long tube under an effective vacuum; during the flight, the droplets cool by evaporation and radiation; the evaporated material moves to the wall of the tube and condenses.

In the present case, the condensate can be assumed to be solid, and of high reflectivity, minimizing the radiative loss of heat. Hence, for a reasonable time of flight and size of droplet, evaporation can be brought to a practical maximum. The tube need not be integral, and might be made of, e.g., iron slats, from which the product is subsequently peeled as strips of pure metal. The height of the prill tower is minimized by lunar gravity. Prill towers 15 to 30 meters high have been used for "shotting" and other purposes on Earth. Given that

$$d = 4.9\ t^2$$

$$t = (d/4.9)^{\underline{1/2}}$$

t = 1.77 seconds for a 15-meter terrestrial tower, and only 2.5 seconds for a 30-meter tower. For a 15-meter tower on the Moon, the time of flight is 4.33 seconds. Proper choice of droplet size (in turn determined by orifice diameter and head) would assure efficient evaporation so that the metal need not be recirculated to excess. Obviously, there will be trade-offs, depending on such variables as feed composition, desired output composition, and sensible/latent heat ratio. There seems no doubt, however, that the desired purity of aluminum can be produced from quite crude input at a reasonable energy cost and with a minimum of imported equipment.

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[4] These need not be at 10^{-3} atm. Calculations can be done for any reasonable T and P.

[5] The prill-tower evaporator is now the subject of a JPL-Caltech-NASA patent application.

5.3 INTERIM REVIEW

To sum up so far; the following lunar-abundant metals can be purified for conventional structural or related purposes:

(1) Iron, by reduction and oxidative refining.[6]

(2) Magnesium and aluminum by one-pass vacuum distillation.[7]

Two lunar-abundant metals remain, one of structural interest (Ti) and the other presenting a challenge in ultrapurification (Si).

5.4 DIFFICULT PURIFICATIONS

5.4.1 Titanium

The purification of titanium by vacuum distillation is simple in principle, but likely to be extremely difficult in practice. While there is a wide spread in vapor-pressure curves, the task of separation must be undertaken at an exceedingly high temperature: it is no less than that of boiling the iron out of molten titanium. This can be seen from the vapor-pressure curves in Figure 5-3.[8]

This is, of necessity (if high-purity titanium is to be obtained), a multipass process, whether a prill-tower evaporator is used, or a more conventional distillation unit. But what is this "conventional unit" to be made of? The boiling point of iron (at 760 torr) is 2750°C. Of the various refractories that might be extracted from the lunar surface, the highest-melting is magnesia, at 2800°C, but titanium reduces MgO in this temperature range (see Figure 5-1). Calcia is the next best, melting at about 2614°C.

It cannot be assumed that calcia has substantial strength close to its melting point. Further, at much above 2700°C, titanium reduces CaO. Worse yet, we can expect that the high vapor pressure of calcium metal, that is, its high fugacity or "escaping tendency," will drive the reduction of CaO at temperatures substantially below 2700°C.

The vapor pressure of (pure) iron is 40 torr at 2223°C and 100 torr at 2360°C. The attack of titanium on CaO might be tolerable in this temperature range, and if titanium could maintain an overall pressure in the range of 40 to 100 torr, we might effect a good separation by distillation. The unhappy fact is that this pressure cannot be maintained at practical temperatures.

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[6] The strengthening of lunar iron to the levels customary in steels is the subject of ongoing research.

[7] As noted, calcium can also be purified and is a potential structural light metal, but it is not conventional, being subject to corrosion in air. This is unimportant in the lunar or space context.

[8] In what follows, read "iron" as "Fe, Cr, and Ni." The vapor pressure curves of these three are close enough to be almost indistinguishable.

There is no containment material that meets the requirements of a container for titanium vapor at a reasonable pressure. The pressure of the titanium vapor emitted by the bottom stage of the container (the "reboiler") must equal or exceed[9] the overall pressure at the higher plates: that is, it must be 40 to 100 torr or higher. But the temperature at which titanium attains 100 torr pressure is nearly 3000°C! Only a few substances will take this temperature (notably tungsten) and it is not guaranteed that they will not slowly dissolve in molten titanium. In any event, they would have to be imported.

5.4.1.1 Solution by "Sparging." Straightforward distillation, enhanced by vacuum, is thus unlikely to work. In such cases, the chemical engineer turns to such techniques as "sparging" or steam distillation, in which a substance of low volatility is carried along by one of high volatility, this last making up the pressure deficits created by the first substance, and thus moving its vapor upward through the distillation tower. Logical sparging substances for titanium would be the noble gases such as He and Ar. Unfortunately, these are extremely rare on the Moon.

There are, however, lunar-abundant substances that might serve: sodium is one. At the temperatures we are talking about, 2000°C and up, sodium might well be regarded as a "permanent gas." We can therefore conceptualize a sparging (or steam distillation) unit along the following lines:

> The unit is a fairly conventional plate tower, built of calcia, and operating at roughly 2200 to 2300°C, with an internal pressure of around 100 torr or so. There is no reboiler *per se*, only a sump for purified titanium. Above this sump, sodium vapor is introduced; it strips the lowest plate of residual iron. The vapor proceeds up the tower, becoming richer in iron, and passes the feed plate. A few plates above that, the overflow liquid contains little titanium, and is substantially pure iron (i.e., there is no point in furnishing further plates). The mixed vapor then proceeds to a condenser operating at 1600°C, where essentially all iron condenses. Part of this iron returns to the top plate of the tower, the remainder leaving the system. The sodium vapor goes on to a condenser operated at 500°C or less, is liquefied, and flows by gravity to the boiler, where it is vaporized for reentry above the sump.

On the Earth, such a process unit would be regarded as sheer insanity, owing to the likelihood of such hazards as sodium fires, reaction of titanium with oxygen and nitrogen, and the attack of moisture on the calcia body. Viewed in a kinder light, as science-fiction rather than absolute madness, the scheme has the following weaknesses:

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[9] In a lower-pressure system, such as this, the hydrostatic head of molten metal on each plate must be taken into account: even under 1/6 terrestrial gravity, this is not trivial, and may amount to 100 torr or more, depending on the number of plates.

(1) How is the system to be sealed against vacuum? Will the sealing material (e.g., graphite gaskets) need to be imported?

(2) How rapidly will attrition of calcia and of any binding or sealing substances occur? What impurities will this introduce into the products? Oxygen, in particular, makes titanium virtually unworkable, and is hence a more obnoxious impurity than most metals.[10]

(3) How will the calcia be manufactured and shaped?

These questions are not unanswerable, but require separate consideration. Hence these details will be dealt with in a separate report, to be issued later.

5.4.1.2 Alternatives. For the moment, we may suggest several alternatives:

(1) Multiple staging of prill-tower evaporators. The disadvantage is high energy consumption.

(2) Processing of the metals as chlorides. Halogens are scarce on the Moon, as the result of several outgassing episodes. Nonetheless, there is a small content of these elements in lunar rock, and the halogens will appear as impurities in electrolytic oxygen, from which they can (and must) be separated before the oxygen is liquefied or otherwise used. A small stock of chlorine, the most abundant halogen, could thus be accumulated for titanium processing.

(Truthfully, one can believe that the pure titanium produced by this route, if it should prove the only practical one, is more likely to be used for the local coinage than for spacecraft. It will be more expensive, in terms of local values, than gold is on Earth. This is a consequence of the scarcity of chlorine on the Moon, and of the fact that any practical process entails losses of the process chemicals.)

(3) Sparging ("steam distillation") with silicon or aluminum. In the last analysis, this scheme has as many potential disadvantages as using sodium, plus further problems, such as the need to heat upper stages of the tower.

The case of titanium will be examined furtner during the coming year's work.

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[10] The addition of small amounts of calcium vapor to the sodium vapor stream should assist in slowing the erosion rate of calcia, and in keeping down the oxygen content of the titanium product. It is simply a matter of Le Chatelier's Rule: displacing the dissociation of calcia in the reverse direction, and hence making free oxygen less available.

5.4.2 Silicon

5.4.2.1 The Purification of Silicon. Compared to the purification of titanium, the purification of silicon appears fairly easy, but has a number of peculiar features. It will be observed from Figure 5-3 that the lines for silicon and iron diverge at higher pressure and temperature.[11] This is an unusual situation, causing complications, as will be shown below. In effect, higher pressures (and hence temperatures) are an advantage in the separation, which is directly contrary to the common observation that better separations can be obtained under vacuum.

The normal situation is that Arrhenius lines (of log P vs 1/T) tend to converge with increasing T and P. This is a consequence of Trouton's Rule and of the Clapeyron-Clausius equation. But Trouton's Rule is not an absolute law. Like other approximations to an ideal, it is frequently violated. If Trouton's Rule were strictly followed, and all unassociated liquids had the same entropy of vaporization, say the traditional 21 calories per degree-mole, then the heats of vaporization of these liquids would simply be the boiling points (in degrees K) times 21. Since the slopes of the Arrhenius lines are proportional to the heat of vaporization, these lines would get steeper as we moved to progressively higher boiling-points, creating the impression, as we looked at the chart, that all the lines "radiate" from a common region at the top of the chart, like sunbeams bursting through a cloud.

Unfortunately, it takes only a slight deviation from Trouton's Rule - some slight variance in the physics or chemistry of the element - to upset this regularity and produce a result such as that seen in Figure 5-3, with the Fe and Si lines converging downwardly rather than upwardly. (The polymerization of Si vapor is the ultimate explanation.) The paradoxical result is that this pair can be separated more easily at high pressure than under vacuum. This conclusion, easily reached from an examination of the figure, is made quantitative in Tables 5-1 through 5-4, which have been obtained from the Newton-Raphson program, assuming Raoult's Law, for overall pressures of 1, 20, 400, and 2000 torr.

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[11] This is the result of the dimerization and trimerization, etc., of silicon in the vapor phase.

Table 5-1. Silicon/Iron: Total Pressure 1 torr

XSi (mole fraction in liquid)	Temperature (^{o}C)	YSi (mole fraction in vapor)
0.05	1779.66	0.114497
0.10	1773.78	0.210511
0.15	1768.63	0.293202
0.20	1764.04	0.365851
0.25	1759.91	0.430674
0.30	1756.15	0.489235
0.35	1752.70	0.542677
0.40	1749.52	0.591861
0.45	1746.56	0.637450
0.50	1743.80	0.679967
0.55	1741.21	0.719827
0.60	1738.77	0.757371
0.65	1736.47	0.792877
0.70	1734.29	0.826577
0.75	1732.22	0.858667
0.80	1730.26	0.889312
0.85	1728.38	0.918654
0.90	1726.58	0.946814
0.925	1725.72	0.960484
0.95	1724.87	0.973898
0.965	1724.37	0.981828
0.975	1724.04	0.987067
0.9825	1723.79	0.990971
0.99	1723.54	0.994854
0.995	1723.38	0.997432
0.999	1723.25	0.999487
0.9995	1723.24	0.9997436
0.9999	1723.22	0.9999487
0.99995	1723.22	0.9999744
0.99999	1723.22	0.99999487
0.999995	1723.22	0.99999744
0.999999	1723.22	0.99999949

Table 5-2. Silicon/Iron: Total Pressure 20 torr

XSi (mole fraction in liquid)	Temperature (°C)	YSi (mole fraction in vapor)
0.05	2092.39	0.275696
0.10	2069.11	0.428211
0.15	2051.70	0.529932
0.20	2037.78	0.604620
0.25	2026.19	0.662814
0.30	2016.26	0.710033
0.35	2007.57	0.749495
0.40	1999.86	0.783223
0.45	1992.93	0.812566
0.50	1986.63	0.838461
0.55	1980.86	0.861585
0.60	1975.55	0.882438
0.65	1970.62	0.901404
0.70	1966.02	0.918777
0.75	1961.72	0.934793
0.80	1957.68	0.949638
0.85	1953.86	0.963466
0.90	1950.26	0.976402
0.925	1948.53	0.982569
0.95	1946.84	0.988552
0.965	1945.84	0.992057
0.975	1945.20	0.994360
0.9825	1944.70	0.996070
0.99	1944.22	0.997766
0.995	1943.90	0.998885
0.999	1943.65	0.999777
0.9995	1943.61	0.9998897
0.9999	1943.59	0.9999783
0.99995	1943.58	0.99998943
0.99999	1943.58	0.99999830
0.999995	1943.58	0.9999988882
0.999999	1943.58	0.9999997781

Table 5-3. Silicon/Iron: Total Pressure 400 torr

XSi (mole fraction in liquid)	Temperature (^{0}C)	YSi (mole fraction in vapor)
0.05	2483.84	0.510090
0.10	2430.03	0.660725
0.15	2395.06	0.740367
0.20	2369.19	0.791611
0.25	2348.70	0.828175
0.30	2331.75	0.856001
0.35	2317.32	0.878134
0.40	2304.78	0.896317
0.45	2293.69	0.911625
0.50	2283.75	0.924766
0.55	2274.76	0.936224
0.60	2266.56	0.946345
0.65	2259.02	0.955381
0.70	2252.05	0.963523
0.75	2245.56	0.970918
0.80	2239.50	0.977681
0.85	2233.82	0.983904
0.90	2228.47	0.989659
0.925	2225.91	0.992382
0.95	2223.42	0.995008
0.965	2221.95	0.996541
0.975	2220.99	0.997546
0.9825	2220.28	0.998291
0.99	2219.57	0.999028
0.995	2219.10	0.9995161
0.999	2218.72	0.9999034
0.9995	2218.68	0.99995170
0.9999	2218.64	0.99999034
0.99995	2218.63	0.999995172
0.99999	2218.63	0.9999990785
0.999995	2218.63	0.9999995172
0.999999	2218.63	0.9999999034

Table 5-4. Silicon/Iron: Total Pressure 2000 torr

XSi (mole fraction in liquid)	Temperature (^{o}C)	YSi (mole fraction in vapor)
0.05	2735.75	0.634013
0.10	2660.69	0.760265
0.15	2614.47	0.831316
0.20	2581.18	0.858864
0.25	2555.25	0.884904
0.30	2534.05	0.904328
0.35	2516.16	0.919548
0.40	2500.70	0.931903
0.45	2487.11	0.942206
0.50	2475.00	0.950978
0.55	2464.07	0.958575
0.60	2454.14	0.965244
0.65	2445.03	0.971168
0.70	2436.62	0.976481
0.75	2428.82	0.981286
0.80	2421.55	0.985664
0.85	2414.74	0.989678
0.90	2408.34	0.993379
0.925	2405.28	0.995125
0.95	2402.30	0.996808
0.965	2400.56	0.997789
0.975	2399.41	0.998432
0.9825	2398.56	0.998908
0.99	2397.71	0.999379
0.995	2397.15	0.9996906
0.999	2396.71	0.9999383
0.9995	2396.65	0.99996916
0.9999	2396.61	0.99999383
0.99995	2396.60	0.999996917
0.99999	2396.60	0.999999384
0.999995	2396.60	0.9999996917
0.999999	2396.59	0.9999999393

Notes to the Tables

(1) The Y values are, of course, overly precise. For ordinary purposes, 3 decimal places would have sufficed in the early sections (low Y). We need the extra digits, however, for a buffer against roundoff error in subsequent computations.

(2) The thermodynamic literature shows a negative free energy of solution for Fe/Si, pointing to a more favorable condition for separation than that indicated by the use of Raoult's Law. (This is wholly analogous to the Al/Si pair.) Hence these figures are only "to hang your hat on." This must be kept in mind in evaluating that which follows.

(3) The temperatures shown in Table I are below the melting point of Al_2O_3, as are most of those in Table II. Distillations at 400 torr could be handled by calcia, and at 2000 torr calcia might also serve. (Magnesia is too likely to be reduced by Si at the upper range of temperature, owing to the high vapor pressure of free magnesium.) Hence we are looking at reasonable processes that could be conducted in locally-available ceramic materials.

(4) At higher pressures, 400 to 2000 torr, the improvement in purity obtainable from each stage of distillation can be described as "an extra nine." Thus 0.99 Si goes to 0.999 Si, etc., approximately. This compares favorably with zone refinement. Of course, the attack of hot Si on the walls of the apparatus will put a limit to the purification process, as here outlined, particularly at high temperatures and pressures.

(5) Other peculiarities will be presented below, in their appropriate places.

5.4.2.2 Alternative Processes. A variety of these might be proposed. For example:

(1) Si might be vaporized as SiO, ensuring a clean separation. SiO disproportionates, on condensation, to Si and SiO_2, as a colloidal mixture. If the condensing surface were calcia, the SiO_2 would fuse with it, producing wollastonite and pure silicon. This deserves further investigation.

(2) Sulfur might be used as a parting agent. SiS and SiS_2 are also volatile.

(3) Iron might be removed as the carbonyl. This requires a source of carbon, specifically carbon monoxide. Since carbon is rare on the Moon, there is an apparent violation of the axiom, which requires explanation.

Of the various human wastes that must be recycled or disposed of in the context of a lunar base, carbon dioxide presents a substantial difficulty in that a considerable greenhouse area (or some equivalent, whether photosynthetic or purely nonbiological) must be provided per person for complete recycling. It is easy enough, however, to remove it from air and to add oxygen in its place. The temptation, therefore, will be to simply dump it. That would be a dismal and sinful waste, however, if it can be put to other uses. Conversion to CO, or even to carbon, would be easy in the metallurgical context: reaction with spiegeleisen in the first case and magnesium in the second.

Some other alternative processes for the purification of silicon might be proposed. For the moment, let us thoroughly examine distillation, so that at least we will have a standard of comparison, not to mention a workable process (when operated in combination with zone-refining and other established techniques).

5.5 EXAMPLES OF COMPUTER DESIGN

Simple programs have been written on the basis of the standard McCabe-Thiele construction, as generally understood in chemical engineering. (Special notes are given below.) Sample results are shown in Tables 5-5 through 5-7; the inputs are for Al/Si at 1 atm, as given earlier; Xp = mol% Al in product.

Table 5-5. Sample McCabe-Thiele Results (Xp = 0.9995; R = 3)

XAl (outflowing liquid)	YAl (inflowing gas)	Plate No., From Top
0.994864	0.996023	1
0.961977	0.971358	2
Flag down		
0.753165	0.814749	3
0.41609	0.573582	4
0.23 etc.	0.427463	5

We are, in the data above, beyond a postulated feed composition, Xf = 0.50, which is achieved in 4 plates.

Table 5-6. Sample McCabe-Thiele Results (Xp = 0.9995; R = 0.5)

XA1 (outflowing liquid)	YA1 (inflowing gas)	Plate No., From Top
0.994864[a]	0.997955	1
0.979906	0.992969	2
0.934530	0.977843	3
Flag down		
0.806082	0.935027	4
0.658854	0.885951	5
0.543641	0.847547	6
0.477701	0.825567	7
0.445962	0.814987	8

[a] This output depends only on Xp, and hence is identical to the plate 1 outflow of the previous table.

Xf is achieved by the seventh plate.

Table 5-7. Sample McCabe-Thiele Results (Xp = 0.9998; R = 0.85)

XA1 (outflowing liquid)	YA1 (inflowing gas)	Plate No., From Top
0.999787	0.999891	1
0.998852	0.999462	2
0.994480	0.997453	3
0.975191	0.988541	4
Flag down		
0.896354	0.952368	5
0.714434	0.868783	6
0.512255	0.775890	7
0.385755	0.717769	8
0.330224	0.692254	9

Xf is reached in 7 to 8 plates.

Notes to the Tables

(1) Since the downflow of liquid from one plate (or from the condenser) is that received by the next lower plate, there is no point in printing this statistic twice. Similarly, the upflowing vapor from a given plate is that received by the plate above (or by the condenser). Hence there are only two, not four, data characterizing a given plate. These can be conceived of as the X-Y coordinates of a point on the Op line that represents the plate.

(2) "Flag down:" most computers incorporate testable switches known as "flags," "lamps," "lights," or "virtual lights," according to the jargon of the manuals; in the present case, one of these "flags" (switches) is used to distinguish between two sections of the vapor curve.

The vapor curve, for this program, has been divided into two regions, "end" (above X = 0.92), and "bend" (below 0.92). The "end" region is approximated by a quadratic under a modified log-log transform. The "bend" region is approximated by a cubic under a special transform.

Thus the notation "flag down" represents the boundary between the two approximations to the vapor curve.

(3) Precision: again, the precision is excessive to guard against round-off error in subsequent processing.

5.6 SUMMARY

In the study of metals separation on the Moon, we have developed a first generation of computer programs that:

(1) Create Raoult's Law phase diagrams from literature vapor-pressure data.

(2) Represent the vapor curves of Y-X plots as empirical transformed polynomials.

(3) Mimic conventional graphic methods of designing distillation plants, while surpassing them in precision.

Often, vacuum distillation is the best way to separate (given lunar conditions) critical pairs of metals. Distillation may be used:

(1) To purify aluminum for structural purposes.

(2) To purify silicon for electronic purposes.

(3) To purify the alkali and alkaline-earth metals. Of these, magnesium and calcium may be used for structural purposes, and all may be used for chemical service.

Distillation may be potentially used to purify titanium for structural purposes. Modified by preliminary chemical steps, such as liquation or reaction with lunar abundant elements, distillation may be potentially used:

(1) To ultrapurify aluminum for Electrical Conductivity (EC)[12] or semiconductor use.

(2) Provide alternate routes to the ultrapurification of silicon.

In all but one of the processes considered in this report, imported materials have been deliberately excluded. (This is the result of an assumption, as noted earlier.) The exception is carbon monoxide, which might be obtained from life-support wastes.

The developments in the coming year are expected to include extensions of these promising beginnings, plus a presentation of a hardening method for "lunar steels" independent of carbon.

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[12] Note that the traditional objection to aluminum as a conductor does not apply on the Moon, since the aluminum will not develop high-resistance oxides at contact points. Copper and silver are not lunar-abundant.

REFERENCES

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SECTION 6

ENERGY SYSTEMS

R. Jones

6.1 REQUIREMENTS

Energy systems may handle energy in various ways to perform four general functions: (1) collection, (2) conversion, (3) storage and (4) transmission. Energy systems may deal with energy in any one of its several forms, i.e., mechanical, thermal, electrical, magnetic, electromagnetic, chemical, and nuclear. Examples of energy systems are solar arrays, batteries, nuclear reactors, electrical lines, electrical power-processing equipment and thermal radiators. The objective of the RUSR energy system is to provide energy in the most economical manner for all the major energy-consuming elements of the eventual operating system.

For the future operating processing facility, one can identify three major categories of operations that will require the use of energy systems: (1) material conditioning and processing, (2) transportation and (3) miscellaneous support operations (crew support, housekeeping, communications, etc.). These operations may be carried on at different locations depending on eventual overall design. For any one operation, the characteristics of these different locations will influence the choice of the appropriate energy system required to perform a certain function. The major possible locations for operations include: (1) the lunar poles, (2) lunar midlatitudes, (3) the surfaces of asteroids, (4) asteroid orbits, (5) deep space, (6) planetary orbits, (7) the surface of Mars and (8) the poles of Mars. The characteristics of these locations insofar as they influence the choice of energy systems are listed in Table 6-1.

The general operations (material processing, transportation and miscellaneous support) will require energy in either electrical or thermal form. Some material-processing and material-conditioning operations will be able to utilize direct thermal power while others will need electrical power. The majority of the transportation and miscellaneous operations will require their energy to be provided in the form of electrical power. For realistic scenarios, only nuclear and solar energy sources can be considered. Waste thermal power from material-processing functions may also be recovered and recycled, but this cannot be considered an energy source. The preliminary energy requirements per ton of process output for the specific processes under each general operation are listed in Table 6-2. Of nearly equal importance to the amount of energy required for each process is the form (electric or thermal) in which that energy is needed. The form of the energy (thermal or electric) is critical to the mass of equipment (power supply) which provides the energy. Thermal energy may be obtained directly from the sun or a nuclear heat source. Electric energy must be converted from some thermal source at an efficiency much less than one. Therefore the mass of the power supply providing 1 joule of electric energy is much larger than the mass of a power

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[1] Metric tons (10^3 kg) throughout this section.

Table 6-1. Location Characteristics

Location	Distance (A.U.)	Solar Intensity (kW/m^2)	Illumination	Gravity (g = 1)	Atmosphere
Lunar Poles	1	1.35	Continuous Sun	.17	No
Lunar Latitudes	1	1.35	Intermittent	.17	No
Asteroid Surfaces	.7 - 2.5	2.76 - .216	Intermittent	.015 - .000015	
Asteroid Orbits	.7 - 2.5	2.76 - .216	Continuous	0	No
Deep Space (L-5)	1	1.35	Continuous	0	No
Earth Orbit (LEO or GEO)	1	1.35	Intermittent	0	No
Martian Poles	1.52	.58	Continuous	.377	Yes
Martian Latitudes	1.52	.58	Intermittent	.377	Yes

supply producing 1 joule of thermal energy. Table 6-2 includes the approximate ratio of electric to thermal energy required for each process. Energy requirements are a basic starting point, but for an operating facility working at some production rate (tons[1]/yr), the time-rate of energy usage, i.e., power, is the key parameter. By far the largest energy/power users are the electrolysis and vapor-phase processes with silicate compaction representing a lower bound on energy/power needs for primary processing. Using the energy requirements of Table 6-2 and the suggested production rates of 1, 10, 100 tons/yr for electrolysis and vapor phase processing and 10, 100, 1000 tons/yr for silicate compaction the electric and thermal power requirements for these processes can be estimated and are presented in Table 6-3.

Table 6-2. Energy Requirements

Operation	Required (Actual Energy per Ton of Output)	Ratio of Required Energy Form (Electric/Thermal)
Processing		
Beneficiation and Comminution[a]	10 kw hr/ton	>> 10
Silicate Compaction[a]	500	.25
Melting Lunar Basalt[a]	500	< 1
Electrolysis[b]	10,000 - 30,000	10
Vapor Phase Processing[b]	50,000 - 100,000	10
Secondary Processing[a]	10	10
Storage and Handling[a]	10	10
Transportation		
Lunar Surface[c]	.05 - .3	>> 10
Electromagnetic Launching[a] (surface to low lunar orbit)	350	>> 10
Miscellaneous Support		
Excavation[c]	.005 - 0.3	>> 10
Housekeeping[a] (communications) (data processing) (command and control)	1 - 5 kW_e [d]	>> 10

[a] JPL Estimates.

[b] Steurer, W. H., "Extraterrestrial Materials Processing", JPL Publication 82-41, April 1982.

[c] Carrier, W. D., "Excavation Costs for Lunar Materials", Space Manufacturing III: Proceedings of the Fourth Princeton/AIAA Conference, October 1979, p. 89.

[d] Minimum electrical power requirements, independent of the required energy per ton of output.

Table 6-3. Power Requirements

Form	Tons/Yr			
	1	10	100	1000
Electric (kW_e)				
Vapor Phase	6 - 10	60 - 111	600 - 1100	
Electrolysis	1 - 3	10 - 30	100 - 300	
Compaction		.1 - 1	1 - 10	10 - 100
Thermal (kW_{th})				
Vapor Phase	.6 - 1	6 - 11	60 - 110	
Electrolysis	.1 - .3	1 - 3	10 - 30	
Compaction		.4	4	40

6.2 POWER-SYSTEM OPTIONS AND TECHNOLOGY

The implementation of the four power-system functions (collection, conversion, storage, and transmission) can take different forms with several different technologies available or potentially available for each function. Tables 6-4 through 6-7, below, contain the most common technology options for each function and three technology levels where appropriate and a representative figure of merit for each technology option and technology level. The three technology levels are: 1) in use, 2) in development (past or current development activity at exploratory to advanced development level) and 3) on the frontier (which means the technology should be physically possible but little or no development is taking place). The energy sources assumed in the following tables were solar or nuclear. Conversion systems are those that convert some form of energy to another. Storage devices are included that upon discharge supply electrical energy. Thermal storage systems might also be of interest.

Table 6-4. Energy Collection Systems

System	Level	kg/kW$_e$	kg/m^2
Microwave Rectenna	On Frontier	.5 - 4	TBD
Laser Receiver	On Frontier	TBD	TBD
Photovoltaic Array (1 A.U.)	In Use	33 - 20	2
	In Development	15 - 8	
Solar Collectors (1 A.U.)	In Development	NA	2
Solar Receiver	On Frontier	2 (kW$_{th}$)	TBD

Table 6-5. Energy Conversion Systems

System	Level	kg/kW$_e$
From Electrical to Electrical:		
Power Conditioning Equipment		
.2 - 2 kW$_e$	In Use	60 - 1.2
10 kW$_e$	In Development	11 - 1.2
100 kW$_e$	On Frontier	3.6 - 1.2
1000 kW$_e$	On Frontier	1.8 - .6
From Photonic to Electrical:		
Photovoltaic Arrays		
5 kW$_e$	In Use	33 - 20
5 - 100 kW$_e$	In Development	15 - 8
>100 kW$_e$	On Frontier	5 - 3
From Chemical to Electrical (fuel not included):		
Primary Fuel Cell	In Use	70 - 40
Regenerative Fuel Cell	In Development	90 - 50
From Solar Thermal to Electrical (includes collector/concentrator):		
Thermionic		
.5 kW	On Frontier	30 - 70
5 kW	On Frontier	30 - 70

Table 6-5. Energy Conversion Systems (continued)

System	Level	kg/kW_e
Dynamic		
Brayton $1 - 15\ kW_e$	In Development	120
Rankine $1 - 15\ kW_e$	In Development	TBD
Stirling $1 - 15\ kW_e$	On Frontier	TBD
Regenerative Fuel Cell	In Development	90 - 50
Magnetohydrodynamics (MHD)		
$1\ MW_e$	On Frontier	TBD
From Nuclear Radioisotope to Electrical (includes heat source):		
Thermoelectric		
$.1 - 10\ kW_e$	In Use	260
$.1 - 10\ kW_e$	In Development	150
Dynamic		
Brayton $1 - 15\ kW_e$	In Development	160 - 130
From Nuclear (Fission) to Electrical (includes the heat source):		
Thermoelectric		
$25 - 400\ kW_e$	In Development	90 - 20
$1000\ kW_e$	On Frontier	20
MHD		
1000 kW	On Frontier	5
10,000	On Frontier	4.0
Brayton		
$5\ kW_e$	On Frontier	100
$100\ kW_e$	On Frontier	20
Microwave Receiver	On Frontier	.5 - 4
	Conceptual	TBD

Table 6-6. Energy Transmission Systems

System	Level	kg/m^2	kg/kW (radiated)
Thermal Radiators	In Use	3 - 11	
Laser Transmissions	On Frontier		TBD
Microwave Transmitter			
2.45 GHz	In Development		TBD
22.145 GHz	On Frontier		TBD

Table 6-7. Energy Storage Systems

System	Electrical Energy Out	Level	kg/kJ
Inertial			
Homopolar Generator	500 MJ	In Development	TBD
Compensated Pulsed	50 MJ	In Development	TBD
Alternator	5 MJ	In Development	3
Flywheel/Alternator	500 MJ	In Development	.002
Inductive			
Superconducting	1 MJ	In Development	1 - 5
	100 MJ	On Frontier	.01 - .1
Capacitive	1 - 10 KJ	In Use	8 - 12
Batteries			
Secondary		In Use	.01
		In Development	.005
		On Frontier	.002

6.3 THE UTILIZATION OF SPACE MATERIALS FOR POWER SYSTEMS

Preliminary indications are that the power subsystem is likely to be by far the most massive subsystem of a RUSR processing facility. As described in Section 8.0, if the ratio of the mass of the entire processing facility to the total facility lifetime product exceeds a few percent, then the use of RUSR materials in Earth orbit (at least) may not be attractive. Therefore any steps that can be taken to reduce the imported mass of the power system may have large benefits to the RUSR program as a whole. The largest fraction of the power system will be the power source itself, most likely either a solar array or nuclear reactor power source. Both of these sources can conceivably supply power from 10 - 1000 kW_e at 100 to 1000 volts DC or AC. In order for a power source component to be considered as potentially replaceable by a component built by the RUSR facility, it should have the following characteristics: 1) the component should form a nontrivial percentage (10 - 20% or larger) of the total source mass, 2) the component should be simple to fabricate from extraterrestrial materials and 3) the component should be easily integrated into the power source.

Considering photovoltaic arrays and nuclear (fission) power sources, certain components appear as candidates. Of course all hydrocarbon components are eliminated. Initially most of the gross structure required to support the power source against lunar gravity would seem to be potentially replaceable by cast basalt or some other material. However, this gross support structure may not be a large fraction of the overall mass.

For photovoltaic arrays the one component that appears to be potentially replaceable is the cover glass which shields the solar cell from U.V. and some particulate radiation. For a 200-W/kg (5-kg/kW) array, the cover glass (sized for the lunar radiation environment) may consume 10 - 20% of the array mass.

A nuclear reactor power source contains the following basic components: 1) reactor, 2) radiation shield, 3) heat transport system, 4) thermal to electric conversion system and 5) a heat rejection system. Of these components the radiation shield appears to be potentially replaceable by RUSR materials. The radiation shield may consume up to 30% (more if human beings are in the immediate area) of the total nuclear power source mass. The heat rejection system is usually a flat plate of metal kept hot by heat pipes or a pumped fluid loop. Perhaps the reject heat could be conducted to the lunar subsurface by a system of buried heat pipes or fluid loops, thereby reducing the heat rejection system mass.

Large current conductors may also be potentially replaceable by extraterrestrial metals. The former suggested items certainly do not exhaust the list of potentially replaceable components, but do serve as a starting point and a first cut at the order of magnitude of potential benefits to the RUSR program when one considers RUSR materials for power-system components in place of "costly imported" components from Earth.

6.4 SUMMARY

For the most common strawman product, i.e., O_2, it is clear that the power required to operate an O_2 facility will be very large and that the power-system mass may dominate the entire facility mass. Therefore power-system technology is of extreme interest to the RUSR program. Section 8, which follows, will describe how the facility-mass to lifetime-product mass must be less than about .05 in order for lunar O_2 to be attractive in LEO. This small ratio implies power-system technology on the order of 1 - 5 kg/kW_e. The production rate which is assumed to be required for a commercially viable facility is of the order of 100 tons/yr which along with the energy requirements imply power levels of between 100 and 1000 kW_e. Space power sources in use today include radioisotope-thermoelectric generators at 200 kg/kW_e and 300 We and photovoltaic arrays at 30 kg/kW and 1 - 5 kW. Photovoltaic arrays at 15 kg/kW_e are awaiting flight demonstration. There are no nuclear reactor power systems currently available, although a NASA/DOE/DARPA R & D program on a 100 kW_e, 20 - 30 kg/kW system is underway. Obviously the RUSR requirements on power system technology are orders of magnitude greater in scale and several times better in performance than what is in use now and still very much bigger and better than what is currently receiving R & D resources. Clearly, power-system technology is of prime interest to the RUSR program, and future results may indicate a need for major power-system technology/system development.

SECTION 7

TRANSPORTATION

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7.1 INTRODUCTION

This section describes the results of analyses of space-based transportation systems for the transport of extraterrestrial materials from their point of origin (e.g., lunar surface) to final delivery point in Earth orbit and transport of equipment and supplies for extraterrestrial processing to the operational site. The emphasis has been on the use of near-term (pre-year 2000) propulsion systems, such as advanced chemical (H_2/O_2) and nuclear electric systems. More exotic systems (mass drivers, nuclear thermal rockets, etc.) were also considered to evaluate the potential to be gained by use of post-year 2000 technologies. Finally, the effects of using extraterrestrial propellants (e.g., H_2 from lunar polar water-ice for chemical systems) were compared to those of propulsion systems deriving all or part of their propellants from the Earth.

In this year's efforts, transportation of materials to and from Earth-orbit and the lunar surface has been emphasized. Transportation for asteroidal materials was also considered, but in less detail. Near-term propulsion systems were emphasized since the technology base being developed for space-based orbit transfer vehicles (OTV's) designed for Earth-orbit transfers can be applied directly to Earth-orbit-to-Moon or asteroid vehicles. This is especially important economically, since the Earth-orbit satellite market will drive and support development of space-based OTV technology independently of, and probably prior to, an extraterrestrial materials program. Thus, the general technology and even specific vehicles will be readily available to a lunar materials program whereas the technology and vehicles required for a Martian materials program would probably need to be developed and supported by that program alone.

7.1.1 Lunar Materials Transportation Scenario

The general operational steps of orbital transfers of the vehicles are described below. In these analyses, we have assumed steady-state operations, i.e., all vehicles are already in place, production facilities are already established and operating, and so on. We have only briefly considered transportation requirements for the initial system set-up this year (see Section 7.2.5); this will be considered in more detail next year. Of course, the cost of the initial set-up would be amortized over many years of steady-state operation.

For these analyses, we have considered only operations from a low Earth-orbit (LEO) typical of a standard Space Shuttle (STS) parking orbit (around 275 km altitude). For example, an OTV and its attached Tanker-Cargo payload module would be serviced and refueled in LEO either by the Shuttle (or its descendant) or by a (manned) space station facility. As mentioned earlier, the OTV's used to transfer lunar materials can be the same as those used for Earth-orbit satellite traffic, or sufficiently similar, so as to permit a commonality of servicing facilities in LEO.

Typically, the Tanker-Cargo module attached to the OTV would be filled with propellant needed by the OTV for its return to Earth orbit from the Moon, or for the needs of a lunar transfer vehicle (LTV) used to ferry cargo between the lunar surface and low lunar-orbit (LLO) typically 100 km above the lunar surface. Once fueled, the OTV would transfer from LEO to LLO. A fully loaded LTV and its attached Tanker-Cargo unit would lift off from the Moon and rendezvous with the OTV in LLO. Propellants and cargoes would be exchanged as needed in LLO between the two vehicles. The LTV and its Tanker-Cargo unit would return to the lunar surface. The OTV and its Tanker-Cargo unit would leave LLO and go either to LEO directly and deliver its payload at LEO, or first go to geosynchronous Earth-orbit (GEO) and deliver its payload at GEO, and then finally return to LEO for servicing and refueling for the next trip.

In the various analyses described below, we have assumed that some lunar material is processed to provide liquid oxygen (LO_2) for the propulsion needs of the baseline chemical (H_2/O_2) OTV and LTV. In fact, lunar LO_2 propellant is essential for any advantageous transport of lunar material to LEO or GEO using chemical vehicles. Additional lunar LO_2 could be produced for delivery to LEO or GEO for use by other chemical propellant OTV's. It is expected that a large demand will exist for LO_2 to satisfy OTV propellant needs for high LEO-to-GEO satellite traffic loads. The LO_2 could also be used for life-support, cold-gas attitude control systems, or other needs. Lunar material could also be processed to produce other useful materials or products (e.g., structural members, shielding panels, etc.), but for this initial analysis we have assumed that the Tanker-Cargo modules are sized to carry LO_2 or liquid hydrogen (LH_2). We have also assumed that there will be no lunar source of hydrogen for chemical propulsion needs so that all the LH_2 must come from Earth. Similarly, we have assumed that all the argon (Ar) required for electric propulsion propellant comes from the Earth. Later in the analyses we show the impact that the presence of lunar LH_2 would have on chemical propulsion systems, and of lunar LO_2 use (instead of Ar) in electric propulsion systems. The ability to eliminate the use of any Earth-supplied propellants can greatly increase the efficiency of the transportation system such that the discovery of a source of lunar hydrogen, such as lunar polar water-ice, would profoundly impact the efficiency and economics of a lunar materials program.

The velocity increments (ΔV's) for the various steps are listed in Table 7-1. High thrust-to-weight (T/W) propulsion systems like chemical or nuclear thermal systems have lower ΔV's than low T/W systems like electric systems. OTV's can use high or low T/W propulsion systems, but an LTV can only use a high T/W propulsion system for lunar takeoff and landing. For example, the Apollo Lunar Module had a T/W of about 0.3. Aerobraking can be employed by compact high T/W systems to reduce the propulsion ΔV required to return to LEO; this results in a saving of 2.3 km/s in equivalent propulsive ΔV.

Table 7-1. Delta V Requirements for Lunar Material Transportation Scenarios

Step	High T/W ΔV (km/s)	Low T/W ΔV (km/s)
LEO ⟷ LLO	3.9	7.80
GEO ⟷ LLO	1.8	2.00
LEO ⟷ GEO	4.3	5.85
LLO ⟷ LEO w/AB	1.6	--
GEO ⟷ LEO w/AB	2.0	--
Lunar Surface ⟷ LLO	1.6	--

LEO = Low Earth-Orbit (275-km alt.)
GEO = Geosynchronous Earth-Orbit
LLO = Low Lunar-Orbit (100-km alt.)
AB = Aerobraking
T/W = Thrust-to-Weight

7.1.2 Transportation Figure-of-Merit

In order to establish a figure-of-merit for evaluating the performance of the various propulsion systems and vehicles, we have made use of a quantity called the Mass Payback Ratio (MPBR):

$$\text{MPBR} = \frac{\text{Total Net Mass of Extraterrestrial Material Delivered to Point of Use}}{\text{Total Mass of Earth Material Which Must Be Put in LEO to Deliver the Extraterrestrial Material}} \qquad (7\text{-}1)$$

This ratio gives the mass multiplication that could be derived by delivering a given mass of Earth-supplied propellant to LEO, using this propellant to run the OTV and LTV, and delivering a mass of lunar material back to LEO (or GEO). These analyses are concerned with propulsion oriented masses, such as propellants; Section 8 discusses the more general case where process equipment and supplies are included in the denominator. This ratio must be significantly greater than unity, i.e., the mass of lunar material returned to LEO must be significantly greater than the mass of Earth-supplied material used to run the system; otherwise the original mass of Earth-supplied material could be used directly in LEO. As shown below, it should be possible to achieve MPBR's of about 2 to 10 using Earth plus lunar propellants. Higher MPBR's can be achieved by the use of all-lunar propellants (e.g., lunar LH_2 and LO_2 for a chemical system).

In Eq. (7-1), the net mass of lunar material delivered to Earth orbit is used since some of the total lunar material may be used as propellant (LO_2) to run the system. In these analyses, only the net mass of Earth-supplied propellants (LH_2, Ar) delivered to LEO (by the STS) were considered; propellant tanks in the STS cargo bay were assumed reuseable and were not included in the Earth-supplied mass total. Note that in a more detailed analysis, the tank weights should be included since a LH_2 tank would be heavier than a LO_2 tank, for a given mass of either propellant. However, for this level of analysis, exclusion of the tankage weights does not introduce a significant error and also eliminates the necessity of assumption of a specific Space Transportation System vehicle (e.g., current STS, Shuttle Derived Vehicle, Single-Stage-to-Orbit Vehicle, etc.). Instead, only a simple mass ratio is used to simplify and clarify the comparisons.

It is possible to directly compare masses when lunar materials are returned to LEO. However, the situation is more complex when the lunar material is returned to GEO, since the STS cannot deliver Earth-supplied material directly to GEO. A MPBR can still be calculated from Eq. (7-1) but the comparison is not entirely valid. Instead, it is more correct to compare the delivery of lunar material to GEO with the delivery of Earth-supplied material to GEO using a LEO-to-GEO OTV. A Mass Ratio (MR) can be calculated for this case also:

MR (Direct GEO Delivery) =

$$\frac{\text{Net Mass of Terrestrial Material Delivered to GEO (GEO Payload)}}{\begin{array}{l}\text{Total Mass of Terrestrial Material (Propellant + GEO Payload)}\\ \text{Which Must Be Put in LEO to Deliver the Payload to GEO}\end{array}} \qquad (7\text{-}2)$$

In this case, the net useable material in GEO is the GEO Payload; the total Earth-supplied mass in LEO includes the propellant for the OTV and the GEO payload. The mass in LEO would also include the mass of an OTV and Tanker-Cargo module if these were one-way (non-reuseable) vehicles. Finally, we can now take the ratio of MPBR (extraterrestrial material at GEO)/MR (direct delivery of terrestrial material at GEO). The result is an "effective" MPBR for GEO delivery that correctly relates the ratio of mass of net extraterrestrial material that can be delivered to GEO to that mass of Earth-supplied material that can be delivered to GEO, with both based on an equal amount of initial Earth-supplied mass in LEO.

7.2 LUNAR MATERIAL TRANSPORTATION USING VEHICLES WITH CHEMICAL PROPULSION SYSTEMS

A series of analyses were performed for transportation of lunar material (e.g., lunar LO_2) from the lunar surface to LEO and GEO using a baseline OTV and LTV with chemical H_2/O_2 propulsion systems and aerobraking. These propulsion systems represent near-term extrapolations of current H_2/O_2 engine technology that would be developed for LEO/GEO OTV's (Refs. 7-1 and 7-2) and could thus be applied to the lunar material scenarios. The vehicles would be derived from the general OTV data base and modified as needed for teleoperation, long service life, or lunar landing. Note that the oxygen required for an H_2/O_2 chemical engine can be made relatively easily from lunar soil as discussed in Sections 3 and 4. If a significant source of lunar hydrogen can be found (e.g., from lunar polar water-ice), it would permit operations without the need for any Earth-supplied propellants, and thus yield an infinite MPBR. The discovery of a significant source of lunar hydrogen for propellants would thus have a major impact. However, even without a source of lunar hydrogen, a chemical propulsion system using relatively easily produced lunar LO_2 is highly advantageous. This is due to the fact that over 80% of the propellant weight in an H_2/O_2 system is oxygen, so the relatively low performance available from chemical systems, as compared to more advanced propulsion systems (e.g., electric), is balanced by the ability to supply a large fraction of the total propellant mass from an extraterrestrial (lunar) source.

7.2.1 Operational Considerations

The use of high T/W chemical or nuclear-thermal propulsion systems permits fairly rapid trip times from LEO to LLO of about three to four days one-way. By contrast, the low T/W electric propelled vehicles described in Section 7.3 would require much longer trip times, from 100 to 200 days one-way. Thus, high T/W propulsion could be used for high-value, high-priority cargos, including manned missions, with low T/W systems used for bulk cargos. For an all-chemical system using lunar LO_2 and Earth-supplied LH_2, the LH_2 would be transported to LEO by the STS or its descendant. Lunar LO_2 from a previous delivery would be used by the OTV to travel to LLO where LH_2 in the OTV Tanker-Cargo unit is used to refuel the OTV and the LTV.

7.2.2 Chemical OTV and LTV Description

As a starting point in our analyses, we have used a conceptual advanced OTV defined in a recent Boeing OTV study (Ref. 2). The OTV is sized to be carried to LEO in the STS cargo bay as are almost all components of the various vehicles described below. This makes it possible to return a vehicle to Earth inside the STS cargo bay if extensive repairs or servicing are needed; however, routine servicing would be conducted at a LEO-based space station for all OTV's.

The OTV is space-based and reuseable. It uses an aerobraking system to reduce the propulsive part of the ΔV required to return to LEO. In the Boeing study (Ref. 2) an inflatable balloon-like ballute was assumed, although several different aerobraking methods might be used (Ref. 7-3 and 7-4). In general, aerobraking to LEO saves about 2.3 km/s in a propulsive ΔV; it is an absolutely vital technology since a chemical OTV cannot achieve a net positive mass return without aerobraking. For these analyses, we have assumed the availability of aerobraking technology. This assumption is based on the probability that the technology will be essential for, and therefore developed for, operational OTV's for LEO/GEO transportation.

