FIFTH ANNUAL MEETING

WORKING GROUP ON EXTRATERRESTRIAL RESOURCES

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Marshall Space Flight Center
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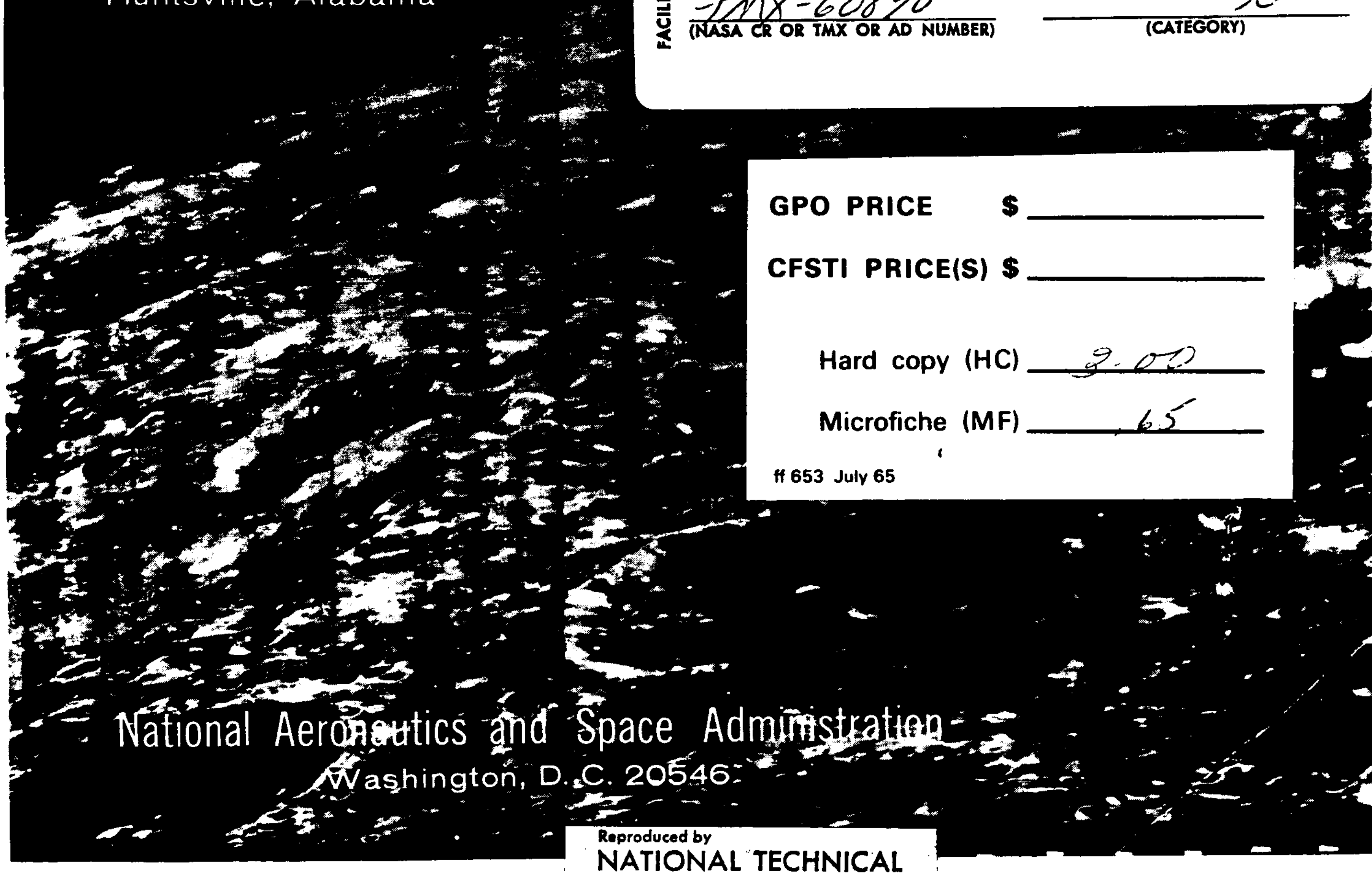

National Aeronautics and Space Administration
Washington, D.C. 20546

The cover photo was transmitted to Earth by the United States Lunar Orbiter I August 23, 1966, at 16:35 GMT.

FOREWORD

The Working Group on Extraterrestrial Resources is composed of people from the National Aeronautics and Space Administration (NASA), the U.S. Air Force, the U.S. Navy, Office of Engineers of the U.S. Army, the Jet Propulsion Laboratory, and the Rand Corporation. It was organized for the following function:

> To evaluate the feasibility and usefulness of the employment of extraterrestrial resources with the objective of reducing dependence of lunar and planetary exploration on terrestrial supplies; to advise cognizant agencies on requirements pertinent to these objectives, and to point out the implications affecting these goals.

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FUTURE TRENDS IN ORBITAL EXPLORATIONS

By

Wernher von Braun
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N68-17351

Dr. Steinhoff, Dr. Edson, ladies and gentlemen.

Thank you, Dr. Steinhoff, for those complimentary remarks, and also for the invitation to speak at this, the Fifth Annual Meeting of the Working Group on Extraterrestrial Resources.

I would like to take this opportunity to personally welcome each of you to Huntsville and to the George C. Marshall Space Flight Center. I am delighted that the agenda for your meeting allows you time to visit some of the laboratories and facilities of the Marshall Center. These laboratories are advancing their particular state-of-the-art in research, large and small manufacturing, quality control and testing techniques. The capability that developed the free world's first satellite is still hard at work on more advanced systems just as exciting as the Explorer I.

At the Marshall Space Flight Center we have been deeply committed in space flight programs. Therefore, it is with pleasure that I speak tonight to this pioneering group seeking the advancement of space flight by the utilization of resources native to the moon and our sister planets.

NASA is making every effort to get the greatest possible return from our space investment. The utilization of extraterrestrial resources can be an economical and practical way of enhancing our space program.

Through studies and experiments undertaken by groups like this, composed of personnel from different government organizations, universities and industrial firms, the advisability of processing native space materials can be determined.

We have, of course, already put to use one extraterrestrial resource. Solar energy has been used to power many Earth satellites, deep space probes and lunar vehicles.

Extraterrestrial raw materials will probably be used for construction when the moon and planets are explored by man. Also, the raw materials may be processed to provide vital life support components and rocket fuel. Our immediate concern in recent years has been focused to a large degree on the moon. However, in the future, Mars and other planets will probably receive a greater portion of our attention.

I am not going to make a speech on extraterrestrial resources, however, for I know I would find myself in the position of the man who survived the disastrous Johnstown, Pa., flood in 1889. This catastrophe made a deep impression on him, and forever afterwards he loved to tell the story of the Johnstown flood and his narrow escape. When the old fellow died and went to heaven, he began right away to tell about the Johnstown disaster to the first group of people he met. He was quite humbled when St. Peter stepped up behind him and whispered, "Before you go any further, I think you should know that Noah is in the audience." You are the experts on extraterrestrial resources, and I hope that your meeting is fruitful for the exchange of information on the continuing studies and experimentation in this field.

Tonight, I would like to discuss NASA's Apollo Applications Program.

The Saturn launch vehicles and Apollo spacecraft have been under development for about five years. The Saturn I has completed its schedule of ten flights, and three unmanned launches of the uprated Saturn I have been conducted to verify the design and performance of the systems.

The uprated Saturn I will be used in the first Apollo spacecraft flights, but the huge three-stage Saturn V will be needed for the trip that will carry man to the moon. The Saturn V with the Apollo spacecraft stands 365 feet tall. The lift-off weight is six million pounds. Of this weight, about 90 percent is fluid, producing seven and a half million pounds of thrust in the first stage alone. When the five huge engines on the first stage are ignited and the Saturn V leaves the launch pad, liquid oxygen and kerosene will be consumed at the rate of 15 tons per second. The enormous payload capability of the Saturn V is difficult to visualize. One Saturn V vehicle can place into Earth orbit in one launch more than the weight of all satellites and deep space probes launched by NASA since the agency was created in 1958. The weight of all the Telstar, Relay and Syncom communications satellites--all the Tiros and Nimbus weather satellites--all the Pegasus and Explorer meteoroid satellites--all the Ranger, Mariner and Surveyor probes of the moon, Venus and Mars--and all of the Mercury and Gemini spacecraft would fill the cargo hold of one Saturn V to little more than 50 percent of capacity.

In 1961, the United States undertook the first step toward a major manned space flight capability by selecting the goal of a manned lunar landing within the decade. The Apollo and Saturn systems have been developed to meet that goal, and the successful Gemini program has provided the necessary early experience in maneuver, rendezvous, docking, extravehicular activity and 14-day flight. The Apollo Applications Program is the second significant step toward that capability and represents an effort even more important and far reaching in its implications than the Apollo effort upon which it is based. The program will carry out a full investigation of man's role in the effective exploitation of the environment of space to meet man's needs on Earth and in long-term exploration of the universe.

Maximum economy will be achieved through the full utilization and extension of Apollo and Saturn developments. The experimental payloads in the Apollo Applications Program are designed to provide significant data on the capabilities of man and equipment, and on their potential, as promptly and as economically as possible. These plans have the flexibility of employing for new missions any spacecraft and launch vehicles currently on order for the Apollo program that are not required to meet Apollo objectives.

The program of investigation and development to be carried forward in the Apollo Applications Program will meet two basic objectives: to make unique contributions to practical applications, operational capabilities, science and technology; and, at the same time, to place the nation in a position to assess, on the basis of valid scientific experimentation and actual experience, the value and feasibility of future space flight and the interrelated roles of manned and unmanned systems.

Apollo Applications experiments will permit important and unique observations of the sun starting in 1969 and will at the same time provide experience essential for design and operation in space of future large telescopes which will open up a whole new chapter in the exploration and understanding of the universe.

Apollo Applications experiments with a wide variety of sensing devices will test the feasibility and utility of advanced types of meteorological observations and earth resource surveys from space. These experiments will provide the data for decisions on future systems -- unmanned or manned -- to derive additional practical benefits from space observations.

Within this framework, specific program elements have been selected that provide the greatest contribution to the nation's space objectives at the lowest cost. These are:

1. The spent, orbiting second stage of an uprated Saturn I will be converted into a habitable. 10 000 cubic foot orbital workshop. Provided with an airlock, the workshop will provide an economical long-duration manned shelter for many experimental activities and will be revisited and reused during the course of the program.
2. The support systems of the basic Apollo command and service modules will be modified for long-duration operations.
3. The lunar module will be modified to serve as a base for manned lunar investigations of up to two weeks.
4. The Apollo-developed lunar mapping and survey system will be used to complete the cartography of the moon.
5. The command module will be modified to carry up to six men for short-duration ferry and resupply missions and will be provided with a land landing-capability, thereby reducing costs and increasing operating flexibility.
6. Specialized payloads will be developed for operation in various orbits and on the moon, including multispectral Earth and weather sensors, biological and biomedical experiments and mobile lunar vehicles and communications systems.
7. A manned solar telescope system, forerunner of long-lived orbital astronomical facilities, will be flown during the peak of solar activity.

The Apollo telescope mount, or ATM, provides a new capability for a variety of solar and stellar scientific experiments to be performed above Earth's atmosphere, where the sun and stars can be clearly observed without being obscured by the atmosphere. For a short time, film can be returned from a space astronomy mission, and the role of the astronaut in astronomical observations can be evaluated. The ATM provides a stabilized platform that will be carried on Apollo Applications missions to accommodate instruments requiring finely controlled pointing. The ATM will be mounted in a structural rack attached to a lunar module ascent stage.

Five experiments using 13 separate instruments to obtain solar data during the period of maximum solar activity have been selected for development of the initial ATM mission. These experiments are: intensity of solar flares, ultraviolet spectrometer, X-ray telescope, solar atmosphere photography and white light coronagraphy.

Instrumentation is mounted in a telescope tube about 80 inches in diameter, 150 inches long and more than a ton in weight. It is quite a massive set of instruments. There are gimbals that permit this tube to move about five degrees to provide for fine pointing.

On this Apollo telescope mount is also a solar cell array that provides the basic power. The RCS system on the lunar module ascent stage will be used for original positioning, as will the guidance and control system that is characteristic of this stage. The stage's environmental control system will be used to provide for the comfort of the crew.

Access is being provided from the ascent stage directly into the telescope tube for the retrieval of film.

The results from five major experiments will be: a coronagraph, a coronal spectroheliograph, a chromospheric spectrograph, and spectrographic X-ray telescope and spectroheliometric ultraviolet telescope views. The spectrographic X-ray telescope is a particularly interesting device because it provides for the first time the ability to image X-rays in a spacecraft so that X-ray pictures of the sun can be obtained.

There is also a spectrometric ultraviolet telescope, the hydrogen-alpha spectroheliograph, and the high resolution X-ray telescope. These are the supporting instruments that are being developed to make these major scientific observing equipments effective, including various alignment systems, display telescopes and the ability to image inside the spacecraft the picture that the X-ray telescope is providing. There are also some proportional counters to give the background count and the direct counting of the local particle density and the hydrogen-alpha display telescope, providing us with the instruments that provide the basic information needed to make these other instruments work effectively.

In addition to the fact that a wide range of electromagnetic radiation will be observed, the observation will be at a much higher resolution than has ever been done before because, for the first time, there is a sizable amount of weight available. This is, of course, a result of the capability of the Saturn vehicles.

This is the most comprehensive array of instruments that has ever been assembled for observing the sun. Results of such a wide spectral view of phenomena that will occur during the next solar sunspot cycle should yield information of considerable value to our understanding of the basic processes of the sun. This in turn can well have marked benefits on our own understanding of how to generate and control energy here on Earth.

The second major element in the Apollo Applications Program is the orbital workshop. This workshop will permit the astronauts to work and perform experiments in the empty hydrogen tank of a spent second stage of the uprated Saturn I. A 65-inch-diameter airlock and docking adapter provides the connection between the Apollo spacecraft and the spent stage. A hatch in the airlock permits the astronauts to go into space without depressurization of the workshop or the spacecraft. In orbital flight, the command and service module docks with the airlock, and the crew activates systems to pressurize the workshop for habitation. The experiment equipment for use in the workshop is stored elsewhere and carried into the workshop for operation.

An extensive list of experiments is planned for operation within the orbital workshop. Some are directed at evaluating the habitability of the workshop for long duration flight, the work capability and mobility of astronauts in zero gravity and the effect of long duration zero gravity on man. Others are directed at engineering and technology experiments that utilize the large enclosed volume of the workshop. There are two rooms in the workshop; each is more than 21 feet in diameter and about 10 feet high. This is quite roomy, compared to the spacecraft that have been put into orbit.

Medical experiments will concentrate on the biomedical effects of long-duration flight on men. A biomedical laboratory is planned and will consist of an Apollo spacecraft module equipped with biomedical and behavioral apparatus to test and record human responses (during long-duration space flights) to various stresses such as: lack of conventional physical exercise, variable gravity and performance of complex tasks.

Bioscience and biotechnology laboratories are planned to extend earlier investigations on various life forms ranging from simple cells to primates. In these laboratories, greater stresses can be applied to specimens than are normally planned for human subjects, and the results will benefit both the bioscience community and manned space flight technology.

The combination of the ATM and the orbital workshop will provide an embryonic space station, or the first step toward a space station, with the capability of reuse and resupply.

Some of the most significant payload packages on which developments are underway are Earth resources payloads. These payload packages will provide observations of Earth in terms of determining geological, agricultural and oceanographic information that will be useful in evaluation of Earth resources.

The look inward at Earth from space can provide a means of monitoring the exploding population and dwindling resources, and assisting with a solution to this crucial problem of providing adequate food, clothing and shelter for everyone.

From space, Earth appears to be a fabulous blue spacecraft, spinning at a dizzying speed of 1000 miles an hour, and racing around the sun at almost 67 000 miles an hour. It is a planet inhabited by three billion people who are using up natural resources, polluting the air and contaminating the water. And this population is doubling today at a frightening rate. The world population doubled only once from the time of Christ until the middle of the 17th century. But it doubled again in the next 200 years, and once again in less than 100 years. At the current rate of growth, the world's population will double within 35 years. That means from six to seven billion people on Earth by the year 2000. And Yale Professor Clement L. Markert says that if the population explosion continues unchecked for 200 years, mankind will face a "beehive existence" with 150 billion people on Earth.

Properly instrumented satellites, together with sophisticated scientific experiments in space, can provide help in solving this problem. Satellites can take land surveys; they can detect calamitous situations, such as insect infestations and drought; and they can determine whether soils in specific locations are suitable for growing certain needed crops.

The clock of history is running at a rapid pace. Our generation may be the last one that has a choice about meeting these problems. We can foresee the needs, and we have the tools and technology for the solutions. Everyone in the world has something to gain by trying to solve the population-food problems, and no one can lose. I believe that we can apply the solutions we have in hand. This great new dimension of space can someday do what nothing else has ever accomplished and bring about true global cooperation among nations.

Our space future is limited only by our ability to use the knowledge we already have and to put into proper perspective the knowledge that we know is yet to come.

Thank you.

ACCELERATING EXPLOITATION OF EXTRATERRESTRIAL RESOURCES

By

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I am indeed honored to be invited as keynote speaker at this Fifth Annual Meeting of the Working Group on Extraterrestrial Resources (WGER). As I examined the program for the next two days and cast about for a suitable keynote to sound, the fact that I have not heretofore been a member of this group and have attended only one of your previous meetings made me somewhat unsure of my ground. After some further consideration, however, and at the risk of being presumptuous, I decided that this very handicap could be put to advantage if, as an outsider, I would attempt to present an inventory of where WGER has been, where it is now, where it is going, and how it might accelerate the use of extraterrestrial resources in the space program.

IS ACTIVITY OF WGER PREMATURE?

Recently, while serving on a committee to plan an important astronautical conference, I suggested that a certain senior communication scientist be asked to give a paper on the future developments which might be anticipated in the area of deep-space tracking nets. The response to the invitation was a decline with the somewhat angry overtone that since decisions had not yet been firmed in detail, it would be wasteful, too speculative and premature to discuss the alternatives. Apparently this colleague does not believe in the value of having the broadest spectrum of technical approaches discussed openly, far ahead of major decisions, for fear that the decisions will thereby become more complex and difficult to make in the process. Nevertheless, those of us who do believe that better decisions will result from open examination of problems still must agree that discussing details of some topics would be premature at this time; for example, details of the propulsion system of a star ship.

Moreover, just now when space budgets are under stress because of national monetary crises and when the Apollo astronaut tragedy has cast a temporary pall upon man's first adventure to a neighboring body, it might indeed seem premature to some that we should be considering how to extract minerals and fluids, how to construct, how to move about, and how to colonize extraterrestrial bodies.

In an attempt to see where we are in WGER, I have traced the development cycle of important technological projects in man's history in Table I. Several distinct stages are noted which are the same stages through which the extraterrestrial resource program must move.

Table I shows that once the auto arrived at the stage of the Cugnot, a three-wheeled steam carriage (1769) which went two and a quarter miles per hour and could only go 100 feet before pausing for a steam pressure rise, it made it through successive stages to the "tool" stage in about 150 years. By the "tool" stage we mean the stage at which society no longer looks at the device itself in amazement but uses it as a tool for aiding more complex projects. The airplane similarly went from Stringfellow's first self-propelled steam-driven model airplane in 1848 to the tool phase in about 70 years.

Controlled ballistic vehicles and space vehicles had science fiction roots in various cultures from the ancient Chinese to medieval Europe [1,2]. The desires and speculations of man remained just that until the formulation of principles of motion by Newton and Kepler and the development of the first models or full-scale prototypes. From these events, as shown in Table I, we see that the ballistic missile incubated in 25 years, the orbital vehicle in perhaps 20 years, the lunar vehicle in perhaps 15 years and the planetary vehicle in perhaps 30 years.

In this regard, it is worth noting that apparently the most accurate mission analysis can bring no acceleration of the project itself if it does not contain within it the germ of new explanation of principle or the design detail and feasibility of a prototype. Jules Verne's book <u>From the Earth to the Moon</u> (1865) was indeed accurate in its depiction of the Apollo mission: launching from Florida; use of an aluminum projectile or spacecraft; a crew of three; use of a tracking station (Long's Peak, Colorado); seven-day astronaut tests of the cabin before flight; contoured g-protective astronaut couches; retrorockets for course correction; and recovery at sea. However, his work, while perhaps inspirational, remained strictly in the realm of fantasy for 100 years.

TABLE I. "INCUBATION" PHASES OF IMPORTANT TECHNOLOGICAL PROJECTS IN MAN'S HISTORY

Project Phases	Automobile	Airplane	Laser	Controlled Ballistic Vehicle	Orbital Vehicle	Lunar Vehicle	Planetary Vehicle	Interstellar Vehicle
1. First Concept; Science Fiction; Physical Principle Understood	Horseless Carriage	Emperor Shun, 2000 B. C. Ornithopter of da Vinci, 1500 Caley Glider, 1810	H. G. Wells "War of the Worlds," 1898	Chinese Rockets 1230 A. D. Laws of Motion, Kepler, 1543 Newton, 1687 Congreve Rocket, 1813	Oberth, 1929 "Wege Sum Raum-schiffahrt"	Greek Lucian, 160 A. D. Europeans, 1600 Jules Verne, 1865 "From the Earth to the Moon"	Father Kircher, 1656 "The Ecstatic Voyage" Oberth 1922, "Rakete zu den Planetenraumen"	Bernard de Fontennelle, "Essays on the Plurality of Worlds," 1689
2. Lab Prototype; Missions Analysis; Trade-Off Studies; Design Definition Phase	Cugnot Steam Carriage, 1769 Early Prizes & Races 1895 - 1905	Stringfellow Model, 1848 Balloon Flights Langley "Aerodrome," 1891	Description by Schallow and Towne, 1957 Prototype by Maimen, 1960	R. H. Goddard, 1926 Peenemunde & U.S. Missile Tests, 1935 - 1955	Sputnik I, 1957 Mercury Studies, 1960	Ranger, 1964 Luna I	Mariner II, 1964 Mariner IV, 1966 von Braun "The Mars Trip"	?
3. First Operational Phase (A Novelty)	R. Olds, 1901	Wright, 1903	1966, Input-Output polarization modulator	V-2, 1942 Atlas, 1957	Vostak I, 1961 Gemini, 1964	Apollo, 1970	1982? 1986?	?
4. Building Block Transportation Phase; (A Tool)	H. Ford's Mass Production, 1910	World War 1, Post-War U.S. Mail	High data rate Communication, 1980 - 1985?	Titan III, 1965 Saturn V, 1968	Large Space Station 1975?	Post-Apollo Lunar Base 1975 - 1985?	Mars Base 1995?	?
Incubation Time	$\sim$150 Years	$\sim$70 Years	$\sim$25 Years	$\sim$25 Years	$\sim$20 Years	$\sim$15 Years	$\sim$30 Years	?

Using Table I for perspective, several conclusions may thus be drawn (with a little license). These are: (1) most projects go through the major development phases 1 to 4 in more or less the same order; (2) the two critical phases, the prototype demonstration and the building-block transportation tool phase take from 15 to 150 years, with a speed-up being observed as time goes on, (3) the duration of the "visionary" or "science fiction" period is several orders of magnitude longer than the "incubation" period from phase 2 to phase 4, and (4) the way to accelerate the project is to apply all efforts to arrive as soon as possible at the stage of the understanding of principles and the successful laboratory prototype.

Specifically, because of lead times of equipment, the necessity for getting extraterrestrial data, and the shortening of incubation periods, it is not too early to focus research and development effort now on the use of extraterrestrial resources. However, the best way to accelerate the project out of the "visionary" stage is to focus on explanation and dissemination of principles, on proof of feasibility, on evolution of prototype devices, and on association of pay-off with other programs of national interest to make the project look like a tool to even the most conservative.

CLASSIFICATION OF PAST EFFORTS OF WGER

To assess what has been accomplished to date by WGER, Table II displays the author's classification of all papers which were published in the proceedings of WGER's second, third and fourth annual meetings. Inclusion of the papers of the fifth (or present annual meeting) used classification as judged by title only [3, 4, 5]. Classes A through E correspond to the beginning of the incubation period or the basics of how things get started. Classes F through I correspond to the analysis, trade-off and enunciation of benefits accruing from exploiting extraterrestrial resources. It is apparent at once that the how-to-do-it classification, B and E, and the nearer-in benefits of F, G and H (the major accelerators of extraterrestrial resource utilization) have been relatively neglected. In the next section, specific suggestions are made for amplifying WGER work in these directions.

TABLE II. DISTRIBUTION OF PAPERS AT WGER MEETINGS BY NATURE OF CONTENT

WGER Meeting \ Classification of Paper		A Basic Physical Principle Permitting ETR* Exploitation	B Detection, Search, Information Theory	C Data on Extraterrestrial Bodies or Environment	D Data, Preliminary Design, Description of ETR Equipment	E Model or Lab Prototype of ETR Equipment	F ETR Applied to First Generation Space Missions	G ETR Exploitation for Earth Benefits	H ETR Exploitation for Scientific Goals	I ETR Applied to Second Generation Space Missions ("tool" phase)
			Incubation				Application			
Second Annual Meeting Alamagordo, New Mexico		2	1	3	1	0	1	0	0	4
Third Annual Meeting Cocoa Beach, Florida		3	1	1	8	0	1	0	0	6
Fourth Annual Meeting Colorado Springs, Colorado		2	0	4	7	1	0	0	1	5
Fifth Annual Meeting Huntsville, Alabama (Judged by title only)		2	0	3	3	0	2	1	1	8
Total Effort	72	9	2	11	19	1	4	1	2	23
	%	12.5	2.8	15.3	26.4	1.4	5.5	1.4	2.8	31.9

SUGGESTIONS FOR IMPLEMENTING WIDER USE OF EXTRATERRESTRIAL RESOURCES IN SPACE PROGRAMS

Perhaps the greatest disparity between use of some of the processes of extraction advocated by WGER and the inclusion of these processes in actual space mission planning lies in class B, the area of detection and search. Table III is a crude indication of the available activity time associated with several classical space missions.

It is evident from Table III that even if the expeditions were equipped with the necessary portable equipment to extract water, propellant, biological materials, or some mineral of value, the appropriate deposits would have to be located within days or weeks if they are tobe of logistic value to the expedition. A paper by E. deGraeve (at the third WGER meeting) was on constant volume "hard" pressure suits [6]. and the experiences of the Gemini astronauts have indicated that unless the "hard" space suit is attained in operational form (and even then) the mobility and work load limitations on astronauts are likely to prevent extensive astronaut research for resource deposits.

The matter is made worse when one looks at terrestrial experience. Consider for a moment the equipment necessary to locate an oil dome--trucks, seismographs, computers back at the laboratory for extensive data reduction, explosives, drilling rigs, etc.

TABLE III. ACTIVITY TIME ASSOCIATED WITH EARLIEST MANNED SPACE MISSIONS

Space Mission	Activity Time on Surface or on Location
Apollo Lunar Trip	2 Days
First Saturn-AAP Orbital Missions	30 Days
First Lunar Surface Base	2 Weeks to 2 Months
First Mars Landing (420-day mission)	2 Weeks to 1 Month
Classical Mars Synodic Base	400 Days

The search for the ill-fated Thresher submarine is another example in which, even though the approximate location was known, the environmental difficulties resulted in a long search time. In short, with ample time, relatively easy logistics, and the ability to repeat trials and trips, Earth-man takes a long time to locate specific minerals and often has a high false-detection rate.

A real effort is needed to devise combinations of lunar and planetary orbital search satellites and rockets, surface rovers and portable hand-held sensors which, together with a totally mathematical information theory and search theory approach, will rapidly and unfailingly lead the expedition to the deposits it seeks to exploit. Some examples will now be given, although the reader is to be cautioned that this paper does not contain the results of suitable feasibility studies and hence the examples are to provoke thought only.

Imagine a crew landing on the moon in the late 1970's or 1980's with the intention and capability to exploit lunar resources, if found. One can imagine the crew launching small reconnaissance orbiters, which would orbit at low altitude (10 to 20 miles) or high altitude search rockets. These would acquire multi-spectral surveys of the surrounding regions. Sensors would include visible light photography, ultraviolet, infrared, possibly electron beam scanning which would generate x-ray emission lines according to the atomic nature of the material found below.

Surface rovers and men using miniaturized detection devices would then prove out the hopeful areas. The surveillance techniques would include soil sampling, coring, neutron logging (by a radioisotope device generating a neutron beam which interacts differently with materials having varied percentages of water and hydrocarbons), direct x-ray line analyzers and holographic recorders. The latter instrument at some future time may not require a coherent beam for taking pictures but may be able to operate in sunlight with a laser being used only for subsequent projection. An early prototype holographic device for astronaut use has now been proposed by Fair and Pernicka of the Martin Marietta Corporation at Denver [7]. The instrument would use a pulsed argon ion laser to illuminate rock samples up to three and a half inches in diameter and would have a resolution of 50 microns. It would weigh 36 pounds, consume 25 watts of average power, have dimensions of 12 × 12 × 30 inches, and would process one sample front and back every three to ten minutes. Clearly this instrument is a "model-T," but it is an important step forward.

Even with the intention of using those data together with the survey data from Ranger, Surveyor, Lunar Orbiter and the early manned Apollo missions, a great flexibility and improved success probability can be provided to the expedition by incorporating mathematical strategies of search, advanced correlation methods and information theory. In the field of bomber-fighter combat, visual detection has long been studied and head-movements and eye-movements have been optimized into the most effective search patterns. The classical work on submarine detection is also indicative of a body of theory waiting to be applied to detection of resources from orbiters or rovers.

In the information theory and in detection of coded messages Shannon [8] has developed an "equivocation function" (Table IV). A message is uniquely decoded

TABLE IV. CONCEPT OF "EQUIVOCATION FUNCTION" DEVELOPED BY C. SHANNON

In Substitution Cipher of English text of Length N,

N = 50, nearly always unique solution to cipher

N = 15, quite a large number of "messages" would fit

N = 8, approx 0.125 of all English sentences are possible

N = 1, any letter possible

Equivocation Functions

$$H_E(K) = \sum_{E,K} P(E,K) \log P_E(K) ,$$

q

E = cryptogram, K = key

$$H_E(M) = \sum_{E,M} P(E,M) \log P_E(M) ,$$

E = cryptogram, M = message

Redundancy

$$D_N = \log [\text{no. messages of length N}] - \underset{\text{uncertainty}}{\underset{\uparrow}{H(M)}}$$

Uniqueness (approx) at

$$N = \frac{H(K)}{D}$$

or separated from other messages with which it might be confused by "listening" to the sender for a long enough time. By "listening" long enough (message length N) the redundancy of the language is overcome and the true message is almost uniquely determined. It should be possible to apply similar concepts to the search for a given resource. In this application the length of message, N, corresponds now to the total amount of data bits and the redundancy corresponds to the multiplicity of stimuli which can give a certain sensor response.

Use of correlation techniques is another approach. Table V shows a correlation chart of the type used in the total economic analysis of a country considering all economic indicators. One could envision something like this connecting the response of many sensors in each of several modes and deducing en bloc the nature of the geological substrata and the proper locale to mine for successful exploitation.

The use of sophisticated techniques of search implies high data processing requirements. Computers of the 1980's, by use of microelectronics and low power circuit techniques, might make the capacity of, say, IBM 7094 computers available in lightweight, small volume, low-power, integrated spacecraft computers.

Sutro of M.I.T. [9] comments on developments for data processing for exobiology mission:

"It is clear that technology supplies us increasingly reliable parts even smaller and able to work on little power so that computers become more and more compact. At the same time the art of using these devices expands greatly so that they have ceased to be the novelties for study and have become, instead, common tools. A number of workers concerned with this art (including us), are attempting to make a true robot, capable of

TABLE V. MATRIX MODEL OF THE BRITISH ECONOMY
(Scientific American, September 1964)

				Production Accounts											
				Commodities				Industries				Consumers' Goods And Services			Government Purposes
				Fuel and Power	Metals, Engineering, Construction	Agriculture, Manufacturing	Services	Fuel and Power	Metals, Engineering, Construction	Agriculture, Manufacturing	Services	Food, Drink, Tobacco	Clothing and Household	Other	Defense
				1	2	3	4	5	6	7	8	9	10	11	12
Production Accounts	Commodities	Fuel and Power	1	0	0	0	0	590	353	274	476	0	647	94	74
		Metals, Engineering, Construction	2	0	0	0	0	235	6077	750	756	0	373	88	637
		Agriculture, Manufacturing	3	0	0	0	0	72	693	4032	882	4046	1363	381	28
		Services	4	0	0	0	0	281	1724	1767	367	1478	878	2569	23
	Industries	Fuel and Power	5	2686	4	12	0	0	0	0	0	0	0	0	0
		Metals, Engineering, Construction	6	5	15 349	74	17	0	0	0	0	0	0	0	0
		Agriculture, Manufacturing	7	12	43	11 769	12	0	0	0	0	0	0	0	0
		Services	8	0	25	4	10 640	0	0	0	0	0	0	0	0
	Consumers' Goods and Services	Food, Drink, Tobacco	9	0	0	0	0	0	0	0	0	0	0	0	0
		Clothing and Household	10	0	0	0	0	0	0	0	0	0	0	0	0
		Other	11	0	0	0	0	0	0	0	0	0	0	0	0
	Government Purposes	Defense	12	0	0	0	0	0	0	0	0	0	0	0	0
		Health, Education, Child Care	13	0	0	0	0	0	0	0	0	0	0	0	0
		Other	14	0	0	0	0	0	0	0	0	0	0	0	0
Income and Outlay Accounts	Indirect Taxes And Subsidies	Indirect Taxes	15	1	49	85	0	53	76	126	604	1221	471	372	10
		Less Subsidies	16	0	0	0	0	-3	0	-257	-111	0	-118	0	0
	Institutional Sectors	Distribution of Property Income	17	0	0	0	0	101	1401	1369	1899	0	524	0	0
		Private Sector	18	0	0	0	0	792	4341	2408	5028	0	86	118	718
		Public Sector	19	0	0	0	0	0	0	0	0	0	0	0	0
Accounts	Commodities	Fuel and Power	20	0	0	0	0	0	0	0	0	0	0	0	0
		Metals, Engineering, Construction	21	0	0	0	0	0	0	0	0	0	0	0	0
		Agriculture, Manufacturing	22	0	0	0	0	0	0	0	0	0	0	0	0
	Industries, Replacements	Fuel and Power	23	0	0	0	0	185	0	0	0	0	0	0	0
		Metals, Engineering, Construction	24	0	0	0	0	0	184	0	0	0	0	0	0
		Agriculture, Manufacturing	25	0	0	0	0	0	0	312	0	0	0	0	0
		Services	26	0	0	0	0	0	0	0	416	0	0	0	0

self-programming within certain limits, one that is capable of handling a large sequence of choices, correcting itself when it makes an error of judgment. This is not to say that we are so rash as to claim that we can build an animal or an animal intelligence of a sort, and one that would permit communication with us ...."

His paper describes :

" three development programs underway at the Instrumentation Laboratory. One is the design of the layered processing of a visual system. The second is the design of a stereoscopic system. The third is a decision and control system. How would a robot reduce data? We propose that the robot carry with it measures of the usefulness to man of the data it acquires, so that it will send back only data which satisfies those measures. For each of the development programs to be described, such measures will be suggested. But a more useful set of measures will have to come from you. "

Thus all samples, all Orbiter data, all Rover data, and the totality of manned logged data might be brought aboard the spacecraft and processed there. Or, alternatively, the data may be relayed to Earth for processing and the subsequent selection of a site for exploitation. If it is necessary to relay the data to Earth for processing during Martian or deep space missions, it may be necessary to increase data rates by employing RTG powered relay space stations between Earth and the mission sites.

This brings us to category (E), making models or prototypes of suggested instruments. Some of the detection instruments mentioned above can and should be made soon in prototype form here on Earth. Present concepts of a lunar drill, for example, allow for a drill which will core-down 25 feet to 100 feet. Now to get at lunar gas pockets, if they exist, might require that depths of 500 feet or more be attained. Present drills for doing this here on Earth weigh in the thousands of pounds. It would be of great interest to devise a very light-weight, highly reliable, space-qualified drill.

A number of papers of the past have indicated preliminary designs of rock-crushing and water extracting equipment. But, now, why not organize a project for producing a flight-weight prototype? Perhaps reliable crushing and coring designs will prove too difficult for the engineer and he will resort to other means of breaking rock into uniformly small bits, such as the use of laser drills or capacitor discharge drills. It is well known that the rapid discharge of energy in these forms into materials will stress the materials beyond their elastic limits. The energy itself should come from solar converters, conversion of surface wind energy (on Mars), or nuclear reactors.

A number of prototype devices have been built in the biological experimentation area, and these should be useful to WGER applications. Examples of such are the implantation devices already tested.

Bumann of M.I.T. [10] described the construction of small mass spectrometers of a resolving power capable of distinguishing integral masses, and notes that small automated gas chromatographs have already been developed in connection with the Surveyor program.

Soffen and Sloan of JPL [11] describe two laboratory bread-boards of remotely operated microscope systems, one of which has the capability of sampling in a dust cloud created by an aerosol can, performing chemical reactions and biological staining, employing a 'flying-spot' scanning device, employing a built-in selector-discriminator, and telemetering the results of biological search to a distant station. Employment of a number of these, perhaps deployed by small rockets, could give a complete picture of the neighborhood. Hostetler [12] also describes the design of a complete, high data rate, automated, biological laboratory powered by RTG units and chemical batteries.

In the present Voyager program, a great emphasis is placed upon designing all hardware to meet NASA sterilization policy that the probability of transporting a live microorganism to the Martian surface is no more than 10^{-3}. Other sources have implied that decontamination of astronauts who visit other bodies will be required upon Earth return. It is possible, then, that all extraterrestrial resource operations might have to conform to sterilization and decontamination procedures. This too will require devices and procedures for which prototypes should be developed and tested now on Earth. Figures 1, 2, 3 show a Martin Marietta concept for a small portable device for sealing lunar or planetary samples in a teflon bag in a sterile manner [13]. A prototype of this device is soon to be built, but ultimately it should be made much smaller and more adaptable.

Another area in which progress could be made here on Earth is in devising advanced fabrication methods involving metal forming by high energy methods and various explosive and diffusion bonding procedures so that we might consciously cannibalize the cargo ships or earlier debris found in the landing area and left behind by earlier expeditions.

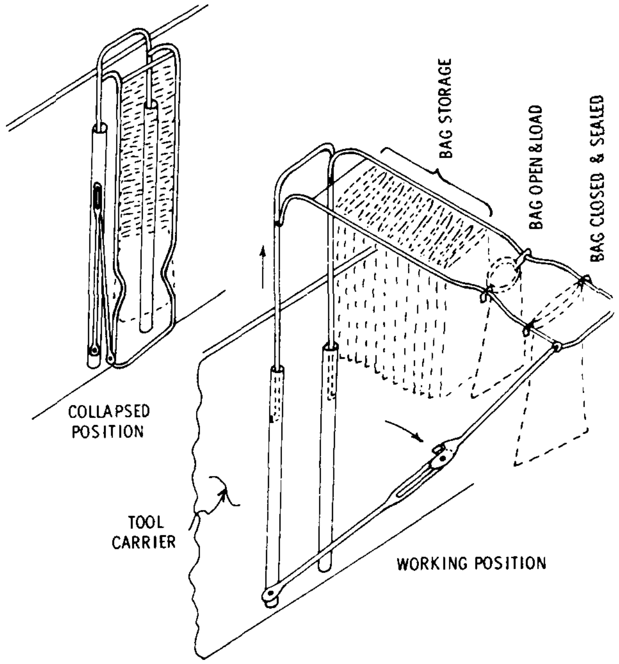

FIGURE 1. BAG DISPENSER

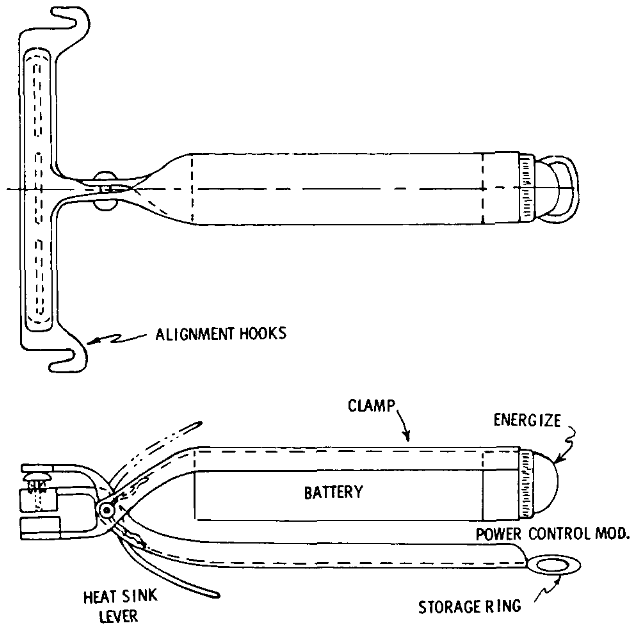

FIGURE 3. BAG HEAT SEALER

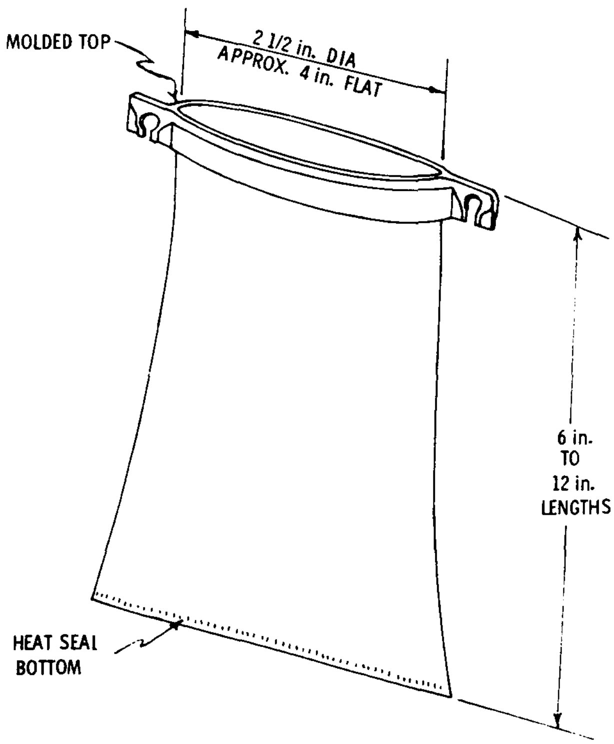

FIGURE 2. SPECIMEN BAG (Material:FEP)

In any case, the production of partial or complete prototypes of the devices described in WGER papers would greatly enhance the planned use of extraterrestrial resources in next generation missions and would serve as a touchstone of reality for further missions analyses and trade-offs concerning potential gains by extraterrestrial resource use. And this leads to the single biggest benefit to be derived from aiming at the target of producing prototypes. If reliable portable prototype devices are built, and if advanced search techniques are developed to make possible the exploitation of other bodies, these same devices would be of great commercial value here on Earth. No doubt significant financial backing could be obtained from industry as well as government and the pressure for progress in extraterrestrial resource use would be irresistible.

"The proof of the pudding is in the eating," and the search techniques plus the prototypes outlined above should be put to the test in simulated trials here on Earth. For example, a helicopter payload of suited astronauts with detection and search devices might be deposited, say, in the Arabian peninsula with the goal of locating and extracting selected products from the area. No doubt the many problems in hardware and soft ware would be enlightening with respect to the corresponding space mission. In such simulations the time deadlines for detection and extraction envisioned in Table III would be imposed to determine the suitability of procedures and equipment. Even so simple a chore as sinking a shaft and exploding a charge to make an underground container for gas or propellants, a procedure suggested by some to avoid carrying storage tankage to the moon, might be simulated against time here on Earth using the prototype equipment. Simulation might also inject the psychological and innovative aspects of the individual into prototype redesign--realistic elements which might otherwise escape the designer. Imagine the fatigue and confusion factors of handling a drill or detector while in a space suit, with dust accretion on the tools, with harsh black and white shadow effects (on moon), or with dust storms (on Mars), and in an unfamiliar gravitational field. Certainly, problems will arise that have not been foreseen by the designer, and corresponding unforeseen innovations by the astronaut will occur. In short, the WGER might more fully espouse simulations of fundamental work situations necessary to extraterrestrial resource processing.

We move now to the categories of (F) through (I) of Table II that apply to the advantages derived from using extraterrestrial resources. Because of the splendid missions analysis and trade-off work in the past and present WGER meetings, I think the case has been made for resource employment for the lunar and planetary missions of the second generation; that is, for those missions in which man goes frequently to these bodies and stays for longer than days or weeks. However, unless man finds more than the mere curiosity of exploration, he may not go often enough to warrant the investment in the necessary search and extraction equipment for many years to come. The solution, it seems to me, is to concentrate extraterrestrial resource application studies on what can be contributed to first generation space missions, to science itself, and, perhaps most important, to the benefit of man here on Earth. I refer to categories (F), (G) and (H) in Table II.

On May 1 through 3, 1967, the American Astronautical Society will hold its annual meeting in Dallas, Texas, to discuss prospects for the commercial utilization of space [14]. Some of the ideas that the AAS has sponsored in this regard definitely use the extraterrestrial resources of vacuum, weightlessness, temperature gradients, and planetary fields, and hence ought to receive the attention of WGER. In addition, the fruition of any of these ideas would add great momentum to the space program as well as to the activities of WGER. Included in these possibilities are (Table VI):

TABLE VI. SOME HIGH PAY-OFF AREAS ON WHICH TO FOCUS EXTRATERRESTRIAL RESOURCE UTILIZATION CONCEPTS

- Large Manned Orbital Communication Centers
- Multi-Spectral Resource Detection and Earth Survey Orbital Systems
- Use of Super-Clean Orbital Environment for Production of Microelectronics and Pharmaceuticals
- Tourism and Medical Research Orbital Systems
- Application to New Scientific Missions

(1) The extension of present satellite communications systems to satellite telephone communications; to large manned orbital communications system satellites with on-station maintenance and repair facilities; to video television, including copy-producing systems; to transmission of mail by automated satellite link; to transmission of library, medical diagnostic, and financial data via satellite link. WGER could contribute toward these valuable data link needs and hence prepare for the data processing problems of extraterrestrial resource extraction.

(2) Application (and test) of extraterrestrial resource search and detection systems to the problems of Earth resource and mineral deposit location. By multi-spectral survey of forests, agricultural areas, geological formations, oceans and atmospheric movements, large amounts of data will be generated which, upon computer processing, will yield commercially valuable results. Such success will speed investment in more prototype equipment of the type envisioned for WGER missions.

(3) There are certain small-sized articles in which the future transportation cost to orbit will be low, relative to the value of the product and for whose manufacture the space environment is helpful. Many antibiotics and drugs must be manufactured under precisely controlled, super-clean environmental conditions, readily available in space. Environmental chambers on Earth which only begin to approach the quality of the space environment are very expensive.

Another such product is microelectronics. Fink [15] notes that electronic packaging density has gone from 10^4 parts per cubic foot to 10^9 parts per cubic foot in the past 15 years. He predicts that in 30 years we shall "grow" complete electronic nerve-like packages having densities of 10^{15} equivalent parts per cubic foot. Manufacturing processes using ion deposition techniques will permit the development of such new classes of devices having their own integrated circuits and logic functions as an integral part of the structure.

These processes, devices, manipulators, insertion procedures, fluid handling techniques, etc., which will be necessary for commercial space applications will also be useful in extraterrestrial resource extraction. WGER could contribute toward these nearer-term space objectives and at the same time prepare for later mission opportunities.

(4) Many writers have indicated the possibilities of orbital tourism and orbital medical facilities. The awesome panoramas, the use of large telescopes, and the curiosities of orbital flight attest to the desire for the former. The interest in cardiovascular problems, physiological phenomena in weightlessness, and study of body fluid flows under new stresses, give unprecedented opportunities for research in medical areas. In both of these possibilities, the first experiments will very likely be with scientist-astronauts. Again, some of the same equipments and bio-status monitoring devices which are pertinent for WGER applications can be developed for these medical and orbital tourism applications.

(5) The scientific firsts possible because of the space, lunar or planetary environment have already been alluded to in earlier WGER meetings. 'At this meeting Greiner is suggesting the use of a lunar crater as the foundation for a large radio telescope. Others have talked of placing large telescopes in orbit, or on the moon, or possibly using the moon as a future sight for ultra high energy accelerators.

Emphasis might also be placed on an exploration of Venus for resources. It might be advantageous to develop balloon-floated exploration devices which can work at high temperature and low optical visibility. In many ways Venus is the greatest challenge to exploration in the solar system. It may ultimately prove to be the major extraterrestrial resources body. It is certainly not too early to start planning a program for exploiting its resources, assuming that its temperature and cloud cover can be dealt with. Much thought on the nature and scope of devices needed for Venus will have to be developed over the next decade. WGER might profitably emphasize investigation of Venus resources in one of its subgroups.

By concentrating research and imagination on the use of extraterrestrial materials and preparing actual designs of devices which will present a unique scientific adventure, WGER can again encourage the use of such resources in space mission planning.

It is evident from Table II that the existence of the Logistic Subgroup, the Environment and Resources Subgroup, the Mining and Processing Subgroup, and the Biotechnology Subgroup has accounted for the better WGER coverage in columns (A), (C), (D), and (I). Perhaps one might suggest that creation of two new Subgroups would help achieve the acceleration opportunities described in this paper. These might be: a Subgroup on Prototypes for Resource Detecting and Extracting; and a Subgroup on Near-Term Space Flight, Scientific, and Commercial Extraterrestrial Resource Application (with special attention to Venus resources.)

The excellent work of WGER also needs more forceful and widespread dissemination. In this regard consideration might be given to the offering of selected WGER papers for publication in journals of related astronautical and scientific journals. Also, joint meetings and occasional affiliations with selected astronautical societies might be encouraged.

CONCLUSIONS AND RECOMMENDATIONS

(1) It is not premature to consider the use of extraterrestrial resources to aid space missions. In fact, steps should be taken to accelerate the progress toward this goal.

(2) The objectives of WGER will be best served by increasing the emphasis on techniques of search, of detection and in situ real-time analysis of resources.

(3) WGER should urge that more prototypes of detection, analysis, retrieval, sterilization, manipulation and bio-astronautical equipments be built and realistically tested.

(4) Most applications to date have hinged on projected heavy space traffic to the moon and the planets. WGER should focus more effort on near-in goals such as selected exploitations which use extraterrestrial resources and related equipments.

(5) WGER should consider establishing new subgroups to emphasize prototype building and near-term applications.

(6) WGER should increase dissemination of its findings and proceedings of its meetings. This can be done by a policy of selective offering of papers to the journals of professional societies and arranging joint meetings with selected astronautical organizations.

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UTILIZATION OF CRATER REFLECTORS FOR LUNAR RADIO ASTRONOMY

By

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INTRODUCTION

The operation of radio telescopes from the lunar surface represents one of the most attractive future missions for the space program. Lunar-based radio astronomy offers unique opportunities for sensitive, high resolution observations at interference levels and frequencies below terrestrial limits. To fully exploit these possibilities using conventional techniques would require large, complex systems feasible only for advanced missions. The hazards and hostility of the lunar environment coupled with the staggering costs of lunar logistics will reduce early radio astronomy missions to relatively modest proportions involving simple equipment and minimal attention. The goals of these preliminary experiments will be limited to determination of the feasibility of lunar radio astronomy and measurement of the lunar environment in sufficient detail to permit definition of more advanced studies.

The possibility of utilizing lunar surface formations to assist early radio astronomy experiments is particularly interesting in that natural features might offer advantages as radio telescope reflectors. Typical dimensions are suitable for sensitive, high-resolution pencil-beam operation at the low frequencies of interest for lunar radio astronomy. The identification of suitably shaped features would permit their utilization in early missions. The role of an astronaut in an initial lunar radio astronomy experiment might be reduced to the mere installation of a receiver at the focus of an appropriate predetermined crater. Such a simple operation could permit immediate exploitation of the uniquely lunar opportunities for low frequency observations at high resolution and reduced interference levels, enabling sophisticated astronomical investigations at a much earlier date in the lunar exploration time frame than would be possible if development of large lunar antenna structures were awaited.

The use of lunar geographic features to exploit the operational advantages of lunar radio astronomy represents an efficient, thorough utilization of the opportunities attendant lunar surface operations for the achievement of post-Apollo scientific goals. The geographical features involved constitute a class of operational resources distinct from the material reserves usually associated with extraterrestrial resources. Complete exploitation of extraterrestrial mission opportunities requires the identification and effective utilization of operational as well as material resources.

These considerations have led to a preliminary evaluation of the feasibility of utilizing natural lunar features as low-frequency parabolic reflectors. Several possible configurations are shown schematically in Figure 1. The approximate location of the focus is indicated for each concept, and the orientation of the beam axis (direction of primary lobe in reception pattern associated with parabolic reflecting surface) is shown to illustrate the angle of observation with respect to the lunar surface for each reflector. In general, the beam direction is parallel to the symmetry of the associated parabolic surface.

The simplest natural reflector would be a symmetric crater in the form of a classical parabolic dish (Fig. 1 - (a)). Observations using such a configuration would be along directions normal to the lunar sphere. An asymmetric crater (Fig. 1 - (b)) would serve if it could be shown to approximate a section from a paraboloid whose axis was oblique to the lunar datum sphere. The broken line in Figure 1 - (b) illustrates the orientation of the complete hypothetical parabolic surface. A flat-bottomed crater whose walls form an annular section of a paraboloid is illustrated in Figure 1 - (c). Such a configuration would exhibit resolution comparable to that of symmetric parabolic craters of similar diameters; some loss in sensitivity would be incurred, however, because of the nonfocusing nature of the flat crater floor.

These three crater reflector concepts can be extended to lunar rilles or grabens by substitution of a symmetry axis perpendicular to drawing in place of the aximuthal symmetry of the craters. Rill reflectors could thus be interpreted as sections of parabolic cylinders. A related concept is illustrated in Figure 1 - (d) by a parabolic embankment at the base of a wall or cliff. The beam axis of such a reflector would be oblique to the lunar surface. Note that each of the last three reflector configurations is characterized by a focal point located at or near the ground.

This paper emphasizes the analysis and evaluation of symmetric parabolic craters, although the specific advantages of several alternate configurations

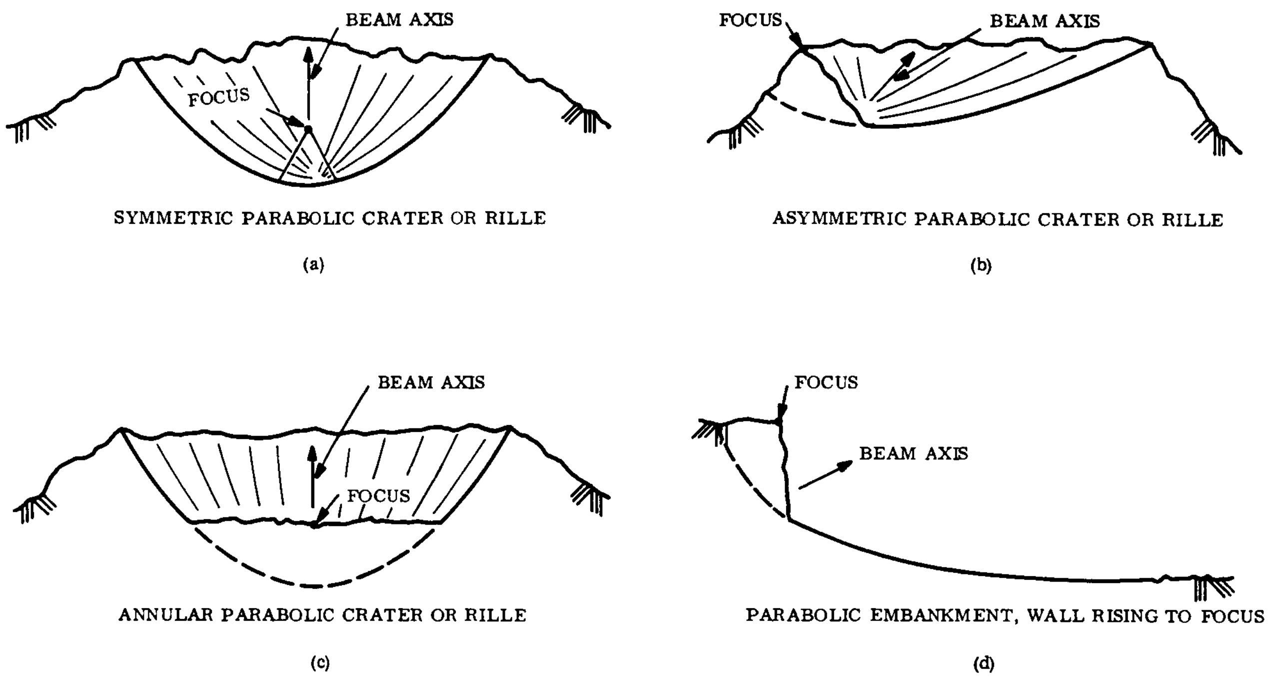

FIGURE 1. NATURAL LUNAR REFLECTOR CONFIGURATIONS

are indicated. The theoretical and practical constraints involved in employing symmetrical craters as low frequency parabolic reflectors will be evaluated first to permit formulation of procedures for examination of lunar surface features to identify suitable craters. The results of this investigation will then be utilized in conjunction with the previously obtained theoretical requirements to evaluate the general feasibility. advantages and problems of crater reflector operation. The results of this analysis permit conclusions to be drawn regarding the usefulness of lunar craters as parabolic reflectors and suggest several techniques for the utilization of lunar craters in radio astronomy.

LUNAR RADIO ASTRONOMY GOALS AND RELATED CONSTRAINTS

The goals proposed for lunar radio astronomy exert a significant influence on the definition of general guidelines for evaluation of crater reflectors. The most desirable operating frequencies, resolutions, and locations for crater reflectors can be identified from such considerations.

Frequencies of Interest

An intelligent assessment of the usefulness and feasibility of lunar crater reflectors requires a knowledge of the frequencies at which the craters would be operated. The latter information is a function of the sources to be studied and the frequency limitations of terrestrial radio astronomy.

A general goal for lunar radio astronomy is the extension of observations to frequencies presently inaccessible from the earth. The limitations imposed upon terrestrial radio astronomy observations by the earth's atmosphere and ionosphere define a window in the electromagnetic spectrum extending from about 5 to 10 000 mc (3cm). The ionosphere renders frequencies below 3 to 8 mc inaccessible from Earth. The upper limits of the radio window are determined by absorption from atmospheric gases. Except for a few narrow transmission windows, the atmosphere is more or less opaque to radio observations at wavelengths from about 3 cm to 10 μ.

From these limitations, the frequencies of interest for lunar radio astronomy studies may be defined. The suggested low-frequency regions have been summarized in Table I. High-frequency bands which are attractive for lunar studies have been omitted because they involve such short wavelengths that the precise reflector surfaces required render these frequencies impractical for use with lunar craters. The suggested frequencies imply that lunar radio astronomy will be called upon to explore the low-frequency region inaccessible from earth all the way down to the cutoff frequency of the interplanetary medium.

TABLE I. SUGGESTED LOW-FREQUENCY BANDS FOR LUNAR OPERATIONS

Source	Suggested Frequency Range (kc - mc)
Marshall Space Flight Center, LESA RFQ, 1965 [1]	1 - 50
LESA Report, North American Aviation, 1965 [2]	30 - 30
Wood's Hole Meeting, National Academy of Sciences, NAS, 1965 [3]	300 - 10
Falmouth Conference on Lunar Exploration and Science, NASA, 1965, [4]	50 - 10

Note that proposed radio astronomy satellite investigations planned for high earth orbit will examine the 1 - 10 mc region in detail well before the lunar radio astronomy time frame [5].

Several proposed lunar radio astronomy studies should be mentioned with regard to their frequency requirements. It has been recommended that observations of solar and planetary radiation, sky brightness and the interplanetary medium be extended to the low frequency limits. Among planetary sources, the Earth and Jupiter offer especially promising possibilities for studies because their intense radiation obviates the need for high sensitivity and fine resolution. In particular, the 725 kc noise bursts from the terrestrial ionosphere detected by Soviet probes [3] raise important questions regarding the interference levels to which lunar radio telescopes in sight of the earth might be subject. Solar bursts will certainly merit increased study at low frequencies from lunar observatories. Galactic noise investigations from 30 kc to 30 mc will be important to evaluation of the feasibility of sensitive low frequency radio astronomy operations. Such studies are especially attractive in view of the relative simplicity of measurements of the galactic noise spectrum. High resolution investigations of the low-frequency spectra of discrete galactic and extragalactic sources, though desirable, are sufficiently difficult to require postponement to the later phases of lunar radio astronomy.

In view of the goals and requirements described above, the frequency range of 100 kc to 10 mc has been chosen as the useful spectrum for low frequency lunar radio astronomy measurements. Crater reflectors must be operable at or below these frequencies to be considered useful for lunar programs.

Resolutions

The general philosophy governing design of lunar radio astronomy facilities maintains that radio telescopes should be diffraction-limited at the frequencies of interest. This constraint defines the irregularities which can be accepted on a reflector surface. The dimensional accuracy maintained prescribes the upper frequency limit of a given reflector, the frequency beyond which resolution is rapidly degraded. To achieve the diffraction-limited operation desired for lunar radio astronomy facilities, tolerances must be maintained to within $\lambda/16$, where λ is the wavelength of observation. At the low frequencies of interest (0.1 - 10 mc), the tolerable deviation from parabolic shape ranges from 19 to 190 meters.

The minimum frequency of operation for a given reflector can be determined from the largest beamwidth which is acceptable. The resolutions necessary for several of the proposed studies may be indicated in general terms. For terrestrial and solar research, beamwidths of about one minute are needed. For galactic and extragalactic investigations, resolutions on the order of seconds of arc are desirable. In general, the beamwidth θ of a parabolic reflector is given by the relation:

$$\theta = 57 \frac{\lambda}{D} \text{ degrees} \qquad (1)$$

where D is the reflector diameter and λ is the wavelength of observation. The minimum diameter aperture with which a resolution θ can be obtained is thus

$$D_{min} = 57 \frac{\lambda}{\theta}, \ \theta \text{ in degrees} \qquad (2)$$

The largest acceptable resolution yet proposed for directional lunar radio telescopes is 11.3 degrees, corresponding to a circular 100 square degree beam [4]. Accepting this requirement as the maximum tolerable beamwidth for a crater reflector operating 100 kc, a diameter of about 13.5 km is obtained from equation (2) as the smallest usable aperture for crater reflectors.

Locations

The selection of optimal sites for lunar radio astronomy is governed to a large degree by the type of research to be conducted. For studies of terrestrial and cis-lunar phenomena, a position near the central portion of the lunar disc is advantageous; galactic, extragalactic and solar research requires sites located toward the limbs to reduce terrestrial interference. For planetary studies, especially with fixed apertures, a location near the equator is desirable in view of the low inclination (5.1 degrees) of the lunar orbit to the ecliptic.

Certainly the most preferable site for radio astronomy purposes is the far side of the moon. Such a location would be shielded from terrestrial magnetospheric and ionospheric radiation as well as the many man-made sources of interference, thus permitting observations over a wider portion of the radio spectrum than has ever before been possible.

The accessibility of the selected sites is of paramount importance. The mobility available for lunar surface operations will be a function of the timing of the mission. Thus, early installations must be located at or near the touchdown point and will be generally situated in the equatorial regions proposed for Apollo landings. Crater locations suitable for association with optical observations in the highlands or with far side operations will not become accessible until the later stages of lunar exploration.

Knowledge of the surface and environmental properties of a particular region acquired from early Apollo landings might be fruitfully applied through followup missions to the same sites. However, such prior information would not be required for exploitation of crater reflectors because crater reflector systems would be justified if they provided a simple means for obtaining initial measurements of these very characteristics, and it is unlikely that utilization of craters would be attempted if initial analysis indicated that natural reflectors would be ineffective under the worst expected lunar conditions.

Except for cislunar research, the crater site must be selected so that the earth and its atmosphere will not interfere with the primary lobe of the crater reception pattern. Assuming that the principal beam of a symmetric parabolic crater reflector is normal to the lunar globe, Figure 2 may be used to estimate the extent of the region from which a crater might view the earth. The angular displacement Ω of the crater site with respect to the center of the lunar disc must be such that the primary lobe of the reflector does not intercept the earth. This lobe extends an angular distance equal to half the beamwidth θ from the crater axis. The limb of the earth is displaced from the earth-moon axis by the earth equatorial parallax α.

MINIMUM CRATER DISPLACEMENT FREE FROM TERRESTRIAL INTERFERENCE: $\Omega = \alpha + \Lambda + \frac{1}{2}\theta = 17.6°$

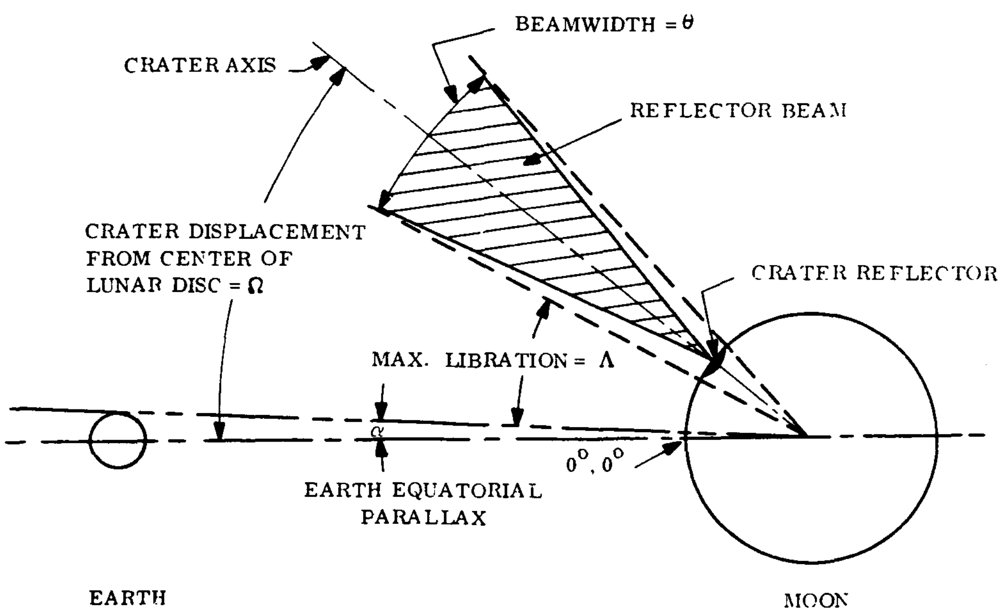

FIGURE 2. CRATER LOCATIONS SUBJECT TO PRIMARY LOBE INTERFERENCE FROM TERRESTRIAL SOURCES

Lunar libration can cause a displacement Λ of a point on the lunar surface by as much as 10.9 degrees [6]. Utilizing the maximum parallax (1.0 degree) and the defined maximum permissible beamwidth for a crater reflector (11.3 degrees), it is found that the angular displacement of a crater from the center of the lunar disc must exceed

$$\Omega = \alpha + \Lambda + 1/2\theta = 17.6 \text{ degrees} \qquad (3)$$

For the present study, the region of terrestrial primary lobe interference Ω was conservatively assumed to extend 25 degrees on each side of the central point of the lunar disc, thus defining a forbidden region for crater reflectors bounded by ± 25 degrees latitude and longitude. An exception to the above restriction was the analysis of a few craters near the center of the lunar disc for possible use in the high priority cis-lunar measurements.

Any crater reflector system must exhibit logistic, construction and operational requirements competitive with other proposed lunar astronomy experiments. There are three major advantages which are expected to be exploited by lunar radio astronomy systems: the possibility of large apertures, interferometer capabilities, and the low noise environment [7]. The antenna systems already suggested for lunar applications reflect this philosophy as illustrated in Table II. However, it appears evident from Table II that no real exploitation of the expected benefits of lunar surface radio observations will be available until the intermediate or advanced stages of the program.

Thus, an investigation of the feasibility of crater reflector systems is justified in view of the early utilization of the advantages of lunar operation which would become possible with simple equipment if crater reflectors were to prove practical. The measurements of terrestrial interference levels and lunar surface electrostatic and electromagnetic properties planned for the initial phase radio astronomy programs could be performed with crater reflectors just as well as with long-wire antennas. But, in contrast to the latter techniques, utilization of suitable lunar craters could yield immediate high resolution low frequency radio astronomy information after the initial analysis of environmental parameters.

In general, craters could provide opportunities for sophisticated radio observations excelled by only the most advanced lunar systems which have been proposed. Yet these natural reflectors would in principle require only a single set-up operation (at the focal point). Thus, there would be no need for the multiple surface operations required for construction of an array 20 km in diameter. This simplicity, in conjunction with the high resolutions attainable using the enormous apertures associated with crater reflectors,

TABLE II. PROPOSED LUNAR ANTENNA SYSTEMS

Development Phase	Mission Constraints	Antenna Type	Operational Range	Source
Initial	Simple systems	Long wires 0 and 3 m above lurain	1 kc - 30 mc	NAA
	Two men on foot	Short wire antennas of various lengths	10 cps - 50 mc	NAA
	Small 1-man vehicle			
Intermediate	Same as above	Short baseline (3 km) interferometer	600 kc - 1.2 mc	NAA
		Long baseline (9 km) interferometer array	300 kc - 1.0 mc	NAA
Advanced	Lunar base	Arrays		
	Far-side operations	100 square degree (at 1 mc) aperture	500 kc - 10 mc	NAS
		20 km aperture, medium-limited	100 kc - 10 mc	NAS
		20 km × 20 km T with single movable element in stem	Unspecified	OSSA
		Reflectors		
		10 m steered parabolic	50 μ - 2 mm	OSSA
		100 ft parabolic, surface accurate to 50 μ	1 mm - 1 μ	NAS
		1000-1500 ft spherical reflector	10 cm - 1 m	OSSA

Sources: NAA = North American Aviation, 1966 [8]; NAS = National Academy of Sciences, 1965 [3]; and OSSA = Office of Space Sciences and Applications, 1965 [7].

indicates the favorable capability of the concept with respect to proposed advanced systems. Utilization of several of the many available lunar craters in conjunction with appropriate data processing facilities could provide the capabilities needed for evolution of crater reflector systems into a sophisticated radio astronomy facility.

ENVIRONMENTAL CONSTRAINTS

The conditions and materials present on the surface of the moon exert a significant influence upon the evaluation of natural reflectors. Surface and subsurface characteristics - especially the electromagnetic properties - will affect the sensitivity and resolution of crater systems. Thus, present estimates of lunar environmental parameters relevant to crater reflector evaluation will be briefly summarized.

Lunar Surface Conditions

The tenuous nature of any existing lunar atmosphere implies that the absorption which limits terrestrial radioastronomy studies will be largely absent on the moon. For the purposes of crater reflector evaluation, the atmosphere can be discounted because its effects, if any, would occur at frequencies well above those suitable for crater systems.

An ionized region will reflect normally incident radiation below a critical frequency, f_c, defined by

$$f_c = \sqrt{\frac{e^2 N}{\pi m}} \qquad (4)$$

where e is the electron charge, m is the mass of the electron, and N is the electron density (per cm^3). A significant lunar ionosphere would play a dominant role in determining the low-frequency limits of lunar surface radio observations. Several estimates of the peak electron density and the associated critical frequency of the lunar ionosphere are listed in Table III.

TABLE III. PREDICTED LUNAR IONOSPHERIC ELECTRON DENSITIES

Source	Maximum Electron Density (cm^{-3})	Critical Frequency (kc)
Weil and Barasch, 1963 [9]	150	110
Herring and Licht, 1960 [10]	400	180
Elsmore, 1957 [11]	10^3 - 10^4	285 - 900
Heffner, 1965 [12]	2.4×10^3 - 2.4×10^5	440 - 4400

No definite experimental verification of the presence of an ionosphere has yet been obtained, although data from Luna 10 [13] indicate the existence of an increased electron flux in the region of the moon. The presence of critical frequencies as high as 4.4 mc (Table III) could severely hamper the utilization of crater reflectors (which appear to be useful only below 310 kc). Evaluation of this crucial constraint must await the receipt of more extensive data on lunar environment.

Even in the absence of an ionosphere, the cutoff frequency of the interplanetary medium, estimated to be near 100 kc [4], will pose an absolute low frequency bound to crater observations. Experimental verification of this limit remains to be obtained.

Another fundamental constraint would be imposed upon a lunar crater radio telescope by its immersion in the interplanetary plasma. The solar wind exhibits irregularities on the order of 100 km in extent [3], causing wide apertures to experience a loss of resolution. Erickson has estimated that the maximum resolution realizable at 1 mc on the lunar surface is limited to 24 seconds of arc by the intensity scintillations and positional oscillation arising from such irregularities [3]. It has been suggested that 20 kilometers is close to the ultimate radio telescope diameter for the 100 kc - 10 mc range [3]. Thus, a 20 km medium-limited aperture apparently poses an upper-bound on crater diameters. Larger craters will permit enhanced response levels because of their greater effective apertures but will yield no finer resolution.

The expected intensities of the radiation incident upon the lunar surface are important to crater reflector analysis but poorly known at present. In general, high flux densities are expected at low frequencies. The spectra of most discrete nonthermal radio sources rise toward the long wavelengths [14] although their intensities will decrease below some as yet unknown frequency. The galactic noise background increases in intensity toward long wavelengths and then begins to fall below about 3 mc [15]. Intense 725 kc radiation from the terrestrial ionosphere has already been noted. The relative magnitudes of these various emissions in the 100 kc - 10 mc range must still be determined experimentally, but their presence emphasizes the need for high resolution pencil beam observations at low frequencies to minimize interference and confusion problems. Such beamwidths could be supplied by lunar crater apertures; the enhanced low frequency intensities of the sources would serve to compensate for reflection losses at the crater surface. Note that the generally high source intensities at the wavelengths of interest should give unfilled apertures such as the annular parabolic crater (Fig. 1 - (c)) sufficient sensitivity to be feasible.

Lunar Subsurface Properties

Present knowledge of characteristics beneath the lunar surface is generally limited to theoretical predictions and estimates based upon terrestrial observation, the outstanding exception being the subsurface soil analysis performed by Luna 13. Currently available data will be summarized here to define the limits of present parameter estimates.

A variety of values for the relative dielectric constant ϵ_r of the lunar surface has been obtained [6]. Microwave experiments yield values of ϵ_r from 1.1 to 1.8; thermal studies imply a dielectric constant of 1.5 to 2.0 and radar investigations yield estimates of 1.8 to 2.8. These values are well below those expected for typical bulk rocks. However, the dielectric constant of granular materials has been found to vary almost linearly with porosity [16], hence the 1.1 to 2.8 range for the lunar dielectric constant is consistent with a porous or granular surface. The porosity dependence may be used in conjunction with various models of the lunar subsurface structure to account for the spread in measured values of ϵ_r. If the surface region is of homogeneous composition (at least to a depth equal to the wavelength used for the measurement), a uniform dielectric constant between 2 and 3 is likely if moderate porosities may be expected. A graded surface layer exhibiting a linear variation of packing density with depth [17] would possess a smoothly increasing dielectric constant to the depth at which the soil reaches a close-packed density (about one meter).

The most recent data support the existence of two regions beneath the lunar surface, each with a different dielectric constant: an underdense surface layer (70 percent porosity) with $\epsilon_r = 1.8$ and an underlying region of solid rock ($\epsilon_r = 4\text{-}5$). Corroborating this model are estimates of a 1 g/cc surface density derived independently from data reported by Surveyor I [18] and Luna XIII [19], the correlation of a dielectric constant of 1.8 with the soil porosity and density derived from Surveyor I data [18], experimental radar results indicating the presence of two subsurface layers, each possessing the previously mentioned values of ϵ_r [20], and cross-disciplinary theoretical analyses of the lunar surface which yield precisely the above model [21]. The impressive array of completely independent analyses highlights the two-layer model of the lunar surface as the most likely to be expected from present knowledge.

The depth and thickness of the hypothetical surface layers are important to the evaluation of crater reflectors, as will be shown. A number of estimates are presented in the compilation of lunar subsurface models given in Table IV.

TABLE IV. LUNAR SUBSURFACE MODELS

Source	Surface Type	Layer Order	Thickness	Nature of Material	Notes
Surveyor I	Mare	Top Bottom	0.02 - 1 m	Granular rock	JPL analysis
Lunar 9	Mare or highland	No layering		Porous lava or slag	Veined by differentiates (Soviet analysis)
Lunar 9	Mare or highland	Top Bottom	1 m	Plowed lava and dust lava	Analysis by T. Gold
Kuiper	Mare	Top Middle Bottom	0.3 - 1 m 10 - 20 m ~50 m	Porous fine fragments Rock froth Lava flows	Well-mixed region
	Highland	No layering		Hydrothermally altered original accreted surface	Undifferentiated except for altered regions
Shoemaker	Mare	Top Middle Bottom	1 mm 1 - 2 m	Finely divided powder Ejecta, like dry soil Rock	Irregular debris/rock interface Shattered and crushed
Urey	Mare or highland Inside craters Outside craters Mare or highland	Top Middle Middle Bottom	10 cm 30 - 40 m 20 - 40 m	Well-mixed powder Fragmented region Iron-rich rock	 Uniform composition to 1 km depth
Hagfors	Mare or highland	Top Bottom	20 - 30 cm	Porous (ϵ_r = 1.8) Rock (ϵ_r 4 - 5)	May contain density irregularities or small objects
TYCHO Report	Between craters	Top Bottom	0.5 - 1.5 m	Graded rubble Cracked lava	Irregular lava/debris interface with magna intrusions
Saair/Shorthill	Bright halo craters	No layering		Apparently bare rock	Thermally anomalous

Estimated thickness of the underdense region ranges from one millimeter to tens of meters. From the dependence of ϵ_r on surface porosity, it is evident that sharp discontinuities in the dielectric constant can be expected as much as 50 meters below the surface at interfaces between the various layers. Such abrupt changes could also exist at mineral/rock interfaces caused by differentiation or magma intrusion.

The conductivity of the lunar surface is even less well-known. Estimates range from a minimum of 3.4×10^{-4} mho/meter [22] to 0.16 mho/meter [23]. Such high conductivities (basalt averages 10^{-8} to 10^{-11} mho/m) indicate a surface enriched in metals and metal oxides, possibly caused by meteoric influx or solar wind sputtering [24].

The available estimates of the electromagnetic properties of the lunar surface thus yield the following limits

$$1.1 \leq \epsilon_r \leq 2.8$$

$$3.4 \times 10^{-4} \leq \sigma \leq 0.16 \text{ (mho/m)}$$

A useful characterization of the electromagnetic properties of a material is the loss tangent $\sigma/\omega\epsilon$, where σ is the conductivity, ϵ is the dielectric constant and ω is the angular frequency. For the frequencies of interest and the above parameter limits

$$2.2 \times 10^{-1} \leq \frac{\sigma}{\omega\epsilon} \leq 1.05 \times 10^{5}$$

Such values of $\sigma/\omega\epsilon$ correspond to substances ranging from lossy dielectrics to reasonably good conductors [25].

The lunar surface electromagnetic parameters affect the reflectivity of the surface and hence the efficiency (effective aperture) of a crater reflector. The reflection coefficient of a dielectric is a sensitive function of both ϵ_r and the loss tangent. For purposes of evaluation, the poorest expected properties have been assumed: $\epsilon_r = 2.0$ and $\sigma = 3.4 \times 10^{-4}$ mho/meter. Reflective characteristics have been evaluated for a perfect dielectric crater, and the effects because of the small but finite conductivity have been subsequently determined.

THEORETICAL PROPERTIES OF PARABOLIC CRATER REFLECTOR

The theoretical effectiveness of a crater radio telescope can be determined by considering the reception properties of a dielectric parabolic reflector. Calculation of the effective aperture of such a configuration determines the maximum efficiency of a crater reflector, and the theoretical aperture power distribution indicates the reception pattern of the crater. The effects of deviation from ideality are then determined, yielding constraints on subsurface inhomogeneity and surface irregularities.

Effective Aperture of Ideal Crater Reflector

The effective aperture of a receiving antenna is defined as the equivalent area over which the aerial removes energy from incident radiation; it is a measure of the efficiency of the antenna system. If the antenna reflective surface is a poor conductor or approximates a dielectric (as in the case for the lunar surface), power will be lost by partial transmission through the surface. The reflection coefficients are functions of both the dielectric constant and the polarization of the incident radiation

$$\Gamma_{\perp} = \frac{\sqrt{\epsilon_r - \sin^2\theta} - \cos\theta}{\sqrt{\epsilon_r - \sin^2\theta} + \cos\theta}$$

$$\Gamma_{||} = \frac{\sqrt{\epsilon_r - \sin^2\theta} - \epsilon_r \cos\theta}{\sqrt{\epsilon_r - \sin^2\theta} + \epsilon_r \cos\theta}$$

where $\Gamma_{\perp}$ is the amplitude reflection coefficient of radiation polarized perpendicular to the plane of incidence, $\Gamma_{||}$ is the corresponding quantity for radiation polarized parallel to the plane of incidence, θ is the angle of incidence, and ϵ_r is the relative dielectric constant. Note that conductive effects have been omitted. Both the angle of incidence and the orientation of the incoming radiation with respect to the plane of incidence will vary over the curved surface of a parabolic crater, and the reflection coefficients will thus be functions of position. To evaluate the efficiency (effective aperture) of crater reflectors, the effects of dielectric constant and oblique incidence must be estimated.

A calculation of the effective aperture of a dielectric parabolic reflector relative to a perfectly conducting paraboloid is given in Appendix A. The results are depicted in Figures 3 and 4 as a function of the dielectric constant ϵ_r and the parameter D/f (D = diameter of parabola aperture, f = focal height from

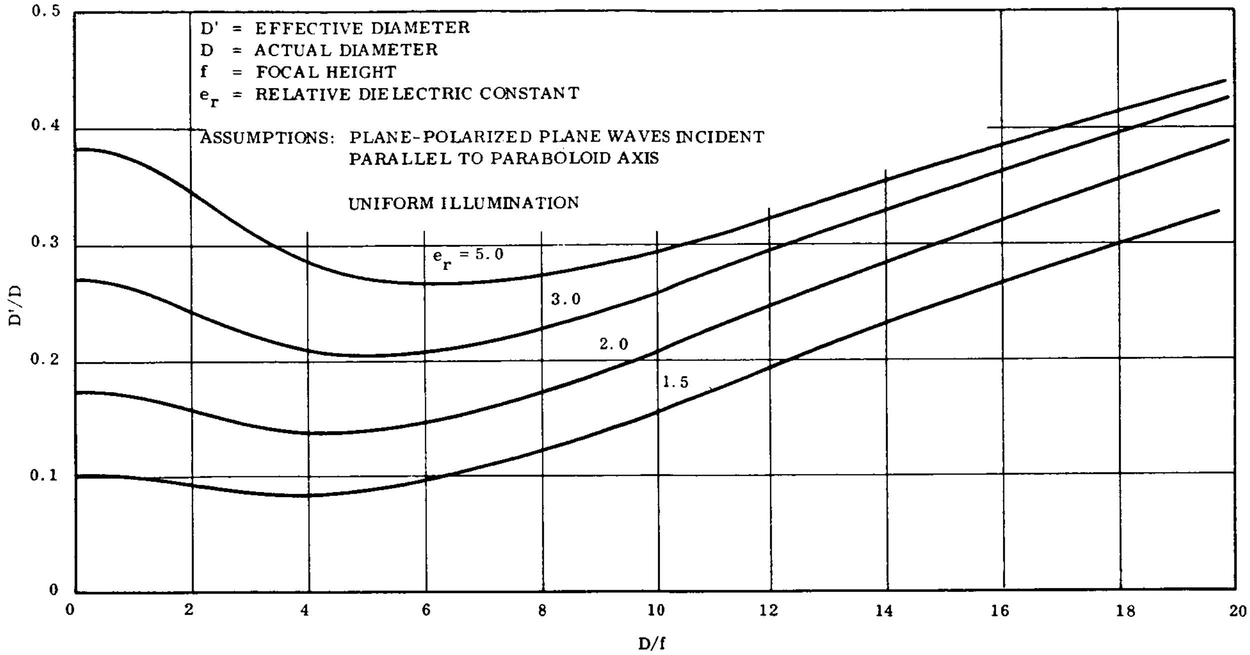

FIGURE 3. RATIO OF EFFECTIVE DIAMETER TO ACTUAL DIAMETER AS FUNCTION D/f

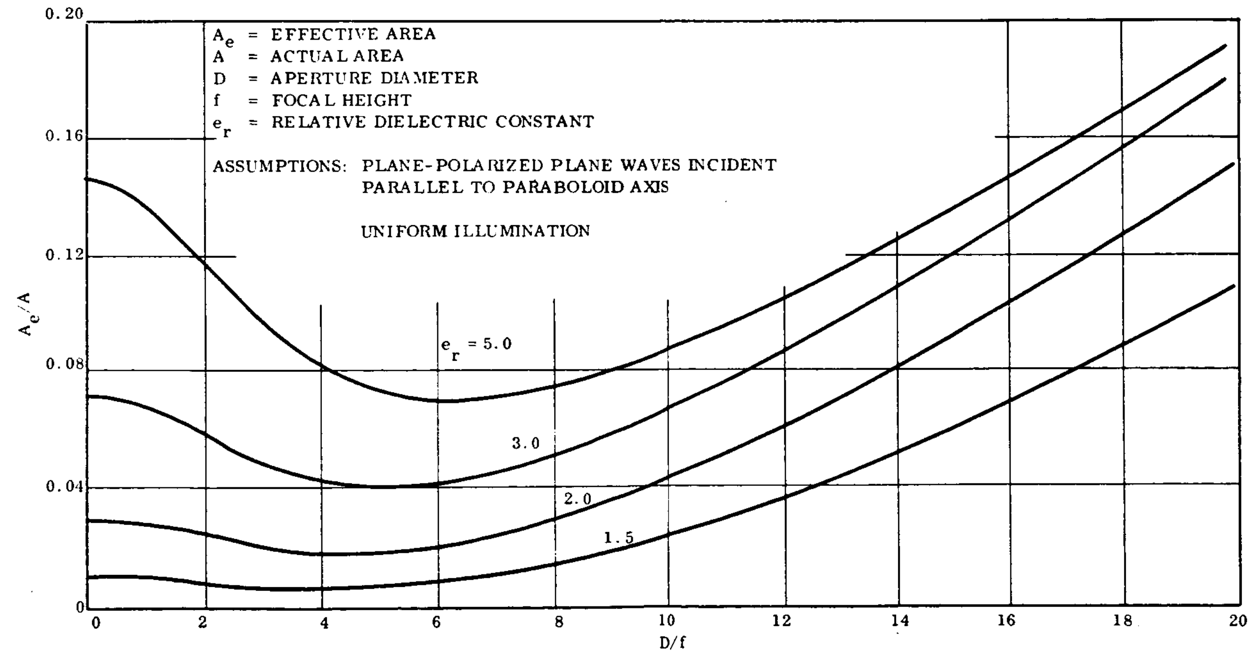

FIGURE 4. RATIO OF EFFECTIVE AREA TO ACTUAL AREA AS FUNCTION OF D/f

parabola vertex). These curves represent the reduction in total amplitude (Fig. 3) and power (Fig. 4) at the focus caused by transmission losses through the reflector. For the expected lunar surface dielectric constant of 2.0, the minimum effective area is only one-fiftieth that of the corresponding conducting parabolic reflector. However, for a 13.5-km diameter crater (the smallest under consideration), such an efficiency is equivalent to a perfect parabolic reflector 1.9 km in diameter, well in excess of present terrestrial capabilities. If solid rock ($\epsilon_r = 5.0$) were as near to the surface as is anticipated for the two-layer model, the minimum expected effective diameter of a 13.5-km crater would be 3.6 km. Thus a dielectric crater reflector appears to be efficient enough under the worst expected lunar conditions to permit sensitive observations.

Aperture Power Distribution

The reception pattern of a parabolic reflector may be determined from the distribution of power across its aperture, assuming a radiation source to be placed at the focus. Calculations of this distribution are given in Appendix B for two feed polarizations [26]. The results are presented in Figures 5 and 6, where the aperture power density, P, has been normalized by the focal height, f, and the power distribution of the feed antenna, U.

Utilization of a vertical feed antenna (parallel to the paraboloid axis) will cause all radiation to be polarized parallel to the plane of incidence. It is evident from Figure 5 that, for a given reflector (constant focal height), the aperture power density decreases linearly from the crater axis to D/f = 4. For those lunar craters which have been found to be parabolic, the maximum D/f has been found to lie between 1.25 and 2.93.

The corresponding distribution for a loop feed antenna (axis coincident with paraboloid axis) is depicted in Figure 6. This configuration causes the radiation to be polarized perpendicular to the plane of incidence at every point on the reflector. It can be seen that the power density is depressed slightly towards the paraboloid axis but increases to a peak at D/f = 4 to 8, depending upon ϵ_r. The power density for D/f = 0 to 3, characteristic of actual parabolic lunar craters is much higher than for a vertical feed antenna.

