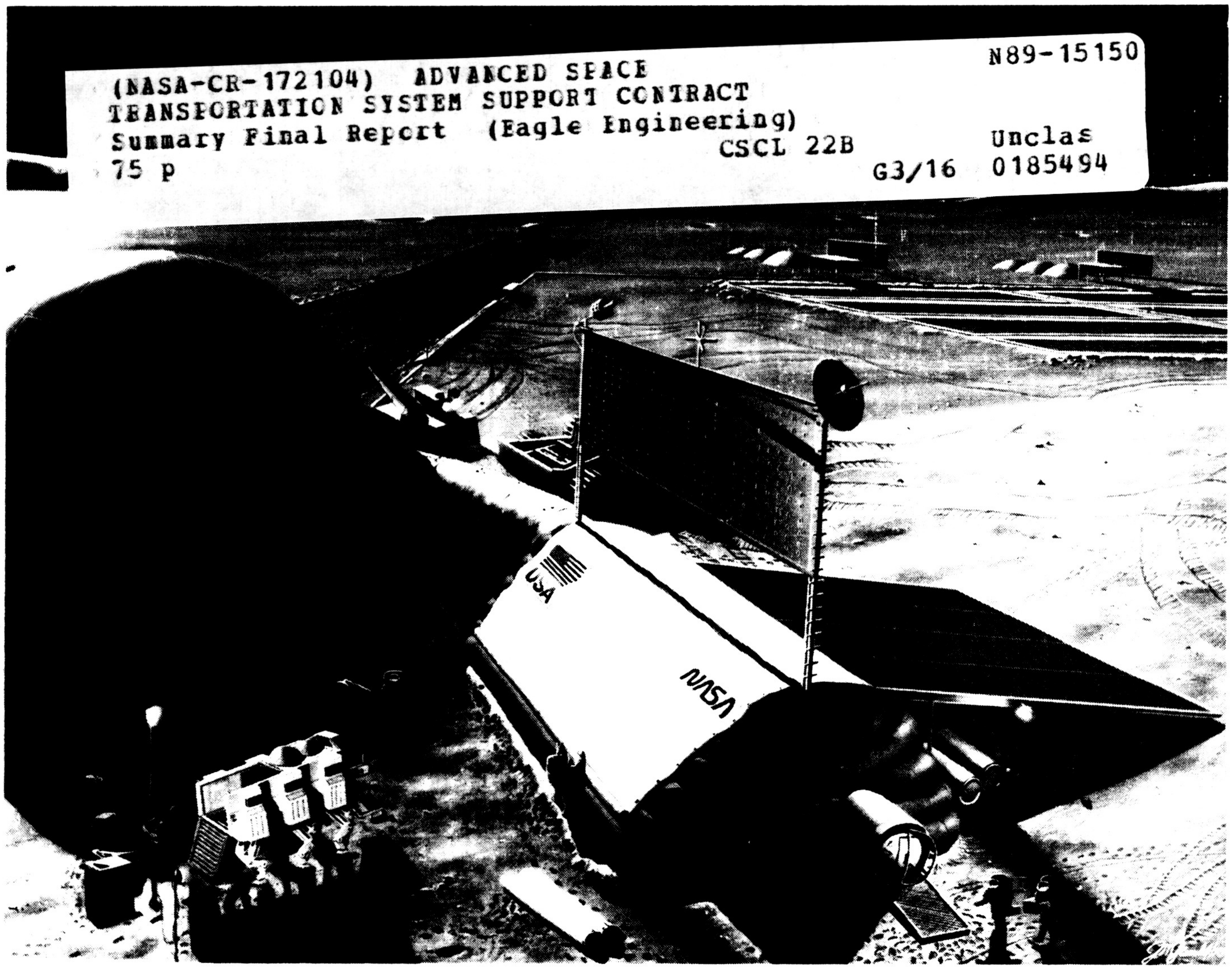
CR # 172104

Advanced Space Transportation System Support Contract Summary Final Report

LBSS

06

(NASA-CR-172104) ADVANCED SPACE TRANSPORTATION SYSTEM SUPPORT CONTRACT Summary Final Report (Eagle Engineering) 75 p
N89-15150
CSCL 22B
Unclas
G3/16 0185494



NASA Contract No. NAS9-17878
Eagle Eng. Report No. 88-210
October 30, 1988

EAGLE

Advanced Space Transportation
System Support Contract

Summary Final Report

October 30, 1988

National Aeronautics and Space Administration
Lyndon B. Johnson Space Center
Advanced Projects Office

Prepared by:
Eagle Engineering, Inc.
Houston, Texas

NASA Contract NAS9-17878
Eagle Engineering Report No. 88-210

Foreword

This report is a summary of work completed from Oct. 1987 through Oct. 1988 under the Advanced Space Transportation System (ASTS) support contract part of the Lunar Base Systems Study (LBSS). The LBSS is a start at building the tools and ideas needed to return to the Moon. Two other studies were performed under the ASTS contract concerning the Mars Rover/Sample Return Mission analysis; summaries are not included here, however. This report contains only lunar base work. The following individuals participated in the lunar studies:

NASA Technical Monitor - John Alred, Ph.D.

Eagle Project Managers - Bill Stump and Barney Evans

Studies Conducted with NASA and Eagle Task Managers:

Task	**NASA Manager**	**Eagle Manager**
Trans-Lunar Software Development	Andy Petro	Jack Funk Gus Babb Chris Varner
Surface Mission Operations Study	Kyle Fairchild	Eric Christiansen
Conceptual Design of Lunar Landing Pad	Mike Roberts	Paul Phillips
Transportation Node Assumptions and Req.	Jonette Stecklein	Bill Davidson
Maintenance and Supply Options	Ann Bufkin	John Hirasaki
Concept. Design of Lunar Oxygen Plant	Kyle Fairchild James Sturm	Eric Christiansen
Concept. Design of Lunar Lander	Jonette Stecklein	Bill Stump
Concept. Design of Surface Transportation	Ann Bufkin	Bill Davidson
Storm Shelter Conceptual Design	Mike Roberts	Bill Gill
Concept. Design of Solar Power Plant	Mike Roberts	Hoyt McBryar
Concept. Design of Transp. Node Station	Jonette Stecklein	Bill Stump Charles Simonds, Ph.D.
Concept. Design of Surf. Const. Equip.	John Alred, Ph.D. Ann Bufkin	Bill Davidson Eric Christiansen
Lunar Applications of Superconductivity	Mike Roberts	Steve Erickson
Lunar Element Cost Estimation	John Alred, Ph.D.	Steve Erickson

David Korsmeyer with the Large Scale Programs Institute wrote and documented the low thrust trajectory program in BASIC.

Other participants from Eagle:

Alexander Adorjan, PhD
Taher Ali Kahn
Michael Amoroso
Carolyn Blacknall
Ronald Blilie
David Carson
Thomas Chambers
Gil Chisholm
Carolynn Conley
Michael D'Onofrio
Mark Dowman
William Engblom
John Euker
Lee Evans
Richard Ferguson
Anthony Hodgson
Paul Jaminet
John Kiker
John Lowery
Karl Maples
Douglas McLeod
Owen Morris
John Nagel
Samuel Nassiff
Gregg Nudd
Christopher Orloski
John Overton
John Philips
Pat Rawlings
Bass Redd
Hildebrando Ruiz
Norman Smith
Michael Stovall
Donald Sullivan
Donovan Teegarden
Scott Udell
John Watson
Steve Woodling
Charles Yodzis
Thomas Zakrzewski
Scott Zimprich

[redacted]

Table of Contents

List of Figures

List of Tables

1.0 Executive Summary

The ASTS contract was subdivided into a number of small studies and trans-lunar software development efforts. The general focus of all studies was on a phase II lunar base, or a lunar base during the period after the first return of a crew to the Moon, but before permanent occupancy. The software effort produced a series of trajectory programs covering low Earth orbit (LEO) to various node locations, the node locations to the lunar surface and then back to LEO.

The surface operations study took a lunar scenario defined in the Civil Needs Data Base (CNDB) and attempted to estimate the amount of space-suit work or extra-vehicular activity (EVA) required on the lunar surface to set up the base. More EVA and crew time was predicted to be needed to set up this base than was easily available. The proposed solution was teleoperation from the lunar surface and from Earth.

The maintenance and supply options study was a first look at the problems of supplying and maintaining a lunar base. Maintenance and supportability must take a higher profile in a lunar base than in any previous program. Spares and consumables are small numbers for short missions (less than 30 days) but dedicated logistics landings may be required for 180 day stays.

A conceptual design for a multi-purpose, single stage lunar lander was produced. The lander was to be returned to LEO for refurbishing and propellant loading after each mission. Numerous trades were examined and Apollo data was collected.

A lunar surface launch and landing facility was conceptually designed, consisting of transponder navigation devices and lighting, surface power and thermal control carts, and a pressurized ingress/egress concept. Ejecta from engine blast was found to be a serious problem requiring the landing area to be some kilometers from the base.

After careful comparison of thirteen different processes, two processes were chosen as candidates for a lunar oxygen pilot plant and plants were conceptually designed. The ilmenite reduction process plant had a landed mass of 25 m tons and required 146 kwe of solar power. The thermal recovery of oxygen and solar wind hydrogen process required 60 m tons landed mass and a 1.7 MWe nuclear source.

The lunar storm shelter study examined the problems of radiation protection on the lunar surface. Dose limits were proposed for various missions and a range of shelter concepts weighed and sized.

The LEO Transportation Node Assumptions and Requirements study attempted to document the assumptions and requirements needed to define a LEO space station supporting a lunar transportation system consisting of orbital transfer vehicles (OTVs) and landers. Requirements were derived based on the assumed activities needed.

A LEO transportation node space station was then conceptually designed to maintain and refurbish two lunar lander/OTV stacks. The station was designed to support eight flights to the Moon per year. The station's maximum loaded weight (with two stacks) is around 1,000 metric tons. The dry weight is around 400 m tons.

Three concepts for lunar surface transportation were examined. An unpressurized rover with a 50 km range weighed 550 kgm empty and required 2.15 kwe of power. A pressurized rover, fuel cell powered, in a train configuration weighed 18 m tons and used 25 kwe peak power to travel 3,000 km in 42 days. A hopper, based on the surface and using the previously designed lander, could travel only 1,000 km from the base. For this reason, descent from orbit was recommended for ranges over 1,500 km.

The Lunar Surface Construction and Equipment Assembly study defined twenty surface construction and assembly tasks in detail. Terrestrial equipment was then surveyed and a variety of different concepts for performing the tasks were identified and compared.

Cost estimates for the development and production of a number of the previously discussed lunar program elements were produced using a widely accepted costing program.

Three applications of superconductivity to a lunar surface base were examined; electrical energy storage, electromagnetic launchers, and magnetic radiation shields.

The initial product of the lunar software effort was a short book with weights of all previous spacecraft and subsystems of interest to aid in spacecraft and subsystem mass estimation.

Seven programs were produced to estimate delta Vs and trans-lunar trajectories:

The first program (LLOFX) calculates in-plane trajectories from a LEO space station to low lunar orbit (LLO) using only two burns. LEO orbit altitude, lunar orbit altitude, and flight time are the inputs to this patched conic style program. The two delta Vs are the output.

The second program (PLANECHG) also calculates LEO to LLO and back trajectories using patched conic methods. It uses three burns however and allows the choice of any altitude, inclination, and longitude of the ascending node for LEO and LLO departure or arrival orbits.

CISLUNAR calculates trajectories from LEO to LLO and back for low thrust vehicles using an integrator. The low-thrust vehicle's characteristics must be supplied and guidance schemes must be manually adjusted. All trajectories must be in the plane of the Moon's orbit.

LANDER calculates ascent and descent trajectories between LLO and the lunar surface using an integrator. Vehicle characteristics must be input.

LIBRATE calculates delta Vs for trajectories from LEO to four of the five Earth-Moon libration points. The libration point on the farside of the Moon (L1) is excluded. LIBRATE also runs trajectories from LLO to L1 and L2. L1 and L2 are the libration points on the Earth-Moon line on the far and near sides of the Moon respectively. Inputs include departure orbit altitude and inclination and flight time.

LP1 calculates trajectories from LEO to and from L1 on the far side of the Moon. Lunar flybys are used.

A short report documents delta Vs from LLO to L4 and L5 and back as a function of flight time. Transfers from LEO to LLO by way of L4 of L5 require on the order of 760 m/sec more total delta V than direct LEO to LLO transfers.

Figure 1.0-1 shows a base moving into the permanent occupancy (phase III) state. A "lunar shack" or single module is first placed on the surface to serves as a construction habitat. A large inflatable habitat is installed later, as well as a lunar oxygen pilot plant, a 100 kw continuous solar power plant, and other systems discussed in later sections. The inflatable is being covered with regolith with a continuous bagging machine. Permanent landing pads are visible in the distance and a small vehicle carrying cryogenic propellants is returning from the landing pad area.

Figure 1.0-1, Lunar Surface Base

ORIGINAL PAGE
COLOR PHOTOGRAPH

2.0 Lunar Surface Operations Study Summary

The purpose of this study was to perform an analysis of the surface operations associated with a human-tended lunar base. Specifically, the study (1) defined surface elements and developed mission manifests for a selected base scenario, (2) determined the nature of surface operations associated with this scenario, (3) generated a preliminary crew extravehicular and intravehicular activity (EVA/IVA) time resource schedule for conducting the missions, and (4) proposed concepts for utilizing remotely operated equipment to perform repetitious or hazardous surface tasks. The operations analysis was performed on a 6-year period of human-tended lunar base operation prior to permanent occupancy. The baseline scenario was derived from a modified version of the civil needs database (CNDB) scenario. The scenario emphasizes achievement of a limited set of science and exploration objectives while emplacing the minimum habitability elements required for a permanent base.

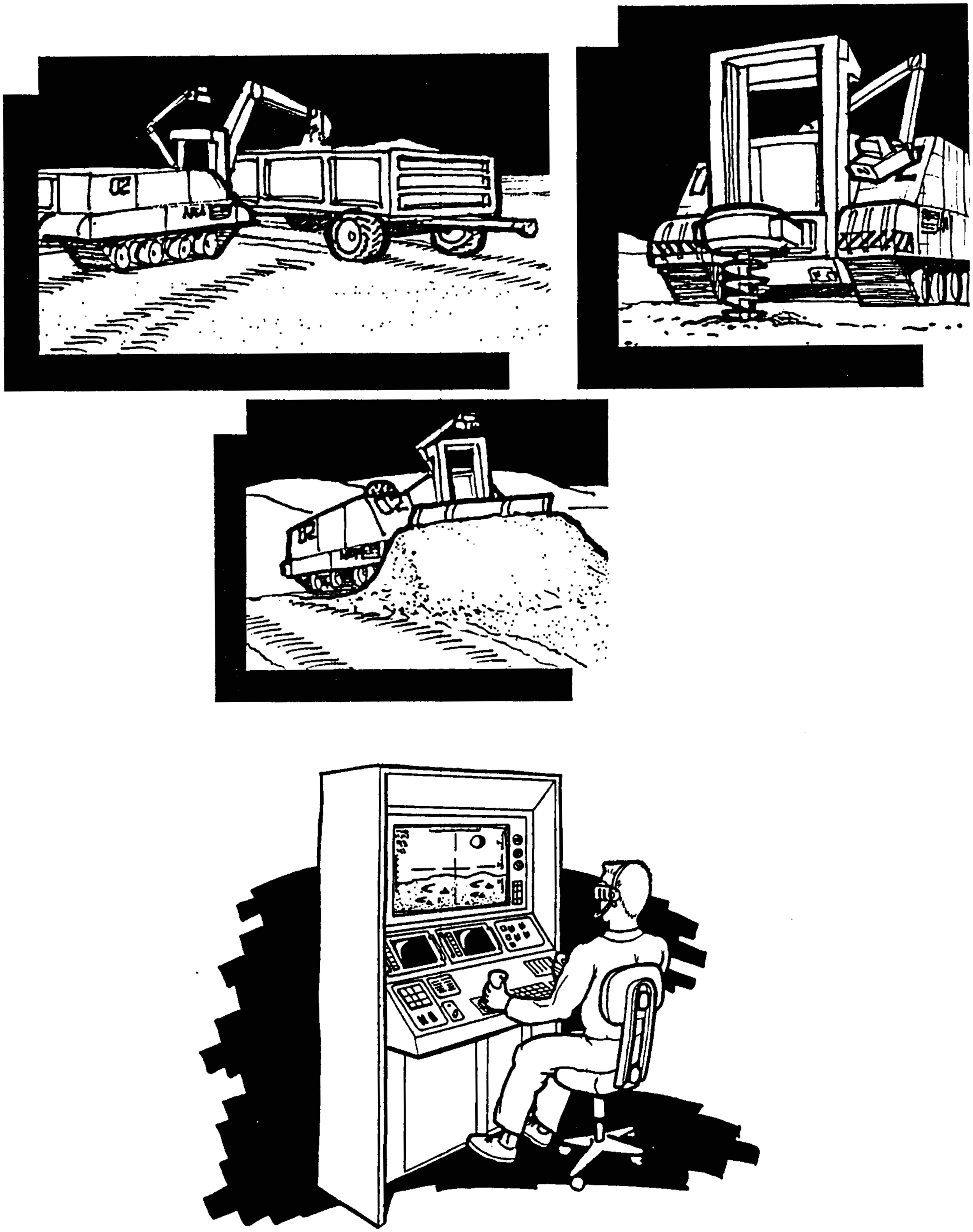
Groundrules defined for the study included: (1) lunar manned and unmanned cargo flight rates were assumed to build from 2 to a maximum of 8 per year in the human-tended base period, (2) initial surface operations used a crew module on top of a lunar lander for habitation and were therefore limited in surface stay times to the life support capability of the lander's crew module, presumed in this case to be 8 days for 4 crew, (3) the operations center shifted to the base and stay times were increased to 24 days after the following surface elements become operational: solar flare radiation shelter, habitation module, interface node, airlock, power system, thermal control system, and communications relay station.

Lunar base crew shift schedules were formulated from Shuttle guidelines and Space Station crew plans, and from them, time allocations for operational tasks were determined. For instance, of the 768 person-hours available on 4 crew, 8-day surface stay missions, only approximately 228 hours were actually available for surface operations after accounting for sleep, meals and personal time, spacecraft housekeeping and systems monitoring, arrival and departure spacecraft checks and preparation activities. Out of this 228 hr. surface operations resource, 6 two person EVA's were planned to provide 72 hrs. of EVA operations. IVA maintenance/refurbishment and ingress/egress activities required to support these EVA's consumed 49 hrs., yielding 107 hrs. for other IVA activities, such as teleoperation of base site surface preparation and construction equipment.

Specific surface operations addressed in this report included IVA support activities for EVA, landing/launch site preparation, cargo handling equipment and activities, radiation shelter emplacement, exposed (non-buried or covered) module emplacement construction equipment and operations, science operations, resource utilization operations, logistics and maintenance activities, manual/telerobotic division of labor, and contingency operations. For instance, the possible methods to provide 700 g/cm^2 of radiation protection (approximately 4 m of regolith overburden) for a solar flare shelter were surveyed, and the EVA/IVA time required for the baseline concept utilizing a bulkhead arrangement was determined.

A major conclusion of the study was that 4 person crews on approximately 1 month missions can accomplish significant science and resource development objectives while constructing a permanent base, but that teleoperation of soil moving and construction equipment from the lunar lander, lunar base, and Earth is required to leverage limited EVA time resources. Teleoperation is particularly important during short duration early missions for site preparation and solar flare shelter emplacement. Figure 2.0-1 illustrates this concept. Technology development in automation and robotics (A&R) applications to surface construction vehicles is considered essential, especially to allow lunar teleoperations from Earth with the imposed communications delay. It was also concluded, after estimating EVA/IVA time requirements for various surface activities, that providing radiation protection for all modules (by burying or covering with soil) should wait until the base is permanently occupied, when sufficient time resources are available. In addition, a concept for a lunar surface telerobotic servicer was proposed to perform inspection and maintenance activities.

Figure 2.0-1, Teleoperation on the Lunar Surface



3.0 Maintenance and Supply Options Study Summary

The purpose of this task was to define the maintenance and supply requirements for a lunar base. Trade studies performed concerned the size of the crew, the impact of lunar stay time intervals, and the options for packaging and shipping of spares and consumables. Design requirements for a logistics supply module were also produced. To accomplish the above, the CNDB Lunar Base Scenario (Ref. 1) was reviewed and a modified version used as a baseline. High level maintenance and supply functions were defined. Some of the design and operational approaches developed for the Space Station were either retained where commonality was desired or modified and applied to the Lunar Base.