It is further assumed that space-based long-life propellant transfer systems and active refrigeration systems for cryogenic propellants such as LH_2 (and to a lesser extent LO_2) will have been developed. Refrigeration systems of this type are under development (Ref. 7-5) since they will be required for future deep space missions and for maintenance of LEO-based propellant depots. Again, we see that a technology important to this program can be expected to be developed prior to and independent of this program.

With the above assumptions and requirements, the baseline OTV used for these analyses has a dry weight of 5.8 metric tons (T = 1000 kg) and a propellant capacity of 32.5 T. The OTV uses two RL10-II engines (Ref. 7-2) operating at an oxidizer-to-fuel ratio (O/F) of 6 and a specific impulse (I_{sp}) of 460 s. Advanced OTV engines with I_{sp}'s of 480 to 500 s may also be available, although their improved performance would be countered by the high initial development costs of advanced engines (Ref. 7-2 and 7-3).

The Lunar Transfer Vehicle (LTV) used to shuttle between the lunar surface to LLO is assumed to be essentially a stretched OTV. The baseline vehicle dry weight is 11.5 T and it has a propellant capacity of around 50.0 T. In order to provide sufficient T/W for lunar liftoff, the LTV uses seven RL10-II engines. The landing gear are estimated to weigh 5% of the lunar weight of the fully loaded vehicle (i.e., 5% * 1/6 "g" * Total Weight). The engines and landing gear could be transported as a single unit to LEO in the STS External Tank Aft Cargo Compartment (ACC). The LTV propellant tankage could be carried to LEO inside the STS cargo bay on the same flight and both components assembled together in LEO. Use of the ACC permits delivery of outsized diameter (8 m) payloads for only a small payload penalty due to carrying the empty External Tank to LEO.

Finally, the Tanker-Cargo units attached to the OTV and LTV are scaled to 5% of the LO_2 (or other payload weight) or 25% of the LH_2 weight that is transported, whichever is the greater. This results in an OTV Tanker-Cargo unit that has a dry weight of 3.5 T and can transport about 70 T LO_2 or 14 T LH_2. and a LTV Tanker-Cargo unit with a dry weight of 5.0 T that can carry about 100 T of LO_2 or 20 T of LH_2. The OTV, LTV, and various Tankers are assumed to have active refrigeration to minimize propellant loss during orbital transfers, and there are propellant depots at LEO, GEO, LLO, and the lunar surface for long-term storage of cryogenic propellants.

7.2.3 MPBR Calculations Approach

These vehicles represent a conservative, near-term set of spacecraft representing the minimum technology of the 1990's. We can now take these vehicles and run them through the lunar material transportation scenarios to determine propellant and payload masses for the various steps, and finally find the MPBR for a given scenario. By fixing the OTV propellant capacity at 32.5 T we effecitvely size the entire system, including the LTV, so as to fit the payload/propellant capacities for the various steps.

To determine the various masses, we use the rocket equation:

$$\frac{M_B}{M_0} = \exp\left(\frac{-\Delta V}{g I_{sp}}\right) \qquad (7\text{-}3)$$

where M_B = Final (dry) mass after engine burn,

M_0 = Initial (wet) mass before engine burn,

g = Gravitational units constant (32 ft/s^2 or 9.8 m/s^2)

and

$$M_0 = M_S + M_{PL} + M_p \qquad (7\text{-}4)$$

with

$$M_B = M_S + M_{PL} \qquad (7\text{-}5)$$

where

M_{PL} = Mass of payload,

M_S = Stage dry mass,

M_p = Propellant mass.

In this case, M_S would be the dry weight of the OTV and its Tanker-Cargo unit (9.3 T) or of the LTV and its Tanker-Cargo unit (16.5 T). The ΔV values would be those from Table 7-1. Specific impulse (I_{sp}) is converted to units of velocity by multiplying I_{sp} in "seconds" by the appropriate value of the constant of acceleration (g = 32 ft/s^2 or 9.8 m/s^2) to obtain units consistent with ΔV.

As mentioned above, the 32.5 T propellant capacity of the OTV sizes the system. Specifically, we assume that the OTV is fueled in LEO. The full 32.5 T of propellant in the OTV propellant tanks is then used to transfer the vehicle from LEO to LLO for a ΔV of 3.9 km/s. The OTV propellant tanks are filled to their 32.5 T capacity in LLO (using terrestrial LH_2 and lunar LO_2) and used to return the vehicle to LEO with aerobraking (ΔV = 1.6 km/s). Alternatively, 27.3 T of the 32.5 T of propellant can be used to travel first from LLO to GEO, where the lunar material payload is deposited, and the remaining 5.2 T of propellant used to transfer from GEO to LEO with aerobraking.

7.2.4 MPBR Results for Return of Lunar Material to LEO or GEO

With the OTV propellant mass specified for each step, it then becomes possible to calculate the propellant and payload masses for the various steps. The results are summarized in Table 7-2 and 7-3. Several options are included depending on whether Earth- or lunar-supplied LO_2 is used to refuel the OTV in LEO for its trip from LEO to LLO, or if it is desired to transport some material to the lunar surface as part of the normal operations. This material could be consumable chemicals or supplies, replacement parts, and so on.

As mentioned earlier, it is necessary to calculate an "effective" MPBR for GEO deliveries based on the mass ratio (MR) that could be achieved for a direct delivery of Earth-supplied material to GEO. The MR's for direct GEO delivery using (1) a one-way non-reusable OTV and Tanker, (2) a one-way Tanker and reusable OTV, and (3) a completely reusable OTV and Tanker are shown in Table 7-4. The MR's range from about 0.20 to 0.26, so that the "effective" MPBR's for return of lunar material to GEO are about 4 to 5 times greater than the simple MPBR found based on initial mass in LEO. Thus, return of lunar material to GEO appears strongly favored over return to LEO, even though in both cases the OTV returns finally to LEO.

From the results in Table 7-2, we see that for this conservative baseline transportation system the MPBR's for return of lunar material to LEO are significant but not spectacular, resulting in about a doubling of the initial mass in LEO. From Table 7-3, the MPBR's for GEO delivery are about one, but when compared to direct GEO delivery, they are effectively about 4 to 5. This suggests that an early highly profitable market might be the supply of lunar LO_2 to GEO for refueling LEO-to-GEO OTV's for their return to LEO. For example, a LEO-to-GEO OTV and its (satellite) payload can be operated in a throwaway (one-way) mode to deliver about 14 T of payload to GEO. However, as shown by case (4) in Table 7-4, by reducing the payload weight by less than 1 T, to accommodate LH_2 for the GEO-to-LEO return leg, it is possible to return the OTV to LEO by using lunar LO_2 at GEO. Thus it is possible to achieve better than one-way OTV delivery performance and still return the OTV to LEO, since six-sevenths of the propellant mass required for the return leg would be available in GEO as lunar-supplied LO_2.

Lastly, the effects of system propellant "closure" should be considered, in which a completely "open" system would derive all its propellant mass from the Earth, whereas a completely "closed" system would derive all its propellant from the Moon. If there is no lunar hydrogen available, the system can never be completely "closed" for a H_2/O_2 propulsion system although use of lunar oxygen can "close" six-sevenths of the propellant mass. Note also that there will be an optimum level of closure for a given mission scenario, as shown in Table 7-2. Elimination of the need for any terrestrial LO_2 propellant (for the OTV LEO-to-LLO step) almost doubles the LEO-delivery MPBR. However, for GEO deliveries, there is a decrease in the GEO-delivery MPBR as more lunar LO_2 is carried to LEO to "close" the system. This is due to increased propellant needed for the GEO-to-LEO step, and thus decreased propellant and payload for the LLO-to-GEO step, as the amount of GEO-to-LEO payload (lunar LO_2) is increased. Also, in a more complete analysis, such as that described in Section 8, it would be necessary to factor in the amount of added lunar factory capacity required to produce the added LO_2 (and LH_2 if available) needed for more complete "closure". Thus, there will be an optimum system "closure" point that may be a function of both propellant needs and lunar processing capability for a given scenario.

Table 7-2. Lunar Material (LO_2) Delivered to LEO
(All weights in metric tons: T = 1000 kg)

Net Earth Equipment & Supplies Delivered to Moon	NO	NO	YES	YES
Earth LO_2 (Propellant) Used in OTV for LEO-to-LLO Step	YES (More Open)	NO (More Closed)	YES (More Open)	NO (More Closed)
Earth-Supplied LH_2 (Propellant)	17.9	17.9	18.9[a]	18.9[a]
Earth-Supplied LO_2 (Propellant)	26.7	0	27.9	0
Total Earth-Supplied Propellant	44.6	17.9	46.8[a]	18.9[a]
OTV LEO-to-LLO LH_2 (Earth)	4.5	4.5	4.6	4.6
OTV LEO-to-LLO LO_2 (Lunar or Earth)	26.7	26.7	27.9	27.9
Net LH_2 (Earth) in OTV-Tanker (for OTV and LTV)	13.4	13.4	14.3[a]	14.3[a]
OTV LLO-to-LEO LH_2 (Earth)	4.6	4.6	4.6	4.6
OTV LLO-to-LEO LO_2 (Lunar)	27.9	27.9	27.9	27.9
OTV-Tanker LO_2 & Payload (Lunar) to LEO	66.9	66.9	66.9	66.9
Net LO_2 & Payload (Lunar) at LEO	66.9	40.2	66.9	39.0
Total LH_2 (Earth) at LLO available for LTV	8.8	8.8	9.7[a]	9.7[a]
LTV LLO-to-Moon LH_2 (Earth)	1.5	1.5	1.5	1.5
LTV LLO-to-Moon LO_2 (Lunar)	8.7	8.7	9.0	9.0
LTV-Tanker LH_2 (Earth) to Moon	7.3	7.3	8.2[a]	8.2[a]
LTV Moon-to-LLO LH_2 (Earth)	7.3	7.3	7.3	7.3
LTV Moon-to-LLO LO_2 (Lunar)	43.9	43.9	44.0	44.0
LTV-Tanker LO_2 & Payload (Lunar) to LLO	103.5	103.5	103.8	103.8
Net Earth Equipment & Supplies on Moon	0	0	0.9	0.9
Total Lunar LO_2 for Propulsion	80.5	107.2	80.9	108.8
Total Lunar LO_2 for Propulsion + Payload	147.4	147.4	147.8	147.8
$\text{MPBR} = \dfrac{\text{(Net Lunar } LO_2 \text{ \& Payload Mass Delivered to LEO)}}{\text{(Total Earth Propellant Mass Initially at LEO)}}$	1.500	2.246	1.429	2.063

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[a] Includes equipment and supplies for lunar surface.

Notes: OTV Dry Wt. = 5.8 T, OTV Payload Tanker Dry Wt. = 3.5 T.
LTV Dry Wt. = 11.5 T, LTV Payload Tanker Dry Wt. = 5.0 T.
Centaur RL10-II Engines (I_{sp} = 460 s, O/F = 6) used on OTV and LTV.
Aerobraking to LEO saves 2.3 km/s in ΔV.

Table 7-3. Lunar Material (LO_2) Delivered to GEO (All weights in metric tons: T = 1000 kg)

Net Earth Equipment & Supplies Delivered to Moon	NO	NO	YES	YES
OTV Refueled (from OTV-Tanker) at GEO	YES	NO	YES	NO
Earth-Supplied LH_2 (Propellant)	17.9	15.8	18.9[a]	18.9[a]
Earth-Supplied LO_2 (Propellant)	26.7	24.8	27.9	27.9
Total Earth-Supplied Propellant	44.6	40.6	46.8[a]	46.8[a]
OTV LEO-to-LLO LH_2 (Earth)	4.5	4.1	4.6	4.6
OTV LEO-to-LLO LO_2 (Earth)	26.7	24.8	27.9	27.9
Net LH_2 (Earth) in OTV-Tanker at LLO (for OTV and LTV)	13.4	11.7	14.3[a]	14.3[a]
OTV LLO-to-GEO LH_2 (Earth)	4.6	4.6[b]	4.6	4.6[b]
OTV LLO-to-GEO LO_2 (Lunar)	27.9	27.9[c]	27.9	27.9[c]
OTV-Tanker LH_2 (Earth) to GEO (for OTV GEO-to-LEO Propulsion)	0.7	0	0.7	0
OTV-Tanker LO_2 (Lunar) to GEO (for OTV GEO-to-LEO Propulsion)	4.5	0	4.5	0
OTV-Tanker LO_2 & Payload (Lunar) to GEO	51.8	41.2	51.8	41.2
Total LH_2/LO_2/Payload in OTV-Tanker to GEO	57.0	(46.4)[d]	57.0	(46.4)[d]
Net LO_2 & Payload (Lunar) at GEO	51.8	41.2	51.8	41.2
OTV GEO-to-LEO LH_2 (Earth)	0.7	0.7	0.7	0.7
OTV GEO-to-LEO LO_2 (Lunar)	4.5	4.5	4.5	4.5
OTV-Tanker LH_2/LO_2/Payload to LEO, Net in LEO	0	0	0	0
Total LH_2 (Earth) at LLO available for LTV	8.1	7.1	9.0[a]	9.7[a]
LTV LLO-to-Moon LH_2 (Earth)	1.4	1.4	1.5	1.5
LTV LLO-to-Moon LO_2 (Lunar)	8.5	8.1	8.7	9.0
LTV-Tanker LH_2 (Earth) to Moon	6.7	5.7	7.5[a]	8.2[a]
LTV Moon-to-LLO LH_2 (Earth)	6.7	5.7	6.7	5.8
LTV Moon-to-LLO LO_2 (Lunar)	39.9	34.3	40.0	34.6
LTV-Tanker LO_2 & Payload (Lunar) to LLO	92.7	77.2	92.9	78.1
Net Earth Equipment & Supplies on Moon	0	0	0.8	2.4
Total Lunar LO_2 for Propulsion	80.8	70.3	81.1	71.5
Total Lunar LO_2 for Propulsion + Payload	132.6	111.5	132.9	112.7

Table 7-3. Lunar Material (LO_2) Delivered to GEO (All weights in metric tons: T = 1000 kg) (Continued)

Net Earth Equipment & Supplies Delivered to Moon	NO	NO	YES	YES
OTV Refueled (from OTV-Tanker) at GEO	YES	NO	YES	NO
MPBR = $\frac{\text{(Net Lunar } LO_2 \text{ \& Payload Mass Delivered to GEO)}}{\text{(Total Earth Propellant Mass Initially at LEO)}}$	1.161	1.015	1.107	0 880
MR Direct (See Table 7-4)				
One-Way Trip	0.261			
Round-Trip OTV only	0.213			
Round-Trip OTV and OTV-Tanker	0.207			
"Effective" GEO MPBR = $\frac{\text{MPBR for Delivery of Lunar Material to GEO}}{\text{MR for Direct Delivery of Terrestrial Material to GEO}}$				
One-Way Trip	4.442	3.883	4.235	3.367
Round-Trip OTV only	5.455	4.766	5.198	4.135
Round-Trip OTV and OTV-Tanker	5.602	4.895	5.339	4.246

[a] Includes equipment and supplies for lunar surface.

[b] Use 3.9 T LH_2 for LLO-to-GEO step, use remaining LH_2 for GEO-to-LFO step.

[c] Use 23.4 T LO_2 for LLO-to-GEO step, use remaining LO_2 for GEO-to-[illegible] step.

[d] Includes 5.2 T propellant in OTV tanks.

Notes: OTV Dry Wt. = 5.8 T, OTV Payload Tanker Dry Wt. = 3.5 T.
LTV Dry Wt. = 11.5 T, LTV Payload Tanker Dry Wt. = 5.0 T.
Centaur RL10-II Engines (I_{sp} = 460 s, O/F = 6) used on OTV and LTV.
Aerobraking to LEO saves 2.3 km/s in ΔV.
See Table 7-4 for description of GEO Direct Supply Options.

Table 7-4. Direct Delivery to GEO using STS and OTV (All weights in metric tons: T = 1000 kg)

Scenario	1. One-Way Trip	1a. One-Way Trip	2. Round Trip OTV Only	3. Round Trip OTV & OTV-T	4. Round Trip With Lunar LO_2
OTV Path	LEO → GEO Left Dry at GEO	LEO → GEO Left Dry at GEO	LEO → GEO → LEO Returns Dry to LEO	LEO → GEO → LEO Returns Dry to LEO	LEO → GEO → LEO Refuel with 3.2 T Lunar LO_2 in GEO, Returns Dry to LEO
OTV-Tanker (OTV-T) Path	LEO → GEO Left Full at GEO	LEO → GEO Left Full at GEO	LEO → GEO Left Full at GEO	LEO → GEO → LEO Returns Dry to LEO	LEO → GEO → LEO Returns Dry to LEO
OTV-Tanker Dry Wt.	0.7	0.7	0.4	0.4	0.8
Net Payload Wt. Delivered to GEO by OTV & OTV-Tanker	13.8	13.8	8.9	8.5	13.2
Total Expendable Vehicle Wt. Delivered to LEO by STS	5.8 + 0.7	0.0[a]	0.4	0.0	0.0
Total Propellant Wt. Delivered to LEO by STS	32.5	32.5	32.5	32.5	32.5 + 0.5(LH_2)
Total Wt. Delivered to LEO by STS	52.8	46.3	41.8	41.0	46.2
MR = $\frac{(\text{Net Payload Wt. at GEO})}{(\text{Total Wt. Delivered by STS})}$	0.261	0.298	0.213	0.207	0.286

[a] Assume OTV and OTV-Tanker returned to LEO at a later date.

Notes: OTV Dry Wt. = 5.8 T, Centaur RL10-11 Engines (I_{sp} = 460 s, O/F = 6) used on OTV.
Aerobraking to LEO saves 2.3 km/s in ΔV.

7.2.5 Delivery of Earth-Supplied Materials from LEO to the Lunar Surface

Several scenarios have been examined for the transport of Earth-supplied materials from LEO to the lunar surface. The first uses contemporary (Centaur) vehicles, the others use the baseline OTV and LTV described above.

7.2.5.1. Transport Using Contemporary (Centaur) Vehicles. An estimate was made of the mass of initial "factory" or exploration equipment that could be delivered to the Moon using current propulsion systems. A single Space Shuttle (STS) launch was assumed, with a total STS payload of 29.5 T. A wide body Centaur (WBC) with added landing gear was used as the single stage for the total mission. No attempt was made to assess weight or performance penalties for additional insulation or refrigeration for the cryogenic propellants, or for propellant boil-off. Based on Centaur performance (Ref. 7-1) (specific impulse = 445 sec) and ΔV's given above, the results are as follows:

"Factory" Payload Wt.	3.5 T
Centaur (WBC) Burn-Out Wt.	2.9
Landing Gear (5% Landing Wt.)	0.4
Landing Wt.	6.8
Expendables	
Main Propellant	17.9 (off-loaded from 20.0)
Others (Attitude Control, etc.)	0.3
Total Vehicle Wt.	25.0
Total Airborne Support Equipment (ASE) Wt.	4.5
Launch Wt.	29.5 T

This represents only a rough estimate, although it does indicate that about a 3.5-T payload could be delivered to the lunar surface for one STS launch. Note that if the "factory" could make H_2/O_2 propellant, it could refuel the WBC and return itself, or a sample return capsule, to Earth.

7.2.5.2 Transport Using the Baseline OTV and LTV. In a more general case, we can consider direct dedicated transport of terrestrial-supplied material from LEO to the lunar surface using vehicles more representative of those in use 20 years from now. Such transport of terrestrial material could be required, for example, for the initial lunar factory set-up, for periodic supply/repair trips, or for delivery of additional lunar-factory modules to the lunar surface. For convenience, the same baseline OTV (at LEO) and LTV (at LLO) described above were used to generate the results shown in Table 7-5.

Three cases were considered to assess the effect of system "closure" of propellant needs. In this context, a completely "open" system would use only terrestrial propellant; a completely "closed" system would use only lunar propellant. Even if lunar LH_2 is not available, the system can be "closed" considerably by the use of lunar LO_2. In the first example shown in Table 7-5, corresponding to a completely "open" system, all the propellant and supplies come from terrestrial sources. Multiple OTV trips are needed to fill the large LTV tanker to capacity, resulting in a ratio of 1 T of terrestrial material delivered to the lunar surface for roughly every 7.3 T of terrestrial propellant and supplies initially in LEO. Increasing the system "closure" so that lunar oxygen is used for all but the OTV LEO-to-LLO step results in a ratio of 1 T of terrestrial material delivered to the lunar surface for every 4.6 T of terrestrial materials initially in LEO. Finally, when only terrestrial LH_2 and lunar LO_2 are used, the ratio drops to 1/3.0. As before, this increased "closure" can only be achieved at the expense of increased lunar material production. In the intermediate case (lunar plus terrestrial propellant LO_2), every 1 T of terrestrial supplies must produce 2.5 T of lunar LO_2 for propulsion, whereas the third case (all-lunar LO_2) requires 12.1 T of lunar LO_2 per 1 T of imported supplies. The trade-offs between increased "closure" and increased lunar propellant production are discussed in more detail in Section 8. In any event, lunar manufacturing processes will need to be highly efficient in terms of requiring terrestrial imports, first because it takes large amounts of terrestrial propellant to transport Earth-supplied material to the lunar surface, and second, because the imported material must generate a large amount of lunar material, such as LO_2, for propulsion.

7.2.5.3 Transport Using the Baseline OTV and LTV for Lunar and Terrestrial Materials. As a final scenario, we consider the delivery of terrestrial materials to the Moon and lunar materials to LEO. For example, in Table 7-2 we see that for lunar material deliveries to LEO it is possible to deliver 0.9 T of terrestrial equipment or supplies to the Moon. This would require 147.8 T of lunar LO_2 production (assuming that the 39.0 T of net payload at LEO is LO_2) so that every 1 T of Earth-supplied materials must produce 164.2 T of lunar LO_2. This can be accomplished with only an 8% reduction in MPBR, but only for a process that requires only 0.6% by weight of product as imported chemical. The situation for GEO delivery of lunar material shown in Table 7-3 would allow delivery of 2.4 T of Earth-supplied equipment or supplies to the Moon for a total lunar LO_2 production of 112.7 T, or 1 T of terrestrial material per 47.0 T of LO_2. This would result in a 13% reduction in MPBR, but the production process would consume 2.1 % by weight of product of Earth-supplied consumables. The impact of production processes requiring Earth-supplied materials is discussed in more detail in Section 8, although these two examples show that any lunar factory will have to be highly efficient.

Table 7-5. Direct Delivery of Terrestrial Material from LEO to the Lunar Surface (All weights in metric tons: T = 1000 kg)

Scenario[a]	All Terrestrial Propellant[b] (Terrestrial LH_2 + LO_2)	Intermediate[c] (Terrestrial LH_2 + LO_2 + Lunar LO_2)	Minimum Terrestrial Propellant (Terrestrial LH_2 + Lunar LO_2)
Terrestrial LH_2 for Propulsion	89.1	8.8	12.6
Terrestrial LO_2 for Propulsion	536.4	27.9	0
Terrestrial Material (Supplies) Delivered to Lunar Surface	100.0	10.1	6.3
Total Terrestrial Mass (LH_2+LO_2+Supplies)	725.5	46.8	18.9
Lunar LO_2 for Propulsion	0	25.0	76.3
$\frac{\text{Supplies Mass}}{\text{Total Terrestrial Mass}}$	$\frac{1}{7.255}$	$\frac{1}{4.634}$	$\frac{1}{3.000}$
$\frac{\text{Lunar } LO_2 \text{ for Propulsion}}{\text{Supplies Mass}}$	0	2.475	12.111

[a] OTV and Tanker dry wt. = 9.3 T; LTV and Tanker dry wt. = 16.5 T; I_{sp} = 460 s, O/F = 6.

[b] Requires 15.5 OTV round trips for every LTV ascent/descent.

[c] Terrestrial LO_2 used for OTV LEO-to-LLO step; lunar LO_2 used for all other steps.

7.2.6 Impact of Lunar Hydrogen Availability on Chemical Transporation Systems' MPBR

As described in Section 9, low concentrations of hydrogen are present in the lunar soil; however, the concentrations are too low to supply the needs of large spacecraft propulsion systems. Thus, it has been assumed that a source of hydrogen for propulsion would be unavailable on the Moon. This in turn forces the use of Earth-supplied LH_2 which then becomes the denominator in the MPBR expression:

$$\text{MPBR} = \frac{\text{Net Mass of Lunar Material Delivered to LEO}}{\text{Mass of Earth-Supplied Material} = LH_2}$$

Any reduction in the amount of Earth LH_2 needed would thus strongly affect the MPBR for chemical propulsion systems. If sufficient quantities of lunar LH_2 were available, an infinite MPBR could be achieved. Finally, note that in the baseline scenarios some Earth-supplied LO_2 is needed for GEO deliveries using chemical propulsion systems so the denominator in the MPBR expression would include both terrestrial LH_2 and LO_2.

7.2.6.1 Terrestrial LH_2 Plus Lunar LH_2 and LO_2. Under the constraint of no lunar LH_2, the baseline chemical OTV and LTV can achieve MPBR's of about 2 to LEO and 1 to GEO. Direct delivery form LEO to GEO using a round-trip OTV has a MR of about 0.2, so the effective GEO MPBR is about 4. If lunar LH_2 were available, the MPBR would increase, the amount of increase depending on how much Earth-supplied LH_2 was needed. Figure 7-1 shows the results of varying the amount of Earth-supplied LH_2 and the resultant MPBR for the baseline H_2/O_2 chemical OTV described above; the baseline scenarios correspond to the lower right of the figure.

If we assume that the LTV requires Earth-supplied LH_2 for its descent from LLO to the lunar surface (1.0 T LH_2 for a 16.5 T dry weight LTV) but has lunar LH_2 available for lift-off (about 7 T), the corresponding MPBR's are about 6, 1.6 and 7 for LEO, GEO and effective-GEO deliveries.

If we further assume a "black box" LTV that requires no Earth-supplied propellant (e.g., mass driver, etc.) so that Earth-supplied LH_2 is needed only for the total OTV round trip (2.7 T + 4.6 T), MPBR's are about 7, 1.7 and 8 for LEO, GEO, and effective-GEO deliveries.

Finally, if only enough terrestrial LH_2 is needed for the outbound LEO-to-LLO steps (1.8 T), the MPBR is almost 31 for LEO deliveries and 3.2 for GEO delivery, or effectively 16 compared to direct GEO delivery. These are MPBR's that would make lunar material return, using chemical propulsion, very promising.

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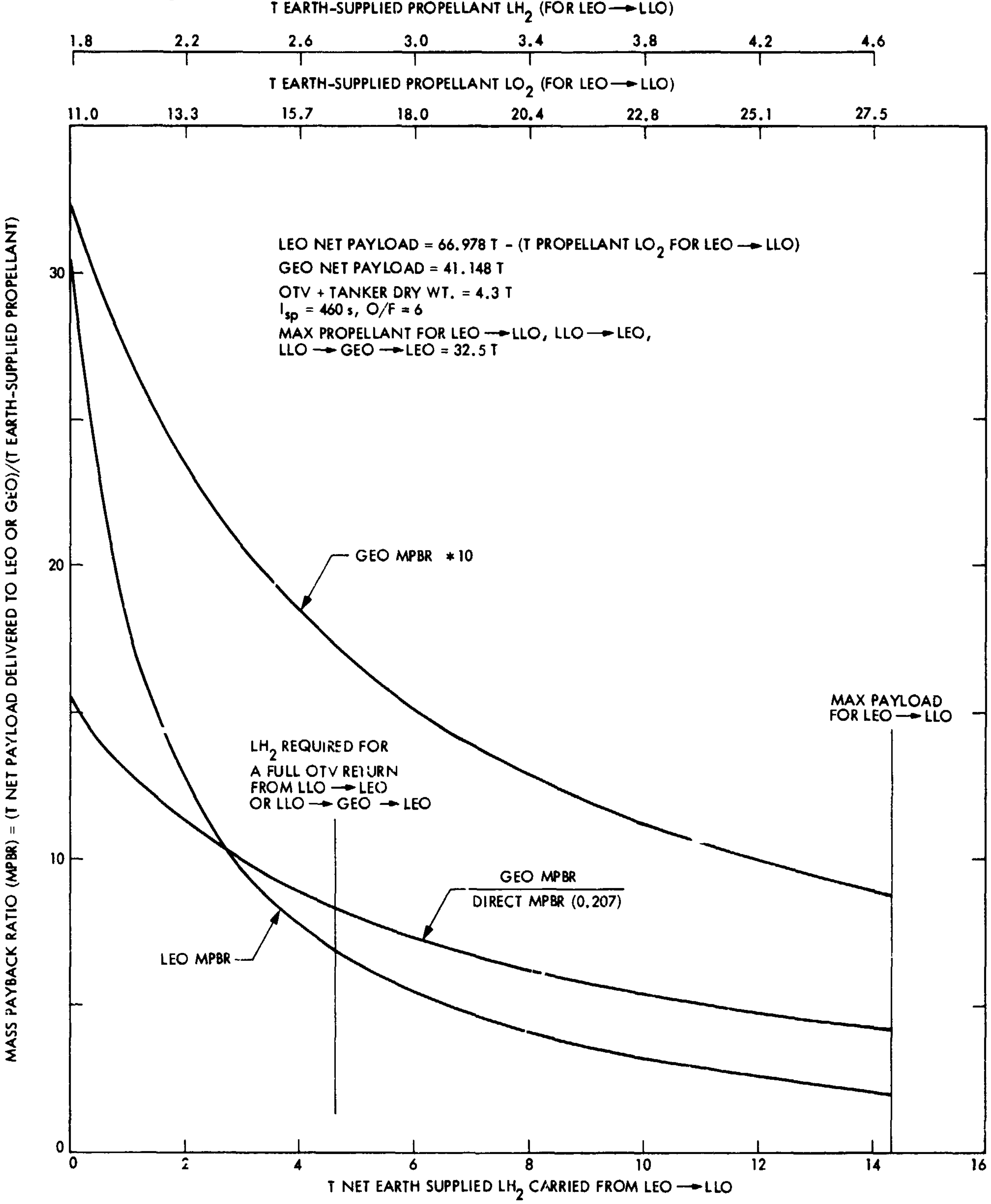


Figure 7-1. MPBR for Various Amounts of Earth-Supplied LH_2 + LO_2

A similar set of options is shown in Table 7-6. In these scenarios, the operation of the OTV and LTV are totally separate. The first example uses a "black box" OTV (e.g., solar sail) that requires no propellants, with a chemical LTV using Earth-supplied LH_2 and lunar LO_2. For a baseline 16.5-T and hypothetical zero-weight LTV, the MPBR's are about 11 and 15. Of course, a zero-weight vehicle is not possible, but future large vehicles and systems might approach this limit for vehicle weights that are a small percentage of the payload and propellant weights, corresponding to stage mass fractions of nearly one. Since the OTV uses no propellant, the MPBR's are the same for LEO or GEO delivery. If a solar sail OTV were actually used, air drag at LEO and the fivefold increase in effective GEO delivery MPBR would favor GEO deliveries.

For a "black box" LTV (e.g., lunar mass driver) and chemical OTV using Earth-supplied LH_2 and lunar LO_2, the MPBR's are less than those found above, since the OTV must provide greater ΔV than the LTV. For LEO delivery, a baseline 9.3-T OTV and a zero-weight OTV give MPBR's of about 7 and 13, showing that LEO delivery can become attractive in some cases. For GEO delivery, the corresponding 9.3-T and zero-weight OTV MPBR's are 1.7 and 10, or effectively 8 and 49. Finally, for a zero-weight OTV, it is possible to return lunar LO_2 to LEO after a GEO delivery, so as to improve the MPBR by eliminating the need for any terrestrial LO_2. This results in an MPBR of about 13, or effectively 61, for GEO delivery.

7.2.6.2 All-Lunar Propellant Scenario. In all the previous discussions, we have assumed that no lunar hydrogen would be available. If lunar hydrogen were available, an infinite MPBR would be achieved. We now consider how much lunar hydrogen and oxygen would be required to deliver lunar material to LEO. As before, a baseline OTV and LTV weighing 9.3 and 16.5 T respectively are used. In addition, a zero-weight pair of vehicles are also used to assess the limit case. The results are summarized in Table 7-7.

In this all-lunar propellant scenario, the LTV lifts off from the lunar surface carrying enough lunar propellant for its return to the surface, all the lunar propellant needed for the OTV round trip, and the payload destined for LEO. The OTV is fueled in LLO and the payload and propellant for the LEO-to-LLO step loaded. The OTV transfers to LEO, deposits its payload, and is refuled with the lunar propellant it has carried from LLO. The OTV returns to LLO empty to await the next LTV delivery.

As shown in Table 7-7, the most surprising result of this analysis is that it requires less all-lunar propellants to deliver a given amount of lunar material payload to LEO than it does for the baseline case using terrestrial LH_2 and lunar LO_2. This is primarily due to the large LEO-to-LLO ΔV that the OTV must undergo. This in turn forces the use of considerable propellant when the OTV is loaded with terrestrial LH_2 for delivery to LLO. The LLO-to-LEO step involves a lower ΔV thanks to aerobraking, so the penalty associated with carrying propellant from LLO to LEO is not as severe as the reverse step.

Table 7-6. MPBR's for Separate OTV-LTV with One Vehicle Using H_2 (Earth) and O_2 (Moon) and the Other Vehicle Not Using Propellant (All weights in metric tons: T = 1000 kg)

H_2/O_2 Vehicle	LTV		OTV		OTV		
Vehicle Dry Wt.	16.5	0	9.3	0	9.3	0	0
Delivery Point	- (LLO) -		- LEO -		- GEO -		
Total Earth LH_2	8.8	7.7	7.3	5.5	7.3	4.9	4.9
Total Earth LO_2 for OTV	-	-	0	0	16.4	1.7	0
Total Earth Mass	8.8	7.7	7.3	5.5	23.7	6.6	4.9
Total Lunar LO_2 Propellant	52.6	46.6	44.3	33.3	27.9	27.9	29.6
Net Delivered LO_2 & Payload	94.8	117.3	50.5	70.8	41.2	66.3	61.8
Total Lunar LO_2 & Payload + Propellant	147.4	163.9	94.8	104.1	69.1	94.2	91.4
Lunar LO_2 at LEO for C .	-	-	16.4	5.4	0	0	1.7
MPBR = $\frac{\text{(Net Delivered Lunar Mass)}}{\text{(Total Earth Propellant Mass)}}$	10.773	15.234	6.918	12.873	1.738	10.045	12.612
$\frac{\text{(Total Lunar } LO_2 \text{ \& Payload + Propellant Mass)}}{\text{(Net Delivered Lunar Mass)}}$	1.555	1.397	1.877	1.470	1.677	1.421	1.479
"Effective" MPBR = $\frac{\text{MPBR Lunar}}{\text{MR Direct (0.207)}}$	-	-	-	-	8.396	48.527	60.928

Notes: H_2/O_2 Engines: I_{sp} = 460 s, O/F = 6.
Examples of vehicles using no propellants would be a Mass Driver Launcher (LTV) or Solar Sail (OTV). Zero-weight vehicle limit could be approached for large systems in which dry weight is a small fraction of the payload and propellant weights.

Table 7-7. Lunar Material Delivered to LEO Using All-Lunar Propellants (All weights in metric tons: T = 1000 kg)

	Baseline Vehicles	Zero-Weight Vehicles
OTV Dry-Wt.	9.3	0
LTV Dry-Wt.	16.5	0
LEO-to-LLO:		
OTV Propellant	12.8	0.0
LLO-to-LEO w/AB:		
OTV Propellant	32.5	32.5
Net Payload at LEO	54.1	76.2
Moon-to-LLO:		
LTV Propellant	52.4	46.4
LLO-to-Moon:		
LTV Propellant	7.0	0.0
Total Propellant	104.7	78.9
Propellant LH_2	15.0	11.3
Propellant LO_2	89.7	67.6
$\frac{\text{Total Propellant}}{\text{Net Payload at LEO}}$	1.935	1.035
Minimum H_2O needed for propellant	135.0	101.7
Excess O_2 (from H_2O) that could be used as payload to LEO	30.3	22.8

(continued)

Table 7-7. Lunar Material Delivered to LEO Using All-Lunar Propellants
(All weights in metric tons: T = 1000 kg) (Continued)

	Baseline Vehicles	Zero-Weight Vehicles
Baseline scenario using terrestrial LH_2 and lunar LO_2:		
Total Terrestrial Propellant LH_2	17.9	13.8
Total Lunar Propellant LO_2	107.2	82.7
Net Payload at LEO	40.2	62.7
$\frac{\text{Total Lunar } LO_2}{\text{Net Payload at LEO}}$	2.667	1.319

NOTES: Centaur RL10-II Engines (I_{sp} = 460 s, O/F = 6) used on OTV and LTV.

Aerobraking (AB) to LEO saves 2.3 km/s in ΔV.

The other noteworthy fact is that if lunar polar water-ice is electrolyzed to produce propellant for an O/F = 6 propulsion system, the hydrogen is the limiting reagent and oxygen in excess of that needed for propulsion is produced. This would not occur if an O/F = 8 engine were used since this matches the stoichiometry of the water molecule. When water is used for propellant, it requires only 2.5 T or 1.3 T of water per 1 T of delivered payload for the baseline and zero-weight vehicle sets.

Thus we see that the early discovery of a source of lunar hydrogen could have a significant effect on operation of a lunar materials program. It would result in commercially attractive MPBR's (>10) using near-term low-cost chemical propulsion systems. The option would also exist for gradually phasing in more advanced electric or nuclear-thermal propulsion systems (described below) using lunar LH_2 for even greater MPBR's, with the potential for eliminating the need for any Earth-supplied propellants.

7.3 LUNAR MATERIAL TRANSPORTATION USING VEHICLES WITH ELECTRIC PROPULSION SYSTEMS

Low T/W electric propulsion transportation systems can only be used between LEO and LLO and not from LLO to the lunar surface. In this analysis, it is assumed that a H_2/O_2 chemical LTV is used to ferry lunar-material products from the lunar surface to LLO where the lunar material is transferred to the electric propulsion OTV for the trip to LEO or GEO. Note that a low T/W results in a higher required ΔV (see Table 7-1) due to gravity losses, but this is countered by the high I_{sp} of electric propulsion systems. Additionally, electric vehicles require long trip times which can have a strong effect on delivery schedules and fleet size. Nevertheless, the high specific impulse and potential for use of lunar LO_2 as propellant make electric propulsion systems very attractive. They represent a near-term advanced propulsion system requiring only moderate development that, like chemical OTV's, may be supported and driven by the needs of LEO-to-GEO satellite traffic.

7.3.1 Operational Scenario

Essentially the same operational scenario described for the all-chemical system is used, with the electric propulsion OTV used in place of the chemically propelled OTV. The electric OTV is fueled in LEO with Earth-supplied Ar as propellant. Earth-supplied LH_2 is also loaded for delivery to LLO for use by the chemical LTV. The OTV transfers to LLO and docks with the LTV. Earth-supplied LH_2 is loaded in the LTV and lunar LO_2 (or any other lunar-material payload) into the OTV. The OTV returns to LEO and the LTV to the lunar surface. Since advanced electric propulsion systems may be able to use oxygen as propellant, the use of lunar O_2 rather than terrestrial Ar is also considered. In this case, the lunar O_2 propellant for the electric OTV would be provided in LLO. Finally, note that aerobraking cannot be used by a vehicle with a low T/W propulsion system since a high T/W impulsive burn is required to raise and circularize the orbit after aerobraking.

7.3.2 MPBR Calculations Approach

In the analyses of chemical OTV performance, a specific vehicle point design was used. In this section, no single electric propulsion system will be assumed for the transportation vehicle; rather, electric propulsion technology level will be used as a variable upon which the figures-of-merit, MPBR and round-trip time, will be dependent. The electric propulsion technology will be characterized by the total power and propulsion specific mass (α,kg/W_e). The specific mass is the combined mass of the electric power supply and the electric propulsion system divided by the power supply power (P). The current state-of-the-art solar electric propulsion (SEP) system has a specific mass of about 0.04 kg/W_e. For the purposes of this analysis we have assumed that specific mass shall range from 0.03 to 0.01 kg/W_e. Both nuclear electric and solar electric power supply projections fall into this range. Power level and system specific impulse shall be free to vary in order to obtain high mass ratios and low trip times. Thruster efficiency shall be dependent on specific impulse as is projected for 50- to 100-cm Ar ion engines as is shown in Figure 7-2.

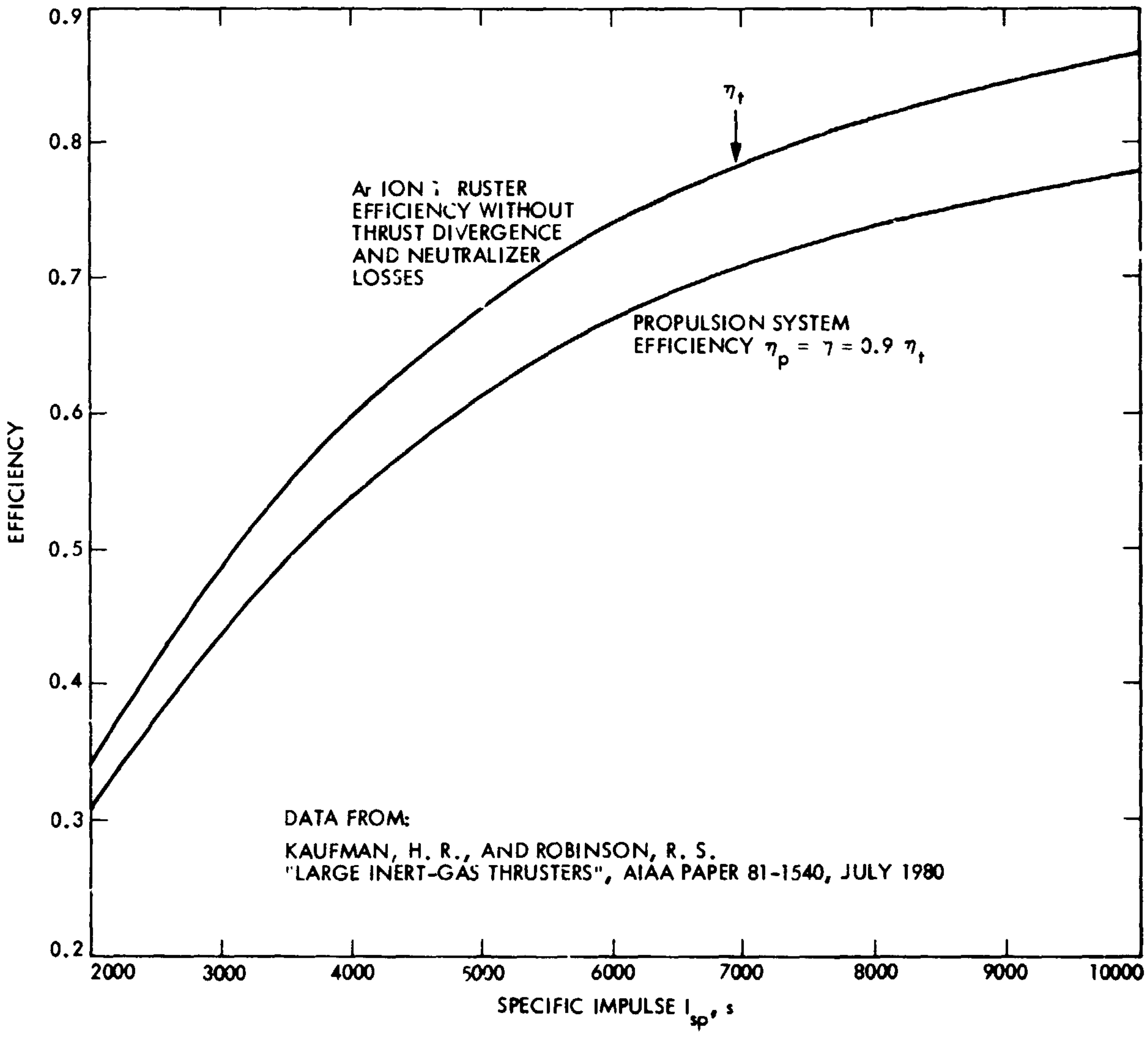
From Table 7-2, we see that a baseline chemical LTV can deliver 94.8 T of product to LLO. This LTV requires 8.8 T of hydrogen in order to complete the round trip from the lunar surface to LLO and back. Therefore, for the electric propulsion vehicle, its payload from LLO to Earth is 94.8 T of lunar O_2 or other product, and its payload from LEO to LLO is 8.8 T of terrestrial LH_2. All the oxygen required for the LTV is assumed to be produced on the Moon. Two delivery locations are considered, LEO and GEO. For each delivery location, two electric propulsion transportation scenarios are considered: 1) all the electric propulsion propellant comes from the Earth (terrestrial Ar propellant), or 2) all the electric propulsion propellant comes from the Moon (lunar LO_2 propellant). For the terrestrial Ar propellant scenario, the MPBR is:

$$\text{MPBR} = \frac{\text{94.8T Net Delivered Lunar Product}}{\text{8.8T Terrestrial } LH_2 + \text{Terrestrial Ar}} \qquad (7\text{-}6)$$

For lunar LO_2 propellant the mass ratio is:

$$\frac{\text{MPBR} = 94.\text{[illegible]} - \text{Lunar } LO_2 \text{ Electric Propulsion Propellant Mass}}{8.8 \text{ T Terrestrial } LH_2} \qquad (7\text{-}7)$$

Electric propulsion propellants traditionally are the inert gases (Ar, Xe, Ne, He), mercury or hydrogen. Mercury will not be considered here due to the possible objections to its use near the Earth's atmosphere. The terrestrial propellant of choice is Ar. The most obvious lunar propellant is O_2. Oxygen electric thrusters will require applied research and substantial technology development. Lunar hydrogen (if available) would make an excellent propellant for plasma engines. Lunar oxygen will be the lunar propellant assumed for this analysis since it represents a propellant whose presence on the Moon would not require a major new discovery, as would a source of lunar hydrogen.



DATA FROM:

KAUFMAN, H. R., AND ROBINSON, R. S.
"LARGE INERT-GAS THRUSTERS", AIAA PAPER 81-1540, JULY 1980

Figure 7-2. Ar Ion Thruster Efficiency vs. Specific Impulse

As shown in Equations (7-6) and (7-7), one needs the propellant masses (M_P) in order to calculate the mass ratios. Propellant masses are easily calculated from the rocket equation as follows:

$$M_P = M_0 \left[1 - \exp\left(\frac{-\Delta V}{g I_{sp}} \right) \right] \qquad (7\text{-}8)$$

where M_0 is the vehicle initial mass, ΔV is the mission velocity requirement, and I_{sp} the propulsion system specific impulse. The mission ΔV's are given in Table 7-1.