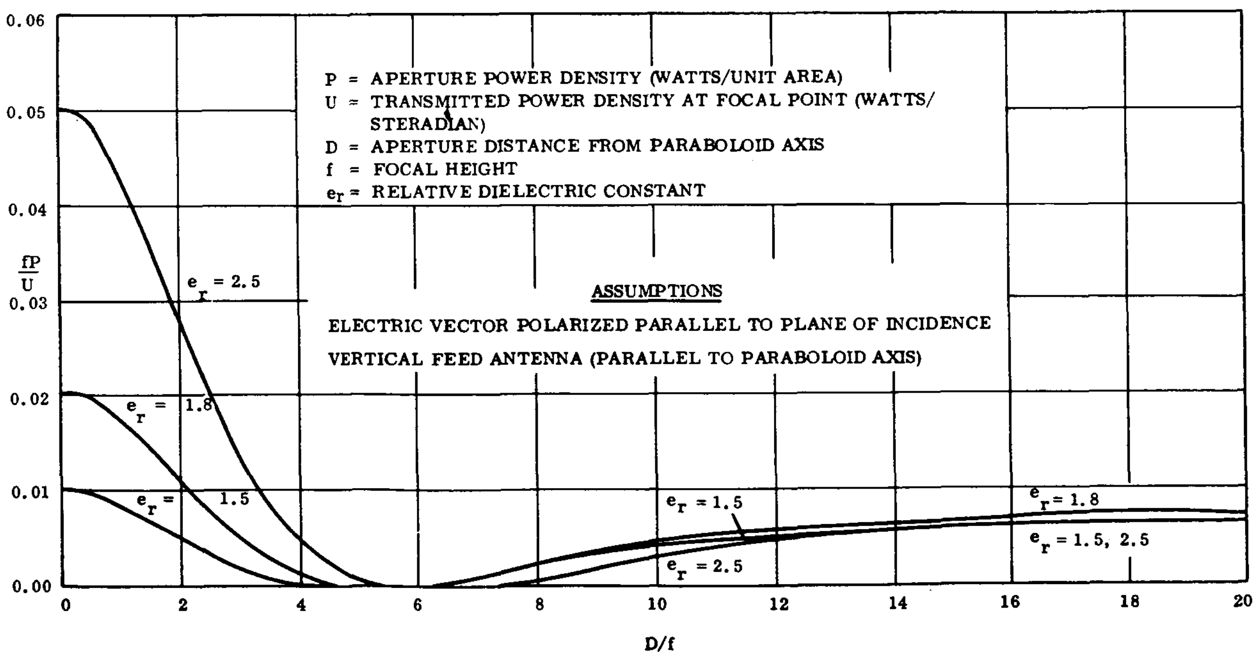

FIGURE 5. APERTURE POWER DENSITY AS A FUNCTION OF D/f FOR A VERTICAL FEED ANTENNA

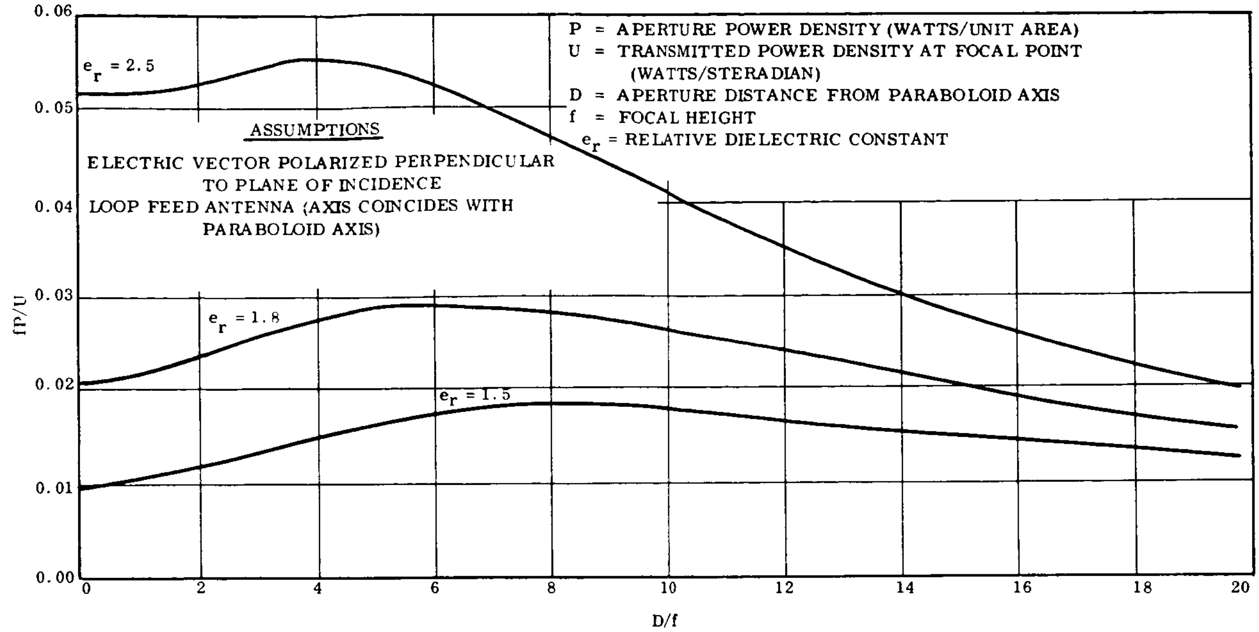

FIGURE 6. APERTURE POWER DENSITY AS A FUNCTION OF D/f FOR A LOOP FEED ANTENNA

The reception polar pattern is the Fourier transform of the aperture amplitude distribution (square root of power distribution). It may be inferred from Figures 5 and 6 using the following information about the properties of aperture distributions [26].

1. For a uniform aperture amplitude distribution, the reception pattern $R(\theta)$ is of the form

$$R(\theta) = \frac{\sin(k \sin\theta)}{k \sin\theta}$$

where k is a constant and θ is the angular displacement from the paraboloid axis.

2. For a triangular aperture distribution, the form of $R(\theta)$ is

$$R(\theta) = \left(\frac{\sin(k \sin\theta)}{k \sin\theta}\right)^2$$

3. Tapering the amplitude distribution towards the edges from a maximum at the center of the aperture tends to broaden the beamwidth of the main lobe (degrading resolution) while reducing sidelobe level.

4. An inverse taper (depressed amplitude at the aperture) yields a sharper primary lobe (and correspondingly enhanced resolution) but increases sidelobe magnitude.

The aperture amplitude distribution for a vertical feed antenna in a parabolic reflector can be represented by superposition of triangular and uniform distributions for D/f = 0 to 3 (Fig. 5). Thus the expression for the lobe pattern of a crater reflector with a vertical feed antenna is midway between (sin x)/x and (sin x/x)^2: a broadened primary lobe with somewhat depressed sidelobes.

The aperture amplitude distribution with a loop feed antenna is nearly uniform except for a slight inverse taper. Thus the reception pattern should closely resemble the form (sin x)/x except for slightly enhanced sidelobes and narrower beamwidth than the (sin x)/x form would indicate. Because of its generally higher sensitivity and resolution, the loop feed antenna seems to be the better of the two configurations analyzed.

Tolerable Subsurface Inhomogeneity

The most critical constraint imposed by the low but finite expected lunar surface conductivity concerns the homogeneity required of the subsurface region to penetrating radiation from discontinuities in the subsurface material. Such reflected waves would possess an appreciable phase shift with respect to the radiation incident at the surface if the penetration depth is a sizable fraction of a wavelength. The existence of stratifications exhibiting large-scale conductivity discontinuities could thus lead to defocusing, loss of effective aperture and a requirement that the maximum frequency of operation be reduced to such a level that the shortest wavelength utilized will be large with respect to the penetration depth.

An evaluation of the depths from which such reflections would be harmful can be obtained by analyzing the penetration in terms of the Rayleigh criterion [27] for tolerable irregularities on a specular surface. Assuming perpendicular incidence, reflection from a depth h will introduce an additional phase shift $2h \times 2\pi/\lambda$, where λ is the wavelength of the radiation. According to the Rayleigh criterion, the maximum tolerable incoherence because of surface irregularities is $\pi/4$. Thus

$$\frac{4\pi h}{\lambda} \le \frac{\pi}{4}$$

or

$$h \le \frac{\lambda}{16} \tag{5}$$

Reflections from random inhomogeneities at depths less than $\lambda/16$ would not be expected to cause appreciable defocusing because the added phase difference would not exceed the Rayleigh criterion. Subsurface discontinuities are thus similar to surface irregularities: large-scale inhomogeneities are intolerable but a random distribution of small irregularities (for example, buried meteorites or an irregular rubble/rock interface) will provide negligible degradation.

The maximum depth at which subsurface reflections are undesirable is a function of the skin depth, δ, is

$$\delta = \sqrt{\frac{1}{\pi f \mu \sigma}} \tag{6}$$

where f is the radiation frequency, μ is the surface permeability, and σ is the surface conductivity. This distance represents the depth at which the amplitude of the penetrating wave has fallen to 1/e (37 percent) of its initial value. For crater reflector evaluation, the effects of the low reflectivity of the dielectric surface must be considered. It has been shown that the effective area of a dielectric parabolic reflector may be as little as two percent of the actual area. The remaining 98 percent of the energy incident upon the reflector will be transmitted into the lunar surface and subject to reflections from subsurface inhomogeneities. Thus the penetration depth which is significant for crater reflectors can be defined as the distance at which the amplitude of the transmitted wave, E_T, is reduced to 1/e of the amplitude, E_R, of the wave reflected from the surface of the crater. This power penetration depth, δ', is defined by the relation

$$E_T e^{-\delta'/\delta} = E_R e^{-1}$$

or

$$\frac{E_T}{E_R} = e^{\frac{\delta'}{\delta} - 1} \tag{7}$$

But the average reflected amplitude per unit aperture is just E_F, calculated in Appendix A, and the average transmitted amplitude $E_o - E_F$ where E_o is the amplitude of the incident radiation. Thus

$$\frac{E_T}{E_R} = \frac{1 - (E_F/E_o)}{E_F/E_o} = \frac{1 - (D'/D)}{D'/D} \tag{8}$$

where E_F/E_o, defined by Equation (A-16), is equivalent to the relative effective diameter D'/D plotted in Figure 3. Solving for δ'

$$\delta' = \left[\ln\left(\frac{1}{D'/D} - 1\right) + 1\right]\delta = k\delta \tag{9}$$

The constant k adjusts the skin depth to compensate for the poor reflectivity of the dielectric surface. Typical maximum values for k are 3.4 ($\epsilon_r = 1.5$), 2.8 ($\epsilon_r = 2.0$), and 2.0 ($\epsilon_r = 5.0$).

At a given frequency, the subsurface material must be homogeneous at depths from h to δ': above h, the added incoherence is negligible; below δ' the amplitude of the reflected wave is negligible. The critical region is depicted as a function of frequency in Figure 7 for the

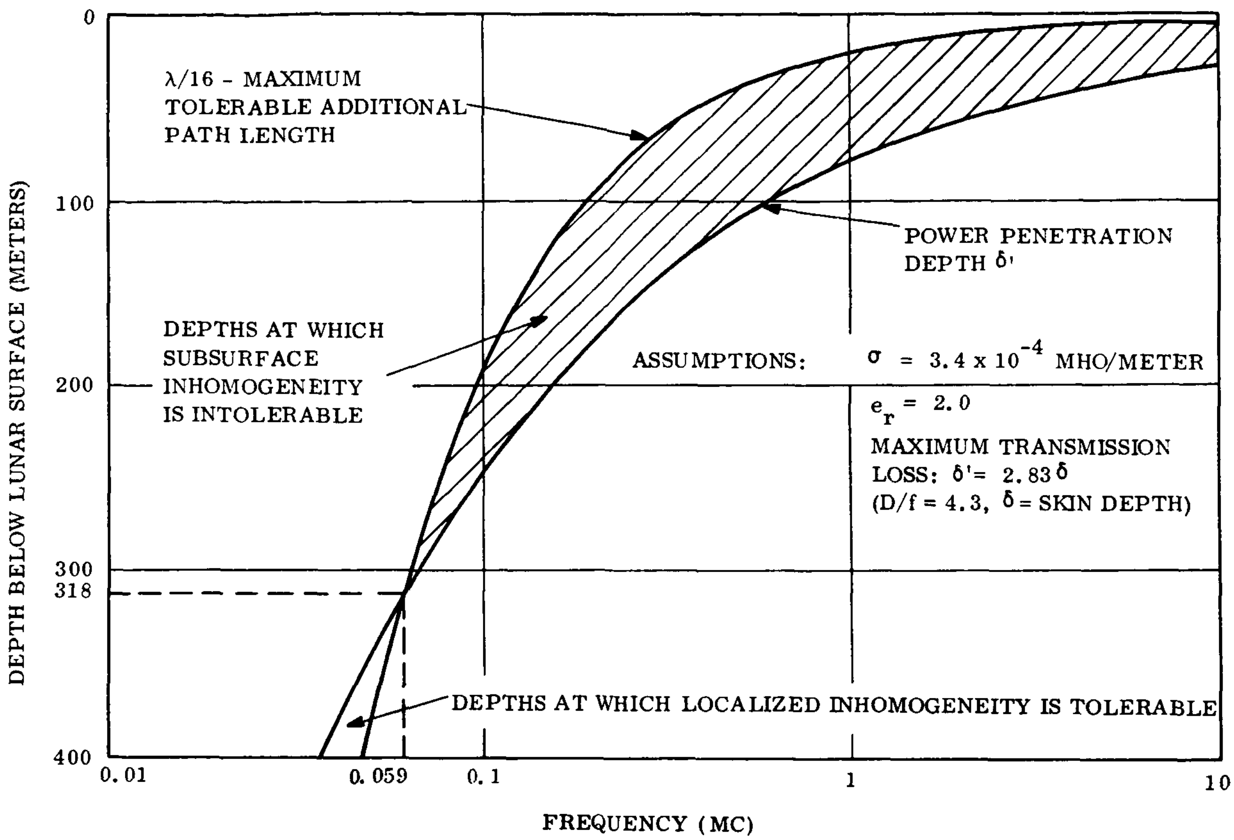

FIGURE 7. DEPTHS AT WHICH DISCONTINUOUS SURFACE PROPERTIES ARE INTOLERABLE

lowest expected lunar surface conductivity (3.4×10^{-4} mho/meter). The graph indicates that below 318 m or 59 kc, no homogeneity constraints exist. Above the latter frequency, the constrained region increases to a maximum thickness of 80 m at 236 kc and then falls to very small values at high frequencies. Typical constrained regions at the frequencies of interest include:

188 to 244 m (100 kc)
19 to 77 m (1 mc)
2 to 24 m (10 mc)

These constraints may be compared with the models of the lunar subsurface listed in Table IV. Many types of discontinuities are possible; soil/debris, debris/rock or rock froth/rock interfaces could exist to depths of up to 50 m. From Figure 7 it is seen that the first 50 m of the lunar surface will be critical only for reflectors operating above 375 kc. Note that thermally anomalous craters are expected to exhibit shallow or nonexistent debris layers [21], thus minimizing the chances for inhomogeneity problems.

Tolerable Surface Irregularities

The effects of surface irregularities are similar to those of subsurface discontinuities in that they determine the maximum frequency at which the reflector may be operated. The Rayleigh criterion may again be employed to define the tolerable limits to irregularities on the reflecting surface, and general expressions for the constraints on surface irregularities are developed in Appendix C. The maximum tolerable vertical irregularity, ϵ_y, is given by

$$\epsilon_y < \frac{\lambda}{16}\left[\left(\frac{R}{2f}\right)^2 + 1\right] \tag{11}$$

and for horizontal irregularities, ϵ_x, one has

$$\epsilon_x < \frac{\lambda}{16}\left[\frac{R}{2f} + \frac{2f}{R}\right] \tag{12}$$

where λ is the wavelength of observation, f is the focal height and R is the radial distance of the irregularity from the paraboloid axis. Appropriate restrictions on ϵ_y and ϵ_x are given in Appendix C. Since the maximum value of R/f is 0.5 to 1.5 for actual lunar parabolic craters (D/f = 1 to 3), the Rayleigh criterion (which implies the necessity for $\lambda/16$ accuracies) is only slightly relaxed. A random distribution of irregularities over the crater surface is acceptable, however, even if some irregularities are larger than the above limits. Equations (11) and (12) are especially useful in evaluating the advantages of asymmetric and annular crater reflectors (Fig. 1) as has been shown by Anderson [26] and Greiner [28].

From equation (11), the maximum frequency at which a crater may be efficiently operated is approximately

$$f_{max} = \frac{18.75}{\epsilon_y}\left[\left(\frac{D}{4f}\right)^2 + 1\right] \quad (\mathrm{Mc}) \qquad (13)$$

where D is the crater diameter, f is the focal height and ϵ_y is the average vertical irregularity in meters. This restriction is one of the most important limitations imposed upon crater capabilities. For operation at 10 Mc, ϵ_y must be less than 3 m. This result emphasizes the constraint that natural reflectors be used only at low frequencies.

IDENTIFICATION AND EVALUATION OF ACTUAL PARABOLIC CRATERS

In summary, the following characteristics are desirable for prospective crater reflectors:

1. Close approximation of parabolic shape.
2. Large actual and effective aperture.
3. Minimal power penetration depth.
4. Homogeneous subsurface composition.
5. Limb and/or equatorial location.
6. Accessible focal feed location.

The latter constraint will be discussed later. The above properties can be used to define a dimensionless qualitative figure of merit, Q:

$$Q = \frac{D}{f}\,\frac{\Delta\nu}{\nu_{min}}\,\frac{D'}{\lambda_{min}}\,\frac{(K\delta)^2}{k} \qquad (14)$$

D = actual diameter

D' = effective diameter

f = focal height

δ = skin depth

k = ratio of power penetration depth to skin depth

λ_{min} = minimum operating wavelength

ν_{min} = minimum operating frequency

$\Delta\nu$ = frequency bandwidth

K = propagation constant

The minimum operating frequency is defined from equation (13) and is a measure of the closeness with which the crater approximates a paraboloid. The effective aperture factor expresses the effects of the surface dielectric constant. Minimum operating wavelength has been defined as that wavelength at which the beamwidth equals 100 square degrees. The factor, k, is defined in equation (9) as the proportional increase in penetration depth caused by poor reflectivity and is an index of subsurface problems. The term $(K\delta)^2$ has been suggested by Anderson [26] as an index of the overall electromagnetic properties of the surface. The formula for this factor is

$$(K\delta)^2 = \frac{4\pi\epsilon f}{\sigma} + \frac{\sigma}{4\pi\epsilon f} \qquad (15)$$

where ϵ is the dielectric constant, σ is the conductivity and f is the frequency of operation.

Utilization of lunar craters as parabolic reflectors is of course contingent upon the existence of craters of suitable shapes. To determine the feasibility of a crater reflector system, available lunar relief maps have been scanned for parabolic craters, and those features which appeared satisfactory have been analyzed in detail.

The maps utilized in this investigation were from the LAC series of lunar topographical charts [29] and include contours of constant elevation measured with respect to an assumed spherical datum surface. The contours are drawn for 300 meter elevation intervals and can be conveniently exploited in the evaluation of crater shapes for reflector applications. A criterion which permits recognition of parabolic shape from a constant elevation contour map can be easily derived from the general equation of a paraboloid :

$$D^2 = 16\ fz \qquad (16)$$

where D is the diameter, z is the height of the given diameter above the vertex, and f is the focal height. Taking the difference between diameters at successive elevations :

$$D_2^2 - D_1^2 = 16\,f\,(z_2 - z_1) = 16\,f\,\Delta z$$

where Δz is the relative difference in elevation between the diameters D_2 and D_1. For the LAC maps, the differential elevations are constant (to within ± 100 m), and thus

$$D^2_{n+1} - D^2_n = \text{constant} = 16\,f\,\Delta z \qquad (17)$$

Thus, if a crater has a parabolic shape, successive diameters must be related by equation (17), and the complete series of level curves corresponding to that crater will yield the constant value $16\,f\,\Delta z$ when analyzed according to equation (17).

A representation of a parabolic crater by successive elevation contours is shown in Figure 8. Analysis of the successive diameters in this drawing verifies the constancy of the term $D^2_{n+1} - D^2_n$. The uniqueness of equation (17) in identifying craters of parabolic shape has been demonstrated elsewhere [28].

Since it is not expected that craters will satisfy equation (17) exactly, techniques must be developed to permit a determination of how well a crater approximates paraboloidal shape. A criterion for recognizing parabolic craters must be obtained that takes account of measurement errors and small real deviations from the desired shape. A statistical analysis of the differences between the squares of successive crater elevation diameters fulfills both requirements.

Assume that for a given crater, there exist N + 1 elevation contours and a corresponding number of diameter measurements. In analogy with equation (17), the quantity X_n can be formed for each pair of successive diameters D_{n+1} and D_n:

$$X_n = D_{n+1}^{\ 2} - D_n^{\ 2} \qquad (18)$$

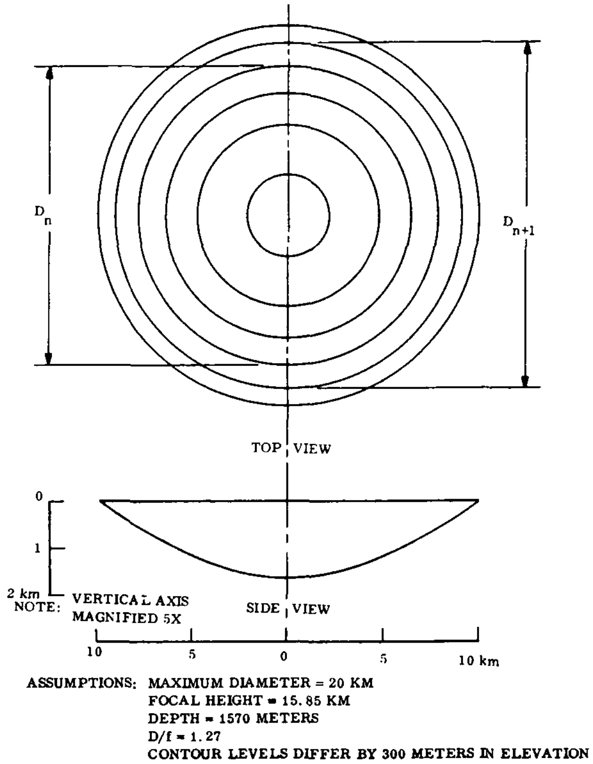

FIGURE 8. SAMPLE ELEVATION CONTOUR DIAGRAM - PARABOLIC CRATER

If the given crater were a paraboloid, each of these N quantities would be equal. In general, the average of all the X_n for a crater is

$$\overline{X} = \frac{1}{N}\sum_{n=1}^{N} X_n = \frac{1}{N}\sum_{n=1}^{N} (D_{n+1}^{\ 2} - D_n^{\ 2}) = \frac{D^2_{N+1} - D^2_1}{N} \qquad (19)$$

Taking note of the uncertainty in the determination of $\overline{X}$ for a given crater, one can write the result of the preceding analysis as

$$\overline{X} \pm \frac{\sigma}{\sqrt{N}} = 16\,f\,\Delta z \quad \text{(if parabolic)} \qquad (20)$$

where N is the number of quantities X_n which were averaged, σ is the standard deviation of $\overline{X}$ and $\sigma/\sqrt{N}$ is the standard error of $\overline{X}$ [30].

If the craters are parabolic, according to equation (17) the N numbers X_n will all be equal, and therefore σ will be zero. However, the process of measuring the contour diameters introduces accidental errors which will produce a non-zero standard deviation for $\overline{X}$ in any set of measurements, even for parabolic craters. The expected contribution to σ from accidental errors can, however, be estimated. Hence a crater will be assumed to approximate a parabolic reflector if the standard deviation of $\overline{X}$ is less than the estimated standard deviation due to accidental errors.

Using equation (18) in conjunction with the usual formulas for the propagation of errors [30], the standard error α_n of the quantity X_n is found to be

$$\alpha_n^2 = 4(\alpha_{n+1}^2 D_{n+1}^2 + \alpha_n^2 D_n^2) \qquad (21)$$

where α_{n+1} and α_n are the standard errors associated with the measurements of D_{n+1} and D_n, respectively. It will be assumed that on the average each measurement has a standard error α_o caused by observational inaccuracies. Thus an upper bound to equation (21) is

$$\alpha_n^2 \leqq 4\,\alpha_o^2\,(D_{n+1}^2 + D_n^2) \qquad (22)$$

If the crater is approximately parabolic, one can use equations (16) and (17) to obtain

$$D_{n+1}^2 = D_n^2 + 16f\Delta z = D_n^2\left(1 + \frac{16f}{D_n^2}\Delta z\right) = D_n^2\left(1 + \frac{\Delta z}{z_n}\right) \qquad (23)$$

It is possible to obtain an average upper bound of 3.22 on the quantity $(1 + \Delta z/z_n)$ by consideration of the actual set of craters analyzed [28]. By substituting this bound and equation (23) into (22), an upper bound for the standard error in X_n is determined:

$$\alpha_n^2 \leqq 4\,\alpha_o^2\,(1 + 3.22) = 16.89\,\alpha_o^2 D_n^2 \qquad (24)$$

The mean square standard error of the set of N values of X_n, using equation (24), is

$$\overline{\alpha^2} = \frac{1}{N}\sum_{n=1}^{N} \alpha_n^2 = \frac{16.89\,\alpha_c^2}{N}\sum_{n=1}^{N} D_n^2 = 16.89\,\alpha_o^2\,\overline{D'^2} \qquad (25)$$

where $\overline{D'^2}$ is the average of the lowest N values of D_n^2. This quantity can be approximated by assuming a parabolic shape for the crater and averaging D_n^2 over the aperture. The result is

$$\overline{D'^2} \cong \frac{1}{2} D_N^2 \qquad (26)$$

where D_N is the second largest diameter. Substituting equation (26) and equation (25)

$$\overline{\alpha^2} \leqq 8.45\,\alpha_o^2 D_N^2 \qquad (27)$$

Since the standard error α of a set of N measurements is related to the standard deviation σ of those measurements by the formula $\alpha = \sigma/\sqrt{N}$, equation (27) can be written in terms of the standard deviation σ_o of the accidental errors involved in the measurements leading to $\overline{X}$

$$\sigma^2 \leqq 8.45\,\sigma_o^2 D_N^2$$

$$\sigma \leqq 2.91\,\sigma_o D_N \qquad (28)$$

But the standard deviation is related to the mean deviation ϵ by the equation [30]

$$\sigma = \sqrt{\frac{\pi}{2}}\,\epsilon$$

Substituting this expression into equation [28] and putting the result in a convenient form

$$\frac{\sigma}{D_N} \leqq 3.65\,\epsilon_o \qquad (29)$$

where D_N is the second largest contour diameter and ϵ_o is the estimated accidental error in the diameter measurements.

In practice, contour diameters for each crater were measured along north-south and east-west directions to permit evaluation of oval shape and a better estimate of irregularities. The parameter σ/D_N has been evaluated separately for the north-south and east-west scans, and the results of the two scans have been averaged:

$$\left(\frac{\sigma}{D_N}\right) = \frac{1}{2}\left(\frac{\sigma}{D_N}\bigg|_{NS} + \frac{\sigma}{D_N}\bigg|_{EW}\right) \qquad (30)$$

Equation (30) provides a measure of the average deviation from paraboloid shape of the crater in question. If it is assumed that the average accidental error ϵ_o associated with a measurement is 0.2 km, equation (29) yields

$$\left(\frac{\bar{\sigma}}{D_N}\right) \leqq 0.730 \qquad (31)$$

Craters which satisfy the condition of equation (31) can be assumed to be approximately parabolic, and the quantity $\overline{\sigma/D_N}$ is an index of the closeness of this approximation.

Several selected craters were evaluated by a technique which eliminates some of the approximations required for obtaining equation (31). From equation (22), the average α^2 is obtained for the set of quantities α_n^2:

$$\overline{\alpha^2} \leq \frac{4\alpha_o^2}{N}\sum_{n=1}^{N}\left(D_{n+1}^2 + D_n^2\right) = \frac{4\alpha_o^2}{N}\left[\sum_{n=1}^{N} 2D_n^2 + D_{N+1}^2 - D_1^2\right] \qquad (32)$$

But the above sum is merely the mean square diameter $\overline{D^2}$ of the first N measurements. Using equation (19) one finds that

$$\overline{\alpha^2} \leq 4\alpha_o^2\left[\overline{2D^2} + \frac{\overline{D^2_{N+1}} - D_1^2}{N}\right] = 4\alpha_o^2\left(\overline{2D^2} + \overline{X}\right) \qquad (33)$$

Transforming to the standard deviation σ and the estimated mean deviation ϵ_o, equation (33) becomes

$$\sigma^2 \leq 2\pi\epsilon_o^2\left(2\overline{D^2} + \overline{X}\right) \qquad (34)$$

Each term in this expression must be evaluated individually for every crater, and thus the equation is not as convenient as equation (31) for the analysis of a large number of prospective reflectors.

If a crater is determined to be parabolic from equation (31), a number of reflector characteristics can be derived. The natural focal height can be obtained from equation (20) with $\Delta z = 0.3$ km for the LAC maps

$$f = \frac{1}{4.8}\left(\overline{X} \pm \frac{\bar{\sigma}}{\sqrt{N}}\right) \qquad (35)$$

where $\bar{\sigma}$ is the average of the standard deviations of of the north-south and east-west scans. The lower frequency limit of the crater may be evaluated from equation (1) utilizing the known crater diameter and the constraint that a 100 square degree beamwidth is the maximum tolerable. A crude estimate of the upper frequency limit of the crater reflector can be obtained using equation (13). It is assumed that the actual average crater irregularity ϵ can be found by applying equation (29) or (34) to the crater data and identifying the resulting mean deviation ϵ_o with the average irregularity ϵ:

$$\epsilon \cong \frac{1}{3.65}\left(\frac{\bar{\sigma}}{D_N}\right) \qquad (36)$$

or

$$\epsilon \cong \frac{\sigma}{\sqrt{2\pi\left(\overline{2D^2} + \overline{X}\right)}} \qquad (37)$$

Note that this is a lower bound to f_{max} since ϵ as given by equations (36) and (37) is probably over-estimated. The numerical coefficient of equation (29) was obtained after many approximations, and its utilization in the definition of an upper frequency limit is questionable. Furthermore, accidental errors are treated in this approximation as actual crater irregularities. However, for lack of any better estimate, the upper frequency limit as obtained from the above expressions will be used in the analysis of crater reflectors. The actual upper frequency limit is probably somewhat higher than that defined by equations (36) and (37).

LUNAR PARABOLIC CRATERS

Using the LAC maps, the lunar surface has been surveyed for the presence of parabolic craters. The portion of the lunar disc which has been studied is indicated in Figure 9. Except for the forbidden area previously defined, all regions of the surface were scanned for which LAC maps were available. The central area of the lunar disc was also examined for craters which would be suitable for cislunar research.

The characteristics sought in the initial selection of prospective craters prior to detailed evaluation of paraboloid shape included the following:

1. Crater diameters were limited to the range 8.5 to 25 miles.

2. The craters must be located either close to the center of the lunar disc or beyond the region bounded by ±25° latitude and longitude.

3. The crater contours must be smooth and very nearly circular.

4. The presence of a central peak is highly desirable.

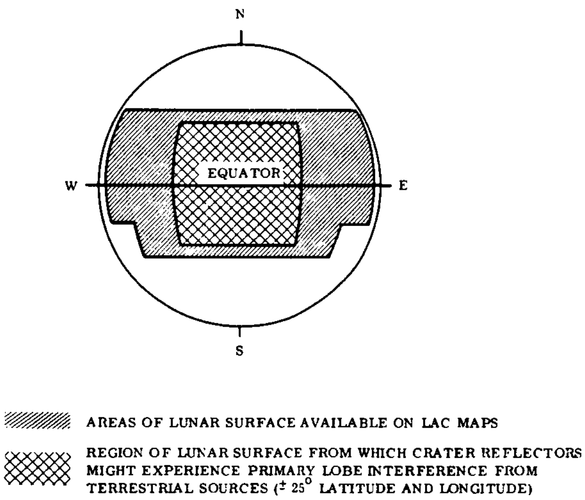

FIGURE 9. REGION OF LUNAR SURFACE EXAMINED FOR PARABOLIC CRATERS

In all, 99 craters were identified which met the four qualifications listed above. Of these, 34 could not be evaluated for paraboloid shape due to the absence of evaluation contours beyond 63°W longitude and 56°E longitude. The remaining 59 craters were rated for parabolic shape using the method outlined previously.

From the criterion given in equation (31), thirty craters were found which approximate a paraboloid to the limits of observational accuracy. The important characteristics of these craters are summarized in Table V where they have been listed in the order of the closeness of their approximation to a paraboloid. The natural focal heights were calculated using equation (35), and the maximum frequency of operation was obtained from equations (13) and (36). The minimum frequency of operation was defined as that frequency for which the beamwidth was 11.3 degrees. Effective diameter was obtained from the parameter D/f and Figure 3.

A number of general results may be observed from Table V. The detailed analysis covered 10.8 percent of the lunar surface, and 29 suitable craters were found in this area (omitting the single usable crater located in the central region). Extrapolating the results over the entire lunar surface, one can expect to find at least 270 suitable craters on the moon. The frequency range covered by the parabolic craters is 50 to 309 kc, within the spectrum of interest for lunar radio astronomy. The effect diameters (2.0 to 4.7 km) are all well in excess of terrestrial capabilities. From the given frequency range and Figure 7,

TABLE V. PARABOLIC CRATERS

No.	Name	Diameter (km)	Depth (km)	Focal Height (km)	$\frac{D}{f}$	$\left(\frac{\bar{\sigma}}{D_N}\right)$	Maximum Frequency (km)	Minimum Frequency (km)	Effective Diameter (km)
1	Neander L	18.0	1.49	8.9 ± 0.5	2.03	0.248	309	84	2.8
2	Santbech A	23.8	2.60	13.0 ± 0.6	1.84	0.279	258	64	3.8
3	Herigonius	14.5	1.70	6.5 ± 0.4	2.23	0.295	265	104	2.2
4	Isodorus D	16.9	2.34	7.7 ± 0.5	2.20	0.398	180	90	2.6
5	Brayley	16.1	2.31	5.5 ± 0.4	2.93	0.411	206	94	2.3
6	Messier G	14.4	1.40	10.0 ± 0.6	1.44	0.443	164	105	2.3
7	E Pickering	15.3	2.38	6.3 ± 0.6	2.43	0.458	167	99	2.3
8	Glaisher	17.7	2.60	10.5 ± 0.8	1.69	0.474	157	86	2.8
9	Colombo E	14.5	1.29	7.1 ± 0.6	2.04	0.488	158	104	2.3
10	Flamsteed A	13.7	1.09	8.0 ± 0.7	1.71	0.545	127	111	2.2
11	Taruntius A	14.5	2.00	9.4 ± 0.7	1.54	0.548	126	105	2.3
12	Proclus A	14.5	1.80	11.6 ± 1.0	1.25	0.550	130	104	2.4
13	Bessarion A	13.7	1.28	5.4 ± 0.6	2.54	0.551	147	110	2.0
14	Vitruvius A	19.2	2.46	6.9 ± 0.8	2.78	0.563	148	79	2.8
15	Piccolomini D	18.5	2.20	6.6 ± 0.6	2.80	0.571	147	82	2.7

TABLE V. PARABOLIC CRATERS (Cont'd)

No.	Name	Diameter (km)	Depth (km)	Focal Height (km)	$\frac{D}{f}$	$\left(\frac{\bar{\sigma}}{D}_N\right)$	Maximum Frequency (km)	Minimum Frequency (km)	Effective Diameter (km)
16	Fontana G	15.3	1.55	9.9 ± 0.8	1.55	0.574	128	99	2.5
17	Proclus	26.8	2.70	15.5 ± 1.4	1.73	0.603	123	56	4.3
18	Kepler	30.6	2.30	14.9 ± 1.5	2.05	0.616	123	50	4.7
19	Galilaei	17.7	2.01	7.6 ± 0.8	2.33	0.616	128	86	2.7
20	Mersenius S	17.2	1.60	8.3 ± 0.8	2.07	0.623	124	88	2.7
21	Neander C	18.5	2.13	9.2 ± 1.0	2.02	0.624	123	82	2.9
22	Mcclure	26.5	1.50	20.7 ± 1.5	1.28	0.628	114	57	4.4
23	Heis	14.5	1.73	5.6 ± 0.7	2.59	0.633	128	104	2.1
24	Diophantus	19.3	2.60	6.9 ± 0.9	2.80	0.643	130	78	2.8
25	Hermann	20.9	1.36	10.6 ± 1.2	1.97	0.646	115	73	3.3
26	Fracastorius B	26.1	2.29	12.6 ± 1.3	2.07	0.662	117	58	3.9
27	Fracastorius C	18.4	1.30	8.6 ± 1.1	2.14	0.682	110	82	2.8
28	Marcrobius A	19.6	3.08	7.5 ± 0.8	2.62	0.688	119	87	2.9
29	Santbech B	15.8	2.04	8.5 ± 0.9	1.86	0.692	109	96	2.5
30	Gutenberg D	20.1	1.26	12.0 ± 1.4	1.67	0.706	105	75	3.2

it can be determined that the existence of subsurface inhomogeneities at depths of 60 to 318 meters would be intolerable. The lunar subsurface models given in Table IV imply that discontinuities will pose few problems. Only four of the parabolic craters possessed central peaks, and the highest peak (Vitruvius A) is only a small percentage of its total focal height. Since all focal heights exceed the respective crater depth, it can be deduced that the central peak of a crater does not approximate the location of the focus (since peaks invariably lie below the crater rim).

The range of D/f for parabolic craters is 1.25 to 2.93, supporting the estimates employed previously for evaluative purposes. A histogram (Fig. 10) of the values of D/f exhibited by parabolic craters emphasizes the narrowness of this distribution.

The locations of the 30 parabolic craters are shown in Figure 11, as are several proposed Apollo and post-Apollo landing sites. The crater reflectors and landing sites are summarized and identified by their respective number and letter designations in Table VI.

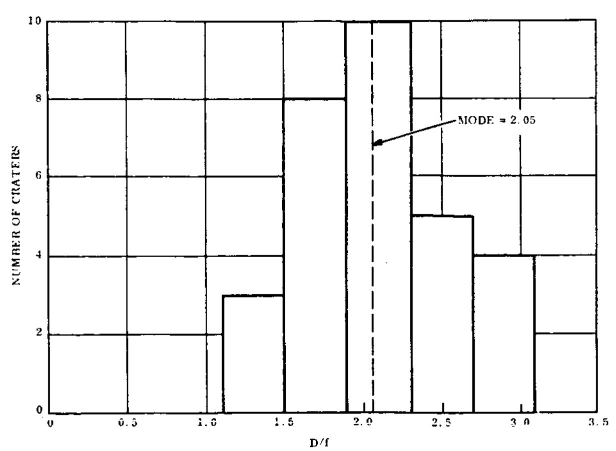

FIGURE 10. FREQUENCY DISTRIBUTION OF D/f FOR LUNAR PARABOLIC CRATERS

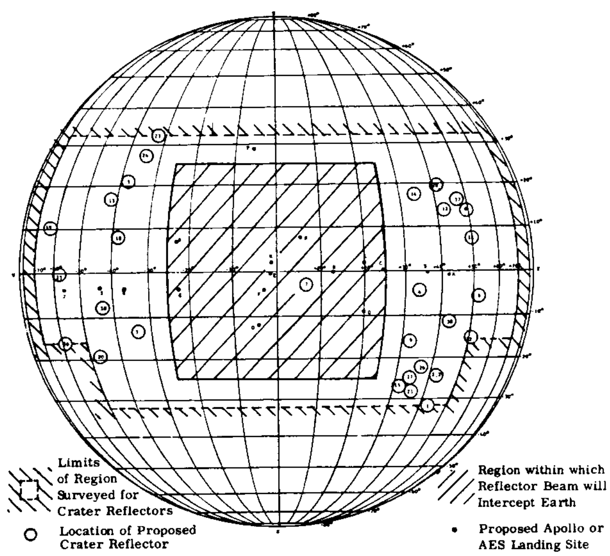

FIGURE 11. LOCATIONS OF PROPOSED LOW-FREQUENCY CRATER REFLECTORS (SITES SUMMARIZED IN TABLE I)

TABLE VI. SUMMARY OF PROPOSED CRATER REFLECTORS AND LANDING SITES
(Key Refers to Figure 11)

Crater Reflectors				Landing Sites			
Key	Name	Key	Name	Key	Location	Key	Location
1	Neander L	16	Fontana G	A	Near Lubbock	Q	North of Kant
2	Santbech A	17	Proclus	B	Between Censorinus and Maskelyne	S	Southwest of Copernicus
3	Herigonius	18	Kepler	C	Near Moltke		
4	Isodorus D	19	Gâlilaei	D	Between Theon Senior and Godin		
		20	Mersenius S				
5	Brayley	21	Neander C	E	Sinus Medii		
6	Messier G	22	McClure	F	Near Flammarion		
7	Pickering	23	Heis	G	Near Lansberg		
8	Glaisher	24	Diophantus	H	East of Flamsteed		
9	Colombo E	25	Hermann	I	North of Flamsteed		
10	Flamsteed A	26	Fracastorius B	J	Southwest O. Procellarum		
		27	Fracastorius C				
11	Taruntius A	28	Macrobius A	K	Sinus Medii		
12	Proclus A	29	Santbech B	L	Mare Tranquillitatus		
13	Bessarion A	30	Gutenberg D	N	Hyginus Rille		
14	Vitruvius A			O	Alphonsus		
15	Piccolomini D			P	South of Archimedes		

From the thirty parabolic craters, seven have been selected as the most promising candidates for further detailed analysis and early utilization. All of the major constraints have been considered: shape, location, diameter, and the useful frequencies of operation. The characteristics of these craters have been evaluated in detail and are summarized in Table VII. The average deviation from parabolic shape has been

TABLE VII. CHARACTERISTICS OF TOP SEVEN CRATERS

Physical Characteristics							
Preference	Name	Diameter (km)	Focal Height (km)	$\frac{D}{f}$	Average Deviation From Paraboloid (km)	Roundness	Thermal Anomaly
1	E. Pickering	15.3	6.3 ± 0.6	2.43	0.154	Excellent	Yes
2	Isodorus D	16.9	7.7 ± 0.5	2.20	0.136	Excellent	No
3	Flamsteed A	13.7	8.0 ± 0.7	1.71	0.173	Excellent	Yes
4	Hermann	20.9	10.6 ± 1.2	1.97	0.203	Excellent	Yes
5	Taruntius A	14.5	9.4 ± 0.7	1.54	0.171	Excellent	No
6	Santbech A	23.8	13.0 ± 0.6	1.84	0.088	Good	No
7	Kepler	30.6	14.9 ± 1.5	2.05	0.193	Poor	Yes
Electromagnetic Characteristics							
Preference	Name	Effective Diameter (km)	Minimum Gain	Minimum Beamwidth	Frequency Range (km)	Homogeneous Depths (m)	Q
1	E. Pickering	2.29	7.30	6.7°	99 - 167	112 - 236	0.78
2	Isodorus D	2.58	7.55	5.7°	90 - 180	104 - 246	1.26
3	Flamsteed A	2.17	8.15	9.9°	111 - 127	148 - 219	0.08
4	Hermann	3.25	7.84	7.1°	73 - 115	163 - 272	0.53
5	Taruntius A	2.33	8.36	9.4°	105 - 126	149 - 223	0.11
6	Santbech A	3.74	8.00	2.8°	64 - 258	73 - 285	6.70
7	Kepler	4.73	7.74	4.6°	50 - 123	152 - 290	2.15

accurately calculated using equation (37). The presence of thermal anomalies at the craters has been ascertained [31-33] as an indication of the possibility of a firm, highly reflective rock surface. The gain was calculated at the lowest frequency of operation and includes effective area losses. The figure of merit Q (equation 14) indicates the superiority of Santbech A as a prospective reflector.

The selected craters and a few of the reasons behind their choice are summarized below in order of preference.

Pickering. - The central location of this crater in conjunction with its excellent approximation of a paraboloid (ranking seventh out of all craters analyzed) makes Pickering the first choice for lunar radio astronomy studies. It is situated near the equator adjacent to four proposed Apollo landing sites at a location which would be ideal for early evaluative studies as well as for cislunar research.

Isodorus D. - This crater is situated near two early Apollo equatorial landing sites, yet it lies beyond the zone of primary lobe terrestrial interference. Thus, exploitation of reduced noise levels is possible. Isodorus D is one of the most parabolic of all craters studied, ranking fourth in the tabulation of Table V. Its highland location between the Mare Tranquillitatis and Mare Fecunditatis would encourage its exploration from the nearby Apollo landing areas at Lubbock and Maskelyne.

Flamsteed A. - Utilization of this crater would permit close approach to the limb (43° E longitude) at an accessible marial location near the Surveyor I landing site. The crater is quite symmetric and a good approximation of a paraboloid (tenth in Table V).

Hermann. - Hermann is located almost exactly on the equator, well within Apollo trajectory capabilities. Its marial location is the nearest to the limb of any of the seven selected craters (57.5°W), and its 20.9-km diameter is nearly the optimum aperture recommended for space applications. Furthermore, the 3.3 km effective diameter will yield high signal levels. The latter characteristic might be compromised to some extent by the crater's poorer approximation of a paraboloid (ranking twenty-fifth in Table V), but this feature is compensated by the lower frequencies which can be studied in view of the large diameter of the crater.

Taruntius A. - This crater is the closest to the eastern limb (50° E) of any of the selected craters and hence will provide an extension of observational capabilities in that hemisphere comparable to that provided by Hermann in the west. The marial location and proximity to the equator make the crater accessible for early missions.

Santbech A. - Although Santbech A exhibits one of the finest approximations to paraboloid shape of any of the craters analyzed, its location in the southern highlands poses some difficulty for its early utilization. However, for more advanced missions, Santbech A is certainly the most attractive crater. Although it is somewhat polygonal, the north-south and east-west scans indicate that its sides will focus incoming waves to approximately the same point. Santbech A is one of the widest of the top seven craters (23.8 km) and possesses an enormous effective diameter of 3.8 km. It might be noted that the sister crater Santbech B is rated twenty-ninth with respect to parabolic shape and might be fruitfully utilized in conjunction with Santbech A in an interferometer system.

Kepler. - The fact that this large, interesting crater can be considered parabolic is indeed surprising. It is ranked eighteenth out of the sixty craters analyzed. As a result of its large diameter (30.6 km), Kepler has a wide usable frequency range (49-125 kc) and a high effective diameter (4.7 km). But most important, Kepler is interesting in itself and thus is a likely candidate for exploration in future lunar missions. Its possible value as a radio telescope reflector thus adds to its attractiveness for AAP studies.

The major problem associated with the implementation of crater reflectors is the mounting of the feed antenna at the focus. The focal heights of 5.4 to 20.7 km characteristic of the parabolic craters of Table V pose a great obstacle to crater utilization. Unfortunately, the use of the central peak of a crater offers no solution to this problem. A Cassegrain system would provide only marginal benefits in view of the low D/f characteristic of craters and the large wavelengths to be used.

Several possible solutions can be tentatively suggested. The evaluation of smaller craters as suitable data becomes available could lead to proportionate reductions in focal height requirements. From Figure 10, the usual value of D/f for large parabolic craters is 2.05. If this relation holds true for smaller diameters, one might search for parabolic craters about twice the size of the largest feasible focal height. The use of uniquely lunar low g construction techniques might increase the latter quantity. The effects of the severe reduction in effective aperture would pose difficult sensitivity problems for small craters, however.

A more promising approach is the use of annular or asymmetric craters (Fig. 1). The focus is in an accessible location for each of these configurations, and the high slopes will result in enhanced reflective efficiencies. Identification of such craters must await more detailed lunar relief maps.

CONCLUSIONS

The foregoing analysis leads to the following conclusions regarding the advantages of a lunar crater reflector system if the feed positioning problem can be overcome:

Minimal Dependence Upon Environmental Unknowns. - Although knowledge of such surface characteristics as dielectric constant and conductivity would be important to the proper performance of many low-frequency, high resolution systems, the need for such preliminary information could be reduced by the utilization of crater reflectors. The latter have been evaluated using the poorest lunar surface conditions expected, and the results indicate that suitable crater reflectors will yield useful information even under such circumstances.

Immediate Exploitation of the Advantages of Lunar Surface Radio Astronomy. - Should the initial analyses of the lunar surface properties and conditions affecting radio astronomy indicate the feasibility of low-frequency observations, a crater reflector system would provide the opportunity for immediate, sophisticated studies exploiting the possibilities of low interference, high resolution observations at frequencies far below the terrestrial radio window. Proper crater choice guarantees a significant return of information independently of the detailed electromagnetic properties and physical nature of the lunar surface. Most importantly, there would be no delay between the initial survey and the complete exploitation of the advantages of lunar operation.

Favorable Capabilities with Respect to Advanced Systems. - The resolutions and frequency ranges afforded by crater reflector systems make them comparable with all but the most advanced techniques proposed for future missions. The addition of suitable data processing facilities could conceivably permit crater systems to evolve into instruments comparable with the most sophisticated proposed low frequency lunar aperture synthesis systems.

Utilization of Unique Lunar Features. - The moon is uniquely suited for exploitation of crater facilities: the reduced gravity will facilitate large focal structures, and the environment should be favorable for low frequency observations. Craters are probably the most common feature of the lunar surface, and there is a high probability of identifying many suitable for reflector applications.

Likewise, a number of problem areas have been identified:

Radiation Intensity. - The magnitude of the incident radiation flux at low frequencies is important to the feasibility of crater reflectors in view of their loss of effective aperture. Reduced radiation intensities could negate the advantages of the large effective diameters of crater reflectors.

Lunar Ionosphere. - The theoretical low-frequency limits imposed by the lunar ionosphere fall within or above the usable crater frequencies. This condition may make impossible astronomical observations using craters.

Surface Irregularities. - The high-frequency limitations of craters can be traced to the effects of surface irregularities, and better data on the irregularity distribution is required. Relaxation of the stringent high frequency limitations imposed by the present analysis is of great importance in view of possible lunar ionospheric limits.

Subsurface Inhomogeneities. - Although present analysis indicates the presence of no homogeneity problems at crater frequencies, the treatment has been greatly simplified. Data from future lunar landings will provide valuable verification of this constraint.

Focal Feed Antenna Placement. - The large focal heights encountered with natural crater reflectors require special consideration. This is clearly the most serious obstacle to the exploitation of crater techniques.

In summary, it may be stated that suitable parabolic crater reflectors exist and would offer significant advantages if the difficult feed placement problem can be overcome. The benefits obtainable by the successful utilization of low-frequency crater reflectors are so great that exploitation of the unique opportunities offered by lunar radio astronomy demands a thorough evaluation of the usefulness of the moon's characteristic feature.

APPENDIX A

EFFECTIVE APERTURE OF A DIELECTRIC PARABOLIC REFLECTOR

Consider the reflection of a ray from a paraboloidal surface. It is assumed that the incident waves are plane-polarized and the x-axis of the paraboloid coordinate system coincides with the $\vec{E}$ vector of the radiation so that $\vec{E}_i = E_o \hat{i}$, neglecting the time dependence. The waves are incident along the $\hat{z}$ axis (normal to the plane of Figure 12(a)) and will be reflected to the focus. For incidence at an arbitrary point on the paraboloid, the electric vector may be resolved into components parallel ($\vec{E}_{||}$) and perpendicular ($\vec{E}_{\perp}$) to the plane of incidence as shown. Note that $\vec{E}_{||}$ is always parallel to the radius vector from the focus, and $E_{\perp}$ is parallel to the direction of the azimuthal angle.

Different reflection coefficients exist for rays polarized parallel and perpendicular to the plane of incidence [34]. For perpendicular polarization, the reflection coefficient for a dielectric is

$$\Gamma_{\perp} = - \frac{\sqrt{\epsilon_r - \sin^2 \theta} - \cos \theta}{\sqrt{\epsilon_r - \sin^2 \theta} + \cos \theta} \qquad \text{(A-1)}$$

where θ is the angle of incidence and ϵ_r is the relative dielectric constant. For the case of parallel polarization, the reflection coefficient becomes

$$\Gamma_{||} = \frac{\sqrt{\epsilon_r - \sin^2 \theta} - \epsilon_r \cos \theta}{\sqrt{\epsilon_r - \sin^2 \theta} + \epsilon_r \cos \theta} \qquad \text{(A-2)}$$

The field $\vec{E}_R$ of a reflected wave is then given by

$$\vec{E}_R = \Gamma_{||} E_{||} \hat{r} + \Gamma_{\perp} E_{\perp} \hat{\varphi} \qquad \text{(A-3)}$$

Except for a phase change, the orientation of $\vec{E}_{\perp}$ is unaltered by reflection. However, the parallel component $\vec{E}_{||}$ is rotated through an angle of $+(\pi - 2\theta)$ as seen from Figure 12(b) where θ is the angle of incidence. Note that the sign of the angle of rotation

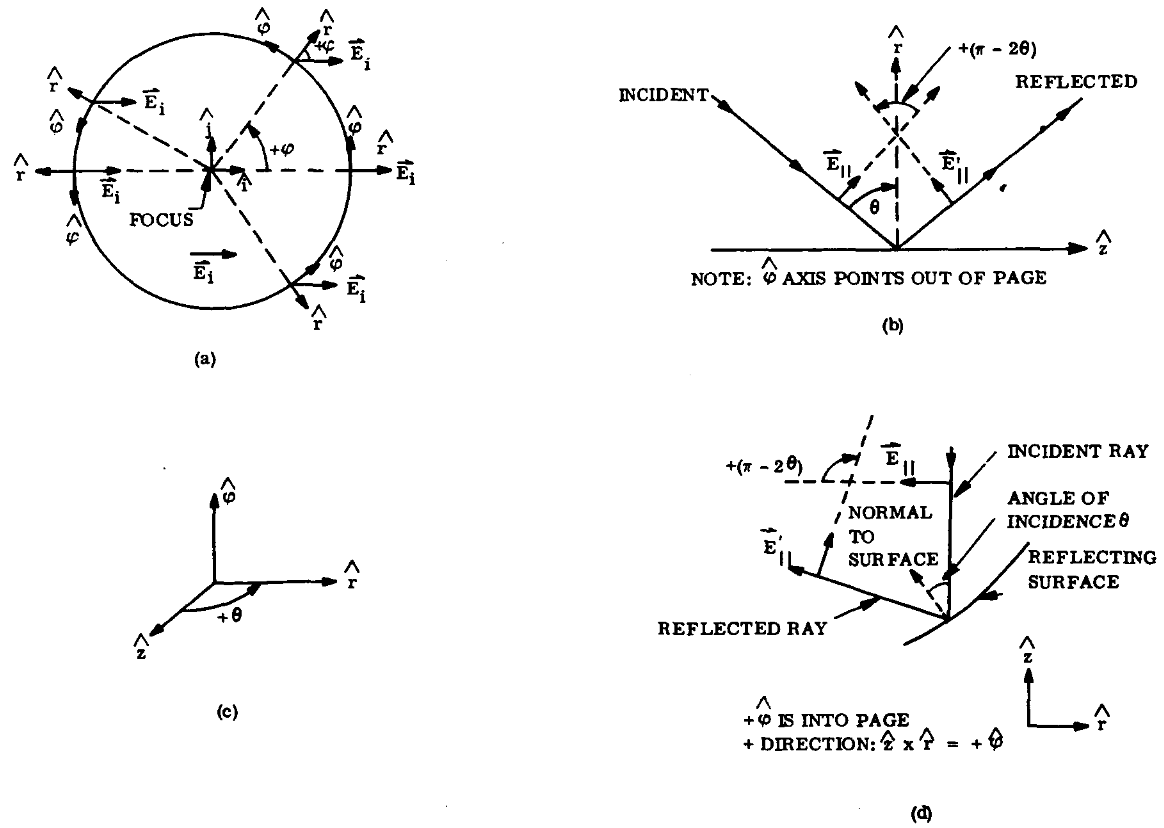

FIGURE 12. REFLECTION GEOMETRY FOR PLANE-POLARIZED RADIATION INCIDENT PARALLEL TO THE AXIS OF A PARABOLIC REFLECTOR

depends upon the direction from which one observes the reflection and the orientation of the incident ray with respect to the coordinate system. These conventions must be clearly defined.

It will be necessary to express the angle of incidence of plane waves propagating parallel to the paraboloid axis in terms of the radial distance r from the axis and the focal height f. The equation of a paraboloid with its vertex at the origin is

$$y = \frac{r^2}{4f} \tag{A-4}$$

From the geometry of Figure 12(d), the angle of incidence θ is given by

$$\tan\theta = \frac{dy}{dr} = \frac{r}{2f} \tag{A-5}$$

Thus

$$\sin\theta = \frac{r}{\sqrt{4f^2 + r^2}}, \quad \cos\theta = \frac{2f}{\sqrt{4f^2 + r^2}} \tag{A-6}$$

Since a parabolic reflector does not destroy phase relationships. the frequency dependence has been omitted from the following calculations.

Consider a small annulus of constant radius in the projected reflector aperture. By equation (A-5), the angle of incidence depends only upon r, and thus θ is constant along the annulus, as are $\Gamma_{||}$ and $\Gamma_{\perp}$. To resolve the electric vector $\vec{E}_i$ into parallel and perpendicular components at any point on the annulus, consider Figure 12(a). At each point, the resolution may be accomplished by resolving $\vec{E}_i$ into the coordinate system $(\hat{r}, \hat{\varphi})$. It is apparent from Figure 12(a) that at any point (r, φ) on the annulus, the coordinate system $(\hat{r}, \hat{\varphi})$ has been rotated by an angle φ from its position at $\varphi = 0$. Thus, to resolve the incident electric vector $\vec{E}_i$ into parallel and perpendicular components at any point φ on the annulus. one rotates the initial $(\hat{i}, \hat{j})$ coordinate system of $\vec{E}_i$ by $+\varphi$ as shown in Figure 12(a).

$$\vec{E}_i = E_o\hat{i} = \begin{pmatrix} E_o \\ 0 \end{pmatrix} \tag{A-7}$$

Resolving E_i at a point φ.

$$\begin{aligned} \vec{E}_i &= R(\varphi)\,\vec{E}_i \\ &= \begin{pmatrix} \cos\varphi & \sin\varphi \\ -\sin\varphi & \cos\varphi \end{pmatrix} \begin{pmatrix} E_o \\ 0 \end{pmatrix} \\ &= E_o \cos\varphi\,\hat{r} - E_o \sin\varphi\,\hat{\varphi} \\ &= E_{||}\,\hat{r} + E_{\perp}\,\hat{\varphi} \end{aligned} \tag{A-8}$$

where $R(\varphi)$ is the rotation matrix, and $\hat{r}$, $\hat{\varphi}$ are unit vectors in the new coordinate system. Using equation (A-3) to obtain the reflected electric amplitude, $\vec{E}'_R$

$$\vec{E}'_R = \Gamma_{||} E_o \cos\varphi\,\hat{r} - \Gamma_{\perp} E_o \sin\varphi\,\hat{\varphi} \tag{A-9}$$

As discussed in conjunction with Figure 12(b), the parallel component is rotated by $+(\pi - 2\theta)$ towards the $\hat{z}$ axis. The sign convention is shown in Figure 12(c) where a positive rotation in the $\hat{r}$ - $\hat{z}$ plane is defined so that $\hat{z} \times \hat{r} = \hat{\varphi}$. Looking towards the $+\varphi$ direction, the reflection in the plane of incidence appears as shown in Figure 12(d). But a rotation of the vector $E_{||}\,\Gamma_{||}\,\hat{r}$ by $+(\pi - 2\theta)$ is equivalent to rotating the $(\hat{z}, \hat{r})$ coordinate system by $-(\pi - 2\theta)$. Thus.

$$\begin{aligned} \vec{E}'_{||} &= R(2\theta - \pi)\;\vec{E}_{||} \\ &= \begin{pmatrix} -\cos 2\theta & -\sin 2\theta \\ \sin 2\theta & -\cos 2\theta \end{pmatrix} \begin{pmatrix} 0 \\ \Gamma_{||} E_o \cos\varphi \end{pmatrix} \\ &= -\Gamma_{||} E_o \cos\varphi \sin 2\theta\,\hat{z} - \Gamma_{||} E_o \cos\varphi \cos 2\theta\,\hat{r} \end{aligned} \tag{A-10}$$

The complete reflected wave, $\vec{E}''_R$ is thus

$$\vec{E}''_R = E''_r\,\hat{r} + E''_{\varphi}\,\hat{\varphi} + E''_z\,\hat{z} \tag{A-11}$$

It is convenient now to resolve E''_R into the initial Cartesian coordinate system by a rotation of $-\varphi$ around the $\hat{z}$ axis:

$$\vec{E}_R = R(-\varphi)\,\vec{E}''_R$$

$$= \begin{pmatrix} \cos\varphi & -\sin\varphi & 0 \\ \sin\varphi & \cos\varphi & 0 \\ 0 & 0 & 1 \end{pmatrix} \begin{pmatrix} E''_r \\ E''_\varphi \\ E''_z \end{pmatrix}$$

$$\begin{pmatrix} \cos\varphi & -\sin\varphi & 0 \\ \sin\varphi & \cos\varphi & 0 \\ 0 & 0 & 1 \end{pmatrix} \begin{pmatrix} -\Gamma_{||} E_o \cos\varphi \cos 2\theta \\ -\Gamma_{\perp} E_o \sin\varphi \\ -\Gamma_{||} E_o \cos\varphi \sin 2\theta \end{pmatrix}$$

$$= (-\Gamma_{||} E_o \cos^2\varphi \cos 2\theta + \Gamma_{\perp} E_o \sin^2\varphi)\,\hat{i}$$

$$-(\Gamma_{||} E_o \cos\varphi \sin\varphi \cos 2\theta + \Gamma_{\perp} E_o \sin\varphi \cos\varphi)\,\hat{j}$$

$$-\Gamma_{||} E_o \cos\varphi \sin 2\theta\,\hat{k} \qquad \text{(A-12)}$$

Since $\Gamma_{||}$ and $\Gamma_{\perp}$ are independent of the azimutnal angle, φ, (A-12) may be integrated over φ to yield the total electric vector existing at the focus due to a small annulus

$$\vec{E}_{R/an} = \int_o^{2\pi} \vec{E}_R \, r \, d\varphi \, dr \qquad \text{(A-13)}$$

The $\sin^2\varphi$ and $\cos^2\varphi$ terms of the $\hat{i}$ component yield the only contribution to the resultant annulus field

$$\vec{E}_{R/an} = \pi E_o \left[\Gamma_{\perp}(\theta) - \Gamma_{||}(\theta) \cos 2\theta \right] r dr\,\hat{i} \qquad \text{(A-14)}$$

The total focal field is obtained by integrating equation (A-14) over all angles of incidence, θ, or equivalently, over all values of r. Using equation (A-6), the total field at the focus $\vec{E}_R$ per unit aperture area is given by

$$\left|\vec{E}_F\right| = \frac{\int_o^R \left|\vec{E}_{R/an}\right| dr}{\int_o^R 2\pi r \, dr} =$$

$$\frac{E_o}{R^2} \int_o^R \left[\Gamma_{\perp}(r) - \Gamma_{||} \cos 2\theta (r) \right] r dr$$

$$= \frac{E_o}{R^2} \left[\int_o^R \frac{\sqrt{r^2(\epsilon_r - 1) + 4\epsilon_r f^2} - 2f}{\sqrt{r^2(\epsilon_r - 1) + 4\epsilon_r f^2} + 2f} \, r dr \right.$$

$$\left. - \int_o^R \frac{\sqrt{r^2(\epsilon_r - 1) + 4\epsilon_r f^2} - 2\epsilon_r f}{\sqrt{r^2(\epsilon_r - 1) + 4\epsilon_r f^2} + 2\epsilon_r f} \; \frac{4f^2 - r^2}{4f^2 + r^2} \, r dr \right] \qquad \text{(A-15)}$$

These integrals have been evaluated, yielding

$$\left|\vec{E}_F\right| = \frac{E_o}{X^2} \left[\frac{2\epsilon_r^2}{\epsilon_r + 1} \ln \left(\frac{\epsilon_r + Y}{(\sqrt{\epsilon_r} - 1)\, Y} \right) \right.$$

$$+ \frac{\epsilon_r - 1}{\epsilon_r + 1} \ln \left(\frac{\sqrt{\epsilon_r}\, Y - \sqrt{\epsilon_r}}{(\sqrt{\epsilon_r} - 1)\, Y} \right)$$

$$+ \frac{\epsilon_r + 3}{\epsilon_r - 1} \ln \left(\frac{\sqrt{\epsilon_r} + \sqrt{\epsilon_r}\, Y}{(1 + \sqrt{\epsilon_r})\, Y} \right)$$

$$- \frac{2(\epsilon_r^2 - \epsilon_r + 2}{\epsilon_r - 1} \ln \left(\frac{\sqrt{\epsilon_r}}{Y} \right) - \frac{2(\epsilon_r + 1)\, Y}{\epsilon_r - 1}$$

$$\left. + X^2 + 2\sqrt{\epsilon_r}\, \frac{\epsilon_r + 1}{\epsilon_r - 1} \right] \qquad \text{(A-16)}$$

where

$$X = \frac{R}{2f} = \frac{D}{4f}$$

$$Y = \sqrt{(\epsilon_r - 1)\, X^2 + \epsilon_r}$$

The ratio $(E_F/E_o)^2$ corresponds to the efficiency of the reflector (the power at the focus divided by the total incident power). Thus, the effective area A' and the effective diameter D' may be related to the actual area A and diameter D through (A-16)

$$\left(\frac{E_F}{E_o} \right)^2 = \frac{A'}{A} = \left(\frac{D'}{D} \right)^2 \qquad \text{(A-17)}$$

The above equation, in conjunction with equation (A-16), leads directly to Figures 3 and 4.

APPENDIX B

APERTURE POWER DISTRIBUTION OF A DIELECTRIC PARABOLIC REFLECTOR

Figure 13 depicts the geometry for the transmission of radiation from the focus F to the aperture r after reflection from the parabolic surface S. The

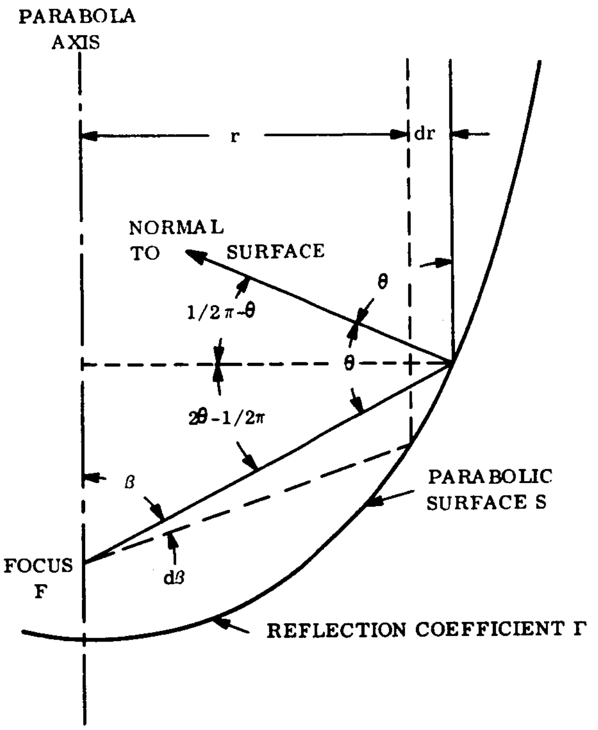

FIGURE 13. GEOMETRY OF APERTURE POWER DISTRIBUTION

surface exhibits a reflection coefficient Γ. Because of the azimuthal symmetry of the problem, only the two-dimensional radial distribution need be considered.

If the feed antenna radiates U watts per steradian, the energy at the aperture element dr is

$$P\,dr = -\Gamma^2\,U\,d\beta \qquad (B\text{-}1)$$

where P is the aperture power density (watts/unit area), Γ^2 is the power reflection coefficient, and β is defined in Figure 13. The minus sign is included since a positive increment $d\beta$ decreases r.

$$\frac{P}{U} = -\Gamma^2\frac{d\beta}{dr} = -\Gamma^2\frac{d\beta}{d\theta}\,\frac{d\theta}{dr} \qquad (B\text{-}2)$$

From the geometry of Figure 13, $\beta = \pi - 2\theta$. Thus

$$\frac{P}{U} = 2\,\Gamma^2\frac{d\theta}{dr} \qquad (B\text{-}3)$$

By equation (A-5),

$$\tan\theta = \frac{r}{2f} \qquad (B\text{-}4)$$

Differentiating (B-4),

$$\sec^2\theta\, d\theta = \frac{dr}{2f}$$

$$\frac{d\theta}{dr} = \frac{\cos^2\theta}{2f} \qquad (B\text{-}5)$$

Substitution of equations (B-5) and (A-6) into (B-3) yields the normalized power density at the aperture

$$\frac{P}{fU} = \frac{\Gamma^2}{1+\left(\frac{D}{4f}\right)^2} \qquad (B\text{-}6)$$

Equation (B-6) has been plotted in Figure 5 with $\Gamma = \Gamma_{\parallel}$ and with $\Gamma = \Gamma_{\perp}$ in Figure 6.

APPENDIX C

SURFACE IRREGULARITY CONSTRAINTS

The constraints on surface irregularities can be analyzed by use of the so-called "Rayleigh criterion" [26]. Consider the parabolic surface illustrated in Figure 14. The parabola is defined by the equation

$$x^2 = 4fy \qquad (C\text{-}1)$$

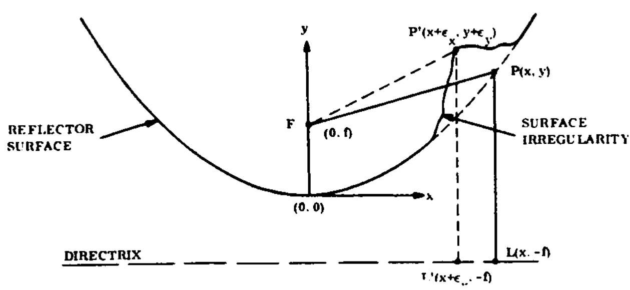

FIGURE 14. REFLECTOR SURFACE IRREGULARITIES

By definition, a parabola is the locus of points P which are equidistant from another point F (the focus) and a line (the directrix). From the geometry of Figure 14, a parabolic surface satisfies the relation

$$\overline{FP} = \min \overline{PL} \qquad (C\text{-}2)$$

Consider now a point P' which is displaced vertically by ϵ_y from the ideal parabolic surface. Applying condition (C-2) to this point (note that $\epsilon_x = 0$ in Figure 14 for this calculation)

$$\overline{FP'} = \sqrt{x^2 + (y + \epsilon_y - f)^2} = \sqrt{(y+f)^2 + 2\epsilon_y(y - f) + \epsilon_y^2} \qquad (C\text{-}3)$$

$$\overline{P'L'} = y + \epsilon_y + f \qquad (C\text{-}4)$$

We define Δ as the deviation from the parabolic condition defined by equation (C-2). Thus

$$\overline{FP'} - \overline{P'L'} = \Delta$$

$$\Delta + y + f = \sqrt{(y + f)^2 + 2\epsilon_y (y - f) + \epsilon_y^2} \qquad (C\text{-}5)$$

Dividing by (y + f)

$$\frac{\Delta}{y+f} + 1 = \sqrt{1 + \frac{2\epsilon_y(y - f) + \epsilon_y^2}{(y + f)^2}} - \frac{\epsilon_y}{y + f}$$

$$\cong 1 + \frac{\epsilon_y(y - f)}{(y + f)^2} - \frac{\epsilon_y}{y + f} \qquad (C\text{-}6)$$

where the last step involves neglecting the term ϵ_y^2 and expanding the square root under the assumption

$$\frac{2\epsilon_y(y - f)}{(y + f)^2} << 1 \qquad (C\text{-}7)$$

Simplifying equation (C-6)

$$\Delta \cong \frac{-2f\,\epsilon_y}{y + f} \qquad (C\text{-}8)$$

The phase shift, $\Delta\varphi$, contributed by the parabolic deviation Δ is

$$\varphi = \frac{2\pi}{\lambda}\,\Delta = \frac{4\pi\epsilon_y f}{\lambda(y + f)} \qquad (C\text{-}9)$$

Applying the Rayleigh criterion ($\Delta\varphi \leq \pi/4$)

$$\epsilon_y \leq \frac{\lambda}{16}\left(\frac{y\ \ f}{f}\right) = \frac{\lambda}{16}\left[\left(\frac{R}{2f}\right)^2 + 1\right] \qquad (C\text{-}10)$$

where equation (C-1) has been utilized with x = R.

Similar analysis for horizontal irregularities ϵ_x yields

$$\epsilon_x \leq \frac{\lambda}{16}\left(\frac{R}{2f} + \frac{2f}{R}\right) \qquad (C\text{-}11)$$

with the conditions

$$\epsilon_x^2 << 1, \quad \frac{\epsilon_x x}{(y + f)^2} << 1 \qquad (C\text{-}12)$$

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ABSTRACT

CURRENT CONCEPTS IN LUNAR GEOLOGY: A REVIEW

By

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The year 1966 was remarkably eventful in terms of increased knowledge of the moon because of the various successful Russian and American lunar probes. This report reviews the current status of knowledge about the lunar surface and summarizes opinion on general problems of lunar geology.

Pre-1966 concepts of the physical nature of the lunar surface have in general been confirmed, at least for mare material. Surveyor I (and recently, Surveyor III), Luna 9 and Luna 13 all landed in or on the border of Oceanus Procellarum and found it to be a relatively smooth surface with many craters and occasional blocks of material. The bulk of the mare material is fragmental, probably to a depth of several meters. Dynamic and static bearing strengths are probably in the range of 5 to 10 psi. No evidence of unusual electrostatic effects was noticed in the transmissions from the various spacecraft.

The problem of the moon's chemical and mineralogical composition remains unsolved. Gamma ray spectra transmitted by Luna 10 indicated, according to Russian scientists, a basaltic composition for both mare and terra areas. However, the many factors involved in data analysis, such as the identical gamma ray spectra of the spacecraft and the moon, make any compositional conclusions subject to revision in the light of later data. The increasing popularity of the theory that the maria are ash flows suggests a composition more silicious than basalt. It has also been suggested that some of the domical highland hills are viscous (i.e., intermediate to acidic) lava flows.

Three general scientific problems are the temperature of the moon's interior, the age of its topography and the origin of its surface features.

The first of these problems is actually a choice of models. Is the moon a relatively cold, inactive body, or is it warm, internally active, and still undergoing major geological evolution? The pendulum of opinion has swung, over the past 10 years, toward the "warm, active" model, even among those favoring an impact origin for lunar craters. During 1966, this trend was accentuated by the discovery of undoubted lunar volcanos (the Marius dome field) and verification from the Lunar Orbiter I orbital parameters that the moon is in hydrostatic equilibrium.

Estimates of the age of the moon's surface have decreased markedly in recent years. Shoemaker estimates that the mare material on the average is about half a billion years old, in contrast to his own earlier estimate of several billion years. Discovery on Lunar Orbiter high-resolution photographs of "patterned ground" indicates that some form of mass wasting may be occurring continuously because the patterned ground is not obliterated by primary and secondary impacts.

Except for the mare material, all major lunar topographic features are circular depressions or are clearly related to such depressions. The origin of craters, therefore, is central to a discussion of the origin of lunar topography in general. The pendulum of opinion has swung somewhat back toward the volcanic theory, but the consensus in the United States still is that most of the surface features are the result of meteoritic or cometary impact, although the material in general is volcanic. The mare material is generally agreed to be volcanic rock of some sort; heavily impacted basalt or volcanic ash are the two main possibilities.

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OBSERVATIONS OF CHANGES ON THE MOON

By

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Diligent search of the literature[1] has revealed a surprisingly large number of reports of lunar transient phenomena (LTP) covering the past 400 years. Two of these were naked eye observations that antedated the invention of the telescope by several decades. Initially, the intention of this paper was to report results of analyses of about 450 reported phenomena, but the collection, reduction and analyses of the data was interrupted and not completed in time. In addition, less than two weeks before this paper was to be given, a report of about 160 observations of Aristarchus covering the past 17 years was published (Bartlett, 1967). Enough information was given in the paper to enable the addition of other data to make some hasty, therefore preliminary analyses for comparison with three of the current hypotheses for the origin of these phenomena.

Before discussing the Aristarchus observations, a brief discussion of the types of phenomena that are observed in the historical reports may be of interest. From more than 550 reports, it has been found that the phenomena are actually confined to relatively few features - about 90 in all. Figure 1 shows the distribution of the 90+ features, plotted on Neison's map (1876). This distribution will be very similar to that recently published by Middlehurst and Moore (1967) because the collection of reports is similar. A careful inspection of the plot will reveal that there is a rather striking relationship to the maria. They tend to ring the maria, being found mostly around the "shores" and the shallow parts of the maria. There even is an inner ring in Mare Imbrium. There is a notable absence of LTP features in the highlands, especially the southern ones. (Top on the map is south, in keeping with the tradition of representing the moon as seen through a telescope, which inverts the image.) Actually, out of the 90+ features showing temporary anomalies, only about a dozen or so account for the large majority of the observations, with Aristarchus alone accounting for nearly half of the total of 550+. Table 1 summarizes these few features, giving the number and kind of observed anomalies. It will be noted that perhaps a majority of the anomalies imply internal activity, probably mostly of a degassing nature, particularly those reports of some kind of obscuration. Shadow anomalies probably indicate gas too. The color phenomena sometimes accompanied obscuration reports; others did not specify obscuration, and may have been caused by luminescence of gases. Other color phenomena may be from activity of surface materials. Studying and working with the data convinces one that several types of phenomena are being observed, and that all types cannot be ascribed to a single origin. Some phenomena may be produced by tidal influences, others by luminescence of gaseous or surface materials. The latter may be excited by either ultraviolet light, or, possibly, solar flare particles that have been accelerated by some mechanism that imparts higher energies than they initially had. An increase of about 10^3 is needed in order to account for the energies observed in lunar events. Some phenomena, however, may be truly of internal origin, not caused by or influenced by any external mechanisms.

If any of the phenomena result from degassing, or even possibly vulcanism, and the evidence is favorable to such an interpretation, then a comparison of their distribution with those of other features that imply lunar vulcanism may be instructive. Figure 2 shows the distribution of (1) LTP's (crosses), (2) dark, flat-floored craters, which are considered to be filled with volcanic material (filled ovals), and (3) craters that resemble terrestrial ring dikes (circles) on a map by Elger, revised by Wilkins (1958). The most striking facts about these distributions are: (1) they are similar to each other and (2) all three tend to be located peripherally to the maria, either on the shore or in the shallow parts. They imply a genetic relationship to the maria. The internal or tectonic nature of these features can be readily understood when other observational evidence of tectonic forces is considered. This is the evidence of isostasy

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[1]The author is indebted to a student aide, William Pala, for some of the research, and to the Lunar Recorders of the Association of Lunar and Planetary Observers, Mr. John Westfall and Mr. Charles Ricker for more recent observations.

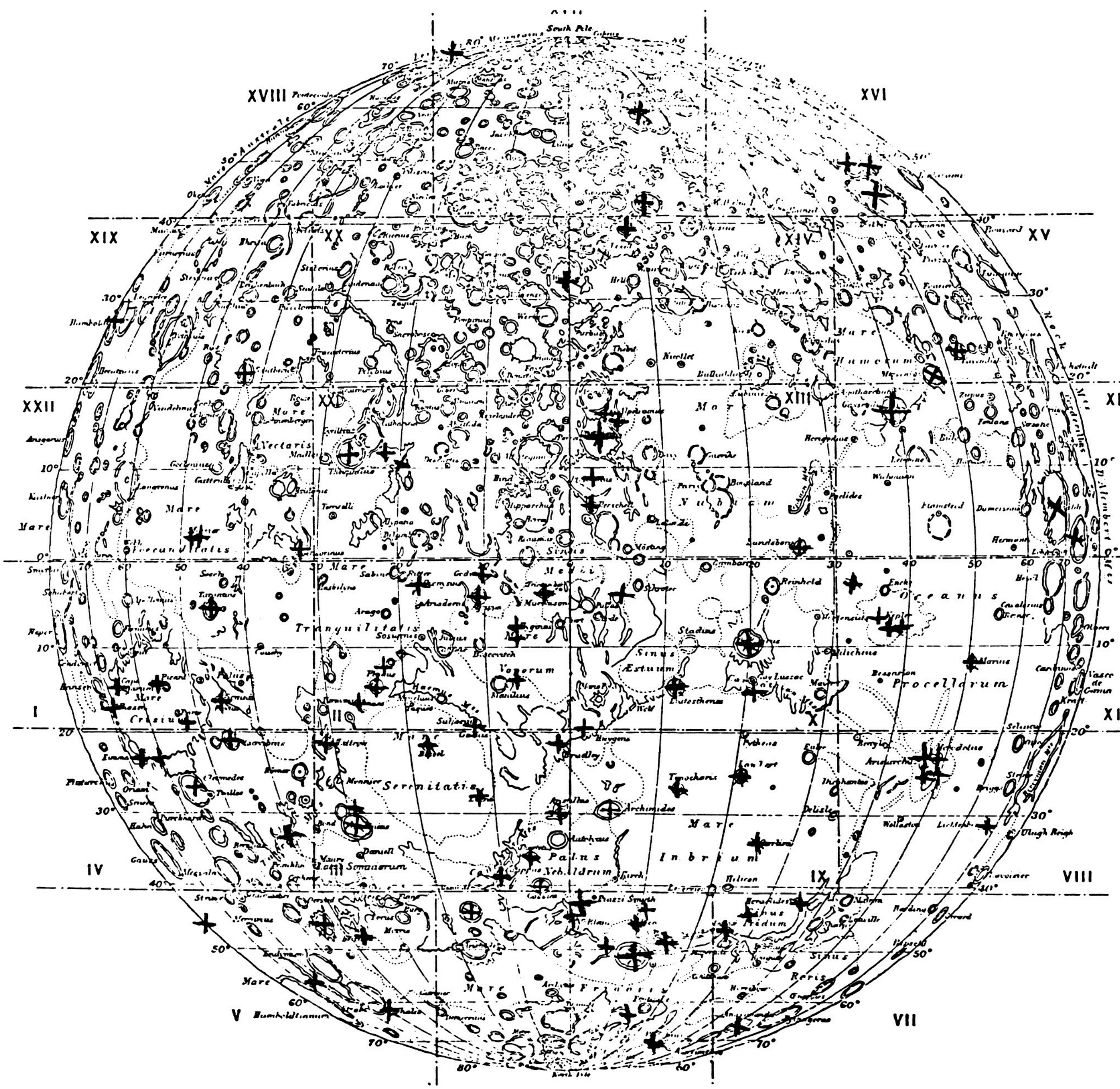


FIGURE 1. DISTRIBUTION OF THE LUNAR TRANSIENT PHENOMENA (LTP) INDICATED BY CROSSES (+) ON NEISON'S 1876 MAP OF THE MOON
(The close association with the edges of the maria is quite apparent.)

TABLE I. SUMMARY OF HISTORICAL REPORTS OF LUNAR TRANSIENT PHENOMENA IN MOST FREQUENTLY REPORTED FEATURES

Feature	Total Reports[1]	Bluish Color	Reddish Color	Obscuration	Brightening	Shadow Anomaly	Spectra	Black Spots	≤20° R	≤20° S	Anomalistic Period ≤0.1 (P) or (A): P	Anomalistic Period ≤0.1 (P) or (A): A	Full Moon $\leq \pm 4^{d}.5$	Full Moon $\leq \pm 2^{d}$
Aristarchus	234[2]	123[3]	30	12	72 (10 in eclipse)	2	4		37/215	6/215	33/142	20/142	139/215	82/215
Plato	28[4]	1	12	11	4				10/24	2/24	0/19	9/19	10/24	6/24
Alphonsus	23		14	3	3	1	3		9/23	5/23	6/22	4/22	8/23	1/23
Gassendi	12		10		1 (meteor?)			1	9/12	0/12	4/10	3/10	12/12	1/12
Schröter's Valley	12	1	7	4	6				9/11	1/11	4/11	0/11	10/11	1/11
Tycho	11			2	5	4			3/8	0/8	3/6	0/6	5/8	5/8
Eratosthenes	6	1		1	4	1			4/6	0/6	0/3	1/3	2/6	0/6
Piton	6		3	5		1			3/6	0/6	0/6	1/6	1/6	1/6
Copernicus	6[5]			1	5			1	2/6	1/6	2/4	0/4	2/6	2/6
Pico and Picob	5	1	1	4					2/5	0/5	0/3	0/3	3/5	2/5
E. of Picard[5]	5			2	3				1/4	0/4	0/2	0/2	1/4	0/4
Schickard	5			3	4				1/3	0/3	2/3	1/3	3/3	3/3
Theophilus	4	1	3	1					3/4	1/4	0/4	1/4	0/4	0/0
Littrow	4			1	1			2	0/4	0/4	0/1	0/1	3/4	2/4

[1] Some reports described more than one kind of phenomenon; e.g., reddish brightening, therefore is listed under both.

[2] 6 with no description.

[3] 108 are Bartlett's observations.

[4] 1 observation is from a drawing in 1540 of the Man in the Moon with a star in his forehead. This may have been Plato.

[5] Directions are in the IAU Convention (Astronautical).

around the maria. Nearly all the maria show evidence of subsidence — where the greatest amount of sinking occurred in the central portions. It may still be operating. Mare Humorum (Fig. 3) demonstrates this evidence of subsidence by (1) the concentric rills (faults) around it, (2) the tilted, invaded peripheral craters, whose seaward walls are lower (obliterated in many cases) than the landward walls, and (3) the cliff on the west side (LAU convention, used in all references to directions hereafter) in which at least one crater on it was distorted by the downward movement along that fracture. Thus, it is not surprising to find these features that provide evidence of internal activity located in these zones of weakness and activity around the maria. There are other features that indicate also vulcanism but have not been included here because of an incomplete survey. These include, in addition to the above-mentioned features, lunar domes, sinuous rills and dark-haloed craters. Their distributions reveal relations similar to the maria shown here for three types. Arthur (1962) shows the distribution of domes and rills.

The observations of Aristarchus by Bartlett that have just been published (1967) are a large, more homogeneous group which can be analyzed with fewer adjusting parameters to be considered than the heterogeneous observations of all features combined. All other observations of Aristarchus can be added for comparison, and significant statistics can be obtained.

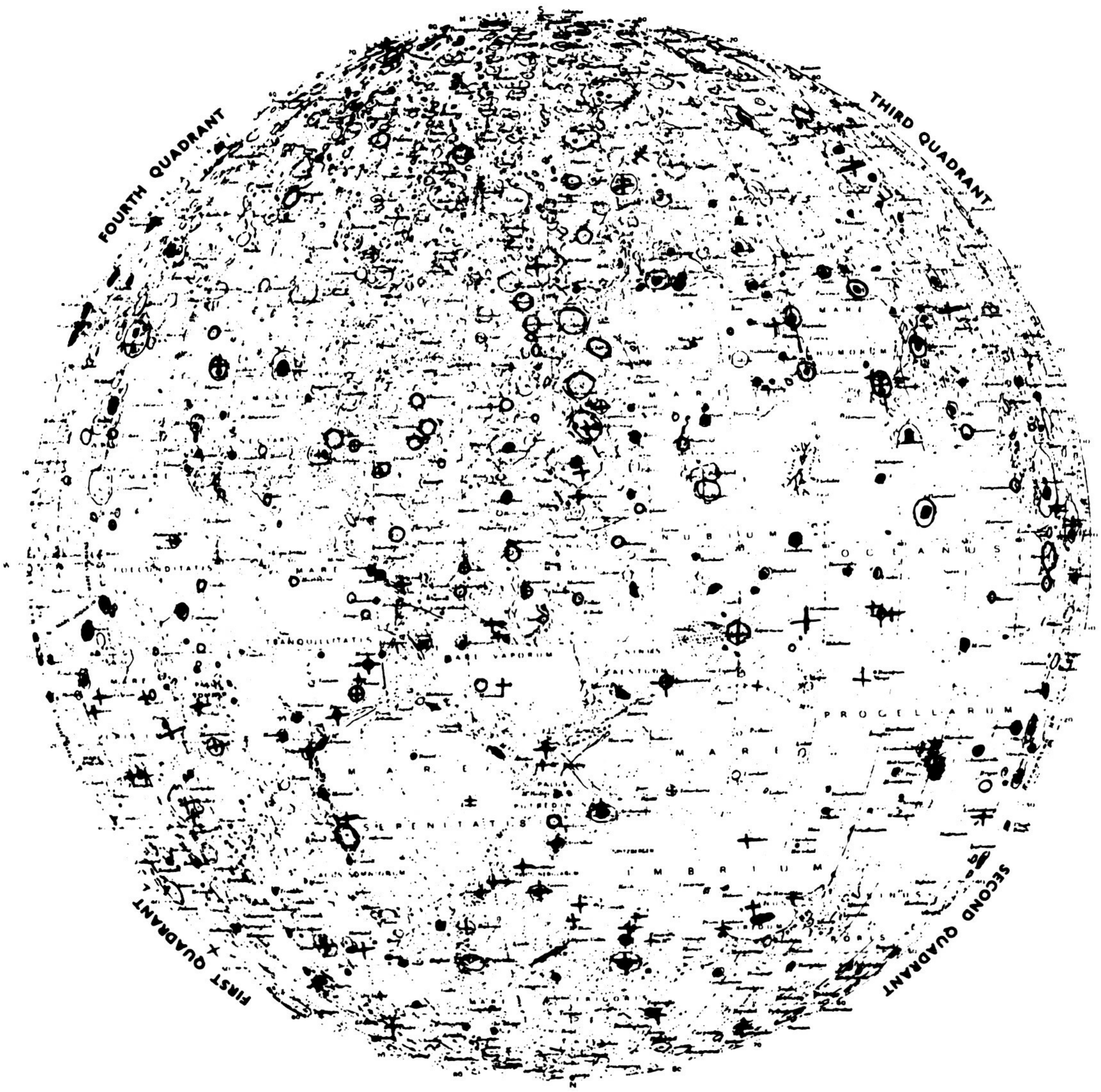


FIGURE 2. DISTRIBUTION OF THREE TYPES OF FEATURES THAT IMPLY OCCURRENCE OF VULCANISM ON MOON

(Black ovals (●) represent the dark, flat-floored craters; circles (O) represent features resembling terrestrial ring dikes; crosses (+) represent Lunar Transient Phenomena. Note the similarity of the three types of features. Sometimes all three types are found in one feature, e. g., Gassendi.)

