Lunar Surface Transportation Systems Conceptual Design

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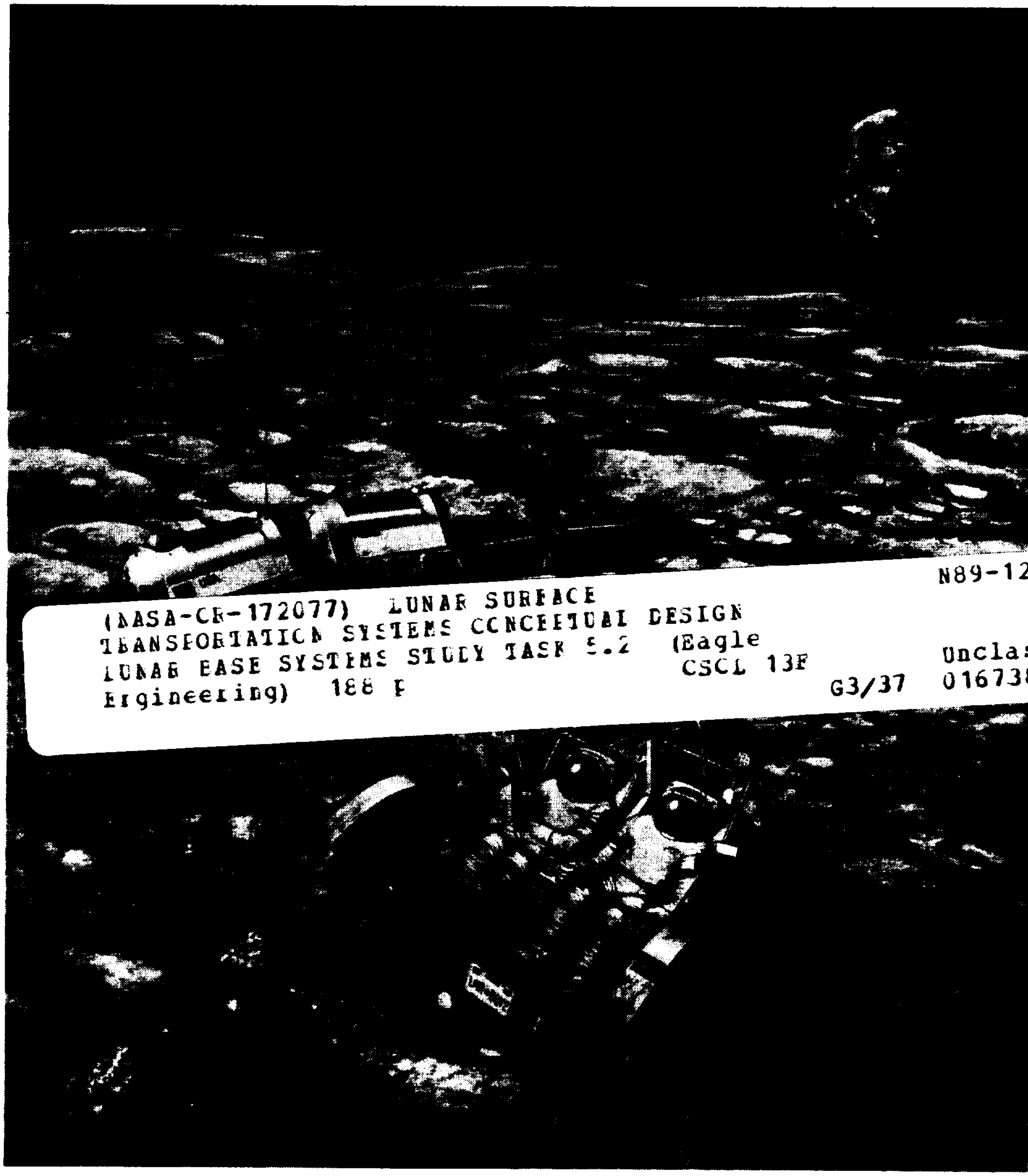

NASA Contract Number NAS9-17878
EEI Report 88-188
July 7, 1988

EAGLE

Lunar Surface Transportation Systems Conceptual Design
Lunar Base Systems Study Task 5.2

Prepared under NASA Contract NAS9-17878 for the
Advanced Programs Office
Engineering Directorate
NASA Johnson Space Center

By
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Houston, Texas
EEI Contract TO-87-57

Task 5.2 Report
EEI Report 88-188
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Foreword

The Lunar Surface Transportation Systems Conceptual Design Task was performed as part of the Advanced Space Transportation Support Contract which is a NASA Johnson Space Center (JSC) study intended to provide planning for a Lunar Base near the year 2000. The task personnel compared surface transportation system concepts; performed trade studies of range, power, payload, and operations of alternate design concepts; and developed conceptual designs. These surface transportation systems designs are necessary to facilitate an integrated review of a complete lunar scenario.