The initial mass of any vehicle is made up of its payload (M_{PL}), propellant (M_P), propellant tanks, and power and propulsion system (M_{PP}) as follows:

$$M_0 = M_{PL} + (1 + TF)\, M_P + M_{PP} \qquad (7\text{-}9)$$

TF is the tankage fraction, assumed to be 5% for LO_2 or Ar. Equations (7-8) and (7-9) can be used to eliminate M_0 and give one equation for the propellant mass as a function of ΔV, I_{sp}, M_{PL}, and $M_{PP} = \alpha_{pp}$*Power. The initial mass of a vehicle leaving LEO for LLO using terrestrial Ar propellant includes the propellant for the LEO-to-LLO leg plus the return (LLO to LEO) Ar propellant.

Once the propellant mass is calculated, the trip time (T) may be obtained by calculating the propellant flow rate ($\dot{m}$) and dividing it into the propellant mass. This procedure assumes that the propulsion system acts continuously (no coasting) and that loading and unloading operations are short (one to five days). The propellant flow rate is calculated as follows:

$$T = \dot{m}\, I_{sp}\, g \qquad (7\text{-}10)$$

$$\eta = \frac{T\, I_{sp}\, g}{2P} = \frac{\dot{m}\, (I_{sp}\, g)^2}{2P} \qquad (7\text{-}11)$$

$$\dot{m} = \frac{2\eta P}{(g I_{sp})^2} \qquad (7\text{-}12)$$

In Eq. (7-11), η is the propulsion system efficiency of converting power supply electric power into thrust power and P is the power supply electric power. The trip time is then given by Eq. (7-13).

$$\text{Trip Time (sec)} = \frac{\text{Electric Propulsion Propellant Mass}}{\dfrac{2\eta P}{(g I_{sp})^2}} \qquad (7\text{-}13)$$

7.3.3 MPBR Results for Return of Lunar Material to LEO or GEO

An example of the general payback system behavior is shown in Figure 7-3 where both figures-of-merit, round-trip time and Mass Payback Ratio are presented as a function of system power with system specific impulse (I_{sp}) and specific mass (α) as parameters. At low power levels, the mass ratio is asymptotic to a moderately high but finite value. The trip time approaches infinity at low power levels. At large power levels the ratio goes to zero and the trip time is asymptotic to a moderately low value. A reasonable operation point is at the power level where trip time is acceptable, i.e., <300 days. This power level is a function of the assumed payload mass (94.8 T) and will change for differing payloads. At this power level one desires as large a MPBR as possible, dependent on the technology level (α) and the specific impulse. In general the dependence of mass ratio and trip time is as follows and vice versa:

Lower specific impulse (I_{sp}) <=> shorter trip time, smaller MPBR.

Lower system efficiency (η) <=> longer time; same MPBR.

Lower specific mass (α) <=> shorter time; larger MPBR.

Lower system power (P) <=> longer time; larger MPBR.

Lower mission energy (ΔV) <=> shorter time; larger MPBR.

Table 7-8 presents the results for all scenarios with round-trip times less than about 1 year for two different technology levels (α = 0.03 and 0.01 kg/W_e). It should be pointed out that in some cases the trip time can double from 1 to 2 years and the Mass Payback Ratio increase by more than a factor of two.

The results presented in Table 7-8 show MPBR's only slightly larger than 1 for electric propulsion systems using terrestrial propellant and a relatively low amount of required near-term technology (α = 0.03); (α = 0.01) raises the MPBR to over 4 (for GEO delivery) and decreases round trip times to a little less than 200 days. Electric propulsion systems using lunar propellant approximately double all the MPBR's up to about 8 for LEO delivery and high technology, while reducing the trip times slightly. The power level for all interesting cases falls in the range between about 5 and 10 MW_e. This power level is many orders of magnitude larger than anything used in space to date. However, just for the operation of the lunar factory, power levels on the order of megawatts may well be required and therefore possibly available for electric propulsion transportation systems.

We can also compare delivery of lunar material to GEO with direct delivery of Earth-supplied material from LEO to GEO using electric propulsion systems, as was done for chemical systems. In this case, a near-term (α = 0.03) electric vehicle could deliver about five times more payload than the propellant required to deliver it so that the direct-GEO MR is about 0.8. Higher technology (α = 0.01) could achieve a MR of around 0.9, significantly higher than a near-term chemical OTV system. With these comparisons, the "effective" MPBR's for delivery of lunar material to GEO range from about 2 to 5 for terrestrial Ar propellant electric vehicles, and 6 to 11 when lunar LO_2 propellant is used by the electric vehicle, with a chemical LTV used in all cases.

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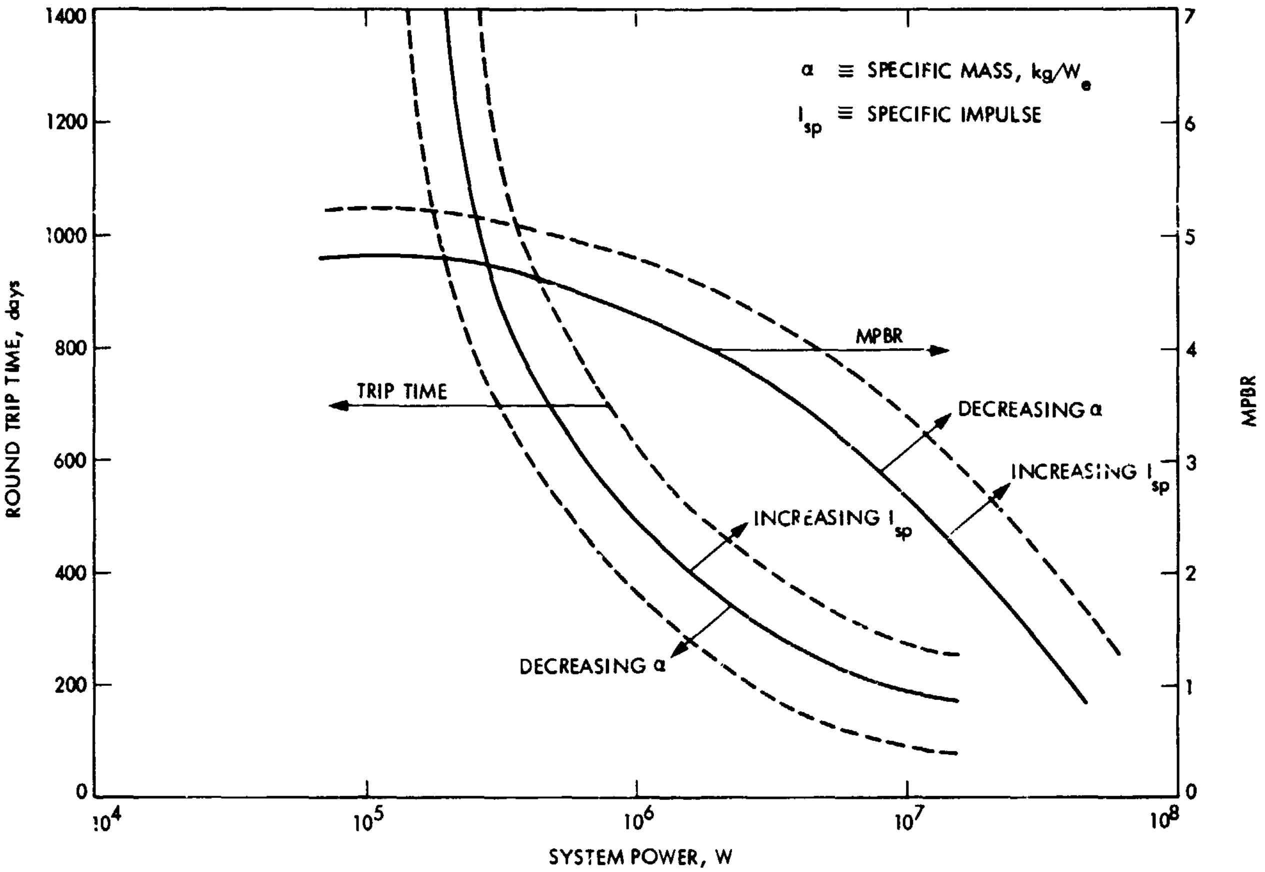

Figure 7-3. Trip Time and MPBR vs. System Power for Electric Propulsion

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Table 7-8 Transport of Lunar Material to LEO and GEO Using an Electric Propulsion OTV
(All weights in metric tons: T = 1000 kg)

Electric Propulsion	Terrestrial Electric Propellant (Ar)				Lunar Electric Propellant (O_2)				Direct Electric to GEO		Direct Chemical to GEO
Delivery Location OTV Path	LEO LEO → LLO → LEO		GEO LEO → LLO → GEO → LEO		LEO LEO → LLO → LEO		GEO LEO → LLO → GEO → LEO		GEO LEO → GEO → LEO		GEO LEO → GEO → LEO
Technology Development Required	α = 0.03 kg/W_e	α = 0.01 kg/W_e									I_{sp} = 460 s O/F = 6
	Low	High	Low	High	Low	High	Low	High	Low	High	Low
MPBR = $\frac{\text{Net Delivered Lunar Payload}}{\text{Earth-Supplied Propellant}}$	1.1	2.7	1.6	4.7	1.7	8.0	5.1	9.5			
Trip Time (Days)	390	197	313	171	390	176	300	170			
Direct GEO MR = $\frac{\text{Net Delivered Earth Payload}}{\text{Earth-Supplied Propellant \& Payload}}$									0.82	0.86	0.21
Trip Time (Days)									428	317	<1
Effective GEO MPBR = $\frac{\text{Lunar Delivery to GEO MPBR}}{\text{Chemical Direct to GEO MR}}$			7.6	22.5			24.8	45.9			
Effective GEO MPBR = $\frac{\text{Lunar Delivery to GEO MPBR}}{\text{Electric Direct to GEO MR}}$			1.9	5.4			6.3	11.0			

Note: Chemical H_2/O_2 LTV used which required terrestrial LH_2.

When compared to the previous analysis for chemical propulsion transportation systems, this analysis of electric propulsion transportation systems shows significant differences. Chemical systems require the development of a new technology (aerobraking) in order to achieve even relatively low ratios (approximately 2) for LEO delivery. Relatively modest technology development is required for electric propulsion systems to provide the same ratio, provided the electric power supply is developed for the lunar facility for either option. In order for chemical systems to provide ratios of 8 or more, the development of aerobraking is required plus the speculative assumption that significant amounts of hydrogen can be found and made available for propulsion on the Moon. Electric propulsion systems can provide ratios of about 8 for LEO delivery with the assumptions of only advanced power supply technology resulting in power supply specific masses of about 0.005 to 0.008 kg/W_e and the development of electric thrusters using readily produced lunar oxygen. Currently there are projections and goals for solar arrays of achieving a specific mass of 0.0033 kg/W_e and reactor power supplies of achieving about 0.01 kg/W_e or better. Efficient oxygen ion thrusters are speculative at this point, but oxygen plasma thrusters appear feasible from an efficiency/I_{sp} point of view.

Finally, Mass Payback Ratios of infinity (no Earth support) can conceptually be achieved by electric propulsion systems using lunar oxygen propellant and mass drivers on the Moon to eliminate the need for any terrestrial LH_2 for a chemical LTV. This would, however, require a sizable initial mass investment on the lunar surface due to the large size of a mass driver.

However, for all these advantages, there are still some potentially serious problems associated with electric propulsion systems. First, as mentioned earlier, the long trip times would probably result in the need to establish a large fleet of electric OTV's. In a detailed cost analysis, it would be necessary to trade the initial cost of many potentially expensive electric OTV's with high MPBR's to that of a few relatively inexpensive chemical OTV's with lower MPBR's. In any event, some chemical OTV's would be needed for priority missions such as manned repair missions.

It is also possible to calculate an MPBR in which the weight of the delivery system (OTV and LTV dry weights) is included. For example, if an OTV/LTV pair can make 10 trips before being replaced, then one-tenth of the OTV/LTV dry weight would be added to the terrestrial propellant mass in the denominator of the MPBR expression to amortize the vehicle weights. Such calculations are considered in Section 8.

A related question to be investigated next year is how the mass of the delivery system (OTV and LTV) compares to the total mass of lunar product. For example, if the rate of lunar [illegible] delivery to LEO were 1000 T per year, a single baseline chemical OTV/LTV pair weighing 25.8 T could easily make the required delivery schedule since they have a one-week round trip time and can deliver about 40 T of lunar material per trip. Thus, a chemical delivery system would only weigh a few percent of the mass of lunar material delivered to LEO. By contrast, a delivery system using an electric OTV (and chemical LTV) capable of delivering around 100 T of lunar material with a one-year trip time would weigh on the order of 50 T. It would thus require ten electric OTV's (and one chemical LTV) to deliver 1000 T per year to LEO. In this case, the mass of the delivery system would be of the same order as the total mass of lunar material delivered to LEO. How these types of considerations may affect the utilization analyses is briefly discussed in Section 8 and will be considered in more detail next year.

Another question involves operations of the electric OTV. A solar electric propulsion (SEP) system cannot be used for repeated LEO-GEO or LEO-LLO trips due to degradation of the solar cells by the Van Allen radiation belts. Also, SEP systems typically are used to supply lower power levels than the 5 to 10 MW_e needed for the OTV's described in these analyses.

This leaves nuclear electric propulsion (NEP) systems using a nuclear reactor as the power supply. Space-based nuclear reactor systems with the required power levels are likely to be developed for Earth-orbital or lunar facilities, but the routine servicing of "hot" NEP OTV's may pose serious challenges for LEO, GEO or LLO orbital stations. Finally, radiation-sensitive cargos would require shielding, although bulk cargos like LO_2 would not require extensive protection.

Nevertheless, even with these disadvantages, the NEP OTV represents a relatively near-term advanced OTV that promises exceptional performance. And, like the chemical OTV, much of the technology base required for this vehicle is likely to be developed over the next 20 years to support Earth-orbit operations.

7.4 LUNAR MATERIAL TRANSPORTATION USING VEHICLES WITH VERY ADVANCED PROPULSION SYSTEMS

Thus far the emphasis has been on near-term propulsion systems, in part because their operational parameters are best known. In this section, we investigate the use of more far-term (post-year 2000) propulsion systems that potentially could be developed for use with a lunar materials program. However, unlike the previously described chemical and electric propulsion systems, these systems would probably <u>not</u> be developed to support LEO-to-GEO orbital operations since their development would entail considerable cost, although they might be developed for other reasons (e.g., manned Mars missions, etc.). This in turn implies that these advanced systems must show a very high MPBR since both their initial R&D and construction cost would be amortized by the lunar materials economy.

The general approach is the same as that used previously. A vehicle of given weight, I_{sp}, etc., is run through the various steps and an MPBR found. In several cases, the operation of the OTV is emphasized since the OTV must supply the largest ΔV for the total mission.

7.4.1 Advanced Propulsion Systems for Earth Orbit-to-LLO OTV's

Two classes of advanced OTV propulsion systems will be considered here. The first involves nuclear-thermal rocket engines in which nuclear energy is used to directly heat a working fluid like hydrogen. The second is the use of a solar sail OTV which could achieve an infinite MPBR since it requires no propellant for its operation.

7.4.1.1 Nuclear-Thermal Propulsion Systems for OTV's. Table 7-9 lists a group of nuclear-thermal advanced propulsion systems (Ref. 7-5). These systems are assumed to use only Earth-supplied LH_2 or nuclear bomblets (Orion) as propellant. The MPBR's are generally low because they derive all their propellant from Earth. By contrast, an H_2/O_2 chemical system only requires one-seventh of its propellant from the Earth. Only the very advanced nuclear fission bomblet (Orion) or the pulsed fusion systems can achieve a MPBR of 10 or more, and this is only for the LEO-to-LLO-to-LEO steps. A "black box" LTV was assumed that used no Earth propellants for its operation (i.e., mass driver). Needless to say, the political implications of the development and use of an Orion pulsed nuclear bomblet system in LEO would not be trivial. Also, one trip by Orion would deliver a 12.5-year production run of lunar LO_2 (6500 T) if produced at a rate of 10 T/week. Finally, the Orion propellant cost would almost certainly negate the 10-to-1 MPBR advantage.

A hydrogen fusion system could provide a MPBR of over 30, but only for a very large system. To maintain perspective, remember that a 3800 T fusion vehicle in LEO (200 T dry wt., 1400 T LH_2, 2200 T tanker) would require 127 STS launches (at 30 T each) to assemble in LEO. Fusion systems, however, could prove attractive for very large scale operations later in the next century. This is especially true since fusion power-plant R&D over the next 20 years could result in technology breakthroughs that are directly applicable to fusion propulsion systems.

7.4.1.2 Solar Sail OTV's. Solar sails have often been suggested for use as interorbit and interplanetary propulsion systems (Ref. 7-7). Since they use only light pressure for acceleration, and no propellant, they have an essentially infinite I_{sp}. However, they have a very low T/W (<10-4) and, like electric propulsion systems, require very long trip times. Solar sails should be relatively inexpensive to produce in orbit, so large fleets may not prove to be economically unattractive.

As in all these schemes, an LTV would be needed and if a baseline chemical LTV were used, the MPBR would be around 11, assuming the use of terrestrial LH_2. If lunar hydrogen were available, or an electromagnetic launcher used, the MPBR would be infinite. Also, the solar sail cannot operate below an altitude of 1000 km due to air drag. Thus, a separate OTV would be needed to travel from LEO (275-km altitude) to the solar sail's lowest orbital altitude and back. Because of this, the sail would be more suited to GEO-to-LLO transfers.

7.4.2 Lunar Surface to LLO Transportation Systems Not Requiring Hydrogen

In several of the scenarios discussed previously, the need for a high T/W chemical LTV placed serious constraints on the overall transportation system MPBR due to the need for Earth-supplied LH_2. One way to increase the MPBR is to assume the discovery of lunar hydrogen. Assuming this is not possible, the two systems described below could be used to eliminate the need for any terrestrial LH_2 by the LTV.

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Table 7-9. Advanced Nuclear Thermal Propulsion Options for Return of Lunar Material to LEO (All weights in metric tons: T = 1000 kg)

	Advanced Nuclear Thermal Engines[a]							Wt. of Propellant (T) for:			
Type	Name	I_{sp} (s)	Thrust (kN)	Wt. (T)	Propellant	OTV Stage Dry Wt. (T)	Lunar Material Returned to LEO (T)	LEO-to-LLO	LLO-to-LEO	Tanker Dry Wt. (T)	MPBR
Solid Core	SNRE (LASL-α)	875	72.6	2.6	LH_2	4.7	57.8	12.1	13.4	2.9	2.267
Particle Core	Particle Bed NRP	1000	250	3.0	LH_2	5.1	67.0	10.6	13.4	3.3	2.792
Closed-Cycle Gas-Core	Large Light Bulb	2080	445	56.8	LH_2	91.7	1362.4	59.8	124.2	68.1	7.400
Fission Pulse	Orion	2500	3500	93.8	0.01-kT A-Bombs	117.0	6493.1	157.0	468.0	324.7	10.389
Fusion Pulse	(Winterberg)	6320	970	--	LH_2	200	44,400	200	1200	2200	31.7

[a] Ref. 7-6.

Notes: "Black Box" (Mass Driver) Places Lunar Material (LO_2) in LLO.
Aerobraked Return to LEO.

7.4.2.1 Nuclear-Thermal Rocket Using Lunar LO_2. One possible near-term LTV using no Earth-supplied LH_2 would be a nuclear thermal rocket (NRP) using O_2, instead of the more usual H_2, as the working fluid. If we assume chemical OTV delivery to LEO, the LTV would need to place 94.8 T LO_2 in LLO for the OTV (44.3 T for OTV propulsion and 50.5 T for net payload at LEO). To do this an NRP-LTV would be needed with a dry weight of 79.0 T, using five NERVA engines (10.4 T weight, 333kN thrust), and a LO_2 tankage equal to 5% of the total LO_2 weight. Five engines are needed to give an adequate liftoff thrust-to-weight (0.27). Assuming an I_{sp} = 200 s for O_2 (corresponding to an I_{sp} around 800 s for H_2), the following results are obtained:

OTV LO_2	44.3 T	8.207%
NRP LO_2	445.0 T	82.438%
Net LEO Payload LO_2	50.5 T	9.355%
Total LO_2	539.8 T	100.000%

Thus, the LO_2-NRP-LTV alone would consume over 80% of all the LO_2 produced (and the chemical OTV another 10%), leaving less than 10% of the total LO_2 for delivery as payload. Finally, this LTV would still give an MPBR of about 7 when coupled with a baseline chemical OTV using terrestrial LH_2 and lunar LO_2. An electric OTV and NRP-LTV both using O_2 could give an infinite MPBR, although the benefit of such a scheme would depend on the difficulty (cost, etc.) of producing O_2.

7.4.2.2 Electromagnetic Launchers (Mass Driver or Rail Gun). Electromagnetic launchers (mass driver or rail gun) can replace the H_2/O_2 Lunar Transfer Vehicle (LTV) assumed in the previous transportation scenarios. Electromagnetic launchers and lunar oxygen electric propulsion systems can conceptually comprise a transportation system which would require no Earth resources and therefore would have an infinite MPBR. This section contains the results of estimates of the mass and power requirements of a lunar-based electromagnetic launcher.

Note that electromagnetic launchers are large and rather complex systems that would require considerable import of terrestrial materials (electromagnets, power production and storage systems, etc.). In principle, a lunar launcher could be made from all-lunar materials, although this would require a highly sophisticated lunar factory to produce the high-technology components of the launcher. A more promising approach would be to use lunar materials for the less technology-intensive components so as to reduce the mass of terrestrial imports required.

A rail gun or mass driver could conceptually launch lunar mass into a trajectory that would intercept LEO, GEO, or LLO. At each of these locations a mass "catcher" is required. This analysis assumes that the lunar mass is launched into LLO where it accumulates until an OTV transports it to the user location. The assumed requirements on the launcher are; 1) a 10 kg payload, 2) 2000 m/sec exit velocity, and 3) 1000 "g" constant launch acceleration.

The independent variable is $\dot{m}$, i.e. average launch rate in kg/s or T/yr. The required average power is:

$$P = \frac{\dot{m}\ (2000)^2}{2\eta}$$

where η is the total efficiency of the launcher. Figure 7-4 shows this average power versus launch rate. The bounds on the power are a high efficiency of 96% characteristic of a mass driver and a lower efficiency of 60% characteristic of a rail gun, although rail gun efficiencies have been projected to be over 90%. The upper bound on launcher mass is a mass driver system as characterized by O'Neill et al. (Ref. 7-8). The lower bound (a factor of 2 lower) is the rail gun estimated mass. Both are a weak function of $\dot{m}$ until $\dot{m}$ is about 10,000 T/yr. The mass driver is more complicated than the rail gun, needing a sophisticated "bucket" for the payload and a deceleration region for bucket retrieval. Because of this deceleration region, the mass driver is about 350 m in length while the rail gun is only 200 m.

A high capacity energy storage system would be needed to accumulate energy between launches. For a delivery rate of about 10 T of lunar material per week (500 T/yr), the average power would be around 40 kW_e, with a launch rate of about one launch every 10.5 minutes.

Such a system could only be used for insensitive bulk cargos due to the high acceleration. Low "g" systems are prohibitively large. For example, a 10 "g" launcher designed to launch 1 T payloads to LLO would be about 20 km in length and have a mass in the 50,000 to 100,000 T range. Finally, some sort of lunar lander would still be required to return empty LO_2 tanks, supplies, etc. to the lunar surface.

One novel approach for using electromagnetic launchers for LTV propulsion would be to fire dust pellets from the lunar surface so as to impact the LTV (Ref. 7-9). The momentum of the dust would be transferred to the LTV (through a shock absorber plate or similar scheme) and the dust would then fall back to the lunar surface. This could be used for landing or takeoff with only a small chemical engine needed on the LTV for final orbital insertion (and emergency back-up in case of lunar launcher failure). However, the peak power required would be very high for launching large LTV's (around 1000 MW for the chemical LTV described previously) and so this method might be restricted to landing small payloads on the lunar surface. This does show, however, the types of novel applications that might be used on the Moon, asteroids (Ref. 7-10), or other airless bodies with low orbital or escape ΔV's.

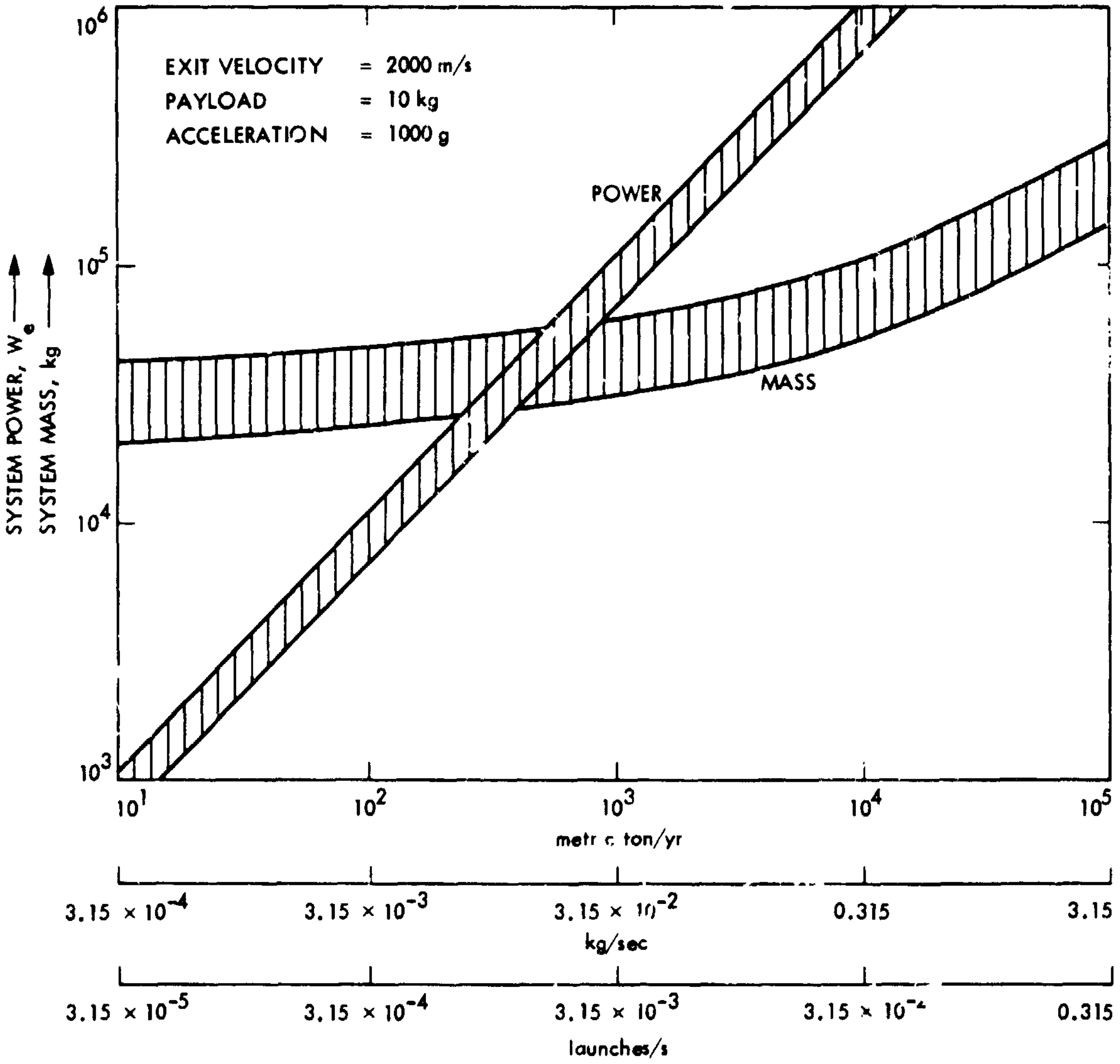

Figure 7-4. Lunar Electromagnetic Launcher Mass and Required Power

7.5 ASTEROID MATERIAL TRANSPORTATION USING VEHICLES WITH CHEMICAL PROPULSION SYSTEMS

This section describes a brief parametric analysis of the Mass Payback Ratio (MPBR) for return of asteroid material to LEO. As in previous analyses we have assumed the use of a baseline H_2/O_2 chemical propellant Orbit Transfer Vehicle (OTV). The engines operate at an O/F = 6 and have an I_{sp} = 460 s; the OTV dry weight is 5.8 T and has a propellant capacity of 32.5 T. Attached to the OTV is a Tanker-Cargo Unit and (optionally) a "Factory" unit which could be used to process the asteroid material at the asteroid. In this scenario, the OTV leaves LEO with the tanker containing 4.6 T of LH_2 propellant for the return trip. The OTV arrives at the asteroid and then refuels with LO_2 (from electrolyzed asteroid soil). Additional LO_2 is loaded into the tanker for use as OTV propellant for the next LEO-to-asteroid trip and still more LO_2 or other material loaded for use back in LEO. Finally, the OTV returns to LEO.

The total Earth-supplied mass required for a running system (OTV and Tanker already in LEO) is thus the LH_2 required for the two steps. As an option, a "Factory" unit or consumables could be carried from LEO to the asteroid and left at the asteriod. For convenience in scaling the system, various "Factory" weights were considered corresponding to a given percent of the total LO_2 production weight. An amortized MPBR can be found for comparing "N" trips with a "Factory" over "M" trips without a "Factory" by:

$$\text{Amortized MPBR} = \frac{N(\text{MPBR with "Factory"}) + M(\text{MPBR without "Factory"})}{(N + M)\ \text{Total Trips}} \qquad (7\text{-}14)$$

The results of this analysis, for various "Factory" weights and LEO-to-asteroid ΔV's are shown in Figure 7-5. In the figure, a "zero-weight" vehicle limit is shown which would approximate a large system with a dry weight that is small compared to the payload weights. Since candidate asteroids have widely varying ΔV's, a range of ΔV's is given in the figure with specific targets indicated. For simplicity, we have assumed equal ΔV's for the outbound and return legs of the trip, with aerobraking being used to lower the return-to-LEO ΔV by 2.3 km/s. In fact, aerobraking might remove even more of the total ΔV since the largest part of the ΔV is required for Earth-orbit insertion and only a small ΔV is needed for escape from an asteroid like 1982 DB (0.2 km/s). The low ΔV needed for escape would also make possible the use of electric launch systems, like rotary launchers (Ref. 7-10), rail guns, or mass drivers, that could be used to inject asteroid material into Earth-intercepting trajectories. Thus, once escape from the asteroid had been achieved, no additional propulsion would be required for return to the Earth's surface. However, for a routine space resource operation, in which asteroid materials and vehicles return to LEO, there may still be a large ΔV on return to Earth orbit that is not removed by aerobraking.

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In addition, there is the advantage of staging in lunar orbit and using a separate Lunar Transfer Vehicle (LTV) for the lunar landing, as was done in the Apollo program. This seems to strongly favor lunar material return for all but possible Earth-Trojan asteroids (L4, L5). Additionally, the Moon is closer for easier teleoperation and shorter trip times, and also is better known. Thus it still appears that the Moon is the most promising candidate for exploitation, at least in terms of the MPBR. However, valuable resources on the asteroids (H_2O or H_2 for propulsion, etc.), advances in aerobraking techniques (to lower the propulsive part of the asteroid-to-LEO ΔV), or the use of asteroid-based electric launchers (rotary launchers, etc.) could still make asteroids a desirable source of material.

7.6 TECHNOLOGY STATUS AND PROJECTIONS

In general, these analyses have concentrated on near-term (pre-year 2000) chemical and electric propulsion systems. This has enabled us to calculate Mass Payback Ratios for transportation systems of well-known characteristics (I_{sp}, dry weight, specific mass α, etc.). This has also resulted in highly conservative estimates of the MPBR, since more advanced systems would have better performance and thus yield higher MPBR's. By using propulsion systems and vehicles typical of LEO-to-GEO OTV's of the next 10 to 20 years, we have established a set of minimum MPBR's that could be obtained using common and inexpensive vehicles for return of extraterrestrial material to Earth orbit.

Thus, H_2/O_2 chemical engines with O/F = 6 and I_{sp} = 460 s, or Ar nuclear electric (NEP) systems with 50 to 100 cm thrusters can be expected to represent state-of-the-art satellite OTV propulsion by the mid-1990's. Several possible technology advances could yield improvements in MPBR, either by increasing I_{sp}, decreasing stage dry weights (and electric propulsion power system specific masses), by increased use of extraterrestrial propellants, or by use of other propulsion systems (mass drivers, nuclear-thermal engines, etc.).

Some of these potential advances are described below, with the MPBR again taken to be the figure-of-merit. However, it should be remembered that many of these areas would need direct support from a lunar material program. Thus, significantly higher MPBR's would be needed so as to amortize R&D costs of these advanced systems, as compared to the near-term OTV's, which are likely to be developed for LEO-to-GEO use.

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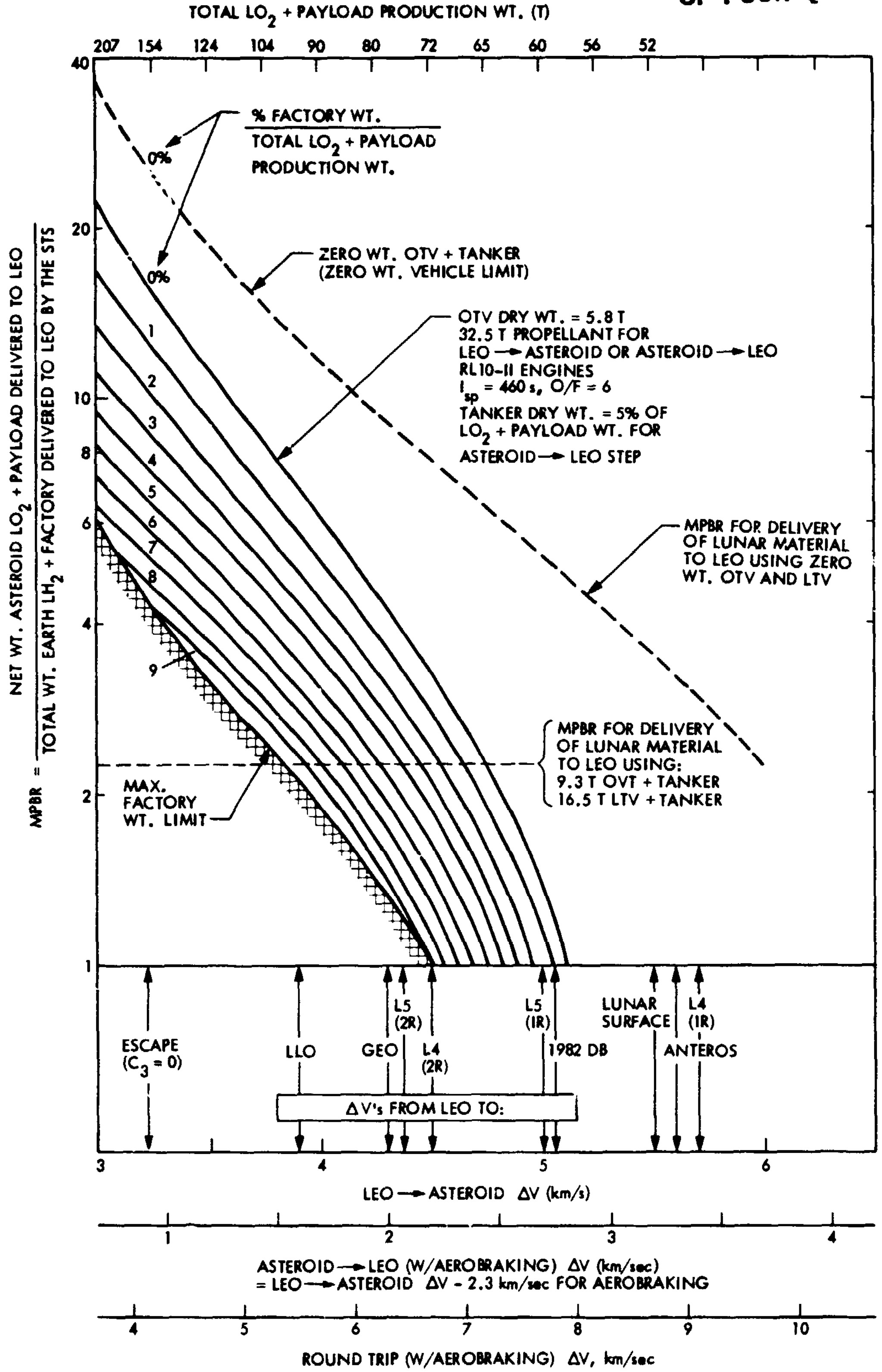

Figure 7-5. MPBR for Return of Asteroid Material to LEO

7.6.1 Chemical Propulsion

The previous analyses of lunar material return to LEO or GEO have used a specific baseline pair of vehicles: a 9.3 T OTV and Tanker, and a 16.5 T LTV and Tanker. The OTV represented a specific point design with a LH_2/LO_2 propellant capacity of 32.5 T. This propellant capacity essentially sized the LTV, since it was a stretched OTV with extra engines and landing gear. Since advances in stage, tanker, and engine weights and performances may be significant over the next 20 to 30 years, a study was conducted to determine the effects on the Mass Payback Ratio for various vehicle weights and for various engine I_{sp} and O/F values. For example, a very large or lightweight vehicle would have approximately zero weight, compared to the mass of payload and propellant. Additionally, increases in engine I_{sp} or O/F can both increase the payload capacity and decrease the amount of Earth-supplied hydrogen required, thus increasing the MPBR.

As in previous work, the OTV is limited to a maximum of 32.5 T of propellant for the LEO-to-LLO, LLO-to-LEO, LLO-to-GEO-to-LEO, LEO-to-GEO, and GEO-to-LEO steps. Again, this sizes the rest of the system in terms of propellant and payload values. The results shown below can be used for other OTV choices by appropriate scaling. For example, if an OTV (and Tanker) had a dry weight of 5.0 T and a propellant capacity of 20.0 T, then the OTV (and Tanker) dry weight would be scaled to 5.0 * (32.5/20.0) = 8.125 T. The 5.0 T/20.0 T OTV would act like a 8.125 T/32.5 T OTV in these analyses. The weight of the LTV (and Tanker) could then be varied to obtain a feel for the effect of LTV weight on system performance.

As before, the figure-of-merit for these analyses is the MPBR. This is defined as the net mass of lunar material (assumed to be LO_2) delivered to LEO (or GEO), divided by the mass of Earth-supplied LH_2 delivered to LEO by the STS that is required to fuel the system. For LEO deliveries, lunar LO_2 is provided for the OTV LEO-to-LLO step, whereas for GEO delivery, the STS must supply Earth LH_2 and LO_2 since the OTV returns dry to LEO after a GEO delivery. In actual operation, it would be possible to use lunar LO_2 separately delivered directly to LEO to fuel the dry GEO-delivery OTV for its trip to LLO. For the heaviest systems, the OTV cannot carry enough terrestrial LH_2 to LLO for all the vehicles, so some lunar LH_2 would also be required, although the intent here has been to minimize or eliminate the need for any lunar LH_2.

A series of parametric analyses were performed to examine the sensitivity of the MPBR to the baseline parameters (vehicle dry weight, I_{sp}, O/F) used above. These results are shown in Figures 7-6 to 7-15. In general, changes in vehicle weight, engine I_{sp}, and high O/F will all be required for major variations in MPBR. Each figure will be discussed in detail below.

7.6.1.1 Variations in Vehicle Dry Weight. Figure 7-6 represents the results for return of lunar material (LO_2) to LEO using a I_{sp} = 460 s, O/F = 6 engine on an OTV and LTV of various dry weights. Maximum MPBR with limiting "zero-weight" vehicles is about 4.5 as compared to a MPBR of about 2.2 for OTV and LTV weights (10 T and 16 T, respectively) representative of the baseline vehicles. The largest effect is due to changes in the OTV dry weight, since it is this vehicle which undergoes the largest ΔV's. With current engine technology, it will be difficult to get a MPBR greater than about 4 for LEO delivery.

Figure 7-7 is similar to Figure 7-6 and shows the MPBR for GEO delivery of lunar material. The zero-weight vehicle MPBR for baseline engines is about 2.5; however, this should be compared to a MPBR of only about 0.3 to 0.4 for zero-weight vehicle direct delivery of terrestrial material to GEO (using a LEO-based OTV). Thus, the effective ratio is MPBR (Lunar)/MR (Direct) = (~2.5)/(~0.3) = ~8. For the baseline weight vehicles, the ratio is about (1)/(.2) for a fivefold increase in mass compared to direct Earth-supply of material to GEO.

7.6.1.2 Variations in Lunar Landing and Takeoff ΔV. Figure 7-8 shows the effect of lunar landing and takeoff on MPBR. A 460-s, 6:1 engine is again used on a zero-weight or baseline 16-T LTV and an OTV of various weights. Takeoff and landing ΔV's of 1.6 km/s have been used in all of the other analyses, although the Apollo Lunar Module had landing ΔV's of up to 2.1 km/s. This figure shows the effect on MPBR when using a 2.1 km/s landing and 1.8 km/s takeoff ΔV. The changes in MPBR as compared to the minimal 1.6 km/s case are fairly small for these large changes in the lunar ΔV's. In perspective, it should be noted that the Apollo landings represented first-time landings with considerable orbital uncertaintly (e.g., mascons) coupled with human intervention to pick a desirable landing point (no boulders, craters, etc.). By contrast, an operational LTV landing would be made (by computer control) to a well-established, well-known landing pad. Any mascons or other anomalies would have been previously discovered and mapped, and a minimum ΔV trajectory established. For this reason, the 1.6 km/s ΔV's for the LTV landing and takeoff will continue to be used.

7.6.1.3 Variations in Direct Delivery OTV Dry Weight and Isp. Figures 7-9 and 7-10 indicate the performance of an OTV for direct delivery to GEO. There is a Mass Ratio performance penalty in using a reuseable OTV as shown in Figure 7-9. This is countered by the up-front cost of a disposable OTV. Similarly, the increases in MPBR using high I_{sp} (>480 s) engines shown in Figure 7-10 might be countered by the high development cost required for these engines. Finally, any roundtrip GEO delivery is quite sensitive to the OTV dry weight, since the GEO-to-LEO step involves a sizable ΔV even with aerobraking (2.0 km/s).

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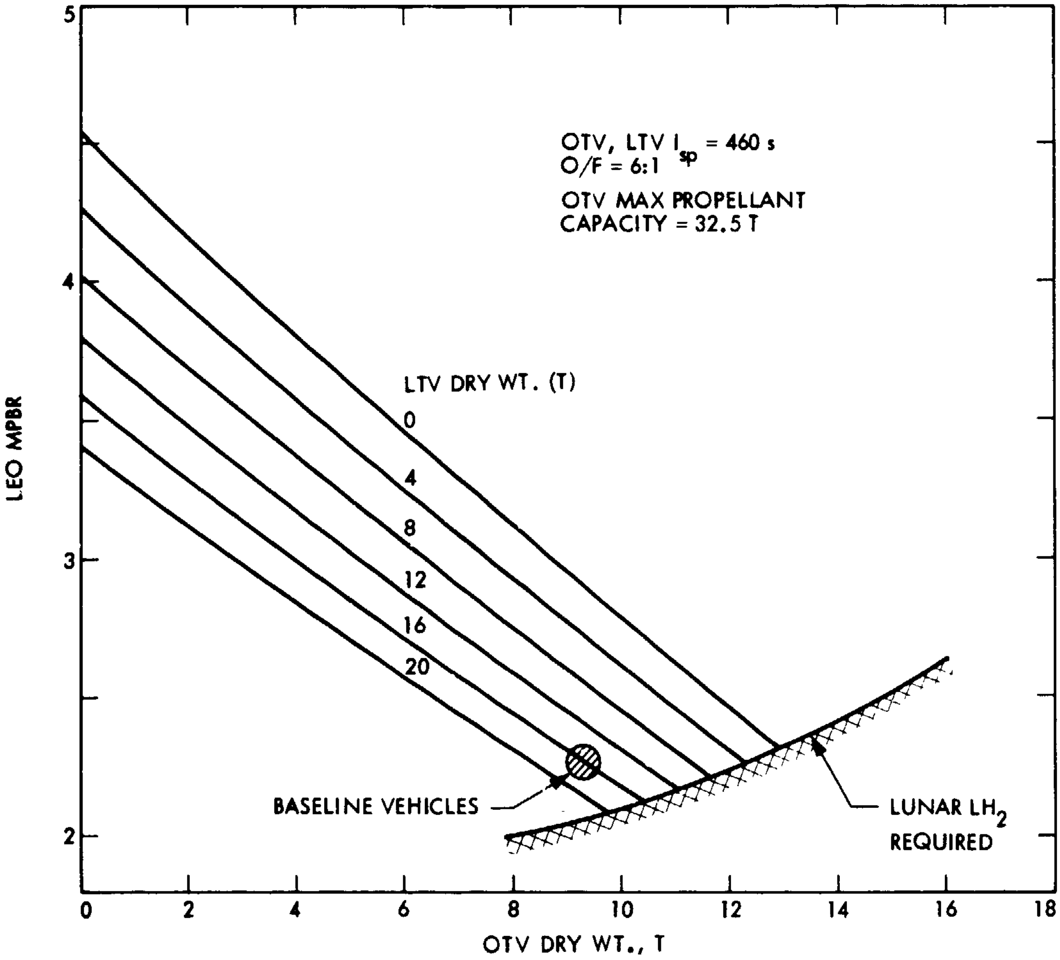

Figure 7-6. MPBR for LEO Delivery vs. OTV and LTV Dry Wts.