FIGURE 3. MARE HUMORUM (120-INCH LICK PHOTOGRAPH)
(Shows evidence of isostasy by tilting of, and some invading of, peripheral craters; concentric faults (rills) indicating stress; and at least one crater on edge of western cliff (right side) indicating distortion from subsidence that occurred after formation of crater. South is at the top as seen through a telescope.)

The homogeneous factors are: (1) one feature or area observed, (2) one type of phenomenon — violet or bluish color with some variations, (3) one observer, (4) the same instruments, (5) the same observing location, and (6) all observations are during the same half period of a lunation, that is, when the feature is in sunlight. This reduces or eliminates the illumination factor, for example.

Three of the current hypotheses were considered in the analyses of the observations. These three are: (1) sunrise or low angle illumination effect, (2) Green's tidal effect (1965) and Speiser's and A. G. W. Cameron's magnetic tail effect (1966). Briefly, the three theories suggest the following effects. Greenacre's and Barr's October 1963 observations of reddish phenomena in the Aristarchus region (1965) ushered in the recent increased interest in the moon. These observations, and some subsequent ones, occurred just after sunrise, and it was suggested that gases that had escaped during the lunar night, fluoresced or luminesced upon receiving the first rays of sunlight (presumably by the ultraviolet part of the spectrum mostly). Perhaps just the low angle of illumination and longer path light made the gases visible. On this hypothesis, one would expect the phenomena to occur very near sunrise (or sunset). I have arbitrarily chosen the limits of ≤ 20 degrees from either the rising terminator (R) or the setting terminator (S). This amounts to about 1.7 days after sunrise or before sunset on Aristarchus because the terminator advances about 12 degrees per day.

Green's hypothesis is based on terrestrial logs of oil well yields and water levels that were found to be highest at times of minimum crustal stress (lunar apogees). Lunar analogy suggests that maximum lunar degassing will occur at the farthest apogees or during the period of maximum eccentricity in the lunar orbit (where the moon's orbital is most elliptical). Minimum degassing will occur at farthest perigees or during the period of minimum eccentricity (where the moon's orbit is most nearly circular). On this hypothesis, one would expect the observations to cluster around the apogee, particularly the farthest ones, and to be absent around the minimums, particularly the most distant ones.

Speiser's hypothesis suggests that the mechanism found to account for the terrestrial aurorae may also produce some of the lunar phenomena. The earth's magnetic tail, with its neutral sheet, accelerates solar particles and focuses them on small regions on the earth. It may also do this toward the moon. Perhaps the increased energies will be sufficient to produce the observed energies of the lunar phenomena. It will have efficiencies of the order of 10^3 or more over the energies of particles arriving at the moon directly from the sun, which are insufficient to produce the observed phenomena. A. G. W. Cameron proposes (private communication) that the magnetic tail's bow shock front (BSF) with its turbulence may accelerate the particles, and, if the moon is within that region at the time, this may produce the observed luminescence. Therefore, under these two hypotheses, one would expect the phenomena to occur mostly within ± 2 days of full moon to be within the magnetopause (MP) or ± 4.5 days of full moon to be within the BSF.

Table II summarizes the data for these three hypotheses. The table gives separately Bartlett's observations of Aristarchus, "All Other" observations of Aristarchus, and the two sets combined. The table lists the observed and expected fractions (numerator = total number for which there is pertinent data) and percents. There were 108 phenomena out of 163 observations by Bartlett, 107 other observations of Aristarchus of all kinds of phenomena. The expected percents and numbers were computed on the following bases. Bartlett's observing period is about 15 days; 20 degrees $\cong$ 1.7 days, so for R and S it will be 3.4 days out of 15, or 23 percent of the observations would be expected to fall within the chosen limits. Half this amount would be expected for one or the other. For the "All Other" Aristarchus observations, the phenomena occur throughout a lunation except for about two days on either side of new moon. Therefore the expectation would be 3.4 out of 25.5 (29.5-4) days $\cong$ 14 percent. Similarly, the expected percent and numbers can be computed for the other kinds of limits. For the tidal hypothesis, the arbitary limits chosen were ± 0.1 anomalistic period (perigee to perigee) from either perigee (P) or apogee (A). The average anomalistic period is about 27.6 days. The Bartlett observing period of 15 days would then include only one, either P or A. The limit of 0.1 period amounts to about 2.8 days and $\pm 2.8 \cong 5.5$ out of 15 = 37 percent expected to fall within the limits. For the "All Other" it is figured over a lunation (25.5 days) and adjusted for the ratio of the period of 25.5 to 27.6 hence the 11 out of $25.5 \times 0.92 = 40$ percent. Perigee data were available for 52 of Bartlett's and 90 of "All Others." Finally, the expected number for the magnetic tail hypothesis is computed as follows: ± 4.5 days = 9 out of 15 for Bartlett's observations; but he does not observe it till sunrise, occurring at about age 11 days, which is about 3+ days before full moon.

TABLE II. ARISTARCHUS OBSERVATIONS

Limits	Bartlett[1]				All Others				Combined			
	Observed	%	Expected	%	Observed	%	Expected	%	Observed	%	Expected	%
$\le 20°$ (R)	19/108	18%	12/108	11%	18/107	17%	7/107	7%	37/215	17%	19/215	9%
≤ 20 (S)	0/108	0%	12/108	11%	6/107	6%	7/107	7%	6/215	3%	19/215	9%
Total	19/108	18%	25/108	23%	24/107	22%	15/107	14%	43/215	20%	39/215	18%
$\le \pm 0.100$(P)	14/52	27%	19/52	37%	19/90	21%	18/90	20%	33/142	23%	30/142	21%
$\le \pm 0.100$(A)	6/52	12%	19/52	37%	14/90	16%	18/90	20%	20/142	14%	30/142	21%
Total	20/52	38%	19/52	37%	33/90	37%	36/90	40%	53/142	37%	60/142	42%
$\le \pm 4^d5$ (B. S. F.)	91/108	84%	54/108	50%	48/107	45%	37/107	35%	139/215	65%	90/215	42%
$\le \pm 2^d$ (M. P.)	57/108	53%	29/108	27%	25/107	23%	17/107	16%	82/215	38%	47/215	22%

[1]Three observations uncertain; two others had typographical errors in dates; corresponding to given colongitude were used here.

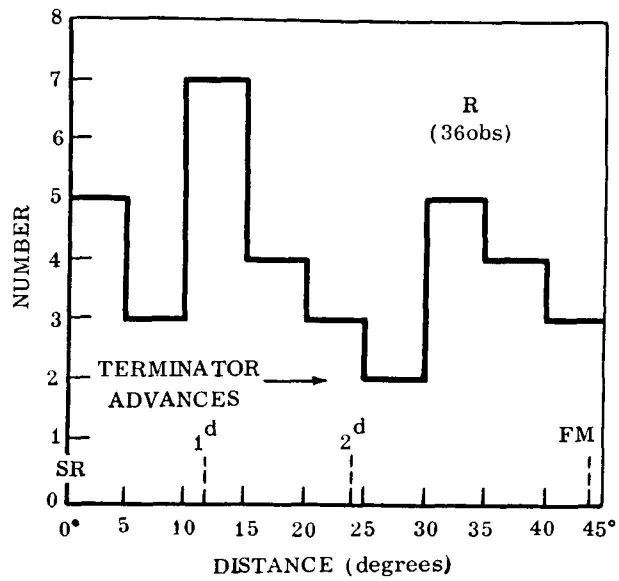
Thus at sunrise, the moon is within the 4.5 day limit, so that the fraction is really ~ 7.5 out of 15 $\cong 50$ percent. This adjustment is not necessary for the MP (± 2 days) nor to the expectation for "All Other" observations. Thus, the percents are $4/15 = 27$ percent, $9/25.5 = 45$ percent, $4/25 = 16$ percent, respectively.

Inspection of Table II reveals the following preliminary statistics: about twice as many phenomena were observed near sunrise as would be statistically expected for Bartlett's, "All Other" and "Combined" observations of Aristarchus. They comprise about 20 percent of the observed phenomena. The number of phenomena that occurred near perigee or apogee is about what would be expected, and these comprise about 40 percent of the observations. The number of phenomena that occurred within the earth's magnetic tail boundaries is about 1.5 to twice the number expected. Those falling within the BSF are about 40 percent and those within the MP are about 20 percent of the observations.

Other results are portrayed graphically. Figure 4 is a histogram of the distribution of Bartlett's observations with respect to distance from the rising (R) and setting (S) terminators. Sunrise, sunset and the phases are also indicated. There is a slight peak of observations around one day after sunrise, and a decided peak just after full moon.

Other results are portrayed graphically. Figure 5 is a histogram of the distribution of Bartlett's observations with respect to the anomalistic period, showing the positions of perigee (P) and apogee (A) in the manner of Middlehurst (1966). There are two peaks, one near perigee and the other about half way between apogee and perigee (not one-half period apart). If one adjusts the observations to acquire symmetry about perigee by averaging those numbers in the intervals of 0.0 to 0.1 and 0.9 to 1.0 (0.0) as shown by the dashed lines, the perigee peak is reduced and the other, dominant one is not significant in the hypothesis. The shape or eccentricity of the orbit should be taken into account too. Figure 6 represents the data for the tidal hypothesis for the years 1961-1966 inclusive for which orbital data were immediately available. Both Bartlett's observations (filled circles) and "All Other" observations (open squares) are plotted. There were 41 Bartlett and 46 "Other" observations in this period. Green's expected times of maximum and minimum emissions and encompassing periods are indicated. The ordinate is the distance of the moon in earth radii, and apogee (A) and perigee (P) are indicated. All observations were superposed on the orbital curve for 1964 (from Woolston, 1961). Some of 1965 is indicated by the dotted portion at the right end. Inspection of this graph shows that the observations are fairly well scattered around all orbits. Some are found at the expected apogee during expected periods of maximum degassing, but an equal number are found at the expected times of inactivity. There does not seem to be strong support for the hypothesis. Generously, about half of the observations occur somewhat near perigee or apogee (within about ±5 days which ≅ 0.2 anomalistic period, where seven days would be one-half way between). Apparently, any tidal effect on the moon is not analogous to the cause of the terrestrial oil and water levels.

Figure 7 is a histogram showing the distribution of Bartlett's observations with respect to the moon's age. The phases, sunrise, sunset and limits of the BSF and MP are also indicated. A large peak occurs



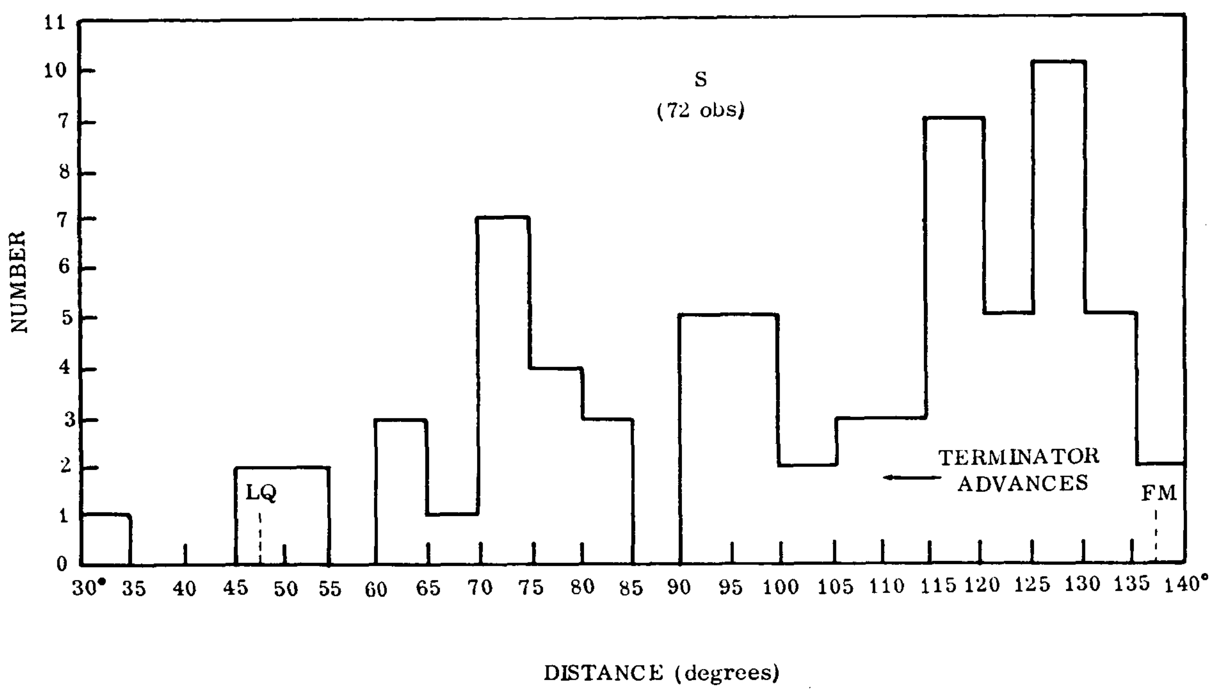


FIGURE 4. DISTRIBUTION OF BARTLETT'S OBSERVATIONS WITH RESPECT TO DISTANCE (IN DEGREE) FROM TERMINATORS

(Upper graph represents data (36 observations) from sunrise on Aristarchus through full moon; therefore, distance is measured from rising (R) terminator. Similarly, lower graph represents data (72 observations) after full moon; therefore, distances are measured from setting (S) terminator. Note strong peak in observations just after full moon -- phases indicated below abscissa. Phenomenon was not observed within 30 degrees of sunset and is a real phenomenon.)

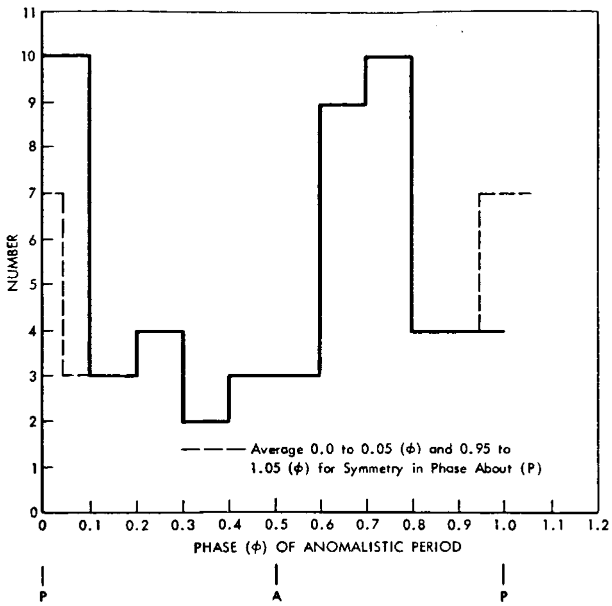


FIGURE 5. DISTRIBUTION OF BARTLETT'S OBSERVATIONS WITH RESPECT TO PHASE (ϕ) OF ANOMALISTIC PERIOD (PERIGEE TO PERIGEE) WITH PERIGEE (P) AND APOGEE (A) INDICATED (Second peak is not at or near apogee and when data are symmetric about perigee peak is reduced and histogram does not strongly indicate tidal effects.)

at age 15 to 17 days at which time the moon is within both the magnetopause and the bow shock front.

The tentative conclusions that I draw from these results are: (1) that there may be a reaction of surface materials and/or gases to the first rays of sunlight, (2) that there may also be an interaction of lunar materials with solar particles that have interacted with the earth's magnetic tail, and (3) that there is not strong evidence that Green's tidal effects operate in the manner suggested by this hypothesis. About half of the observations in the period 1961 through 1966 are found near perigee or apogee if generous allowances for departures are made. The latter results conform more nearly to the results that Cameron and Gilheany (1967) obtained from the Moon Blink project, than those obtained by Middlehurst from the historical observations of all phenomena.

The foregoing suggests that all mechanisms may be influencing the moon and that more detailed analyses and further refinements should be attempted to try to ascertain which phenomena are dominated by which mechanism. It was found that frequently two and many times the three limits overlapped. The sorting out of the influences may be very difficult, if not impossible. Some of these analyses are in progress for all the historical data too, and, when this is published, all of the data will be given in tabular form.

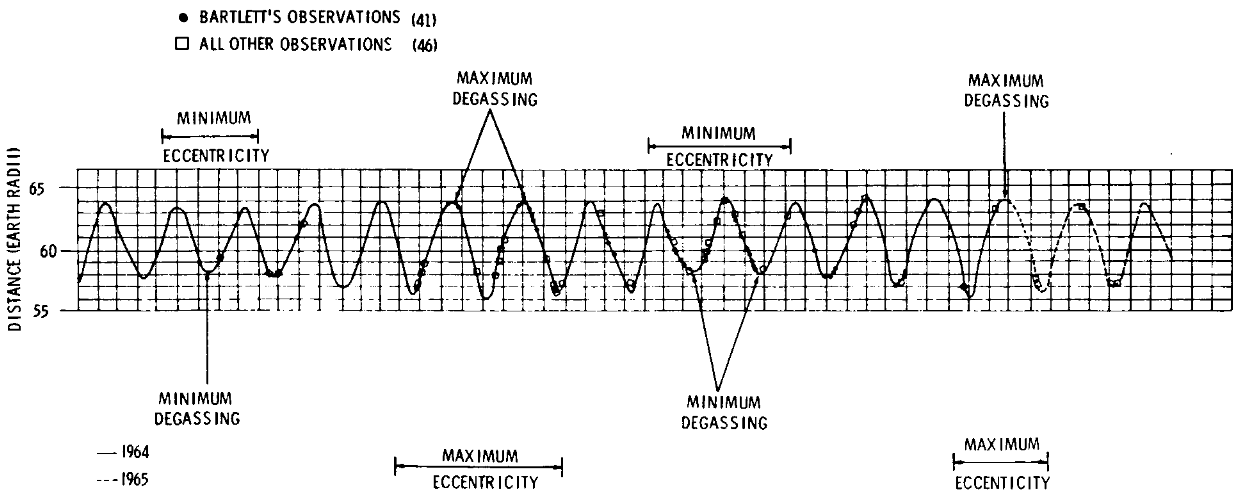


FIGURE 6. SUPERPOSITION OF ARISTARCHUS OBSERVATIONS DURING PERIOD 1961-1966 INCLUSIVE ON 1964 LUNAR ORBIT

(Portion of 1965 is represented by dashed curve at right end. Curves show variation of eccentricity of lunar orbit. Periods of maximum and minimum degassing based on Green's tidal hypothesis are indicated. Note that observations are to be found in all parts of orbit and that as many are found in expected times of inactivity as in times of expected activity. About half of the 86 total observations in this period occur near perigee or apogee if generous allowances are made.)

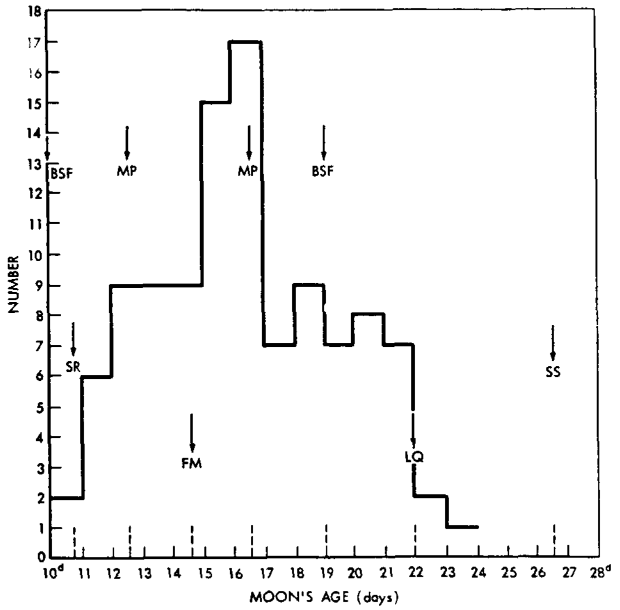


FIGURE 7. DISTRIBUTION OF OBSERVATIONS WITH RESPECT TO MOON'S AGE (IN DAYS) (Magnetic tail magnetopause (MP), bow shock front (BSF), sunrise (R), sunset (S) and phases are indicated on histogram. Note strong peak just after full moon when moon is within both magnetopause — boundary of the earth's magnetic tail — and the bow shock front of the magnetic tail.)

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ABSTRACT

SEALAB II: LESSONS FOR LUNAR BASE OPERATIONS

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Sealab II was a deep-submergence exercise carried out at a depth of 205 feet off La Jolla, California, from 28 August to 12 October 1965. It resembled certain lunar base concepts in duration, complement and other aspects. The purpose of this paper is to present conclusions about lunar base operations derived from the Sealab II experience.

The objective of Sealab II was to prove that men could live and work for extended periods at a depth of 205 feet. One crewman stayed down for 30 continuous days; others spent 15 days. The complement at any one time was 10 men. The Sealab itself was a steel cylinder 12 feet wide and 57.5 feet long, divided into four separate areas: entry, galley, laboratory and living space.

The lunar base concept with which the Sealab was compared is the Lunar Exploration Systems for Apollo I (LESA I), a modular concept designed for transportation on a Saturn V Lunar Logistics Vehicle. The basic shelter module would be 22 feet in diameter, with a living space about 10 feet high, and would be intended to support three men on the moon for three months.

Sealab II was successfully completed, with about 75 percent of the planned experimental tasks accomplished. Conclusions suggested for LESA I include the following:

1. The chief reason for establishing a lunar base - to increase the ratio of lunar stay time to travel time - was demonstrated and confirmed by Sealab II. The time necessary for decompression from the 205-foot depth was demonstrated to be 30 hours for stay times of more than a few hours. The specific reason for saturation diving sea floor bases was to increase the useful time on the bottom relative to this decompression time.

2. Base maintenance and housekeeping occupied most of the available man-hours during Sealab II; there was very little off-duty time. This suggests for lunar bases a need for (a) real-time simulations, (b) enough time in the schedule for scientific work, and (c) a reduction in the time necessary for equipment maintenance and operation.

3. Space on Sealab II was barely adequate and was an obstacle to daily operations and storage. This suggests the desirability of larger volume shelters for lunar operations than currently planned, and, in particular, the desirability of flexible (i.e., inflated) shelters rather than rigid structures.

4. Preparation for trips outside the Sealab took more than an hour. This emphasizes the need for reduction in time necessary for lunar base EVA.

5. The crew was under considerable pressure from the intentionally overfull schedule and averaged about four hours sleep per night. This indicates the need for careful planning of lunar base activities to avoid overloading the crew, especially since a LESA I crew would have to make the return trip to Earth at the end of their stay.

6. Although continuous surface support was necessary for Sealab II, operational decisions were made by the on-site commander. This suggests similar responsibility for a lunar-base commander.

7. The success of Sealab II, intentionally conducted under difficult and hazardous conditions, implies that there is nothing inherently impossible about minimum lunar bases, such as

LESA I, and strongly indicates the desirability of long stay times instead of brief excursions.

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ABSTRACT

LUNAR ORBITER PHOTOGRAPHY

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Lunar orbiter photography is one of the major activities being undertaken under the NASA lunar exploration program. This photography provides detailed information about the topography of widely separated areas on the lunar surface for use in the selection of suitable landing sites for automated Surveyor spacecraft and manned Apollo spacecraft. It also contributes to an understanding of processes responsible for the formation of the various types of features on the lunar surface.

The acquisition of detailed lunar surface photographs is the primary mission of the lunar orbiter program. This program calls for the placement of a series of five automated spacecraft into close-in orbits of the moon. Each spacecraft carries a complex photographic system consisting of a dual-lens (medium and high resolution) roll-film camera, film processor and film readout. It is the job of the spacecraft to place the photographic system in the orbit required to photograph a number of selected sites under restricted lighting conditions and to transmit the photo data in video form to earth for photo reconstruction.

To date, three lunar orbiter missions have been successfully carried out. They provided the first close-up look at candidate landing sites and features of scientific interest with resolutions at least two orders of magnitude better than achievable by earth-based telescopic photography. They have also given us a detailed look at the far side of the moon as well as the first pictures of the earth from the immediate vicinity of the moon.

Plans for the fourth lunar orbiter mission call for photographic coverage of about 85 percent of the near side of the moon at a resolution of about 65 to 90 meters. The last of the lunar orbiter missions will have detailed one-meter photographic coverage of a number of specific sites of scientific interest as its objective.

THE ANTARCTIC ANALOGY FOR LUNAR EXPLORATION

By

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SUMMARY

This paper considers the relationship between the exploration of Antarctica and the proposed program of exploration of the moon. The discussion centers on comparison of the environment, scientific missions and the role of man in exploration. Particular emphasis is placed on evaluation of the capability of man in extreme environments to participate in and support the mission objectives. Specific operational elements of locomotion, shelter, logistics and construction techniques are discussed in some detail for both Antarctica and the moon. The importance of operational capability and flexibility in assuring the success of the scientific mission is shown to be a vital element of exploration programs. A number of tables summarizing reference data are included; several conclusions derived from the analysis are presented.

INTRODUCTION

The primary use of comparing the exploration of Antarctica with exploration of the moon lies in answering questions of "how" to do it rather than "why" one should do it. Many observers of the lunar exploration scene attempt to base justification for exploration of the moon on reasons similar to those which led to exploration of Antarctica. There may be academic interest in doing this but there is little practical value. Though exploration of Antarctica has been useful in a scientific sense, the payoff from these Antarctica activities is yet to come. Exploitation in the economic sense of the word has not yet been accomplished, nor does it appear feasible at this time. Exploration of the moon will proceed in a similar manner with no prospect of an economic return at the present time.

However, by examining proposed techniques and approaches to how one might proceed to explore the moon in the light of Antarctic experience, one can derive some insight into the associated problems, together with possible solutions. For this reason it is worthwhile to consider this topic in the light of our anticipated requirements to explore the moon's surface.

When asked what made a fine explorer, Steffanson is said to have replied, "A great explorer has the minimum of adventures. In other words you think out everything that could happen to you ----- and then you are prepared for most of the things that happen to you." If this is the criterion for a great terrestrial explorer it is even more the criterion for a great lunar explorer. It is the purpose of the NASA study of the Antarctic Program to develop an insight into the things that could happen on the moon, so that lunar explorers can be prepared for them.

In developing the analogy between Antarctic exploration and lunar exploration, a major emphasis should be placed upon such specific elements as the environment, scientific missions, logistics and various subsystems such as locomotion, shelters and construction techniques. It is the purpose of this paper to examine these topics briefly in order to provide both a basis for provocative thinking and also a reference for current lunar operations studies.

SCIENTIFIC MISSIONS

Comparison of Antarctic and proposed lunar scientific missions discloses rather close agreement. In each case, the concentration of effort is on the geosciences with geology predominant. Geophysics and atmospheric physics rank next in importance, though terrestrial and marine biology are quite important in Antarctica. Though geographic and geologic investigations have always been of major importance in Antarctica, the emphasis on this discipline stems partly from the difficulty in obtaining the data. Ford [1] points out that "by no means is there yet a final geological or structural map of Antarctica, nor will there be one in the near future (and there well may never be one), since 98 percent of the bedrock is covered by glacial ice." A similar mask does not cover the lunar surface; yet environmental constraints combine to impose a difficult handicap on manned geologic exploration activities.

The total surface area of the moon can be computed as 38×10^6 km^2 and that of Antarctica as 14×10^6 km^2. Thus the near side is 1.36 times larger than Antarctica. This similarity in size relates well the problem of areal extent as one factor in determining the geology of the moon. Ford points out a significant element of beginning exploration of such large areas when he says, "Interpretations of Antarctic geology change rapidly, as field and laboratory work progress, for every new rock looked at and every new range of mountains studied makes important contributions to the geologic map of Antarctica."

Wiggins [2], in describing topographic surveys of large region, has divided them into three principal stages which he calls traveller's tales, exploratory surveys and systematic surveys. Much the same classification could be extended to all surface surveys including geologic, biologic and geomorphic--to name a few--for both the Antarctic and the moon.

Since topographic and geologic surveys will comprise a major portion of the mission-related activities of early lunar exploration, and further, since surveys of this kind have occupied scientific investigators in the Antarctic for the last half century or more, it is pertinent to describe the similarities and differences which exist between surveys conducted in these two environments on two separate, but cosmologically related bodies.

Traveller's tales include those data which are derived from areas not physically visited but seen from a distance by viewers reporting with a greater or lesser (often lesser) degree of objectivity. Pre-Ranger and Surveyor data derived from telescopic observation of the moon fall into this category, as do the hills (Flamsteed Crater) observed on the photographic mission of Surveyor I. Captain James Cook's Expedition No. 2 of 1772 to 75 had as its purpose the discovery of the Antarctic and though Cook never saw Antarctica proper, he did substantiate the existence of a vast body of land, the Continent of Antarctica at the South Pole during the circumnavigation of the Southern Ocean.

Exploratory surveys cover the entire spectrum of scientific disciplines and include maps, sketches, pictures and scientific observations made during and after travel through an area. This phase of lunar exploration will be ushered in with the first manned landing on the moon. In the Antarctic a large number of these surveys have been conducted of which the Scott, Amundsen and Byrd expeditions are the best known. The characteristic element of these exploratory surveys is the vast amount of new information obtained which, though often times inaccurate and imprecise, opens up vast new areas of scientific knowledge. Though these data lack some of the rigor of scientific discipline, later surveys and additional journeys usually refine the results of these early exploratory surveys. Operation Highjump (1947) ended the period of United States exploratory surveys.

Systematic surveys include well marked ground features acceptable for positive recovery and confident acceptance by subsequent users. Maps produced by surveys of this kind include topographic, geologic and geotectonic features and evidence upon which broad conclusions and analyses can be based. Lunar bases and stations, even those of the most temporary nature, will permit systematic surveys of the moon to be accomplished. With the British-Swedish-Norwegian Expedition of 1949 to 52 followed by IGY and Operation Deepfreeze, systematic surveys became an accomplished fact in the Antarctic. The U. S. Antarctic Research Program (USARP) dates from 1958 to the present time and reflects the combined effort by the National Science Foundation and the Department of Defense on a long-term commitment to Antarctic research. Table I compares proposed lunar exploration systems with some early Antarctic expeditions.

TABLE I. COMPARISON OF PROPOSED LUNAR EXPLORATION SYSTEMS WITH ANTARCTIC EXPEDITIONS

	ANTARCTICA	
Traveller's Tales	Cook, No. 2 (1772-75)	Ranger, 1965 Surveyor 1, 1966
Exploratory Surveys	Scott (1901-04) Byrd No. 2 (1933-35)	Apollo (1968-70)
Systematic Surveys	High Jump (1946-47) Deep Freeze (1955-Present)	AAP (1970-74) Follow On Exploration
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ROLE OF MAN

Man's presence in Antarctica has been vital to the success of all expeditions. His function has led to whatever achievement, scientific or otherwise, that can be attached to these expeditions; not only from the earliest beginnings, but also at the present time. In the case of the moon, however, sophisticated lunar landing probes and orbital spacecraft have preceded man's direct arrival, reducing the element of the unexpected to a large degree, at least as far as his capability for survival is concerned. Though man will participate directly and personally in lunar exploration, it is nonetheless true that he will be aided in his scientific activities by a wide variety of highly developed scientific instruments which will be used to supplement visual and photographic data and sample acquisition on the moon. Early polar explorers were limited by the scientific instruments then available, by what they could carry with them and by the capabilities of the individuals themselves. It must be recognized also that these early expeditions were not organized specifically for scientific purposes, much less so than Apollo for that matter. Lunar exploration must thus include orbital and remote sensor surveys of the surface and cislunar space. At the point in time that man arrives on the lunar surface he will be far more knowledgeable and informed regarding the environment and his functional tasks than were the early Antarctic explorers. This does not mean that the role assigned to man will require less education, training and preparation; indeed the exact opposite is true. Antarctic experience has taught that the trained observer is essential to the accomplishment of the scientific mission.

Antarctic research and exploration has occupied man in varying degrees of effort for the past century, with the major effort taking place in the last several decades. The lunar exploration program as visualized at this time differs in several ways from Antarctic experience. First it must be recognized that the effort is being greatly compressed in time, with the early probes and manned landings to be followed by more ambitious programs of extended exploration. Early Antarctic expeditions were concerned chiefly with keeping alive; the science and discovery aspects occupied a sub-role, which though important, had to take second place to survival. Secondly, man in the Antarctic has had to perform lower orders of work at every phase and level of the mission than is expected to be the case on the moon. This fact stems both from the availability of man and from the lack of automation of his functional tasks in Antarctica. Even today in Antarctica, with modern scientific gear, his presence is required to constantly monitor, check, and calibrate, his instruments. Such a waste of effort cannot be tolerated on the moon, at least not until such a time that economies of travel and staytime are much improved. Conservation of man's time on the moon means that the man-machine tradeoff must be carefully analyzed. His function in accomplishment of the scientific operations of lunar exploration, as in the Antarctic today, will doubtless be strongly oriented toward and concerned with non-routine operational checks and scientific observations. Much of the data obtained from Antarctic research is recorded and stored for shipment back to the universities and research laboratories supporting the program. Early lunar science will be conducted along much similar lines.

A third major comparison on how man is used in these two environments concerns the interface between the operational support function and the scientific support function. Directives of the Department of Defense and the National Science Foundation have established a working relationship in the U. S. Antarctic Program, evolved from the days of Opera-Deepfreeze and the Dufek-Gould organization, which is based upon a separation of these two functions in the accomplishment of research and exploration missions. The command structure of the DOD is based upon a commander each for the U. S. Naval Support Force, Antarctica and for the U. S. Antarctic Support Activities. The former has responsibilities for planning and implementation of logistics support, the latter for administrative and operational control of military personnel and facilities.

The National Science Foundation has responsibility for coordination and management of the integrated U. S. scientific program in the Antarctica. This is accomplished at the local level in the Antarctic by the Station Scientific Leader who directs the scientific personnal and program activities at each station. His counterpart from the DOD is the Officer-in-Charge. Day-to-day scientific activities are established by the Station Scientific Leader who calls upon the Officer-in-Charge for the required operational and logistic support.

Lunar operations, at least early in time, will be conducted by personnel who combine both the operational and scientific functions. Thus, the "science" function must be imbedded in the mission in such a way that operational control is exercised by the astronauts in the conduct of the mission simultaneously with the science role being accomplished by the same personnel. Early Antarctic exploration depended upon the support of whaling captains with little emphasis on science (the first American Geologist, James Eights, did not appear in Antarctica until 1830); it was not until the first Byrd expedition that the roles and responsibilities of the scientist became firmly established. Much the same trend can be predicted for lunar exploration, where the close relationship of the scientific and operational functions characteristic of early missions will give way to complete separation of these functions later in time. This can only take place as the logistics and personnel transportation systems to the moon are increased in capability to the point where several teams having long staytimes (6 to 12 months) can be maintained on the lunar surface.

MAN AND THE ENVIRONMENT

Table II depicts an abbreviated comparison of the environments of the moon and Antarctica. In comparing these data, it must be recognized at once that direct transfer of Antarctic experience to the lunar case is impossible and only general comparisons can be made. Certainly the environmental differences are great, yet the stress conditions are similar in many ways. Comparison of stress and energy expenditures are valid indices for assessing the effect of the environment on man.

Table III presents a comparison from Edholm [3] of the percent of time utilized in certain physical activities for both the Antarctic and lunar cases. Lunar data [4] are not conclusive, since mission plans are still being studied extensively. It can be noted, however, that the heavy work fraction is much less for the lunar case than Antarctica. In other words man's involvement will be upgraded to more sophisticated tasks where brute labor is not a requirement. The extraordinary physical accomplishments of early explorers in the Antarctic in their fight against nature will not be duplicated on the moon. Appreciation of this fact stems from the hazards of the ambient atmosphere (space); life or survival-compromising situations must be avoided. The nature of the physical exertions of polar exploration can be understood from the fact that even during sledge operations, the Antarctic explorer spent only three to 14 hours on traverse per day, though these traverses were sometimes several months or more in duration. Energy expenditures during these operations exceeded 5200 kcal per day and unless a carefully controlled diet was followed, the individual lost weight. Unregulated food consumption under these conditions can go to 9000 kcal per day. American experience in the

TABLE II. ENVIRONMENTAL COMPARISON

SURFACE TERRAIN	TOPOGRAPHY	TEMPERATURE	ATMOSPHERE
	ANTARCTICA		
Snow Ice	Rough	Cold	Reduced Pressure
-Sastrugi -Hummocks	-Crevasses -Mountain Ranges	-6 Mo. Cycle D-N -Max. -32° F to Min. -120° F	-10,000 Elevation 522 Torr = 0.7 AMT -Dry
-Bearing Capacity 1 PSI Soft Snow 5 PSI Hard Snow			
	MOON		
Soil	Rough	Cold-Hot	Vacuum
-Boulders	-Craters	-14 Day Cycle D-N	-14^{-14} Torr
-Rubble	-Rilles	-Max. +250° F to Min. -250° F	-Cosmic Rays
-Dry -Bearing Capacity (Est.) 5 PSI Soft 10 PSI Hard	-Domes -Mountains		-Meteoroids

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TABLE III. PERCENT TIME SPENT IN VARIOUS ACTIVITIES

	U. K.		ANTARCTICA	
Activity	Miners	Clerks	Station	Sledge Traverse
Lying	32	32	35	53
Sitting	34	35	40	8
Standing	7	18	6	9
Walking	11	8	8	9
Light Work	4	6	9	7
Heavy Work	11	1	2	14
Total	100	100	100	100

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Antarctic at the present time is based on a base camp food intake requirement of 3600 to 4000 kcal per day, and 4400 kcal per day for scientists during the Antarctic winter.

Metabolic requirements for man in the Apollo program are very conservative, though not unrealistic, since much of his activity is expected to be sedentary. Extrapolation of metabolic requirements obtained on Earth to spaceflight environments is not simple, since many variational effects must be considered. The effects caused by pressurized suits and reduced gravity are perhaps the most significant. The Apollo Command Module is designed for a metabolic rate of 3000 kcal/day (12,000 BTU) per man [5]. Lunar EVA is suggested as requiring a metabolic safe of 400 kcal/hour (1600 BTU) showing a strong increase over normal military walking requirements averaging about 250 kcal/hour [5]. The same reference reports on the evaluation of three different space suits during walking exercises, in both the pressurized (3.5 psig) and unpressurized modes as follows:

Type	Pressurized		Unpressurized	
	kcal/hr	BTU/hr	kcal/hr	BTU/hr
1	475	1875	200	792
2	645	2530	248	946
3	800	3150	280	1100

This tabulation shows that significant increases in metabolic energy expenditure will result from working in pressurized suits because of the restrictions to movement imposed by the suit.

Comparison of stress conditions and effects is more difficult than comparisons of energy. Synergistic effects of cold and long night cycles have been studied in the Antarctic, but not to the degree that they might have been. Results of studies made by the U. S. Navy in connection with Operation Deepfreeze indicate that diurnal rhythms are commonly altered voluntarily during the long winter night to provide eight hours sleep in two periods of four hours each. The normal pattern of eight consecutive hours is then resumed when spring returns. There is no evidence to show that diurnal rhythms are permanently modified by six month-long periods of light and darkness, thus the lunar 14 day cycle is of no consequence in this respect.

Effects of cold and types of illness, including mental and emotional, have also been studied in Antarctica. The types of illness and diseases are the same as those experienced by western man elsewhere on Earth and the incidence of any one or the other is probably not significant, though the psychology of stress under extreme environments is not well understood.

With respect to the effect of reduced gravity, it can be hypothesized that additional muscular forces will be required for certain activities to provide restoring effects normally provided by gravity. Opposing this view is the expectation that less work will be required in the reduced gravity field. The magnitude of contribution of these effects upon the individual is uncertain at the present time, though it is believed that a one-sixth gravity field produces effects much nearer to full gravity than to zero-gravity.

In concluding this discussion of man and the environment, it should be noted that little research has been accomplished in Antarctica on studying the effect of combined environmental conditions on induced stress in man. This is unfortunate since the current era of dependable rapid air transportation has eliminated the severe stress condition associated with Antarctica. Loneliness, cold, long nights and remote location all served to shape and modify the individual explorer during the period of exploration preceding World War II. At this point it is not clear to what degree these data will be useful in predicting man's behavior and response to the lunar environment.

LOGISTICS

W. von Braun and G. Woodcock have stated that "logistics and operations are the principal limiting factors in determining the scope of possible future activities on the Moon," [6]. Dufek in writing about

Operation Deepfreeze has stated that the one departmental job on the staff which would touch all other departments was logistics [7].

Logistics for Antarctic operations are characterized by the capability for resupply through air operations. This capability has made Antarctic exploration not only less hazardous, but has also acted to reduce the emphasis on life support considerations, thus permitting greater attention and effort on scientific objectives. Though exploration and scientific research and experimentation are still being conducted in the Antarctic, the period of wide utilization or exploitation has not developed. Inspection of Table IV shows that logistics requirements for Antarctic exploration varied greatly with the time period and staytime. A typical logistic data point for Byrd's second expedition according to best available records indicates that about 9.1 kg (20 lbs) per man per day of equipment and expendables were required for this operation. Conversely, sledge operations, initiating from a shipbased point of origin required about 4 kg (9 lbs) per man per day of equipment and expendables, excluding the weight of sledge and dogs. Roughly 1 kg of food per man per day (excluding water derived from snow) supported the team members.

Lunar logistics are in a slightly different category, since the problem is more one of priorities for the competing mission-related equipments in terms of weights and volumes. In other words the logistics required to support a given lunar staytime are strongly influenced by the mission requirements. This can be illustrated by comparing geological reconnaissance survey missions with radio and optical astronomy experiments, in which the mission equipments would vary considerably in weight and volume. This is one reason why it is difficult to postulate lunar logistics requirements, though an attempt is made in Table V. Note that the logistic requirements are expressed in terms of payload landed on the lunar surface and do not include the hardware elements necessary to transport the equipment from earth to lunar surface. Increasing staytime reduces the design point logistic requirements, due primarily to the fact that the ratio of expendables to permanent facilities equipment varies with duration on the surface, the proportion of expendable supplies increasing with staytime.

The National Science Foundation has provided data on the logistics requirements for an Antarctic scientific outpost, the Byrd VLF substation, in an attempt to define the logistics support required for a two-man station. The total initial material, stores and equipment provided to this station is computed as 105 500 lbs or 48 000 kg representing an estimated volume of 8000 cubic feet. These materials included three vans, two generators and considerable electronic gear, but

TABLE IV. LOGISTICS REQUIREMENTS FOR POLAR EXPEDITIONS

Expedition	Purpose	No. Men	Equipment Wt. (Metric Tons)	Staytime (Days)	KG/Man/Day
Cook, No. 2 (1772-75)	Discover Antarctica	203	750	1112	3.3
Scott (1901-04)	Explore Antarctica	43	440	530	7.8
Amundsen (1911)	Discover South Pole	6	4 Sledges 52 Dogs	93	3.9
Byrd, No. 2 (1933-35)	Explore Antarctica	115	500	473	9.1
Deep Freeze (1965-66)	Antarctic Research	1000 S 200 W	26,500	150 S 210 W	135 W/POL 35 WO/POL

S: Summer
W: Winter

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TABLE V. LOGISTICS REQUIREMENTS FOR LUNAR EXPEDITIONS

Expedition	Purpose	No. Men	Equipment (Metric Tons)	Staytime (Days)	KG/Man/Day
AAP-ALSS	Apollo Support Base	2	3.5	14	114
LESA 1	Lunar Base	3	11.4	90	42
LESA 2	Lunar Base	6	22.8	180	21
LESA 3	Lunar Base	12	45.5	360+	10.5
LESA 4	Lunar Base	18	91	720+	7

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did not include the wonderarch, support timber, arch ends, POL (petroleum, oil, lubrication) bladders (2-25 000 gallons). The initial POL supply was 50 000 gallons of DFA (diesel fuel, arctic). Resupply for the current season, 1966 to 67 came to 17 500 gallons DFA and 2.5 tons miscellaneous cargo. These figures can be manipulated to show that for two men over an assumed six month staytime and assuming 20 000 gallons of DFA consumed during this period, an average logistic support requirement of 750 pounds (340 kg) per man per day was required during the first season. Obviously the bulky and generous logistics support characteristic of Antarctic operations today cannot be used as a straight-line index to lunar operational requirements where weight and volume are severely constrained. Antarctic operations are less sensitive to volume constraints and bulk is not a major factor governing their selection. Thus a comparison of Antarctic logistics compared with lunar logistics is largely a comparison of bulky, heavy and generous supplies as opposed to compact, light and limited supplies and equipment for lunar operations.

SHELTER

In the general sense, the purposes of shelter systems is to produce a controlled environment for man's habitation and working space. To this end a large number and wide variety of shelters have been developed for Antarctica and proposed for the moon. Before examining these in any detail it should be noted that shelters can be categorized in several ways as above and below ground, fixed and portable, or preconstructed and locally constructed. In this review it is not possible to examine all of these types in great depth of detail, yet it must be recognized that many types exist and have or can have widely varying degrees of utility in either environment. Examination of the various techniques for producing a controlled environment discloses that in general, the degree of sophistication increases rapidly with increase in staytime; the major difference between the Antarctic and the moon is that this sophistication increases much more rapidly in the lunar example, at least as far as present systems seem to indicate.

The Antarctic reflects a hazardous and what might be termed an "extreme" environment, yet not to the same degree of hazard as space. For this reason simple devices such as tents, snow trenches and caves constructed of indigenous materials are very satisfactory for survival since they can be constructed to provide interior temperatures considerably above ambient without additional heating. Shelter from high-velocity winds is a great advantage because of the relief from wind cooling. Charts used by the National Science Foundation indicate that the equivalent temperature at -20° F and calm air decreases to -67° F at 20 mph and to -85° F at 40 mph. Local materials can be used on the moon but not in an emergency survival sense except possibly as protection against meteoroids. Their use in later, longer staytime missions could afford economies in logistics since sub-surface, cut-and-cover techniques could provide thermal protection and temperature control. Lining an underground cave with a plastic film would provide additional control of the air supply for limited periods. Sophisticated Antarctic shelters include a popular preconstructed "Jamesway" hut which can be erected in a minimum of time under most Antarctic conditions at any designated location. Larger pre-constructed shelters are used as base camps for logistic support purposes. Eventually it can be expected that semi-permanent shelters will be erected at these Antarctic logistics staging areas to provide long-term shelter capability. It should be noted here that early and mid-term post-Apollo exploration of the moon envisions a short staytime capability at any one location, for the primary reason that the early scientific mission experiments are not strongly time dependent. Thus long staytime shelters are not an early requirement providing the land-anywhere mission criterion is followed. In Antarctica the logistics staging area concept has been followed throughout most of the exploration phase. This approach has developed as a natural progression from the early expeditions, as a result of "implied" national areas of interest among the several nations involved and lastly as a result of a combined requirement for long-staytime bases for scientific installations and logistics staging, particularly when the Antarctic winter reduces surface transportation capability to a minimum.

Table VI presents summary data on several proposed lunar shelters having varying personnel and staytime capability. The 30 day shelter is designed to

TABLE VI. LUNAR SHELTER-LAB WEIGHTS

	Staytime - Days		
	30	180	365
Fixed Weights	Weight in Kilograms		
Structure	610	4535	5800
ECS/LS	235	1350	1350
Power Subsystem	355	4535	6350
Astrionics	160	270	270
Crew Provisions	330	1060	1060
Support/Adapter	---	1730	1730
Fluid Storage	300	----	----
Spares	---	400	400
Expendables			
Food/Water	170	1100	2200
Hydrogen/Oxygen	670	8000	16000
Hygiene/Waste	40	100	200
PLSS	245	----	-----
Scientific Equipment	1575	6650	----
Total Weight	4690	29730	35360

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be compatible with a LM descent stage and will provide shelter and mission support for three men. Its chief merit stems from its economical design from the standpoint of weight, power requirements and life support and its ability to provide considerable mission support and experimental capability. It is envisioned as a shelter which might find application during the early post-Apollo exploration period.

The six months and one year staytime shelters can meet the mission and life-support requirements for three to six men for the indicated staytime. The weights shown include allowances for both solar cells and fuel cells to produce an estimated power demand of four to six kilowatts during both the lunar day and night. Fuel cell operation is assumed to produce sufficient quantities of water for both personnel and cooling. In the operational mode expendables for the 365 day configuration would be off-loaded to be compatible with the capability of the uprated Saturn V launch vehicle configuration and a direct descent stage. Expendables would be supplied by a second unmanned flight which would deliver the mission support and scientific equipment.

CONSTRUCTION METHODS AND TECHNIQUES

Construction methods and processes in the polar regions are characterized by two major criteria; namely, utilization of pre-constructed and pre-fabricated structures and structural elements, and utilization of ice and snow where its use fills a practical need. Thus shelters, repair facilities, depots and stations are largely of the former type, whereas runways are of the latter type. Cut-and-cover shelters, already alluded to, are combinations of both of these fundamental criteria.

Construction of runways is largely a matter of grading to establish alignment and elevation compaction. The snow/ice surface has a bearing capacity suitable for large landing loads, either for wheeled aircraft or ski-equipped planes. Methods used in construction of this kind is merely an adaptation of ordinary temperate-climate techniques employing standard winterized equipment. At the present time construction in and with snow and ice has limited applications, simply because its temperature sensitivity prohibits more extensive use for more sophisticated applications.

Construction requirements on the moon have not been identified with any certainty, primarily because of the unknown character of the surface and its composition and hence uncertainty about its application to man's needs and also because of a lack of certainty regarding the need for long staytimes at a particular location. Further, the ability of man and machines to do useful work is not well understood. Construction methods will be limited by man's ability to do useful work in a spacesuit without compromising his own safety. In all likelihood his efficiency will be reduced considerably, if data presented earlier in this paper relative to his energy output are any indication. Tasks assigned to man must be carefully planned and simulated on Earth prior to attempting them on the lunar surface. Additionally they should be as few and simple as possible with maximum prefabrication and preassembly performed prior to Earth launch. Besides the doubt concerning the materials encountered and their eventual use, there is considerable doubt about the ability of construction equipment to function well in the lunar environment. Light, strong, reliable and easily maintainable equipment possessing high compatibility with both the surface and atmospheric environment on the moon are indicated [8].

Other considerations include maintenance and repair, deployment, checkout and runup of mechanized equipment. Two approaches are fundamental in this regard. One is the use of automatic equipment to checkout and pinpoint elements of equipment or subsystems needing repair or service. The other is the requirement that as much as possible of this equipment and their subsystems should be standardized, even at a cost in weight. Smith [9] has emphasized that Antarctic experience demonstrates that the successful explorer must develop and learn his operational capability in the polar environment. This may be done at a sacrifice in mission accomplishment, at least early in time. Further, he stresses that there must be a search and rescue capability and the above mentioned repair and maintenance capability. How these capabilities will be developed on the lunar surface in an early operational mission is a question of continuing complexity and even perplexity.

If, as recent data seem to suggest, the lunar surface is more closely related to terrestrial equivalents, then the solution may be to develop both a construction and a logistic staging and support capability on the moon early in the exploration period in order to assure that these objectives can be met. Certainly this will not be done until the moon has proved to be a body whose extended and expanded exploration merits such an investment.

TRANSPORTATION

The need for transportation on the moon is as evident as the need for transportation in Antarctica; though the type of transportation systems which will eventually be developed and so find useful application is much less evident in the lunar example. Certainly, the classic image of the bearded explorer trudging behind his dog team and sledge is a thing of the past in Antarctica, yet transportation in Antarctica reremains the backbone of all logistic and scientific missions.

The use of mechanized surface and air transportation systems in Antarctica began early but developed slowly at first, primarily because of slow technology growth during the first half of this century. Byrd's first Antarctic Expedition 1928 to 1930 carried both aircraft and a "Snowmobile." A number of useful flights were made in the aircraft but the "Snowmobile" traveled only 75 miles. It became clear that the conversion of an ordinary automobile to skis and tracks was not the answer to surface travel [10]. The famous "Snow Cruiser" was developed for Byrd's third (1939) expedition. It weighted 33 tons loaded, each wheel was 10 feet in diameter and exerted a contact pressure of 56 psi (3.9 kg/cm^2). Since the bearing capacity of hard snow is about 5 psi (Table II) it is apparent that the vehicle could not operate on the surface. It lies buried today, two miles from its initial starting point of travel. Subsequent to World War II, mechanization developed rapidly, starting with the wartime "Weasel" and progressing to specific Antarctic-designed vehicles. The Sno-Cat is an illustration of the kind of innovation that developed, where the flotation characteristics are enhanced through pontoon-type tracked assemblies rather than conventional tracks. These were the vehicles used by the successful Hillary-Fuchs Trans-Antarctic Expedition of 1958.

Table VII indicates that whereas locomotion and transportation systems in Antarctica are well standardized and reflect a good spectrum of capabilities, this is not as true for lunar systems. As has been mentioned the dog sledge is no longer used for practical purposes except in New Zealand. Tracked vehicles are widely used in many applications by geological field parties and for logistics purposes. Fixed wing and rotary wing aircraft are the primary mode of personnel and cargo transportation.

Proposed lunar transportation systems include not only the wheeled vehicle for surface transport, the counterpart to Antarctic tracked vehicles, but also flying vehicles to complete the spectrum of potential transportation systems. In addition, it must be recognized that the capability for orbiting the moon constitutes an additional increment of delivery not available to the Antarctic explorer. If the destination is known with certainty, the need for long surface traverses to more men and cargo is not required.

TABLE VII. COMPARISON OF LOCOMOTION TECHNIQUES

	Antarctica	Moon
Local Range 400 KM	- Dog Sledge - Tracked Vehicles - Helicopter	- Wheeled Vehicles - Flying Vehicles?
Intermediate Range 400 - 1000 KM	- Dog Sledge - Tracked Vehicles - Fixed Wing Aircraft	- Wheeled Vehicles - Flying Vehicles
Long Range 1000 KM+	- Fixed Wing Aircraft	- Wheeled Vehicles? - Ballistic Vehicles?

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Table VIII, developed from data furnished by the National Science Foundation, portrays the complete spectrum of surface and air transportation systems in use in Antarctica today. Large masses and long range are characteristic of these vehicles as well as their low induced surface loadings, on the order of one to 1.5 psi. The Weasel and smaller Sno-Cats have been largely replaced by the Nodwells, bigger Sno-Cats and the motor toboggan. The motor toboggan is the equivalent of the proposed lunar LSSM and has wide flexibility and versatility as either sole or secondary means of transportation.

Rolli-trailers can be used in conjunction with the large Sno-Cats. These trailers carry fuel in the large, low pressure rubber wheels, and bulk cargo in the body suspended between the tire-bladders. Using this combination, long traverses have been made practicable. The size, weight and cost of Sno-Cats limit their use to level terrain, but by adding the motor toboggan to the Sno-Cat all the basic requirements of field support can be met. Thus the larger Sno-Cats (743 and 843) combined with the motor toboggan (K-12) reflect essentially the same capability in Antarctica as the MOLAB or MOBEX combined with the LSSM would on the moon.

Fuchs [10] emphasizes that aircraft are an indispensable part of any Antarctic expedition today, though Byrd must be recognized as a pioneer in the practical application of aircraft in the Antarctic. As Table VIII shows, the stable of aircraft is a wide one varying from large cargo craft to small helicopters. Their speed and cross-country capability make them indispensable for reconnaissance, logistics support and

TABLE VIII. ANTARCTIC VEHICLES

Vehicle Designation	Operating Mass KG	Payload KG	Speed KM/HR	Length M	Width M	Range KM	Vehicle Use	Personnel Capacity
			Tracked					
Nodwell RN-10	1180	450	30	3.20	2.00	120	Personnel-Cargo Taxi	3
Trackmaster 4T2	2000	1100	55	3.60	2.40	220	Light Reconn-Local	2
Sno-Cat 443	1900	750	24	4.95	1.90	200	Personnel Transport 6-Man	5-8
Sno-Cat 743	3300	1000	24	6.10	2.30	320	Transport 8-Man; Living 2-Man	N/A
Sno-Cat 843	9500	2700	27	7.60	2.90	320	4-Man Laboratory	N/A
Toboggan K-12	210	-	40	2.36	0.85	10	Field Reconn-Personnel	1
			Flying					
Globemaster C-124	84 000	32 000	370	40	-	3700 Max. Cargo	Cargo Transport	Crew: 7
Hercules C-130	56 000	14 000	550	30	-	2800 Max. Cargo	Cargo Transport	Crew: 4
Constellation R7V	66 000	15 500	570	35.4	-	5800	C/P Transport	Crew: 6
Helicopter HUS-1A	6400	6100	225	11.3	-	500	C/Personnel Transport	Crew: 2
Helicopter HUL-1	1200	450	120	13.2	-	260	C/Personnel Transport	Crew: 1

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aerial mapping functions. The effects of weather, terrain and maintenance requirements make them costly to operate, however. The larger the aircraft, the greater the support requirements in terms of ships, men and money.

Helicopters are widely used in the Antarctic for reconnaissance, remote landing of survey parties, establishing geodetic control and for emergency rescue. Table VIII shows that the types in current use are limited to short ranges and light payloads.

Proposed lunar surface and flying vehicles are tabulated in Table IX, [11]. Note that the surface vehicles are wheeled, rather than tracked and that the flying vehicles are rocket powered rather than having internal combustion or turbine engines. These are design differences; the operational and functional uses are nearly similar, however, since the mission requirements for lunar exploration are quite parallel with Antarctic objectives. Thus one should not be tempted to draw comparisons between specific vehicles, but should attempt to compare the operational uses of the vehicle.

As has been pointed out earlier, there is a need on the moon for small, light one- or two-man reconnaissance vehicles such as the LSSM as well as for extended range personnel, cargo and laboratory vehicles. The need for flying vehicles has not been completely established, primarily because of the technology development necessary to assure safe, reliable operation coupled with high fuel expenditures associated with them. It can be expected, however, that lunar transportation will be a vital necessity for manned exploration, a requirement which will increase as man's scope of activities and staytimes on the moon mature. These requirements have been the object of much study both in NASA and in industry, and will continue to be studied in order to refine the requirements and functional specifications to the best degree possible prior to hardware development.

CONCLUSIONS

There is no doubt that the Antarctic is a part of the future of man. Its vast expanse has been only partially mapped, yet extensive sources of coal, uranium and other ores are known to exist. Clearly, after an investment of some 20 billion dollars, so is the moon a part of human destiny. It is not unreasonable to expect that the moon may even eclipse Antarctica in value and significance, and conversely it is reasonable to expect that the much greater cost associated with the lunar assault should produce equivalent greater benefits.

TABLE IX. PROPOSED LUNAR VEHICLES

Wheeled

Vehicle Designation	Operating Mass KG	Payload KG	Speed KM/HR	Wheel Base M	Width M	Range KM	Vehicle Use	Personnel Capacity
Go-Cart	288	10	5	1.83	1.83	12	Local Reconn-Expl.	1
LSSM	865	320	8	3.90	2.34	30	Local Reconn-Expl.	1
MOLAB - 14D	3400	320	10	5.36	4.47	400	Short Traverse	2
MOBEX - 28D	5555	700	10	7.72	5.08	800	Extended Traverse	3
MOBEX - 90D	8445	1500	10	7.72	5.08	3425	Extended Traverse	3

Flying

			Flying Time, HR	Max. Alt. KM	Propellant Mass, KG			
F1A	64	51*	0.07	7	31	8	Local Exploration-Rescue	1
F1C	309	1130	0.38	68	200	50	Local Exploration-Rescue	1
F2A	287	143	0.19	16	117	20	Intermediate Range-Rescue	2
F2C	530	204	0.42	53	292	100	Intermediate Range-Rescue	2
F3A	529	238	0.31	30	265	50	Intermediate Range-Rescue	3
F3C	2066	623	0.51	167	1485	400	Long Range-Rescue	3
F3D	4209	1160	0.59	250	3235	800	Long Range-Rescue	3

*In addition to crew and passengers

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This paper has attempted to examine these two programs in terms of their similarities and dissimilarities. These are portrayed in Table X. In addition,

TABLE X. ANTARCTIC ANALOGY

Antarctica	Moon
1. Environment	
a. Snow	a. Soil - Rock
b. Crevasses	b. Rilles - Craters
c. Reduced Pressure - Cold	c. Vacuum - Cold
d. Long Day-Night Intervals (6 mo.)	d. Long Day-Night Intervals (14 Days)
e. Dry	e. Dry
2. Locomotion	
a. Tracked Vehicles	a. Wheeled Vehicles
b. Sledges	b. Flying Vehicles
c. Aircraft	
3. Shelter	
a. Indigenous Materials - Tents	a. Pre-Constructed
b. Below Ground	b. Above Ground
c. Crude	c. Sophisticated
4. Logistics	
a. Bulky	a. Compact
b. Plentiful	b. Limited
5. Construction	
a. Simple	a. Difficult
b. Some Pre-Fab	b. All Pre-Fab
c. Capability Important	c. Not too Important
6. Early Missions	
a. Geology	a. Geology
b. Geophysics	b. Geophysics
c. Mineral Exploration	c. Geochemistry
7. Role of Man	
a. Vital	a. Designed into Mission
b. Loose Control	b. Tight Control

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the following important conclusions appear warranted:

1. The role and function of man in the mission loop is vital and important. Competition for "scientific" manhours in the total mission will be strong during early missions, and decrease as staytimes and operational flexibility increases.

2. Logistics is a persistent concern in both Antarctica and on the moon.

3. The importance of mobility as an element of exploration cannot be overstressed. Antarctic exploration has taught that as operations expand in scope and complexity, vehicle requirements become more specialized to serve more specific needs.

4. Hazardous environments demonstrate that the successful explorer must develop and learn his operational capability before reaching out too far too fast.

5. Equipment must be safe, simple and reliable and possess repair/maintenance capability.

6. A slow careful buildup of operational capability is essential--despite increased cost in doing so.

7. Antarctica is a good example of man's conquest of a large and hazardous continent. This experience can be valuable in planning lunar operations and missions since it provides a clue to the understanding of "how" to proceed, though little understanding about "why" lunar exploration should be attempted.

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UTILIZATION OF PLANETARY ATMOSPHERES FOR POWER AND PROPULSION OPERATIONS

by

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INTRODUCTION

With current rocket technology, a very small fraction (of the order of 0.001) of the mass sent from the earth will be available as useful payload and supporting equipment at the near planets, Venus and Mars. This suggests that the use of materials indigenous to the planets for power and propulsion could result in significant reductions in gross system mass launched from earth and hence could offer significant reductions in mission cost. The present study is a preliminary, exploratory examination of possible advantages derived from using the atmospheric gases of Mars or Venus as an extraterrestrial resource. Since factors such as increased operational complexity and increased total system development cost will clearly detract from the desirability of utilizing planetary atmospheres, only large predicted mass advantages, say greater than a factor of two, are likely to be classed as significant.

Our current interest in planetary atmospheres as sources of chemical energy stems from recent experiments conducted at the Jet Propulsion Laboratory by R. A. Rhein [1, 2]. Carbon dioxide and nitrogen are thought to be the major constituents of both the Mars and Venus atmospheres, although the proper proportions of the two gases are still largely undetermined [3]. Rhein measured ignition temperatures at a pressure of one bar for a wide variety of powdered metals in mixtures of CO_2 and N_2. He found ignition to occur at temperatures well below expected stoichiometric combustion temperatures for many of the powders studied. Thus, for these powders, combustion will be sustained, once initiated. From this, one can envision the prospect of simple combustors for obtaining energy from planetary atmospheres.

This prospect has stimulated interest in two distinct applications which comprise the specific subjects of this study. The first application uses atmospheric gases as oxidizer and working fluid in a gas turbine designed to produce either power for ground operations or thrust for aircraft flight. The second calls for the conversion of atmospheres to liquefied oxidizers which will supply the rocket engines of vehicles designed for leaving the planet.

ATMOSPHERIC MODELS AND FUEL SELECTION

Simple models are assumed for the characteristics of the Mars and Venus atmospheres. Because the proportions of CO_2 and N_2 are so poorly known, the two limiting compositions of pure CO_2 and pure N_2 are adopted for each planet. The surface atmospheric pressures assumed for Mars and Venus are 10 millibars and 10 bars, respectively. For both planets, the surface temperature is assumed to be 530°R. Measurements from earth-based instruments and deductions from the Mariner II data imply a surface temperature on Venus of 1260 ± 90°R [3]. However, since the present study is necessarily premised on the existence of survivable conditions, it is assumed that specific locations will be found on the Venusian surface where the temperature approximates 530°R.

The energy obtainable from the various metal powders burning in CO_2 or N_2 is displayed in Table I [2] in terms of energy per unit mass of metal consumed. For comparison the right hand column shows the energy available from selected bipropellant and monopropellant reactions in terms of total reactants consumed. The large potential advantage in specific energy afforded by burning metals with the indigenous atmospheres is evident.

Of those metals presented in Table I, beryllium releases the highest energy (11.0 kcal/gm of fuel or 3.2 kcal/gm of reactant) when burned with CO_2 and the second highest energy (5.0 kcal/gm of fuel or 2.4 kcal/gm of reactant) when burned with N_2. Boron does show greater available specific energy than beryllium when reacted with N_2. However, Rhein

TABLE I. ENERGY OBTAINABLE FROM PROPELLANTS IN ATMOSPHERES OF NITROGEN AND OF CARBON DIOXIDE

Energy (kcal/g)	CO_2 Atmosphere*	N_2 Atmosphere*	Reactions w/o Use of Atmosphere**
11.0	$Be + CO_2 \rightarrow BeO, C$		
10.0			
9.0			
8.0			
	$B + CO_2 \rightarrow B_2O_2, C$		
7.0			
	$Al + CO_2 \rightarrow Al_2O_3, C$		
6.0		$B + N_2 \rightarrow BN$	$Be + O_2 \rightarrow BeO$
			$Li + F \rightarrow LiF$
5.0		$Be + N_2 \rightarrow Be_3N_2$	
4.0			$B_2H_6 + OF_2 \rightarrow BF_3, H_2O$
3.0		$Al + N_2 \rightarrow AlN$	$H_2 + F_2 \rightarrow HF$
2.0			
			$N_2H_4 + N_{204} \rightarrow H_2O, N_2$
1.0			$N_2H_4 \rightarrow NH_3, N_2$
			$H_{202} \rightarrow H_2O + O_2$
0.0			

* Energy Given Per Gram of Metal Burned
** Energy Given Per Gram of Total Reactants

could not obtain ignition of ultra-fine boron powder in nitrogen up to the maximum temperature used in his experiments (~2100°R). He found that beryllium ignited on contact with CO_2 at room temperature and ignited with N_2 at 1940°R. On the basis of these observations, beryllium has been selected for this study to represent the possible advantages of systems burning powdered metals in the atmospheres of Mars and Venus.

UTILIZATION OF ATMOSPHERES FOR POWER OPERATIONS

A gas-turbine system is selected as a technique for producing the power or thrust. This system is then compared with alternate devices which do not make use of the atmosphere.

A simple, air-equivalent, open Brayton cycle is assumed. Air-equivalent implies that the combustion products have a negligible effect on the composition of the working fluid, i.e., the working fluid has the same composition as the inducted planetary atmosphere (air). The fuel supply for the gas turbine is finely powdered beryllium metal and the working fluid is either CO_2 or N_2. A schematic of the open Brayton cycle engine is presented in Figure 1. In this cycle, air is compressed isentropically from 1-2; part of the air is burned in a primary combustor and the products are thermally mixed with the secondary air at constant pressure from 2-3; the mixture is expanded isentropically to environmental pressure from 3-4.

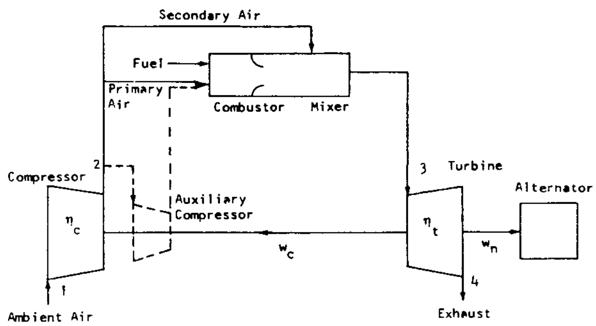

FIGURE 1. OPEN BRAYTON CYCLE

In Figure 1, w_c refers to the work required to drive the compressor and w_n refers to the work available to drive the alternator. As neither the compression nor the expansion process is isentropic, typical compressor and turbine efficiencies, $\eta_c = 0.83$ and $\eta_t = 0.85$, are assigned. The compressor pressure ratio for the secondary air (p_2/p_1) is assumed to be 7.2 and the turbine inlet temperature (T_3) is taken as 2660°R. These values are consistent with accepted design practice for modern aircraft gas-turbine engines. Lastly the alternator efficiency is taken as 0.90.

Since the ambient pressure is very low on Mars (10 millibars assumed), additional auxiliary compression of the primary air is assumed so that the combustor operates at one bar pressure, the value used in Rhein's experiments. The combustion products mix with the secondary air at the main compressor discharge pressure (p_2). The auxiliary compression is depicted by the dashed lines in Figure 1. A significant amount of additional work is needed to compress the primary air on Mars; this is reflected in higher fuel comsumption when compared to the consumption computed for operation on Venus.

The foregoing cycle model can now be combined with the assumptions to yield the following values for specific fuel consumption, i.e., beryllium consumption:

Environment	Specific Fuel Consumption (lbm/kWh)
Mars, CO_2	0.87
Mars, N_2	1.74
Venus, CO_2	0.77
Venus, N_2	1.44

It is of interest to note that, if the Venus surface temperature were 1260°R as implied by current interpretations of data, no net power could be generated under the assumption used herein.

It remains now to develop some rough estimates of the power system dry mass needed in the computation of the total mass that must be carried from earth. Simple mass scaling relationships have been derived using the assumptions listed under the discussion of size scaling appearing later in this section. The result is that the masses of the turbine, compressor and alternator are to a first order independent of atmospheric density. The combustor-mixer mass, a small part of the total, is derived to be inversely proportional to the square root of the density. Thus, since the dry mass turns out to be a small contributor to the total mass carried from earth, a satisfactory assumption is that the dry mass of the system to be used on Mars or Venus is nearly the same as that of a unit designed to yield the same power on earth. However, the size of the power generator would be very much different, as will be discussed subsequently.

The results of the preceding assumptions and analyses are displayed in Figure 2 in terms of the power produced per unit mass carried from earth, as a function of the duration of power generation. As noted in the previous paragraph, the dry system mass is not an important contributor to total mass for the operating times shown. From this it follows that the value of the power per unit mass carried is nearly inversely proportional to the operating time, and the value at any given time is inversely dependent on the specific fuel consumption. Note that two curves are shown for each planet, bracketing the possible atmospheric composition from pure N_2 to pure CO_2.

Curves for competing power systems are shown for comparison in Figure 2. The monopropellant hydrazine turboalternator system is an example of a chemical system with reactants completely supplied from earth. The specific fuel consumption assumed is 7.6 lbm/kWh. The more energetic bipropellants would show about 50 percent increase in power-to-mass ratio over the monopropellants, but even then the competing chemical systems will be at least a factor of three heavier than a beryllium-powder gas-turbine utilizing the planetary atmospheres.

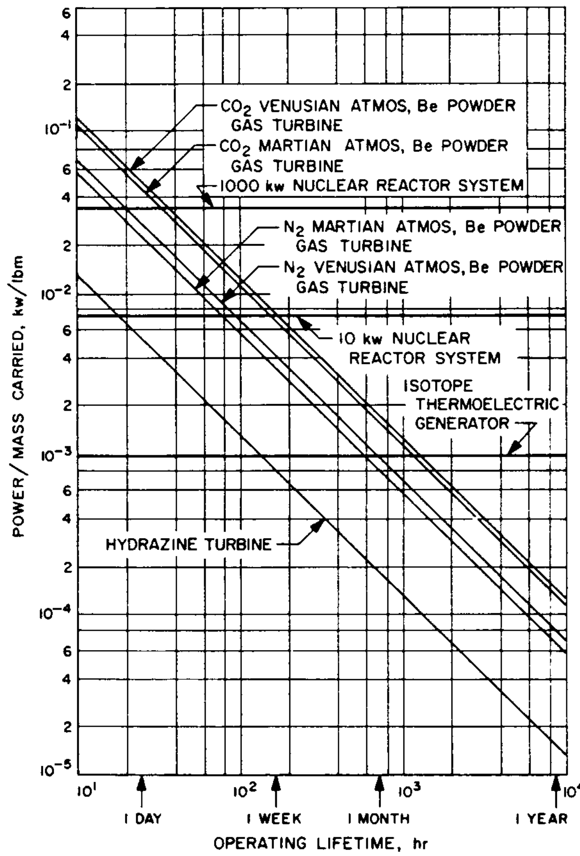

FIGURE 2. POWER PER UNIT MASS VERSUS DURATION OF POWER GENERATION

Another class of competitors is the nuclear power systems, characterized by large system mass but negligible mass consumption. The systems shown in Figure 2 are the radioisotope thermoelectric generator with a specific mass of 1000 lbm/kW, characteristic of units in the 5 to 30 watt power range [IV], and two nuclear reactor systems at 10 kW and 1000 kW output with assumed specific masses of 147 and 26 lbm/kW respectively. The 1000 kW class of power systems would most likely find use only for missions with extended operating lifetimes, measured in months; and in this circumstance, as seen in Figure 1, the nuclear systems will offer very large mass advantages over the beryllium fueled gas-turbines. However the gas-turbines show promise of significant mass savings over the 10 kW-class nuclear reactor system for lifetimes up to the order of days. Gas-turbines also show significant savings over the 0.01 to 0.1 kW-class isotope systems for lifetimes up to the order of weeks. The desire to avoid the presence of nuclear radiations may favor use of chemical systems for operating times even longer than those evident in Figure 2.

A unique problem related to the use of air breathing engines on Mars stems from the extremely low atmospheric density near the surface. This requires the design of hardware with very large dimensions, though not necessarily with large mass, as was stated previously. To gain some insight as to the dimensions required, rough scaling laws have been developed from the following assumptions:

1. Velocity, temperature and pressure-ratio profiles throughout the engine are independent of inlet air density
2. The size of solid particles generated in the primary combustor will be the same for Mars and earth (assuming use of the auxiliary compressor on Mars)
3. The number of particle collisions required for the transfer of heat from the primary combustion products to the secondary air is independent of pressure level
4. Power output is directly proportional to mass flow rate.

The resulting scaling proportionalities are then:

$$\text{Inlet Diameter} \propto \left[\frac{\text{Power}}{\text{Inlet Air Density}}\right]^{\frac{1}{2}}$$

$$\text{Combustor/Mixer Length} \propto \left[\frac{1}{\text{Inlet Air Density}}\right]^{\frac{1}{2}}$$

Since the surface density of the Martian atmosphere is assumed to be one-hundredth that of earth, the preceding relations show that a gas-turbine designed for Martian operation would be about 10 times as large in linear dimensions as an earth-based unit of the same power rating. As an example, a currently available earth-based gas-turbine engine drives a 27 kW generator and has the dimensions of about 1.5 feet by 2 feet by 2 feet. On Mars, the corresponding dimensions may become 15 feet by 20 feet by 20 feet. Thus physical size will be a major design consideration, but it does not in itself preclude the effective use of gas-turbines on Mars for power levels of the order of say 10 kW. Conversely, the high surface air densities assumed for Venus yield gas-turbine designs more compact than corresponding earth-based machines.

UTILIZATION OF ATMOSPHERES FOR ROCKET PROPULSION

The second distinct use of the planetary atmospheres considered here involves liquefaction of the constituents, N_2 and CO_2, for use as rocket oxidizer to be burned with powdered beryllium fuel. For the concept to be of any utility, the mass of oxidizer produced in usable form must exceed the mass carried from earth to produce it, i.e.,

$$R \equiv \frac{\text{mass of oxidizer produced}}{\text{mass carried from earth}} > 1.$$

If the converse were true, one might better carry all propellants from earth.

A concept that appeared attractive at first brush is that of accumulating and condensing mass from the atmosphere while in orbit about the target planet. The spacecraft would fly in an elliptical orbit with periapsis just below the outer fringes of the atmosphere. The kinetic energy and the heat of condensation of the mass collected on each pass through periapsis would be radiated to space during the remainder of the orbit. The insurmountable problem associated with the concept comes from the need to conserve momentum. In order to maintain the original orbit during each pass through the atmosphere, thrustors must produce force which exceeds the ingested momentum of the accumulated mass by the amount of the spacecraft drag.

Thus the thrust force, which is equal to the expelled mass flow rate ($\dot{m}_e$) times the effective exhaust velocity (v_e), must exceed the mass accumulation rate ($\dot{m}_a$) times the flight velocity (v_o). That is, $\dot{m}_e v_e > \dot{m}_a v_o$ or on rearranging:

$$\frac{\dot{m}_a}{\dot{m}_e} < \frac{v_e}{v_o}$$

Thus if more mass is to be accumulated than is expelled, the effective exhaust velocity must exceed the flight velocity. But the flight velocity at periapsis of orbits around Mars or Venus is of the order of 30 000 fps and the highest exhaust velocity possible for chemical systems is only on the order of 15 000 fps, making the net accumulation of mass from orbit quite impossible.

A more realistic method for mass accumulation employs a stationary condensation plant on the planet's surface. The achievable mass ratio (R) is computed based upon the simple condensation cycle illustrated in Figure 3. The cycle is made up of the following

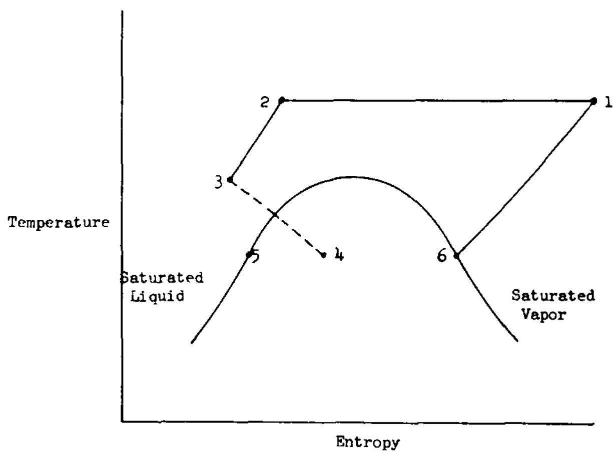

FIGURE 3. CONDENSATION CYCLE

processes: Atmospheric gases are isothermally compressed from 1 to 2. Isothermal compression is approached by the use of many compression stages with intercooling to the atmosphere between each. The gases are then cooled isobarically from 2 to 3 by the uncondensed vapor from the preceding pass, which is concomitantly heated from 6 toward ambient temperature at 1. Finally the gases undergo adiabatic expansion to ambient pressure from 3 to 4 producing condensed liquid at 5 and saturated vapor at 6.

The only power required by the cycle is used to drive the ambient gas compressor. The high pressure is taken as 40 bars for nitrogen liquefaction and 80 bars for carbon dioxide. The high pressure points are assumed the same for both Mars and Venus; thus gases must be compressed by about a factor of 1000 more for Mars than for Venus. This makes the compression work very much greater in the case of Mars, as is evident in the following table of the mass of condensate produced per unit of compression work:

Environmental Condition	Specific Condensate (lbm/kWh)
Mars, CO_2	2.4
Mars, N_2	2.7
Venus, CO_2	33.4
Venus, N_2	38.5

By combining the power system masses presented in Figure 2 with the specific condensate figures just listed, one can compute the mass of propellant produced per unit of mass carried from earth as a function of the time allowed for the accumulation of condensate. The results of such computations are presented in Figures 4 through 7 for Mars and for Venus, each with assumptions of pure N_2 and pure CO_2 atmospheres.

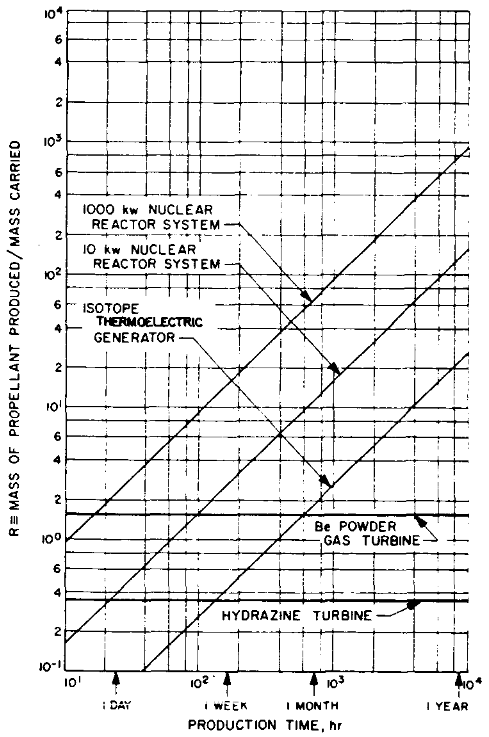

FIGURE 4. PROPELLANT PRODUCED PER UNIT OF MASS VERSUS TIME - MARS, N_2 ATMOSPHERE

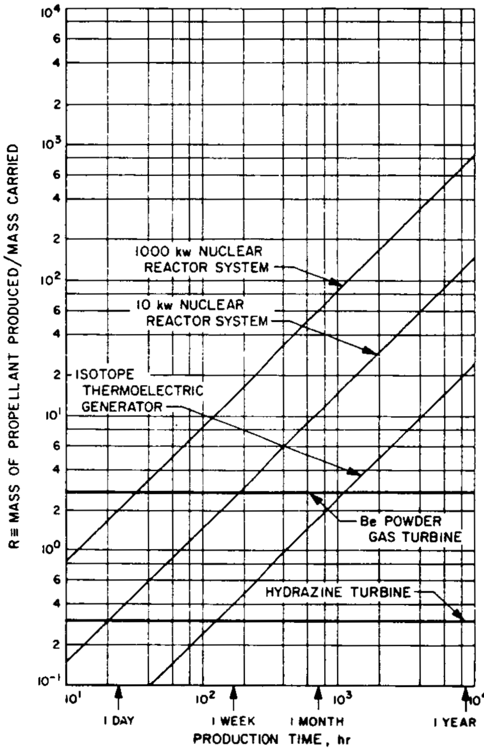

FIGURE 5. PROPELLANT PRODUCED PER UNIT OF MASS VERSUS TIME - MARS, CO_2 ATMOSPHERE

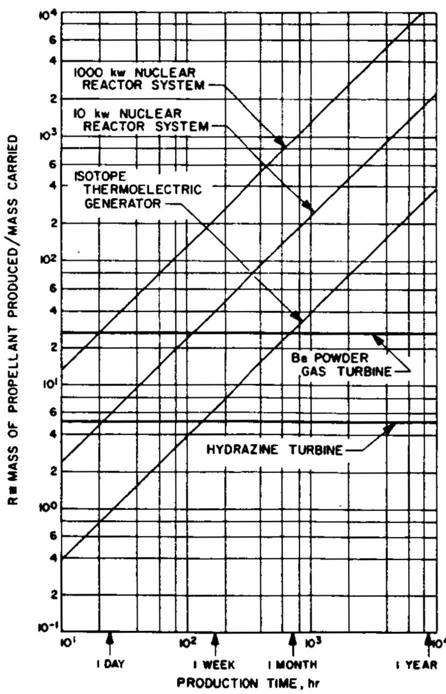

FIGURE 6. PROPELLANT PRODUCED PER UNIT OF MASS VERSUS TIME-VENUS, N_2 ATMOSPHERE

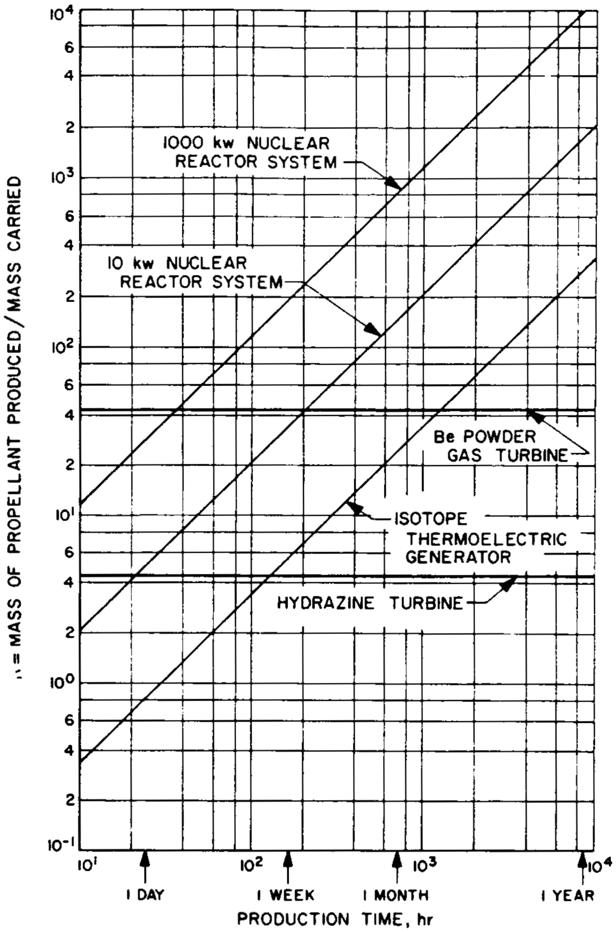

FIGURE 7. PROPELLANT PRODUCED PER UNIT OF MASS VERSUS TIME-VENUS, CO_2 ATMOSPHERE

Component masses for the compressor system have been included in the above calculations, but as with the power systems these contribute little to the total system mass. Consequently for the chemical systems, the values of R are nearly independent of the time allowed for production, reflecting mainly the mass condensed per unit mass of reactant consumed. On the other hand, the nuclear systems having fixed total mass will produce increasing quantities of condensate as production time is increased.

As mentioned previously, mass accumulation from the atmosphere can only be useful if R is greater than one; moreover, as will be seen in the subsequent discussion, R must exceed values of the order of 10 before any interest for rocket operations with beryllium powder fuel becomes evident. With this assertion in mind, one can see from Figures 4 and 5 that condensation of the Martian atmosphere by chemical power offers no interest. Also production periods on the order of one month are required with moderate-sized nuclear systems before interesting R values are developed. For the assumed Venusian conditions, high R values are readily attained with either the beryllium-powder gas-turbines or the nuclear systems (Figs. 6 and 7). On Venus, if allowable production times are less than about a week, the beryllium-fueled system would be the proper choice.

Let us now turn to the operation of a rocket system using beryllium powder as fuel and condensed CO_2 or N_2 as oxidizer. Figures 8 and 9 show values of the payload of a single stage rocket normalized by the mass carried from earth to supply the stage and its propellant. This would be the same as the usually presented payload-to-initial-mass ratio except for the fact that some of the initial mass (in the form of oxidizer produced on Mars) is "cheaper" by the factor R than if it were supplied directly from earth. Thus for $R = \infty$, the oxidizer mass is "free" in the sense that no mass is delivered from earth to obtain it -- the oxidizer is available from "gas pumps" on Mars. On the other hand, for R = 1, the results are the same as if this same oxidizer had been carried from earth with no production at Mars. The intermediate value, R = 10, is taken to be reasonably attainable. Hence the curves for R = 10 in Figures 8 and 9 represent reasonably attainable payloads.