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List of Abbreviations

BALTRAN	Ballistic Transportation Vehicle
bps	bits per second
CASE	Checkout and Service Equipment (for EMU)
cg	Center of gravity
cm	centimeters
CNDB	NASA Headquarters Civil Needs Data Base
CPM	Critical Path Method (of Project Management Review)
DIPS	Dynamic Isotope Power System
DRM	Design Reference Mission
ECLSS	Environmental Control and Life Support System
EMU	Extravehicular Mobility Unit
EVA	Extravehicular Activity
E-Ascent	Expendable Lunar Ascent Stage
E-Lander	Expendable Lunar Lander Stage
ft	feet
ft-L	foot-Lamberts
HEDRB	High Energy Density Rechargeable Batteries
HUT	Hard Upper Torso
Hz	Hertz (cycles per second)
IVA	Intravehicular Activity
kg	kilograms
km	kilometers
kw	kilowatts
kwh	kilowatt-hours
LaRC	NASA Langley Research Center
lbs	pounds
LCVG	Liquid Cooling and Ventilation Garment
LEO	Low Earth Orbit
LH2	Liquid Hydrogen
LiOH	Lithium Hydroxide
LOTRAN	Local Transportation Vehicle
LO2	Liquid Oxygen
LRV	Apollo Lunar Roving Vehicle

List of Abbreviations
(Continued)

LSE	Lunar Surface Element
m	meter(s)
MET	Apollo Modularized Equipment Transporter
MOSAP	Mobile Surface Applications Traverse Vehicle
MSDB	Missions and Supporting Elements Data Base
MT	metric tons
N	Newton
NERVA	Nuclear Engine for Rocket Vehicle Application
OMV	Orbit Maneuvering Vehicle
OTV	Orbit Transfer Vehicle
OTV-A	Orbit Transfer Vehicle Flight with No Crew (Automated)
OTV-M	Orbit Transfer Vehicle Flight in Manned Configuration
RF	radio frequency
RTG	radioisotope thermoelectric generator
STN	Space Transportation Node
wh	watt-hours

Executive 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 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 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 suspension 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 1 Lunar Transportation Surface Vehicles Near Base

Figure 2 Lunar Transportation Ballistic Flight Vehicle At Pole

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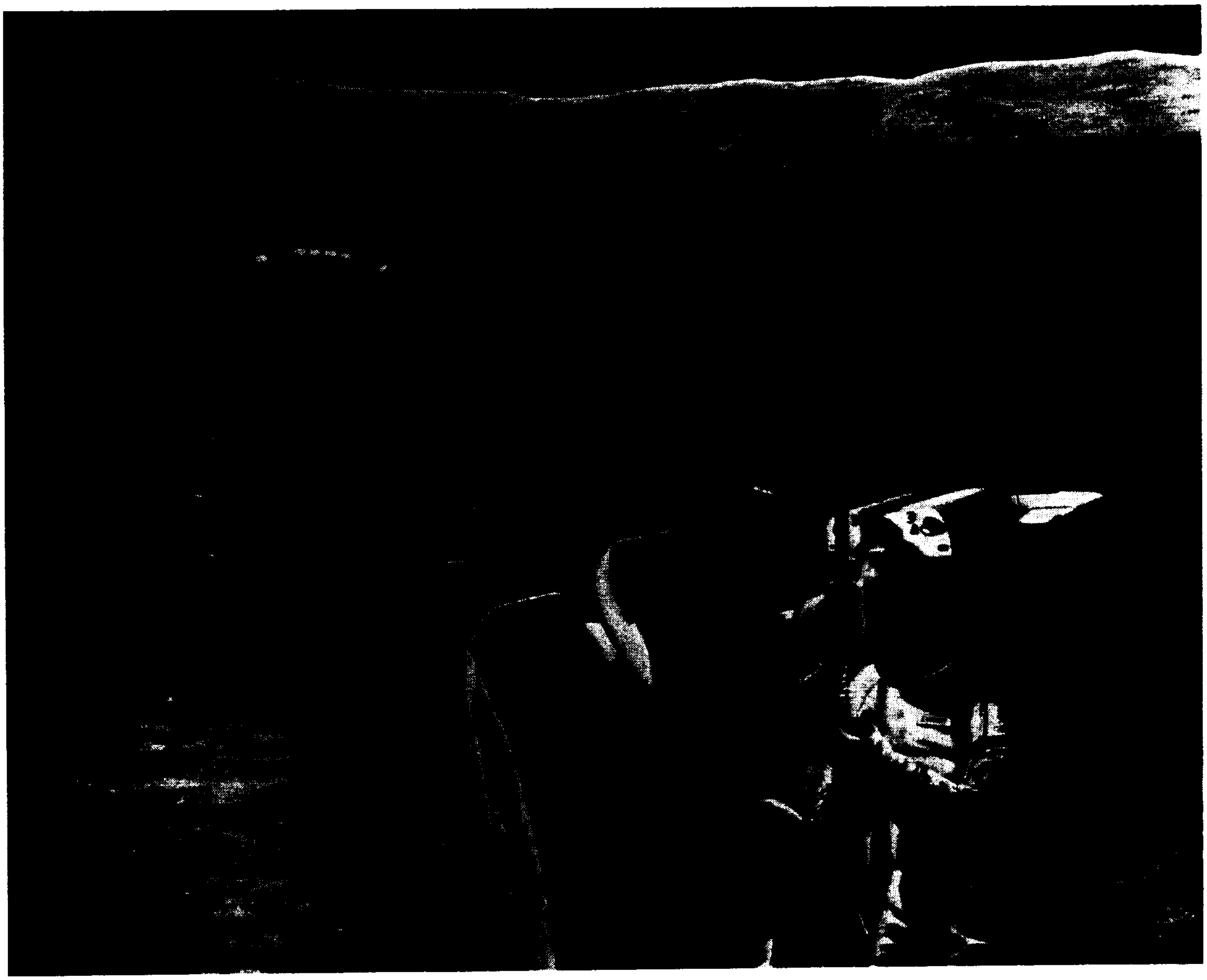