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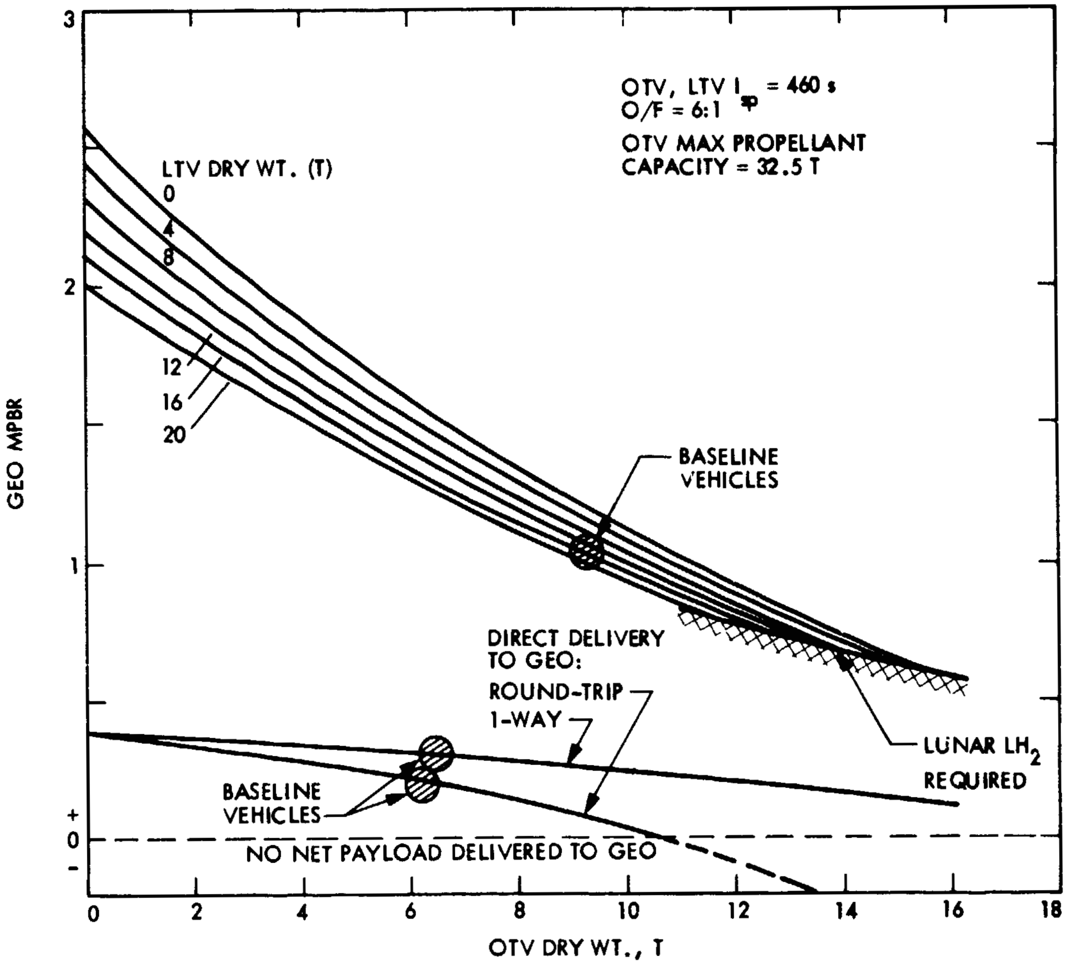

Figure 7-7. MPBR for GEO Delivery vs. OTV and LTV Dry Wts.

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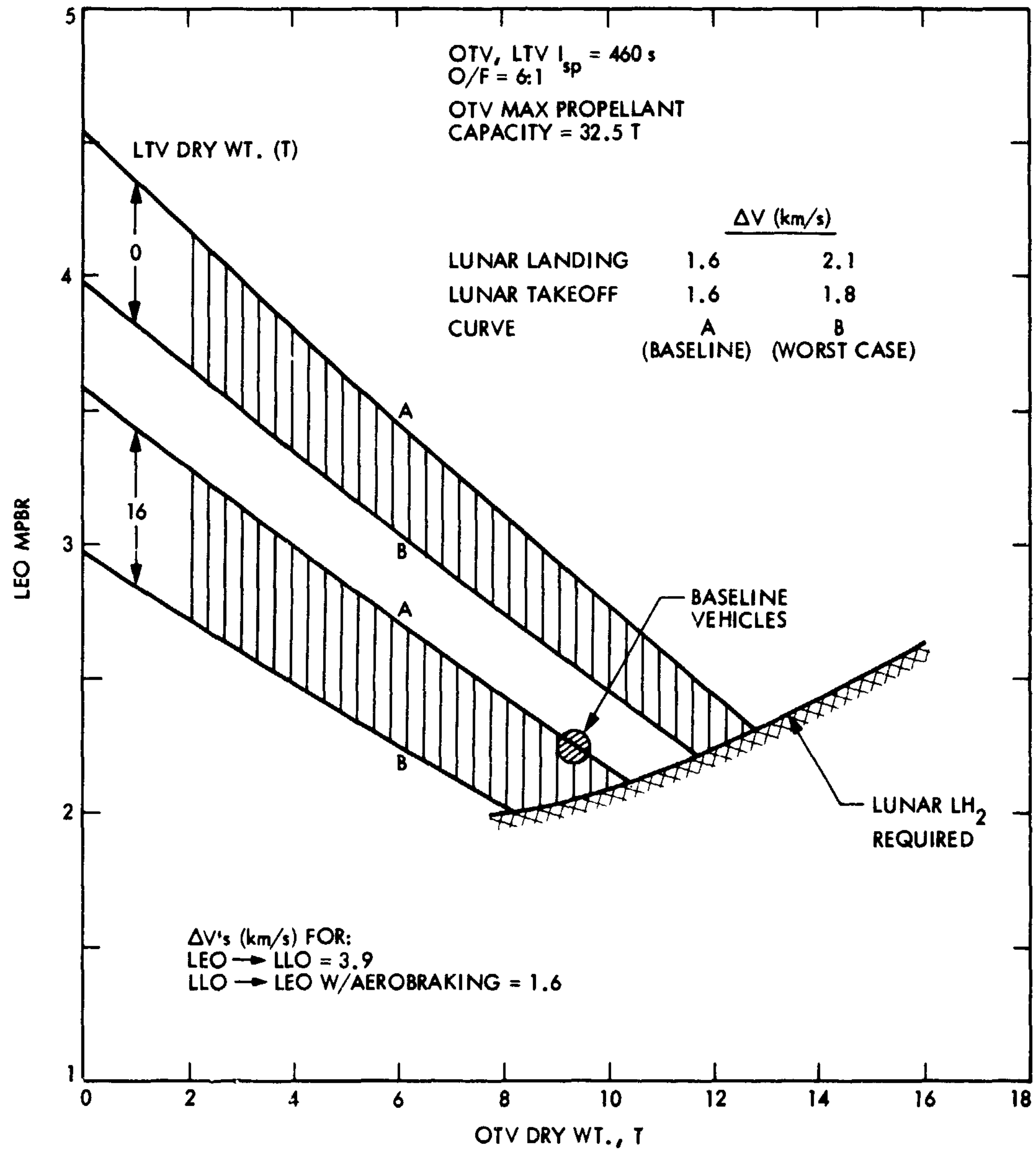

Figure 7-8. MPBR for LEO Delivery vs. OTV and LTV Dry Wts. and ΔV's for Lunar Landing/Takeoff

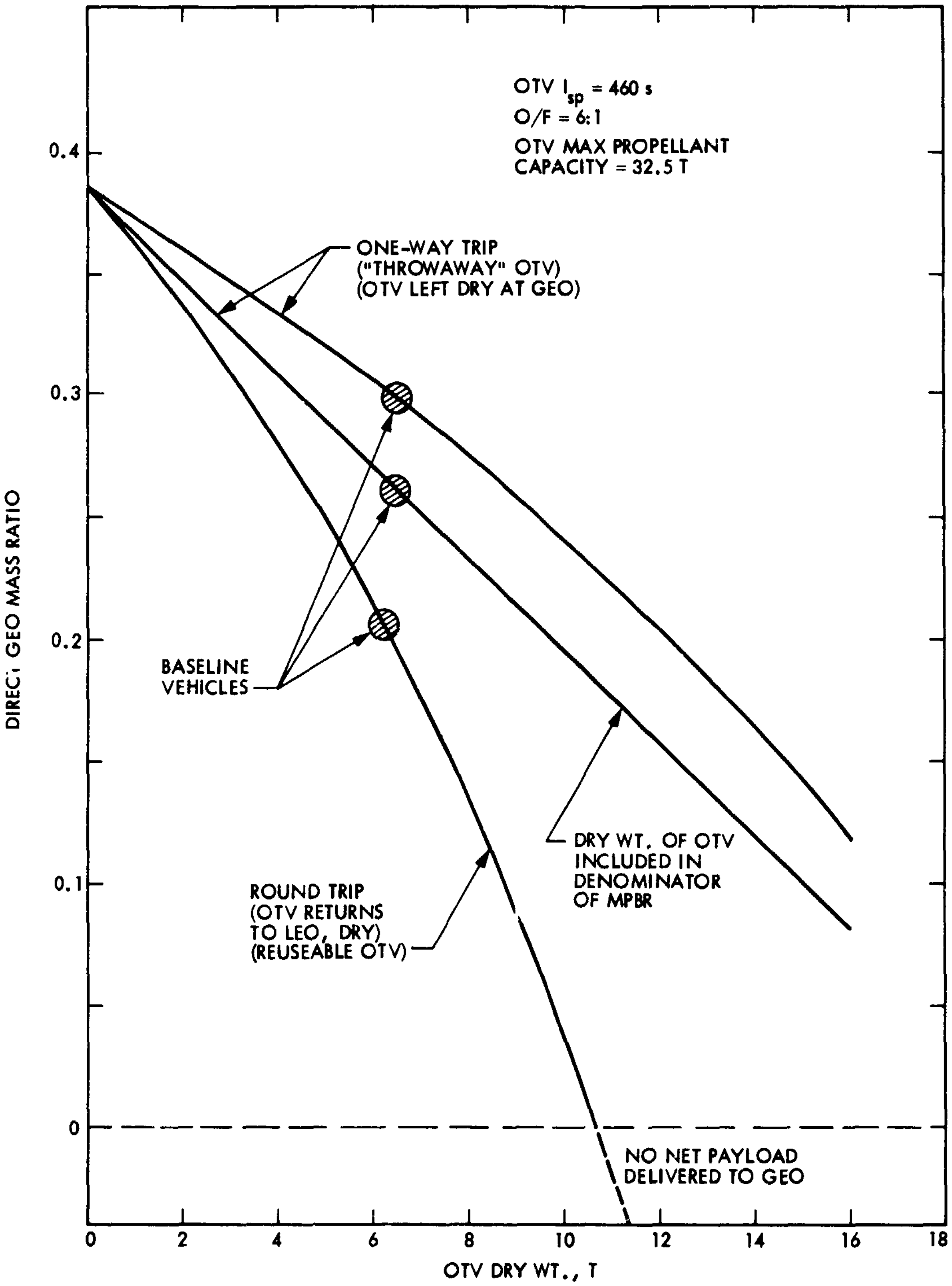

Figure 7-9. MR for Direct GEO Delivery vs. OTV Dry Wt.

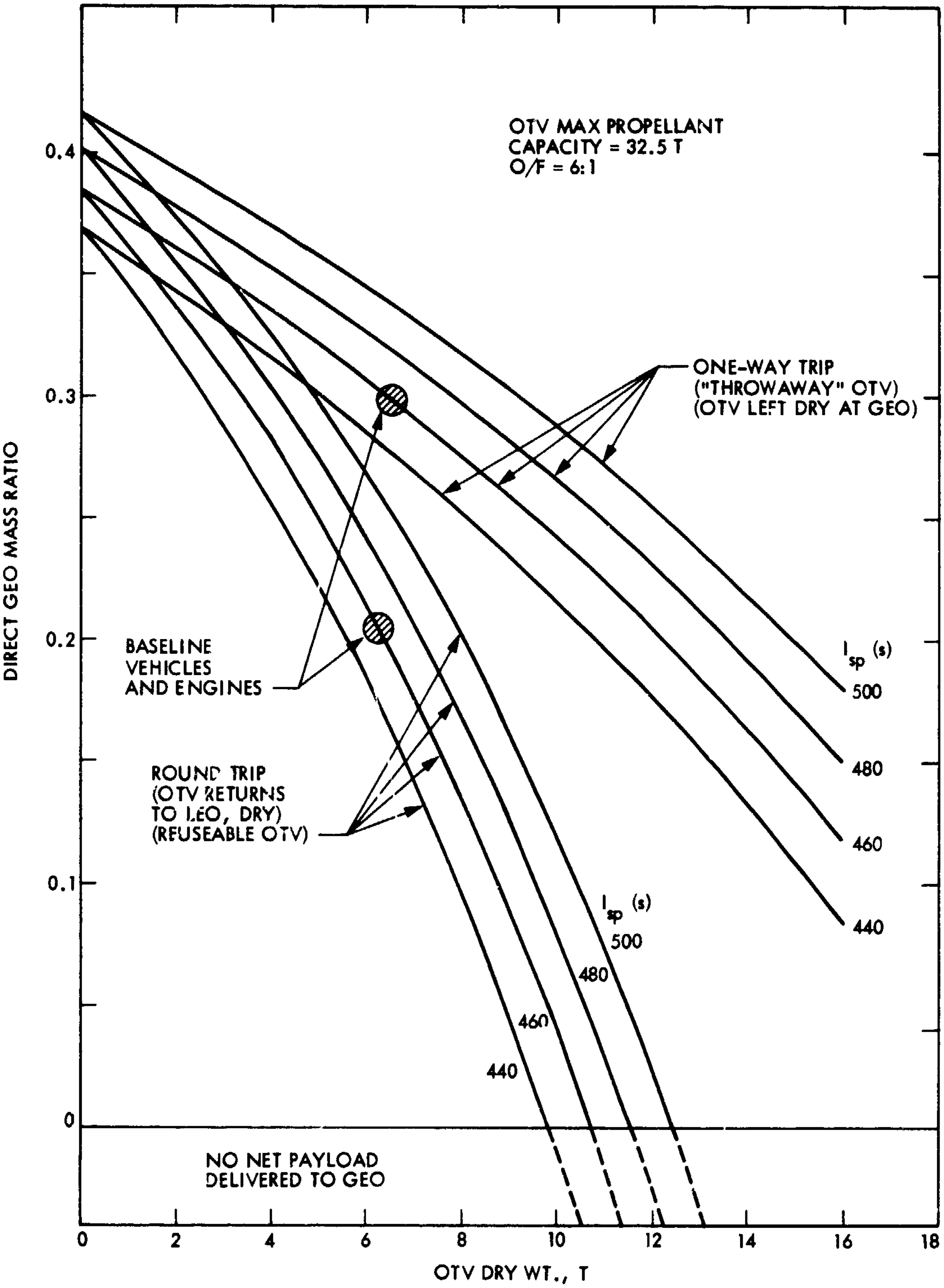

Figure 7-10. MR for Direct GEO Delivery vs. OTV Dry Wt. and Engine I_{sp}

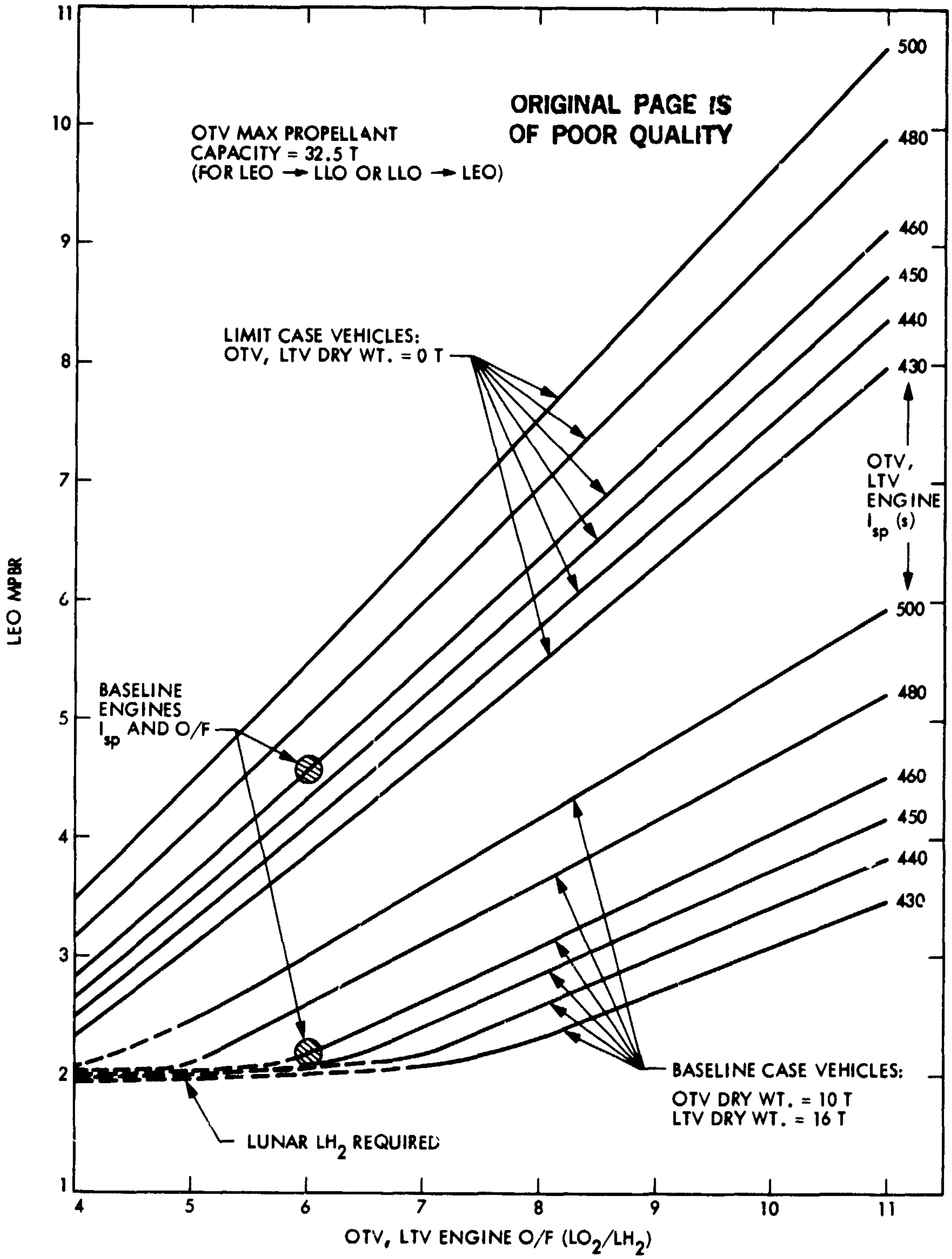

Figure 7-11. MPBR for LEO Delivery vs. OTV and LTV Engine I_{sp} and O/F

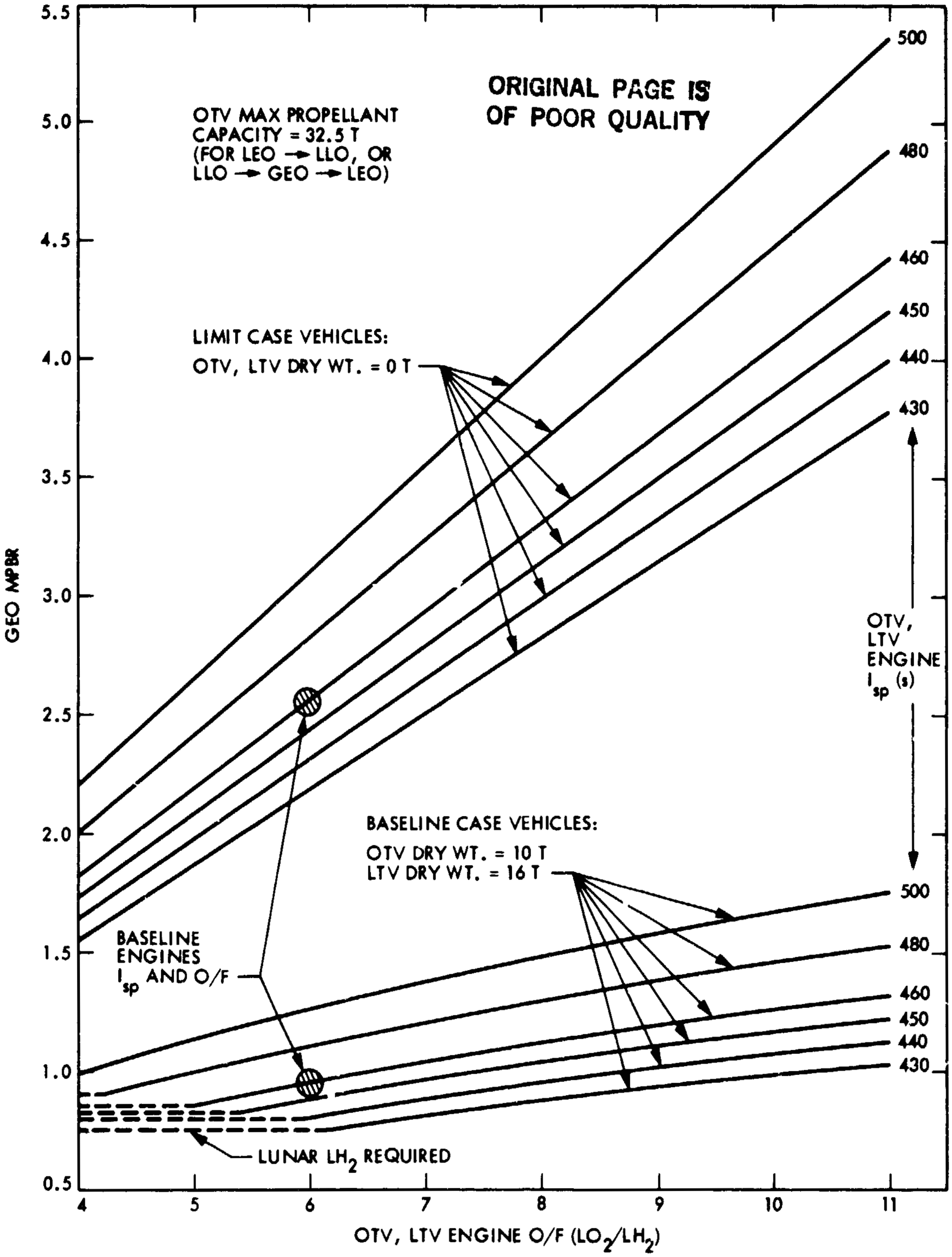

Figure 7-12. MPBR for GEO Delivery vs. OTV and LTV Engine I_{sp} and O/F

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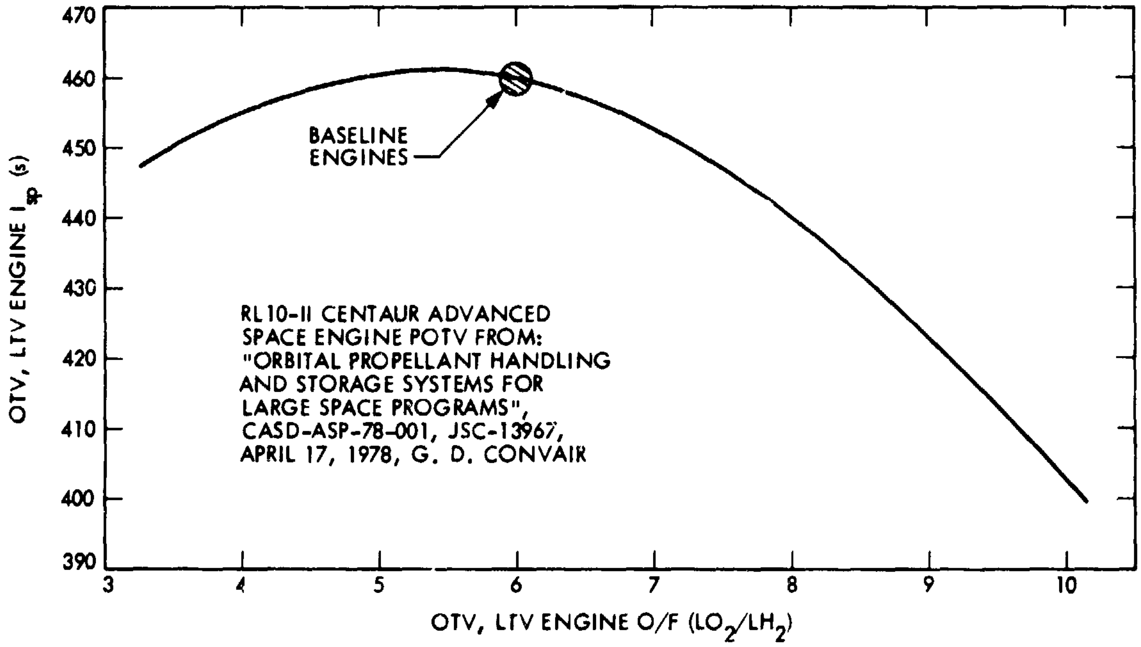

Figure 7-13. RL10-II Engine I_{sp} vs. O/F

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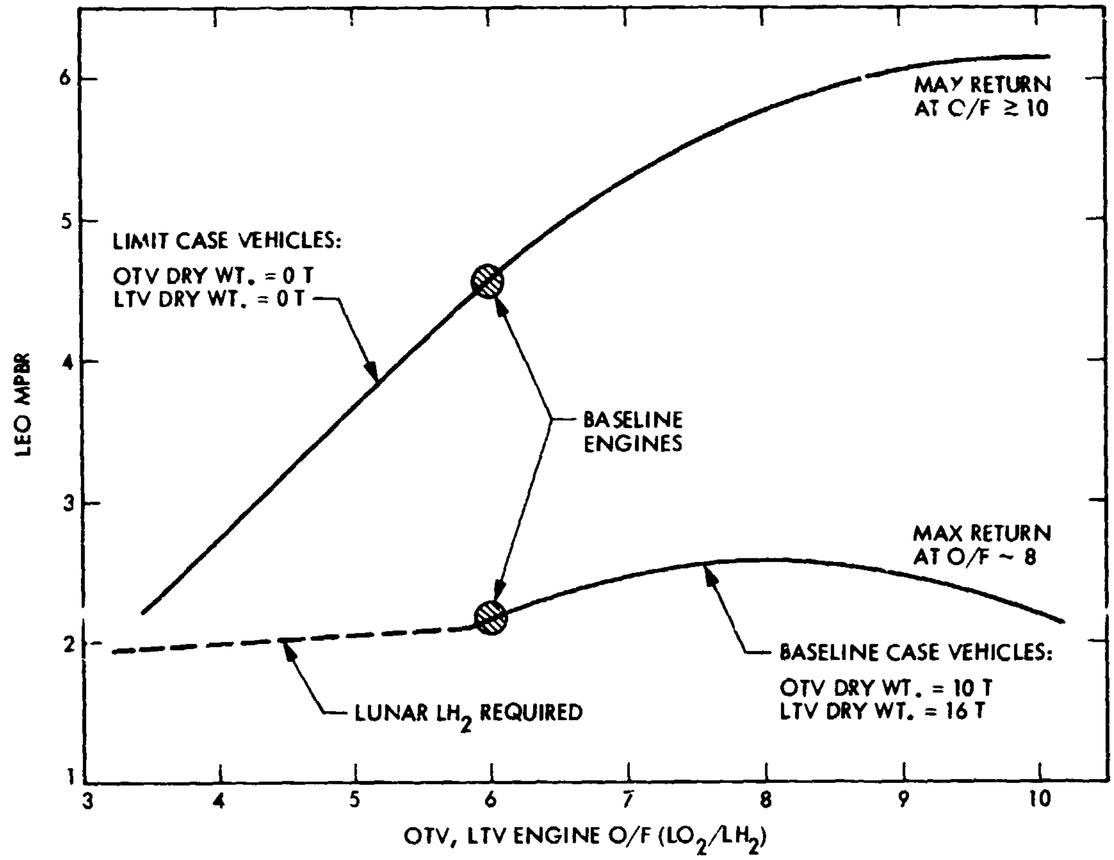

Figure 7-14. MPBR for LEO Delivery vs. Engine Performance

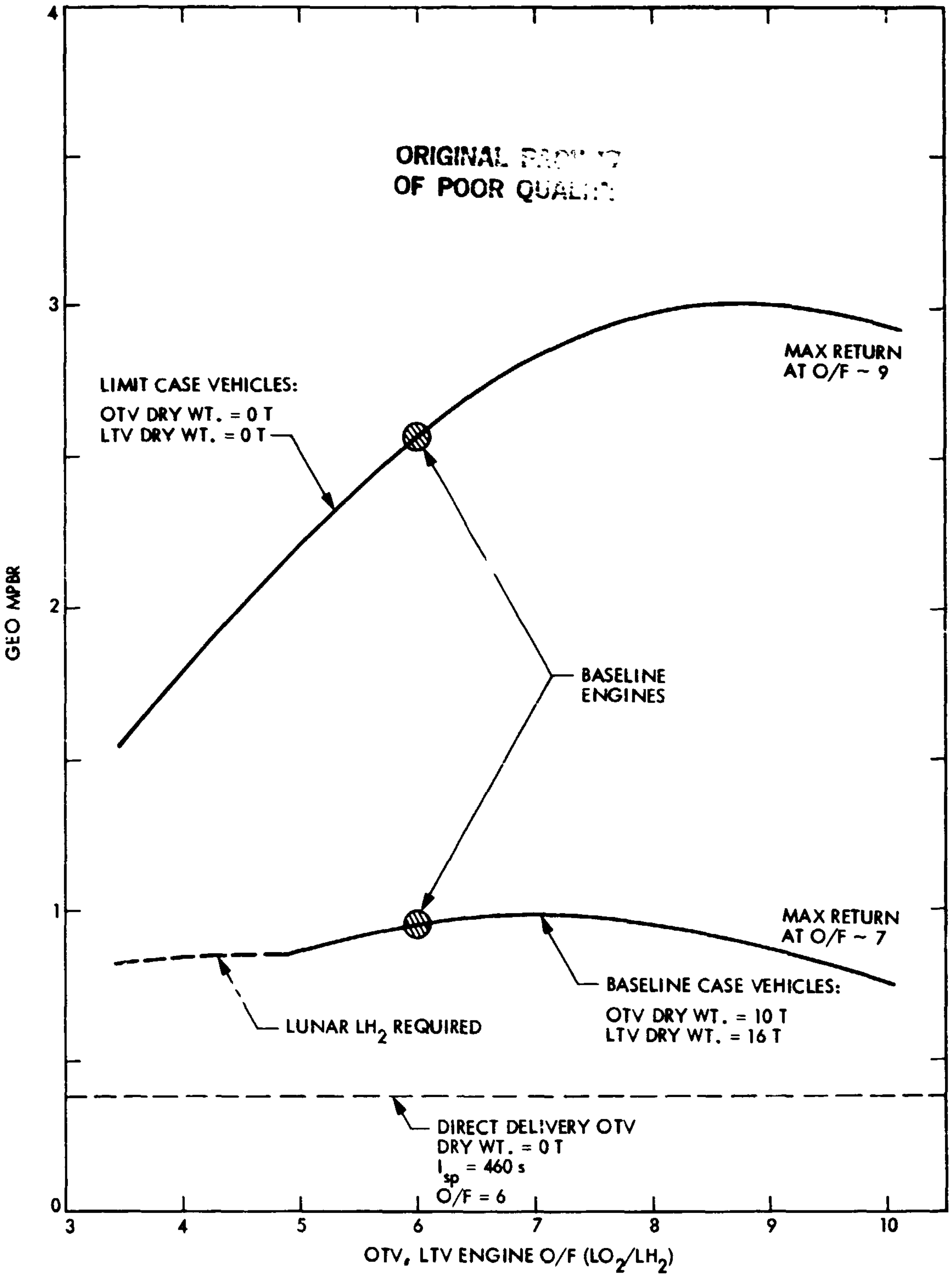

Figure 7-15. MPBR for GEO Delivery vs. Engine Performance

7.6.1.4 Variations in I_{sp} and O/F. Figures 7-11 and 7-12 show, for LEO and GEO delivery, the effects of engine I_{sp} and O/F on the MPBR for two sets of OTV and LTV dry weights. A zero-weight OTV and LTV, and a baseline 10-T/16-T OTV/LTV are again used. MPBR's for both LEO and for GEO delivery of lunar material show significant increases with I_{sp} and O/F. Any increase in I_{sp} results in the need for less propellant. An increase in O/F decreases the amount of LH_2 supplied from the Earth, thus increasing the MPBR. However, this does require more lunar LO_2 so that the additional machinery required to produce this additional LO_2 will have an effect on establishing an optimum for the total system. Finally, real engines have a definite trade-off between I_{sp} and O/F as described below so that it is not possible to freely vary I_{sp} and O/F.

7.6.1.5 RL10-II Engine I_{sp} and O/F. Figure 7-13 shows the variation in I_{sp} with O/F for an RL10-II Centaur engine. Above an O/F of about 5.5, an increase in O/F (which increases MPBR) results in a decrease in I_{sp} (which decreases MPBR). Thus there is a competition between these two factors, as shown in the next two figures.

7.6.1.6 Variations in I_{sp} and O/F Using an RL10-II Engine. Figures 7-14 and 7-15 show, for LEO and GEO delivery, the effect of actual engine performance on the MPBR. As before, a set of limiting zero-weight and baseline 10-T/16-T OTV/LTV vehicles are used, with the actual RL10-II engine I_{sp} for a given O/F used. For the baseline weight vehicles, there is a broad maximum near the stoichiometric water O/F of 8. For zero-weight vehicles, there is a strong maximum at higher O/F values. Engines designed to use LH_2 and LO_2 derived from water electrolysis (O/F = 8) in LEO will probably be developed independently of a lunar materials program. Such engines could be used to improve lunar material delivery, although the slight improvement in MPBR may not justify their development costs for this program alone. Nevertheless, these O/F = 8 engines could be used as-is on the OTV and LTV, with the possibility of eventually using LH_2 and LO_2 derived from electrolysis of lunar water-ice, thereby eliminating the need for any LH_2 from Earth.

7.6.1.7 Summary. From these analyses, it appears that a strong drive to decrease vehicle weights may not be warranted, since MPBR is fairly insensitive to the OTV and LTV vehicle weights for the size of vehicles used in this study. For example, a given percent decrease in OTV and LTV weights will produce a roughly equal percent increase in MPBR, using the baseline 10-T/16-T OTV/LTV pair as a base case as shown in Table 7-10. However, decreasing the weight of reuseable vehicles like these will be difficult. By contrast, there appears to be more promise in pursuing increases in engine I_{sp} by using advanced H_2/O_2 engines. As shown in Table 7-10, a 4% increase in I_{sp} from 460 to 480 s, typical of predicted performance for advanced H_2/O_2 engines (Refs. 7-2 and 7-3), can produce a 16 to 19% increase in MPBR. By the year 2000, 500 s I_{sp} engines should be available (Refs. 7-2 and 7-3), which could increase the MPBR by over 30% compared to near-term 460 s I_{sp} engines. However, the economic tradeoffs between increased I_{sp} and increased engine cost are not always clear; current LEO-to-GEO OTV studies tend to favor low-cost near-term 460 s I_{sp} engines, although their conclusions are somewhat mission-model dependent. Note that mission models and cost assumptions may

Table 7-10. Technology Trades for Lunar Material Transportation Using Chemical Propulsion (All weights in metric tons: T = 1000 kg)

Lunar Material Delivered To:	LEO			GEO			Comments
OTV Dry Wt. (T)	10	8	6	10	8	6	Decrease Vehicle Dry Wt.
LTV Dry Wt. (T)	16	12	8	16	12	8	I_{sp} = 460 s, O/F = 6
% Decrease in OTV Dry Wt.	--	20%	40%	--	20%	40%	
% Decrease in LTV Dry Wt.	--	25%	50%	--	25%	50%	
MPBR	2.176	2.590	3.054	0.956	1.184	1.456	
% Increase in MPBR	--	19%	40%	--	24%	52%	
Engine I_{sp} (s)	460	480	500	460	480	500	Increase Engine I_{sp}
% Increase in Engine I_{sp}	--	4.35%	8.70%	--	4.35%	8.70%	(O/F = 6)
MPBR	2.176	2.586	2.996	0.956	1.106	1.265	OTV Dry Wt. = 10 T LTV Dry Wt. = 16 T
% Increase in MPBR	--	19%	38%	--	16%	32%	

change significantly over the next 20 years, so it is not possible at this time to predict which approach is the better in terms of overall cost. Finally, adjusting the oxidizer-to-fuel ratio to higher O/F values does not appear worthwhile since the MPBR gain is small for realistic vehicle weights. Additionally, operation of an engine under leaner than usual conditions would result in higher than normal combustion temperatures and thus a shorter service life.

Thus there are a number of technology improvements that can increase the MPBR by relatively small factors which can be important, but not dramatic, for chemical engines constrained to operate on terrestrial LH_2 and lunar LO_2. By contrast, if lunar LH_2 is available, these engines can readily operate on all-extraterrestrial propellants to give infinite MPBR's, and do so for a low initial cost. Finally, high T/W chemical vehicles will always be needed for high priority, high cargo value missions, such as manned missions, where fast trip times, low radiation doses, or man-rated propulsion systems are required.

7.6.2 Electric Propulsion

The state of the art for primary electric propulsion in the U.S. is represented by the J-series 30-cm mercury ion thruster, its functional model power processor, and the detailed designs for an integrated total propulsion subsystem (Refs. 7-11 and 7-12). However, this propulsion system is not flight qualified. In the late 1960's electrothermal thrusters (resistojets and arcjets) were nearly at the same technology readiness level as the J-series system but that level has not been maintained. Electromagnetic thrusters (primarily the magnetoplasmadynamic, MPD) are in the research phase. Projections for the potential performance of future thruster systems are speculative, but are available. Performance parameters include: efficiency, lifetime, power per thruster, specific impulse range, propellants, and system specific mass. Only electrothermal, ion and MPD technologies will be considered further. Electrothermal thrusters operate best on NH_3 and H_2. Ion engines are developed for mercury and are being developed for Ar and Xe. MPD thrusters can probably operate well on the widest selection of propellants. Note that oxygen electrothermal and ion thrusters are not practical, whereas oxygen and hydrogen can best be utilized as propellants in MPD thrusters. Thus, MPD thrusters could use Earth-supplied H_2 or lunar O_2 as propellant. Electrothermal thrusters are limited to specific impulses below about 2000 s. Ion engines can potentially operate on Xe at I_{sp}'s of 1000 s to 10,000 s and more using argon with mercury ion thrusters generally falling in between. MPD thrusters can operate at low specific impulses around 2000 s using Ar and at 10,000 s or above using H_2. Figure 7-16 presents projections for system efficiency (including power processing) versus I_{sp} for several potential thruster technologies. Data are also given for the J-series solar electric propulsion system (SEPS) technology at an I_{sp} of 3000 s and a curve for the efficiency of electrothermal devices for I_{sp}'s between 800 and 2000 s (Ref. 7-13). Note that thruster size and power levels vary with the specific impulse in Figure 7-16. MPD thrusters operated on Hg and Xe would be expected to show similar improvements over operation on Ar as do the ion engines. Figure 7-17 presents the propulsion system specific mass as a function of specific impulse (Ref. 7-13). In Figure 7-17, projections for a pulsed MPD system are shown; for comparison, a steady-state MPD thruster system might have a specific mass of 1 to 4 kg/kW. Figure 7-18 presents the potential levels of power per thruster for the three generic technologies. Steady-state MPD thrusters operate around 1 MW_e while a single ion thruster may be able to consume from 10 kW_e at an I_{sp} of 2000 s to 1 MW_e at an I_{sp} above 6000 s. The lifetime for an electrothermal thruster is at least 1000 hrs, is projected to be 15,000 to 20,000 hrs for ion thrusters, and is unknown for MPD thrusters.

7.6.3 Very Advanced Propulsion Systems

Very advanced propulsion systems have the potential for yielding very high MPBR's either by providing high engine I_{sp}'s or by eliminating the need for terrestrial propellants (e.g., solar sails and mass drivers). Such advanced propulsion systems are generally large and complex; their high initial development cost would be offset by their high MPBR. Many advanced systems could be developed with sufficient funding, although some, like Orion (fission bomblet) might face serious political obstacles to their use.

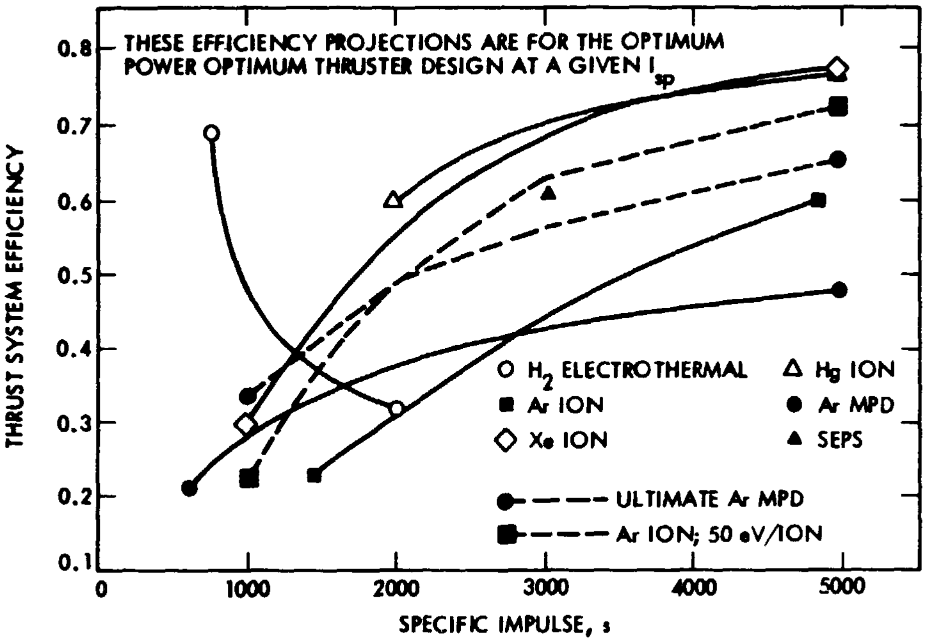

Figure 7-16. Projected System Efficiency vs. Specific Impulse

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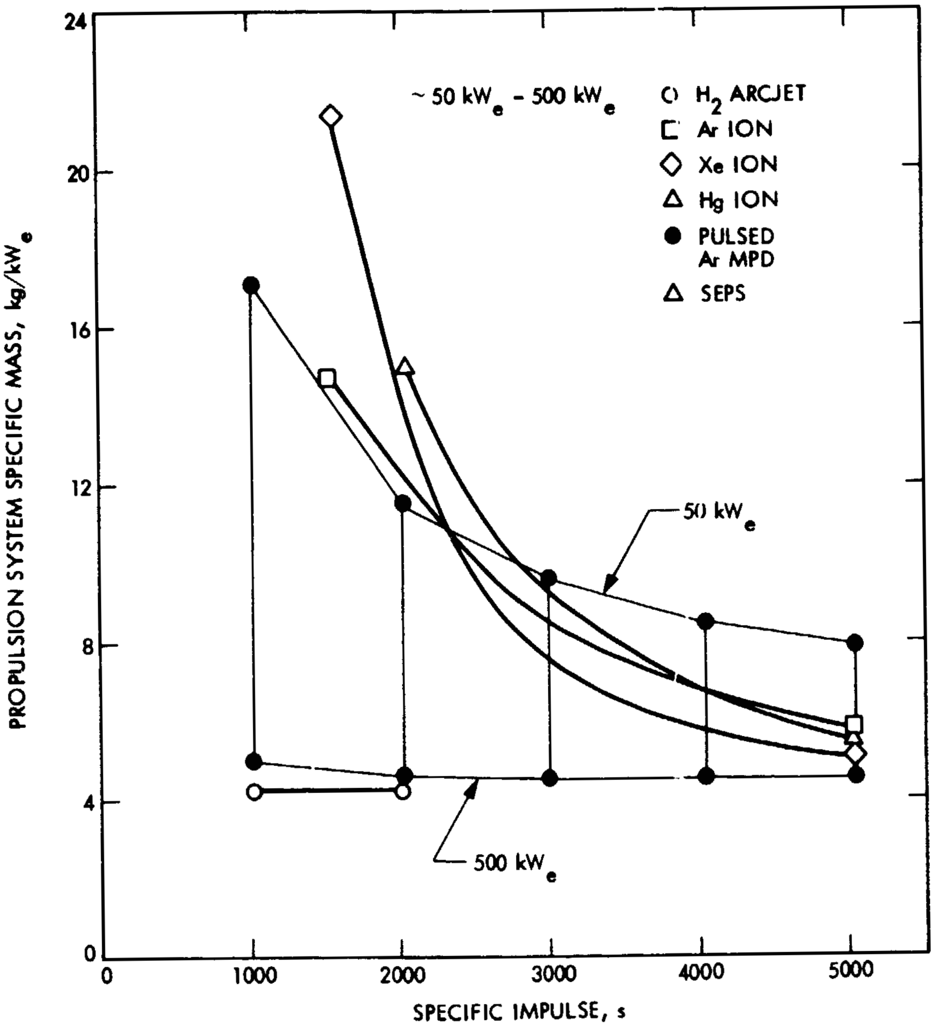

Figure 7-17. Propulsion System Specific Mass vs. Specific Impulse

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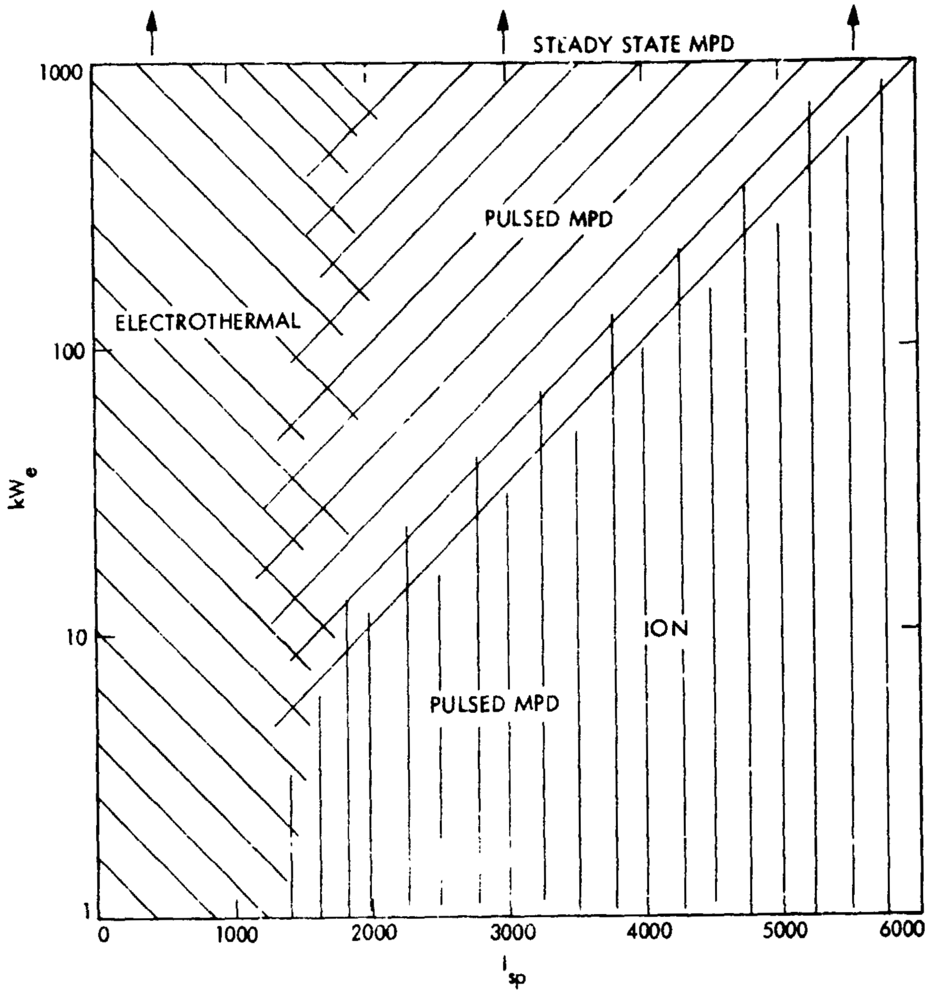

Figure 7-18. Power per Thruster After Technology Development

7.6.3.1 Electromagnetic Launchers (Mass Driver or Rail Gun). Electromagnetic launchers, rail guns and mass drivers, are only conceptual at this time. Their fundamental principles have been proven. The required exit velocities for lunar launch have been achieved by rail guns but not in a long rail, low acceleration, high payload mass configuration. Many engineering problems need to be addressed. No definitive projections can be made for lifetime, efficiency or system mass.

