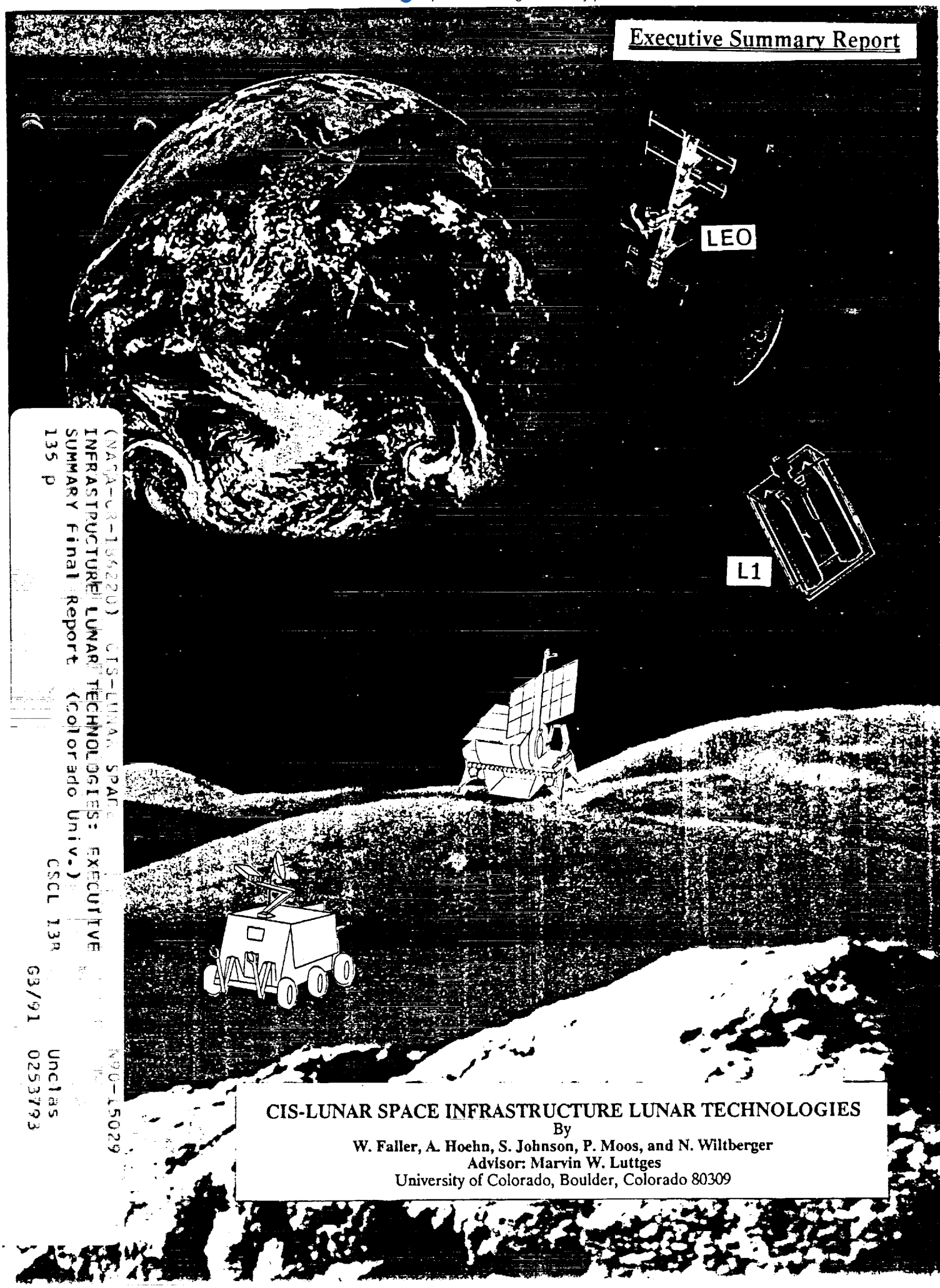
Executive Summary Report



CIS-LUNAR SPACE INFRASTRUCTURE LUNAR TECHNOLOGIES

By
W. Faller, A. Hoehn, S. Johnson, P. Moos, and N. Wiltberger
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Final Report to USRA - Summer 1989 - Overview

STRUCTURE OF THE FINAL REPORT:

The Final Report package consists out of the following 4 'stand-alone' documents:

Abstract
Executive Summary
Summary Report
Appendices

ABSTRACT:

Continuing its emphasis on the creation of a cis-Lunar infrastructure as an appropriate and cost-effective method of space exploration and development, the University of Colorado explores the technologies necessary for the creation of such an infrastructure, namely (1) Automation and Robotic, (2) Life Support Systems, (3) Fluid Management, (4) Propulsion and (5) Rotating Technologies. The technological focal point is on the development of automated and robotic systems for the implementation of a Lunar Oasis produced by Automation and Robotic (*LOAR*). Under direction from the NASA Office of Exploration, automation and robotic have been extensively utilized as an initiating stage in the return to the Moon. A pair of autonomous rovers, modular in design and built from interchangeable and specialized components, is proposed. Utilizing a "*buddy system*", these rovers will be able to support each other and to enhance their individual capabilities. One rover primarily explores and maps while the second rover tests the feasibility of various materials-processing techniques. The automated missions emphasize availability and potential uses of Lunar resources, and the deployment and operations of the *LOAR* program. An experimental bio-volume is put into place as the precursor to a Lunar Environmentally Controlled Life Support System. The bio-volume will determine the reproduction, growth and production characteristics of various life forms housed on the Lunar surface. Physicochemical regenerative technologies and stored resources will be used to buffer biological disturbances of the bio-volume environment. The *in situ* Lunar resources will be both tested and used within this bio-volume. Second phase development on the Lunar surface calls for manned operations. Repairs and re-configuration of the initial framework will ensue. An autonomously-initiated manned Lunar Oasis can become an essential component of the United States space program. The Lunar Oasis will provide support to science, technology and commerce. It will enable more cost-effective space exploration to the planets and beyond.

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Executive Summary Report

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INTRODUCTION:

Continuing its emphasis on the creation of a cis-Lunar infrastructure as an appropriate and cost effective method of space exploration and development, the University of Colorado explores the technologies necessary for the creation of such an infrastructure. A critical part in the development of a practical space infrastructure is envisioned to be the function of *LOAR* (Lunar Oasis produced by Automation and Robotic) project as a rudimentary resource storage and exploration base on the Lunar surface. Preparation for such an oasis as a prelude to manned presence an be achieved using a variety of automation and robotic technologies. The technological focal point is on the development of Lunar Robotic Vehicles, *LRV*, and an Experimental Bio-Volume, *EBV*, for the implementation of these activities. These high impact technologies in turn can have immediate positive influences on the capabilities of U.S. industry.

LUNAR OASIS produced by AUTOMATION AND ROBOTIC:

An effective space infrastructure would provide the basic facilities, equipment, and service necessary to conduct operations and science in space. Flexibility, cost efficiency, timely availability, are all characteristics of a well established infrastructure. In order to initiate the foundation of an infrastructure on the Lunar surface, a modest investment in Earth originated resources is delivered to the Lunar surface. Through robotic activities, potential resources for future exploitation may be identified. In addition, life support consumables and the capacity to process bio-materials is staged. This beginning of an infrastructure may enable the performance of future science missions and extend life support capabilities. Identification and verification of potential resources will allow the assessment of future infrastructure expansion on the Lunar surface. (See Figure 1).

LUNAR ROBOTIC VEHICLE:

The vehicles will be designed and built using many existing technologies with proven success rates and reliability. Advanced robotic and automated system designs will be utilized, such that the next generation of robotic systems will no longer simply observe and interact passively with the environment. Rather, these systems will be capable of autonomous interaction with, and manipulation of the environment.

Using "*buddy system*" operations, the rovers will support each other and enhance individual rover capabilities and survivability. The first rover would provide on-site verification of lunar resources previously identified by the *LPO*. With the second rover on hand, they would survey areas inaccessible to the *LPO* such as lava tubes and they would analyze core samples for subsurface mineral and volatile compositions. Topographic information, enhanced by rover surveys, would establish the accessibility of available lunar resources. The best routes to high resource concentration sites would be established. Minor route surface improvements would be undertaken and common staging points for future resource recovery operations would be identified. Prototype resource processing units would be specified by onboard resource analysis equipment. The robotic vehicles would be able to deploy a far side radio observatory and other scientific instrumentation. Site improvements such as roads and radiation shields are also readily achieved. Since the modular nature of the robotic vehicles will allow interchanging of specialized components such as an analyzer module for a processing module, the rovers would scavenge valuable Lunar resources. Trace amounts of water in the regolith, for example, could be collected and stored for later human consumption.

Design Guidelines:
(1) *Highly redundant:* Multiple sensors, effectors and control systems. (2) *Self-similarity:* Modular components and functional exchanges. (3) *Existing technology:* Reliability tested, low cost, accurate integrations. (4) *Simple mission profiles:* Planned, mapped, structured tasks. (5) *Environmental and situational predictability:* Extensive sensing, internalized maps, "learning". (6) *Modular system upgrades:* Phased technology demonstrations (7) Common power bus, standard hardware templates, standard mechanical and electrical connectors.

Table 01: Design Guidelines for Lunar Rover Vehicles, *LRV*.

Specifications (see also Figure 2):	
Mass:	1,500 - 3,000 kg
Power:	Dynamic Radio-Isotope Generator, *DIPS*
Locomotion:	four independent 1 Horsepower drives
Modular:	Interchangeable design
Capabilities:	extensive sensing capabilities geological analysis
Operation:	"*buddy system*" operating system automation / teleoperation control

LIST OF SYMBOLS / ACRONYMS:

A/R	Automation and Robotic
DIPS	Dynamic Isotope Power System
EBV	Experimental Bio-Volume
ECLSS	Environmentally Controlled Life Support System
ELV	Expendable Launch Vehicle
FTIR	Fourier Transform Infrared Spectral analyzer
ICP	Inductive Coupled Plasma analyzer
ISPV	Specific Impulse delivered in Vacuum
LECLSS	Lunar Environmentally Controlled Life Support System
LEL	Lunar Expendable Lander
LEO	Low Earth Orbit
LH2	Liquid Hydrogen
LO	Lunar Orbit
LOAR	Lunar Oasis produced by Automation and Robotic
LOX	Liquid Oxygen
LPO	Lunar Polar Orbiter
LRAT	Lunar Remote Access Technology
LRV(s)	Lunar Robotic Vehicle(s)
LS	Lunar Surface
MR	Mass Ratio of initial to final mass of rocket
RTG	Radioisotope Thermoelectric Generator
XRF	X-Ray Fluorescent Spectrometer

Equipment:	Pulsed Laser Ranging Infrared Proximity Sensing Semi-Dexterous Manipulators Control Systems - conventional computers Navigation Control - Neural Network Architecture Construction Tools
Analyzer:	Gas Chromatography X-Ray Fluorescent Spectrometer

Table 02: Specifications for Lunar Rover Vehicles, *LRV*.

THE EXPERIMENTAL BIO-VOLUME, EBV:

The continuous long-term support of life in the non-terrestrial Lunar setting establishes confidence in subsequent use of the bio-volume as a safe manned habitat. The support of life on the Moon will stimulate public interest in the exploration and development of space as a future habitable environment. The experimental bio-volume is envisioned as a novel but highly useful prelude to manned habitation on the Lunar surface. The *EBV* will provide quantitative data on the life cycles of biological systems in the Lunar environment. It will field test key technologies for the implementation of a *LECLSS*. The *EBV* will further provide on site testing of a refittable module suitable for fulfilling the immediate life support requirements of the *LOAR* project. A unique opportunity to characterize an environmentally controlled system with respect to remote terrestrial performance should be attained. And, the use of biological interactions in extracting and using *in situ* resources can be evaluated. The tandem mission scenarios of the *LRVs* and *EBV* will provide the ground-based verifications and databases necessary to establish the feasibility and baseline technologies required for the development of manned habitation. Leading to full Lunar surface access for science, technology and commerce.

The reconfigurable bio-volume will consist of two major parts: The airlock and the experimental volume. The airlock (the previous oxygen tank) will provide access to the pressurized interior (see Bio-volume Atmosphere Control). The experimental volume (the previous hydrogen tank and the intertank section) will provide space for experiments as well as control and monitoring equipment. During the automated phase of the *LOAR* operation biological experiments for a Lunar *ECLSS* will be conducted. Manned sorties starting in 2004 can utilize the volumes to provide temporary shelter and life support consumables. The proposed mission scenario requires three *LEL* landings prior to any human mission. Each of the three bio-volumes will provide reconfigurable volumes for the development of life supporting modules, a laboratory area and control centers. Activities within the experimental volume will include plant growth, biological waste treatment (bioreactor) and small-scale animal experiments (aquaculture and soil processing). The use of Lunar soil as a growth medium and the recovery of trace elements from the soil via microbial acidification and extraction will be investigated. The *EBV* will provide the first non-terrestrial *ECLSS* capabilities. Plants, animals and microorganisms will live in a physicochemically buffered symbiosis. Robotic arms will provide internal manipulation of hardware for all experiments. Buffer volumes will be maintained for carbon dioxide, oxygen, nitrogen and water. Capitalizing on local resources (trapped volatiles such as water, carbon from carbonaceous meteorites) the buffer volumes can potentially be increased prior to human visits. Results obtained from the experiments will elucidate important questions in the design of a *LECLSS*. A cut-away view is shown in Figure 3.

Design Guidelines:
(1) Highly redundant growth chambers. (2) Autonomous: Self contained power and regulatory systems. (3) Multiple sensors and experimental monitoring systems. (4) Phased technology demonstrations: Lunar processing, in situ resource recovery. (5) Utilization of available *in situ* resources: volatiles, carbon, ambient temperature, sunlight. (6) Modular adaptability for manned habitation.

Table 03: Design Guidelines for the Experimental Bio-Volume, *EBV*.

Specifications:	
Module:	recycled from cryogenic propellant tank.
Airlock :	former oxygen tank: L = 1 m, D = 4.5 m, 18.5 m^3.
Bio-volume :	former *LH2* tank: L = 5 m, D = 4.5 m, 60 m^3.
Mass empty:	3,500 kg
Volume:	75.0 m^3 total; 50.0 m^3 Hydrogen tank; 12.5 m^3 intertank; 18.5 m^3 oxygen tank
Power:	10 kW_e photovoltaic (30 m^2 solar array, regenerative fuel-cell back-up).
Heat rejection:	30 m^2 radiators.
Control:	Automated control system
Equipment:	Intertank space for temperature sensitive equipment; pre-installed plumbing. Internal / External Robotic Arms. "Dark Room" Growth Chambers (mushrooms, sprouts, etc.) Aquaculture Tank Waste Processors: biological processors, composter, soil conditioner. Physicochemical Support Equipment Lighting

Table 04: Specifications for the Experimental Bio-Volume, *EBV*.

SYSTEM INTEGRATION:

Careful design of the lunar landing system will allow for maximal use of all delivered materials. The main propellant tanks will be prepared for conversion into an autonomous, life supporting bio-volume. Additional tankage volumes (attitude control thrusters) will be used for storage of produced volatiles. Expended lander material may be scavenged and be put to good use. Lunar materials gathered by the rovers may be introduced into the Experimental Bio-Volume, *EBV*, to test the bio-processing feasibility of *in situ* resources, such as carbonaceous meteorite materials. Left-over nitrogen gas from the attitude thrusters will contribute to the initial atmosphere in the Bio-Volume. Excess

propellants (hydrogen and oxygen) will be converted into water and used as initial atmosphere. The landing configuration of the Lunar Lander with the Lunar Rover Vehicle and the tankage volume and external payload are shown in Figure 4.

Scenario Assumption:
- Results from Lunar Polar Orbiter available prior to launch. - Shuttle-C or equivalent for transportation into Low-Earth-Orbit, *LEO*. - Technology for long-term cryogenic storage (weeks) available. - small variable-thrust cryogenic thrusters (20 kN max. thrust). - dynamic radio-isotope power generators, *DIPS*, for the Lunar Rover Vehicles, *LRV*. - Communication links at all time between Lunar Base (*LOAR*) and the vehicles, *LRVs*. - A polar site with continuous sunlight availability desirable, but not required.

Table 05: Assumption for the Implementation of the lunar oasis, *LOAR*.

COST AND COMMERCIAL CONSIDERATIONS:

Business as usual will just not do! The cost of doing business with large administrative overhead drains financial resources which otherwise could be applied to achieving results. By establishing less rigorous reliability requirements and accepting a certain degree of failure, costs may be minimized. Perhaps such a scenario could be best implemented by small companies with academic partners. To avoid administrative overhead due to concerns of technology transfer, no international interaction will be sought.

Estimated Cost:	
Launch cost	1.0 Billion
Research and Development	2.5 Billion
Production	1.0 Billion
Operations	0.5 Billion
Total Cost	**5.0 Billion**

Table 06: Estimated Cost for Implementation of the lunar Oasis, *LOAR*.

Commercial Considerations:

Private industry will be jointly vested in homesteading the Lunar surface. Resources identified and staged through Lunar based operations may be of value to the infrastructure in the future. For example, sufficient life support expendables may be stored such that week-long human stays may be facilitated. The "Moon-Steading" partners would receive financial compensation for the use of these resources. At the same time, the new waste products added to the system will allow the bio-system mass to expand and accommodate larger future manned missions (and greater financial returns). Similarly, Lunar resources identified and utilized for infrastructure support also would yield a return on the investment made.

CONCLUSION:

The assessment of available Lunar resources and the establishment of an Experimental Bio-Volume, *EBV*, on the Moon should be feasible at low cost through Automation and Robotic technologies. Two similar modular robotic vehicles along with three Bio-Volumes, converted from propellant tanks, can be established on the Lunar surface. Comprehensive sample analysis, enabled through Automation and Robotic technologies will verify the availability and accessibility of Lunar resources. In addition, these vehicles can be used to perform construction type activities for site preparations and establishing routes to resource locations. The operation of these vehicles is complemented by the "*buddy system*" whereby the vehicle rovers provide aid to each other, assist in complex activities and share information gathered. The Bio-Volume will establish the ability to support life on the Moon. Physicochemical systems will supplement the biological systems in order to maintain proper ecological balance. Buffer volumes will be established to accommodate variations in Bio-Volume operations and support future human missions. This program may be implemented by relatively inexpensive means if reductions in reliability are accepted and administrative overhead is severely restricted. This may be possible by utilizing small corporations in concert with United States science and technology interests within academia.

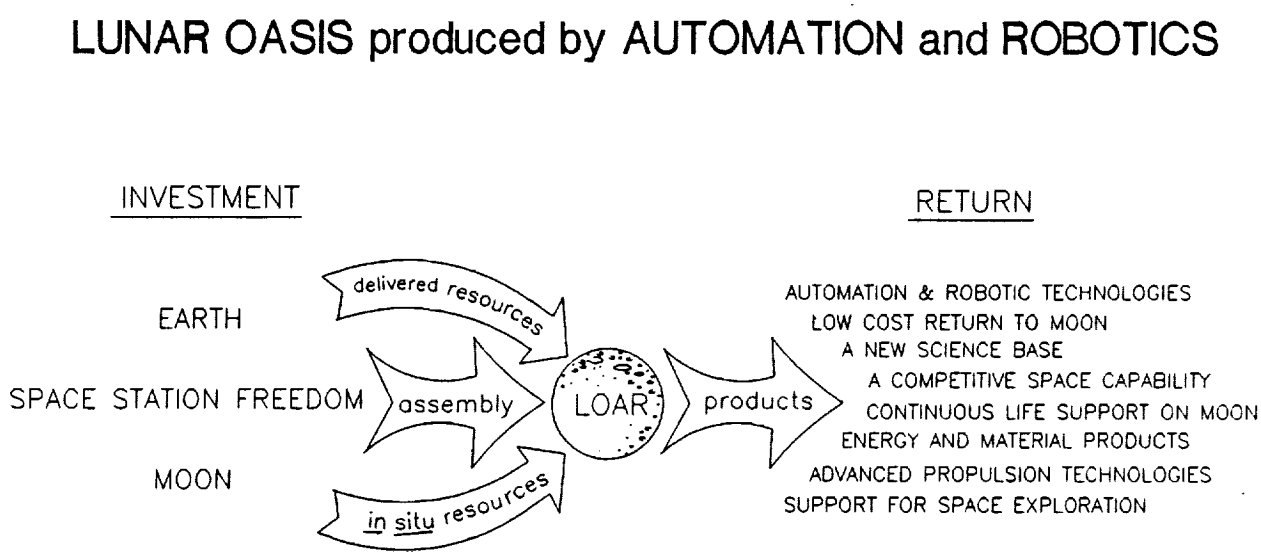


Figure 1: *LOAR*: Investment and Return.

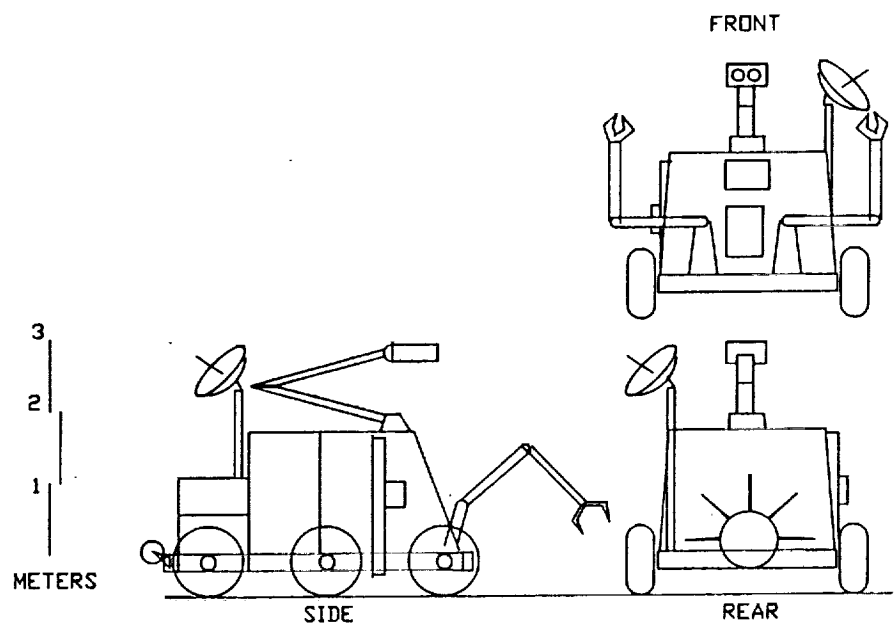


Figure 2: The Lunar Robotic Vehicles, *LRV*

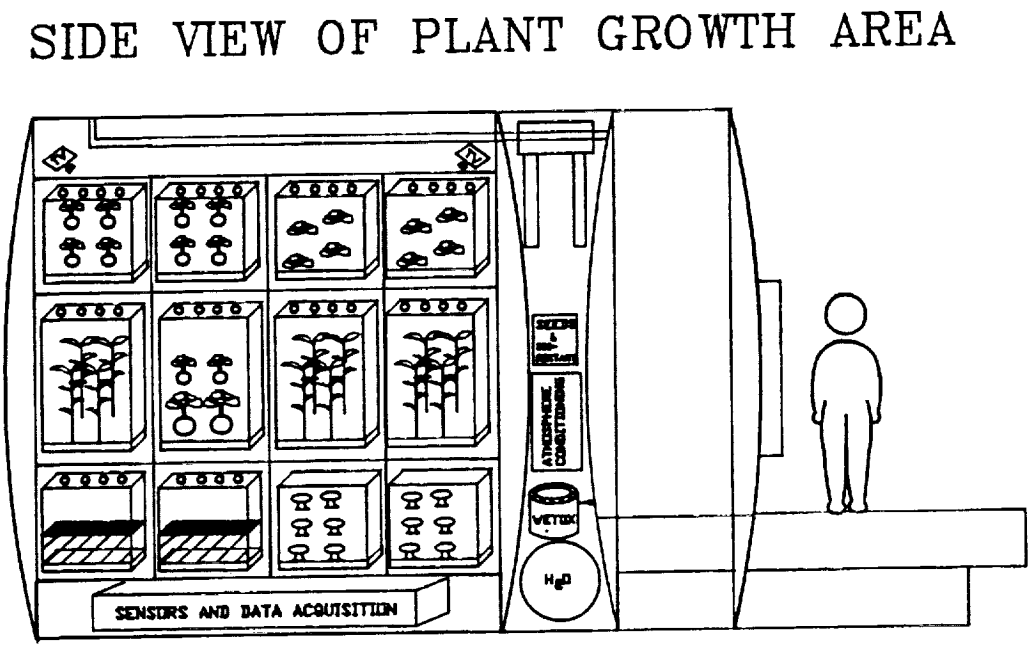


Figure 3: Schematic of Experimental Bio-Volume (Human for Size Comparison Only).

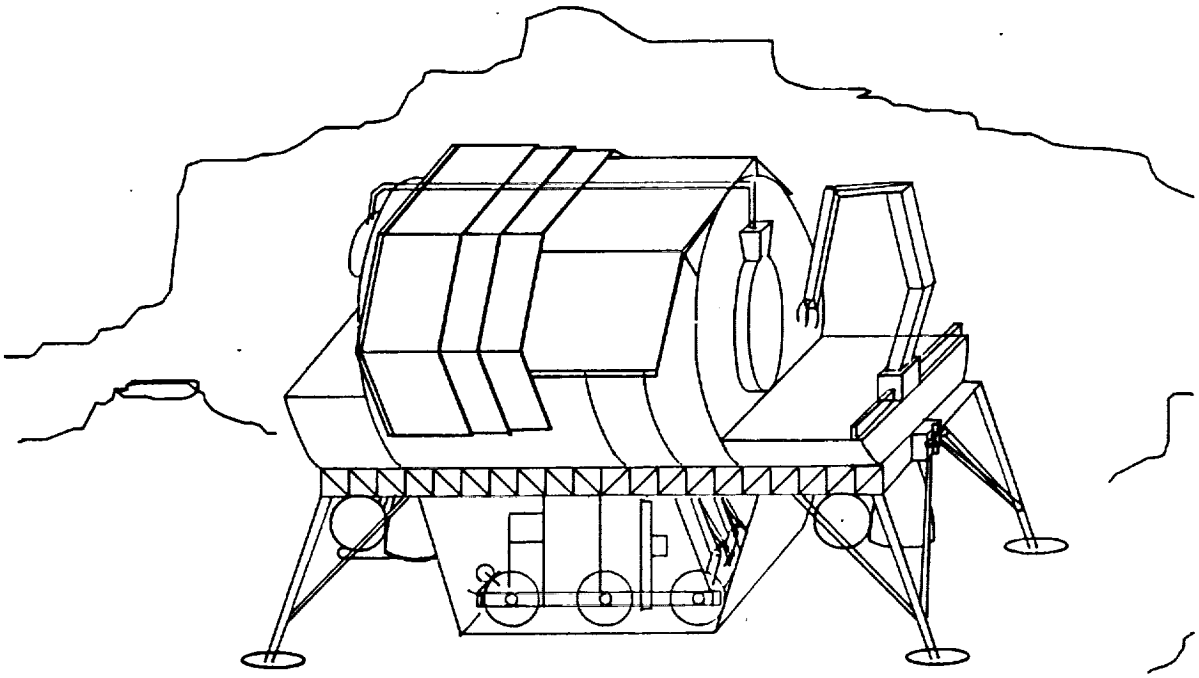
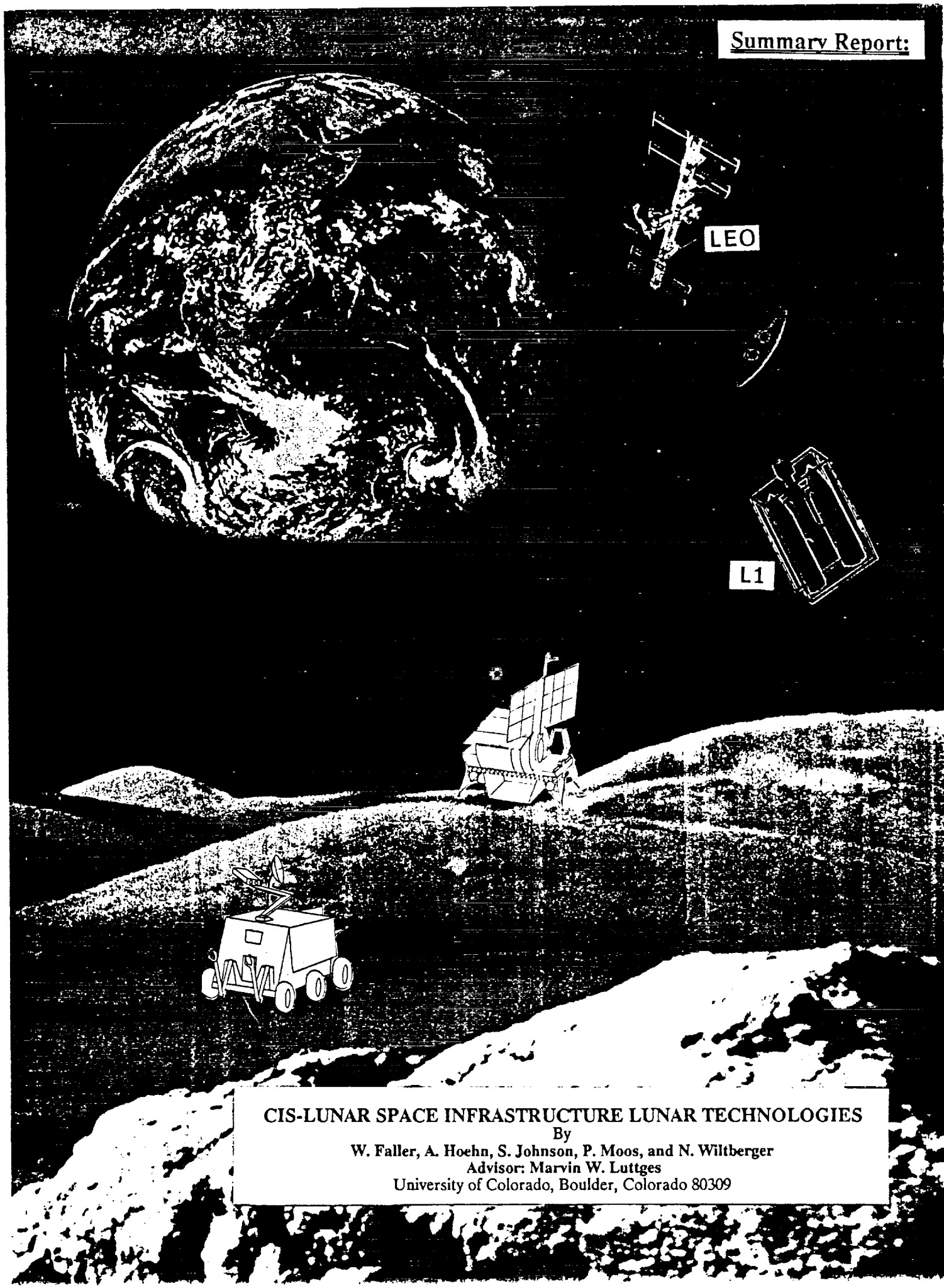


Figure 4: The landing configuration of the Lunar Expendable Lander, *LEL*, with the Lunar Rover Vehicle, *LRV*, and the tankage volume and external robotic arm. Solarvoltaic cells and heat radiator are still undeployed.



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Summary Report:

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I. INTRODUCTION:

The United States Space Programme must have a safe and cost-effective means of enabling missions in the space environment. Pioneering the Space Frontier and the Leadership Report both emphasize the need for activities that will catalyze future mission scenarios such as the "*Exploration of the Solar System*" and "*Humans on Mars*" (Ref. 2). The development of a cis-Lunar infrastructure has been proposed as the foundation for implementing such future space missions(Ref. 3) (see Figure 1).

A key element in the development of the cis-Lunar infrastructure will be a habitable manned outpost on the Lunar surface. Implementation utilizing automation and robotic will develop the framework for the Lunar outpost and will have immediate benefits to U.S. industry. Manned activities will be supported by the *LOAR* (Lunar Oasis produced by Automation and Robotic) programme and will provide an enhanced level of capability to the overall infrastructure (see Figure 2).

The success of the Apollo missions, due in part to the Lunar Surveyors and Rovers, has shown the feasibility and desirability of coupling automated and manned missions. The *LOAR* project will demonstrate the enduring value of advanced robotic missions followed by manned missions. Automation and robotic (*A/R*) coupled with manned operations are a lower cost yet safe and effective means for implementing many elements of the space programme.

A Lunar Oasis by definition must provide respite from the rigors of the harsh Lunar environment. A secure habitat with the capability to sustain an atmosphere, to store both delivered and Lunar recovered consumables and to support communication would be of high value. Full deployment or even partial completion of the Lunar Oasis will enable more productive and innovative human missions on the Lunar surface. Automation and robotic technologies would be used to deploy a primitive oasis on the Lunar surface prior to human presence. Upon arrival, humans could finalize the oasis, deploy science experiments, explore along robotically prepared trails and initiate higher level Lunar processing activities.

To initiate the *LOAR* project three major hardware elements are to be delivered to the Lunar surface: Two robotic vehicles and an autonomous bio-volume (see Figure 3).

The development of two Lunar robotic vehicles (*LRVs*) and an experimental bio-volume (*EBV*) provides the requisite conditions for implementation of the *LOAR*. The *LRVs* will develop the preliminary information and hardware deployment framework required for the establishment of the Lunar outpost. Implementation requires that specific remote sensing information be gathered by the Lunar Polar Orbiter (*LPO*). Databases on regolith composition and location, resource and volatile availability, and navigable terrain will be crucial to search strategies utilized by the automated vehicles. Continuous monitoring via telemetry will enable ground-based verification of Lunar activities. Manned teleoperation is anticipated only in the case of "*unrecognized*" circumstances and emergencies. The vehicles will systematically search the Lunar surface for valuable resources and will carry out the deployment of hardware and scientific experiments. The bio-volume will be set up to provide information on Lunar life-support systems. It will be convertible to an interim habitable module for the *LOAR* supporting early manned sorties to the Lunar surface (see Figure 4).

Ground-based topographic maps and resource deposit location maps created by the robotic vehicles and Lunar Remote Access Technology (*LRAT*) will verify and enhance the maps generated using *LPO* remote sensing information. Prescribed vehicle interactions

with the environment coupled with the field testing of a variety of processing and resource recovery techniques will provide the experimental data necessary to develop detailed resource recovery and processing operations. The resource and operations databases will enable manned use of the *LOAR* to enable efficient and effective Lunar science, technology and commerce activities.

The continuous long-term support of life in the non-terrestrial Lunar setting establishes confidence in subsequent use of the bio-volume as a safe manned habitat. The support of life on the Moon will stimulate public interest in the exploration and development of space as a future habitable environment. The experimental bio-volume is envisioned as a novel but highly useful prelude to manned habitation on the Lunar surface. The *EBV* will provide quantitative data on the life cycles of biological systems in the Lunar environment. It will field test key technologies for the implementation of a *LECLSS*. The *EBV* will further provide on site testing of a refittable module suitable for fulfilling the immediate life support requirements of the *LOAR* project. A unique opportunity to characterize an environmentally controlled system with respect to remote terrestrial performance should be attained. And, the use of biological interactions in extracting and using *in situ* resources can be evaluated. The tandem mission scenarios of the *LRVs* and *EBV* will provide the ground-based verifications and databases necessary to establish the feasibility and baseline technologies required for the development of manned habitation. Leading to full Lunar surface access for science, technology and commerce.