The requirement to design for maintainability must play a much larger role in this program than has been the case in the past. A very complex system must be maintained at a great distance by a handful of people. This is without precedent in space work.

A phased approach was taken for both maintenance and logistics operations to support the Lunar Base. Maintenance and logistics operations were minimized during the early phase of the base build up and gradually increased as the equipment and crew complement allowed.

In the first period of 8 day surface stay times, the mode of operation is similar to that of the Apollo missions where the crew operated out of the lander and depended on subsystem redundancy instead of planned maintenance. Logistic activities during this period are primarily limited to delivery of items to the lunar surface and temporary storage of spares and consumables. The next period of 24 day surface stay time missions allowed operation from a pressurized habitat module which enabled lunar base crew interaction with onboard systems which tracked maintenance and logistic activities and hardware. A minimum level of maintenance activities was envisioned for this period with the availability of limited spares and maintenance capabilities. In the final period of 180 day surface stays and permanent occupancy of the lunar base, scheduled maintenance and routine logistics are incorporated into the daily activities.

The crew size trade led to a baseline crew size of 4 which permitted the minimum weight and volume impact to the transportation system but limited the amount of crew time available to perform lunar surface operations. Comparisons between crew skill and specialty mix to surface operational activities led to a preference for a larger crew size to provide the necessary specialty mix and crew hours available to perform the tasks envisioned to support lunar base buildup and operation. A crew size of 6 provided maximum operational flexibility (3 shifts) with a minimum of weight penalty.

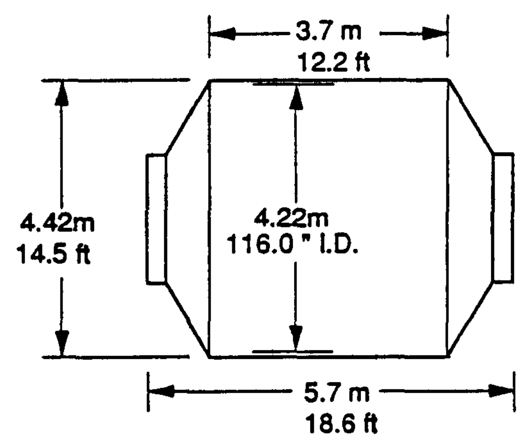
Investigations into extending the crew surface stay times showed that stay times for both short and medium duration missions could be increased with a small delta weight impact for consumables. These impacts amounted to 52 kg. (113 lbs.) to extend the surface stay time from 8 days to 11 days, and 307 kg. (675 lbs.) to extend the surface stay time from 24 days to 42 days for a crew of 4. In both cases, the impacts are small when compared to performing an additional mission to gain the operational crew time for lunar surface operations. Factors not considered in the stay time extension study were the effects of crew fatigue and limitations due to Earth/Moon orbital mechanics. Optimum launch windows to and from the Moon occur at intervals of approximately 9 days for low latitude bases. The issue of surface stay time should be reevaluated after a detailed study of launch windows and abort opportunities is performed.

The study on packaging and supply options indicated that for the earlier missions of 8 and 24 day surface stay times, the consumables could be carried with the manned missions. The baseline scenario did not include provisions for spares for equipment delivered to the lunar surface. Spares will make up a large fraction of the maintenance and supply mass delivered to the lunar base. Mass estimates indicate that dedicated cargo flights carrying only spares may be required. For the later manned missions where permanent Lunar Base occupancy in considered, logistic modules should be developed to deliver spares and consumables to the lunar surface. A conceptual design for the logistic supply module evolved to three modules, a pressurized supply module, a tank module, and pallet modules. These concepts are shown in Figure 3.0-1.

Additionally, a need for a facility to temporarily store spares and disposed reusable materials was identified.

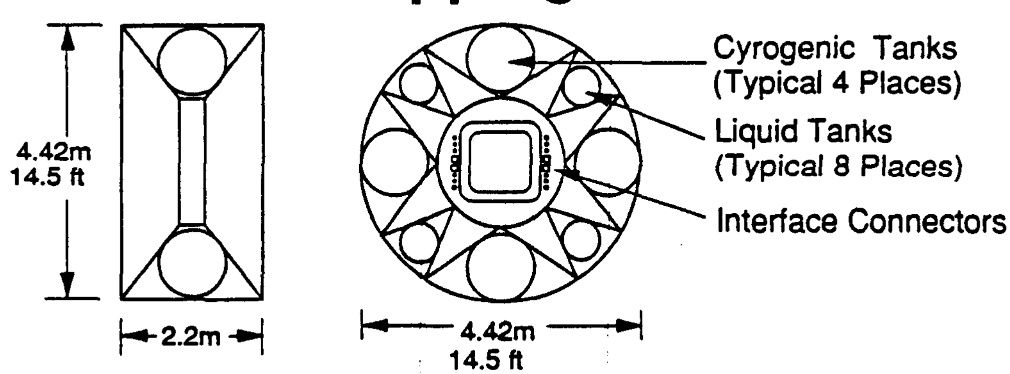
Figure 3.0, Logistics Modules

Logistics Supply Module



Module Length	Structure and Hardware Mass	Subsystems	Payload	Total Mass
5.7 m 18.6 ft	3,525 Kg 7,771 lbs	528 Kg 1,165 lbs	10,838 Kg 23,894 lbs	14,891 Kg 32,830 lbs

Fluid Shipping Module



Module Length	Struct. Mass	Subsystems	Cryo & Liquids	Total Mass
2.2 m 7.2 ft	305 Kg 673 lbs	1,220 Kg 2,690 lbs	3,051 Kg 6,726 lbs	4,576 Kg 10,089 lbs

4.0 Lunar Lander Conceptual Design Study Summary

This study prepared a conceptual design for a lunar lander to support a small lunar surface base. One lander, which can land 25 metric tons, one way, or take a 6 metric ton crew capsule up and down was desired. The initial idea was to build a reusable lander, suitable for minimizing the transportation cost to a permanent base, and use it from the first manned mission on, taking some penalty and perhaps expending expensive vehicles early in the program in order to avoid building multiple types of landers and focusing the effort on a space maintainable, single-stage, reusable vehicle.

A single stage lander is feasible from low lunar orbit. Initial calculations do not show large weight penalties (15-30%) over two-stage vehicles. A lander capable of multiple roles, such as landing cargo one way or taking crew modules round trip is possible with some penalty (5 to 10%) over dedicated designs. The size of payload (lander plus cargo) delivered from Earth to lunar orbit may vary by as much as a factor of two however, complicating orbital transfer vehicle (OTV) design.

A single type of engine usable for several different size landers appears to be possible. Different size landers and radically different payloads may require multiple trips with the OTV delivery vehicle(s) and storage of the first payload in lunar orbit, or a performance penalty due to additional tankage mass carried for small payload missions.

A four engine design for a multi-purpose vehicle, with total thrust in the range of 35-40,000 lbf (12 to 13,000 lbf per engine) and a throttling ratio in the 13:1 to 20:1 range is proposed. Initial work indicates a regeneratively cooled, pump-fed engine will be required due to difficulties with regenerative cooling over wide throttling ranges with pressure-fed systems. The engine is the single most important technical development item. Reuse and space maintainability requirements make it near or beyond the current state of the art. Study and simulation work should continue until this engine is defined well enough such that long lead development can start.

Initial calculations indicate low lunar orbit offers the lowest low Earth orbit (LEO) stack mass. Low altitude lunar orbits are unstable for long periods of time. The instability limit may set the parking orbit altitude.

LEO basing for the lander appears possible, with some penalty in LEO stack mass (10-25%) over a scheme that bases the lander in low lunar orbit (LLO) or expends it. The lander will require a special OTV to aerobrake it into LEO however. Loading all propellants from Earth on the lunar surface does not appear to be practical because of the additional propellant needed to land this propellant on the lunar surface. An additional lander mission is needed to land the propellant on the lunar surface. This may change when propellant can be produced on the lunar surface.

The lander must be designed from the start for ease of maintenance, and simplicity. Design features, such as special pressurized volumes will be needed to make the vehicle maintainable in space. Space maintainability and reusability must be made a priority.

Liquid oxygen/liquid hydrogen propellants show the best performance, but hydrogen may be difficult to store for long periods of time in the lander on the surface. Earth storable and space storable propellants are not ruled out. Liquid hydrogen storage over a 180 day period on the lunar surface at the equator needs study. A point design of a liquid oxygen/liq. hydrogen lander needs to be done in order to have a good inert mass data point that shows the performance gain is real.

Figure 4.0-1 shows this lander in lunar orbit, about to dock with a large (single stage) OTV. The OTV is designed to return the lander to the Space Station for servicing. The OTV delivers the lander to low lunar orbit, single stage, and waits in orbit for it to return. The OTV tanks are sized to hold 118 m tons of LO_2/LH_2 propellants.

Important features of this lander include:

1) Airlock/servicing tunnel down center of lander to allow easy access on surface and pressurized volume for LRUs. Many engine connections can be made and broken inside the pressurized volume.

2) Removable crew module. The lander is flyable without the crew module.

3) Lander fits in 30 foot heavy lift vehicle shroud with landing gear stowed.

4) Electro-mechanical shock absorbers on landing gear.

5) Emergency ascent with one or two crew possible without crew module. Crew would ride in suites in airlock/servicing tunnel.

Figure 5.0-3 shows this lander being serviced on the lunar surface and illustrates how the airlock/servicing tunnel allows pressurized access to a surface vehicle. An engine is being removed in the figure.

Figure 4.0-2 shows the lander on the surface at a lunar pole. The lander may also serve as a suborbital "hopper" if propellant loading on the surface is provided.

Table 4.0-1 is a weight statement for three reference missions a multi-purpose lander might be required to perform.

Figure 4.0-1, Single-Stage Reusable Lunar Lander and Single-Stage OTV in Lunar Orbit

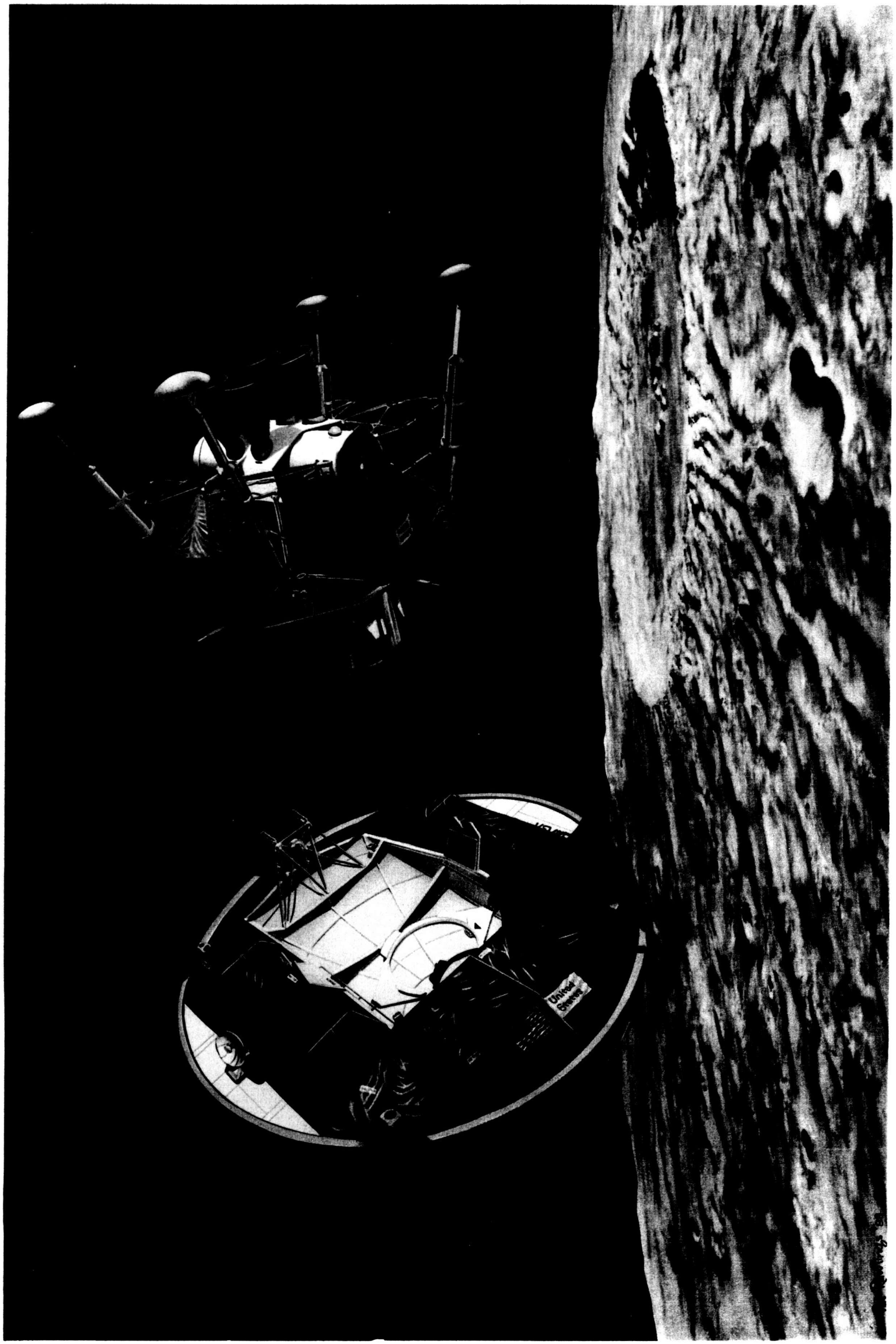


Figure 4.0-2, Single Stage Lander at Lunar Pole

Table 4.0-1, LO_2/LH_2 Multi-purpose Lander Weight Statement

All masses are kg, all Delta Vs, km/sec, Isp = 450 lbf - sec/lbm

Delta V, Ascent	0	*2.28	*2.28
Payload, Ascent	0	6,000 returned to LLO	0, Inert mass
Delta V, Descent	2.10	2.10	2.10
Payload, Descent	25,000	6,000	14,000
Total Inert Mass	9,823	9,823	9,823
Structure	1,681	1,681	1,681
Engines	822	822	822
RCS Dry	411	411	411
Landing Syst.	784	784	784
Thermal Prot.	2,017	2,017	2,017
Tanks	3,025	3,025	3,025
DMS (GN&C)	150	150	150
**Elect. Power	478	478	478
Airlock/Tunnel	455	455	455
Total Prop. Mass	25,251	32,395	30,638
Ascent Prop.	0	11,334	7,240
Descent Prop.	22,597	18,137	20,486
Unusable Prop. (3%)	678	884	832
Flight Perf. Res. Prop. (4%)	904	1,179	1,109
Usable RCS	858	689	778
Unusable RCS (5%)	43	34	39
Flight Perf. Res. (20%)	172	138	156
Deorbit or Gross Mass (less payload)	35,074	42,218	40,461
Deorbit or Gross	60,074	48,218	54,461

* Delta V = 1.85 + .43 km/sec for a 15° plane change in a 93 km circular orbit.

** Electrical power provided for 3 days only, (2kw). 100% redundant fuel cells have dedicated redundant tankage.

5.0 Lunar Base Launch and Landing Facility Conceptual Design Summary

The purpose of this study was to examine the requirements for launch and landing facilities for early lunar bases and to prepare conceptual designs for some of these facilities. The emphasis of this study is on the facilities needed from the first manned landing until permanent occupancy. Surface characteristics and flight vehicle interactions are described, and various facility operations are related. Specific recommendations for equipment, facilities, and evolutionary planning are made, and effects of different aspects of lunar development scenarios on facilities and operations are detailed. Finally, for a given scenario, a specific conceptual design is developed and presented.

Launch and landing facilities and their growth rate depend on the base development scenario. The major emphasis of the base, the rate of emplacement of facilities, and the design of the flight vehicle will all play major roles in the requirements for facilities. Resource utilization bases will require more and different landing facilities than will science or habitation bases. The more rapidly some base capabilities are achieved, the more rapidly landing facility capabilities are required. Vehicles that require extensive surface-based servicing will require leveled permanent landing areas. These permanent reusable landing pads are not needed or desired before major resource export of vehicle servicing activities take place. For some lunar base scenarios, permanent landing pads may never be needed.

Based on the calculations done during this study, the effects of engine blast are significant. While they are not critical or life threatening, they must be considered. Equipment within 50 meters of a landing may experience severe damage due to the impact of fairly large grains of lunar soil. Equipment over 400 meters away will require only minimal protection. At 1 to 2 kilometers blast effects are very small. Figure 5.0-1 shows representatives particle trajectories. Figure 5.0-2 shows the estimated number of impacts as a function of distance.

Landing pads can be designed without general regard to the specific landing site because overall surface conditions are fairly uniform across the entire lunar surface. Landing pads, whether prepared or not, should be about 100 meters across. The area just outside this circle to 200 meters across should not include any major obstructions such as boulders or expended landers. Lunar derived gravel may be used to stabilize prepared landing pads.

With few exceptions, lunar landing facilities and equipment are present on the lunar surface for other reasons before they are needed for landing operations. Landing equipment and facilities will probably not be major drivers of delivery schedules and missions plans.

More work is needed concerning blast effects, vehicle servicing on the surface, site planning and development, and safety and rescue operations. More design definition is needed for surface stabilization methods, cryogen storage and transfer facilities, servicing and maintenance equipment, and other items.

The launch and landing facilities of a permanently occupied base need to be defined. This study was limited to the initial lunar base, and the facilities needed for extensive permanently occupied or Phase III bases have only been reviewed in a cursory fashion.

Figure 5.0-1, Lofted Particle Trajectories

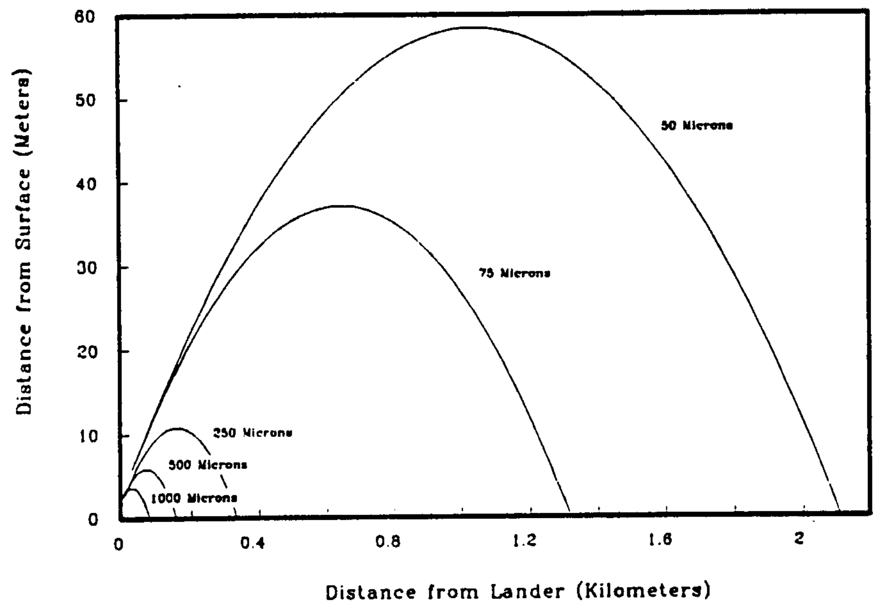


Figure 5.0-2, Impacts as a Function of Distance

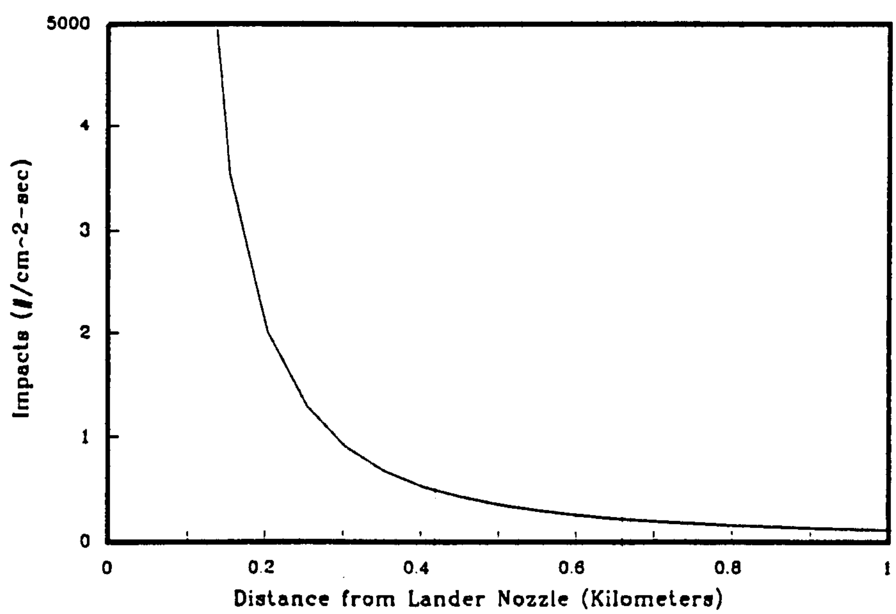


Figure 5.0-3 is an illustration of some of the landing facilities as they might appear near the end of the Phase II Lunar Base. The landing has taken place just after the lunar dawn. Earth appears over the Rook Mountains in the east across the bed of the Lake of Spring (Lacus Veris). Throughout the next month it will dip below the horizon, only to appear again after a few days. Above Earth, the Sun moves slowly higher in the lunar sky. The lander sits in the middle of the 100-meter diameter gravel landing pad where it is being readied for its stay on the lunar surface. Inside, crewmembers are shutting down the flight systems and configuring the lander for its layover.