7.6.3.2 Fission/Fusion Thermal. Solid and particle core nuclear (fission) rockets already have an extensive technology data base. Liquid and gas core nuclear rockets are more conceptual and would require extensive development. Considerable effort went into the Orion (fission bomblet) concept, although much of the study results remain classified. Fusion rockets remain strictly conceptual, although the well-funded fusion powerplant R&D program may yield technology that would be applicable to fusion propulsion systems (Ref. 7-6).

Similar technology crossover may be gained from current interest in particle bed nuclear reactors for use as power supplies. Since a particle bed nuclear rocket has good performance and permits dumping of the radioactive particles prior to servicing, it appears to be a likely candidate for development. However, the use of terrestrial LH_2 limits any of the nuclear-thermal rockets to fairly low MPBR's, with only the very large fusion systems having an MPBR greater than 30 for transport of lunar material. As with chemical rockets, a source of lunar LH_2 would eliminate the need for terrestrial LH_2, although all the propellant mass would be LH_2 instead of only one-seventh as in chemical engines.

7.6.3.3 Solar Sail. A solar sail using a mass driver lunar catapult could provide an infinite MPBR. However, sails are very slow and could only operate between GEO and LLO due to air drag at LEO (Ref. 7-2). However, although the sail itself cannot operate at LEO, it could insert a lunar material payload capsule into a LEO-aerobraking trajectory after leaving LLO. An ablative aerobraking capsule made from lunar materials might be a relatively simple structure amenable to lunar fabrication. By using a solar sail, the only propellant required would be those for the LTV and the small amounts required for the aeromaneuvering capsule. Finally, for transport of propellants or supplies from LEO to LLO and the lunar surface, a chemical OTV could transfer the materials from LEO to a higher orbit (>1000 km) (Ref. 7-2) to rendezvous with the solar sail, which would then transport the materials the rest of the way to LLO. The reverse procedure could also be used to return lunar material to LEO.

Current work on solar sails is being actively supported by private organizations; there is no current major NASA-funded work (Ref. 7-7). The relatively low cost of solar sails, compared to conventional propulsion systems, makes it feasible for non-government groups to pursue the R&D of sails. In addition, the low initial investment required for purchase of a solar sail should make this system attractive to commercial users. Finally, NASA-funded work on lightweight structures for space applications may yield results applicable to solar sails, since both sails and gossamer structures share a common technology base.

Major technology and operational issues facing solar sails include the weight of sail material and structures, stability of materials and wrinkles in the sail sheeting, deployment of erectable sails brought up from Earth versus sails assembled and fabricated in orbit, and finally, sail control, stability, and maneuverability (Ref. 7-7). Nevertheless, the sail represents an attractive GEO-to-LLO OTV with no propellant needs and low initial and operational costs. The large sails needed to transport the large payloads of a lunar material economy would almost certainly require an on-orbit fabrication facility, although the technology base required for on-orbit manufacture should be well in place by the year 2000.

7.7 CONCLUSIONS

Ultimately, the question is not technical feasibility; the basic technology has existed since Apollo 11 for return of lunar material to Earth. Instead, the problem is to choose which propulsion options will maximize the figure(s)-of-merit used to characterize the overall system. Based on a simple MPBR figure-of-merit, the following conclusions can be drawn from the results of this year's efforts:

(1) Chemical propulsion systems using near-term technology (aerobraking space-based H_2/O_2 OTV's, I_{sp} = 460 s, O/F = 6 engines) can transport lunar material to LEO or GEO for net gains in mass. As usual, GEO delivery is effectively better than LEO delivery, since direct GEO delivery has such a low Mass Ratio (MR). The MPBR's found for the baseline chemical system are highly significant since they are large enough to be potentially attractive without the need to develop any new major technologies. This is especially important for the chemical vehicles, since it can be assumed that they will be readily derivable from the pre-existing OTV technology base of the late 1990's.

(2) There are several options available that can provide significant (but not dramatic) improvements in MPBR. Several samples are given in Table 7-11. These could take the form of the use of other propulsion systems such as a relatively near-term electric propulsion system using 50- to 100-cm Ar electric thrusters. Alternatively, the chemical systems could use either improved technology (e.g., higher I_{sp}) or new operational techniques (e.g, lunar hydrogen). Note that, as far as propulsion is concerned, the discovery of a lunar hydrogen source would profoundly affect the MPBR's by eliminating the need for H_2 from the Earth for either chemical or electric propulsion systems.

(3) For asteroid materials, only Earth-Trojan asteroids appear to offer significant promise on a mass basis, although the presence of valuable resources (H_2 from H_2O, etc.) might change this evaluation.

Table 7-11. MPBR's for Various Propulsion Options for Return of Lunar Material to Earth Orbit
(All weights in metric tons: T = 1000 kg)

OPTION	OTV	LTV	MPBR for Delivery to:	
			LEO	GEO
LH_2, Ar from Earth; LO_2 from Moon	Chem	Chem	2	1
	Ar-Electric	Chem	1-3	2-5
	Chem	Mass Driver	7	2
	Solar Sail	Chem	11	11
LH_2 from Earth; LO_2 from Moon	O_2-Electric	Chem	2-8	5-10
Min. LH_2 from Earth; LH_2, LO_2 from Moon	Chem (Earth LH_2, Lunar LO_2)	Chem (Lunar LH_2 + LO_2) or Mass Driver	7	8
	Chem (Earth and Lunar LO_2, Lunar LO_2)	Chem (Lunar LH_2 + LO_2, or Mass Driver	31	16
Direct GEO Delivery	Chem	--	--	0.2
$MR = \frac{\text{Net Delivered Earth Payload}}{\text{Earth Propellant and Payload}}$	Ar-Electric	--	--	0.8-0.9

Notes: Chemical H_2/O_2 Engines; I_{sp} = 460 s, O/F = 6.
Electric Vehicle MPBR range given for α = 0.03-0.01 kg/W_e.

(4) Most advanced systems, like nuclear-thermal rockets, are less effective than chemical systems since all the nuclear rocket propellant (LH_2) would come from the Earth, assuming no lunar LH_2. An LO_2 nuclear rocket LTV and chemical OTV combination would need no Earth-supplied LH_2 for the LTV, but would consume over 90% of the total lunar LO_2 production for propellant with the remaining 10% left as useful payload. Finally, only very large terrestrial propellant nuclear systems (Orion, fusion) would give MPBR's greater than 10, although they might very well become available and attractive after the year 2000.

(5) Specific technologies and vehicle components required for a lunar materials program would be directly adaptable from OTV's developed in the next few decades to satisfy LEO-to-GEO satellite traffic demands (independently of a lunar materials program). Some specific technology areas of interest to this program might include:

(a) Aerobraking for High T/W OTV's.

(b) Large, High Power Ar or O_2 Ion and Plasma Electric Thrusters.

(c) High I_{sp}, High O/F Chemical Engines.

(d) Mass Driver for Lunar Launchers.

(e) 5-10 MW_e Space Nuclear Power Systems for Electric Propulsion.

(f) Space-Based Zero-G Refrigeration and Fluid Transfer Technology for Cryogenic Propellants.

(g) Teleoperated Spacecraft for Rendezvous, Docking, and Lunar Landing.

(6) Finally, there is a non-technology issue that vastly overshadows all others. This would be the discovery of a source of lunar hydrogen (see Section 9) which could yield infinite MPBR's by eliminating the need for any LH_2 from Earth. The same would be true for any source of extra-terrestrial H_2 (H_2O, CH_4, NH_3, etc.) and could thus make asteroids, Mars and its moons, or comets and their nuclei potentially important H_2 sources. Granted, we may never drag icebergs back from Saturn's rings, but the need for H_2 might someday economically justify such grandiose schemes. Failing the discovery of lunar H_2, electric propulsion systems could be developed that use O_2 so as to eliminate the need for H_2 for transport of time-insensitive bulk cargos between LLO and Earth orbit, although some terrestrial LH_2 would be needed for transport of high-priority cargos using chemical OTV's.

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SECTION 8

UTILIZATION ANALYSIS

William F. Carroll

8.1 CONCLUSIONS

It requires 7 times more energy to go from the Earth's surface to low Earth orbit (LEO) than it does from the Moon to LEO. LEO is particularly important since that is the orbit in which initial large-scale utilization of space is likely to occur--in the form of manufacturing, micro-g science experiments, astronomical and astrophysical observations, staging of spacecraft for geostationary or interplanetary orbits, etc. At times, it requires even less energy to go from some asteroids to LEO than from the Moon to LEO. Launch energies to most higher Earth orbits or to interplanetary trajectories are even more favorable to payloads of lunar or asteroidal origin.

Therefore, there may be advantages to be gained from the use of extraterrestrial materials in large-scale space operations. The conditions under which such advantages are possible have been examined. The basis for the evaluation is an examination of the mass of Earth-origin supplies and equipment to produce and transport extraterrestrial materials compared to the mass of conventional Earth-origin space hardware or materials that might be displaced.

Some of the conclusions which have been reached as a result of the analyses are:

- Lunar Products for Use in LEO or GEO:
  - Lunar O_2 or some nonconventional form of propulsion (e.g., an electromagnetic launcher) is required for transportation in order for any product to be practical.
  - Products whose performance is directly proportional to mass (e.g., lunar O_2 for propulsion, lunar products for radiation shielding) appear to have a high probability of significant advantage over terrestrial competition.
  - Structural metal-alloy parts are not likely to be viable early candidates and will be viable only if their performance closely approaches terrestrial competition.
  - Components made from unrefined or partially refined lunar surface material ("lunar bricks") whose function is primarily structural are very unlikely to be competitive.

    - Products which require production facilities and processing aids or reagents on the Moon in excess of a few percent (i.e., 1%-3%) of the life cycle product mass are very unlikely to be competitive. This would appear to preclude consideration of (for LEO use):
        - Conventional silicon photovoltaics, which require extensive facilities to process, purify, and form the high-purity silicon crystals and to fabricate the photovoltaic devices and arrays.
        - Wet chemical processes unless the facility mass is small, and reagent recovery and recycling approaches 100%.
- Asteroidal products for use in Earth orbit require less transport energy than from the Moon. While this difference may improve the potential advantages of extraterrestrial materials, it is unlikely to result in conclusions which are significantly different from those regarding lunar products.
- In the case of lunar products for use on the Moon, the conditions under which an advantage can be shown over terrestrial competition are significantly less constrained.
    - Lunar products are competitive even with a performance disadvantage of 20 to 100 compared to terrestrial counterparts, which appears to allow the use of:
        - Ceramic structural parts made from lunar surface material.
        - Metal structural/mechanical products with relatively "crude" composition and shape.
    - For products where performance is equivalent to terrestrial competition, total imports. including production facility, energy supply, and expendable processing aids/reagents can approach the life cycle mass of the product. (This condition applies only for products with a large potential market on the Moon.)
- Manned operations will be practical only for initial exploration and possibly for nonroutine maintenance; even with these constraints, manned operations will be practical only for a large quantity (e.g., 100 tons[1]/year) production.
- Extraterrestrial materials will be practical only for operations requiring large quantities (tons/year) of individual products.

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[1] Metric tons (10^3 kg) throughout this Section.

8.2 ANALYTICAL METHODOLOGY

There may be advantages to be gained from the use of extraterrestrial materials in large-scale space operations, including the advantage of significant reduction in costly launch of mass from Earth. Numerous studies and papers have been completed on the pros and cons of extraterrestrial materials including economics and the effects on world energy status (for example, Refs. 8-1 through 8-6).

It is not the intent here to critique or criticize any of these studies or their results. The intent is to examine and compare some options for products and the processes for their production. The basis for this comparison is extraterrestrial vs. terrestrial mass for equivalent function or performance. While the principal emphasis has been on lunar-derived products for use in LEO, similar calculations but in less detail have been made for lunar products for GEO and for asteroidal products for both LEO and GEO.

The comparison here is based strictly on mass differences and is therefore independent of Earth-to-LEO launch-vehicle technology and of current or future assumptions regarding payload cost.

Obviously excluded from this analysis are important factors like inflation, return on investment (ROI), societal benefits, national defense strategy, etc. Because space materials production will be a long-lead-time endeavor, conventional economic considerations (e.g. ROI) will tend to shift the balance in favor of conventional use of terrestrial materials. Societal factors such as preservation of the environment or the desire to establish space industry might shift the break-even toward favoring extraterrestrial materials.

The intent here is not to make a decision or recommendation regarding the benefit of space materials. The intent is to examine the sensitivity of the tradeoff to various factors so that the research program addresses the appropriate key issues and generates information that can lead to objective decisions in the future.

8.2.1 Mass Comparison

In the analyses, comparisons (difference or ratio) are made between the total Earth-origin mass necessary to produce and transport extraterrestrial products and the Earth-origin mass required to perform the same function without extraterrestrial materials. For convenience and consistency, mass in low Earth orbit (LEO) is used as the baseline quantity, independent of location of use.

Since the products of relatively simple extraterrestrial processing facilities are likely to be significantly different than the products of the elaborate terrestrial industrial infrastructure, the appropriate comparison of mass per performance is included. In some of the examples, the performance ratio required to break even is the important result of the comparison (i.e., how good does the extraterrestrial product have to be to be competitive?)

The total mass associated with production and transport of extraterrestrial products is defined to include the total Earth-origin mass which must be delivered to LEO in order to:

A - Carry out "indirect" missions as necessary in order to: find and assay resources (raw materials); develop and verify processes (including installation and operation of pilot plants as required); and install, assemble, and start-up processing and power equipment.

B - Operate, maintain, and repair processing and power equipment.

C - Provide Earth-origin components of the processing equipment.

D - Provide Earth-origin components of the power supply.

E - Provide supplies (e.g., expendables and consumables).

F - Provide equipment to transport processing equipment, power supply, and supplies from LEO to the operating location.

G - Provide propellants to transport processing equipment, power supply, and supplies from LEO to the operating location.

H - Provide equipment to transport the extraterrestrial products from point of production to point of use.

J - Provide propellants to transport the extraterrestrial products from point of production to point of use.

The total Earth-origin mass which would be required to be delivered to LEO in lieu of extraterrestrial materials includes:

U - Mass of the competing terrestrial product.

V - Equipment to transport terrestrial products from LEO to point of use (if other than LEO).

W - Propellants to transport terrestrial products from LEO to point of use (if other than LEO).

It is useful to also define:

K - The mass of extraterrestrial product delivered. (Note that only the equipment necessary for production and transport--not the extraterrestrial material itself--must be launched from Earth into LEO. Therefore K does not appear in Equation 1 below. The terrestrial competition (U) does require launch and therefore appears in Equation 8-1.)

M - The total mass which must be put into LEO in order to find, produce and deliver extraterrestrial product or products for some mission or function; therefore $M = \Sigma A\text{-}J$.

η - Product performance ratio (comparison of the performance per unit mass of an extraterrestrial product to that of terrestrial competition). For equal performance $\eta = U/K$ (see Section 8.3 for discussion).

Note that there has been no attempt to use the usual letter/definition association (e.g., F/fuel, P/power) in the above designations. Instead, to facilitate mental association of the large number of terms in subsequent discussions, the initial part of the alphabet, A-K, has been assigned to factors associated with space materials and later segments of the alphabet, U-W, represent factors associated with the competing conventional Earth materials.

For clarity, it is useful to express the central theme of this analysis in both narrative and mathematical terms. For this study, it is defined that there is a potential advantage to be gained from the use of extraterrestrial materials when, for equal function or performance:

The Earth-origin mass of facilities, equipment, supplies, etc. to produce and transport an extraterrestrial product
- is equal to or less than -
the mass of conventional Earth-origin space hardware that can be displaced.

or: The number of Shuttle (or other launch vehicle) launches to put Earth-origin equipment, facilities, supplies, propellant, etc. in space to produce and transport an extraterrestrial product
- is equal to or less than -
the number of launches to put conventional equipment in space to do the same job

or:

$$\Sigma A\text{-}J \leq \Sigma U\text{-}W \quad (8\text{-}1)$$

Note that the mass of the extraterrestrial product, K, does not appear in Equation 8-1. Mathematically, Equation 8-1 compares what it takes to produce the extraterrestrial product to what it takes to do the same job with conventional Earth-origin products. In practice, some of the factors necessary to produce and transport extraterrestrial products are proportional to the mass of such products. Therefore, the Earth-origin mass is influenced by, but does not include the mass of the extraterrestrial product.

Rigorous simultaneous treatment of all of the terms in Equation 8-1 is possible and will likely be done in the future. However, at the current status of the program, maximum perspective can be gained by examination of the individual and selected groups of terms.

If we initially consider that production and use of extraterrestrial materials will be practical only in large quantities, and that they will be produced by automated, reliable equipment, we can temporarily ignore "indirect" missions (A) and maintenance (B). The impact of these terms is examined later in Section 8.9.

If we further consider the case of conventional chemical propulsion, with the necessary reusability, then the equipment to transport the production facility (F), the extraterrestrial products (H), and the terrestrial competition (V) will be small in comparison to the other terms and can be neglected in the initial considerations. For some cases involving "advanced technology" (e.g., electric propulsion, electrostatic accelerators) these terms cannot be ignored and are appropriately considered in the analyses and results described in Section 8.5.

All of the remaining terms can be considered to be proportional to either the mass of the extraterrestrial product or to the mass of the terrestrial product displaced. It is convenient to use the equivalent lower case letters to represent the factors as a function of extraterrestrial or terrestrial product mass. For example, while C has been defined as the total equipment mass, then c is the mass of equipment per mass of product delivered and $C = cK$; similarly $W = wU$. With these definitions we can expand and rewrite Equation 8-1 for conditions where there is an advantage to extraterrestrial materials as:

$$K(c + d + e + g + j) < U(1 + w) \qquad (8\text{-}2)$$

Both the difference and ratio of masses are valid and can provide useful comparisons in the analysis; the obviously equivalent equation is:

$$\frac{U(1 + w)}{K(c + d + e + g + j)} > 1 \qquad (8\text{-}3)$$

Since the performance ratio (η) has been defined as the ratio of masses for equal performance, $\eta = U/K$, then the equations become

$$\eta(1 + w) > c + d + e + g + j \qquad (8\text{-}4)$$

Using an analogous definition, m is the mass in LEO per unit mass of product delivered and $M = mK$. Therefore (still considering A, B, F, and H insignificant),

$$m = c + d + e + g + j \qquad (8\text{-}5)$$

Thus, from Equations 8-4 and 8-5 there is a potential advantage for extraterrestrial materials whenever m is less than $\eta(1 + w)$. For the specific case of lunar O_2 delivered for LEO, which will be used extensively in these analyses, there is a potential advantage whenever m is less than 1.0. It is important to note that the definition is "potential advantage"; whether the margin of advantage is sufficient to warrant use of space materials depends on many factors.

8.2.2 Lunar Oxygen as Propellant

For the case of lunar O_2 is not only a candidate product, it can also serve as a propellant. The output of a small initial facility on the Moon can be used to transport equipment and supplies from LEO to the Moon for facility expansion. Furthermore, in combination with terrestrial H_2 or as propellant for electric propulsion, it can be used to transport O_2 or any other payload to LEO, GEO, or elsewhere.

For the cases of lunar O_2 for transport propellant as well as payload, it is useful to redefine the propellant terms defined above as follows:

g' = terrestrial propellant for transport of equipment (and supplies) from LEO to the Moon, tons per ton of delivered equipment.

g'' = lunar O_2 for transport of equipment and supplies from LEO to the Moon, tons per ton of delivered equipment.

j' = terrestrial propellant to transport extraterrestrial products to the point of use, tons per ton of extraterrestrial product.

j'' = lunar O_2 for transport of extraterrestrial products to the point of use, tons per ton of extraterrestrial product.

It is also convenient to define some additional terms for use in this analysis:

g^* = the mass of equipment and supplies in LEO that must be transported to the Moon to produce g'' (the O_2 required for transport).

χ = the mass of equipment, power supply, and supplies on the Moon, tons per ton of O_2 produced.

β = total Earth-origin mass (equipment, supplies, plus propellant) in LEO per ton of equipment and supplies delivered to the lunar surface. Since lunar O_2 can be used, β can be a function of χ.

Note that while all of the terms used earlier were defined in terms of mass in LEO, the terms g'', j'', and χ are defined as mass on the lunar surface.

Three specific cases have been examined (see Section 7.2) for transport from LEO to the lunar surface: 1) all terrestrial propellants; 2) an intermediate case using all terrestrial hydrogen and a combination of lunar and terrestrial oxygen and 3) minimum terrestrial propellant, with terrestrial hydrogen and all oxygen from the Moon. The latter two cases assume an initial small facility to produce oxygen to transport the equipment to increase the production capacity. For all cases, it is assumed that the transport vehicle masses are insignificant and that the propellant masses dominate. Then an incremental increase in facility on the Moon is χ and the equivalent mass in LEO is $\chi\beta$. The mass in LEO is also equal to the equipment plus the propellant, and in the latter two cases, plus the equipment and supplies to produce the required quantity of lunar O_2 or:

$$\chi\beta = \chi + \chi g' + \chi g^{*} \qquad (8\text{-}6)$$

or

$$\beta = 1 + g' + g^{*} \qquad (8\text{-}7)$$

The analyses of transport of equipment and supplies to the Moon (see Section 7.2) yield results in terms of lunar O_2 propellant on the Moon per ton of equipment and supplies transported (defined above as g'') rather than equivalent mass in LEO, g^{*}, which appears in Equations 8-6 and 8-7 above. Therefore it is necessary to establish the relationship between g'' and g^{*}. If we assume that the efficiency of an initial small facility will be the same as that of a full-scale one, and that transport from LEO to the Moon is optimized, then the definitions of χ and β above apply for most of the facility installation and the relationship can be defined as:

$$g^{*} = \chi\beta g'' \qquad (8\text{-}8)$$

then Equation 8-7 becomes:

$$\beta = 1 + g' + \chi\beta g'' \qquad (8\text{-}9)$$

or

$$\beta = \frac{1 + g'}{1 - \chi g''} \qquad (8\text{-}10)$$

For the three cases examined, propellant requirements calculated as shown in Section 7.2 and illustrated graphically in Figure 8-1 were:

$g_1' = 6.25$; $g_1'' = 0$ (all terrestrial)

$g_2' = 3.6$; $g_2'' = 2.5$

$g_3' = 2.0$; $g_3'' = 12$ (Minimum terrestrial)

therefore for the three cases, the equivalent values of β are:

$$\beta_1 = \frac{7.25}{1 - 0} = 7.25$$

$$\beta_2 = \frac{4.6}{1 - 2.5\chi}$$

$$\beta_3 = \frac{3.0}{1 - 12\chi}$$

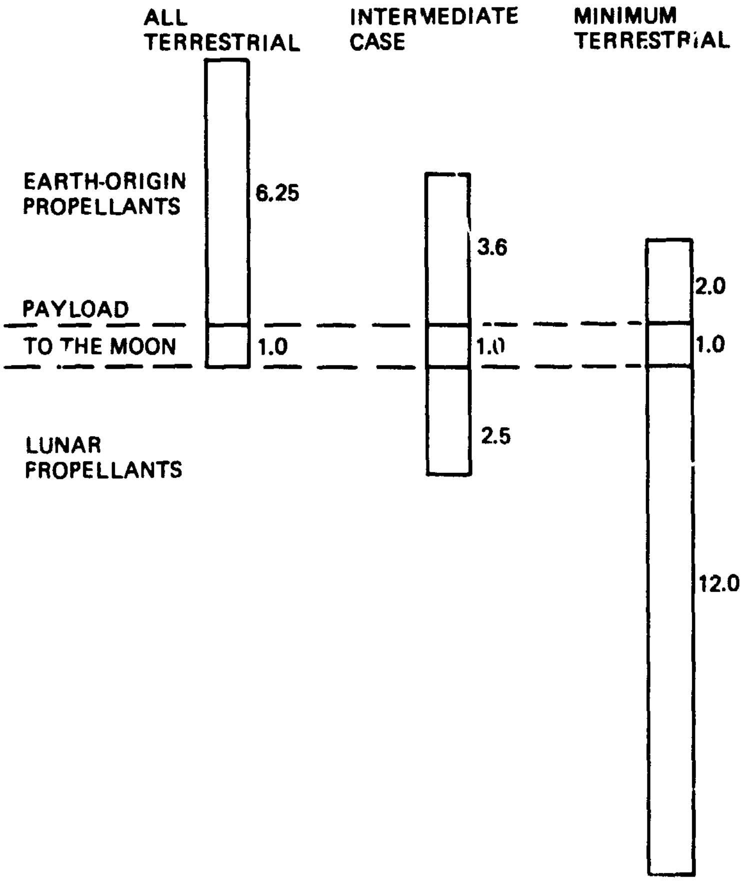

Figure 8-1. Transportation of Equipment and Supplies to the Lunar Surface--Propellant Source Options

The benefit of reducing terrestrial propellant by increasing the use of lunar propellant depends on how "difficult" it is to produce the lunar material--that is, how much equipment and supplies are required. The effects are illustrated in Figure 8-2, which shows the values of β as a function of χ. The minimum terrestrial case (β_3) is obviously the best when the mass of equipment and supplies required for production is small but becomes less attractive rapidly as the mass required for production increases. Using some terrestrial oxygen (β_2) reduces the sensitivity to the mass of the factory. The selected intermediate case, β_2, is illustrative but is not necessarily the optimum.

8.2.3 Cost Considerations

While mass, not cost, is the basis of this analysis, some simplified discussion of cost is appropriate to indicate that comparison of launch mass is a valid basis to initially compare and limit options.

The three major cost elements of space activities are acquisition (including design, fabrication, test, etc.), launch, and mission operations support. The analytical approach described here deals with launch mass and therefore launch costs. However, the calculations, comparisons, and conclusions are independent of the absolute value of the launch cost.

Launch costs have been, and will continue to be a major fraction of the total cost of any space activity. While the $/kg for launch are expected to decrease with time, factors such as larger system size and reduced testing associated with opportunity for repair will tend to reduce acquisition costs (per unit mass) as well.

It is useful to compare the magnitude of some estimated launch costs and some example acquisition costs.

The out year Shuttle cost in current year dollars is ~$\$10^8$ mission ($100M) (Ref. 8-7) with a projected payload capability of 30 metric tons or a launch cost of $3,300/kg. If we assume, as a limit case for this comparison, an order of magnitude improvement via a combination of vehicle, payload, and operating improvements, future launch costs may be in the range of $\$3 \times 10^2$ to $\$3 \times 10^3$/kg.

Acquisition costs of one-of-a-kind, high-performance spacecraft in the pre-Shuttle era were in the range of $\$10^5$ kg to well over $\$10^6$/kg. Many factors, including larger system size, experience and the opportunity for repair will tend to reduce unit acquisition costs. Obviously costs will not approach the $10/kg to $\$10^2$/kg typical of products like mass-produced automobiles, consumer electronics and commercial aircraft.

Some photovoltaics are currently in the hundreds of dollars per watt (>$\$10^4$/kg). However, the goal of $0.70/watt for commercial terrestrial photovoltaics appears achievable. Thus, with technology developed for terrestrial photovoltaics and for the large quantities required for processing extraterrestrial materials, $10/watt or ~$\$10^3$/kg seems a reasonable projection. Since a photovoltaic power plant is likely to represent a significant fraction of the mass of extraterrestrial processing facility, the estimated $\$10^3$/kg is a useful comparative value.

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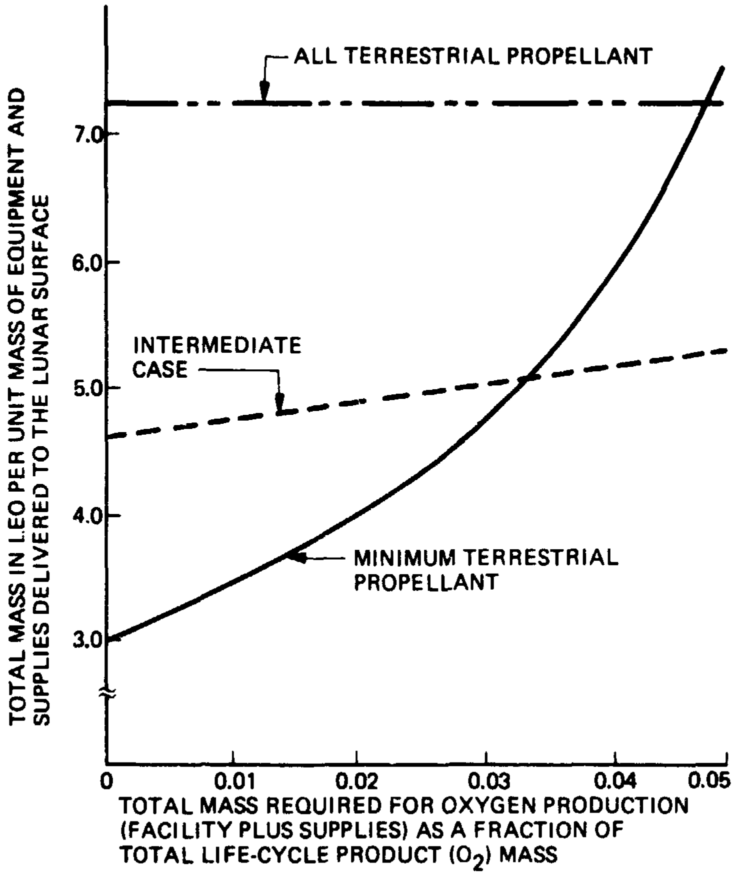

Figure 8-2. Transportation of Equipment and Supplies to the Lunar Surface, Sensitivity to Oxygen Production Facility Mass

Past and current acquisition costs of communications satellites indicate a range of \$2 x 10^4 to \$5 x 10^4/kg in orbit (Refs. 8-10, 8-11). Since the mass of such satellites includes a large fraction of high-reliability electronics and precision pointing and station keeping capabilities, the acquisition costs could represent an upper bound of probable cost of processing equipment.

Since it is necessary to launch 3-5 kg into LEO for every kg to be delivered to GEO, for the projected Shuttle launch costs \$3 x 10^3), launch costs (3-5 times \$3 x 10^3) approach those of acquisition. Even if the limit case of \$3 x 10^3/kg could be achieved, acquisition costs would likely decrease so that launch costs would still be a significant fraction of the total.

Thus, comparison of launch mass is a valid, but not necessarily sufficient, basis for comparison. Until other factors are more solidly established through the current research, launch mass represents the only valid basis for comparison.

8.3 PRODUCT PERFORMANCE CONSTRAINTS

8.3.1 Product Performance Ratio

The products derived from processing of extraterrestrial materials in space are likely to be different in one or more characteristics from the terrestrial counterparts which they are intended to displace. In order to evaluate potential utility, it is useful to use the Product Performance Ratio (η) defined in Section 8.2.

Simply, the ratio, η, can be used as a quantitative definition of 1) how "good" a particular extraterrestrial product is, relative to its terrestrial competition, or 2) how good a product or class of products must be to be competitive.

For example, if the extraterrestrial product for a particular function (e.g., load bearing) requires four times the mass for the equivalent function by a terrestrial product, the performance ratio $\eta = U/K = 1/4 = 0.25$. The numerical value of the performance ratio is the same, independent of whether the factor of 4X is the result of material properties (e.g., inability to achieve optimum alloy composition or heat-treatment) or is the result of inability to maximize the use of material properties such as by removal of excess material from lower stressed regions during part fabrication.

If we consider the structural performances of components fabricated from unrefined or partially refined lunar surface material, compared to high-performance aerospace components producible on Earth (e.g., metallic alloy and/or fiber-reinforced composite parts), the performance ratio will obviously be very small.

At the opposite extreme, a propulsion system would use lunar-produced oxygen just as effectively as it would the terrestrial counterpart, and thus lunar oxygen would have a performance ratio of 1.0. Similarly, radiation shielding made from lunar material might have a performance ratio approaching 1.0.

In the intermediate case of components made from lunar-derived metals, the conclusions are less obvious. In this regard, it is useful to note that there is typically a factor of 2 or greater difference between unalloyed metals and their structural counterparts and another factor of 2 to 6 between heat-treated and nonheat-treated conditions. Furthermore, for conventional metallic space hardware significant improvement in function per unit mass is typically achieved by removing excess materal in lower-stressed regions.

8.3.2 Lunar Materials for Use in LEO

Considering the general case of lunar materials for use in low Earth orbit (LEO), some general constraints can be established. Since the terms have been defined as mass in LEO, for the specific case of use in LEO, the equipment and propellant for delivery of terrestrial material, V and W in Equation 8-1, become zero.

Several cases have been examined for transportation of lunar products to LEO using conventional chemical propulsion and aerobraking for LEO orbit insertion. Details are described in Section 7.2 and are summarized in Table 8.1. For the baseline, Case 3, each ton of payload delivered from the Moon to LEO requires 0.44 tons of terrestrial H_2 (and 2.67 tons of lunar O_2) for propulsion. As a first approximation we can assume that the equipment to produce and transport the payload and lunar oxygen (c, d, e, and g in Equation 8-4) is insignificant. Since j, the remaining term in Equation 8-4, includes 0.44 terrestrial H_2 the minimum performance ratio at which lunar material would have an advantage is:

$$\eta > 0.44$$

To examine the possible effects of improved chemical propulsion technology, a limit case (see Section 7.2, Case 4 in Table 8-1) has been used. For this limit case, it has been assumed that the transport equipment mass is zero and therefore terrestrial and lunar propellants are required only to transport themselves and the payload. Under these circumstances, j in Equation 8-4 is 0.22 and if we continue to assume that the other terms are insignificant, then the minimum break-even performance ratio is:

$$\eta > 0.22$$

Under these circumstances, the following conclusions appear reasonable.

- Products whose performance is directly or nearly proportional to mass are the most likely to show an advantage.
    - Lunar 0^2 for space-vehicle propulsion, including orbit raising
    - Lunar products for radiation shielding
- Structural metal alloy parts may be viable candidates, but only if they can be made in ways so that their performance closely approaches terrestrial competition.

Table 8-1. Selected Examples of Lunar-to-LEO Transportation

Case	Conditions	Propellants				Lunar Payload to LEO	Payload Ratio[b]
		Terrestrial		Lunar			
		O_2	H_2	O_2	H_2[a]		
1	Baseline OTV All Terrestrial Propellant[c]	321[c]	53.1	0	0	66.9	0.18
2	Baseline OTV, LTV Combined Terrestrial and Lunar O_2	26.7	17.9	80.5	0	66.9	1.50
3	Baseline OTV, LTV Minimum Terrestrial Propellant	0	17.9	107.2	0	40.2	2.25
4	Limit Case Zero Dry Mass OTV and LTV	0	13.8	82.7	0	62.7	4.54
5	Assume Lunar H_2 Available[a]	0	0	89.8	14.9	54.1	∞

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[a] Lunar H_2 has been postulated (Ref. 8-7), but its existence is uncertain; results shown for illustration only.

[b] Payload ratio is defined as tons of payload delivered (from the lunar surface to LEO) per ton of terrestrial-origin propellant required in LEO to make the delivery.

[c] Requires multiple OTV trips and staging in LLO.

> Components made from unrefined or partially refined lunar surface materials, whose function is primarily structural, are very unlikely to be competitive with terrestrial counterparts.

Products which have an advantageous performance ratio, but which are required in relatively small quantities (e.g., oxygen for life support at ~300 kg/person year), will not be advantageous unless produced and transported in conjunction with a high-volume product. Otherwise, "indirect" missions (A), transport equipment (F and H), etc., will be too large in proportion to the quantity of the product required.

The significance of alloying, heat-treating, and final machining of metal components becomes evident in comparison to the minumum break-even ratios of 0.22 and 0.44. While alloying, heat-treating, and machining extraterrestrial metal products is possible, additional facilities, energy, supplies and transport (terms C, D, E, F, and G in Equation 1) would be required.

8.3.3 Lunar Materials for GEO

The equivalent break-even performance-ratio values for lunar material in GEO are somewhat more advantageous. As shown in Section 7.2 with current conventional chemical propulsion, $\eta > 0.2$ is possible. Even with advances and optimistic estimated $\eta > 0.1$, conclusions similar to these for Lunar/LEO hold. There would be a larger margin for O_2 and radiation shielding, and somewhat more margin for nonoptimum metal parts. Structural use of lunar ceramic parts remains highly unlikely.

8.3.4 Lunar Materials for Lunar Use

Obviously, for lunar products to be used on the Moon, constraints are significantly different. However, the more advantageous conditions must be considered in the context of probable large-scale lunar operations and the resulting "market" for lunar products. For lunar use, the propellant for product transport, j in Equation 8-4, is zero (or at least relatively insignificant) and Equation 8-4 can be rewritten as:

$$\eta > \frac{c + d + e + g}{1 + w} \qquad (8\text{-}11)$$

If we define χ' as tons of equipment and supplies in place on the Moon to produce any product per ton of any product (previously χ had been specifically related to oxygen) then the mass in LEO ($\chi'\beta$) is the mass necessary for production (c + d + e) plus the mass for propellant, g, or $\chi'\beta = c + d + e + g$. We can also consider that the transport of the competing terrestrial product can be considered equivalent to the transport of equipment for production or in terms of tons of terrestrial product to the Moon, $\beta = 1 + w$. Then Equation 8-11 becomes

$$\eta > \frac{c + d + e + g}{1 + w} = \frac{\chi'\beta}{\beta} = \chi' \qquad (8\text{-}12)$$

As will be shown later in Section 8-8, relatively elaborate facilities and power (χ') for production of lunar O_2 at significantly less than 0.05 appears feasible. For production of structural silicate shapes, facility mass to product mass <0.01 appears possible. Therefore, on the Moon, lunar products are competitive even with a performance disadvantage of 20 to 100 compared to terrestrial products.

From Equation 8-11 it is evident that there is a potential advantage to extraterrestrial materials so long as the ratio of the mass of imported equipment and supplies to the mass of product is equal to or less than the product performance ratio (η). Thus for high-performance products, processes and products requiring major facilities (e.g., photovoltaics, wet chemical processes) may become viable. The advantage is possible only if the volume of the market for the product is large enough; otherwise the fixed mass items like "indirect" missions (A), operations (B), and transfer vehicles (F) are likely to become dominant factors.

For example, it is useful to consider an optimistic market of 500 metric tons per year of lunar O_2 for low Earth orbit which in turn generates a market for lunar silicon photovoltaics to produce the oxygen. The power plant to produce the 500 tons/year plus another ~1000 tons of O_2 to deliver the payload for LEO would require a capacity of ~3 x 10^3 kw_e. With current photovoltaic technology such a plant would have a mass of 40-50 tons; with advanced technology, it might be as small as 10-20 tons. Such a level may or may not be sufficient to justify installation of a plant to produce photovoltaics on the Moon (see Section 8-9).

8.4 CONVENTIONAL PROPULSION--MOON TO LEO

So far we have considered only the terrestrial propellants required for transportation and established minimum performance based on that simplification. It is useful to examine in more detail the conditions under which it is advantageous to transport lunar products from the Moon to LEO. The following calculations are based primarily on lunar oxygen transported to LEO for orbit raising. The assumptions are current state of the art H_2/O_2 propulsion technology with the addition of aerobraking for all return to LEO trajectories. Delivery to LEO was examined for minimum terrestrial propellant (all H_2) from Earth and all O_2 from the Moon as a function of the "dry weights" of the Moon-LLO and the LLO-LEO transfer vehicles, as described in Section 7-2.

If we assume the payload to be oxygen, with η = 1.0, and the mass of the transfer vehicles to be insignificant to the total mass (except as they affect transport propellant consumption), then there is an advantage to the lunar product when:

$$\eta = 1 > m = \chi\beta\,(1 + j'') + j' \qquad (8\text{-}13)$$

With conservative values for transfer-vehicle dry weights from detailed studies, j' is 0.44 tons of terrestrial hydrogen in LEO per ton of lunar oxygen delivered and j" is 2.67 tons of lunar O_2 consumed for every ton of O_2 delivered. Equation 8-13, is plotted in Figure 8-3, which shows m as a

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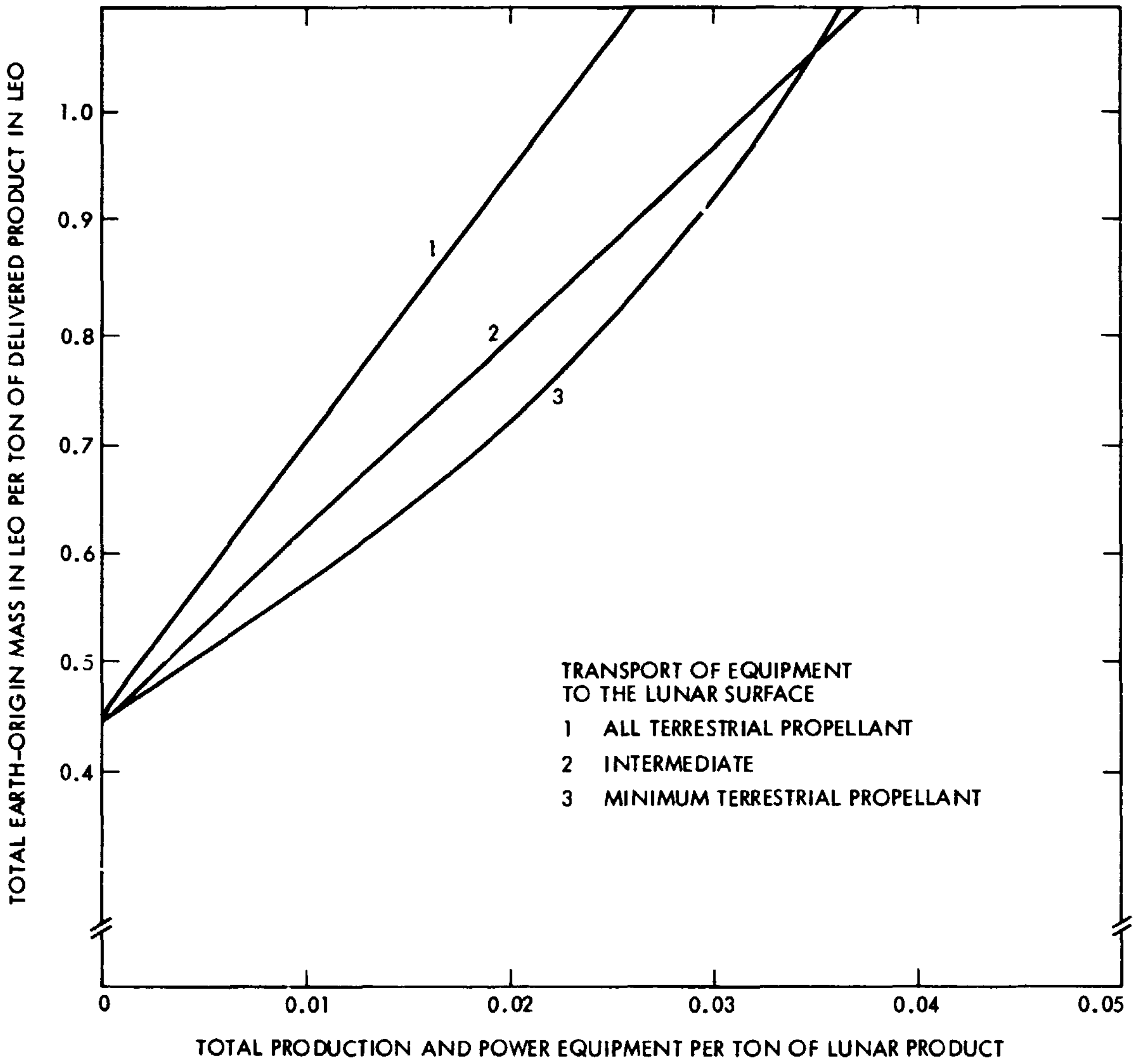

Figure 8-3. Potential Advantage of Lunar Payload Deliver to LEO:
Nominal Propulsion System Case

function of χ (facility and supplies per ton of product) for the three values of β described in Section 8-2 above. The range of probable values of the parts of the facility and the total, χ, are discussed in Section 8-8 of this report.

For the limit case of zero "dry weight" for the LLO-LEO and LLO/lunar surface transfer vehicles, one ton of lunar O_2 payload delivered to LEO requires 0.22 tons of terrestrial H_2 and 1.32 tons of lunar O_2 for transport. Equation 8-13 is plotted in Figure 8-4, which shows m as a function of χ, with these limit values of j' and j" and for the three values of β. Note that the β values are not lower limit cases.

It is possible to make some order of magnitude calculations for power and processing equipment, as discussed later in Section 8.8. However, reduction in the uncertainties in these quantities must be a major goal of the research project. In Equation 8-13, the lunar surface mass, χ, is compared to terrestrial origin mass that the lunar product can displace. Note that the value χ has multipliers, β, ranging from 3 to 7.25 and (1 + j") which has values from 2.3 to 3.7. The same multipliers (~7 to ~30) apply also to uncertainties in χ.

Similar calculations and conditions are possible for products other than O_2 propellant. Processed lunar material for use as radiation shielding might have a performance ratio (η) near 1.0 and would likely require less elaborate facilities and power supply than O_2. Some metal alloy shapes might approach 1.0; the facilities and power are uncertain, but probably not significantly greater than for O_2 production.

8.5 NONCONVENTIONAL PROPULSION--MOON TO LEO

The conclusions and constraints shown in the Sections above are based on conventional H_2/O_2 propulsion technology, with the essential addition of aerobraking for return to LEO. Since the use of extraterrestrial materials is at least a decade or two away, it is reasonable to examine the impact of major improvements in propulsion on the viability of such use.

It is entirely possible that some time in the future, a variety of scenarios may be practical like: electromagnetic-launcher delivery directly to L-points or to LEO; nuclear propulsion with lunar propellant; metal/oxygen rockets; matter/antimatter propulsion. However, for this study, somewhat more conservative advanced technology assumptions are more appropriate.

Lunar-surface to LLO using an electromagnetic launcher and LLO to LEO using electric propulsion are examined individually, with conventional chemical propulsion for the other leg in each case. In addition, a combination of electromagnetic launch to LLO and electric propulsion from LLO to LEO was examined.

The mass of transport equipment, considered insignificant for chemical propulsion, can become a major factor for these cases. For purposes of this analysis, it is assumed that the entire mass of the electromagnetic launcher is transported from Earth, while in fact, it might be possible to construct portions of it from lunar materials. The mass of the electric propulsion system is dominated by the power supply for the thruster and can be reasonably estimated.