II. RATIONALE:

A critical element in a practical space infrastructure is envisioned to be the *LOAR*. It is a rudimentary resource storage and accumulation site, an exploration base on the Lunar surface and a near-Earth technology demonstrator. Preparation for such an oasis as a prelude to manned presence can be achieved using a variety of automation and robotic technologies. These high impact technologies in turn can have immediate positive influences on the automation and robotic capabilities of U.S. industry. Initial costs and public investments can be kept relatively modest without safety risks, high reliability costs and time-sensitive concerns (see Figure 5).

1. Investment and Return

Using automation prior to manned missions offers a number of advantages. The *LRVs* and *LRAT* can be utilized to implement the remote sensing verification and resource availability studies. The automated mission scenarios take advantage of the fact that *A/R* hardware is inexpensive to transport, does not require life-support systems and is insensitive to unanticipated mission delays and constraints. Manned presence following implementation via automated and robotic systems provides a logical step forward in *LOAR* development. The processing techniques and biological system databases can be used in the design and certification of life support requirements necessary for man-rated missions. Starting with autonomously demonstrated resource recovery and processing techniques, manned operations can capitalize on the initial framework established by the *LRVs* and *EBV*. Scientific and technological spin-offs such as advanced robotic could be utilized by industry, even during the initial Earth-based testing of *LOAR* technologies. Commercial interests in low volume, low power food production and waste handling systems would benefit almost immediately from *LOAR* technologies. Controlled environmental research would provide valuable clues to the dynamic alterations that appear to be occurring on Earth.

The implementation of Lunar activities beginning with the *LRVs* and *EBV* will provide the field testing and initial development of the *LOAR* project for utilization within the cis-Lunar infrastructure. As seen in Figure 5, the *LOAR* plays a critical role in providing both immediate and long-term returns from the public investment. As described below, the activity can become a major focal point for precursor missions that enable a more productive, less risky human activity in the space environment. As a programmatic matter, the precursor mission may be more dependent on academic and industrial support than a manned, NASA mission support.

MISSION GOAL:	Explore space in an effective and low cost manner.	
INTERIM OBJECTIVES:	(1)	Establish a continuous progression of programmes leading to space exploration, utilization, and eventually habitation.
	(2)	Develop space science, technology, applications and commerce for the benefit of humans on Earth.
	(3)	Utilize space activities as the focus for industrial, technological and commercial growth for future generations in the United States.

2. Mission Scenario:

The *LOAR* mission scenario focuses on a phased automated and manned activity to be undertaken on the Lunar surface. Robotic and automated systems provide a starting point for the development of enabling technologies, and the establishment of an operational Lunar framework. The *EBV* supports and evaluates biological systems such as plants, primitive unicellular life forms, and invertebrates on the Lunar surface. Interactive physicochemical systems are used to provide environmental buffers and to aid in *in situ* resource use. *LRV* activities include ground-based verification of remote sensing topography. Gas chromatography, X-ray fluorescent spectrometry (*XRF*), Fourier transform infrared spectral analysis (*FTIR*) and inductive coupled plasma analysis (*ICP*) of recovered soil samples will aid in the detailed quantification of remote sensing information. Candidate locations for the recovery of volatiles are searched both by the *LRVs* and *LRAT*. The *LRAT* can be used for the exploration of spatially restrictive craters, caves, and lava tubes. Manned operations take advantage of the information gathered on the effectiveness of resource recovery methods for Lunar regolith. Initial automated evaluations can be done for rock cooking, Ilmenite reduction by hydrogen, and centrifugal separations of the Lunar soil into baseline components. The production of materials such as Lunar composites, fiberglass and concrete for use in the implementation of fiberglass winding technologies for making "utility" volumes or for the construction of Lunar buildings is explored. A general database on space based *LECLSS* is established and used to satisfy future manned mission requirements. Scientific locations of interest are mapped and scientific apparatus, such as the dark side synthetic-aperture radio observatory are deployed. Manned operations will utilize any resources that have been recovered, and will instigate any repairs and modifications necessary for the conversion of the bio-volume to a man-rated habitation module. Manned operations complement the *LOAR* project with the establishment of a habitable manned outpost on the Lunar surface, and the development of a key element in the cis-Lunar infrastructure. A permanently manned Lunar base may arise from these initial infrastructure developments.

III. MISSION IMPLEMENTATION:

1. Prototype Field Testing:

The first step in the implementation of a phased automated and manned scenario should involve extensive prototype field testing. Development of the vehicles is envisioned to start in 1994, leading to prototype demonstrations in 1998. Crater National Park (Arizona) has been chosen to determine the functional operating characteristics of the robotic vehicles. Bio-volumes can be developed at university centers. Both activities are to be fully completed prior to delivery on the Lunar surface. The natural break point for manned operations will be determined based on such testing. Paradigms and control systems for *LRV* remote sensing verification, resource exploration and recovery, and resource transport will be tested. The functional operating parameters of the subsystems to be utilized in remote sensing verification will be defined. Topographic maps can be verified using pulsed-laser ranging. Soil calibration can be accomplished through sample retrieval, gas chromatography, *XRF*, *FTIR* and *ICP*. The available sensor systems for monitoring both the internal state of the vehicles and external interactions with the environment can be field tested. A determination can then be made on the completeness of available sensory information. System upgrades can be implemented to fill any voids in the required sensory information. Field testing of control system integration with pulsed-laser ranging and infrared proximity sensing will document the operational accuracy of spatial localization for both the *LRVs* and robotic arms. Strain gauges and magneto-resistive systems utilized in the robotic arms will be tested to determine the quality of cutaneous and kinesthetic information, respectively. The functional control algorithms of all subsystems will be updated based on information gathered during field testing. The power requirements for the vehicles and subsystems will be determined for a broad range of mission scenarios. Extensive field testing will allow the operational capabilities of the *LRVs* and *LRAT* to be defined, and an appropriate manned intervention scenario to be chosen. The extent to which automated control can be utilized will be determined, and functional manual override systems tested. Teleoperation will be utilized for repairs and control of the *LRAT* during exploration and remote sensing verification. The completion of field testing in 2000 will define the degrees of freedom available for autonomous interactions with the environment and the capabilities of teleoperation.

Model crisis situations can be contrived in a highly-monitored environment in order to determine the extent and effectiveness of a "*buddy system*" operating protocol (see Figure 6).

Classically the "*buddy system*" is an arrangement in which two individuals operate individually but may work together as a team in critical situations. In this scenario an otherwise disabling or dangerous situation can be avoided due to the intervention and support of the other team member. Field testing of manned intervention will enable hands-on human training. Estimates on the capabilities of manned teleoperated repairs will be elucidated, and programmed crisis recognition routines will be experimentally verified. Nominally, teleoperated intervention, communication constraints withstanding, are anticipated for most repairs. Manned intervention on the Lunar surface during the automated missions would occur only in the event of *LRV* failure or "unrecognized" circumstances. The "*buddy system*" should not only enable more complicated tasks to be undertaken, but should provide an effective manner in which to perform these tasks. The implementation of the "buddy system" via teleoperated links could, for example, allow a wheel to be changed using the manipulator arms. Similarly a tow-line could be utilized if one vehicle became stuck, or a jumper cable could be installed to supplement the power of a single vehicle. Further, the quantity of information gathered during the mission will be doubled, memory dumps will provide redundancy in the stored information as well as

shared information between vehicles. The teleoperated "*buddy system*" should play a crucial role in the effective initiation of Lunar activities utilizing automation and robotic.

Field testing of the bio-volume, *EBV*, evaluate life support system hardware and show balanced compatibility of plants, fish and primitive life forms. The role of physicochemical systems and storage volumes will be assessed. The experimental field testing will provide the necessary control studies and experimental verification of the operating systems to compare Earth based to Lunar based biological experimentation. The types of plants, fish, algae, bacteria, and fungus to be utilized can be determined. The volumes and power dedicated to each subsystem should be determined such that an optimal balance between the biological and physicochemical systems is attained. Subsystems can be monitored to determine power requirements, water and nutrient requirements, and growth rates. Further, the extent of interactions with the vehicles will be determined, and the potential degree of local resource utilization defined. Manned interactions and modifications of the *EBV* necessary to provide habitat modifications of the *LOAR* can be extensively planned and structured. Modular designs will be essential in the development of the subsystems in order to permit rapid manned upgrades of system technologies. Information gathered will impact the *LOAR* programme, Space Station Freedom and terrestrial-based *ECLSS* programmes. Extensive earth based testing of fully controlled life-support systems will allow the baseline capabilities of the *EBV* to be defined. A detailed scenario for manned re-configurations can then be determined.

2. Lunar Polar Orbiter (Remote Sensing)

The site chosen for the implementation of the *LOAR* project relates directly to the initial missions and future development scenarios. The *LPO* information will be crucial in order to determine the areas of maximum sunlight, resource availability, and navigable terrain. The polar regions offer the possibility of nearly continuous lighting. Continuous satellite monitoring of the Lunar poles will enable the determination of the variable lighting and power use schedules. The thermal and lighting characteristics available at a polar site should play a major role in determining a landing site. The north pole has been more extensively mapped, while the south pole faces the scientifically intriguing galactic center. A polar site offers the advantage of at least 6 months continuous sun light (maximum elevation above the horizon is 1.5°). Areas continuously shadowed, caves and lava tubes, are sites for the potential existence of volatiles. Maximum sunlight is likely to be the most applicable for manned outposts, since at least some portion of the power requirements will be fulfilled utilizing solarvoltaic arrays. The availability of natural sunlight for growing plants would also reduce the power requirements of the *EBV*. Potentially a site can be located that will provide continuous sunlight throughout the year[Ref. 5].

The implementation of Lunar activities will begin with the launch of the Lunar Polar Orbiter (*LPO*) scheduled for 1994. High resolution (1 meter) topographic maps coupled with candidate identification of available resources will provide the initial world models (maps) upon which the automated exploration and search strategies will be built. Each mission scenario will require a particular database of sensory information. High resolution topographic maps, 1 meter or less, will provide the information for creating internal maps and markers for autonomous navigation of the Lunar surface. Further, elevation changes and the relative hardness or softness of the Lunar regolith will be estimated in order to develop safe and efficient navigable routes on the Lunar surface. Gamma-ray, X-ray, IR and Neutron spectrometers will provide the initial mapping of candidate areas for deposits of both volatiles and recoverable *in situ* resources. The information obtained by remote sensing will determine not only the initial landing site, but

the operating characteristics and mission scenarios undertaken at each of the various exploration sites. The availability of local information or "ground truth" will allow more detailed and accurate remote sensing to be accomplished, therefore more detailed and accurate world models can be developed. Periodic updates of the robotic systems should increase the efficiency and safety of progressive more ambitious missions. *A/R* coupled with the *LPO* will enable manned missions to take advantage of accurate topographic maps and resource locations.

3. Surface Delivery

Following the evaluation of remote sensing data from the *LPO*, the *LRVs* and *EBV* will be delivered to the selected landing site on the Lunar surface by Lunar Expendable Landers (*LEL*). Initial deployment is scheduled for the years 2000 - 20001. The unique features of the *LEL* includes the maximum use of all hardware and consumables delivered with the payload. Cryogenic propellants (liquid hydrogen, *LH2*, and liquid oxygen, *LOX*) rather than existing storable propellant technology has been selected due to potential use in the *LOAR*. Leftover propellants will be combined into water for use in regenerative fuel cells. The attitude control jets, fueled by nitrogen, will be scavenged and used as the inert gas in the bio-volume. The principle advantage of this propulsion system is the reconfiguration and use of the tank structures in the bio-volume. The hydrogen tank is equipped with rack-type structures for the biological experiments, and the ceiling track for the robotic arms. The oxygen tank will serve as the airlock for bio-volume operations. The intertank section is a safety buffer between the *LOX* and *LH2* tanks, and serves as a storage volume for the temperature sensitive equipment and biological experiments. The robotic arms utilized in the bio-volume will have to remove the hydrogen bulkhead in order to get access to the hydrogen tank. The landing configuration of the *LEL* includes an approximate dry mass of 8,770 kg (lander structure, engines, tank, bio-volume support equipment and *LRV*). With the characteristics given for the chosen propellants and the mission, a total wet mass of 33,300 kg to be launched into Low Earth Orbit (*LEO*) (see Figure 7).

Delta-V from *LEO* to *LO*	4,100. m/s
Delta-V from LO to *LS*	1,900. m/s
Total Delta-V	6,000. m/s
I_{sp} vac	4,500. m/s
Dry Mass on *LS*	8,770. kg
Propellants *LO* to *LS* (*MR* = 1.52; Delta-V 1,900 m/s)	4,600. kg
Wet Mass in LO	13,370. kg
Propellants *LEO* to *LO* (*MR* = 2.49, Delta-V 4,100 m/s)	19,900. kg
Total Mass Delivered to *LEO*	33,270. kg

Table 01: Mission and Mass Assumption for the LEL / Surface Delivery Summary. A break-down of the mission and mass assumptions used for the design of the Lunar Lander using *LH2* and *LOX*. Additional data in the Chapter: Hardware.

The propellant tanks (with the bio-volume support equipment attached on the outside) and the lander structure (with the *LRV* payload bay, engines and *LRV*) will be launched in the Shuttle-C (D=4.5m * L=22m, -> 35,000 kg payload into *LEO*). In orbit

the lander and tank module will be reconfigured into the Lunar transfer and landing configuration as shown in Figure 7. The *LEL* consists of a truss structure with landing gear, two cryogenic engines (20 kN thrust each, variable thrust, restartable) and distributed attitude control jets. Trans-Lunar shipment and the Lunar landing sequences for the *LEL* will be controlled via automated computer systems. The solar arrays, heat radiators and external robotic arm are permanent installations on the outside of the tank. The *LEL* assembly will be protected from dust during landing by a removeable cover. Micrometeorite "bumper" shielding is built into the individual components. Beginning activities on the Lunar surface will require that the *LRV* deploy solar arrays and heat radiators. Remaining propellants will be combined into water for storage.

4. Remote Sensing Verification

Utilizing the mobile search routines developed during field testing, the ground truth of remote sensing information will be determined using the first *LRV*. The vehicle will establish a home base at the landing site by deploying a set of triangulation beacons. Starting from this central hub, gas chromatography can be utilized for the calibration testing of Lunar soil, and pulsed-laser ranging can verify the topographic maps. Pulsed-laser ranging should also allow the vehicle to determine relative position to known landmarks and the triangulation beacons. Stereoscopic television cameras will provide Earth-based verification of search and movement strategies, as well as the capability for teleoperated intervention and control of both the vehicles and robotic arms. Navigation strategies will take advantage of the world model developed from the remote-sensing topographic maps. Determination of the routes and search orders will be via neural network computation. A three-dimensional version of the two-dimensional Hopfield model(Ref. 4) should allow for rapid, accurate calculations of shortest routes and pathways. The sites and topographic markers would form the nodes of such a model. A two-dimensional model solves the traveling salesman problem on the order of 1000 times faster than conventional computer systems. A three-dimensional architecture should provide real-time strategies. Verification of the pathway integrity would rely on expert system diagnostics that compare the defined routes to available topographic maps. Sample retrieval can be done via robotic arms utilizing infrared proximity sensors and a sample coring device. The robotic arms should be able to place the samples inside a closed module containing a number of separate storage compartments on a rotating track. The analyzer package on-board the *LRVs* must be capable of providing information on the chemical and mineralogical composition of the Lunar soil and volatiles. An important mission driver is the search for and retrieval of volatiles trapped within the soil and recoverable by heating. The analyzer package should consist of the following:

(1) X-ray fluorescent spectrometer (*XRF*) - analysis of trace elements.
(2) Gas chromatography - analysis of heat released gases from rock samples.
(3) Fourier transform infrared spectral analyzer (*FTIR*) - volatile searches.
(4) Inductive coupled plasma analysis - analysis of bulk chemistry.
(5) X-ray diffraction - mineralogical analysis of soil.

Table 02: Scientific Instruments on-board the *LPO*: Possible and recommended analysis instrumentation of the Lunar Rover Vehicle for initial resource analysis.

Teleoperated *LRAT* tethered to the main vehicles will provide access to craters, caves and lava tubes in the search for volatiles during the verification of remote sensing. Information on soil hardness and spectrographic analysis information will also be obtained both for calibration and verification of remote sensing information. By 2004 the more

detailed Lunar topographic and resource maps should provide manned missions with the databases required for a more effective and efficient development of a habitable manned outpost on the Lunar surface (see Figure 8).

5. Volatile Availability / Recovery

Following remote sensing calibration by the *LRVs* and *LRAT*, the question of volatile availability at the Lunar poles in shadows, caves, craters and lava tubes will be reasonably well resolved. Areas indicated as potential resource deposits via remote sensing will be searched. These sites can be located using the same search and navigation strategies as utilized for verification of the remote sensing, and will in all likelihood parallel such efforts. Gas chromatography of samples taken from the sites can not only serve as verification, but should provide an estimate of the quantity and recoverability of the resources. Available resources of interest are expected to be water, carbonaceous meteorites, metal deposits, and regions of regolith high in Ilmenite, silicon dioxide, or iron. Exploration for volatiles and resource availability in less accessible areas will have been carried out by the teleoperated *LRAT*. Utilizing existing high-survivability cart designs and tethers for connection with the main vehicles areas several hundred meters from the main vehicles will be explored. The *LRAT* are anticipated to provide science information on the interior of craters, caves and lava tubes. Highly accessible local resources will be collected utilizing specialized digging appendages and a 10 meter core drill. Recovered volatiles will be transferred into a trailer-towed N2 tank for transport to the Lunar Oasis and *EBV* for storage. The collection and mapping of high-quality resource deposits should enhance future manned missions. Manned missions should be able to plan on the *in situ* resources available to the *LOAR*, as well as on consumables which must be shipped to the Lunar surface.

6. Experimental Processing

Experimental processing will be carried out by both a robotic vehicle designated for processing and the bio-volume facility. The *LRV* will perform small-scale tests on experimental processing techniques such as material separation, fiberglass, concrete, and composite production from Lunar regolith. A 0.05 m^3 silicon-carbide ceramic furnace, (1500°C) powered by the vehicles and *RTG* waste heat, will test "regolith cooking" for volatile extraction. Vibratory screens and electromagnetism are anticipated to be the separation techniques employed to increase the expected yield from raw materials (e.g. Ilmenite separation for reduction by hydrogen to water).

In addition to feasibility testing, the *LRVs* will produce one specific Lunar composite. The *LRVs* will collect Lunar rocks and utilizing an earth-derived resin poured onto the rocks should be able to create a composite material. To protect the polymers from ultraviolet radiation the composites will be coated with a UV resistant paint. The composites can then be utilized in the exploration of building techniques for use on the Lunar surface. The *EBV* will perform moderate-scale regolith cooking for volatile recovery. Zonal centrifugal acid separation techniques should be explored for the recovery of other valuable components of the regolith such as the silicates and metal oxides. The high grade materials can be stock-piled and/or utilized in the *EBV* as inputs to the life-support system. For example, Ilmenite might be reduced utilizing the methane produced in the bioreactor. Outputs from this process would be carbon dioxide and water which could be stored and/or utilized in the biological experiments. Microbial acidification of Lunar regolith into the soil components suitable for higher plant growth will also be explored. Such microbial action

can be considered a first step in microbial resource recovery such as carbon extraction from carbonaceous meteorites.

7. Biological Experimentation

The utilization of local resources and the viability of primitive life forms will be tested on the Lunar surface. Information should be gathered impacting on the development of life-support systems not only for space based operations but for potential utilization on the Earth. A percentage of the *EBV* will be dedicated to a higher plant growth chamber, a dark box for the growth of fungus, spores, and sprouts, a soil composter, a *WETOX* digester, an aquaculture chamber and a bioreactor. Outputs from the higher plants will include transpired water, oxygen, and biomass. Outputs from the fungus, spores, and sprouts can potentially be fed into the bioreactor. The bioreactor will test waste processing by bacteria, algae, and primitive animals such as worms. The worms should perform soil conditioning and serve as a digestive mechanism for plant biomass. The bacteria and algae should remove the gaseous, liquid and small solid contaminants from the plants. The outputs of the bioreactor will include carbon dioxide, nitrates, and other chemicals which can be used to nourish the higher plants. The physicochemical systems working in concert with the biological systems must produce an atmosphere capable of sustaining life. Atmospheric control in the *EBV* coupled with plant growth and waste processing should provide valuable information in the development of *LECLSS* technologies (see Figure 9).

IV. HARDWARE:

1. Surface Support Equipment

Prior to manned missions to the *LOAR*, the following requisite equipment is envisioned to be in place: (1) Power Supply, (2) Heat Rejection System, (3) Storage Volume, (4) Airlock Access to *EBV*., (5) Communication Link.

(1) Power System Options:

Two different power requirements have to be fulfilled:

1) continuous power for the stationary Experimental Bio-Volumes, *EBV*.

2) power for the travelling Lunar Rover Vehicles, *LRV*, for locomotion, data acquisition and processing, and for analysis / processing of local resources.

Power systems being evaluated and compared:
Battery Power, *BP*. Regenerative Fuel Cells, *RFC*. Solarvoltaic Power, *SVP*. Solar Dynamic Power, *SDP*. Radioisotope Thermal Generator, *RTG*. Dynamic Isotope Power System, *DIPS*. Nuclear Reactor, *NR*.

Requirements and Constraints:

Vehicles (operating from polar site)	Bio-Volume (fixed at polar site)
nominal 5 kW_e electric power	nominal 10 kW_e electric power
changing position (moving)	fixed position close to poles (stationary)
sun not always visible (mountains, day/night cycle, seasons)	sunlight available for at least 6 month during year (polar site; max. sun elevation 1.5° above horizon.
position far away from base possible	n/a

Table 03: Requirements and constraints for power system for *LRV* and *EBV*.

Power System:	P_{max} [kW_e]	P_{spec} [W_e/kg]	Effic. [W_e/W_{th}]	Life [years]
Battery Power, BP.				
Regenerative Fuel Cells, RFC.				
Solarvoltaic Power, SVP.		50-60	20.0-30.0%	
Solar Dynamic Power, SDP.			25.0%	
Radioisotope Thermal Generator, RTG.	0.5 - 5.0	5.2 (9)	4.2- 6.6%	10 years
Dynamic Isotope Power System, DIPS.	1-10	> 6.5	18.0-24.0%	7 years
Nuclear Reactor (static; SP100)	> 10	22-33	6-12.0%	> 7 years
Advanced SP100	500-1,000	80		
Nuclear Reactor (dynamic)	1-100 MW	125		

Table 04: Power Systems Options: Systems may only be useful up to or from a certain power load on. In addition, all waste heat has to be rejected by a designated heat rejection system. When comparing system data, it is important to know whether the heat rejection system mass is already included in the 'mass per kW_e' numbers (in the above cited numbers for the nuclear power systems, radiator mass is included; data mostly from Ref. 6).

Energy Storage:	P_{max} [kW_e h]	P_{spec} [kg/(kW_e h)]	Life [years]
Battery Storage, Ni-Cd		50	
Battery Storage, Ni-H_2		33	
Regenerative Fuel Cells, *RFC*.		35	
NASA - long-term battery goal		10	
NASA - long-term *RFC* goal		2-3	

Table 05: Energy Storage Systems: For power systems that operate intermittently, for peak power or for back-up emergency power, energy storage has to be provided.

Conclusion:

Based on the mission requirements, a Dynamic Isotope Power System, *DIPS*, has been selected for the Lunar Rover Vehicles, *LRV*. The Experimental Bio-Volume, EBV, is powered by solarvoltaic arrays.

The *DIPS* has been selected for the *LRV* because of its higher efficiency (> 22% thermal to electric power) compared with *RTG*s and the greatest mobility of the *LRV*s. It will provide power independent from the location of the vehicles on the Moon and possible shading by mountains or the horizon. Energy storage as the main power source has been rejected as not feasible due to the desired life time and operation radius of the vehicles. Returning to the base (*EBV*) for refueling is also not desirable in the anticipated mission and operation plan. Storage would require excessive volumes / mass on the vehicles. With the *DIPS*, however, main problem would be safety concerns during launch. An alternative power source would be a solarvoltaic power system. A disadvantage to this alternative is that operation near the equator will only be possible for up to 14 days during the Lunar day; operation near the poles might not be feasible due to the low elevation of the sun above horizon and the possible shading by mountains during operation.

For the *EBV*, in contrary, the use of solarvoltaic power is the appropriate choice. With proper site selection based on Lunar Polar Orbiter remote sensing, a site providing at least 6 month continuous sunlight will be selected. There is a possibility that regions near the poles will have continuous sunlight during the whole year (on high, unshaded mountains or plains; also possible are light reflectors as relay stations). Minimal power storage with batteries would exist for emergencies. The considerations for a proper site selection can be seen in Figure 10.

(2) Heat Rejection Systems:

In space it becomes necessary to reject all the waste heat away for proper thermal control. Ultimately, all electric power will most likely be transformed into heat. Additionally, the waste heat of the power generation system has to be rejected, too (maybe as little as 2 times to up to 20 times the electric output energy of the power system). Heat 'transporting systems' (cooling loops, heat pipes, etc.) have to be installed between the heat rejection system and the heat source.

Power System	temperature for lowest mass	typical temperature
Thermoelectric Nuclear Power	775-875 K	750-950 K
Nuclear + Brayton Cycle	475 K	400-600 K
RTG	575 K	

Table 06: Typical temperature ranges for various power generating systems and the corresponding heat rejecting system. Typical temperatures are determined by the nature of the process and material constraints. Temperature for lowest mass are based on current technology for radiator material and system process. From Angelo (Ref. 6)

The only heat sink available is ultimately deep space. On a limited basis, the Lunar surface may be used. No other heat sink is available on a continuous basis. Parameters describing the heat rejection system are therefore: the radiating surface area, A, the temperature at which heat is radiated, T, and the thermal environment, T_a, to which heat is

radiated. Heat radiated is proportional to the exchange area, A, as well as the temperature difference (T^4 - T_a^4). The system area and therefore the total system mass can be reduced dramatically with higher radiator temperatures.

The following systems have been compared for the Lunar Rovers and the Experimental Bio-Volume:

Passive Radiators
Flash Evaporators
Liquid Droplet Radiators
Lunar Soil Mass Dump Cooling
Passive Lunar Soil Cooling

Requirements and Constraints for Heat Rejection System:

Vehicles	Bio-Volume
5 kW_e electric 20 kW thermal to be rejected changing position, mobile system continues thermal load for *RTG/DIPS* all sun elevations and orientations	10 kW_e 10 kW thermal to be rejected fixed position, stationary system power load varying with power usage sun always at horizon (polar site)

Table 07: Requirements and constraints for heat rejection system. The mobility of the vehicles and therefore changing environmental conditions are a severe constraint.

Rejecting System:	Temperature range	Mass per $kW_{thermal}$
Passive Radiators	200-1,500 K	
Flash Evaporators		
Liquid Droplet Radiators	550-1,000 K (T_{in}) 250-350 K (Si-oil)	
Lunar Soil Mass Dump Cooling		
Passive Lunar Soil Cooling		

Table 08: Comparison of Various Heat Rejection Systems.

Passive radiators have been selected for the Lunar Rover Vehicles arranged around the *DIPS* power generator together with further radiating surfaces on top of the vehicles and adjustable radiating surfaces on the sides. For operation near the poles, lowest radiating temperature (deep space) is vertically up. Radiators on the side may 'see' the sun (max. elevation above horizon 1.5°). The side radiators can either radiate heat vertically up (polar operation) or sideways (equatorial operation, operation during transportation). Of special concern is radiating the heat during transportation from Earth to the Lunar surface. Heat is generated by radioactive decay independent from actual electric power usage. Therefore, for the 5kW_e *DIPS*, 20-25 $kW_{thermal}$ have to be radiated continuously. During transportation, only the side radiators of the vehicle are available. Additionally, the coolant circuit of the *LRV* has to be connected to the coolant circuit of the *EBV* in order to use the heat radiating surfaces of the *EBV*.

(3) Storage Volume:

Following the philosophy of maximum resource utilization, the different propellant tanks of the Lunar Lander will be used for storage of processed and produced fluids. Storage is required for liquids (water) and gases (nitrogen, oxygen, carbon dioxide, methane, hydrogen). Wherever possible, the nitrogen tanks (attitude control thruster tanks) will be used for this purpose.

During the operation of the *EBVs* and *LRVs*, consumables may be produced and stored in a sufficient manner to fulfill human safety requirements for following missions. These buffer volumes will help to reduce the otherwise necessary additional mass to be delivered to the Lunar surface during the first human sortie missions.

(4) Airlock Access:

The proposed *EBV* will have hatches in each of the bulkheads of the main propellant tanks (hydrogen and oxygen). After configuring the *EBV* from the tanks, only the hatches in the oxygen tank (now airlock) will be used as airlock access to the Bio-Volume. Losses of gas have to be minimized during airlock operation. The airlocks have to be operated by the robot arms. The outside hatch to the hydrogen tank (now experimental volume) will only be used for the initial configuration and transportation of material stored outside the volume. The intertank bulkhead of the hydrogen tank has to be removed by the internal robot arm in order to give the robot access to the full volume. This is a critical point in the operation. Failure to remove the bulkhead will not allow configuration and operation of the EBV as foreseen.

(5) Communication Link:

Three communication links have been identified: 1) Earth to *EBV*; 2) Earth to *LRV*, 3) *LRV* to *EBV*. Due to the conditions on the Moon, communication is more or less limited to line-of-sight. Therefore special efforts are needed to ensure continuous communication links between all components of the *LOAR*. Possibilities include:

1) multiple communication satellites in Lunar orbit.
2) multiple communication satellite in a Halo-orbit around the Earth-Moon libration point, *L2*, on the far side of the Moon.
3) multiple relay stations on the surface to insure Earth view.

Discussion and Conclusion:

Satellite(s) around the libration point L2 have clear advantages. Only a small number of satellites (2-3) would be needed. The Halo-orbit would enable a constant view to Earth and the far side of the Moon (latitude limited). Therefore, the *L2*-satellite would allow for far-side exploration by the *LRV*s. On the other hand, communication via satellites in *L2* would disturb the radio-silence in the Moon's shadow desired by radio-astronomers.

Line of sight between the *EBV* and the libration point satellites is in general guaranteed. Exploration along and across the terminator between far and near side might not be possible when using the libration point satellites (locations not visible neither from satellites nor from Earth). Exploration on the near side, at a sufficient distance from the

poles, will be possible using direct communication link to Earth (comparable to the Apollo-program).

2. Surface Delivery - Lunar Lander:

Table 9 lists the performance criteria for delivering payloads to the Lunar surface. Two different systems have been analyzed: 1) a cryogenic system, using Liquid Hydrogen, *LH2*, and Liquid Oxygen, *LOX*, and 2) a Earth-storable system, using Mono-Methyl Hydrazine, *MMH*, and Nitrogen Tetroxide, *NTO*.

Delta-V from *LEO* to *LO*	4,100. m/s
Delta-V from LO to *LS*	1,900. m/s
Total Delta-V	6,000. m/s
$I_{sp\ vac}$ Hydrogen-Oxygen (*LOX-LH2*; 460 s)	4,500. m/s
$I_{sp\ vac}$ MM-Hydrazine - Nitrogen Tetroxide (*MMH-NTO*; 346 s)	3,400. m/s

Table 09: Requirements and Constraints for Lunar Lander Propulsion System.

Main Thruster Selection:

a) general philosophy
- restartable
- throttlability 35% to 100%
- two thrusters, one on each end
- $V_{max\ landing}$ = 1m/s, target: 0 m/s.

b) Minimum Thrust:
- hovering above surface with minimum mass (= dry mass)
- dry mass = 8770 kg

$g_{moon} = 1.62\ m/s^2$
$F_{min} = 0.5 * 8770\ kg * 1.62\ m/s^2 = 7104\ N/_{engine}$
$F_{min} = 0.35 * F_{norm}$; $F_{norm} = 20\ kN/_{engine}$

c) max. acceleration in Lunar Orbit (LOX-LH2):
$m = m_{dry} = 8770\ kg$; $F = F_{max} = 2 * 20\ kN = 40\ kN$
$a_{max} = F_{max} / m_{dry} = 4.6\ m/s^2 = 0.47 * g_0$

d) max. acceleration in Low Earth Orbit (LOX-LH2):
$m = m_{wet} = 33{,}270\ kg$; $F = F_{max} = 40\ kN$
$a_{max} = F_{max} / m_{wet} = 3{,}0\ m/s^2 = 0.31 * g_0$

e) Assumption / Development Needs:
- small, throttlable cryogenic thrusters do not exist to date.
- long-term cryogenic storage (weeks) will be needed, if hydrogen-oxygen thrusters will be used for the outlined mission.
- Hydrazine-Nitrogen Tetroxide thrusters in the required thrust range are readily available.

Attitude Control System:
- 4 thruster (2 pairs) on each side: 4 * 4 = 16 thruster. - 3-axis maneuverability - thrust per thruster: 450 N - propellant: 265 kg - fuel: thermal augmented nitrogen thruster from high pressure storage.

Trade-Off between Storable and Cryogenic Propulsion System:

	Cryogenic *LOX-LH2*	Storable *NTO-MMH*
$I_{sp\ vac}$	4,500 m/s	3,400 m/s
$m_{payload}$	8,770 kg	8,770 kg
propellant *LO-LS*	4,600 kg	6,565 kg
mass in *LO*	13,370 kg	15,335 kg
propellant *LEO-LO*	19,900 kg	35,885 kg
total mass in *LEO*	33,270 kg	51,220 kg

Table 10: Mass Trade-Off between an all-cryogenic propulsion system, using liquid oxygen and liquid hydrogen, and a all-storable propulsion system, using Mono-Methyl Hydrazine and Nitrogen Tetroxide. A hybrid system, using different propulsion systems for Lunar transfer and Lunar descent, have not been considered. Tankage volume for the cryogenic system is twice the volume for the storable system, which also increases the mass of the cryogenic tanks (also more insulation required). This mass increase has not been included in the calculations. Note that the planned Shuttle-C launch capability is 45,000 kg into Space Station orbit.

	Cryogenic *LOX-LH2*	Storable *NTO-MMH*
oxidizer density	1,140 kg/m^3	1,447 kg/m^3
fuel density	71 kg/m^3	878 kg/m^3
mixture ratio	6:1	2:1
$m_{propellant\ total}$	24,500 kg	42,450 kg
m_{ox}	21,500 kg	28,300 kg
m_{fuel}	3,500 kg	14,150 kg
V_{ox}	18.4 m^3	19.6 m^3
V_{fuel}	50.3 m^3	16.1 m^3
V_{total}	68.7 m^3	35.7 m^3

Table 11: Required tank volumes for cryogenic versus storable propellants. The cryogenic system requires much larger tank volumes due to the low density of hydrogen. However, this increase of volume is desirable to use the tank as the Bio-Volume.