The pressurized vehicle in the foreground is connected to the lander, waiting to take the crew back to the lunar base. The transfer tunnel has been connected between the lander and the vehicle to allow the crew easy access in and out.

Beneath the lander an astronaut begins the process of changing an engine by removing and relocating an old engine with a mounting rig. Changing modular line replaceable units is the first form of flight vehicle servicing to take place at the lunar base.

To the right of the lander, a Propellant Refill Vehicle is being used to scavenge hydrogen remaining in the tanks. This might be done as quickly as possible before the Sun heats the tanks and boils away the fuel. Still further, a supplemental cooling cart has been connected to the lander's thermal control system. The radiator on this cart will help keep the lander and its systems cool during the lunar day.

A crane removes a small canister containing the personal items of the arriving crew along with some small experiments and supplies.

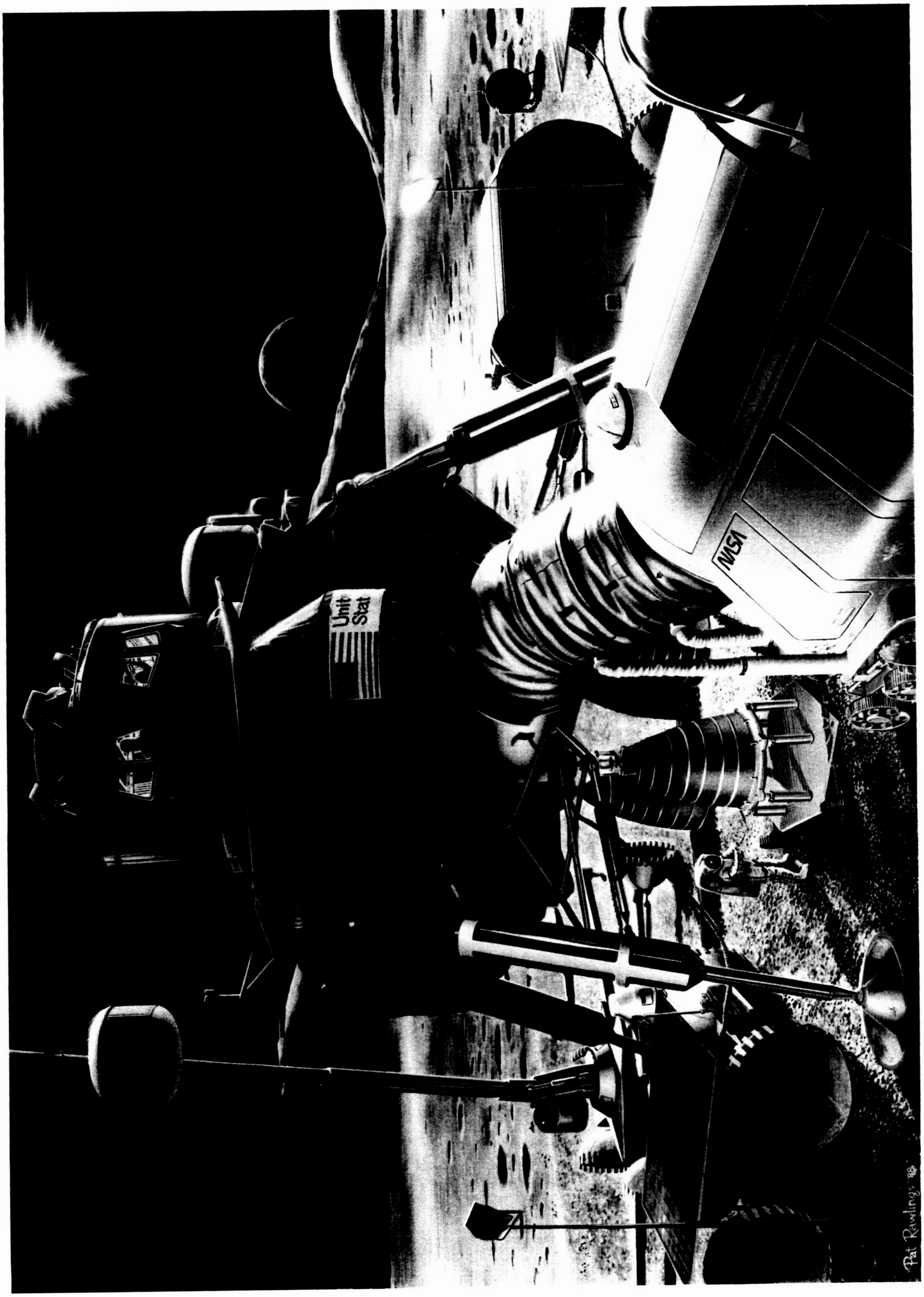
To provide electrical power to the lander while it is on the pad, a power cart has been moved between the lander legs on the left. Using the fuel cells and the solar panel on top, the power cart will support the lander for the next month.

Draped around the back legs of the lander, a thermal and meteoroid blanket is ready to be lifted over the lander. When all other preparations have been finished the blanket will be pulled over the lander to protect it from the bombardment of micrometeoroids. It will shield the lander from the Sun during the lunar day and help keep it warm during the night.

At the edge of the pad, an astronaut inspects one of the three landing pad markers for damage. Ejecta from repeated landings may have damaged the reflector, the light, or the radar transponder mounted on the marker. The transponders are vital in guiding the lander to an accurate landing. The reflector appears translucent in the intense sunlight. The mesh surface decreases the pressure effects of the engine exhaust as the lander passes over.

The landing pad is surfaced with packed gravel which is a bi-product of the mining operations taking place on the lunar surface. Roads and other smooth level surfaces can also be surfaced with this gravel.

Figure 5.0-3, Lunar Lander on Surface



6.0 Lunar Oxygen Pilot Plant Conceptual Design Summary

The primary objective of this study was to develop conceptual designs of two pilot plants to produce oxygen from lunar materials. A lunar pilot plant will be used to generate engineering data necessary to support an optimum design of a large scale production plant. Lunar oxygen would be of primary value as spacecraft propellant oxidizer. In addition, lunar oxygen would be useful for servicing non-regenerative fuel cell power systems, providing requirements for life support, and to makeup oxygen losses from leakage and airlock cycling.

Numerous processes to produce oxygen from lunar materials have been proposed. Thirteen different lunar oxygen production methods are described in this report. Comparisons are complicated because many variations of each process exist, and some produce multiple byproducts with potential uses at a later stage of lunar base development. Based on process simplicity and well understood reaction chemistry, hydrogen reduction of ilmenite was selected for conceptual design studies. Based on recovery of an important "byproduct", a second process pathway to oxygen, extraction of solar-wind hydrogen from bulk lunar soil, was also selected for conceptual design. Thermal recovery of solar-wind hydrogen liberates water, which is subsequently electrolyzed to produce oxygen (water is a reaction product of hydrogen and ilmenite contained in the soil), as well as hydrogen. Thus, hydrogen recovery offers a process that produces both oxidizer and fuel propellants for lunar landers and other spacecraft.

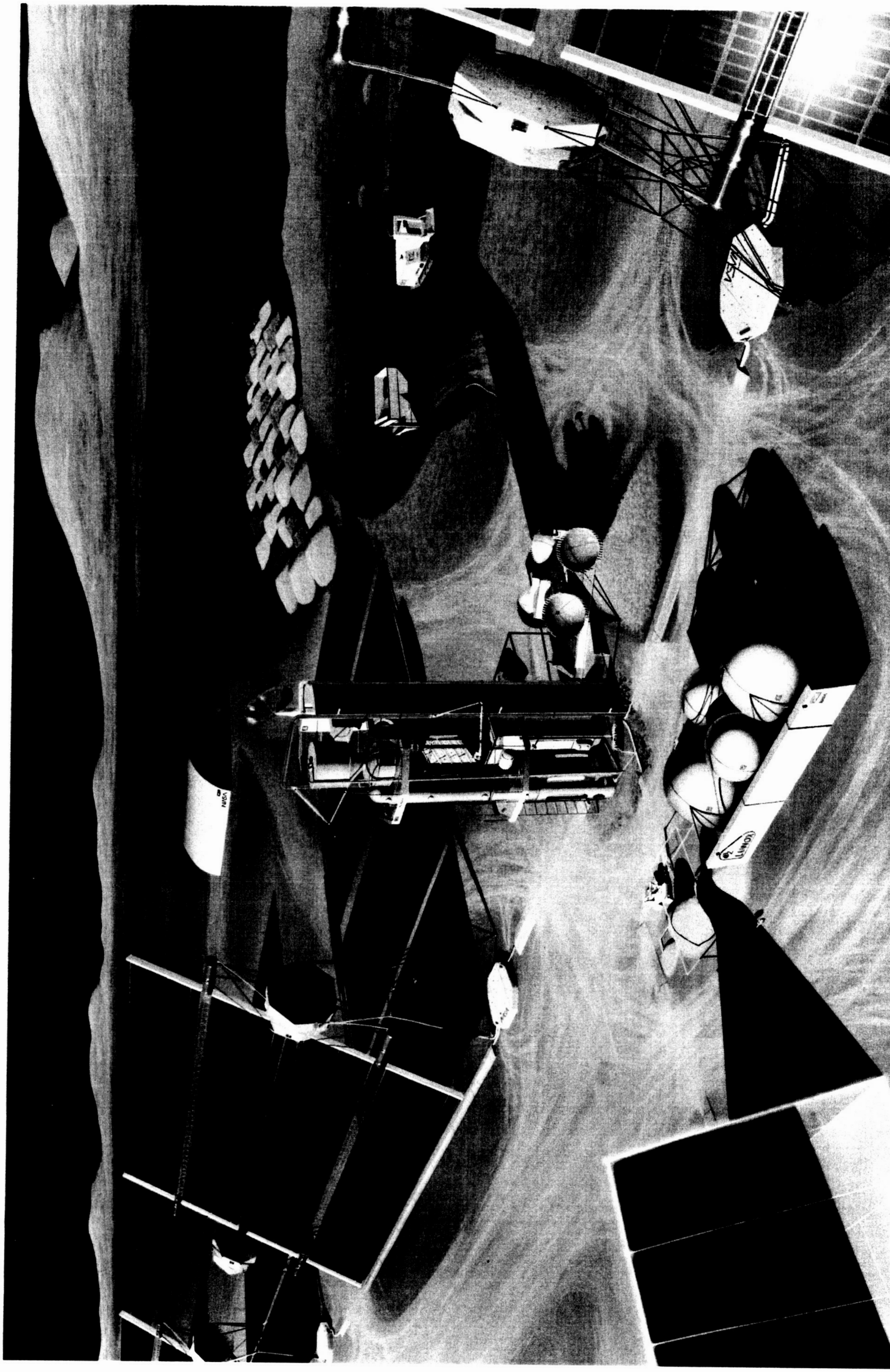
Computer models of both processes were prepared that utilize equipment scaling relations, mass and energy balances, and thermodynamic relationships to estimate mass and power requirements for oxygen production plants. Trades and sensitivity analyses were performed with these models. Studies on the hydrogen reduction of ilmenite process included:

- Evaluation of feedstock alternatives: high-titanium mare soil or basalt
- Effect of solar and nuclear-electric power sources.
- Effect on pilot plant mass/power to simply vent the product oxygen gas instead of liquefying and storing it (since the pilot plant is a research tool).
- Comparison between delivering a series of small self-contained, modular production plants to increase oxygen production versus constructing a single, large plant.
- Difference between using unbeneficiated feedstock or using magnetic or electrostatic separation to feed an ilmenite concentrate to the reactor.
- Sensitivity of process mass and power to oxygen production rate.
- Sensitivity of process mass and power to feedstock conditions such as ilmenite abundance in soil or ilmenite grain size in basalt.

Figure 6.0-1 shows a 2 metric ton/month LOX pilot plant conceptual design, employing hydrogen reduction of ilmenite. Figure 6.0-2 defines the pieces in Figure 6.0-1. Plant mass is 24.7 metric tons (54,400 lb_m) including a power system that uses solar photovoltaic arrays to provide 146 kwe for the process and for regenerating fuel cell reactants. Baseline plant operating strategy is mining and continuous processing during the lunar day, and no mining with processing units on hot standby during the lunar night. Figure 6.0-3 shows a schematic of the process. The major process equipment is delivered to the lunar surface in an integrated package that manifests easily into a Shuttle payload pallet with outside dimensions of 14' diameter x 45' long. However, additional volume is required to deliver the power systems. Since it is assumed that the purpose of the pilot plant is to provide long-term, 1/6-g equipment performance data, the plant will be operated for continuous periods without on-site human attention. Thus, extensive automation and robotics applications are anticipated for the pilot plant, such as teleoperated mining vehicles and equipment servicers. These would have numerous applications in other areas of lunar base operations.

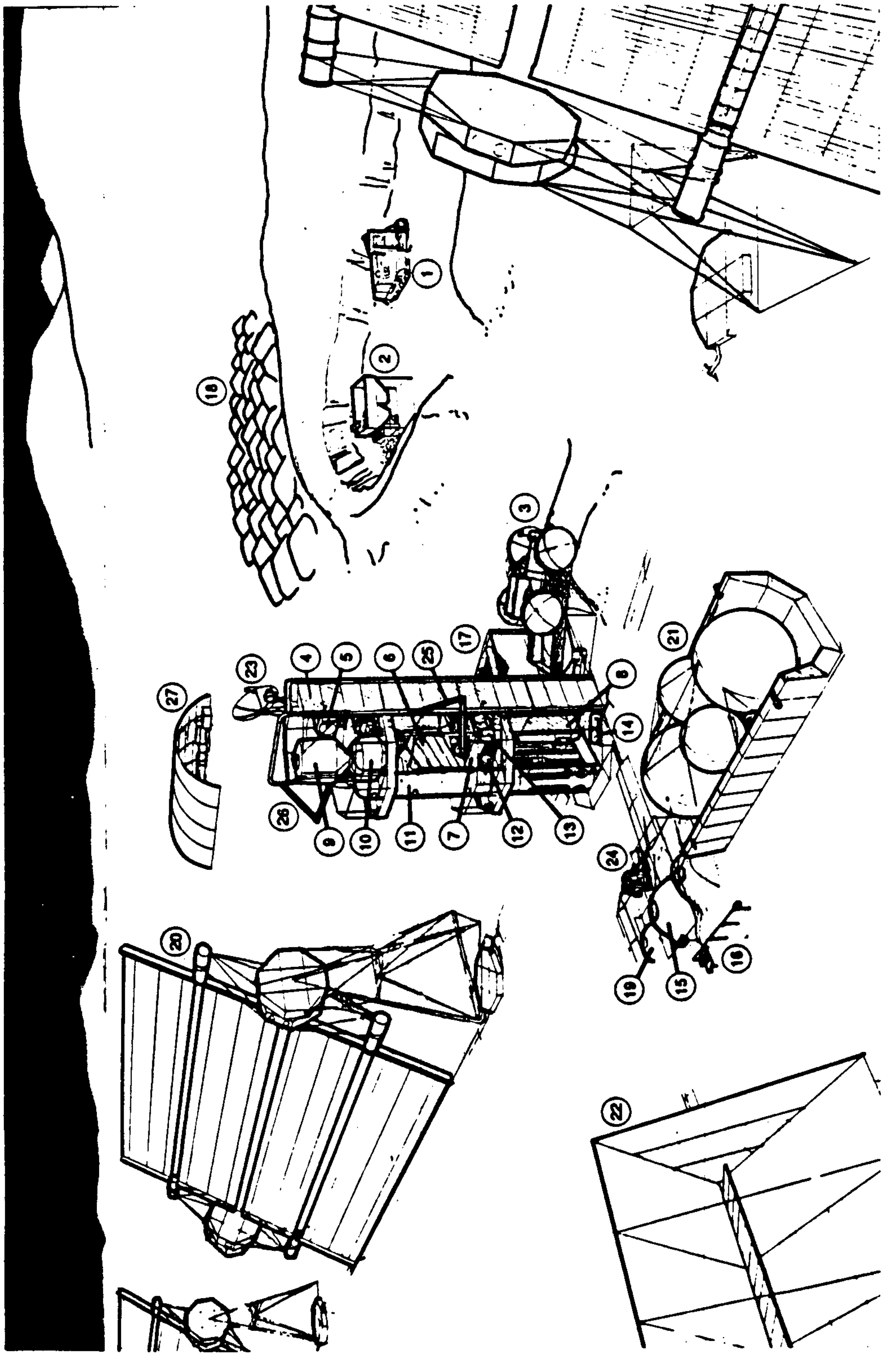
Studies of the optimum temperature for solar-wind hydrogen extraction and the sensitivity of plant mass/power to production rates were also completed. Mass of a pilot plant designed to produce 2 metric ton/month LOX and 1.2 metric ton/month LH_2 is 60 metric tons (132,200 lb_m). The mass estimate includes a nuclear power plant providing 1.7 MWe for the process

Figure 6.0-1, Ilmenite Reduction Process Lunar Oxygen Pilot Plant



ORIGINAL PAGE
COLOR PHOTOGRAPH

Figure 6.0-2, Explanation of Oxygen Pilot Plant

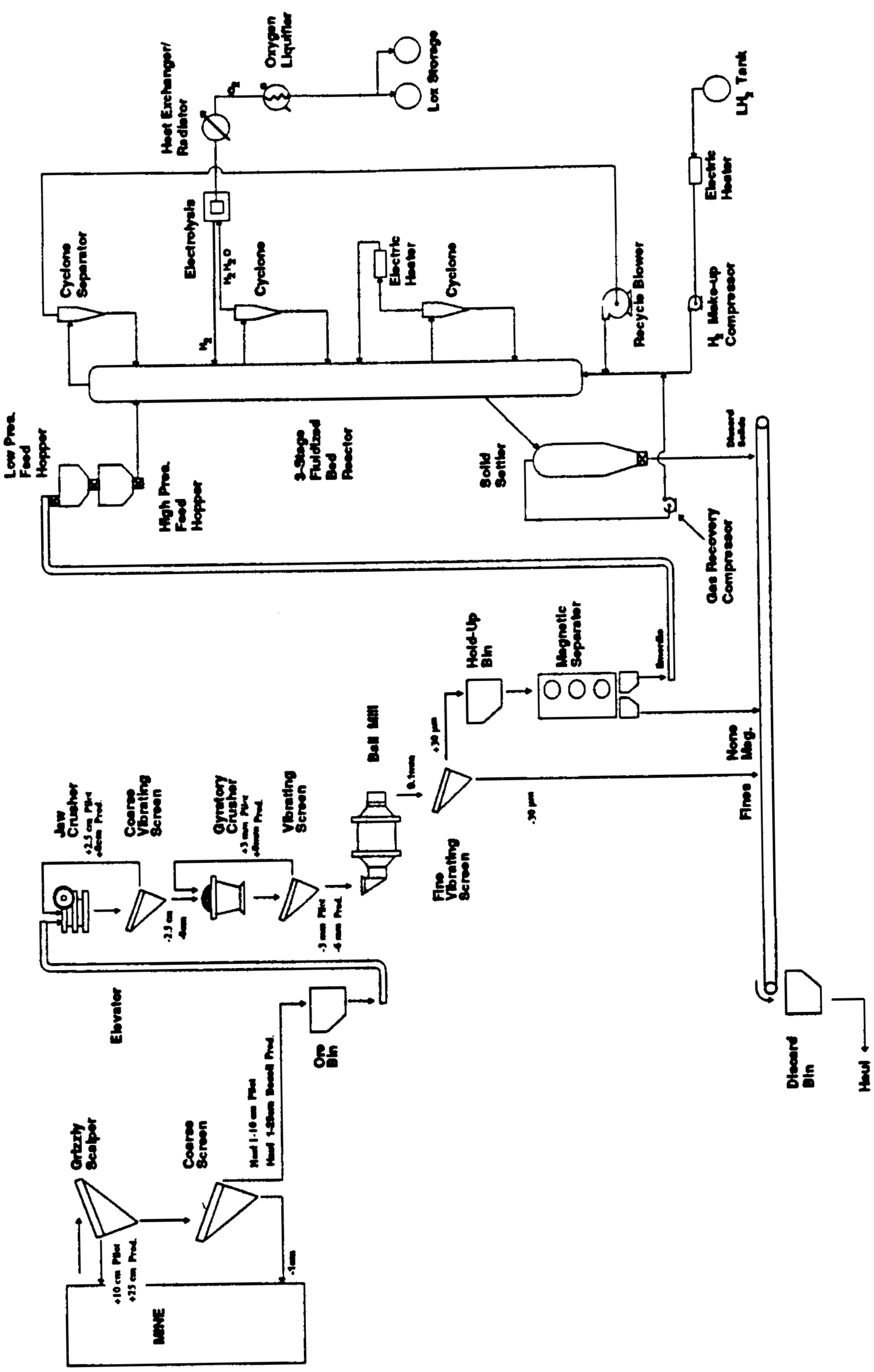


Location: Lacus Veris (87.5°W, 13°S)
View Facing North-East

1. Excavator (Front-End Loader)
2. Pit Scalper
3. Hauler
4. Support Structure (Payload Bay Pallet)
5. 3-Stage Crushing/Grinding Circuit
6. Vibratory Screen (Fines Removal)
7. Hold-up Bin
8. High-Intensity Magnetic Separator
9. Low-Pressure Reactor Feed Hopper
10. High-Pressure Reactor Feed Hopper
11. 3-Stage Fluidized Bed Reactor
12. Electric Gas Heater
13. Solid-State Electrolysis Cell
14. Oxygen Liquefier
15. Buried Oxygen Storage Tanks
16. Liquid Oxygen Loading Station
17. Tails Discharge Bin
18. Tailings Piles
19. Makeup Hydrogen Storage Tank
20. Photovoltaic Power System (Sun-Tracking)
21. Regenerative Fuel-Cell Gaseous-Reactant Storage Tanks
22. Radiator with Fixed Sun-Screen
23. Communications: High- and Low-Gain Antennas
24. Telerobotic Servicer on Lunar Surface Mobile Platform
25. Telerobotic Servicer on Remote Manipulator Arm
26. Spare Remote Manipulator Arm
27. Equipment Repair and Spares Storage Shed