Lunar Surface Transportation Systems Conceptual Design

1.0 Introduction

The Advanced Programs Office of the NASA Johnson Space Center (JSC) has conducted a study to document conceptual designs of surface transportation systems.

1.1 Task Statement

The vehicle systems studied in this task relate to moving people and/or equipment to accomplish local base activities and long distance objectives. These systems can also be used to map and survey future mining and resource processing sites. Construction equipment activities will be addressed in a separate task and report.

The original task statement has been revised to specify the study scope as follows:

Determine environmental criteria necessary to evaluate trade study options. Survey and compare different concepts for surface transportation systems. Include discussions on locomotion methods, power systems options, surface lighting considerations, pressure vessel options, communications, and life support systems. Conceptual designs for a pressurized manned vehicle and an unpressurized manned/remote small vehicle will be defined. Conceptual engineering drawings of these concepts will be developed. Feasibility of a lunar flying vehicle for very remote sites should be addressed.

1.2 Task Organization

The task activities have been planned to compare different vehicle system approaches for implementing lunar transportation vehicles and to utilize the comparative analyses to provide three conceptual designs.

The strategy for the study is to provide analyses and designs which are applicable to current advanced program planning, but are not directed at mission targets so specific as to be invalid when program evolution changes mission definitions. The JSC Advanced Programs Office is defining a candidate Lunar Base Scenario. The transportation requirements of the generalized missions and flight schedules of the scenario have been studied and expressed in generic baseline definitions for study guidance. This reference baseline information for lunar surface transportation activities to be supported by vehicles in this study is provided in Section 2.0.

A small effort has been completed to survey earlier lunar surface transportation systems documentation. The findings of this survey are provided in Section 3.0.

The transportation vehicle system has been separated into thirteen topics for purposes of performing comparative analyses of the relative merits in alternative designs. At this embryonic stage of mission definition, the analyses generally identify advantages and disadvantages of certain features. Identification of the best design approaches must be deferred until later design iterations when more specific, integrated mission specifications are appropriate. Section 4.0 is the documentation of the vehicle systems comparison analyses.

A conceptual design for each of three different classes of lunar surface transportation was developed. The designs for the local vehicle; the longer range, pressurized surface application vehicle; and the lunar ballistic sortie vehicle provide definition of internal systems and general dimensions. The design information is developed in Section 5.0.

The closing summary and conclusions are provided in section 6.0.

2.0 Task Guidelines

2.1 Study Baseline Lunar Terrain Guidelines

This section defines the terrain parameters which affect the surface propulsion, navigation, and communications systems of vehicles moving on the lunar surface. The vehicles are assumed to operate in recent lunar sites of interest which can be characterized by data from previous landings and photos of other sites. Two of the four sites lie on flat mare surfaces surrounded by mountains (Lacus Veris and Taurus Littrow), one lies purely in flat mare (Nubium), and one is a rugged highlands region (South Pole). Data of the type and quality required to plan detailed traverses is available for only Taurus Littrow, site of the Apollo 17 landing. These data consist of Apollo 15 and Apollo 17 Pan camera pictures with a resolution of less than 5 meters (16 ft) and the metric camera pictures with a resolution of about 20 meters (66 ft). The latter was coupled to the laser altimeter and provides the best geodetic data base for the moon. Data on the Lacus Veris and Nubium landing sites is limited to Lunar Orbiter IV imagery with a resolution of 60-65 meters (197-213 ft). The imagery of the South Pole is limited to Lunar Orbiter IV images with most of the region in shadow. These images suggest that the South Pole is extremely rugged highlands terrain.

Because the lunar soil has a relatively constant bearing strength, mobility will not be constrained by the presence of unusually soft soil anywhere. The principal barriers that are expected are steep slopes and boulder fields at the rims of fresh craters, portions of the walls of rills, and parts of fault scarps. This section defines aspects relating the impact of terrain on lunar surface transportation vehicle design. These lunar terrain topics are: 1) the mixture of slopes likely to be encountered, 2) the presence of barriers to movement, 3) soil bearing strength, and 4) surface topography.