Discussion and Selection of Systems:

The cryogenic hydrogen-oxygen system has been chosen as the preferred system due to the dramatic reduction in mass to be delivered into *LEO* and due to the advantages for system integration and reusability of left-over propellants. Total mass launched to *LEO* is approximately 1/2 when compared to a Earth-storable system. Leftover hydrogen-oxygen may be combined into water for use in the Bio-Volume; oxygen will be used for the initial atmosphere within the *EBV*. Hydrogen may be used for an experimental Ilmenite reduction experiment. The increased complexity resulting from the use of cryogenic fuels is a disadvantage for this selection (cryogenic temperatures). Also, cryogenic thrusters with the described thrust range (20 kN thrust) do not exist.

The *MMH*ydrazine-*NTO* system has a lower specific impulse than the cryogenic system and the total mass required is almost two-fold when compared with the cryogenic system. Use of left-over propellants is more limited. Nitrogen Tetroxide could be decomposed catalytically into oxygen and nitrogen for the initial atmosphere of the *EBV*. Mono-Methyl Hydrazine is less likely to be decomposable into useful, non-toxic gases. The use of Hydrazine, instead of *MMH*, would alleviate the decomposition concerns, however, the reduction in specific impulse would present a propellant mass penalty.

MMH and *NTO* have to be stored at Earth ambient temperatures (*MMH* above freezing point of water). Temperature-wise, the *MMH-NTO* system is more compatible with the idea of a "*Wet Laboratory*", where the interior of the propellant tanks has pre-installed equipment for the Experimental Bio-Volume, *EBV*. Volume-wise, the cryogenic system requires twice the volume of the *MMH-NTO* system due to the low density of hydrogen. However, for the future use of the Bio-Volume, the larger volume is actually desirable. It should be noted that the total mass of the *MMH-NTO* system would exceed the launch capability of the Shuttle-C launcher system in its current configuration.

Another possibility is a hybrid system, with separate systems for Lunar transfer (Earth Orbit to Lunar Orbit) and Lunar landing. This would increase the system complexity and it is less likely that such a system could be launched in the proposed manner. Re-use of all mass delivered to the Lunar surface would also be more difficult than with the selected all-cryogenic system.

Delivery Missions Required:

Payload	*Launch-Vehicle*	*Mass into LEO*
LRV-1, EBV-1	1. Shuttle-C flight	33,270 kg
LRV-2, EBV-2	2. Shuttle-C flight	33,270 kg
EBV-3, Consumables, Analyzer, Support Equipment	3. Shuttle-C flight	33,270 kg

Table 12: Required launches for implementation of *LOAR*. Payload mass from Earth surface into *LEO* include the hardware and the cryogenic propellants for Lunar transfer and Lunar landing. With Earth storable propellants (*MMH-NTO*), the initial mass in *LEO* would exceed the payload capability of the Shuttle-C (45,000 kg into Space Station orbit) and two launches per Lunar Lander would be required.

3. Vehicles:

The vehicles will be designed and built using many existing technologies with proven success rates and reliability. Advanced robotic and automated system designs will be utilized, such that the next generation of robotic systems will no longer simply observe and interact passively with the environment. Rather, these systems will be capable of autonomous interaction with, and manipulation of the environment.

Design Guidelines:
(1) *Highly redundant:* Multiple sensors, effectors and control systems. (2) *Self-similarity:* Modular components and functional exchanges. (3) *Existing technology:* Reliability tested, low cost, accurate integrations. (4) *Simple mission profiles:* Planned, mapped, structured tasks. (5) *Environmental and situational predictability:* Extensive sensing, internalized maps, "learning". (6) *Modular system upgrades:* Phased technology demonstrations (7) Common power bus, standard hardware templates, standard mechanical and electrical connectors.

Table 13: Design Guidelines for the Lunar Rover Vehicles, *LRV*.

General System Recommendations:

(1) The baseline models in space robotic development were the Lunar Roving Vehicles of the Apollo era, as well as the Mars Viking Lander. The *LRV* mass of 1,500 kg each is twice that of the Lunar rovers. Each vehicle should be capable of towing/winching approximately 1.5 times their vehicular weight and manipulating 250 kilograms. The vehicles are anticipated to be six-wheeled flat platforms of approximately 10.5 square meters (4.2 m length, 2.5 m width), with attachment sites for the modular components. Advanced materials such as aluminum alloys and composites should be utilized to increase the strength to weight ratio. Composites can be made highly resistant to general wear, and laminated to minimize their coefficients of expansion.

(2) The vehicles will be similar and modular in design. Specialized components will be utilized for specific tasks. Functional overlap between the following subsystems is anticipated: Power supply systems (*RTG* and *DIPS*, battery back-up), thermal control systems (heat rejectors, 5 to 10 times the electric power), meteoroid protection shields, the guidance and control systems, antennas, the electric drive motors (3 KW each, 4 HP total), on-board computers for autonomous operation (maneuvering, geological analysis) and the manipulator arms for self-maintenance as well as for surface operations. Robust systems with an emphasis on parallel delivery of information, sensory input, power, and computation through multiple pathways will provide redundancy in all levels of operation.

(3) Task and mission scenarios will be designed to place minimal requirements on the computational systems. Accordingly all mission scenarios and tasks should be designed in a fashion conducive to robotic implementation.

For geologic analysis, the modular packages include drilling equipment for core samples and automated laboratory equipment (see Table 14: Geologic Analysis Systems). For materials processing, the modular packages include mechanical separators for material sorting and mechanical testing rigs for material evaluations (see Table 15: Materials Processing Systems).

GEOLOGIC ANALYSIS SYSTEMS:	
Sample collection:	Drill, manipulator arm, teleoperated *LRAT*
Sample preparation:	Core-drill, cutter, polisher
Physical properties:	Density, hardness, temperature, thermal conducvity, melting point, magnetism, solubility (consumables water and acids)
Optical Analysis:	Polishing facility, microscope
Composition:	Gas Chromatography / thermal release of volatiles X-ray fluorescent spectrometer for analysis of solids
Geophysical data:	Magnetometer, gravimeter, vibrator, seismometers

Table 14: Geological Analysis Systems for *LRV*.

MATERIALS PROCESSING SYSTEMS:	
Pre-processing:	Sieves, separator, storage, transportation
Sample analysis:	Strength tests, loading/bending, out-gassing, hardness
Materials Processing:	Lunar composites, aggregate binding, environmental endurance, fiberglass, fiberglass winding technologies.
Additional Requirements:	(1) consumables for analyzer laboratory (2) spare parts for vehicles and modules (3) manipulation of landed volumes / material-crane.

Table 15: Materials processing Systems for *LRV*.

VEHICLE SPECIFICATIONS:	
Total mass with payload	1,500. - 3,000. kg
RTG for 5 KW_e	800. kg (2m * 2m * 0.25m)
DIPS for 5 KW_e	500. kg (1.32m * D0.65m)
Structure	500. kg
Scientific Instruments	300. - 2,000. kg
Radio / communication	100. kg
Computer / navigation	100. kg
Computer / analyzer / data processing	100. kg
Propulsion power	3. kW_e (4 HP) for electric motors
Payload power	2. kW_e
Maximum slope	20.° or 36%
Maximum speed	10. km/h

Table 16: Vehicle Specifications for the Lunar Rover Vehicle, *LRV*.

3. Bio-Volume:

The reconfigurable bio-volume will consist of two major parts: The airlock and the experimental volume. The airlock (the previous oxygen tank) will provide access to the pressurized interior (see Table 19: Bio-volume Atmosphere Control). The experimental volume (the previous hydrogen tank and the intertank section) will provide space for experiments as well as control and monitoring equipment. During the automated phase of the *LOAR* operation biological experiments for a Lunar *ECLSS* will be conducted. Manned sorties starting in 2004 can utilize the volumes to provide temporary shelter and life support consumables. The proposed mission scenario requires three *LEL* landings prior to any human mission. Each of the three bio-volumes will provide reconfigurable volumes for the development of life supporting modules, a laboratory area and control centers. Activities within the experimental volume will include plant growth, biological waste treatment (bioreactor) and small-scale animal experiments (aquaculture and soil processing). The use of Lunar soil as a growth medium and the recovery of trace elements from the soil via microbial acidification and extraction will be investigated. The *EBV* will provide the first non-terrestrial *ECLSS* capabilities. Plants, animals and microorganisms will live in a physicochemically buffered symbiosis. Robotic arms will provide internal manipulation of hardware for all experiments. Buffer volumes will be maintained for carbon dioxide, oxygen, nitrogen and water. Capitalizing on local resources (trapped volatiles such as water, carbon form carbonaceous meteorites) the buffer volumes can potentially be increased prior to human visits. Results obtained from the experiments will elucidate important questions in the design of a *LECLSS* (see Table 18: Bio-volume Specifications).

Design Guidelines:	
(1)	Highly redundant growth chambers.
(2)	Autonomous: Self contained power and regulatory systems.
(3)	Multiple sensors and experimental monitoring systems.
(4)	Phased technology demonstrations: Lunar processing, *in situ* resource recovery.
(5)	Utilization of available *in situ* resources: Volatiles, carbon, ambient temperature, artificial light.
(6)	Modular adaptability for manned habitation.

Table 17: Design Guidelines for *EBV* Design.

BIO-VOLUME SPECIFICATIONS:	
Module:	recycled from cryogenic propellant tank.
Airlock (oxygen tank):	L = 1 m, D = 4.5 m, volume = 18.5 m^3.
Bio-volume (*LH2* tank):	L = 5 m, D = 4.5 m, volume = 60 m^3.
Total Mass:	3,500 kg (empty tank = 1,250 kg, equipment = 2,000 kg, buffer = 250 kg).
Total power:	10 KW_e (30 m^2 solar array), regenerative fuel- cell back-up.
Heat rejection:	30 m^2 solar radiator.

Table 18: Bio-Volume Specifications.

BIO-VOLUME ATMOSPHERE CONTROL:	
Volume:	3 m^3 (intertank).
Mass:	300 kg.
Power:	2 KW_e.
Atmosphere:	Control temperature, humidity, composition, temperature, pressure, nitrogen 75 - 95%, oxygen 5 - 25%, carbon dioxide 25 - 5000 ppm.
Physicochemical systems:	
CO_2 absorption:	Solid Amine process.
O_2 absorption:	Salcomine process.
CO_2 conversion into H_2O:	Sabatier.
Oxygen generation:	Electrolysis (H_2O into O_2 and H_2).
Humidity / Temperature control:	cold-plate condenser, heat exchanger.

Table 19: Bio-Volume Atmosphere Control.

The contents of the bio-volume consist of multiple-start, stored biological systems including complex, "wild-type" lyophilized cultures of microorganisms. Adapted soil and water cultures will range from N_2 and CO_2 fixing organisms to soil acidification / digestion organisms. Higher plant seeds will be maintained in sufficient amounts to employ a variety

of re-seeding technologies. Using "batch mode" operations, materials will periodically be changed-out between growth and production environments. Human consumables will be stored for subsequent manned missions. The net N_2 fixation losses will be restored from cold jet tankage supplies and carbon fixation losses from derived Lunar carbon resources. Lunar soil exhausted of carbon reserves by composter activities will be returned to the Lunar surface. Other bio-volume contents and systems are listed (see Table 20).

BIO-VOLUME SYSTEMS:
Higher plants. Bioreactor - algae, bacteria. Fungus, spores, sprouts. Aquaculture. *WETOX* digester. Atmospheric control system / humidity control. Hydroponic nutrient delivery system - for higher plants. System controller and data acquisition system. Buffers - gasses, liquids.

Table 20: Bio-Volume Systems. Selected species for initial Experimental Bio-Volume operation.

The bio-volume uses the buffer and process rate mixes arising from 5 major systems. The composter and *WETOX* systems focus on long-term and short-term, respectively, elemental extractions and bio-conversion. The bioreactor provides rudimentary gas release and fixations. Finally, the higher plant and aquaculture systems provide consumable products, most of which are candidates for storage (see Table 21: Bio-volume Commitments).

BIO-VOLUME COMMITMENTS:	
Higher plants:	Volume: 8 m^3 growth space. Power: 0.7 KW_e/m^2 = 5.6 KW_e. Temperature: 20 - 25°C. Hydroponic growth media. Artificial lighting. Transpiration water recovery (centrifugal, dew point precipitator). Lunar Soil (processed).
Bioreactor:	Volume: 2.0 m^3. Temperature: 20°C. Algae, bacteria, microorganisms. Gas fixation / release. Filtered recoveries / separation of species.

Aquaculture:	Volume - 0.5 m^3. Temperature: 15 - 25°C. Eggs, brine shrimp, live cultures. Bacteria, micro-algae. Controlled P_{CO2}, P_{O2}.
Composter:	Volume: 1.0 m^3. Temperature: 10 - 30°C. Pressure: 300 kPA. Mixing: Auger. Atmosphere: High O_2.
WETOX:	Volume: 0.5 m^3. Temperature: Ambient to 700°C (microwave heat). Pressure: Ambient to 7 MPa.
Buffer volumes:	Gasses, liquids (approx. 10 m^3, total). Plant materials such as seeds, spores, lyophilized algae and bacteria (1 m^3). Carbon sources (CO_2, recovered meteorite carbon).

Table 21: Bio-Volume Commitments.

V. COST AND COMMERCIAL CONSIDERATIONS:

Business as usual will just not do! The cost of doing business with large administrative overhead drains financial resources which otherwise could be applied to achieving results. By establishing less rigorous reliability requirements and accepting a certain degree of failure, costs may be minimized while performance is realized through redundancy. Perhaps such a scenario could be best implemented by small companies with academic partners. To avoid administrative overhead due to concerns of technology transfer, no international interaction will be sought. To encourage investments by the private sector, companies could retain technology exclusivity. Both business and academia could benefit while providing a low cost Lunar mission. Many safety issues as noted above, will be put aside since manned missions will come later. An estimate of the budget for the implementation and operation of *LOAR* under these conditions is given in Table 22.

Estimated Cost:	
Launch Cost	1.0 Billion
Research and Development	2.5 Billion
Production	1.0 Billion
Operations	0.5 Billion
Total Cost	**5.0 Billion**

Table 22: Estimated Cost for Implementation and operation of the Lunar Oasis, *LOAR*.

Commercial Considerations:

Private industry will be jointly vested in homesteading the Lunar surface. Resources identified and staged through Lunar based operations may be of value to the infrastructure in the future. For example, sufficient life support expendables may be stored such that week-long human stays may be facilitated. The "Moon-Steading" partners would receive financial compensation for the use of these resources. At the same time, the new waste products added to the system will allow the bio-system mass to expand and accommodate larger future manned missions (and greater financial returns). Similarly, Lunar resources identified and utilized for infrastructure support also would yield a return on the investment made.

VI. CONCLUSION:

The initiation of Lunar activities via automation and robotic will enable effective low-cost implementation of the *LOAR* project prior to manned operations. Databases containing the topographic features of the Lunar surface, and the availability and recoverability of volatiles and Lunar resources are completed. Future manned missions spanning the period of 2004 - 2010 finish and elaborate the fully manned *LECLSS*. Utilization of available resources should be optimized. Potentially, the shipment of consumables from the Lunar surface to supplement the cis-Lunar infrastructure can be undertaken. This might be a commercial activity. The propulsive system utilized in the delivery of Lunar resources is projected to be a mass launcher. The far-side synthetic aperture radio observatory as well as remote seismic and gravitational experiments should be enhanced. Manned re-configuration of the experimental bio-volumes into the habitable *LOAR* should allow manned sorties to complete the deployment of scientific apparatus or repair and service them as necessary. The development and growth of an effective cis-Lunar infrastructure is anticipated to rely heavily on *A/R* and manned projects such as *LOAR* (Figure 10).

Preliminary data from the *EBV* will allow a quantitative comparison between the capabilities of a terrestrial life-support system and a Lunar life-support system. Information gathered will not only impact space-based life-support systems, but should impact on the development and implementation of *ECLSS* technologies on the Earth. Commercial interests in life-support systems and advanced robotic will stimulate the design of these systems, as well as providing potential Earth-based industrial utilization of the technologies developed. The phased mission scenario emphasizes a shift in space based operations to a more cost-effective, efficient system in which automation and robotic coupled with manned missions can play a crucial role in the development of the cis-Lunar infrastructure. The *LOAR* programme highlights the possibilities of this cooperative type endeavor for future activities in the space environment.

To permit implementation at modest cost and with maximum technical impact, the *LOAR* is projected to cost 5 billion dollars. Technical manpower will be leveraged through academia involvement and business/industry competitiveness internationally will be assured by corporate involvements. The *LOAR* will be implemented by these entities and subsequently interfaced with the NASA manned space programme elements.

VII. REFERENCES:

(1) National Commission on Space; Pioneering the Space Frontier. New York: Bantam Books, 1986.
(2) Ride, Sally K., Leadership and Americas Future in Space.
(3) Buck, C.A., Johnson, A.S., McGlinchey, J.M., Ryan, K.D., Luttges, M.W., The Development Of A Cis-Lunar Space Infrastructure, NASA/USRA Conference 1988.
(4) Hopfield, J.J., Tank, D.W., Computing With Neural Circuits: A Model, Science 233:625-633.
(5) Burke, James D., Merits of a Lunar Polar Base Location, In: Lunar Bases and Space Activities of the 21st Century. W.W. Mendell (ed.). Lunar and Planetary Institute, Houston.
(6) Joseph A. Angelo, Jr., Ph.D., David Buden, Space Nuclear Power, Orbit Book Company, Inc., Malabar, Florida, 1985.
(7) SSI Special Report: Lunar Prospector Probe Project. Space Studies Institute, Princeton, N.J.

FIGURES FOR SUMMARY REPORT:

Figure 1: The cis-Lunar Infrastructure: Showing its key elements the Space Station Freedom at *LEO*, the *L1*-Station and an Orbital Transfer Vehicle. Robotic Vehicle, *LRV*, and Experimental Bio-Volume, *EBV*, pictured on the Lunar surface.

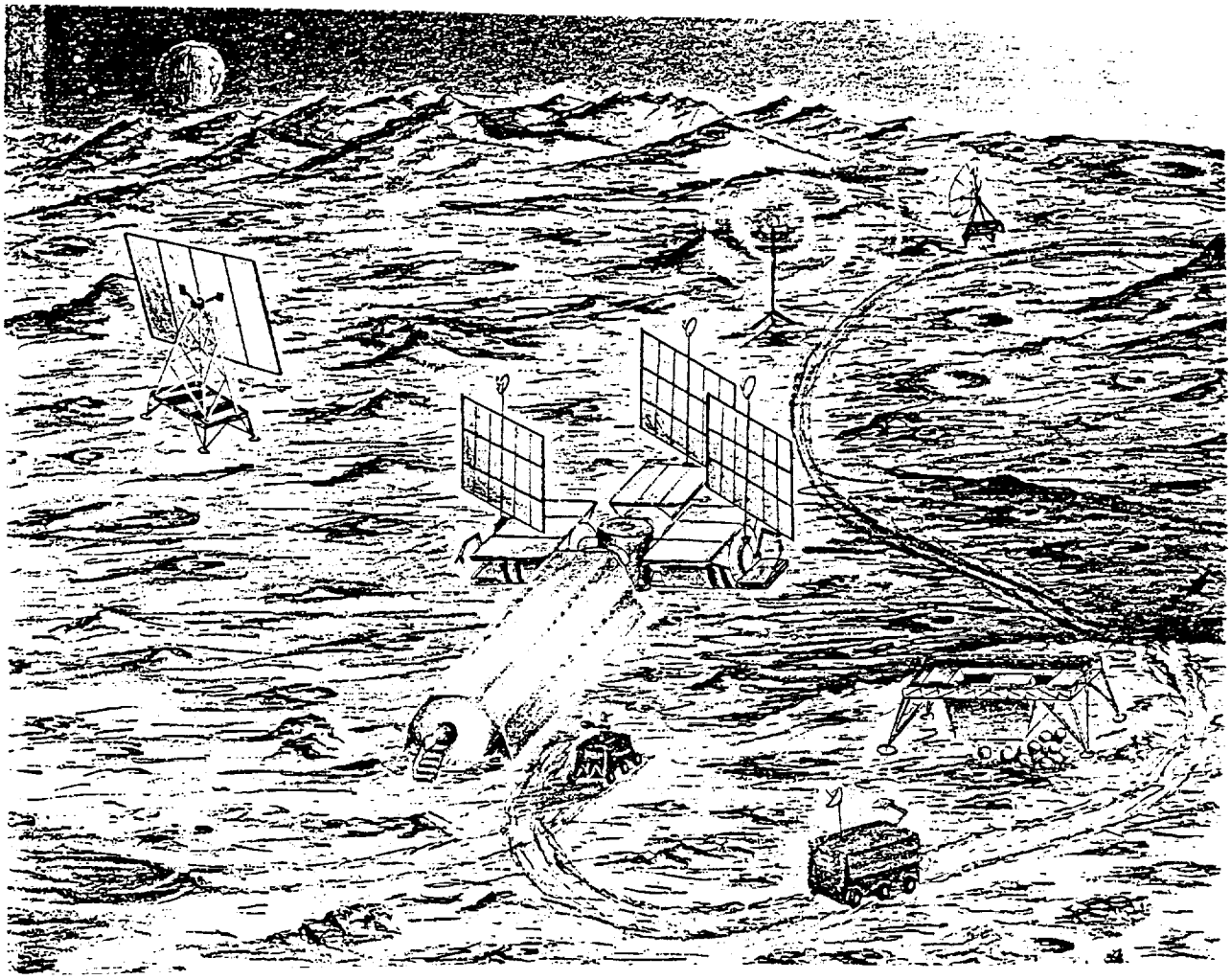


Figure 2: The envisioned *LOAR* following manned operations. Shown are the three combined *EBV*s, together with an additional habitation module. The two *LRV*s are shown in the foreground.

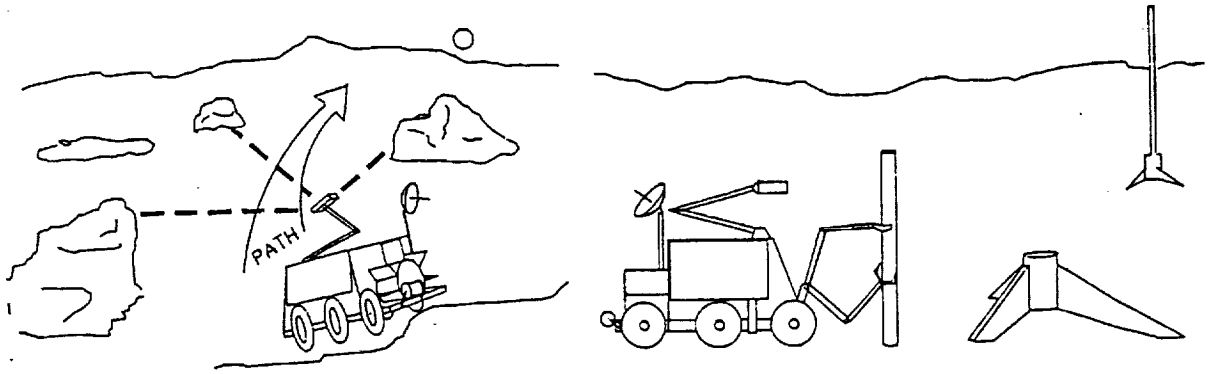


Figure 3: The Lunar Rover Vehicle, *LRV*. Vehicle on left is shown utilizing pulsed-laser ranging and its neural network for navigation on the Lunar surface. Vehicle on the right is shown deploying one of the triangulation beacons.

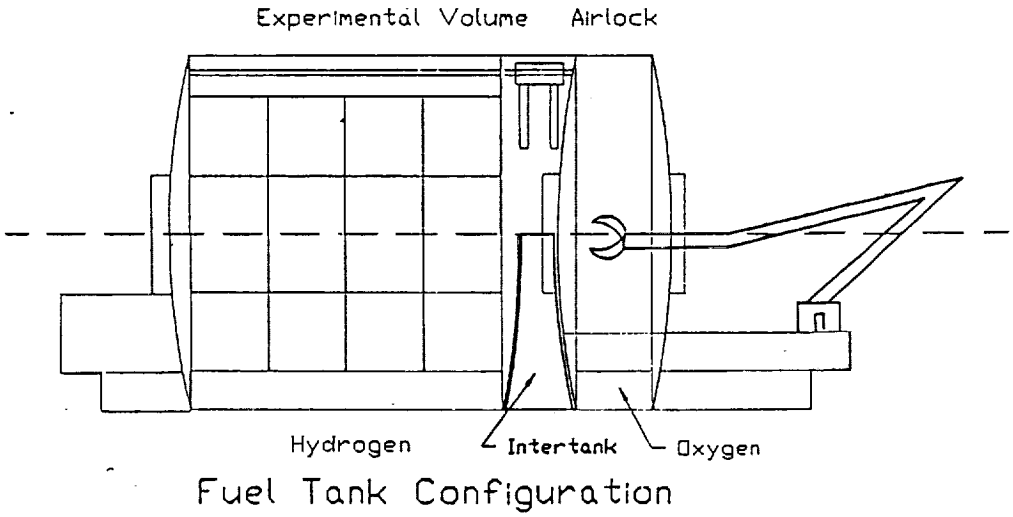


Figure 4: The experimental bio-volume, *EBV*, is shown prior to re-configuration. Used as the propellant tanks during delivery, it will be re-configured into the experimental bio-volume.

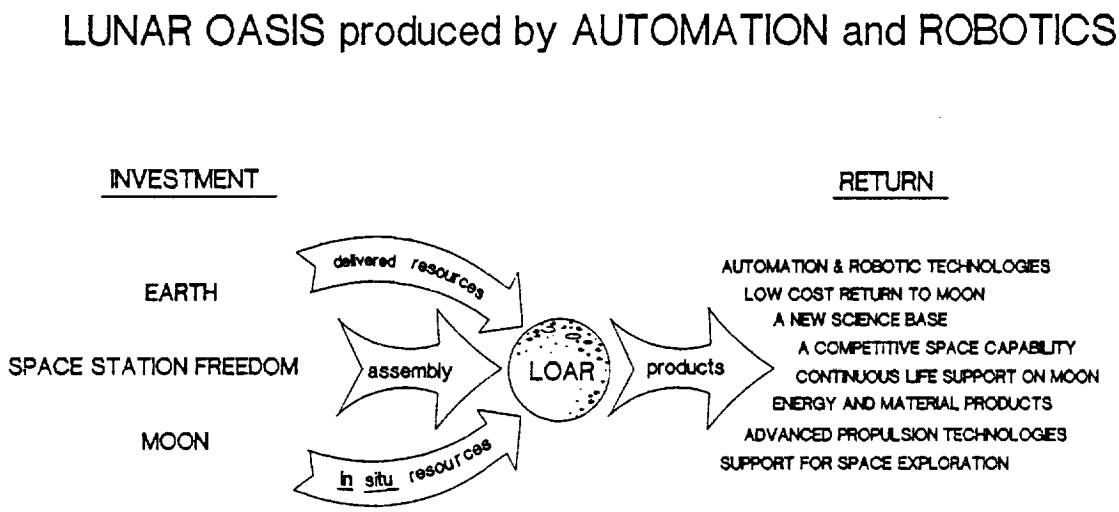


Figure 5: The investment and anticipated returns from the *LOAR* programme.

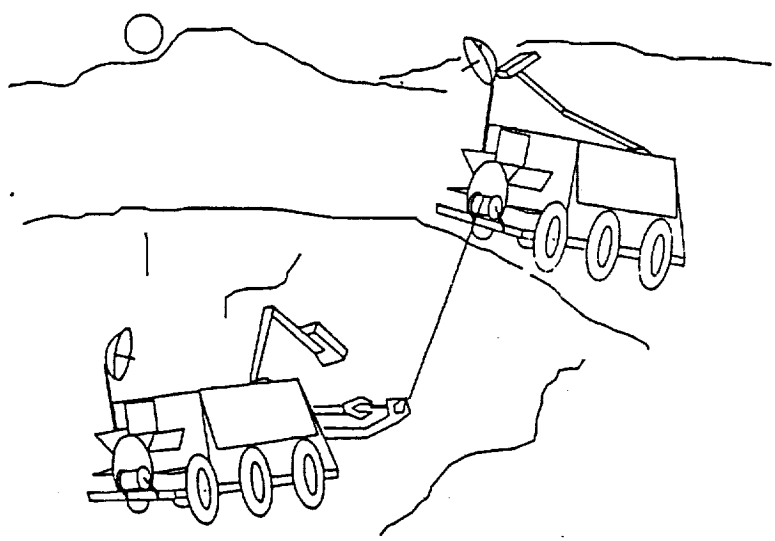
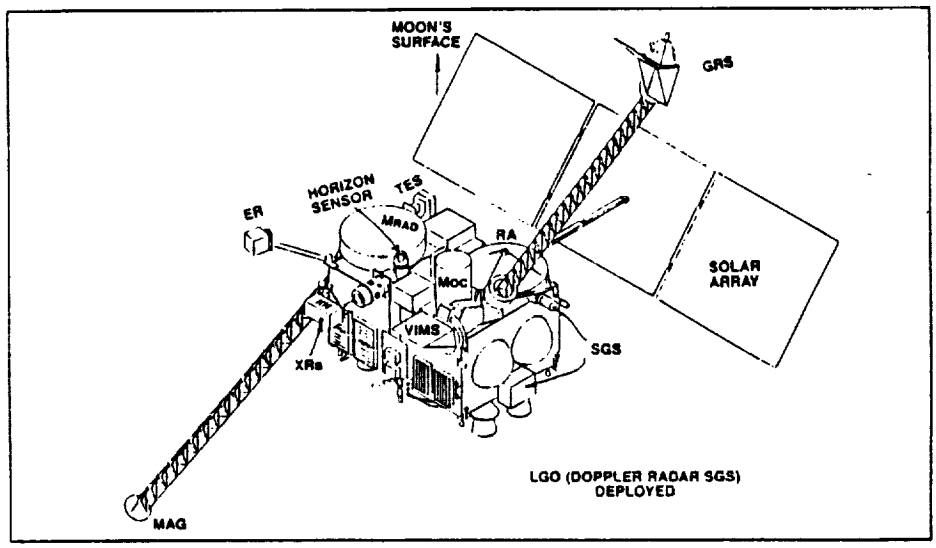


Figure 6: The "*Buddy System*": Utilizing '*Manual Override*' through teleoperation, the two vehicles can assist each other when required.

INSTRUMENT/EXPERIMENT CANDIDATES
(FOR AN ORBITING SPACECRAFT)

INSTR./EXPER.	ABBREV.	MEASUREMENT(S)
HIGH RESOLUTION IMAGER	(IMAGER)	SURF. MORPHOL., ALBEDO DISTRIBUTION
GAMMA RAY SPECTROMETER	(GRS)	SURF. ELEMENT. COMPOS. (U, Th, K, Si, Al, Mg, Fe, Ti, Ca, S, O, H, C)
X-RAY SPECTROMETER	(XRS)	SURF. ELEMENT COMPOS. (Si, Al, Mg; Poss. Fe, Ca, Ti, K)
RADAR ALTIMETER	(ALTIM)	SURF. TOPOGRAPHIC (ELEVATION) PROFILES
MAGNETOMETER	(MAG)	MAGNETIC FIELDS (INTERNAL)
ELECTRON REFLECTOMETER	(ER)	MAGNETIC FIELDS (SURFACE)
GRAVITY (DSN DOPPLER)	(GRAV-DSN)	SUBSURF. MASS DISTRIBUTION
GRAVITY (WITH SUBSATELLITE RETROREFLECTOR)	(GRAV-SUB)	SUBSURF. MASS DISTRIBUTION
VISIBLE/IR MAPPING SPECTROM	(VIMS)	SURFACE MINERAL COMPOSITION
THERMAL/IR MAPPING SPECTROM	(TIMS)	SURF. MINERAL COMPOS. & TEXTURE
MICROWAVE RADIOMETER	(MRAD)	SUBSURFACE TEMPERATURE GRADIENT



NASA lunar observer and experiments

Figure 7: Possible Instrumentation of the Lunar Polar Orbiter, *LPO*, (from Ref. 7).

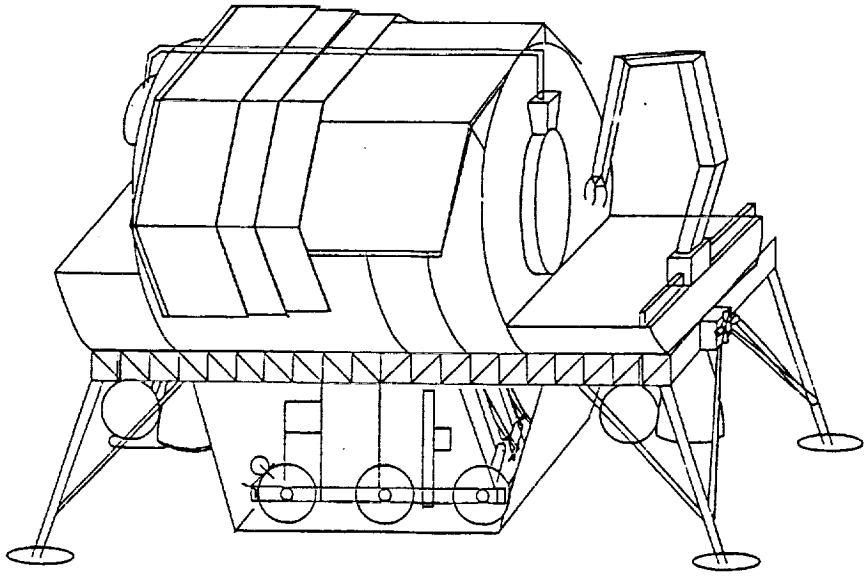


Figure 8: The landing configuration of the proposed Lunar Expendable Lander, *LEL*.

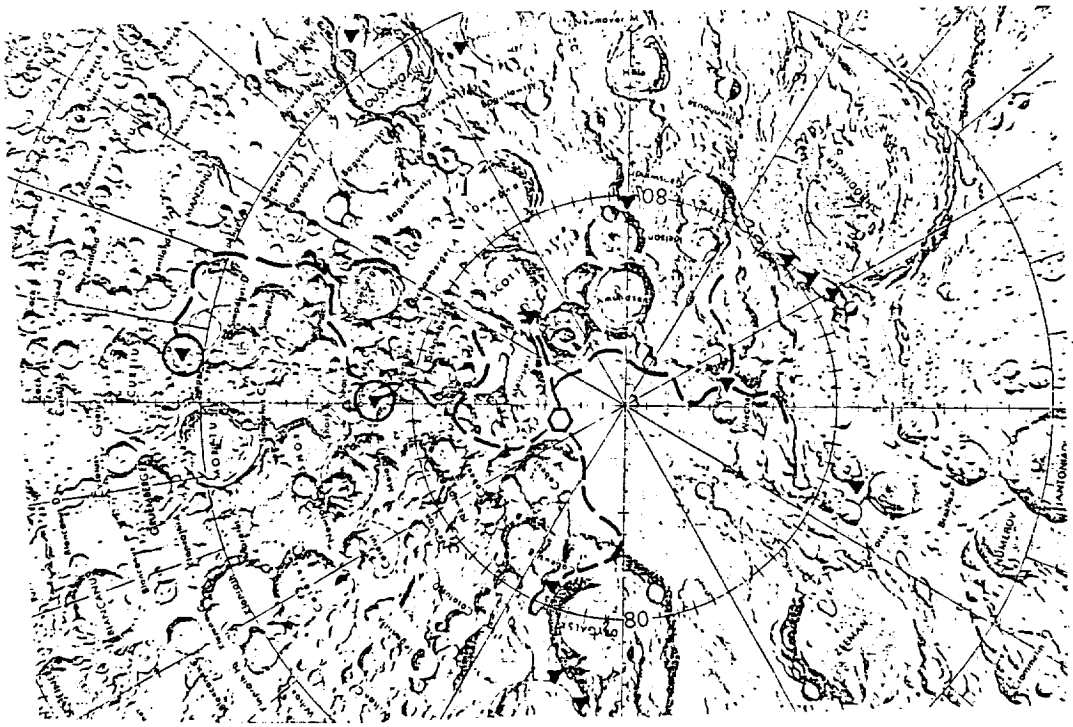


Figure 9: Possible map of the Lunar South Pole showing the Lunar Oasis (hexagon), volatiles and resources identified by the *LPO* (triangle), verification of LPO information by *LRV* (circle/triangle) and navigable pathways (dashed line).