ORIGINAL PAGE IS
OF POOR QUALITY

Figure 6.0-3, Schematic of Proposed Ilmenite Reduction Process



7.0 Lunar Storm Shelter Conceptual Design Summary

Extended occupancy on the lunar surface will require redefinition of the allowed radiation exposure of crewmen performing lunar missions. It is proposed that the radiation dose be divided into three parts as follows:

1) Extra-Vehicular Activity (EVA) Exposure during quiescent solar periods (no solar flare activity) While performing EVA type operations and while in transit to and from the lunar surface, or other unprotected conditions. A period of continuous low level of known radiation exposure will occur. It is proposed that the dose limit for these exposure be set at **5 REM**.

2) Emergency exposure While on an extended EVA mission, or during other unprotected conditions, in the event a solar flare occurs and the crewman cannot return to the main solar base where more complete radiation protection is provided, a short period of high level exposure will occur. For this emergency exposure a dose limit of **20 REM** is proposed, delivered in a period of 24 hours or less.

3) Exposure within the permanent lunar shelter While within a well shielded habitat, radiation exposure corresponds to the natural radiation background on Earth. It is proposed that this limit level be set at **0 REM**.

Under the worst possible conditions the total dose received by any crewman is limited to the sum of the quiescent and emergency doses or 25 REM. To accomplish this level of dose control, quiescent EVA exposure must be limited to 5 REM by measurement and control of individual exposures. Sufficient shielding must be provided for the case where a solar flare is encountered while on an EVA operation to limit the dose in that period to 20 REM, and the main lunar shelter must be shielded to a level that produces an Earth equivalent background radiation level during all periods that a crewman is not performing an EVA operation. Note that the main shelter is not merely a storm shelter, but that it also eliminates the quiescent radiation dose in order to maximize the allowable dose received during EVA operations.

In this paper no attempt is made to correlate with any specific lunar program or mission. Instead, some of the options that should cover the range of possible missions are considered. The lunar missions could have durations of a week, a month, or six months and periods of occupancy up to years. Solar flare protection is the primary consideration for the shorter missions up to a month. For the longer missions, the requirement to reduce the constant galactic cosmic ray dose is the primary radiation protection consideration resulting in a heavily shielded habitat. The un-shielded galactic cosmic ray dose is on the order of 20 to 50 REM/year. Exploration of the lunar surface, and the establishment of remote scientific stations adds additional complications to the radiation problem. Several options to cover the range of missions have evolved as follows:

Buried Lunar Base This base provides a radiation environment equivalent to the background radiation encountered on Earth, and is required for missions of six months or more. A four man base is estimated to require 4,000 cu.ft. interior volume. The resupply time is taken as 180 days. The minimum shielding requirement is 785 grams/cm^2, which provides a dose from galactic cosmic rays similar to that on Earth for people living at an altitude of 9000 ft. above sea level. The thickness of the shielding requires that the density of the lunar material as placed upon the shelter be known. Because of tamping problems on the lunar surface the density might be as low as 1 gm/cm^3. This density would require a shield thickness of 7.85 meters. If the upper estimated density of lunar material, 3 gm/cm^3 is used, the shield thickness is 2.62 meters. Actual thickness will be determined by on site measurements, while burying is underway. The construction of a buried shelter assumes that construction equipment is on the lunar surface, and should require a number of flights to implement. Thus it is not a candidate for early lunar missions.

Earth Fabricated Solar Flare Storm Shelter This type of shelter is considered applicable for missions of up to about 30 days duration. It is considered capable of supporting four men for a period of up to 10 days, while a solar flare is in progress. Because the total exposure time to Galactic Cosmic Rays is for not more than 30 days, the total dose from these rays will be less than 5 REM under the most pessimistic assumptions. The storm shelter needs only to protect against a worst case solar flare by reducing the dose to 20 REM. Such a shelter would require a shield thickness of 59 gm/cm^2 of aluminum or a wall of thickness of 8.70" (22 cm). The mass of such a shelter is estimated at 14.7 tons. No provision in this estimate has been made other than interface connections for power, air, communications and control. A small, self-contained waste disposal device is needed. The capability to deliver this shelter to the Moon and to offload, level and connect to the life support, power, and other systems was not addressed.

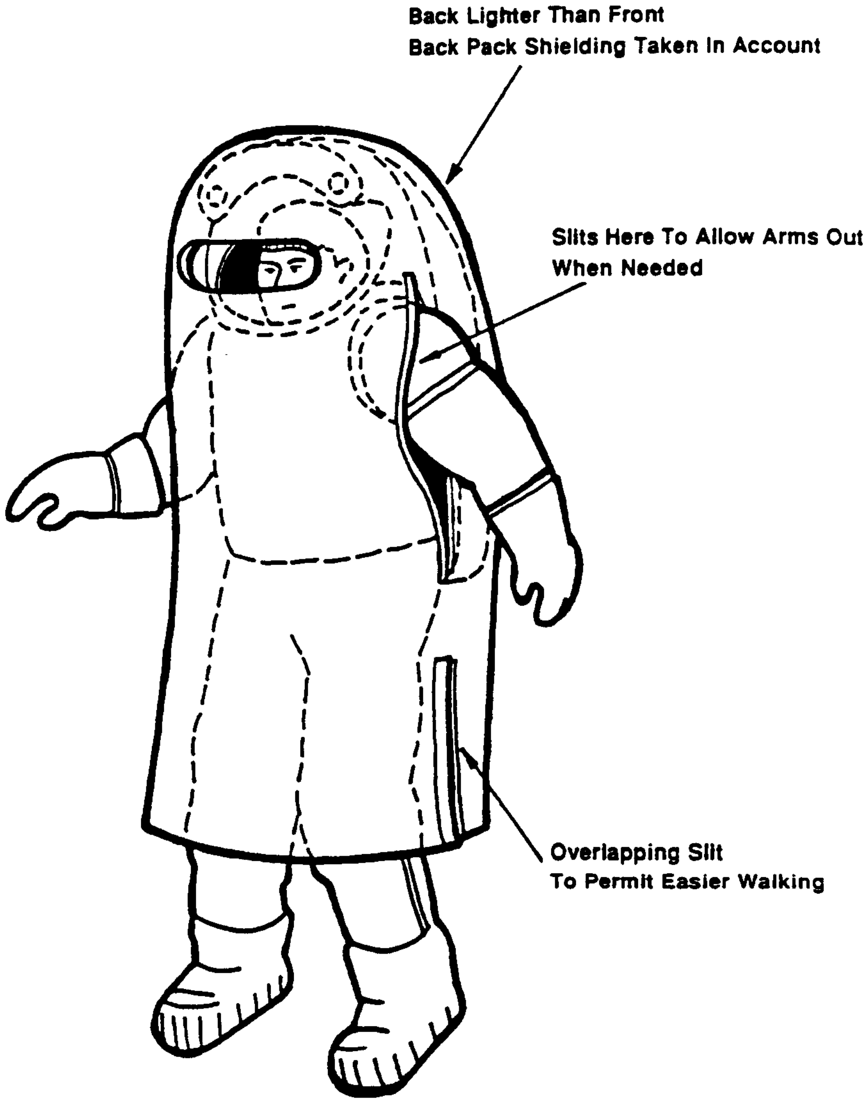
An alternate to the thick walled, 4 man storm shelter would be to deliver a thin walled shelter and a small earth moving device. Assuming that the average wall thickness is 3/16", the weight of the storm cellar module delivered from Earth should be ~500-600 lbs (225-275 kg) including only the aluminum shell. For solar flare protection only, covering with lunar soil could be accomplished with a small teleoperated earth mover. Assuming loosely packed lunar soil with a density of 1 gm/cm^2 , a soil cover of about 2 feet (61 cm) would be required. This would require moving 815 to 850 ft^3 of soil depending on burial depth.

Should neither of the above solutions prove feasible, then the mission should be planned for periods of low solar activity. The available solar flare data indicate that no major and very few small flares (which would not impact dose limits) are encountered when the sunspot number is less than about 35. The sunspot activity is below this level for about 4 years as one cycle ends and the next cycle begins.

Lightly Shielded Vehicles on the Lunar Surface These vehicles consist of a pressurized flyer or lunar rover, which are operated under shirt sleeve conditions. In the event of a solar flare, the time needed to return to either of the previously discussed shelters may equal the time to deliver in excess of 90% of the total dose from a solar flare (i.e. 6 to 8 hours). These vehicles are assumed to carry a crew of one or two. The mass of shielding to produce a dose of not more than 20 REM for a two man arrangement is ~6 tons of aluminum. The mass is noted as a guide for vehicle design. Since incorporating this amount of mass may not be feasible an inflatable structure which can be buried by a backhoe blade on the surface vehicle should be investigated. The required burial depth is on the order of 2 feet as discussed above.

Partial Protection Garment For operations performed in the spacesuit and also in an unpressurized lunar rover, the return time to a safe shelter is estimated not to exceed 3 hours. Vital repairs may require exposure to high radiation fields. For these conditions a concept for a partial protection garment is described. This garment weighs 375 lbs. (170 kgms). On the lunar surface it is equivalent to carrying 63 lbs (29 kgms) on Earth. It is capable of reducing the radiation level from 5 to 7 times. A trade study is needed between mobility and weight, and detailed work may eliminate unneeded shielding around the back pack area. Figure 7.0-1 illustrates the partial protection garment.

Figure 7.0-1, Partial Protection Garment



8.0 Space Transportation Nodes Assumptions and Requirements

This study was an effort to document upper level Space Transportation Node (STN) assumptions and requirements. The STN is an LEO space station supporting the transportation system for an early lunar base.

Assumptions are one category of design guidance provided to an engineering design team. They are those guidelines that are conceived through supposition and legislated by policy because insufficient information or time is available for explicit verification. Requirements are the other design guidance category. They are derived by analysis of the functional task of interest or known by prior experience. The assumptions and requirements of interest to this task are the upper level specifications which bound the architectural concepts and state the functional performance demands on the systems. While numerous conceptual configurations for space transportation nodes can be found in the literature, the intent of this task was to develop and document STN assumptions and requirements without any preconceived model of a design configuration. A later, related study (see section 9.0) was tasked to develop an STN design concept which satisfied the upper level assumptions and requirements of this study task.

The task activities were planned to produce results which were relatable to space station development, responsive to the synthesized models for the initial years of the lunar base, and organized to accommodate continued development.

In the Space Station Program, the requirements documentation tree begins with JSC 30000, JSC 31000 and the Architectural Control Documents (ACD's). Therefore, the STN requirements documentation is patterned after the JSC 31000, Space Station Projects Requirements Document. Documents with detail such as the Space Station Interface Control Documents and Contract End Item Specifications are not appropriate at this phase of program planning.

The assumptions and requirements are obtained from discussions with appropriate personnel and by analysis of a space transportation reference baseline. Essentially, the requirements analyst is performing the earliest stage of system engineering design. The task is to determine, thinking as a designer, what data must be known to perform specific engineering designs at this level of detail. The assumptions and requirements are identified in the thought process of considering what activities the STN must perform for each particular mission and vehicle passing through the node.

The Civil Needs Data Base Option 3, Phase 2 initial years was used for the lunar missions scenario. Due to the fluid nature of a space program definition at this early planning stage and the probable change in detail data, the requirements were formulated based on generalized missions and flight schedules synthesized from three representative years of this lunar missions scenario.

The documentation of results is organized into the three sections of Source References, Assumptions and Groundrules, and STN Requirements. Data base methods were chosen as the medium for recording the results. The use of data bases allows the identification of links between references, assumptions, and requirements. The data bases also enable flexibility and ease in reviewing and analyzing the results. Table 8.0-1 shows an example section of the requirements database.

Table **8.0-1**, LEO Space Transportation Node List of Requirements

WBS No.: 1.01 Requirement ID: 1
STN Element: Mgt/Integr Assumptions: Ref:

The General Requirements in Section 2.1 of JSC 31000 also apply to the LEO STN; except for the induced environment restrictions due to user accommodation.

Rationale: The JSC 31000 general Space Station requirements appear to be appropriate for the STN and represent more planning and analysis effort than is available for derivation of similar requirements in this study. The limitations of induced environment due to accommodating applications users are not appropriate for the STN and are one reason for the need of an STN separate from the Space Station.

WBS No.: 1.01 Requirement ID: 2
STN Element: Mgt/Integr Assumptions: Ref: 3

The LEO STN orbit parameters must enable efficient payload delivery from Earth, allow transfer to lunar trajectories, insure no collision with the space station, and minimize space station viewing interference.

Rationale: By definition, the transportation node should be located in the optimum position in the transportation path. However, the transportation facility must not interfere with the important objectives of the Earth orbit base, the Space Station.

WBS No.: 1.01 Requirement ID: 3
STN Element: Mgt/Integr Assumptions: 3.05,3.06 Ref: 3

The STN shall have the capability to accommodate one docked space shuttle while supporting a lunar flight departure or arrival.

Rationale: The lunar crews do not arrive until the lunar flight vehicle is substantially ready for departure and the Shuttle must remain until the lunar flight has departed so the lunar crew could be returned to Earth in the event of a failure to launch.

WBS No.: 1.01 Requirement ID: 4
STN Element: Mgt/Integr Assumptions: 1.01b Ref: 3

The LEO STN orbital orientation is to be optimized for spacecraft systems design. There are no mission pointing or orientation requirements.

Rationale: The STN is not subject to design compromises related to diverse earth orbit applications interests. The STN orientation is to be designed to facilitate the best possible support to transportation activities transitioning from Earth orbit to translunar trajectories.

9.0 Transportation Node Space Station Conceptual Design Summary

A low Earth orbit space station was conceptually designed to support a reusable transportation system for lunar flights. Figure 9.0-1 illustrates the overall concept showing a departing stack and arriving heavy lift tanker. Figure 9.0-2 shows a three view. This Space Transportation Node (STN) station is oriented exclusively toward the assembly, refurbishment, maintenance, propellant loading, checkout, and repeated reuse and launch of cargo and piloted vehicles going to the lunar surface.

Up to eight flights per year to the lunar surface are to be supported. The transportation system consists of a large single-stage reusable OTV that delivers a single-stage reusable lander/launcher to low lunar orbit (LLO). The OTV waits in orbit for the lander to return. Both then aerobrake back to the low Earth orbit (LEO) station using separate aerobrakes. Both vehicles are reloaded with propellant and refurbished at the LEO station. Though a specific transportation system is used, a range of different transportation system (vehicle) options can be accommodated. The emphasis however, is on reusability for space-maintainable vehicles.

The station supports two stacks, each consisting of an OTV, lunar lander/launcher, and a payload. The single stage reusable lander/launcher delivers 25 m tons one way to the lunar surface or a 6 m ton crew capsule round trip from low lunar orbit (LLO). A stack departing LEO weighs on the order of 200 m tons, including 158 m tons of cryogenic propellant.

The dry weight of the station, without propellants or OTVs and landers is approximately 400 metric tons. 182 m tons of cryogenic hydrogen and oxygen propellant is stored in four tanks. The storage uses liquid acquisition devices to acquire the propellant for transfer to the OTVs and landers. Passive thermal control is used and boil-off is used for orbital make-up propellant. With two stacks fully loaded with propellant, and the storage tanks full, the station has a maximum weight of approximately 1,000 m tons. Table 9.0-1 shows a summary weight statement.

75 kilowatts of continuous power is provided by a Phase 1 Space Station photo-voltaic system. 75 kw of heat is rejected via a Space Station thermal control system. The station subsystems are in general taken directly from or derived from the Freedom Space Station design.

The two stacks can be assembled or serviced in parallel. Each stack is docked to a rotating fixture that turns to allow 360° access to the entire stack from a manipulator running up and down the truss. The rotating fixture also allows pressurized access to the lunar crew module or cargo from the STN interior.

The two major assumptions of the design are; 1) the fully reusable, space-maintainable OTV and lander, and 2) the high maximum flight rate (8/year). These assumptions require careful examination in future work.

The advantages and disadvantages of a low lunar orbit station were also examined as part of this effort. A LLO STN in lunar equatorial orbit would allow the OTV and landers to always deliver maximum payload which may be required in some lunar oxygen schemes to achieve reasonable mass efficiency. These scenarios generally assume a lunar based and maintained reusable lander/launcher however and are probably not practical until well after a permanent lunar base is established. As the inclination of the lunar orbit goes up, the number of opportunities to arrive and depart the Moon without excessive delta-V penalties goes down. For these higher inclination lunar orbits an LLO STN adds another constraint that further complicates the window problem. Delta V plots were generated that indicated inclinations of 10° and less can be made essentially equal to equatorial in their accessibility for a 15% penalty in LEO stack mass.