2.1.1 Surface Slope Distribution

Published slope data is available for all of the candidate Apollo landing sites as well as a large number of other areas of the moon [refs 48 and 70]. The data presented in Figures 2.1.1-1 through 4 and Table 2.1.1-1 comes from Moore and Tyler [48], and Wu and Moore [70]. Note that although there are extreme variations in the long wavelength portions of the slope spectrum, the shortest slopes wavelengths of 25 meters (82 ft) are relatively constant for the mare or highland plains at 4-6°. Figure 2.1.1-3 presents the slope data in a graphical manner which emphasizes the obvious: slopes are less on the mare than in the uplands or highlands, with the highland plains unit, the Cayley plains, being intermediate between the two. For example at Apollo 17, the landing site was in the flat mare floored valley with average slopes of 5-7°. In contrast, the slopes of the flanking North and South Massif have slopes of 20-30°. Figures 2.1.1-2, 3, and 4 show that the slopes over 20° make up less than 2% of even the most rugged regions on an areal basis. Thus it is possible to plan traverses to avoid the steep slopes. For example, the only major terrain impediment at the Apollo 17 landing site was the Lee-Lincoln Scarp. However, a pass with modest slope, the "Hole in the Wall," was identified in the Apollo 15 pictures and that pass was used by the Apollo 17 Lunar Rover Vehicle. In future missions, similar planning can essentially eliminate the limits on mobility due to steep slopes, assuming that adequate photography is available.

2.1.2 Barriers to Movement and Surface Roughness

The empirical observation of the Apollo program was that local surface roughness which might affect the mobility of a vehicle came exclusively from recent impacts, associated with the bright rayed craters. These events throw out large and small angular blocks for distances of several crater diameters. With time, these blocks are comminuted to fine lunar soil by micro-meteorite impacts which also darken the soil. The process is extremely slow by terrestrial standards, a few million years are required simply to round off the corners of the boulders several meters across such as those seen at Apollo 17 Station 6. The best documentation of an ejecta block field on the moon was that of South Ray Crater, a 2.5 million year old crater 0.5 km (1640 ft) across at the Apollo 16 landing site. Blocks from this event littered the south half of the landing site. South Ray was approached within seven crater diameters or 3.5 km (2.2 miles) where the blocks covered a few percent of the surface. Conditions probably become impassable only within one crater radius or 250 m (820 ft) from the rim. Only a very small number of craters are young bright rayed craters less than 100 million years old.

2.1.3 Soil Mechanics

The lunar surface consists of a fine grained soil with a significant amount of material finer than 0.05 mm (0.002 in). The fragments are mostly silicate mineral fragments and glass with a fraction of a percent metallic iron. The soil at all points studied in detail by Apollo, Surveyor, Luna, and Lunikhod spacecraft consisted of a porous zone a few centimeters thick at the surface which graded into progressively more and more compacted material with depth. Soil thickness is generally related to the age of the rocks nearest the surface. The older the rocks the thicker the soil. However, there is significant local variation in the thickness of the soil due to the presence of craters over a hundred meters across which penetrate into bedrock. In general, the soil layers are 2 to 5 meters (6.6 to 16.4 ft) thick on the mare. The soil in highlands areas lacks a well defined base because the bedrock consists of coarse rubble and breccias disrupted by craters tens of kilometers across.

The physical properties of the soil are dominated by its degree of comminution by micro-meteorites and its packing. Grain size effects and the abundance of small glass bound fragments called agglutinates play a more critical part in soil physical properties than chemical or mineralogical composition of the bed rock. Grain size and composition effects are in turn dominated with the effect of packing. The first observation from Apollo core samples is that the packing density is very loose at the surface and increases sharply in the top few centimeters. The second observation from Apollo core samples is that soil agglutinate content decreases and grain size increases with depth [Figure 2.1.3-1, from reference 46]. Craters which are surrounded by light colored material have sharp well defined rims and an abundance of blocks of bedrock. Near these fresh craters, the grain size of the soil is generally coarser than dark colored soils away from such craters. The process of destroying the blocks, comminuting the soil, and building up the agglutinate content is very slow. The young fresh crater, Cone, sampled by Apollo 14 is about 25 million years old. Tycho, the large bright crater readily visible from earth using a pair of binoculars, is thought to be about 75-110 million years old.

The definition of requirements placed on vehicles by the soil bearing strength and related factors should be treated generally for the entire moon since the dominating factors vary over a scale of several hundred. Table 2.1.3-1 summarizes the soil physical properties for the Apollo 14 through 17 landing sites and is taken from reference 46. For reference, an astronaut boot or the Apollo lunar module both place a stress on the surface of about a pound per square inch (0.69 N/cm^2 or 6.9 kN/m^2). Such stresses result in penetration of the lunar surface of less than a centimeter to a few centimeters. The angle of internal friction of lunar soil is also summarized in Table 2.1.3-1. The angle of

36° to 42° is equivalent to the angle of repose for loose soil such as on the side of a mountain. The tangent of the angle is equal to the coefficient of internal friction, 0.73 to 0.90. The cohesion of the soil is 0.01 to 0.1 N/cm^2 and like other properties increases with packing density and depth.