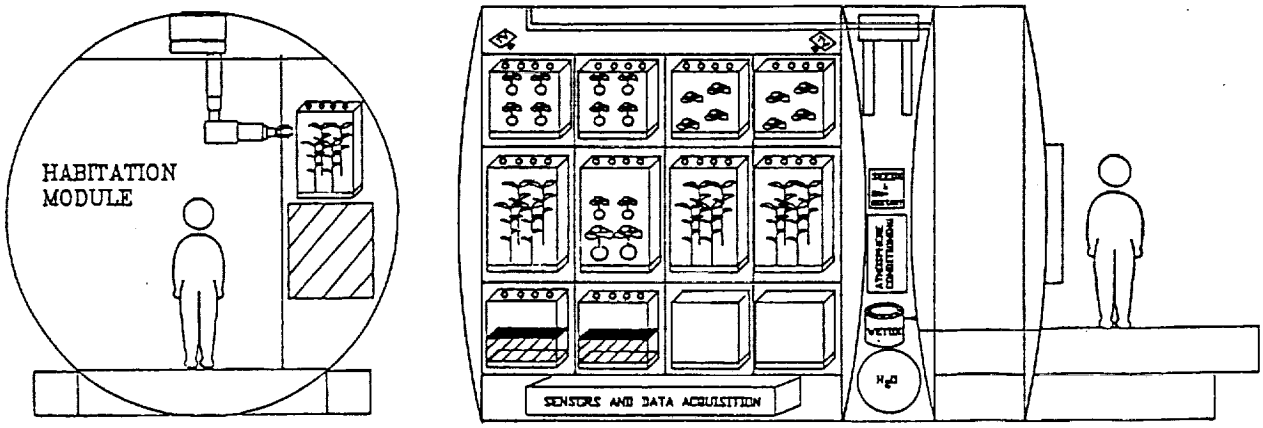


Figure 10: The Experimental Bio-Volume, *EBV*, following re-configuration and implementation of the biological systems. Shown are the higher plant growth chambers.

SITING OF A LUNAR BASE

Design Driver		Polar North/South	Equator Far/Near	Earth Terminator
Mission:	*Resource:He^3*	little/no	yes	yes
	Resource:Ilmenite	maybe	Far:little Near:yes	little
	Resource:Hydrogen	little/no	solar wind	solar wind
	Resource:Volatiles	trapped in shade	unlikely	unlikely
	Tech.Demonstrator			
	Commercial Potential			
Mission:	*Science*			
	Exploration	very little known	F:little known N:more known	little known
	Astrophysic/Sky	S:little known N:more explored	F:EM Shielding N:Earth EM noise	Earth noise
	IR-astronomy	cryogenic in shade	artificial cooling	artificial cooling
Power:	*Solar Power Avail.*	0.5 year day/night	14 day day/night	14 day day/night
	Solar Tracking	360 Degree tracking	180 Degree tracking	180 Degree tracking
	Heat Sink	high Delta T/crater	no shade at full sun	lattitude dependent
	Energy Storage	for 0.5 years	for 14 days	for 14 days
Safety:	*Solar Wind*	craters for shielding	no natural shielding	no natural shielding
	Solar Flares	craters for shielding	no natural shielding	no natural shielding
	Cosmic Radiation	no benefits	no benefits	no benefits
	Meteroids	no benefits	N:Earth shielding	no benefits
	Communication	14d Earth visible	N:Earth always F:earth not visible	mostly visible
	Accessibility	always from polar orbit	always from equatorial	limited to certain launch window
	Corrosion		H_2-embrittlement? temperature variation	H_2-embrittlement temperature variation
	Operation		F:disturbing EM-silence	
Environment:	*Visibility*	shade/dark	long twilight	long twilight
	Temperature Var.	little	high (14d)	high (14d)
	dust	same	same	same
	illumination	constant, long shade	changing	changing
Flexibility: Expandability:	*small area on moon* *(limited to 80km diam.)*	ample space	ample space	

Figure 11: Siting considerations for the selection of the base location. Based on the advantages, a polar site has been selected as the best location for the 2 Experimental Bio-Volumes, *EBV*. Possible continuous availability of sunlight, accessibility to most Lunar features were the main drivers.

Appendix I:

CIS-LUNAR SPACE INFRASTRUCTURE LUNAR TECHNOLOGIES

By
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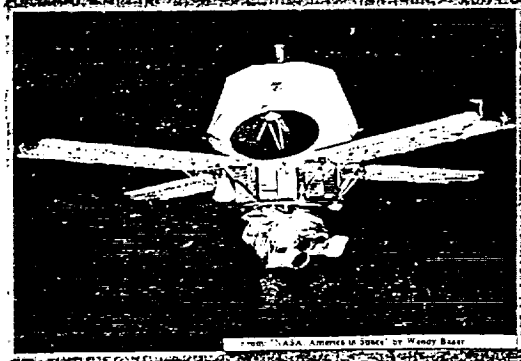
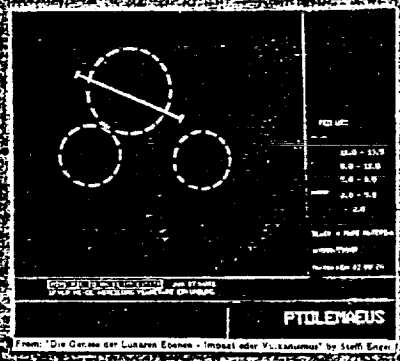
Appendix 1: Anticipated Timeline for Implementation of LOAR

Timeframe 1994 - 1999: Preparation Phase

LUNAR POLAR ORBITER



From: "NASA: America in Space" by Wendy Baker



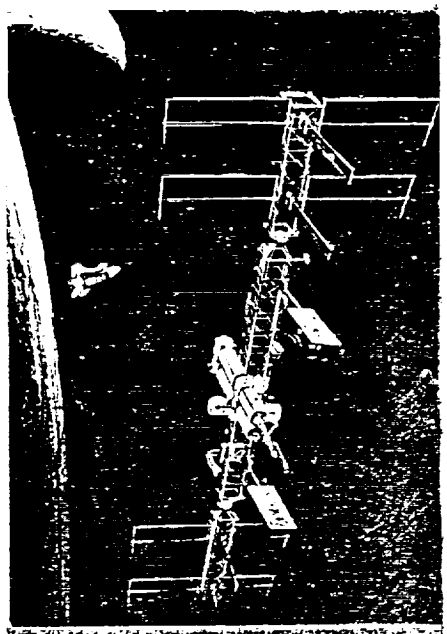
From: "Die Geheimnisse der Lunaren Ebenen - Imbrium oder Vulcanicum" by Steffi Engel

From: "The New Atlas of the Universe" by Patrick Moore

LUNAR POLAR ORBITER - The purpose of the LPO is to perform remote sensing of the Lunar surface in a polar orbit. Topographic information, identification of resource location, and areas in sunlight will be determined.

REMOTE SENSING - LPO remote sensing capabilities will identify potential caches of Lunar resources.

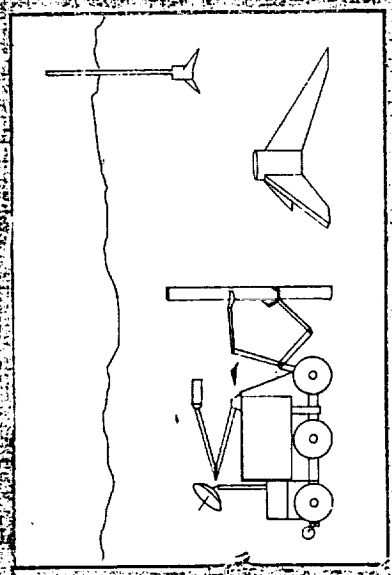
SITE SELECTION - Based on topographical information and resource location information from the LPO, the South Pole of the Moon was chosen as the site for the LOAR.



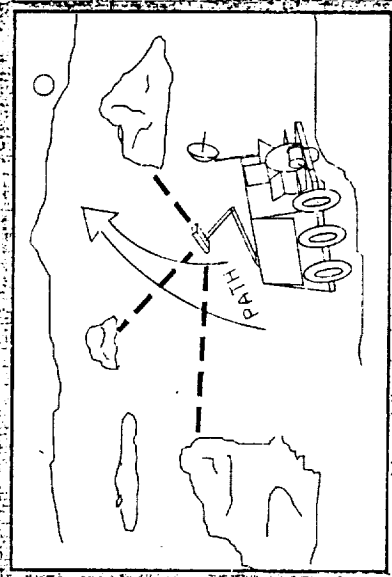
FREEDOM - Crews based at Space Station Freedom may assist in mission orbital assembly operations. Later, it may receive life support supplies from a mature LOAR.

Timeframe 1994 - 1999: Preparation Phase (continued)

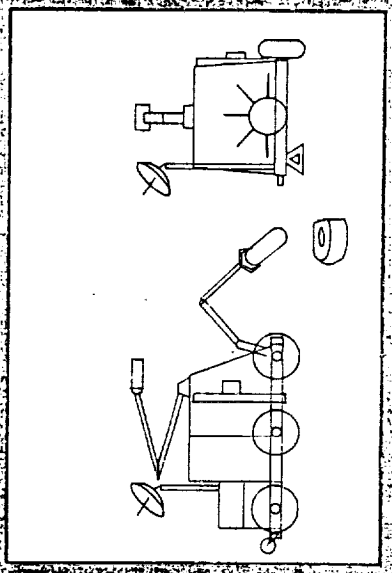
VEHICLE R & D



CONSTRUCTION TASKS - The vehicles will perform a variety of tasks ranging from roadway improvement, to moving soil to erect a radiation shield, to equipment assembly and set up. Example shown here is the construction of a triangulation beacon tower.



NAVIGATION - Using topographical data of the terrain, the practicality of vehicle self navigation through neural networks will be tested. Extensive field testing will occur at Crater National Park in Arizona, USA



BUDDY SYSTEM - During field testing, crisis situations can be created in a highly monitored environment to determine the extent and effective capabilities of the "buddy system" operating protocol.

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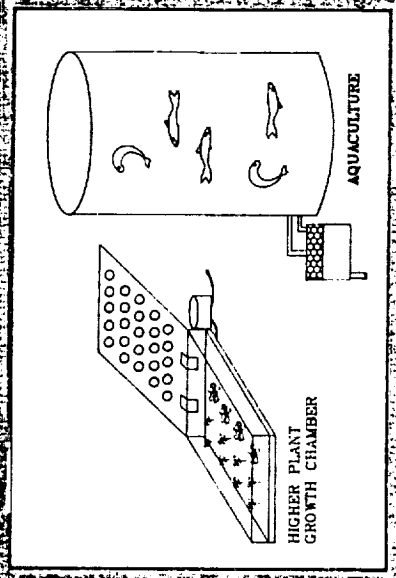
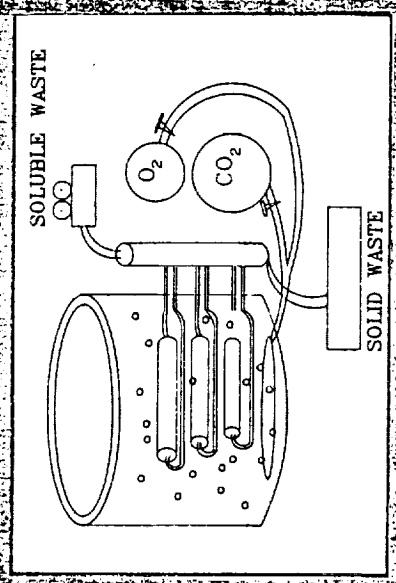
Timeframe 1994 - 1999: Preparation Phase (continued)

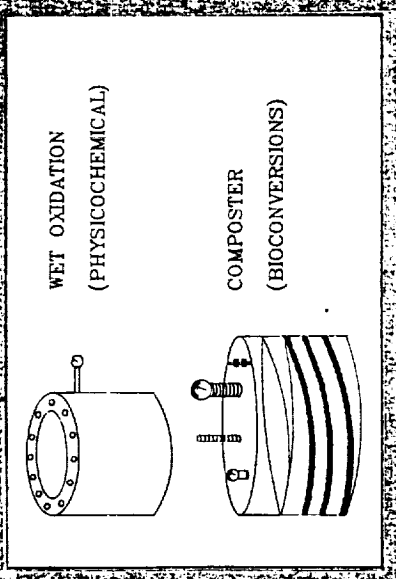
BIO—VOLUME R & D



BIOLOGICAL SUPPORT - Plant growth and aquaculture systems will be tested to determine which types of plants and organisms are put into the Bio-volume for experimentation on the lunar surface.

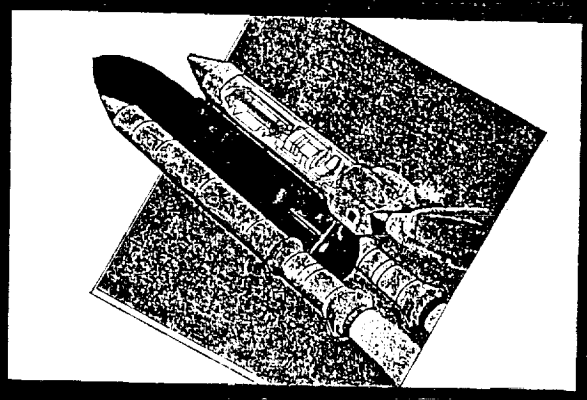


RUDIMENTARY BIOPROCESSING - Microorganisms will be utilized to convert waste into forms that may be reintroduced into the Bio-volume system as plant nutrients.



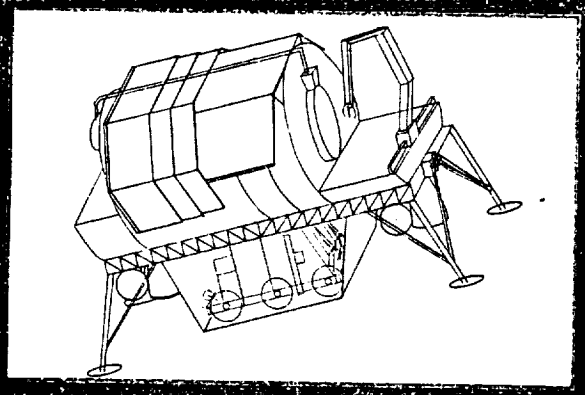
PROCESSORS - Waste processing is an integral part of a complete biological life support system. Ground based testing of waste processors and their integration into the Bio-volume is essential.

Deployment

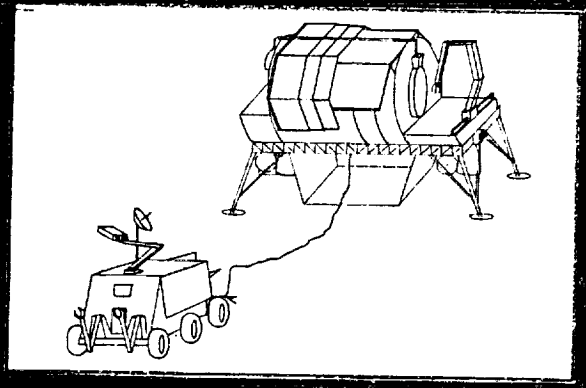


1) Shuttle-C launch

Launch	Payload	
1	Bio-volume	Vehicle
2	Bio-volume	Vehicle
3	Bio-volume	Extra equipment

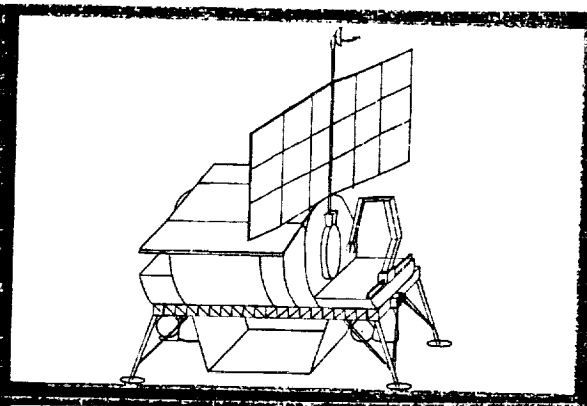


2) Lander and tank mated
3) Land on Lunar surface

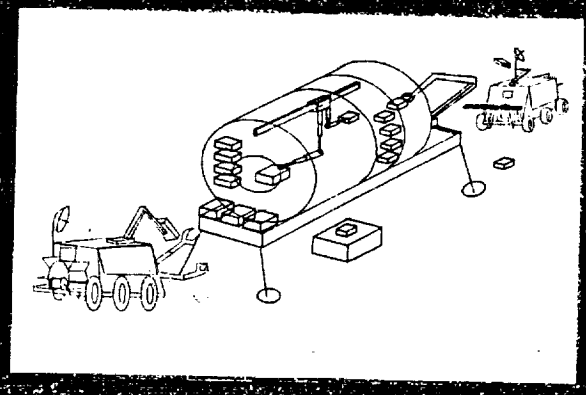


4) Vehicle moves out
5) Vehicle still provides power
6) Combine remaining hydrogen and oxygen and store water

Year 2000: Deployment



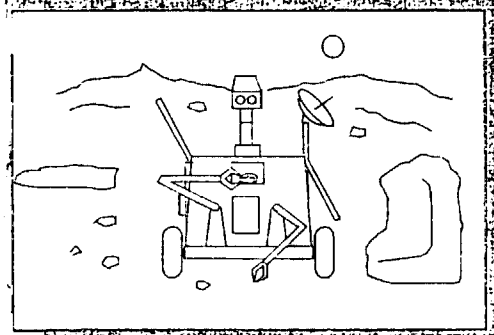
7) Deploy solar array and radiators
8) Shift to solar power
9) Release vehicle
10) Power internal robotic arm
11) Remove internal hydrogen tank wall

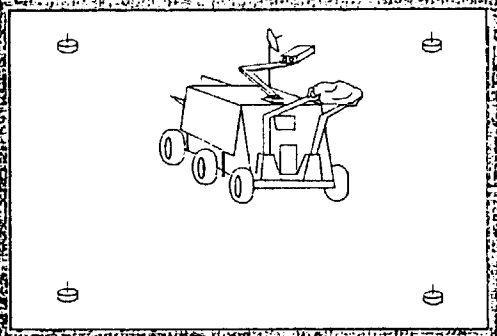


12) Vehicle places supplies up to external robot arms
13) Internal robot installs equipment
14) Close hatch

Timeframe 2000 - 2003: Explorer Phase

EXPLORER PHASE





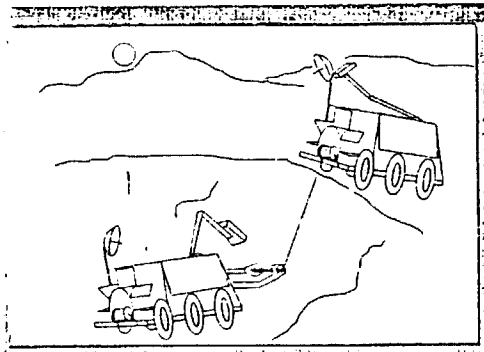
RESOURCE ANALYSIS - The vehicles will perform chemical analysis using gas chromatography of the Lunar soil for the purpose of remote sensing verification and the indentification of localized resources.

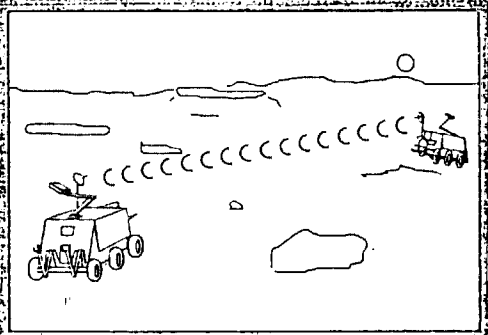
SEISMIC/SHOCK ANALYSIS - Vehicles will 1) deploy the seismophones, 2) determine their distances by pulsed-laser ranging, 3) use radio waves to initiate data collection, 4) drop the rock, and 5) collect seismophones and read data.

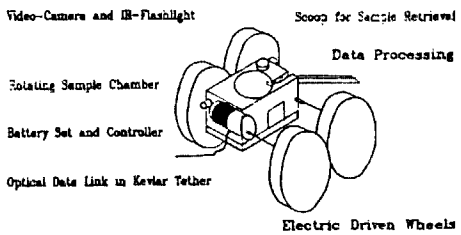
RESOURCE VERIFICATION/ACCESSIBILITY - Concentrated deposits of raw materials will be identified. Accessibility and possible routes will be determined.

—— Good Route
— — — Difficult Route

■ Water/Volatiles
■ Ilmenite
■ Carbonaceous Meteorites







RESCUE/BUDDY OPERATIONS - An otherwise disabling or dangerous situation can be avoided due to the intervention and support of the other vehicle. For example a tow line can be utilized if a vehicle becomes stuck.

INFORMATION SHARING - Vehicles will perform information sharing using radio transmissions. Information to be shared will include resource locations, navigable pathways, topographical maps, and distress signals.

ACCESS VEHICLES - " To remotely go where no vehicle has gone before. " The LRAT will be small tethered vehicles used to retrieve samples from areas which are inaccessible to the explorer vehicles such as lava tubes, craters, and caves.

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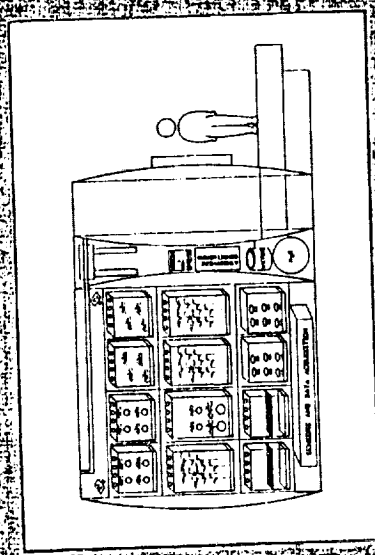
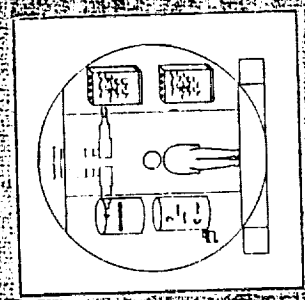
Timeframe 2000 - 2003: Explorer Phase (Continued)

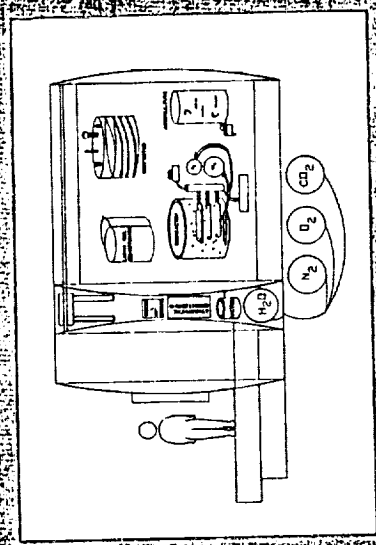
BIO-VOLUME ACTIVITIES



PLANT GROWTH AREA - The viability of plant species in the Lunar environment will be determined. Plant selection is based on possible human consumption. Species range from higher plants to fungi and algae.



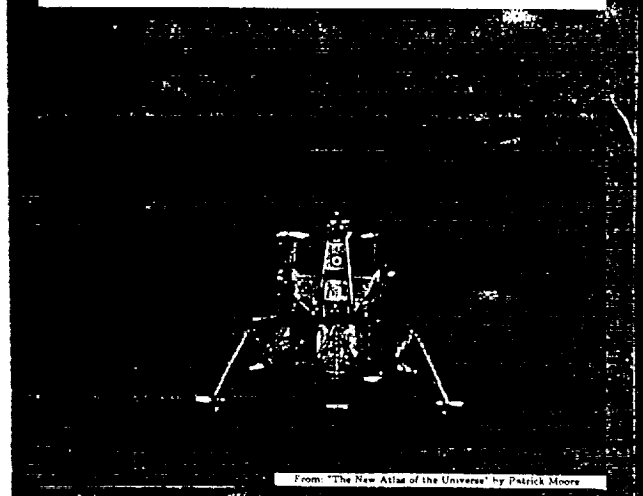
SUPPORT EQUIPMENT - The bio-volume is "gardened" by the multi-purpose robot mounted on a ceiling rail. Physicochemical life support systems will buffer and stabilize the biological systems.



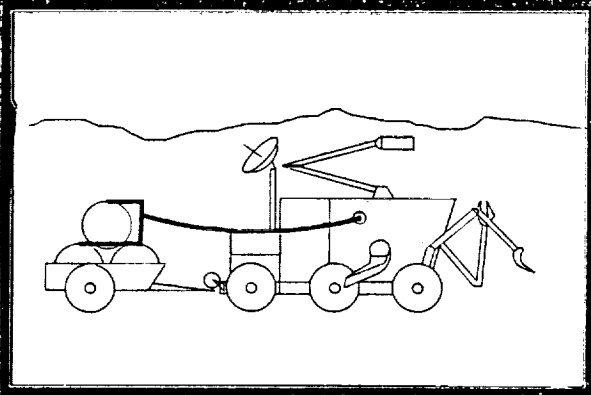
WASTE PROCESSING - Buffer volumes will be maintained for all essential consumables. Produced biomass will be recycled. Recovery of trace elements and the conditioning of Lunar soil for plant growth will be conducted.

Year 2004: First Humans Arrive

First Humans Arrive

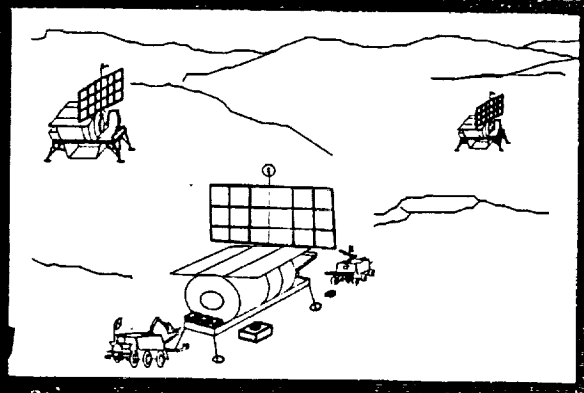


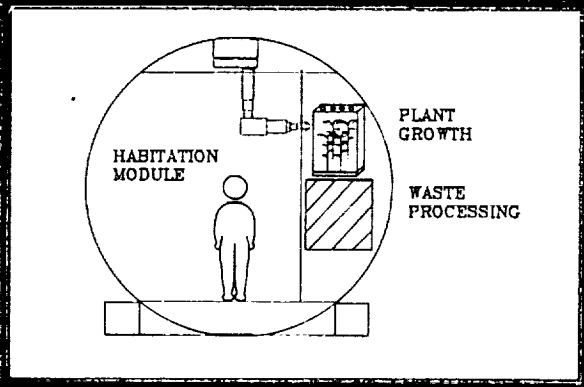
From: "The New Atlas of the Universe" by Patrick Moore



MAN RETURNS - Man returns 35 years after humans first set foot on the Lunar surface. Additional supplies and scientific equipment are also transferred to the Lunar Oasis to complement existing resources.

PROCESSOR CONVERSION - The analyzer of one vehicle will be replaced by a processing unit (on front). Nitrogen tanks scavenged from the lander may be used as storage tanks for by-products of processing.



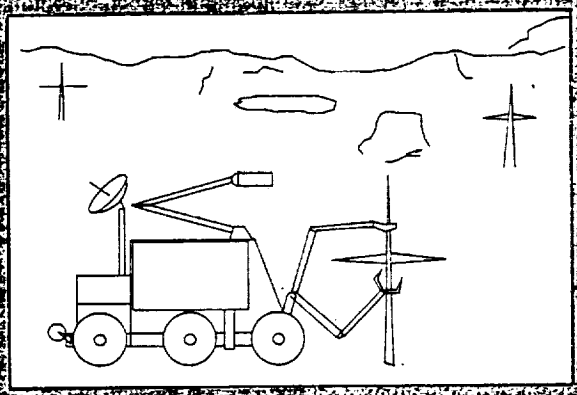


RESUPPLY - Additional equipment and supplies are staged into the bio-volumes. Equipment such as processors will have been optimized based on the results of sample analysis and availability.

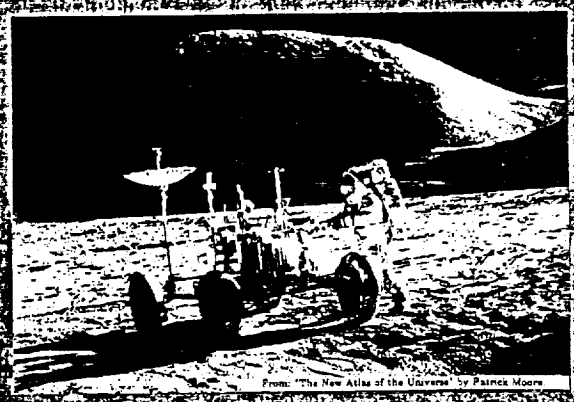
HUMAN INTERFACE - Astronauts will use stored consumables to extend their stay times. Waste products from the crew will be added to the bio-volume mass, and processed later.

Timeframe 2004 - 2009: Human Sortie Missions

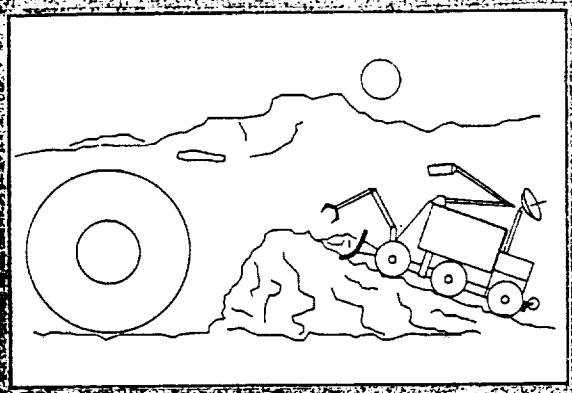
HUMAN SORTIES



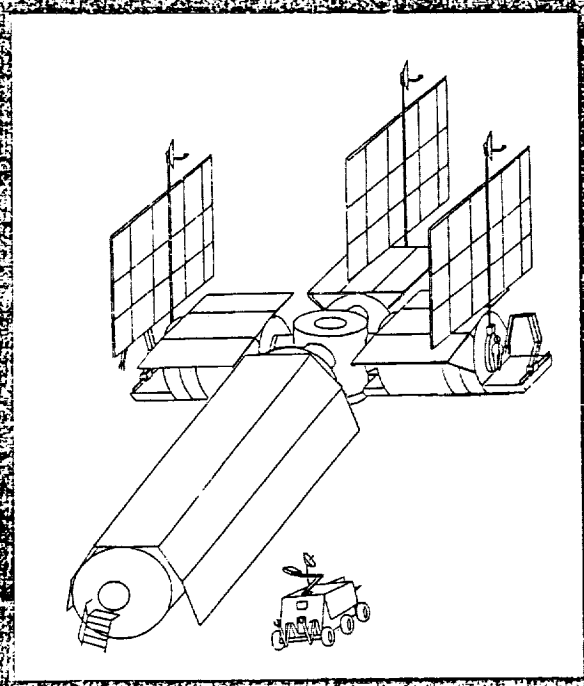
RADIO OBSERVATORY - The robotic vehicles can deploy other scientific experiments at various locations on the Moon. Pictured here is a rover constructing a synthetic aperture radio obervatory on the far side of the Moon.



MANNED EXPLORATION - Previous robotic surveys have identified the most interesting sites. Preferred travel routes minimize astronaut transit times and increase their stay times at choice sites.

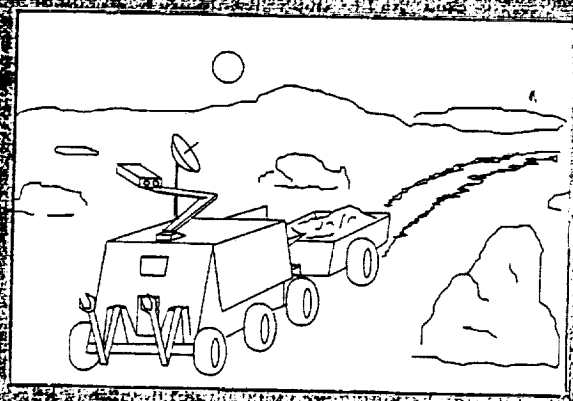


SHIELDING - Longer mission durations on the Lunar surface imply greater risk of exposure to the solar wind. Meter thick walls will be constructed to protect the crews.

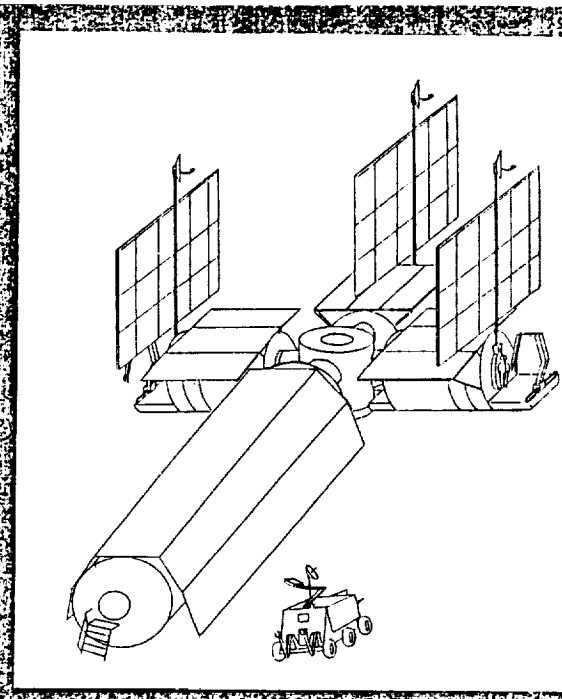


MAIN BASE - The three bio-volumes are transported to the site with the best lighting and joined by a common node. A larger habitation module has been delivered and attached.