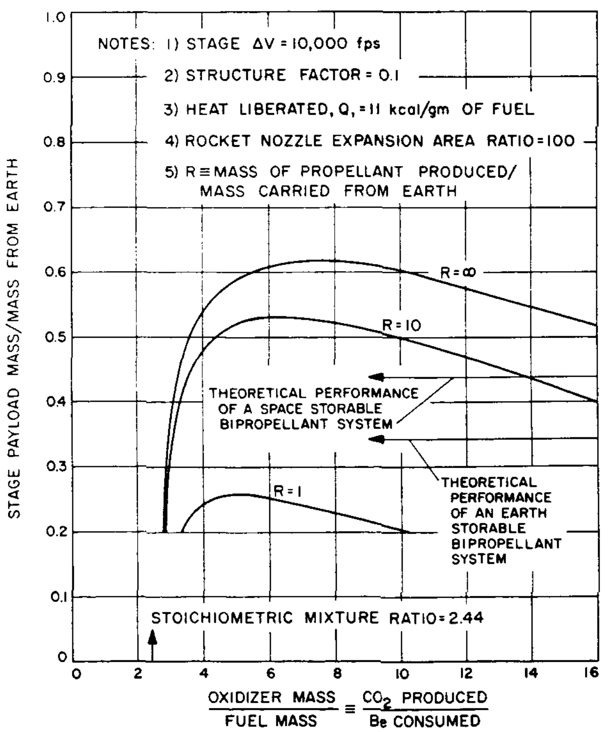

FIGURE 8. PAYLOAD RATIO AS FUNCTION OF MIXTURE RATIO FOR BERYLLIUM POWDER GAS TURBINE-*STOICHIOMETRIC MIXTURE* RATIO OF 2.44

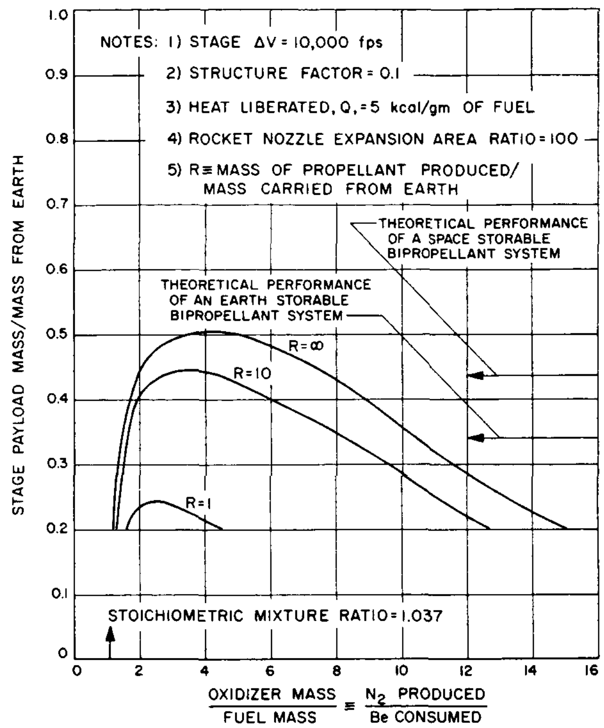

FIGURE 9. PAYLOAD RATIO AS FUNCTION OF MIXTURE RATIO FOR BERYLLIUM POWDER GAS TURBINE - STOICHIOMETRIC MIXTURE RATIO OF 1.037

The payload ratios in Figures 8 and 9 are plotted against the mixture ratio, i.e., oxidizer produced to beryllium powder carried. At stoichiometric mixture ratio, only condensed products are formed under equilibrium combustion; hence the specific impulse will be zero, there being no simple way to convert the thermal energy to kinetic energy by means of a conventional expansion nozzle. If mixture ratio is increased, excess oxidizer gas is provided which can serve as a working fluid in the expansion nozzle. Two assumptions are made to simplify the analysis of the mixture of condensed species and a working fluid: first the condensed products are in temperature and velocity equilibrium with the working gas at all points in the expansion, and second the specific heat of the condensed species is equal to the specific heat at constant pressure of the working gas. The first assumption leads to maximum attainable values of specific impulse for any mixture ratio. Checks with more complete methods show that the second assumption introduces errors of less than 10 percent in specific impulse.

Payload becomes negative when the specific impulse falls below a value necessary to deliver only the empty stage structure to the required stage velocity (here taken as 10 000 fps). Thus, for each value of R, payload goes to zero at two mixture ratios: one near stoichiometric, for the reasons just given, and the other at a large value of excess oxidizer whereas the energy per unit mass of propellant falls off. The maxima in payload seen in Figures 8 and 9 occur between these extremes.

With the above discussion in mind, we can now review the implications found in Figures 8 and 9. It can be seen that approximately a 2 to 1 reduction in the mass launched from earth is afforded by producing oxidizer on the surface when compared to bringing all propellant from earth (compare the maxima for R = 1 and R = 10). In this comparison the same propellant combination is assumed, whether the oxidizer is carried from earth or produced on the planet. However, a decision to utilize planetary atmospheres will be governed instead by comparisons with the performance of either "space-storable" (typically OF_2-B_2H_6) or "earth-storable" (typically N_2O_4-N_2H_4) bipropellants. Values of payload-to-initial-mass ratio for these classes of propellants are shown on the figures. The comparisons are made on a common basis of maximum attainable performance, but it should be expected that the portion of maximum theoretical values attainable in practice will be lower for the beryllium powder systems because of the additional loss mechanisms associated with the condensed particles in the expanding fluid.

Figure 9 shows that, if the atmospheres were found to be mostly nitrogen, there would be little gained by using condensed atmospheres when compared to the conventional use of carried space-storable propellants. Even when compared to earth-storable propellants, the Be-N_2 combination offers, at R = 10, a theoretical mass advantage of only 1.3:1.

The case for the use of condensed atmospheres looks somewhat more convincing if CO_2 is the primary constituent. Here the mass advantages, at R = 10, over conventional use of earth-storables is about 1.6:1 and over space-storables about 1.2:1.

Realization of the maximum advantages of using propellants derived from a planet's atmosphere lies in the distant future when permanent nuclear power stations are established on the surface. Then effective production times will become very large resulting in R values approaching infinity. In this case, Figure 8 shows that the combination of Be and accumulated CO_2 offers a theoretical mass advantage over the earth-storables and space-storables of 1.8:1 and 1.4:1 respectively. And Figure 9 shows that the Be and accumulated N_2 offers a mass advantage of 1.5:1 over the earth-storables and 1.2:1 over space-storables.

Even though Figures 8 and 9 are constructed on the basis of the low back pressure of the Martian environment, the comparisons among propellant systems are equally applicable for Venus. The high back pressure on Venus will reduce similarly the engine performance of the beryllium powder systems and the other propellant systems.

The mass advantages of using accumulated propellant in upper stages of the rocket system will be even less than those computed for the first stage. In fact, because of the relatively low specific impulse values derived for the beryllium powder combination, less mass delivered from earth is required with conventional propellants in the upper stages. Thus, the results presented for a first stage in Figures 8 and 9 represent essentially all that can be gained from using condensed atmospheres in rockets leaving the planet.

We see from the above discussion that under no circumstances do the predicted mass advantages of using condensed atmospheres as rocket propellants meet the arbitrary factor-of-two criterion for further interest. The closest approach to the criterion would occur with the advent of permanent nuclear-powered stations and with the discovery that atmospheric compositions are nearly pure CO_2.

CONCLUSIONS

Air-breathing power systems operating in the Martian or Venusian atmospheres promise large reductions in the mass launched from earth when compared with other chemical power systems. When compared with nuclear systems, the air-breathing systems are competitive if operational lifetimes are of the order of weeks or less. Gas-turbine power units designed for use on Mars will be on the order of 10 times larger in linear dimensions than comparable units designed for use on earth, while units designed for Venus will be smaller.

The concept of accumulating condensed atmospheric gases for use as rocket propellant in a return rocket flight from Mars or Venus may not offer sufficient mass savings to warrant the complexities involved. This conclusion is based on the use of powdered beryllium as the rocket fuel. The discovery of some other chemical reactant that releases high energy with CO_2 or N_2, that does not produce condensed products, and that can be readily transported in space may call for re-examination of this conclusion.

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THE CONTRIBUTION OF LUNAR RESOURCE PRODUCTION TO A PROGRAM OF LUNAR EXPLORATION AND SCIENCE

By

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SUMMARY

Expendable requirements associated with a projected program of lunar science and exploration have been identified and analyzed for compatibility with the utilization of lunar resource production. A possible production scheme has been suggested and its economic justification assessed. The marginal economies realized as a result of the many favorable assumptions on which this analysis is based and the sensitivity of the economics to the optimism of the estimates lead the author to suggest that currently envisioned programs of lunar science and exploration cannot be expected to utilize lunar resource production effectively.

BACKGROUND INFORMATION AND PURPOSE

The Logistics Requirements subgroup of the Working Group on Extraterrestrial Resources has devoted its energies to the question of economic justification for development of lunar resources to support space flight operations. In these studies, mission demands for candidate lunar resource items (water, oxygen, hydrogen) were postulated and analyzed. Generally, the market analysis has been optimistic and far removed from the realistically foreseeable space program to be largely incontestable as a model for resource demands.

In the recent past, the National Aeronautics and Space Administration has intensified its interest in the development of a realistic, long-range space flight plan and has conducted in-house and contracted studies toward that goal. To assess the practical role of resource production in relation to the projected space program, this report will analyze the lunar resource situation in relation to one specific, rather aggressive, lunar program option described in great depth in the Mission Modes & Systems Analysis (MIMOSA) for Lunar Exploration [1]. This study analyzes a wide variety of lunar mission candidates to assess the desirability of several program options and to select exploration and flight equipments capable of fulfilling the requirements of a postulated scientific program in a timely but evolutionary manner.

EXPLORATION AND SCIENCE PROGRAM DESCRIPTION

The MIMOSA study recommends a general plan of lunar exploration and exploitation which highlights for NASA management a number of key decision points controlling the aggressiveness of the resulting program and generating a number of program options. The most aggressive option, MIMOSA Program III, was chosen as the mission model for this analysis because it provides the environment most favorable for consideration of implementation of lunar resource production. (Under the terms of the MIMOSA contract, economies which might be achieved through utilization of lunar resources were not assessed.)

The lunar program chosen for analysis consists of four distinct operational phases of exploration and science. The program, of course, is initiated with the "landing" phase which includes the completed and current programs culminating in one or more successful manned lunar landings (Ranger, Surveyor, Orbiter, Apollo). The next phase in a progressive program has been termed the "reconnaissance" phase and is characterized by exploration equipment of the Apollo Applications Program type. During this phase, manned missions will be conducted at widely separated sites of high scientific interest. Staytimes for the two astronauts will approach 14 days, and mobility devices will extend their radius of operation considerably. In the specific program selected for study, four such mission - one per year - are shown (Fig. 1). In addition to the surface missions of interest in the analysis of lunar resource applicability, the program envisions three manned-lunar-orbiter mapping and survey missions within the same time period.

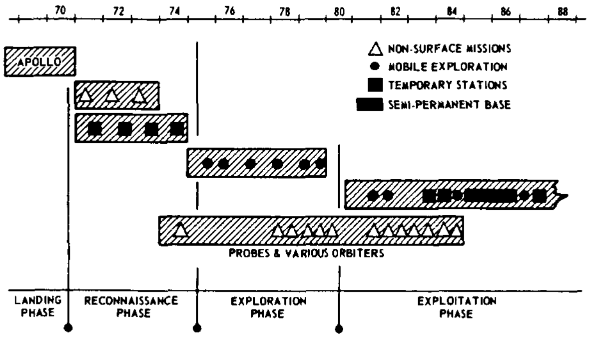


FIGURE 1. REFERENCE PROGRAM OF LUNAR EXPLORATION AND SCIENCE

The third phase of the program postulates a serious geoscientific assault aimed at an understanding of the moon and its relationship to Earth and the solar system. This, the "exploration" phase, presumes that extensive exploration of the lunar surface and its major features would be accomplished by conducting long-range surface traverses with mobile laboratories. It is expected that three astronauts would comprise the surface exploration teams, utilize highly capable roving vehicles, travel hundreds of kilometers and stay for periods on the order of 90 days. Six missions of this class are implied, as shown in Figure 1, during this phase. Supporting unmanned orbiters and probes are also considered in the referenced MIMOSA program.

The last phase postulated in the MIMOSA effort is referred to as the "exploitation" phase and envisions a scientific activity which evolves from lunar oriented experiments into a program concerned more with the exploitation of the moon as a platform in space for "outward" looking astronomical observatories, for long term biomedical research, for geochemical and materials research laboratories, and as a base for space flight technology development. Included witnin the time-confines of this era are four missions of the "exploration" type charged with the additional objective of providing detailed investigations of the subsurface lunar "crust" through deep drilling and extended active-seismic surveys. Three temporary laboratories are scheduled with a crew of six scientist-astronauts. These temporary stations (3 to 6 months) begin the astronomical work with medium sized telescopes and establish the basis for the culminating mission in this plan: the extended semi-permanent lunar base. This particular plan considers a 12-man station which is maintained for a period approaching two years through logistic resupply and crew rotation. Table I details the crew size, staytime, and traverse length for each of the above missions.

TABLE I. POST-APOLLO MISSION SUMMARY MANNED LUNAR SURFACE MISSIONS

	Year	Crew	Staytime (Days)	Traverse (km)
Reconnaissance Phase				
1 AAP Mission	1971	2	13	195
2 AAP Mission	1972	2	13	195
3 AAP Mission	1973	2	13	195
4 AAP Mission	1974	2	13	195
Exploration Phase				
1 Mobile Exploration	1975	3	13	80
2 Mobile Exploration	1976	3	13	80
3 Mobile Exploration	1977	3	33	400
4 Mobile Exploration	1978	3	82	500
5 Mobile Exploration	1979	3	67	900
6 Mobile Exploration	1979	3	74	1600
Exploitation				
1 Mobile Exploration	1981	3	103	1800
2 Mobile Exploration	1982	3	65	1200
3 Temporary Station	1983	6	94	2000
4 Temporary Station	1984	6	178	1000
5 Mobile Exploration	1984	3	57	1200
6 Semi-Permanent Base	1985	12	660	240
7 Mobile Exploration	1986	3	23	800
8 Temporary Station	1987	6	178	2100

It is important, for purposes of this analysis, to delineate the specific technology assumptions, implied in the mission model, which influence the decision on resource production. The mobile exploration vehicle is considered to be powered by an oxygen-hydrogen fuel-cell. Life support systems utilized in this vehicle represents an open ecological approach. Atmosphere selected is a 5 psia mixture of oxygen and nitrogen. Fuel cell product water is sufficient to satisfy much of the water requirements. Waste water is dumped overboard. Lithium hydroxide is proposed as the carbon dioxide removal concept (partially used backpack LiOH changes are proposed for use in the cabin systems) and is compatible with missions involving high spacesuit activity; no oxygen recovery is attempted.

The shelters associated with the semi-permanent base envision a partially closed ecology. The atmosphere assumed is a 5 psia oxygen-nitrogen mixture. The systems are powered, in this concept, by a nuclear power plant and thus adequate power can be postulated for more sophisticated recovery systems. Partial water reclamation is practiced: urine, wash-water, and condensate are recycled, but feces are not processed for water recovery. Oxygen recovery from the removed carbon dioxide is proposed with a Sabatier reactor and water electrolysis system.

The "back-pack" or portable life support system postulated across the entire program is equivalent to the Apollo program PLSS, in concept. Water boiling is utilized as a heat rejection system and lithium hydroxide is utilized as a carbon dioxide removal medium. No recovery of these expendables is attempted.

The program defined above is generally described as a rather aggressive view of probable scope of the future lunar exploration and science program. In the view of many persons, this program is so optimistic that it is unimaginable as a serious contender for space flight resources. Being ambitious, however, this program constitutes a perfect candidate for assessment of resource production potential. If resource development cannot be justified in light of the optimistic "market" analysis implied by this plan then it certainly cannot be justified, economically at least, for a more austere program.

The scope of the program just described can best be visualized in terms of "man-days" of lunar surface staytime. A total of 12 400 man-days are accumulated over the entire program; 8000 man-days are contributed by the 12-man two-year lunar base. Approximately one-third of these man-hours can be devoted to scientific or experimental tasks; the remainder is associated with personal time (eating, sleeping, relaxation, etc.) and housekeeping tasks (communications, maintenance, monitoring, etc.). Figure 2 shows the time history of the build-up of this surface time.

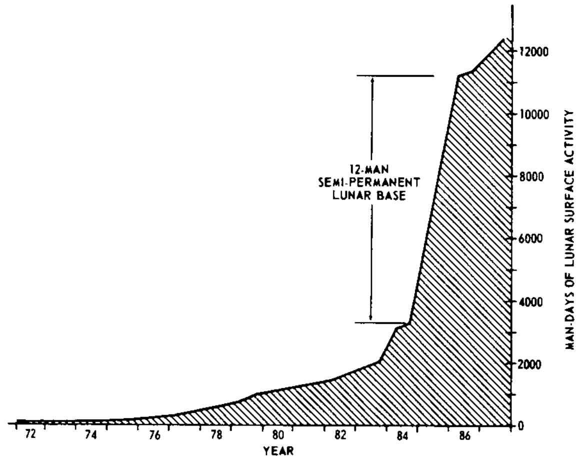


FIGURE 2. CUMULATIVE HISTORY - LUNAR SURFACE ACTIVITY

EXPENDABLES REQUIREMENTS LUNAR EXPLORATION AND SCIENCE

Each of the missions of Figure 1 and Table I was analyzed in light of the specific hardware approach envisioned, and requirements were calculated for each of the following expendables: food, water, oxygen, nitrogen, hydrogen and lithium hydroxide. The resulting demands for expendable items are shown on Figures 3 and 4. Figure 3 depicts the cumulative demand for each expendable as it varies with time during the course of the entire program. A comparison with Figure 2 indicates, as would be expected, that some demands are directly proportional to the number of man-days spent on the moon; other demands

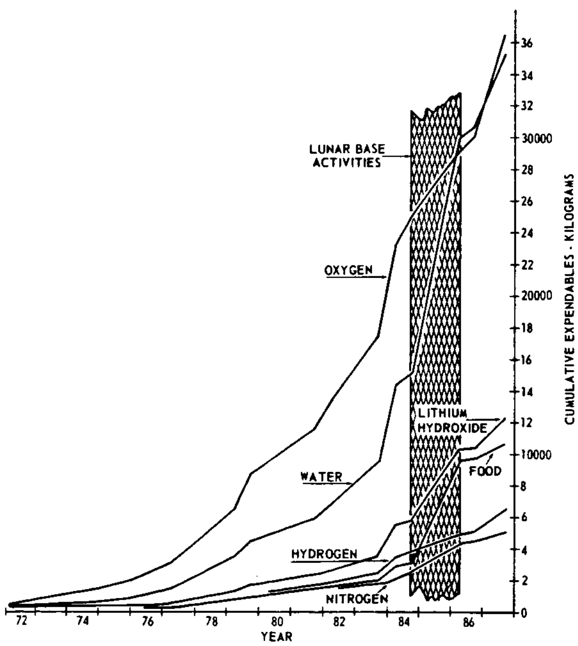


FIGURE 3. CUMULATIVE HISTORY OF EXPENDABLE DEMANDS - TOTAL LUNAR PROGRAM

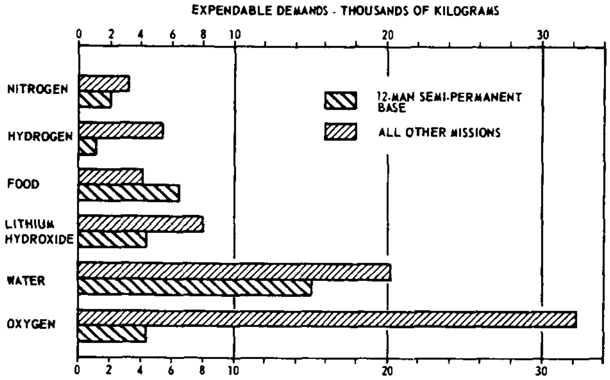


FIGURE 4. TOTAL DEMAND FOR SPECIFIC EXPENDABLES

are more sensitive to the other factors considered: extra-vehicular activity, traverse length or specific technology assumptions. Table II lists the total demands established for each expendable accumulated

TABLE II. TOTAL EXPENDABLE DEMANDS ALL MISSIONS

	Kilograms	% Total
O_2	36 500	34.3
H_2O	35 200	33.1
LiOH	12 300	11.6
Food	10 700	10.1
H_2	6 500	6.1
N_2	5 100	4.8
Total	106 300	100.0

over the entire program. Oxygen and water are the critical items and, at about 35 percent each, make up 70 percent of the total logistic burden. Lithium hydroxide and food make up the next group, and each represents about 10 percent of the total. Hydrogen and nitrogen complete the list with about 5 percent each.

Figure 4 charts these same expendables but splits the requirements into two categories: (1) the demands associated with the semi-permanent 12-man lunar base and (2) all other demands. The expendable demands associated with the non-base activity are, in general, considerably more acute than for the base operations despite the fact that only one-third of the total man-days are represented in the non-base efforts. Obviously, these demands are influenced heavily by the equipment and technology postulations associated with these phases of the total program. The fuel-cell expendables linked to the long-range traverse missions are prime examples of this influence; 62 percent of the oxygen requirement shown is associated with power production. The lunar base, on the other hand, utilizes nuclear power and avoids this particular burden.

In considering lunar resource potential it is not, however, practical to consider production of resources during short-staytime missions nor especially during roving-type exploration missions. We must therefore turn our attention to the lunar base demands when we seek to capitalize on lunar resource production. Table III lists the total demand for each expendable utilized at the base. Water is seen as the prime requirement representing almost half of the total demand. The requirement for oxygen has been dramatically reduced because of the more sophisticated oxygen recovery system assumed for these shelters. In surveying the list of necessary expendables for lunar resource candidates, food and lithium hydroxide are quickly eliminated. Nitrogen is presumed to be unavailable. Hydrogen and water can be developed as lunar resources if water, ice or permafrost is discovered or if water can be found in lunar rock materials. Oxygen production can come as a by-product of hydrogen production, if water bearing deposits are located. Oxygen can also most probably be produced from non-water bearing lunar rock if the lunar rocks are similar to the terrestrial ferro-magnesian silicates. Feasible chemical processes have been identified for the reduction of these silicates into metal oxides, silicon and oxygen [2, 3].

It is believed by the author that the oxygen logistic demand is not sufficiently acute to warrant the development and operation of a silicate reduction plant despite the fact that there is high confidence that the necessary natural resource will be located in great abundance. Since water is the most obvious requirement for lunar manufacturing (Table III) the role that lunar resources will play in this lunar program will pivot around the question of the availability of water or water-bearing minerals on the lunar surface. If it is postulated that water will be located in sufficient supply several production options are presented: (1) water production only can be attempted, (2) water and oxygen can be produced (oxygen production through water electrolysis - the by-product hydrogen vented) or (3) water, oxygen and hydrogen can be produced.

TABLE III. EXPENDABLE DEMANDS
12 MAN, SEMI-PERMANENT BASE

	Kilograms	% Total
H_2O	15 000	45.1
Food	6 500	19.5
LiOH	4 400	13.2
O_2	4 300	12.9
N_2	2 000	6.0
H_2	1 100	3.3
Total	33 300	100.0

Since, for the particular problem being discussed, the oxygen and hydrogen requirements are not balanced in the molecular ratio (8:1) conducive to efficient electrolysis, the hydrogen mass requirement is rather modest, and the liquefaction of hydrogen would dramatically increase the power requirements of the production plant, it is deduced that the candidate expendables for lunar production are water and oxygen. Recall that the assumption must be made that water is available from water-bearing rock minerals or some other natural form.

RESOURCE PRODUCTION CYCLE CONCEPTUAL DEFINITION

With a decision to assess the economic producibility of lunar water and oxygen resources, we can estimate the magnitude of the manufacturing problem and assess its implications in logistics and operational burdens imposed on the lunar complex. A total demand of 15 000 kilograms of water has been estimated for the lunar base over the 660-day mission duration assumed; in addition, the lunar station requires 4300 kilograms of liquid oxygen during the same time period. The portion of this requirement derived from the provisioning of the emergency power system (fuel cells) and the allotment for roving vehicle power requirements is assumed to be delivered as part of the initial logistics load for the base. The difference, 3200 kilograms, which is devoted to metabolic make-up, leakage and extravehicular activity, is assumed to be produced at the base facility. Since the oxygen is to be produced from water electrolysis, the total water requirement must be increased to 18 600 kilograms.

As a means to implement the analysis, a lunar "ore" model had to be chosen. It was assumed that lunar rock could be located near the base site which had a 2 percent adsorbed water content. With the known demand and the assumed quality of the ore, the entire production mass and energy flow diagram of Figure 5 can be generated. It is noted that a total of

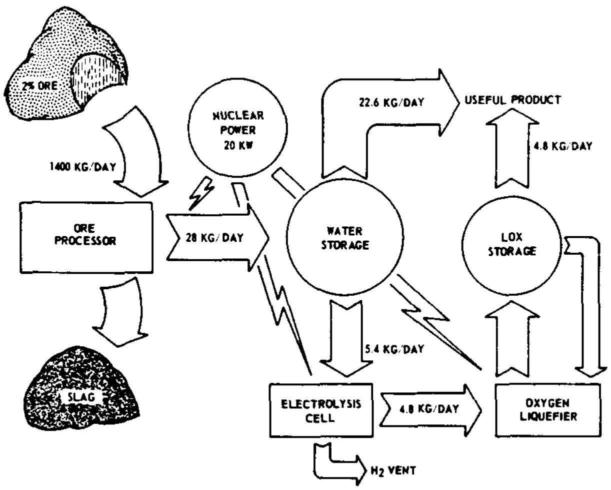
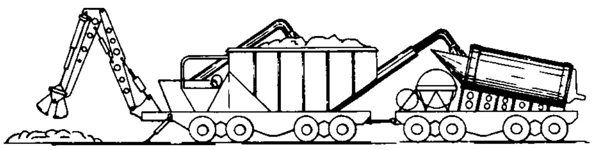


FIGURE 5. CONCEPTUAL WATER & LIQUID OXYGEN PRODUCTION CYCLE MASS FLOW ESTIMATES

925 000 kilograms of ore are required to support the stated product demands - more than 2 million pounds! This ore must be located, excavated, crushed, processed and the resulting slag dumped into a refuse area. An estimated power requirement to support the process is shown on Figure 5. The total power demand must support the ore extraction, grinding or crushing, etc., the water boil-off in the production process [4], the water electrolysis [5], and the necessary oxygen liquefaction process [5]. For the assumed conditions, the power estimate is within the bounds of both feasibility and practicability for the timeframe proposed. It is extremely sensitive, however, to the quality and character of the assumed ore model.

The rock handling requirements previously stated are substantial when viewed in the total; however, when broken down into weekly production units, they appear within reason. One week's demand for ore approximates 10 000 kilograms; at an assumed specific gravity of 1.5, this would amount to 6.6 cubic meters (8.6 cubic yards) of rock. It is estimated that such a production "batch" could be handled by two men working for an 8-hour shift, one day per week. The rest of the cycle could be automated to be monitored from the lunar base and would operate for the full week on a semi-continuous basis. Transfer of products to the lunar base would be accomplished manually as part of the weekly cycle. A scheme for conceptually implementing this process is depicted on Figure 6. Weight estimates, considered to be optimistic, for the equipments required to implement the concept are shown in Table IV.



- TRANSPORTABLE PLANT
- MANUAL MINING
- AUTOMATIC PROCESSING
- "EXTENSION CORD" POWER
- SPACESUIT OPERATION
- MANUAL FLUID TRANSFER
- WEEKLY "BATCH" MINING
- CONTINUOUS PROCESSING

FIGURE 6. MINING & PROCESSING EQUIPMENT CONCEPT

TABLE IV. EQUIPMENT MASS ESTIMATES - WATER AND OXYGEN PROCESSING PLANT

	Subsystem Mass (kg)	System Mass (kg)
Nuclear Power Module		5560
20 kWe Plant	4730	
Distribution	830	
Mining and Crushing Module		4780
Vehicle	3850	
Back-Hoe	365	
Crusher	395	
Conveyors	170	
Ore Processing Module		4215
Trailer	2180	
Electrolysis, Liquefaction and Storage	535	
Processor	1500	
Subtotal		14 555
20% Spares		2911
Total Mass		17 466

CONCLUDING REMARKS

For the particular mission model, ore model and process plant assumptions, selected lunar resource production of water and oxygen can be considered to be competitive on an economic basis with the total transportation of all expendables from Earth. This conclusion is drawn, admittedly somewhat naively, from the approximate equivalence of the mass of the required expendables and the mass of the necessary resource processing equipment.

Certain major assumptions and results should be reiterated at this point. If any of these conditions tend to become less favorable, the economic competitiveness of lunar resource production will vanish. Consider first the requirements themselves; the driving incentive for any production at all rests with the large expenditure of water. That water demand is largely due (62%) to the use of a water subliminator to provide cooling for the space-suited astronaut. If a closed thermal control loop can be developed for the back-pack (over the next 15 years) the water requirement, and the incentive for resource development, will drastically diminish. Alternatively, if extravehicular activity associated with fixed base operations should be reduced, then the desirability of resource production would also be reduced.

Consider also the possibility that the processing equipment mass estimates, or power estimates, are generously optimistic, that equipment life-times approximating two years cannot be achieved, or that equipment reliability cannot be made compatible with the crew safety requirements. Any of these situations would quickly force the marginal equivalence of logistic demands in favor of transporting all expendables from Earth.

The assumption of a lunar "ore" with a 2 percent water content (adsorbed) is fairly optimistic; water chemically bonded to the rock as a hydrated mineral would require a considerably greater energy input in the processing module. Conversely, of course, a higher percentage of water in the ore, or actual location of permafrost or ice deposits, would add to the economic feasibility.

A factor which has yet to be mentioned is the substantial reduction in the output of scientific activity imposed by the additional astronaut time associated with ore location and mining and the operational duties of keeping the process plant in operation. Since astronaut time on the lunar surface is extremely expensive, the normal tendency of the mission planner is to try to reduce the amount of non-scientific astronaut activity; the reliance on resource processing as proposed herein violates this basic principle.

Since the assumed mission model is as optimistic as realistic planning will permit and since the case for the reliance on lunar resource production as a source of lunar base expendables is weakly justified, the author must conclude that lunar resource production, in support of a credible program of lunar exploration and science during the 1970's and early 1980's, cannot be expected to be economically justifiable.

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FUEL PRODUCED FROM SPACECRAFT MATERIAL

By

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ABSTRACT

In the far future, the material of the moon or space will probably be used to refuel space vehicles thereby reducing the cost of space exploration. But what are the prospects for using indigenous materials in the immediate future, let us say before 1975? The prospects are good if a new approach toward utilizing extraterrestrial resources is taken by the working group. This approach would involve work in two new areas of propellant synthesis. In the propellant mass area, sources of mass other than the moon should be considered; in the propellant energy area, unconventional methods for energizing material should be investigated.

The purpose of this paper is to suggest new areas for the working group to investigate and to provide systems data supporting these suggestions. These new areas involve the use of waste material in the vicinity of the spacecraft as a possible fuel ingredient and the development of unconventional methods for energizing material. To help define these new areas of investigation it will be necessary to assess the objectives of the working group, determine what has been done to date to meet these objectives, identify holes that exist in work to date, propose a program to fill these holes and, finally, determine what savings this program can realize.

WORK GROUP OBJECTIVES

To identify new work areas for the working group to undertake, a cursory systems analysis was initiated. Figure 1 is a functional flow diagram of events leading up to a major objective of the working group: to reduce the cost of space exploration. Three items are listed under this major objective: reducing logistics requirements, waiting for technology improvements and reducing the pace of space exploration. The first item is the one toward which the group is directing most of its efforts. However, the two other items are included as alternatives, not so much to propose them as firm objectives but rather to indicate that if one is concerned with the fundamental problem of reducing the cost of space exploration there may be ways of doing this other than reducing logistics requirements.

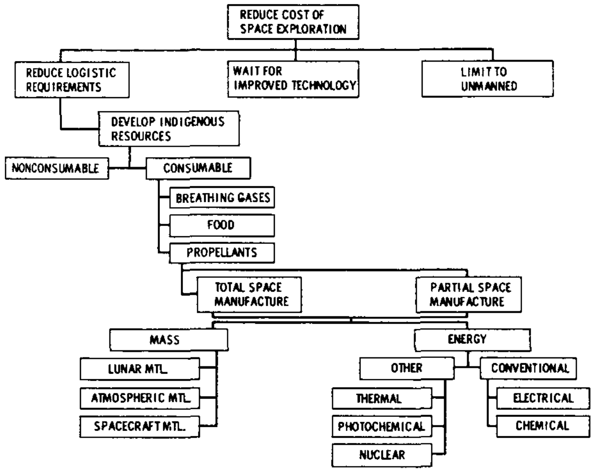

FIGURE 1. OBJECTIVES OF WORKING GROUP ON EXTRATERRESTRIAL RESOURCES

Propellant Resource Identification

Referring again to the flow diagram, the method for reducing logistic requirements is to develop indigenous resources which can be put into two categories, consumable or nonconsumable. Which one of these might warrant further investigation? Consumable resources are much more attractive for study than the nonconsumable ones because of the large quantities of material involved. These larger quantities of material will greatly expedite the rate at which the development and manufacturing costs of expensive lunar processing equipment can be amortized. If we have the option of selecting consumable or nonconsumable materials, it appears that consumable materials would offer greater promise from a quantity standpoint.

In the flow diagram under consumable materials are listed three items: breathing gases, food and propellants. One sees, however, that flow follows only from propellants. The question then arises -- aren't breathing gases and food consumed? Technically, they are not. A breathing gas such as oxygen when entering the body is actually converted through metabolism to water and CO_2 which theoretically can be reclaimed because it is still in the spacecraft. The same reasoning applies to food. When it comes to propellants, however, this situation does not exist. Once a propellant is fired out of a nozzle, the molecules have left the spacecraft forever. Propellant is a truly consumable material.

Total and Partial Space Manufacture

Assuming, therefore, that propellant is the type of material we wish to produce, it is seen that two types of space manufacture are possible. They are total and partial space manufacture. An example of total space manufacture is the conversion of expected lunar or spacecraft water into hydrogen and oxygen fuel. In this case, the entire space ingredient, water, is used. An example of partial space manufacture is demonstrated in the process Rocket Research Corporation is developing under contract to NASA in which space-produced feces are added to propellant ingredients transported from earth. In this case, only part of the propellant is produced from indigenous materials.

Possible Sources of Space Propellant Mass

Let us now see where indigenous materials for use either in partial or total space manufacture may be located in space. Figure 1 shows the three prime locations of space matter: the moon, atmospheres (either earth or planetary), and the material in the vicinity of or within a spacecraft. To date, the working group has been concentrating its efforts on lunar material used in total space manufacture. An example of this effort is equipment and systems techniques that convert expected lunar water into hydrogen and oxygen fuels. However, the prospects of developing lunar materials as fuel [1, 2, 3] either for crew rotation or for Mars missions seem quite remote for the immediate future, as indicated in the references and appendix of this report. As for the case of using earth or planetary atmospheres as a fuel source, extensive work is presently in progress for the earth case because indigenous earth atmospheric material is a basic ingredient for all air breathing engines. Because of the extensive work now in progress by other groups in developing concepts for using the earth's atmosphere and because the WGER is more concerned with advanced space concepts, the investigation of the earth's atmosphere as a fuel source will not be considered further. However, as for planetary atmospheres, Dipprey [4] describes the case for using planetary atmospheres as a possible fuel ingredient. It is obvious, however, from just a planning standpoint that this concept will not be realized before 1975.

While the prospects for using indigenous lunar or planetary atmospheric material in the near future appear remote, the case for using material indigenous to the spacecraft appears more promising, primarily because of the early date at which manned orbiting laboratories are expected to fly. Because the working group has directed no effort toward the latter area of activity, it is recommended that the group concentrate some of its effort on the possible use of indigenous spacecraft material as a possible fuel ingredient.

POSSIBLE SOURCES OF SPACE PROPELLANT ENERGY

The mass aspect of space propellant synthesis appears to have been covered. What about energy, the second item involved in synthesizing a fuel in space. Figure 1 shows that the conventional way of energizing a material (for example, water) is to electrolyze the substance into its component elements with current obtained from a nuclear or solar electric system. However, these energy sources are relatively inefficient and, because power is a major constraint in space manufacture especially in the near future [1], it appears appropriate to investigate other possibly more efficient methods for energizing these substances. Three other methods for directly energizing materials are thermal, nuclear and photochemical. In thermal energizing, water can be dissociated with thermal energy, as shown by the data in Table I.

TABLE I. PERCENT DISSOCIATION OF H_2O

Temperature (°C)	Pressure in Atmospheres			
	0.1	1.0	10	100
	% Dissociation			
1500	0.043	0.02	0.009	0.004
2000	1.25	0.58	0.27	0.125
2500	8.84	4.21	1.98	0.927
3000	28.4	14.4	7.04	3.33

However, large amounts of heat and appropriate equipment are required to separate the resulting hydrogen and oxygen gases. When using nuclear systems for energizing water, the phenomena whereby water is dissociated in the presence of a nuclear reactor is well understood because water is used as a moderator or coolant in a nuclear reactor. In the photochemical area, light mainly in the visible wave lengths converts the water in plants to oxygen and carbohydrates through photosynthesis. However, short wave length photosynthesis is not so straightforward. As a result, an apparatus [5] was assembled in 1963 which successfully dissociated H_2O with ultraviolet light (Fig. 2). Because of the critical space power problem, especially in the near future, it is recommended

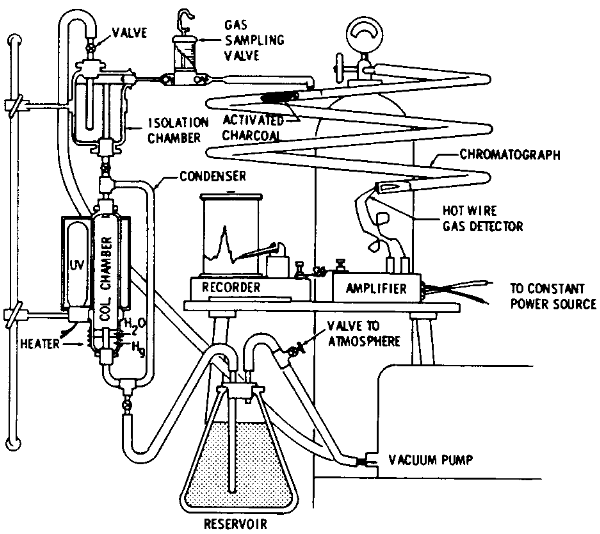

FIGURE 2. APPARATUS USED TO DISSOCIATE H_2O BY ULTRAVIOLET LIGHT

that quantitative work be started on unconventional methods of energizing material from both a feasibility and cost effectiveness standpoint.

Experimental Results

Experimental work has been done in the area of partial space manufacture by Rocket Research Corporation. Judge [6] shows the gel nature of the formulated propellant that uses the mass of feces, carbon black and urine to increase impulse. In operation, the waste material which would include feces, personal hygiene waste residues, wicks and filters would be collected in the waste collector and storage unit. Propellant ingredients added to the spacecraft before launch would be mixed with this waste material in space and fired through a liquid injection motor.

Logistic Savings

Figure 3 shows a comparison of two propulsion systems designed to produce the same total impulse.

(65%) (10%) (20%) (5%)
(1051 LB) MONEX + $\underbrace{\text{URINE + FECES + CARBON BLACK}}_{\text{(FROM SPACE)}}$ → 300,000 LB SEC IMPULSE
(FROM EARTH)

(100%)
(1343 LB) BIPROPELLANT → 300,000 LB SEC IMPULSE
(FROM EARTH)

FIGURE 3. MANNED SPACE STATION APPLICATION

This impulse is required for orbital functions during a 90-day, four-man MORL mission. The monopropellant system uses spacecraft waste but the bipropellant system does not. Rocket Research Corporation estimates the weight savings for the monopropellant system to be 292 pounds out of a total subsystem weight of 1343 pounds.

CONCLUSIONS

The objectives of the working group were assessed by means of a functional analysis to determine new areas for working group investigation. Early utilization of indigenous resources appeared to be an area warranting further analysis. In synthesizing propellants in space, detailed work in both the energy and mass aspects of this propellant should be initiated. In the energy area, unconventional space energizing methods should be analyzed, and, in the mass area, sources of indigenous material from places other than the moon should be investigated.

Investigations showed that experimental results are available which describe how human wastes have been used to formulate a monopropellant that could be compounded in space. Estimated logistic savings by this method amounted to 290 pounds out of a total subsystem weight of approximately 1300 pounds for a 90-day, four-man MORL mission.

RECOMMENDATIONS

The use of waste material in the vicinity of, or within a spacecraft, as a possible fuel source appears to be the most fruitful area for working group investigation if early utilization of extraterrestrial resources is desired. By no means, however, is human waste material the only candidate for space manufacture. Spacecraft equipment might be designed for subsequent use as an indigenous propellant ingredient. Spent stages might also be used as a fuel for chemical, ion or sputtering engines [7].

There are a wide variety of spacecraft materials which may be used as a fuel source. Typical items include:

1. Used food containers
2. Spares containers
3. Disposable clothing and cleaning pads
4. Interstage and other structure.

It is conceivable that if the concept of spacecraft produced propellant proves feasible items such as those listed above could be specially designed to perform a second function, spacecraft propulsion.

APPENDIX

FEASIBILITY OF UTILIZING LUNAR RESOURCES BEFORE 1975

Background

Initial working group activity was directed toward developing concepts such as the one shown in Figure 4. This concept [8] consists of a device to convert expected water bearing lunar rock into hydrogen and oxygen fuel after processing through a propellant manufacturing plant consisting of a rock crusher, a water evaporator, a water condenser, and a water electrolizer and liquifier.

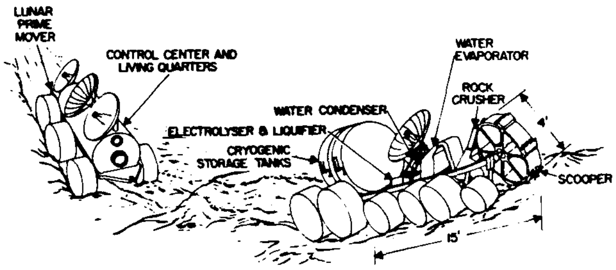

FIGURE 4. LUNAR PROPELLANT PLANT COMPLEX

About two years ago it was felt that while there was extensive work being performed on concepts, there was very little work being done to determine the cost effectiveness of these concepts. Emphasis for the past two years was therefore placed upon determining the cost effectiveness of lunar manufacture.

FEASIBILITY OF LUNAR MANUFACTURING BEFORE 1975

Last year, in Paul [1], Gorrell [2] and Segal [3], the cost effectiveness of lunar manufacture was determined for the main uses of lunar fuel: fuel to refuel a vehicle returning astronauts from the moon and fuel to refuel a manned Mars vehicle. For the latter case, Gorrell [2] showed that if nuclear upper stages were used, lunar fuel would not be warranted. If chemical rather than nuclear upper stages were used instead, the break-even point for lunar manufacture would not occur until four manned Mars missions were completed. Because our first manned Mars mission is not expected to occur until the late 1970's, it is obvious that under the ground rules of this study, lunar manufacture would not be warranted.

A second possible use for lunar fuel would be to refuel a lunar-landed vehicle that would rotate astronauts directly back to earth. As was stated by Paul [1], the break-even point for lunar manufacture, even under the most optimistic circumstances would not occur before 1975. As for the last possible use of lunar fuel, LEM refueling before crew rotation via lunar orbit rendezvous, a more detailed discussion will ensue, as this mode is the one that may occur the earliest of the three mentioned.

Manufacture and transport curves are plotted in Figure 5. The lunar manufacture mode was selected most conservatively because the lunar raw material was assumed to be bare ice. If water expected to exist in lunar rock was selected, or even permafrost, processing costs would increase, resulting in an increase in the slope of the manufacturing curve. The manufacturing curve is also conservative because development costs were not included. To help surmount this problem, these costs were considered parametrically in the following manner: (1) a development cost was assumed and (2) the manufacturing curve was displaced parallel to itself and redrawn to establish a crossover or break-even point between the manufacture

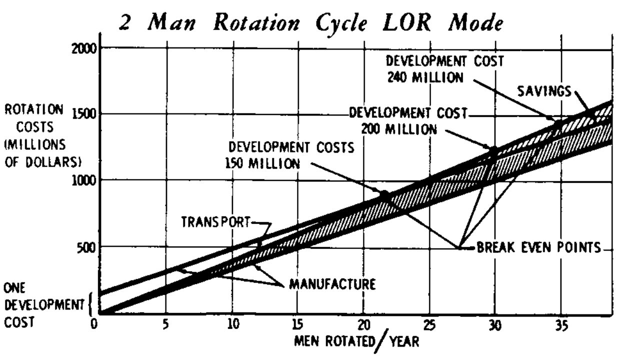

FIGURE 5. DEVELOPMENT COSTS AFFECTING BREAK-EVEN POINT

and transport curves. Then second and third development costs were assumed and other parallel lines drawn to establish two more break-even points. These break-even points were then cross-plotted in Figure 6.

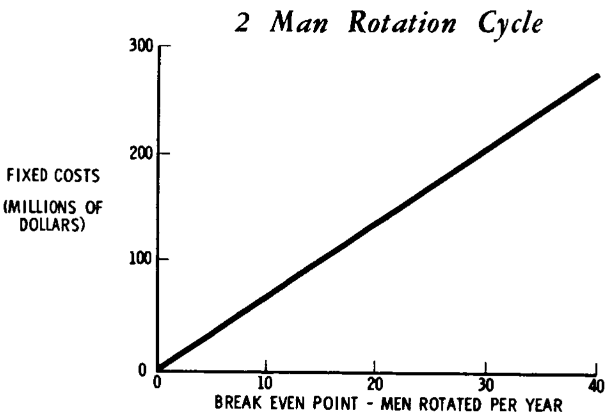

FIGURE 6. FIXED COSTS AFFECTING LUNAR FUELING COSTS

The next thing that was done was to make a rough estimate of development costs to determine which part of the curve shown in Figure 6 would apply. Paul [1] estimates $400 million as the cost of liquification, electrolyzing and storing H_2 and O_2. For this study, a conservative approach would be to cut this $400 million figure in half. Referring again to Figure 6, it is seen that the intersection point between the $200 million development cost and the break-even point is approximately 30 men per year. This means, then, that with a development cost of $200 million it would require the rotation of 30 men per year to have lunar manufacture costs break even with the direct transport costs.

How many men do we expect to rotate by 1975? In the prospectus and in [9], a lunar base which would start in 1975 and extend through to 1985 or 1990 would have somewhere between 6 and 12 men per year. If again we are conservative and take the 12-man case for 1975 and if we assume that these 12 men must be rotated every six months, rotation requirements would be 24 men per year. This 24 man per year rotation requirement, which is a very conservative estimate, is still under the 30 men per year for the $200 million development cost. It appears, then, that according to present planning, insufficient numbers of men are expected to be rotated before 1975 to make lunar manufacture cost effective.

Referring again to the three modes of using lunar fuel, namely fuel for manned Mars missions, fuel for rotating crews with a new direct flight vehicle and fuel for rotating crews using the conventional lunar orbit rendezvous Apollo mode, it appears that the utilization of lunar resources will not occur before 1975.

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DECISION THEORY AND PLANNING OF EXTRATERRESTRIAL RESOURCE UTILIZATION

By

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In "Science and Optimal Decisions" by C. West Churchman, the first sentence reads, "Probably the most startling feature of twentieth century culture is the fact that we have developed such elaborate ways of doing things and at the same time have developed no way of justifying the things we do." Equally startling perhaps is the observation that very seldom are we forced to give any elaborate justification of our acts. However, space exploration seems to have been singled out for an unusual degree of explanation, justification of its social importance, and criticisms of its value. The difficulty in relating the potential achievements of space exploration to the life of the average citizen is, no doubt, the source of this demand for justification, particularly in view of the large amounts of money that are expended. Although large amounts are not yet spent on exploitation of extraterrestrial resources, the motivation for this paper was an investigation of the value of current efforts which seem premature to many observers. A new program always calls for more justification than an established program since it competes for funds with projects more easily understood.

If current studies do have utility, it is in the long range planning of and decision making for space programs. This was the motivation for the examination of resource utilization studies in the light of decision theory. Decision theory [1, 2] deals with the following.

1.	"States"	$\{\theta\}$	factors that affect mission success, i.e., resource availability, technological advances, budget support
2.	Potential "experiments"	$\{e\}$	probes, development flights, logistics studies, systems analyses, R&D programs
3.	"Outcomes" of experiments	$\{z\}$	knowledge of environment, expected performance, logistics requirements
4.	"Probabilities"	$\{p_e(z, \theta)\}$	probability that a state exists when the result of experiment e is z
5.	"Decisions"	$\{d^e, d^t\}$	the decision to conduct experiment e or to make the terminal act t
6.	"Utility"	$\{U(t/z)\}$	the expected value of the act t in light of the knowledge z

It is not the purpose of this discussion to formulate a mathematical model that will optimize the future space program. There are too many variables and unknowns and the theory is not sufficiently developed. The purpose is to examine the formulation of a program within the logical framework provided by decision theory, particularly to examine how certain steps may be justified.

If the meaning of resources is taken in its most general sense, then utility of the space program lies in the ability it gives mankind in exploiting extraterrestrial resources. Ultimately it means that man will be freed of the constraint of living in a finite environment. The absence of any final outcome of the space program is one of the features that makes an application of decision theory difficult. However, we can work the terminal action in the context of meeting some intermediate goal; fairly ambitious and far downstream in order that extraterrestrial resources play an important role.

Consider the following missions proposed for the 2000 to 2010 time period:

1. A Venus complex consisting of a manned orbiting station and unmanned surface installation.

   Objective — Climate control experiments with the results eventually to be applied on Earth. Development of capability to utilize natural resources of Venus.

   Approach — Create a cloud between Venus and the sun to shade portions of the planet's surface. Ground and atmospheric installations used for data acquisition, experimental use of resources, planetary exploration.

2. Mars Station

   Objective — Prepare basis for eventual colonization.

   Approach — Establish manufacturing plants, conduct resource surveys and exploration.

3. Lunar Colony

Objective — Provide a normal productive life for a considerable body of people on the moon.

Approach — Create a large enclosed region containing surroundings suitable for all the needs of a colony of people.

The terminal decision, d^t, for each of these plans will be to proceed with the mission according to some definite plan or to cancel the effort completely. The decisions to experiment will be decisions to fly probes and developmental missions, conduct logistics studies, and systems analyses, etc.

We will trace developments leading to the missions, depicting in a scenario from the experiments possible results of the experiments and the decisions to be made. The Venus mission will be treated in the greatest detail.

Application of Decision Theory to this mission runs into an immediate difficulty. The utility function which expresses the worth of conducting such a mission is not defined, much less is it generally accepted. Moreover, opinions of the expected benefits of the mission will undoubtedly change from year to year. Consider, for example, a series of events such as the following:

1970 - International Space Congress has Venus exploration as theme.

1973 - Venus probe discovers H_2O in lower atmosphere.

1976 - Meteoroid hits New York City killing 750 people.

1981 - Manned orbit of Venus achieved successfully.

1984 - Drastic solution to the world's food shortage needed. Climate control proposed.

1986 - Planetoid misses Earth by 75 000 miles.

1988 - International law pertaining to space successfully completes trial period and becomes permanent. The international space program receives a 40 billion dollar annual budget. Much debate is carried on as to how to spend the budget.

1989 - Climate control experiments unsuccessful on Earth.

1993 - Ageing process of human beings can be controlled. Death rate drops to 25 percent of the 1975 rate.

1995 - Mars resources prove limited. Self sufficiency of the lunar colony cannot be achieved in the near term.

1996 - Planetoid misses Earth by 10 000 miles. Much concern over future encounters results.

1998 - Climate control experiments on Earth have unexpected results with harmful side effects.

1999 - Propulsion developments cut costs of space transport sharply.

2000 - Much discontent results from overcrowded conditions on Earth--lack of new frontiers. Food problems still not solved.

Each of these events affects the utility function of either the decision to carry out the Venus mission or some auxiliary mission such as a meteoroid capture which would provide some needed technology. There will be periods of gradual change and times when world events cause instantaneous changes in utility. When related to dollars the utility function could have the following appearance (Fig. 1):

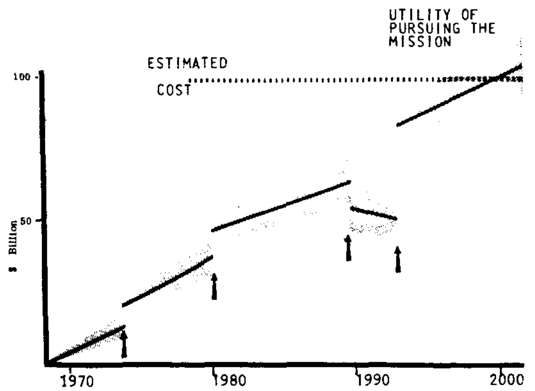

FIGURE 1. UTILITY FUNCTION

When the utility exceeds the expected cost, a terminal decision to proceed with the mission will be made. The choice of mission plan will be made with the objective of maximizing utility and minimizing cost.

There are two utility functions at any given time. The current estimate of the value of the utility function at the time of mission execution (100 billion dollars on the chart above), and the present utility of taking measures to further success of the mission.

The "present utility" of this mission is probably worth a few thousand dollars. It is enough to warrant study of the mission on a limited basis and is based entirely on the great future potential (unless the pleasure of making such studies is given value). When at some future time the present utility function is very much larger, decisions to conduct experiments in the form of probes, research and development programs, etc., will be made.

The expected cost of the mission depends on the state, θ, which, in this example, is the set of all the relevant factors that bear on the feasibility, the success and the resource requirements of the mission. Our knowledge of the state also changes from time to time. This change can be caused by an experiment, e, that we conduct deliberately to improve our knowledge or it can be caused by external events (not considered in formal decision theory), which are not under control of the mission planner.

Consider the following sequence of related events and experiments:

1967 - Resources studies of Venus made (experiment).

1971 - Venus probe detects water in the lower atmosphere (experiment).

1975 - Atmospheric model of Venus developed (experiment).

1976 - It is determined that no artificial satellites of Venus greater than 0.1 mile diameter exist (experiment).

1977 - Large scale meteoroid survey instituted (experiment).

1980 - Nuclear pulse propulsion development started (related event).

1981 - Manned orbiting station of Venus discovers topographic features on planet including volcanoes (experiment).

1986 - Artificial methods of inducing violent volcanic action developed (experiment).

1986 Energy sources developed to vaporize meteoroids (external event).

1987 - Successful test of nuclear pulse system (related experiment) in modifying a 1 000 000 ton meteoroid orbit. Meteoroid and planetoid detection stations established in the vicinity of Earth (related event).

1990 - Prototype of proposed Venus Power Plant run successfully (experiment) using internal planetary heat.

1992 - Transportation cost from Earth surface to low Earth orbit is thirty dollars a pound (related event).

1995 - Unmanned ground station on Venus discovers native metals, ammonia, etc. Physical climatology studies made (experiment).

1996 - Climate control becomes a well developed subject without much experimental work having been done. Proposals for control of climate of Venus are shown to be practical (experiment).

1998 - Environmental control experience on Earth and on lunar and Mars bases can be applied to Venus Project (external event).

1998 - Automation of manufacturing process has become highly developed (related event). Prototype plants using Venus type raw materials tested (experiment).

1999 - Studies show that induced volcanic action would interfere with other activities on Venus (experiment).

2000 - International annual budget for space is 50 billion dollars (related event).

Each of the experiments above was conducted on the basis of improving the "present" utility function as known at the time of the decision. The decision (Fig. 2) to make the experiment, d^e, is not essentially different from the terminal decision, d^t, to execute the mission since, in fact, to tag one mission as terminal and others experimental is purely artificial. The last mission in our list will clearly provide experimental results that will be used to determine additional missions. Also the experiments to be conducted in the 70's and 80's must be done without any clear idea of what the final mission will be. Therefore space planners will be faced with a long series of decisions to be made on the basis of an ill defined utility, and on the knowledge gained from previous experiments and external events.

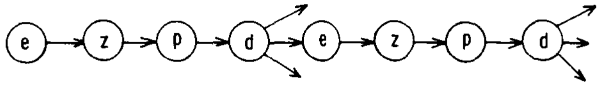

FIGURE 2. POSSIBLE "STATES" FOR VENUS MISSION

Consider possible "states" for the Venus mission. As explained earlier these include all facts relevant to the execution of the mission.

1. Various models of the atmosphere exists [5]. One such model is approximated by (no claims are made for its accuracy, but this is one of the most hostile):

$$T = -10\,h + 750$$

$$P = 30\,e^{-.067h}$$

where
- h = the altitude in kilometers
- T = the temperature in degrees Kelvin
- P = the pressure in atmospheres.

2. If the temperature is to be changed to

$$T = -h + 300$$

and assuming the atmosphere to have the heat content of carbon dioxide, one calculates (Fig. 3) that some two million calories must be removed from a column of the atmosphere one square centimeter in cross section. Adding a quarter million calories to be removed from the upper layer of rock or sand, a total of 2.25 million calories need to be radiated from each square centimeter of the surface of Venus. Venus currently radiates about 0.4 cal/cm^2-min to remain in equilibrium with the solar heat input (3.8 cal/cm^2-min 40 percent absorbed over 0.25 the area over which the heat is re-radiated by Venus.)

3. If all the sunlight was cut off from Venus $2.25 \times 10^6 \div 0.4 = 5.6 \times 10^6$ minutes, 10.5 years would be required to cool Venus to the desired temperature. This seems to be within the limits of human patience as regards waiting for the results of an expenditure of effort. The time would change if large amounts of water were to be condensed or the planet's radiation pattern changed.

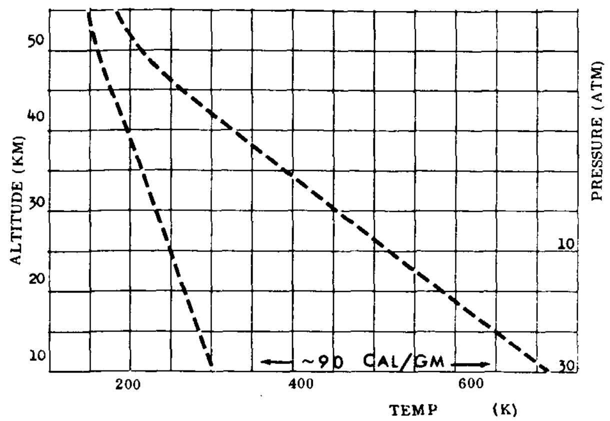

FIGURE 3. RATE OF HEAT REDUCTION VERSUS ALTITUDE AND PRESSURE

4. Procedures for reducing the sunlight reaching Venus or increasing the heat radiated by Venus are:

   a. A cloud of magnetic particles 4×10^6 inches in diameter and having the density of iron would require 10^8 tons of material at a minimum to completely shade Venus.

   b. Create a reflector at an orbit about Venus. A band on either side of a great circle of Venus could be shaded. This band would be most effective if located to include the orbit plane of Venus. More material would be required than in "a" for the same degree of shading but problems of stability are less troublesome. In addition to cutting the heat input to the planet, atmospheric currents would form, cloud patterns would change and the heat radiated from the planet would be different. An extensive physical climatology investigation would be required to determine the effect on the climate.

If a synchronous orbit is feasible for the reflector, a circular region could also be kept in perpetual shade. A similarly complex analysis would again be required to determine the effect.

c. Create dust clouds in the upper atmosphere to change the emissivity of the planet. The analysis is even more complex than in "b" above.

5. All the alternatives involve moving large quantities of material. Some possibilities present themselves.

a. A small satellite of Venus may exist still undetected. It could be pulverized or vaporized to form a cloud, or it could be transformed into a thin sheet (for this purpose it would be convenient if it were solid metal).

b. A large meteoroid or planetoid could be captured and given the desired orbital characteristics. Impetus for the technological development might be external to this project as indicated in the scenario discussed earlier. Feasibility of capturing a planetoid is discussed in [3] which also contains a bibliography. A 150 000 pound package featuring a 43.8 megaton nuclear pulse propulsion system is presented in this paper as a typical requirement for the capture of 5×10^8 ton planetoid.

There is some evidence that a planetoid hits Earth about once every ten million years [4]. Using a capture cross section of 16 million miles radius instead of the 4000 mile earth radius and assuming random arrivals we have the following:

Number of Capture Opportunities per Year	0	1	2	3	4
Probability of Occurrence	0.2	0.32	0.25	0.14	0.05

The 10^8 ton minimum weight needed at the libration point to completely shade Venus is about equal to the amount of meteoric material falling on the Earth in one year and is roughly one-tenth the size of Hermes which passed within 400 000 miles of Earth some years ago. Because of the instability of the libration point, we might have to wait for a nickle-iron meteoroid so as to be able to apply a magnetic field. For an orbital cloud any meteoric material would do.

c. The material could be obtained from Earth. This seems uneconomical even with optimistic forecasting of propulsion developments since we are talking about a quantity roughly equal to the U. S. annual steel production. Use of nuclear pulses, suitable propulsion for meteoroid capture, seem out of place for Earth launchings. It is probable that moving the material from the moon would require less energy and rougher methods could be used.

d. Material from the surface of Venus might be used. Volcanic activity might be induced with atomic explosions to create dust clouds in the upper atmosphere. An effect such as the cooling that the eruption of Krakatoa in 1883 produced on Earth might result. A very complex analysis is required to determine the results of such methods. Lofting a very large quantity of material into orbit from the surface of Venus is difficult to see at this stage.

e. Part of Deimos, the outer moon of Mars could be transferred to the libration point of Venus. Again nuclear pulses could be used for propulsion. Energy requirements would be greater than for the capture of some planetoids but it is possible that advantages would result from the better definition of mission requirements.

At this point, we are clearly in no position to decide to proceed with the mission, much less to decide which of the alternatives to follow. We are at the beginning of the sequence of decisions to experiment, make studies, develop technology, etc., that will enhance man's capabilities in space. The role of utilization studies of extraterrestrial resources at this very early stage (before we even know what the resources are) is to assist in the decision making process. They are part of the set of experiments, e, used to uncover knowledge of the state, θ, which determines the success of the mission.

The analysis supporting the final decision will be similar to that used in current logistics studies. The cost of the mission is indicated by the gross weight of material needed to support the mission. For this mission the net weight, W_N, is given by

$$W_N = W_S + W_O + W_R$$

where

W_S = the weight requirements of the surface station (unmanned)

W_O = the weight requirements of an orbiting station (manned)

W_R = the weight of the reflecting material

The gross weight is obtained by multiplying each of the above terms by a "transportation factor" that accounts for the weight required to move the net material to its desired position. This factor, T, is of the form

$$T = 1/(X-Y)$$

where

X = the ratio of structure to total weight

Y = $e_{xp}\ (-\Delta V/gI_{sp})$ as given in the rocket equation

These materials are obtained from the earth, some extraterrestrial source, and perhaps the planet Venus (Fig. 4). The weight required at the planet's surface can be written

$$W_S = E_S + P_S$$

where

E_S = that part of W_S originating on Earth.

P_S = that part of W_S of local origin.

Then, the gross weight to be supplied from Earth is

$$GW_S = T_S(E_S + x_S P_S)$$

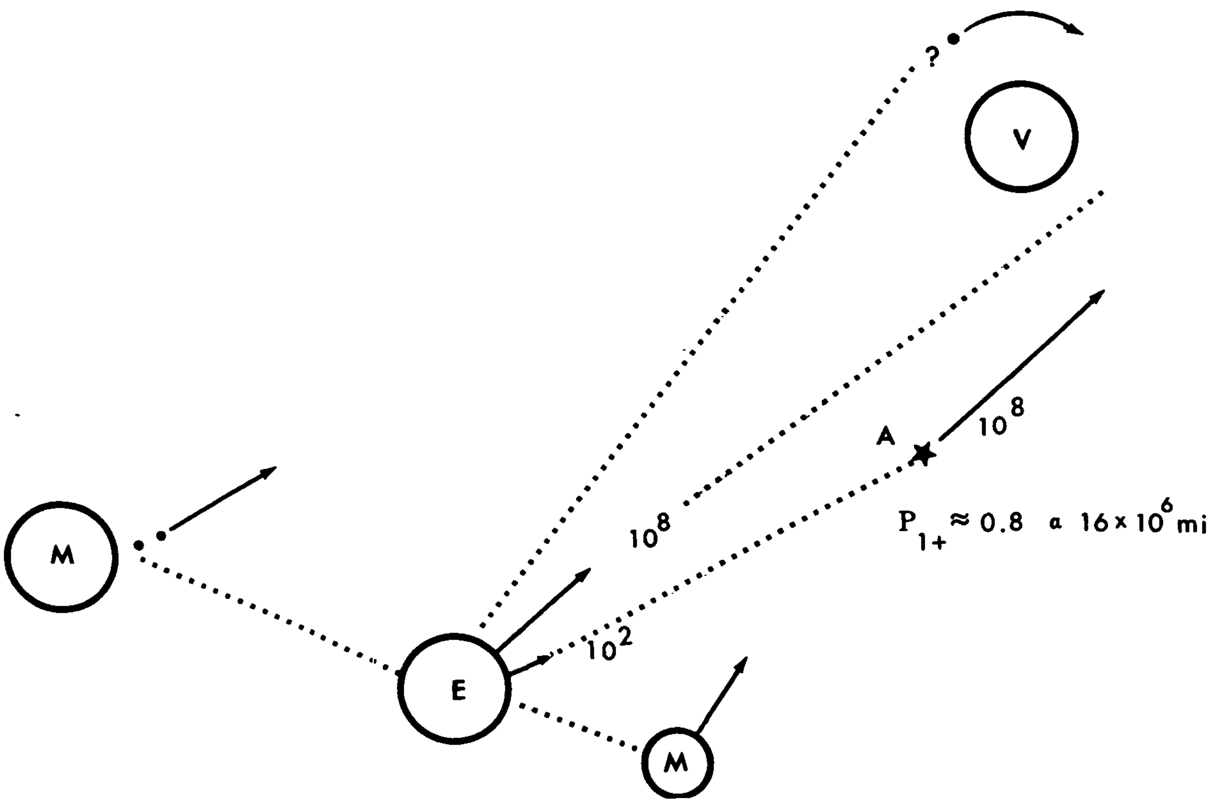

FIGURE 4. SOURCES FOR MATERIALS

where

T_S = transportation factor.

$x_S P_S$ = the weight of equipment to be supplied from Earth that makes it possible to use local resources in the amount, P_S.

The relationship of the various elements is shown in Figure 5.

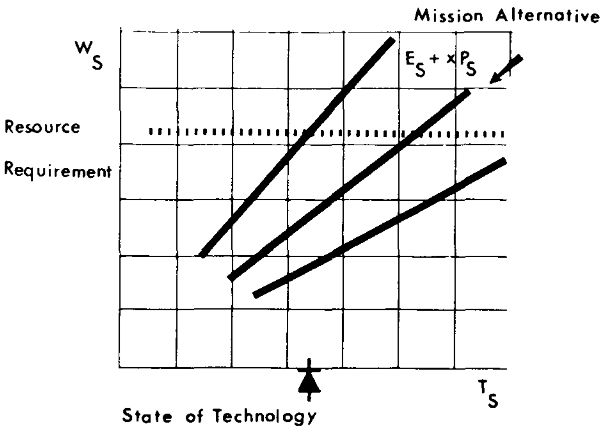

FIGURE 5. RELATIONSHIP OF VARIOUS ELEMENTS

The state of space technology at the time of the mission is represented by the value of the transportation factor, T_S, the mission alternative is represented by the net weight, $E_S + x_S P_S$, and the resource requirements by the gross weight, GW_S, to be launched from Earth. This "model" applies to the lunar colony and Mars base as well as the Venus ground station. It can be used for a variety of decisions:

1. To pick a mission alternative
2. To proceed with the mission because available resources match requirements
3. To inaugurate an R&D program with the object of decreasing T_S or $E_S + xP_S$
4. To conduct experiments or logistics studies
5. To improve knowledge of the values of E_S, x or P_S

The model for the entire mission is complicated by the presence of several transportation factors.

$$GW = T_S(E_S + x_S P_S) + T_O(E_O + x_O P_O) + T_R x_R W_R$$

where the new notation is similar to that already defined. W_R, for example, could be the 5×10^8 tons which is the weight of the meteoroid discussed earlier. The $x_R W_R$ is the 150 000 pounds needed to capture the meteoroid and $T_R x_R W_R$ is the gross weight to be launched from the Earth's surface. A simple two dimensional diagram no longer exists, but the application to decision making is the same.

What has been presented is, of course, a very much condensed version of the analysis. Each term is a function of hundreds of parameters, and the expansion to show dependence on a certain parameter may very well be needed to make a given decision. Also, a better indication of cost than gross weight will finally be used. The purpose here is to explore the application of the analysis rather than develop it. In particular, justification of current studies of extraterrestrial resources has been placed in the context of long-term series decisions on space developments, and their value in exposing the utility of many potential experimental projects has been discussed.

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IMPLICATIONS OF USE OF LUNAR PROPELLANTS IN SPACE TRANSPORTATION SYSTEMS

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ABSTRACT

It is suggested that operational as well as economic advantages will eventually result in the evolution of a completely reusable space transportation system. To assess the potential of lunar manufactured propellants, this paper infers their availability for use with this future system.

A simplified analysis, using the rocket equation, is made on typical space stages representing S-II, S-IVB and Centaur stages in size, operating from orbiting terminals around the Earth and moon. The terminals are logistically supported by reusable surface to orbit vehicles. Payload fractions for several mission altitudes and mission modes are determined and compared.

The cost of a hypothetical planetary program, logistically supported from earth, is estimated. From this estimate, a reference or break-even cost for lunar propellant production is established. Operation below the break-even point could result in substantially reduced program costs were logistic support to originate from the moon.

The results should serve as a guide for improved models of lunar mining and processing and system synthesis in support of space transportation.

INTRODUCTION

In the past ten years, we have witnessed tremendous advances in space activities and have built a great capacity for future progress. Those unthinkable tomorrows have become real todays. But the task to fulfill the imaginative objectives of the National Aeronautics and Space Act has only started.

Because of its enormity, the aerospace program infringes on most other aspects of our national life and competes with them for resources. It behooves us, therefore, to be alert to subtle scientific or technological revelations and to give timely and serious consideration to plausible approaches to space exploration that exhibit potential economic and operational advantages. Simply stated with regard to the space transportation system, the goal is to drastically reduce the fare per pound of payload.

We have standardized some vehicles and eliminated others in our efforts to economize. We have launched multiple spacecraft on a single vehicle and plan to combine some programs in one spacecraft where it is possible. All of these efforts are undertaken to reduce the cost of transportation.

As spacecraft weight and costs increase and missions require greater energy, we note increased interest in reusability and the technology base associated with such vehicles [1].

A quick glimpse at plans for advanced programs [2] gives a hint as to a possible trend in space operation. The objective of Apollo applications is to develop the capability to allow men to work outside the spacecraft for long periods to transfer cargo, personnel and propellant from one craft to another (Fig. 1).

Ultimately, we see the manned workshops evolving into orbital launch facilities or shuttle terminals to serve orbital, lunar and planetary programs with reusable transportation from Earth orbits. Even reuse of our present upper stages can be visualized, thus allowing us to capitalize on the price paid for their quality and reliability. Plans also suggest that logistic support for these stations will evolve and result in an advanced system featuring reusability, land landing and mission versatility [2], as illustrated in Figure 2.

Despite the fact that we have come a long way since the beginning of our assault on this new frontier, space, we have hardly started. Only the most optimistic and visionary among us catch a glimpse of, and understand, the opportunities that await us in the years ahead. Certainly our transportation system will continue to be hard-pressed to meet future needs within its allotted resources.

FIGURE 1. ARTIST'S CONCEPT OF TRANSFER OF CARGO

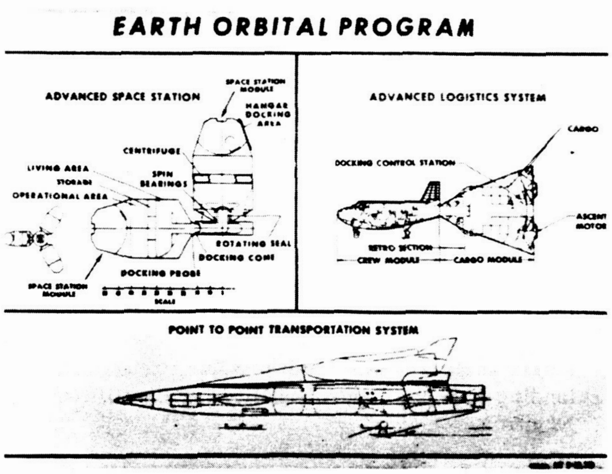

FIGURE 2. EARTH ORBITAL PROGRAM

The Working Group on Extraterrestrial Resources has done considerable work concerned with utilization of lunar resources. Exploration, mining and processing have been dealt with in great detail in the previous proceedings [3]. We can, with some encouragement from this work, envision a lunar base, self-sustaining for the most part, and eventually capable of producing propellants.

Perhaps orbital terminals around the moon could evolve similar to those foreseen about earth. To assess the potential of lunar propellants, this report infers their availability for use with a space shuttle system and estimates an expenditure that could be considered reasonable for the production of propellants on the moon.

DISCUSSION

The space shuttle system consists of three items: (1) a surface to orbit and return vehicle, (2) the orbiting terminal and (3) the space shuttle vehicle. An Earth-based and lunar-based system operating in two modes are compared. In mode 1, the space shuttle vehicle carries a payload to the mission and returns with the same weight to the terminal (simulating an exchange of modules). This mode is designated as a "round-trip payload" (Fig. 3).

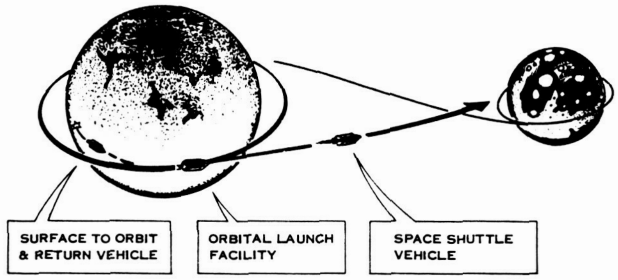

FIGURE 3. SPACE SHUTTLE SYSTEM

In mode 2, the space shuttle vehicle carries a payload to the mission and returns to the terminal empty. Mode 2 is designated as a "one-way payload/ empty return." Estimates of the possible payload fractions that could be achieved by the surface-to-orbit and return vehicle and the space shuttle vehicle have been computed. It is not surprising to find that the lunar surface-to-orbit and return vehicle can deliver a payload to 100 nautical miles which is approximately nine times greater than the Earth-launched vehicle, as shown in the Appendix to this report. This condition is to be expected because of the absence of atmospheric drag and the reduced gravity force of the lunar mass. Similarly, a comparison of the payload fraction of the space shuttle between the terminal and mission altitude brought few surprises. There is an approximate two-fold gain of the synchronous altitude and more than a three-fold gain at 208 000 nautical miles for mode 2 missions (Table I).

TABLE I. SUMMARY OF RESULTS*

	Vehicle A					
Mission	300	700	2000	6700	19 350	208 000
Round Trip						
Earth Launch Payload	893 054	727 979	439 074	162 662	66 211	64 574
Moon Launch Payload	64 574	65 551	71 427	99 652	162 967	414 000
One-Way Empty Return						
Earth Launch Payload	939 000	841 000	627 000	333 000	170 000	165 000
Moon Launch Payload	167 000	170 000	180 000	232 000	333 000	605 000
	Vehicle B					
Round Trip						
Earth Launch Payload	285 971	234 337	141 525	53 575	22 886	22 355
Moon Launch Payload	22 357	22 666	24 544	33 500	53 962	133 701
One-Way Empty Return						
Earth Launch Payload	300 000	269 000	202 000	109 000	58 000	57 000
Moon Launch Payload	57 000	58 000	62 000	78 000	110 000	195 000
	Vehicle C					
Round Trip						
Earth Launch Payload	41 139	33 762	20 504	7939	3555	3479
Moon Launch Payload	3479	3523	3792	5071	7993	19 630
One-Way Empty Return						
Earth Launch Payload	43 200	38 800	29 200	16 000	9000	8800
Moon Launch Payload	8800	9000	9500	12 000	16 000	28 300

* NOTE: All launches are initiated from 100 nautical mile orbit.

When the total system is tied together, and the payload fraction plotted, results are extremely interesting (Fig. 4).

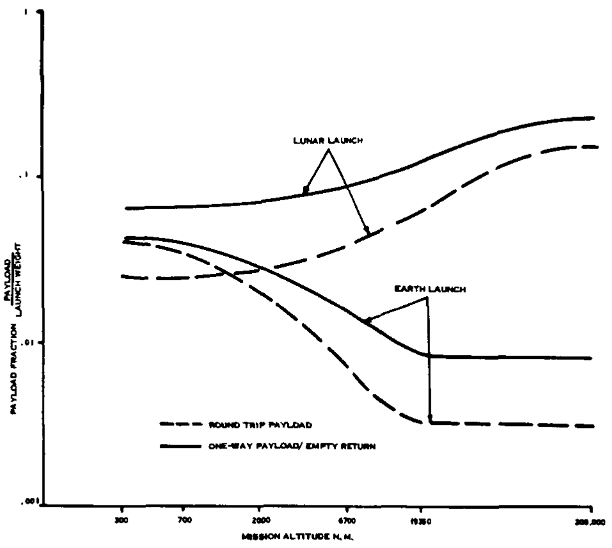

FIGURE 4. EARTH SURFACE AND LUNAR SURFACE LAUNCH PAYLOAD FRACTIONS VERSUS MISSION ALTITUDE

One can see immediately that a substantial advantage for the lunar-based system extends in the case of the round-trip payload mode to between 1000 and 2000 nautical miles above the earth, and in the case of the one-way payload/empty return mode, down to 300 nautical miles. In this mode, apparently, Earth-launch payload fractions never show an advantage. For example, assume one-million-pound vehicles are launched from the surface of the Earth and from the lunar surface. The Earth-based system will deliver approximately 3.9 percent of its launch weight as payload, or 39 000 pounds. The lunar-based system on the other hand will deliver 6.5 percent or 65 000 pounds to the 300 nautical mile Earth orbit.

Although this is a significant payload gain, it is doubtful that this mission from the moon could be cost effective. The lunar manufactured propellants will be much more costly than those produced on Earth. How much more, we do not know, but by looking at the other extreme, the lunar libration point, we hopefully can establish at least in a general way how much we could pay to make logistic support from the moon attractive for higher energy missions.

Gillespie has discussed the advantages of the Earth-moon libration point as a marshalling yard for trips to the planets in a previous Working Group on Extraterrestrial Resources paper [4]. We see a substantial difference in the payload fractions for this mission. The payload fraction for lunar launch is approximately 30 times greater than that from Earth.

Just let us suppose that we become involved in planetary operation at some future date and it is necessary that our transportation system provide approximately 220 000 pounds of propellants to the libration point each month for a five-year period (Fig. 5).

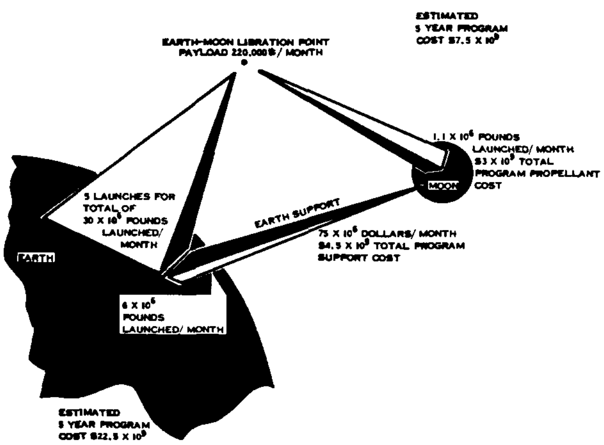

FIGURE 5. COMPARISON OF POSSIBLE LOGISTIC SUPPORT COSTS FROM EARTH AND MOON

Let us also entertain the hope that cost reductions resulting from the introduction of reusable vehicles will reflect a decrease of the present 150 million to 180 million dollar cost of Saturn V type vehicles (6×10^6 pounds launch weight) to a cost of 75 million dollars. Five flights of this type of vehicle, or approximately 30×10^6 pounds launched from the earth, versus one flight of a much smaller vehicle, possibly a modified S-II stage (1.1×10^6 lb) launched from the surface of the moon would be needed each month to supply the required amount of propellant to the libration point. It may well be that these stages are already in space, left over from other programs and available with minimum refurbishment to play a role in logistic operations from the moon.

The total five-year program, or 60 missions logistically supported from Earth, is estimated to cost 22.5×10^9 dollars. From this estimate, we should be able to establish a break-even or reference cost for lunar propellant production were logistic support to originate from the moon.

A rate of 1×10^6 pounds of propellant per month for a total of 60 months would be required for lunar support operations. The systems developed for the extraction of propellant at this production rate and for this period of time with integral storage and propellant loading capabilities would require a high degree of automation and reliability.

It is envisioned that the operation of this plant, and the launch operations and services could be within the capability of six-man crews, rotated to Earth each month by a Saturn V type of support vehicle that would provide spare parts and other special help not already available at the lunar base.

The cost of this effort (75×10^6 dollars per month) is not directly related to the propellant production but must be considered a legitimate cost associated with a lunar-based logistic operation. This leaves 18×10^9 dollars which could be used to design, develop and install a plant for propellant production which would allow logistic support from the moon at no greater cost than that from an Earth base. To account for possible misjudgment in estimating, a contingency factor of 1.5 might be introduced to adjust this to 12×10^9 dollars.

Our goal, however, is a savings, a real pay-off for this method of operation. If it can be shown that these costs come out somewhat lower, one might consider it sufficiently promising to proceed, but if the costs are very much lower, perhaps 75 percent less, this approach could be considered highly desirable. We must remember that discussions such as these are not attempts at precision; they constitute hypotheses about what one considers the important features of a concept. Seventy-five percent or a 3×10^9 dollar cost for the production of propellants on the moon to support planetary operations represents a reasonable goal.

If achieved, the support of our hypothetical program from the moon, including Earth-based assistance, would cost 7.5×10^9 dollars against the 22.5×10^9 dollars estimated for Earth-based logistics.

CONCLUDING REMARKS

Certainly the cost of operating from the moon is extremely sensitive to our program projection, and it would appear prudent to look at a range of numbers to see what the possible range of cost is.

Let us propose this cost for lunar-based propellant production only as a reasonable projection, possibly a challenge, and invite our colleagues concerned with extracting the hydrogen and oxygen from the lunar mass to examine and define the problems and obstacles that relate to the propellant quantity, rate and costs that have been discussed here.

It is hoped that these parameters, which may be characteristic of future lunar-based logistic operation in support of planetary programs, will be helpful as a guide for improved propellant extraction modeling and system synthesis.

APPENDIX

Surface to Orbit and Return Vehicles

The Earth and lunar surface to orbit return vehicles may be easily compared. In the Earth to orbit case, we can select the present payload fraction of the Saturn V, which is approximately 4.5 percent [5], and use it to represent what one would envision a future reusable system could attain.

In the lunar surface to orbit case where the vehicle would possibly resemble a modified version of the S-IVB or S-II stages, one would expect to achieve a payload fraction estimated at approximately 40 percent, as shown in Table II.

This means, for example, that a vehicle weighing 100 000 pounds departing the Earth surface could be expected to place a 4500-pound payload in a 100 nautical mile orbit and return to Earth, as compared to a vehicle of the same weight leaving the surface of the moon, which could be expected to place 40 000 pounds in a 100 nautical mile orbit about the moon and return to the lunar surface.

TABLE II. DETERMINATION OF ESTIMATED LUNAR SURFACE TO ORBIT REUSABLE VEHICLE PAYLOAD FRACTION

Assume Use of Modified S-II Stage

Launch Weight 11 000 000 lb

▲V Required to Lunar Orbit 100 nm	6000 ft/sec
▲V Gravity Losses	1000 ft/sec
Total ▲V Required to Orbit	7000 ft/sec
▲V Required to Return to Surface	7000 ft/sec

▲V	$g I_{sp}$	LNMR	MR	$\frac{MR - 1}{MR}$	W_o *	W_f **
To Orbit 7000	13.685	0.5115	1.6678	0.4000	1 100 000	440 000
	Weight in Orbit				660 000 of Which 460 000 lb Is Payload	
To Lunar Surface 7000	13.685	0.5115	1.6678	0.4000	200 000	80 000
	Weight of Vehicle Empty				120 000	
	Payload Fraction				$\frac{460\ 000}{1\ 100\ 000}$ = 41.8% ***	

* Launch weight
** Propellant weight
*** 40% used as approximate value

Space Shuttle Vehicle

A simplified analysis using the rocket equation is made on three space shuttle vehicles representing S-II, S-IVB and Centaur stages in size (vehicles A, B, and C, respectively).

The energy required, ▲V, (Tables III and IV) is that associated with Hohmann transfer trajectories. An impulsive incremental velocity is given to the vehicle while in a circular parking orbit. At apogee of the transfer orbit, an incremental impulsive velocity of sufficient magnitude is added to circularize at the mission altitude. To return to a lower altitude, the maneuver is reversed [6].

The rocket equation

$$\Delta V = g I_{sp} \text{ NL MR}$$

is solved for weight of propellant required for each maneuver.

$$\text{Mass Ratio} = MR = \frac{\text{Lift-off Weight}}{\text{Burn-out Weight}} = \frac{W_o'}{W_o' - W_f}$$

$$W_f = W_o' \frac{MR - 1}{MR}$$

For this simplified exercise, payload weight is defined as all weight over the weight of the empty vehicle and propellant required to complete the mission.