Data on the Apollo Lunar Roving Vehicle (LRV) indicates the amount of electrical power required to overcome the resistance of rolling over the moon. The Apollo LRV has a loaded mass of 708 kg (1,561 lbs). Figure 2.1.3-2 gives the power drawn from the LRV batteries. Using approximate numbers, the rover required 60 wh/km (1,800 wh over 28 km) on Apollo 15, 80 wh/km (2,880 wh over 35 km) for Apollo 17, and 100 wh/km (2,700 wh over 27 km) for Apollo 16. The higher power draw of the Apollo 16 mission reflects the highland terrain, which was more rugged than that traversed at Apollo 15 or 17.

2.1.4 Surface Topography

It is assumed that all traverses whether for science, resource exploration, or base logistical support will be preplanned to some extent. Initially, traverses will have to be planned and practiced with the thoroughness of Apollo J mission traverses. Once the operating characteristics of the vehicles are well known, planning more typical of terrestrial exploitations should be sufficient, where the crew need only be given a detailed traverse plan, a navigation system update of key reference points, and maps showing the planned traverse. Such a level of planning is sufficient to eliminate the possibility of having the traverse plan affected by insurmountable scarps or dense boulder fields which require a slow circuitous path around the obstructions.

Navigation within a few kilometers of the base is easily accomplished using landmark tracking, probably supplemented by data derived from line of sight communication between the base and the transportation vehicle. Planning traverses of significant distance is greatly enhanced by knowing what the terrain will be like in advance. Such data is typically recorded on topographic maps whether in hard copy or digital format. The data which is needed includes both contour lines, displaying the elevation and slope data, and data on the presence of small scale features such as ejecta from fresh young craters. Navigating traverse vehicles will certainly be done relative to landmarks on the ground, whether the vehicle is controlled by a human driver or some type of automated system. Furthermore, the detailed planning of traverses requires maps of sufficient quality to identify slopes which exceed the capabilities of the vehicle or areas with blocky ejecta from recent craters which would require a serpentine traverse path around the blocks. In essence, operating traverse vehicles will require the same quality data used for similar activities on earth such as geological surveys in remote wilderness areas. Those data are equal to those required to produce topographic maps approximately the quality of the standard 1:24000 scale maps available for most of the United States from the U.S. Geological Survey. Such maps have all points located laterally within 61 m (200 ft) and vertically within about 3 m (10 ft) in areas of low relief such as mare. The maps will certainly have to be prepared by photogrammetric techniques with the map locations tied together with a benchmark system. Such a system would have a small number of positions known with great precision and accuracy and a far larger number of positions known to a lower lever of precision. The requirements are different from those required for landing sites because the absolute geodetic reference frame is not particularly significant for traverse vehicles. It is only the relative elevation differences of points (bench marks) that must be established within a few feet. These requirements imply the existence of data of a type that exceeds that defined for the Lunar Geoscience Observer. The amount of territory that must be accurately imaged is only that accessible or visible to the traverse vehicles.

Figure 2.1.1-1 Comparison Between Algebraic Standard Deviation and Mean Absolute Slopes for Lunar Slope-Frequency Distributions (from Ref. 48)