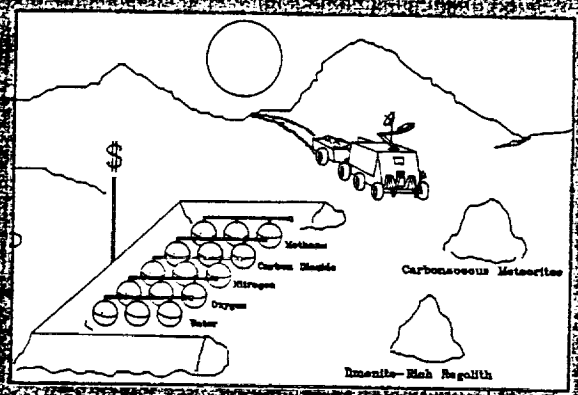
Timeframe 2004 - 2009: Human Sortie Missions (Continued)



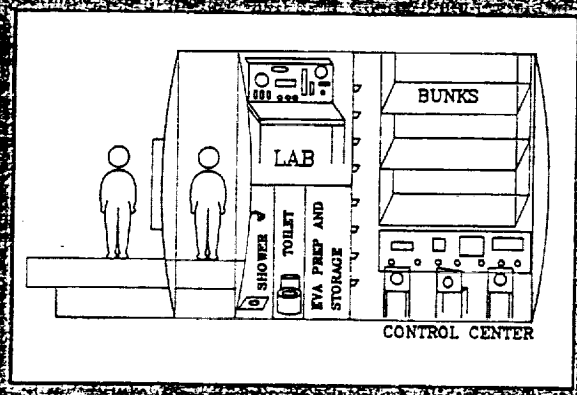
RESOURCE GATHERING - Choice raw materials, such as carbonaceous meteorites and water, may be brought back to the base for processing or storage.



MAIN BASE - The three bio-volumes are transported to the site with the best lighting and joined by a common node. A larger habitation module has been delivered and attached.



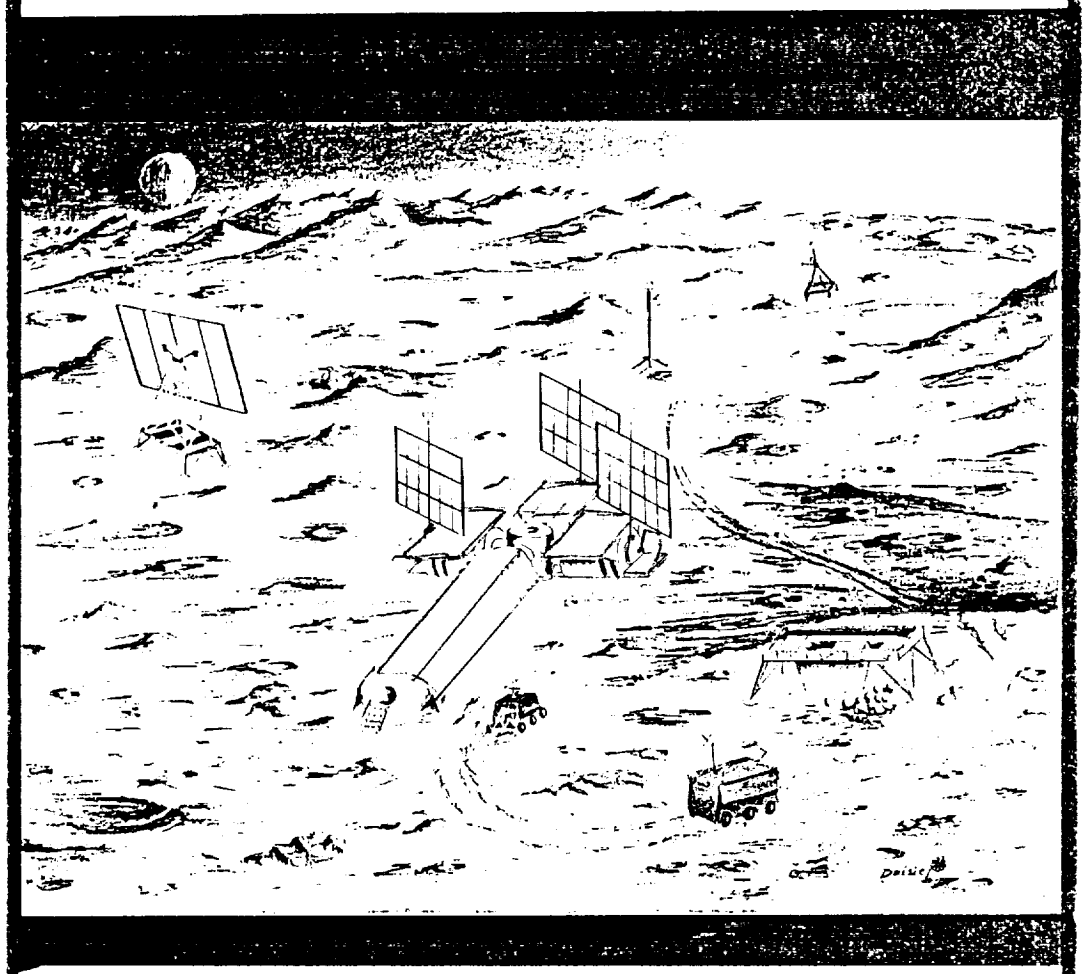
RESOURCE STOCKPILES - Raw materials are transported to the base for biological or chemical processing. Many biological products are stored for future use by man. Water is scavenged from the Lunar soil for bio-volume use.



BIO-VOLUME CONVERSION - Early on, a portion of one bio-volume may be modified for human habitation. Additional life support consumables may be transferred between bio-volumes for crew support.

Year 2010: Mature Lunar Oasis

Lunar Oasis



MATURE LOAR - Many capabilities have been expanded and developed at the mature Lunar Oasis. Power capacity has grown with base activity. A radiation safe haven has been provided. Extensive life support consumables stored at the site also enhance long duration missions. Long duration missions will enable an even more detailed examination of the Moon than provided by the vehicles. Raw materials will be processed on much larger scales, retrieving water and other products from the Lunar soil. Raw fiberglass from regolith processing may be used to construct larger storage tanks, bio-volumes, or manned habitation modules. Expandable membranes will be fiber wound to produce such volumes.

Year 2010: Mature Lunar Oasis (continued)

INFRASTRUCTURE SUPPORT - The mature LOAR may be able to exploit Lunar resources on larger scales. The economical processing of Lunar oxygen may yield an alternative propellant supply point for some space activities. Lunar fiberglass wound tanks may be used to transport propellants into space through the use of a mass launcher. As the biomass increases, some life support consumables may be transported to provide the life support needs of space stations at L1 and LEO. Waste products from these stations may be returned to the moon for processing and recycling. Thus, the mature LOAR may become a vital support facility in a cis-Lunar space infrastructure.

Appendix 2: Class Participants and Acknowledgements

Space Habitation Class Fall 1988:

Teaching assistant: Steve Johnson
Al Abeyta, Todd Aten, Paul Benfield, Todd Bergren, David Bowman, Mark D'Amara, Carl Devillier, Joy Folkvord, Kevin Graham, James Hardaway, John Hayase, Jim Hayes, Alex Hoehn, Christian Kalar, Katie Link, Gerald Marfoe, Dave Mell, Kris Mickus, Philip Moos, David Newell, Rob Pearson, John Ragunath, Kelly Rengstorf, Stephen Shibel, John Stenner, Shane Schwartz, Teresa van Hove, Nancy Wiltberger.

Winter Conference January 1989

Teaching Assistant: Steve Johnson
Paul Benfield, Joy Folkvord, Kris Mickus.

Space Habitation Class Spring 1989:

Teaching Assistant: Steve Johnson
Gregg Allison, Fred Baker, Paul Benfield, Sam Budoff, Kenneth Center, Will Faller, James Ferriday, Joy Folkvord, Michael Forte, Brent Helleckson, Alexander Hoehn, Christine Hollop, Dural Horton, Morgan Jones, Robert Kelso, Frank Lane, Matt Lindsey, Joseph Lopez, Mike Loucks, Dave Mesnard, Kris Mickus, Gary Mills, Phil Moos, Alexander Munro, Jeff Peterson, Henry Rackley, Kelly Rengstorf, Matt Scarlett, Steven Shibel.

Summer Conference June 1989

Teaching Assistant: Steve Johnson
Will Faller, Alex Hoehn, Nancy Wiltberger.

Acknowledgement:

We appreciate the support from the NASA / USRA Advanced Mission Design Program. We would like to thank Bob MacElroy, at NASA-Ames, for his support to our program. In addition, we would like to thank all the people at NASA and in industry who provided information to members of our class in support of our design efforts.

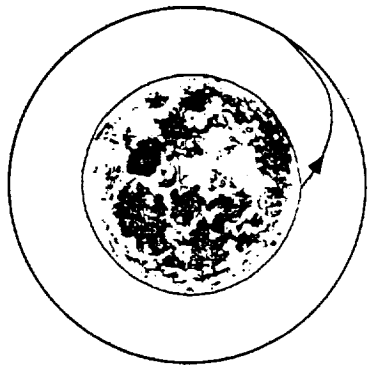
Special thanks to Professor Marvin W. Luttges and Prof. D. Kennedy for their timeless efforts and constructive comments. We would also like to thank all the people from industry, NASA and the public who attended the Critical Design Reviews, and helped with their comments to improve our design work.

Thanks to all the others who helped and whom we did not mention specifically. We appreciate your help. Thanks.

Space Habitation
Spring Semester 1989

Propulsion Design Group

Lunar Surface to Lunar Orbit Propulsion Options

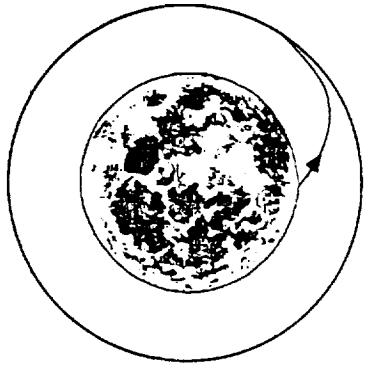


MForte
BHelleckson
MJones
RKelso
MLoucks
DMesnard

Space Habitation
Spring Semester 1989

Propulsion Design Group

Lunar Surface to Lunar Orbit Propulsion Options



Michael Forte
Brent Helleckson
Morgan Jones
Robert Kelso
Mike Loucks
Dave Mesnard

Spring 1989 Space Habitation Design Propulsion Group Final Paper

ABSTRACT

A logical extension of human activity in space beyond low Earth orbit (LEO) includes the development of a permanently occupied lunar base. The cis-lunar space infrastructure will require increasing amounts of propellant for transport operations and costs may become prohibitive without a more cost-effective means of producing and transporting propellant.

In this paper, several methods and systems for transporting processed propellent material from the lunar surface to lunar orbit will be explored. Metrics will be assigned to a cost structure and a ten-year lifecycle cost will be determined for each system.

The Electro-Magnetic Mass Driver (EMD) is offered as an option for a non-chemical rocket launching system. This system must be augmented by a traditional rocket-powered apogee kick motor and cargo orbit transfer vehicle to take the material back to LEO.

In addition, the propulsion group designed and constructed a demonstration of Mass Driver technology; this effort is described.

INTRODUCTION

Propulsion requirements for the cis-lunar infrastructure will need to be carefully integrated into the overall mission plan if the most cost-effective systems are to be realized. Propulsion systems are inherently expensive and will control the scope and direction of the return and settlement of the Moon.

The most successful mission plan will be the one that is the most flexible or will offer the most reasonable alternatives when changes in objectives are required. This flexibility is not always possible, especially with more complicated systems such as rocket motors and transportation systems. For this reason, as many candidate technologies as possible should be examined early in the mission scenario.

Provided it is cost-effective to do so, propellants, mainly oxygen, can be manufactured on the lunar surface and brought to low Earth orbit to reduce the cost of maintaining the expanding transportation requirements.

The cost structure presented in this paper takes into account development, manufacture, transportation, operations, and recurring costs. While this system is simplified, the method has been applied equally to all candidate systems so that a fair comparison can be made.

The feasibility of manufacturing and storing propellants and of construction on the lunar surface are not analyzed in this paper, but are recognized as critical to the success of this proposal.

RATIONALE

Historically, spaceflight has been enabled and paced by developments in propulsion systems. Current programs are heavily dependent upon, and shaped by, available launching methods.

Furthermore, future mission plans call for much larger launch systems based upon current technology, and for exotic propulsion schemes not yet proven in space. The technological challenges posed by propulsion, and its pivotal role in space exploration and development demand a thorough and on-going examination of propulsion options. Pioneering the Space Frontier, Pathfinder and many other studies on space exploration and development, identify the cost of propulsion as a major impediment to all in-space activity.

The Challenger tragedy has emphasized the importance of a diversified fleet of robust propulsion systems, whose functions are overlapping yet designed to carry missions appropriate to the propulsion technology employed (i.e. human vs cargo).

Operational launch systems from early solid rocketry to the LOX/hydrocarbon-LH2 Saturn V booster have been designed mainly for earth launch and have been characterized by the following:

- Mission specific
- Non-reusable
- Reasonably reliable
- High development and operational costs

The Space Transportation System (STS) represents the state-of-the-art, and is an initial attempt at a generic, reusable, reliable, low-cost propulsion system. Interestingly, the STS utilizes systems derived from all of the historical propulsive systems (i.e. solid, LOX/LH2, hypergolic, and cold gas) except those based upon hydrocarbon fuels.

The cis-lunar infrastructure proposed by Buck et. al, requires a transportation system capable of providing access from earth to low earth orbit (LEO), geosynchronous orbit (GEO), cis-lunar space, lunar orbit (LO), to the lunar surface (LS), and back again. The propellant required for transportation within this region accounts for a large fraction of the total mass flow required. The proposed infrastructure assumes that propulsion systems and their propellants were to initially be lifted from earth.

Due to the relatively low lunar gravity (1/6 that of earth), it is possible that significant savings in the amount and cost of propellant delivered to cis-lunar space may be realized by supplying propellants from the lunar surface . The methods of propellant production and of launching cargo (i.e. propellants) from the lunar surface into cis-lunar space play a crucial role in determining when and if such savings can be attained.

To date, the only U.S. propulsion system that has delivered cargo and personnel from the lunar surface to cis-lunar space is the hypergolic (i.e. self-igniting) rocket engine on the Apollo Lunar Module (LM). While this system reliably performed its intended function, it may not be the most efficient method for transporting cargo at all phases of cis-lunar development. Therefore, additional options must be analyzed. This leads in to the problem statement:

> "Given the requirements and deveolpment schedule of a cis-lunar infrastructure, what is the most appropriate method of launching cargo (e.g. propellant) from the lunar surface to lunar orbit."

The problem statement presumes the following:

- A definition of the word "appropriate."
- Knowledge of competing propulsion options.
- Knowledge of types and amount of cargo expected.
- An assumed cis-lunar development sequence and timeline.

This paper defines the appropriate propulsion technology as the one with the lowest lifecycle cost. The type of cargo is that of lunar-produced propellant in the form of liquid oxygen or water. It is assumed that this

propulsion system will be available when the cis-lunar propellant requirements reach 500,000 Kg/yr.

Candidate Technologies

For the purposes of our study, we have chosen four systems as candidates for propulsion from the lunar surface. The chosen systems include three conventional systems: hypergolic fuels, cryogenic fuels, and Solid fuels, plus an untried system, the electromagnetic mass driver.

Hypergolic fuels (UDMH/N2O4/MMH).

Launch systems utilizing hypergolic systems are the simplest of all combustion systems due to the nature of the fuel. Hypergolic fuels ignite on contact and thus require no complex ignition system. The most common fuels of this type are the bipropellants Unsymmetrical DiMethyl Hydrazine (UDHM) and Nitrogen Tetroxide (N2O4). Also used are combinations of Monomethyl Hydrazine (MMH) and N2O4. The Lunar Excursion Module used in the Apollo program used a 50-50 mix of UDMH and MMH known as Arazine 50. A similar system is in use today on the Space Shuttle Reaction Control System (RCS), which is used for attitude control. Hypergolic fuels are extremely dangerous and difficult to handle due to their caustic and explosive nature. Although the specific impulse of hypergolic systems are lower than that of other systems, these systems offer high reliability, simplicity and reusability. There are no plans to process hypergolic fuels on the lunar surface, thus any system using hypergolic fuels would need to bring propellants from the Earth.

Cryogenic Fuels (LO2/LH2).

Propulsion systems using cryogenic (cooled) fuels provide the highest specific impulse of all fuels in use today. The most common fuels are liquid hydrogen and liquid oxygen. Cryogenic systems tend to be complicated, and require high-speed turbines to mix the fuels at high pressure in a combustion chamber. The complexity of cryogenic systems is contained mostly in the turbine technology and the cryogenic storage systems required to keep the fuels at low temperatures. The Space Shuttle Main Engines (SSME's),and the RL10A engine used on the Centaur family of upper stages are examples of cryogenic systems. Both the SSME's and the RL10A use liquid hydrogen and liquid oxygen. These systems offer high

thrust with high complexity and can be reusable. Oxygen may be processed on the lunar surface, at this time however, it appears that the hydrogen for such a system would be brought from Earth.

Solid Fuels (AP/Al).

Solid fuels have long been used as ICBM propulsion systems due to the ability of solids to be stored relatively easily. The most common solid fuels are Ammonium Perchlorate (AP) and Aluminum (Al) mixtures. The fuels are normally bound in a case with Hydroxyl Terminated Polybutadine (HTPB) and are fired with igniters from the top of the casing. Solid fuel systems currently in use include the Shuttle Solid Rocket Boosters (SRB's), the Star-48 motor used in the Payload Assist Module (PAM), and the SRM-1, used in both the Inertial Upper Stage (IUS) and the Transfer Orbit Stage (TOS). The Martin Marietta Titan III and IV series of boosters also make use of solid motors. Solid motors offer easy storage and handling, medium reliability with medium thrust and are in general not reusable (the exception being the Shuttle SRB's). To utilize such a system on the moon would likely require all of the fuel to be brought from Earth.

Electromagnetic Mass Drivers (EMD's).

EMD's use electromagnetic coils to accelerate mass to escape velocity down a track. EMD systems require no propellant but instead use electricity and electromagnetic forces to generate thrust. Such systems, which are not currently in use, will be large in size, but will be reliable (no moving parts) and reusable. Such a system would initially require a large amount of mass delivered to the lunar surface but once in place would require only mass containers in which to launch cargo. EMD systems offer low cost, and current technology along with high power requirements and untested systems. In order to keep the size of such systems within reasonable ranges and reach orbital velocities, accelerations on the order of 500 g's (500 times the force of gravity on Earth) are required. This eliminates realistic EMD systems from launching biological cargos such as humans.

Shuttle Reaction Control System Derivative (RCS)

The Reaction control system of the Space Shuttle utilizes UDMH/MMH and N2O4 as bipropellants for attitude control of the Space Shuttle orbiter.

The RCS jets each provide 900 lbs of thrust and are manufactured by the Atlantic Richfield Corporation (ARC). Each thruster is rated for 10,000 restarts or approximately 100 shuttle flights. Using similar technology to the Shuttle RCS system, a reusable hypergolic system could be constructed to lift material off of the lunar surface. This system would consist of three clusters of six ARC 900 lb engines, for a total of 18. The main expense in the development of such a system would be the large tanks required for propellant storage. The estimated mass can be calculated from the rocket equation, assuming vehicle weight to be about 1/8 of payload weight. The estimated hardware cost of the vehicle would be in the range of $30 million. Using the $10,000/kg figure and the estimated weight, a cost of $43 million was calculated. A development cost of $140 million was used, assuming approximately 4.5 times the hardware cost ($30 million) for development.

The RCS derivative system would be a reusable system, rated for 100 flights. All fuel for the RCS derivative would be launched from the Earth.

Transfer Orbit Stage Derivative

The Transfer Orbit Stage (TOS) is a commercial upper stage vehicle built by Orbital Sciences Corporation and Martin Marietta using solid rocket motor technology. The TOS is designed for launching medium-capacity satellites (between the capacity of PAM and Centaur) from LEO to Geosynchronous orbit or to interplanetary trajectories. The propulsion system of the TOS is based on the solid propellant first stage of the IUS vehicle, which is the United Technologies/Chemical Division SRM-1. The TOS derivative system explored here consists of using the existing TOS vehicle with minor modifications to allow for firing from the lunar surface. A- TOS derivative system would be an expendable one, due to the prohibitive cost of refurbishing this type of a vehicle. The listed cost for the TOS is $14 million, the cost used for this report is $24 million assuming $10,000 per kilogram of space hardware.

We have assumed a development cost for such as system to be minimal ($5 million) since the vehicle has already been developed.

Centaur G-Prime Derivative

The General Dynamics Centaur G-Prime is a high-energy transfer stage designed for use on the Space Shuttle. Originally designed for missions such as the Galileo Probe, the Centaur G-Prime program has since been cancelled, and currently none are scheduled to fly on STS. The basic systems, however, are pertinent since all are derivatives of the Centaur upper stage used on the Atlas and Titan launch vehicles. The Centaur

vehicle uses 2 Pratt-Whitney RL10A-3-3A engines powered by cryogens LO2 and LH2 with a 5/1 mixing ratio of fuel to oxidizer. A system derived from Centaur would be appropriate for the lunar surface especially if lunar oxygen was available. The Centaur vehicle is not designed to be reusable and has a listed price of $66 million. This price compares very well with the standard $10,000/kg price for space hardware. We have assumed a development cost of $10 million since the vehicle has already been designed. This is identical for the assumption used on the TOS vehicle, except that $5 million extra has been added for the development of a system to pressurize the vehicle while it is being transported to the moon unfueled. It is assumed that the Centaur derived vehicle uses lunar oxygen.

Orbital Transfer Vehicle Derivative

The Martin Marietta Orbital Transfer Vehicle (OTV) chosen for this trade-off is the one described in Willcockson's "Applying the OTV to Lunar Logistics." The OTV is still a conceptual vehicle, although clearly it will likely become a standard vehicle at some point in the next twenty years. The OTV will use advanced cryogenic propellant systems with an estimated mixing ratio of 10/1 and a projected specific impulse of 475 seconds. The engines used on this system will be derivatives of the Centaur's RL10A engines, modified for reusability. The calculated hardware cost is estimated to be $37 million. We have adopted Willcockson's estimates for development cost of $800 million. The vehicle is designed for approximately 100 uses, and it is assumed to use lunar oxygen.

Electronic Mass Driver (EMD)

The Electromagnetic Mass Driver (EMD) traded off in this study is the configuration described by Snow, Kubby and Dunbar, 'A Small Scale Lunar Launcher For Early Lunar Material Utilization'. Proposed is a system using small Lunar Oxygen Transfer Vehicles (LOTV's) which could also be used for processed water. The LOTV's are "smart" vehicles, with an attitude control system and an apolune kick motor. The LOTV's would rendezvous with an orbiting tanker vehicle which would transfer the propellants to a freighter for return to LEO.

The system described by Snow et. al requires launch to a 100 km lunar orbit. The system uses a 'pull-only system', and can launch at a rate of once per 1/2 hour, during peak power periods. This allows for 400 LOTV launches per lunar day.

The track is 330 meters long and the LOTV is accelerated at 4834 m/sec^2 in order to reach lunar orbitital velocity in this length. Each LOTV can be used approximately 100 times. Loading on the bottles is estimated to be 10,800 psi lengthwise along the cylindrical section.

The development cost of such a system is estimated by Andrews and Snow in 'The Supply of lunar Oxygen to Low Earth Orbit' to be $2 billion.

EMD's offer low operational costs, high reliability and safety, along with high development costs.

The Electromagnetic Mass Driver employs a variety of large-scale systems that are untried on the scale required for implementation. Snow, Kubby and Dunbar estimate the power requirements for such a system to be 181 kw with a launch rate of 400 per lunar day or every 30 minutes at peak power. With modern solar cells, this would require a field of solar cells on the order of 2500 square meters. This itself amounts to about 2.5 tons of mass, for power system alone.

The homopolar storage system is an electromagnetic energy storage device which can be used in place of, or in addition to, capacitors, which are required for the large voltages needed by the EMD. Snow, et. al. estimate the total mass for this system, including the power requirements to be 138 metric tons.

The system also requires a processing facility to create the cargo. It is assumed that this system is already in place by the time the mass driver becomes necessary. Additionally, the alternative systems (i.e. TOS, OTV, etc..) would also require the processing facility.

Since the system has relatively high power requirements, it must dissipate large amounts of heat, requiring numerous radiators. This technology is still in development, as is indicated by the recent problems with the Space Station Radiator experiment on Discovery in March.

Note that the LOX processing plant is assumed to exist This could easily be replaced by an H2O plant or other processing facility.

Trade Off Study and Cost Model

Although parameters such as reliability, simplicity, technology development required, etc., are considered qualitatively, only lifecycle cost is considered quantitatively. To accomplish a trade-off study based upon lifecycle cost, a definition of metrics and assumptions must be made. For the purposes of this study, the lifecycle is defined to be 10 years in duration and each propulsion system must be capable of placing 500,000 kg of mass into lunar orbit each year. All cost figures are in 1989 dollars.

Lifecycle cost is calculated, via the assumptions described previously, by summing development costs (i.e. design, hardware, and set-up), transportation costs (i.e. Δv from earth's surface to lunar surface), lunar launch costs (Δv from lunar surface to lunar orbit, and operational), and lunar return costs (Δv required to return to the lunar surface). Appendix A contains the calculations, formulae, assumptions, and limitations of the cost model

The results of the cost model indicate that the electromagnetic mass driver yields the lowest 10-year lifecycle cost of all of the options examined. A limited sensitivity analysis indicates the major factors affecting the ranking of the options are the amount of re-supply mass delivered from earth throughout the lifecycle, and the cost of transporting that mass from the earth to the lunar surface (see Appendix A).

Based upon the actual propellant requirements, technology availability, and level of lunar surface activity, the mass driver described could become feasible as early as the 2010-2015 time frame.

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Appendix A

Propulsion Option Cost Model Calculation

Propulsion Option Cost Model
Calculation Appendix

Overview

Hardware/Software

The cost model used to perform the propulsion option trade-off was created on a Macintosh II using Microsoft Excel 1.5. The plots were generated on Ablebeck's Kalidegraph 1.1 software.

Assumptions

Although parameters such as reliability, simplicity, technology development required, etc., are considered qualitatively, only lifecycle cost is considered quantitatively. To accomplish a trade-off study based upon lifecycle costs, a definition of metrics and assumptions must be made. For the purposes of this study, the lifecycle is defined to be 10 years in duration and each propulsion system must be capable of placing 500,000 kg of mass into lunar orbit each year. All cost figures are in 1989 U.S. dollars

The cost model utilizes the following variable and formulae to determine the projected lifecycle cost of a given system.

Cdev = development cost (i.e. Cdes + Chdw + Cset)

where: Cdes = nonrecurring engineering cost
Chdw = hardware cost (@ $10,000/kg dry mass)
Cset = lunar set-up costs ($20 million ea. system)

Ce>ls = transportation cost from earth to the lunar surface (i.e. Cops + $26,400 * mass of the earth-produced propellant required to arrive at the lunar surface)

Cls>lo = transportation cost from the lunar surface to lunar orbit (i.e. Cops + $26,400 * mass if the earth-produced propellant required to arrive on lunar orbit)

where: Cops = lunar operating costs (i.e. 25% of Chdw)

Clo>ls = transportation cost from lunar orbit back to the lunar surface (i.e. $26,400 * mass of the earth-produced propellant required to arrive at the lunar surface)

Clc = lifecycle cost = cost of launching 500,000 kg/yr for 10 years

where: Clc = Cdev + Ce>ls + Cls>lo + C lo>ls

Propulsion Option Cost Model
Calculation Appendix

Results and Critique

As the following plot, "Propulsion Systems Cumulative Costs," indicates, the mass driver, under the stated assumptions yields the lowest 10-year lifecycle cost of the options examined.

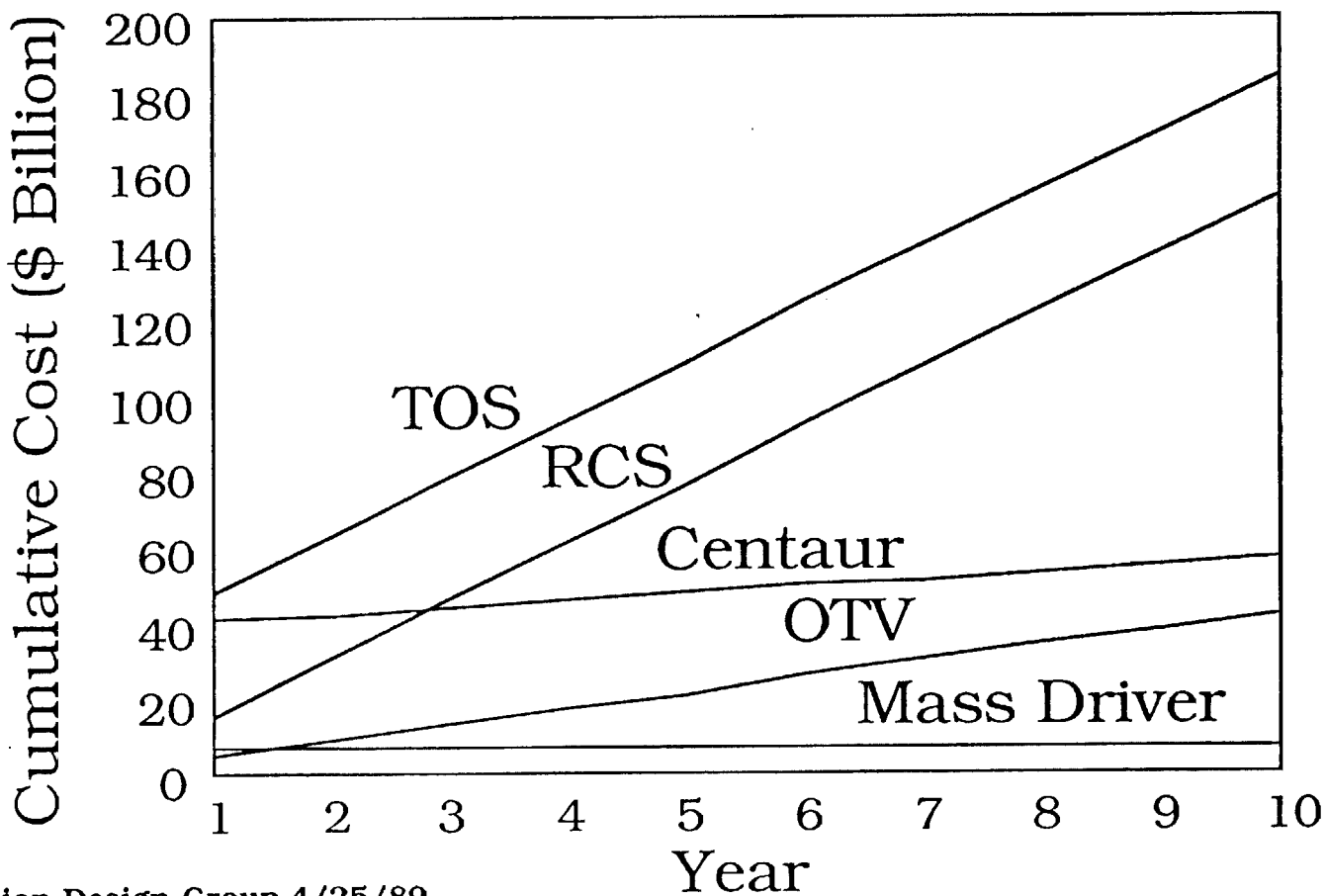
Upon analyzing the results of the cost model, a sensitivity analysis was performed to determine which variable most affected the outcome of the model. It was discovered that reasonable assumptions for hardware, development, and set-up costs were insignificant compared to the cost of transporting hardware, propellant, etc., from earth to the lunar surface. As the plot, "Variation of Lifecycle Costs with Launch Costs," indicates, a factor of 5 to 10 reduction in earth-to-lunar surface launch cost (i.e. from an assumed $26,400/kg) may not allow the initial investment in the mass driver to be recovered in a lifecycle of 10 years. Also, this analysis makes no allowance for the cost of producing propellant on the moon. This indicates that further study is in order to determine the most appropriate expenditure of research money, be it mass driver development or launch-cost reduction.

Along with the estimated launch costs, the operational costs require further investigation. The assumption of 25% of hardware costs may not be the most appropriate measure of operational costs.

Finally the cost impact of the utilization of lunar-produced, fiber-wound tanks similar to those proposed by the Fluid Technology Design Group must be examined.