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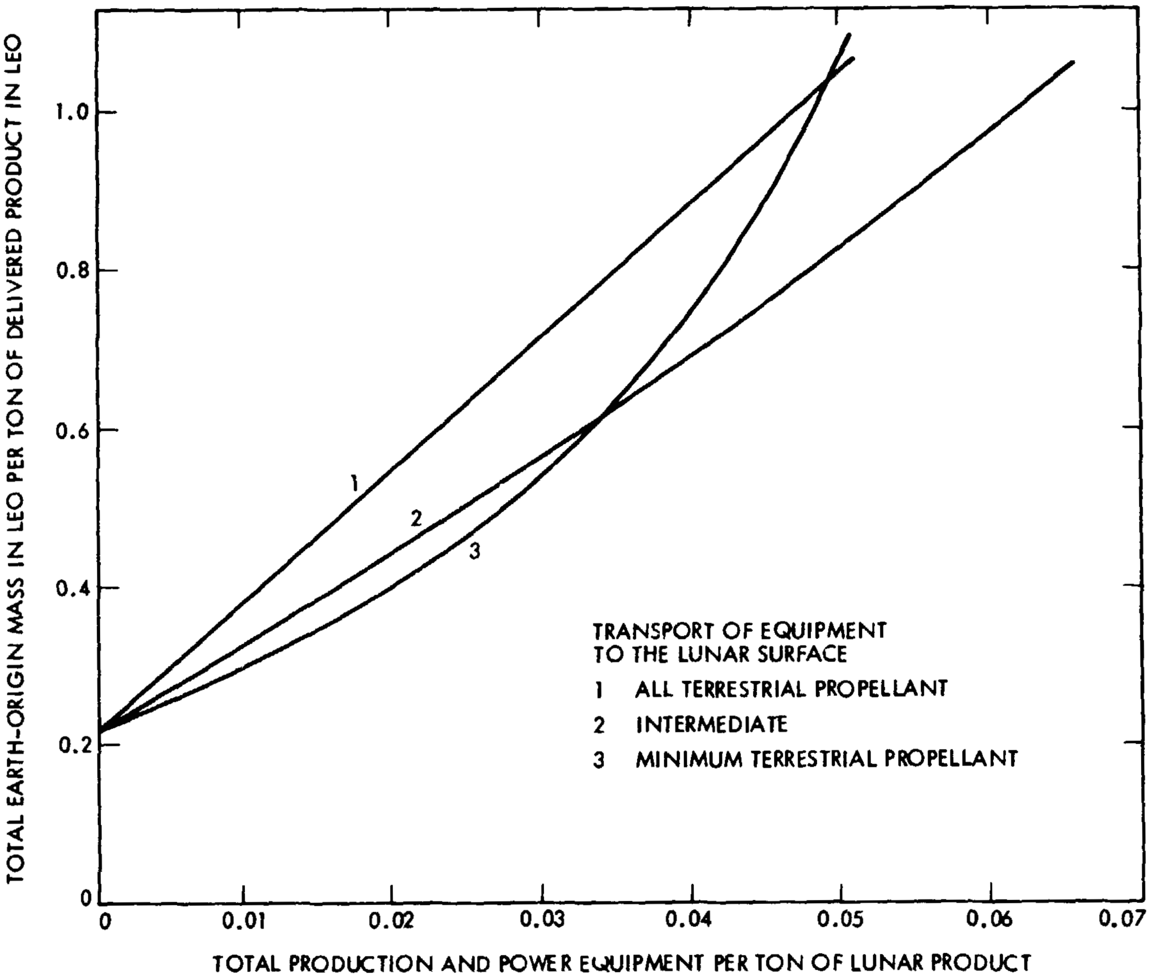

Figure 8-4. Potential Advantage of Lunar Payload Delivery to LEO; Limit Case Propulsion Systems

Since the mass of transport equipment and propellant can be related to the mass of delivered product, we can use Equations 8-2 through 8-4, retaining the definition of terrestrial propellants (j') and lunar propellants (j") for product delivery, and redefining transport equipment (H) for product delivery as:

H' = mass of LEO/LLO orbit transfer vehicle; h' = transfer vehicle mass per mass of product (including the power system in the case of the electric version).

H" = mass of lunar-surface/LLO transfer vehicle; h" = transfer vehicle mass per mass of product (including the power system in the case of the electromagnetic launcher).

With these definitions for transport of lunar material to LEO (w = 0), Equation 8-4 becomes:

$$n > \beta(c + d + e) + h' + \beta h'' + j' + j'' \qquad (8\text{-}14)$$

(The use of β, the ratio of total mass in LEO to mass delivered to the Moon, is convenient in lieu of considering F and G for equipment transport.)

8.5.1 Electromagnetic Launchers (Mass-Driver and Rail-Gun Systems)

The details of candidate mass-driver and rail-gun systems are described in Section 7-2. Using values from Figure 7-4, some extremes of total launcher plus power supply mass can be estimated. Since the launcher mass is relatively insensitive to product mass, it must be considered relative to some typical, expected annual launch rates and lifetimes. A nominal value of h" for any large-quantity market, excluding the power supply is:

$$h'' = \frac{\text{60 tons equip.}}{\text{500 tons O}_2\text{/yr x 5 yrs}} = \text{.024 tons equip./ton O}_2 \text{ delivered}$$

At the other, optimistic extreme, 500 tons of O_2 plus metal "by-product" are launched, and using the lower bound of the launcher-mass uncertainty band, the mass might be as low as (excluding the power supply):

$$h'' = \frac{\text{30 tons equip.}}{\text{1000 tons matl/yr x 10 yrs}} = \text{.003 tons equip./ton matl delivered}$$

The power consumption for the mass driver is a function of delivered mass. Using the upper limit of the uncertainty band of 100 kW for a product mass of 10^3 tons/year and current space photovoltaic technology of 66 W/kg, the power generating equipment is 1.5×10^{-3} tons per ton of product launched. Using the lower limit of the uncertainty band and more optimistic values of future space power technology, the value becomes even smaller. Thus, the total lunar-surface/LLO transport equipment (h"), including power supply, can be estimated to be 3×10^{-3} to 2.4×10^{-2} tons/ton of lunar product.

For the case of lunar O_2 to LEO, assuming that the entire launcher is transported like the processing and power equipment, we can rewrite Equation 8-14 for conditions with advantage to lunar O_2 as:

$$n = 1 > m = \chi \beta(1 + j'') + h' + \beta h'' + j' \qquad (8\text{-}15)$$

In this case, calculations (see Section 7-2) show that, for terrestrial H_2, $j' = 0.145$ and, for lunar O_2, $j'' = 0.88$, using realistic dry-weight mass of the LLO/LEO transport vehicle. The curves in Figure 8-5 show the potential advantage of lunar material (right side of Equation 8-15) m as a function of χ for the three values of β described in Section 8.2.2, but excluding consideration of the launcher mass (h") and assuming h' is negligible.

Since the launcher is a large mass, relatively insensitive to the mass to be launched (see Figure 7-4), the viability of such an option depends strongly upon an establishment and maintenance of sufficient product volume to warrant the installation. The curves in Figure 8-5 assumed h" in χ Equation 8-15 to be negligible; Figure 8-6 shows the vital sensitivity to annual product mass. Figure 8-6 shows values of m as a function of χ (Equation 8-15) for β_1 only, and for selected values of annual product, including the β_1 curve for h" = 0 from Figure 8-5. It should be noted that the launcher is sized (h") to transport (1 + j") from the lunar surface to LLO, but the mass is "amortized" only for payload delivered to LEO.

8.5.2 Electric Propulsion for LLO/LEO

Details of electric propulsion options and variables are discussed in Section 7.2. If we assume chemical Moon to LLO transport and examine the single optimistic ("high-technology") case of payload/terrestrial propellant mass ratio of 7.7, it will illustrate the impact of total equipment for extraterrestrial material production and transport and therefore can serve as an example for other calculations that may be desired in the future.

The terms in Equation 8-14 for this case are:

c + d + e = process equipment and supplies as defined in Section 8.2

$$h' = \frac{\text{30-ton vehicle}}{\text{60 tons/load x 2 loads/yr x 5 yrs}} = \frac{\text{0.05-ton vehicle}}{\text{ton product}}$$

h" = assumed to be negligible

j' = 0.13 (terrestrial propellant for lunar-surface to LLO)

$$j'' = \frac{\text{LS to LLO + LLO to LEO}}{\text{payload}} = \frac{52{,}800 + 27{,}040}{67{,}760} = 1.96$$

For this case, the appropriate version of Equation 15, for an advantage of lunar oxygen to LEO becomes:

$$n = 1 > m = \chi\beta(1 + j'') + h' + j' \qquad (8\text{-}16)$$

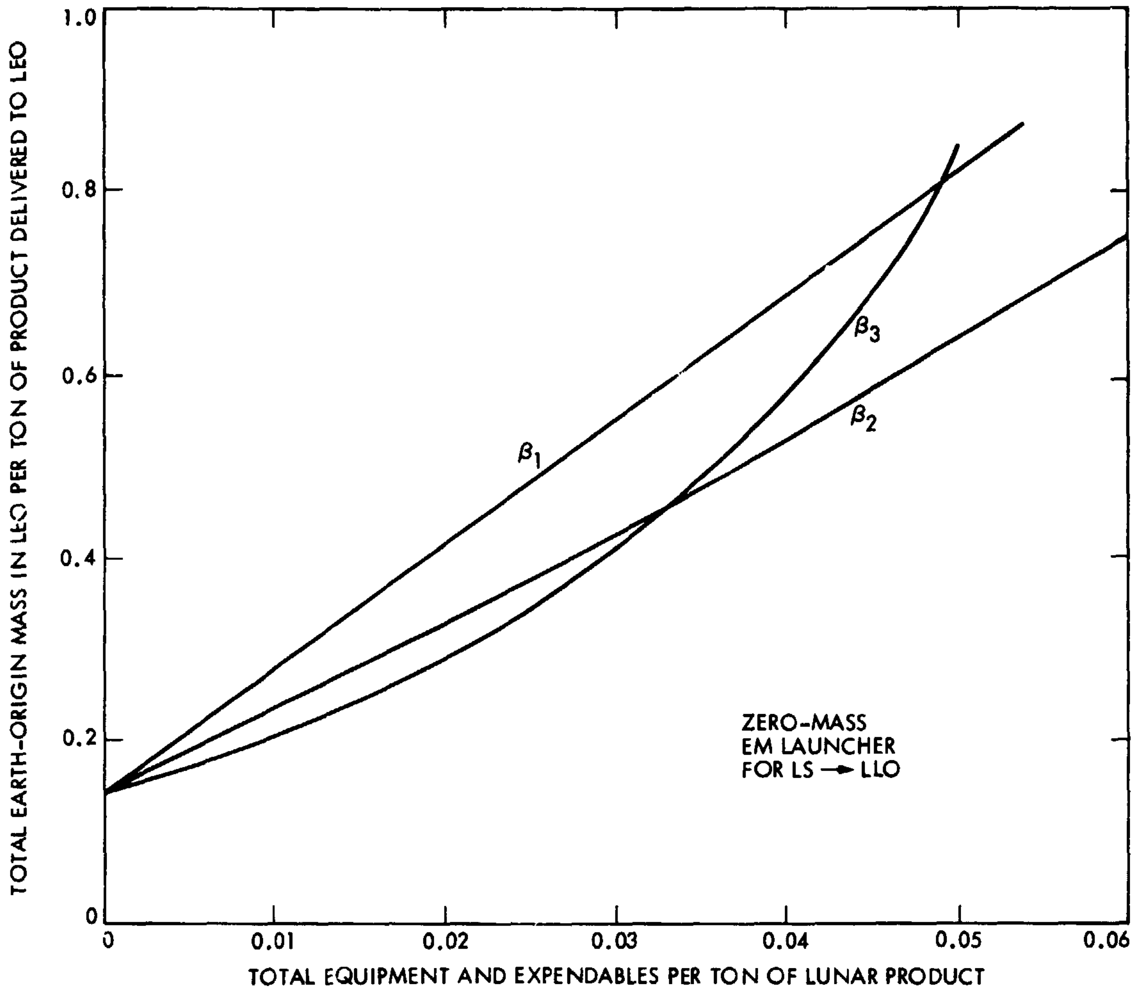

Figure 8-5. Potential Advantage of Lunar Payload Delivery to LEO; Zero Mass Electromagnetic Launcher for Lunar-LLO and Conventional Chemical Propulsion and Aerobraking for LLO-LEO

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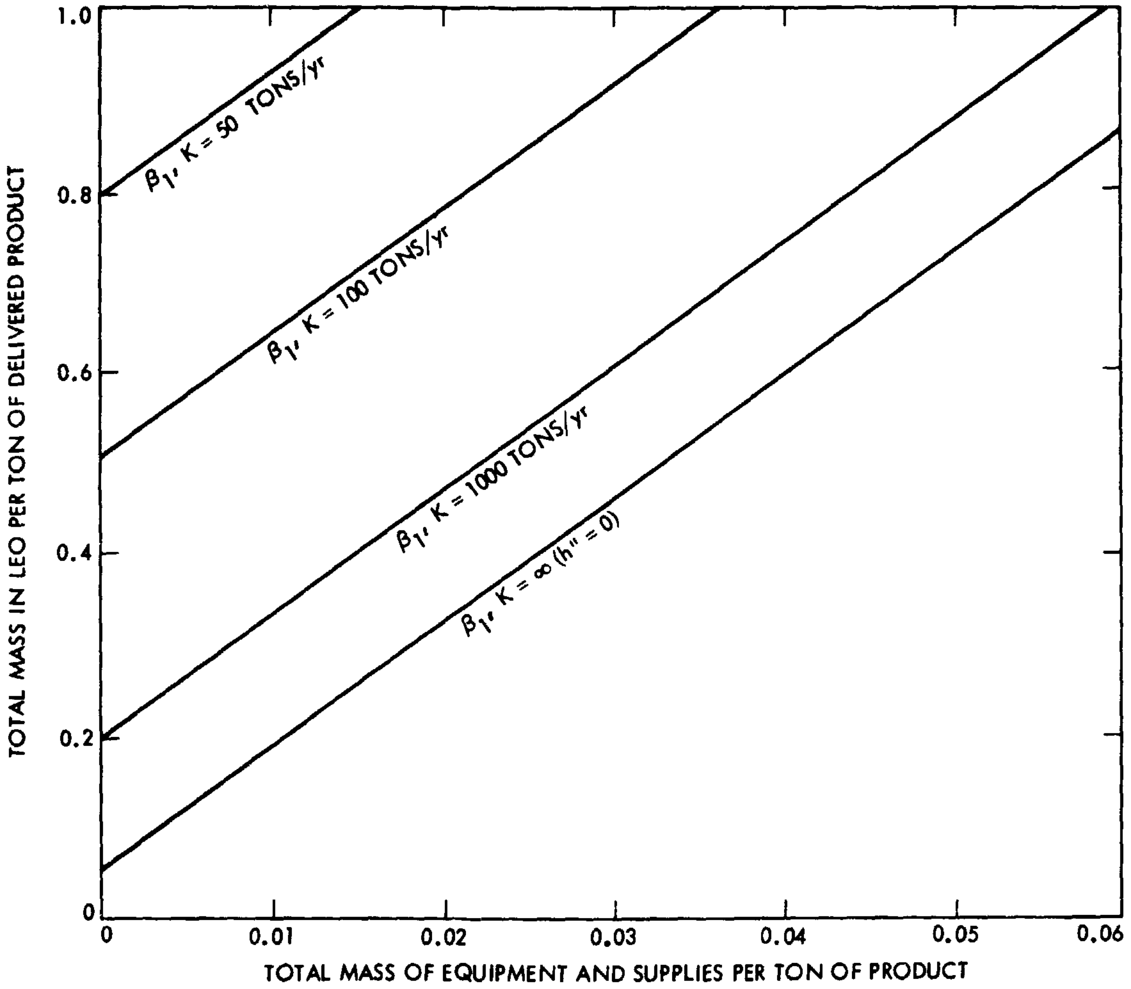

Figure 8-6. Potential Advantage of Lunar Payload to LEO Using Electromagnetic Launch for Moon to LLO and Chemical Propulsion and Aerobraking for LLO to LEO

Values of m as a function of χ for this case and for the three values of β are shown in Figure 8-7. As expected from the numbers above, the results are not much different than the zero "dry-weight" chemical case described in Section 8.4 (Figure 8-4).

8.5.3 Electromagnetic Launch and Electric Propulsion

Theoretically, with the combination, once production, power, and maintenance equipment, expendables, and transport vehicles are in place, the extraterrestrial-product/terrestrial-propellant ratio is infinite. It is useful to examine the actual potential which might be achieved with this combination as a limit case.

Considering an advantage for lunar oxygen to LEO, Equation 8-14 becomes:

$$\eta = 1 > m = \chi\beta(1 + j'') + h' + \beta h'' + j' \qquad (8\text{-}17)$$

In this case

$j' = 0$ (no terrestrial propellant required)

j'' = lunar oxygen for electric propulsion transport = 0.40

h' = 0.05 for 5-year lifetime; 0.025 for 10-year lifetime

h'' = launcher and power supply for (payload + j''); strongly dependent on product quantity (see Section 8.5.1).

Values of m as a function of χ with the above numbers in Equation 8-17 are shown in Figure 8-8 for β_1 only and for selected values of annual payload. As was evident in Figure 8-6, the potential benefit is strongly dependent on product quantity; production well in excess of 100-200 tons/year for 10 years is essential for reasonable return for the electromagnetic launch systems indicated.

The curves in Figure 8-8 are based on $\beta_1 = 7.25$ (see Section 8.2). Some additional advantage is possible for other values of β similar to the curves in other figures (for example 8-3). In addition, there may be further advantage to an initial small facility and subsequent expansion via equipment transported with the electric-propulsion module. Such an approach could result in a significant reduction in the long-term net value of β and since β is a multiplier in the two largest terms in Equation 8-17, a significant improvement in the advantage of lunar payload to LEO.

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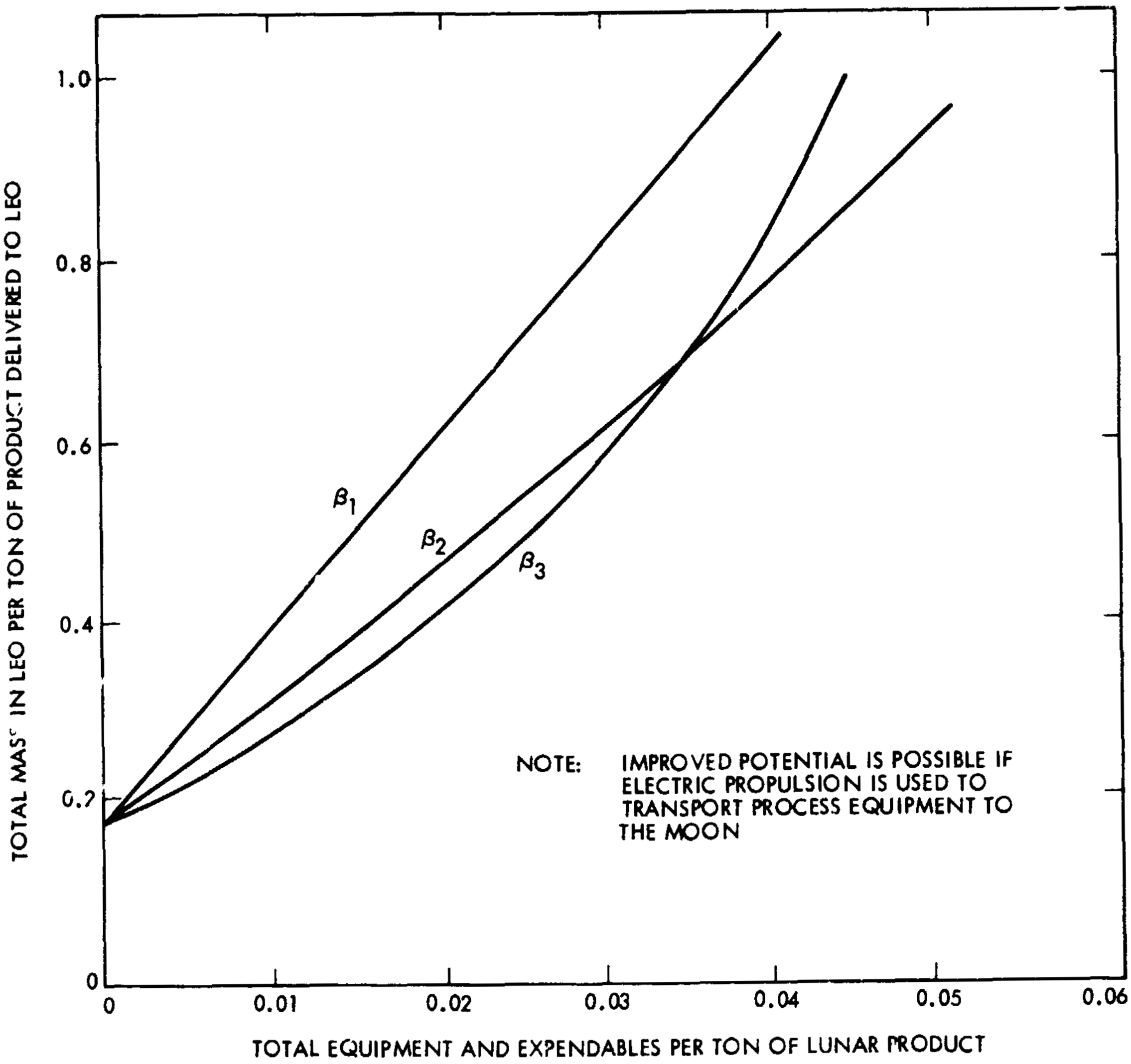

Figure 8-7. Potential Advantage, Lunar-Product Delivery to LEO with Electric Propulsion, Lunar O_2 Propellant, Conventional LS → LLO

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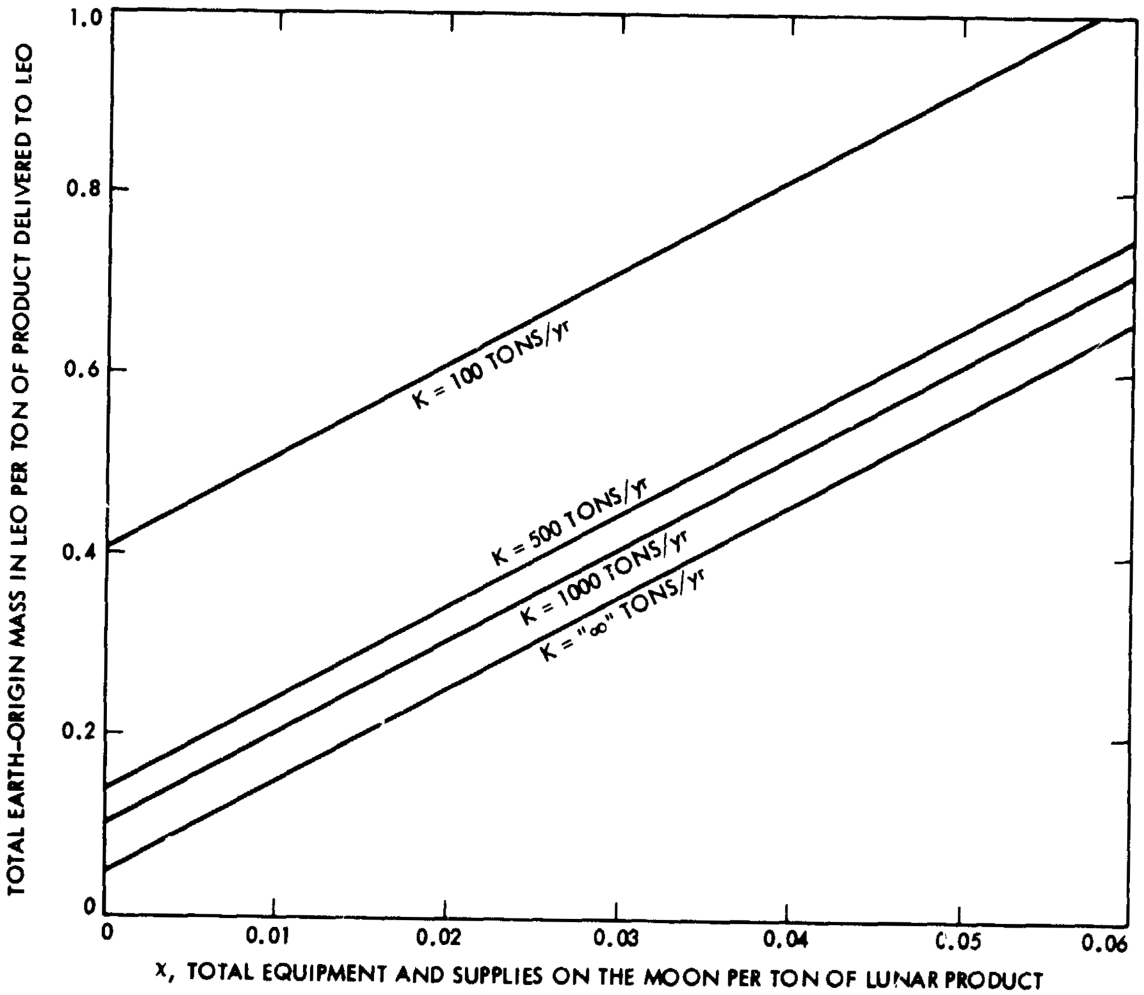

Figure 8-8. Potential Advantage, Lunar-Product Delivery to LEO with Electromagnetic Launcher and Electric Propulsion

8.6 EXTRATERRESTRIAL MATERIALS FOR USE IN GEO

Potential candidate extraterrestrial materials for use in GEO include metals and/or silicates for construction and radiation shielding and oxygen for return of reusable LEO/GEO OTVs and for orbit charge/trim, life support, and fuel-cell power generation. The large-quantity needs are most likely to be for construction.

For materials delivered to GEO, all terms in Equation 8-1 can be significant, depending on the type of propulsion to be used.

If we assume large-quantity product use in GEO, the transport vehicles can be assumed to be small enough to be considered insignificant except as they affect propellant consumption. Under these circumstances, Equation 8-4 can be rewritten in terms of break-even ratio as:

$$n > \frac{c + d + e + g + j}{1 + w} \qquad (8\text{-}18)$$

and using the terms defined following Equation 8-4 at the end of Section 8.2.2 as:

$$n > x\beta \frac{(1 + j'') + j'}{1 + w} \qquad (8\text{-}19)$$

Among the several cases examined for transport to GEO (see Section 7.2) three yield particularly useful results for illustration of the relative merits or value of extraterrestrial materials.

1) Using a single-loop mission profile of LEO/Moon/GEO/LEO with minimum terrestrial propellant (H_2 only) and all O_2 from the Moon, the net deliverable is negative. This result is somewhat surprising until one considers that the vehicle and oxygen for the GEO/LEO and the next outbound LEO/LLO must be injected into and back out of GEO.

2) With all terrestrial H_2, terrestrial O_2 for the outbound (LEO to LLO) leg, lunar O_2 for the remainder (LLO to lunar surface, lunar surface to LLO, LLO to GEO and empty OTV return to LEO) yields a positive return of 1.16 tons of payload in GEO per ton in LEO. (Since w in Equations 8-18 and 8-19 is significant, the benefit of lunar payload is much larger than 1.16; the effective payload ratio, as described in Section 7.2, is 4 or 5.)

3) Using two parallel mission profiles, with empty OTV return from GEO to LEO like #2 above, and parallel supply of lunar O_2 for the outbound leg direct to LEO as described earlier (Section 7.2), one ton of GEO payload requires only 0.57 ton of terrestrial propellant and 3.45 tons of lunar O_2.

Making the somewhat optimistic assumption that "self-contained" lunar products can be injected into low lunar orbit with electromagnetic accelerators, the terrestrial and lunar propellants associated with the lunar orbit-to-surface-to-orbit round trip can be eliminated. Under the circumstances, the potential advantage of lunar material increases significantly.

There is similar advantage for the use of solar-electric propulsion for the LLO-to-GEO transfer. Solar-electric shows greater advantage over chemical to GEO than to LEO because the chemical systems cannot use aerobraking at GEO. The difference between the advantage of lunar products in GEO and LEO is significant but not dramatic, and appears to be partially counteracted by a somewhat lower probability of requirement for large quantities of materials. The differences do not appear sufficient to modify most of the general conclusions regarding the types of products which may find markets.

Some recently suggested processes for fabricating silicon solar cells (private communication D. Bickler, et al., JPL) indicate that conventional silicon solar cells for lunar use might be competitive. Competition in LEO still appears unlikely. Facility requirements are too uncertain to define viability in GEO so further investigation appears warranted and will be undertaken in the future.

8.7 ASTEROID MATERIALS

There are three general classes of asteroids: the majority in the main belt between Mars and Jupiter; the "Apollo" or "near-Earth" asteroids in elliptical orbits which approach or cross the orbit of Earth; and the postulated "Trojan" asteroids in Earth-Sun libration orbits.

The major advantage of the asteroids as sources of extraterrestrial materials is the probability that some contain carbon and chemically combined useful "volatiles" including CO_2 and H_2O which can be extracted and used as reagents or to make propellants.

The main belt asteroids do not appear to be attractive sources of material unless they can be proven to contain volatiles that can be used to make propellant. They require relatively large transfer energies and transit times. Furthermore, the large distance from the Sun would require development of processing energy from some other source.

The Earth-crossing asteroids have the advantage of low transfer energy but the countering disadvantage that: low energy opportunities occur infrequently (e.g., 2-5 year intervals); the large and varying Earth-body distance eliminates the possibility of practical teleoperation, requiring highly reliable, fully autonomous operations; and the large and varying Sun-body distance degrades the efficiency and utility of solar energy for processing.

The major disadvantage of the "Trojan" asteroids is the uncertainty of their existence. While they can theoretically exist, they have never been observed. In addition to the possibility of useful volatiles, they have the advantages of low transfer energy, intermediate between the Apollo bodies and the Moon; nearly constant Sun distance, and thus the possibility of practical solar processing energy; and nearly continuous opportunities.

While the net transfer energies to the asteroids are lower than those to and from the Moon, transport studies (see Section 7.2) indicate that the mass transfer advantage to LEO is not significantly better than that from the Moon. This conclusion stems in part from the fact that lunar transfer can take advantage of low lunar orbit (LLO) staging; that is, the terrestrial H_2 for the return LLO to LEO leg can be left in LLO and not carried to the lunar surface and back up to LLO. Similarly, lunar oxygen for the LLO to lunar surface leg of transporting processing equipment and supplies can be staged in LLO rather than carried to LEO and back.

As a result of the transfer energy similarities, the general conclusions and constraints described earlier for the Moon apply to asteroidal materials as well, with small, quantitative differences. Should reasonably accessible asteroids be identified with proven, large quantities of "volatiles" that can be made into propellants, the advantages of processing and transport of products to LEO, GEO, or the Moon, would be dramatic.

8.8 FACILITIES AND SUPPLIES

Examination of the break-even intercepts in Figures 8-3 through 8-8 make it obvious that the total mass on the lunar surface to produce products for LEO, including processing equipment, power supply and expendables, etc., cannot exceed approximately 0.05 tons/ton of product. In fact, for any advantage to extraterrestrial materials in LEO, the total must be significantly less than that value.

8.8.1 Energy Supply for Processing

Small solar and nonsolar (e.g. RTG) power systems have been widely used in space, and scale-up to larger systems has been extensively studied. Thus, current and potential future technology are reasonably well understood; these factors are described in detail in Section 6. Solar photovoltaic technology is estimated to be currently capable of ~66 W/kg and has been extrapolated, with appropriate development, to be 200 W/kg by the mid 1990s.

Current radioactive thermoelectric generators (RTGs) are ~4 W/kg; future space reactors in the 100-kW electric range are projected to be in the 10-50 W/kg range.

Analytical studies indicate that using solar concentrators made from local extraterrestrial materials and with major development programs, advanced systems might approach 1 kW/kg. While this value is obviously speculative, it is a useful limit case to consider since it might be approached with extensive development, but is almost certain not to be exceeded.

Anywhere on the Moon except at the poles, solar-power systems will be able to operate ~1/2 of the time or ~4000 hours per year. Since the performance numbers already include solar-intensity and photon-to-electrical conversion factors, the expected output will be ~ 4000 kWh_e (kilowatt hours, electric) per year, per kW of capacity.

Using the free energy of formation of SiO_2 from Si and O_2 as the limit of energy required for dissociation at 100% efficiency, the lower limit of electrical power required for electrolysis is $\sim 8 \times 10^3$ kWh/ton of O_2.

The amount of equipment on the Moon for power generation per ton of produced oxygen, previously defined as d can be calculated from:

$$d = \frac{8 \times 10^3 \text{ kWh/ton of } O_2}{\dfrac{4 \times 10^3 \text{ kWh}_e}{\text{kW capacity yr}} \quad \text{power system performance } \dfrac{\text{kW}}{\text{kg}} \quad \dfrac{10^3 \text{ kg}}{\text{ton}} \quad \text{process efficiency} \quad \text{system lifetime yrs}}$$

Using pessimistic values of: power-system performance at 0.1 kW/kg, system lifetime of 5 years, d at 100% process efficiency is 0.004 and at 10% efficiency is 0.04. With power-system performance projected for the late 1990's of 0.2 kW/kg and a lifetime of 10 years, the values are 0.001 at 100% efficiency and 0.01 at 10%. Using the limit case of 1 kW/kg and assuming 10-year lifetime, values of d would be further reduced by a factor of 5. Values of d as a function of process efficiency for the three cases are shown in Figure 8-9.

Calculations in Sections 8-2 through 8-6 have shown that there is an advantage to use of extraterrestrial materials only when the total mass of production and power equipment and expendibles is less than a few percent of the total product mass. Comparison of break-even points in Figures 8-3 through 8-8 and the scale of power-supply mass in Figure 8-9 clearly demonstrate that process efficiency and energy technology are key issues in the viability of lunar O_2 production.

It is important to note that the current and mid-term technology curves are for photovoltaics with inherent solar to electric conversion efficiency of 10-15%, assuming that all of the dissociation energy is provided electrically. With this built-in scaling factor, relatively high efficiencies (potentially greater than 100%) are possible if a major fraction of the energy can be provided via the more efficient direct photon to thermal conversion.

In contrast to O_2 production requiring $\sim 10^4$ kWh/ton of product, conversion of lunar material into useful mixed silicate shapes requires <500 kWh/ton of product. For such products, the process efficiency and thus, process energy requirements, are therefore important but certainly less critical.

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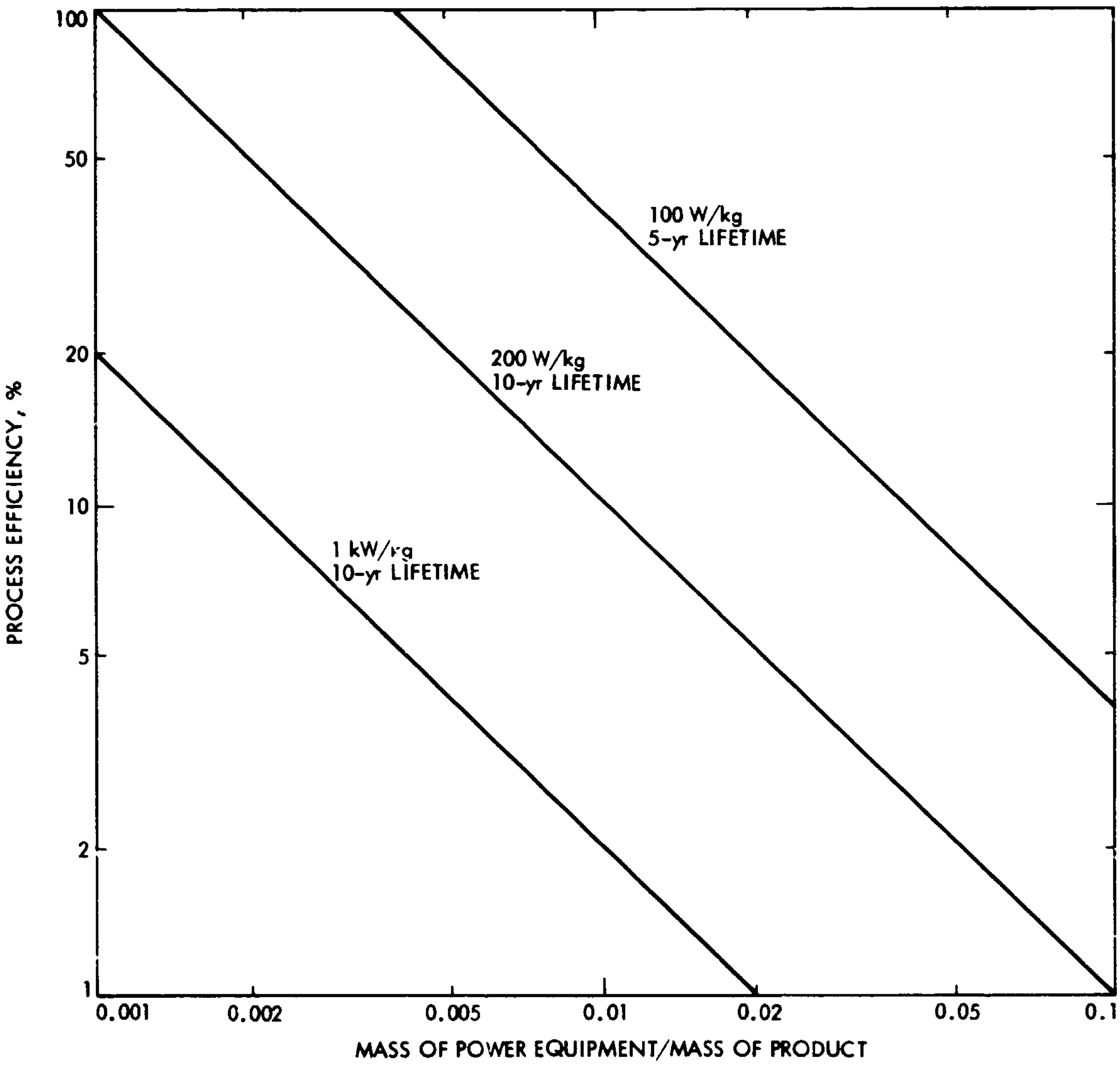

Figure 8-9. Power-System Requirements for Lunar Oxygen

8.8.2 Processing Equipment

The Earth-origin mass of equipment for processing extraterrestrial materials will no doubt be the subject of a significant design and development effort prior to eventual operational development. The mass relationship to analogous terrestrial equipment is not immediately obvious. Classical aerospace design and materials selection will tend to reduce mass for equivalent function; the requirement for reliable, long-term unattended operation will tend to lead to increases. Mass/function equivalent to terrestrial equipment appears to be a reasonable assumption pending specific equipment conceptual design.

In spite of the uncertainties, it is useful to make rough order-of-magnitude (ROM) estimates of equipment mass in order to focus technology. These can serve as specific guides for technical evaluation and emphasis within the program. A more important application is initial evaluation of the ultimate viability of candidate process/product combinations.

In previous sections, it has been shown that total power and processing equipment must be <5% of the lifetime product mass. It is useful to conceptually consider process equipment in the context of the time required to produce a product mass equivalent to the equipment mass. At 0.1% of the annual[2] mass, the time for equivalent mass is a few hours, as might be typical of bulk material-handling (e.g. Earth-moving) equipment. At 10% of the annual production mass, the time for production of equivalent mass is ~1 month, which might be typical of a large machine tool producing small precision parts.

A ROM estimate of electrolysis equipment has been made for the equipment concept shown in the FY'82 annual report (Reference 8-6), Figure VI-13, p. 98, as follows:

- 500 tons LO_2/year at 40% O_2 in soil requires processing of 1250 tons soil/year
- 1250 tons soil/ year 4000 operating hours/ year requires processing 300 kg soil/ hour
- 300 kg soil/hour requires ~0.1 to 0.2 m^3 soil/hour
- With throughput in 10 hours, process volume is 1-2 m^3
- Container with 1 cm thick walls, ~10^5 cm^3
- Container at density of tungsten, 20 gm/cm^3, ~2 tons

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[2] Note that we are using annual product mass; other comparisons are based on life cycle product mass with life cycles of 5-10 years assumed.

If such a 2-ton container represents 1/10th of the total electrolysis facility and such a facility has an operating lifetime of 10 years, then (ROM):

$$c = \frac{20 \text{ tons}}{500 \text{ tons/yr} \times 10 \text{ yrs}} = .004$$

Such a value seems to indicate that the processing facility (<u>not</u> including power supply) for electrolytic production of lunar O_2 may be well within the limits of potential practicality. Obviously, improved estimates will be needed when better understanding of the process and the requisite facility are better established.

8.8.3 Process Expendables/Consumables

Several investigators have proposed or examined processes requiring significant quantities of process aids such as fluxes, working fluids, consumable electrodes, etc. In preliminary electrolysis experiments, Kesterke (Ref. 8-12) used 65-75 weight flux by weight and found it impossible to recover or recycle any of it.

The analyses in earlier sections have shown that the maximum mass of equipment and expendables for lunar products for use in LEO cannot exceed a few (e.g. 3-5) percent of the product mass. Thus, with requirements for power (Section 8.8.1) and production facilities (Section 8.8.2) it would appear that net expendables should not exceed, and preferably should be significantly less than, 1% of the product mass. If we consider the conditions used by Kesterke, the molten materials contained ~4 kg of flux per kg of O_2. Thus, a recycling efficiency of ~99.75% would be required to keep consumption down to a net 1% of product (O_2) mass. Such a level would be difficult with multistage terrestrial facilities, and unlikely for the simple facilities that will be practical on the Moon.

Processes which use volatile gases or fluids (e.g., wet chemistry) in closed systems must be protected from damage from micrometeorite impact, which would result in catastrophic loss of the working fluid.

Obviously, for products made from lunar materials and intended for use on the Moon, these constraints are significantly less severe (see Section 8.3.4).

8.9 EXPLORATION, INSTALLATION, AND MAINTENANCE

In the analyses described above, we have assumed that several of the factors (exploration, installation, etc.) are amortized over large quantities of material and can therefore be considered insignificant. If only small quantities of extraterrestrial materials are required, even very conservative exploration, installation, or maintenance will tend to overshadow and cancel any advantages.

The impact of these factors can be illustrated by examining lunar operations and products. For the analysis it is useful to consider both manned and unmanned operation options and to use the following approximate quantitative values:

- Manned operation for exploration, installation, and maintenance equivalent to the Apollo system, which had the capability of delivering two men with instruments and tools to the lunar surface and returning them to Earth. Total starting mass in LEO ~100 tons (Ref. 8-13).
- Instrumented operation for exploration, etc., equivalent to the Surveyor spacecraft, which had observation and simple teleoperated manipulation capabilities. Would probably require addition of mobility or mobile-vehicle payload (like the Surveyor roving vehicle) so that the mass might be 2-5x that of Surveyor. Total mass of Surveyor vehicle in parking orbit (LEO) was approximately one ton (Ref. 8-14).

At an extraterrestrial materials market level of 500 tons of lunar O_2 delivered to LEO per year for 5-10 years, several manned missions for exploration and equipment installation would not have a major impact. However, at a level of 100 tons of product per year at a 2:1 advantage (i.e., 100 tons of O_2 delivered to LEO for every 50 tons of equipment and supplies) a manned maintenance visit once in two years would completely cancel any advantage. Obviously if the maintenance visit included service of some other lunar operations (e.g., astronomical laboratory) and the "charge" to extraterrestrial materials operation reduced accordingly, the impact would be less.

Even at a high optimistic level of 500 tons per year, an annual manned maintenance visit represents a significant fraction of the displaced terrestrial-origin mass. Therefore, for the foreseeable future, plans for manned intervention must be for nonroutine maintenance only; all routine and even simple nonroutine maintenance must be done remotely. Relatively frequent (e.g., 1-3/year) use of Surveyor-class vehicles (2-5 tons in LEO) carrying teleoperable equipment for installation or maintenance will be practical for large-quantity production.

Even Surveyor-class operations become impractical for smaller quantities of material. If we consider oxygen for Space Station life support, at 0.84 kg (1.84 lbs) per person-day or ~300 kg/person-year (Ref. 15), an annual Surveyor-scale maintenance vehicle would be impractical for supplies of oxygen to a 10- 20-man Space Station. Even if lunar O_2 could be delivered at a 2:1 advantage, more than 30 years would be required to recover the investment of a single Apollo-type mission for exploration and startup.

8.10 IMPACT OF AVAILABLE VOLATILES

The analyses and discussions above are based primarily on a knowledge of available space resources which is supported by some level of physically measured evidence. Lunar samples, returned by the various Apollo missions, have been extensively examined and indicate a Moon--near the surface of the areas explored--virtually devoid of volatiles. There is evidence (Ref. 8-16) that at least some of the carbonaceous chondrite asteroids contain a few percent water.

The ready availability of volatiles--hydrogen, carbon, or even sulfur or phosphorus compounds--in large quantities would have an important effect on the entire strategy and limits regarding the use of extraterrestrial materials. Such availability would have an even more profound effect on the entire world effort to explore and utilize space.

Strategies for ground observations and exploration missions to determine the possibility and extent of extraterrestial volatiles are described in Section 9, following.

8.10.1 Utility of Volatiles

The principal use of large-quantity volatiles (if found) will most likely be the production of propellants, for LEO to GEO orbit raising, for Moon to Earth-orbit transfer of propellants and other lunar products, for interplanetary orbit insertion, etc. In addition, they are likely to find important use as reagents or processing aids for production of extraterrestrial products, for conversion into useful extraterrestial products (e.g., plastics), and for life support.

While the simplified example of Case 5 in Table 8-1 shows a "payload ratio" of infinity for the availability of lunar H_2, the real benefit of lunar volatiles would be somewhat less, but still dramatic when requirements for production and power equipment are considered. The quantitative impact will be examined in future studies.

Similarly, the effect of a small initial facility to produce H_2 and O_2 to transport equipment and supplies for expansion of the lunar facility has not been examined but will likely have the effect of further improved margins.

It is important to note that volatiles must be present in significant quantities and concentrations to have the type of engineering impact expected. For example, H_2 from implanted solar-wind protons is present in the lunar regolith to the extent of ~10 ppm. As established by experiments on the Apollo samples, the H_2 is easily released by moderate heating. However, if, in processing lunar soil to extract O_2 all of the available H_2 is extracted, it will amount to much less than 1% of the H_2 propellant required to launch the lunar O_2 to LLO (using H_2/O_2 chemical propulsion). This comparison is not to imply that the trapped solar wind will not be extracted and used, only that it is not present in sufficient quantities to affect transportation.

8.10.2 Sources

Proof of existence of lunar polar water-ice would have an important effect on any program to extract and use extraterrestrial materials. It would be the most accessible source for early exploration.

The small amount of water on some carbonaceous chondrates is attractive but would require means to identify the water-bearing asteroids, that they be accessible, and that means for extraction in microgravity be developed.