Figure 9.0-1, LEO Transportation Node Space Station for Lunar Base Support

ORIGINAL PAGE
COLOR PHOTOGRAPH

Figure 9.0-2, LEO Transportation Node Space Station for Lunar Base Support (three view)

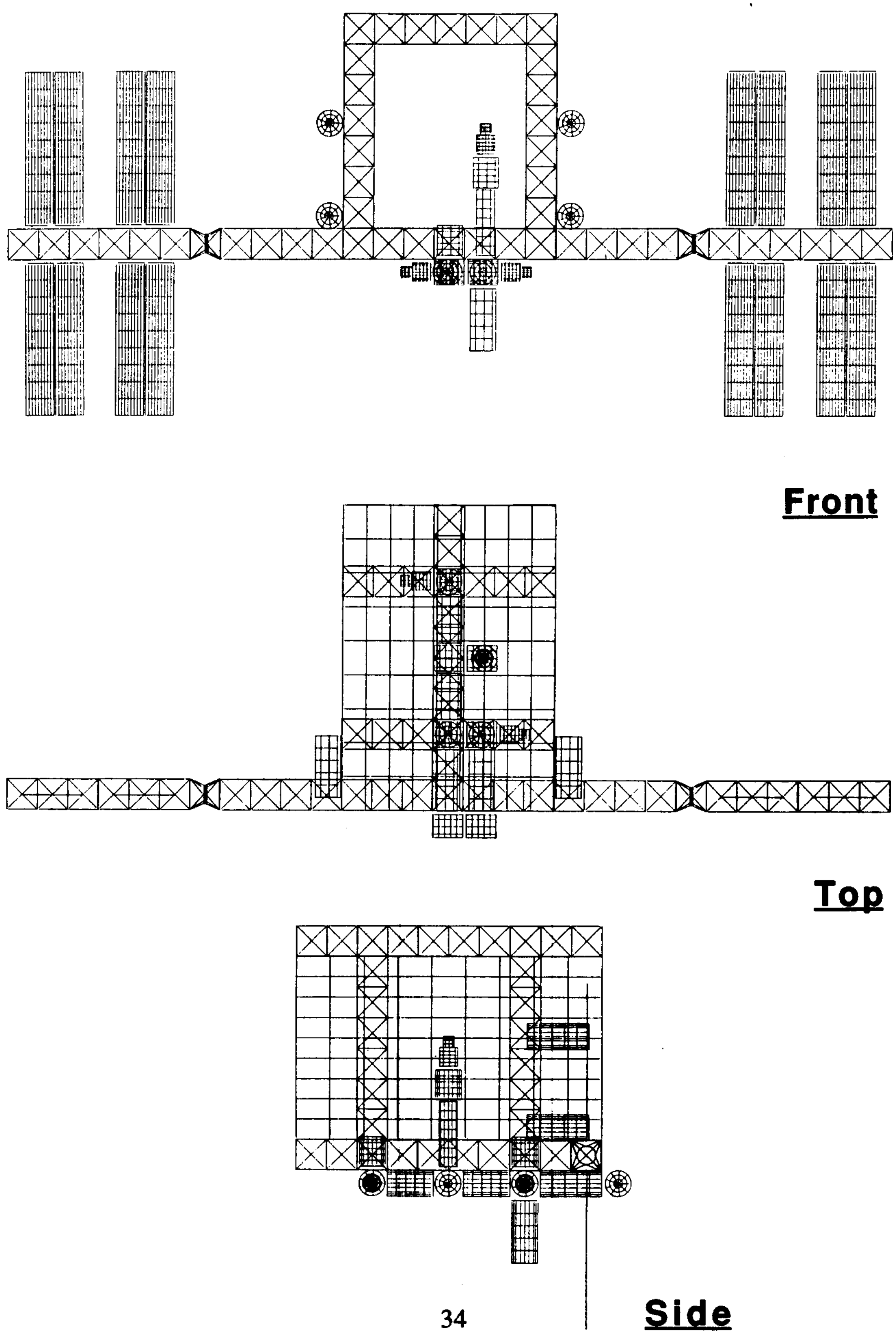


Table **9.0-1,** LEO Transportation Node Space Station Summary Weight Statement

Elements	**lbs**	**kgms**
Hangar	48,257	(21,935)
Propellant Storage (4 Tanks, dry)	122,652	(55,751)
Transfer lines, Interfaces, Other prop. related (wet)	37,975	(17,222)
Remote Manipulator System with Transporter (2)	7,200	(3,273)
Truss	14,963	(6,801)
Power Supply	57,059	(25,936)
Habitation Module 1 (Active)	45,838	(20,835)
Habitation Module 2 (Quiet)	41,260	(18,755)
Workshop Module	127,640	(12,564)
Workshop Module	227,640	(12,564)
Pressurized Logistics Module	16,845	(7,657)
Node 1 (forward starboard)	34,283	(15,551)
Node 2 (forward port)	34,283	(15,551)
Node 3 (starboard)	34,283	(15,551)
Node 4 (port)	34,283	(15,551)
Node 5 (for vert. with rotating fixture)	34,283	(15,551)
Node 6 (starboard)	34,283	(15,551)
Node 7 (port)	34,283	(15,551)
Node 8 (Hangar Control)	34,283	(15,551)
Node 9 (Aft)	34,283	(15,551)
Node 10 (Aft Vert. with rotating fixture)	34,283	(15,551)
Airlock 1 (Hyperbaric)	8,254	(3,744)
Airlock 2	8,254	(3,744)
CETAs	4,209	(1,913)
Thermal Control (Rad. & Pallets)	8,092	(3,678)
Cupola 1	3,000	(1,364)
Cupola 2	3,000	(1,364)
Cupola 3	3,000	(1,364)
Tunnel	3,058	(1,390)
GN&C Pallets (2)	9,648	(4,385)
RCS Tank Pallets (2)	8,922	(4,055)
Utility Trays	29,878	(13,550)
Antennas	1,348	(613)
Lander OTV Propellant Boom	2,000	(909)
RCMs (6)	1,782	(810)
Total (Dry)	**884,604**	**(401,686)**
Stored Cryogenic Propellant	400,000	(182,000)
Total (Wet)	**1,284,604**	**(583,686)**
Loaded OTV Stacks (2)	877,684	(398,947)
Total (Gross)	**2,162,288**	**(982,633)**

10.0 Lunar Surface Transportation Systems Conceptual Design Summary

Conceptual designs for transportation vehicles to perform three different baseline mission types were produced.

To transport crews of two to four, unpressurized, on trips of up to 50 km, a six-wheeled, articulated vehicle was chosen. This vehicle, shown in Figure 10-1, has an unloaded mass of 550 kg and is powered by four lithium metal sulfide batteries with 196 kg total mass and storing 21 kwh. The maximum power requirement for this vehicle is predicted to be 2.15 kw, with 1.6 kw required for locomotion.

To transport crews of four on traverses of up to 1,500 km from the base, a pressurized vehicle shown in Figure 1 is proposed. The vehicle is powered by shuttle-type hydrogen/oxygen fuel cells storing up to 7,000 kwh of energy. configured for a 3,000 km traverse, the total train weighs 17,600 kg and requires 25 kw peak power. Environmental control is essentially open loop with used consumables returned to the base for regeneration. The 1,500 km mission would involve numerous stops and crew excursions in suits. A trip time of 42 days is planned.

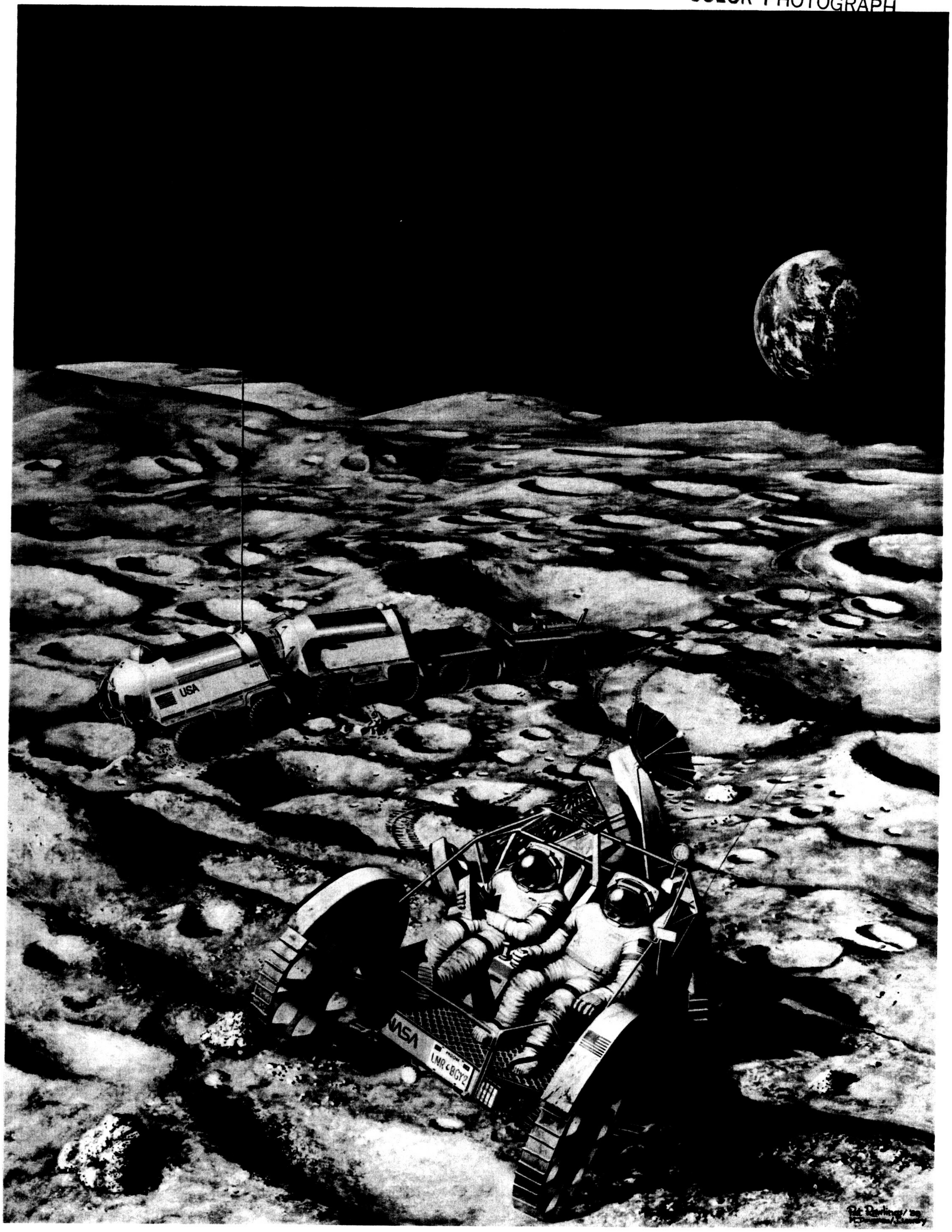
To transport crews beyond 1,500 km to the opposite side of the moon, the baseline lunar lander (see Figure 4.0-2) descending from orbit is proposed. A ballistic flyer, which would fly from the base to the opposite side of the Moon and return was also studied, but high Delta V requirements (essentially twice that required to descend and ascend to low lunar orbit) make this vehicle large and impractical for near term scenarios. The difference is that the ballistic flyer must carry sufficient propellant for the trip out and the trip back, whereas the lunar lander is assumed to refuel in lunar orbit between each trip to the surface. If the baseline lander was used as a ballistic transport from the base to points on the surface and back, tank size would limit its range to less than 1,000 km from the base.

A variety of subsystems were reviewed for each of these vehicles, including: power, propulsion, locomotion, thermal control, pressure vessels, airlocks, extra-vehicular activity (EVA) systems, life support, lighting, communication, radiation protection, and emergency breakdown. Selection criteria were developed. Numerous useful rules of thumb were recorded.

Future work should concentrate on refining the conceptual design of the vehicles in terms of practical and operational considerations. For example, both vehicles need more improvements to accommodate the rugged "off-road" service. The unpressurized vehicle steering, articulation, and suspensions need more conceptual design work. The pressurized vehicle design which is actually a train of vehicles requires more study to confirm the locomotion performance on lunar terrain. Finally, the subsystems for the pressurized vehicle all need a second iteration of design study to achieve proper vehicle integration.

Figure 10-1, Lunar Transportation Surface Vehicles Near Base

ORIGINAL PAGE
COLOR PHOTOGRAPH



11.0 Conceptual Design of a Lunar Base Solar Power Plant Summary

The objective of this study was to determine the viability of a solar powered lunar base. Study guidelines required that power plant growth occur incrementally with an ultimate capacity of 100 kW continuously. The latitude of the base was given as 18° South based upon criteria other than power systems siting.

The study required selection of the most rational static and dynamic conversion systems for comparison. Based upon trade-off analyses, one of these was to be selected for a more in-depth analysis of the impacts of a solar power system on a lunar base development.

Several photovoltaic cell technologies were investigated in several configurations of fixed and tracked arrays. From these trade studies, a fixed flat array of gallium arsenide cells was selected for comparison with a solar dynamic system. Lunar night energy is supplied from a Regenerative Fuel Cell (RFC) system with H_2 and O_2 stored at pressures up to 20,673 kpa (3,000 psi).

Rankine, Brayton and Stirling cycle engines were considered as dynamic candidates. Operation of any of these engines during the Lunar night with thermal energy storage was determined to be impractical if the storage media is required to be transported from the earth. As a result, energy storage is based on the use of an RFC which would be identical to that required for the photovoltaic system. Alternator output would be required to be converted to DC for the RFC. Based primarily on weight, the Stirling cycle was selected for comparison with the photovoltaic system.

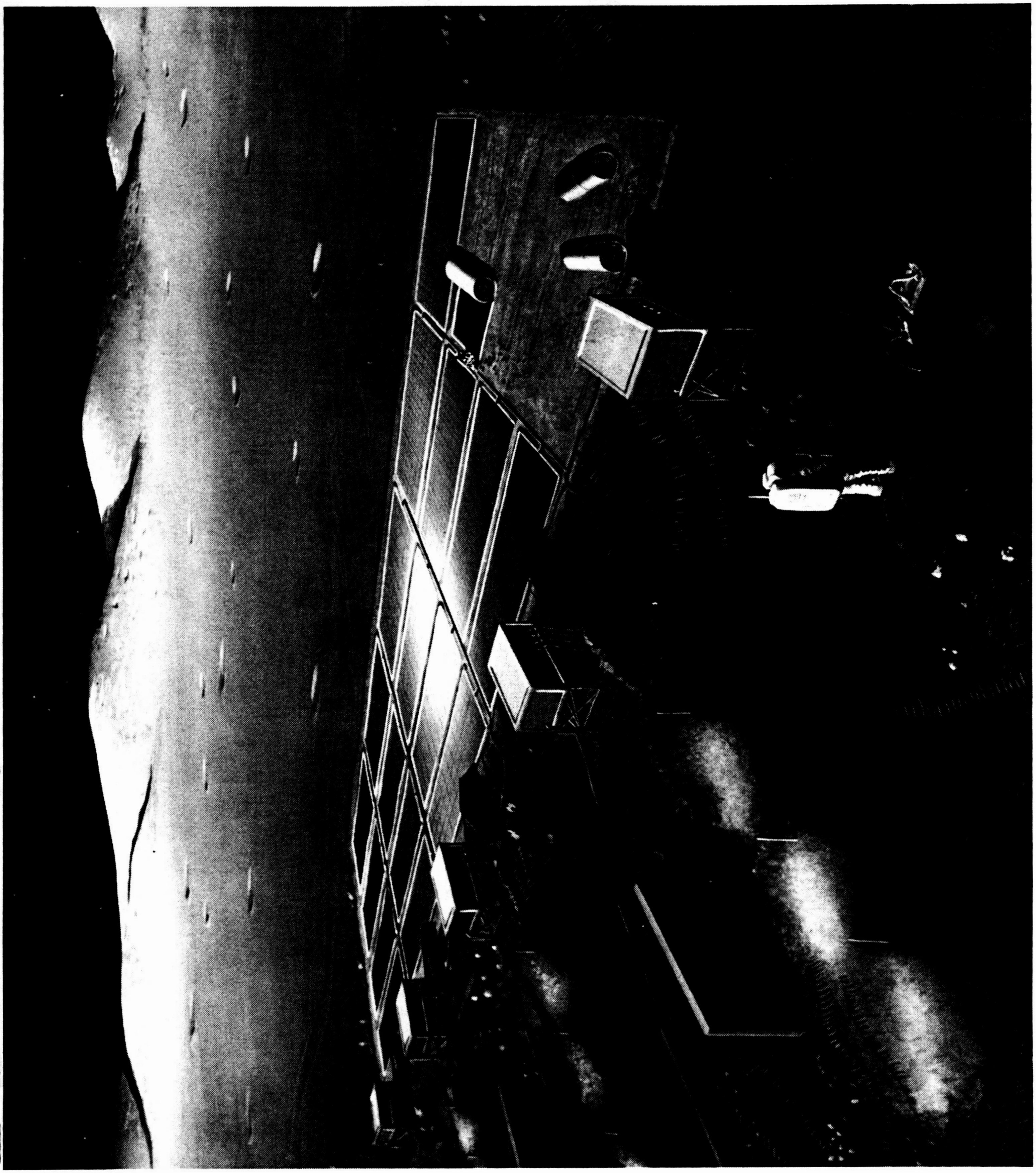
The primary obstacle to the successful use of solar power systems on the lunar surface is the extremely long period in each cycle in which solar energy is unavailable. As a result, the energy storage system constitutes over 90 percent of the photovoltaic and over 50 percent of the dynamic system weights. The photovoltaic power generation portion of the system would weigh approximately one-eighth that of the corresponding Stirling generation equivalent, and is therefore the logical candidate for the initial Lunar Solar Power Plant.

The first power plant module to be deployed would carry a portion of the reactants as gaseous H_2 and O_2. The amount that would be necessary would be determined after operational timelines were developed. The total mass to be transported would remain constant and the first 25 kW module would only require 12.5 MT and 60 m^3 of payload capacity. Landed mass for a 100 kW continuous power system is 50 MT. The most difficult part of the power plant installation appears to be the placement and burial of the reactant tanks which is required for thermal and micrometeorite protection. Equipment handling and trenching machinery must be landed prior to the first power plant module.

Figure 11.0-1 depicts a roll-out, flat-plate solar array configuration for a 100 kW (net) system. The fuel/electrolysis cell modules and associated tanks are sized for 25 kW each, a size well suited for logistics and incremental power plant buildup. A fifth segment is shown being deployed illustrating power plant growth. There are no known limitations to the size of a power plant so constructed. Being modularized allows for flexibility in siting individual segments so that "mini" power plants may be located closer to the users. Thus it seems reasonable to consider the photovoltaic/regenerative fuel cell power system for lunar surface applications into the multi-megawatt power range or until nuclear sources becomes available.

ORIGINAL PAGE
BLACK AND WHITE PHOTOGRAPH

Figure 11.0-1, Lunar Base 100 kwe Solar Power Plant



12.0 Surface Construction and Assembly Equipment Summary

This study was initiated to develop requirements for equipment to be used in constructing and assembling a permanently manned lunar base. A survey of lunar construction and assembly tasks was made and the requirements that these tasks place on construction equipment were identified. Major construction tasks that are most likely to occur during construction of a lunar base primarily involve cargo and soil handling operations such as:

- Unloading cargo from lunar landers.
- Transporting loads.
- Lifting and positioning loads.
- Preparing the lunar surface for a base site, landing pads, and roads.
- Providing quantities of lunar soil for habitat radiation protection.
- Assembling large structures.

Cargo handling system requirements include maximum cargo size and weight that must be managed, transportation distance to be traversed, and maximum lift and reach that these systems must have to unload and position the cargo. The maximum weight that cargo unloading and transporting equipment must be able to handle is fixed by the payload capability of the lunar lander which, for a lander concept recently studied (3), is on the order of 25,000 kg. Cargo size influences transporter size as well as road width. Sizes of a number of potential cargo elements for a lunar base are reviewed in this report. Cargo size has generally been constrained to the Shuttle payload bay envelope and the maximum cargo dimension identified was 4.5 m x 14 m. However, availability of heavy lift launch vehicles could allow delivery of payloads with greater diameter. Required lift and reach is dictated by lunar lander dimensions and cargo manifesting configuration.

Soil handling system requirements include the quantity of soil needed for radiation protection, and the required grade or slope and the amount of soil to be moved to prepare the base site, landing pads, and roads. Lunar terrain and crater density as well as the size of the base elements effects the magnitude of these jobs and the type of equipment needed. For instance, a partially buried inflatable spherical habitat would require such a large excavation that blasting will probably be necessary which means a mobile drill unit will be needed. A base constructed of buried modules would not require excavating or blasting.

An important element missing from the requirements is the amount of time allowed in the schedule for completing the various construction tasks and the quantity of crew time (extra- and intravehicular activity time) available to support the activities. Timing has a direct bearing on the required size and number of equipment, and can influence the type of construction equipment selected. More study is required to better define both task and timing requirements.

The severity of the lunar environment (dust, vacuum, deep thermal and long diurnal cycles) and its remoteness dictates that lunar construction equipment concepts should have the following goals:

- Versatility: The systems can be made capable of performing multiple tasks by attaching different implements.

- Commonality: A modular design and common subsystem approach should be pursued where practical to reduce spares and maintenance requirements.

- Reliability: Dust-control, lubrication, and maintenance will be important design considerations.

- Low Weight: Although the equipment must be rugged for reliability, lunar materials (soil or rocks) could be used as counterweights and/or ballast to improve the stability and/or traction of the equipment and to reduce the machine's Earth launch weight.

- Telerobotics: The systems should be capable of both manual and teleoperated operation to potentially reduce EVA requirements.

Terrestrial construction equipment functions and capability are described. Several versatile machine combinations are commonly used on Earth construction sites, such as a backhoe front-end loader machine and a boom crane with multiple attachments for hoisting, grappling, and excavating.