Propulsion System Cumulative Costs

10 Year Lifecycle



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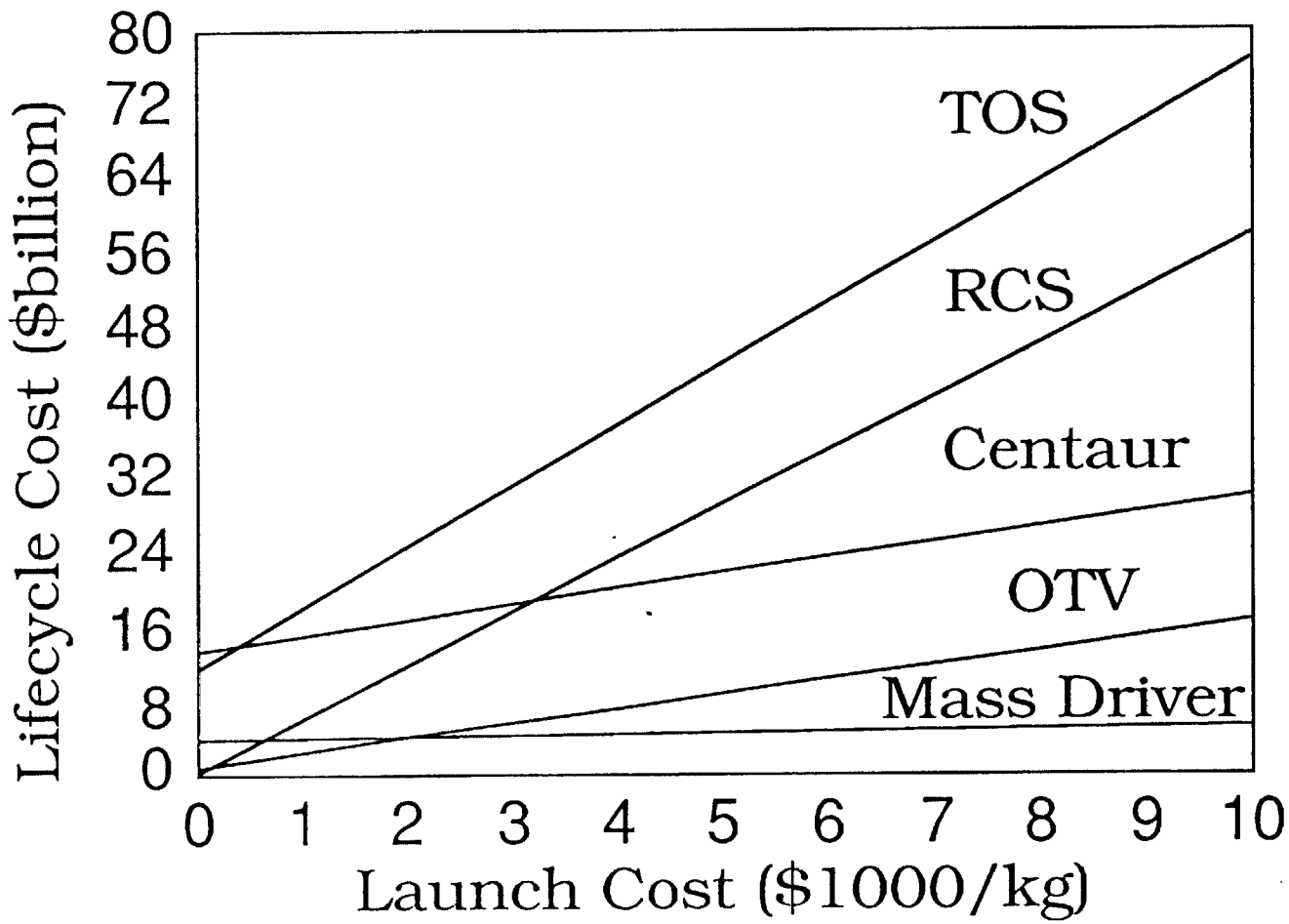
Variation of Lifecycle Costs with Launch Costs



Propulsion Design Group 4/25/89

Trade off

Launchers	Mass Driver	TOS	Centaur	OTV Derivative	STS RCS Derivative
Ce>ls ($/kg)	26,400	26,400	26,400	26,400	26,400
Cls>lo ($/kg)	70	30,199	3,551	8,151	30,366
Cops ($/LTV)	695,000	8,810,000	16,480,000	9,250,000	10,750,000
Clo>ls ($/deorbit)	0	0	0	26,400,000	55,440,000
Cdev ($/LTV)	3,535,000,000	35,354,972	66,102,850	212,480,000	100,600,000
Cdes ($/program)	2,000,000,000	10,000,000	10,000,000	800,000,000	140,000,000
Chdw ($/kg)	10,000	10,000	10,000	10,000	10,000
Cset ($/program)	20,000,000	20,000,000	20,000,000	20,000,000	20,000,000
MILTV (kg/MILTV)	139,000	3,524	6,592	3,700	4,300
MLTV (kg/LTV)	25	3,524	6,592	3,700	4,300
Mlp (kg/launch)	0	21,586	3,475	3,300	20,700
Mp (kg/launch)	100	19,162	30,475	10,700	18,000
Mdp (kg/deorbit)	0	0	0	1,000	2,100
nLTV (LTV)	500	261	164	5	3
nly (launches/yr)	5,000	26	16	47	28
ndy (deorbits/yr)	5,000	0	0	1	1
LLTV (launch/LTV)	100	1	1	100	100
t (yr)	10	10	10	10	10
LOX (kg/yr)	500,000	500,000	500,000	500,000	500,000
Lifecycle cost ($)	$7,882,100,000	$184,497,696,483	$57,153,675,144	$42,466,850,467	$152,979,038,889
Cost per kg of LOX to lunar orbit	$1,576	$36,900	$11,431	$8,493	$30,596

Trade off

Mass Driver	Life Cycle Cost	Year 1	Year 2	Year 3	Year 4	Year 5	Year 6	Year 7	Year 8	Year 9	Year 10
Ce>ls ($/kg)	26,400	26,400	0	0	0	0	1	7,092,750,000	48,600,514,560	41,173,691,550	5,551,096,262
Cls>lo ($/kg)	68	68	68	68	68	68	68	68	68	68	68
Cops ($/LTV)	675,000	67,500	67,500	67,500	67,500	67,500	67,500	67,500	67,500	67,500	67,500
Clo>ls ($/deorbit)	0	0	0	0	0	0	0	0	0	0	0
Cdev ($/LTV)	3,495,000,000	3,495,000,000	0	0	0	0	0	0	0	0	0
Cdes ($/program)	2,000,000,000	2,000,000,000	0	0	0	0	0	0	0	0	0
Chdw ($/kg)	10,000	0	0	0	0	0	0	0	0	0	0
Cset ($/program)	20,000,000	0	0	0	0	0	0	0	0	0	0
MILTV (kg/MILTV)	135,000	135,000	0	0	0	0	0	0	0	0	0
MLTV (kg/LTV)	25	25	25	25	25	25	25	25	25	25	25
Mpl (kg/launch)	0	0	0	0	0	0	0	0	0	0	0
Mp (kg/launch)	100	100	100	100	100	100	100	100	100	100	100
Mpd (kg/deorbit)	0	0	0	0	0	0	0	0	0	0	0
nLTV (LTV)	500	0	0	0	0	0	0	0	0	0	0
nly (launches/yr)	5,000	5,000	5,000	5,000	5,000	5,000	5,000	5,000	5,000	5,000	5,000
ndy (deorbits/yr)	5,000	5,000	5,000	5,000	5,000	5,000	5,000	5,000	5,000	5,000	5,000
LLTV (launch/LTV)	100	100	100	100	100	100	100	100	100	100	100
t (yr)	10	1	1	1	1	1	1	1	1	1	1
LOX (kg/yr)	500,000	500,000	500,000	500,000	500,000	500,000	500,000	500,000	500,000	500,000	500,000
Lifecycle cost ($)	$7,396,500,000	$7,092,750,000	$33,750,000	$33,750,000	$33,750,000	$33,750,000	$33,750,000	$33,750,000	$33,750,000	$33,750,000	$33,750,000
Cumulative Cost		$7,092,750,000	$7,126,500,000	$7,160,250,000	$7,194,000,000	$7,227,750,000	$7,261,500,000	$7,295,250,000	$7,329,000,000	$7,362,750,000	$7,396,500,000

Trade off

TOS	Life Cycle Cost	Year 1	Year 2	Year 3	Year 4	Year 5	Year 6	Year 7	Year 8	Year 9	Year 10
Ce>ls ($/kg)	26,400	26,400	0	0	0	0	1	7,092,750,000	48,600,514,560	41,173,691,550	5,551,096,262
Cls>lo ($/kg)	30,199	30,199	30,199	30,199	30,199	30,199	30,199	30,199	30,199	30,199	30,199
Cops ($/LTV)	8,810,000	881,000	881,000	881,000	881,000	881,000	881,000	881,000	881,000	881,000	881,000
Clo>ls ($/deorbit)	0	0	0	0	0	0	0	0	0	0	0
Cdev ($/LTV)	35,354,972	35,354,972	0	0	0	0	0	0	0	0	0
Cdes ($/program)	10,000,000	10,000,000	0	0	0	0	0	0	0	0	0
Chdw ($/kg)	10,000	0	0	0	0	0	0	0	0	0	0
Cset ($/program)	20,000,000	0	0	0	0	0	0	0	0	0	0
MILTV (kg/MILTV)	3,524	3,524	0	0	0	0	0	0	0	0	0
MLTV (kg/LTV)	3,524	3,524	3,524	3,524	3,524	3,524	3,524	3,524	3,524	3,524	3,524
Mpl (kg/launch)	21,586	21,586	21,586	21,586	21,586	21,586	21,586	21,586	21,586	21,586	21,586
Mp (kg/launch)	19,162	19,162	19,162	19,162	19,162	19,162	19,162	19,162	19,162	19,162	19,162
Mpd (kg/deorbit)	0	0	0	0	0	0	0	0	0	0	0
nLTV (LTV)	261	261	0	0	0	0	0	0	0	0	0
nly (launches/yr)	26	26	26	26	26	26	26	26	26	26	26
ndy (deorbits/yr)	0	0	0	0	0	0	0	0	0	0	0
LLTV (launch/LTV)	1	1	1	1	1	1	1	1	1	1	1
t (yr)	10	1	1	1	1	1	1	1	1	1	1
LOX (kg/yr)	500,000	500,000	500,000	500,000	500,000	500,000	500,000	500,000	500,000	500,000	500,000
Lifecycle cost ($)	184,497,696,483	$48,600,514,560	$15,099,686,880	$15,099,686,880	$15,099,686,880	$15,099,686,880	$15,099,686,880	$15,099,686,880	$15,099,686,880	$15,099,686,880	$15,099,686,880
Cumulative Cost		$48,600,514,560	$63,700,201,440	$78,799,888,321	$93,899,575,201	$108,999,262,081	$124,098,948,961	$139,198,635,842	$154,298,322,722	$169,398,009,602	$184,497,696,483

Trade off

Centaur	Life Cycle Cost	Year 1	Year 2	Year 3	Year 4	Year 5	Year 6	Year 7	Year 8	Year 9	Year 10
Ce>ls ($/kg)	26,400	26,400	0	0	0	30,199	30,199	30,199	30,199	30,199	30,199
Cls>lo ($/kg)	3,551	3,551	3,551	3,551	3,551	3,551	3,551	3,551	3,551	3,551	3,551
Cops ($/LTV)	16,480,000	1,648,000	1,648,000	1,648,000	1,648,000	1,648,000	1,648,000	1,648,000	1,648,000	1,648,000	1,648,000
Clo>ls ($/deorbit)	0	0	0	0	0	0	0	0	0	0	0
Cdev ($/LTV)	66,102,850	66,102,850	0	0	0	0	0	0	0	0	0
Cdes ($/program)	10,000,000	10,000,000	0	0	0	0	0	0	0	0	0
Chdw ($/kg)	10,000	0	0	0	0	0	0	0	0	0	0
Cset ($/program)	20,000,000	0	0	0	0	0	0	0	0	0	0
MILTV (kg/MILTV)	6,592	6,592	0	0	0	0	0	0	0	0	0
MLTV (kg/LTV)	6,592	6,592	6,592	6,592	6,592	6,592	6,592	6,592	6,592	6,592	6,592
Mpl (kg/launch)	3,475	3,475	3,475	3,475	3,475	3,475	3,475	3,475	3,475	3,475	3,475
Mp (kg/launch)	30,475	30,475	30,475	30,475	30,475	30,475	30,475	30,475	30,475	30,475	30,475
Mpd (kg/deorbit)	0	0	0	0	0	0	0	0	0	0	0
nLTV (LTV)	164	164	0	0	0	0	0	0	0	0	0
nly (launches/yr)	16	16	16	16	16	16	16	16	16	16	16
ndy (deorbits/yr)	0	0	0	0	0	0	0	0	0	0	0
LLTV (launch/LTV)	1	1	1	1	1	1	1	1	1	1	1
t (yr)	10	1	1	1	1	1	1	1	1	1	1
LOX (kg/yr)	500,000	500,000	500,000	500,000	500,000	500,000	500,000	500,000	500,000	500,000	500,000
Lifecycle cost ($)	57,153,675,144	$41,173,691,550	$1,775,553,733	$1,775,553,733	$1,775,553,733	$1,775,553,733	$1,775,553,733	$1,775,553,733	$1,775,553,733	$1,775,553,733	$1,775,553,733
Cumulative Cost	11,431	$41,173,691,550	$42,949,245,283	$44,724,799,016	$46,500,352,748	$48,275,906,481	$50,051,460,213	$51,827,013,946	$53,602,567,678	$55,378,121,411	$57,153,675,144

Trade off

OTV Derivative	Life Cycle Cost	Year 1	Year 2	Year 3	Year 4	Year 5	Year 6	Year 7	Year 8	Year 9	Year 10
Ce>ls ($/kg)	26,400	26,400	0	0	0	3,551	3,551	3,551	3,551	3,551	3,551
Cls>lo ($/kg)	8,151	8,151	8,151	8,151	8,151	8,151	8,151	8,151	8,151	8,151	8,151
Cops ($/LTV)	9,250,000	925,000	925,000	925,000	925,000	925,000	925,000	925,000	925,000	925,000	925,000
Clo>ls ($/deorbit)	26,400,000	26,400,000	26,400,000	26,400,000	26,400,000	26,400,000	26,400,000	26,400,000	26,400,000	26,400,000	26,400,000
Cdev ($/LTV)	212,480,000	212,480,000	0	0	0	0	0	0	0	0	0
Cdes ($/program)	800,000,000	800,000,000	0	0	0	0	0	0	0	0	0
Chdw ($/kg)	10,000	0	0	0	0	0	0	0	0	0	0
Cset ($/program)	20,000,000	0	0	0	0	0	0	0	0	0	0
MILTV (kg/MILTV)	3,700	3,700	0	0	0	0	0	0	0	0	0
MLTV (kg/LTV)	3,700	3,700	3,700	3,700	3,700	3,700	3,700	3,700	3,700	3,700	3,700
Mpl (kg/launch)	3,300	3,300	3,300	3,300	3,300	3,300	3,300	3,300	3,300	3,300	3,300
Mp (kg/launch)	10,700	10,700	10,700	10,700	10,700	10,700	10,700	10,700	10,700	10,700	10,700
Mpd (kg/deorbit)	1,000	1,000	1,000	1,000	1,000	1,000	1,000	1,000	1,000	1,000	1,000
nLTV (LTV)	5	5	0	0	0	0	0	0	0	0	0
nly (launches/yr)	47	47	47	47	47	47	47	47	47	47	47
ndy (deorbits/yr)	1	1	1	1	1	1	1	1	1	1	1
LLTV (launch/LTV)	100	100	100	100	100	100	100	100	100	100	100
t (yr)	10	1	1	1	1	1	1	1	1	1	1
LOX (kg/yr)	500,000	500,000	500,000	500,000	500,000	500,000	500,000	500,000	500,000	500,000	500,000
Lifecycle cost ($)	42,466,850,467	$5,551,096,262	$4,101,750,467	$4,101,750,467	$4,101,750,467	$4,101,750,467	$4,101,750,467	$4,101,750,467	$4,101,750,467	$4,101,750,467	$4,101,750,467
Cumulative Cost		$5,551,096,262	$9,652,846,729	$13,754,597,196	$17,856,347,664	$21,958,098,131	$26,059,848,598	$30,161,599,065	$34,263,349,533	$38,365,100,000	$42,466,850,467

Trade off

STS RCS Derivative	Life Cycle Cost	Year 1	Year 2	Year 3	Year 4	Year 5	Year 6	Year 7	Year 8	Year 9	Year 10
Cs>ls ($/kg)	26,400	26,400	0	0	0	925,000	925,000	925,000	925,000	925,000	925,000
Cls>lo ($/kg)	30,366	30,366	30,366	30,366	30,366	30,366	30,366	30,366	30,366	30,366	30,366
Cops ($/LTV)	10,750,000	1,075,000	1,075,000	1,075,000	1,075,000	1,075,000	1,075,000	1,075,000	1,075,000	1,075,000	1,075,000
Clo>ls ($/deorbit)	55,440,000	55,440,000	55,440,000	55,440,000	55,440,000	55,440,000	55,440,000	55,440,000	55,440,000	55,440,000	55,440,000
Cdev ($/LTV)	100,600,000	100,600,000	0	0	0	0	0	0	0	0	0
Cdes ($/program)	140,000,000	140,000,000	0	0	0	0	0	0	0	0	0
Chdw ($/kg)	10,000	0	0	0	0	0	0	0	0	0	0
Cset ($/program)	20,000,000	0	0	0	0	0	0	0	0	0	0
MILTV (kg/MILTV)	4,300	4,300	0	0	0	0	0	0	0	0	0
MLTV (kg/LTV)	4,300	4,300	4,300	4,300	4,300	4,300	4,300	4,300	4,300	4,300	4,300
Mpl (kg/launch)	20,700	20,700	20,700	20,700	20,700	20,700	20,700	20,700	20,700	20,700	20,700
Mp (kg/launch)	18,000	18,000	18,000	18,000	18,000	18,000	18,000	18,000	18,000	18,000	18,000
Mpd (kg/deorbit)	2,100	2,100	2,100	2,100	2,100	2,100	2,100	2,100	2,100	2,100	2,100
nLTV (LTV)	3	3	0	0	0	0	0	0	0	0	0
nly (launches/yr)	28	28	28	28	28	28	28	28	28	28	28
ndy (deorbits/yr)	1	1	1	1	1	1	1	1	1	1	1
LLTV (launch/LTV)	100	100	100	100	100	100	100	100	100	100	100
t (yr)	10	1	1	1	1	1	1	1	1	1	1
LOX (kg/yr)	500,000	500,000	500,000	500,000	500,000	500,000	500,000	500,000	500,000	500,000	500,000
Lifecycle cost ($)	152,979,038,889	$15,833,203,889	$15,238,426,111	$15,238,426,111	$15,238,426,111	$15,238,426,111	$15,238,426,111	$15,238,426,111	$15,238,426,111	$15,238,426,111	$15,238,426,111
Cumulative Cost	30,596	$15,833,203,889	$31,071,630,000	$46,310,056,111	$61,548,482,222	$76,786,908,333	$92,025,334,444	$107,263,760,556	$122,502,186,667	$137,740,612,778	$152,979,038,889

Trade off

Cumulative Costs

Year	Mass Driver	TOS	Centaur	OTV Derivative	STS RCS Derivative
1	7,092,750,000	48,600,514,560	41,173,691,550	5,551,096,262	15,833,203,889
2	7,126,500,000	63,700,201,440	42,949,245,283	9,652,846,729	31,071,630,000
3	7,160,250,000	78,799,888,321	44,724,799,016	13,754,597,196	46,310,056,111
4	7,194,000,000	93,899,575,201	46,500,352,748	17,856,347,664	61,548,482,222
5	7,227,750,000	108,999,262,081	48,275,906,481	21,958,098,131	76,786,908,333
6	7,261,500,000	124,098,948,961	50,051,460,213	26,059,848,598	92,025,334,444
7	7,295,250,000	139,198,635,842	51,827,013,946	30,161,599,065	107,263,760,556
8	7,329,000,000	154,298,322,722	53,602,567,678	34,263,349,533	122,502,186,667
9	7,362,750,000	169,398,009,602	55,378,121,411	38,365,100,000	137,740,612,778
10	7,396,500,000	184,497,696,483	57,153,675,144	42,466,850,467	152,979,038,889

Appendix B

Space Habitation Implementable Technology

Electromagnetic Mass Driver Technology Demonstration Description

Mass Driver Technology Demonstrator Overview

The goal of the propulsion group's hardware design effort for the Spring of 1989 was to produce a technology demonstration of an electromagnetic mass driver. The following criteria were used in the design process: it had to be simple, easy to construct, and inexpensive. The first criterion was the driving factor in the design: we opted to not include some of the features of full scale mass drivers such as push-pull coils and capacitors for energy storage.

Technology Overview

A mass driver works on the principle of electromagnetic attraction between coils of current-carrying wire and a magnetized "bucket". The B field of the magnet is normal to the current in the wires, producing an il x B force in the forward direction. With a single coil on, a magnet would accelerate towards this coil, and if this coil is not turned off, the magnet would oscillate about the center of the coil, until it finally came to rest in the coil; this is why a sequencing circuit is needed. This circuit senses where the magnet is in the tube, and turns the coils off as the magnet passes them.

Three types of coil configurations are possible: pull, push and push-pull. The pull system involves activating the coil ahead of the magnet, attracting the magnet. When the magnet passes the center of the attracting coil, that coil is turned off, and the next coil is turned on to attract the magnet; only one coil is on at a time. The push system sequences the coils similarly, but activates the coil on behind the magnet, repelling the magnet. The push-pull is a combination of the two methods: here two coils are on at once, the one in front of the magnet attracting, and the coil behind repelling.

Design Specifics

The propulsion group chose to build a mass driver with the following characteristics:

- 14 Coils of Copper, each with 130 Turns
- Push or Pull, but not both
- Photo-diode / Photo-transistor sensors to "see" the magnet
- Counter Circuitry for sequencing of coils

The goal was to demonstrate the electromagnetic acceleration of a magnet by sequencing of the coils thus demonstrating the technologies

involved in building a full scale mass driver. To this end, we decided to build 14 coils - a sufficient number to demonstrate the sequencing of coils. Photo-transistors were chosen as the sensors to properly sequence the coils - they are simple, cheap, and are much more reliable and less obtrusive than mechanical switches. Instead of building a separate circuit for each coil, we decided to build a master counting circuit to sequence the coils. This both reduced the number of chips required, and provided more flexibility for future improvements to the mass driver.

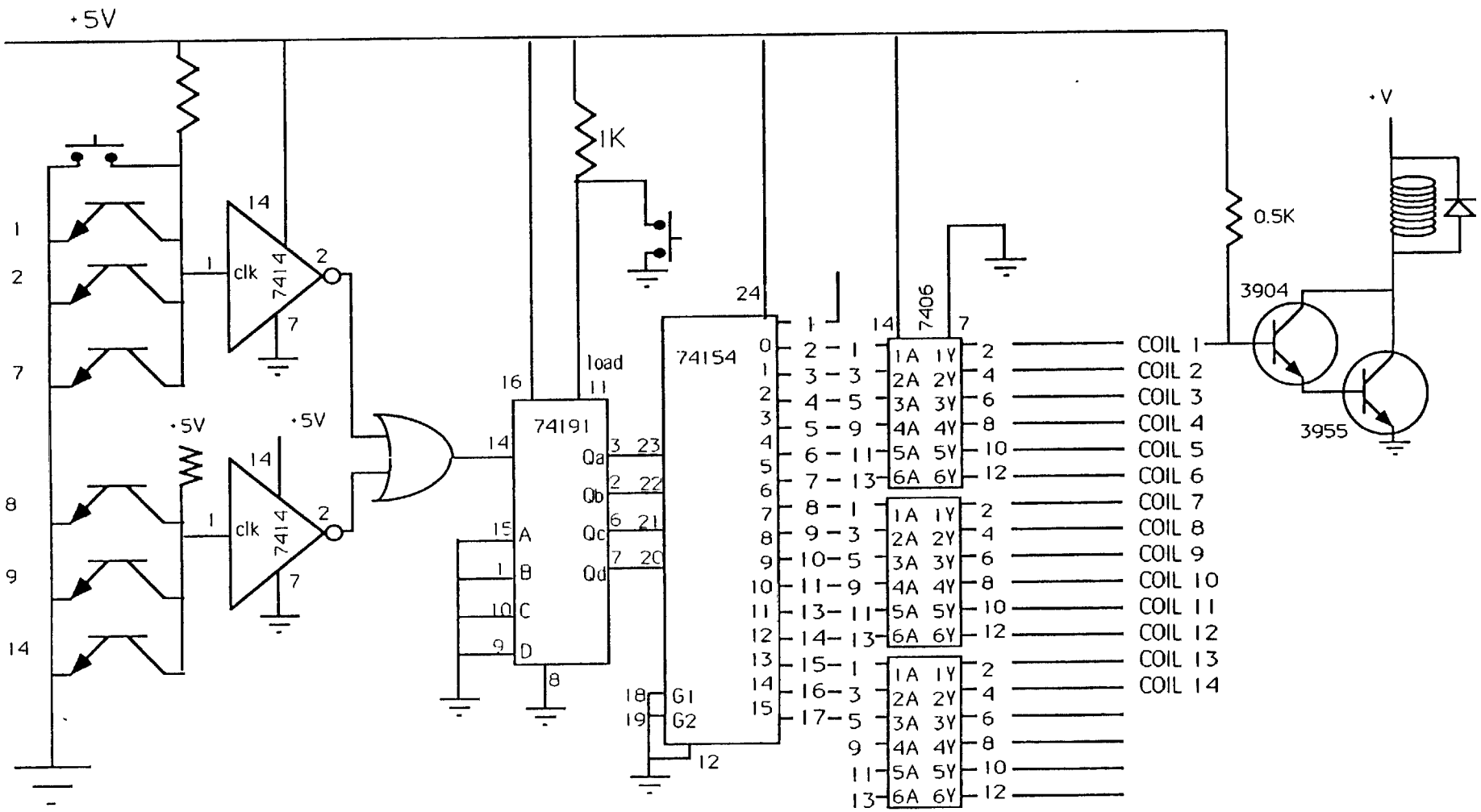
Sequencing Circuit

The controlling circuit consists of 4 main areas: 14 sensors, one counter, a 4-16 converter, and drivers for 14 power transistors (see circuit diagram). When the magnet passes in front of a photo-transistor, a voltage drop is provided to the input of a Schmidt trigger (7414), which then shapes this pulse (makes it more square). This is then connected to the input of a 4 bit counter (74191) that keeps track of the number of the current coil; this counter increments this number by 1 on the leading edge of the input pulse. The output of the counter is connected to a 4-16 converter (74154), which has 16 outputs, only one of which is active (depending on the counter value: if the counter is 1, the first line is active...). This active line is the active coil. Fourteen of these lines are connected to coil driver circuitry, which takes this one line and turns on that coil. This consists of an open collector driver (7506), and a Darlington transistor pair (3904 and 3955). This sends 4.5 amps through the coils at 5.5 volts.

Design Critique

The counter circuit accomplished the sequencing of coils, and allows avenues of expansion for the mass driver design. One of the problems with mass drivers to present has been velocity control. If a computer were interfaced with the counter circuit, it could measure the time between pulses, and compute velocity, and then control it by varying the current in the remaining coils. Energy storage through such devices as capacitors is critical for mass drivers launching objects; this is the one area that our design did not address; this, however, was not in our goals. The propulsion group's mass driver technology demonstrator accomplished it's design goals, and was therefore considered a success.

Electromagnetic Mass Driver Circuit



Circuit by Morgan Jones

Fluids Design Group
Final Report

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Space Habitation
May 8, 1989

Introduction

Fluid technology is a key part of almost any complex technical system and the U.S. space program is no exception. Fluid technology has been, and will continue to be a critical enabling technology in the U.S. and other nations' space programs. The most obvious large scale use of fluids is the use of fuels for rockets and spacecraft. The mass and volume of these fluid propellants and their associated tankage is frequently a design driver in spacecraft and mission design. Propellants comprise a major portion of the fluids used in space and propellant resupply will be essential for space station maintenance, satellite refueling, and Orbital Transfer Vehicle refueling.

The management of water, gases, and biological materials is critical for life support systems. In addition, fuel for power systems and working fluids for cooling systems are important applications within the unique space environment.

An example of the critical role that fluid technology can play is shown by the use of hydrazine fluid propellant in station-keeping propulsion for communication satellites. Each year several communication satellites, costing approximately $100 million dollars, are literally thrown away because they are close to exhausting their hydrazine supply. The technology is currently not available to resupply these satellites, which would save these huge investments. The technology is also not available to accurately gauge the quantity of hydrazine aboard these satellites. To make sure there is sufficient hydrazine left to properly deorbit the satellite, the operators of these spacecraft must deorbit them months earlier than they would if they had a very accurate way of determining the amount of hydrazine left. Since the hydrazine supply usually limits the satellite life to a few years, a few months reduction in useful life is a significant impact.[1]

Fluid Types

Numerous types of fluids have been used in the space program for a wide variety of purposes. Figure 1 is a breakdown of the types of fluids that have been used or are currently being considered for use in the U.S. manned space flight program. The most important consideration in using fluids in space is the low effective gravity when the vehicles are not under acceleration. This makes phase separation more complicated than on the earth's surface. Phase separation refers to the ability to control and and predict the location of the various phases of a multi-

TYPES OF FLUIDS USED IN SPACE

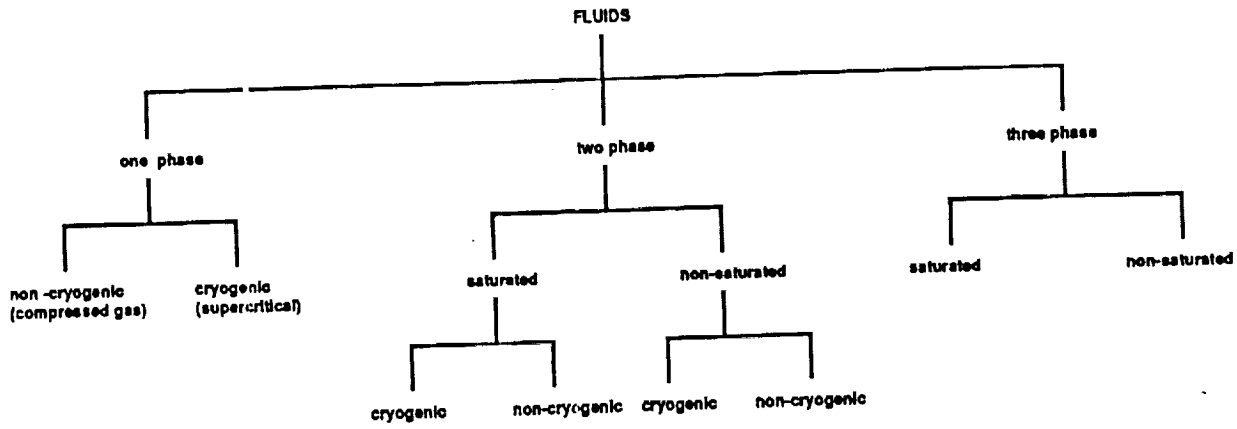


Fig. 1

phase system. In a gravity field as on earth we can usually count on the densest fluid phase being on the bottom of a tank and the least dense being on top. Phase separation is required for very basic operations such as transferring liquids out of tanks, venting the gas from a tank and gauging the quantity of liquid in a tank. Therefore, a key attribute of fluid systems used in spaceflight is the the number of phases.

Single phase systems are the simplest since the phase separation issue is avoided. As a result, single phase systems were used earliest in the space program. Single phase systems consist of compressed gases at either ambient or cryogenic temperatures. The latter are usually called supercritical cryogenic systems since the pressure must be kept above the critical point to maintain a single phase. The major advantage of a single phase system is that the fluid in the the tank is fairly homogeneous, although even single phase systems in low gravity experience some stratification of the fluid into regions of different densities and temperatures. The basic homogeneity of the fluid in a single phase system allows the fluid density to be determined with a single temperature and pressure measurement. This homogeneity also allows the fluid to be removed from the tank in a predictable way. The major disadvantage is that tanks and other components must be able to withstand high pressures. This makes tankage for single phase systems much heavier than multiphase systems and raises safety issues. Due to a long history of the use of high pressure fluid systems on earth, the safety issues are routinely dealt with.

Two phase systems usually consist of a liquid and gas phase and can be either saturated or non-saturated. In a saturated system, the liquid is

in equilibrium with its vapor and therefore is at the boiling point of the fluid. An example of a saturated system would be liquid nitrogen and nitrogen gas at 77 K and one atmosphere. In a non-saturated fluid system the fluid is below its boiling point, and the gas consists of something besides the liquid vapor. An example of a non-saturated system would be liquid water and vapor in a tank with gaseous nitrogen at 300 K and one atmosphere pressure.

In a gravity field or in a spacecraft that is accelerating, the location of the two phases with respect to each other is quite predictable. However, in low gravity the shape, number, and location of the gas bubbles in the liquid is somewhat unpredictable. Reynolds and Satterlee[2] have outlined some of the basic behavior of liquid and gas systems in very low gravity based on classical fluid mechanics. Minimum surface tension energy would force the formation of one bubble instead of several. The bubble would be spherical except where it is attached to the wall. Attachment to the wall would occur if the contact angle of the liquid-gas interface on the wall surface is greater than approximately 53 degrees[3]. Most liquid gas systems have contact angles less than this on metal. The problem is that real systems often exhibit different behaviors from ideal conditions. Real systems often see brief accelerations as large as a milligravity during spacecraft coast periods and even larger during maneuvers. Real systems also have temperature gradients and surface tension driven convection.

The primary disadvantage of using two phase fluid systems in space is having to develop new techniques for venting, transfer, and gauging. The problem with venting is that there is no simple way to make sure that only gas is released from the tank. The problem with transfer is the opposite in that there is no simple way to make sure that only liquid is released from the tank. The problem with gauging is that most gauging systems require the liquid and gas to be in specific orientations with respect to each other.

There are two approaches to dealing with the uncertain location of the gas bubble in a two phase fluid. One is to develop techniques that actively locate the liquid and gas and the other is to develop techniques that are independent of the bubble location. An example is quantity gauging in which surface tension devices can be used to locate the liquid. Radiographic techniques can then measure its depth, or the total amount of gas in the tank can be determined by measuring the system compressibility, independent of bubble location.

The primary advantage of two phase systems is that they can operate at low pressures, therefore minimizing the system weight. As the amount of fluids taken into space increases, the need to reduce the system weight per weight of fluid will cause two phase systems to be used increasingly. Present and future fluid technology development will need to concentrate on developing techniques of venting, transferring, and gauging two phase fluids in low and unpredictable gravity.

Three phase fluids are systems in which the gas, liquid, and solid exist simultaneously. By definition, three phase fluids exist only at the triple point, although pressurant gases can be used to raise the pressure of a system without causing condensation. The advantage of three phase systems is that when the solid density is higher than the liquid, tank weight can be reduced. The disadvantage of three phase systems is that transfer, venting, and gauging are even more complex than in the two phase case, and there are additional problems with ground safety and handling. So far, no three phase fluid systems have been used in space, but slush hydrogen is being considered for use in the national aerospace plane.

The fluids used in space are grouped by use in figure 2. It is obvious that there are a great many different fluids used. Liquid hydrogen and oxygen are used currently in large quantities for propulsion and will probably be used in even larger quantities in the future. This is because of the high specific impulse that can be achieved with this fuel combination. Hydrogen and oxygen are also important as fuels for electric power generation and oxygen is used for life support. The military is also planning on using hydrogen and oxygen as reactants

FLUIDS USED IN SPACE

PROPULSION	LIFE SUPPORT	ELECTRICAL POWER	MATERIALS PROCESSING
hydrogen	nitrogen	oxygen	nitrogen
oxygen	oxygen	hydrogen	oxygen
hydrazines	water		argon
nitrogen tetroxide			
kerosene			
helium			
hydogen peroxide			

HEAT TRANSFER	OPEN CYCLE COOLING	FIRE SUPRESSION
nitrogen	superfluid helium	halon
amonia	liquid nitrogen	
freons		

Fig. 2

for chemical lasers. If earth satellites are refueled in orbit, this would require a large amount of hydrazine and a smaller amount of nitrogen tetroxide. Water is needed for life support and can be turned into hydrogen and oxygen with heat or electric power. Nitrogen is used for life support, small propulsion systems, and instrument or power system cooling.

Superfluid Helium

Superfluid helium is very useful as a coolant for various scientific instruments such as infrared telescopes or experiments with relativity.[4,5,6] Some of the properties of superfluid helium (helium II) are very different from any other fluid, and deserve special notice since these properties make superfluid helium well suited for use in space. In fact, these properties reduce considerably the problems of two phase venting, transfer, and gauging.

Helium II is another phase of liquid helium (isotope 4) that occurs below a temperature of 2.17 K and an absolute pressure of 38 torr. It behaves as if it is made up of two fluid components whose relative concentration varies with temperature.[7] One component, the normal fluid, behaves like conventional liquids while the other component, the superfluid behaves as if it has no viscosity or entropy. The concentration of the superfluid component is zero at 2.17 K and 100% at 0 K while the normal component makes up the balance of the liquid. All of the observable properties of Helium II can be explained by this 'two phase model.' One property is the apparent extremely high thermal conductivity of the bulk liquid. This conductivity has been estimated to be 800 times that of copper and makes helium II a good isothermal coolant for space instruments. The high thermal conductivity also allows a very simple method of quantity gauging through the use of heat addition and specific heat. Helium II liquid will also evaporate within a porous plug if there is a low pressure on one side of the plug and liquid on the other. The liquid will not flow through the plug, providing a method for venting of helium dewars in low gravity. Porous plugs can also be used as a kind of osmotic or thermomechanical pump with no moving parts for use in transfer operations.