Proof of the hypothesis that some Earth-approaching asteroids are extinct comets with large volatile cores (Ref. 8-17) and identification of such bodies which are accessible would provide very attractive targets for early exploration.

The means of finding and assaying potential sources of volatiles are discussed in the next Section (Section 9.).

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SECTION 9

RESOURCE EXPLORATION MISSIONS

R. L. Staehle

9.1 BACKGROUND

Early analysis indicates that the use of certain nonterrestrial resources may be commercially advantageous within two decades. Oxygen from the Moon presently appears to be the most likely candidate for an initial space-resource product--in this case, a product to be used for propellant for orbit transfer vehicles (OTVs) carrying high-value cargo from low Earth orbit (LEO) to geostationary Earth orbit (GEO). Such OTVs would utilize hydrogen as a fuel, to be burned with oxygen; the latter represents about 60% of the gross loaded mass of an OTV, and about 4 times the useful payload mass.

Strong advantages have been identified (by Frisbee, Carroll, and Staehle), if hydrogen could be found along with oxygen in useful accessible quantities. At this time there is no known source of both hydrogen and oxygen that meets both the favorable concentration and the accessibility criteria (Mars is rather inaccessible, while known lunar hydrogen is at a very low concentration). However, there are good reasons to suspect undiscovered hydrogen sources on the Moon and among the asteroids.

Other resources of potentially high value may also exist (Table 1), particularly among the near-Earth asteroids. During FY82, Resource Exploration Mission activities have been focused on identifying the potentially important early resources, identifying the key uncertainties regarding their presence in accessible locations, and proposing methods of resolving those uncertainties. Results of this work occupy the remainder of this section.

If resource utilization is to begin in 20 years, the general resource sites need to be known in 10 years, and specific sites in 15. If an operational capability option is to be preserved for the years 2000-2005, then required resource exploration missions must be in progress by 1990-92 (for general sites) and 1995-98 (for specific sites). Consequently, the key missions must be identified during the next year and defined over the next 2-3 years. One mission (for lunar polar ice) has been identified and is described later, and much of the groundwork has been laid for possible asteroid missions.

Table 9-1. Operational Uses for Volatile Elements

Propulsion, especially O_2 and H_2, possibly with C

- Using LOX/LH_2, 60% of GEO-bound LEO mass is LOX for advanced chemical propulsion
- LOX/LH_2/CH_x reduces tankage mass, increases propellant mass
- Total mass 2-3 times useful payload mass for most common operations
- 100-700 t[a]/yr by 2000 if traffic trends as predicted
- Possible later interest in Ar for electric propulsion

Life Support

- 1-2 kg O_2/person/day if CO_2 not recycled
- N_2 leakage makeup (less than O_2 consumption)
- 10-20 t/yr by 2000 if 40 people continuous, recycled CO_2

Other

- Process reagents
- Military lasers
- TBD
- Demand uncertain

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[a] t = metric tons

9.2 KEY RESOURCE UNCERTAINTIES

As presently planned, space materials processing techniques are to receive the largest portion of Research into the Utilization of Space Resources (RUSR) funding over the next few years. At this time, processing research must be fairly broadly aimed for two reasons. First, with space materials processing research in its infancy, it is difficult to select the most promising technologies so soon. Second, our level of ignorance about the available resources to be processed is high, so that processing research must prepare for a broad spectrum of resource options (Table 9-2).

The present research program, based on known natural availability, is nevertheless planned with the flexibility to take advantage of any newly discovered resources (Tables 9-3, 4, 5). Four particular uncertainties regarding resource availability are key issues in focusing processing technology development. The sooner these uncertainties can be resolved, the sooner a sound space resource utilization scenario may be constructed, allowing all relevant technology areas to be focused much more effectively than is possible today. These four uncertainties, in descending order of priority, are:

1) Is there water (or other volatile ices containing H or C) at the lunar poles (Ref. 9-1)?

2) Do Earth Trojan asteroids exist, and if so, what are their orbital parameters, what resources do they harbor, and how large are they (Ref. 9-2)?

3) Are there easily accessible asteroids (i.e., low inclination "near-Earth" asteroids) with water or with substantial quantities of free metals at or near their surfaces (Ref. 9-3)?

4) Do transient lunar phenomena indicate the existence of substantial gas or volatile ice deposits on the Moon, and if so, what gases of resource importance are available (Ref. 9-4)?

A positive answer as early as possible to any of these questions would be of great value by providing a focus for early resource utilization work. The importance of each resolved uncertainty is summarized below.

1) Lunar water: with electrolysis, water provides a source of both hydrogen and oxygen. Hydrogen appears to be absent on the rest of the Moon in 0.1% or higher concentrations. Hydrogen can be employed in very effective high-thrust rocket engines (with oxygen), and can be used as a reducing agent in processing lunar soil for certain metals. (Concentrations of carbon, another possible reducing agent, also appear to be absent on the Moon. Carbon dioxide ice could conceivably be mixed in with any water ice deposits as a clathrate (Ref. 9-5). Having a lower sublimation point, CO_2 is less susceptible to the polar cold-trap effect (Ref. 9-1). In addition, the electrolysis of water is a much simpler and better understood process for obtaining oxygen than is oxygen-extraction from rocks and soil. The availability of both fuel

Table 9-2. Accessibility in the Inner Solar System

Location	Minimum ΔV to LEO[a] (km/sec)	Time of Flight (ballistic)	Frequency of Opportunity (ballistic)	Distance from Earth (light time)	Distance from Sun (AU)	Rotation Period
1982 DB	0.1-0.5	2-8 mo	2-10 yr	0.5-20 min	0.9-1.2	? hr
Near-Earth Asteroids	0.5-2	2-20 mo	2-5 yr	1-20 min	0.8-2	? hr
Earth Trojans[b]	≤1.4-2	0.8-2 yr	continuous	4-10 min	1	? hr
Phobos Deimos	~1.5-2	0.5-2 yr	1-2 yr	5-20 min	1.4-1.7	8-31 hr
Lunar Surface	3.2	3-5 d	continuous	~2 s	1	28 d
Mars Surface	5.6	0.5-2 yr	1-2 yr	5-20 min	1.4-1.7	25 hr
Earth Surface	9.1	10-15 min	continuous	-	1	24 hr

[a] With aerobraking in Earth atmosphere prior to orbit insertion

[b] Speculative

All sites shown are of possible interest. If water exists near the lunar surface (see later charts), the Moon will almost certainly become the first site of commercial resource extraction. The first three accessibility criteria (ΔV, TOF, FOO) are the most important, and since no body excels in all three, complex commercial tradeoffs will exist. Note that transit energy varies as the square of ΔV, while propellant required is an exponential function of ΔV.

Table 9-3. Known Volatiles Sources of Potential Operational Importance

Body	Volatile Elements of Operational Interest				
	O	H	C	N	Ar
Near-Earth Asteroids	probable	?	?	?	no
Earth Trojans[a]	?	?	?	?	no
Phobos/ Deimos	probable	?	?	?	no
Moon	30-50%	10-100 ppm	trace	no	?[b]
Mars	abundant	polar caps H_2O vapor	CO_2 atmos.	2.7% atmos.	1.6% atmos.
Earth	yes	yes	yes	yes	yes

[a] Not discovered

[b] See S. Jovanovic and G.W. Reed, Jr., "Trapping of Gases and Low Temperature Volatiles in Lunar Samples," Conference on Planetary Volatiles, Abstracts, Lunar and Planetary Institute, October 9, 1982.

Table 9-4. Speculative Volatiles--Sources of Potential Operational Importance (Ordered by Anticipated Operational Significance If Discovered)

- Lunar polar water (ice)
  - Suggested by Watson, Murray, Brown (Ref. 9-13)
  - No evidence for or against
  - Disagreement on quantity and loss mechanisms
  - "Ideal" source for H and O, simple extraction and processing
- Earth Trojan asteroids
  - Existence postulated (Weissman and Wetherill), theoretically stable orbits (Dunbar)
  - Low trajectory energies, low thrust:weight
  - Long transit time but continuous flight opportunities
- Near-Earth asteroids
  - 59-60 known[a]
  - At least 4 with ΔV comparable to lunar surface, probably dozens within detection limits[a]
  - Speculative composition parallels meteorites
- Lunar subsurface gas
  - Highly speculative and controversial
  - H and C suggested
  - Specific sites suggested
  - No high-quality evidence
  - Difficult extraction, potentially simple processing
  - Possible contemporary source of polar ice

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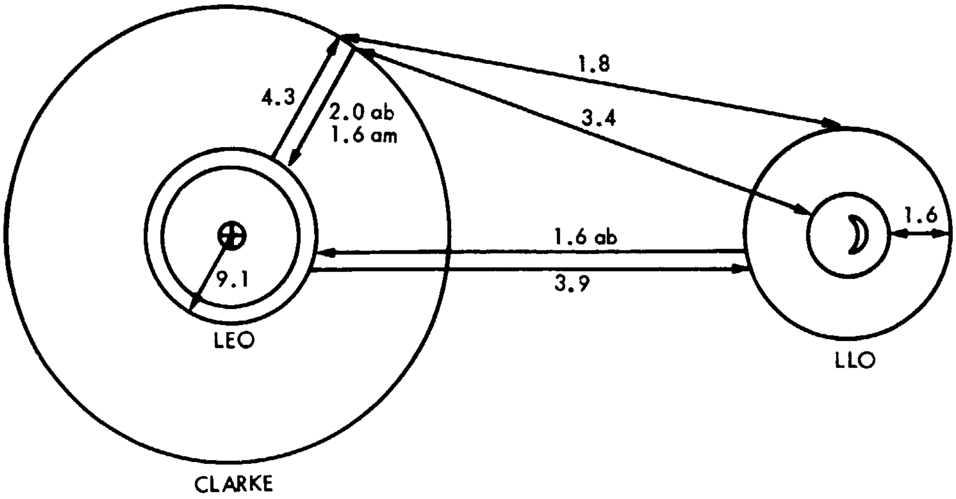
[a] As of October 1982.

Table 9-5. Resolving the Key Resource Uncertainties

- Lunar ice
    - Polar orbiter
    - γ-ray spectrometer, electromagnetic sounder (100-150 MHz)
    - Very simple spacecraft, large launch-mass margins
    - Alternative to integrate with science-oriented LPO or POLO
- Earth Trojans
    - Ongoing ground-based search (Dunbar, Helin)
    - Simple search spacecraft possible if no telescopic success
- Heliocentric, near-Earth asteroids
    - Ongoing search and discovery (e.g., Helin, et al.)
    - Ongoing spectral characterization (e.g., McCord, et al.)
- Lunar gas releases
    - Gaseous origin of lunar transient phenomena not confirmed
    - No ongoing confirmation effort
    - No ongoing spectral instrument readiness



LEO = LOW EARTH ORBIT 200–400 km
CLARKE = GEOSTATIONARY EARTH ORBIT (GEO) 35,900 km
LLO = LOW LUNAR ORBIT 100–200 km
ab = WITH AEROBRAKING (DRAG ONLY)
am = WITH AEROMANEUVERING (LIFT AND DRAG)

Fig. 9-1. Cis-Lunar Accessibility (km/sec) and oxidizer on the Moon in an efficient and tested combination would tremendously relieve transportation burdens during any early lunar resources utilization effort (see Figure 9-1). Because of these advantages, mining and processing technology development would best be redirected in the event of a major lunar volatiles discovery. Early discovery would reduce the costs of research and redirection, and would probably result in an earlier capability for commercial space resource utilization.

2) Earth Trojans (ETs): if they exist, asteroids in stable, librating Earth-Sun orbits would require the least energy to reach from Earth of any surface in the Solar System, including the Moon's (Ref. 9-6). Also, such asteroids would move only very slowly in the rotating Earth-Sun system, permitting launch and return trajectories largely insensitive to launch date. These advantages mean that the existence of ETs could substantially reduce logistical burdens of space resource utilization. If ETs contain any useful resources (oxygen at least would be present, along with perhaps carbon, metals, or water--probably with variability among different bodies), then they could become the strongest candidates for early utilization. A determination of ET composition would then permit focusing the mining and processing technology development on those materials most accessible from Earth.

3) Near-Earth asteroids with metals or water: based on meteorite analysis and some uncertain spectral data, a number of near-Earth asteroids may possess substantial quantities of natural nickel-iron alloy. Free metals are much easier to process than those in silicate rocks, and any metals can be formed into more versatile structural members than other space resource products. Easily accessible asteroids with high free-metal content would suggest a greater emphasis on metal mining and processing technologies, though it is not yet clear if the structural performance of asteroidal metals is sufficient to compete with metals produced on Earth using complex and highly evolved processing techniques, which would be prohibitively costly to employ in space.

Again, based on meteorite analysis, carbonaceous chondrite asteroids may contain from 1 to 20% water. Advantages would be similar to finding water on the Moon, except that transit times are much longer and ballistic opportunities infrequent. At certain times, transit energies can be much lower than from the Moon for at least 6 known asteroids. Compositions of these asteroids have not been determined with any real certainty.

4) Transient Lunar Phenomena (TLP): near-surface lunar gas reservoirs might be tapped using equipment already understood by the terrestrial natural gas industry. These methods are radically different from those now under study for acquisition of volatile elements on the Moon. If TLPs are in fact caused by gas releases, the most likely constituent appears to be hydrogen (Ref. 9-7), though CO_2, H_2O, CH_4 and He are also possibilities by terrestrial analogy. Ready access to any of these volatile compounds except helium would make lunar resource utilization much more practical than presently envisioned with the absence of volatiles concentrations. Confirmation of gas-originated TLP, determination of their constituent gases and estimation of their volume would probably result in a redirection of processing technology development. At the very least, this information about TLP, which may be obtainable within the next 5 years, would indicate a plausible source for gases cold-trapped at the lunar poles. This in turn would add extra impetus to an early lunar polar volatiles detection mission.

9.3 GEO TRAFFIC

If water is found on the Moon, its primary initial market is likely to be as propellant (H_2 and O_2) for LEO to GEO transfer vehicles (OTVs).

9.3.1 Historical Traffic to GEO

The first successful geostationary satellite, the Hughes Syncom 3, was launched August 14, 1964. Hughes has since become the leading manufacturer of communications satellites, and recently was awarded the Intelsat VI contract, valued in excess of $1 billion.

Including a conservative estimate as of April 1982 (excluding Soviet traffic and the effect of the Ariane L-05 failure), by the end of 1982, 150 satellites will have been placed in GEO with a total mass of 82 t (t = metric tons). Of this mass, 40% is in the form of civilian operational communications satellites. All of these can be thought of as commercial, though they are frequently operated by government-owned companies when outside the United States. Another 21% of the total mass is contributed by operational military communications satellites (U.S., U.K., and NATO). The balance consists of classified military satellites (probably mainly for ballistic missile early warning and electronic intelligence) and satellites for commercial and military experimentation, weather observation, and science. The civilian communications fraction appears to be increasing, probably to remain between 50 and 65% over the next 3-5 years.

GEO traffic is plotted in Figure 9-2 with totals listed in Table 9-6.

Table 9-6. Non-Soviet GEO Deliveries by Calendar Year
(Estimated as of April 1982)

Year	Total No.	Total No. Cumul.	Civilian Communications[a] Year	Civilian Communications[a] Cumul.	Military Communications[a] Year	Military Communications[a] Cumul.	Total to GEO[a] Year	Total to GEO[a] Cumul.
1964	1	1	0	0	0	0	39	39
1965	0	1	0	0	0	0	0	39
1966	8[b]	9	0	0	363	363	704	743
1967	7	16	261	261	499	862	1490	2233
1968	13[b]	29	150	411	363	1225	893	3126
1969	6	33	290	701	854	2079	1893	5019
1970	4	39	302	1003	129	2208	1131	6150
1971	5[b]	44	1414	2417	2429	4637	3843	9993
1972	3	47	1698	4115	0	4637	1698	11691
1973	6[b]	53	999	5114	1132	5769	5331	17022
1974	7	60	1477	6591	129	5898	3000	20022
1975	9	69	2272	8863	0	5898	6115	26137
1976	13[b]	82	4223	13086	310	6208	8394	34531
1977	14[b]	96	1086	14172	1506	7714	7725	42256
1978	12[b]	108	3042	17214	2463	10177	7551	49807
1979	6[b]	114	572	17786	3587	13764	7019	56826
1980	5	119	1615	19401	1974	15738	4216	61042
1981	12[b]	131	5216	24617	987	16725	10292	71334
1982	19[b]	150	8650	33267 (40%)	1000	17725 (21%)	11500	82834

[a] Total mass in kilograms.

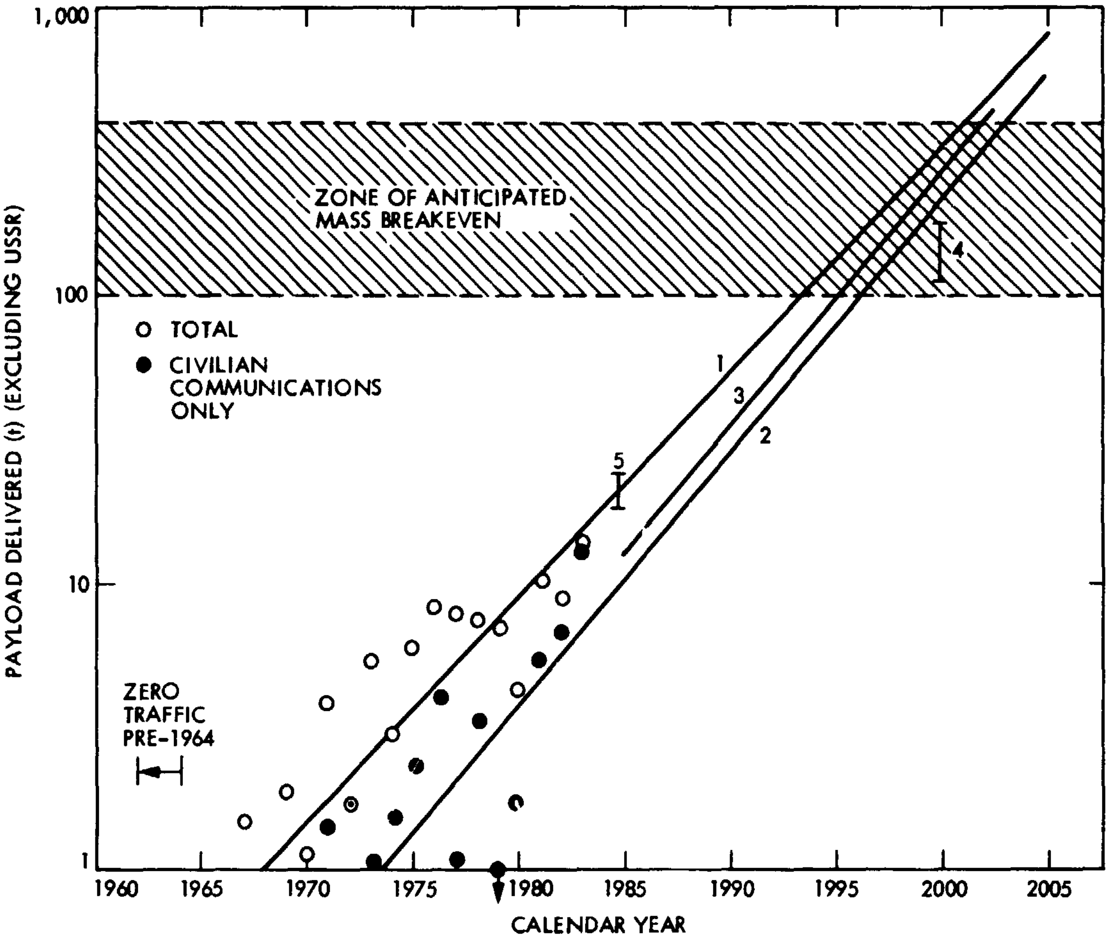
[b] Includes total number of individual satellites on multiple launches, except on classified launches. The Ariane and Shuttle result in multiple launches in all years beyond 1981.

9.3.2 Future Traffic to GEO

The following graph (Fig. 9-2) and charts note the factors influencing traffic to GEO. The graph shows historical traffic (plotted points), point projections for the years 1985 and 2000 (vertical bars), and logarithmic trend-line projections based on past traffic with various assumptions about the mix of satellite users.

No 20-year projection will be accurate, but the growth rate shown is consistent with earlier rates in communications and commercial aviation. Possible factors which could hasten or inhibit future traffic growth are summarized in Tables 9-7 through 9-9. Competition from fiber optics could reduce demand for satellite traffic serving overseas and domestic point-to-point telephone and data traffic, but provide little threat to direct-broadcast, video-teleconferencing and mobile applications.

Present and evolving geostationary delivery methods are cited in Table 9-9.



HISTORICAL TRAFFIC TO GEO IS SHOWN BY PLOTTED POINTS, WITH LOGARITHMIC FITS SHOWN AS LINES WITH EXTRAPOLATION AT CONSTANT GROWTH RATE. OBVIOUSLY NO SUCH EXTRAPOLATION SHOULD BE CONSIDERED A SOLID PREDICTION, BUT MAY SERVE AS AN INDICATOR OF PLAUSIBILITY WHEN CONSIDERED ALONG WITH INFLUENCING FACTORS.

SHADED REGION INDICATES EXPECTED LEVEL OF MASS BREAKEVEN LEVEL WHEN CAPITAL AND OPERATIONAL MASSES ARE CONSIDERED FOR USE OF LUNAR OXYGEN IN OTVs, AS DISCUSSED IN REF. 9-39. 1 = TOTAL TRAFFIC TO GEO. 2 = CIVILIAN COMMUNICATIONS EQUIPMENT TRAFFIC ONLY. 3 = ADJUSTED EXTRAPOLATED TOTAL, ASSUMING CIVILIAN COMMUNICATIONS MAKES UP 80% OF TOTAL TRAFFIC AND FOLLOWS LINE #2. 4 = PREDICTED CIVILIAN COMMUNICATIONS TRAFFIC ALONE FOR THE YEAR 2000 (REF. 9-8). 5 = TOTAL TRAFFIC SCHEDULED FOR 1985 ASSUMING STS MANIFEST DESCRIBED IN REF. 9-40, PLUS ARIANE TRAFFIC OF 4.3 t, JAPANESE TRAFFIC OF 1 t, CHINESE TRAFFIC OF 0.5 t, WITH 4.5 t ADDED FOR MILITARY SATELLITES NOT ANNOUNCED TO GEO IN MANIFEST. 1983 TRAFFIC IS ESTIMATED ASSUMING ONLY ONE TDRSS LAUNCH AND NO CLASSIFIED MILITARY SATELLITES.

Fig. 9-2. Annual Traffic to GEO

Table 9-7. GEO Traffic Vital Statistics

Total 71 t, 131 satellites, 1964-1981, excluding USSR

$$\left.\begin{array}{l}\text{Operational civilian communications} = 24.6\text{ t}\\ \text{Operational military communications} = 17.7\text{ t}\end{array}\right\}\ \sim 60\%$$

1985 scheduled launches ~30-40 + total[a]

1981, 82 actual and scheduled launches ~10-12 t total[a] each year

Communications satellite operator group rapidly growing

- New U.S. corporate carriers
- European government/corporate carriers
- Developing countries and regional groups

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[a] Not considering Ariane L-05 failure

Table 9-8. Factors Influencing GEO Traffic

- Customer demand
  - Traditional point-to-point (telephone, data, TV, via Intelsat, Comsat General, AT&T, RCA, ANIK, PALAPA, etc.)
  - Maritime
  - Direct broadcast
  - Mobile service (land commercial, aeronautical)
  - Video teleconferencing
- Launch vehicle throughput capacity
  - Capacity-limited through 1986-1990
  - Possible private launch vehicles
  - Congressional vs commercial STS fleet size selection
  - STS upper stage selection (cost vs. mass-to-orbit)
  - Uprated Ariane (4 and 5) development
- Launch vehicle reliability
  - Insurance costs, risk exposure
  - Uncertainty of Ariane L-02, L-05 failures
- Regulatory uncertainty
  - International Allocations of slots and frequencies
  - Domestic allocations
  - Direct TV broadcast treaties
- Other uncertainties
  - Petroleum prices
  - Interest rates
  - Technological advances

Table 9-9. GEO Transportation

Present options (prices dependent on launch year, customer services, use as ROM only)

U.S.: Delta (600 kg, $30M), Atlas Centaur (1100 kg, $50M), STS/PAM-D (600 kg, $15-25M), STS/PAM-A (~1200 kg, ~$40-50M), STS/IUS (~2,300 kg, ~$130M)

E.C.: Ariane (~1,000 kg, ~$20-30M)

Japan: Delta-derivative

$3,000/kg to LEO on STS

Developing options

Shuttle: IUS too expensive for commercial users, Centaur capability 5,000-6,000 kg, $100M?

Ariane: Aggressive marketing, development of growth capability, possible cost reductions. 2,000-3,000 kg 1986-1988

Future possibilities

Orbit-based reusable orbit transfer vehicle (OTV)	Likely together
Aero-assisted OTV	

Ariane to 5,000 kg

Solar- or nuclear-ion low thrust (revenue loss, solar cell degradation)

Tether-assist (>20 y)

9.3.3 Propellant Estimate for GEO Traffic in the Year 2000

The following is a preliminary estimate of the mass of propellant required to supply orbit transfer vehicles (OTV) carrying communications equipment from LEO to GEO in the year 2000. These numbers are very crude and based on several unverifiable assumptions, but represent a first step in estimating commercial interorbit traffic for the time period when space resources could conceivably be entering the marketplace. Any estimate of turn-of-the-century traffic is highly speculative, just as a 1960 model of 1978 traffic would have been.

This propellant estimate is highly dependent on a crude mission model generated at the Johnson Space Center as part of research on the Space Operations Center concept (Ref. 9-8). In the communications category for the year 2000, this model shows 18 equivalent STS flights per year--10 domestic and 8 foreign. Elsewhere in the reference it can be shown that approximately one-third of each STS-equivalent load is actual communications spacecraft mass, with the balance consisting of upper stages, propellant, cradles, and airborne support equipment (Table 9-10). Taking an STS load to be 29,500 kg (~65,000 lbs) this implies a GEO payload mass of (18 x 19, 500 ÷ 3) = 177,000 kg = 177 t. Taking this same payload and assuming it to be boosted by an expendable Centaur-derived OTV (I_{sp} = 460 sec, propulsion and tankage mass = 0.1 m_p, 3% unuseable propellant, ΔV = 4.3 km/sec), this results in a need for 369 t of propellant per year. At a 6:1 mixture ratio, this implies 316 t of O_2, and 53 t of H_2. (See also Table 9-11.) If this were all tankered to orbit on separate Shuttle flights (certainly not the case with an expendable OTV, but an illustrative comparison, anyway), the launch charges would be $1.4 billion dollars at $90 million/flight, with $1.1 billion dollars for O_2 alone.

A reuseable OTV would reduce expended hardware costs, but increase the total propellant requirement. An aerobraking vehicle, which uses drag but no lift to reduce orbital energy, requires a return ΔV of 2.0 km/sec (Ref. 9-9). According to mass estimation techniques given in Ref. 9-9, along with some reasonable OTV assumptions (at a delivered payload mass of 8,000 kg per flight), the total (up and down) propellant mass is 2.8 times the payload mass delivered to GEO.

Based on the original GEO payload mass, this yields 426 t of O_2 and 71 t of H_2, for an O_2 launch cost of about $1.4 billion dollars under cost assumptions identical to the previous expendable case. If this is below, the cost for a space resource operation to deliver the same amount of propellant, then, after accounting for all the questionable assumptions, the economic door to commercial space resource operations is at least part way open. Keep in mind that total system costs for resource operations are being compared here only to Shuttle launch costs. Even if all the assumptions were correct, a much more detailed analysis would be required to determine cost effectiveness. Still more complex factors would enter an estimate of commercial viability (as opposed to cost effectiveness).

By the year 2000 it is at least conceivable that transportation expenditures for LEO-GEO propellant will be on the order of $1 billion annually. Actual traffic will depend on a great many external factors. Several of the mechanisms which could force traffic down are:

Table 9-10. GEO Transportation

Physical parameters

LEO ~ 370 km, $i = 28.5^{\circ}$; GEO ~ 35,900 km, $i = 0^{\circ}$

ΔV_{up} = 4.3 km/sec = characteristic velocity

$$\Delta V_{down} = \begin{cases} \text{4.3 km/sec = all propulsive} \\ \text{2.0 km/sec = aerobraking (drag only) [best choice]} \\ \text{1.6 km/sec = aeromaneuvering (lift and drag)} \end{cases}$$

rocket equation:

$$\frac{\text{propellant mass}}{\text{delivered mass}} = \frac{m_p}{m_f} = \exp\left(\frac{\Delta V}{c_{eff}}\right) - 1$$

c_{eff} = effective exhaust velocity ~3.4-4.5 km/sec

Actual stages

m_f = tankage + guidance + structure + useful payload

$$\frac{m_{LOX}}{m_{LH_2}} = 6$$

$$\frac{m\ (\text{propellant})}{m\ (\text{useful payload})} \sim \begin{array}{l} \text{2.5 expendable} \\ \text{3.8 aerobraking, all propellant from LEO} \\ \text{3.2 aerobraking, return propellant delivered to GEO} \end{array}$$

Table 9-11. GEO Transportation Propellant Requirements

Civilian communications only, c_{eff} = 4.5 km/sec, Aero-assist OTV (4.3, 2.0 km/sec), m_{LOX}/m_{LH_2} = 6, no boiloff provision

No terrestrial propellants after initialization

Year	Useful Payload to GEO (t)	Terrestrial LEO (t)		Lunar Hydrogen and Oxygen Production (t)		Lunar Hydrogen and Oxygen to LEO (t)		Lunar Hydrogen and Oxygen to GEO (t)	
		LOX	LH_2	LOX	LH_2	LOX	LH_2	LOX	LH_2
1995	62	202	33	477	79	160	27	10	1.7
2000	163	531	88	1260	209	421	70	27	4.4
2005	427	1390	231	3290	547	1100	184	70	11.5

Comparison of terrestrial vs lunar masses of required propellant. Production on lunar surface is greater than sum of propellants delivered to LEO and GEO orbit by amount equal to that used in cis-lunar transportation of propellant product.

1) launch vehicle capacity limitations (Ref. 9-10)
2) saturation of orbital slot allocations
3) saturation of satellite-served communications markets through overcapacity or high price
4) political blockage of GEO utilization

On the other hand, there are several factors which could tend to increase GEO traffic--among them are:

1) substantial reduction in the price of launch services
2) substantial increase in Third World demand and ability to pay for domestic communications services
3) new uses for GEO developed during the next decade
4) major rise in petroleum prices, with consequent demand for extensive video teleconferencing (Ref. 9-10)

We should note that space communications is today one of the fastest growing industries of its size. Such a growth rate may or may not be sustainable over the next two decades. The broad-based communications industry has to date been capable of attracting very large amounts of capital (assets of the Bell System are in the $100-billion class) because communication is a "must have" service among nearly all consumers, and it facilitates substantial productivity improvements. Because communications equipment dominates the LEO [illegible] traffic load, traffic in the intermediate future necessarily depends on the ability of space communications to open new markets and to continue satisfying present markets.

9.4 FINDING WATER AT THE LUNAR POLES[1]

9.4.1 Statement of Problem

What is the fastest and least expensive method of ascertaining whether or not useful quantities of accessible water or other volatile ices exist at the lunar poles? This section documents our preliminary answer to that question. As discussed in the next section, the availability of significant quantities ($>10^3$t) of volatile ices on the Moon would have a profound effect upon future space operations requiring propellants or reduced metals (Ref. 9-11). A section on science background summarizes several references which support the credibility of possible lunar water. Based on what might exist in the permanently shadowed polar craters, we discuss several possible methods of detecting these subsurface volatiles. We then present a conceptual mission design (spacecraft, trajectory, instruments) as a suggested solution to the stated problem. A very rough cost estimate is presented using comparisons with other missions. Finally, we mention some other opportunities from the past and future to find lunar water.

9.4.2 Rationale

The energy required to reach LEO from the Moon's surface is about one-seventh of that required to launch from the ground to LEO. Obviously, at today's throughput, the energy advantage of going to the Moon is outweighed by the capital investment (even in terms of mass alone) required to establish lunar operations. We already know that the Moon holds accessible oxygen (about 50% of soil, by mass) which can be liberated with available solar energy (Ref. 9-12.) It has been shown in sections 7 and 8 that OTVs can be provided with lunar oxygen in LEO with about a 2:1 advantage over terrestrial oxygen in terms of the mass which must be carried from the ground to LEO (i.e., one Shuttle launch can be made to do the work of two). This scenario involves all-chemical propulsion and hydrogen from Earth only. Additional leverage can be obtained with the use of low-thrust or (much later) electromagnetic launchers, for some mission phases.

However, if hydrogen becomes available on the Moon, the leverage is increased to a factor of 5 or greater, drastically reducing the required throughput to support a given level of interorbit and escape operations in space on the most difficult leg of the transportation network. With both fuel and oxygen relatively seven times more accessible in energy terms, the scale of space operations can be magnified by a large factor with small increases in operations costs (assuming amortization of fixed and capital costs over a large throughput). If hydrogen is available on the Moon, then, theoretically, only useful payload and associated airborne support equipment needs to be brought from Earth to LEO, which effectively means that instead of 20-25% of GEO-bound launch-vehicle payload capability being devoted to equipment arriving at the final destination, 100% of payload can be "useful" equipment. This

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[1] The principal spacecraft and trajectory work was done by Chauncey Uphoff, with assistance from David Skinner and others who are referenced. The original problem was posed by Robert Staehle, who also wrote the sections on rationale and scientific basis. Other work, including alternate concepts and mission complexity were prepared jointly.

assumes a reusable space-based OTV. The leverage is, of course, greater for missions of higher characteristic velocity where a larger fraction of payload must be devoted to propellant. And with a local source of propellant, operations from the Moon (e.g., returning to Earth) no longer require the costly, high-throughput supply line of propellant from Earth.

At present, and for some time to come, the first leg of space transportation, getting off the Earth's surface, is the most expensive per unit of useful payload delivered to common destinations such as the Moon, GEO, asteroids, and Mars orbit. This arises for two reasons. First, the vehicles for Earth to LEO transit are intrinsically more complex because of high characteristic velocity, high surface gravity, and the variety of flight regimes. Second, the amount of mass carried on this first leg always exceeds the amount carried to the final destination, because of aerodynamic shrouds or the Shuttle Orbiter itself, and because of terrestrial propellants required to get beyond LEO, midcourse equipment, etc.

At Shuttle launch costs of about $3,000/kg, we can afford to spend perhaps 80-90% of that amount initially for every kilogram which can be taken off the Shuttle and delivered through another portion of a transportation network. If lunar hydrogen and oxygen are available and there is a market to deliver 200 t/yr of equipment to GEO, we can afford to spend approximately $(0.9) \times (200\ t) \times (2.8) \times (\$3 \times 10^6/t) = \$1.5 \times 10^9/yr$ to deliver the required lunar-derived OTV propellant to LEO.

Without a detailed cost analysis and projection of competing launch costs, it is not clear if a 200 t/yr market is large enough to support commercial lunar propellant operations. But lunar hydrogen will lower the market breakeven level in any case by increasing the saving in launch vehicle throughput, to support a given level of OTV operations. In addition, if water permafrost exists, the extraction of hydrogen and oxygen is simpler (by electrolysis of water) than the extraction of oxygen alone from lunar regolith (by molten electrolysis of rock). Liquefaction of hydrogen is not simple, but perhaps this could be done mostly in LEO, with the propellant payload carried to LEO as water. Production of hydrogen on the Moon could be limited to that necessary for the trip to LEO.

Space station operations could be affected substantially by the use of lunar hydrogen and oxygen. Not only would OTVs be serviced, but the likely OTV-derivative vehicles used to support lunar operations would also be serviced. At least one station would take on an even greater operations-oriented role than presently envisioned. If successful in reducing the cost of space operations, then more users could be expected, increasing the quantity and frequency of space station operations in spite of a reduction in launch vehicle operations.

A second advantage of lunar hydrogen in large quantities is its possible use in the reduction of lunar metal oxides (of which the soil is composed) into free metals and oxygen. Many terrestrial processes are based on the use of H as a reducing agent (frequently in the form of an acid). Other elements rare on the Moon, such as F, Cl, C, or N, may be required in combination with H to employ the most advantageous reduction processes, so finding H alone may not be as important as in the case of propellant manufacture. Free metals can apparently be produced by energy-intensive processes not involving any reagents.

The principal advantages, for metal production, of finding indigenous reducing agents would be first to lower the energy requirements, and second to enable utilization of well-understood terrestrial processes. Because less sophisticated alloys would be available from lunar sources than from terrestrial vendors, it may be some time before lunar metals are competitive with terrestrial metals in LEO on a mass basis. For applications on the lunar surface, however, structural performance of lunar metals may be a small fraction of that of common terrestrial products, while still maintaining a commercial advantage because of vastly reduced transportation costs.

9.4.3 Scientific Basis (Summarized in Table 9-12)

To maintain the proper perspective, it is important to remember that there is not a single item of real evidence for or against the existence of ice on the Moon. At this time the arguments on either side of the issue can only be speculative. The proper instrumentation has not been flown over the lunar poles. Since the ice, if it exists, resides at the bottom of permanently shadowed craters, it was undetectable by the cameras which Lunar Orbiter carried.

Hydrogen is already known to exist on the Moon, but only at levels of 10-100 ppm in lunar regolith. In this case the source is solar wind implantation. This concentration is insufficient to be operationally important because of the amount of soil which must be contained and heated to liberate significant quantities, i.e., 10^4-10^5 kg soil per 1 kg hydrogen.

The first direct speculation about lunar ice was published in 1961 by Kenneth Watson, Bruce Murray, and Harrison Brown (WMB), all of Caltech (Ref. 9-1). Their work (Ref. 9-13) appears to have been prompted by Urey, Herzberg, and Kuiper, who asserted that, while volatiles might be present, water was out of the question (Herzberg suggested SO_2) (Refs. 9-14 and 9-15). WMB asserted that "water, in the form of ice, is the only common volatile that could be stable for a period of time comparable to the age of the Moon," and that "absence of lunar ice would indicate an extremely small amount of chemical differentiation of the lunar mass." Apollo results have since indicated substantial differentiation, and WMB went on to examine the sources and loss mecnanisms of water, assuming it was released from all over the lunar surface during and after differentiation. In fact, WMB viewed the existence of water as a tool in measuring the degree of differentiation, and made no mention of its possible operational importance. (They were scientists, besides which, just getting to the Moon seemed hard enough in 1961).

In 1962 E. J. Opik (Ref. 9-16) recognized the WMB arguments for stability of water at the poles, but suggested a solar wind source of hydrogen creating water chemically which accumulated "up to 100 meters equivalent thickness" in permanent shadow. The solar wind source has not been verified. In 1963, V. S. Safronov and Ye. L. Ruskol (Ref. 9-17) pointed out some shortcomings of WMB and Opick, namely that ". . . the traps were filled to capacity at the very start of the epoch of liberation," but that "the ice existing in these traps . . . is to a practical effect preserved. It has for a long time been impossible to add to this ice." From a practical point of view, this simply means that there can still be plenty of ice present, though its quantity may be limited from the perspective of 100 years of utilization. Conservation may become important, but is unlikely to be an overriding concern for the first few decades of utilization.

Table 9-12. Is There Water at the Lunar Poles?

No real evidence, for or against, exists regarding the presence of lunar polar ice.

Permanently shadowed craters exist at latitudes higher than ~70°, covering about 0.5% of total lunar surface. Surface temperatures are 120 K and below, at which several meters of water ice are stable over billions of years (Watson, Murray, Brown, 1961).

Source of water: presumably outgassing during and after differentiation. Released water does not achieve escape immediately, and some migrates to cold polar surfaces from all latitudes.

Theoretical loss mechanisms (sublimation, meteoroid impact, solar wind bombardment, etc.) remove a small fraction of total expected _if_ proposed sources are accurate.

If ice exists, it will almost certainly be all H_2O, and will be present in operationally large (i.e., many decades' supply) quantities.

The ability of a γ-ray spectrometer to detect hydrogen, and therefore water, at the lunar poles was noted in 1975 by Metzger, et al. (Ref. 9-18), while Burke noted its potential operational utility for polar base siting in 1977 (Ref. 9-19). It is not known whether either of these references were the first ones presented for their respective concepts.

In light of Apollo results, James R. Arnold undertook in 1979 to review the WMB work (Ref. 9-20). "The result (was) to confirm, so far as a paper analysis can do so, the essential correctness of the ideas of WMB." Since then, Arnold indicated (Ref. 9-21) that he is aware of two further analyses, one by Hodges (Ref. 9-22) and one by Lanzerotti, et al. (Ref. 9-23), in 1980. Arnold said Lanzerotti was very negative on the existence of ice because of sputtering by solar wind, but Arnold "does not believe" this conclusion given the analyses which he and others have done of this loss mechanism. Hodges showed that the area of sufficiently low equilibrium temperatures in permanent shadows was less than the WMB and Arnold estimates, because of greater reflection and reradiation from illuminated areas into the shadowed areas. Even the 4/5 reduction in suitable area attributed by Arnold to Hodges would be operationally insignificant. A single crater with a 1-km^2 floor having a 100-m layer of ice/regolith and an ice concentration of 500 kg/m^3 (i.e., about 15% of regolith mass) would contain over 10^7 t of water.

Based on the reviewed literature, it appears that if water is indeed at the poles, there will be plenty to supply projected space operations for at least several decades.

9.4.4 Instrument Selection

Several instruments were considered as candidates for a simple water detector. Multispectral imaging was one of the first candidates to come to mind but was rejected after some consideration because of the difficulty of supplying a light source in the permanently shadowed regions near the poles. Several possibilities for providing light were considered, including a laser beam, a magnesium bomb, and even a large reflector to reflect sunlight into the craters. On cursory examination all these schemes appeared too complex for further consideration. The X-ray spectrometer was similarly rejected because it requires solar X-ray illumination.

One possibility that might receive attention in the future is the concept of a penetrator to imbed itself into one of the craters. Navigation requirements for such a mission and the complexity of the device itself, required to achieve positive results with certainty, would be very expensive and would require difficult technology development. Because of the possibility of water ice being covered by either a thin layer of regolith or impact dust, all the detection mechanisms depending on reflected light were rejected.

Three other possible detection methods were considered. A surface rover could carry the required instruments and could even drill down several meters to collect samples. However, this seemed a much too complex and thorough method to answer the basic question. A rover would seem more appropriate for mapping out a single potential resource site which appeared promising based on other data. An earlier suggestion (Ref. 9-24) was to launch a water or OH 'sniffer' on a collision trajectory with a pole. Just in front of this probe, several projectiles or 'bombs' would be launched, impacting the surface perhaps 30 sec to 1 min ahead of the sniffer, which would then pass through the ejecta clouds. A tuned instrument would result in a brief, low-resolution data item indicating whether or not water was within a few meters of any of the various impacts. Two problems here are the relatively high probability of a false negative answer because of inopportune impact sites or chance surface obstructions such as boulders, and secondly, that the method would give little indication of the actual location or extent of any water deposit indicated. Consequently, this method cannot indicate whether or not a quantity detected is "useful" in engineering terms. A final, similar method was suggested (Ref. 9-25), and suffers for the same reasons. This was to launch a cluster of explosives or passive impactors and attempt to detect water using an in-place experiment on an Apollo Lunar Surface Experiments Package (ALSEP), the Lunar Atmospheric Composition Experiment (Ref. 9-26). This option was not investigated because it cannot answer the basic question, but it is thought that the ALSEP stations have been irreversibly turned off.

The two principal instrument candidates were a γ-ray spectrometer and an electromagnetic sounder, which would include a VHF transmitter and receiver in the range of 100-150 MHz which could penetrate the water ice and would give an excellent reading for the depth of the ice by yielding multiple reflected signals off sharp subsurface gradients in the dielectric constant. This however is not an unambiguous determination of ice, and while a dual or triple frequency or a frequency-modulated signal might lend more information, there are still many things that might give the same kind of return signal. The γ-ray spectrometer was ultimately selected as the principal instrument for the detection of water. The major reason is that it requires no input source as it detects gamma rays which are generated by incoming cosmic rays interacting with hydrogen. The determination of water is almost unambiguous, using the γ-ray spectrometer, and the final reason for the selection is that there is a flight spare from one of the Apollo missions available and in good condition here at JPL (Ref. 9-27).

The final selection of the instruments was then a combination of the electromagnetic sounder, in order to yield a good depth measurement in case water is found, and, of course, the γ-ray spectrometer for the positive identification of the water. The exact nature of the γ-ray spectrometer is not specified here. The Apollo instrument (Ref. 9-28) is of the scintillation type and is less sensitive than solid state devices now under development but not yet available. The Apollo γ-ray spectrometer would be able to detect water up to approximately 1 m beneath the surface (Ref. 9-27), even if 1 m of dust or regolith is above the water. The new instrument being developed at JPL would be able to detect water several meters beneath the surface and would have much better resolution (Ref. 9-27), but would require expensive development over the next two years. The γ-ray spectrometer, in combination with an electromagnetic sounder to yield depth and corroborative absorption characteristics, is a combination that would yield a very high confidence in detection of water. The following preliminary spacecraft description assumes the use of the Apollo γ-ray spectrometer and the standard electromagnetic sounder using a yagi antenna in the range of 100-150 MHz (Ref. 9-29), similar to that flown on Apollo 17 (Ref. 9-30). At 100 MHz the sounder can detect ice-rock interfaces to a depth of 10-15 m. Greater penetration could be achieved with lower frequencies, but might not be considered easily accessible for early commercial operation.

9.4.5 Trajectory and Midcourse Correction Requirements

It is assumed that the spacecraft will be launched from the Shuttle using a solid rocket in a spin-stabilized mode. One possibility is the McDonnell Douglas PAM D, a substantial solid rocket that is compatible with the Shuttle and the Shuttle spin table capability (Ref. 9-31). A low-energy Earth-to-moon trajectory requiring about 115 hr for the transfer has been selected. This trajectory requires a launch C_3 of from -1.9 to -2.1 km^2/s^2, depending upon the particular geometry and launch opportunity selected. Figure 9-3 shows the launch mass capability of the PAM D injection module, showing that a total launch mass of up to 800 kg can be expected, although it is anticipated that much less mass will be required for the suggested mission.

Translunar midcourse correction requirements will be dictated by the errors incurred during the translunar insertion (TLI) and can be expected to range from 80 to 120 m/sec (3σ). Execution errors of this first midcourse maneuver (expected to be required in the time period from 5 to 15 hr after launch) will propagate to the Moon and cause errors of the order of 500 km in periapsis altitude. A second midcourse maneuver will therefore be required to remove these execution errors and can be expected to occur from 70 to 90 hr into the transfer.

Orbit insertion requirements will be of the order of 800 m/sec for insertion into a 100-km circular orbit. Orbital trim maneuvers of 20 to 40 m/sec will subsequently be required to adjust the position of the perilune and the eccentricity for maximum orbit lifetime.