A preliminary comparison was made of equipment options to perform the lunar construction/-assembly task set. More work is needed before an optimum set of equipment can be selected with confidence. Figures 12.0-1 and 12.0-2 show representative concepts. The comparison did indicate that one possible set of equipment that could perform the lunar tasks would consist of the following major equipment elements:

- Mobile boom crane. The boom crane would be used to hoist cargo off landers and surface transporters, place soil over habitation elements for radiation protection, and provide a backup to the soil excavator. Crane attachments needed for these operations include a hoisting hook and cargo sling, dumpable soil transfer bucket, and a pile-driving ram to emplace anchors.

- Soil excavator and surface grader/leveler. Capability for excavating/grading could be provided by a front-end loader using a multi-application bucket which can be used as a shovel, bulldozer blade, or scraper. For deeper excavations, a front-end loader and a backhoe machine mounted on a single prime mover tractor is a possibility. A compactor roll attachment can be provided for the prime mover tractor, and pulled to compact the lunar surface. Other excavators, such as a bucket wheel excavator, should also be examined in more detail.

- Haulers. Several flatbed cargo transporters are required with mounting cradles for constraining large cargo elements. Soil transport trucks will be required if large soil volumes must be moved in short time periods.

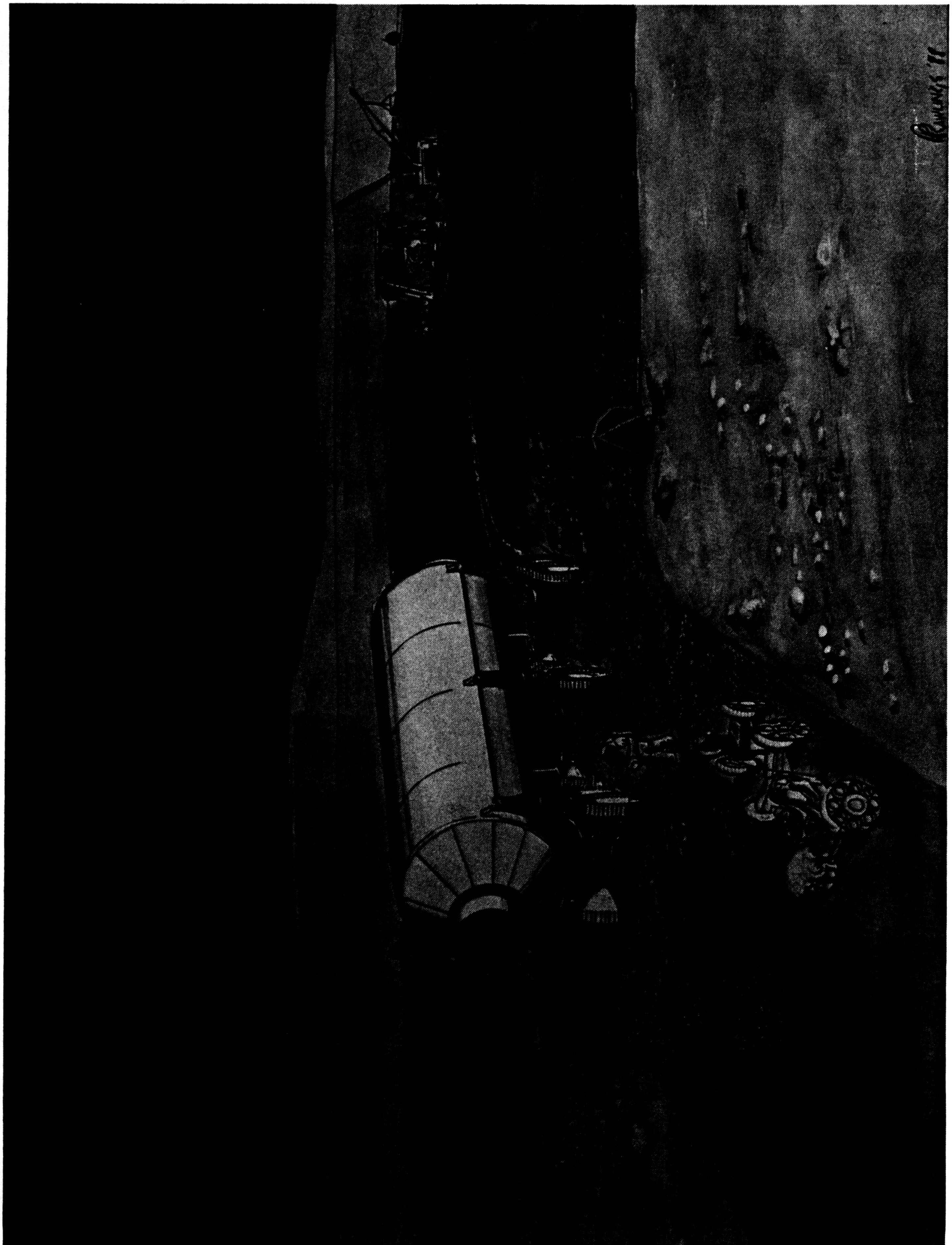
- Auxiliary Equipment. Miscellaneous equipment needed for the job set includes a ramp or chute for contingency lander unloading operations, jacks for lifting a lander (in the event a lander needs to be moved), a local transportation vehicle (LOTRAN) for crew transport, and rock and soil drills for blasting large boulders or large excavations if needed. Requirements for blasting need more definition. A small drill rig could be attached to a prime mover to provide mobility. A drill device used for scientific coring could double as a construction tool.

It is recommended that conceptual designs be developed for several construction equipment elements to accomplish a well defined set of tasks within a preliminary schedule. Trade studies are required to better define the primary power source, propulsion means (wheels vs. tracks), and actuator/control systems (hydraulic, electric, mechanical linkage). More detailed study of requirements for teleoperation of these vehicles from a lunar base and from Earth is also needed.

Figure 12.0-1, Crane with Bucket, Front-End Loader, and Hauler

Figure 12.0-2, Crane Unloads Module on to Transporter

ORIGINAL PAGE
BLACK AND WHITE PHOTOGRAPH



13.0 Lunar Application of Superconductivity Study Summary

Superconductors are materials that exhibit zero electrical resistance when cooled to temperatures below a critical value characteristic of the material. Until recently, these temperatures have been in the 1-40°K range, which requires that the material be cooled using expensive liquid helium. The past two years have seen advances in the development of high-temperature superconductors that have led to a resurgent interest in their potential applications. Materials are being discovered that achieve superconducting characteristics at temperatures over 100°K, allowing cooling with relatively inexpensive liquid nitrogen. Research is continuing to find a room-temperature superconductor that will require no refrigeration.

Magnets constructed of superconducting materials are finding many applications because of their ability to store large amounts of energy in their coils in the form of electrical current with virtually no energy loss. They have the added advantage of being dischargeable in durations ranging from a fraction of a second to many hours or days. Three applications of superconductor magnet technology at a lunar base are discussed in this report: magnetic energy storage during the lunar day for usage during the night; electromagnetic rail launchers to propel lunar-derived oxygen or raw materials into lunar orbit; and magnetic shielding to protect lunar inhabitants from radiation.

The practical usage of superconducting magnetic energy storage (SMES) devices to store large amounts of energy has been proven through the development of prototypes by several research institutes and utility companies. Current technology makes them competitive with capacitor storage but not yet with fuel cells. SMES devices are practical as energy storage units only at stored levels above 500 MWH (1.5 MW delivered for 14 days), which corresponds to lunar energy storage requirements at the early settlement phase (100 person) or perhaps for propellant plants.

Several geometries are considered for the coil, and a toroid shape is recommended to eliminate fringe fields. Such a device would be about 200 meters in diameter and 30 meters high to store 500 MWH.

Current terrestrial designs require cooling of the coil with liquid helium or liquid nitrogen in order for the devices to achieve their superconducting characteristics. Low temperature on the Moon may eliminate this need, though, making SMES even more competitive. Devices situated above the surface can be shaded with artificial shadows, producing operating temperatures of about 110°K. Superconducting materials already exist that work at this temperature, but higher current densities than those experimentally achieved are necessary before these can be used for practical energy storage. An above-ground SMES device would require strong containment, probably with steel, to withstand the large hoop stresses generated by the coil. The device could be buried in lunar bedrock to contain these forces, but a subsurface temperature of 230°K would require the discovery of a new superconductor material with a critical temperature higher than this.

Some advantages of SMES over other technologies include high efficiency, high reliability, and high energy density.

Superconductor technology has made possible the development of electromagnetic launchers (EML), which require the delivery of high energy in short bursts of power. EML designs have been proposed that are capable of accelerating 1,000 kgs at 1,000 gravities to 12.3 km/s. These devices offer several advantages over rocket launch systems, including a higher payload fraction, greater launch rate, lower launch unit cost, and greater reliability.

Two families of EML are discussed: railgun and coaxial. Railgun devices consist of two parallel rails connected to a pulsed direct current with a projectile propelled between the rails by Lorentz forces generated in the projectile's armature. Coaxial devices use a linear synchronous motor to accelerate payload buckets containing energized coils along an assembly of fixed coaxial coils several kilometers in length. Coaxial EML's offer several advantages over railguns, a including larger bore diameters for the projectiles, greater efficiency, longer life span, and the ability to operate well at moderate accelerations. A problem with coaxial EML's is the generation of large internal voltages that could cause arcing.

The development of high temperature superconductors has led to the application of quenching methods to a lunar electromagnetic launcher. This method uses successive switching on and off (quenching) of adjacent coils to propel the projectile. Problems with this approach that need further attention include the requirement to withstand high magnetic induction, high current densities, and large stresses. The advantage over other coaxial designs is the elimination of the need for superconducting switches. Quenchguns may be able to accelerate 1000 kg payloads to 1.7 km/s. Figure 13.0-1 shows a concept for an EML on the lunar surface inside a shielded, pressurized volume.

Further study needs to be performed to determine the tradeoffs of electromagnetic launchers versus reusable landers for oxygen and raw material transport.

A major concern of lunar mission planners is the exposure of astronauts to high levels of cosmic radiation, and occasional high-radiation dosages from solar flares. To safeguard the astronauts, radiation shielding must be considered for lunar habitats. The large magnetic fields generated in superconducting magnets may be beneficial for trapping charge particles away from the habitats. Two approaches are considered here: toroidal magnetic shielding and plasma core shielding.

The cosmic ray spectrum must be cut off at 10-15 GeV/nucleon to achieve an acceptable dosage rate of 5 rem/year. It is estimated that on the order of 4.5 metric tons/m^2 of passive shielding is required to provide this protection. If active magnetic shielding is used, current densities of 10^9 amps/m^2 are required to achieve this cutoff, a value approachable only with the use of superconducting coil materials.

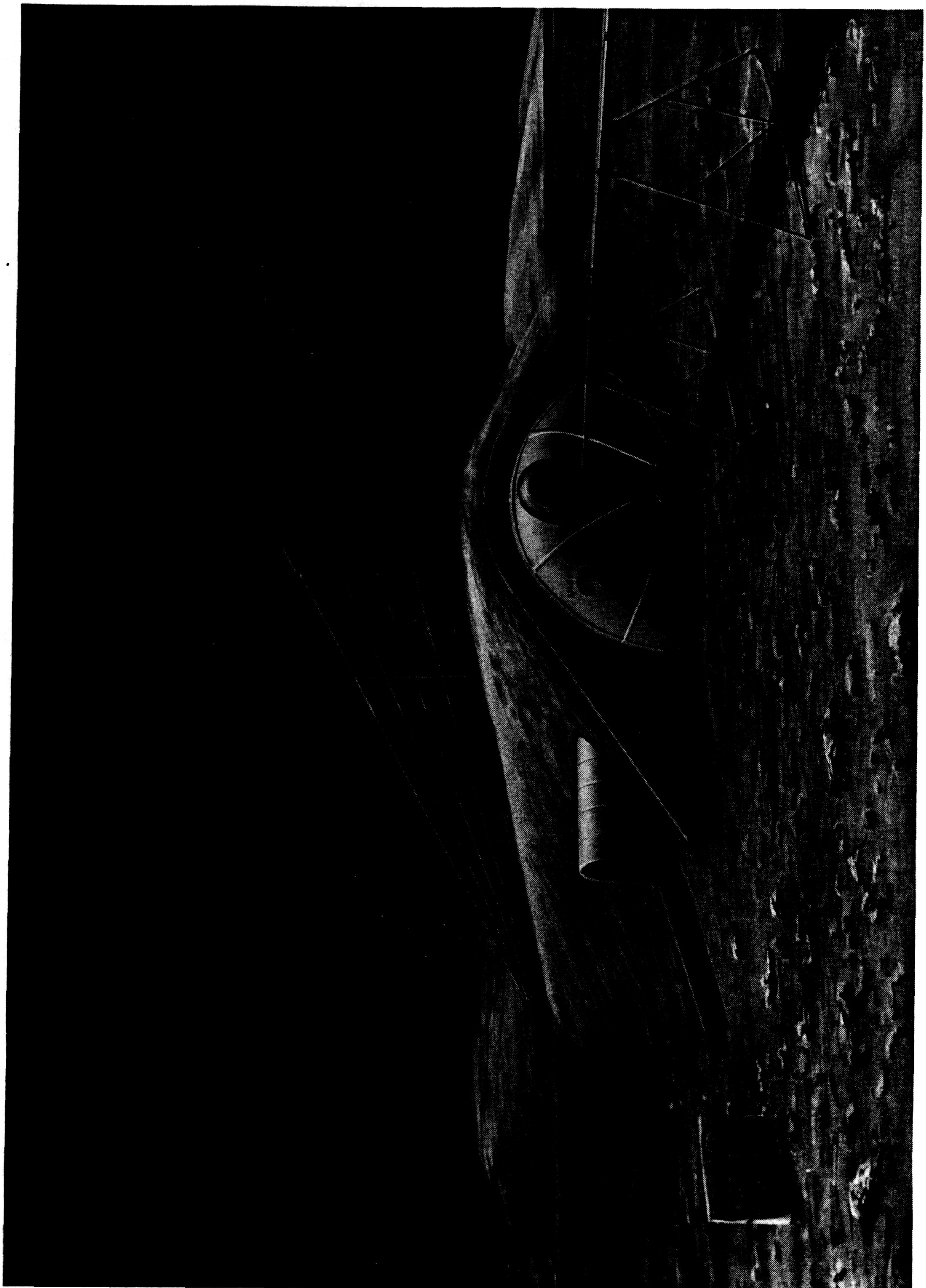
Three configurations are considered for magnetic shielding:

- An unconfined field dipole, which is toroidal in shape producing a shield outside the torus,
- A confined field double torus, which places one toroid inside another and traps the particles between them, and
- A hybrid of the two in which a deformed toroidal winding is used to produce a spherical shape.

The hybrid magnetic shield is the most attractive configuration because it leads to the lowest dosage rate and has the lowest system mass.

A plasma core shield creates a huge electric field around the habitat that deflects positively charged cosmic rays back into space. The attraction of surrounding electrons could neutralize the device, but superconducting magnets are arranged to create an electron well that traps these electrons. The advantage of a plasma core shield over a magnetic shield is that no electrons occur near the exterior surfaces of the habitat. The major disadvantage is the extremely high voltages required, which may cause arcing between the habitat and ground. To overcome the arcing problem, a system that combines passive mass shielding and low-power magnetic shielding may be desireable.

Figure 13.0-1, Electromagnetic Launcher on the Lunar Surface



14.0 Lunar Base Scenario Cost Estimates

This report describes the estimated development and production costs, in constant 1988 dollars, of each of the systems conceptually designed under the Advanced Space Transportation Support Contract. In addition, estimates were derived for a unit cost (dollars per kilogram) to transport the systems from Earth to the Lunar surface and for a unit cost (dollars per EVA and IVA hour) to set up the systems on the Lunar surface. These estimates do not include the cost of spares, consumables, new facilities for system development and production, or ongoing operations on the lunar surface.

The ASTS contract did not include provisions for designing crew habitation and laboratory modules, nor for costing them. However, a price tag for the entire lunar system would not be complete without their inclusion. Solely for the purpose of providing a more complete picture of lunar system costs, gross cost estimates were made for these modules, using a cost estimating relationship developed for estimating space station module costs [14]. The projected pressurized volume is 658.17 m^3, and includes two habitation modules, one laboratory, one node, and two airlocks. The projected cost for pressurized volume is $4028/$m^3$, or $2,651,000,000 for the entire system. The development to production cost ratio was assumed to be 3:1 for these modules.

Table 14.0-1 summarizes the total system hardware costs, and Table 14.0-2 summarizes the unit costs for transport and setup.

Table **14.0-1**, Summary of Lunar Base Scenario Estimated Costs ($Millions)

System	Development	Production	Total
Lunar Lander	$ 1,415	$ 649	$ 2,064
Lunar Oxygen Pilot Plant	732	122	854
Unpressurized Lunar Rover	140	47	187
Pressurized Lunar Rover	474	184	658
Solar Power Plant	314	118	432
Logistics Module	242	108	350
Storm Shelter	241	70	311
Transportation Node	7,219	2,361	9,580
Surface Construction Equipment	350	79	429
Fuel Cell Cart	70	13	82
Supplemental Cooling Cart	45	7	52
Orbital Transfer Vehicle	1,464	1,059	2,523
Low Earth Orbit Launcher	4,162	13,166	17,328
Lunar Landing Pad	581	104	685
Surface Habitats/Labs	1,988	663	2,651
Total	$19,437	$18,750	$38,186

Table 14.0-2, Summary of Lunar Base Scenario Transport and Setup Costs

Operation	Unit Cost
Transport	
Earth-Lunar Surface	\$ 23,732/kg
Setup	
EVA	\$ 84,237/hour
IVA	\$ 29,483/hour

As a point of comparison, the Apollo program cost $93 Billion in 1988 dollars.

15.0 Spacecraft Mass Estimation, Relationships, and Engine Data Summary

At the first of the software task a variety of information was collected to aid in weight estimation. The book produced contains a collection of scaling equations, weight statements, scaling factors, etc., useful to someone doing conceptual design of translunar spacecraft. It provides rules of thumb and methods for calculating quantities of interest. Basic relationships for conventional--and several non-conventional--propulsion systems (nuclear and solar electric, and solar thermal) are included. The equations and other data have been taken from a number of sources and are not all consistent with each other in level of detail or method, but provide useful references for early estimation purposes. Table 15.0-1 exemplifies the data collected.

Scaling equations are presented on two levels: overall vehicle sizing and subsystem sizing. The equations for overall vehicle sizing are quick and simple. They should be used when extreme accuracy is not a prerequisite. When higher fidelity is required, and time is not an overriding concern, the vehicle can be sized by subsystem, using the subsystem sizing equations and relationships.

Vehicle subsystems can be broken down in any number of ways. To prevent confusion, a list of general subsystems discussed throughout this book is presented here:

- Propellant
- Engines
- Avionics
- Structures
- Aerobrakes and Heatshields
- Environmental Control and Life Support
- Crew
- Power and Electrical
- Landing and Docking
- Propellant Tanks
- Insulation and Thermal Protection
- Attitude Control

The relationships and other numbers collected here are primarily for Orbital Transfer Vehicles (OTV's) operating between Low Earth Orbit (LEO) and Low Lunar Orbit (LLO), and for lunar surface lander/launchers.

Table **15.0-1,** Rocket Engine Performance (LO_2/LH_2)

Liquid Oxygen/Liquid Hydrogen Engines

Application	Thrust (vac) lbf	Engine Mass lbm	Isp (vac) sec.	Mixture Ratio
NASA M-1 Booster (proposed)	1,500,000	20,000	428	5
Booster (proposed)	588,400	7,775	446	7
Space Shuttle Main Engine	470,000	7,000	453	6
Saturn IB & V Second Stage Vehicle	230,000	3,480	422	5.5
Centaur - Upper Stage Vehicle (RL-10)	15,000	292	444	5
Advanced Space Engine (proposed)	7,500	870	485	-
RCS (proposed)	1,250	31	427	4.5
Advanced OTV (proposed)	500	44	456	6
Space Propulsion/RCS (proposed)	25	3.8	400	3

16.0 Earth to Low Lunar Orbit and Back (all in plane) Trajectory Program (LLOFX)

The program LLOFX calculates in-plane trajectories from an Earth-orbiting space station to Lunar orbit in such a way that the journey requires only two delta-v burns (one to leave Earth circular orbit and one to circularize into lunar orbit). The program requires the user to supply the space station altitude and lunar orbit altitude (in kilometers above the surface), and the desired time of flight for the transfer (in hours). It then determines and displays the Trans-Lunar Injection (TLI) delta-v required to achieve the transfer, the Lunar Orbit Insertion (LOI) delta-v required to circularize the orbit around the Moon, the actual time of flight, and whether the transfer orbit is elliptical or hyperbolic. Return information is also displayed. Finally, a plot of the transfer orbit is displayed.