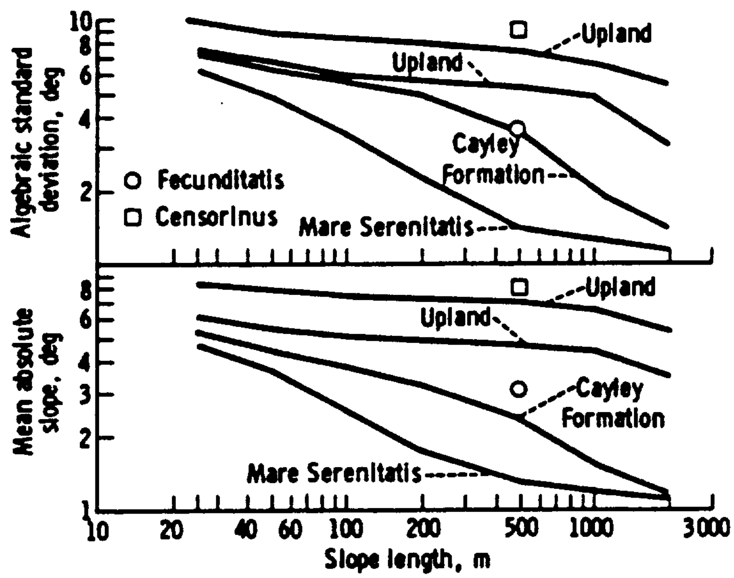
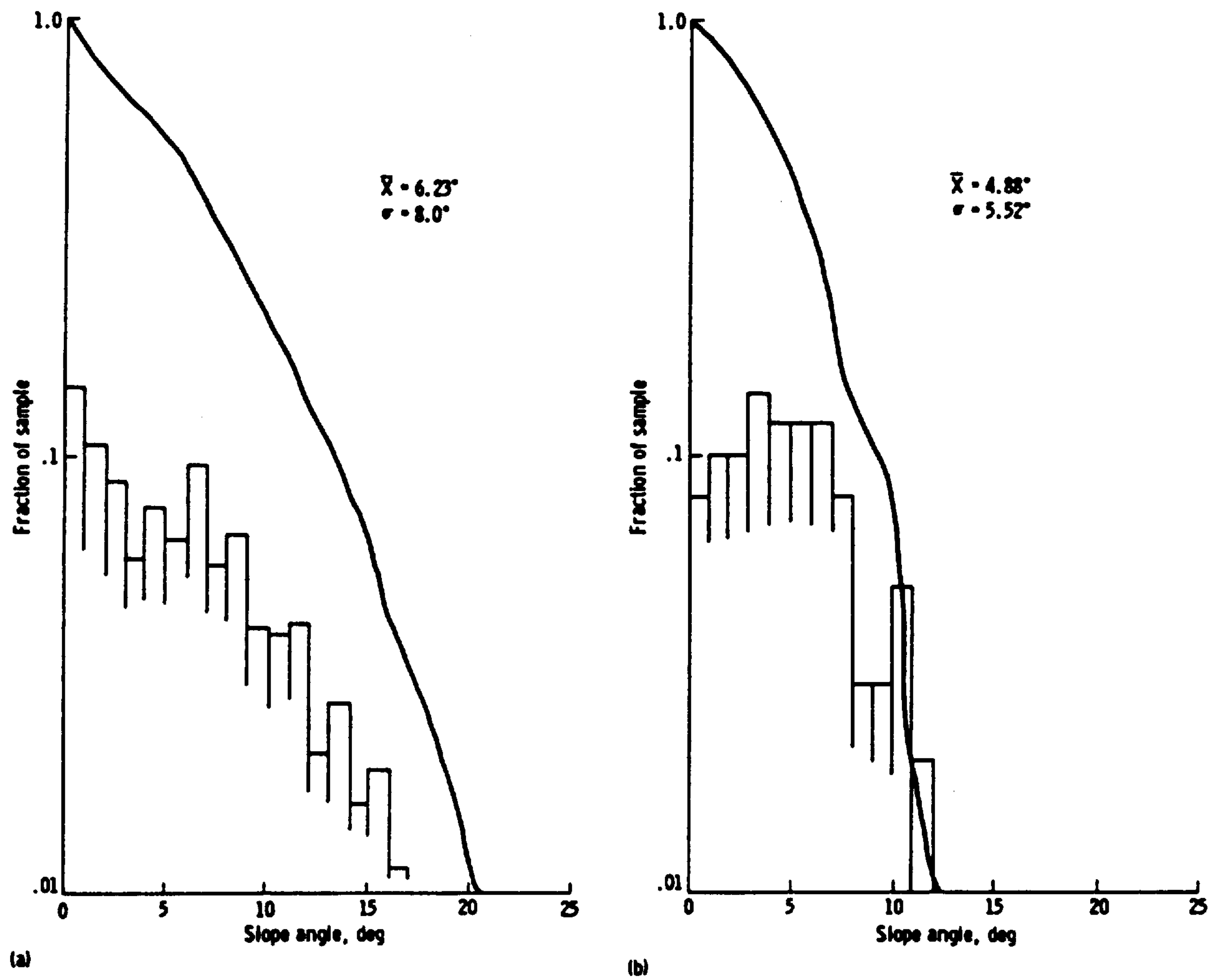
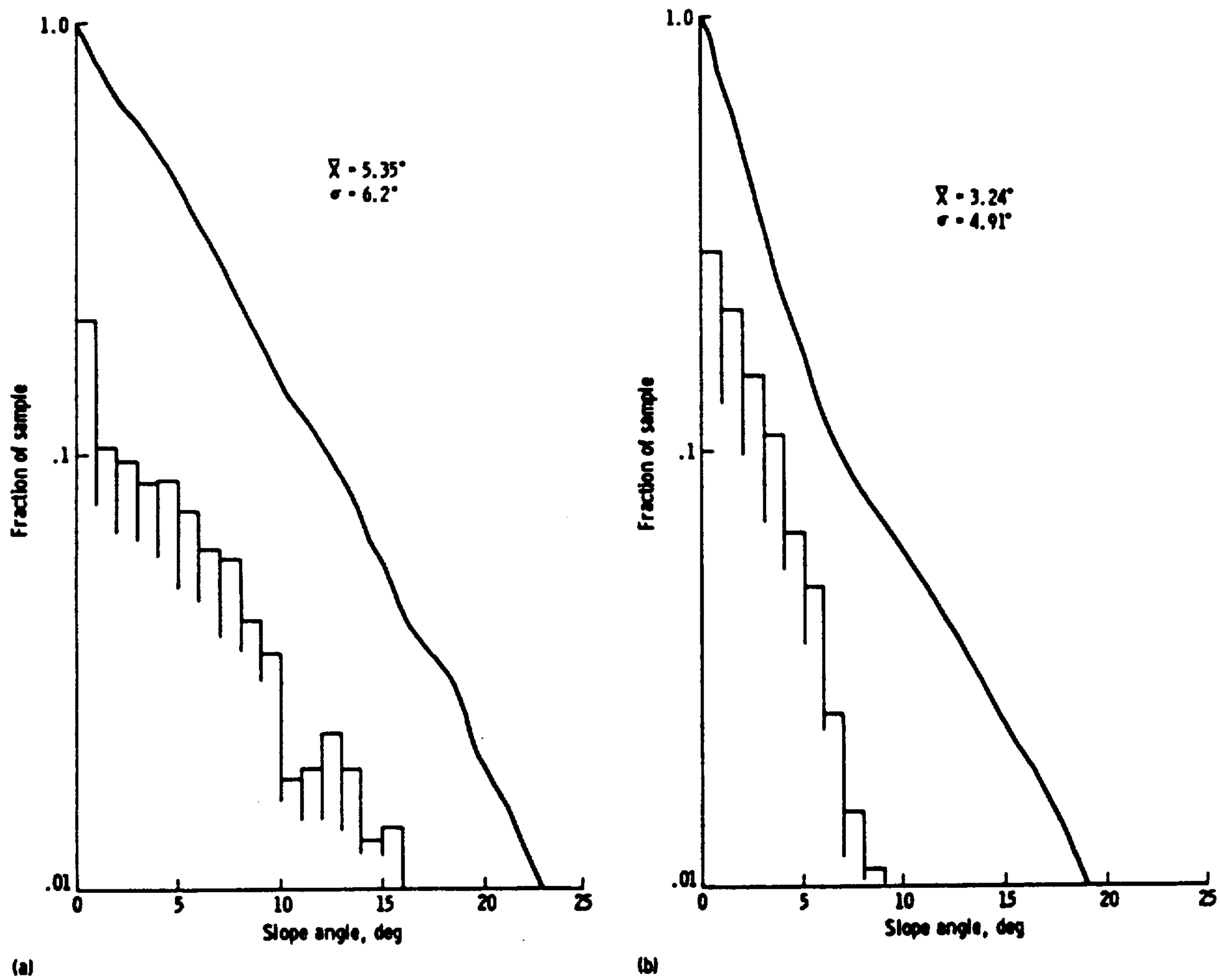