History

Fluids in the Mercury and Gemini Programs

The fluids in the Mercury program were limited to life support, launch fuels and attitude control gases. The Mercury program used the Atlas/Redstone launch vehicle using liquid oxygen (LOX) and RP1, a hydrocarbon, as the two components. Rough attitude control was provided by pressure fed hydrogen peroxide. Instrument cooling was provided by circulation of the O_2 around the instruments. The fluids in the Gemini program also consisted of life support, launch fuels, and attitude control in addition to power supply. The systems were more complex due to longer stays in space. The launch vehicle for the Gemini program was primarily the Titan II using hydrazine and unsymmetricaldimethylhydrazine (UDMH) with nitrogen tetroxide (NO_4) as the oxidizer. Life support and power were provided by supercritical H_2 and O_2. When combined, they provided power for the fuel cells as well as drinking water, with excess O_2 used for breathing purposes.

The Apollo Years

The primary applications of fluids handling during the Apollo program included the use of liquified hydrogen and oxygen as propellant in the Saturn V launch vehicle and cryogenic materials for life-support systems.[8,9] Fluids such as nitrogen tetroxide, hydrazine, and supercritical helium were used as propellants for the command, service, and lunar modules.[10] The state of development regarding the properties of low gravity fluids had reached a level where storage of cryogenics for the Apollo missions needed to be expanded to long-term space flights.[11] As a result, numerous problems pertaining to reduction of boil-off of cryogenic fluids remained to be solved during the Apollo program. Additionally, gauging technology of two phase fluids in low gravity environments existed only in the preliminary phases.

Various venting methods being explored or proposed at that time included a) using surface tension to position the vapor space in a known location (ullage-positioning), b) dielectrophoric devices used electrostatic fields to attract the liquid to a specific location in the tank, c) centrifuging, and d) thrusting of the craft to settle the liquid. Orbital tests were proposed to gain further knowledge of fluid behavior in space and to research the various suggested venting methods for the two phase fluids of the Apollo program.[12]

The Space Shuttle

Fluid handling technologies aboard the Space Shuttle may be found in the propulsion, power, and life-support systems. The on-orbit propulsion system consists of an orbital maneuvering system (OMS) for major orbital changes, and a reaction control system to make small adjustments or corrections to the Shuttle's attitude. The OMS consists of two pods located on either side of the Shuttle's vertical fin. The propellants utilized are monomethyl hydrazine as a fuel and nitrogen tetroxide as an oxidizer. Pressurization is maintained in the propellant system by gaseous helium. The OMS engines utilize gaseous nitrogen to operate the engine valves and to purge the fuel lines after a burn to prevent freezing.

Unique handling of these propellants is necessitated by the micro-gravity environment in which they operate. Flexible bladders were not considered to be durable enough for a reusable vehicle. Instead, the propellant tanks utilize mesh screens which rely on the capillary action of the propellant along these screens to capture the fluid. Once an OMS burn is initiated, the acceleration will cause the fluid to move to the aft end of the tank, where it is drawn out. The tanks were thus designed such that the aft end of the tank would always contain fluid.[13]

The gauging of these Shuttle tanks is also quite unique. A probe runs along the central axis of the tank divided into a forward and aft section. Each section determines the fluid level by electrical capacitance within the probe, which reflects the amount of the probe immersed in fluid. Due to the micro-gravity environment, the gauging system updates only when at least one OMS engine is burning. A gauging problem exists when the fluid quantity is between 30 and 40 percent. This 'ungaugable region' lies between the forward and aft probe where the system determines a computed quantity based on measured propellant flow rates.

The Shuttle power system produces the water utilized on board the Shuttle. The power system generates electricity by chemically combining oxygen and hydrogen (which are stored as supercritical cryogens), the by-product of this reaction being water. The power system produces enough of water to supply both the thermal control and life support systems, and then some. Excess water is simply dumped overboard.[14] The water supply is pressurized by nitrogen gas to move the water throughout the Shuttle.

The life-support system draws from two liquid oxygen supplies and two nitrogen gas supplies. The fluids handling problems seem to have been solved by eliminating fluids wherever possible.

The fluid management techniques aboard the Space Shuttle meet the requirements that the Shuttle demands. However, if a more advanced and prolonged presence in space is a goal of the US, new advances in micro-gravity fluid handling must take place.

The Space Station

The fluids to be used on the space station are very similar to those used in past and present applications. Improved fluids management techniques, however, will enable these fluids to be utilized much more effectively. It is envisioned that supercritical nitrogen will be used to provide an artificial atmosphere similar to that found on earth. Liquid hydrogen and oxygen will be used as propellants for thrusters. Additionally, two phase fluids will be used for satellite servicing.[15]

Fluid Demand and Supply

Fluid resupply is the process of transferring fluid from one spacecraft to another. This has been done on only a very limited scale so far, but will become increasingly important in the future for applications such as resupplying satellites, the space station, and the Orbital Transfer Vehicle. Figure 3 is a summary of the demand for fluid resupply in earth orbit over the next 13 years.[16] Note that in 1999 the fluid demand is projected to increase significantly. This is due to the projected start of operations of the Orbital Transfer Vehicle which will need to be supplied with liquid hydrogen and liquid oxygen on orbit. Note that 86% of the mass of the liquid oxygen and hydrogen propellant combination is oxygen. Therefore, after 1999 the majority of the mass of orbital fluid demand is projected to be oxygen.

To date, all fluids resupplied to orbit have come from earth. Figure 4 shows why this may not the best way to perform all of the resupply in the future. This figure shows the relative energy required to get from the earth and moon in terms of the equivalent energy needed to haul a mass against earth's gravity at the surface. It can be seen that the earth sits at the bottom of a 4000 mile deep "gravity well" with low earth orbit 2000 miles out of this gravity well. The moon sits at the bottom of a gravity well only 180 miles deep.[17] Once objects on the moon are launched into lunar orbit, very little energy is required to get

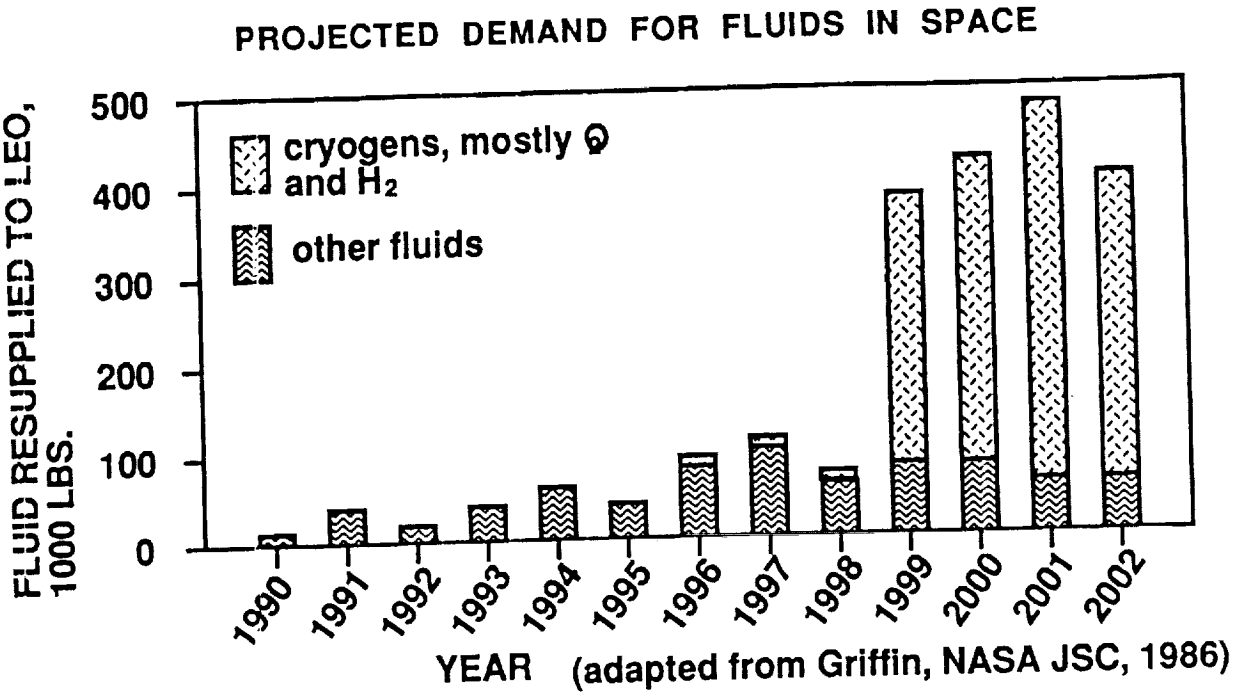

PROJECTED DEMAND FOR FLUIDS IN SPACE
FLUID RESUPPLIED TO LEO, 1000 LBS.
500
400
300
200
100
0
cryogens, mostly O2 and H2
other fluids
1990
1991
1992
1993
1994
1995
1996
1997
1998
1999
2000
2001
2002
YEAR (adapted from Griffin, NASA JSC, 1986)

Fig. 3

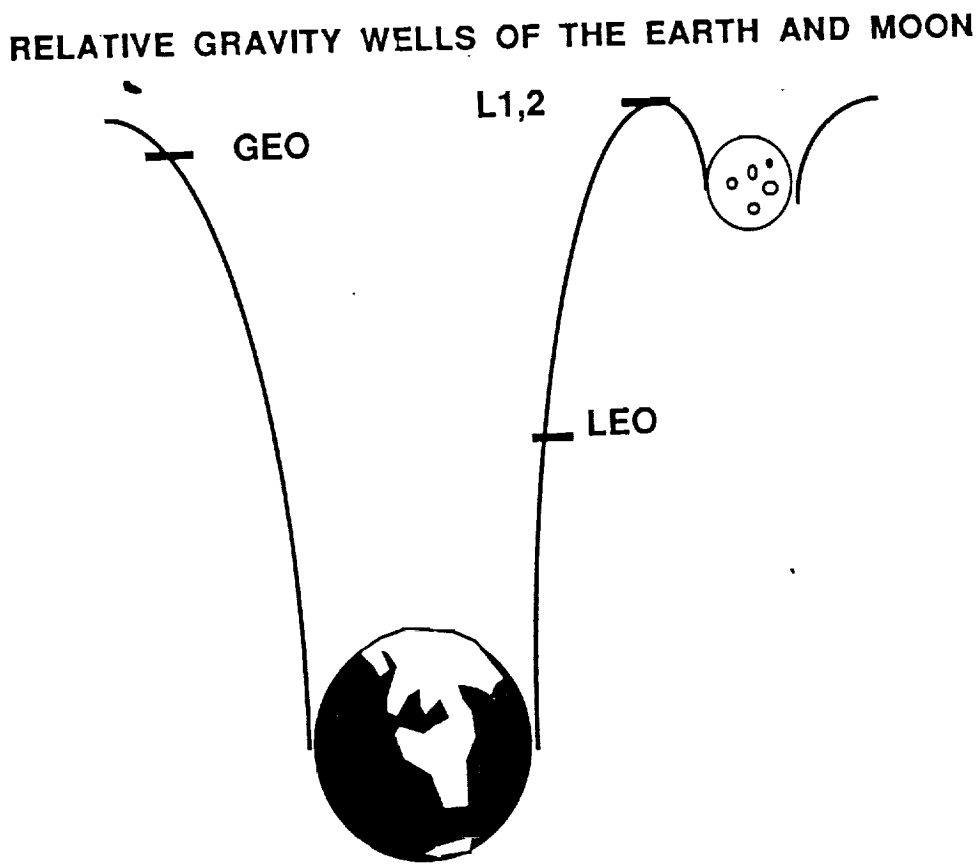

RELATIVE GRAVITY WELLS OF THE EARTH AND MOON
GEO
L1,2
LEO

Fig. 4

objects from the moon to low earth orbit. Since oxygen and hydrogen are available on the moon, and a large amount of oxygen will be required at low earth orbit, we decided to investigate means for efficiently getting lunar hydrogen and oxygen to earth orbit.

Design Concept and Applications

Having identified the potential benefit for delivering fluids to space from the Moon, how might such a concept be implemented? To escape the costs of launching from the Earth's gravity well, it seems logical that not only should fluids be supplied from the Moon, but the tanks to transport them should be produced there as well. Such a process would minimize the amount of mass required to be launched from Earth. To incorporate these concepts, the fluids design group has proposed to manufacture filament-wound utility tanks on the lunar surface primarily from lunar materials for distribution and support throughout the infrastructure.

The core of these tanks are bladders supplied from Earth. These bladders and the manufacturing equipment are the only materials that would be supplied from Earth. The initial cost to deliver this manufacturing equipment is relatively large, but is justified by the low cost of producing tanks once implemented.

Filament winding technology is expanding currently on Earth and this lunar concept takes advantage of the silicates stockpiled by the Automation and Robotics Design Group in the production of water. Earth-supplied bladders are inflated on the lunar surface and then filament-wrapped with glass fibers produced from these stockpiled silicates. The end result of this process is an array of multi-use utility vessels. Current filament-winding techniques utilize a rigid inner tank and thus this concept of winding an inflated bladder represents a new technology.

The strength of this utility tank design concept is its variety of applications. Filament-wound tanks are used in a multitude of roles on Earth ranging from fire-fighting and scuba oxygen tanks to low weight fuel tanks in aircraft. Filament-wound tanks were first used in space on Skylab.

Applications for these proposed tanks range from fluids delivery to LEO and GEO, to a variety of storage tanks, habitation modules, and closed-ecological-life-support-system (CELSS) vessels on the lunar surface.

As a method of fluid transport, these tanks will be launched by a mass-driver system directly from the lunar surface. Such a mass-driver has been proposed by the Propulsion Design Group. In addition, with the implementation of a lunar base, the high cost of delivering structures from the Earth again justifies this concept as one method of providing structures for the Moon.

These applications will be examined in greater detail and in general, the size and strength of these tanks are highly variable and provide the shell for innumerable uses.

Satisfying the Fluid Demands of LEO

In order to give a better understanding of the advantages and disadvantages of our concept a trade-off table was constructed which compares the different forms of supplying O_2 and H_2 to LEO from the earth and from the moon vs. factors including cost, launch mass, implementation time, and reliability.

SATISFYING THE FLUID DEMANDS OF LEO

	WATER		CRYOGENS		GASES	
	EARTH	MOON	EARTH	MOON	EARTH	MOON
COST	1	3	1	2	1	2
LAUNCH MASS	1	3	1	3	1	3
IMPLEMENTATION TIME	3	1	3	1	3	1
RELIABILITY	3	2	3	1	3	2
TOTALS	8	9	8	7	8	8

3=Most favorable
2=Moderate/Possible
1=Poor/Negative

Fig. 5

Regardless of the form in which the fluids will be shipped, there are several advantages and disadvantages depending on where they originate. Since any space mission's primary cost is in direct proportion to launch mass, these two factors are closely related. From the earth, as mentioned earlier, any mass must climb out of a very deep gravity well. Due to the amount of fluids which must be supplied, increasing almost four-fold by the year 1998[16], shipping fluids from the earth will prove costly. This being the prime consideration for our idea, using filament-

wound tanks supplied from the moon will drastically reduce these launch costs. Once our initial factory has been launched the only other launch cost is supplying bosses and bladders to the lunar surface. This could be accomplished with other payloads or supplied in a single separate payload since each combination for the fluid transfer tanks will have a mass around 10.0 kg.

However, our idea of supplying fluids from the lunar surface, has a few disadvantages. Due to the timeline for initial robotic occupation of the lunar surface and consequently the date at which our winding facility would be launched and become operational, it will be approximately thirty years before we can begin delivering fluids to LEO. If they are shipped from the earth they could theoretically be delivered immediately but, once again, at a significant cost. If they are shipped from the earth this method will also be more reliable. Since our facility is completely automated, if an unforeseen or catastrophic problem arises the automated mechanisms might not be able to solve the problem, and the presence of man allows for better trouble shooting and testing of each package before delivery. On the earth these problems would be significantly reduced.

Other factors of consideration for delivering fluids to LEO are the form in which they are shipped. Although the fluids, once they reach LEO, will be used in cryogenic form there are other possibilities in which to ship them to LEO. Shipping them as cryogens, which at first would seem easier since it reduces the amount of processing required at LEO, has many inherent difficulties. From the earth it would be slightly easier than from the moon due to the shorter transit time, but cryogens require extensive amounts of insulation in order to keep them at the extremely cold temperatures of around -200 °C.[18] This insulation adds to the launch costs but does allow very dense amounts of fluid containment. From the moon, long transit times of several weeks using energy efficient transfer orbits would require extensive insulation and active vs. passive cooling systems, which this would add to the complication of an already automated system. As mentioned earlier, venting of these cryogens also becomes difficult in a micro-gravity environment.

As with cryogens, shipping fluids as high pressure gases also requires additional weight. This increase in weight arises from the fluid containment vessel needing to be structurally adequate to handle pressures in the range of 400 atm.[19] This adds to the cost of launch either from the earth or the moon. Although filament wound tanks can

weigh almost 50% less than titanium tanks for a given pressure load the masses required are still large. Also, due to the increased pressure, the danger of a tank leaking or exploding is increased as well.

Water, although perhaps not the best method of shipping H_2 and O_2 from the earth, has several advantages when shipped from the moon. Being a by-product of the ilmenite processing of the Automation & Robotics Design Group it requires no additional processing. This also reduces the required tank weights and increases reliability since refrigeration systems, insulation, pressure check systems, venting systems, and added tank strength are not needed. The water can be shipped in liquid or frozen form or a mixture of each. Due to the high reflectivity of the filament wound tanks the inside temperature will stay around 20 °C.

In summary, shipping H_2 and O_2 in the form of water from the lunar surface offers the best tradeoff in terms of launch costs and simplicity and reliability of form vs. cryogens or gases. The drawbacks are that it won't be available for around thirty years nor is it as reliable as shipping fluids from the earths surface due to the presence of manned intervention in testing or if a problem in production arises. At the bottom of our table we have added the columns, in an unweighted form, to show the advantages and disadvantages of all of these factors.

Design of Transport Tanks

The specifics of our fluid transport tanks are constrained by various factors. Since they will be catapulted off the lunar surface they must conform to the payload specifications of the magnetic launcher proposed by the Propulsion Design Group. These requirements are that the payload must not exceed 125.0 kg., the diameter must be less than 0.5 m, and the tanks must be able to withstand a launch acceleration of 490 g's. By knowing the loading of various factors (tension, compression, collapse pressure, etc.), the maximum required thickness and tank mass can be determined. Subtracting this from the total mass of 125 kg will give the mass of H_2O.

In order to simplify initial calculations the length of the tank was assumed to be 3/2 of the diameter, thus being initially 0.75 m. Since both ends are hemispherical this also simplifies volume and area calculations. Behind the raised ring the tank walls are under tension, the water column mass at 490 g's for this length giving a wall thickness

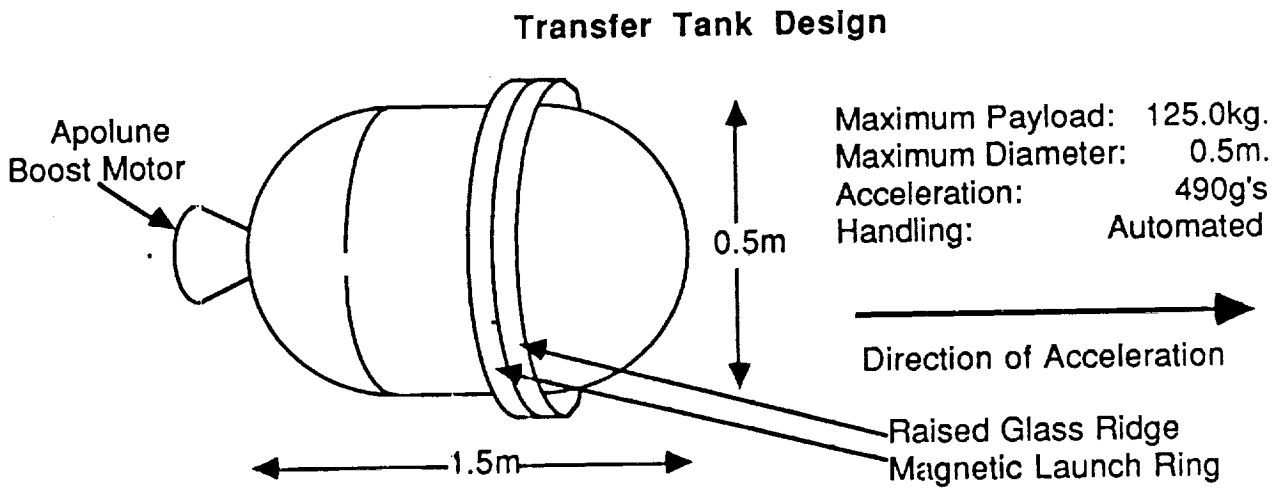


Fig. 6

of 0.1 cm (S2 glass has a strength of 2,818 N/cm^2 (40,000 psi) in a glass fiber matrix form). The compression of the glass in front of the ring requires a much larger thickness so that it will not cave in under its own weight. The front end will contain a high pressure gas, possibly CO_2, another by-product of the ilmenite processing, which will prevent collapse and allow the water space for expansion should freezing occur.

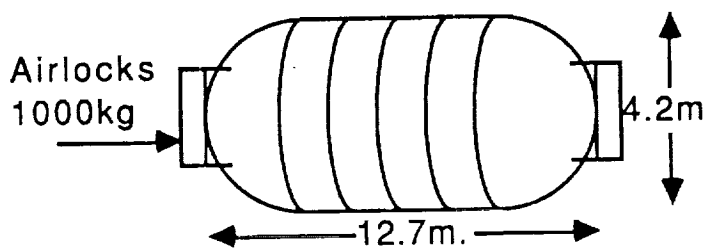
However the external collapse pressure was found to be the critical determiner of wall thickness. This pressure results from the handling mechanisms of our automated facility or of the tank resisting its own weight while resting on the lunar surface. Assuming a safety factor of 6 for extremely rough handling of 1.94 N/cm^2 (27.5 psi) gives a wall thickness of 0.70 cm. This figure for external pressure, 1.94 N/cm^2 (27.5 psi), was found from the tank resisting its own weight on the earth's surface, which also gives the safety factor of 6. As a result, the tank mass is 31.3 kg, with an H_2O mass of 91.3 kg. The size is reduced from 0.5 by 0.75 m to 0.48 by 0.72 m. and the tension loading, due to the increased wall thickness, is now 2,380 g's. This also allows for extremely large g spikes introduced by the magnetic launcher.

Lunar Habitation Modules

As mentioned before, another possible use of these tanks is for manned habitation or utility volumes on the lunar surface. Due to the nature of our process it is very simple to increase tank size. On the earth, filament wound structures have already been made with diameters in excess of 28 ft.[20] Our facility will be able to handle tanks 4.2 m in

diameter by 12.7 m in length. These dimensions were chosen because they allow us to construct tanks of the same size as the Shuttle's cargo bay. This gives the flexibility of interfacing systems designed for use in habitation modules at Space Station Freedom to be used in our tanks and the ability to interface tanks as well. Habitation modules made on the lunar surface could easily be placed in the space station or vice-versa.

HABITATION MODULES: INITIAL DESIGN



Wall thickness: 6.0cm.
Module mass: 15,000kg.
Shielding: 4.0m lunar soil

Fig. 7

As in the case of the transfer tanks, the collapse pressure becomes the critical factor in determining wall thickness and tank weight. Assuming they are made for manned habitation they must be shielded from such factors as cosmic radiation and solar flares. About 4.0 m of lunar soil piled on top of the tanks gives the necessary amount of shielding. This weight, on the lunar surface, requires a wall thickness of 6.0 cm and a mass of 15,000 kg. This weight, along with the fact that bladders and airlocks, acting as bosses, are supplied from the earth, allows us to construct another trade-off table showing the advantages of using filament wound tanks as habitation modules on the moon.

SUPPLYING MODULES FOR THE MOON

	Manufactured on the moon	Delivered from earth
Launch Cost per module	$46 million	$182 million
Implementation Time	later than 2015	Immediate
Mass Launched per module	1730kg.	7000kg
Reliability	Moderate	High
Launch Risk	One Time	Repetitve

Fig. 8

As in the previous trade-off table mass is directly related to cost. Assuming a total of ten modules, either delivered or manufactured, and a cost of $26,000/kg to the lunar surface, we get $46 million per module if they are manufactured or $182 million if they are delivered. For our concept we assumed a total facility mass of 7,300 kg plus 1000 kg/module for the bladder, airlocks, and other equipment needed that could not be made on the moon. For modules delivered from the earth we assumed the more exotic materials such as beryllium could be used, cutting the habitation module mass to half that of the filament wound module. Even assuming this mass savings, our concept still offers significant cost reductions.

Yet, the same disadvantages still exist, those of reliability and implementation time. As above, the factor of man intervening for a crisis or for quality control gives the delivered method an advantage. Our facility cannot produce these modules immediately, but since the permanent presence of manned habitation on the lunar surface is not predicted until after our facility is operational, implementation time is not a real disadvantage. Launch risks, in terms of possible failure or loss of life, are again advantages since the frequency of earth launches for our concept is one-time vs. repetitive risks for the delivered method.

Glass Manufacture

Glass to be used in the fiber winding process will be facilitated by the operation of a hybrid solar-electric furnace detailed in a 1986 proposal by Clemson University.

In principle, the furnace will intake a regolith derivative processed by the Automation and Robotics Design Group, melt it on a 1200 °C hot knob, and maintain a molten state within a graphite composite crucible. Approximately 75 square meters of mirrors will provide the solar power required to melt the incoming silicates at a rate of 350 kg/hr. After the regolith has been melted, it may be held in that phase indefinitely by resistive heating elements in the base of the crucible.

At the specified maximum glass melting rate, a habitation module 4.2 m in diameter and 12.7 m in length with a wall thickness of 6.0 cm may be wrapped in under 43 hours. Fluid transfer tanks could conceivably be produced at the rate of 260/day, limited only by the output

capability of the regolith preparation facility and the need to remove the finished tanks and insert new bladders.

The weight of the glass processing components (crucible, hot knob, mirrors, and fiber spinner head) is a total of 7300 kg.

In order to maintain structural integrity of the glass, it may be desirable to coat the fibers with a metal derived from the regolith. As Clemson outlined in their paper, the best choice would seem to be iron/titanium based on its lunar abundance, structural properties, and the temperature at which its eutectic point occurs. An iron/titanium alloy in a 23/67 weight percentage ratio has a melting point at 1068 °C, which is ideal for wrapping onto glass with a melting temperature of 1150 °C.

Integration Timeline

Integration of manufacture of the filament wound tanks discussed relates closely with the development of the Automation and Robotics Design Group's mining and processing of the lunar surface. The baseline time period begins at the onset of their rover operation. Over the following five years, waste silicates from the process of H_2O production will be stockpiled and stored for later use at the rate of 200 kg of silicates per day. Over the next five years, the necessary equipment for the manufacture of the tanks will be delivered to the lunar surface. This includes the required filament-winding equipment, further lunar processing equipment, and machinery to transfer the materials and finished products from station to station. Integration of the manufacture of the transfer tanks also depends upon the ability of a mass driver to be developed. Such a mass driver would launch the transfer tanks toward earth orbit.

In order for a filament winding concept of transfer, storage, and habitation vessels to be realized, extensive research must be performed in various technological areas..

Future Research

We have identified the following areas in which further research will be required to allow filament wound tanks to be built from lunar materials and for those tanks to be effectively used in the space infrastructure.

Materials

Considerable further work is needed on processes for making glass fibers and matrix materials automatically from lunar materials. The processes developed so far have not resulted in long, continuous fibers. Continuous fibers will be needed for adequate strength of the filament wound tanks and for efficient handling by automated equipment. The use of glass and metals as a matrix material need to be investigated further in order to determine the best approach. The bonding of these matrix materials to each other as the tank is wound needs to be developed. This area of research is the most long term and should begin soonest.

Bladder Technology

The methods and materials for building inflatable bladders will need to be investigated and developed. A likely candidate material for the bladder would be polyimide film, since it seems to fit the requirements of being a strong, flexible and stable material and has a considerable history of use in space.[21] Methods for forming and bonding this material into inflatable bladders need to be demonstrated.

Winding Technology

Once the fiberglass and bladder components have been developed, machines and methods for using them in an automated winding process can be developed. Currently, the winding processes for winding filament-wound tanks are already fairly automated,[22] but work needs to be done on automated methods of starting and stopping the winding process and for fixing problems such as snagged or broken fibers.

Propulsion and Control

Methods of launching tanks from the lunar surface as described in this and the Propulsion Design Group's report will require considerable hardware demonstrations and detailed engineering.

Fluids Handling and Gauging

The transfer, venting, and gauging of two and three phase fluids in low gravity is required for this and numerous other space applications. It is and should continue to be a significant area of space research.

Summary

The use of lunar materials for filament winding on the Moon has tremendous potential for support of a cis-lunar infrastructure. Filament-winding techniques as proposed here would have several applications, including manufacture of transport tanks, habitation modules, and storage vessels. Because the majority of material for such vessels is derived from the lunar surface, significantly reduced mass would need to be sent to the lunar surface. As a result, the overall costs are reduced. Because use of lunar resources has a large cost advantage, they should prove to be an integral future resource in the space environment.

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ROTATING TECHNOLOGIES REPORT TO SPACE HABITATION

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8 May 1989

OUTLINE:

I. Introduction

Rationale
History
Current Status

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Lunar Soil Types/Composition
Lunar Soil Trade-Off

III. Material Separation & Extraction

Separation Process
Zonal Centrifugation
Separator Parameters

IV. Critique

Enabling Technologies

V. Conclusion

Rotation Technology Development
Overview

I. Introduction:

The return to the lunar surface is a goal that has been advocated by the National Commission on Space and Dr. Sally Ride in their reports to the NASA Administrator. This return to the Moon would not simply be for quick visits but the development of a non-terrestrial outpost or base. This base would be dependent on a strong foundation. The University of Colorado Advanced Mission Design Program defined the basic characteristics and evolution of a near-Earth space infrastructure. The infrastructure included a transportation system, L1 space station, and a permanently manned, self-sustaining lunar base. This design did not specify many of the elements that may be required and specific requirements of the support systems were not included. Since many of the problems that arise in space are a direct result of the microgravity environment, the implementation of an artificial gravity force may minimize some gravity related concerns. Artificial gravity may be produced by rotation. The same centrifugal force may be used for

"surface" separation activities. Therefore, rotating systems can be an integral part of the cis-lunar infrastructure.

In general, we defined rotating systems to be those which can generate centrifugal force. This can include anything from a centrifuge to a spinning space station, and includes tethers. From this definition, we briefly examined several applications including space station laboratory hardware, a variable-g research facility, a fluids depot in LEO, momentum transfers via tethers and mineral extraction on the lunar surface.

Rationale:

As space based activities, both manned and unmanned, increase, the use of rotation technology benefits expansion in four major areas. These include material/phase separation, flexible on orbit experimentation, gravity threshold determination and various tether applications.

Material/Phase Separation refers to primarily two potentially cost saving systems; a fluids depot in LEO, and material extraction from non-terrestrial bodies. A fluids depot in LEO could store propellent that could be used to service satellites thus increasing their useful life, or refuel orbital maneuvering or transfer vehicles (OMV's and OTV's). Such a depot could also store aqueous solutions for ECLSS or CELSS applications and other fuels not derived from the Earth.

To truly develop an independent base on the Moon or, in the future, Mars, *in situ* material extraction is a key technology to eliminate consumable resupply from the Earth. By processing the raw materials found, building materials, consumables and possible fuels can be stock-piled until the base is habitable.

Flexible on Orbit Experimentation is the active experimentation that will become more common as human presence is extended within the infrastructure. Here, laboratory tools such as weighing balances, centrifuges and separation technologies will be required. Some of these tools already use rotation technology but it could be applied to all of these areas for the space researcher.

Gravity Threshold Determination follows from the implementation of a variable-g research facility. Advanced planning documents such as Pioneering the Space Frontier have expressed a need for such a facility to determine the amount of gravity required for the reversal of human microgravity maladaption. Other tests on plant growth cycles and possible materials processing could also make use of this facility.

Tether Applications could be widespread in the infrastructure. In theory, tethers could be used to provide artificial gravity, momentum transfers and electrodynamic forces. For example, the artificial gravity produced could be used to predictably settle the phases of fluids in the fluids depot (see Figure 1). Space station could be equipt with tethers so OTV's could be launched as the shuttle is lowered to deorbit. Finally, tethers that contain a conductive core could be used to accelerate or decelerate space craft by interacting with Earth's magnetic field.

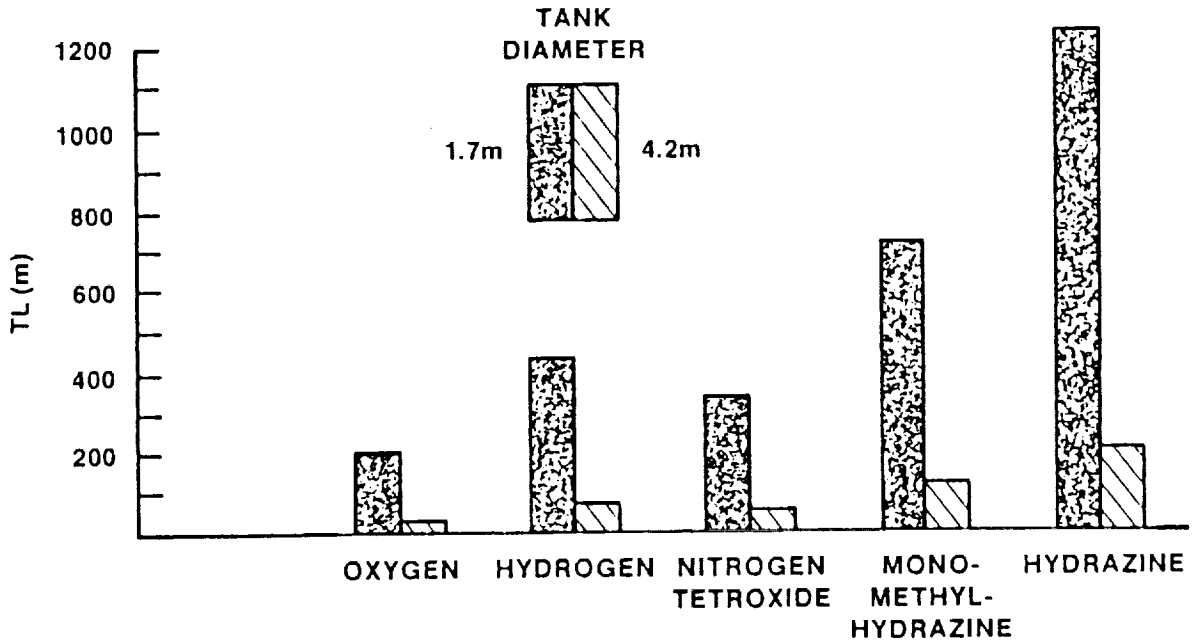

Figure 1. Fluid settling properties of Various Liquid Propellants Under Conditions of Artificial Gravity - Required Tether Length Versus Propellant.

History:

Rotation technology has been used since the onset of the space program. The largest showing has been in the form of spin stabilized satellites. These satellites have a passive stability by rotation about their major axis of inertia and an inherent restoring force. A satellite is also stable about its minor axis if no energy is dissipated. Unfortunately, the very first satellite the U. S. placed in orbit was spun about its minor axis and energy was lost due to its antennae and it began to tumble.