Calculation of the trajectory takes advantage of the fact that the Moon travels at great velocity in orbit around the Earth (1.02 kilometers per second). The vehicle's circular orbit about the Earth is turned into an elliptical transfer orbit that intercepts the Moon's orbit. This transfer orbit is rotated ahead of the Earth-Moon line in such a way that, as the vehicle enters the Moon's Sphere of Action (SOA) ahead of the Moon, the high velocity of the Moon in the direction of the vehicle causes the vehicle to appear to be headed back toward the Moon (from a Lunar point of view). This program identifies the eccentricity, size, and rotation of the transfer ellipse or hyperbola that causes the velocity vector of the vehicle (in Lunar coordinates) to correspond to an orbit passing in front of the Moon with a perigee at the Lunar orbit altitude supplied by the user.

17.0 Earth to Low Lunar Orbit and Back (Variable Geometry) Trajectory Program (PLANECHG)

The program PLANECHG calculates velocities for Earth-to-Moon or Moon-to-Earth trajectories. The flight to be analyzed originates in a circular orbit of any inclination and altitude about one of the bodies, and culminates in a circular orbit of any inclination and altitude about the other body. An intermediate ΔV and plane change occurs at the lunar sphere of influence (SOI), the region where the vehicle is near its lowest velocity in the trajectory, and therefore where it is able to make the plane change with the lowest ΔV. This results in a three-burn trajectory.

A given flight may penetrate the SOI at a number of points. Each point has associated with it a unique set of ΔV's and total velocity. This program displays the velocities, in matrix form, for a representative set of SOI penetration points. An SOI point is identified by projecting Lunar latitude and longitude onto the SOI. The points reported for a given flight are defined by the user, who provides a starting longitude and latitude, and an increment for each. A matrix is built with ten longitudes forming the columns and 19 latitudes forming the rows. This matrix is presented in six different reports, each report containing different velocity or node information in the body of the matrix.

The technical and user documentation describes the inputs provided by the user to define the flight profile and the contents of the six reports that are produced as outputs. Instructions to execute this program, and a look at the structure and details of the program code are also included. Table 17.0-1 is an example of output, the first report.

Table 17.0-1, PLANECHG Example Output

```
VELOCITY MAP FOR OUTBOUND TRAJECTORIES NODE OPTION, 2
DATE  7-SEP-88   TIME  08:35:41
JUILIAN DAY 2451545.   PERIGEE ALT (NMI) EARTH = 250.  MOON =  60.
TRANSLUNAR FLIGHT TIME (HR) =  60.0  INCL EARTH  30.0  INCL MOON =  30.0
```

ALON> ALAT	0.0	-10.0	-20.0	-30.0	-40.0	-50.0	-60.0	-70.0	-80.0	-90.0
90.0	0.	0.	0.	0.	0.	0.	0.	0.	0.	0.
80.0	0.	0.	0.	0.	0.	0.	0.	0.	0.	0.
70.0	0.	0.	0.	0.	0.	0.	0.	0.	0.	0.
60.0	0.	0.	0.	0.	0.	0.	0.	0.	0.	0.
50.0	0.	0.	0.	0.	0.	0.	0.	0.	0.	0.
40.0	0.	0.	0.	0.	0.	0.	0.	0.	0.	0.
30.0	16011.	15685.	15512.	15563.	15857.	16333.	16926.	17556.	18216.	18876.
20.0	15698.	15254.	14950.	14946.	15305.	15894.	16583.	17309.	18041.	18764.
10.0	15444.	14868.	14351.	14213.	14757.	15518.	16320.	17125.	17917.	18689.
0.0	15369.	14745.	14100.	13727.	14544.	15394.	16238.	17070.	17883.	18672.
-10.0	15506.	14964.	14518.	14428.	14883.	15591.	16367.	17160.	17948.	18719.
-20.0	15829.	15427.	15173.	15180.	15496.	16033.	16687.	17391.	18113.	18831.
-30.0	16357.	16099.	15976.	16032.	16277.	16682.	17206.	17781.	18398.	19026.
-40.0	0.	0.	0.	0.	0.	0.	0.	0.	0.	0.
-50.0	0.	0.	0.	0.	0.	0.	0.	0.	0.	0.
-60.0	0.	0.	0.	0.	0.	0.	0.	0.	0.	0.
-70.0	0.	0.	0.	0.	0.	0.	0.	0.	0.	0.
-80.0	0.	0.	0.	0.	0.	0.	0.	0.	0.	0.
-90.0	0.	0.	0.	0.	0.	0.	0.	0.	0.	0.

```
MINIMUM VELOCITY PRINT NODE OPTION 2
DATE  7-SEP-88 TIME  08:35:41

FLIGHT TIME =  60. LAT =    0.0 LON =  -30.0 LUNAR AINC =  30.0 EARTH AINC =  30.0
BODY    VX      VY     VZ     VEL    GAMA    AZM     AINC    ANODE  TIME  DVCIR
MOON   3641.   700.  -199.  4400.  -84.8  120.0    30.0    150.0   7.5   3248. DVPHER = 229.
EARTH  3799.   793.  -336.  3895.   80.1   120.0    30.0 3.2   52.5  10250. DVTOTAL =  13727.
```

18.0 Low Lunar Orbit to Surface and Back Trajectory Program (LANDER)

LANDER is a computer program used to predict the trajectory and flight performance of a spacecraft ascending or descending between a low lunar orbit of 15 to 500 nautical miles (nm) and the lunar surface. It is a three degree-of-freedom simulation which is used to analyze the translational motion of the vehicle during descent. Attitude dynamics and rotational motion are not considered.

The program can be used to simulate either an ascent from the Moon or a descent to the Moon. For an ascent, the spacecraft is initialized at the lunar surface and accelerates vertically away from the ground at full thrust. When the local velocity becomes 30 ft/s, the vehicle turns downrange with a pitch-over maneuver and proceeds to fly a gravity turn until Main Engine Cutoff (MECO). The spacecraft then coasts until it reaches the requested holding orbit where it performs an orbital insertion burn. Figure 18.0-1 is an example of ascent output.

During a descent simulation, the lander begins in the holding orbit and performs a deorbit burn. It then coasts to periapsis, where it reignites its engines and begins a gravity turn descent. When the local horizontal velocity becomes zero, the lander pitches up to a vertical orientation and begins to hover in search of a landing site. The lander hovers for a period of time specified by the user, and then lands.

Newton-Raphson iteration techniques are used to optimize the pitch-over maneuver and the MECO time for proper orbit insertion. Integration is performed using a Runge-Kutta fourth order integrator. This integrator has been verified with launch simulations of the Titan and Conestoga launch vehicles. LANDER receives input, presents output, and does all calculations in English units. The basic coordinate system is spherical. The Moon is modelled as a spherical body of uniform gravity having no atmosphere and no gravitational harmonics.

Even though the output for a descent simulation appears to start at orbit and end at the surface, the mathematical calculations are performed in reverse. The program actually initializes the lander at the lunar surface and proceeds to simulate an ascent using negative mass flow. After the proper orbit has been achieved the data is recognized and printed in the proper chronological sequence for a descent. Note: that this "reversed flight" is only characteristic of the descent simulations. Apollo descent simulations used this same "reversed" technique to decrease the required number of iterations needed to find a satisfactory trajectory.

Figure 18.0-1, LANDER Example Output

ORIGINAL PAGE IS
OF POOR QUALITY

LANDER Ascent Simulation of the Apollo 15 Lunar Ascent Module

The following inputs are supplied at the program prompts:

IS THIS TO BE AN ASCENT OR DESCENT SIMULATION ?
Answer: A

LANDING SITE LATITUDE (-90 TO +90)
Answer: 26.1011

LANDING SITE LONGITUDE (0 TO 360)
Answer: 3.6527

INERT WEIGHT <LB>
Answer: 5326

PROPELLANT WEIGHT <LB>
Answer: 5589'

THRUST <LBF>
Answer: 3500

SPECIFIC IMPULSE <S>
Answer: 306

HOVER TIME <S>
Answer: (Not Applicable)

PAYLOAD WEIGHT <LB>
Answer: 0

TIME TO MAIN ENGINE CUT-OFF (MECO)? <S>
Answer: 460

HOLDING ORBIT PERICYNTHION <NM>
Answer: 50

HOLDING ORBIT APOCYNTHION <NM>
Answer: 50

FLIGHT PATH ANGLE AT PITCH-OVER ?
Answer: 85

HOLDING ORBIT INCLINATION ? (0 TO 360)
Answer: 26.2

DO YOU WISH TO SEE THE TRAJECTORY OF EACH ITERATION ?
Answer: N

OUTPUT:

Time <s>	Altitude <ft>	Range <nm>	Velocity <ft/s>	Gamma <deg>	Heading <deg>	Thrust <lbf>	Weight <lbm>
0.	0.	0.	2.	90.00	87.63	3500.	10915.
5.	71.	0.	27.	90.00	87.63	3500.	10858.
10.	267.	0.	52.	84.34	87.63	3500.	10801.
15.	590.	0.	78.	81.47	87.63	3500.	10743.
20.	1038.	0.	104.	78.58	87.63	3500.	10686.
25.	1612.	0.	131.	75.71	87.63	3500.	10629.
30.	2310.	0.	159.	72.87	87.63	3500.	10572.
35.	3129.	0.	187.	70.08	87.63	3500.	10515.
40.	4067.	0.	216.	67.35	87.63	3500.	10457.
45.	5120.	0.	246.	64.68	87.63	3500.	10400.
50.	6286.	0.	276.	62.07	87.63	3500.	10343.
55.	7559.	1.	308.	59.55	87.63	3500.	10286.
60.	8936.	1.	340.	57.09	87.63	3500.	10229.
65.	10412.	1.	373.	54.72	87.63	3500.	10172.
70.	11981.	1.	407.	52.42	87.63	3500.	10114.
75.	13639.	1.	443.	50.21	87.63	3500.	10057.
80.	15380.	2.	479.	48.08	87.63	3500.	10000.
85.	17199.	2.	516.	46.02	87.63	3500.	9943.
•							
•							
•							
325.	109036.	75.	3315.	3.18	87.63	3500.	7198.
330.	109926.	78.	3392.	2.91	87.63	3500.	7140.
335.	110758.	81.	3470.	2.65	87.63	3500.	7083.
340.	111532.	85.	3549.	2.41	87.63	3500.	7026.
345.	112250.	88.	3628.	2.18	87.63	3500.	6969.
350.	112912.	91.	3708.	1.96	87.63	3500.	6912.
355.	113520.	95.	3789.	1.76	87.63	3500.	6855.
360.	114075.	98.	3871.	1.57	87.63	3500.	6797.
365.	114579.	101.	3953.	1.39	87.63	3500.	6740.
370.	115032.	105.	4036.	1.22	87.63	3500.	6683.
375.	115438.	109.	4120.	1.06	87.63	3500.	6626.
380.	115797.	112.	4205.	0.92	87.63	3500.	6569.
385.	116111.	116.	4291.	0.78	87.63	3500.	6511.
390.	116384.	120.	4377.	0.66	87.63	3500.	6454.
395.	116616.	124.	4464.	0.55	87.63	3500.	6397.
400.	116812.	128.	4553.	0.45	87.63	3500.	6340.
405.	116973.	132.	4641.	0.36	87.63	3500.	6283.
410.	117102.	137.	4731.	0.28	87.63	3500.	6225.
415.	117203.	141.	4822.	0.21	87.63	3500.	6168.
420.	117278.	145.	4913.	0.15	87.63	3500.	6111.
425.	117331.	150.	5006.	0.10	87.63	3500.	6054.
430.	117365.	154.	5099.	0.06	87.63	3500.	5997.
435.	117384.	159.	5193.	0.03	87.63	3500.	5940.
440.	117393.	164.	5288.	0.01	87.63	3500.	5882.
445.	117395.	168.	5384.	0.00	87.63	3500.	5825.

```
Ideal Performance Delta Velocity is:  6254.70 <ft/s>

   Boost Orbit:

   Apocynthion                      --    36.6000 <nm>
   Pericynthion                     --    19.3000 <nm>
   Inclination                      --    26.2500 <deg>
   Longitude of the Ascending Node --   283.0000 <deg>
   Argument of Pericynthion         --    91.2600 <deg>
   Eccentricity                     --     0.0089 <nd>

  Velocity Required at Apocynthion to Achieve
  the Holding Orbit (  37. X  50. )      :     42.50 <ft/s>

  Fuel Required for the Apocynthion Burn :     24.94 <lbm>

  Weight After Apocynthion Burn          :   5754.00 <lbm>

  Weight of the Payload Placed in Orbit  :    428.00 <lbm>

************ SIMULATION COMPLETE ***************
```

19.0 Earth to Low Lunar Orbit and Back (all in-plane) Low Thrust Program (CISLUNAR)

CISLUNAR is a stand alone program designed to generate the trajectory of a low-thrust spacecraft travelling in Earth-Moon space. The program allows the creation of functional trajectories dependent upon the supplied spacecraft characteristics. The trajectory generation is a user interactive process. The program user must modify the necessary control values to create a satisfactory trajectory. Figure 19.0-1 illustrates example output.

Initial screen display information shows the spacecraft's default characteristics. These characteristics can be modified by the user at the beginning of each run. The program prompts the user for the direction of the trajectory generation by asking whether the initial orbit is about the Earth or the Moon. This sets the direction flags for the rest of the program. Initial altitude, velocity, and orbital position of the spacecraft must be entered. The altitude must be input for the program to continue. The velocity will default to the circular velocity at the input altitude.

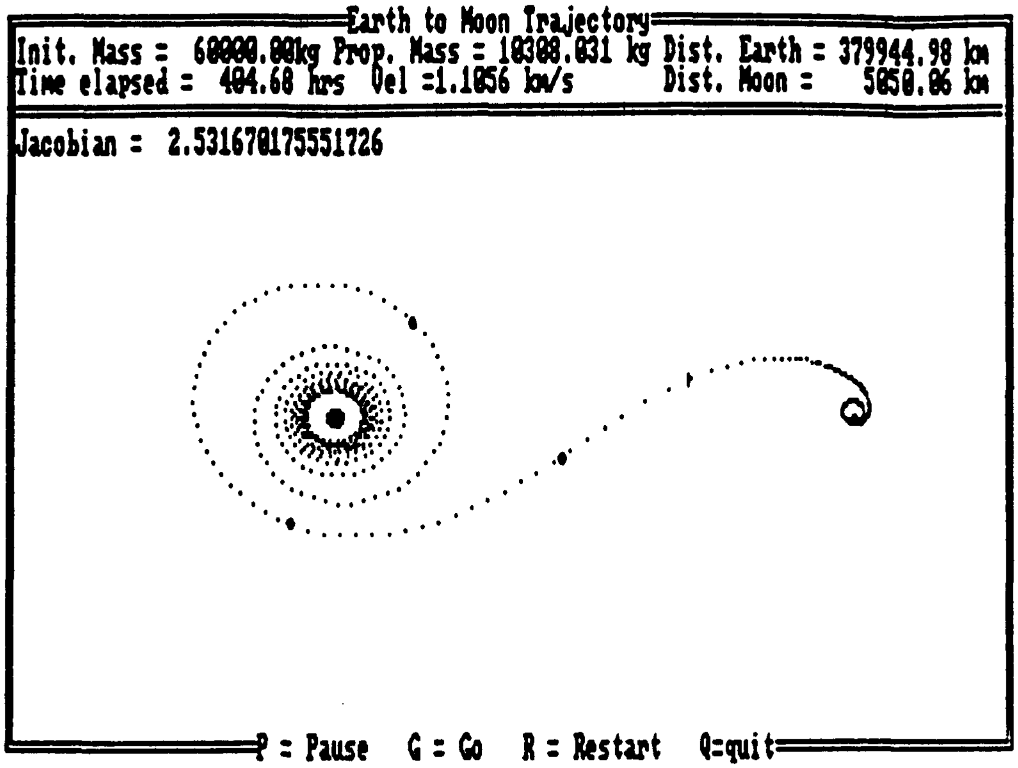
Four guidance controls are specified, Jac1, Jac2, Jac3, and Range. These four values govern the thrusting of the spacecraft during the final escape and translunar portions of the trajectory. Jac 1 indicates the spacecraft is nearing the end of its spiral escape from the initial orbit; the engines shut down, and thrusting ceases unless the spacecraft is in the proper quadrant for transfer injection. Jac 2 is the control that determines whether the spacecraft can achieve a cislunar trajectory. Ideally the Jacobian Constant at Jac 2 has a value of $3 \pm .2$ km/s. After reaching Jac 2, the spacecraft thrusts continuously. Jac 3 is the final constraint on the amount energy for the spacecraft to have during transfer. Following Jac 3, the spacecraft does not thrust. Range is the control that determines the distance from the initial planet that the capture guidance to the target planet is begun. This is the point at which reverse thrusting begins.

Markers for these four controls show up on the trajectory as each of them is passed. For Jac 1 - Jac 3, a small circle will indicate that this control has been reached. The passage of Range is indicated by a small vertical line. The visual representation of the controls is helpful to understand and plan a modification of the controls. The markers do not appear in FORTRAN versions of the program.

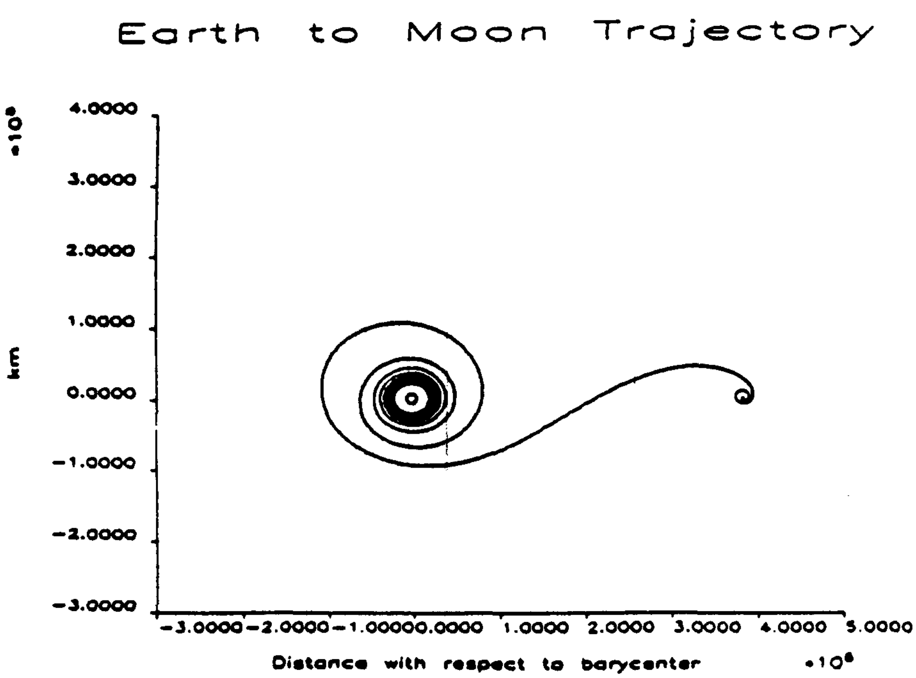
In the development of trajectories for low-thrust cislunar OTVs, little attention has been directed at the guidance and control of the spacecraft. The premise that the guidance of the vehicle and the determination of the appropriate trajectory are unrelated is false. Rather guidance and trajectory determination are closely related problems which by necessity must be treated with equal importance.

Figure 19.0-1, CISLUNAR Example Output

CISLUNAR Output: BASIC Version



CISLUNAR Output: FORTRAN Version



20.0 Earth to L2, L3, L4 and L5 and LLO to L1 and L2 and Back Trajectory Program (LIBRATE)

The program LIBRATE calculates velocities for trajectories from Low Earth Orbit (LEO) to four of the five known libration points (L2, L3, L4, L5), and from Low Lunar Orbit (LLO) to libration points L1 and L2. Libration Points (LP) are defined as locations in space that orbit the Earth such that they are always stationary with respect to the Earth-Moon line. Libration point #2 (L2) is located between the Earth and Moon where the gravitational attraction from both bodies are equal. L1 and L3 are located behind the Moon and Earth, respectively, such that the pull of the Earth and Moon together just cancel the centrifugal acceleration associated with the libration point's orbit. L4 and L5 are located half-way between the Earth and the Moon and 60° off the Earth-Moon line to the left and right, respectively. Hence, the Earth, Moon, and all libration points, lie in the same plane.