Figure 2.1.1-2 Upland Distribution of Slope Values, North of Vitruvius (from Ref. 70)



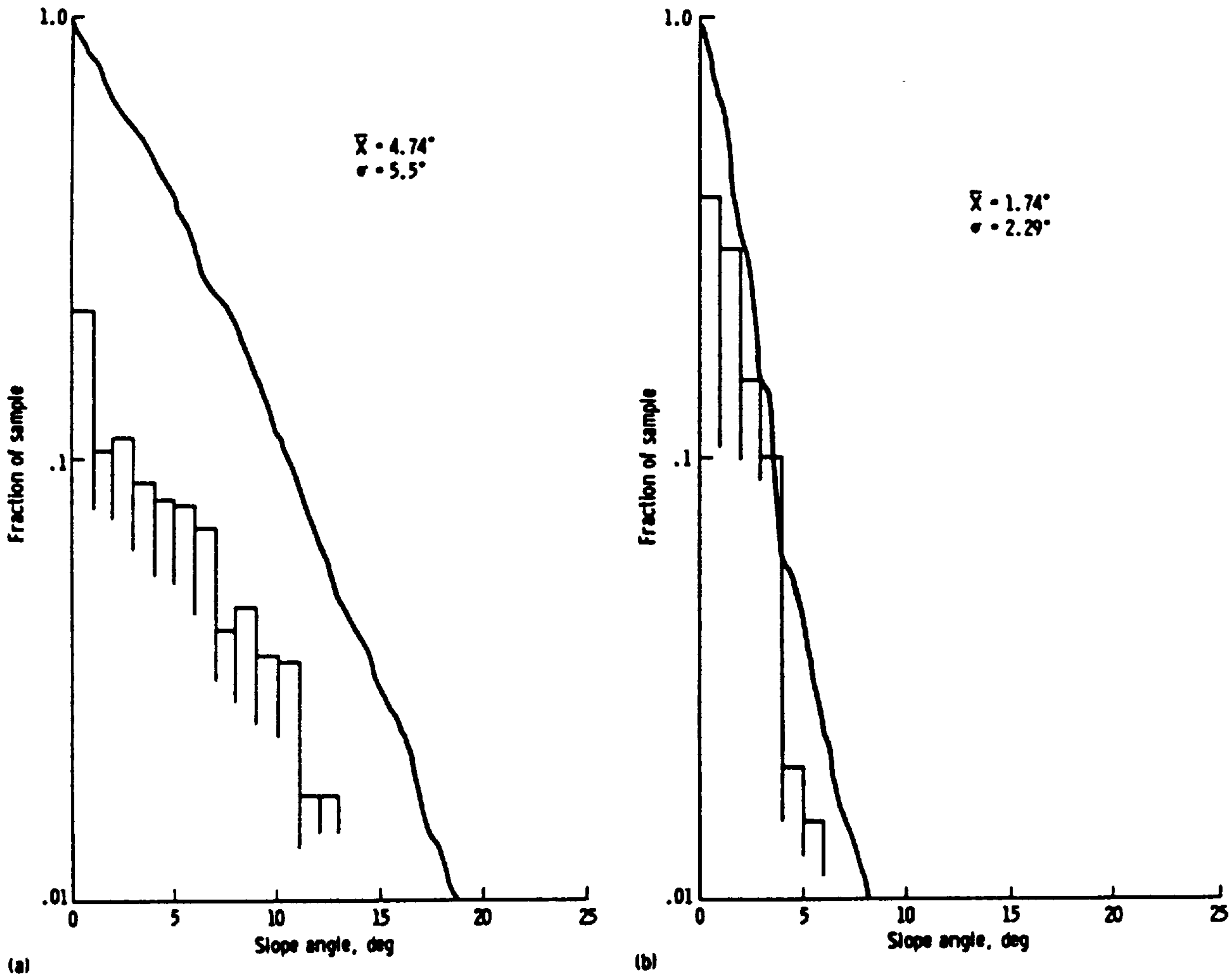
Frequency distribution of absolute slope values for upland surface north of Vitruvius. Bars represent fraction of sample for 1° increments of slope angles. Solid lines indicate cumulative fraction of sample with absolute slopes larger than angle indicated. The quantity X is the mean absolute slope and σ is algebraic standard deviation of distribution. (a) Slope length ΔL 25.2 m. (b) Slope length ΔL 201 m.