Because of these requirements, it has been assumed that the orbit insertion maneuver will be made using a solid rocket motor and that the midcourse correction and orbit trim maneuvers will be accomplished using a monopropellant hydrazine system of I_{sp} = 220 sec, and that the total ΔV required for the midcourse and trim system will be 200 m/sec. Additional propellant can be loaded if desired in order to provide orbit sustenance maneuvers to lengthen the lifetime of the orbiter. The assumptions above are quite similar to those made for the lunar polar orbiter study (Ref. 9-31).

9.4.6 Spacecraft and Subsystem Assumptions

The two major experiments require a total power of from 45 to 50 W (30 W for the electromagnetic sounder, 5 W for the Apollo γ-ray spectrometer, plus 10 W for the γ-ray spectrometer heating requirements). An S-band transponder capable of supporting a 5 kbits/s data rate will require approximately 17 W. Although this may sound large it is assumed to have this value so that the full data rate may be transmitted at any time via an omni-directional antenna. This relieves the spacecraft of the burden of pointing a medium- or high-gain antenna at Earth. Thus a total power requirement for the spacecraft, including 15 W for thermal control and engineering housekeeping, is under 90 W.

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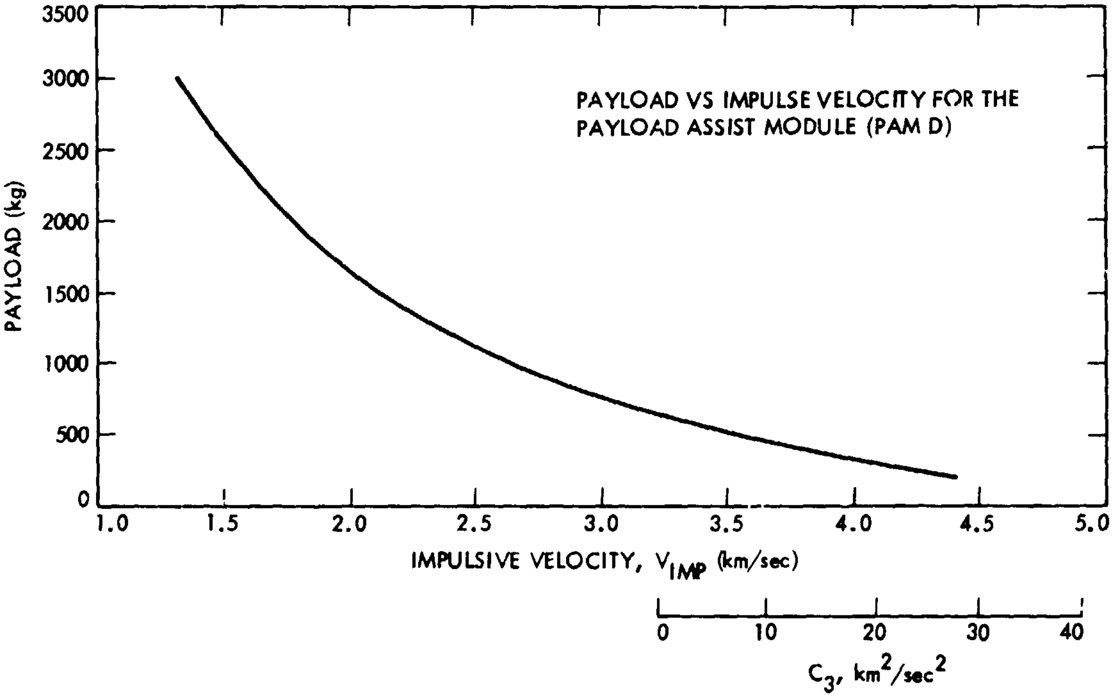


Fig. 9-3. Injection Module Performance (PAM D)

9.4.7 Orbit Selection

The orbit selected must be fairly near the poles but need not be exactly 90 degrees. Orbits of 85 or 95 degrees are much less likely to have a short lifetime than orbits exactly at 90 degrees. The orbit sustenance requirements for nearly polar orbits are considerably less than those for truly polar orbits. This is because in a truly polar orbit the spacecraft comes over exactly the same lumps at the poles each revolution. Some "frozen" orbits have been identified for several models of the lunar gravity, and it is possible to find orbits with periapsis near the south pole that have very long lifetimes. Nevertheless, one would wish to examine both poles, and some orbit sustenance maneuvers will be required. Insofar as the spacecraft needs a midcourse correction system in any case for delivery to lunar orbit, it is not a major expense to load more hydrazine on board in order to provide orbit sustenance capability.

The spacecraft will be placed into an 85- or 95-degree orbit by firing along the velocity vector at closest approach to the Moon, and the orbit will have a periapsis of about 100 km and an apoapsis altitude from 100 to 500 km. The spacecraft will remain in the spin-stabilized mode with the instruments mounted perpendicularly to the spin axis so that each revolution of the spacecraft will allow the instrument field of view to sweep across the Moon directly below at the poles. Another possibility which would yield nearly full coverage of the Moon during the mission is that the spin axis can be reoriented normal to the orbit plane; the instruments mounted as described above would then sweep along the ground track once each spacecraft revolution. The signal so received could then be calibrated according to a horizon sensor or a signal returned from the electromagnetic sounder. Adequate coverage will be available given even a very slow spin rate, but it is expected that the spacecraft would spin at approximately 60 rpm, yielding one full sweep per second during any pass over the poles. Expected duration of the mission is approximately 3 months. Very little attitude and orbit sustenance fuel would be required for such a mission.

Mass summaries are presented in Tables 9-13 and 9-14. Mass estimation was assisted by David Skinner's software.

Table 9-13. Spacecraft Mass Estimate[a]

Equipment	Mass (kg)
Components	93.3
Solar Array	10.5
Harness	10.7
Structure	35.1
γ-Ray Spectrometer Boom	5.0
Thermal Control	6.7
Satellite Adapter	7.3
Mission Equipment	22.6
Total Spacecraft Mass	191.2

[a] C. Uphoff and D. Skinner

Table 9-14. Mission Mass Statement[a]

	M (kg)	ΔM (kg)	ΔV (m/s)
Mass of Spacecraft in Orbit	191.2		
Orbit Sustenance	200.2	9.0	100
Lunar Orbit Insertion			
Solid Rocket Inerts	207.0	6.8	
Solid Rocket Propellant	275.0	68.0	810
Earth-Moon Midcourse and Attitude Control	288.2	13.2	100

Injection Mass Required ($C_3 = -2\ km^2/s^2$) = 288.2 kg
Shuttle/PAM-D Injection Capability = 800

[a] C. Uphoff

9.4.8 Mission Cost and Complexity (Summarized in Table 9-15)

If restricted to the original objective of searching for large, accessible deposits of ice on the Moon, the mission we examined is very simple. The small spacecraft accommodates only two instruments with identical objectives, no pointing variations (except by virtue of spacecraft spin), and apparently only two modes of operation, i.e., both on or both off. The PAM D was suggested as an injection stage only because it will be Shuttle-qualified and in routine use, thereby reducing test and integration costs. In 1981 dollars, PAM D hardware costs are listed as approximately $3.0 million, general mission specific costs are under about $2.1 million (Ref. 9-31), and we estimate optional mission specific costs payable to the stage contractor as being under $2 million. Shuttle launch charges, which are not historically covered by the mission budget (but must be paid somewhere) would be under $25 million. Using the Apollo γ-ray spectrometer and an off-the-shelf solid insertion stage and hydrazine system, $20 million would seem more than adequate to build the spacecraft, even by traditionally expensive methods at JPL. A flight team of 60 people with a 2-month mission and 9 months training (quite generous) yields approximately $6 million in personnel costs, with another $6 million allocated for mission-specific ground equipment and facilities (most of which would in fact be available as surplus or on standby from other projects). Data analysis could require another $5 million if done with extreme thoroughness and little cost consciousness. This yields a total around $50 million exclusive of launch charges, or $75 million inclusive.

Should a mission like this be flown, the greatest temptation would be to add "a few" instruments, use up the PAM D payload margin, and make it into a scientific "Lunar Polar Orbiter". This would almost certainly drive the costs into the Mariner Mark II class of $200 million. While such a mission's constituency would be broadened, the past record of post-Apollo lunar mission proposals makes this mission look like a poor chance in the face of scientific interest in other bodies. Whether a resource mission under $100 million can be funded is an entirely different question--no one has ever tried. The payoff could be very large indeed, or the mission could yield only a little poorly correlated Apollo-type data to that which is already shelved and unanalyzed.

9.4.9 Lunar Science Mission Candidates

Because as proposed they would have carried sensitive γ-ray spectrometers, both the NASA Lunar Polar Orbiter (LPO) (Ref. 9-32, 9-33) and later ESA-NASA Polar Orbiting Lunar Observatory (POLO) (Ref. 9-34) could have detected near-surface (within 1-2 m) water at the poles. Neither proposal was authorized as a funded program, and it does not appear that either is likely to be revived soon. A recent announcement was made of Soviet intentions to launch an LPO/POLO-type mission in 1986-87, in which other countries, including the U.S. and France, have been invited to participate (Ref. 9-35). In the current political climate, U.S. participation appears out of the question. French participation is a distinct possibility, but any such involvement is as yet unfunded, and space science is not a top French priority in astronautics (Ref. 9-36).

Table 9-15. A Summary of the Proposed Lunar Water-Discovery Mission

Mission Characteristics
Spin-stabilized, 300-kg spacecraft
2-month mission
Apollo spare γ-spectrometer, a simple EM sounder derived from Apollo
Minimal new facilities
Cost and Complexity
Single mission objective
Two instruments, low data rate, on or off, omni-antenna
In-production PAM-D standard injection stage, STS-qualified, large mass margin (500 kg)
Short mission, no instrument sequence planning
Low data quantity, modest processing required
Frequent launch opportunities, compatibility with other Shuttle payloads
Total project cost ~$70M

[a] From C. Uphoff, R. Staehle, and D. Skinner

If the French do participate in the Soviet mission, they will most likely provide several high-quality instruments, including a γ-ray spectrometer. If the Soviets go it alone, as Prof. Jacques Blamont feels they are prepared to do--though perhaps with a year's delay (Ref. 9-36)--it is not clear whether proper instrumentation of high enough quality would be included. A false negative answer regarding the presence of water could be a likely result. We have seen no evidence of Soviet operational interest in lunar ice. The operational significance of an ice discovery was new to Blamont, and might not yet have occurred to the appropriate Soviet planners.

9.4.10 Conclusion and Recommendation

Our work suggests that at least one resource-oriented mission whose results could be profound can be conducted for much less than the cost of traditional solar system exploration missions. Another mission concept of comparable complexity, designed to discover Earth Trojan asteroids, has also been reported (Ref. 9-37). The key to low cost is a single, narrowly focused mission objective. Nothing is added to the mission which is not essential to achieving the objective. In reality, the discipline required to improve this condition may prove beyond the capability of a government-sponsored project because of the varying political motives of all the groups and committees which must endorse a successful proposal.

If space resource operations are to be viable in 20-30 years, and if we are to retain the option for early utilization of as yet unconfirmed resources which could profoundly affect space operations, then the confirmation of those resources (if they exist) must come within the decade so that other developments may proceed accordingly. If discovery occurs later, after massive investments in the development of more difficult technologies, the space resource industry could be substantially set back by the competitive threat posed by new, easily accessible competing resources available at lower cost. The resulting loss write-offs could retard development of both traditional and now-speculative resources.

9.5 EARTH TROJAN ASTEROID DISCOVERY

Another mission with a focused-objective philosophy similar to that for the lunar polar-ice mission was studied in 1979, to search for Earth Trojan asteroids (Ref. 9-37). Cost and injection stage can probably be similar to the lunar mission. The spacecraft carries a single, simple optical instrument (a 0.25-degree-field-of-view telescope with a photomultiplier, but no imaging) which scans the sky on the far side of the spacecraft from the Sun. A simple technique results in Earth Trojans appearing in two different locations, while fixed stars appear in one location, allowing the signals to be separated. Precision ground observations are targeted by spacecraft discoveries in order to obtain asteroid orbital elements.

The following two charts (Figs. 9-4 and 9-5) illustrate the Earth Trojan discovery mission. Ground-based search operations are sufficiently marginal (because of sky brightness and unfavorable viewing geometry) that this mission could discover asteroids of great importance (if they exist) even if ground searches turn up nothing. Ground searches are appropriate, however, for several more years, because they should be capable of detecting larger objects which could exist. The ground-based orbit-determination observatory, to follow up any spacecraft discoveries, can employ a much narrower field of view than required for search observations.

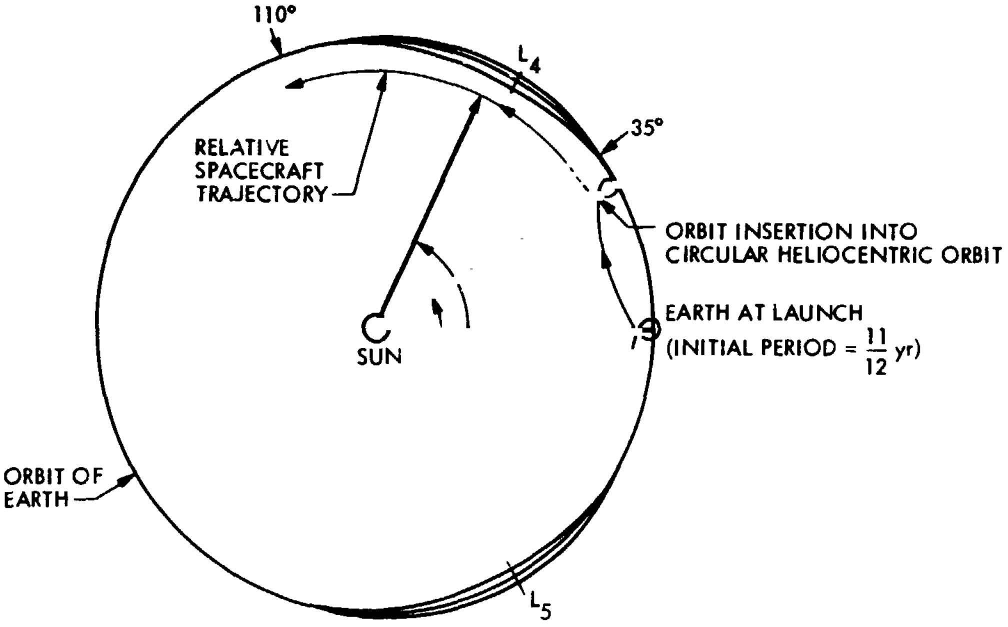
9.6 NEAR-EARTH ASTEROIDS

Asteroids which come near Earth's orbit while in their own independent, heliocentric orbits, are referred to as near-Earth asteroids. Among these objects, a few have a low enough inclination and eccentricity to be relatively accessible by occasional ballistic trajectories. Neal Hulkower and David Bender, both of the Advanced Projects Group, have identified about five known asteroids which, during infrequent opportunities, are easier to return from than the Moon, in terms of energy (see, for example, Ref. 9-38).

With funding from advanced planetary mission offices of NASA Headquarters and under the oversight of Robert Staehle in the RUSR group, a brief consulting study was performed by consultant Jerome Wright. The task was to loosely define an asteroid research program aimed at resource utilization, rather than purely scientific understanding. Wright noted generalized selection criteria for favorable asteroids, and a plausible sequence of missions leading to establishment of resource operations. Wright also noted particular problems with the infrequency of opportunities to the most accessible asteroids.

While near-Earth asteroids remain of interest, the transit time, communications and solar distances, and the limited number of favorable transfer opportunities combine to make them look less attractive for early resource operations than the Moon or Earth Trojans (if discovered).

C-4



TRAJECTORY TO SEARCH THE ENTIRE L4 REGION IS SHOWN IN THE ROTATING SUN-EARTH COORDINATE SYSTEM. SPACECRAFT TRANSFERS IN 11/12 OF A YEAR TO 30 DEGREES AHEAD OF EARTH AND AN INTERIOR ORBIT, THEN MOVES AHEAD FASTER THAN EARTH, SCANNING THE ENTIRE REGION OF INTEREST IN ABOUT 4 YEARS. L5 SCHEME IS SIMILAR, EXCEPT THE SEARCH STARTS AT ABOUT THE 250-DEGREE POSITION, PROGRESSING TOWARD EARTH.

Fig. 9-4. Earth Trojan Search Trajectory (Ref. 9-6)

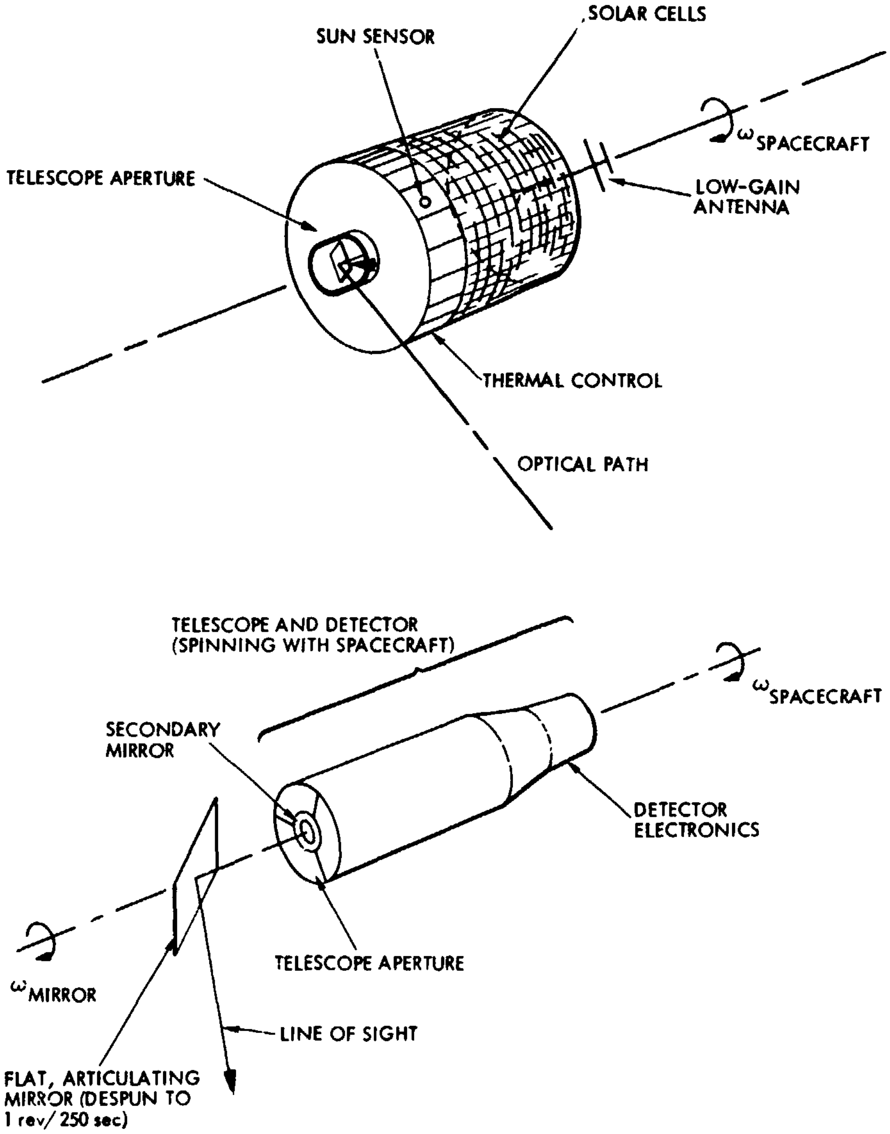


Fig. 9-5. Earth Trojan Search Concept

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APPENDIX A

ASTEROID PROGRAM

ASTEROID PROGRAM

A Report for
Jet Propulsion Laboratory

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1982 September

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CONTENTS

INTRODUCTION

The basic objective of this study is to describe an asteroid exploration program which could lead to the exploitation of asteroidal resources as an eventual goal. The study identifies mission possibilities and science objectives consistent with this goal.

This report describes three aspects of the asteroid program. First is a description of the program concept and its specific objectives. Second is a description of a set of mission projects which can comprise the program. Third is a discussion of specific mission types, with emphasis upon the first mission.

This study has been conducted for the Jet Propulsion Laboratory, through Section 312, Advanced Projects Group. It has been a four man-week effort (JPL consulting agreement KO-740366). It has been done in coordination with a complementary study for The Planetary Society.

Asteroids are a major source of nonterrestrial resources, which can be used at the asteroids or transferred elsewhere. Energy costs for use of near-Earth asteroids are lowest for all nonterrestrial sources. The energy requirements for getting to the best asteroids is essentially the same as going to geosynchronous orbit. The return velocity increments typically are a few hundred meters per second, and less than 100 m/s in the best cases. Material can be returned using compact rotary launchers. The primary penalties for using asteroids are the time associated with transfers and the restricted opportunities.

Asteroid resources will be used. At issue is what specific resources are available, what difficulties are associated with mining and transport operations, and when and where specific resources are needed.

PROGRAM DEFINITION

PROGRAM CONCEPT

The program described here is a coherent set of projects directed toward the goal of asteroid usage capability. It defines a set of objectives for each project which supports the program goal.

The program is assumed to start in the mid-1980's with commitment to the first mission. The first launch would then occur in the late 1980's or early 1990's. The program is defined to end with acquiring the capability to begin major usage of resources. Exploratory missions within this program might make use of some resources, but this is not here considered a major usage.

Programs which are science oriented and resource oriented have much in common during the early years, in the discovery and Earth-based analysis of asteroids. However, a significant bifurcation will occur eventually because the science objectives are to understand the asteroids as a system, to understand the evolution of that system and how it relates to solar system evolution. Resource objectives focus on how to use the asteroids which are conviently located relative to potential human activity sites in space. Thus most asteroid scientists are concerned with the main belt, while resource investigators are concerned with near-Earth asteroids. These two groups support mission concepts which are significantly different from each other.

A coherent, long-term mission planning and design effort needs to be established for a near-Earth asteroid program. Continuity of effort and consistency of goals is essential for it to be effective. The work needs to cover mission definition within an overall program. Manned, robotic, and orbiter/sampler modes need to be investigated.

Economic arguments for the program should be realistic. What resources are used, where and when they are used, depends upon economics, not popular ideas or wishful thinking. Earth is not running out of resources, it is only cheap sources that are being used up. Asteroidal resources are not needed to save civilization on Earth, but they can be very important where they are cost effective. Costs on Earth, per ton, presently range from 8 M$ for gold and platinum, to 160 k$ for silver, to 440 $ for steel, and 60 $ for scrap steel. It seems very unlikely that asteroidal iron could be brought to Earth for prices comparable to that of terrestrial steel.

Transportation costs to orbit will be about 3 M$ per ton in 1985, which far exceeds the intrinsic cost of basic materials on Earth. There is clearly a potential major savings from using asteroidal materials in Earth orbit, in contrast to their use on Earth. This means that asteroidal resource usage should not be considered completely out of context with potential operations in Earth orbit, where essentially all of the returned material might be used.

If bulk structural material is needed in orbit during the late 1980's and 1990's, then Shuttle External Tanks are almost certainly the cheapest sources of up to a few thousand tons. The tanks could also provide a few hundred tons of volatiles (oxygen, hydrogen, and water). Asteroidal, or other nonterrestrial, resources will probably not be needed until large scale operations are started in Earth orbit. At that time, they could be low cost sources of a wide variety of materials.

The overall availability of funding is, and probably will remain, very tight. This clearly impacts the number and scope of possible asteroid missions, putting a premium on an ability to accomplish specific goals with the least cost. An important result of the near-term funding limitations is that crucial preliminary mission analysis might not be sufficiently funded. This work is important not only to identify opportunities, but to characterize the missions and determine the most cost effective manner for conducting the missions. If this work is not adequately done, the attractiveness of the missions might not be sufficiently well described to generate interest in getting the missions under way.

PROJECTS

The individual projects are each based upon a specific mission associated with near-Earth asteroids or a specific ancillary task supporting a mission or technology development. The projects can be placed into four general categories, as follows:

1. Detection and orbit determination is the most primary activity. This needs to be enhanced for a major increase the number of known near-Earth asteroids.

2. Analysis of asteroids and meteorites using Earth-based observations and laboratory work for characterization

3. Exploration of selected near-Earth asteroids in any of several mission modes to characterize specific asteroids.

4. Development of capabilities to process and use asteroidal materials.

Only mission projects are discussed further.

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PROGRAMMATICS

Schedules

A serial program based upon including each mission type, with project starts at the completion of the preceding mission, would require at least 20 years. This means extending the program until 2010 or later, which seems unnecessarily long. The program might better be based upon overlapping projects and less than the full set of mission types. No attempt is made here to show a particular program structure; it could be meaningfully done only after substantial analysis. Estimated schedules for individual projects are listed in the following table.

```
--------------------------------------------------
                   Time From Project Start, years
PROJECTS           ...1...2...3...4...5...6...7...8
Rendezvous         .............-L...
Sample Return      ...............-L-.......
In-situ Automated  .......... .....-L-......
Manned, fast       ..................-L..
Manned, slow       ...................-L--.......
--------------------------------------------------
```

L: nominal launch time
-: range of variation in launch

Costs

Cost estimates were made based upon analogies with other projects of roughly similar scope. The estimating was approximate, without extensive analysis. The slow manned mission will necessarily require far more hardware, almost entirely new, than the fast mission based upon using modified Apollo designs. Projects made sufficiently ambitious can exceed the larger estimates.

COST RANGES, M$, FY82 (rough estimates)		
Rendezvous, minimum	70	90
expected	80	120
Sample Return	300	600
In-situ Automated	500	1000
Manned, fast	4,000	8,000
Manned, slow	10,000	20,000

MISSIONS PLAN

The basic types of missions to asteroids are rendezvous, sample return, in situ, and manned. These types provide different levels of capabilities with which to meet the objectives, with increasing levels of cost, complexity, and capabilities.

Orbital mapping is a primary requirement to determine volume, mass, surface properties, and some elemental abundances. This would be the objective of a rendezvous mission, but would be included in a sample return or in situ mission if there was no precursor rendezvous mission to that asteroid.

For an objective of resource usage, a single near-Earth asteroid would probably be selected according to its orbital characteristics and its type. Type will probably be relatively unimportant except that it is desirable to use a water-bearing asteroid. Pure metallic asteroids will probably be avoided because they cannot be processed with foreseeable technology. A series of missions could be sent to the selected asteroid for mapping, sampling, and possibly trial processing of materials. The orbital characteristics should assure suitability for mining operations over a minimum of a few decades.

For resource usage, the transfer from the asteroid to Earth is more important than the outbound trip. The phasing of return opportunities is an important factor, perhaps as much as the energy requirements. The phasing and return energy requirements should be analyzed for several asteroid candidates to better understand the practicality of resource usage. It appears to be the case that many proponents of asteroid usage have concentrated upon energy requirements and ignored the phasing, which has a strong impact upon the practicality of using a particular asteroid during a span of a few decades.

The transfer of asteroidal material to Earth appears to best be done by use of electrically driven rotary launchers which throw packages of perhaps a few tons mass. These packages would have a course correction capability and a thermal shield to withstand aerodynamic braking in Earth's atmosphere. The packages would be claimed in Earth orbit.

Mission studies should be supported for work determining the long-term suitability of specific asteriods for resource operations. While the geometry of the orbit of an asteroid might not change much over a 50 year span, for example, the asteroid can drift in its relative phasing with respect to Earth. This can preclude short, low-energy transfers for extended periods of time. For near-Earth asteroids, optimum trajectories are usually determined, but there are no studies for defining specific missions. There is also nothing being done for defining very low-cost missions using low-mass spacecraft. There was some recent work on a possible asteroid mission using Mariner Mark II, but that has been oriented toward main belt asteroids.

Asteroid 1982DB is a very good target for missions of various scopes, including eventual resource usage. An apparently good mission opportunity exists in 1990-91, but the characteristics have not yet been quantified. Both a moderately fast rendezvous and a round-trip mission might exist here. An excellent opportunity apparently exists in 2001. This opportunity includes a fast, low-energy transfer mission, and apparently a six-month round-trip mission. Thus a modest precursor mission could be done in 1991, followed by an advanced mission ten years later. This two-mission set might meet the rather ideal goal of achieving resource usage capability by around the end of the century. However, there is no current funding for a mission study. There is no funding either for tieing in an asteroid like 1982DB with an ultimate objective of a manned mission and resource usage. Even modest funding (a few tens of k$ per year) will still allow some important definitive work to be done. Such work done at this somewhat early point can serve to focus later work, and even induce additional mission work which might not otherwise come about.

If a mass launcher with a 500 m/s capability was placed on this asteroid, over a span of several years it could be used approximately 20% of the time to send material to Earth. This does not allow for additional constraints which might be applied, such as restricting the arrival times at Earth. The determination of specific departure windows requires further analysis.

MISSIONS DESCRIPTIONS

SELECTION CRITERIA

There might be a set of particular missions which best meets the overall objectives. The basic functions of orbital mapping, sample return, and in situ analysis might be combined into two missions, or even one. One or more of the functions might be delayed until a manned mission is done. These missions might include more than one asteroid. However, the total cost might be minimized by concentrating upon the single most attractive near-Earth asteroid.

Both celestial limitations and funding limitations are also major factors, along with program objectives, in determining which missions should be accomplished. A clear definition of asteroidal resource objectives is needed to determine the specific characteristics of the missions.

The ideal characteristics of resource asteroids are listed in estimated order of importance:

1. Low inclination, which improves the span of opportunities and reduces the general energy requirements. This is important for resource shipments to Earth.

2. An apsis near 1.0 AU, typically the perihelion. This is important for low-energy requirements and fast transfers.

3. Relative phasing determines the ability to use the best transfers. Good synchronization is important to allow repeats of the best phasing conditions. The relative phase angle can be defined as the central angle between the Earth and the asteroid when the asteroid is at its 1. AU apsis. The best transfer conditions usually occur when this angle is only a few degrees.

4. The second apsis should not be too far from 1. AU to keep energy requirements moderate. A good range would be between approximately 0.6 and 2. AU.

Current Best Choices

	Inclination	Periapsis
1982DB	1.4 deg	.95 AU
Anteros	8.7	1.06
1982HR	2.8	.81
Eros	10.8	1.13

TRAJECTORIES

Estimated trajectories for the 2001 opportunity with 1982DB are listed. The total velocity increments are shown for all propulsion events, including Earth departure. These are approximate, and require confirmation through additional analysis.

40 day	Minimum Time	5.0 km/s
190 day	Minimum Energy	4.4 km/s
6 month	Round Trip	9.0 km/s

RENDEZVOUS

A rendezvous mission has the least cost and complexity of the set of possible missions. It has the shortest development time and can be done with a small spacecraft. The mission objective would be orbital mapping, although a landing followed by modest surface analysis could be included. The mapping could be satisfactorily done with a set of 2 or 3 instruments. The relatively low spacecraft mass can allow use of fast transfer trajectories.

SAMPLE RETURN

A sample return mission could be done over a wide range of costs depending upon the complexity of the mission. A simple, low-cost return capsule could be used which would provide course correction and thermal protection for aerodynamic braking at Earth. Using aerodynamic braking, the capsule could be sent into Earth orbit or direct to the surface. This mission could be done using fast (6 month) or slow (2 year) trajectories.

Sampling should be done following the orbital mapping, which should be done during the same or a precursor mission. Several samples from the asteroid would be preferred.

AUTOMATED IN SITU

The automated in situ mission has a broad choice for its scope. Even a modest effort will be relatively expensive, and have limited flexibility. A semi-automated system using macro commands might be best to avoid time-lag problems with teleoperators. Technological and cost problems would probably preclude using a fully automated (robotic) system. It would be desirable for the system to work at several sites on the asteroid. A sample return capsule and its loader can be made for less cost than any substantial automated system, thus the sample return might more effective unless the in situ system can process enough of the surface to justify the additional cost.

MANNED

The manned mission clearly is the most complex, expensive, and flexible, with the ability to meet the widest range of objectives. A manned mission might be necessary to initiate the resource extraction processes.

The first manned mission probably should be limited to 6 to 8 months, using the fast round-trip trajectories. For this type of mission, modified Apollo hardware could probably be used to meet most of the hardware requirements. This would provide a substantial savings in development costs. In favorable cases, as with 1982DB, the return to Earth is sufficiently short (3 weeks) to allow return in only the Apollo Command and Service Module. A large quantity of material could be sent back separately on a low-energy transfer.

The longer mission is approximately two years in duration, with a stay time at the asteroid of several months. This would allow extensive operations, including setting up a processing facility. The total velocity increment would be somewhat less than for the fast round trip, but substantially more mass would have to be sent to support the mission.

OBJECTIVES

The general objectives for the exploration of an asteroid are clearly to gain sufficient information for establishing the technology and capability to use its resources. Additionally, there are science objectives which include the morphology and evolution of the target asteroid and asteroids in general.

The information needed for resource usage includes mineralogy, elemental abundances, distributions across and through the asteroid, and the structure of the asteroid. The early missions will probably not have more than capabilities for surface mapping of elements, morphology, some surface mineralogy, and density determination. Obtaining the other information will require sample return and extensive in situ investigations.

INSTRUMENTS

Optical Camera

CCD type, with spectral/polarization filters. Optical mapping yields volume and topography. from which density can be obtained. Reflectance spectroscopy yields mineralogy of the surface. Polarimetry yields surface texture and particle size range; it can be used to corroborate mineralogy.

Gamma-ray Spectrometer

This spectrometer yields the elemental composition. When used with the x-ray instrument, they can analyze all major elements, radioactive elements (Th, U, K), and some trace elements including H.

X-ray Fluorescence Spectrometer

X-ray fluorescence yields elemental composition. This instrument might be combined with the gamma-ray spectrometer.

IR Radiometer

This instrument can provide surface temperatures and albedo meausrements, and can provide some compositional information on volatiles.

SPACECRAFT

LOW-COST ORBITER

A low-cost, low-mass orbiter could be used for a simple rendezvous mission. It could be spin stabilized, and carry a set of 2 or 3 instruments to accomplish topographical, mineralogical, and elemental mapping. It could be compatible with a landing on the surface at the end of the mission.

ORBITER

A more sophisticated orbiter might be 3-axis stabilized, and carry a larger instrument set. It could be designed for multiple landings on the surface, or carry a lander or sample return capsules.

LANDER

A lander could be used for surface analysis, and could provide a platform for loading and launching sample return capsules. It could be built with a capability of moving to additional sites on the surface.

SAMPLE RETURN

The return capsules could be simple spin-stabilized units able to provide the departure impulse and course corrections. They might be designed to carry on the order of one kilogram samples back to Earth or Earth orbit using aerodynamic braking.

SURFACE STATION

A sophisticated station could provide extensive in situ operations both for analysis of the asteroid and launching of material to Earth.

MANNED

Manned spacecraft could be based upon Apollo hardware for a six-month mission. The Command and Service Modules might be essentially unchanged in general design and function. The Lander would be the upper stage of the Lunar Module, with landing pads and without the large propellant tanks. A Cabin Module, similar in function to the Apollo-Soyuz Adapter, could be used for living quarters, storage, and docking port. The short return trip to Earth could be done with only the Command and Service Modules.

ASTEROID PROGRAM Objectives and Hardware

The two-year mission might be able to make only very limited use of existing designs for the major structural units. The mission objectives would be extensive, involving considerable hardware for asteroid operations.

PROPULSION SYSTEMS

The propulsion systems needed for exploitation of near-Earth asteroids can economically be based upon use of ballistic transfers. Thus the systems would use short impulses, based upon chemical propellants or electric motor (rotary) launchers. Low-thrust systems are usually not suitable for fast transfers.

RECOMMENDATIONS

1. Specific opportunities for missions to 1982DB should be analyzed.
2. Phasing and return energy requirements should be analyzed for several asteroid candidates to better understand the practicality of resource usage.
3. Mission studies should be supported to determine the long-term suitability of specific asteroids for resource operations, coupled with studies of specific modes of conducting operations to meet various objectives.

SOURCES

This work included input from David Bender, Neal Hulkower, John Neihoff, and Robert Staehle.

APPENDIX B

AN ASTEROID PROGRAM FOR RESOURCE USAGE

AN ASTEROID PROGRAM
FOR RESOURCE USAGE

1982 August 20

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INTRODUCTION

OBJECTIVE: DESCRIBE AN ASTEROID EXPLORATION PROGRAM WHICH COULD LEAD TO EXPLOITATION OF ASTEROID RESOURCES

MAIN ITEMS:

1. PROGRAM CONCEPT AND OBJECTIVES

2. A SET OF PROJECTS (MISSIONS, ETC.) FOR THE PROGRAM

3. SPECIFIC MISSIONS, EMPHASIZING THE FIRST MISSION

THE ENERGY REQUIREMENTS FOR GETTING TO THE BEST ASTEROIDS IS ESSENTIALLY THE SAME AS GOING TO GEOSYNCHRONOUS ORBIT. THE RETURN VELOCITY CHANGES ARE TYPICALLY A FEW HUNDRED m/s. AND LESS THAN 100 m/s IN THE BEST CASES. MATERIAL CAN BE RETURNED USING COMPACT ROTARY LAUNCHERS DRIVEN BY ELECTRIC POWER.

THIS OUTLINE WAS PREPARED WITHIN THE SCOPE OF A ONE MANMONTH STUDY FOR JPL. IT COMPLEMENTS WORK DONE FOR THE PLANETARY SOCIETY, WHICH EMPHASIZED THE INVOLVEMENT OF NON-NASA ORGANIZATIONS.

PROGRAM CONCEPT

THE CONCEPT OF THE PROGRAM IS THAT OF A COHERENT SET OF PROJECTS DIRECTED TOWARD THE GOAL OF ASTEROID USAGE

THE PROGRAM MIGHT START IN THE INITIAL PHASE IN 1982 WITH PREPARATORY SCIENCE AND ENGINEERING WORK. THE FIRST MISSION MIGHT BE IN THE LATE 1980'S OR EARLY 1990'S. THE PROGRAM IS DEFINED TO LAST UNTIL THE ACHIEVEMENT OF A RESOURCE USAGE CAPABILITY.

CONTINUITY OF EFFORT, CONSISTENCY OF GOALS, AND CONSISTENCY OF MANAGEMENT IS NEEDED TO MAINTAIN AN EFFECTIVE PROGRAM.

THE OVERALL AVAILABILITY OF FUNDING IS, AND PROBABLY WILL REMAIN, VERY TIGHT. THIS IS AMPLIFIED BY THE RATHER LOW PRIORITY WHICH ASTEROID MISSIONS AND APPLICATIONS CURRENTLY HAVE. THIS WILL AFFECT BOTH THE PRELIMINARY WORK WHICH IS NEEDED AS WELL AS THE TYPES OF MISSIONS WHICH WILL BE DONE. A BENEFICIAL CHANGE IN PRIORITIES CAN ALWAYS BE HOPED FOR, AND MIGHT BE BROUGHT ABOUT THROUGH DELIBERATE EFFORT.

RESOURCE USAGE

ASTEROIDAL RESOURCES CAN BE USED IN MANUFACTURING AND CONSTRUCTION BOTH AT THE ASTEROID AND IN EARTH ORBIT. IT APPEARS UNLIKELY THAT BASIC ASTEROIDAL MATERIALS WILL BE COST EFFECTIVE ON EARTH.

TRANSPORTATION COSTS TO LOW EARTH ORBIT IN THE MID-1980'S WILL BE APPROXIMATELY 3 M$ PER TON, COMPARED TO 160 K$ FOR SILVER AND 440 $ FOR STEEL ON EARTH. THUS THE VALUE OF ASTEROIDAL MATERIALS APPEARS TO BE IN SPACE, NOT ON EARTH.

THE MOST COMMON USABLE MATERIALS FROM ASTEROIDS ARE:

SILICATES
METALS, ESPECIALLY IRON
VOLATILES, INCLUDING OXYGEN AND WATER
CARBON COMPOUNDS

PROJECTS

INDIVIDUAL PROJECTS ARE BASED UPON SPECIFIC MISSIONS, SCIENCE ACTIVITIES, OR TECHNOLOGY DEVELOPMENT. THESE ARE SHOWN HERE IN FOUR CATEGORIES.

1. DETECTION AND ORBIT DETERMINATION IS THE MOST PRIMARY ACTIVITY. THIS NEEDS TO BE ENHANCED TO GREATLY INCREASE THE NUMBER OF KNOWN EARTH-APPROACHING ASTEROIDS.

2. ANALYSIS OF ASTEROIDS AND METEORITES USING EARTH-BASED OBSERVATIONS AND LABORATORY WORK.

3. EXPLORATION OF SELECTED EARTH-APPROACHING ASTEROIDS IN ANY OF SEVERAL MISSION MODES.

4. DEVELOPMENT OF A CAPABILITY TO PROCESS AND USE ASTEROIDAL RESOURCES.

MISSIONS

PRIMARY TYPES:

RENDEZVOUS: MINIMAL COST AND COMPLEXITY, SHORT DEVELOPMENT TIME, ADEQUATE GAIN WITH 2 OR 3 INSTRUMENTS.

SAMPLE RETURN: COULD BE DONE OVER A WIDE RANGE OF COSTS DEPENDING UPON COMPLEXITY AND ACCEPTED RISK. A HIGH RISK RETURN COULD BE COUPLED WITH A BASIC RENDEVZOUS MISSION.

AUTOMATED IN SITU: A ROBOTIC TYPE MISSION, NECESSARILY COMPLEX, INFLEXIBLE CAPABILITY, AND HIGH COST. MIGHT BE MORE SUITABLE FOR A SERIES OF MISSIONS.

MANNED: PROBABLY BEST IF LIMITED TO 6 TO 8 MONTH MISSIONS, MAKING USE OF APOLLO OR OTHER ALREADY DEVELOPED HARDWARE. THIS HAS THE GREATEST FLEXIBILITY AND SCIENCE GAIN.

OBJECTIVES: SURVEY MINERALOGY AND STRUCTURE OF ASTEROID TO DETERMINE RESOURCE UTILITY, AND SCIENCE RESULTS TO AID UNDERSTANDING OF ASTEROID EVOLUTION.

TARGET CHARACTERISTICS

IN ESTIMATED ORDER OF IMPORTANCE:

1. LOW INCLINATION: IMPROVES SPAN OF OPPORTUNITIES AND REDUCES THE GENERAL ENERGY REQUIREMENTS. IMPORTANT FOR RESOURCE SHIPMENTS TO EARTH.

2. PERIHELION NEAR 1. AU: IMPORTANT FOR LOW ENERGY REQUIREMENTS AND FAST TRANSFERS.

3. PHASING: GOOD SYNCHRONIZATION IS IMPORTANT TO ACHIEVE THE BEST TRANSFERS. REPEAT PERIOD INTEGER = N*P/(P-1).

4. LOW APHELION: REDUCES ENERGY REQUIREMENTS AND PERIOD (AND THUS THE TIME FOR A 360 DEGREE MISSION).

MISSION CHOICES

CURRENT CHOICES:	1982DB	i = 1.4 deg	q = .95 AU
	ANTEROS	i = 8.7 deg	q = 1.06 AU
	1982HR	i = 2.8 deg	q = .81 AU
	EROS	i = 10.8 deg	q = 1.13 AU

THE CHOICE FOR AN EARLY MISSION SHOULD ALSO BE SELECTED TO BE A GOOD CANDIDATE FOR MATERIAL RETURN OVER SEVERAL DECADES. THIS REQUIRES ATTENTION TO PHASING. ASTEROID TYPE MIGHT NOT BE A STRONG DRIVER EXCEPT TO AVOID A PURE METALLIC TYPE IF A WIDE RANGE OF MATERIALS IS NEEDED.

1982DB MISSIONS

1982DB HAS THE LOWEST ENERGY REQUIREMENTS OF ALL KNOWN ASTEROIDS. THE PHASING FOR THE 1991 PASSAGE APPEARS EXCELLENT BASED UPON PRESENT ORBIT ESTIMATES.

INDICATED OPPORTUNITIES - NOT YET CONFIRMED

1991 JAN 40 DAY MINIMUM TIME 5.0 km/s

1991 JAN 190 DAY MINIMUM ENERGY 4.43 km/s

1990 JUL 6 MONTH ROUND TRIP 9.0 km/s
PHASES: 160 DAY OUTBOUND / 25 DAY STAY / 20 DAY RETURN
WINDOWS: 10 DAYS FOR LAUNCH AND RETURN

THESE POTENTIAL OPPORTUNITIES ARE DEPENDENT UPON THE PHASING GEOMETRY DURING THE 1991 PASSAGE. IMPROVED ORBITAL ELEMENTS FROM ADDITIONAL OBSERVATIONS AND PERTURBATION EFFECTS MIGHT BE DETRIMENTAL TO THE PHASING.

THE DETERMINATION OF THESE POTENTIAL OPPORTUNITIES BY THE AUTHOR WAS MADE USING PRIME RIB PLOTS (TRUE ANOMALIES OF EARTH AND ASTEROID) AND A PHASING ANALYSIS PROGRAM WRITTEN BY THE AUTHOR.

FUTURE WORK

INITIATE THE EARLY PROJECTS IN THE PROGRAM

STUDY OF BEST OPPORTUNITIES OF ALL CANDIDATE ASTEROIDS, ESPECIALLY A PHASING ANALYSIS

CONFIRMATION OF 1982DB OPPORTUNITIES IN 1990-91, AND EXPANSION OF ANALYSIS IF VALID

PURSUE THE POSSIBILITY OF A SAMPLE RETURN OR MANNED MISSION IN 1990?

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15. Supplementary Notes

16. Abstract This report covers the second year of a multiyear research program on the processing and use of extraterrestrial resources. The research tasks included: 1) silicate processing, 2) magma electrolysis, 3) vapor phase reduction, and 4) metals separation. Concomitant studies included: 1) energy systems, 2) transportation systems, 3) utilization analysis, and 4) resource exploration missions. Emphasis in fiscal year 1982 was placed on the magma electrolysis and vapor phase reduction processes (both analytical and experimental) for separation of oxygen and metals from lunar regolith. The early experimental work on magma electrolysis resulted in gram quantities of iron (mixed metals) and the identification of significant anode, cathode, and container problems. In the vapor phase reduction task a detailed analysis of various process concepts led to the selection of two specific processes designated as "Vapor Separation" and "Selective Ionization." Experimental work was deferred to fiscal year 1983. In the Silicate Processing task a thermophysical model of the casting process was developed and used to study the effect of variations in material properties on the cooling behavior of lunar basalt. Preliminary indications are that the power source is likely to be the most massive subsystem of an extraterrestrial materials processing facility. Based on a simple mass pay-back ration figure-of-merit, chemical propulsion systems using near-term technology and oxygen produced at the Moon can transport lunar products to low earth orbit for net gains in mass. In the Utilization Analysis Task, lunar oxygen was identified as the most important early product to be derived from space resources. Four important questions were examined by the Resource Exploration Missions Task: 1) water at the lunar poles, 2) Earth Trojan asteroids, 3) accessible "near-earth" asteroids, and 4) lunar transient phenomena as indicators of gas deposits.

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