The flight to be analyzed departs from a circular orbit of any altitude and inclination about the Earth or Moon and finished in a circular orbit about the Earth at the desired libration point within a specified flight time. First, the departure orbit is made into a more eccentric orbit (ellipse or hyperbola) with an initial ΔV in order to reach the libration point while meeting the flight time constraint. The less the desired flight time, the more eccentric the orbit, and the larger the initial ΔV required. The least eccentric elliptic orbit would require the minimum ΔV and the maximum flight time. A second ΔV is then needed once the elliptic or hyperbolic flight path has reached the libration point in order to change the velocity vector of the eccentric trajectory to that of the libration point's orbit (circularize). So, the more eccentric the orbit, the larger the velocity change. This second burn must also account for the inclination of the eccentric trajectory with respect to the Earth-Moon-LP plane.

This program produces a matrix of the ΔV's needed to complete the desired flight. The user specifies the departure orbit (location and altitude), and the maximum flight time. A matrix is then developed with 10 inclinations (with respect to the Earth-Moon-LP plane), ranging from 0° to 90°, forming the columns, and 19 possible flight times, ranging from the flight time (input) to 36 hours less than the input value, in decrements of 2 hours, forming the rows. This matrix is presented in three different reports including the total ΔV's and both of the ΔV components discussed above. Table 20.0-1 shows example output.

LIBRATE was derived in part from the PLANECHG program, discussed in a different, more detailed documentation report. Therefore, for a more in-depth look at many of the equations, variables, and conventions used in LIBRATE, PLANECHG documentation may be consulted.

Table 20.0-1, LIBRATE Example Output

```
MAP OF TOTAL VELOCITY NEEDED FOR EARTH TRANSFER TRAJECTORY TO LIBRATION POINT #2
                    RUN DATE 23-SEP-88   RUN TIME 13:47:50
                    CIRCULAR ORBIT ALTITUDE   250.0
```

INCL> FLIGHT TIME (HR)	0.0	10.0	20.0	30.0	40.0	50.0	60.0	70.0	80.0	90.0
80.0	12167.	12180.	12226.	12299.	12394.	12507.	12631.	12762.	12895.	13026.
78.0	12190.	12206.	12251.	12323.	12417.	12528.	12652.	12781.	12913.	13044.
76.0	12223.	12238.	12282.	12353.	12446.	12556.	12678.	12806.	12937.	13066.
74.0	12264.	12279.	12323.	12392.	12484.	12592.	12712.	12839.	12968.	13096.
72.0	12316.	12330.	12373.	12441.	12531.	12637.	12755.	12880.	13007.	13133.
70.0	12378.	12392.	12434.	12501.	12588.	12692.	12808.	12930.	13055.	13179.
68.0	12453.	12466.	12507.	12571.	12657.	12758.	12871.	12991.	13113.	13235.
66.0	12540.	12553.	12592.	12655.	12738.	12836.	12946.	13063.	13183.	13302.
64.0	12642.	12654.	12692.	12752.	12832.	12927.	13034.	13148.	13265.	13381.
62.0	12758.	12771.	12807.	12865.	12941.	13033.	13136.	13247.	13360.	13473.
60.0	12891.	12903.	12938.	12993.	13067.	13155.	13254.	13361.	13471.	13580.
58.0	13042.	13053.	13086.	13139.	13210.	13294.	13390.	13492.	13598.	13704.
56.0	13212.	13223.	13254.	13304.	13371.	13452.	13544.	13642.	13744.	13846.
54.0	13403.	13413.	13443.	13491.	13554.	13631.	13718.	13812.	13910.	14008.
52.0	13617.	13627.	13655.	13700.	13760.	13833.	13916.	14005.	14098.	14192.
50.0	13857.	13865.	13892.	13934.	13991.	14060.	14138.	14223.	14312.	14401.
48.0	14124.	14132.	14157.	14196.	14250.	14315.	14389.	14469.	14554.	14639.
46.0	14422.	14430.	14453.	14490.	14540.	14601.	14671.	14747.	14827.	14907.
44.0	14755.	14762.	14784.	14819.	14866.	14923.	14989.	15060.	15135.	15211.

21.0 Earth to L1 and Return Trajectory Program (LP1)

The program LP1 calculates outbound and return trajectories between low earth orbit (LEO) and libration point #1 (L1). Libration points (LP) are defined as locations in space that orbit the Earth such that they are always stationary with respect to the Earth-Moon line. L1 is located behind the Moon such that the pull of the Earth and Moon together just cancel the centrifugal acceleration associated with the libration point's orbit.

The outbound flights depart from a circular orbit of any altitude and inclination about the Earth and culminate in a circular orbit about the Earth at libration point #1 within a specified flight time. The flight involves three burns.

First, the departure orbit is made into a more eccentric orbit (ellipse or hyperbola) with an initial ΔV in order to reach the lunar sphere of influence (SOI), a region where the vehicle is near its lowest velocity in the trajectory. The SOI is a spherical region whose surface normally includes all the points at a distance of 11% of the Earth-Moon distance from the Moon's center. However, in order to simplify the calculations this radius was increased to include L1, enlarging the sphere radius to 15% of the Earth-Moon distance. This change in SOI radius should not change the results significantly. A given flight may penetrate the SOI at a number of points identified by projecting lunar latitude and longitude onto the SOI. For each flight the program will calculate a set of possible trajectories associated with a set of SOI penetration points--a matrix of longitudes and latitudes.

Next, a second burn is calculated involving a "flyby" of the Moon from the SOI point above the front side of the Moon to L1 behind the back side. The SOI penetration point and L1 will always be the same distance from the center of the Moon. From the geometry of the trajectory it is apparent that the Lunar "flyby" perigee altitude, supplied by the user, will occur midway between the SOI point and L1. Consequently, the geometry of the orbit will force true anomaly, flight path angle, and absolute time to or from perigee passage to be the same for both the SOI and L1 points. There are two paths between these points, posigrade and retrograde. LP1 calculates only posigrade "flyby" trajectories since retrograde orbits that pass through the perigee altitude are not always possible while there is always a posigrade solution. The retrograde trajectories warrant more study in future work. Since L1 is constantly rotating with the Moon, this trajectory is iterated until L1 is reached.

The third burn is simply a circularization of the trajectory at L1 about the Earth. The velocity vector is corrected to that of L1. Once the SOI to L1 trajectory has been established, the Earth-SOI flight is iterated until the total transfer time, including the transfers from LEO to the SOI, and SOI to L1, match the user's flight time constraint (an input value). This is done for a matrix of SOI penetration points, as mentioned earlier.

The return trajectories, which start at L1 and finish in the specified LEO orbit within the specified flight time, are calculated similarly. For instance, the "flyby" trajectory is calculated first, starting at L1 and finishing at the SOI via a posigrade orbit calculated using the same geometric simplifications described above.

After the user has defined the trajectory as outbound or return, the Earth orbit altitude and inclination, and the total flight time, LP1 produces matrices which display the total ΔVs, the three component ΔVs (described above), the "flyby" trajectory inclinations, and the "flyby" azimuth angle at the SOI for the resulting flight from Earth to L1 for a representative set of SOI points. These points are defined by the user, who provides a starting longitude and latitude, and an increment for each. The matrix is built with 10 longitudes forming the columns and 19 latitudes forming the rows.

The documentation describes the input required from the user to define the flight and the contents of the six reports that are produced as outputs. Table 21.0-1 shows an example report.

LP1 was derived from the PLANECHG program (also produced under this contract) with the major addition of the FLYBY subroutine. Therefore, the documentation for PLANECHG may be used as a reference for many of the equations, variables, and conventions used in LP1 (except in the FLYBY routine).

Table 21.0-1, LP1 Example Report

```
VELOCITY MAP FOR EARTH TO L1 FLYBY TRANSFER TRAJECTORIES NODE OPTION 0
                        DATE  7-NOV-88   TIME  14:46:12

JUILIAN DAY 2451545.   PERIGEE ALT (NMI) EARTH = 250.  MOON = 10000.
TRANSLUNAR FLIGHT TIME (HR) = 200.0  INCL EARTH  25.0   INCL MOON PAGE3

ALON>     0.0    -5.0   -10.0   -15.0   -20.0   -25.0   -30.0   -35.0   -40.0   -45.0
ALAT
 90.0   14701.  14819.  14862.  14902.  14938.  14970.  14998.  15023.  15043.  15059.
 80.0   13866.  13863.  13863.  13866.  13872.  13881.  13893.  13907.  13924.  13943.
 70.0   13690.  13672.  13663.  13661.  13667.  13682.  13705.  13737.  13778.  13826.
 60.0       0.  13542.  13508.  13488.  13485.  13499.  13532.  13585.  13657.  13746.
 50.0       0.  13493.  13420.  13367.  13341.  13345.  13383.  13457.  13568.  13713.
 40.0       0.      0.      0.  13314.  13248.  13229.  13264.  13358.  13513.  13729.
 30.0       0.      0.      0.      0.  13211.  13157.  13180.  13290.  13492.  13790.
 20.0       0.      0.      0.      0.      0.  13126.  13130.  13248.  13492.  13876.
 10.0       0.      0.      0.      0.      0.  13125.  13111.  13231.  13505.  13953.
  0.0       0.      0.      0.      0.      0.  13143.  13118.  13234.  13514.  13985.
-10.0       0.      0.      0.      0.      0.  13173.  13147.  13255.  13516.  13955.
-20.0       0.      0.      0.      0.      0.  13216.  13198.  13294.  13517.  13884.
-30.0       0.      0.      0.      0.  13352.  13276.  13273.  13355.  13532.  13809.
-40.0       0.      0.      0.  13491.  13406.  13363.  13372.  13439.  13570.  13763.
-50.0       0.  13696.  13610.  13542.  13497.  13480.  13497.  13549.  13639.  13765.
-60.0       0.  13723.  13677.  13645.  13627.  13627.  13644.  13681.  13738.  13813.
-70.0   13844.  13820.  13804.  13794.  13791.  13797.  13811.  13834.  13865.  13904.
-80.0   13984.  13978.  13975.  13975.  13977.  13982.  13989.  14000.  14012.  14028.
-90.0   14836.  14881.  14923.  14962.  14998.  15030.  15059.  15083.  15104.  15121.

                     MINIMUM VELOCITY PRINT NODE OPTION 0
                        DATE  7-NOV-88 TIME  14:46:12

FLIGHT TIME = 200. LAT =   10.0 LON =  -30.0 LUNAR AINC =  10.0 EARTH AINC =  25.0
BODY    VX     VY     VZ      VEL   GAMA    AZM     AINC    ANODE  TIME  DVCIR
MOON    100.  2304.   -96.    900.  -38.3    90.0    10.0   240.3 109.8   1402. DVPHER =   1679.
EARTH   447.   672.  -279.    853.   35.3   114.9    25.0     8.5  90.2  10030. DVTOTAL = 13111.
```

22.0 Trajectory Analysis of Transfers from L4 and L5 to LLO

Libration Points 4 and 5 are, respectively, locations in space 60° behind and ahead of the Moon in its orbit. Theoretically, there is no tendency for an object to leave these locations relative to the Earth and Moon, and if displaced, the gravitational stability tends to return the object to the libration point.

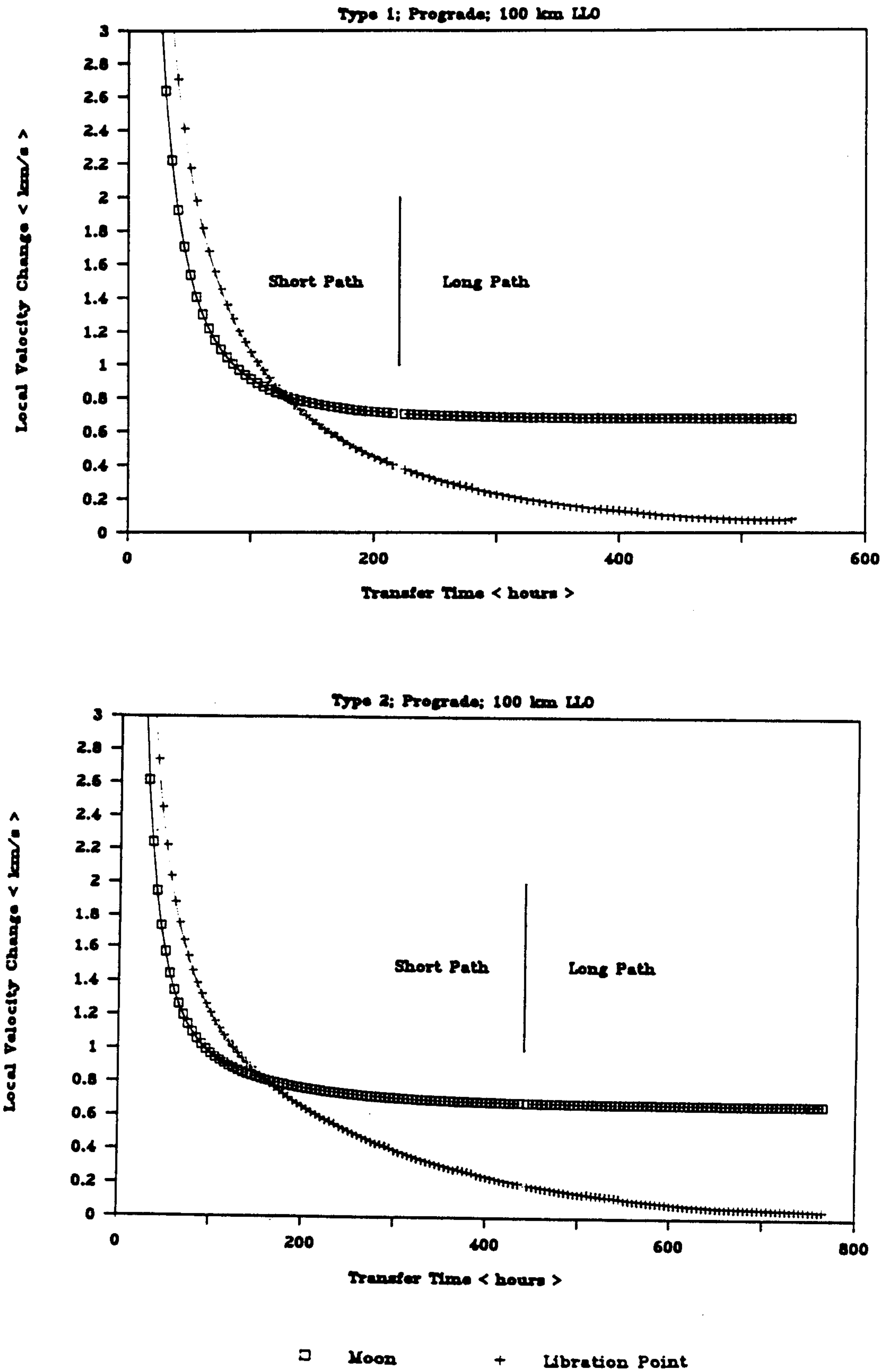
A Full Orbit Hohmann Transfer provides the most economical method of travelling between the Moon and these libration points. The flight time for this transfer is 395 hours (16.5 days) for flight from L4 to the Moon and from the Moon to L5 ("Type 1" flights). For flights from L5 to the Moon or from the Moon to L4 ("Type 2" flights), the flight time is 565 hours (23.5 days). The total velocity change required by the spacecraft is 757 m/s for the "Type 1" flights and 737 m/s for "Type 2" flights. For "Type 1" flights a 677 m/s velocity change is required at the Moon and a 80 m/s velocity change is necessary at the libration point. "Type 2" flights require 677 m/s at the Moon and 60 m/s at the Libration Point.

For short flight times Gauss' solution to Lambert's Problem is sufficiently accurate to predict the flight time and Delta V's. Figure 22.0-1 shows delta V plots for these shorter transfers. But for flight times approaching the period of a lunar orbit, the errors in the flight time make it necessary to use other solution techniques such as those presented by Broucke in his analysis of the Restricted Three Body Problem.

It is estimated that to fly to the Moon by way of L4 and L5 requires 758 m/s more Delta V than by direct transfer to LLO. Consequently, it appears unlikely that L4 and L5 will be used as transportation nodes for a lunar base.

If a program is to be written to perform this analysis, it is recommended that a Lambert Problem solution be used as the "driver" for converging an "N" Body Problem integration which would take into account the mass of the Moon and the perturbations of the Sun.

Figure 22.0-1, Delta Vs for Short Flight Times



23.0 List of All ASTS Contract Deliverables

1. Advanced Space Transportation System (ASTS) Support Contract, Summary Final Report, NASA Contract No. NAS 9-17878, Eagle Rep. No. 88-210, Oct. 30, 1988.

2. Lunar Surface Operations Study, NASA Contract No. NAS 9-17878, Eagle Rep. No. 87-172, Dec. 1, 1987.

3. Maintenance and Supply Options, NASA Contract No. NAS 9-17878, Eagle Engineering Rep. No. 87-173, May, 1988.

4. Lunar Lander Conceptual Design, NASA Contract No. NAS 9-17878, Eagle Engineering Rep. No. 88-181, March 30, 1988.

5. Lunar Base Launch and Landing Facility Conceptual Design, NASA Contract No. NAS 9-17878, Eagle Engineering Rep. No. 88-178, March 25, 1988.

6. Conceptual Design of a Lunar Oxygen Pilot Plant, NASA Contract No. NAS 9-17878, Eagle Engineering Rep. No. 88-182, July 1, 1988.

7. Lunar Storm Shelter Conceptual Design Summary, NASA Contract NAS 9-17878, Eagle Engineering, Rep. No. 88-189, May 1, 1988.

8. Space Transportation Nodes Assumptions and Requirements, NASA Contract No. NAS 9-17878, Eagle Engineering Rep. No. 87-174, April 18, 1988.

9. Transportation Node Space Station Conceptual Design, NASA Contract No. NAS 9-17878, Eagle Engineering Rep. No. 88-207, Sept. 30, 1988.

10. Lunar Surface Transportation Systems Conceptual Design Summary, NASA Contract No. NAS 9-17878, Eagle Engineering Rep. No. 88-188, July 7, 1988.

11. Conceptual Design of a Lunar Base Solar Power Plant, NASA Contract No. NAS 9-17878, Eagle Engineering Rep. No. 88-199, Aug. 14, 1988.

12. Lunar Surface Construction and Assembly Equipment Study Summary, NASA Contract No. NAS 9-17878, Eagle Engineering Rep. No. 88-194, September 1, 1988.

13. Lunar Base Applications of Superconductivity, NASA Contract No. NAS 9-17878, Eagle Engineering Rep. No. 88-218, Oct. 31, 1988.

14. Lunar Base Scenario Cost Estimates, NASA Contract No. NAS 9-17878, Eagle Engineering Rep. No. 88-211, Oct. 30, 1988.

15. Spacecraft Mass Estimation, Relationships, and Engine Data, NASA Contract No. NAS 9-17878, Eagle Engineering Rep. No. 87-171, April 6, 1988.

16. LLOFX, Earth Orbit to Lunar Orbit Delta V Estimation Program, User and Technical Documentation, NASA Contract No. NAS 9-17878, Eagle Engineering Rep. No. 88-212, April, 1988.

17. PLANECHG, Earth Orbit to Lunar Orbit Delta V Estimation Program, User and Technical Documentation, NASA Contract No. NAS 9-17878, Eagle Engineering Report No. 88-214, Sept. 20, 1988.

18. Lander Program Manual, NASA Contract No. NAS 9-17878, Eagle Engineering Report No. 88-195, Sept. 30, 1988.

19. CISLUNAR Program Manual, NASA Contract No. NAS 9-17878, Eagle Engineering Report No. 88-209, Sept. 30, 1988.

20. Velocity Deltas for LEO to L2, L3, L4 and L5 and LLO to L1 and L2, NASA Contract No. NAS 9-17878, Eagle Engineering Report No. 88-208.

21. LEO to L1 Trajectory Program, NASA Contract No. NAS 9-17878, Eagle Engineering Report No. 88-219, Oct. 30, 1988.

22. Low Lunar Orbit to L4, L5, and Back, NASA Contract No. NAS 9-17878, Eagle Engineering Report No. 88-216, Oct. 30, 1988.

23. Mars Rover/Sample Return Mission Requirements Affecting Space Station, NASA Contract No. NAS 9-17878, Eagle Engineering Report No. 88-183, March 31, 1988.

24. Risk Analysis of Earth Return Options for the Mars Rover/Sample Return Mission, NASA Contract No. NAS 9-17878, Eagle Engineering Report No. 88-196, July 13, 1988.