TABLE III. EARTH ORBIT DEPARTURE AND RETURN

						Vehicle A				Vehicle B				Vehicle C			
						Round Trip Payload		One-Way Payload Empty Return		Round Trip Payload		One-Way Payload Empty Return		Round Trip Payload		One-Way Payload Empty Return	
Mission Altitude (nm) Maneuvers	ΔV REQ	$g\,I_{sp}$ 32.2 (425) = 13 685	LNMR	MR	$\frac{MR-1}{MR}$	W_o' Light Off Weight	W_f' Prop Weight	W_o' Light Off Weight	W_f' Prop Weight	W_o' Light Off Weight	W_f' Prop Weight	W_o' Light Off Weight	W_f' Prop Weight	W_o' Light Off Weight	W_f' Prop Weight	W_o' Light Off Weight	W_f' Prop Weight
300																	
Transfer to Apogee ALT	350	13 685	0.025575	1.0259	0.025246	1 100 000	27 780			350 000	8836			50 000	1262		
Circ at ALT	350	13 685	0.025575	1.0259	0.025246	1 072 230	27 070	939K		341 164	8613	300K		48 738	1230	43.2K	
Retro to 100 nm																	
Perigee	350	13 685	0.025575	1.0259	0.025246	1 045 160	26 386	106 160	2680	332 551	8396	32 551	832	47 508	1199	4308	109
Circ at 100 nm	350	13 685	0.025575	1.0259	0.025246	1 018 774	25 720	103 480	2612	324 155	8184	31 730	801	46 308	1169	4199	106
						▲993 054		100 868		▲315 971		30 928		▲45 139		4093	
700																	
Transfer to Apogee ALT	980	13 685	0.071611	1.0743	0.069161	1 100 000	76 077			350 000	24 206			50 000	3458		
Circ at ALT	940	13 685	0.068688	1.0711	0.066380	1 023 923	67 968	841K		325 794	21 626	269K		46 542	3089	38.8K	
Retro to 100 nm																	
Perigee ALT	940	13 685	0.071611	1.0711	0.066380	955 955	63 457	114 955	7631	304 168	20 191	35 168	2334	43 452	2884	4653	309
Circ at 100 nm	980	13 685	0.068688	1.0743	0.069161	889 498	61 519	107 324	7422	283 977	19 640	32 834	2271	40 568	2806	4344	300
						▲827 979		99 902		▲264 337		30 563		▲37 762		4044	
2000																	
Transfer to Apogee ALT	2580	13 685	0.188527	1.2075	0.171843	1 100 000	189 027			350 000	60 144			50 000	8592		
Circ at ALT	2300	13 685	0.168067	1.1830	0.154691	910 973	140 920	627K		289 856	44 838	202K		41 408	6405	29.2K	
Retro to 100 nm																	
Perigee ALT	2300	13 685	0.168067	1.1830	0.154691	770 053	119 121	143 053	22 129	245 018	37 902	43 018	6654	35 003	5415	5803	898
Circ at 100 nm	2580	13 685	0.188527	1.2075	0.171843	650 932	111 858	120 924	20 780	207 116	35 591	36 364	6249	29 588	5084	4905	843
						▲539 074		100 144		▲171 525		30 115		▲24 504		4062	
6700																	
Transfer to Apogee ALT	5560	13 685	0.406284	1.5012	0.333866	1 100 000	367 253			350 000	116 853			50 000	16 693		
Circ at ALT	4240	13 685	0.309828	1.3632	0.266432	732 747	195 227	333K		233 147	62 118	109K		33 307	8874	16K	
Retro to 100 nm																	
Perigee ALT	4240	13 685	0.309828	1.3632	0.266432	537 520	143 212	204 520	54 491	171 029	45 566	62 029	16 527	24 433	6510	8433	2247
Circ at 100 nm	5560	13 685	0.406284	1.5012	0.333866	394 308	131 646	150 029	50 090	125 463	41 888	45 502	15 192	17 923	5984	6186	2065
						▲262 662		99 939		▲83 575		30 310		▲11 939		4121	
19 350																	
Transfer to Apogee ALT	8080	13 685	0.590427	1.8048	0.445922	1 100 000	490 514			350 000	136 073			50 000	22 296		
Circ at ALT	4850	13 685	0.354402	1.4254	0.298442	609 486	181 897	170K		193 927	57 876	58K		27 704	8268	9K	
Retro to 100 nm																	
Perigee ALT	4850	13 685	0.354402	1.4254	0.298442	427 589	127 611	257 589	76 876	136 051	40 603	78 051	23 294	19 436	5800	10 436	3115
Cir at 100 nm	8080	13 685	0.590427	1.8048	0.445922	299 978	133 767	180 713	80 584	95 448	42 562	54 757	24 417	13 636	6080	7321	3265
						▲166 211		100 129		▲52 886		30 340		▲7535		4056	
208 000																	
Transfer to Apogee ALT	10 300	13 685	0.752648	2.1226	0.528880	1 100 000	581 768			350 000	185 108			50 000	26 444		
Circ at ALT	2700	13 685	0.197296	1.2181	0.179049	518 232	92 789	165K		164 892	29 524	57K		23 556	4218	8.8K	
Retro to 100 nm																	
Perigee ALT	2700	13 685	0.197296	1.2181	0.179049	425 443	76 175	260 443	46 632	135 368	24 238	78 368	14 032	19 338	3462	10 538	1887
Circ at 100 nm	10 300	13 685	0.752648	2.1226	0.528880	349 268	184 721	213 811	113 080	111 130	58 775	64 336	34 026	15 876	8396	8651	4575
						▲164 574		100 713		▲52 355		30 310		▲7479		4075	

LN = Natural Log
MR - Mass Ratio

□ Round Trip Payload + Vehicle Weight Empty
▲ One-Way Payload

TABLE IV. LUNAR ORBIT DEPARTURE AND RETURN

						Vehicle A				Vehicle B				Vehicle C			
						Round Trip Payload		One-Way Payload Empty Return		Round Trip Payload		One-Way Payload Empty Return		Round Trip Payload		One-Way Payload Empty Return	
Mission Altitude (nm) Maneuvers	ΔV REQ	32.2 (425) 13 685	LNMR	MR	MR-1 / MR	W_o' Light Off Weight	W' Prop Weight	W_o' Light Off Weight	W' Prop Weight	W_o' Light Off Weight	W' Prop Weight	W_o' Light Off Weight	W' Prop Weight	W_o' Light Off Weight	W' Prop Weight	W_o' Light Off Weight	W' Prop Weight
300																	
Depart 100 nm LO*	2500	13 685	0.18268	1.2004	0.166944	1 100 000	183 639			350 000	58 430			50 000	8347		
Retro at 100 nm EPA**	10 150	13 685	0.74169	2.0995	0.523969	916 361	479 913	167K		291 570	152 693	57K		41 653	21 814	8.8K	
Circ at ALT	350	13 685	0.02538	1.0259	0.025246		11 019			138 875	3506			19 839	501		
Retro to 100 nm EPA**	350	13 685	0.02538	1.0259	0.025246		10 742	258 429	6524	135 369	3418	78 369	1979	19 338	488	10 536	266
Inject to Moon	10 150	13 685	0.74169	2.0995	0.523696	414 687	217 133	251 905	131 922	131 951	69 103	76 390	40 005	18 850	9872	10 270	5378
Retro to 100 nm LO*	2500	13 685	0.18268	1.2004	0.166944	197 554	32 980	119 983	20 030	62 849	10 492	36 385	6074	8978	1499	4892	817
						▲164 574		99 953		▲52 357		30 311		▲7479			
700																	
Depart 100 nm LO*	2500	13 685	0.18268	1.2004	0.166944	1 100 000	183 639			350 000	58 430			50 000	8347		
Retro at 100 nm EPA**	9520	13 685	0.69565	2.0050	0.501247	916 361	459 323	170K		291 570	146 148	58K		41 653	20 878	9K	
Circ at ALT	940	13 685	0.06869	1.0711	0.066380	457 038	30 338			145 421	9653			20 775	1379		
Retro to 100 nm EPA**	940	13 685	0.06869	1.0711	0.066380	426 700	28 324	256 700	17 040	135 768	9012	77 768	5162	19 396	1287	10 369	688
Inject to Moon	9520	13 685	0.69565	2.0050	0.501247	398 376	199 685	239 660	120 165	126 756	63 536	72 606	36 393	18 109	9077	9681	4852
Retro to 100 nm LO*	2500	13 685	0.18268	1.2004	0.166944	198 691	33 170	119 495	19 949	63 220	10 554	36 212	6045	9031	1508	4828	806
						▲165 521		99 546		▲52 666		30 167		▲7523		4022	
2000																	
Depart 100 nm LO*	2500	13 685	0.18268	1.2004	0.166944	1 100 000	183 639			350 000	58 430			50 000	8347		
Retro at 100 nm EPA**	7920	13 685	0.57874	1.7838	0.439399	916 361	402 648	180K		291 570	128 116	62K		41 653	18 302	9.5K	
Circ at ALT	2300	13 685	0.16807	1.1830	0.154691	513 713	79 467			163 454	25 285			23 351	3612		
Retro to 100 nm EPA**	2300	13 685	0.16807	1.1830	0.154691	434 246	67 174	254 246	39 330	138 169	21 373	76 169	11 783	19 738	3053	10 238	1584
Inject to Moon	7920	13 685	0.57874	1.7838	0.439399	367 072	161 291	214 916	94 434	116 795	51 320	64 386	28 291	16 685	7331	8654	3803
Retro into 100 nm LO*	2500	13 685	0.18268	1.2004	0.166944	205 781	34 354	120 482	20 114	65 475	10 931	36 095	6026	9354	1562	4851	810
						▲171 427		100 368		▲54 544		30 069		▲7792		4041	
6700																	
Depart LO*	2500	13 685	0.18268	1.2004	0.166944	1 100 000	183 639			350 000	58 430			50 000	8347		
Retro at 100 nm EPA**	4940	13 685	0.36098	1.4347	0.302990	916 361	277 648	232K		291 570	88 343	78K		41 653	12 620	12K	
Circ at ALT	4240	13 685	0.30983	1.3632	0.266432	638 713	170 174			203 227	54 146			29 032	7735		
Retro to 100 nm EPA**	4240	13 685	0.30983	1.3632	0.266432	468 539	124 834	236 539	63 024	149 081	39 720	71 081	18 938	21 297	5674	9297	2477
Inject to Moon	4940	13 685	0.36098	1.4347	0.302990	343 705	104 139	173 515	52 573	109 361	33 135	52 143	15 799	15 623	4734	6820	2066
Retro into 100 nm LO*	2500	13 685	0.18268	1.2004	0.166944	239 566	39 914	120 942	20 191	76 226	12 725	36 344	6067	10 889	1818	4754	794
						▲199 652		100 751		▲63 500		30 277		▲9071		3960	
19 350																	
Depart LO*	2500	13 685	0.18268	1.2004	0.166944	1 100 000	183 639			350 000	58 430			50 000	8347		
Retro at 100 nm EPA**	2420	13 685	0.17684	1.1934	0.162058	916 361	148 504	333K		291 570	47 251	110K		41 653	6750	16K	
Circ at ALT	4850	13 685	0.35440	1.4253	0.298393	767 857	229 123			244 319	72 883			34 903	10 415		
Retro to 100 nm EPA**	4850	13 685	0.35440	1.4253	0.298393	536 934	160 218	203 934	60 853	171 436	51 155	61 436	18 332	24 488	7307	8488	2533
Inject to Moon	2420	13 685	0.17684	1.1934	0.162058	376 716	61 050	143 081	23 187	120 280	19 492	43 104	6985	17 181	2784	5955	965
Retro into 100 nm LO*	2500	13 685	0.18268	1.2004	0.166944	315 666	52 699	119 894	20 016	100 788	16 826	36 119	6030	14 397	2403	4990	833
						▲262 967		99 878		▲83 962		30 089		▲11 993		4157	
208 000																	
Depart LO*	2500	13 685	0.18268	1.2004	0.166944	1 100 000	183 639			350 000	58 430			50 000	8347		
Retro at 100 nm EPA**								605K				195K				28.3K	
Circ at ALT	2700	13 685	0.19730	1.2181	0.179049	916 361	164 074			291 570	52 205			41 653	7458		
Retro to 100 nm EPA**	2700	13 685	0.19730	1.2181	0.179049	752 287	134 696	147 280	26 370	239 265	42 858	44 265	7926	34 195	5830	5895	1055
Inject to Moon																	
Retro into 100 nm LO*	2500	13 685	0.18268	1.2004	0.166944	617 591	103 103	120 910	20 185	196 507	32 806	36 339	6067	28 365	4735	4840	808
						▲514 488		100 725		▲163 701		30 272		▲23 630		4032	

* Lunar Orbit
** Earth Perigee Altitude

LN = Natural Log
MR = Mass Ratio

□ One-Way Payload Empty Return
▲ Round Trip Payload + Vehicle Weight Empty

It would include such weight as unused propellant and propellant reserves and/or possible upper body structures.

The specific impulse was conservatively selected at 425 seconds, with a view towards a 5 percent increase in I_{sp} time.

This increase would tend to balance any additional propellant requirements for mid-course corrections that were not included in the basic calculations. However, one example selected shows an approximate 5 percent payload loss for a 200 feet per second correction enroute and on return for a round-trip payload to 19 350 nautical miles (Earth synchronous orbit), as shown in Table V.

Boil-off losses were considered negligible and were not included in these rough calculations.

Two types of payload transfers were examined: (1) round-trip payload and (2) one-way payloads with the vehicle returning empty.

TABLE V. ESTIMATED PAYLOAD ADJUSTMENT FOR MIDCOURSE CORRECTION ROUND TRIP PAYLOAD TO 19 350 NAUTICAL MILES*

	▲V	$\frac{MR-1}{MR}$	W_o' Lift Off Weight	W_T Weight of Prop
Depart LO**	2500	0. 16694	1 100 000	183 634
Midcourse	200	0. 01449	916 364	13 278
Retro at 100 nm	2420	0. 16206	903 086	146 354
Circ at Alt	4850	0. 29839	756 732	225 801
Ret to 100 nm	4850	0. 29839	530 931	158 425
Inject to Moon	2420	0. 16206	372 506	60 368
Midcourse	200	0. 01449	312 138	4523
Retro Moon	2500	0. 16694	307 615	51 353
	Payload + Vehicle Weight Empty		256 262	
	Payload, No Midcourse Correction			162 967
		With Midcourse Correction		156 262
			% Payload Loss =	$1 - \frac{156\ 262}{162\ 967}$
				= 5% Approx

* I_{sp} = 425 sec; vehicle weight = 1 100 000 lb

** LO = lunar orbit

The maximum one-way payload was iteratively arrived at by working the problem until the weight that was returned approximated (within 1000 pounds), the weight of the vehicle empty.

The missions examined were arbitrarily selected (except for Earth synchronous orbit, 19 350 nautical miles, and libration point at lunar altitude, 208 000 nautical miles, which are of particular interest).

The payload figures for vehicle B have been reduced to payload fractions by dividing them by the launch weight of the vehicle. These vehicle payload fractions are considered representative and are combined with the lunar and Earth surface to orbit payload fractions, 0.40 and 0.045 respectively, to arrive at the curves shown in Figure 4 and tabulated in Table VI.

TABLE VI. TOTAL SYSTEM PAYLOAD FRACTIONS VERSUS MISSION ALTITUDE*

Mission Altitude	300	700	2000	6700	19 350	208 000
Earth Launch						
Round Trip	0.037	0.030	0.018	0.007	0.003	0.003
One Way/Empty Return	0.039	0.035	0.026	0.014	0.007	0.007
Lunar Launch						
Round Trip	0.026	0.026	0.028	0.038	0.062	0.153
One Way/Empty Return	0.065	0.066	0.071	0.089	0.126	0.223

* Derived from vehicle B payloads

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LUNAR GRAVITY AS A POWER SOURCE

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One of the intrinsic quantities that is associated with any body in space is gravity. The magnitude of the gravitational force varies directly with the mass of the body. Even the moon, with its very small mass when compared to most other bodies in the universe, provides an acceleration due to gravity of 1.62 m/sec^2. This gravity can be put to use by our astronauts when they reach the moon to provide power, not only for life support, but also for other programmed activities of lunar exploitation.

The purpose of this paper will be to focus attention on the gravity of the moon as a source of power and suggest that it be seriously considered. Therefore, all the ramifications and detailed analysis that usually accompany any subject dealing with projected use of the moon will not be discussed at this time. It is hoped that this source of indigenous power will look enticing enough so that in the future it can be thoroughly investigated and its cost effectiveness ascertained, as compared to other sources of power that will be put forth for use on the moon.

In the past, many techniques for generating power on the moon have been devised. One of these is the rather speculative, but very intriguing and practical use of geothermal areas. In the last two years the use of this source of power has gained great momentum; personnel working on Project Moon Blink have observed transient phenomena on the moon which can be explained only as being caused by volcanic processes. Several other techniques have involved the transportation of power sources from the Earth. The latter sources usually look more interesting to most people because they have already been developed and used on the Earth and are reliable, whereas the former source is still speculative. The geothermal source will, of course, remain speculative until we either have astronauts on the moon who find such areas or else one of our lunar probes like Orbitor with its high resolution cameras, or Surveyor with its sampling equipment sends back information which can be uniquely associated with geothermal processes.

Whatever the suggested source of power, or no matter how speculative or far-fetched it may seem at the present, it should still be considered as having some potential. Ultimately, the deciding factor will be either cost, availability, desirability, or necessity.

Gravity is one of the known factors associated with the moon. As a power source it is neither speculative as are geothermal areas, nor does it have to be transported from Earth as would nuclear power plants or batteries. Gravity is also a constant quantity that can be used day in and day out whereas those sources, such as solar cells, which rely upon the sun to generate power can be used only in 14-day increments and then must either be shut down or make use of batteries during the lunar 14-day night.

Naturally, some of the equipment necessary to make use of the moon's gravity will have to come from the Earth. It can be very simple equipment in the beginning of exploitation and expanded to be more complex as time goes on and as more material can be brought from the earth. In other words, it can be used to produce power with very little material initially, but can grow to any desired point.

Any method of utilizing gravity requires that material possessing a high potential energy may be taken to a place of lower potential energy and in the process release the difference of energy in the production of power. The potential energy of material can be defined as being proportional to the elevation or altitude of the material, multiplied by its mass and the pull acting on it because of gravity. As the moon's gravity is a small number, large masses of material must be moved over small changes of elevation or smaller amounts must be moved through large changes of elevation in order for any given quantity of power to be produced. The moon has a tremendous amount of relief almost anywhere on its surface and therefore the last criterion can be the easiest to meet.

Let us look at the surface of the moon to determine the kinds of relief that are available. Baldwin, in his book "The Surface of the Moon," gives an excellent coverage of this subject. Just a few examples here will serve to illustrate what changes of elevation might be expected. Several mountain ranges exist on the moon. The Pyrenees average 6000 feet in height and the Altai Mountains are from 4000 to 6000 feet high with some of the peaks rising to 11 000 and 13 000 feet. There are valleys which are very deep. Alpine Valley has a depth of 10 000 feet and Rheita Valley averages 5000 feet in depth with a maximum of 11 000 feet.

Without going into a discussion of the origin of the craters, they also show great changes in relief and because they are so numerous, they will be of greater advantage than either the mountains or the valleys. There are large craters like Clavius which has a diameter of 146 miles and depth of 16 100 feet. Its rim extends above the surrounding area by 5000 feet. Other craters of small dimensions exist on down to ones such as Piazzi Smyth with a diameter of 6 miles, a depth of 3500 feet, and a rim of 2100 feet high. Then there are craters numbering in the thousands and probably hundreds of thousands which are of the order of a mile in diameter, a thousand or so feet in depth and have rims a few hundred feet high. Lunar probes have shown that there are countless small craters. In general, the smaller the crater the greater is the ratio of its depth to diameter. Craters under 18 miles in diameter have sides that slope approximately 30 degrees, while in larger craters the slope decreases to about 10 degrees.

When making use of the lunar relief with its gravity, the smaller craters look most promising. There are two reasons for this: (1) they have steeper sides, although vertical cliffs with a 1000-foot relief should be found quite extensively, and (2) they are not only numerous, but are evenly distributed over the surface of the moon and should therefore be quite accessible.

There are three main methods of using the gravitation field of the moon. They are, in order of greatest usefulness: (1) tramways which transport rocky material from an elevated place of working to a lower dump area, (2) pipelines with rock suspensions in fluids systems, and (3) fluids systems themselves in connection with solar radiation. Each method has its own particular problems and disadvantages. Each will be discussed in the reverse order from which they appear above.

The fluid-solar radiation pipeline system must take advantage of as large a change in specific gravity as possible. At the bottom of the closed pipeline cycle the fluid is heated, decreasing its specific gravity, and at the top it must be cooled by radiation to increase its specific gravity so that gravity can now act on the difference in specific gravity to pull the fluid back down the pipe. There it can be used to turn turbines and produce electrical power. Very few fluids have a specific gravity greater than one and water is a good example. The greatest change in specific gravity that can be obtained from water, as with any fluid, is when it is converted from the liquid to the gaseous phase or vice versa; the specific gravity difference being approximately one.

Several drawbacks to this type of power generation are as follows:

1. The number of fluids that can be used under the range of temperature conditions that exist on the moon is quite small.

2. Solar energy, as in several of the other proposed power generating devices, can be used only on alternating 14-day periods.

3. As stated above, most fluids can, at most, yield a specific gravity difference of only about one which limits gravitational possibilities.

4. The small force produced by gravity on a liquid such as water is almost nullified by flow friction when small pipes are used.

All of the stated drawbacks can be circumvented in one way or another, but, in turn, new disadvantages are brought into being. For instance, in item (3) a liquid with a large specific gravity could be used. One that might come readily to mind is mercury. However, mercury does not quite fit into the temperature range of the moon. Its boiling point is about 356°C, whereas the sub-solar point of the moon at noon is only 134°C. Assuming that this problem might be solved, let mercury be used for the system and further explore the possibilities. It has a specific gravity of 13.5 which certainly would appear acceptable if the total difference of 13.5 were used. But any fluid, at least for the present, for use in the pipeline would have to be taken from the Earth and the weight involved would be so great as to preclude its feasibility. Likewise, the flow friction in item (4) can be decreased by using larger pipes, but this adds greater weight requirements for the pipe and the liquid being used. Although this method of utilizing lunar gravity appears rather extreme, it should not be discounted. As our technology advances, currently insurmountable problems could conceivably be resolved by this method.

The second method is that of using a liquid pipeline system with solids placed in suspension to provide the specific gravity difference. Some of the problems that were encountered in the previous method are still present although others are eliminated. Flow-friction problems are intensified because of the added solids, but the liquid now does not have to be heated to change states, and reliance upon the sun is obviated. However, new problems are encountered. Getting the solid material into and out of the system brings up many questions. This method appears to be more interesting than the fluid-solar pipeline and can be of value in disposing of waste material from mining operations. At present, mining has a top priority position among the projects that will be carried on in the initial stages of lunar exploitation.

The first method listed, that of utilizing lunar gravity, is by far the most intriguing and offers the greatest possibility for success. Tramways of one type or another have been used for many years to transport material and passengers. This method usually requires energy to be put into the system but in several cases in which changes of elevation are involved, a regenerative braking arrangement is used to create electrical power. Some examples of both types are given below.

1. Probably one of the largest users of tramlines and one that most people are familiar with are ski resorts. In this case most of the transportation is uphill and therefore energy is being put into the system.

2. At the Kennecott Copper Mine at Bingham, Utah, the ore trains which come down around the open pit have regenerative braking which produces electrical power that can be sold to the Utah Power and Light Company.

3. Union Carbide uses a tramline at Pine Creek, California, for bringing material down the mountain from their mine which is able to produce power to run their mining operation.

4. Published data from the operation of the Wesport Stockton Coal Company in New Zealand shows that they produce 288 horsepower in surplus power from traming coal from one of their mines down to the sea in five distinct tramline sections.

These examples serve to illustrate that the method is workable on the Earth at the present time and can offer great possibilities for generating power on the moon. On the moon a tramline can be installed along with any mining operation, and because of numerous craters in all areas, plenty of available dump room at a lower elevation can be found.

A 50-kilowatt power plant would require only a vertical drop of 1000 feet (about 350 m) in elevation and roughly a flow rate of 10 kilograms of material per second being poured into the tramline buckets. For this short distance only a loading terminal and a dumping terminal would be necessary. If longer distances were to be used, a few trestles, in addition to the two end-terminals, would be required. Because of the decrease in gravity over that of Earth, fewer trestles for any given length would be needed. Where vertical drops could not be used and the steepness of the slopes decreased, greater distances are required to gain the 1000-foot drop in elevation and keep the capacity of the plant up to 50 kilowatts. Several parameters can be varied to change the output of the plant. Bucket sizes can be varied, the flow rate can be changed, and the amount of elevation change can be increased or decreased.

A tramline has the advantages of: (1) being very efficient, (2) not requiring excessive maintenance, (3) having very few moving parts, and (4) being either simple or complex as the situation demands.

The power output is high for the amount of equipment used. All that is really necessary is a cable, buckets, moorings and some type of electrical generating device. For the early stages of lunar exploration, simplicity of use, ease of installation, and reliability, coupled with low-weight requirements are absolutely necessary. For this reason, tramlines making use of gravity can be of great importance.

In summary, there are three types of systems that can be used with lunar gravity for producing power. They are fluid-solar pipeline systems, fluid-solid suspension systems, and tramlines. Of these, tramlines offer the greatest advantages and appear to be very practical for early lunar use.

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ROCKET PROPELLANTS FROM GEOTHERMAL EMISSIONS

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SUMMARY

There is a definite possibility that active volcanism may still exist on the moon. If so, the associated geothermal fluids will supply a useful source of propellants, power, and life support. In the present paper, possible compositions of lunar geothermal fluids are considered. Based on this study, a "standard composition" was assumed for use in processing studies. This mixture consists of water (49 percent), carbon dioxide (41 percent), hydrogen chloride and hydrogen sulfide (3.5 percent each), and hydrogen and carbon monoxide (1.5 percent each). The performance of various rocket propellants that could be manufactured from these and several other geothermal gases was investigated. As a result of this work plus cursory mission studies, four processes were chosen for more detailed study. These were: (1) hydrogen for nuclear rockets, (2) hydrogen and oxygen for chemical rockets and fuel cells, (3) methane and oxygen, and (4) methanol and oxygen. Flow charts were developed for the processes, and energy and mass flow requirements were given. Life support water, oxygen, and power were byproducts of each processing scheme. Technically, the feasibility of producing useful quantities of rocket propellants on the moon from geothermal emissions was established.

INTRODUCTION

A growing body of evidence indicates that remnants of past lunar volcanism may still exist. If such is the case, the associated geothermal fluids could provide useful power, life support fluids, and rocket propellants. These commodities would be most valuable in support of lunar exploration and exploitation and for follow-on space ventures. The potential benefits that could be derived from such emissions are sufficiently great to warrant study in the area of lunar geothermal utilization.

The prime objective of the present studies is to explore those rocket propellants that might be obtained from geothermal emissions and how such propellants could be produced. To do this, it is first necessary to estimate what fluids might be present on the moon. Propellant combinations that might possibly be manufactured from these fluids were evaluated in light of their rocket motor performance. From this information, plus that derived from a brief examination of possible lunar missions requiring rocket propellants, four propellant manufacturing processes were chosen for further detailed study. Flow sheets were developed for these processes along with mass and energy balances. Process equipment was not designed to the point where weights could be determined, and economic studies were not undertaken.

No attempt will be made in this paper to establish the existence of lunar geothermal deposits. The topic has been discussed widely throughout the literature and was summarized by Dr. Carl F. Austin at the Fourth Annual Meeting of the Working Group on Extraterrestrial Resources [1].

The present studies are an outgrowth of applied research work in the field of geothermal utilization at the Naval Ordnance Test Station (NOTS), China Lake, California. The lunar propellant studies also served to fulfill Master's thesis requirements [2] for one of the authors (R. D. Fulmer) at the University of California at Los Angeles.

GEOTHERMAL FLUIDS

A geothermal fluid is defined for purposes of this paper as a water-rich accumulation of gases with associated brines, heated by and derived at least in part from magmatic and metamorphic activity. Assuming such emissions do exist on the moon, what components are likely to be present? To determine this, terrestrial geothermal emissions were first studied.

The assumption is made that (1) the Earth and the moon have common rock types and a common origin for their magmatic activity, and (2) the basic geological processes associated with magmatic defluidization will be similar in both environments [2]. With these assumptions, some idea of the lunar geothermal fluid composition can be obtained by extrapolating the terrestrial compositions to the moon with due consideration being given to basic differences in environment.

Terrestrial geothermal emissions are characterized by very high water or steam content--often running above 98 percent by weight of the total fluid. A majority of this water results from circulating ground waters that are heated either by (1) heat transfer with the parent body of molten rock (magma) or (2) contact and mixing with the less voluminous fluid components expelled by the cooling magma. Since the existence of extensive ground water systems on the moon is quite unlikely, any geothermal gases seen in the lunar environment should be a result of the magmatic defluidization. As a result, the water content will be considerably lower than that observed on Earth. D. E. White [3] estimates that around five percent or less of the water seen in terrestrial geothermals results from magmatic defluidization.

With the exception of water, much of the remaining nonaqueous gas content observed in terrestrial geothermals probably results from the magmatic defluidization. A few components may be contributed by chemical and physical reactions with the conduit system. Figure 1 shows percentages for each of the primary geothermal gases as observed for a number of terrestrial emissions. The data are a further summary of the White and Waring study [4]. As can be seen, CO_2 is usually the dominant nonaqueous gas. This gas along with H_2S, HCl, CO, H_2 and HF appears to be magmatic in origin and, thus, a likely candidate for lunar geothermals. A magmatic origin is difficult to establish with certainty for any gas, but geochemical studies plus physical evidence, such as gas and fluid inclusion in igneous rocks, provide clues.

Sulfur dioxide is very often observed in geothermal emissions and may in some cases be magmatic in origin. Much of the SO_2 observed, however, originates as a result of chemical reaction between H_2S and atmospheric oxygen. The origin of methane in volcanic emissions is also open to some question. Although much of what is observed on Earth results from thermal distillation of organics in the rock structure, fluid

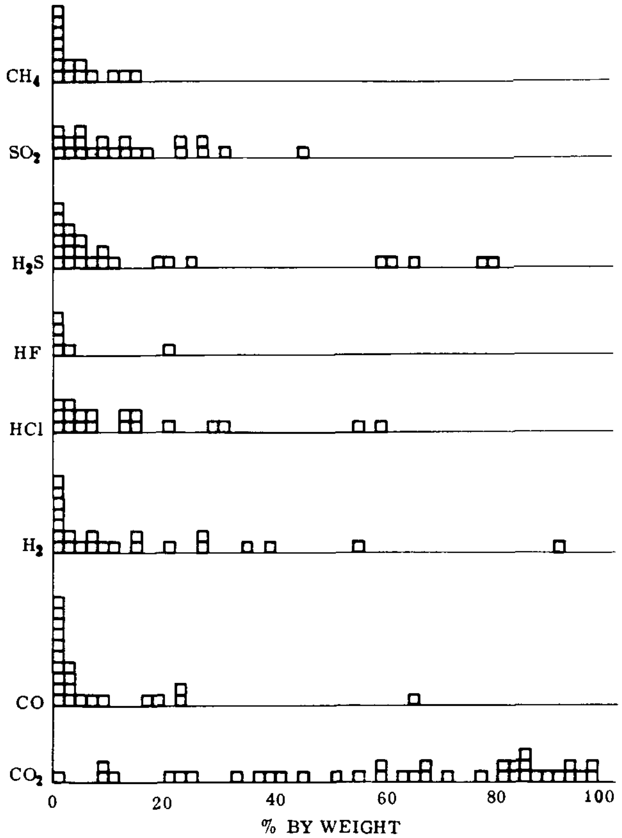

FIGURE 1. COMPOSITION OF NONAQUEOUS FLUIDS FOR SELECTED TERRESTRIAL GEOTHERMAL DEPOSITS

inclusions in igneous rocks indicate that methane may at times be magmatic in origin and, thus, a possibility for lunar geothermal fluids. The spectral observations of carbon bearing gas emissions in lunar crater Alphonus by N. A. Kozyrev [5] indicate the possibility of perhaps even higher chain hydrocarbons if, as assumed here, these gases can be credited to geothermal emissions. A number of other gases, such as the rare or noble gases, have also been observed in minor quantities on occasion.

The lack of extensive circulating ground waters on the moon, as assumed in this study, should result in a much hotter emission than exists on Earth. After even a relatively short geologic time, the conduit system would likely reach an equilibrium condition whereby little heat should be lost from the system. Thus the gas at the surface could be at nearly the temperature of the defluidizing magma. As a result of the high temperature, some of the higher boiling point sublimates and salts should be more in evidence. On Earth these compounds are often observed around hot volcano vents, but are absent in most of the water-rich geothermal deposits where the heat source is at a significant depth and abundant water is present. In the latter case, the sublimates are cooled at depth by circulating water and then disseminated within the Earth. Although relatively little data are available on the magmatic origin of the compounds, the chlorides of ammonia and alkali group metals have often been observed and are thought to be magmatic. Likewise at Niland, California, very deep chloride rich geothermal brines were tapped which contained many of the metallic ore elements including significant quantities of silver and copper. Traces of mercury and abundant boron (boric acid) are both widely associated with geothermal emissions.

At several places in the literature, authors propose the existence of large quantities of sulfur on the moon. This conclusion results from observation of the large occurrences of sulfur found in terrestrial volcanic areas. A majority of this sulfur, however, results from the oxidation of hydrogen sulfide by atmospheric oxygen. With little if any free oxygen available on the moon, the volcanic sulfur output will be reduced drastically.

In addition to the chemical composition of the geothermal fluids, knowledge of their thermodynamic state is also required before processing studies can be considered. Temperature was mentioned earlier in discussing sublimates. The other parameter of interest is pressure.

The maximum fluid pressure possible at any depth is equal to the static load of the bed rock above the point of interest plus the shearing strength of the rock structure [6]. Pressure higher than this would result in fissuring of the overlying rocks and an eruption. Because of the lower lunar gravity, the maximum possible pressures at any well depth will be lower than at equivalent depths on earth (assuming equal rock shearing strengths). Well pressures on earth rarely reach the maximum value. The condensing effects of the ground waters very likely influence this situation. With little or no circulating ground waters on the moon to cool and quench the gases, one would expect higher geothermal gas pressures near the surface. This would result in a larger number of gas eruptions and associated craterings on the moon than is experienced on Earth.

Based on results of the study, a reasonable but somewhat arbitrary lunar geothermal fluid composition was chosen upon which to base processing studies. This was done by first assuming a terrestrial composition composed of 95 percent water with five percent nonaqueous gases. Using the White [3] maximum figure of five percent for the amount of magmatic (and in turn lunar) water included in the total geothermal water, it was determined that about 49 percent of the lunar geothermal fluid would be water, and the remainder nonaqueous gases. The choice of composition of the nonaqueous gases was based to some extent on the relative quantities shown in Figure 1 and the relative frequency with which they were observed on Earth. The composition chosen included:

49. 0% H_2	3. 5% H_2S
41. 0% CO_2	1. 5% CO
3. 5% HCl	1. 5% H_2

A temperature of 700° C was chosen for the fluid. This is intermediate in the crystallization range of granite, given by White [3] as 900°C during early magma crystallization to 500° C during final stages of defluidization. A pressure of 90 psi was chosen. This pressure corresponds to a lunar static rock load at a depth of about 500 feet. The density of lunar rock is not precisely known other than from gross averages, nor will the well pressure developed at any lunar site necessarily equal the static load. The presumption is made, however, that a well of reasonable depth can achieve a 90 psi working pressure.

PROPELLANT EVALUATION

From the chemical elements that might be found in lunar geothermal emissions, a list of possible rocket propellants was developed. Because almost all the elements composing the major propellants can also be found in geothermal emissions at one time or another, it is obvious that the list can be quite extensive. By a cursory examination of the processes involved in manufacturing most propellants, the list was reduced considerably. Only those propellants that can be manufactured with no (or minor) expendable chemicals imported from Earth were considered. Likewise, processes involving very complex or extensive operations were not considered. Fuels chosen for performance calculations were: (1) hydrogen, (2) methane, (3) methanol, (4) carbon monoxide, (5) water, and

(6) boron. Possible oxidizers were: (1) oxygen, (2) fluorine, (3) chlorine, and (4) chlorine trifluoride. The fluorine and chlorine trifluoride, of course, depend upon the presence of hydrogen fluoride. As noted in the previous section, HF is not very common in terrestrial geothermals and when present is usually in minor amounts. Both fluorine and boron would be difficult to produce on the moon using present commerical techniques. Fluorine (from HF) was considered in the study because of its outstanding performance. Boron was included because of its use as a rocket propellant and because boric acid has been widely produced commercially from geothermal fluids throughout the world.

Various propellant combinations were examined with the aid of the Propellant Evaluation Computer Program of NOTS [7]. This thermochemical program calculates: (1) the adiabatic flame temperature and mole fraction of chemical species in the rocket combustion chamber under conditions of thermal equilibrium, and (2) the temperature, flow velocity, and mole fraction of species at the rocket nozzle exit based on one dimensional, isentropic expansion from the combustion chamber. Values of exit parameters are calculated for both shifting and frozen chemical equilibrium expansion processes (equilibrium corresponding to infinite and zero chemical reaction rates, respectively). The propellant composition, combustion chamber pressure and nozzle exit pressure are the required input data.

Of primary interest to this study was the specific impulse (I_{sp}) of the propellant calculated by the propellant performance and is defined as the amount of thrust produced per unit weight flow rate of propellants. Calculations were made over a range of oxidizer-to-fuel (O/F) ratios to determine the maximum value of I_{sp}.

Results of the propellant evaluation study are given in Figure 2. The combinations examined are listed along with their maximum I_{sp} (represented in a bar-type chart). Oxidizer-to-fuel ratio at maximum I_{sp} is also tabulated. As would be expected, hydrogen is an excellent fuel, and fluoride and oxygen are excellent oxidizers. Chlorine is quite disappointing as an oxidizer, especially in view of the fact that the chloride ion is quite common in geothermal deposits. Likewise, hydrogen sulfide, another common geothermal product, does not appear very satisfactory.

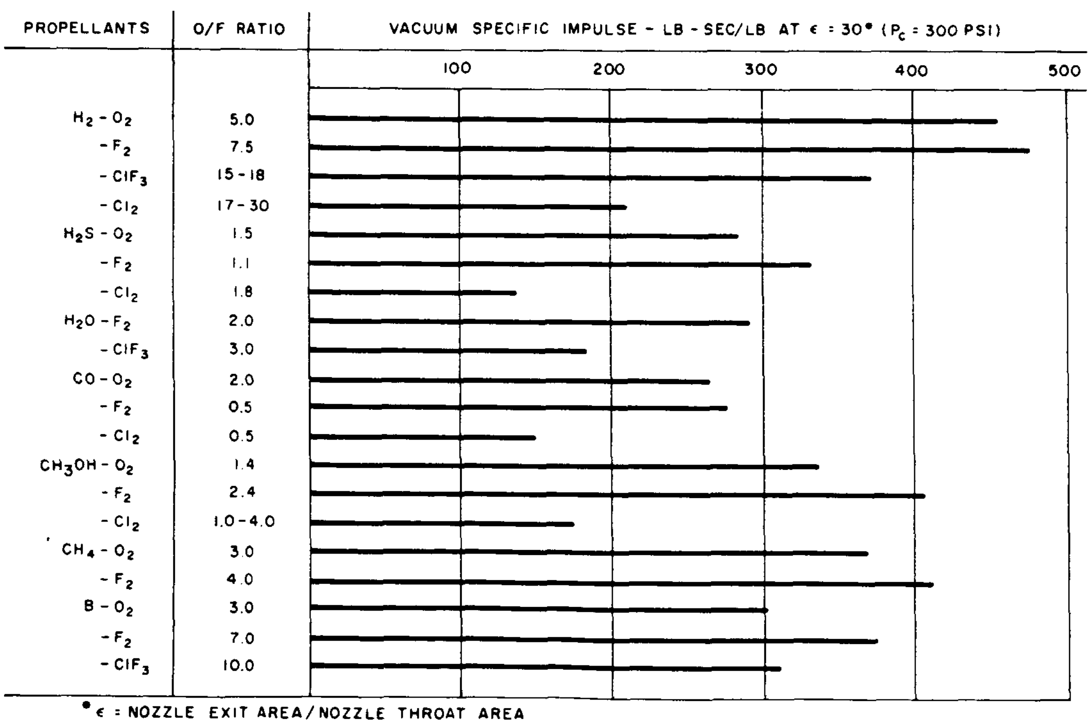

FIGURE 2. MAXIMUM VACUUM I_{sp} FOR GEOTHERMAL PROPELLANTS

MISSION CONSIDERATIONS AND PROCESS SELECTION

To select propellants for processing studies, it was necessary to examine some rocket missions that might require propellants manufactured on the lunar surface.

Three main missions were selected:

1. Lunar surface exploration and exploitation.
2. Further space exploration.
3. Return trips to Earth.

In surface exploration and exploitation, both land and "flying" vehicles will be necessary. The land vehicles will most likely be powered by fuel cells and/or internal combustion engines of some type. In both bases, the hydrogen-oxygen combination would be well suited. Methanol-oxygen fuels cells are well on the way toward operational status and would eliminate at least one cryogenic material. Methane would also be a likely material for fuel cell operation at some date in the future.

The flying vehicles would be rocket powered and could also use the hydrogen-oxygen combination. A storable fuel such as methanol, however, would be easier to handle than cryogenic hydrogen. If they were readily available in large quantities, chlorine and/or hydrogen sulfide could be used as propellant where long ranges and high I_{sp} were not needed.

For further space efforts, such as Mars exploration missions, either hydrogen for nuclear rockets or hydrogen and oxygen for chemical rockets would be most beneficial. Both could be manufactured from the geothermal gases.

For return trips to Earth, hydrogen-oxygen would be a likely choice. However, if the hydrogen-oxygen combination were in demand for space flight purposes, it might be possible to design transport vehicles to operate on by-product chlorine and hydrogen sulfide. Further detailed analysis is needed in this area before a firm judgment can be made.

From the above considerations and earlier examinations of the manufacturing techniques, four processes were chosen for closer study. These were:

1. Manufacture of hydrogen for nuclear rockets.
2. Manufacture of hydrogen and oxygen for chemical rockets and fuel cells.
3. Manufacture of methane and oxygen for chemical rockets.
4. Manufacture of methanol and oxygen for chemical rockets and fuel cells.

As will be noted later, chlorine and hydrogen sulfide are relatively easy to manufacture as by-products in the above schemes.

PROCESSING SCHEMES

The following schemes can be sensitive to the assumed geothermal inputs (composition, pressure and temperature). A standard working composition is necessary, however, to allow detailed examination of the problem. The results provide insights into the overall problems of generating rocket propellants from geothermal fluids and establishing a departure point from which other compositions can be explored.

In each of the schemes considered, all process power was derived from the thermal potential of the geothermal fluid. Likewise, sufficient life support water, oxygen, and power were provided as a by-product to sustain a base population of 20 people (only three or four of this number would be required for the propellant plant). The life support requirements were estimated to be: 1 000 lb/day water, 100 lb/day oxygen, and 10 kilowatts of electrical power.

Although equipment weight was not calculated as such, efforts were made to keep total equipment weight and complexity to a minimum. In many cases, choices in techniques were necessary. Where it was obvious, those requiring less power and equipment were chosen. In many cases, however, more detailed trade-off studies are necessary.

In general, the schemes were power limited. It was necessary to process more geothermal fluid than needed strictly for the chemical content alone. Although this is not overly serious because the fluid is essentially free, an attempt was made to use processes requiring a minimum of energy.

Well drilling procedures will not be discussed here. Dr. Carl F. Austin discussed this problem briefly in his paper [1] presented to the Fourth Annual WGER meeting. For the present, it will be assumed that a geothermal well is available which produces the standard composition fluids in quantities sufficient to the needs.

Hydrogen Production for Nuclear Rockets

The scheme shown in Figure 3 was developed to produce hydrogen for use in nuclear rockets from the standard lunar geothermal fluid. As noted earlier, life support oxygen, water, and power were also generated.

The fluid emerges from the well at the standard pressure of 90 psi and a temperature of 1300° F. It passes through a boiler-heat exchanger (B1) where heat is transferred to a secondary water-steam loop. Power is extracted using a turbine (T1) and a Rankine cycle. Heat is rejected at R1 with orienting radiators. Exit temperatures from this boiler run around 300° F with the pressure still in the 90 psi range. At this temperature and pressure, essentially all of the high boiling point materials have condensed in the boiler. These materials are continuously removed from the large boiler tubes with screw-type scrapers. These scrapers clear the walls of condensed materials and carry them to the downstream ends of the tubes where they are separated and collected. Although not discussed to any degree in this paper, these solids may provide valuable by-products. The temperature is maintained sufficiently high to prevent water condensation in this section.

The fluid then passes into boiler B2 where the geothermal water is condensed and the temperature lowered to about 285° F. The heat is transferred to a secondary loop; this time SO_2 is used as the secondary working fluid. Again a Rankine cycle with radiator reject is used for the power take-off and heat rejection. It should be noted that in both power cycles, the secondary working fluid is available from the geothermal fluids (SO_2 being derived from the H_2S). In condensing to the liquid form, the water also absorbs the HCl from the fluid.

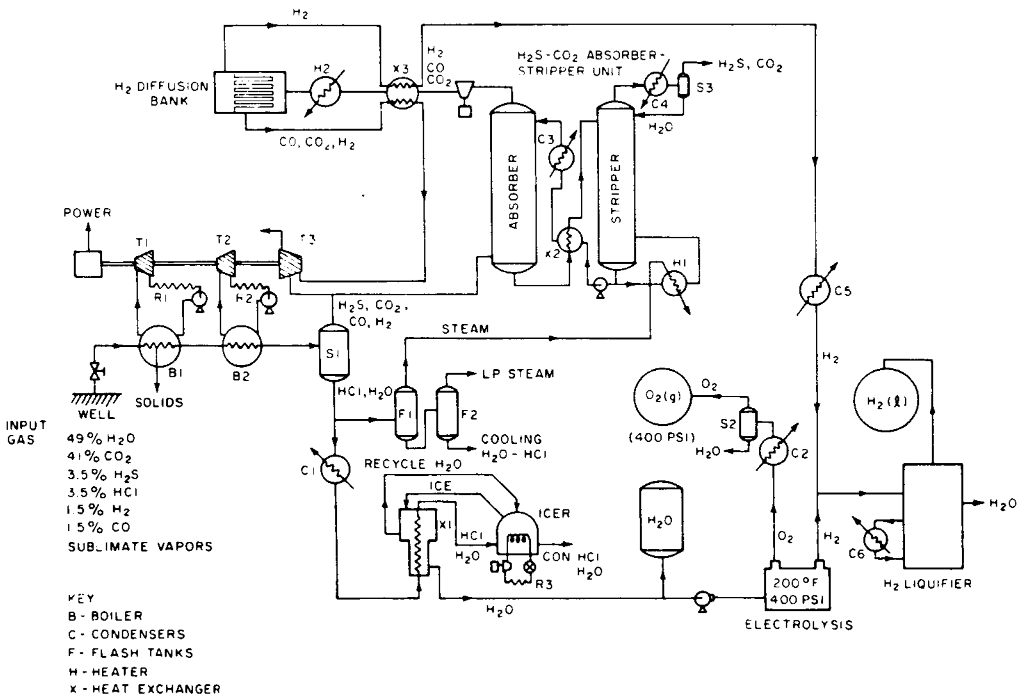

FIGURE 3. PRODUCTION SCHEMATIC: H_2 PROPELLANT AND LIFE SUPPORT FLUIDS

Separation of the gas and fluid mixtures occurs in S1. The liquid stream composed of 6.7 percent aqueous HCl solution exits from the separator. A majority of the fluid is flashed to vapor at F1. The resulting steam (about 10 percent of the total input) is used for process heat. Very little of the HCl goes into the gas phase. The remaining HCl from F1 is sent to F2 where it is further flashed down to a pressure somewhat below one pound per square inch. The temperature of the resulting HCl solution is lowered to about 80° F. The steam or vapor is rejected, but the HCl solution makes an excellent cooling solution. Because the solution is at its boiling point at 80° F, it can absorb large quantities of heat in changing phase upon heating. Sufficient quantities of this fluid are available to take care of all cooling needs other than those associated with the heat rejection at boiler B1 (R1), at boiler B2 (R2) and at the icer (R3). Sufficient surplus is available to take a portion of the heat load at R1, if a mixed heat rejection process appears desirable.

The remaining HCl solution is cooled in cooler C1 and heat exchanger X1 to a temperature of about 40° F. It enters the icer which is similar to a flake ice machine. A large refrigerated cylinder revolves with its lower segment immersed in the HCl solution (now cooled to 15° F). Pure ice forms on this lower portion of the cylinder. As the ice film emerges from the liquid, the adhering HCl solution is washed off with small quantities of recycle water. At the top of the cylinder's revolution, the ice is scraped from the cylinder. It is fed by screw conveyor to the X1 heat exchanger where it is dumped into an ice-water slurry and is used to cool the incoming HCl stream. The HCl is concentrated in the icer to about 33 percent and rejected.

If chlorine is desired as a by-product, the concentrated HCl is sent to electrolysis cells where chlorine and hydrogen are liberated. Diluted HCl is then returned to the icer cycle. However, for the present purposes chlorine is not needed, and the hydrogen generated, as in the case of water electrolysis, requires more energy expenditure than processing the elemental hydrogen in the geothermal fluid directly.

Part of the water from X1 is stored for life support and process water needs (make-up). The remainder is sent to a high pressure electrolysis unit where it is broken down into oxygen and hydrogen. For the reason mentioned above, only enough water is electrolyzed to provide the required life support oxygen. Electrolysis strictly for producing hydrogen is not attractive here. The oxygen is cooled, the water separated out, and the gas stored at a pressure of about 400 psi.

Returning to the gas stream exiting from separator S1, it is apparent that a portion of this gas is run through a turbine to generate additional power. It is then rejected to the environment. (The three turbines are shown on a common shaft primarily for convenience. This arrangement is not necessarily optimum). The process gas stream is sent to an absorber-stripper unit where essentially all of the H_2S and about 75 percent of the CO_2 is removed. Sodium carbonate will be imported from earth initially. Small amounts of make-up carbonate will be necessary over a period of time.) After the carbonate solution has absorbed the required amount of H_2S and CO_2, it is regenerated at reduced pressures using steam stripping [8]. Hydrogen sulfide is absorbed faster then the CO_2. At some minimum liquid-gas contact time, essentially all the H_2S and an amount of CO_2 equivalent to about 20 percent of the H_2S flow is absorbed. By increasing the size of the column, hence the contact time, larger quantities of CO_2 can be absorbed.

The gas exiting from the absorber is now composed of about 74 percent CO_2, 13 percent CO and 13 percent H_2 by weight (75 percent by volume H_2). The gas is compressed to 300 psi. It is heated first by gases from the H_2 diffusion bank and finally by geothermal fluids directly from the well to increase the stream temperature to 900° F.

A palladium alloy diffusion bank is used to separate out pure hydrogen [9]. The CO_2, CO and residue hydrogen that does not diffuse through the membranes is recycled through the heat exchanger and is relieved of some thermodynamic energy by passing it through turbine T3 along with a portion of the gases from separator S1.

The hydrogen stream, now at a pressure of about 10 psi, is passed through heat exchanger X3 where it is cooled to 510° F. In cooler C5 the temperature is lowered further to about 90° F. The H_2 is then sent to the liquifying unit, where, along with the hydrogen from the electrolysis unit, it is liquified, converted to parahydrogen, and stored.

Among the questions that might arise as a result of this process is whether it is better to process the geothermal fluids for the 1.5 percent free hydrogen or use electrolytic methods. In the first approach, an absorber stripper unit, a diffusion unit, and a compressor, plus a number of heat exchanger units, are required. This is balanced against substantially larger power generating equipment, icer and electrolysis units plus a large surplus of O_2. As will be seen shortly, the electrolysis approach is used in the next scheme where both hydrogen and oxygen are needed. Comparing the two processes, based only on hydrogen production, it turns out that for the size plants considered, it required 6.95 kW/lb H_2 for the diffusion technique compared to 28.5 kW/lb H_2 for the electrolysis scheme. Likewise, it is necessary to process 112 pounds of weil fluids (including that strictly for power generation) to obtain one pound of hydrogen in the diffusion scheme. With the electrolysis approach, 420 pounds are necessary. A closer examination of the problem with economic trade-off studies would be necessary before a final decision could be made.

Mass and energy balances were made for a plant to produce:

1. 1500 pounds of hydrogen per day.
2. 1200 pounds of life support and process water per day.
3. 100 pounds of life support and utility oxygen per day.

At this rate of production it would take about three months to generate enough hydrogen for a round trip to Mars (using the Cole and Segal [10] figure of 155 000 pounds hydrogen for nuclear rockets). Admittedly, this will be quite excessive at first but might not be too unrealistic during that period when extensive exploration of the near solar system becomes a reality.

To produce the 1500 pounds of hydrogen/day (62.5 lb/hr) and necessary life support fluids requires a total of nearly 7000 pounds of well fluids per hour. Power requirements are 435 kW. This breaks down as follows:

Icer	1 kW
Electrolysis	11 kW
Hydrogen liquification and conversion	362 kW
Compression of hydrogen gas for diffuser	31 kW
Pumps, actuators, instrumentation, and other utility power (five percent of above total)	20 kW
Life support	10 kW
	435 kW

The power is generated strictly from the geofluids as stated before. The breakdown on power generation is as follows:

Power from B1 boiler loop	233 kW
Power from B2 boiler loop	158 kW
Power from direct expansion of gases from separator S1	22 kW
Power from direct expansion of gases from hydrogen diffuser	22 kW
	435 kW

Hydrogen-Oxygen Production for Chemical Rockets

The flow schematic for production of hydrogen and oxygen is shown in Figure 4. As will be noticed, a number of the individual processes are common with the hydrogen production schemes discussed previously. However, in this case, the hydrogen is all produced by electrolysis. This, of course, results from the much higher oxygen requirements. Because of the electrolysis process, much more power is needed. This in turn requires a relatively large flow of well fluids strictly for power generating purposes.

As in the hydrogen process, the fluid from the well is sent through a boiler where heat is transferred to a secondary power loop and higher boiling point components are removed from the flow. In this case, however, the geothermal fluids from the boiler are sent directly through turbine T2. They are expanded down to the point (about 7 psi) where a quantity of water equal to about 10 percent of the total fluid has been condensed. (A higher condensate content results in turbine problems.) The mixture is separated in S2 and the vapor phase returned to the turbine line (T3) where the gases are further expanded to about one pound per square inch. The water condensed in T3 is separated in S2 and is used for cooling purposes.

Most of the water and HCl from S1 is sent into the water-icer process discussed earlier. As before, some of the water is stored for life support functions and the remainder is passed to electrolysis cells. A portion of the oxygen generated is stored in gaseous form for use in local life support. The remainder is liquified for propellant or fuel cell uses. Likewise the hydrogen from the electrolysis cell is liquified and stored.

Again a mass and energy balance was run on a plant capable of producing enough rocket propellant

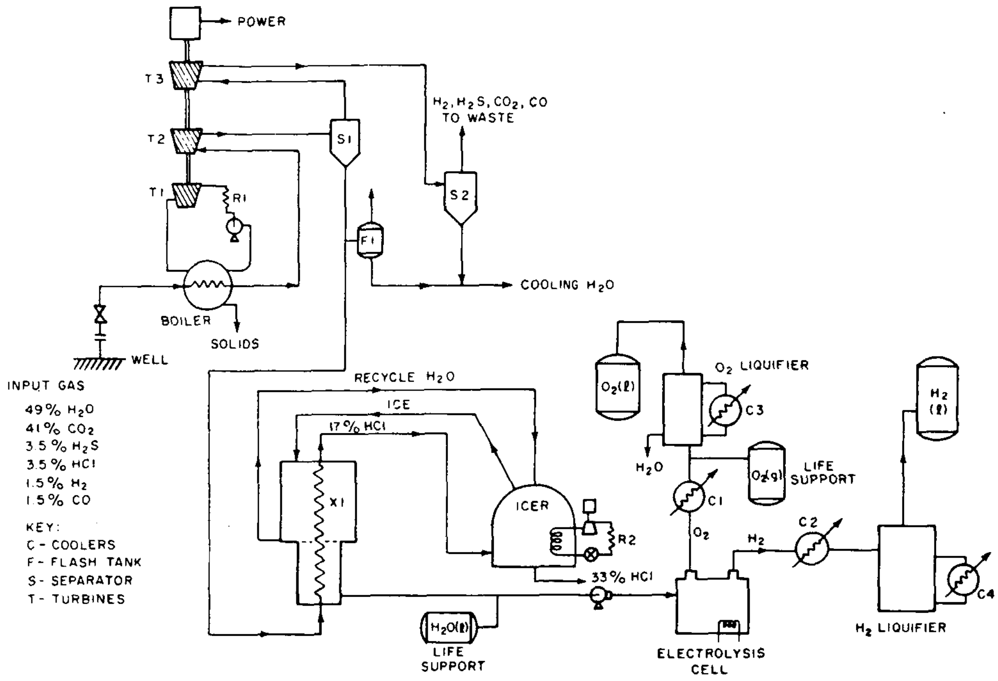

FIGURE 4. PRODUCTION SCHEMATIC: H_2 AND O_2 PROPELLANTS AND LIFE SUPPORT FLUIDS

for a round trip to Mars about every three months. This requires a total of 400 000 pounds of propellant according to Cole and Segal. For a set rate of 4000 lb/day this would require about 100 days. For a 24-hour day this is 167 lb/hr. Referring to Figure 2, the correct O/F ratio for maximum I_{sp} is about 5. This means 137 pounds of oxygen and 28 pounds of hydrogen are required per hour. To produce these propellants requires a total of 11 700 pounds of geofluids.

The energy required to generate these propellants is as follows:

Icer	6 kW
Electrolysis	592 kW
Hydrogen liquification	162 kW
Oxygen liquification	123 kW
Pumps, actuators, instrumentation and other utility power (five percent of above total)	37 kW
Life support	10 kW
	930 kW

Power generated from the geofluids is as follows:

Power from boiler loop (T1)	396 kW
Power from turbine T2	340 kW
Power from turbine T3	184 kW
	930 kW

The production of oxygen through the decomposition or reduction of carbon dioxide was examined. It did not appear competitive with the water electrolysis scheme presented here.

Methane-Oxygen Production for Chemical Rockets

The flow schematic for methane-oxygen production is given in Figure 5. It combines many of the features of the processes discussed earlier. The methane is produced by the Sabatier reaction.

$$CO + 3H_2 \rightarrow CH_4 + H_2O$$

$$CO_2 + 4H_2 \rightarrow CH_4 + 2H_2O$$

The geothermal gases are well suited for this purpose.

Initially the first increment of power is extracted by the same boiler-solid separator unit using a secondary loop. The fluid is then sent directly through turbine T2 as in the hydrogen-oxygen process because, unlike the hydrogen process, it was not necessary to maintain a high pressure in the gas stream for downstream processes. As noted earlier, only a portion

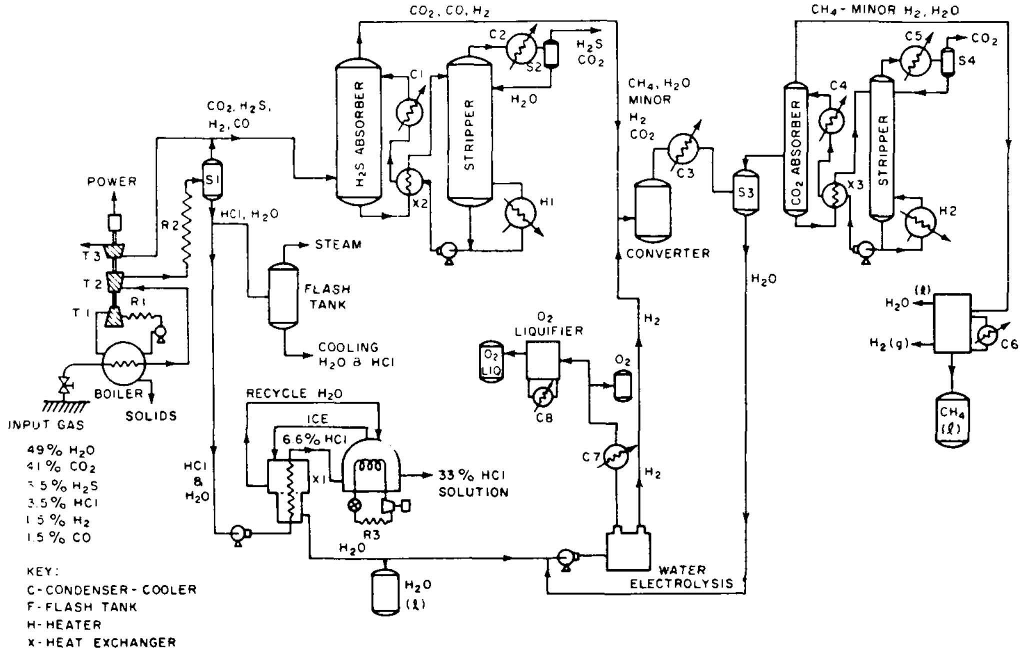

FIGURE 5. PRODUCTION SCHEMATIC: CH_4 AND O_2 PROPELLANTS AND LIFE SUPPORT FLUIDS

of the water can be condensed in a direct expansion process from the gas stream. Because it was necessary for most of the water to be removed from the gas stream and because the stream temperature was still relatively high (approximately 150° F), a radiator (R2) was used to condense the water. The mixture was separated at S1. Part of the liquid was again flashed to produce cooling water. The remainder was cycled through the now familiar water purification and electrolysis system. Sufficient oxygen was generated to satisfy both life support and propellant needs. Fortunately, the hydrogen generated from the electrolysis was quite close to the amount needed for supplementing the elemental hydrogen in feed for the Sabatier process.

A portion of the gas from separator S1 is sent through an absorber-stripper unit. This time the units are designed to pull out nearly all of the H_2S and a minimum of CO_2 (less than one percent of the total).

The feed now consists of CO_2, CO and H_2. The hydrogen from the electrolysis unit is introduced into the stream. It may be necessary to run this feed gas through an iron oxide box at this point to remove any H_2S that may remain because the catalysts in the Sabatier process are quite sensitive to poisoning by this chemical. The mixture is reacted. A nearly 100 percent conversion is obtained on the first pass. The residual CO_2 can be scrubbed out as shown and the methane gas liquified. If the amount of CO_2 is fairly small, it may be more advantageous to condense it in the methane liquifying process.

The methane-oxygen process was set up to produce a total of 300 lb/day of methane and 900 lb/day of oxygen (plus the usual life support water and oxygen). The power required was 176 kW; the geofluid needed was 2700 lb/hr.

Methanol-Oxygen Production for Chemical Rockets

The methanol process also contains elements of each of the former processes, as shown in Figure 6.

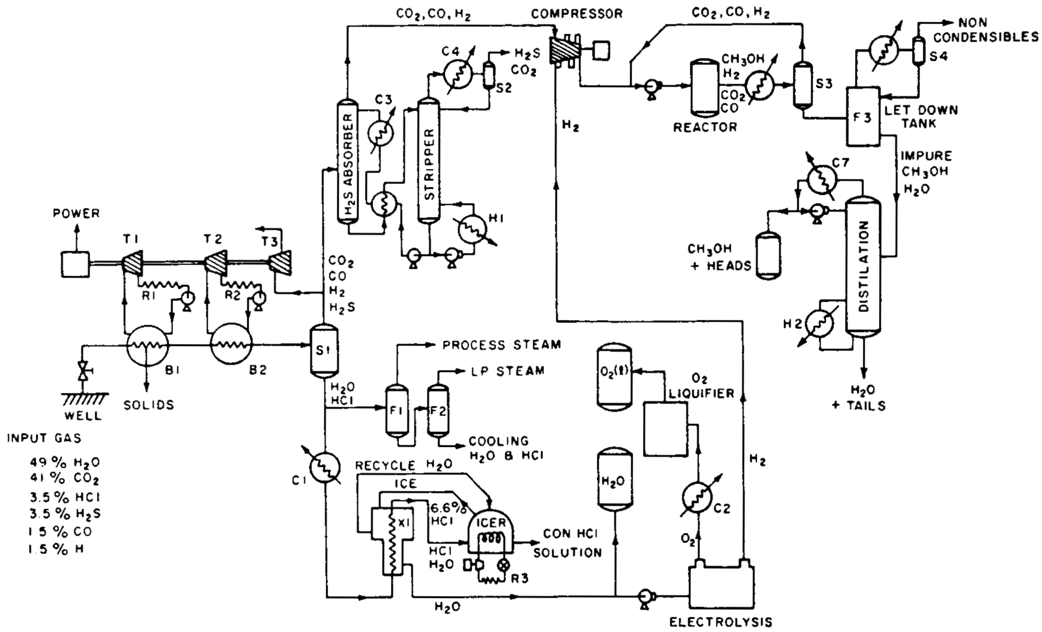

FIGURE 6. PRODUCTION SCHEMATIC: CH_3OH AND O_2 PROPELLANTS AND LIFE SUPPORT FLUIDS

The process is quite similar to that used for the synthetic production of methanol from CO_2 and H_2 in industry [11].

The basic equations are

$$CO + 2H_2 \rightarrow CH_3OH$$

$$CO_2 + 3H_2 \rightarrow CH_3OH + H_2O$$

The power generating cycle is identical to that used in the original hydrogen production scheme: two secondary loop power systems with a portion of the gas rejected through a third turbine (stream pressure is maintained). Likewise the water purification and electrolysis system is identical to that used before in the methane process. The gas is stripped of H_2S with minimum CO_2 loss and together with the hydrogen input from the electrolysis unit is compressed to around 4400 psi.

From the compressor, the gases go to a catalytic reactor where 12 to 15 percent of the feed is converted to methanol. The mixture of CH_3OH, H_2, CO, and CO_2 is cooled and processed through a liquid-gas separator. The gaseous components are recycled while the methanol-water mixture goes to the letdown tank. Here the pressure is lowered to around 20 psia with the flash vapor recondensed. The mixture is distilled.

In the present schematic, only one column or tower is used. This gives a mixture off the top of the column that is primarily methanol with less than 0.5 percent of other lower boiling point hydrocarbons (ethers, etc.). For a purer methanol fraction, it is necessary to use another distillation column to further process this fraction. The small amount of impurities should not appreciably affect the performance of chemical rockets; for fuel cells, impurities may be more critical.

The high boiling point fraction from the bottom of the distillation column contains water plus less than half a percent of lower boiling point impurities (such as the higher chain alcohols). As with the low boiling fraction, this can be further processed in another distillation column. However, for present purposes it is dumped. It takes very little power to process the same amount of water in the already available icer unit.

As with the methane process, the energy and mass balances were based on a production rate of 1200 pounds of propellant per day. At the optimum O/F ratio for maximum I_{sp} (Fig. 2), this would break down to 720 pounds of oxygen and 480 pounds of methanol. The power requirements were 143 kW. An input flow rate of 2150 lb/hr was required.

PROCESS COMPARISONS

The choice of propellant manufacturing process will, of course, depend heavily on the product desired and economics plus other factors not considered in this study. However, it is interesting to compare the four processes discussed with respect to (1) power required for a given production rate and (2) input geothermal fluids required to make one pound of product. Of necessity, the small life support needs are included.

Process	Power/ Production Rate kW/lb/hr	Input Rate/ Production Rate
Hydrogen	6.95	112
Hydrogen-Oxygen	5.57	70
Methane-Oxygen	3.52	54
Methanol-Oxygen	2.86	43

PROCESSES INVOLVING DIFFERENT FLUID COMPOSITIONS

As the fluid composition and thermal state change, the techniques and processes used to obtain propellants of interest will also vary. No attempt will be made to cover the wide range of possibilities. Several of the more interesting or pertinent possibilities will be mentioned.

Processes with Predominantly CO_2 Emission

A distinct possibility exists for a flow that is dominantly CO_2 with minor quantities of water (as low as one percent or less) and other gases. Such flows have been observed on Earth occasionally. The processes just discussed would still be applicable with some modifications provided the other assumed gases were still there in usable (though perhaps small) quantities. A larger flow might have to be processed in some cases to obtain sufficient water.

If, however, the flow is quite deficient in the other gases and consists primarily of CO_2 and the minor water, it might be worth processing the CO_2 for oxygen. The water would be condensed, separated and a portion electrolyzed. The resulting hydrogen could be used to extract oxygen from the CO_2 using one of the H_2-CO_2 reduction schemes [12]. The overall process would supply water and oxygen for life support plus oxygen for propellant use.

If the well flow is large, it might be better to process the fluid for the water content (and in turn for hydrogen and oxygen) and use the CO_2 for power generating purposes only. Reducing the CO_2 requires more energy and equipment than merely electrolyzing water. As before, it would also be possible to combine the hydrogen from the electrolysis and the CO_2 to produce either methane or methanol.

Processes with High H_2S Emissions

Conceivably, a fairly high H_2S flow might be obtained. If circumstances dictated, it would be possible to process the H_2S for hydrogen by first reacting the hydrogen sulfide with oxygen to obtain water. The water, in turn, would be electrolyzed to produce hydrogen and oxygen. In the case where the flow consisted of primarily CO_2 and H_2S, the previously discussed CO_2 reduction process could be combined with the H_2S process in a complementary fashion to produce a balance of oxygen and hydrogen.

Processes with HF Emission

If hydrogen fluoride is present in the geothermal fluids, the realm of fluorine oxidizers may be opened. The first problem will be in separating out the HF, if HCl and other soluble components are present along with water. Once the HF is separated, the next difficulty is in breaking down the HF into its elemental components. Ideally, direct electrolysis of aqueous HF solutions (as can be done with HCl) would be most convenient. Although this approach is being studied with some promise for success by D. N. Bennion (University of California at Los Angelse) electrolysis from an anhydrous HF bath is still required. Conventional manufacture of anhydrous HF would require continuous importation of process chemicals from Earth, thus doing away with the benefits of lunar manufacture. One new process was noted in the literature [13] whereby anhydrous HF is recovered from even very diluted aqueous solution of HF by amine extraction. An alkali fluoride is added to form the bifluoride; this is crystalized by cooling. The crystals are calcined to liberate HF and restore the KF. The KF and amine solutions can be recycled. Such a process where only small amounts of make-up chemicals are required might possibly be used to advantage on the moon.

Processes with Low Total Emission

If the flow rate of available geothermal fluids is limited, power from sources other than the primary flow may be necessary. Recovery of some of the geothermal heat within the rock structure could provide the additional energy needed. Waste fluids might be reinjected in the rock structure, heated, and recovered through the well. For better control, heat transfer devices (such as heat pipes) might be inserted into the the ground to heat a circulating working fluid for power extraction.

CONCLUSIONS

The use of geothermal emissions for generating rocket propellants, life support fluids and power on the moon appears technically feasible and very promising, provided the geothermal fluids exist. The present paper illustrates what can be done with one particular flow. Further investigation of what could be done with other fluid compositions would stimulate additional thinking. Study into the possible sublimate products that might be obtained as economic by-products of the processes is also warranted. Economic studies are also needed to show how the geothermal products compete with Earth-delivered propellants and propellants manufactured on the moon using other processes, such as rock processing. As further information becomes available on the existence and composition of possible deposits, the course of action will become better defined.

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EARLY POST-APOLLO CRYOGENIC FUEL PRODUCTION SYSTEM

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INTRODUCTION

Extensive exploration of the lunar surface - a primary goal of the Apollo Applications Program (AAP) - should be preceded by a comprehensive analysis of the role of water in mission operations and logistics. The importance of water and its constituents to space exploration as a source of life support, as a coolant, as a propellant and as a source of electric power has long been recognized, and the identification and evaluation of lunar water resources is an important goal of early lunar missions. Conservation of water and its regeneration into hydrogen and oxygen have been incorporated into the design philosophy of the more advanced Apollo bases [1]. However, the uncertainties underlying discovery and exploitation of lunar water resources, questions which motivate the AAP lunar surface program, emphasize the need for ensuring the presence of an efficient, reliable supply of water and its constituents during the early post-Apollo period.

In the absence of a priori knowledge regarding the location, quantity, and accessibility of indigenous lunar water, the regeneration of logistic water produced during fuel cell operation would appear to be an efficient mode of operation. Regardless of the source, the conservation of water whenever possible appears to be a prudent philosophy for lunar operations in view of the crucial role of that commodity in lunar missions. Complete exploitation of available supplies will require facilities for production of cryogenic hydrogen and oxygen from water reserves; indeed, the existence of an efficient regeneration system could itself influence the planning of AAP lunar surface missions.

The purpose of this paper is to present in some detail the design and performance characteristics of a system capable of producing liquid hydrogen and liquid oxygen from water while operating on the surface of the moon.

PROJECTED LUNAR CRYOGENIC REQUIREMENTS

The final design of a suitable water reclamation system will be a function of specific mission and program goals, their attendant cryogenic requirements, and the status of lunar resources. However, several general considerations are apparent. The specifications and time frame of the intended user will determine cryogenic production rates, storage volumes, product pressure, the available power source, vehicles available for lunar logistics, and the sources of water that might be utilized. Indeed, the frequency at which lunar roving vehicles are deployed will be fundamental to evaluation of the economic feasibility of fuel regeneration.

A preliminary estimate of early lunar cryogenic fluid requirements provides an indication of the need for a post-Apollo cryogenic fuel system and the economic advantages that might be realized through exploitation of such a facility. The proposed Lunar Exploration Systems for Apollo (LESA) bases exhibit significant cryogenic demands (Table I). However,

TABLE I. EARLY LUNAR CRYOGENIC FLUID REQUIREMENTS

LESA Lunar Base Requirements				
LESA Base	LOX (lb)	LH_2 (lb)	Water (lb)	Provision for Regeneration
I	4594	389	300	No
II	6636	723	600	No
III	8100	501	2367	Yes
IV	20 250	611	7101	Yes

Lunar Roving Vehicle Requirements					
Duration (days)	Crew Complement	Total Mileage	Locomotion Cryogenics (total lb)	Life-Support Cryogenics (lb-LOX)	Total Cryogenic Fluids (lb)
14	2	250	300	130	430
28	3	500	600	375	975
90	3	1000	1195	820	2015
90	3	1500	1790	820	2610

the earliest shelters will probably require inclusion of complete cryogenic supplies in view of the limited power sources expected to be available for those missions. Regeneration of fuel cell water will become practical only with the presence of significant nuclear power sources on the lunar surface. However, the very existence of such facilities might appear to obviate or greatly reduce the need for fuel cell power at lunar bases. Nevertheless, water regeneration facilities have been included in LESA base concepts III and IV, primarily to permit reuse of the expended fuel products of roving vehicles and auxiliary base power supplies. Mobility requirements and the subsequent prolonged proximity to manned operations suggest fuel cells as the most practical source of power for lunar roving vehicles. Despite the presence of plentiful nuclear energy at advanced lunar bases, fuel cells could provide a ready source of emergency of supplemental (peak load) power. The availability of cryogenic oxygen for emergency life support demands is an additional advantage accruing from the inclusion of auxiliary fuel cell power. The cryogenic requirements projected for LESA base concepts III and IV in Table I thus reflect the utility of water regeneration even to advanced programs.

It is expected that the earliest major users of lunar cryogenics will be roving vehicles, probably of the MOLAB type. Conservative estimates of the cryogenic requirements for several roving vehicle missions (including only locomotion and environmental control demands) are presented in Table I. Assumptions implicit in these projections include an energy requirement of 0. 585 kilowatt-hour per mile for a soft soil traverse [2], 50 percent overall system drive efficiency, and a 20 percent obstacle factor. Cryogenic requirements for the more ambitious missions are sufficient to justify consideration of fuel regeneration techniques, especially since the base demands previously discussed are likely to exist concurrently. The mobile exploration procedure suggested in the LESA studies [3]--utilization of multiple short radial excursions from a central base or shelter--is especially suited to the exploitation of fuel regeneration because fuel loads for any given traverse can be minimized by exchanging fuel cell product water for regenerated cryogenic hydrogen and oxygen on each return to the central shelter.

The foregoing examples have stressed the earliest requirements for water conservation and regeneration. Of course, the advent of more advanced modes of lunar transportation would very likely signal increased demands for cryogenic fluids. The production of cryogenic liquids for lunar refueling of interplanetary spacecraft will necessarily include a subsystem for conversion of water resources into hydrogen and oxygen propellants, regardless of the process employed to extract the water from the lunar surface. Thus, an efficient cryogenic fuel regeneration system would be expected to have an expanding utility and range of application as the lunar program evolves.

GENERAL DESIGN APPROACH

The foregoing considerations indicate the need for an early post-Apollo cryogenic fuel production system. In view of the uncertainties associated with specific mission applications and the nature of the lunar surface, the following general assumptions have been made.

1. Water is present on the lunar surface either as fuel cell product or from natural sources.

2. Water will be conserved whenever possible.

3. Maximum power available on the lunar surface in the early post-Apollo period will be 30 kilowatts (electrical).

4. Early post-Apollo systems will be jointly limited by power and weight.

The much-debated question of the presence of natural lunar water was not considered. Instead, reclamation of logistic water was emphasized, and the ultimate utilization of lunar resources was left as an extension for a later date. The capability of the system to include a substantial water supply of its own relaxes the dependence upon available water sources. The assumed post-Apollo power limit will be discussed later. Assumption 4 requires that neither power nor weight trade-offs be given priority in this general study, although in a specific case either philosophy might be advantageous.

Specific constraints that were imposed on the system design include:

1. Packaging on an LM truck with minimal deployment requirements.

2. Adaptability to a variety of presently unspecified users and missions.

3. Utilization of power sources and vehicles expected to be available for the early post-Apollo period.

4. Reliance upon available, operational hardware.

Although packaging within the LM envelope places major limitations upon the system, realistic utilization of a regeneration facility during the early post-Apollo period dictates such an approach. Similar considerations motivate the emphasis upon use of available hardware and supporting systems. The constraint on system flexibility has resulted in a design which is as attractive for regeneration of logistic water as it is for inclusion with any of the proposed techniques for manufacturing fuel from lunar materials. This goal has been achieved by the creation of a flexible, partially modular configuration characterized by efficient, long-lived operation and minimal astronaut attention.

The key subsystems and their interrelation are pictured on the flow sheet of Figure 1. The electrolysis subsystem is based upon a reverse-cycle fuel cell developed by General Electric from its Gemini fuel cell hardware. The liquefaction subsystem operates on a modified Claude cycle using the hydrogen product from the electrolysis unit as the working fluid. The hardware requirements of the liquefaction subsystem are filled by compressors, expansion engines, and heat exchangers based upon presently operational equipment. Separate heat-rejection subsystems, operating at different temperature levels, dissipate the waste heat from the electrolysis and liquefaction units. The electrical requirements are filled by a nuclear power supply typical of early post-Apollo lunar capabilities. Liquid hydrogen and liquid oxygen storage sufficient to contain 28 days of production is provided.

As can be seen from Figure 1, there are a number of system interfaces which offer opportunities for optimization of the overall weight, power demands, and production rate of the unit. Among the most important trade-off parameters are the current density of the electrolysis cell, the product pressure of the liquefier, the liquid storage pressure, and the temperature levels of the two heat-rejection systems. The following sections of the paper discuss the major components of the unit and present the parametric data used in the optimization of the overall system.

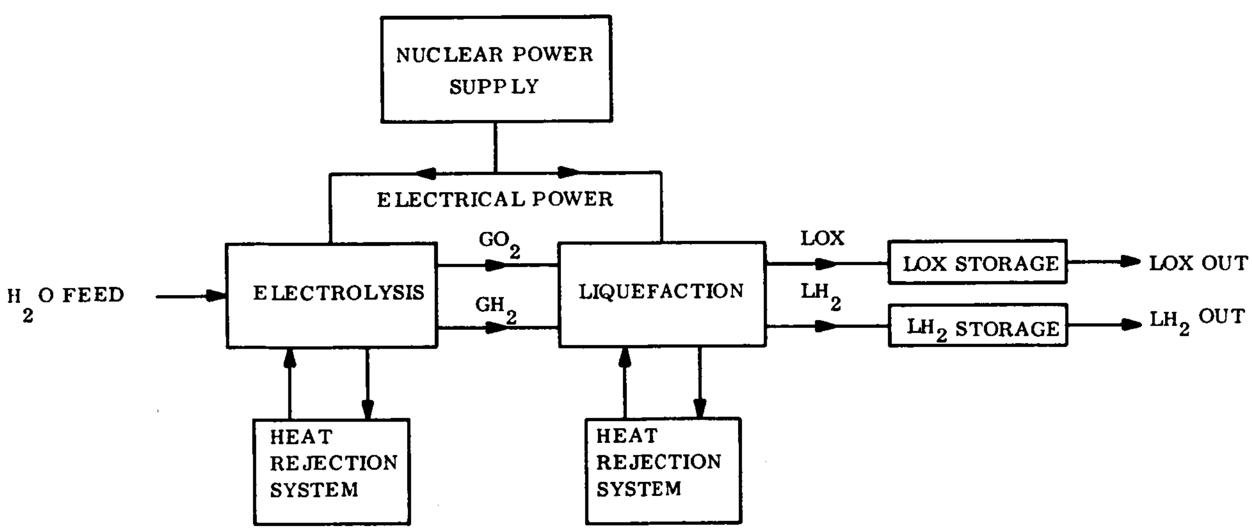

FIGURE 1. SYSTEM FLOW DIAGRAM

ELECTROLYSIS SYSTEM

The first step in the regeneration procedure consists of electrolytically dissociating the incoming water into gaseous hydrogen and oxygen. The minimum voltage, V_o, required for initiation of this process is determined by the change in Gibbs free energy (ΔG) characteristic of the chemical reaction:

$$V_o = \frac{\Delta G}{nF} = 1.23 \text{ volts}$$

where n represents the number of electrons transferred per molecule of water electrolyzed and F is the Faraday constant. In practice, the presence of diverse loss mechanisms -- surface energy barriers, concentration gradients associated with mass transport of the reactants, ohmic losses arising from electron and ion currents, etc., -- combine to raise the reaction potential above V_o [6]. For example, an electrolytic system utilizing a platinum catalyst in conjunction with a sulfuric acid electrolyte exhibits a minimum measured reaction potential of 1.485 volts [5].

The irreversibilities associated with the electrolysis process and the subsequent increase in effective reaction potential are translated into power losses under actual operating conditions. The efficiency, η, of an electrolytic cell can be defined as the minimum chemical energy required for electrolysis of a given quantity of water divided by the energy actually supplied to the cell. For a prescribed production rate, η is the quotient of the rate, P_o, at which chemical energy must be supplied and the electrolysis cell power input, P:

$$\eta = \frac{P_o}{P} = \frac{V_o I_o}{VI} \qquad (1)$$

In equation (1), V_o and I_o are, respectively, the reaction potential and the current demand associated with the ideal electrolytic process; V and I represent corresponding quantities for actual operating conditions. Because the currents, I_o and I, are equal under the assumption of a common production rate, equation (1) reduces to

$$\eta = \frac{V_o}{V} \qquad (2)$$

The irreversible losses associated with electrolysis are generally functions of electrolysis cell current density and, less strongly, cell temperature. In particular, ohmic losses are a linear function of current density, but irreversibilities arising from concentration gradients are significant only at high current densities. Thus, the effective reaction potential, V, is a function of cell current density (Fig. 2) [5]. This relationship is reflected in the cell efficiency defined

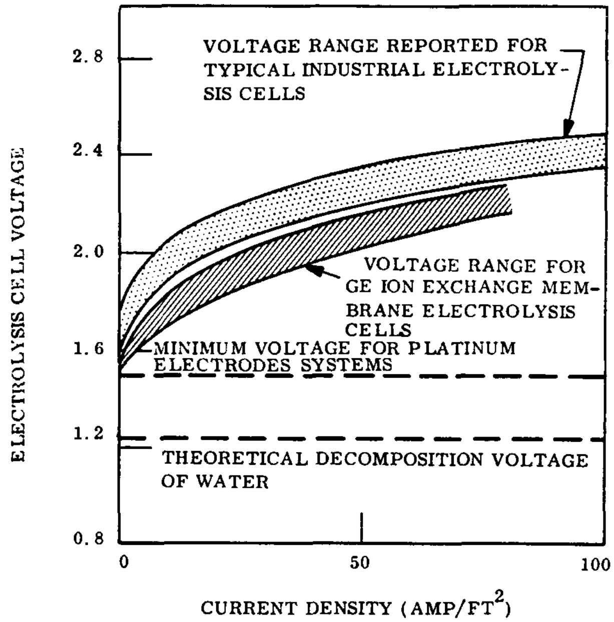

FIGURE 2. EFFECTIVE ELECTROLYSIS CELL VOLTAGE AS A FUNCTION OF CELL CURRENT DENSITY

by equation (2) and illustrated in Figure 3 as a function of current density. High efficiencies can be obtained at low current densities if one is willing to accept the weight penalty associated with increasing the number of cells to maintain a given gas production rate. Thus, a trade-off exists between production rate and system efficiency. The selected operating point determines the power and hardware mass necessary to achieve a given production rate, the heat-rejection load, and the associated radiator weight and area. Furthermore, utilization of high current densities may affect system lifetime.

Another consideration in the selection of an electrolysis unit for lunar operation is the maintenance of

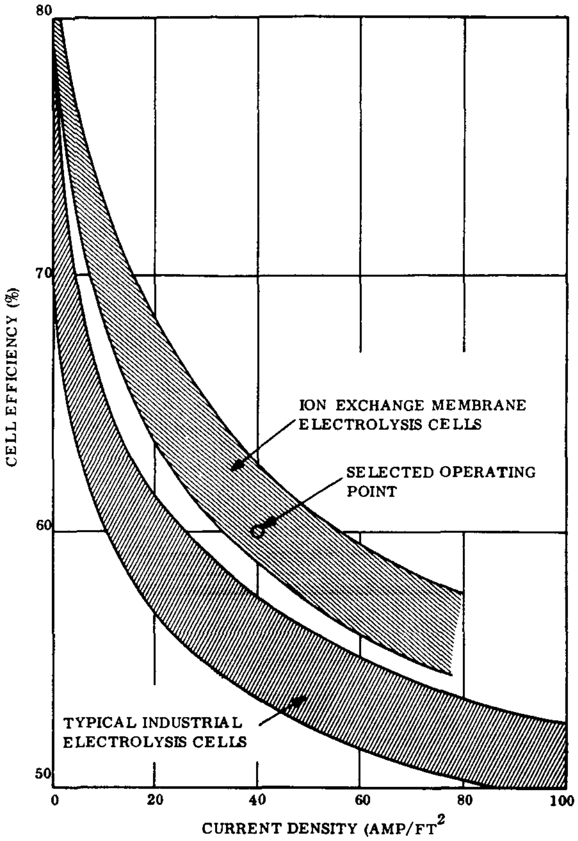

FIGURE 3. ELECTROLYSIS CELL EFFICIENCY AS A FUNCTION OF CELL CURRENT DENSITY

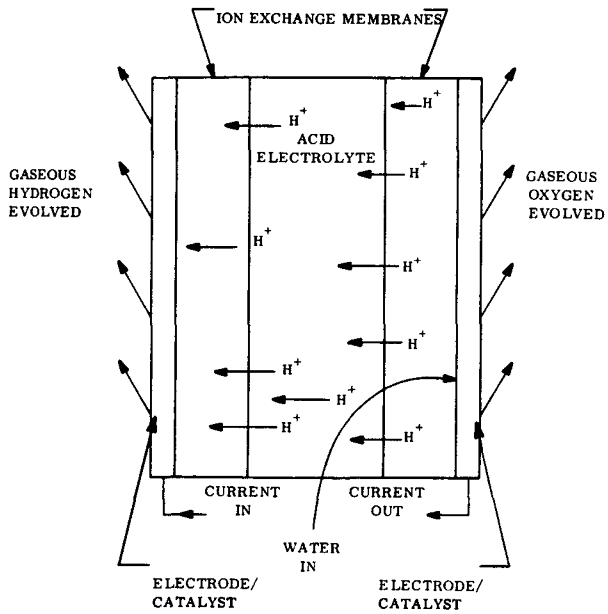

FIGURE 4. ELECTROLYSIS CELL OPERATION

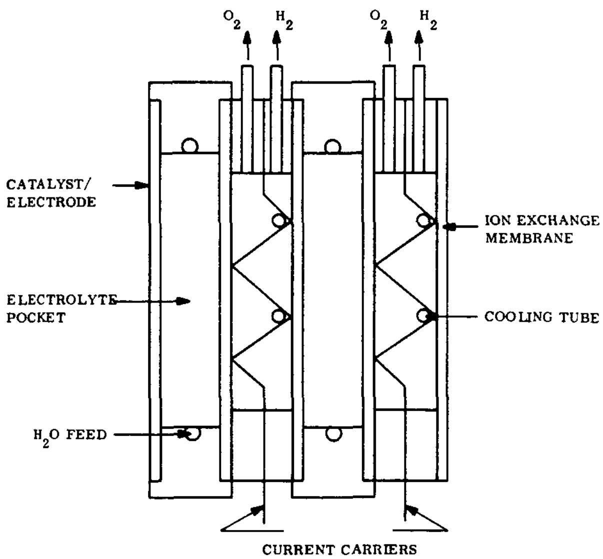

FIGURE 5. GENERAL ELECTRIC ELECTROLYSIS CELL/MODULE CONFIGURATION

a stable interface between the electrolyte and the evolved gas while permitting rapid gas removel under reduced g conditions. Conventional industrial electrolytic units that rely upon the buoyance of evolved gas bubbles for collection and removal of electrolysis products will be inefficient in lunar applications.

To satisfy these joint requirements for efficient gas production and reduced g design, General Electric has developed an electrolysis cell based upon the fuel cell configuration used in the Gemini flights. An acid electrolyte is confined between ion exchange membranes. The ion exchange membrane (IEM) permits selective migration of hydrogen ions while providing gas/water separation at reduced or zero g (Fig. 4). Gas is evolved at the catalyst-coated outer surface of the IEM which makes contact with a corrugated metal current carrier (Fig. 4). The latter element provides physical separation of the evolved gases while supplying electrical power to the catalyst/electrode. The current carrier is cooled at its points of contact with the IEM to maintain cell temperature between 70 and 90° F.

The enhanced efficiency of this electrolysis cell with respect to typical industrial units is illustrated in Figure 3. The current density selected for the present study represents a compromise between electrolysis system mass, power requirement, heat rejection load, and gas production rate in accordance with the assumed general design constraints. Operation at 60 percent efficiency for the chosen current density is regularly achieved in the laboratory.

A group of 16 electrolytic cells was wired in series and assembled as a module (Fig. 5). The gases produced by each cell were passed through integral condensers where entrained moisture was removed before delivery to the liquefaction system. The condensed water vapor was recycled into the electrolyte pocket by capillary action (assuring efficient phase separation under reduced g conditions). Gas delivery pressure (80 psia for the present design) may be set at any desired level by adjustment of the appropriate relief valves. Differential pressure regulators maintain the oxygen and hydrogen gases one psi above the water pressure to prevent water from passing through the IEM. A cooling system is necessary to reject the heat developed during cell operation and to condense moisture from the product gases. Freon 11 is used as the coolant, maintaining the electrolysis cells at about 80° F.

At the selected current density, a production rate of 0.167 pound O_2 per module per hour is achieved. Thus 36 modules of 16 cells each are required for the production level of 6 pounds O_2 per hour (0.75 pound H_2 per hour) selected as optimal under the assumptions and constraints postulated for the entire regeneration system. The characteristics of the complete unit include a weight of 1080 pounds, a volume of 10.2 cubic feet, and an 18.8 kilowatt power requirement.

The relationship between electrolysis power demands and oxygen production rate is illustrated in Figure 6. Electrical requirements for a 72-module electrolyzer, included in Figure 6 for purposes of comparison, exhibit the expected reduction in power level at a given gas production rate because of reduced current densities and subsequent higher efficiency. At the assumed operating level of 60 percent efficiency, the total electrolytic system weight penalty (including heat-rejection requirements) is 301 pounds per kilowatt decrease in total electrolysis system (electrolytic cells plus heat pump for cell thermal control).

In summary, the proposed electrolysis system appears to satisfy the requirements concomitant to its utilization in an early post-Apollo fuel regeneration facility. The unit is adapted from space-qualified Gemini hardware and exhibits the enhanced efficiency required by lunar power and heat rejection limitations. The ion exchange membrane at reduced g eliminates inefficiencies which would otherwise accompany the presence of bubbles within the electrolyte. The cells presently exhibit good lifetimes, and the utilization of high current densities and operating temperatures without degradation in lifetime appears feasible in the near future. The modular nature of the system permits convenient sizing for any desired redundancy and gas production rate while eliminating the possibility of catastrophic failure. Thus, in accordance with the general design constraints, the electrolysis unit is adaptable to a variety of lunar missions while relying upon operational, space-qualified hardware.

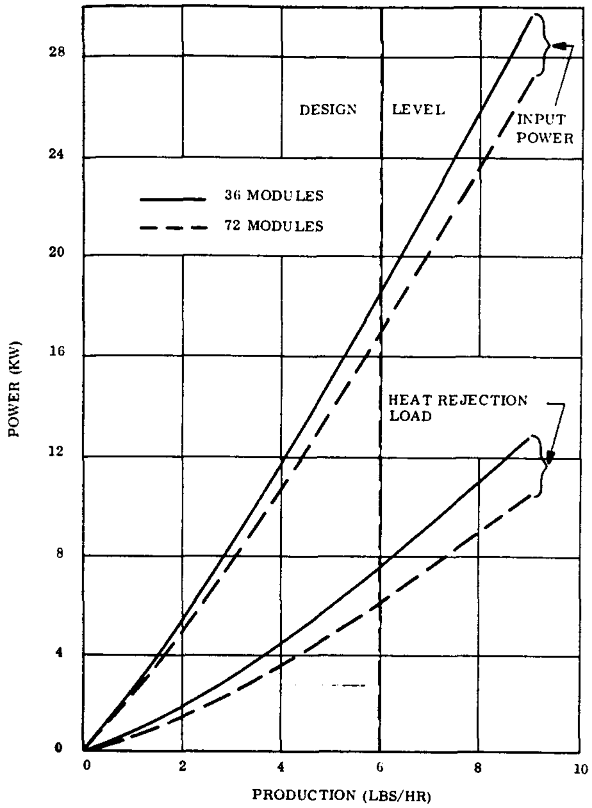

FIGURE 6. ELECTROLYSIS POWER REQUIREMENTS

Liquefaction Subsystem

A number of factors were considered in arriving at the optimum hydrogen-oxygen liquefier for this application. Among these are the following:

Minimum Electrical Power Requirement. - The specific weight of the power-generating equipment could be quite high -- perhaps 1000 pounds per kilowatt in the 30-kilowatt size range. Liquefier weight will be quite small compared with this figure; therefore, every effort must be made to minimize liquefier power consumption.