During Gemini XI, a tether was used to link the capsule to the Agena upper stage. This demonstrated the feasibility of tethered rotating systems in space. The tether kept the capsule and booster rocket proximal while studying a two body system.

Variable gravity research began early by studying greater than one gravity or hypergravity on pilots. The next data point came with the Mercury program seeing microgravity effects for short durations. Artificial gravity in space to date consists of one-g centrifuge studies on the Cosmos Biosatellites and ESA's Spacelab. During the Cosmos flights (Ilyin, 1983), various life forms ranging from bacteria to rats and primitive to higher plants were studied. The centrifuged experimental groups were compared with non-spun experimental groups in space. These groups were then compared with synchronous control groups based on the Earth. These ground based controls imitated the space based experimentals in all of the significant factors except for the different gravity environment. Upon return, physiological, morphological and biochemical studies were done on the biomaterial. These studies showed a reduction in space maladaption in the centrifuged rats compared with the weightless rats. However, there were still a number of pathological changes in the centrifuged rats that need further investigation. Effects on the germination of seeds were studied on Spacelab under similar conditions.

Current Status:

Tether Applications are not novel ideas but there is a lack of actual experimentation so the technology can be considered to be in its infancy. Considerations must be given to the materials the tether is made of since atomic oxygen is highly reactive and present in relatively high concentrations in the upper atmosphere. The stability and control of a tethered system must also be investigated further. Rumor has it that a recent shuttle mission deployed a tether that was corroded by atomic oxygen and became tangled, but no information was found that discussed this experiment. Hopefully, the upcoming Tethered Satellite System (TSS) experiments will dispel rumors and provide much needed information on tethers in use.

Gravity Threshold Determination has not progressed much beyond the Cosmos satellites and Spacelab centrifugation experiments. Upon entry into microgravity, there is a cephalic fluid shift which signals the body to excrete fluids. This results in electrolytic and hormonal imbalances and possible changes in excitable tissue (muscle, nerve). In time, there is muscle atrophy, bone degeneration and demineralization. Muscle atrophy and bone degeneration occur primarily in the load bearing muscles and bones. Similarly, the load on the heart is reduced resulting in deconditioning. The calcium lost by bone contributes to the electrolyte imbalance and may manifest itself by forming kidney stones.

About 50% of the astronauts experience Space Adaptation Syndrome. They feel disoriented, vertigo, nausea, and may vomit. This is the result of a neurovestibular and visual signal mismatch. These effects are usually of short duration and probably pose a bigger problem when returning from the microgravity environment if one must safely land a reusable vehicle.

Still, a human is not well suited to long duration stays in this hostile environment and some of the progressive effects (muscle deconditioning and bone decalcification) may never return to the pre-flight baseline even after rehabituation in a nominal gravity environment. Therefore, it is necessary to determine the limiting amount of acceleration required to minimize microgravity maladaption. Presently, there is a lack of data between zero and one-g to determine this threshold.

On Orbit Experimentation consists of prepackaged, autonomous experiments that do not allow for variations in the experimental parameters. Many times these experiments require intervention for trouble shooting but complete protocols are usually not carried out by astronauts on board. The lack of suitable laboratory tools is one of the primary reasons for the lack of active experimentation. Appendix A details a simple centrifugal weighing balance to show the feasibility of using rotating technologies for such purposes.

Material/Phase Separation applications integrated best with the other design groups for the development of a Lunar Surface Infrastructure (LSI) this semester. The major focus became material separation on the lunar surface. This technology is highly developed on Earth but there has been little use in space. As has been mentioned previously, this technology is key to the development of an independent base.

II. Lunar Resources:

Utilization of Lunar Regolith:

The lunar soil contains the raw materials to produce at least four desirable products. These include water, fibergalss, cement anbd propellants. Each of these

products has a direct application to not only the Lunar Surface Infrastructure, but also the general cis-lunar space infrastructure.

Water production from ilmenite reduction requires a hydrogen source for this procedure. It could be in the form of molecular hydrogen resupplied as a consumable, salvaged from the lunar fines, or in a different form such as methane. The chemical equation follows:

$$\underset{\text{(ilmenite)}}{FeTiO_3} + \underset{\text{(hydrogen)}}{H_2} \rightarrow \underset{\text{(titania)}}{TiO_2} + \underset{\text{(iron)}}{Fe} + \underset{\text{(water)}}{H_20}$$

Silicon dioxide (SiO_2) or silica is plentiful in much of the soil and in the anhydrous, hard vacuum lunar environment, fiberglass can be processed to form a material that has interesting mechanical properties. For example, the projected ultimate tensile strength of lunar fiberglass (3.0 GPa) is nearly three orders of magnitude greater than Earth based fiberglass products (see Table 1) but will be brittle. This can be minimized by coating the fibers in a metal matrix. Without water, the silicon and oxygen elements remain covalently bound. If water is introduced, the Si-O bond is hydrolysed and the strong covalent bond is replaced by a hydrogen bond resulting in a weakening of the material.

$$\text{-Si-O-Si-} + H_2O \rightarrow \text{-Si-OH : HO-Si}$$

The reason fiberglass is the glass material of choice is the brittleness of the glass products lends itself best to numerous small cross-sectional area fibers in a more flexible composite matrix (Blacic, 1985).

-	T (GPa/10^6psi)	ρ	E (GPa/10^6psi)	T/ρ (GPa/10^6psi)	E/ρ (GPa/10^6psi)
Aluminum	0.17/0.02	2.7	70/10.2	0.06/0.009	25.9/3.76
Magnesium	0.20/0.03	1.7	45/6.5	0.12/0.017	26.5/3.84
Iron	0.28/0.04	7.9	196/28.4	0.04/0.006	24.8/3.60
Titanium	2.3/0.33	4.6	119/17.3	0.50/0.073	25.9/3.76
Alloy Steel	2.3/0.33	8.2	224/32	0.28/0.041	27.3/3.90
Soda-lime Glass (Earth Environment)	0.007/0.01	2.5	68/9.9	0.003/0.004	27.2/3.95
Lunar Glass (Space Environment)	0.007/0.01-3.0/0.44 or greater?	2.8	100/14.5?	0.003/0.004-1.07/0.16	35.7/5.19?

T = ultimate tensile strength
ρ = specific gravity
E = Young's modulus

Table 1. Mechanical Properties of Lunar-Derived Materials (Blacic, 1985)

Calcium oxides are the necessary ingredient for cement. By combining CaO with silica, alumina, and aggregate and then hydrating the mixture, concrete can be produced. The water required for hydration could be supplied by ilmenite reduction

or must be resupplied as a consumable. The hydration reactions follow (Van Vlack, 1985).

$$Ca_3Al_2O_6 + 6H_2O \rightarrow Ca_3Al_2(OH)_{12}$$

$$Ca_2SiO_4 + xH_2O \rightarrow Ca_2SiO_4 \cdot xH_2O$$

$$Ca_3SiO_5 + (x+1)H_2O \rightarrow Ca_2SiO_4 \cdot xH_2O + Ca(OH)_2$$

The different cements have different strengths as shown in Figure 2. These strengths are dependent on the level of hydration. On the lunar surface, the hard vacuum may make the hydration reaction more complex. Free moisture in any structures built may evaporate but the bonded water should not.

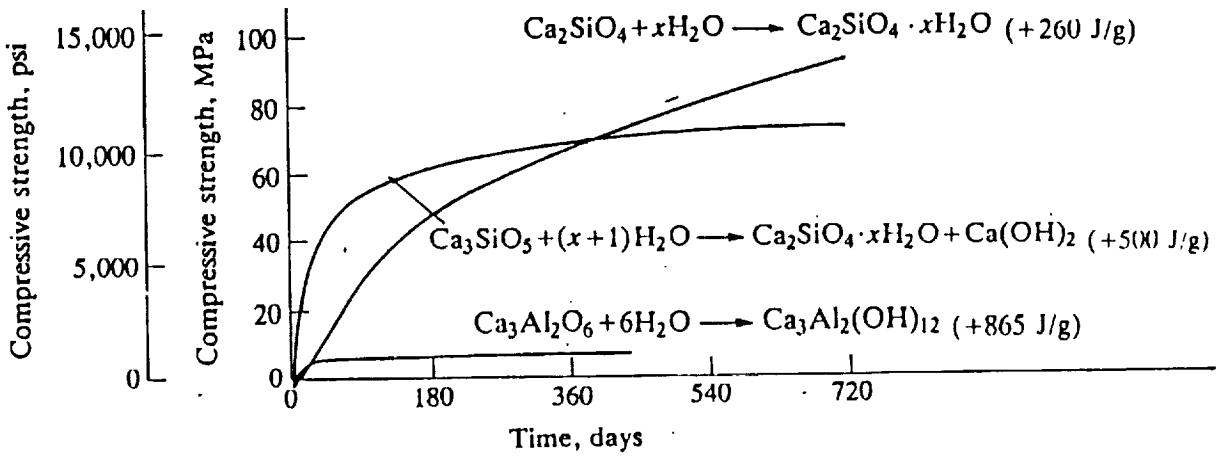

Figure 2. Cement Hydration and Strength

Concrete has several properties that are advantageous for use as building material on the Moon (Table 2). Little energy is required when compared to other construction materials. It has low thermal conductivity but higher heat capacity yielding a heat resistant construction material (Lin, 1985).

Materials	α (10^{-6}/°C)	k (W/m K)	E_{rq}' (GJ/m^3)
Aluminum Alloy	23	125	360
Mild Steel	12	50	300
Glass	6	3	50
Concrete	10	3	3.4 (4.0)*

*H_2O is made from ilmenite.

Table 2. Typical Properties of Construction Materials
where alpha is the thermal capacity, k is the thermal conductivity and E_{rq}' is the energy required for processing.
(Lin, 1985)

Propellants could also be processed from the lunar regolith. An example already given is ilmenite reduction to form water. Water can be electrolysed to form hydrogen and oxygen.

$$2H_2O \rightarrow 2H_2 + O_2$$

Therefore, water produced may be used as an input to a biological lunar CELSS and for oxidant.

Another propellant that could be produced is silane (SiH_4). Silane is the product of energy intensive magnesium oxide (MgO) and silica processing which would require hydrochloric acid (HCl) resupply. Both MgO and SiO_2 would need electrolysis or carbothermal reduction to free the Mg and Si. These elements must then undergo another energy intensive reaction to yield dimagnesium silicide (Mg_2Si). Finally, this can react with HCl to yield magnesium chloride ($MgCl_2$) and silane.

$$2MgO \rightarrow 2Mg + O_2;\ SiO_2 \rightarrow Si + O_2;\ 2Mg + Si \rightarrow Mg_2Si$$

$$Mg_2Si + 4HCl \rightarrow 2MgCl_2 + SiH_4$$

To reduce the resupply of HCl, one more electrolysis reaction requiring water could follow to recycle HCl.

$$2MgCl_2 + 2H_2O \rightarrow 2Mg + 4HCl + O_2$$

Liquid silane is best used as a bipropellant with liquid oxygen. The combination is hypergolic and assuming a mixture ratio of 1.80 ($LO_2/LSiH_4$) in a pressure-fed system, it may give a 345 second specific impulse (Rosenberg, 1985). Table 3 lists some properties of these potential Lunar propellants.

Propellant	Melting Point °F	Melting Point °C	Boiling Point °F	Boiling Point °C	Specific Gravity (liquid)
Oxygen, O_2	-361	-218.4	-297	-183	1.142 (-297°F)
Silane, SiH_4	-301	-185	-169.4	-111.9	0.68 (-301°F)
Methane, CH_4	-296.7	-182.6	-258.3	-161.3	0.46 (-296.7°F)

1. SiH_4 is thermally stable to *ca.* 800°F.
2. O_2/SiH_4 is hypergolic.
3. SiH_4 is a liquid at the nbp of O_2.

Table 3. Physical Properties of Potential Lunar Propellants (Rosenberg, 1985)

Lunar Soil Types/Composition:

Apollo sample analysis has shown that the lunar regolith consists of four soil types. Olivine, pyroxene, ilmenite, and plagioclase feldspars [$(Ca,Na)Al_2Si_2O_3$] make up the vast majority of the regolith components (almost 100%). The feldspar is the primary type of anorthosite rock and is coated with glass. Olivine, pyroxene and ilmenite are the fine grade soils that are best suited for mineral separation and processing.

Compound	Mare, wt %	Highland, wt %
Analyses of Typical Lunar Olivine		
SiO_2	37.36	37.66
TiO_2	0.11	0.09
Cr_2O_3	0.20	0.15
Al_2O_3	<0.01	0.02
FeO	27.00	26.24
MnO	0.22	0.32
MgO	35.80	35.76
CaO	0.27	0.16
	<0.01	<0.01
Total	100.97	100.40
Analyses of Typical Lunar Pyroxenes		
SiO_2	47.84	53.53
TiO_2	3.46	0.90
Cr_2O_3	0.80	0.50
Al_2O_3	4.90	0.99
FeO	8.97	15.42
MnO	0.25	0.19
MgO	14.88	26.36
CaO	18.56	2.43
Na_2O	0.07	0.06
Total	99.73	100.39
Analyses of Typical Lunar Ilmenite		
SiO_2	0.01	0.21
TiO_2	53.58	54.16
Cr_2O_3	1.08	0.44
Al_2O_3	0.07	<0.01
FeO	44.88	37.38
MnO	0.40	0.46
MgO	2.04	6.56
ZrO	0.08	0.01
V_2O_2	0.01	<0.01
Na_2O	<0.01	0.13
Total	102.16	99.37

Table 4. Apollo Sample Analyses
(Rosenberg, 1985)

Table 4 shows the soil composition for both lunar mares and highlands. This data may be important for lunar base site selection. Also, remote sensing or robotic exploration and testing may show regions rich in specific soil types. This could be used for efficient processing of a selected product. For example, fiberglass

production from an ilmenite rich region would be a poor choice since both olivine and pyroxene have higher silica concentrations.

Lunar Soil Trade-Off:

To have the most efficient processor for the desired products, we determined which of the soils would be best for the products previously described. The soils were ranked on a scale from zero (0) to three (3) with 0 being infeasible and 3 being the best choice. The values were assigned by examining the concentrations of usable minerals in each soil type. The soil that had the highest concentration of a particular compound was assigned a 3 signifying that it was the best choice for that given mineral. The intermediate values were assigned by the same method.

Lunar Soil Trade-off

0-infeasible
1-poor
2-acceptable
3-best

Product	Olivine	Pyroxenes	Ilmenite
H_2O	1	2	3
Fiberglass	2	3	1
Concrete	1	3	0
Propellant	3	2	1

Conclusion: Ilmenite/H_2O Production
Pyroxenes/Fiberglass & Concrete
Olivine/Propellant

Table 5. Lunar Soil Trade Off

From the Table 5, it can be seen that ilmenite is the best soil type for water production. Fiberglass and concrete are best processed from pyroxene and silane propellant is best synthesized from olivine. In a more general sense, a lunar base would have the best possibility of development in a pyroxene rich region. Here, many of the structural components needed could be produced plus, water processing capabilities are acceptable.

III. Material Separation & Extraction:

Separation Process:

The proposed method for lunar regolith extraction and separation into the desired minerals for further processing is centrifugal separation as seen in Figure 3. The separator consists of a vibratory screen sieve that will separate the regolith by size so only the fine grade soil is allowed to continue. The soil will be funnelled into a screw auger. As the soil is moved into the separator, hot sulfuric acid (H_2SO_4) is added as a solvent for FeO, TiO_2, CaO and MgO. SiO_2 is insoluble and Al_2O_3 is

only very slightly soluble in sulfuric acid. This mixture passes into a conical shaped chamber. This shape causes a pressure head to develop and begins separating the mixtures components by density. The mixture then enters a zonal centrifuge rotor that completes the separation of the minerals by density. Once separated, the purified material passes into another screw auger which passes through a heating (approximately 340°C) zone to distill the sulfuric acid from the processed material to be reused again in the first screw auger. The distillation process will minimize the need for sulfuric acid resupply since this is a consumable of the procedure. The final step is the expulsion of the material into the lunar environment for vacuum drying.

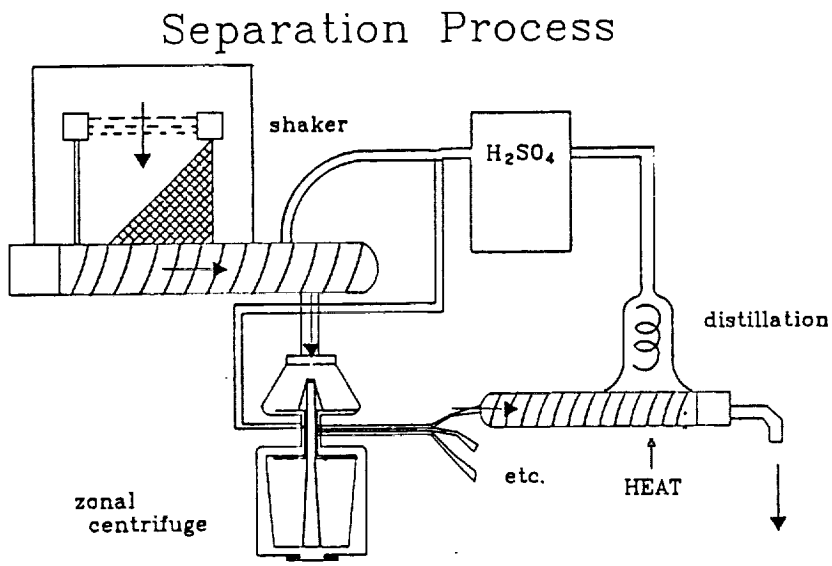

Figure 3. Separation Process

The regolith input will need a cover since the screen shaker will most likely create a cloud of dust and flying particles, and the dynamics of the particulates on the lunar surface is unknown. This separation is similar to the action of combine farm equipment when separating the stalk from the fruit of crops. Also, the dust could contaminate the purified materials during the vacuum drying stage. This cover would have an outlet shoot to expel the aggregate material. If a low grade concrete was needed, this aggregate could be melted together using a heliostat mirror or as the aggregate for higher quality concretes.

Zonal Centrifugation:

Density gradient separations can be done either by sedimentation rate or differences in density. To separate quantities of material, zonal rotors are commonly used. Zonal rotors are basically large cylindrical containers that have a rotating seal so fluid can be transfered in or out of the rotor while it is still spinning. The rotor needed for our separator would be tapered so the material being spun will have the tendency to flow without turbulence toward the top of the rotor. The mixture that enters the zonal rotor will separate by density. The densities of the materials is different enough so separation should be easily obtained. The materials

separate from the center of the rotor outward. Therefore, the least dense material will be near the center and the most dense material will be near the outer edge.

material	density (kg/m^3)
pure H_2SO_4	1.8
SiO_2	2.4
CaO	3.3
MgO	3.6
Al_2O_3	3.97
TiO_2	4.05
FeO	5.2

Table 6. Densities of Raw Materials

From Table 6, it can be seen that silica will float near the center of the rotor and there will be another floating section as a result of the alumina.

Density dependent valves will be placed at strategic locations within the rotor. These are the most critical components of the rotor head. Each of the valves is connected to a separate screw auger and will only be activated when the material of the correct density opens the valve. This way, the processed materials will remain pure.

Separator Parameters:

The rotor of the separator centrifuge will be made out of aluminum and lined with a sulfuric acid resistant layer of Teflon. The rotor has a wall thickness of 3 cm to withstand the load. This thickness was determined using the equations for a thick walled pressure vessile and using a safety factor of 3. The minimum radius is 5 cm and maximum radius is 23 cm. The hieght of the zonal centrifuge is approximately 16 cm. Therefore, this rotor is capable of holding approximately 25 liters of material at once.

Assuming one metric tonne of water is required for a LCELSS per year, and that 27 kg of ilmenite are reduced per day, to yield 6.7 kg of water per day in a 60% efficient system from lunar soil that is 10% ilmenite, 150 liters of material must pass through the centrifugal separator per day. If this system is running 20 hours per day, six "batches" of material must be processed per day with each batch having over three hours to separate. This is more than enough time for density separation with the rotor spinning at a maximum of 20,000 rpm. This rotation rate could yield approximately 100,000 g at the maximum radius.

Because of the size of the rotor, it has a mass of 150 kg. The motor and peripherals (screen shaker, screw augers, etc.) have a mass of 850 kg bringing the total system mass to 1000 kg. The rotor plus the motor assembly need 2 m^3 volume and the peripherals add another 5 m^3 so the complete system requires 7 m^3. Finally, the system uses 20 kW electric power. The power, mass and volume requirements were obtained by scaling up from a Beckman L8-55 Preparative Ultracentrifuge.

IV. Critique:

Enabling Technologies:

A critique of this design shows the technologies needed to implement the separator. First, reliable density dependent valves are the most enabling technology. Without these valves, the system would not be able to operate automatically. Zonal centrifuges used in a laboratory setting are human tended during the addition and extraction of the material being separated. These valves make this centrifuge a combination continuous flow and zonal centrifuge.

Other valves must be autonomous to control material flow rates. Such valves could be used for periodic purging of the system by sulfuric acid. The rotation rate of the rotor must also be monitored to remain stable.

Using sulfuric acid as the solvent brings additional concerns. All of the separators parts that come into contact with the sulfuric acid must be resistant to corrosion. Since this is a consumable that needs to be resupplied, the containment of the acid during transportation will require triple layer vessels for safety purposes. Once on site, the containment of the acid within the separator is slightly less critical since vacuum drying is expected to be the major loss of acid.

To minimize the resupply need of sulfuric acid, technologies to extract the volatile sulfur from the lunar regolith and subsequently synthesize the acid on site would be beneficial.

Finally, it is highly probable that manned sorties will be needed to maintain the system. This may be in the form of trouble shooting or replacing parts. For this reason, the system should be as modular as possible.

V. Conclusion:

Rotation Technology Development:

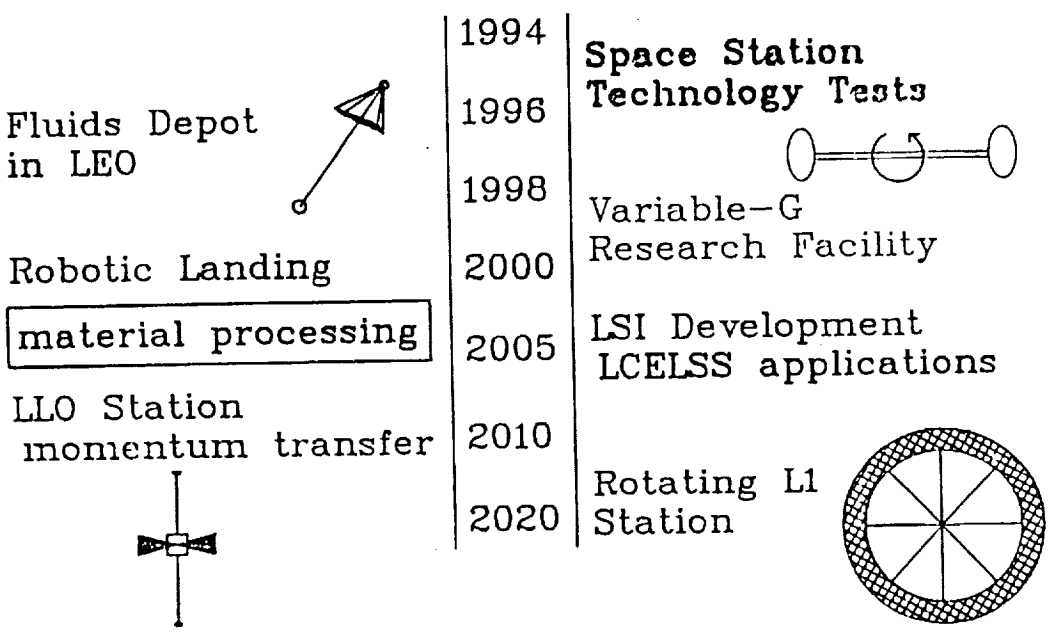

Figure 4. Time Line.

Many of the technologies applicable to support of lunar surface activities (see Figure 4) should be tested on the space station. Some specific technologies such as the laboratory tools would be beneficial at this point. On orbit refueling is a priority in the cis-lunar infrastructure because of its potential cost savings. Thus, the fluids depot in LEO could use rotation technology to assist in fluids management. To begin some actual lunar gravity experimentation, the variable-g research facility would be needed next. This facility could also be used to determine the gravity threshold.

After lunar remote sensing, a robotic landing could begin work on the initial stages of a lunar outpost. With the robotic landing, the material separator discussed above could begin stock-piling useful materials for the development of a Lunar Surface Infrastructure. As this system develops, rotation technology may be needed for biomass separation in a LCELSS. If the gravity on the Moon is not conducive to full cycle plant development, a centrifuge could be used to orient the plants during germination or as long as required.

With a low lunar orbiting station, material to be sent to LEO could be launched from the lunar surface to the LLO station. From the LLO station, orbital transfers could be launched utilizing tethers.

Finally, a rotating station at L1, as described by previous Advanced Mission Design Classes, could be the hub of activity between the Earth and the Moon. It may serve as a reacclimation facility or an interplanetary staging point.

Overview:

The return to the Moon has been the goal of the design groups. To accomplish such a task, and begin the process of developing a truly independent base, the rotation technologies design group focused on the extraction and separation of useful material from the lunar regolith.

To realistically develop a lunar base, a strong cis-lunar infrastructure is needed. Rotation technologies can be beneficial by supplying the tools for active on orbit experimentation, systems for material/phase separation, a variable-g research facility, and orbital transfers via tethers.

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Appendix: Technology Demonstrator.

The requirement for extensive on-orbit experimentation in the proposed cis-lunar infrastructure necessitates the development of experimental hardware designed to operate in a micro-gravity environment. Several devices employing rotating systems and their resultant centripetal force could be advantageously used in the preparation of materials for on-orbit experiments. One such device is a centrifuge for the weighing of experimental materials. A inexpensive prototype of a weighing centrifuge was constructed for this semester's technology demonstrator.

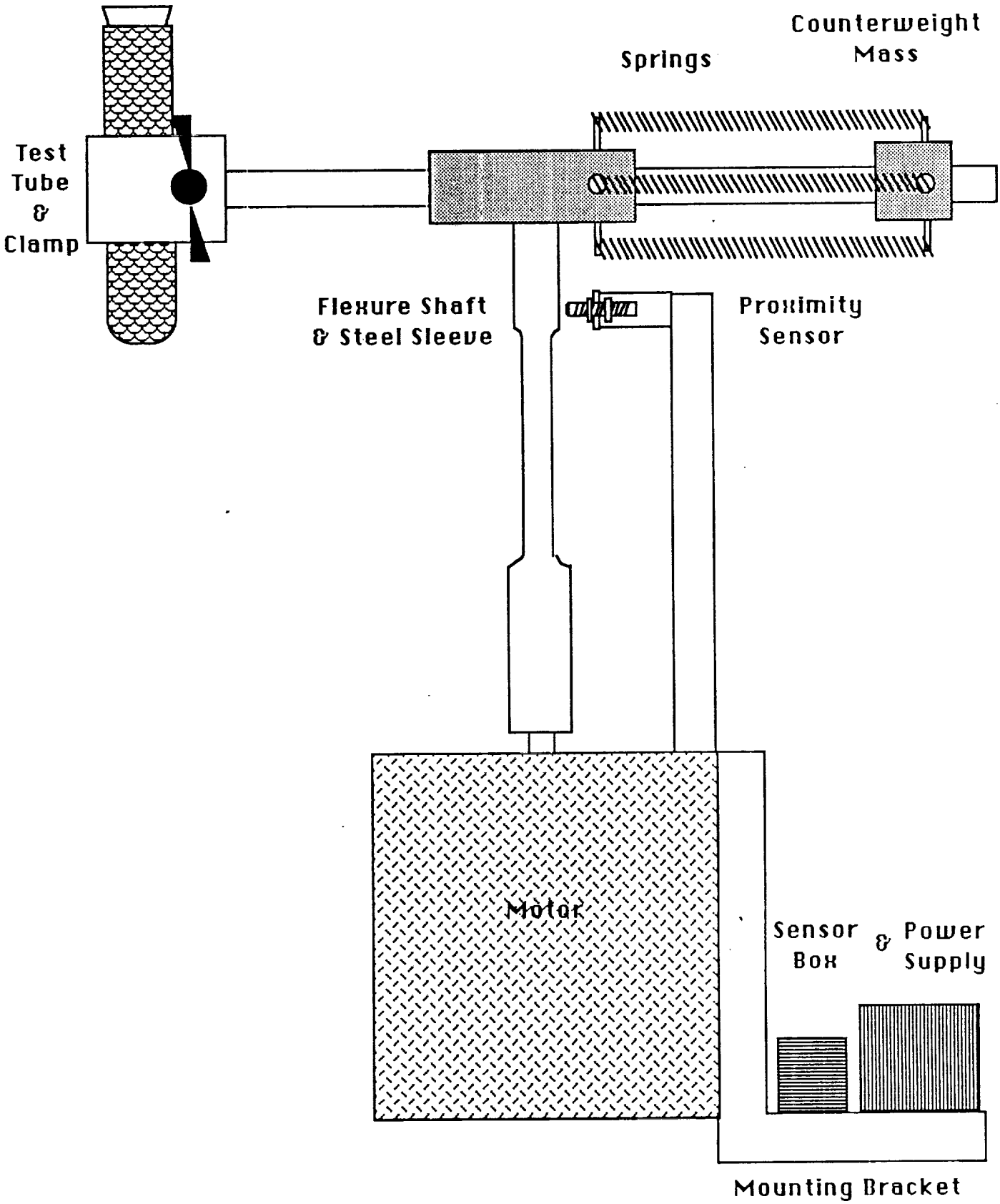
The weighing of materials in the presence of a gravity field is well understood, and devices to perform this task have been in existence for several centuries. The triple beam balance is perhaps the most familiar example of such a device. The operation of the triple beam balance involves the comparison of moments produced about the balance fulcrum. Here, known masses are placed at certain specified distances from the fulcrum, until the balancing of the unknown. mass is achieved. The weight can then be read off the calibrated scale associated with the counterweight masses. In the absence of a gravity field, different methods must be employed to generate forces which scale as mass, and which can be measured or compared in order to give an indication of the mass of the material in question. As mentioned previously, our instrument uses centripetal forces generated by rotation to give an indication of balance, and ultimately the mass of a quantity of material.

As shown in the accompanying figure, our device employes a cantilevered, flexible shaft to translate a force imbalance into a displacement which can then be measured with a capacitance-type proximity sensor. These forces are the centripetal forces experienced by the mass of the fluid to be measured, and by the counterweight mass on the other end of the rotating beam. In operation, the magnitude of these forces and the position of the counterweight mass relative to the axis of rotation are governed by the rotation speed (omega), as these simple equations indicate:

$$F=ma=-kr=mw^2 r$$

Furthermore, the forces which are compared to give an indication of "balance" are:

Centrifuge For Mass Determination



$$F = (m w^2 r)_{unknown} \qquad F = (m w^2 r)_{counterweight}$$

These forces act in opposite directions at the upper end of the cantilevered flexure shaft. The vector resultant of these forces produces a deflection of the shaft which then produces a sinusiodal signal in the proximity sensor. By selecting the appropriate rotation rate, the resultant force acting on the end of the flexure shaft can be made zero, and the position of the counterweight mass at that time can be related to the mass of the material which is being measured.

Our centrifuge balance was designed to weigh either liquids or fine solids. As can be seen from the figure, this material is contained in a fixed volume, a test tube. Consequently, the radial position of the centroid of the material as it settles against the outermost side of the test tube varies with the amount of material present. The calibration of the balance will have to account for this geometrical effect in some manner. The solution adopted was to calibrate the counterweight balance position using known volumes of water (known mass) in the test tube. A calibration equation relating the balance point mass indication to an adjusted value based on the ratio of the density of the settled fluid to that of water can be derived from geometrical considerations. An adjustment for use with fine solids is more difficult, and requires knowledge of packing factors and degree of settling. A similar instrument with a geometry which would eliminate this problem may need to be developed for solid weighing. To aid in the calibration, a series of marks was placed on the counterweight beam, and the marks and the counterweight were painted white for ease of visualization with a strobe light. The resulting calibration data and instrument parameters are presented in the second figure.

The operation of the device is as follows:

1. A quantity of liquid or fine solid is placed in the test tube and the test tube is then sealed with a rubber stopper.

2. The centrifuge is then spun up to an omega which places the counterweight at the extreme outer position on the support rod.

3. By slowly decreasing the power to the motor, the counterweight can be made to move inward.

4. When the amplitude of the proximity sensor signal is minimized, the balance point is reached.

5. The position of the counterweight mass is determined by visual inspection, aided by a strobe light.

6. The position of the counterweight mass is related to the unknown mass of the material by reference to calibration data.

In practice, the utilization of this apparatus requires two operators, since the device utilizes no feedback to control rotation rate. Consequently, the balance state is not achieved statically, but exists in transition from one out-of-balance regime to another. The operation requires that the location of the counterweight mass position be accomplished in a very short period of time, and this necessitates the need for two operators. This is obviously a serious drawback, and any really useful instrument must have features which make its operation possible and easy for one person. One nice feature of our apparatus is that no electronic information or power transfer is required across a rotating interface. All information is transmitted by indirect (reflection, capacitance) means.

Possible modifications of our baseline concept could significantly improve operation. The essential improvement for any device would be to add a bearing assembly to eliminate noise resulting from poor mounting of the flexure shaft to the motor. This was a significant noise source in our device, and therefore there was a sinusoidal signal of small amplitude at the balance point. Ideally, this signal should be made to virtually vanish at balance. After this simple modification is made, numerous options exist for improvement. A balance indication which is insensitive to rpm is the next improvement. The counterweight positioning can be accomplished using a D-C stepper motor and lead screw arrangement as shown in the next figure. This arrangement requires the need for a slip ring to transfer information and power across a rotating interface. The signal for balancing can then be held constant (for a given rpm) and can still be measured from outside the rotating system. The desirable feature of this system, as in our demonstrator, is that the

Technology Demonstrator Calibration

Volume: 0 - 100 ml.

Mass of Counterweight: 130 g.

Nominal Rotation Rate: Approx. 150 - 200 rpm.

balance indication goes to zero at the balance point. The mass can therefore be determined to a resolution determined by the system noise. Another possible improvement is to allow the flexure shaft to flex, and then measure the deflection at a specified rpm. A counterweight with a fixed set of positions, similar to the larger weights on a triple beam balance, can be employed to increase the range of operation of the device. The accuracy of such a device depends primarily on the degree of control of the rotation rate, and the ability to correlate the sensor signal to the unknown mass. Proximity sensors (capacitance or optical) can be utilized to eliminate the need for information and power transfer across a rotating interface, and are probably the best choice. Finally, the ability to add material to the test volume while in operation is a very desirable feature and some method of accomplishing this task so that a given mass of material can be made up is needed for any reasonably efficient system. One such possibility for fluid metering to the test volume is shown schematically in the figure.

With suitable modifications and refinements, a centrifuge for the weighing of materials for use in laboratory experimentation and eventually commercial production should find widespread use throughout the cis-lunar infrastructure.

Centrifuge For Mass Determination