Minimum Weight and Volume. - Each refrigerator component must be made as light and small as possible without sacrificing reliability.

Heat-Rejection Requirements. - Both the amount of heat rejected by the liquefier and the rejection temperature must be considered to keep the radiator area within reasonable limits. In general, power requirement, heat rejection, and compressor size will decrease approximately in proportion to the absolute temperature, while radiator area per unit of heat flux varies inversely with the fourth power of the temperature.

Reliability, Lifetime, and Maintenance. - These requirements strongly influence the selection of components for the liquefier. Rotating compressors and expanders are highly reliable, light in weight, and have long operating life because they have a minimum of wearing parts. On the other hand, they are less well-developed (at the present state of the art) in the sizes required for this liquefier and are usually less efficient than reciprocating equipment. Furthermore, hydrogen is difficult to compress in centrifugal compressors because of its low molecular weight.

Liquid Product Pressure. - The liquid product pressure affects the product temperature and hence the power requirement of the liquefier. Raising the product pressure lowers the liquefier power consumption.

Contaminant Removal. - Contaminants, such as carbon dioxide and water, must be removed from the feed gas to prevent plugging of the low-temperature components of the liquefier. Physical adsorption is the technique used to deal with this problem.

Hardware Performance. - Design values used in this analysis are based on current equipment performance in the approximate sizes required, extrapolated to what might reasonably be expected with two to three years of development.

With these factors in mind, the design embodied in the flow sheet of Figure 7 was conceived. The

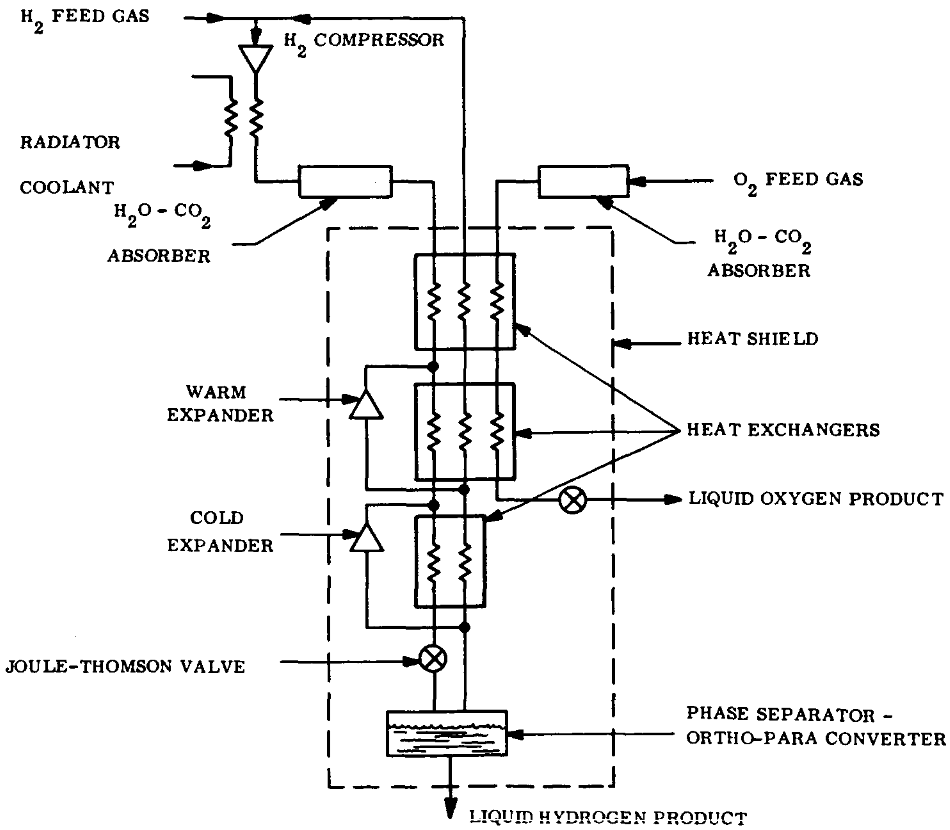

FIGURE 7. HYDROGEN-OXYGEN LIQUEFIER FLOW DIAGRAM

thermodynamic cycle reflected in Figure 7 is most appropriate for reciprocating machinery. In this cycle, all the refrigeration is provided by the two expanders in the hydrogen stream. The hydrogen feed gas, combined with recycled hydrogen from the liquefier, is compressed, and the heat of compression is removed by coolant from the liquefaction radiator. The high-pressure hydrogen then passes through an adsorption system to remove contaminants and through a series of heat exchangers, where it is cooled by counterflow of recycle gas. At two points, part of the high-pressure stream is tapped off and run through expanders, where energy is extracted in the form of work. The remainder of the gas expands through a Joule-Thomson valve. Part of the gas liquefies and is removed as liquid product. The remainder recycles through the heat exchangers to cool the high-pressure stream. Oxygen produced by the electrolysis subsystem is fed to the liquefier at electrolysis cell pressure; it is cooled and completely liquefied by heat exchange with the hydrogen.

Should turbomachinery developments over the next several years indicate a preference for turbomachinery, a modified thermodynamic cycle using neon as the working fluid is believed to offer significant advantage. The low molecular weight of hydrogen limits the compression per stage in a centrifugal compressor to a ratio of 1:1 or less; it also limits the expansion ratio for a single-stage turboexpander. A pressure ratio of about 10 is required for reasonable efficiency in the cycle of Figure 7; thus a centrifugal compressor of at least 24 stages would be required. A regenerative turbocompressor would require fewer stages but would be less efficient, thereby raising power and heat-rejection requirements. With neon as the working fluid, a maximum compression ratio of 2. 3 per stage at the limiting values of 80 percent stage efficiency and 1400 feet per second tip speed can be attained, requiring only three stages for a 10:1 ratio. The neon cycle will attain reasonable efficiency at a 3:1 ratio, permitting the use of fewer stages and a more conservative compressor design.

To arrive at an optimum system design, parametric data on the power and heat-rejection requirements of the liquefier subsystem were developed. The following discussion applies to the reciprocating machinery cycle of Figure 7.

Figure 8 is a plot of the power and heat-rejection requirements of the liquefier as a function of product pressure. The decrease in power with increasing

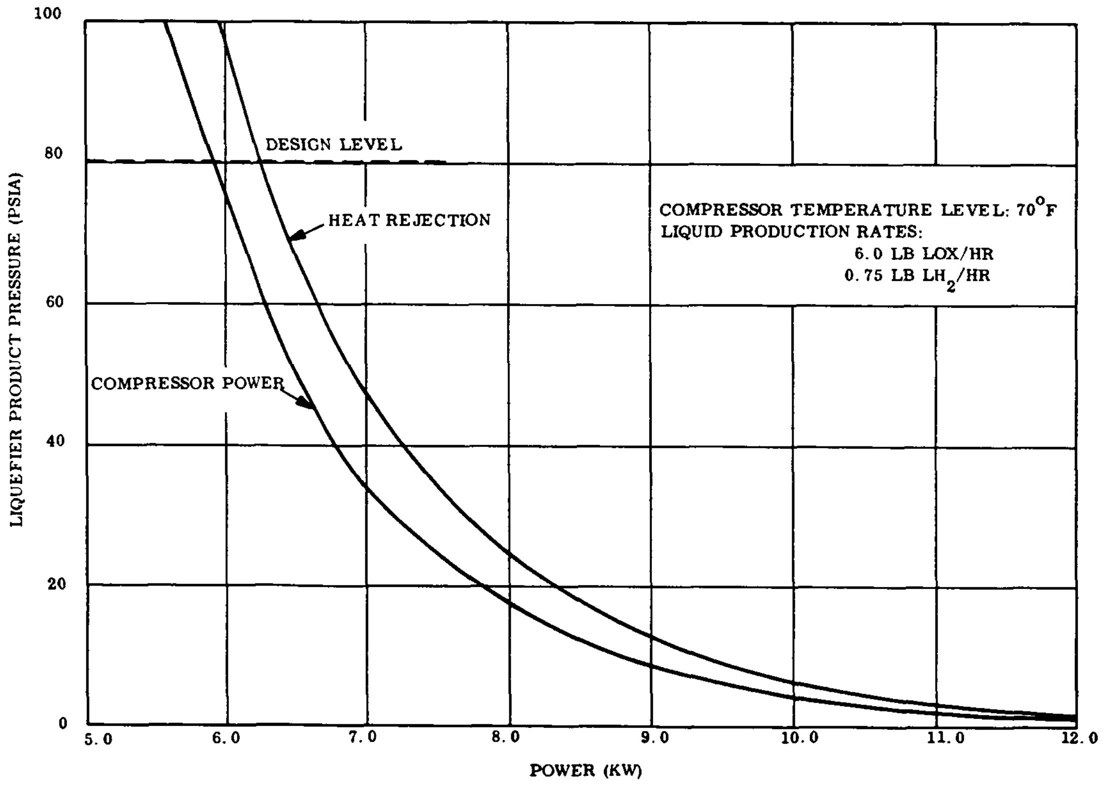

FIGURE 8. EFFECT OF PRODUCT PRESSURE ON LIQUEFIER POWER AND HEAT-REJECTION REQUIREMENTS

pressure is based on two factors. First, as the pressure increases less heat needs to be extracted to liquefy each pound of gas. Second, the liquefaction temperature increases with increasing pressure. Because

$$\frac{\text{Input Power}}{\text{Unit Refrigeration}} \quad \alpha \quad = \quad \frac{\text{Compression Temperature - Product Temperature}}{\text{Product Temperature}}$$

the input power will decrease with increasing pressure. This effect is much more significant for hydrogen than oxygen because the product temperature for hydrogen is much closer to absolute zero. Thus, small changes in the hydrogen product temperature result in significant changes in the foregoing ratio and, hence, in the power requirements.

The selection of 80 psia as the liquefier product pressure represents a compromise between low power, reasonable boiloff losses from storage, and acceptable tankage weight penalties caused by elevated pressure.

Figure 9 presents the weight of the hydrogen-oxygen liquefier as a function of the hydrogen and oxygen liquefaction rate, based on an O_2/H_2 weight ratio of 8, a compressor temperature of 70°F, and a product pressure of 80 psia. Weight is estimated on the basis of currently available components developed and redesigned for minimum weight and size. The design point ultimately selected is indicated for reference.

In Figure 10 the power and heat-rejection requirements of the liquefier are depicted as a function of the liquefaction rate, based upon a system optimized for a production rate of 6 pounds of lox/per hour.

The weight, volume, and power requirements of the liquefaction subsystem, optimized with respect to the complete fuel regeneration facility, are summarized in Table II. Component weights and volumes are estimated on the basis of currently available components redesigned for minimum weight and size. Power and heat-rejection requirements are based on a 60 percent overall adiabatic compressor efficiency and a 70 percent overall adiabatic expansion engine efficiency.

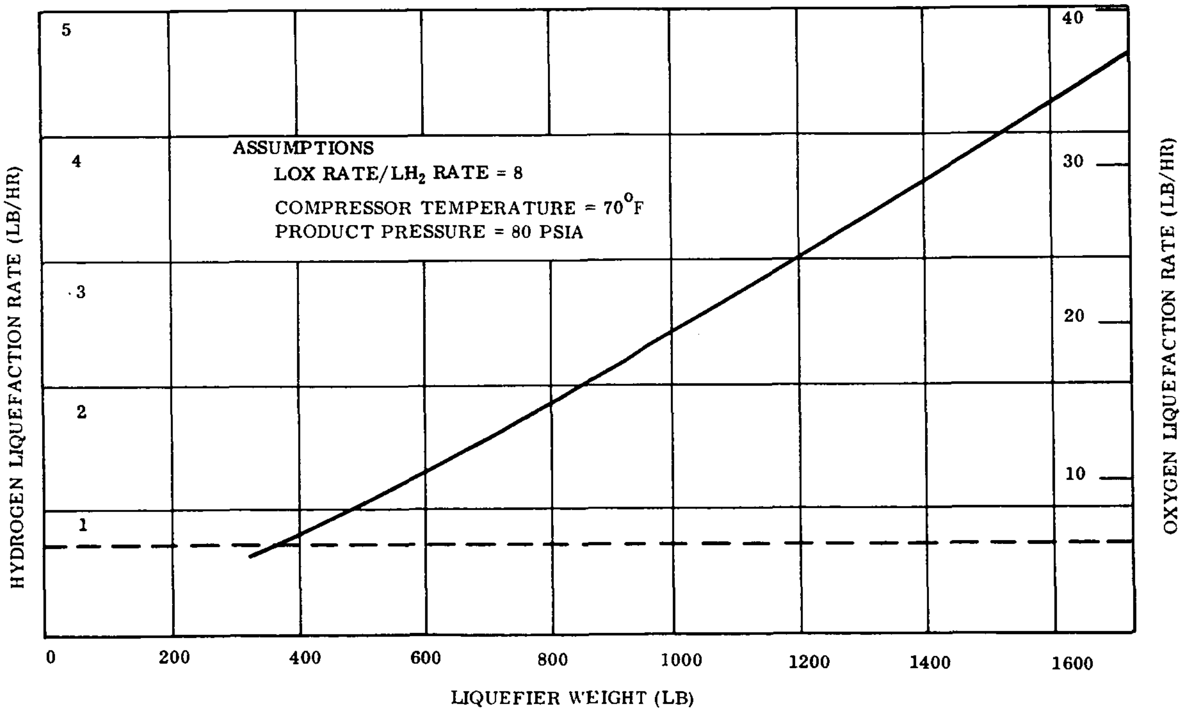

FIGURE 9. LIQUEFIER WEIGHT AS FUNCTION OF MAXIMUM LIQUEFACTION RATE

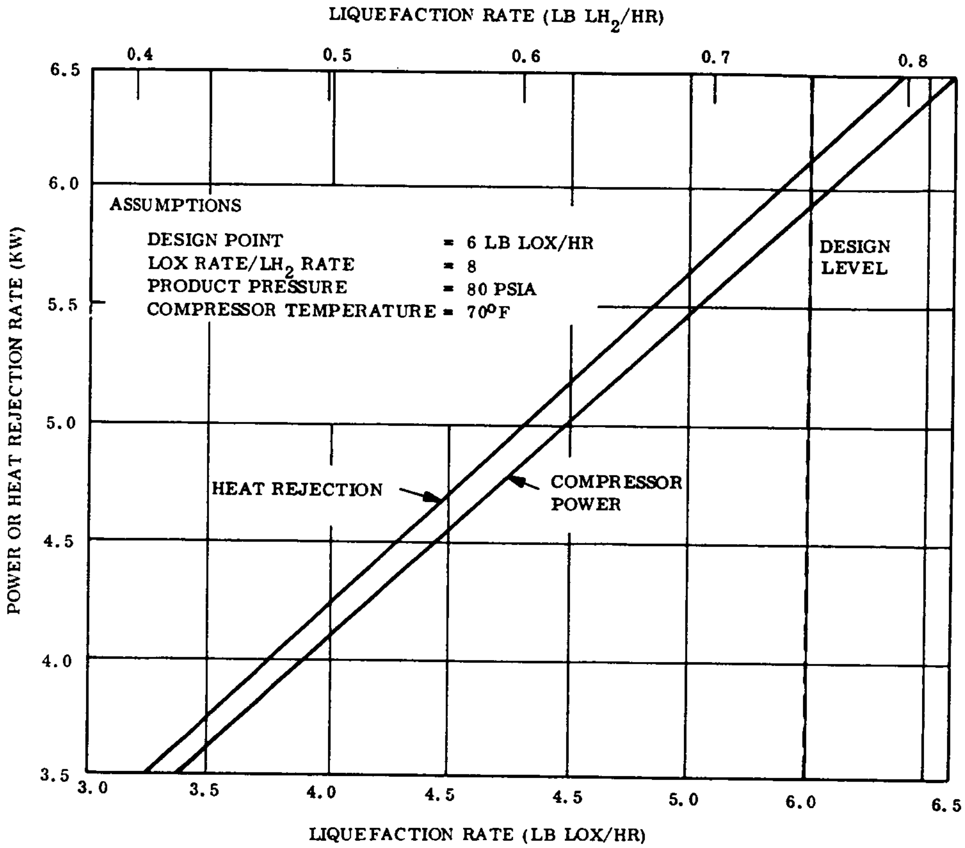

FIGURE 10. EFFECT OF PRODUCTION RATE UPON LIQUEFIER POWER AND HEAT-REJECTION REQUIREMENTS

TABLE II. LIQUEFIER WEIGHT-VOLUME-POWER BREAKDOWN

Operating Conditions

Fluid	Flow Rate (lb/hr)	Pressure (psia)			Temperature (°R)	
		In	After Compressor	Out	In	Out
O_2	6	80	---	80	530	198.5
H_2	0.75	80	800	80	530	49.5

Liquefier Components

	Weight (lb)	Volume (ft^3)
Compressors	160	8
Expansion Engines	70	
Heat Exchangers	80	2
Controls, Piping, Etc.	60	—
Totals	370	10

Power Requirement 5.9 kw (electrical)

Waste Heat 6.25 kw

Storage Subsystem

Figure 11 depicts the interfacing of the liquefier with the liquid hydrogen and liquid oxygen stórage vessels and the tank filling and recycle boiloff recovery system. Hydrogen and oxygen gases entering the liquefier from the electrolysis cells are joined by the boiloff from the liquid storage vessels. Gaseous recycle from tank-filling operations plus recycle boiloff which cannot be continuously accommodated at the liquefier are directed to a low-pressure gaseous surge system. From there they are withdrawn through an auxiliary compressor as rapidly as they can be accommodated at the liquefier.

The 80-psia liquid storage pressure is determined by the liquefier outlet pressure. Both the liquefaction and the storage subsystems must be considered in the overall optimization of the operating pressure level. The effects of product pressure on liquefier power were considered previously. The influence of storage pressure upon boiloff rate and tankage weight are illustrated in Figures 12 and 13.

The boiloff losses from the hydrogen and oxygen storage vessels are a function of the heat leak, the stored fluids' latent heat of vaporization, and the

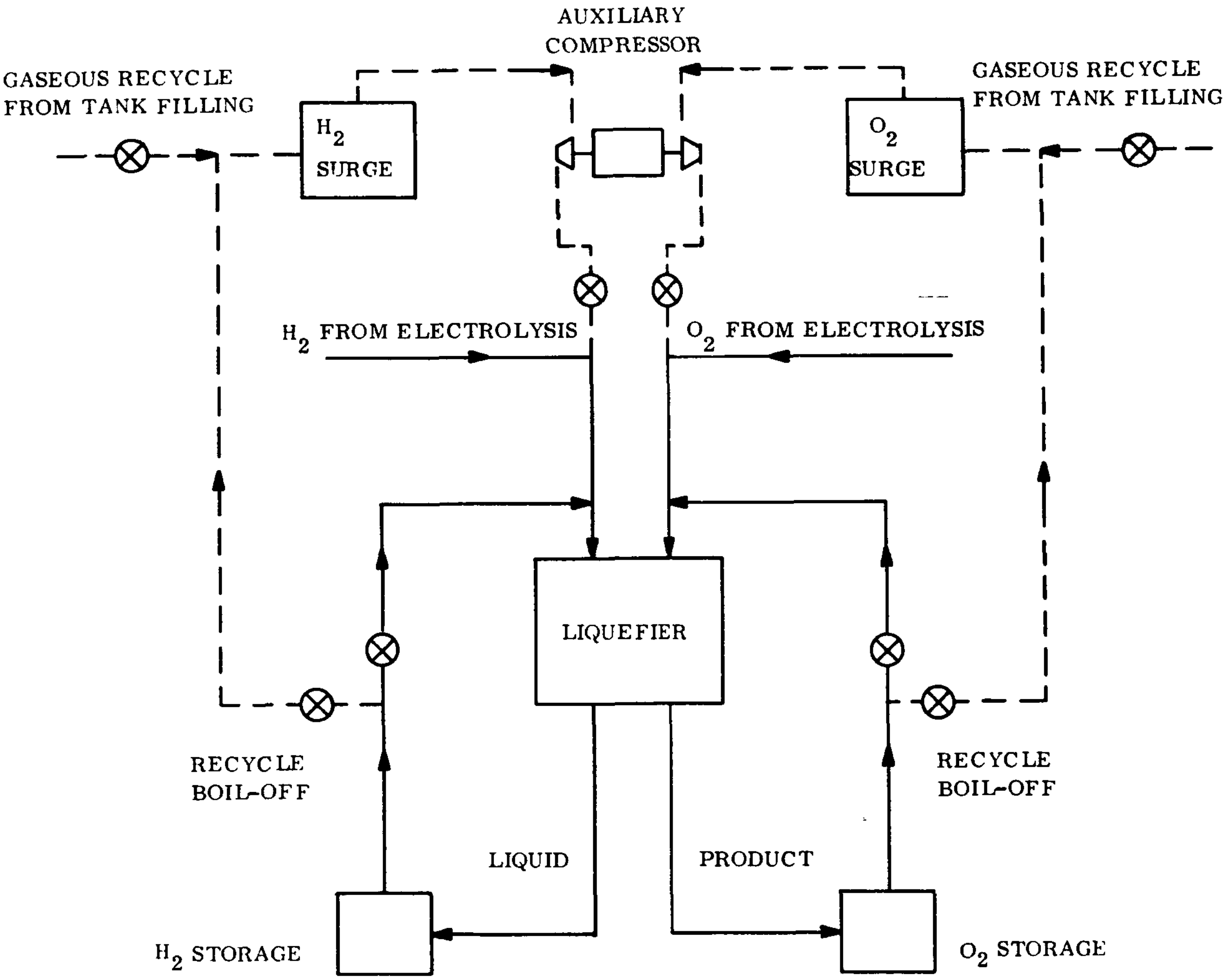

FIGURE 11. LIQUEFIER STORAGE COMPLEX

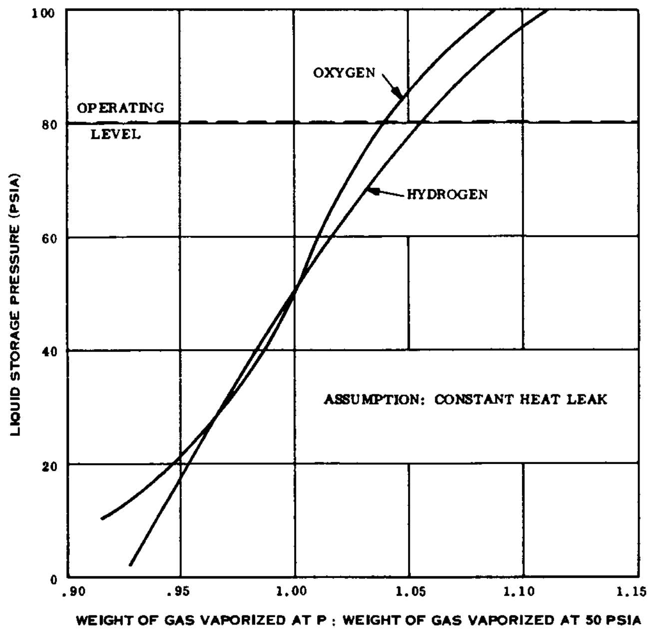

FIGURE 12. EFFECT OF STORAGE PRESSURE ON BOILOFF RATES

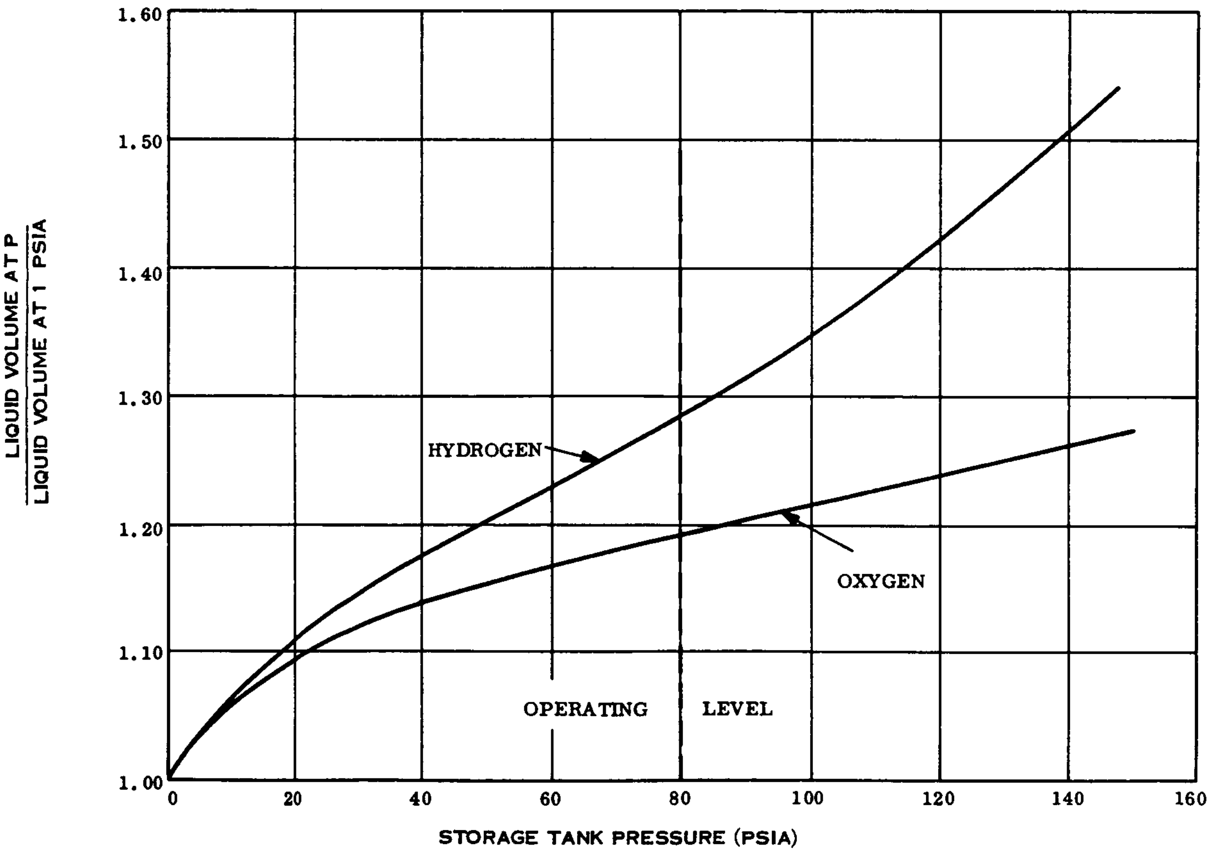

FIGURE 13. EFFECT OF STORAGE PRESSURE ON LH_2, LOX SPECIFIC VOLUME

saturated liquid and vapor densities of the stored fluid. Because the latent heats and densities are functions of pressure, the boiloff losses from the tank are likewise pressure dependent, as indicated in Figure 12. The dependency is such that boiloff rates increase as storage tank pressure increases. The magnitude of the effect is similar for both oxygen and hydrogen. Taking the average of the two curves, we see that the ratio of loss rate at 100 psia to that at one psia is 1.10/0.91 = 1.21. Thus, the loss rate at 100 psia storage pressure is 21 percent greater than at one psia. Figure 10, however, shows that the liquefier power requirements at 100 psia are only 46 percent of the power requirements for producing one psia liquid. Therefore, for the same power, 117 percent more product can be made at 100 psia than at one psia.

Densities of most fluids decrease as their temperature is increased. Because increased storage pressure results in higher liquid temperature, it is not surprising that the stored liquid density decreases as storage pressure increases. The effect is shown quantitatively in Figure 13. At 140 psia, it is seen that the specific volumes of hydrogen and oxygen are respectively 43 and 25 percent greater than the corresponding specific volumes at one psia storage pressure. Tankage weights will reflect the increased fluid volumes.

The liquid storage vessels are sized for a full 28-day production, corresponding to 1100 gallons for the hydrogen tank and 480 gallons for the oxygen tank. Use of superinsulation around each of these vessels reduces the LH_2 boiloff rate to 8 percent of the feed rate (0.06 pound per hour) and the lox boiloff rate to 2 percent of the feed rate (0.12 pound per hour).

Heat-Rejection Subsystem

The rejection of heat on the lunar surface is a problem which must be carefully considered in the design of a system suitable for operation on the moon. The lunar surface possesses a very low thermal conductivity; hence, the presence of a vehicle on the moon will locally alter the lunar surface temperature. Furthermore, those portions of the vehicle viewing the lurain will exhibit increased temperatures. There is little reflected radiation in the visible range, but a considerable quantity of heat is radiated from the moon. Lunar surface temperatures range from 250° F at the subsolar point to -250° F just before lunar dawn. Thus, in designing lunar radiators, one must consider both the extremes in temperature over full day-night operation and the interaction of the lunar surface with the radiator. The absence of a lunar atmosphere leaves radiation as the major mode of heat dissipation.

The electrolysis cells are responsible for the largest single heat-rejection load in the present system. These units must be maintained at a temperature of 85° F for optimum lifetime and efficiency, and thus the electrolysis heat load must be rejected at a low temperature (55° F). However, radiation at low temperatures is inefficient. Since it is desired to utilize the available surface area of the LM truck payload envelope for the radiators, radiator area will be limited to about 450 square feet.

Another important heat-rejection requirement is the minimization of overall system input power. Suggested power supplies for the cryogenic fuel production system have a capacity of about 30 kilowatts, so techniques such as use of a heat pump to increase radiator efficiency will be power limited.

Figure 14 indicates the heat load to be dissipated by each of the major components of the cryogenic fuel system as well as the total heat-rejection requirement of the system. The curves are based on the system design level of 80 psia product pressure and a liquefier compressor temperature of 70° F. Heat load is a function of the latter parameter; hence, there will be a decrease in heat-rejection requirements during the lunar night as the feed gas temperature falls.

To satisfy the constraints outlined above, the heat-rejection system depicted in Figure 15 was proposed. The heat rejected by the liquefier unit is at a relatively high temperature (206° F for a liquefier coolant inlet temperature of 70° F), and hence it is possible to efficiently dissipate the heat load directly from the liquefier radiator. The electrolysis heat rejection, however, is at a low temperature (55° F). In order to package the radiator as compactly as possible, a heat pump is utilized to raise the temperature at which the electrolysis heat load is rejected. Use of the heat pump decreases the required radiator area to the point where the electrolysis radiator will fit in the LM truck payload envelope.

The specifications of the heat-rejection subsystem are detailed in Table III. The lifetime is comparable with that of the rest of the cryogenic fuel production system. A regenerative heat exchanger is used for control of the coolant temperatures since it appears to provide the most efficient technique for maintaining a constant radiator outlet temperature to prevent fluid freezeup and drastic viscosity increases during the lunar night.

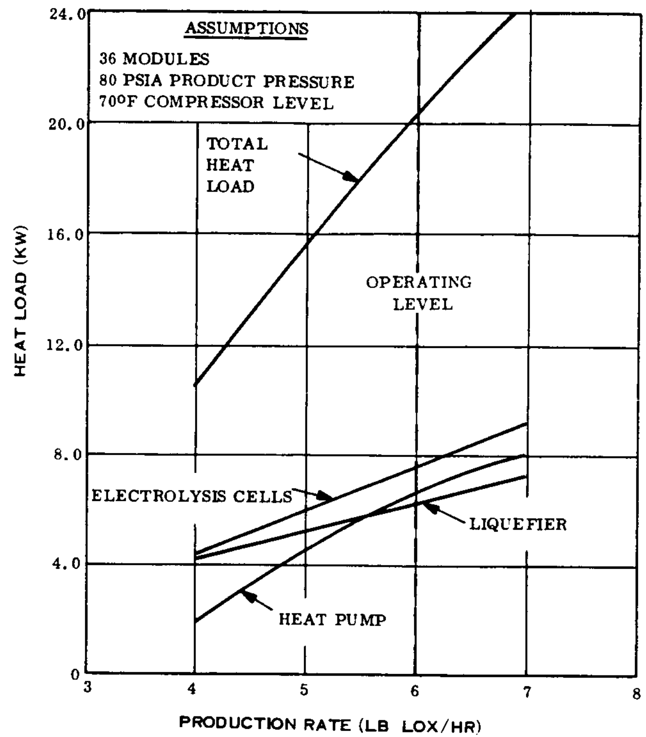

FIGURE 14. TOTAL HEAT-REJECTION REQUIREMENTS

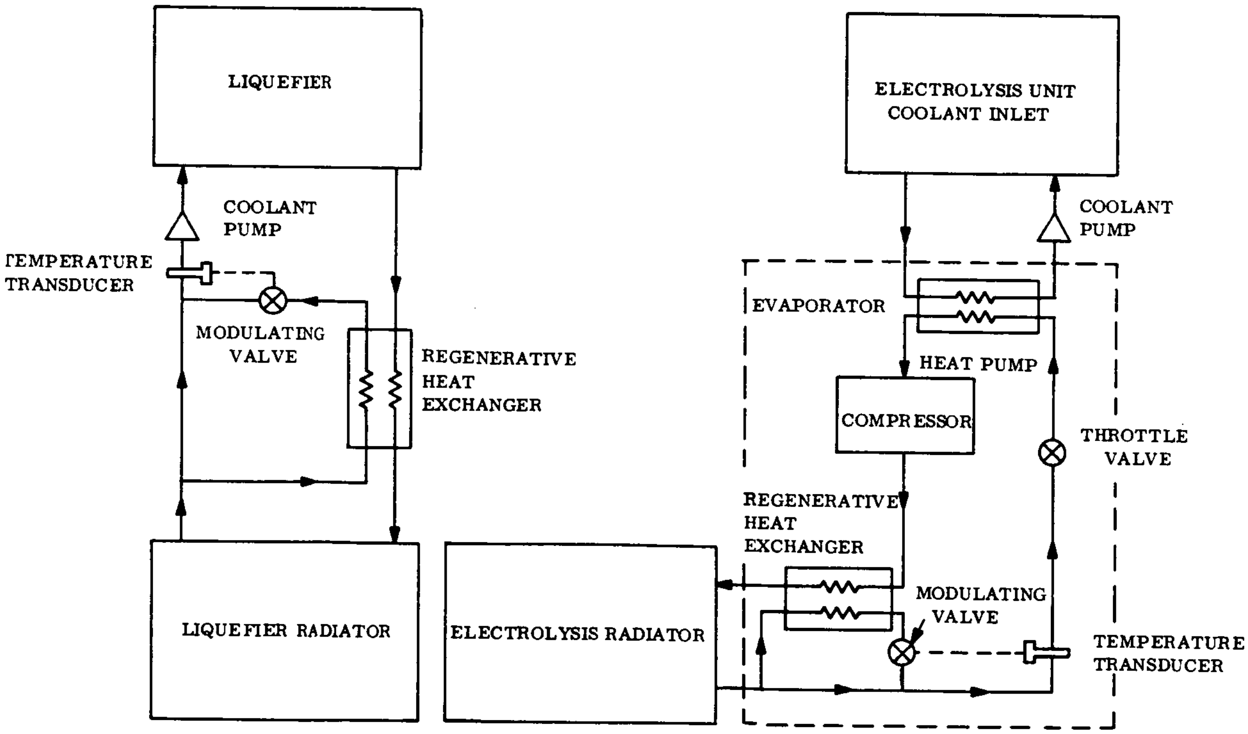

FIGURE 15. HEAT-REJECTION SYSTEM

TABLE III. HEAT REJECTION SUBSYSTEM SPECIFICATIONS

System	Coolant	Radiator α/ϵ	Coolant Temp T_{in} (a)	Coolant Temp T_{out} (b)	Radiator Area (ft^2)	Radiator Weight (lb)
Liquefier Cooling	Coolanol 15	0.2/0.9	206	70	203	101
Electrolysis Cooling	Freon 11	--	55	40	--	--
Heat Pump	Freon 11	0.2/0.9	202	40	440	220

Radiators

Aluminum Structure
2-Year Lifetime at 0.99 Survival Probability
Regenerative Heat Exchanger for Day-Night Temperature Control

Lunar Surface Shield

Radius = 3 × LEM Payload Radius = 279 inches
α/ϵ = 0.2/0.9

(a) T_{in} = temperature at radiator or evaporator inlet (°F)

(b) T_{out} = temperature at radiator or evaporator outlet (°F)

The interaction of the radiators with the lurain may be minimized in several ways, among them optimizing the attitude of the radiator with respect to the surroundings as a function of time and locally modifying the lunar surface by the use of a ground shield with a low solar absorptance-to-emittance ratio. The use of a ground shield three times the diameter of the maximum payload envelope of the LM and characterized by $\alpha/\epsilon = 0.2/0.9$ is a simple procedure which will materially increase the radiating capacity of the system.

The choice of coolant is based upon many factors, the most prominent being:

1. Saturation pressure, freezing point, flash point.
2. Low pump-power requirement for a fixed film temperature drop.
3. Low pump-power requirement for a fixed absolute temperature rise.

A survey of potential working fluids indicates that the best compromise coolants are Freon 11 for use below 200° F and Coolanol 15 above 200° F.

The best radiator coatings presently available have a value of $\alpha/\epsilon = 0.2/0.9$.

The electrolysis and liquefier radiators are depolyed from a telescoping storage configuration; thus, the available radiator area is about 450 square feet for each cooling system. Since only half of the allotted area is required for the liquefier system, the surplus liquefier radiator area could be used for electrolysis heat rejection to reduce the high heat-pump power required for the present system.

POWER SUPPLY SYSTEM

Power requirements for the cryogenic fuel production system are depicted as a function of lox production rate in Figure 16. The heat-rejection demands illustrated therein include both heat pump compressor and liquefier coolant pump power. Moreover, 250 watts have been added to the total system power to account for instrumentation, surge pumps, and similar miscellaneous demands. The liquid product pressure is assumed to be 80 psia, and liquefier compressor temperature is taken as 70° F. The latter parameter has a minor influence upon total system

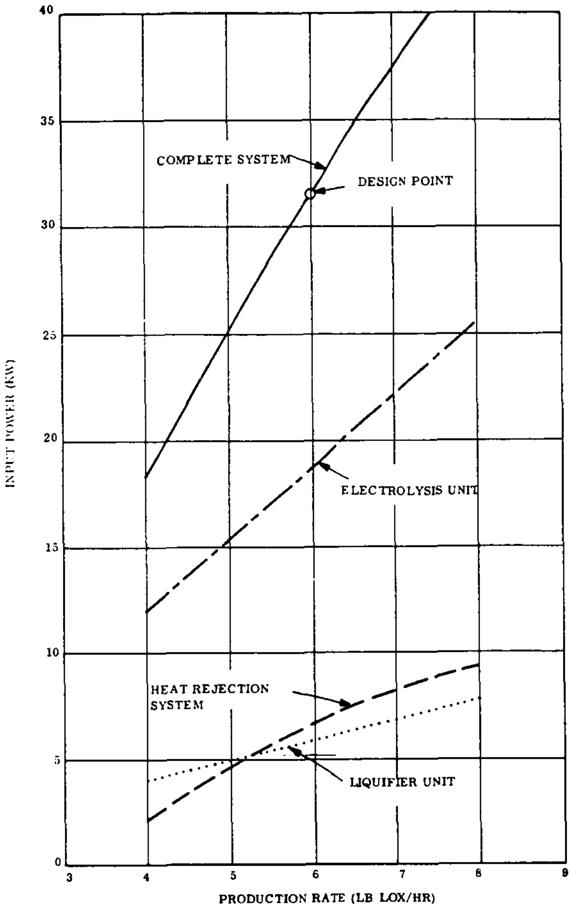

FIGURE 16. TOTAL POWER REQUIREMENTS

power, averaging about 170 watts per 10° F increase in compressor temperature. Cryogenic production rate is the dominant factor in determining system power demands, the requirements being about 5.25 kilowatts per pound of lox produced per hour.

At a given production level, the power demand can be reduced somewhat by use of additional electrolysis modules if the weight penalty is acceptable. The enhanced efficiency due to lower electrolytic cell current density will reduce the electrolysis power requirement as well as the corresponding heat-rejection load (thus diminishing heat-pump power). Furthermore, the liquefier radiator utilizes only half of its allotted 440 square feet. Use of the surplus area for electrolysis heat rejection will significantly reduce heat-pump power requirements.

Power demands during the lunar night will be below daytime levels. The lower temperature of the supply water and the corresponding reduction in compressor temperature could reduce liquefier power by as much as 700 watts. Heat-pump power will also decrease as a result of the lowered sink temperature.

Thus, it appears that a power supply of 30 kilowatts would be sufficient for the cryogenic fuel production system at a production rate of 6 pounds of lox per hour and 0.75 pound of LH_2 per hour. Such a power level appears to be realistic for early post-Apollo lunar surface operations, and systems utilizing the SNAP-8 reactor in thermoelectric and mercury Rankine power conversion cycles at the 30-kilowatt level have been proposed for that time frame. The pertinent characteristics for typical examples of such power sources, derived from several studies made by General Electric [6, 7], are summarized in Table IV. The indicated specific weights (exclusive of shielding) are representative of present technologies. The estimated lifetime of the thermoelectric

TABLE IV. CHARACTERISTICS OF TYPICAL EARLY POST-APOLLO POWER SYSTEMS

	35-kW Hg-Rankine	30-kW Thermo-electric
SNAP-8 Reactor	540 lb	540 lb
Power Conversion System	3950	4350
Heat Rejection System	5200	6900
Power Conditioning	350	350
Thermal Shroud	120	120
Activation Power Supply	445	445
Total Weight	10 605 lb	12 655 lb
Shield Weight (50-foot exclusion distance)	20 600 lb	20 600 lb
Specific Weight (exclusive of shielding)	305 lb/kW	420 lb/kW
Projected Lifetime	10 000 hr	30 000 hr

system is that projected for the early post-Apollo period. Sufficient redundancy has been included in both examples to ensure the indicated lifetimes.

Since the mass of the power source to be used with the cryogenic fuel production system is important for evaluating the economics of lunar fuel regeneration, a summary of the specific weights of a wide variety of possible energy sources has been included in Table V. This list has been adapted from the comprehensive data compiled by G. B. Stafford [8] with the addition of information on nuclear power systems in the 30- to 35-kilowatt range specifically applicable to the cryogenic fuel production system. Note that data on solar power sources were derived under the assumption of a (terrestrial) 300-mile circular equatorial orbit, which necessitates inclusion of provision for energy storage during periods of darkness. Thus, the specific weights of these sources are not inductive of lunar designs; they are included in view of the absence of more accurate information.

TABLE V. SPECIFIC WEIGHTS OF POSSIBLE LUNAR POWER SOURCES (AFTER STAFFORD [8])

Power Source	Pounds per kW (1965)	Pounds per kW (1972)	Comments
Solar Sources			
Silicon Solar Cell			Batteries included for energy storage
Panel Mounted	275	90	Orientation to ± 15°
Body Mounted	535	150	No orientation
Thin-Film Solar Cell			Batteries included for energy storage
Panel Mounted	---	85	Orientation to ± 15°
Body Mounted	---	344	No orientation
Solar Thermoelectric			Batteries included for energy storage
Flat Plate	300	140	Orientation to ± 15°
High-Temp Concentrating	715	330	Orientation to ± 0. 5°
Solar Thermionic	286	139	Batteries included for energy storage Orientation to ± 0. 1°
Solar Dynamic			Orientation to ± 0. 5°
20-kW Mercury Rankine	---	85	
20-kW Brayton	---	180	
Nuclear Sources (Shielding Included)			
Reactor: 30-kW Thermoelectric	1110*	274	Shielding: 50-foot exclusion distance assumed for 1965 data; unspecified for 1972
Reactor: 35-kW Dynamic	890*	530*	
Nuclear Sources (Lunar Materials Used as Shielding)			
Isotope Thermoelectric	900	300	Curium^{244} fuel
Isotope Thermionic	365	85	Polonium^{210} fuel, 1-year lifetime
Isotope Dynamic			Polonium^{210} fuel
3-kW Mercury Rankine	200	180	
1.5 kW Brayton	400	320	
Reactor Thermoelectric	420*	<274	
Reactor Dynamic			
35 kW	250	100*	
100 kW	---	23	
Reactor Thermionic	---	14	1000-kW level

* Data from General Electric studies[6, 7] and personal communications. All other specific weights from Stafford.[8]

The choice of a suitable power system is governed less by weight comparisons than by the availability and practicality of the system for the mission time frame. With this guideline, the 30-kilowatt power requirement for the cryogenic fuel production system seems to be in step with the proposed space-qualified power supplies expected to be available for the early post-Apollo period.

CHARACTERISTICS OF THE COMPLETE SYSTEM

The specifications for the complete cryogenic fuel system are compiled in Table VI. The production rates of 6 pounds of lox/per hour and 0.75 pound of LH_2/per hour are the maximum flows which appear feasible under the joint constraints of minimum system weight and an early post-Apollo power source. The weight-volume breakdown indicates a gross mass of 4153 pounds. Included in the surge system weight are both surge pump and tankage requirements. Similarly, the 10 percent weight allowance incorporates the water feed pump, thermal shroud, radiator deployment mechanism, structural support, and insulating blanket around the inside of the radiator, and miscellaneous hardware. Exclusive of storage requirements, which are greatly dependent upon the mission for which the regeneration unit is intended, the system mass factor is 522 pounds per pound of lox produced per hour. A power penalty of 93.2 pounds of regeneration hardware per kilowatt is incurred. incurred.

TABLE VI. SYSTEM CHARACTERISTICS

Specifications	
Power Requirement	31.5 kW
Production Rate	6 lb lox/hr
	0.75 lb LH_2/hr
Boiloff Recycle Rate	0.12 lb lox/hr
Storage Pressure	80 psia
Storage Capacity	1 month production
Surge Capacity	20 hours production
Liquefier Radiator Temperature	206° F
Electrolysis Radiator Temperature	202° F
Lifetime	10 000 hours

Weight-Volume Analysis	Weight (lb)	Volume (ft^3)
Electrolysis Cells	1080	10
Liquefier Unit	370	10
Storage Tanks	1020	211
Surge Tank System	180	44
Heat Rejection System	562	61
Instrumentation	500	25
10% Allowance	441	86
	4153	397

Operational Requirements	Electrolysis Unit	Liquefier Unit	SNAP-8 System
Startup Time (Man-Hours)	1	6	10
Operation Time (Man-Hours/Day)	0	0	3
Maintenance (Man-Hours/Year)	4	8	-

The system is designed to operate with minimal attention, as indicated in Table VI. Initially, one hour will be required to bring the electrolysis unit on line and six hours to cool down the liquefier. Commencing liquefier cool-down during the lunar night would speed up the process. Both subsystems should require no attention after start-up. Operator time indicated for the power source is purely diagnostic and could be reduced or eliminated with proper exploitation of automatic controls.

No power supply maintenance appears possible in view of the difficulty of reactor repairs. The liquefier-electrolysis maintenance of 12 man-hours per year consists primarily of monitoring and occasionally purging individual electrolysis cells as well as replacement of liquefier compressor rings. The surge tanks permit liquefier or electrolysis shutdown for up to 20 hours without interfering with the rest of the system. If the electrolysis unit were temporarily inoperative, the liquefier could draw from the surge gas supply. Conversely, if the liquefier is disconnected, gas produced by the electrolysis unit may be stored in the surge tanks. Note that the 10 000-hour design lifetime is a conservative projection of present capabilities to the early post-Apollo time frame.

Plan and side elevation drawings of a complete cryogenic fuel regeneration system packaged in an LM truck are illustrated in Figure 17. Components are arranged for proper location of the center of mass, and cryogenic supplies are paralleled with the surge lines to permit efficient recycle of boil-off and flash losses. The electrolysis and liquefaction radiators are depicted prior to deployment. During operation, the liquefaction radiator would be raised above the electrolysis radiator, providing 440 square feet of usable area for each system.

If necessary, the lunar module control unit could be relocated to free the central portion of the payload volume for a water storage tank. Such a vessel would provide an integral water supply that could greatly increase the flexibility of the system by eliminating dependence upon external water sources. About 2850 pounds of 7000-pound LM truck payload capacity is not used by the cryogenic fuel system; thus, a 2500-pound integral water supply would be feasible. These reserves would be more than sufficient to meet most of the lunar roving vehicle fuel demands indicated in Table I.

The packaging of a cryogenic fuel production unit and its power source on the same LM truck is an intriguing possibility. Although many problems exist in mounting and isolating the two systems, a rough estimate of the capabilities of such a unit can be obtained. Neglecting storage requirements, the mass-power burden, M_R, for the regeneration unit is 93.2 pounds per kilowatt. The specific weight, M_P, of several power systems can be obtained from Table V. Then the power supply capacity, P, which can be included in a 7000-pound LM payload in conjunction with a cryogenic fuel system sized for the available power is given by

$$P = \frac{7000}{93.2 + M_P} .$$

The rate of lox production is P/5.25, where 5.25 is the regeneration system power requirement per pound of lox produced per hour. Figure 18 illustrates the cryogenic production rates possible for a self-contained LM-packaged system using any of several proposed nuclear power sources. The advantages of using lunar materials for shielding are evident: cryogenic production rates well in excess of the present capability could be achieved by delivery of a single LM truck utilizing an advanced (100 pound per kilowatt) dynamic reactor in conjunction with indigenous lunar shielding.

SYSTEM EVALUATION

The advantages of the proposed fuel production system are twofold: general benefits accruing from the existence of a lunar cryogenic facility and the particular advantages of the suggested unit.

Pre-eminent among the latter qualities is the efficiency of the overall fuel production approach employed: the use of ion exchange membrane electrolysis cells in conjunction with an optimized high-pressure Claude cycle liquefier results in minimal power demands consistent with projected lunar power and heat-rejection limitations. Other specific advantages include convenient scaling of the production rate as larger power sources become available and total weight low enough to permit packaging of the system on a lunar module truck with sufficient surplus payload capacity for inclusion of a substantial water supply. Maximum conservation of fluids is achieved by the use of surge tanks and boiloff recycle. The system is designed around available space-qualified hardware whose flexibility, lifetime, and efficiency permit integration with a great variety of lunar missions.

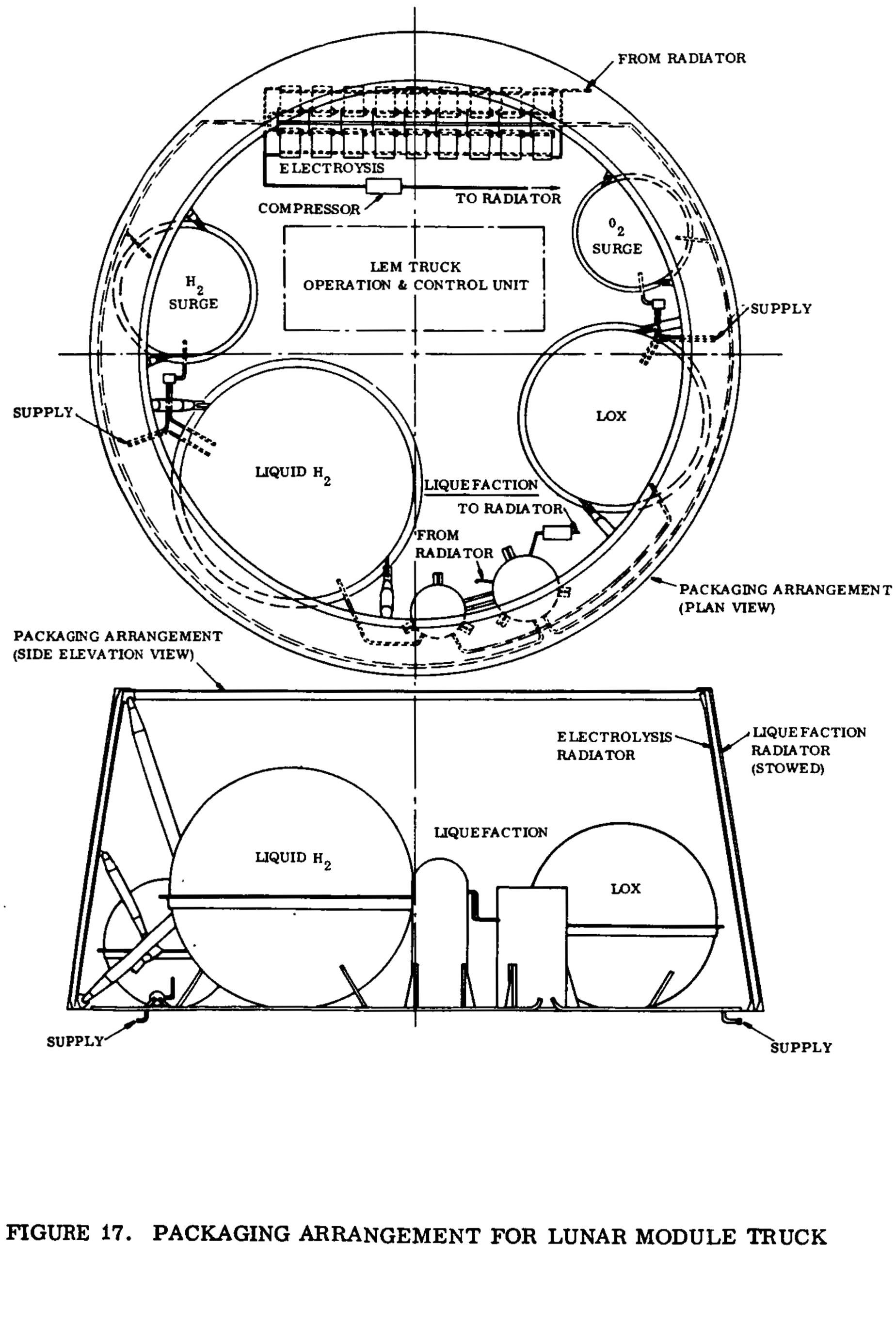

FIGURE 17. PACKAGING ARRANGEMENT FOR LUNAR MODULE TRUCK

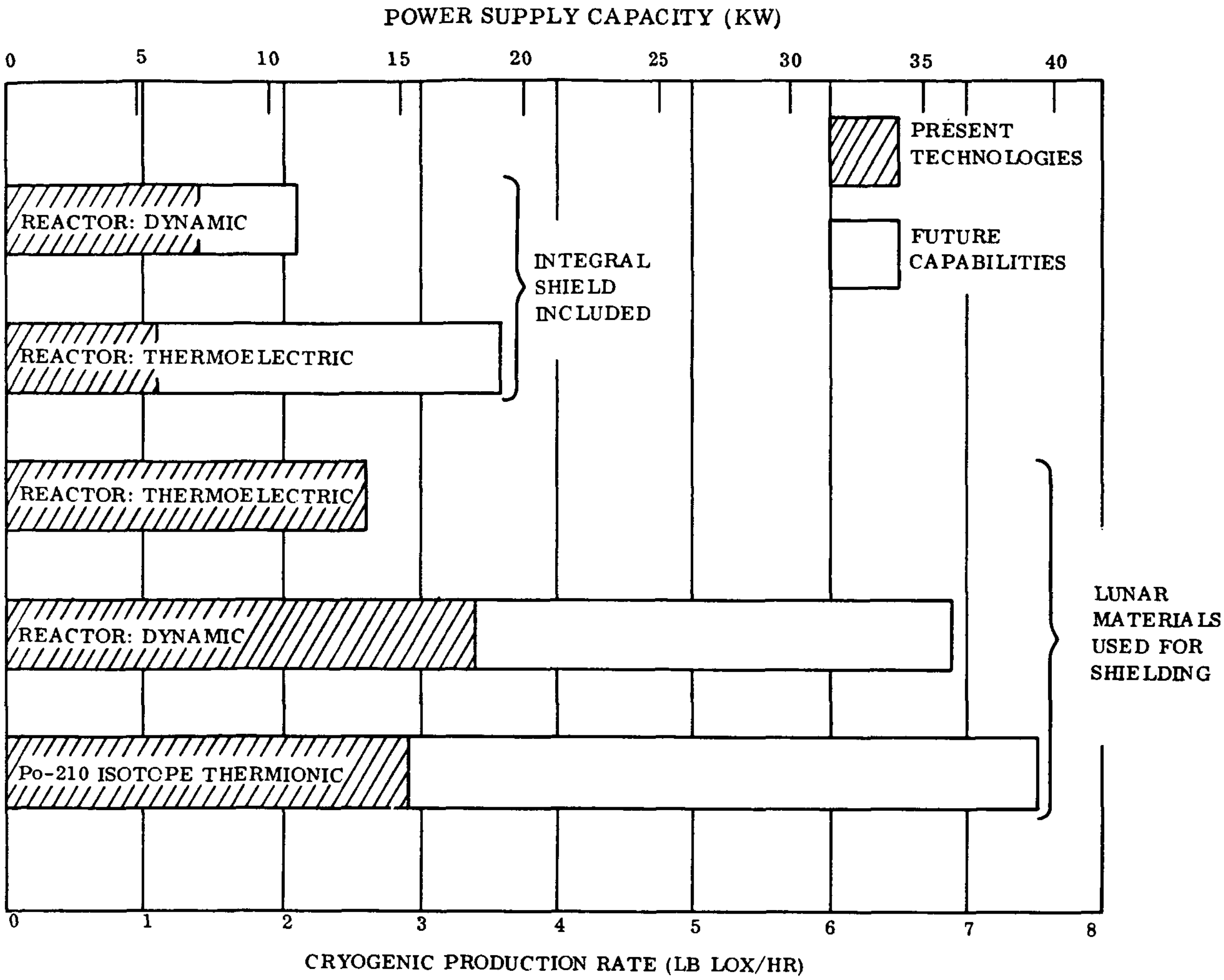

FIGURE 18. CRYOGENIC PRODUCTION CAPABILITY AND POWER SUPPLY CAPACITY/ DELIVERABLE BY A SINGLE LUNAR MODULE TRUCK

A convenient figure of merit for evaluating the utility of a lunar fuel regeneration facility is the ratio

$$K = \frac{\text{System lifetime}}{\text{Payout time}} = \frac{L}{T} \tag{3}$$

The payout time is the period required for the system to produce its own weight in cryogenics. Thus, K is an index of the advantage of lunar fuel production over transportation of cryogenics to the moon. If K is greater than unity, utilization of lunar fuel regeneration can be expected to reduce the cost of lunar logistics.

The proposed system is designed for a total net cryogenic production rate (corrected for boiloff recycle) of 6.57 pounds per hour. The system payout time is 5382 hours if the weight of a SNAP-8 mercury Rankine power supply with integral shield (Table V) is included. For the designed system lifetime of 10 000 hours, K equals 1.9, indicating a significant advantage over transportation of cryogenics from Earth. Utilization of more advanced power sources and/or lunar shielding will greatly improve the figure of merit.

The existence of an efficient regeneration system adapted to lunar operation underscores the importance of conserving early post-Apollo water reserves, whether logistic supplies or natural resources. Mission philosophy should reflect the availability of regenerated cryogenic fuels for use in closed-cycle operations. It is felt that the presence of economical regeneration facilities will render unattractive the proposed practice of using water as an expendable evaporant, especially in early post-Apollo applications.

Roving vehicle design must also reflect the advantages of utilizing regenerated fuel. For example, tank capacities could be reduced to exploit the existence of resupply capabilities. On the other hand, the power reduction from the use of high product pressures during liquefaction indicates that minimal power supply weight penalties and efficient fuel regeneration will require that vehicle tanks withstand high stresses.

The economics of the lunar refueling of interplanetary and logistic spacecraft depend upon the efficiency of cryogenic fuel production, the availability of lunar water resources, the existence of suitable material processing facilities and techniques, and the presence of effective power supplies [9]. An indication of the utility of lunar propellant production may be obtained by extending the figure of merit, K (equation 3), to the complete cryogenic fuel production process.

$$K = \frac{\text{Total mass of propellants manufactured on moon}}{\text{Hardware delivered to moon for production and maintenance}} = \frac{W}{H} \quad (4)$$

Equation 4 accounts for all fuel production hardware: mining apparatus, water-extraction plant, power supplies, cryogenic fuel production system, etc. The total mass of propellants manufactured by the facility is:

$$W = L \cdot D$$

where L is the system lifetime (hours) and D is the average hourly cryogenic demand (pounds per hour). The hardware requirement consists of the fuel manufacturing plant, fluid storage facilities, and the power source. The complete manufacturing hardware burden (mining and rock-handling equipment, water refinery, electrolysis-liquefaction system, etc.) may be characterized by the factor

$$M = \frac{\text{Pounds of manufacturing hardware}}{\text{Pounds of cryogenic fuel production per hour}} \cdot$$

If the plant is scaled to the demand, the total manufacturing hardware weight is $M \cdot D$. Cryogenic storage requirements are given by

$$S = G \cdot F \cdot D \cdot L$$

where G is the specific weight of cryogenic tankage, F is the fraction of the total fuel production which is stored, and L and D are defined above. The power weight penalty, N, is

$$N = M_P \cdot R \cdot D$$

where M_P is the specific weight of the power source and R is the manufacturing power demand per pound of fuel produced per hour. Combining these definitions with Equation 4,

$$K = \frac{L \cdot D}{M \cdot D + G \cdot F \cdot D \cdot L + M_p \cdot R \cdot D} = \frac{L}{M + M_p R + G \cdot F \cdot L}. \quad (5)$$

This expression could be utilized with the projected characteristics of any lunar propellant facility to evaluate the economics of lunar fuel production: It has been applied to the present system to illustrate the effectiveness of the unit for large-scale propellant production. The system parameters yield:

$$K = \frac{10\ 000}{464 + 4.66\ M_p + 0.595 \cdot 0.0672 \cdot 10^4} = \frac{10\ 000}{864 + 4.66\ M_p}. \quad (6)$$

This expression has been plotted as a function of M_P in Figure 19 along with the estimated specific weights of several proposed power sources. For the most advanced power supplies, the system will produce nearly eight times its weight in cryogenic fluids during its 10 000-hour lifetime.

The utility of the proposed system for large-scale cryogenic propellant production can be evaluated using a study by Cole and Segal [10] of the logistic savings realizable with various modes of lunar refueling for a typical lunar logistics mission. The latter is assumed to consist of the round-trip delivery of a 125 000-pound payload between the Earth and the moon. The total payload required in Earth orbit has been evaluated as a function of the figure of merit, K, and the mode of lunar propellant resupply: lunar surface refueling, lunar orbit refueling (with a shuttle to transfer fuel from the lunar surface), and no refueling.

The results are depicted in Figure 20 [10]. Without refueling, the mission requires the delivery of 1450 kilopounds of payload into near-Earth orbit. Lunar refueling can provide a 19 percent saving in logistic weight even under present technologies (dynamic reactor with integral shield), and a 36 percent reduction is possible using the most advanced nuclear power sources. The point at which lunar fuel production just becomes economical is K = 1.25 (assuming a two-stage vehicle for direct flight and a single-stage vehicle for the refueled mission). This figure of merit implies system practicality for power source specific weight less than 1530 pounds per kilowatt. Below power burdens of 475 pounds per kilowatt, lunar orbit refueling would be the optimal resupply mode if a shuttle were available to convey propellants from the moon's surface to lunar orbit.

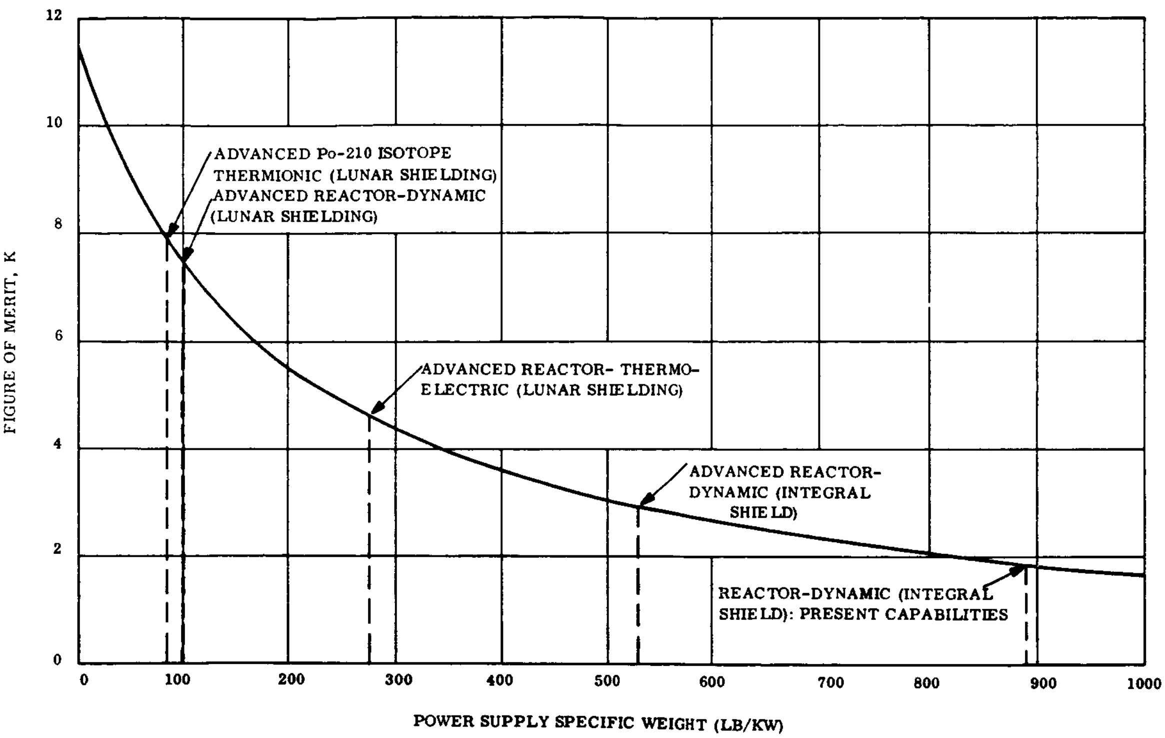

FIGURE 19. CRYOGENIC FUEL PRODUCTION SYSTEM FIGURE OF MERIT AS A FUNCTION OF POWER SUPPLY SPECIFIC WEIGHT

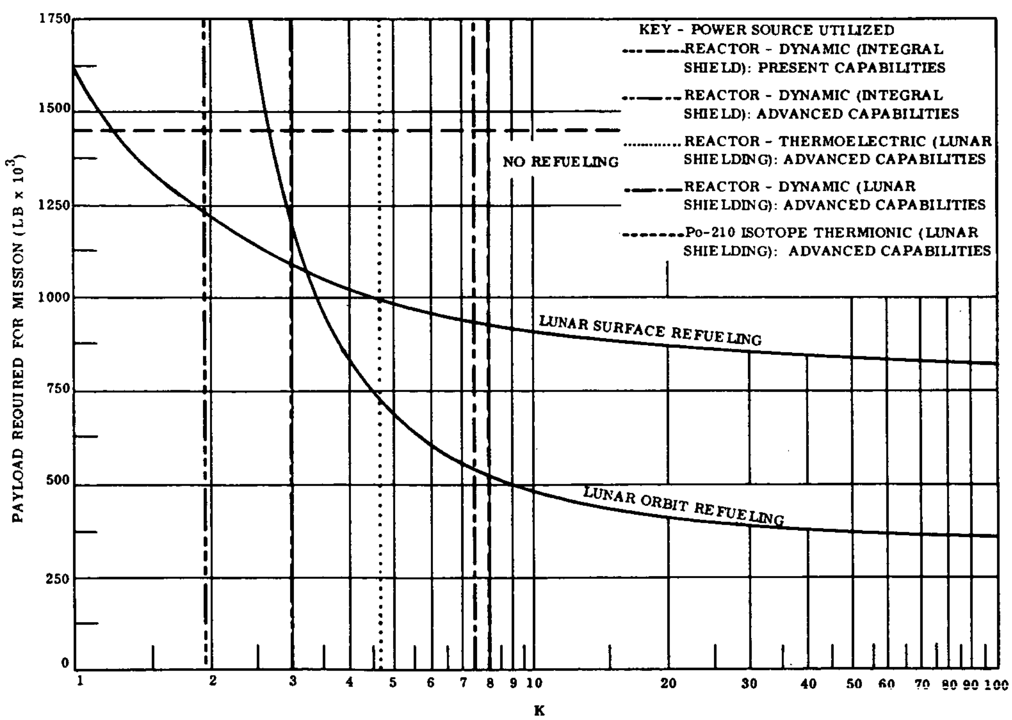

FIGURE 20. TOTAL PAYLOAD REQUIRED FOR LUNAR LOGISTICS MISSION AS A FUNCTION OF REFUELING MODE AND EFFICIENCY

CONCLUSIONS

The advantages and feasibility of lunar cryogenic fuel production for the early post-Apollo period have been demonstrated. Using existing operational hardware, a system having a cryogenic production rate of 6 pounds of lox per hour and 0.75 pound of LH_2 per hour is practical under the joint constraints of minimal weight and early post-Apollo power limitations (30 kilowatts). Early availability is enhanced by utilization of the LM truck for delivery to the lunar surface, as illustrated in Figure 21. Power supply capacity available on the lunar surface in the early post-Apollo period will greatly influence system size and production rate. Yet, even with the use of integral shielding and current reactor technologies, the lunar cryogenic fuel system can produce nearly twice its weight in fuels. This production rate can easily be extended to higher levels as power sources are improved and new missions are identified.

Acknowledgement. - The foregoing analysis was conducted under the direction of Dr. Rodney W. Johnson while he was manager of Lunar and Planetary Systems Development at General Electric's Missile and Space Division. His timely suggestions and encouragement are gratefully acknowledged.

Liquefaction and storage system design was conducted by Air Products and Chemicals Inc., Allentown, Pennsylvania.

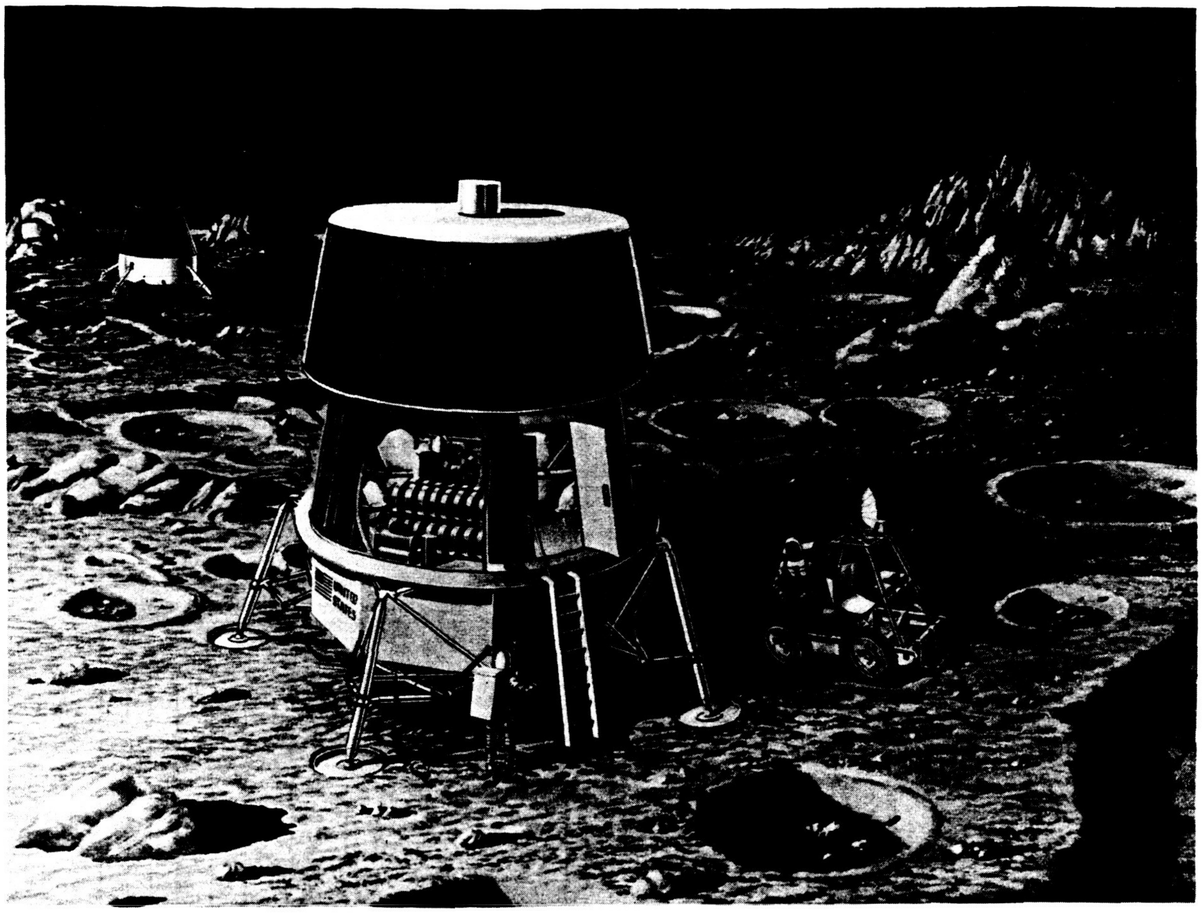

FIGURE 21. EARLY POST-APOLLO CRYOGENIC FUEL PRODUCTION SYSTEM

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SURFACE MINING ON THE MOON

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INTRODUCTION

The members of the Working Group on Extraterrestrial Resources are aware that there are many potential advantages in the use of indigenous minerals in connection with the space program over the alternative of transporting everything from the Earth.

The safety and flexibility of lunar colonies and expeditions will be greatly enhanced by availability of lunar sources of supply. If return fuel and oxidizer are available on the moon, colonies will no longer be dependent solely upon rigid adherence to flight schedules from Earth. In the event of a lunar plant or supply breakdown, fuel and oxygen can be supplied from the Earth; if there are Earth launch failures or delays, local supplies will insure that the colony is not endangered. If indigenous fuel materials are not available, a long term supply from Earth may be required for lunar storage.

The technolcgical requirements for establishment of mining and processing facilities on the moon will not only tax man's ingenuity to the utmost, but will produce an abundance of scientific data which will benefit present technology. This will also enhance the advanced technologies required for later and longer-term expeditions to Mars and to other planets of the inner solar system.

The use of lunar resources, if they exist, will increase man's understanding of the geology and structure of the moon. The exploration for the mining of minerals should add significantly to the detailed knowledge to be gained from Apollo and post-Apollo scientific expeditions.

The effects of vacuum conditioning on granular rock materials are being investigated in the Space Sciences Laboratory of Marshall Space Flight Center. Some results of experiments are presented in this paper which concern the effects of grain packing factors, shape and size characteristics, and their surface roughness. Presented also are some results from certain atmospheric and vacuum experiments to determine the change in coefficient of friction between metals and rock material. This is related directly to the experiments on the adhesive properties and bearing strengths of such material in a vacuum environment. These experiments and studies are providing a better understanding of the mechanical properties of the material expected to be present on the surface of the moon. Results from these and other experiments are necessary for the proper design of lunar mining systems.

This report to the Working Group on Extraterrestrial Resources evaluates some of the known problems and possible techniques for surface mining of water deposits on the moon.

Finally, if it is found that indigenous lunar resources can compete economically with the same or similar materials transported from the Earth, they will be utilized. If, however, this cannot be shown, the chances for the mining and use of these materials, even if deposits are present, are not probable and must await changes in supplies and economic position of Earth materials.

FACTORS AFFECTING LUNAR SURFACE MINING

Environment

Vacuum. - The most pronounced effect upon lunar surface operations will be extremely low pressure and the physical effects associated with or caused by this near-vacuum atmosphere. Any surface mining operation will involve the suited astronaut working in this environment.

The maximum atmospheric density is known to be less than 10^{-6} of Earth atmosphere at sea level [1]. Estimates of the lunar surface pressure range from 1.3×10^{-10} N/m^2 to below 1.3×10^{-12} N/m (1×10^{-12} to

1×10^{-14} torr). The work of Elsmore with the Crab Nebula is cited by de Wys [2] who states that the lunar atmospheric density is 10^{-13} that of Earth.

Some of the probable atmospheric constituents include water, carbon dioxide, and hydrogen. Carbon has been identified by Kosyrev from spectrograms of the crater Alphonsus [2].

As will be discussed later, the vacuum environment will probably impose severe constraints on the design of systems and equipment to be used in mining. The frictional and adhesive characteristics of lunar material will affect the power requirements of mining systems and ore transportation equipment. Lubrication of machinery, rock breakage and removal, and the stability of artifical slopes are all affected by the vacuum environment. The problem of lubricating bearing surfaces exposed to vacuum is well recognized. Slope stability is indicated by a material's angle of repose. The mine face or overburden slope should be more stable than the same slope angle on Earth since the factors of vacuum adhesion, lunar gravity, and extreme particle roughness favor greater stability [3].

The astronaut must be protected, both from the hostile native environment and from the induced hazards associated with mining, such as machinery, blasting, rock falls, etc. Protection from mining hazards must be accomplished not only with the suit, but through proper design of equipment, techniques, and procedures to insure astronaut safety while he is performing the necessary related tasks. If the mining operation is large enough to require mobile equipment, pressurized astronaut compartments will be required. The danger from rock debris is too great to allow the astronaut to be in an exposed location without protection while mining operations are in progress. If the lunar surface program is extensive enough to require a mining operation, even on a limited scale, subsurface shelters should be available to give protection during rock blasting or other highly hazardous periods.

Thermal. - The maximum and minimum observed temperatures of the lunar surface have been 389 ± 4°K and 95 ± 5°K, while 210°K is estimated at a depth below which diurnal solar heating is negligible [4]. Aside from the more obvious problems associated with the astronaut and his equipment, the thermal environment could impose some unusual mining problems. Assume that the ore is covered by an overburden which has been stripped of its water to some depth by the interaction of vacuum, temperature fluctuations, solar wind, etc. The ore may be turned into overburden by exposure to these environmental conditions; this may require that the actual working face of the open pit be covered or otherwise protected. The depth to which ore could be turned into overburden would depend upon the extent to which the particular rock is susceptible to outgassing. Mining performed at night or in permanently shaded areas may reduce the gaseous emissions from the rock and allow an ore of higher water content to be extracted.

Solar Wind and Cosmic Debris. - The lack of any appreciable atmosphere allows the alpha particles and protons of the solar wind to strike the lunar surface with full force. Infrared and ultraviolet radiation is more intense than here on Earth. Whatever hard shelters are provided for vacuum and thermal protection should be adequate for the solar wind except during periods of high solar flare activity. No unique problems are expected with mining systems or operations because of the solar wind. Interruptions from time to time for personnel protection during flares may be an exception.

The high speed collision of meteoroids and micrometeoroids with the surface could be a real danger to the astronauts and to the surface-mounted equipment. The magnitude of the danger has not been fully determined primarily because the frequency, velocity, and mass of these impacting materials are not known. The particle sizes range upward from micron or submicron size to perhaps many meters, while their velocities could vary from 10 to 72 km/sec [5]. Estimates have been made [6] that the chances of a one-gram particle hitting an astronaut on the lunar surface are less than one in 3×10^{14} Earth days. A strong erosional process could result from the impact of much smaller particles. It has been estimated [7] that an area of 1 to 10^4 cm^2 of the moon is struck each second by a particle weighing more than 10^{-10} g. The resulting erosion, either of native material or exposed equipment and structures, would be a product of this "lunar weather." This would give the appearance of a finely etched surface and would not be of major consequence except for unprotected optical surfaces or precision machined parts of machinery.

Lunar Gravity. - Acceleration due to gravity on the moon is about 1.62 to 1.63 m/sec^2, or approximately 0.166 that of the Earth. The Earth's variable tidal attraction on the moon reduces the lunar gravity on the near side by 3×10^{-5} m/sec^2 and increases it by the same amount on the far side. Lunar mining equipment would be designed for the mean gravity, as any anomalies should not be large enough to affect power requirements or load-handling capabilities.

Lunar mining techniques will probably be modifications of conventional methods. The lower lunar gravity will allow increased drawbar pull and decreased rolling resistance of mobile equipment such as ore loaders, transporters, and other rolling machinery.

Vibration and Shock. - Seismic activity would result from moonquakes and meteorite impacts or explosive shock caused by scientific experiments performed by the astronauts. Quakes and the shocks of impacts are potentially troublesome for mining operations. The stability of foundations, mine working faces, overburden slopes, and process equipment could be modified or destroyed. The magnitude of the forces involved must be determined from seismological data gathered from the moon.

Surface Structure and Strength

Texture. - Photographic evidence from Surveyor I, the Ranger series spacecraft, and terrestrial observations is beginning to give a better understanding of the surface on the moon. Admittedly, Surveyor I data do not answer the question of the entire surface, but do give a view of one local area.

Earlier estimates of a rubble surface texture [3] are supported by the Surveyor pictures. The surface material is fragmented, porous, and partly smoothed by some process. The granular soil-like material shown on the photographs appears to range from fine grain to coarse. Some of the results of a preliminary analysis [8] are shown in Figure 1 as the cumulative frequency distribution of particles on the lunar surface. The material disturbed by the footpad of the spacecraft exhibits a coarser texture than the undisturbed material. The coarser, undisturbed material may be agglomerates of finer material. This indicates that the granular portions of the surface have a distinct amount of cohesion. It is, in some respects, qualitatively similar to a damp, fine-grain soil on Earth.

The photographs show the surface to be littered with coarse blocks and fragments. The distribution of the larger blocks appears to be fairly random, but local concentrations can be seen on the flanks of recognizable craters. Most of these blocks are probably debris from within the crater and are thought to consist of relatively strong rock [8].

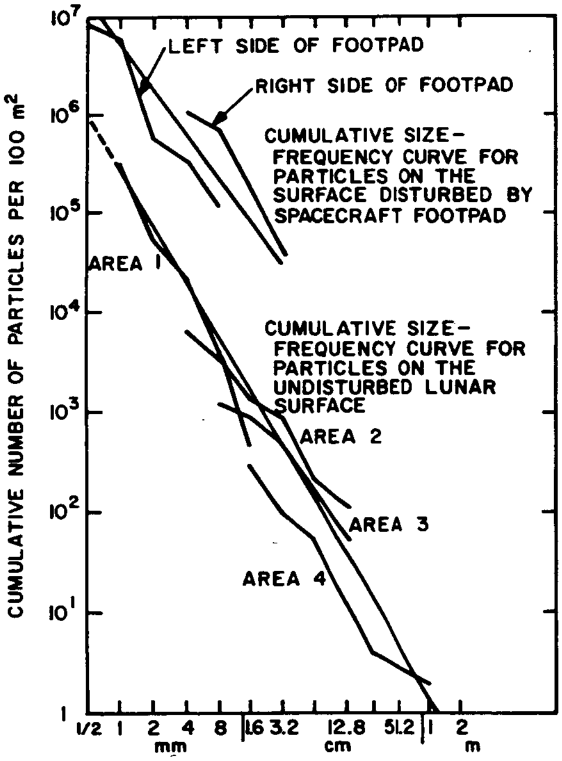

FIGURE 1. CUMULATIVE FREQUENCY DISTRIBUTION OF PARTICLES ON THE LUNAR SURFACE [8]

The implication from the Surveyor I data is that relatively solid rock lies beneath the surface rubble. This is the potential ore from which water could be extracted to support extensive lunar exploration. If the depth of rubble is not too great, surface stripping would not be as difficult as opening an underground mine and could be done with fewer men and less equipment.

Strength. - There are indications [8] that this surface has a static bearing capacity on the order of 4×10^5 dyn/cm^2 or 5 lb/in.^{2}. A man who weighs about 80 kg (180 lb) would exert a pressure, while standing, of about 2.1×10^5 dyn cm^2 or 3 lb/in.^{2} here on Earth. If this strength increases with depth, which is reasonable to assume, stable support foundations can be constructed. Considering the small depth to which the footpads penetrated the surface material, vehicle sinkage and rolling resistance should not be a significant problem.

However, other areas could have quite different surface characteristics and engineering properties. If some erosional and transportation process has been operating on the moon, there could be areas of local deposition of material which are supported by an openly dendritic structure. If such a soil profile exists, its strength could be so low as to preclude normal surface operations. This soil structure need not be composed of dust or micron-sized particles to have this characteristic if the vacuum adhesion or cohesive tendency is strong enough to bond individual pieces as they are deposited. If such areas do exist, it is expected that they would be local in nature and not abundant.

MATERIAL BEHAVIOR IN VACUUM

The behavior of material in a lunar environment is of interest to designers of lunar equipment and planners of lunar missions. This is of particular interest to mining engineers because of the necessity to handle and transport mass quantities of surface and subsurface material. There are many methods of mining which could be used on the lunar surface. Some of these are discussed in detail later in this paper. The method of mining selected for use on the lunar surface will be determined after a large number of variables have been evaluated. Some of these will be equipment weight, power requirements, astronaut participation, lunar surface and subsurface composition, and the effect of the lunar environment on the properties of these materials.

There is a large effort being exerted by a number of scientists to evaluate the effect of a lunar environment on the physical and mechanical properties of simulated lunar materials. The program is complicated because of the large number of variables to be considered and tests often necessary to establish reliability in the observed results. Ultra-high vacuum is probably the most influencing parameter of the lunar environment on the mechanical characteristics of the surface material. Consequently, this is usually the first variable to be studied in the laboratory and appears to be of major importance in lunar mining problems.

Of particular interest to those concerned with lunar surface mining is the effect of the vacuum on the coefficient of friction between engineering materials and the lunar surface material. Also of importance is the possibility of lunar surface material adhering to engineering materials with which it comes in contact when disturbed by operations on the lunar surface. These two problems are being studied and a detailed discussion follows.

Friction. - The effect of a vacuum environment on the coefficient of friction of two metal surfaces has been studied in detail. However, despite this widespread interest in metals, very little study has been directed toward the coefficient of friction of metals on simulated lunar materials.

Some studies have been made to determine friction and wear of various combinations of aluminum, steel, basalt, and rhyolite slabs at ambient pressure, in an argon atmosphere, and at a vacuum level of 10^{-7} N/m^2 (10^{-9} torr) [9]. This study was oriented toward the wear portion of the program. The coefficient of friction was studied as a function of speed, load, and pressure. Figure 2 [9] shows how the coefficient of friction varied with load, generally decreasing as the load increased, for aluminum on aluminum and aluminum on basalt. The coefficient of friction was larger in vacuum than in atmosphere in each test regardless of the load applied.

An experimental program is currently in progress to study the effect of vacuum on the coefficient of friction of metals on granular nonmetallic materials. The metallic materials are aluminum 7075 and steel AISI-1020. The nonmetallic materials are quartz and basalt in the size ranges 37 to 44 microns and 250 to 500 microns in diameter. Selected tests are also performed on a 50 percent mixture (by volume) of these sizes.

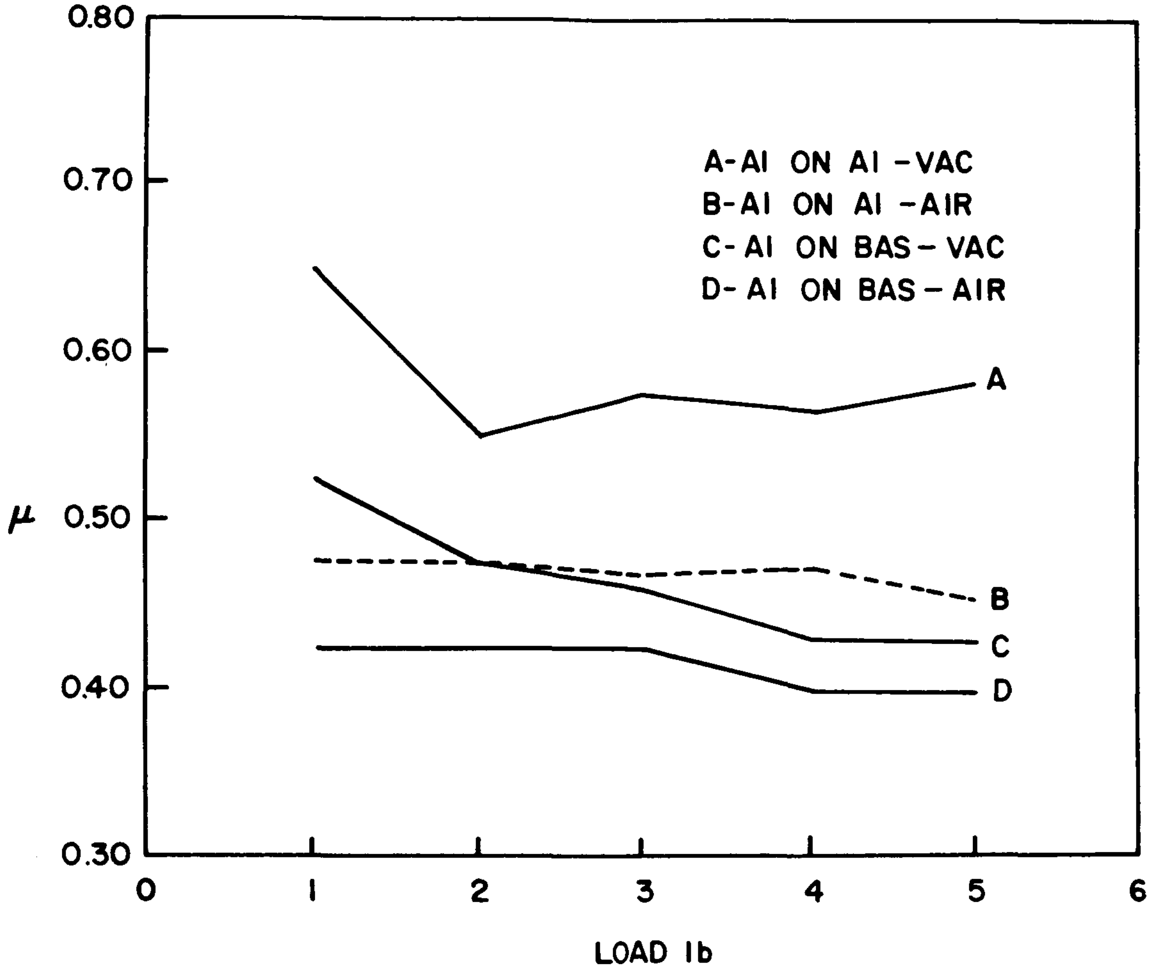

FIGURE 2. COEFFICIENT OF FRICTION VERSUS LOAD [9]

The tests were performed in a vacuum of 10^{-5} N/m^2 (10^{-7} torr) during the first phase of this program. The coefficient of friction is higher in vacuum than in atmosphere for all tests completed during this phase. However, there are many variables which influence the measured values. Some of these cause an increase while others cause a decrease in the measured coefficient of friction.

Tests were made in vacuum at a temperature of 160° C. The coefficient of friction was lower than that in identical tests at ambient temperature but higher than in atmospheric tests at ambient temperature. This would be expected because as the sample is heated, gas molecules are driven out, resulting in a higher net pressure at the surface of the sample than would be measured at some other location inside the chamber. Also, the measured value for the coefficient of friction is a function of the surface roughness of the metallic sample. In some tests the same metallic sample was used in pre-vacuum, vacuum, and post-vacuum. In all cases, the post-vacuum results were higher than pre-vacuum, with the vacuum results higher than either. The values obtained in this phase of the program are given in Table I [10].

In an effort to evaluate the results listed in Table I, and to determine the single effect of vacuum on the coefficient of friction, a series of tests was performed using aluminum 7075 on quartz 250 to 500 microns in diameter.

TABLE I. COEFFICIENT OF FRICTION FOR METALLIC AND NONMETALLIC MATERIALS

	Quartz			Basalt		
	37 to 44μ	250 to 500μ	Mixture	37 to 44μ	250 to 500μ	Mixture
Steel AISI-1020						
ATM	0. 21	0. 09	-	0. 28	0. 15	-
VAC	0. 29	0. 16	-	0. 62	0. 29	-
Aluminum 7075						
ATM	0. 23	0. 18	0. 25	0. 31	0. 21	0. 22
VAC	0. 34	0. 58	0. 39	0. 41	0. 24	0. 26

In this phase, the coefficient of friction obtained in atmosphere was 0. 15 compared to 0. 18 in the previous phase. The value obtained in a vacuum of 10^{-2} N/m^3 (10^{-7} torr) was 0. 30 compared to 0. 58 in the previous phase. This value of 0. 30 is little more than half the previous value because the metal sample was used in atmospheric tests prior to the vacuum tests. However, the values of Table I can be taken as upper limits for the vacuum tests.

In tests using metallic hemispheres in contact with basalt and quartzite solids. the coefficient of friction increases from atmosphere to a moderate vacuum of 10^{-4} to 10^{-6} N/m^2 (10^{-6} to 10^{-8} torr) [11]. The coefficient of friction on quartz increased from 0. 19 to 0. 29 for steel AISI-1020, and from 0. 16 to 0. 25 for aluminum 7075. Likewise, for basalt, the coefficient of friction increased from 0. 14 to 0. 28 for steel AISI-120 and from 0. 21 to 0. 28 for aluminum.

Experimental results have shown that the increase in moderate vacuum of the coefficient of friction between engineering materials and nonmetallic materials is not severe. Unfortunately, these results cannot be extrapolated with any confidence to a lunar vacuum. However, one can be certain that as the vacuum increases, the coefficient of friction will increase.

Adhesion. - There is general agreement that a layer of dust is present on the lunar surface. However, the thickness of this layer has not been established; estimates range from kilometers [12] to a few millimeters [13]. Likewise. there are two primary hypotheses of the origin of the lunar craters—impact and volcanic. Either origin would result in the deposit of some fine granular material on the lunar surface and, in fact. both mechanisms have probably been active on the moon.

The very high quality photographs of the moon taken by Rangers VII, VIII, and IX and more recently by Surveyor I (Fig. 3) leave little doubt that at least least some areas of the moon are covered by granular material.

The presence of a fine layer of dust on the surface of the moon could be an obstacle to mining operations, affecting man, vehicles and machinery. The degree to which it will be an obstacle depends primarily upon its mechanical properties. Recent experiments indicate that the mechanical properties of a granular material in a lunar vacuum will be dominated by cold welding of the material.

Single crystal minerals have been contacted in a vacuum of 10^{-8} N/m^2 (10^{-10} torr) [14]. Some of the tests were made with crystals cleaved in air, while others were made with crystals cleaved in vacuum. The crystals cleaved in air exhibited adhesive forces up to about 0. 4 gram, while the crystals cleaved in vacuum have had forces of a magnitude larger. Fine powdered rock including chondrite, tektite, obsidian, basalt, andesite, undite, and pyroxenite was sieved in a vacuum and adhered to the sample holder and wires leading to the apparatus [15]. Samples of fine powdered basalt and slightly larger aluminum powder were shown to adhere at a vacuum of 10^{-8} N/m^2 (10^{-10} torr) [16].

Experiments have been conducted using basalt, pumice, crushed glass and glass spheres to determine the adhesive forces in an oil-free vacuum as high as 10^{-9} N/m^2 (10^{-11} torr). The test apparatus consisted of a 304 stainless steel drum, a variable speed motor, and a magnetic rotary seal. The drum was 25 cm long and 15 cm in diameter. The drum contained 4 slats 1. 3 cm high extending the length of the drum. One end

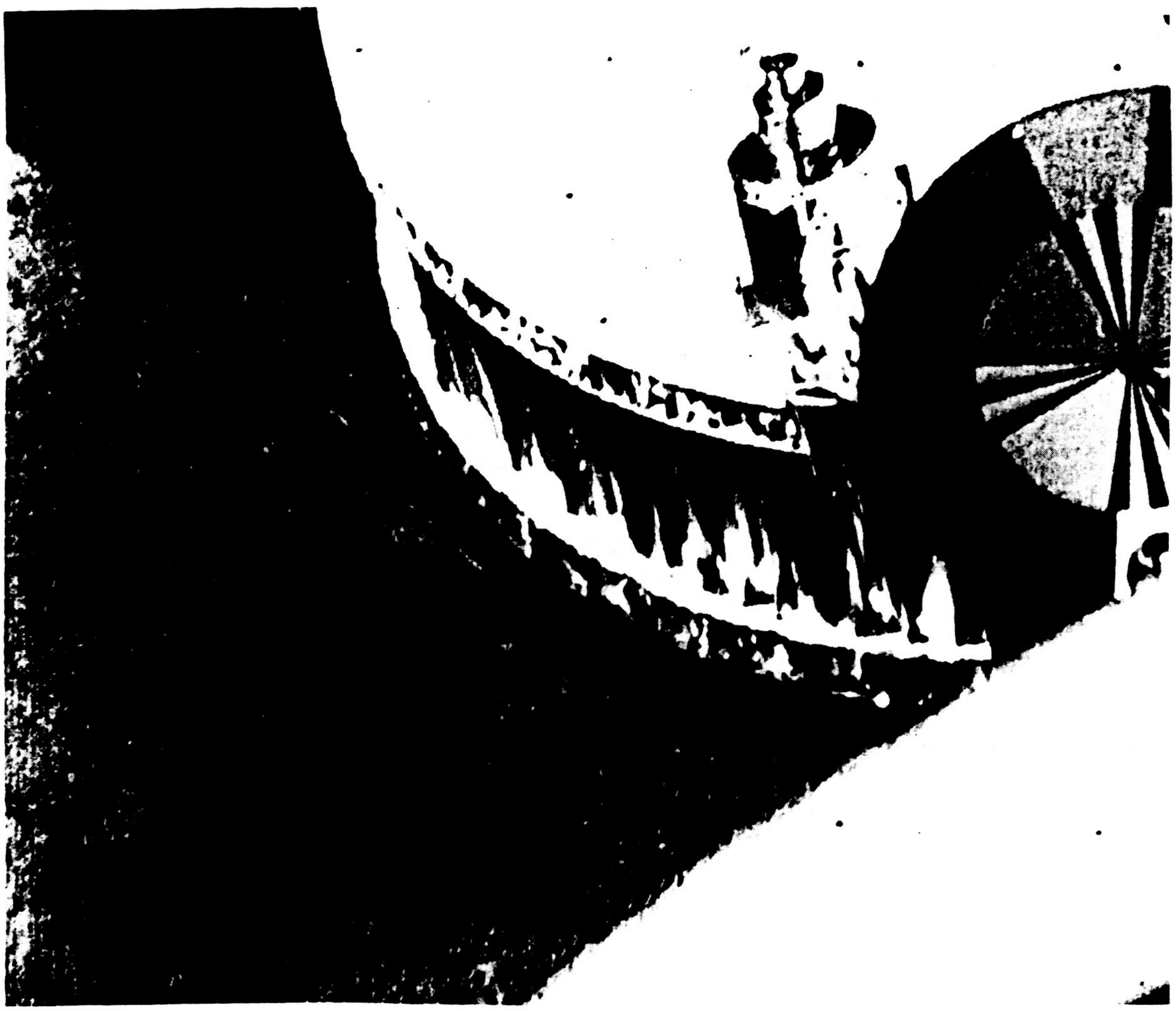

FIGURE 3. SURVEYOR I SPACECRAFT FOOTPAD

of the drum is closed by a plate containing 4 ports of fine mesh wire to allow gas to escape. The other end is closed by a glass window to allow viewing of the sample while the experiment is in progress.

The results will be discussed briefly here and are given in detail by Fields [17]. Basalt (180 g) in size ranges 10 to 20, 20 to 37, and 37 to 62 microns in diameter was tumbled at 1 rpm until 100 percent adhesion occurred. With each sample, adhesion began immediately with rotation of the drum and 100 percent adhesion occurred in 15 minutes for the two smaller sizes, and in a slightly longer time for the larger size. Figure 4 shows the drum with basalt immediately after bringing the sample to atmospheric pressure with dry nitrogen.

The initial pumice tests indicated very weak adhesive forces with about 20 percent of 180 grams adhering after rotating 24 hours. However, later tests have indicated that this is a gas emission problem. If the pumice is exposed to vacuum of approximately two weeks instead of three days, the adhesive forces are considerably stronger.

Samples of identical composition but different shape characteristics were studied to determine the effect of surface configuration on the adhesive forces. Crushed glass and glass spheres were each tested in the size range 20 to 37 microns. Approximately 99 percent of the crushed glass adhered after 30 minutes with no further increase after 24 hours. Approximately 30 percent of the glass spheres adhered after 24 hours.

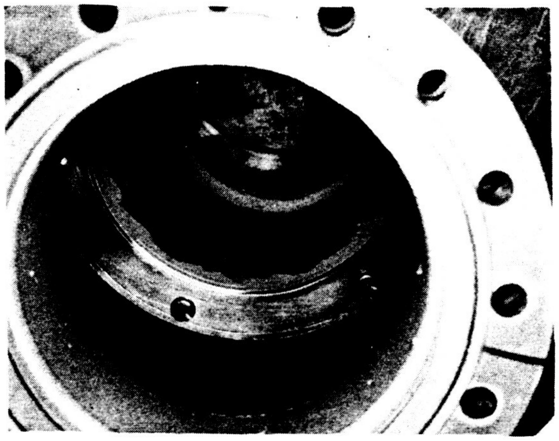

FIGURE 4. BASALT ADHERED TO STAINLESS STEEL DRUM IMMEDIATELY AFTER RELEASE TO ATMOSPHERIC PRESSURE

MINERAL DEPOSITS

Just as fundamental as lunar surface environmental problems are those of the kind, grade, extent, structure, and location of the deposits that may contain water or other useful minerals. At this point, we have no direct evidence of the presence of mineral deposits at all. The presence of water or any other mineral deposit must be inferred at the present time and inferences must, in turn, depend upon assumptions regarding lunar origin, thermal history, chemical composition, and extent of near-surface chemical differentiation. No attempt will be made to explore here the many possibilities that have been suggested and reported.

At the present time, when considering mining problems and techniques, it will be necessary to cover the entire range of possible chemical differentiation of the lunar crust. In a previous report [6] twelve deposit models were examined. They ranged from strictly impact origin to both intrusive and extrusive volcanic models. Ten of them were taken from the work of Salisbury [18, 19] and of Westhusing and Crowe [20]. Nine of the twelve models are associated with the lunar maria. Of the nine, four imply that the maria consist of bedded tuff layers, three that they are filled with successive lava flows, and two could be interpreted as a filling from one thick lava outpouring. One lunar upland model consists of a single rubble blanket over granodiorite; a second, of interbedded, lenticular rubble layers over granodiorite; and a third, a rubble layer alone with imbedded serpentine boulders.

If lunar mineral deposits occur in the form of local mineral enrichments, as do Earth deposits, a program for their discovery, evaluation and exploitation will be necessary. If they can be associated early with surface features (maria edges, crater bottoms, crater rims, lunar domes) or with visible or hidden structural features (fracture systems, fluid vents, etc.) or if they are found to occur in groups or geographic districts, as on Earth, the problem of discovery and evaluation will be greatly simplified. Unless concentration does occur to some extent, drilling or other exploration to evaluate the deposit will be useless and mining will become merely the random removal of country rock.

Water occurring in free form should be more economical to mine and process than chemically combined water. If free water occupies pores, shear zones or small fractures in hard or rough rock, mining it will be almost as difficult as for hydrous minerals in similar rock. Permafrost zones in tuff and similar free water deposits would be much less difficult to mine.

In free water deposits, ice may be accompanied by other frozen volatiles. Some of these may introduce problems of corrosion of mining equipment. For example, if titanium alloys are used in the construction of lightweight equipment, and chlorine were present with the ice, severe damage to the titanium parts could occur. Although chlorine attacks titanium very slowly when wet, dry chlorine gas (less than 0.5 percent water) "may react at room temperature," according to one source [21], while another reference [22] states "rapid attack, ignited and burned" (less than 0.1 percent water at 30°C). In any mixture of chlorine and water, the evaporating chlorine would likely be dry because the pressure of water at that temperature is only a little more than 0.1 mm of mercury.

Some other corrosive substances are bromine, ammonia and steam, calcium chloride, chlorine triflouride, fluoroboric acid, fluorine fluorosilic acid, hydrochloric acid (concentrated), hydrofluoric acid, mercury (371°C), phosphoric acid (boiling, concentrated), potassium hydroxide, sodium bifluoride and sulfuric acid (less than room temperature) [21]. Most of these substances attack titanium only at moderate rates. In general, titanium and its alloys may be attacked if the lunar chemistry proves to be strongly reducing. "Titanium provides excellent resistance to general and localized attack under most oxidizing, neutral and inhibited reducing conditions. It also remains passive under mildly reducing condition, although it may be attacked by strongly reducing or complexing media" [21].

Other surface mining problems may be related to the lunar day and mineral deposit location. It will be assumed that all early lunar landings and activities, except possibly lunar exploration, will be limited to the landing belt, 10 degrees on either side of the lunar equator. Here, a surface mine for approximately half the lunar day would be subjected to the direct rays of the sun with results noted above, if the ore is free water. Ores consisting of hydrous minerals should not be affected.

For surface mining operations, however, there would be certain advantages to night operation, independent of ore composition. Some of the advantages of night-time operations will be: uniformly low temperature conditions; uniform lighting from the Earth; absence of solar wind; possible absence or reduction of rarified ionized gases of the thin lunar atmosphere; possible absence or reduction of a postulated surface electric charge; no cooling needed for space suits and/or machine cabs or structures; and better heat dissipation from bearings, motors, etc.

Possible disadvantages of night operations are: the necessity to provide artificial lighting, at least part of the time; the high heating load for space suits, cabs, and structures (objectionable only if primary power is solar and battery storage is required for night operations); and the increase in freezing of working fluids, if any are used.

Equipment Problems

Many problems will be encountered in the selection and design of lunar mining equipment. The first automobile resembled a buggy, so it is quite likely that the first lunar mining equipment will be Earth models, modified for the lunar environment and mining conditions. This is one assumption on which the present work on mining systems is based. After some lunar mining experience, undoubtedly original and different equipment designs will be developed. If lunar mining is preceded by a considerable period of lunar exploration, there will be opportunity for new designs to be applied to transportation vehicles and methods and to drilling techniques. Any underground shelter construction activities, prior to mining, would also give some experience in problems of fragmentation, digging, loading and limited-scale excavation. These activities may, however, resemble underground mining more than surface methods.

Problems considered to date may be classified as power supply; mode of traction for moving equipment; equipment control methods; materials for equipment construction; and secondary or service materials, such as for ballast or counterweights, maintenance facilities, roads, bins, crusher foundations, control systems, and power distribution systems.

Power Supply. - Although no fundamental studies of types of equipment power have been made, it is assumed that alternating current electrical power will be used. Most small surface mining equipment on Earth has gasoline or diesel engines, while larger shovels and draglines depend upon diesel-electric or electric power. No great trouble is anticipated in substituting electric motors for diesel engines in small equipment. Substitution of electric motors for diesel or gasoline engines should result in a new weight reduction.

Electrical energy may be generated centrally and distributed by lines to stationary installations or through trailing cables to moving equipment. If fuel cells or other compact generation equipment is used, cells and motors may be individual for each piece of equipment. Detailed consideration of these problems is beyond the scope of this paper.

Mode of Traction for Moving Equipment. - Present small equipment moves on treads or on wheels, usually rubber-tired. Larger shovels and draglines are crawlers or walkers. Mitchum [23] concluded that tracked vehicles are less efficient than wheeled ones. They are heavier and less reliable and their only superiority, which is slight, is in soft, dry granular material. It is assumed that tracked vehicles will be used for mining equipment, because (1) Surveyor I photographs indicate the probable presence of such material; (2) wheels must be rather large to cross cracks easily negotiated by equal-sized crawlers vehicles; and (3) the practicability of lunar use of pneumatic tires is questionable.

Equipment Control Methods. - Mining equipment may conceivably be controlled by: (1) an operator in a space suit, much as in operations on Earth, a method probably practical only for lunar night work; (2) an operator in an enclosed, Earth-environment cab mounted on each piece of equipment, whether bulldozer, loader, shovel, or dragline; (3) remote, automatic controls from an Earth-environment shelter especially erected at the mine or from the crew living shelter which may or may not be located at the mine; or (4) remote, automatic control from the Earth. The last two control methods probably will require continuous television viewing.

At least one mining system using control at the mine and a number of others using either of the first two methods will be considered. The necessity to study the first two in detail has been eliminated by limiting our study to lunar night work. Neglected also are real "remote" control systems.

It is readily acknowledged that systems (1) and (2) will use more lunar-based labor than will the others and that this large cost item may constitute its greatest handicap.

Materials for Equipment Construction. - Excavating equipment is constructed almost entirely of steel and cast iron. Since transportation of equipment from Earth to moon probably will be the greatest single charge against the utilization of indigeneous lunar resources [24], some way must be found to reduce weight without impairing the usefulness of the equipment.

Large draglines, and perhaps some shovels, make some use of aluminum, especially in booms. One manufacturer writes of a particular model: "We might also add that the 120-foot boom consists of 45-foot steel butt, 38-foot aluminum insert and 37-foot aluminum top."* Thus, 75 feet of the boom length were made of aluminum. It weighed 8780 pounds, which is probably far less than equivalent steel parts would weigh. Cabs and other parts normally subjected to comparatively low stresses may also utilize aluminum.

Parts of machines carrying maximum stresses often cannot be made of aluminum or magnesium. Many of these, however, may be replaced by titanium or titanium alloys. Beryllium alloys are also a possibility. One manufacturer [25] reports yield strengths of 9.99×10^8 N/cm^2 (145 000 lb/in^2) for Ti-6Al-4V alloy sheet and plate, and 10.07×10^8 to 11.31×10^8 N/m^2 (146 000 to 164 000 lb/in^2) for extruded shapes. This compares with about 6.89×10^8 N/m^2 (100 000 lb/in^2) for heat-treated constructional steels [26]. Titanium may not be so desirable with regard to some other properties, but it is said to have superior strength-to-weight ratios, excellent elevated-temperature performance, good corrosion resistance and unusual erosion resistance [27]. Nothing was found in manufacturers' data on properties at low temperatures like those of the lunar night, but the mean thermal coefficient of expansion (32 to 212 F) is 4.9×10^{-6} °F (8.82×10^{-6} °K) for Ti-6Al-4V alloy [21] as compared to about 7×10^{-6} °F (12.60×10^{-6} °K) for 0.2 C steel; thus, titanium should not be as strongly stressed by rapid temperature changes as is steel.

One manufacturer** lists weights of five bulldozers, three crawler loaders, and one wheeled loader with the weights of steel used in their construction. The bulldozers averaged 88 percent steel, the crawler loaders, 89.4 percent, and the wheeled loader, 91.3 percent. All steel can hardly be replaced by titanium or aluminum. Titanium, for example, probably does not have the wear resistance of the steel used for bulldozer blade cutting edges, although one manufacturer says"...titanium has been successfully flame-plated with tungsten by the Linde process,"*** a treatment that should make it much more abrasion-resistant. The weights of Earth-designed equipment will be estimated if Ti-6Al-4V alloy is substituted for much of the steel and if electric motors are substituted for diesels. The latter substitution may not bring any real weight reduction if heat dissipation radiators have to be added to electric motors. These probably would be made largely of aluminum, and it is doubtful if the heat dissipation systems for electric motors would weigh as much as the brass cooling radiators of gasoline and diesel motors. Also, if fuel cells or batteries must be added to the electric motors, there may even be some increase in weight over diesel engines. Trailing cables will add weight for this method of electric power distribution.

It must be remembered that lifting, but not necessarily scraping or loading, of soil or broken rock on the lunar surface should require only one-sixth the power required for the same volume lifted on Earth. Thus, the power required for equivalent volumetric performance of equipment may well be much less on the moon than on Earth. If lunar materials are also of lower density than Earth materials, even less power will be required.

<u>Secondary or Service Material</u>. - The final category of secondary or service materials is not discussed in this paper except as follows: There is every reason to believe that there should be on or near the lunar surface a considerable accumulation of meteoritic nickel-iron. Whether it occurs in discrete, large, or small masses, or as condensed vapor from the heat of impact, dispersed throughout the lunar soil, is not known. If it can be collected, it should be ideal for ballast or counterweight in construction equipment needing such material. A given volume of it, of equivalent density, should be six times as effective for this use as the same volume would be on Earth. Some large equipment manufacturers do not ship ballast, but at the time of equipment erection it is provided locally from scrap iron or concrete. If counterweights must be monolithic, small meteorite iron masses in concrete would be ideal. Green [28] has suggested case basalt as a possible concrete substitute.

The second comment is prompted by the necessity for an ore chute or bin as the minimum auxiliary facility required at a mine. If mining and haulage can be combined into one operation, such as with a scraper-loader, the bin or chute could be placed at the processing plant and ore hauled there directly. If ore is loaded into vehicles (trucks) by a front-end loader, shovel, or dragline, or if it is merely stockpiled at the mine and loaded and shipped later, no chute or bin may be needed. As will be seen later, however, any

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scraper mining system or any automatic control system will require some kind of chutes or bins. If the ore is to be crushed at the mine, at least a chute and a foundation will be required. As will be shown in illustrations following, it is proposed that metal posts be used for such structures. In Earth soil, these could be placed by a piledriver; this would be possible also in the lunar soil shown in Surveyor I photographs provided it is deep enough and there are no large boulders, meteorites, iron masses, etc., to prevent drilling. If these conditions or solid rock exist, posts could still be placed in drilled holes.

TECHNIQUES

Mining Systems

After learning something specific about the occurrence of lunar minerals, and certainly after some mining experience, surface mining systems may be developed which are entirely unlike any now used on Earth. Until that time, however, little can be done except to apply variations on Earth methods which seem suitable to current concepts regarding lunar mineral deposits.

High unit costs and limited water demand for fuel or other purposes will require that early lunar water mining be simple, on a small scale, and at a cost equal to or lower than that of transporting water from Earth. If accessible supplies of ice can be found in caverns, lava tunnels, etc., near the lunar equator, no water will be mined, in the usual sense, but it will merely be "harvested." If, however, water is found only under the lunar surface, at depths too great to be affected by the lunar diurnal temperature cycle, mining logically can be contemplated. As the demand for water, oxygen, and hydrogen for lunar colonies and for fuels increases, it is reasonable to expect the size and sophistication of mining operations also to increase.

Virtually nothing has been published on "conventional" mining methods applicable to lunar conditions. Heyward [29] has discussed the general problem, pointing out the separate functions involved in mining and processing indigenous lunar resources: rock-breaking, loading, size reduction, transportation, processing and purification, conversion, and storage. Some work has appeared on indirect methods of mining, similar to the Frasch process and to solution mining [30, 31], tunneling methods [32] and a general review of published work on lunar mining [33].

The work reported here deals with the more conventional surface, strip, or open cast mining systems. Indirect systems and mining by drilling are also conducted from the surface but involve the removal of very limited quantities of overburden.

Systems for surface mining of water may be classified as follows:

1. Scraper methods
    a. Stationary engine-operated scraper
    b. Scraper drawn by a tractor or other vehicle
    c. Scraping ore and overburden by bulldozer
    d. Removing overburden and ore by scraper-loader
2. Methods involving scraping of all or part of the overdurden and digging of ore or ore and part of overburden
    a. Overburden removed wholly or partly by scraper, bulldozer, or scraper-loader, and ore dug and loaded by front-end loader or hoe
    b. Overburden wholly or partly removed by scraping and ore or ore and harder overburden dug by power shovel
    c. Same methods as above except for ore removal by dragline
    d. Overburden and ore both dug and removed by shovel or dragline
3. Methods involving fragmentation of ore and of part or all of overburden prior to digging and removal
    a. Overburden removed wholly or partly by scraping and ore and part or all of overburden fragmented for removal by shovel or dragline
    b. All ore and all overburden fragmented before removal by shovel or dragline
    c. All ore and overburder fragmented (digging and loading may be done with front-end loader or hoe.)

It will be noted that the methods have been roughly arranged in order of increasing hardness or toughness of overburden and ore. Surveyor I photographs proved that at least part of the lunar suface is covered, to an undetermined depth, by material which probably can be scraped and removed. Under the impact hypothesis, there must be a considerable thickness of rubble both on the maria and terrae that may be scrapable if it contains no or few large boulders or has not become firmly cemented in some way. If the maria are younger than the terrae the rubble blanket there probably is thinner. If the maria are volcanic tuff flows or falls, the material may be scrapable to considerable depth unless extensively intruded by lava.

The classification of mining methods is also arranged according to increasing size of mining equipment. Small shovels and draglines will weigh less than bulldozers or loaders, and cable-drawn scrapers are even lighter.

Of the methods listed, only 1a can be made mechanically automatic or operated by one man from a single on-site location. The other methods will require an operator in a space suit or in a pressurized cab on each machine or electronic controls installed in each machine and operated from a central control tower or booth. High resolution television viewing may be required in remote control systems if overburden and ore are difficult to distinguish.

In the classification, it is assumed that hardness or toughness of overburden and ore may increase with depth and that ore will be harder than at least a part of the overburden. This assumption appears reasonable for hydrous mineral ores, and probably for lean free water ores.

Definition of Model Water Deposit

Six hypothetical ore deposits have been defined for use in comparing suitability and first order relative costs of various mining systems. Only one basic model will be used in this paper. It will be used as two deposits with differing depths of overburden.

Deposit. -

a. The ore is a uniform permafrost zone, 3 m (9.84 ft) thick in a friable volcanic tuff (Fig. 5).

b. The zone lies at a uniform depth of 10 m (32.8 ft) below a generally level mare surface (similar to Surveyor I terrain). Overburden is a dry "semiwelded" tuff with an average bulk density of 1.25 g/cm^3 (78 lb/ft^3).

c. The deposit location is within 10 degrees of the lunar equator.

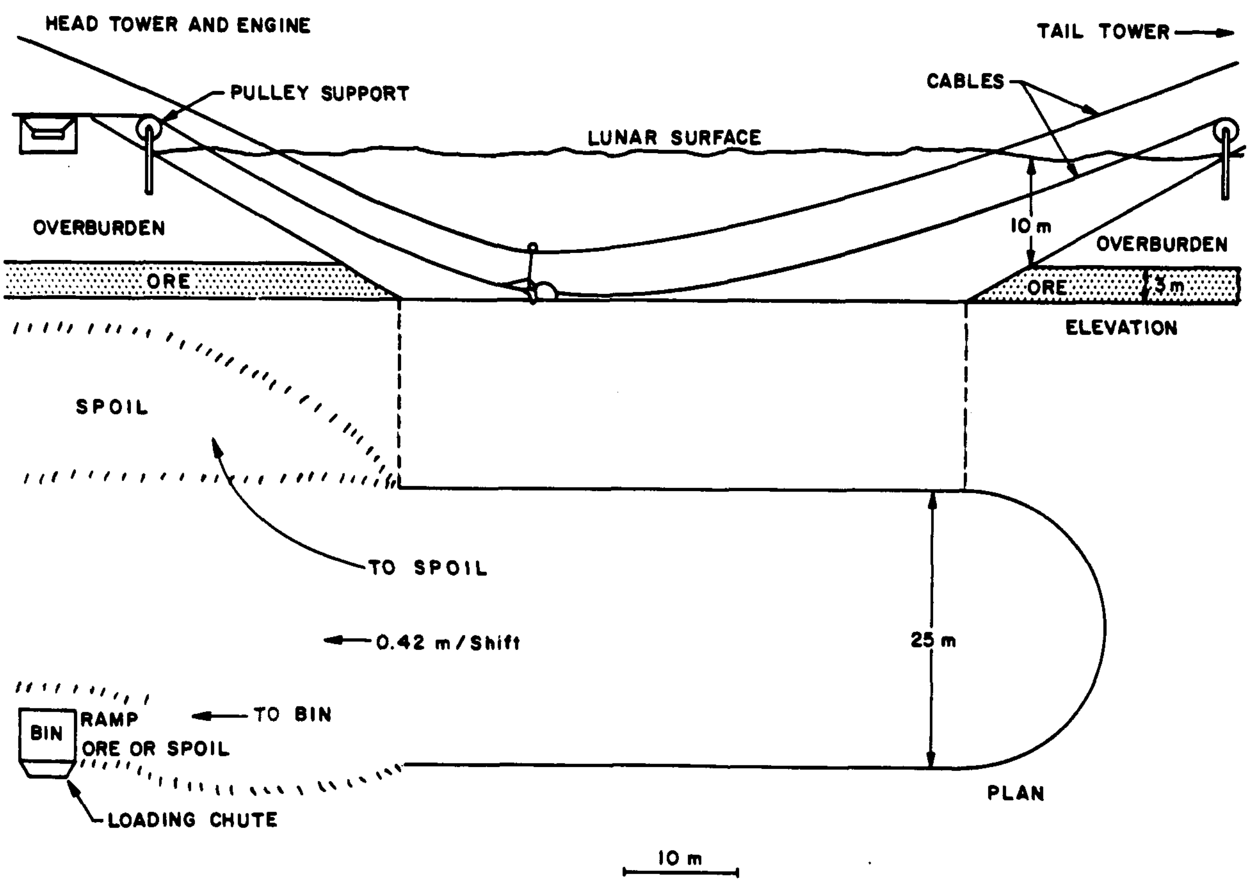

FIGURE 5. SCHEMATIC DIAGRAM OF A PROPOSED ROPE-AND-SCRAPER MINING SYSTEM

d. The ore contains 2 percent free water and has a bulk density, in place, of 1.20 g/cm^2 (75 lb/ft^3) [34].

Mining Schedule. - One schedule suggests a lunar oxygen demand of 4546 kilograms per month (10 000 pounds per month) or 54 552 kg/yr (120 000 lb/yr) by 1976 and double this quantity in 1982 [24]. The 1982 quantity will be used for estimates. It is equivalent to 122 728 kg (270 000 lb) of water per year. On this basis, ore averaging 2 percent water, by weight, must be mined and processed at the rate of 6 136 367 kg (13 500 000 lb) per year, if extraction is 100 percent efficient. It shall be arbitrarily assumed that 6 818 200 kg (15 000 000 lb or 7500 tons) is the basic annual quantity to be mined from either of the above deposits.

a. Mining will be assumed to be limited to one 8-hour shift per 24 hours and to lunar night only. The lunar night is equal to 13.66 Earth days and there are 13.37 lunar nights per Earth years. This gives 182.63 shifts per year. For simplification of calculations, 180 shifts per year will be used. If mining is also done during the lunar day, mining may be expanded to 360 shifts.

b. For 180 shifts, the mining rate will be 37 879 kg (83 333 lb) ore per shift.

c. In the assumed deposit, the ore volume that must be mined is 31.566 m^3 per shift. The thickness (and volume) of overburden to be moved is 3 1/3 times that of ore so that the volume of ore and overburden to be moved per shift totals 136.8 m^3.

d. Alternate calculations will be made for the same ore thickness but for an overburden depth of 30 m or 10 times that of the ore thickness. This deposit will require removal of 347.2 m^3 of material per 8-hour shift.

e. If a pit width of 25m (82 ft) is arbitrarily assumed, the distance that the pit must be advanced each shift to remove 31.566 m^3 of ore, 3 m high and 25 m wide, will be only 0.42 m. In a year, this will be only 75.6 m or three times the pit width. With a low rate of pit lengthening, haulage distances will increase only slowly. In a relatively narrow pit, stationary scraper systems will require infrequent shifting of towers or pulleys.

Mining Capacities of Some Equipment

Figure 5 shows a rope-and-scraper system for mining the ore zone at either 10- or 30-m depth. This system is similar to larger scale tower excavators on the Earth. Figure 6 is taken from a publication advertisting this kind of system [35]. Such a system should be operable by one man from a central booth.

The capacity of a scraper system will be determined by its rate of transport rather than by its rate of digging. The transport rate is a function of scraper or bucket capacity and of the cycle frequency. Cycle frequency may vary slightly for overburden and ore. Figure 5 shows overburden deposited on a spoil pile with the ore dragged to a chute. The ore-to-chute distance is almost constant and that for waste disposal gradually increases.

Figure 5 indicates that at the stage of mining shown (another bin must be installed to the left of the one shown and the ramp extended to it), the distance of ore transport is about 30 m. If the other bin is installed 50 m farther left, the distance of drag will be about 80 m. If one assumes an average drag distance for ore of 50 m, and for spoil a longer one of 70 m, the average is about 60 m. The lowest speed listed for a small tractor was 1.2 miles per hour or 1.932 kilometers per hour. Adopting this speed for dragging, recognizing that the return drag to the pit for both ore and spoil is unproductive, and assuming that dumping time will require half the time required for a one-way trip, the equivalent drag distance per productive load is 150 m. This gives 12.88 loads per hour or 103 loads per shift.

One scraper manufacturer [36] advertises scrapers from 66 to 213 cm (26 to 84 in) wide. Calculations suggested that these scrapers are too small to mine the deposit, so dragline buckets were investigated. It was found that a 1.34 m^3 (1 3/4 yd^3) dragline bucket can be used to mine the deposit and at the dragging speed to give 103 trips, 138.0 m^3 of material per shift can be moved and the smaller deposit can be mined at the required rate. A 3.37 m^3 (4.41 yd^3) bucket would be needed to mine the deeper overburden deposit at the same dragging speed.

The calculated costs for this system, using the two dragline buckets, are not shown here but will be included in the summary table. The weight of a rope and scraper system is in the head tower, tail tower, rope and dragline bucket. No light metal alloys were substituted in weight calculations for this system.

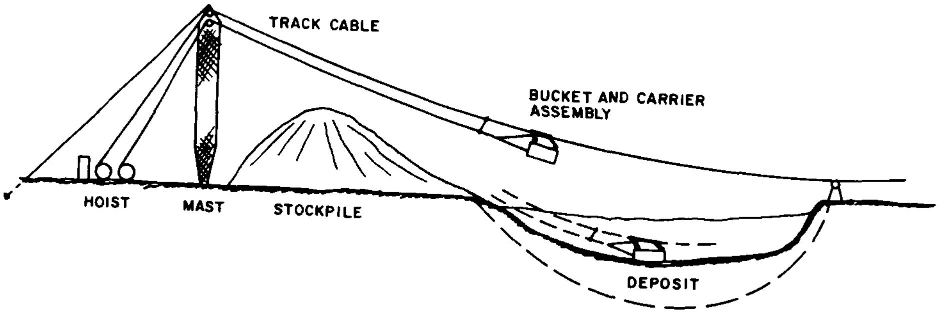

FIGURE 6. SCHEMATIC DIAGRAM, SLACKLINE CABLEWAY EXCAVATOR [35]

An alternative and much more flexible scraping system would be a scraper drawn by a tractor. This method probably will require an operator in a space suit or in a tractor cab. It should also prove the ideal method, if it becomes necessary to mine a large number of small, scattered deposits rather than a large one.

The pit layout could be similar to that shown in Figure 5, but with much more latitude as to location of spoil piles and loading chutes.

The limiting factor in material transporation capacity for this mining method probably will be very close to that of the previous one. It is possible that the average length of haul could be shortened and the haulage might be increased by using a larger tractor than one with a 1.93 km/hr (1.2 mph) speed.

A very similar mining method is one in which ore and overburden are scraped by a bulldozer to loading chute or spoil pile.

A very wide variety of bulldozers is available. The limiting capacity factor again will not be digging rate, but transporting rate. Using the assumption of material piled against the blade at a 45-degree angle, Table II shows the capacities of the smaller dozers on which information was collected. It also shows the capacities of each at 1.93 km/hr (1.2 mi/hr) and at the fastest low gear speed of any of the bulldozers 2.73 km (1.7 mi/hr). The minimum number of bulldozers required to handle both 10 and 30 m of overburden, plus 3 m of ore, at the low speed, are also shown.

TABLE II. WEIGHTS, DIGGING, AND SCRAPING CAPACITIES OF SMALL BULLDOZERS (CRAWLER TYPE)

Machine No.	Power	Blade Length	Blade Height	Transport Capacity		Capacity per Shift (1)		Capacity per Shift (2)		No. Dozers to Handle		Weight (5)	
		(in.)	(in.)	m^3	yd^3	m^3	yd^3	m^3	yd^3	10 m	30 m	kg	lb
1	(3)	80	24	0.378	0.494	38.9	50.9	55.1	72.1	4	9	2828	6228
2	(3)	80	24	0.378	0.494	38.9	50.9	55.1	72.1	4	9	3301	7272
3	(3)	83.9	28	0.539	0.705	55.5	72.6	78.7	102.9	3	7	3953	8707
4	(4)	110	28	0.706	0.924	72.8	95.2	103.1	134.9	2	5	4998	11 009
5	(4)	121-5/8	27-3/4	0.740	0.968	76.2	99.7	108.1	141.3	2	5	5148	11 340
6	(4)	96	33.2	0.867	1.134	89.3	116.8	126.7	165.6	2	4	5147	11 336
7	(4)	124-3/4	34	1.182	1.546	121.8	159.2	172.6	225.7	2	3	7054	15 538

(1) 103 trips per 8 hours at 1.2 mph

(2) 146 trips per 8 hours at 1.7 mph

(3) Gasoline or diesel

(4) Diesel

(5) With substitution of Ti-6A1-4V alloy and A1 for part of steel

Considerably larger bulldozers are made than those listed in Table II, but they are also much heavier. A mining installation using this system will need a minimum of two machines in order to keep production going in the event maintenance is needed on the principal mining machine.

A pit layout similar to that of Figure 5 may be preserved if overburden and ore are dug, transported, and dumped by front-end loaders. Table III gives data for 16 small front-end loaders. The bucket capacity (level rather than heaped, where both are given) is the average of the range given by the manufacturer. It is evident that for equivalent capacity, rubber-tired models are considerably higher than crawlers. For consistency, however, only the crawler models will be used for most comparisons. If wheeled vehicles prove satisfactory for haulage on the lunar surface, there exist additional possibilities for weight reductions. Even if rubber tires prove impractical, large, light metal wheels should prove sufficiently strong and much lighter than crawlers.

Table III illustrates that the small loaders move materials faster than bulldozers -- in some cases, with the same tractors. No one crawler-loader, however was sufficiently large to handle the 10-m overburden deposit.

Both the deposits can also be mined by shovels and draglines. Because the "reach" of each of the machines is limited and their mobility restricted,

TABLE III. WEIGHTS, DIGGING, AND TRANSPORTATION CAPACITIES OF SMALL FRONT-END LOADERS

Machine No.	Power	Bucket Capacity (7)		Capacity per Shift (1)		Capacity per Shift (2)		No Loaders to Handle		Weight (6)		Traction
		m^3	yd^3	m^3	yd^3	m^3	yd^3	10 m	30 m	kg	lb	
1	(3)	0.99	1.30	102.4	133.9	145.1	189.8	2	4	1683	3708	R
2	(3)	1.11	1.45	114.2	149.4	161.9	211.7	2	3	2128	4687	R
3	(4)	1.06	1.38	108.6	142.1	154.1	201.5	2	4	3154	6948	R
4	(3)	0.62	0.81	63.8	83.4	90.5	118.3	3	6	3432	7560	C
5	(3)	1.25	1.63	128.4	167.9	182.0	238.0	2	3	3465	7632	R
6	(3)	1.25	1.63	128.4	167.9	182.0	238.0	2	3	3661	8064	R
7	(3)	0.76	1.00	78.8	103.0	111.6	146.0	2	5	4086	9000	C
8	(5)	0.86	1.13	89.0	116.4	126.2	165.0	2	4	4846	10 674	C
9	(5)	1.53	2.00	157.5	206.0	223.3	292.0	1	3	5459	12 024	R
10	(5)	1.72	2.25	177.2	231.8	251.2	328.5	1	2	5532	12 186	R
11	(3)	1.77	2.31	181.9	237.9	257.9	337.3	1	2	5680	12 510	R
12	(5)	1.00	1.31	103.1	134.9	146.3	191.3	2	4	5720	12 600	C
13	(3)	1.77	2.31	181.9	237.9	257.9	337.3	1	2	6194	13 644	R
14	(5)	1.53	2.00	157.5	206.0	223.3	892.0	1	2	6397	14 090	R
15	(5)	1.15	1.50	118.1	154.5	167.4	219.0	2	3	6789	14 954	C
16	(5)	1.15	1.50	118.1	154.5	167.4	219.0	2	3	7028	15 480	C

(1) 103 trips/ 8 hr at 1.2 mph
(2) 146 trips/8 hr at 1.7 mph
(3) Gasoline or diesel
(4) Gasoline only
(5) Diesel only
(6) With substitution of Ti-6Al-4V alloy and Al for part of steel
(7) Average of range given by manufacturer
(R) Rubber tired
(C) Crawler

some kind of ore haulage will be needed. To place the ore mined by shovels in bins located as they were in the previous examples would, however, require only very short hauls. If it is assumed that ore, regardless of the system of mining, must be hauled one kilometer to the preparation plant, bins are unnecessary for any of the nonscraping methods, including shovels and draglines, because the ore may simply be piled on the surface for later loading and hauling. In this case, systems using front-end loaders, shovels, and draglines will not need to be charged with any greater haulage costs than will scraper methods, unless elaborate preparation of the area for ore piling is needed.

Draglines have greater capability for mining below machine level than do shovels although the latter can make "box" or first cuts, after which all mining can be at machine level and above in a deposit like that shown in Figure 5. The shovel will, however, generally work in the pit bottom while the dragline can work from the surface. With 10-m thick overburden, all the machines shown in Table IV should be able to mine all material in one bench. For 30-m overburden depth, removal in two benches will be necessary.

Loads that can be picked up by draglines decrease as boom length becomes greater. On the moon, both

TABLE IV. CAPACITIES, WEIGHTS, AND OTHER DATA ON FIVE SMALL SHOVELS AND NINE DRAGLINES

Machine No.	Power	Bucket[1] Capacity		Max Height[2]		Oper Radius[3]		Capacity[4] per Shift		Weight[5]	
		m^3	yd^3	m	ft	m	ft	m^3	yd^3	kg	lb
SHOVELS											
1	(7)	1.44	1.88	9.0	29.5	9.0	29.5	552	902	31 160	68 550
2	(7)	2.29	3.00	11.8	38.8	12.1	39.8	881	1440	46 770	102 900
3	(7)	2.68	3.5	26.8	88.0	11.5	37.6	1028	1680	57 360	126 200
4	(7)	1.91	2.5	7.6	24.8	9.5	31.0	734	1200	49 600	109 400
5	(8)	1.91	2.5	9.1	30.0	11.3	32.0	734	1200	57 000	125 600
DRAGLINES											
1	(7)	1.44	1.88	29.0	95.0	13.7	45.0	732	902	34 680	76 300
2	(7)	2.48	3.25	35.4	116.0	16.8	55.0	954	1560	42 100	92 200
3	(7)	2.68	3.5	35.7	117.0	16.8	55.0	1028	1680	48 260	106 400
4	(7)	1.72	2.25	-	-	-	-	864	1080	50 670	111 700
5	(8)	1.82	2.38	-	-	-	-	698	1142	52 300	115 300
6	(7)	2.01	2.63	-	-	-	-	772	1262	54 100	119 200
7	(8)	2.20	2.88	-	-	-	-	846	1382	58 900	129 900
8	(7)	1.15	1.50	18.3	0.60	10.7	35.0[6]	441	720	26 200	57 250
9	(7)	2.68	3.50[6]	21.3	0.60	12.2	40.0[6]	1028	1680	58 540	128 800

(1) Average of range given by manufacturer

(2) Not given in literature of one manufacturer; for draglines, some data probably are for maximum boom length, other for optimum

(3) For shovels only; maximum digging depth for longest boom, for draglines

(4) For cycle time of 60 sec

(5) Adjusted for 10% non-steel, 65% Ti-6Al-4V alloy and 15% Al or a factor of 0.62 × Earth wt

(6) Not given but estimated from other models of similar weight

(7) Diesel

(8) Electric

NOTE: Some machine weights apparently include counterweights. One manufacturer's weights include "maximum ballast."

boom weight and equivalent volume load are less than on Earth so that boom lengths and consequently mining depths may be correspondingly greater. The same conclusions with regard to boom lengths and mining heights should be true of shovels, but to a lesser degree.

The capacities shown in Table IV indicate that both the shallow and the deep deposit can easily be handled by even the smallest draglines and shovels. The larger machines have considerable excess capacities under the assumptions made -- in many cases, more than double. The No. 1 shovel and the No. 1 and No. 8 draglines are not grossly oversized in the deposits chosen.

FRAGMENTATION-BLASTING ON THE MOON

The determination of the blasting pattern, the quantity of explosives, and the detonation sequence for effective rock breakage are the main problems that must be solved for efficient blasting of any rock material. The trial and improvement method of developing satisfactory procedures is expensive but feasible in Earth operations; however, it would be essentially impossible for lunar operations. In addition, there is no way of determining if a method so developed is the most efficient or economical.

The present state of knowledge concerning the mathematical calculation of the explosive charge has been developed to a fairly high degree, but much remains to be done. Most formulas to date have been based on a particular usage, with no general formula available.

This section of the report will attempt to trace the development of current formulas and modify them for lunar use.

General Case. - The force generated by an explosive must break as well as move the rock; it has to overcome the resistance against breakage and against gravitational forces. Belidor recognized this dual resistance of rock as early as 1725 [37, 38]. He stated that one part of the charge was proportional to the strength of the rock and the other proportional to the volume excavated. He proposed the formula

$$Q = a\,V^2 + b\,V^3 \tag{1}$$

where Q = explosive charge

V = burden (least line of resistance)

a = constant based on rock strength

b = constant based on gravitational force.

Because early interest centered about military applications (i.e., the formation of craters), Belidor's work was quickly forgotten and replaced by the classical cube-rule (proposed earlier by Vauban) which is based on the law of conformity. Simply stated, the explosive charge is proportional to the volume of the crater produced. The cube-rule is expressed as

$$Q = k\,v^3 \tag{2}$$

where k = constant based on rock type.

Based on the results of extensive research since World War II, Langefors and Kihlstrom [39] have developed the following general formula which is an extension of Belidor's formula to include a "through" or "swell" component:

$$Q = k_2\,V^2 + k_3\,V^3 + k_4\,V^4 \tag{3}$$

where k_2, k_3, and k_4 are constants dependent on rock type. They additionally reported that for ordinary bed rock

$$Q = 0.01\,V^2 + 0.40\,V^3 + 0.004\,V^4 \tag{4}$$

where Q is in kilograms and V is in meters.

A close look at equation (3) shows the following facts of interest:

1. For small blasts ($V < 1$ m) the strength of the rock is most important.
2. For relatively large high explosive blasts ($1 < V < 10$ m) the volume of material to be excavated is of greatest importance.
3. For larger high explosive or nuclear blasts ($V > 10$ m) the amount of throw is of considerable importance.

Kochanowsky's work [37], based on the results of numerous test blasts, had also suggested the variability of rock resistance as related to the size of the blast. He made the following postulations:

1. The larger the piece of rock to be blasted, the greater the points of weakness and, therefore, the smaller the specific strength.

2. The larger the burden, the further the material must be thrown for effective excavation.

These considerations explain the past failures resulting from the use of the cube-rule to predict the results of large blasts from the analysis of small test blasts.

The equation developed by Langefors and Kihlstrom is the general prediction formula for crater blasts, and as such gives reliable results when applied with the appropriate constants for each rock type.

Assuming ordinary lunar rock to be similar in strength and mass to Earth bed rock, the prediction equation for crater blasts can be applied to determine if high explosives could be used economically for blasting rock.

Based on the preceding assumption, the first term of equation (4) would remain the same, but the second and third terms would have to be modified to account for the change in gravity. Since lunar gravity is approximately 0.166 that of Earth gravity, the second term would be $0.07\ V^3$ and the third $0.0007\ V^4$. This indicates a greater proportion of the explosive will be required to break the rock in lunar operation than on Earth. Rewriting the equation, we have

$$Q_L = 0.01\ V^2 + 0.07\ V^3 + 0.0007\ V^4 \qquad (5)$$

where Q_L is lunar explosive charge in kilograms.

A more meaningful comparison can be made if the equations are converted to calculate the specific charge (kg/m^3). This can be done by dividing through by V^3 to obtain:

$$q = \frac{0.01}{V} + 0.4 + 0.004\ V \qquad (4a)$$

$$q_1 = \frac{0.01}{V} + 0.07 + 0.0007\ V \qquad (5a)$$

where q is specific charge on Earth (kg/m^3) and q_1 is specific charge on moon (kg/m^3).

Calculation of the specific charge required for burdens ranging from 0.01 to 100 meters has been made and shown in Table V for comparison. Also included is a ratio of the lunar specific charge to the Earth specific charge (n).

TABLE V. SPECIFIC CHARGE CALCULATIONS FOR EFFECTIVE FRAGMENTATION ON THE EARTH AND MOON

Burden (meters)	Specific Charge (kg/m^3) Earth (q)	Moon (q_1)	Ratio q_1/q (n)
0.01	1.400	1.070	0.764
0.10	0.500	0.170	0.340
1.00	0.414	0.081	0.196
10.00	0.441	0.078	0.177
100.00	0.800	0.140	0.175

The calculations shown in Table V indicate that the most efficient blasts are those conducted with burdens between 1 and 10 meters on both Earth and moon. Also it will be noted that as the burden increases, the ratio, n, decreases and approaches 0.166.

It is a well-known fact that in Earth operations, the specific charge for bench blasts varies from approximately 0.2 kg/m^3 for soft rock to 0.6 kg/m^3 for hard rock. Assuming the same relationship for bench blasting on Earth and moon as for crater blasting, it can be predicted that the specific charge for lunar bench blasting will vary from approximately 0.04 kg/m^3 for soft material to 0.12 kg/m^3 for hard material.

Assuming that most of the "ores" to be mined initially will have a mass of approximately 2 454 kg/m^3, it will require between 16.5 g/metric ton (0.033 lb/ton) mass to 49 g/metric ton (0.098 lb/ton) mass of explosive to effectively fragment and excavate the ore.

Because of a lack of a definite cost figure for sending supplies to the moon, Table VI has been developed to give an idea of the quantity of explosives in kilograms required to produce one kilogram of water for various grades of "ore".

TABLE VI. ESTIMATED AMOUNT OF EXPLOSIVES REQUIRED TO PRODUCE WATER ON THE MOON

Grade of Ore %	Explosives required (kg/kg of water) Soft "Ore"	Hard "Ore"
1	0.0017	0.0049
5	0.0003	0.0010
10	0.0002	0.0005
100 (ice)	0.00002	--

From Table VI, it can be concluded that the relative cost of explosives necessary to produce water on the moon is small compared to the cost of shipping water to the moon, assuming a constant cost per pound for supplies.

If a chemical explosive or blasting "agent" can be developed that will be stable and safe to use in the lunar environment, it should be feasible to use explosives for fragmentation and excavation.

In later work, the conclusions approximated in Tables V and VI will be applied to specific explosives and to their use in selected model deposits.

SUMMARY

The feasibility of mining on the moon depends upon many factors, some of which cannot be well defined at this time. The two largest uncertainties are the availability of quantities of desirable native materials which could be used for the production of fuel components and water, and the magnitude of the lunar and planetary program which will determine the extent to which lunar resources will be used or needed.

Laboratory experiments and test programs have indicated that the physical characteristics of lunar surface materials may differ in several respects from those of terrestrial rocks and soils. The vacuum environment has been shown to have a pronounced effect upon porosity, bearing strength, adhesion, and the frictional characteristics of simulated lunar surface materials.

Studies have been made of several surface mining methods which could be adapted for use on the moon. The requirements for power, manpower, and ore production were considered in rejecting certain systems commonly used on Earth. Crew safety, weight and delivery costs, simplicity of operation, and mine operations were other factors which further restricted the remaining systems to the ones presented.

These preliminary experiments, studies, and tests have indicated that surface mining on the moon may be advantageous if this nation's space program becomes large enough to require extensive lunar exploration or planetary exploration from a lunar base or colony.

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TERRESTRIAL BY-PRODUCTS OF LUNAR ORIENTED RESEARCH IN EXPERIMENTAL PETROLOGY

By

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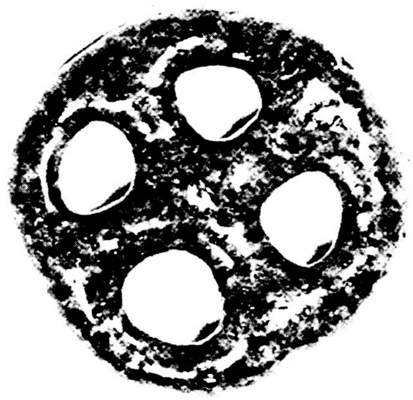
For more than six years, Northrop Corporation and NASA OART have been supporting the study of complex rock materials under extreme pressures and temperatures, thus developing the technique and skill for handling and analysis of complex materials.

The prime purpose of that study was to develop the basic knowledge needed for beneficial use of lunar rocks. Several results, which may have been incidental to that study, establish a better understanding of complex solids in three fields of terrestrial interests: (1) materials, or complex metal oxides; (2) high speed transportation, or re-entry temperature systems; and (3) geophysics, or elastic wave propagation through rock media at elevated temperature and pressure. Each one of these fields will be discussed briefly in an attempt to show the way in which the lunar study has yielded terrestrial by-products of significance.

COMPLEX METAL OXIDES

With the prospects of landing man on the moon and the probability that the major raw material on the moon is one kind or another of common silicate rock, the need for common rock utilization came into focus. The new problem is not how to combine ceramic and metallic properties, but how to add some metallic properties to a common silicate rock. The products of such beneficiation could be a single phase, a multiphase, or a composite of interlocking grains having properties not possessed by the original rock. A systematic study of the behavior and properties of common rocks under conditions of high pressure and temperature yielded much basic data that is needed to develop processes for melting, casting, heat treating, forming in the plastic state, and sintering of rock powders. Figure 1 shows two rocks cast into shape and one bent in the plastic state below the melting temperature of the rock.

(a) BASALT 1080 °C 1 1/2 HOURS BENDING BELOW THE MELTING



(b) OBSIDIAN DUST 1300 °C 1 1/2 HOURS MELT AND CAST

(c) OBSIDIAN DUST 1075 °C 12 HOURS MELT AND CAST

FIGURE 1. TWO ROCK CASTS AND ONE ROCK BENT IN THE SOLID STATE

The art of using natural rock material for production of useful utensils is one of the oldest human trades on Earth. Beginning thousands of years ago with porous ceramics made from naturally occurring clay silicates, this art took a big jump forward as early as the seventh century with the invention of a dense white "china porcelain." The raw material for this new ceramic was primarily naturally occurring kaoline, feldspar, and quartz. When fired at high temperatures, this complex mixture formed a multiphase composite of crystals, glass, and gas inclusions. It was not until after the beginning of the twentieth century that the next step forward was taken with the development of single phase sintered ceramics, made of pure oxides of metals such as aluminum, beryllium, hafnium, thorium, and zirconium. These new products have high chemical and abrasion resistance, low density, and high refractory properties. In the past decade another step forward was taken to combine metal oxides with metal oxides or with their metals and develop end products having some properties of the constituents and some new properties of their own. The common, abundant and accessible raw material that was used for the earlier ceramics gave way to more scarce material thus raising the price of the final products.

While considerable effort was invested by many laboratories in enhancing the properties of metals and pure oxides, the naturally occurring common rock was gradually left behind as unsuitable for the requirements of modern metallurgy or ceramics. Current studies of beneficial uses of common rocks are directed toward the exploitation of the moon. One could, however, outline beneficial terrestrial uses based on these studies. The following is a list of typical items that can be considered for manufacturing from common rocks for terrestrial use:

1. High temperature thermal insulators - flat plates and curved fixtures
2. High temperature electrical insulators - flat plates and curved fixtures
3. Wear resistant flat plates - chemical and mechanical abrasion
4. Wear resistant conduits - chemical and mechanical abrasion
5. Architectural surfaces - wall panels, floors, and table tops
6. Ornamental objects - sculptures, bookends, etc.

In a sense, this is a ceramic line of products except for the "new" raw material, common rock. The description of Kopecky and Voldan [1] is an excellent example of the commercial utilization of common basalt rock products. It should be obvious to the reader that the beneficial use of rocks is by no way limited to lunar rocks and that new common rock complex ceramics may turn out to be just as useful on Earth as on the moon.

RE-ENTRY TEMPERATURE SYSTEMS

One of the interesting characteristics of many rocks is that they retain in texture and structure the evidence of the environments and processes to which they have been subjected. The temperature, pressure and time history, and the composition combine to give the rock its characteristics. A plutonic rock, for example, generated in the depths of the Earth looks distinctly different from a volcanic rock of the same composition. In the same way a feldspar exposed to conditions exceeding 30 kilobars and 1400°C will develop garnets and kyanites which disclose the prevailing pressures and temperature.

One of the problems studied in relation to lunar magma has been the correlation between the physical environmental conditions imposed on a variety of rocks and the recognizable characteristics of the rock end-product. This requirement for correlation dictated the need to use a sample which is large enough for both x-ray and petrographic analyses to be conducted after exposure to the experimental conditions. Figure 2 shows a laboratory generated series of characteristics induced by laboratory treatment of a basalt sample from Hokook, Israel, as function of temperature and of time sustained at that temperature. Of these three parameters (time, temperature, induced characteristics) any two parameters approximate the third one. The characteristics of the rocks can thus be used as approximate indicators of the adjacent environmental conditions. The geologic interpretations of the history of the Earth are partly based on such clues to the paleo-environments. Experiments such as those conducted on the Hokook basalt were repeated on a whole range of silicate rocks, and similar experiments were conducted at elevated pressures. The correlation of the petrographic characteristics of the end products with the experimental conditions is being catalogued with the view that as it becomes possible to examine lunar rocks, their genetic history can be reconstructed on the basis of the acquired data.

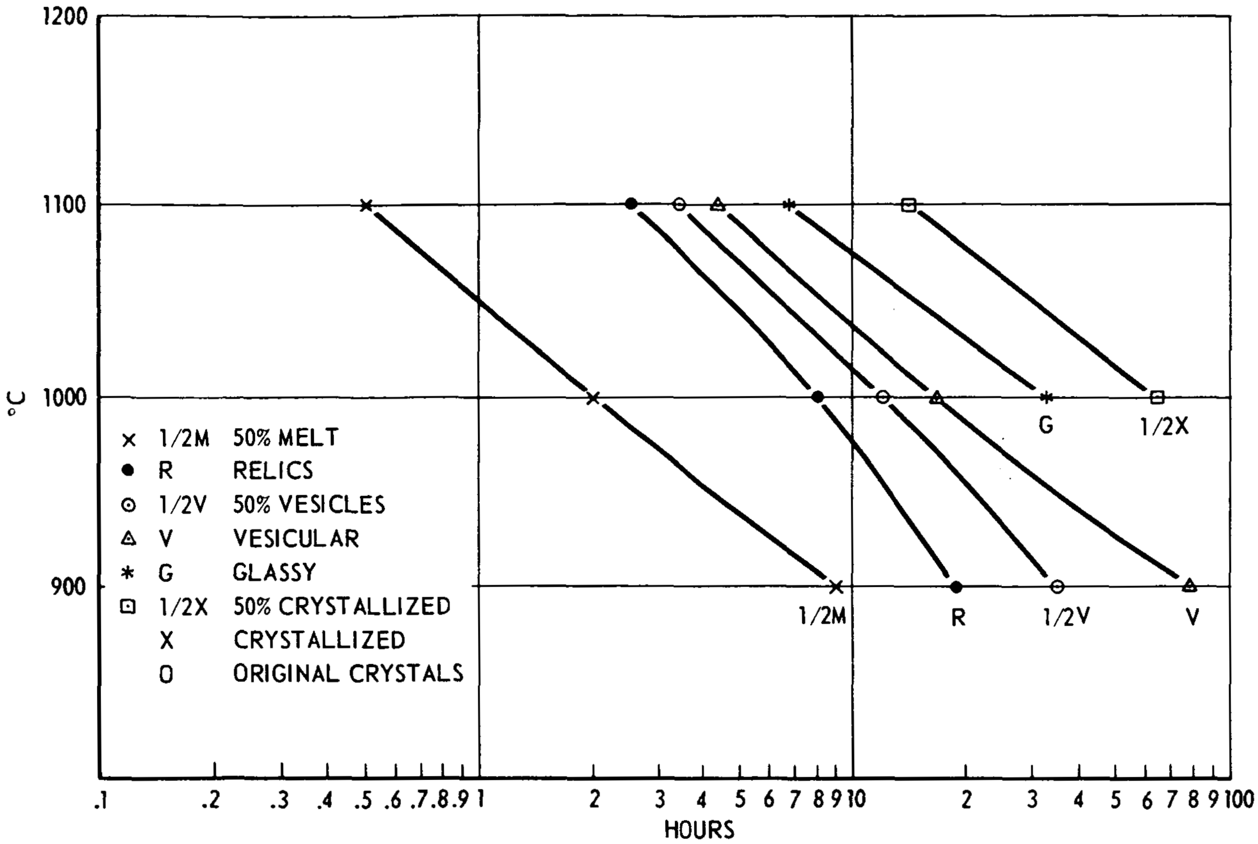


FIGURE 2. CHARACTERISTICS DEVELOPED IN HOKOOK BASALT (ISRAEL) AS FUNCTION OF TEMPERATURE AND TIME SUSTAINED AT TEMPERATURE (LINDBERG FURNACE - 1 ATMOSPHERE)

This type of analysis can be applied to the study of meteorites and to the study of the re-entry of bodies into the Earth's atmosphere. A thorough understanding of the physical and chemical changes that occur during the entry of a body into the atmosphere requires data on the pre-entry size of the body, its precise path through the atmosphere versus time, the effect of the atmospheric composition on the hot body, and the time-temperature-composition correlation with the physical and chemical changes that take place during deceleration. Some re-entry data should be obtained by using crystalline silicate rocks as artificial meteorites where every crystal that survives ablation can be utilized as a sensing device, thus recording the temperature history at each point during the fall.

Figure 3 shows the calculated temperature gradient that develops in a sphere made of dunite weighing one kilogram and having a specific gravity of 3.3, when dropped from an altitude of 150 kilometers with an initial velocity of zero. Figure 4 shows the rate of heating of the same sphere. Both figures indicate that even a small meteorite at free fall could develop high temperatures at least near its surface. Calculated temperature data can be compared with data derived from examination of the artificial meteorites and when enough data have been collected to allow reliable interpretation of the temperature distribution during entry, the re-entry temperature pattern can be examined with greater detail than can be supplied by several conventional sensors positioned on the re-entry body.

This characteristic of a crystalline rock as a multisensor device has been the essence in the study of lunar magma, but it is by no means unique or limited to the extraterrestrial realm.

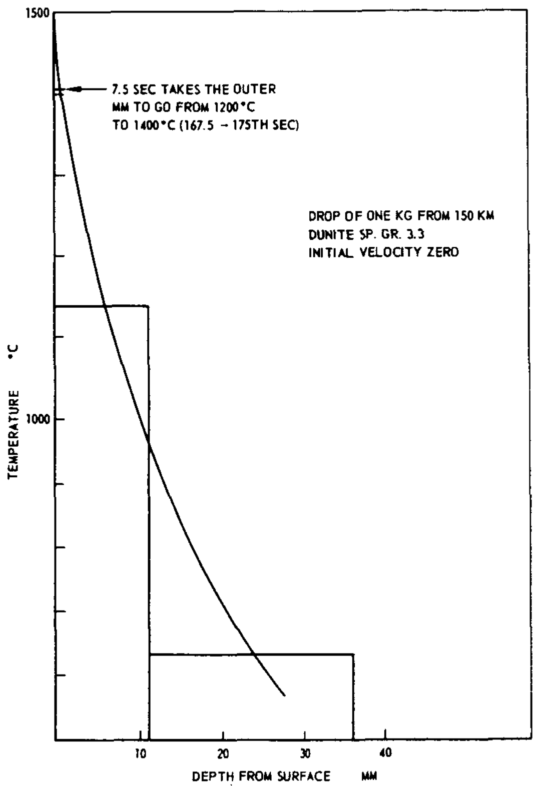


FIGURE 3. CALCULATED TEMPERATURE GRADIENT IN DUNITE ARTIFICIAL METEORITE

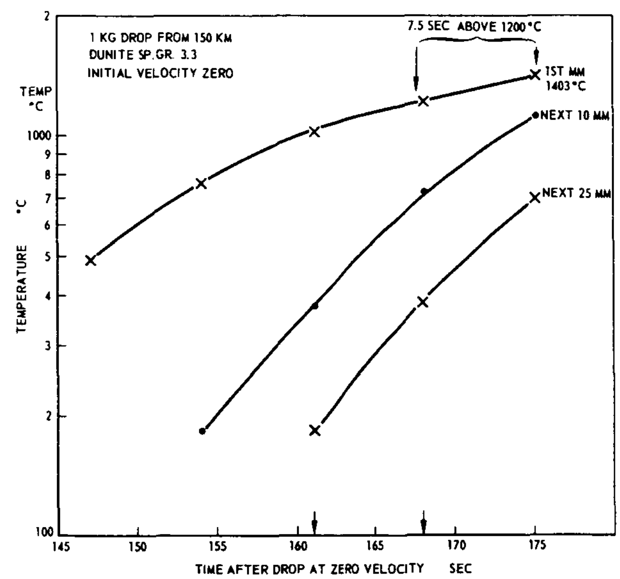


FIGURE 4. RATE OF HEATING OF DUNITE ARTIFICIAL METEORITE

ELASTIC WAVE PROPAGATION THROUGH ROCK MEDIA AT ELEVATED PRESSURES AND TEMPERATURES

One of the difficult problems in the determination of melting in a rock while it is under high pressure is the result of two characteristics of the experiments. One, the bulk of material which surrounds the sample is essential to contain the pressure, but it also makes the sample inaccessible during the experiment and conceals it from direct observations; two, the complex nature of most rock samples and the multiplicity of reactions that may take place in them during a run make the data obtained by conventional electric probing techniques to measure melting (such as differential thermal analysis or resistivity changes) overwhelmingly detailed and too complicated to be simply correlated with the melting event. So, rather than determine melting in situ, one is forced to resort to a quench technique where a "post mortem" petrographic analysis of the sample determines whether melting had occurred.

One promising technique to provide an in situ indication of melting while under pressure is an ultrasonic probing of the elastic characteristics of the rock. Compressional waves and shear waves are dependent on the bulk properties of the medium. Shear waves, for example, attenuate more rapidly as the solid rock gradually becomes more plastic and are completely absorbed when it melts. Figure 5 shows the geometry of the laboratory configuration, and Figure 6 shows a basalt sample and heater arrangement used by Dr. R. D. Tooley of Northrop Corporation in velocity measurements under high pressure and temperature.

The most extensive velocity measurements of rocks under pressure are those of Birch [2] and Simmons [3]. These measurements were made in a hydrostatic pressure vessel. The measurements are very precise but were made only to 10 kilobars. Hughes and Cross [4] have worked at both high pressure and elevated temperatures. Their pressure and temperature limits were five kilobars and 300°C. Ahrens and Katz [5] reached 40 kilobars and room temperature. There are no simultaneous high pressure and high temperature values in the literature above these values. All the work done by the above cited references was strictly directed toward geophysical research. Their published experience was relied on in the attempt to develop an in situ measurement of rock melting as part of a lunar magma study. Now let us re-direct our attention to the Earth and examine one example of application of the combined petrographic-seismic studies of terrestrial interests.

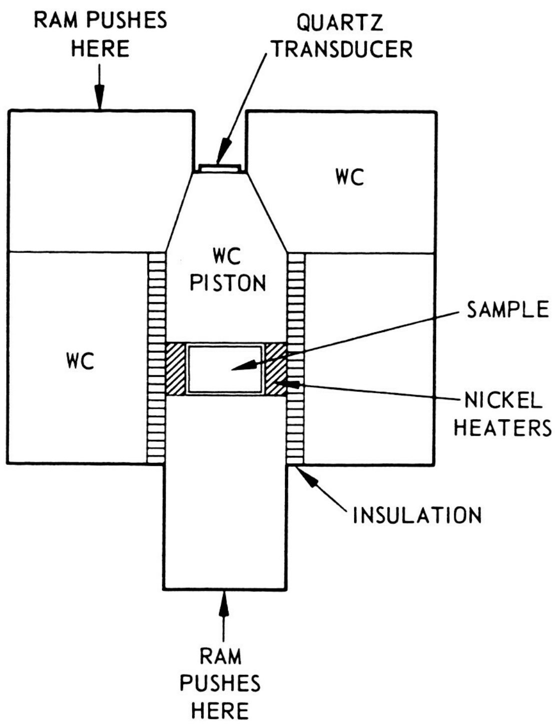


FIGURE 5. GEOMETRY OF THE LABORATORY CONFIGURATION USEFUL IN MELTING DETERMINATION IN SITU

Petrographic analyses of tholeitic basalt were run by Dr. E. Azmon (Northrop) and Dr. R. Gaal (University of Southern California) to determine the changes in percentages of glass and crystals after the specimens had been subjected to various temperature-pressure combinations for a duration of five minutes. The results of these measurements were plotted (Fig. 7) as percent crystals versus the ratio of temperature to pressure. Figure 8 which follows shows that elastic wave velocities in the rocks can be plotted against the same temperature to pressure ratio. If a temperature to pressure ratio can represent conditions at some depth point below the surface of the Earth, then the corresponding data on percent crystals, wave velocities, or many other physical parameters may be related to the rocks at that point. It is convenient to use temperature over pressure rather than specific volume as a common denominator for the above correlation because both temperatures and pressure can be easily measured in the laboratory on a routine basis. Conducting both petrologic and seismic experiments side by side on cuts of the same rocks make the genetic state of the rock, at specific pressures and temperatures, a known parameter which corresponds to measurable wave velocities. Conducting seismological

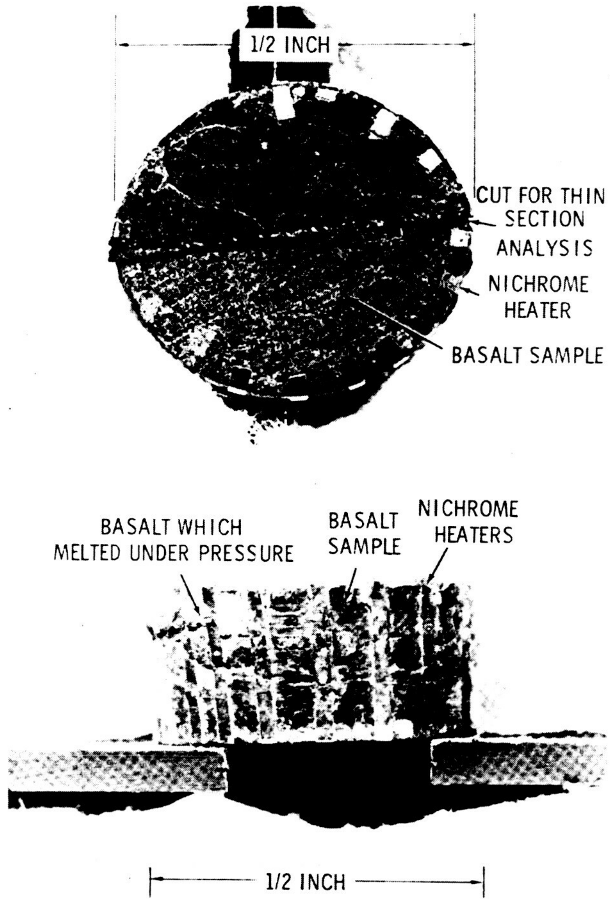


FIGURE 6. BASALT SAMPLE HEATER ARRANGEMENT USED BY DR. R. D. TOOLEY IN VELOCITY MEASUREMENTS UNDER HIGH PRESSURE AND TEMPERATURE

exploration in the field gives us the wave velocities. The estimates of temperatures, pressure (depth), and identity (composition and genetic state) of the rock medium must agree with the laboratory correlation. relation.

This combination of petrology and seismology in laboratory studies is a by-product of a lunar study, but by no means is it unique or limited to the extra-terrestrial realm.

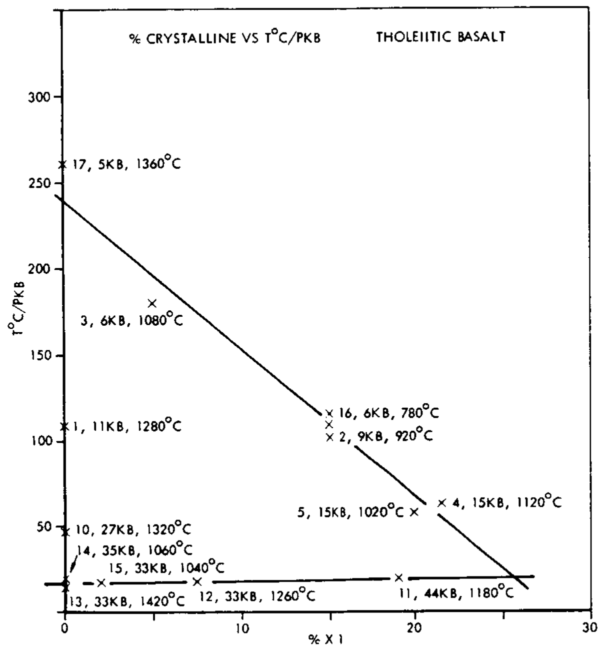


FIGURE 7. PERCENT CRYSTALS RELATION TO THE RATIO OF TEMPERATURE TO PRESSURE

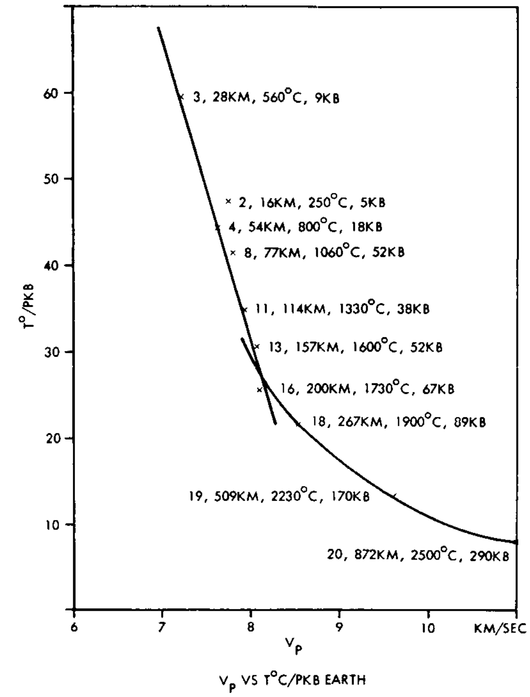


FIGURE 8. VELOCITY OF P WAVES RELATION TO THE RATIO OF TEMPERATURE TO PRESSURE

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POTENTIAL USE OF EXTRATERRESTRIAL RESOURCES IN LIFE SUPPORT SYSTEMS

By

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SUMMARY

Extraterrestrial resources include those of geological origin, the atmosphere, solar and cosmic radiation, and possibly life. The uses of potential resources are discussed with respect to the functions of life support systems to provide food, water, atmosphere, optimum temperature, energy, and a means of waste disposal.

INTRODUCTION

The concepts of space exploration, space travel and colonization of the extraterrestrial bodies have never ceased to entrance the science fiction writer and captivate his reader's imagination. To propose possible uses of extraterrestrial resources for life support systems gives me license to join the ranks of the science fiction writer if only as a science proposal writer (the difference being one of degree). The science fiction writer has infinite freedom of thought with no demand on him for reality. The science proposal writer is one who must labor under the inhibition of immediate realities and possible attainments of engineering applications of the biological, chemical, and physical sciences. However inhibited, nothing can substitute for the excitement, stimulation, and reward received by those working toward the actual fulfillment of new concepts.

Succinctly stated, extraterrestrial resources include those of geological origin, solar and cosmic radiation, and atmosphere and indigenous life. The utilization of these resources will be considered with respect to requirements of food and water, gaseous atmosphere, solid and liquid waste disposal, optimum temperature, and energy generation.

FOOD AND WATER

The feeding of man at a space station will be in the initial stages a situation in which man is supplied foods from Earth. Later, as the stations evolve and become larger, much of the food supply will undoubtedly be produced at the space station. Food sources of terrestrial origin will include freeze-dried compressed rations and/or completely liquid diets supplemented perhaps with higher energy synthetic foods. Foods produced at the space stations will include conventional foods of plant origin; microorganisms including algae, bacteria, fungi and protozoa; undifferentiated plant and animal tissues; and those chemically synthesized. The chemical synthesis of foods is a longe-range projection, particularly for extraterrestrial conditions. Of all the systems considered to date, hydroponics for the production of conventional foods and mass algal culture offer the only dependable methods of food production for proposed space situations. As their culture is a photosynthetic process, there is the added bonus of maintaining the organic energy level of the system through the regeneration of high energy food and oxygen from low energy gases, liquid and solid waste products. Next most promising is the mass culture of hydrogenous and undifferentiated plant tissue. The use of undifferentiated animal tissues and other microorganisms will develop more slowly but will gain importance as it lends a variation in flavor, texture, and quality of the foods in the diets of the astronauts. The use of animals is questionable, particularly since their growth is extremely expensive in respect to requirements for oxygen, complex organic nutrition, space, more exacting hygenic practices, and attending personnel.

There are several avenues by which extraterrestrial resources can be used to support food production and water reclamation. At space stations, materials of geological origin may serve as sources of inorganic nutrient for plant and tissue production. In the hydroponic culture of conventional food plants, rock and sand may serve as root support media, as aggregate and cement for vessels, and as structural supports for vessels. Media should be available for liquid-gas stripping columns, for surface reaction columns, for filters, and as chemical sources for chemical food synthesis. Materials may also be found that will be useful in ion-exchange reactions, particularly for water purification and concentration of plant nutrients.

Indigenous life, both of animal and plant origin, may ultimately serve as food. Indigenous life will require the same careful nutritional evaluation for wholesomeness that all new terrestrial foods are given. Life on extraterrestrial bodies may not follow the same metabolic pathways or use the same basic elemental building blocks we experience here on Earth. Physiological saline is required of most living organisms to maintain isotonic conditions, sodium chloride serving as the source of saline. This is quite understandable as all life is thought to have originated from the seas which are rich in sodium chloride. However, in life systems which may have evolved from other environments, sodium fluoride might possibly serve in the same function. Life utilizing sodium fluoride for maintaining physiological saline would not be acceptable as food for humans of earthly origin.

The use of solar radiation energy of electromagnetic wavelength for photosynthetic processes has frequently been proposed. Unfortunately, it appears impossible to filter out cosmic and undesirable solar radiations while permitting the visible spectrum to pass through a filter medium. The products resulting from the interaction of heavy particles and cosmic radiations can be radioactive. There are also secondary forms of ionizing radiation resulting from these interactions. Their concentrations are relatively low, too low for sterilization of waste products, but may be too high for viable organisms because of their relatively long life and the accumulative effect. Someday, plants and animals may possibly adapt a radiation resistance similar to that inherent or developed through mutation in certain bacteria and algae. There are algal strains that live on the spent fuel rods from atomic reactors. These algae have been cultured and found virtually immune to gamma radiations. At present then, while the concentration of solar and cosmic radiations is too low for complete destruction of viable organisms, the potential long-term effect of radioactive particles appears to be a real danger.

Atmospheric conditions of vacuum and extreme cold, particularly as experienced on the moon, might have direct application to food production and water reclamation. The extreme cold may serve to preserve foods and to desalt waste waters by freezing. If freeze-drying should be employed for foods produced on the moon for further space exploration, the energy presently used for freezing and to produce a vacuum under terrestrial conditions becomes free in the lunar environment. This same concept has application to vacuum distillation for water purification and water from viable plant tissues. The feasibility of using near vacuum low temperature (300 to 500°C) plasma conditions for the chemical synthesis of food is presently being considered.

GASEOUS ATMOSPHERES

If maximum benefits are to be derived from photosynthetic culture of organisms and plants, the conversion of carbon dioxide and water into oxygen and high energy organic material needs to be realized. It has been proposed that silicates might serve as a source of oxygen. Materials of geological origin may serve as chemicals and media for filters, gas absorption, and carbon dioxide and liquid stripping from closed system atmospheres. The use of alumina in air dryers makes it possible to concentrate carbon dioxide and moisture in backwash gases while flow-through gases are freed of these fluids.

SOLID AND LIQUID WASTE DISPOSAL

The waste management system will depend upon the extent to which the ecological system is closed. Should it be desirable to retrieve water, minerals, and organics, waste management systems will become an integral part of the food and water regeneration systems. If not, the function of the systems will be to stabilize the wastes.

Conditions of vacuum atmosphere and temperature extremes would serve for freeze dehydration and/or heat sterilization of wastes. Materials of geological origin would serve as media for trickling filters, as sand for filters, and as process vessels.

OPTIMUM TEMPERATURE

The extreme temperature differences known to exist on the moon (-230 to 235° F) will require means for cooling and heating to maintain man's environment

[1]. The heat from energy generation, natural radiation, cooking, and physiological processes can be conserved for periods when the station is in a dark phase, a time of intense cold. Warm air and liquids during periods of excess heat can be pumped into a deep well, the soil and rock absorbing the heat while cooling the air and liquids. When there is a deficit of heat the process continues but in reverse. Cold fluids can then receive heat from the well reservior. Certainly, lunar materials should be studied for insulating properties against extreme temperature conditions and harmful solar radiations for possible use in the development of shelters, clothing, and vehicles. In these same applications, the materials may also serve as ballast.

ENERGY GENERATION

Although one does not usually associate power generation with environmental life support systems, it is directly and significantly applicable to this operations. Power is required to operate pumps, compressors, gas exchangers, lighting, and electrochemical units. As the atmosphere on the moon is void of gases, materials may exist which will support combustion. Raw materials may be found that can be directly used for batteries and fuel cells. The existence of radioactive materials has frequently been postulated. Solar radiation is presently being used for solar batteries. J. Green [2] suggests that the generation of electricity through systems coupled with a solar furnace may compete on a weight basis with that of atomic reactors for moon station applications.

CONCLUSIONS

Extraterrestrial resources can and will find applications in life support systems. The extent to which they will be used will depend upon their intrinsic value, ingenuity used in their application, and requirements with respect to the life support system, shipping costs, and necessary processing. It should be apparent from the foregoing discussion that even such a negative condition as a vacuum atmosphere can in itself be put to useful work.

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FIRE HAZARDS IN SPACECRAFT ATMOSPHERES

By

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If man is to live, travel, and work in extraterrestrial space, he must carry his atmosphere with him. This need not be the normal atmosphere of the Earth's surface. A range of atmospheric compositions has been shown to be capable of supporting human life for extended periods of time. Engineering considerations suggest that atmospheres containing a higher portion of oxygen than normal air at a total pressure somewhat less than 14.7 psia will be desirable for extraterrestrial use.

The increased fire hazard inherent in the use of oxygen-enriched atmospheres was recognized at the outset of the manned space flight program. The safety of a man in space, however, depends on the precise functioning of a number of complex systems, often with conflicting requirements for reliable operation. The choice of a spacecraft atmosphere results from careful evaluation of several possible solutions to satisfy these requirements. Recent tragic accidents suggest that the assumptions underlying this choice should be reexamined. This paper is concerned only with the fire hazard problem and its bearing on the choice of a spacecraft atmosphere.

Classical fire hazard control theory teaches that three conditions must be met in order to produce a fire -- a source of oxygen must be present, a supply of combustible fuel must be present, and a source of ignition energy must be provided. The elimination of an oxygen source appears to be impossible since no atmosphere is known which will sustain life for a significant period and which will not also support the combustion of a wide range of materials. By careful selection of materials, the amount of combustibles present can be kept to a minimum, but complete elimination is not possible. Noncombustible substitutes for many essential materials have not been developed. Food, fuel, and the astronauts themselves cannot be made fireproof. Potential ignition sources can also be kept to a minimum through careful design; but with requirements for electrical power, moving parts and and reactive chemicals, and even external hazards such as meteorite penetration, the possibility of an ignition occurring cannot be completely eliminated.

Because the possibility of a fire in space cannot be ruled out, consideration should also be given to the development of methods of combating such fires. No adequate and proven methods are available at this time.

Certain unusual characteristics of spacecraft fires will influence the choice of methods to be used in combating them:

1. The fire will take place within a confined volume; personnel will be located within the fire space; it will not be possible to fight the fire from "outside."

2. Fires in oxygen-enriched atmospheres will develop very rapidly; automatic detection and initiation of extinguishment procedures will be needed.

3. The oxygen supply will be limited in contrast to a terrestrial fire which may be fed by an unlimited supply of air; a vacuum may surround the fire capsule.

4. The fire may occur in a reduced or zero gravitational field; this could affect both the development of the fire and the behavior of the fire extinguishment system.

5. The method of extinguishment must not only control the fire, but it must not be dangerous to the astronaut, and it must not damage operating systems essential to a safe return to earth.

An experimental program to help define the fire hazard in extraterrestrial atmospheres was conducted under a program [1, 2] sponsored by the USAF School of Aerospace Medicine, Brooks Air Force Base, Texas. A fire in a space simulator chamber at Brooks prompted the study. The initial program considered only the atmosphere in use in the chamber at the time of the fire, five psia of oxygen. Later, oxygen-nitrogen and oxygen-helium atmospheres were studied briefly. Materials used in the construction and operation of the simulator chamber received particular attention. Because the fire occurred on the 27th day of a planned 30-day experiment, it was suggested that long exposure of the materials to the oxygen atmosphere might have contributed to the speed with which the fire spread. This question was investigated.

Two criteria were selected as indicative of the hazard associated with the use of a given material in a particular atmosphere -- the ease with which the material was ignited and the rate of flame spread over the surface.

The ease of ignition was determined by exposing a sample of the material to a constant thermal radiation flux and noting the time to appearance of a visible flame. This does not provide a measure of the true ignition energy because no correction is made for losses caused by reflection, transmission, or convection. It does provide a convenient means of comparing the effects of different atmospheres on the ignition process and also provides a rough indication of the relative ease of ignition of the different materials.

Flame spread rates were obtained from the analysis of motion pictures of burning strips of material. Where possible, sample strips one-half inch wide and four inches long were used, with flame propagation in the horizontal plane. Under conditions more representative of a real fire situation, the flame spread rate would be enhanced by the interaction of adjacent burning surfaces and by buoyant convection in the upward direction. The flame spread rates given here must be considered as minimum rates, but the method permits reproducible measurements and makes possible the comparison of different materials and the observation of the effects of different atmospheres.

The samples of the materials used in the construction of the simulator chamber were frequently irregular in shape and of indefinite or composite make-up, making it difficult to obtain precise measurements and to interpret results. A second series of materials of better defined chemical compositions and more uniform physical form was used to obtain data for detailed interpretation.

Data on the ignition of materials in air and in oxygen at five psia are presented in Table I. As may be seen, slightly less energy was required to ignite most of the samples in the oxygen atmosphere than was required for ignition in air. In the case of asbestos tape, burning was confined to the organic sizing material used to bond the asbestos fibers. Detection of the ignition point was difficult, and large experimental errors could occur. Storage in oxygen for up to 30 days appears to have no significant effect on the energy required for ignition in most cases. The paint samples, however, show a substantial increase in energy requirements after oxygen storage. This could result from either a loss of solvent or a curing of the paint resin during the long storage period.

TABLE I. EFFECT OF OXYGEN AND STORAGE IN OXYGEN ON ENERGY REQUIRED FOR IGNITION OF VARIOUS MATERIALS

Material	Energy Required for Ignition (cal/cm^2)		
	In Air	In 258 mm Oxygen	
		Before Storage	After 30-day Storage
Masonite	18.8 ± 0.8	13.3 ± 1.0	16.4 ± 1.8
Pine wood	19.1 ± 0.8	18.1 ± 1.7	15.0 ± 2.8
Hard wood	23.5 ± 2.0	22.6 ± 2.0	19.2 ± 2.5
Index card	33.4 ± 0.8	24.5 ± 1.6	32.4 ± 3.0
Masking tape	22.4 ± 0.5	20.2 ± 0	21.4 ± 1.9
Cotton shirt fabric	13.0 ± 0	14.8 ± 1.2	17.0 ± 0.3
Brown food packet	6.6 ± 0.7	3.3 ± 0.7	4.4 ± 0.3
3-M velvet paint (beige)	10.0 ± 0.4	8.8 ± 0.5	11.9 ± 0.3
Capon paint (ivory)	30.0 ± 2.5	35.8 ± 0.6	69.5 ± 5.0
Grey paint	19.2 ± 0.3	19.5 ± 0	33.6 ± 4.0
Asbestos tape	59.9 ± 1.6	18.8 ± 0.3	31.0 ± 2.0

NOTE: Intensity of irradiation was 13.25 cal/cm^2 sec

Data on the ignition of materials in mixed atmospheres are presented in Table II. In the case of dense materials of low thermal diffusivity such as wood and paper, the composition of the atmosphere has little effect on the energy required for ignition. In the case of the atmosphere containing 80 percent helium, increased convective heat loss from the irradiated surface does have a significant effect. The increased surface area of the cotton fabric accentuates this heat loss, and the sample fails to ignite in the 80 percent helium atmosphere. In the case of the plastic-coated wire and the painted metal surface, the metal provides an additional heat sink to drain energy from the ignition site. With these materials, conditions for ignition are marginal in this experiment, and the effect of increased heat loss is apparent even at the lower helium concentrations.

It appears that the composition of the atmosphere has little effect on the true ignition energy; but through its effect on the rate of dissipation of energy from a heat source, it may have a significant effect on the probability of ignition occurring. Similar observations have been made by others in studies of hot wire ignition in mixed atmospheres.

TABLE II. ENERGY REQUIRED FOR IGNITION OF MATERIALS IN VARIOUS ATMOSPHERES (cal/cm^2)

Atmosphere	Air	20% O_2 80% He	46% O_2 54% N_2	46% O_2 54% He	70% O_2 30% N_2	70% O_2 30% He	100% O_2
Pressure (mm Hg)	760	760	380	380	258	258	258
Wood	25 ± 1	109 ± 11	25 ± 2	24 ± 0.5	25 ± 1	22 ± 1	23 ± 1
Paper	32 ± 1	39 ± 0.5	25 ± 2	26 ± 0.5	26 ± 0.5	25 ± 0.5	25 ± 1
Cotton Fabric	13 ± 0.5	NI	12 ± 0.5	17 ± 0.5	15 ± 0.5	16 ± 0.5	15 ± 0.5
Plastic Wire	20 ± 1	NI	16 ± 1	NI	17 ± 1	46 ± 1	16 ± 1
Painted Surface	30 ± 1	NI	56 ± 5	70 ± 4	61 ± 3	57 ± 5	36 ± 1

NOTE: Intensity of irradiation was 13.25 cal/cm^2 sec

Table III presents data on rates of flame spread over the surfaces of various materials in air and in oxygen at five psia. It will be observed that a number of these materials failed to propagate combustion in air. These tests were conducted with small isolated samples burning in the horizontal direction. Many of these materials will burn in air when the flame propagation is in an upward direction or in more complex configurations where interaction between opposing burning surfaces is possible. Only the fully halogenated materials, Kel-F and Teflon, failed to burn in the oxygen atmosphere. Even these materials are known to burn, however, in higher pressures of oxygen or if brought to a sufficiently high temperature by a separate energy source such as a more readily combustible fuel.

TABLE III. EFFECT OF OXYGEN AND STORAGE IN OXYGEN ON FLAME SPREAD RATES FOR VARIOUS MATERIALS (EDGES NOT INHIBITED)

Material	Flame Spread Rate (in./sec) In Air	In 258 mm Oxygen: Before Storage	In 258 mm Oxygen: After 30-day storage
Butyl rubber	0.006	0.40 ± 0.04	0.31 ± 0.04
Canvas duck	NP	0.25 ± 0.05	
Cellulose acetate	0.012	0.28 ± 0.12	0.24 ± 0.12
Kel-F	NI	NI	NI
Natural rubber	0.010	0.61 ± 0.05	0.61 ± 0.08
Neoprene rubber	NI	0.32 ± 0.04	0.25 ± 0.05
Nylon 101	NI	0.19 ± 0.05	0.15 ± 0.01
Plexiglas	0.005	0.35 ± 0.01	0.24 ± 0.01
Polyethylene	0.014	0.25 ± 0.05	0.36 ± 0.06
Polypropylene	0.010	0.35 ± 0.01	0.36 ± 0.12
Polystyrene	0.032	0.80 ± 0.20	0.51 ± 0.01
Polyvinyl chloride	NI	0.10 ± 0.01	0.06 ± 0.01
Silicone rubber	NI	0.14 ± 0.01	0.14 ± 0.01
Teflon	NI	NI	NI
Viton A	NI	0.003 ± 0.002	0.01 ± 0.005

NOTE: All samples except canvas duck, 3 by 1/2 by 1/8 in.; canvas duck, 3 by 1/2 by 1/80 in.

NP -- No sustained propagation of flame

NI -- No ignition of material

In the case of the materials which burn in air, the rate of flame spread in oxygen was one to two orders of magnitude greater although the partial pressure of oxygen was increased only from three to five psia. Thirty-day storage in the oxygen atmosphere had little effect on the flame spread rate. This is not unexpected because surface equilibration with oxygen would be expected to occur rapidly while any slow oxidation of the material which might affect its flammability would be expected to proceed at approximately the same rate in air and in 258 millimeters of oxygen.

Flame spread rates for materials used in the construction of the simulator chamber are shown in Table IV. Most of these materials failed to sustain combustion in air in the small sample configurations tested. Most of them, however, burned vigorously in five psia of oxygen, the only exceptions being Teflon, Teflon-insulated wires, and glass wool. Where comparisons can be made, the rate of flame spread was increased

TABLE IV. EFFECT OF OXYGEN AND STORAGE IN OXYGEN ON FLAME SPREAD RATES FOR VARIOUS SPACE CABIN MATERIALS (EDGES NOT INHIBITED)

Material	Flame Spread Rate (in./sec)		
	In Air	In 258 mm oxygen	
		Before Storage	After 30-day Storage
Aluminized Mylar tape		1.95	
Aluminized vinyl tape	NI	3.1 ± 0.4	3.0 ± 0.4
Asbestos insulating tape	NI	0.08	0.05
Chapstick	NI	1.82	
Cotton shirt fabric	NP	1.50 ± 0.05	2.10 ± 0.3
Electrical insulating resin	NI	0.27	0.20
Electrical terminal board	NI	0.06 ± 0.01	0.06 ± 0.01
Fiberglas insulating tape	NI	4.2 ± 0.6	2.0 ± 0.8
Foam cushion material	0.19	12.4	11.3
Foamed insulation	0.002	2.2 ± 0.2	3.0 ± 0.3
Food packet, aluminized paper	NI	0.28 ± 0.05	0.26 ± 0.05
Food packet, brown aluminum	NI	0.7 ± 0.30	0.8 ± 0.20
Food packet, plastic	0.33	0.55	0.47
Glass wool	NI	NI	NI
Masking tape	0.17	1.82	
Paint, 3-M velvet	NI	0.15 ± 0.01	0.31 ± 0.02
Paint, Capon ivory	NI	0.38 ± 0.04	0.35 ± 0.02
Paint, Pratt & Lambert, grey	NI	0.60 ± 0.2	0.24
Pump oil	NI	0.89	
Refrigeration oil	NI	0.82 ± 0.07	
Rubber tubing	0.03	0.24	0.25 ± 0.05
Silicone grease	NI	0.92	
Solder, rosin core	NI	0.18	0.25
Sponge, washing	0.07	8.1 ± 0.1	10 ± 2
Teflon pipe sealing tape	NI	NI	NI
Teflon tubing	NI	NI	NI
Tygon tubing	0.18	0.50 ± 0.05	0.52 ± 0.05
Wire, MIL-W-76B, orange	NI	0.57 ± 0.05	0.54
Wire, MIL-W-76B, blue	NI		0.57
Wire, MIL-W-76B, yellow	NI		0.54
Wire, MIL-W-16878, green	NI	NI	NI
Wire, MIL-W-16878, black	NI	NI	NI
Wire, MIL-W-16878, yellow	NI	NI	NI
Wire, MIL-W-16878, white	NI	NI	NI
Wire, misc., white, 3/32 dim.	NI	0.33	0.25
Wire, misc., black, 3/16	NI		0.40
Wire, misc., brown, 7/32	NI	0.51 ± 0.05	
Wire, misc., yellow, 7/64	NI	0.89	
Wire, misc., yellow, 5/32	NI	0.41	

NOTE: NP -- No sustained propagation of flame

NI -- No ignition of material

by from one to three orders of magnitude in oxygen. Plastic foam materials exhibited the highest flame spread rates of any of the materials tested and may be considered to represent particularly hazardous materials for use in spacecraft. Thirty-day storage in the oxygen atmosphere does not appear to have a significant effect on the flame spread rates of these materials.

Flame spread rates for various materials in air, pure oxygen, oxygen-nitrogen, and oxygen-helium atmospheres are compared in Table V. The data indicates that flames spread faster in the helium-oxygen mixtures than in the nitrogen-oxygen mixtures. The flames also spread faster as the percentage of oxygen in the mixture is increased. Correlation of the observed flame spread rates with the physical properties of the gas mixtures was sought. A reasonably consistent correlation was obtained by plotting the rate of flame spread for a given material against the logarithm of the heat capacity of the gas mixture per mole of oxygen. Representative plots are shown in Figures 1 to 4. The scatter of data is sufficiently small to permit straight line extrapolation to zero flame spread rate. This intercept defines a critical inert gas dilution level for the material beyond which the atmosphere ceases to support flame spread. This correlation is expressed as:

$$\log \frac{C_{p(crit)}}{C_{p(O_2)} + n\, C_{p(x)}} = kr$$

where n is the number of moles of inert gas per mole of oxygen, k is a slope factor which appears to be slightly dependent on the nature of the inert gas and r is the flame spread rate. Critical concentrations of nitrogen and helium necessary to suppress flame propagation are shown in Table VI. A higher concentration of helium than of nitrogen is required to suppress combustion or effect an equal reduction in flame spread rate. This correlation suggests that polyatomic gases with high heat capacities such as CF_4 or SF_6 would be more effective diluents than nitrogen or helium in suppressing combustion.

The effect of pressure on the rate of flame spread was investigated briefly, as shown in Table VII. Data from other sources also supports the conclusion that the rate of flame spread in the horizontal plane is relatively independent of pressure. Flame spread rates in an upward direction, supported by buoyant convection, have been found to increase with pressure.

We have also observed that the rate of flame spread over a horizontal surface is the same as the

TABLE V. FLAME SPREAD RATES FOR MATERIALS IN VARIOUS ATMOSPHERES [1] (in./sec)

Atmosphere	Air[2]	Air[3]	20% O_2 80% He	46% O_2 54% N_2	46% O_2 54% He	70% O_2 30% N_2	70% O_2 30% He	100% O_2	100% O_2
Pressure	760 mm	760 mm	760 mm	380 mm	380 mm	258 mm	258 mm	258 mm	258 mm
Wood	-	0.025 ±0.025	0.04 ±0.005	0.12 ±0.02	0.18 ±0.03	0.18 ±0.03	0.27 ±0.03	0.35 ±0.03	
Paper	-	0.08 ±0.01	0.30 ±0.06	0.42 ±0.02	0.63 ±0.05	0.55 ±0.05	0.74 ±0.06	0.90 ±0.07	
Cellulose Acetate	0.012	0.008 ±0.002		0.11 ±0.01	0.15 ±0.02	0.20 ±0.03	0.18 ±0.02	0.30 ±0.01	0.28 ±0.12
Cotton Fabric	0	0.10 ±0.01	0.17 ±0.01	0.9 ±0.3	1.1 ±0.1	1.8 ±0.2	1.2 ±0.2	3.2 ±0.2	1.5 ±0.05
Foam Cushion	0.19	0.14 ±0.02	0 0	2.7 ±0.8	2.1 ±0.3	6.1 ±0.5	6.0 ±0.6	13 ±1	12.4 ±0.5
Plastic Wire	0	0	0	0.25 ±0.01	0.35 ±0.02	0.48 ±0.01	0.60 ±0.02	0.84 ±0.03	0.89
Painted Surface	0	0	0	0.21 ±0.01	0.27 ±0.01	0.32 ±0.02	0.42 ±0.06	0.45 ±0.05	0.38 ±0.04

1. ± indicates average deviation
2. Previously reported SAM-TR-65-78
3. This investigation

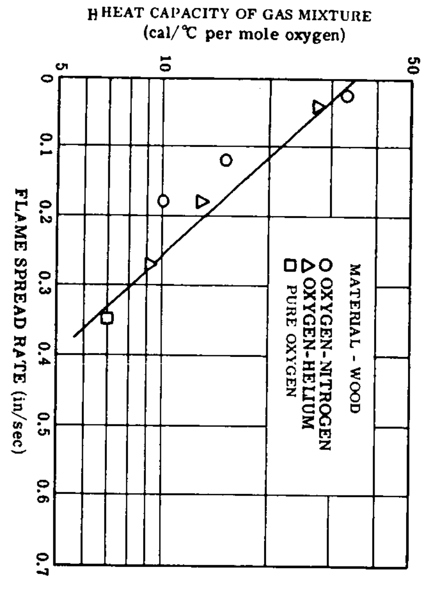

FIGURE 1. FLAME SPREAD RATE VERSUS HEAT CAPACITY OF ATMOSPHERE

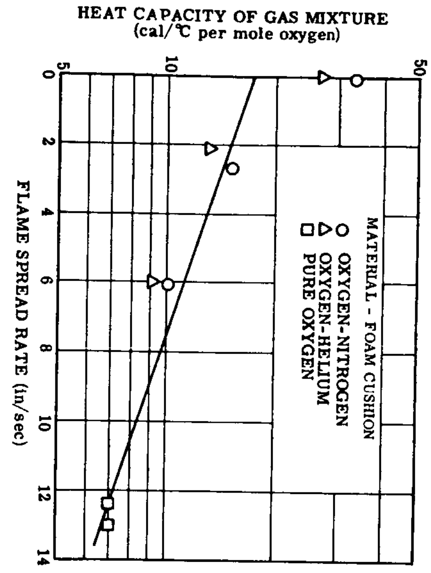

FIGURE 3. FLAME SPREAD RATE VERSUS HEAT CAPACITY OF ATMOSPHERE

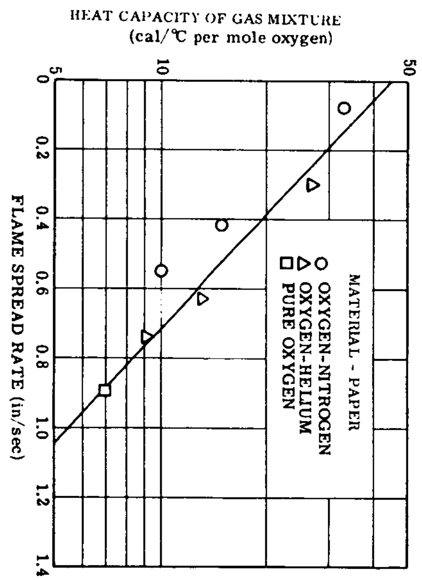

FIGURE 2. FLAME SPREAD RATE VERSUS HEAT CAPACITY OF ATMOSPHERE

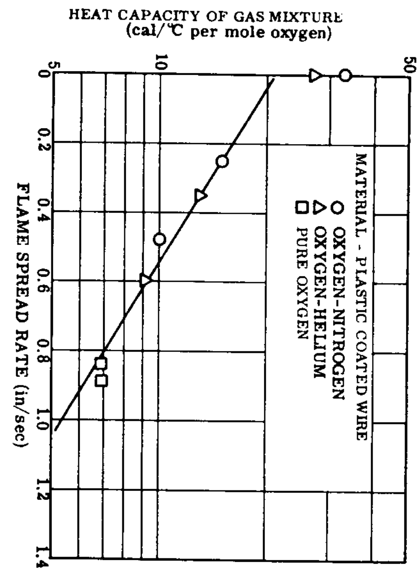

FIGURE 4. FLAME SPREAD RATE VERSUS HEAT CAPACITY OF ATMOSPHERE

TABLE VI. CRITICAL FLAME SPREAD CONDITIONS

Material	$C_{p(crit)}$ cal/°C mole O_2	Critical Inert Diluent Concentration	
		mole % N_2	mole % He
Wood	35.0	80.2	84.8
Paper	45.0	84.5	88.4
Cellulose Acetate	27.0	73.3	80.1
Cotton Fabric	36.0	80.6	85.4
Foam Cushion	17.5	60.3	68.0
Plastic Coated Wire	21.2	65.0	74.0
Painted Surface	27.0	73.3	80.1

TABLE VII. EFFECT OF PRESSURE ON FLAME SPREAD RATE

Material	Atmosphere	Pressure (mm Hg)	Flame Spread Rate (in./sec)
Wood	46% O_2 - 54% He	380	0.18 ± 0.03
Wood	46% O_2 - 54% He	760	0.18 ± 0.02
Paper	46% O_2 - 54% He	380	0.63 ± 0.05
Paper	46% O_2 - 54% He	760	0.64 ± 0.08
Wire	70% O_2 - 30% He	258	0.60 ± 0.02
Wire	70% O_2 - 30% He	380	0.60 ± 0.02

rate of vertical spread in the downward direction, as shown in Table VIII. Because these rates do not appear to be affected by buoyant convection, they should be applicable to the gravity-free environment. Much speculation has appeared on the nature of combustion in the absence of a gravitational field, but adequate experimental observations are lacking. It has been suggested that a fire would be self-extinguishing under gravity-free conditions, smothering its own combustion products. This suggestion is based on brief observations of point flames such as a candle or cigarette lighter where fuel is brought to a fixed flame holder. The latter does not provide a suitable model for a fire in a spacecraft where flame propagation over a fuel surface would move toward fresh oxygen supplies.

TABLE VIII. COMPARISON OF HORIZONTAL AND VERTICAL FLAME SPREAD RATES

	Flame Spread Rate (in /sec)	
	Range	Average
Vertical position -- ignition on top		
Inhibited flat strip (0.5 in. wide)	0.054 - 0.055	0.054
Horizontal position -- ignition in center		
Flat sheet (2 by 2 in.)	0.047 - 0.050	0.048
Flat sheet (4 by 4 in.)	0.053 - 0.054	0.054

NOTE: Material was 1/8-in. cellulose acetate sheet

It is also unlikely that spacecraft atmosphere would be free of forced convection currents, even in the absence of buoyant convection. Thus, there is no reason to believe that the fire hazard would be less severe in a gravity-free environment than in the normal gravitational field.

The following conclusions are offered concerning the nature of the fire hazard in spacecraft:

1. The fire hazard will be greatly enhanced because of the use of an oxygen-enriched atmosphere, the conditions of confinement, and the vulnerability of vital operating systems to fire damage.

2. The possibility of a fire occurring can be reduced but not eliminated by careful attention to design and the selection of materials. Noncombustible substitutes for certain essential materials must be developed.

3. The presence of an inert gas diluent will not reduce significantly the probability of a fire occurring.

4. The presence of an inert gas diluent will reduce the rate of flame spread and fire buildup, thus providing more time for combating a fire. Nitrogen will be more effective than helium for this purpose.

5. No proven system for combating fires in spacecraft is available at the present time.

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2. Huggett, C.; von Elbe, G.; and Haggerty, W.: The Combustibility of Materials in Oxygen-Helium and Oxygen-Nitrogen Atmospheres. SAM-TR-66-85, December 1966.

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LUNEX II: A SIMULATED EIGHTEEN DAY MANNED LUNAR MISSION

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INTRODUCTION

In earlier studies [1] a series of simulations was conducted using a variable-volume simulator to establish preliminary cabin free-volume design criteria for lunar surface vehicles. Using performance and physiological measures [1, 2, 3, 4] in evaluating vehicle interior volumes during a series of three-hour, ten-hour and 72-hour simulations, a minimum free volume was determined which did not seriously compromise either subject performance or physical well being. The data, obtained on two-man crews performing representative lunar mission segments, provided critical design data for overall size, weight and translunar stowage space required for a lunar surface vehicle. These parameters in turn affect the amount of vehicle space available for scientific equipment and lunar surface sample collection.

The LUNEX II was designed to simulate a lunar mobile laboratory, one of several concepts NASA is evaluating for manned lunar exploration. This lunar vehicle is intended to house a two-man crew in a shirt-sleeve environment for a nominal 14-day lunar mission with a maximum staytime of 21 days. During this mission the crew would conduct a scientific lunar exploration requiring extravehicular excursions in pressurized suits on the lunar surface. Crew members would also perform on-board scientific system system monitoring, and routing housekeeping tasks samples. In addition, vehicle driving, navigation, system monitoring and routine housekeeping tasks would need to be performed. The stationary LUNEX II simulator housed two test subjects for a full-term 14- to 21-day lunar surface mission requiring the performance of tasks related to the lunar mobile laboratory concept. No attempt was made to provide a completely closed ecological system or to simulate unusual environmental conditions.

The test subjects were highly motivated NASA engineer-scientists. The LUNEX II simulator cabin interior incorporated the previously determined minimal cabin volumes into an integrated workspace design as an initial approach to establishing realistic design criteria.

The principal purposes of this simulation were:

1. To validate the results of previous short-term studies by using a vehicle of minimal volume and by evaluating crew performance during a two- to three-week simulated lunar surface mission by means of behavioral and physiological tests.

2. To develop and validate manned system design criteria for lunar surface cabin interiors.

Shirt-sleeve, ventilated, and pressurized suit conditions were evaluated in the context of simulated normal and emergency activities.

This paper covers the data presented at the Fifth Annual Meeting of the Working Group on Extraterrestrial Resources, Huntsville, Alabama, on March 1 - 3, 1967. More comprehensive reports can be obtained from the interim technical documents previously published. (Man System Criteria for Extraterrestrial Roving Vehicles, Phase IB - The LUNEX II Simulation, 12504-ITR2; LUNEX II: A Study on Manned Lunar Exploration, presented at the Seventeenth International Astronautical Congress, Madrid, Spain, October, 1966.)

STUDY METHODS

The LUNEX II simulator is divided into two compartments -- a driving and workspace/living area having 3.26 cubic meters (115.3 cubic feet) of free workspace and an airlock designed so that, under normal mission conditions, its free workspace volume is 1.36 cubic meters (48 cubic feet). The airlock provides 1.86 cubic meters (65.9 cubic feet) during emergency conditions requiring pressure suit use. (The airlock was designed so that under emergency conditions requiring the use of inflated pressure suits the suit and backpack storage space in the airlock would be available for use). The driving and workspace/living area provides 1.63 cubic meters (57.6 cubic feet) per man. The two cabin compartments have a total free volume under normal conditions of 4.61 cubic meters (163.3 cubic feet) with an additional 3.96 cubic meters (140.3 cubic feet) of equipment/storage space. These volumes are summarized in Table I. The crew/cabin space is designed to provide the crew with maximum free work space in the seated position. An unrestricted standing position was not provided because the ceiling height was only 166 centimeters (65.4 inches) which is 90 percent of the standing height of a 95th percentile man [5]. The LUNEX II is basically a cylindrical plywood/masonite structure having an inside diameter of 2.1 meters (7 feet) and a floor area of 2.47 square meters (26.6 square feet), as shown in Figure 1.

Because the simulation was concerned only with cabin interior space habitated by the test subjects and the space necessary for storing scientific and life support equipment used daily, the unused spaces below the floor and above the ceiling were omitted. Assuming a completely cylindrical vehicle, the entire body would have a volume of 11.08 cubic meters (392.9 cubic feet), with 2.51 cubic meters (89.3 cubic feet) independent of the crew occupied area allotted for vehicle system equipment and accessory gear not directly associated with crew/task activities.

A driving station occupied the front of the vehicle and was used for simulated driving and monitoring tasks. Part of this area was used during sleep periods. Two eccentrically pivoted chairs were located in the driving station, either of them capable of being turned 180 degrees to form a seat usable in the central workspace station. Stowage space was provided under and to the outside of each chair. Sliding writing boards and storable arm rests were available to each subject when facing forward in the driving seats.

The workspace/living area served as the primary area for performing scientific tasks, preparing and consuming meals, and sleeping. Workspace was available on each side of the center aisle. A stowable work surface extended across the center aisle accessible to both subjects. A small traveling stool could be pulled from its wall stowage space for sitting in the center aisle area. The maximum standing height in the center aisle was 166 centimeters (65.4 inches). This area served as the sleeping space when a recessed upper bunk was pulled out.

TABLE I. LUNEX II VOLUMES

<table>
<tr><th>Duty Station</th><th colspan="2">Free Volume*</th><th colspan="2">Storage/Equipment Space**</th><th>Totals</th></tr>
<tr><td>Driving Area</td><td>1.70m³
(60.2 ft³)</td><td>Subtotal
3.26 m³
(115.3 ft³)</td><td>0.72m³
(25.4 ft³)</td><td>Subtotal
2.79m³
(98.5 ft³)</td><td rowspan="2">6.05m³
(213.8 ft³)</td></tr>
<tr><td>Work Space/Living Area</td><td>1.56 m³
(55.1 ft³)</td><td></td><td>2.07m³
(73.1 ft³)</td><td></td></tr>
<tr><td>Airlock***</td><td></td><td>1.36m³
(48.0 ft³)</td><td></td><td>1.18m³
(41.9 ft³)</td><td>2.54m³
(89.9 ft³)</td></tr>
<tr><td>Totals</td><td></td><td>4.62m³
163.3 ft³)</td><td></td><td>3.97m³
140.4 ft³)</td><td>8.59m³
(303.7 ft³)</td></tr>
</table>

* These volumes represent the actual Lunex II free volume--excluding all irregular projections and including all space accessible to the crew with the exception of storage space.

** These volumes represent space occupied by the equipment inside the Lunex II (such as the chairs) and all interior space usable for equipment storage.

*** Under emergency conditions the airlock volume could be increased to 1.86 cubic meters (65.9 ft³) with the airlock storage space being reduced accordingly to 0.68 cubic meters (24.0 ft³).

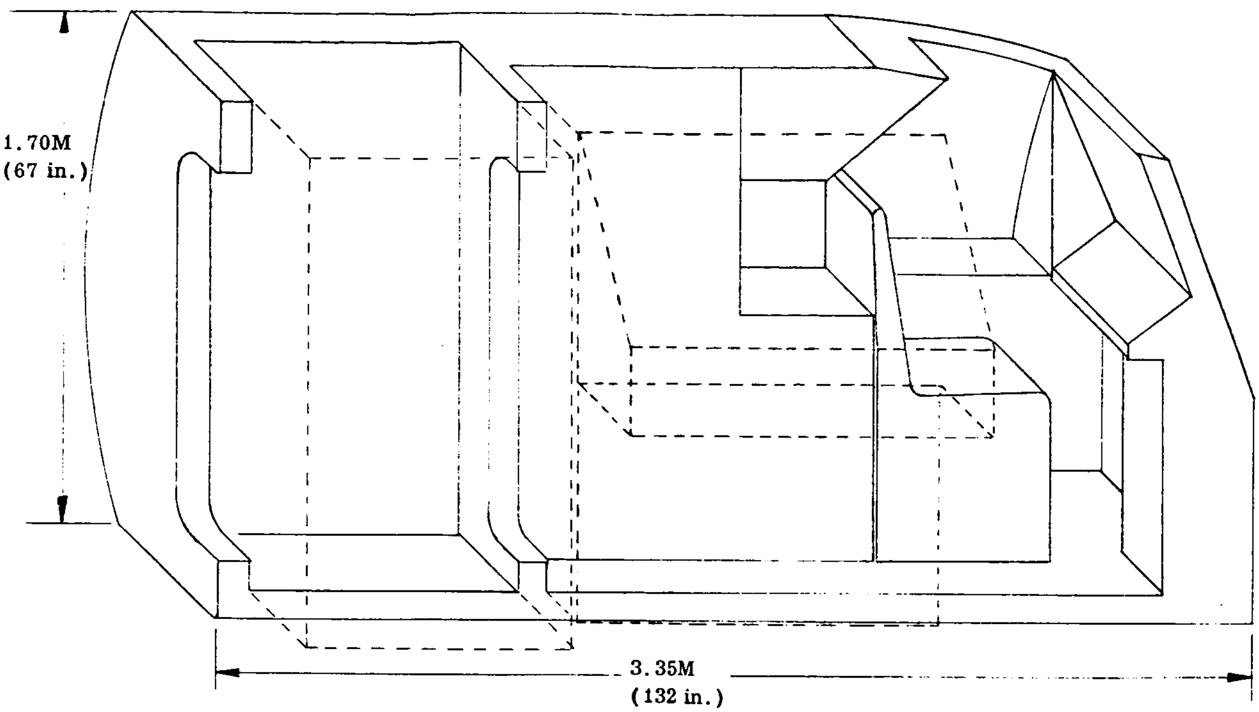

FIGURE 1. INTEGRATED WORK SPACE LAYOUT BASED ON MINIMUM VOLUMES

The airlock provided space for pressure suit donning and doffing and, in addition, personal waste elimination was accomplished in special facilities located in the airlock. Urine and fecal wastes were passed by a sliding compartment to outside experimenters for analysis. A double-acting door (55.8 × 127 centimeters), capable of opening either into the airlock or the cabin, separated the airlock from the living/work-space area. The outer airlock door (76.2 × 127 centimeters) opened to the exterior of the vehicle. The airlock hatches were opened and secured by 12-inch handwheels with 55-degree travel to the locked position and 120-degree travel to the unlocked position (that is, 65 degrees of overthrow). A constant 12-inch-pound torque was required to operate the outer airlock hatch and six inch-pounds were required to operate the inner hatch.

Subjects

Two experienced engineer-scientists from the NASA Marshall Space Flight Center served as subjects. They were Michael J. Vaccaro, designated as commander during the simulation and hereafter referred to as Operator 1 and Haydon Y. Grubbs, co-commander, hereafter referred to as Operator 2. Both men are vitally active in studies associated with lunar surface missions and were highly motivated test subjects. Operator 2 was a senior pilot with approximately 1000 hours of jet flying time. Both were experienced in simulation studies and in the background and purpose of this particular lunar surface vehicle simulation. Each subject had had extensive experience in the use of various NASA pressure suits and support equipment.

Task Descriptions

The following tasks (Fig. 2) were among those performed during the study.

Driving Tasks. - The driving task was a pursuit tracking problem displayed on a dual-beam oscilloscope. The display presented two spaced vertical lines, representing a "road path," which oscillated horizontally as they were driven by three sine-wave function generators in parallel. The subject controlled a small dot on the scope by means of side-to-side

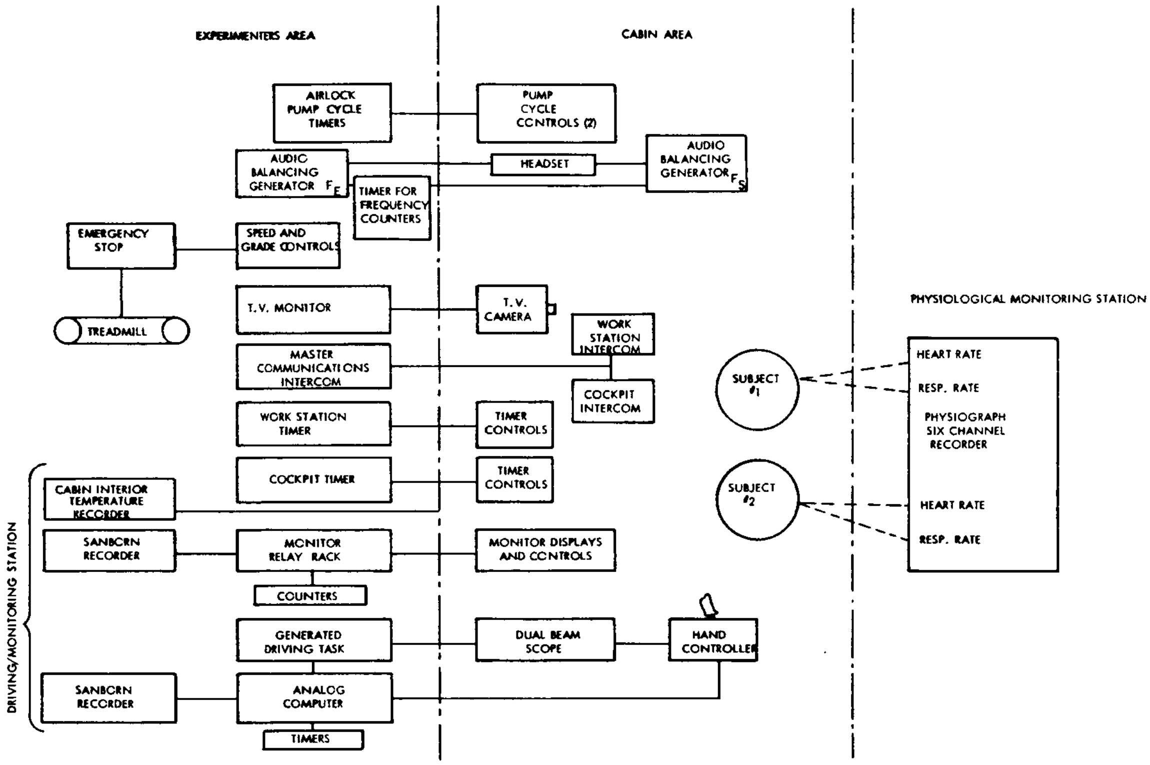

FIGURE 2. INTERRELATIONSHIP OF SIMULATION SUPPORT EQUIPMENT

motions of a modified developmental Apollo side stick control having two degrees of freedom in a single axis. The subject's task was to keep the dot centered on the simulated "path." Subject errors were recorded by an analog computer as the absolute integral error resulting when the dot was outside the roadway. Total integral error was also displayed on a Sanborn recorder. Based on pre-simulation evaluations, three error categories were derived relating the integral error to location on the navigation map. Subject time-off-course was recorded directly by means of a clock timer. Each driving task consisted of eight 5-minute presentations. Two frequencies were displayed to the subject, the order of their presentation during eight driving subtasks being systematically varied.

Monitoring Tasks. - While one operator was driving, the other operator performed monitoring tasks. These tasks consisted of: (1) monitoring the driver's time "off course" by responding to a push button which was lighted whenever the driver left the "roadway" and (2) an associated change-no-change pattern recognition problem. The latter problem presented various light patterns on a nine-light panel, the operator being required to scan the pattern for a "change" or "no-change," and then make the appropriate push-button response. This task represented monitoring of vehicle subsystems.

Two monitoring tasks were time-shared. Off-course time monitoring error, Δt, was recorded as the difference between the time the monitor's switch was depressed, t_m, and the time the driver was off course, t_o, per five minute driving task:

$$\Delta t = t_m - t_o .$$

This particular recorded parameter warrants a brief description of the procedure used in handling the data for this task. Δt has a finite lower limit because of the monitor's reaction time. Thus t_m is always greater than t_o provided the monitor does not release the switch while the driver is still off-course; t_o has a maximum upper limit of 300 seconds for the worst possible driving condition (that is, if the driver should be off course the full five-minute driving period). To account for variations in driving proficiences, the monitoring error was considered as a percentage:

$$\text{Percentage of off-course time} = \frac{\Delta t}{t_o} \; 100 .$$

Considered in this manner, it is obvious that a 100 percent monitoring error means that the monitor switch was depressed twice as long as the time the driver was off course, that is if:

$$\frac{\Delta t}{t_o} = \frac{t_m - t_o}{t_o} = 1, \text{ then } t_m = 2t_o .$$

Because t_o has a possible lower limit of zero (the driver never being off course), the ratio, $\frac{\Delta t}{t_o}$, approaches infinity for the perfect driving condition for any finite monitoring time, t_m. These conditions were observed in the data as is noted in the interim reports associated with this study.

Navigation Task. - The subjects were provided with maps of the intended lunar traverse for the 14- to 21-day mission. As the mission progressed, the subjects' position was determined by the experimenters as a function of driving task performance. The experimenters kept track of the vehicle on a large master map. The position of the vehicle in relation to various map features was determined by the experimenters in terms of lines-of-bearing. Simple triangular objects representing the lunar terrain features were mounted at the appropriate angles on the rim of a large disc mounted on the top of the LUNEX II simulator. The subjects then performed a navigation task to determine their position. The navigation task was a time-shared task. The subjects initiated the task by starting a timer and indicated task completion by switching the timer off. A war surplus panoramic periscope had to be mounted through the ceiling in the center of the disc to perform the sightings and stowed at the completion of each sighting task. The subjects' task was to set up the periscope and determine the angles to the appropriate terrain features. The accuracy of the subjects' angular measurements was evaluated. Based on these angular measurements, the subjects "fixed" their location and transmitted their position in map reference coordinates to the experimenters. The "charting error" was the difference between their determined location and the actual location.

Audio Balancing. - To observe the subjects' response to stimuli other than visual and tactile, the subjects were required to balance a Wheatstone bridge to four-decimal-place accuracy by audio means. This task simulates a physical chemical procedure commonly used to determine the ionic resistivity of unknown solutions (in this case, an imaginary lunar dust sample in solution) by comparing the sample resistivity with known standards.

A stereo headset connected to two variable-audio-frequency generators (one earphone to each generator) was used by the subject to compare and match two audio stimuli. Digital counters were used for accurate frequency recording. The frequencies presented to the subject were selected from a range of 800 to 100 cps, divided into 10 approximately equivalent intevals. Each frequency was presented at an intensity of 70dB once per experiment in a randomized order.

Sample Measurement. - Precise scientific work can be expected of astronauts on a lunar mission. To simulate such precise work, the subjects were required to measure the outside diameter of optically calibrated aluminum discs, using a traveling microscope. The accuracy and time of the measurement were recorded. Accuracy to four decimal places was required of the subjects.

Geophysical Tasks. - The University of Minnesota Geophysics Department provided a variety of tasks designed to represent the type of geological activity expected on early lunar missions. These tasks were not intended to imitate actual lunar mission tasks but to represent simple yet realistic geological activities. The tasks required the collection of as many different rock and mineral categories as possible, retaining only one sample of each category. The tasks further required that the traverse of the lunar mission be accurately mapped with respect to prominent known terrain features and that new terrain features be mapped as they are observed. The careful collection of rock and mineral samples so that the scientific value of the return payload is maximized while minimizing its weight involves both macroscopic and microscopic analysis. The macroscopic analysis began with the subject in the inflated suit making judicious rock sample collections during his extravehicular activity. Nearly 100 rock samples, representative of the types geologists expect to find on the moon, were located in the area outside the LUNEX II simulator. The subjects' task was to visually sort the samples and select 20 rocks appearing to represent discrete categories. These 20 samples were returned to the LUNEX II where analysis of the samples was performed. In each of the 20 samples, there was no more than eight categories. The subjects' task was to determine which eight should be returned to Earth. Microscopic analysis of mineral samples required the analysis of mineral grains by use of a binocular microscope and a polarizing microscope. Mineral crystals requiring sorting by category and by properties were provided. These crystals had to be identified by sampling and microscopic counting. The minerals mounted on petrographic slides required analysis by the determination of anisotropic, isotropic, and opaque light transferring properties using the polarizing microscope. The subject's task was to sort crystals and to distinguish distinct mineral samples mounted on petrographic slides.

In association with the navigation tasks, specific terrain features were revealed at irregular times and had to be located on the subjects' map. For example, a particular mountain was shown during one navigation period. The angle to the mountain was noted and a ray drawn from the LUNEX II location through the mountain location. After several driving periods, the same mountain was again shown and a ray drawn. The intersection of the rays located the new terrain feature on the subjects' map. In this way, single-dimension plotting of simulated lunar terrain features was possible.

Physiological Measures

In addition to evaluating crew performance by integrated behavioral and psychophysiological tasks, selected physiological factors were evaluated. The maximum oxygen consumption measurements (obtained by requiring the subjects to work on a treadmill during indirect calorimetry) were performed by the University of Minnesota Laboratory of Physiological Hygiene. These formed the basis for evaluating relative task work loads using calibration curves relating oxygen consumption to heart rate and respiratory rate derived for each subject during the presimulation period. Heart rate and respiratory rate were monitored continuously during the day using a two-lead telemetry system and a polygraph. Mean heart and respiratory rates were obtained during performance of tasks. These rates, when considered as a percent of each subject's maximum steady-state rate obtained during maximum oxygen determinations, permit comparison of relative task work loads [4]. A measurement of each subject's maximum oxygen consumption immediately after the simulation provided a direct evaluation of physical fitness changes in the subjects during the simulation.

Urine samples were analyzed for 17-ketosteroids and 17-hydroxycorticosterone (cortisol) as well as for glucose, ketones, pH and protein. Body weight and food and water consumption were recorded throughout the experiment as well as urine volumes and the wet and dry weight of the feces. The temperature of the simulator cabin, as well as the intake and output temperature of suits when worn, was also monitored.

Simulation Time Lines

The initial crew time line was derived from previous studies [1] and from examining MOLAB time lines generated in related studies. The simulation activities were initially guided by the 24-hour repeating activity sequence. The subjects alternated between sequence A and B on a daily basis. These activities fully occupied a 16-hour workday. As the simulation progressed, this activity sequence evolved to an operational time line with experimentally determined task activity time constraints. Based on this evolution and on the mission goals to accomplish maximal scientific exploration and optimal and efficient scientific analysis and sample collection, a functional simulation time line was established. The meals and sleep periods were held constant through the simulation.

CONCLUSIONS

This study has demonstrated that careful workspace layouts can make a small vehicle volume habitable and functional using the tasks simulated. The results obtained can be used in the definition of operational workspace and stowage areas for the use and stowage of actual mission hardware.

1. A lunar surface vehicle with a cabin free volume of 3.26 cubic meters (115.3 cubic feet), a nominal airlock volume of 1.36 cubic meters (48 cubic feet), and a ceiling height of 166 centimeters (65.4 inches) is adequate to house two men (47 to 93 percentile with respect to height) performing simulated lunar surface mission tasks for 18 days.

2. Simulated driving, monitoring, navigating, sample measurement, and audio balancing tasks could be performed throughout the simulation with satisfactory accuracy and no adverse trends. No unusual differences between the two subjects' performance levels were observed.

3. Realistic geophysical tasks could be successfully performed in the simulator by subjects relatively untrained in geology.

4. Performance during emergencies indicated that:

   a. Power assistance is required for the rescue of a simulated totally disabled crew member.

   b. Emergency procedures are critical to mission success. Further study of the emergency procedures and techniques is recommended.

   c. An airlock having a volume of 1.86 cubic meters (65.9 cubic feet) and a ceiling height of 166 centimeters (65.4 inches) will adequately contain two operators in inflated pressure suits, one operator being totally immobilized. Furthermore, an airlock of these dimensions permits two crew members to simultaneously don pressure suits and backpacks and inflate their Apollo state-of-the-art pressure suits.

   d. The cabin volume, driving station volumes, and chair locations were adequate for operating in the inflated pressure suit condition using the cabin air supply system. Performance of the driving and monitoring tasks while wearing suits deteriorated.

   e. A 55.8 by 127 centimeter (22 × 50 inch) inner hatch space could not accommodate subjects wearing pressurized suits and backpacks.

   f. A single crew member could successfully perform simulated tasks within the minimum cabin volume for at least seven hours even if one crew member was temporarily disabled.

5. Task time-line analyses suggest that to increase the time available for inside scientific tasks, no more than three meals a day should be required. Time is also saved if subjects simultaneously don and doff pressure suits and exchange inside-and-outside vehicle tasks.

6. It is recommended that extravehicular tasks be alternated by day with inside tasks to permit maximum task performance efficiency.

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SPACE CABIN TOXICOLOGY

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INTRODUCTION

Man's dependence on the earthly atmosphere requires careful regulation of the atmospheric environment in sealed cabins or extraterrestrial shelters [1]. Contaminants that can be generated either by construction materials, equipment, weapons, or crew can constitute a major limiting factor on habitability and mission duration.

The factors that can aggravate the contaminant concentration are far more numerous than those which can alleviate the problem (Table I). The limited volume of usable atmosphere in space systems allows for very little latitude in air pollution control. Air purification and life support equipment are being heavily taxed with increasing mission profile and can, per se, change the total contaminant picture by incomplete processing of toxic materials [2, 3].

TABLE I. IMPORTANT FACTORS INFLUENCING ATMOSPHERIC CONTAMINATION

Aggravating	Beneficial
Continuous Generation and Exposure	Leak Rate of Cabin
[1]Reduced Pressure	Materials Selection
[1]Volume/Man Ratio	Preconditioning of Materials
[1]Power and Weight Limitation	
Filter Characteristics	
Complexity of Contaminants	
[1]Multi-Stress Environment	
[1]Escape Lead Time	

[1]Not significant in nuclear submarines

The state-of-the-art in environment toxicology does not allow valid predictions of human tolerance to any toxic materials for prolonged continuous exposure [4]. Moreover, the bizarre mixture of any contaminants always carries the threat of potentiation of toxic effect by individual constituents within the mixture [5]. Exotic environments such as low barometric pressure, oxygen-rich atmosphere and the multitude of physiological stresses are still unknown quantities which can have a profound influence upon man's resistance to chemical insults. Similar to our air pollution problems on Earth, freak coincidences of relatively harmless factors could lead to severe biological embarrassment. It is also quite obvious that the problems on nuclear submarines are far less serious than in spacecraft environment [2]. Finally, truly uninterrupted 90-day continuous exposure in ambient pressure air to contaminant concentrations not exceeding the Threshold Limit Value (TLV) has resulted in 100 percent mortality of animals with certain chemicals such as hydrazine, decarborane, etc. [6], as shown in Table II.

TABLE II. SUMMARY OF MORTALITY RATES DURING 90-DAY CONTINUOUS EXPOSURE

Compound[1]	Monkeys		Rats		Mice	
	# Dead / # Used	% Dead	# Dead / # Used	% Dead	# Dead / # Used	% Dead
N_2H_4	2/10	20	48/50	96	98/100	98
UDMH	1/10	10	3/50	6	6/100	6
NO_2	0/10	0	9/50	18	13/100	13
$B_{10}H_{14}$	6/10	60	25/50	50	82/100	82
Controls	1/10	10	0/50	0	0/100	1
CC_{14}	1/10	10	0/50	0	0/100	0
Phenol	0/10	0	0/50	0	0/100	0
Indole	2/10	20	5/50	10	22/100	22
H_2S	0/10	0	12/50	24	26/100	26
Me·SH	4/10	40	5/50	10	43/100	43
Mixture[2]	16/20	80	32/50	64	99/100	99
Controls	0/19	0	2/50	4	38/200	19

[1]At TLV Concentrations
[2]Indole, H_2S, Me·SH, and Skatole

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RESEARCH FACILITIES

A unique inhalation exposure facility has been built at the 6570th Aerospace Medical Research Laboratories to study the effects of low atmospheric pressure and oxygen-rich atmospheres on the characteristics of truly uninterrupted long-term exposure to toxic gases and vapors that are encountered in the atmospheres of space cabins. This facility became operational in September 1964 and consists of four dome-shaped altitude chambers with the necessary air lock, contaminant feed, and drainage systems to facilitate continuous exposure of a large number of various species of animals without interferring with either the pressure or contaminant concentration.

These "Thomas Domes" were conceptually designed by the author [7] and were engineered, installed and are currently operated by the Aerojet-General Corporation under a contract with the Air Force. One of the desirable features of this facility, usually not encountered in other exposure chambers, is the capability to simulate environmental parameters that are peculiar to space cabin atmospheres. Within the four-fold structural safety factor these domes can be operated with a pressure differential of 10 psi (from one to one-third atmosphere) and with either single-gas, oxygen, or mixed gas, oxygen-nitrogen or oxygen-helium, environments. For both safety considerations and versatility, the upper portion of the dome is mated to the lower portion by an O-ring arrangement, thus providing both a quick escape capability for the operators in case of a fire, and an unrestricted access during the loading of animals, cages, and life support and specialized test equipment before and after experiments. An operating console continuously records oxygen flow rate, total pressure, temperature, and humidity within each dome.

The middle section of the dome consists of one-inch thick, shatter-proof glass paneling that assures unobstructed visual and photographic observation of the experimental subjects. The black ceiling, walls and floors facilitate both photographic work, through elimination of reflectance, and sensory deprivation studies. The facility is highly automated and can be operated by a single technician during the night-time hours. At the beginning of each experiment, the basic parameters such as contaminant flow, temperature, humidity, and mixed versus single gas atmosphere are adjusted in the basement of the facility, and from then on the entire operation is automatically controlled. The control console has manual override features and both visual and audible alarm systems.

During exposure studies, chamber technicians enter the dome once daily through a vertical airlock system to clean cages, feed animals and obtain biological samples. Wastes are hosed down through a special drainage system that can handle the pressure differentials involved. There is also a water spray ring which can be activated from the outside to prevent accumulation of excreta during the 24-hour cycles. All dead animals are removed promptly and necropsied immediately.

A liquid oxygen storage cylinder with 68 000-pound capacity supplies gaseous oxygen to the facility through a converter. All crucial support equipment, such as air conditioning units, chillers, oxygen heaters, vacuum pumps, compressors for instrument air, etc., are redundant and, in case of a general power outage, a 160 kw diesel generator can supply the necessary ac power requirements.

The safety and occupational medicine aspects of this system require an altitude indoctrination training course for all chamber operators and other scientific personnel who must enter the altitude exposure chambers. Chamber technicians who must remain at altitude for prolonged intervals to perform the considerable physical work of lifting feed bags, cleaning cages, collecting biological samples, etc., are denitrogenated for a 60-minute period by breathing 100 percent oxygen to prevent occurrence of the "bends." A 30-minute period has been found satisfactory for the shorter entries required to remove dead animals. No operator may make more than four entries within a single 24-hour period and after each day of entry, he may not re-enter for another 24-hours. During approximately three years of operation, there have been no serious medical problems because of altitude exposure.

With the oxygen-rich environment, safety considerations are primarily governed by the increased fire hazard. Automatic and manually operated water deluge and CO_2 fire extinguishing systems in each dome can be activated by either the console operator or the technician within the dome. Whenever technicians enter the dome, an overhead crane is engaged at the hook-lift portion of the dome top, and should an emergency arise, the dome can be repressurized by a three-inch dump valve within 15 seconds. After pressure equalization, the top section can be lifted to facilitate rapid removal of personnel or animals. This dump valve is redundant and can be operated from either outside or inside the dome.

A triple-safe communication system includes a hard-line transistorized intercom using standard Air Force headgear and lip contact microphone, a hard-line system with amplifier and loud speaker system to alert the inside operator to any emergencies of which he may not be aware, and a wireless "walkie-talkie." All three systems are battery operated, as are the emergency lights which are automatically activated by a general power failure.

To avoid possible sources of fire, all fixtures are of explosion proof design, and 24 pairs of shielded cable are available for transporting biological signals to physiological instrumentation recorders located outside the domes.

RESEARCH OBJECTIVES

The mission of the facility is to provide both fundamental experimental capabilities in space cabin toxicological qualification of space cabin materials. Because there is mutual Air Force and NASA interest in these areas, the toxicological information generated serves both military and civilian space requirements. A cost sharing Air Force-NASA research effort is currently exploring the following fundamental and practical questions:

1. Does a 5 psia oxygen rich atmosphere cause pulmonary irritation or functional impairment during long-term continuous exposure?
2. Will such an atmosphere influence or modify tolerance to atmospheric contaminants?
3. Can cabin materials be screened for potential toxic effects in a practical and timely fashion?

EXPERIMENTAL DESIGN

To answer the basic questions, three major groups of animal experiments were conducted in the past two years. Group I experiments consisted of an eight-month continuous exposure of large numbers of various species to five psia 100 percent oxygen and to five psia 70 percent oxygen 30 percent nitrogen basic cabin atmospheres. In addition, 90-day continuous exposure of 12 trained primates working on performance tasks was also performed with both types of cabin atmosphere.

Group II experiments consisted of comparative toxicity studies on pulmonary irritants and systemic poisons. Large numbers of various species were continuously exposed for two weeks to graded high concentrations of such agents and for 90 days to their representative industrial Threshold Limit Value (TLV) concentrations in both ambient air and in various cabin atmospheres.

Group III experiments consisted of analytical characterization of volatile contaminants generated by various cabin materials and continuous exposure of rats and mice for one week and 60-day duration to such gas-off products in a five psia 100 percent oxygen environment.

In the following discussion of results, these experimental groups will be treated as separate entries.

RESULTS

Group I Experiments

Group I experiments were concerned with fundamental cabin atmospheres. The basic criteria for environmental parameters, animal loads and evaluation of toxicity are condensed in Table III. While the eight-month experiments represent a classical toxicological approach, the three-month study is solely for the evaluation of animal performance. Those enzymes studied in the classical experiments were transaminases, lactic dehydrogenase, and alkaline phosphatase.

Controls for these experiments consisted of similar numbers of animals held at ambient air conditions in the animal holding facilities with the exception of the 12 trained primates which served as their own controls on performance measurements.

A number of ancillary experiments were also conducted to serve as "positive controls." These consisted of continuous exposure to pure oxygen atmospheres at ambient pressure and gradually reduced pressure (760, 725, 695, 650 and 600 mm Hg), and utilized four monkeys (Macaca mulatta) per group [8]. These animals furnished the classical morphological and clinical laboratory baseline materials for oxygen toxicity criteria. Again for the sake of positive biochemistry controls, large groups of rats were exposed to pure oxygen at ambient pressure and mitochondrial enzymes, and morphology was studied [9, 10, 11, 12, 13, and 14].

TABLE III. CRITERIA FOR GROUP I EXPERIMENTS

Exposure:	Continuous	
Atmospheric Composition:	Single Gas Oxygen or Mixed Gas ($68\%\ O_2$ - $32\%\ N_2$)	
Pressure:	5 psia	
Temperature:	72° F	
Relative Humidity:	50% (corrected for altitude)	
Gas Flow:	Dynamic (no recirculation)	
Animal Load:	8-Month Study	3-Month Study
	4 Monkeys 8 Beagles 65 Rats 40 Mice	Performance Testing of 12 Monkeys
Laboratory Procedures:	Hematology Blood Electrolytes Proteins (Serum) Enzymes and Isoenzymes	Hematology Blood Electrolytes Proteins (Serum) Enzymes and Isoenzymes
Behavior:	Spontaneous Activity	Continuous and Discrete Avoidance Tests
Growth Rate:	Pre- and Postexposure	Pre- and Postexposure
Pathology:	Gross and Histopathology	

Both single and mixed gas fundamental five psia atmospheres appear to be well qualified by animal toxicity tests for long mission durations. In general there was a paucity of abnormal clinical findings with both. The five psia pure oxygen atmosphere did produce some morphological changes in rat liver and kidney mitochondria, but these changes were most impressive at the end of the first week of exposure and then tended to return to normal past 90 days of exposure. Therefore, these are definitely reversible and, most likely, adaptive changes. In addition, the significance of these ultrastructural aberrations is as yet unclear. No mitochondrial enzyme changes could be detected with the eight-month exposure to five psia oxygen; neither was there an uncoupling of oxidative phosphorylation in the mitochondria, a highly characteristic effect of oxygen at ambient pressure in positive control studies. Final statistical analysis of all clinical laboratory and pulmonary morphometric electron microscopy data has revealed a complete absence of any significant deviation from the normal baseline.

The eight-month exposure to the five psia mixed gas atmosphere was completed on January 3, 1967, and the pathology reports are not yet available. The only obvious abnormality in clinical laboratory results which can be seen on first glance is an inverted albumin-globulin ratio in several beagle dogs. The significance of this finding has yet to be determined. On several occasions, some control dogs exhibited a downward trend on this test, but the mean average A/G ratio in control beagles stayed well within the normal limits.

The trained primates have performed well in both five psia oxygen and mixed gas environments for the 90-day test duration.

To summarize the conclusions from Group I experiments:

1. By extrapolation from the results of the eight-month animal exposures in this laboratory and interpretation of available information from shorter duration human experience in life support simulators [15], it appears now that both fundamental atmospheres are suitable to maintain life and preserve performance on extended missions of several months duration.

2. Strictly based on animal data, the 5 psia oxygen atmosphere has shown no ill effects

whatsoever, and all the parameters studied, including subcellular morphology and mitochondrial chemistry, are within normal limits at the end of the eight-month exposure.

3. There is some uncertainty about the significance of the inverted A/G ratios in dogs observed in the eight-month mixed gas study. Final evaluation will be possible only after thorough analysis of histopathology and electron microscopy data. It may be necessary to repeat this experiment to verify the reproducibility of this change. Fortunately, the trend toward an inverted A/G ratio has shown up as early as three months into the prolonged exposure, and this may shorten the length of the second experiment.

4. Last but not least, these animal experiments have demonstrated their great value in conjunction with the manned simulator experiments. Complete absence of pathology in the animal biopsy and necropsy materials tremendously increases the confidence in extrapolation of tolerance from animals to humans.

Group II Experiments

Group II experiments encompassed the comparative toxicity of pulmonary irritants and systemic poisons at ambient air versus cabin atmosphere and the study of cumulative effects from continuous exposure.

The basic animal loading criteria and evaluation of toxicity are similar to those used in Group I experiments previously noted in Table III.

The toxic agents were chosen for the following considerations:

1. There was abundant information on toxicity of repeated daily eight-hour exposure in both humans and animals.

2. The mode of action was well defined; for example, nitrogen dioxide represented a pulmonary irritant without systemic effects; ozone was known to cause both pulmonary irritation and systemic effects; and carbon tetrachloride represented no pulmonary irritant properties but was regarded purely as a systemic poison at the concentrations employed.

3. Since the TLV values for industrial exposure were well established and proven by occupational medicine experience, study of these agents could answer two basic questions: (1) how will oxygen enriched atmosphere influence toxicity during continuous exposure and (2) what is the magnitude of cumulative effects from continuous exposure?

Although these studies are not yet complete, Tables IV and V depict both the accomplished and planned comparative toxicity studies. As in Group I experiments, 90-day performance evaluation of trained primates was also included here, but only at the TLV concentration to prevent loss of these expensive trained animals.

TABLE IV. EXPOSURE PLAN, COMPARATIVE TWO-WEEK CONTINUOUS EXPOSURE STUDIES

Contaminant (mg/M³)	Ambient Pressure Air pO_2 = 148	Ambient Pressure Air O_2 Enriched pO_2 = 260	5 psia 68% O_2 32% N_2 pO_2 = 148	5 psia 100% O_2 pO_2 = 260
NO_2 (38)	X	X	X	X
(81)	X			X
O_3 (8)	X	X	X	X
(15)	X			X
CCl_4 (35)	X			X
(576)	X			X

Note: X denotes completed studies

These high concentration exposures were primarily designed to study the influence of these atmospheric mixtures on the toxicity of NO_2, O_3 and CCl_4 The doses were picked to assure mortality or severe pathological changes during two-week continuous exposure.

TABLE V. EXPOSURE PLAN, COMPARATIVE 90-DAY STUDIES

Contaminant TLV (mg/M³)	Ambient Pressure Air	5 psia 68% O_2 - 32% N_2	5 psia 100% O_2
NO_2 (9.0)	X	-	X
O_3 (0.2)		X (12 trained primates)	X
CCl_4 (32.0)	X		X

Note: X denotes completed studies

Early in these studies when comparing data from ambient air and five psia 100 percent oxygen exposures, marked differences were noted in toxicity under these conditions. This brought up the question of whether increased oxygen concentration on decreased total pressure is the responsible influencing factor. To clarify this point, ambient pressure air was enriched with oxygen to the point where the partial pressure of oxygen (pO_2) in the mixture was equal to the pO_2 in the five psia 100 percent oxygen experiments, that is, 260 mm Hg.

It would be impossible to present a detailed analysis of all test results in this presentation. They are, however, published in the Proceedings of the First and Second Conferences on Atmospheric Contaminants in Confined Spaces [16, 17].

As a brief summary of findings from these comparative experiments, the following conclusions can be made:

1. There is a clear-cut difference in the toxicity of pulmonary irritants at high concentration under various environmental conditions. Ozone and NO_2 are most toxic in ambient pressure air and least toxic in 5 psia 100 percent oxygen atmosphere during a 2-week continuous exposure. Mixed gas, 68 percent oxygen - 32 percent nitrogen at 5 psia pressure and oxygen enriched air (pO_2 = 260 mm Hg) at ambient pressure behaved differently; NO_2 was less toxic in mixed gas atmosphere at 5 psia than in O_2 enriched ambient air, while the reverse was true with ozone. Still, both of these atmospheres seemed to reduce significantly the toxicity of pulmonary irritants when compared to plain ambient pressure air environments. This was evidenced by either the absence of mortality or prolonged survival times. Thus, it appears now that both oxygen rich and reduced pressure environments reduce the toxicity of pulmonary irritants, and a combination of both seems to have the most protective effect.

2. In the 90-day studies, the TLV concentration of CCl_4 was definitely toxic in both ambient pressure air and 5 psia oxygen atmospheres. The two-week high concentration CCl_4 studies indicated a more pronounced toxic effect in 5 psia oxygen than in ambient pressure air. In the case of ozone, one dog died on day 30 of the ambient pressure air exposure, but no animals were lost in the 5 psia oxygen environment. Even with NO_2, the TLV concentration seems to be marginally toxic under both conditions. Therefore, generally it can be concluded that 8-hour industrial TLV's are unsafe concentrations for prolonged continuous exposure. This statement does not necessarily imply that ensuing pathological changes are irreversible, but simply points out that they should not be used as design criteria for the control of atmospheric contaminants in space cabins. As a matter of interest, ozone at the TLV concentration in a 5 psia mixed gas atmosphere did not produce any performance decrement in the trained primates.

3. The most important implication of these findings is that the basic cabin atmosphere of 5 psia 100 percent oxygen is not likely to enhance the toxicity of pulmonary irritants or systemic poisons upon long confinement to the cabin if levels are kept within a safe fraction of the TLV values. What precisely this fraction should be is hard to predict; very likely 1/10 of the TLV may be safe in most instances and even oversafe in many instances. Straightforward compensatory extrapolation for the 3-fold increase in daily exposure ($8 \times 3 = 24$ hr) will not yield a sufficiently safe design factor. Ultimately, levels must be set on the basis of mission-length-oriented animal exposure studies and careful extrapolation of these results to humans. This approach is satisfactory for present systems where mission duration is not in excess of 90 days. With the advent of interplanetary missions, more efficient ways of extrapolation will be necessary to permit the use of shorter than mission length animal exposures in the prediction of safe atmospheric contaminant levels in the spacecraft. Obviously, there are not enough facilities in the country to allow the toxicologist to conduct a one-year-long animal experiment on each contaminant before a level is recommended for design purposes; neither will the engineering leadtime allow for such delays in the biological leadtime.

Group III Experiments

Group III experiments were concerned with the analytical characterization and toxicological screening of gas-off products from cabin materials.

Analytical Characterization. - There are many construction materials used in today's sophisticated spacecraft cabins and the attending environmental control or life support systems. Those groups of materials which are most likely to produce volatile contaminants are listed in Table VI.

There are more than 600 various materials used in the construction of the crew cabin. Rigid criteria have been set up for qualification of these materials before use in the Apollo and MOL systems. Again, these criteria cannot be reviewed here, but interested readers can obtain the information from the references in this paper [16, 17].

TABLE VI. VOLATILE CONTAMINANT SOURCES

Adhesives
Elastomers
Electrical Insulation
Finishes
Coatings, Paints and Varnishes
Markings, Inks
Foams
Greases, Lubricants
Moldings
Plastics and Laminates
Potting and Sealing Compounds
Thermoplastics

The analytical identification of volatile contaminants is the final phase of the chemical acceptance program. To illustrate the categories of volatile products from the previously listed groups of cabin materials, Table VII depicts both major types and the most frequent sources that were positively identified.

TABLE VII. TYPES AND SOURCES OF VOLATILE CONTAMINANTS

Major Types of Gas-Off	Typical Sources and Products
Inorganics	Paints and Coatings
Alkanes	Carbon Monoxide
Alkenes	Solvents
Alcohols	Plasticizers
Alkyl Halides	Resins
Aryl Halides	Ammonia
Benzene & Homologues	Ethylamine
Carboxylic Acids	Silicon Greases
Aldehydes	Tetrachlorobenzene
Ketones	Polyurethane Foams
Aliphatic Nitrogens	Carbon Dioxide
Silicon Compounds	Lubricants
	Chlorine Substituted Fluorocarbons (up to C_6)

To further illustrate the complexity of contaminant mixtures in the atmosphere, analysis was made of concentrated freeze-out samples from one of the Thomas Domes during the performance of the eight-month five psia 68 percent oxygen 32 percent nitrogen mixed gas study (Table VIII).

The surprising findings in this analysis are: (1) the relatively high contaminant content of the atmosphere in view of the fact that the oxygen source always meets the required purity standards as prescribed in the military specification for aviator's breathing oxygen and (2) that such accumulation occurs because we are not dealing with a recirculating system but with a 20CFM open flow design, which should prevent substantial accumulation of contaminants. Many contaminants in this mixture are obviously of biological origin because of the animal loading in the dome.

TABLE VIII. ANALYSIS OF ATMOSPHERE OF THOMAS DOME NO. 4 CONTAINING DOGS, MONKEYS, RATS, AND MICE IN FIVE PSIA 70 PERCENT O_2 30 PERCENT N_2

Compound	Level in MGS/M^3
[1]Combined Methyl and Ethyl Amines	12
[1]Methane	3
[1]Acetone	4
[1]Carbon Monoxide	2
Methylene Chloride (supply gas)	8
Benzene	0.2
Toluene	0.1
Xylene	0.2
[1]Diethyl Ketone	2
Methyl Isobutyl Ketone	0.1
Dimethyl Formamide (tentative identification)	0.004
[1]Phenol (tentative identification)	0.008
Carbon Disulfide	5
[1]Hydrogen Sulfide	5
[2]Solid Ammonium Salts - estimated	50 mg/m^3

[1]Biological origin possible
[2]Result of interaction with ammonia

Another great potential source of unexpected contaminants is malfunctioning equipment. Overheating of electrical insulation and neoprene ducting for air-conditioning have already been identified during simulator runs as real problem areas. One malfunction can trigger a chain reaction of other malfunctions resulting in a truly vicious circle and acute health hazard to the crew. This has actually occurred and has led to abortion of the Project MESA experiment (Manned Environmental Systems Assessment), and was a mystery for a long time until recently solved by R. A. Saunders of the U. S. Naval Research Laboratory, Washington, D. C. [17]. During the manned trial, because of the increased humidity of the atmosphere, water condensed in the aluminum canister containing sodium superoxide. This started a chain of events:

1. The moisture generated sodium hydroxide from the sodium superoxide.
2. Sodium hydroxide generated hydrogen in contact with aluminum.
3. To stop the steady generation of hydrogen into the atmosphere, the crew increased the flow rate through the catalytic hopcalite burner to a faster rate than specified.
4. The increased flow rate dropped the temperature of the catalyst bed, making it inefficient in combusting organic materials in the atmosphere.
5. The chamber had been cleaned before use with a relatively harmless solvent (trichloroethylene). Because of incomplete combustion in the catalytic bed, dichloroacetylene was produced. Dichloroacetylene is highly toxic and made the crew so sick they had to abort the mission on the third day.

The analytical identification and measurement of trace quantities of contaminants is a difficult and time-consuming task. Cabin materials are sealed in large glass bottles, stored in a five psia oxygen atmosphere and allowed to off-gas for periods of 30, 60 and 90 days. The atmosphere containing the volatile contaminants is then analyzed by a combination of gas chromatopography, mass spectrometry, and infrared techniques [16, 17, 18]. In the case of environmental samples collected from cabins and simulators, several steps of sample concentration by freeze-out techniques are necessary to obtain sufficient material for analysis [16].

Toxicological Screening. - Two types of studies are conducted by continuous exposure of 25 rats and 25 mice to gas-off products from cabin materials generated at 155° F in a five psia oxygen environment.

The acute exposure is of one week duration to a mixture of 10 to 12 various cabin materials; the chronic exposure is of 60-day duration, to a large mixture of about 100 to 120 materials that have been previously screened and accepted in smaller batches in the acute one-week studies.

A dome is used as an environmental envelope at five psia pressure and maintains dynamic 20 CFM flow of pure oxygen. Small animal cages housing the 25 rats and 25 mice are inserted into a closed loop recirculating life support system consisting of CO_2 removal, humidity trap, and constant temperature ovens heating the materials. The loss of oxygen converted to CO_2 by the animals is made up from the dome atmosphere by an automatic pressure-sensitive valve admitting fresh oxygen into the closed loop.

The reason for screening materials in mixtures of one to 120 at a time is two-fold. First, the tests have to be completed within the milestone schedules of system development. Secondly, employing large mixtures of materials enhances the detection of serious synergistic effects which cannot be seen if materials are screened singly.

If a mixture shows toxic effects as determined by clinical observation, depressed growth rate, or gross and histopathological abnormalities, it is systematically taken apart until the culprit material or combination of materials is found which is responsible for the toxicity. This can be a very time-consuming process, especially if the large mixture shows adverse effects, while the previously screened smaller batches do not demonstrate any activity.

Thus far we have run into only two small mixtures requiring this approach. In one instance the weight loss of animals could not be duplicated in the subsequent experiment. On the second occasion, a group of 12 materials which showed toxicity was subdivided into batches of six; one of these smaller batches again exhibited overt toxicity and was further broken down into two subgroups of three materials. At that time, a new supply of these materials was ordered from the various vendors, and the fresh supply failed to cause toxic effects. This variation between production batches of materials is not uncommon at all and has also been noted in the analytical gas-off studies. Consequently, one is faced with the frustrating task of testing a large number of production runs on the same material before the truth is found, if ever. To avoid such trials and tribulations, perhaps it may be wise to obtain, in advance of the actual hardware construction, sufficient quantity samples of the very same materials that are to be used in the cabin. Trends to accomplish this are promising. The Apollo system has established a good material screening program, and all future systems are certain to follow suit. This will lead eventually to a good catalogue of tested and accepted materials that can be relied upon not to create serious health hazards in the closed atmosphere.

HAZARD EVALUATION AND ENGINEERING CONSIDERATIONS

The crucial factors in contaminant build-up are generation rate, removal rate, equilibration time, and the maximum permissible concentration of a contaminant that can be tolerated for the total mission duration without ill effect on health or performance. Interrelation of all these factors can be expressed by the following mathematical equations.

Assuming perfect mixing, a constant contaminant generation rate, and a constant leak rate, the concentration of contaminant at any point in time can be calculated by using the following equations:

$$c = \frac{w}{b}\left[1 - e^{\frac{-(bt)}{a}}\right] \quad \text{where} \qquad (1)$$

c = contaminant concentration in mg/m^3
w = amount of contaminant generated in mg per day
b = leak rate in cubic meter per day at a particular cabin pressure
a = total effective gaseous volume in cubic meters
t = time in days
e = 2.718 .

When equilibrium conditions are reached, the above equation can be simplified because

$$\left[1 - e^{\frac{-(bt)}{a}}\right] = 1 \text{ and, therefore,}$$

$$c = w/b \; . \qquad (2)$$

The time period required to reach a certain percent of equilibrium values can be calculated according to the formula

$$t = \frac{Ka}{b} \qquad (3)$$

where K is a constant with a numerical value of 3 for 95% equilibrium and 4.6 for 99% equilibrium.

The above equations are useful for the following purposes:

1. To calculate leak rates (controlled dumping) or removal (purification) rates needed to prevent buildup of a contaminant above the maximum allowable level, assuming that contaminant generation rate and total mission duration are known.

2. To calculate the Maximum Allowable Quantity of a certain material aboard, assuming a known generation rate, a known allowable concentration, a known removal rate, and a known mission duration.

3. To calculate Maximum Permissible Mission Duration if the contaminant generation and removal rates are unacceptable for toxicity reasons, assuming that an absolute "ceiling" concentration of a contaminant has been defined.

For example: Given a contaminant generation rate of 0.26 mg/m^3/day and a cabin volume, a, of 8.49 m^3, w = 8.49 × 0.26 or 2.21 mg/day. Because the leak rate is known to be 1 lb/day (or 1.04 m^3 O_2 at five psia per day), b = 1.04. Therefore, the contaminant concentration at which the cabin will equilibrate, c, is

$$c = \frac{w}{b} = \frac{2.21 \text{ mg/day}}{1.04 \text{ m}^3\text{/day}} = 2.1 \text{ mg/m}^3$$

Ninety-nine percent of this equilibrium concentration will be reached

$$t = \frac{Ka}{b} = \frac{4.6 \times 8.49 \text{ m}^3}{1.04 \text{ m}^3\text{/day}} = 37.5 \text{ days}$$

Should the value for c exceed the maximum allowable level for the mission duration from a toxicity view, and neither the generation rate, w, nor the leak rate, b, can be altered, the limiting factor is now time, t, which must be recalculated using equation (1) for the acceptable c value.

To summarize the salient features of the contaminant build-up hazards, the following axioms are derived from these equations:

1. Given a certain cabin volume, the time of equilibration of contaminant concentrations in the atmosphere is independent of the final consideration attained, if contaminant generation and removal rates are held constant. Under certain conditions, the virtual (but not the true) rates of contaminant generation can become constant because the decreasing rate of gas-off from materials is balanced by the progressive loss of filtering efficiency.

2. The magnitude of the final equilibrium concentration of contaminants is directly proportional to the generation rate and inversely proportional to the removal rate in sealed cabins.

3. Contaminant concentration rises rapidly at first and then approaches a constant value (equilibrium concentration) at finite time.

From a practical standpoint, the design engineer has primarily to worry about contaminant removal rates to keep the atmosphere habitable on long duration missions. Removal rates depend on what contaminants are lost together with the cabin atmosphere as the result of outboard leak and the amount of contaminants adsorbed on the various filter beds.

Leak rates are the most effective disposal methods for contaminants. They also have the greatest impact on the rate at which contaminants accumulate in the cabin (Table I). Because large leak rates are undesirable from a logistic standpoint, other means of contaminant elimination must be found. To simplify calculations, "equivalent leak-rate times" can be used for rating filters, scrubbers, or other air purification equipment. This equivalent leak-rate time (ELRT) can be defined as the volume of atmosphere that has been "absolutely cleaned of contaminants" in one day's time with specific consideration for the efficiency of the purification unit. For example: A filter operating at 50 percent efficiency would remove only one-half of the contaminants present in a unit-volume of air passing through it. On the second passage through, it would remove one-half of the remainder, etc.

The problem with filters is that their efficiency decreases with time as the filter bed saturates with contaminants and as the flow rates through the filter drop because of particles obliterating the free passage of atmosphere.

With increasing time, filter efficiency decreases in an exponential fashion that is quite similar to the build-up of contaminants. Consequently, the inefficiencies of filters on long-duration missions will be greatly aggravated by the contaminant build-up if there is no substantial outboard leak or controlled dumping. Moreover, nuclear submarine experience indicates that adsorbent beds become saturated with boiling hydrocarbons within two to three days and permit the low boiling point materials to pass through, even displacing these. Until specific adsorption coefficients become available for particular materials and periods of time, generous safety factors should be used in calculating the amount of adsorbents.

SUMMARY

Space cabin toxicology is a new and challenging area of research in the conquest of extraterrestrial support of life. The most unique problem of this new branch of toxicology is the truly uninterrupted continuous nature of exposure to chemical toxicants. The fundamental research in the past two years has answered the most urgent basic questions.

1. Continuous exposure can lead to a "summation of interest" type of toxic effect because daily recuperative periods from exposure are non-existent.

2. The exotic atmospheric environment will influence the outcome of toxic damage; both reduced barometric pressure and oxygen rich atmosphere are influencing factors.

3. All cabin materials can and must be screened by analytical and biological methods to insure the health and performance of the crew in future manned space missions.

Recognition of these facts came none too soon. The toxicological problems involved in intraplanetary exploration are staggering and will require revolutionary approaches and many breakthroughs in the state-of-the-art in toxicology, pharmacology, aerospace medicine, and bioenvironmental engineering. Undoubtedly, these improvements will have many spin-off benefits to modern medicine, industrial hygiene, and environmental pollution control in everyday life. The better understanding of chronic toxic effects, new approaches to physical therapy, and contributions to the solution of community air pollution problems are just a few areas where this research already has produced significant and immediately applicable fallout benefits.

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METABOLIC DEMAND DURING MAN'S LUNAR ACTIVITY IN SPACE SUITS

N68-17370

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The prediction of bioenergetics of human beings in the lunar environment is now beginning to receive a great deal of attention as the date set for a manned lunar landing draws near. In addition, the recent EVA experience in the Gemini program has emphasized the necessity of understanding the effects of reduced-gravity environments on both human performance and bioenergetics.

The early history of study, theory, and initial research into problems was reviewed by Roth [1] in a recent NASA publication. All the early work, and indeed most of the recent work, has been accomplished with unsuited subjects and with a wide variety of simulation techniques. The simulation devices have included gas-filled balloons [2, 3], elastic cables [4] inclined planes [5, 6, 7] and gimbal simulators [8, 9]

Most of the research into subgravity environments has been directed specifically at the problem of weightlessness. Consequently, the exercise most investigated has been upper torso work. These investigations [8, 10, 11, 12] have uniformly concluded that exercise in a weightless environment results in heightened levels of energy expenditure over those found at Earth gravity. Also, other workers have pointed out that work apparently becomes more difficult when traction is reduced [13, 14, 15].

The first controlled experiments performed to establish metabolic rates occurring in walking under lunar and other reduced-gravity conditions were conducted by Wortz and Prescott in 1965 [9]. These experiments, conducted with a gimbal simulator having six degrees of freedom, indicated a substantial and systematic reduction in metabolic rates during walking in environments of one-fourth, one-sixth and one-eighth gravity (Fig. 1). These observations apparently contradicted what had been observed in earlier weightless simulations. Consequently, Prescott and Wortz [8] conducted another series of experiments, systematically varying the reduced-gravity simulation while conducting both upper torso work and calisthenics. They found no change in the energy expenditures for these different calisthenics conducted at one, one-half, one-sixth and zero gravity. They did find, however, an increase in energy expenditure for each of two different upper-torso work conditions. The energy expenditures were systematically increased with the simulated lowering of the gravity environment. The results of these walking and upper-torso experiments are summarized in Figure 2.

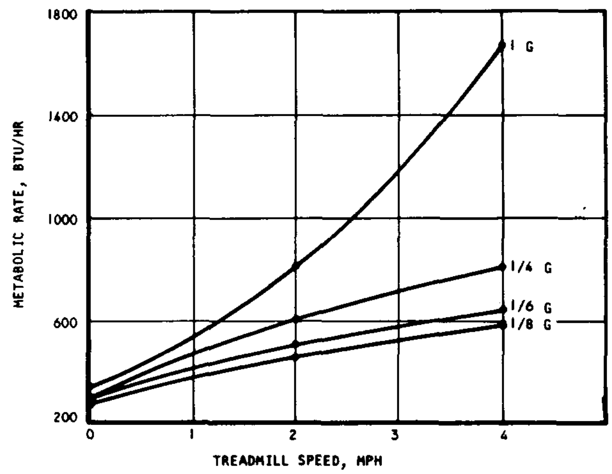

FIGURE 1. METABOLIC RATE AS FUNCTION OF TREADMILL SPEED, SIX-DEGREE-OF-FREEDOM SIMULATOR

Figure 2 shows that no changes in metabolic rate were found for calisthenic maneuvers, regardless of the level of reduced-gravity traction simulation. There is a systematic increase in metabolic rate for upper-torso exercises involving external work as traction decreases, and there is a systematic decrease in metabolic rates during walking at decreasing simulated gravity levels. It is suggested that the seemingly conflicting results can be accounted for by the following considerations. First, in calisthenics there is no external work involved, regardless of the

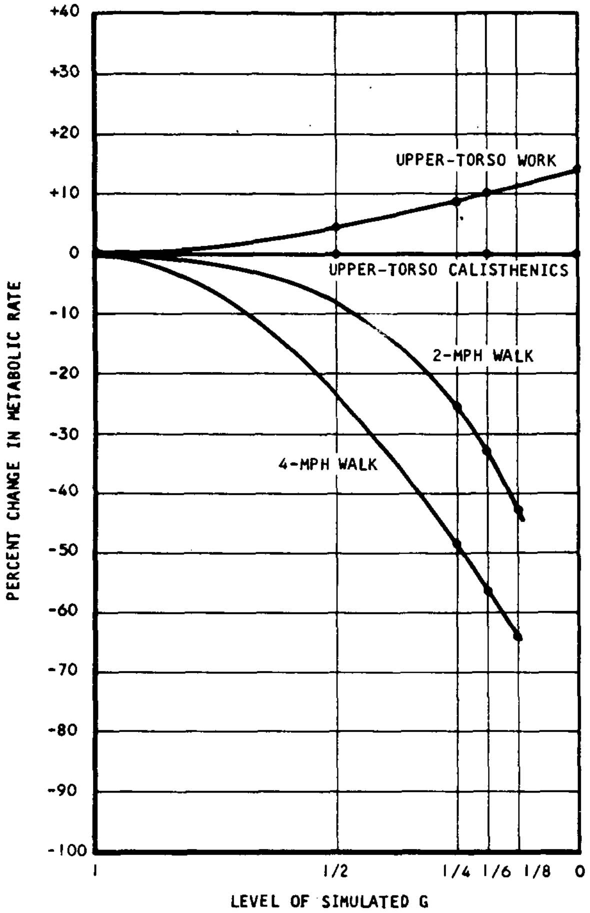

FIGURE 2. SUMMARY OF ENERGY COST DATA OBTAINED IN PREVIOUS INVESTIGATIONS AT AIRESEARCH

traction environment. Second, in the upper-torso work, the subject is required to do additional muscular work to produce the reactive forces that are normally supplied by friction at one gravity. This additional muscular work is sometimes supplied by the other hand and arm, or by combinations of arms and legs in various bracing positions that result in higher metabolic rates. Third, walking in reduced-gravity environments is physically equivalent to carrying less weight; consequently, the work is reduced as a function of the lowered weight. Company-funded experiments now in progress at AiResearch are attempting to verify these explanations.

In 1966, Kuehnegger [16] reported metabolic rates of approximately 800 Btu/hr and 1200 Btu/hr for subjects walking and loping in a Mk IV pressure suit carrying a 72-pound pack at two and four miles per hour, respectively, at simulated one-sixth gravity, using an inclined-plane simulator. He also reported that a metabolic rate of 1600 Btu/hr was associated with a loping gait at a velocity of six miles per hour and 2000 Btu/hr at seven miles per hour.

However, a program now being conducted at AiResearch for NASA Manned Spacecraft Center under contract NAS 9-6494 is showing somewhat different results in the levels of energy expenditure. This program is a comparison of two potential lunar suits, the RX-2 and the A5-L, at simulated one-sixth gravity using an inclined-plane simulator. The data for both the RX-2 and the A5-L indicate mean metabolic rates of 675 Btu/hr and 1500 Btu/hr for walking at one-and-a-half and four miles per hour, respectively.

Figure 3 provides a comparison of the current data on energy expenditures associated with walking for both suited and unsuited subjects. From this figure it can be seen that the data consistently support the hypothesis of a reduction in energy expenditure below that observed at one gravity for the exercise of walking under lunar gravity conditions. It would also appear from the comparisons of the one-sixth gravity data on the RX-2 and the one gravity data on the G2C [17] that the metabolic rates reported by Kuehnegger [16] are quite low.

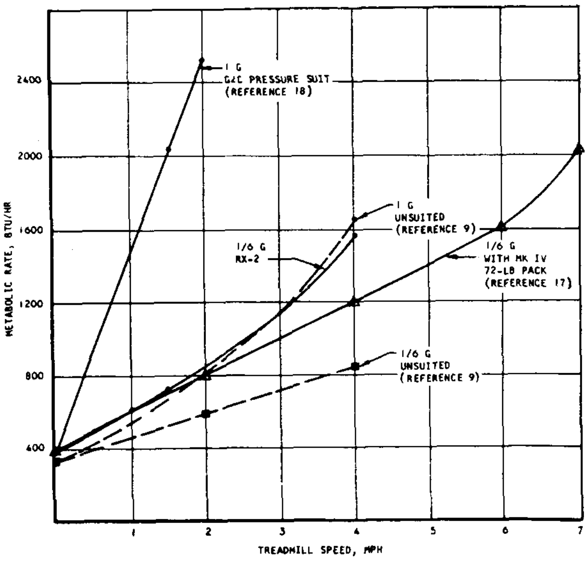

FIGURE 3. METABOLIC RATE AS A FUNCTION OF TREADMILL SPEED UNDER VARIOUS CONDITIONS OF LOADING

Recent experiments at AiResearch on upper-torso work in reduced-gravity environments have led to the observations that some work normally accomplished at one gravity cannot be accomplished at one-sixth gravity without **restraints** or work aids. For example, the work of lifting a weight over a pulley in Earth gravity is quite simple, and one can normally lift a weight equal to or greater than one's own weight. At simulated one-sixth gravity traction, however, subjects working as shown in Figure 4 could not exert a force of 15 foot-pounds without pulling themselves over onto their faces. Such a work mode is functionally analogous to sliding out an equipment rack. This type of work could not be accomplished with ease by pressure-suited subjects at loads much in excess of five foot-pounds.

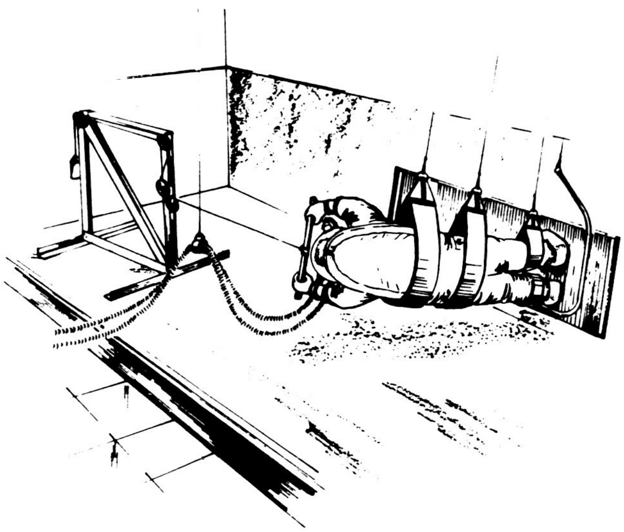

FIGURE 4. UPPER-TORSO WORK AT ONE-SIXTH GRAVITY WITH RX-2 SUIT

Although the information presently available is sketchy, some conclusions may be drawn. These are:

1. Metabolic rates during walking on the lunar surface will be substantially lower than at one gravity.

2. The metabolic cost for upper-torso work will be higher on the lunar surface than at one gravity.

3. Upper-torso tasks that can be performed at one gravity may be impossible to perform on the moon without restraints, tethers, tie-downs, or other systems designed to provide adequate reactive forces.

During the next year, the information available for the prediction of metabolic rates on the lunar surface will greatly increase. AiResearch will conduct an extensive test program for NASA Langley Research Center under Contract NAS 1-7053. More than 1000 tests will be conducted, utilizing G2C suits with modified articulated joints. The independent variables to be investigated are types of simulation, effects of various surface conditions and inclines, as well as weights of packs and locomotive velocities. Consequently, a fairly thorough understanding of major aspects of the metabolic demand during man's lunar activity in space suits should be achieved by the next meeting of this working group.

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METHODS TESTING FOR GEOLOGIC EXPLORATION OF MOON

By

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Film Presentation

All of us at this conference are concerned with the exploitation of extraterrestrial resources by men who are in a foreign environment and who are wearing cumbersome clothing. We need to use men to drive the steam shovels with which we will mine water and other minerals. But first, mineral deposits must be found, and one way to do this is through geologic reconnaissance.

Some geologic reconnaissance is scheduled for the Apollo program. On Earth, geologists typically take a pickup truck or a jeep, a pad of paper, some bags, and are off to the field. Then, weeks or months later, they return with copious notes and many rocks. But this method is not going to work for Apollo. For this reason, half of the Branch of Astrogeology of the United States Geological Survey is working on the problem of how to get geologic information from the surface of the moon onto a map on Earth, preferably in real time during a mission, and also how to obtain samples.

In the Surface Planetary Exploration Group of the United States Geological Survey , we are developing these methods. We have developed a scientific missiontype of control center, the Communications, Data Reception, and Analysis (CDRA) facility which is described in a ten-minute 16-mm color and sound film report, "Early Apollo Investigations, Field Test 8 (Production Nr. 6605-2)". This film report is available on loan from the Motion Picture Services, Publications Division, U. S. Geological Survey, Room 2647, Interior Building, Washington, D. C. 20240. The film emphasizes the development of new methods of geologic reconnaissance for extraterrestrial use.

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