Figure 2.1.1-3 Upland Distribution of Slope Values, Cayley Plain at Apollo 16 Landing Site (from Ref. 70)



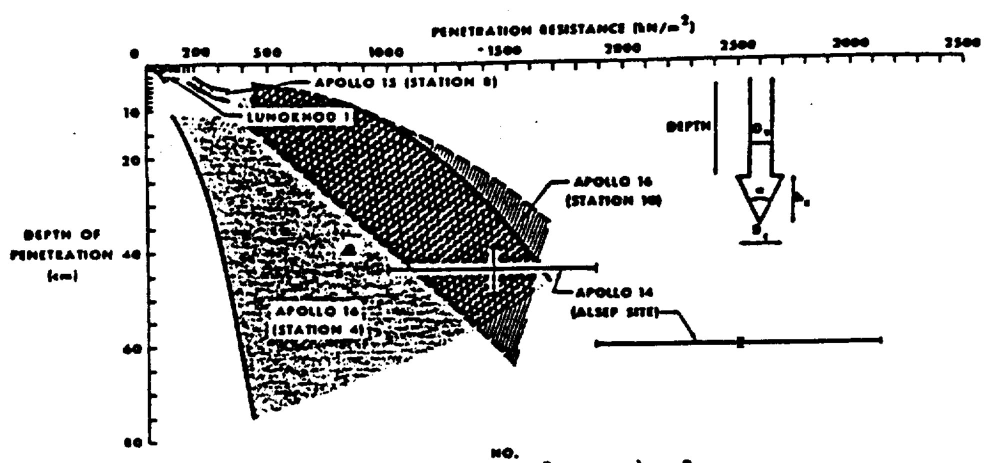
Frequency distributions of absolute slope values for Cayley Plain at the Apollo 16 landing site. Bars represent fraction of sample for 1° increments of slope angle. Solid lines indicate cumulative fraction of sample with absolute slopes larger than angle indicated. The quantity $\bar{X}$ is mean absolute slope and σ is algebraic standard deviation. (a) Slope length ΔL 25.1 m. (b) Slope length ΔL 201 m.

Figure 2.1.1-4 Mare Distribution of Slope Values, Mare Serenatatus (from Ref. 70)



Frequency distribution of absolute slope values for surface in Mare Serenitatis. Bars represent fraction of sample contained in 1° increments of slope angle. Solid lines indicate cumulative fraction of sample with absolute slopes larger than angle indicated. The quantity $\bar{X}$ is mean absolute slope and σ is algebraic standard deviation of distribution. (a) Slope length ΔL 25 m. (b) Slope length ΔL 200 m.

Figure 2.1.3-1 Penetration Resistance of Lunar Surface at Various Locations (from Ref. 46)



MISSION	NO. OF TESTS	D_c (cm)	α	h_c (cm)	D_t (cm)
APOLLO 14	3	0.95	30°	1.78	0.95
LUNOKHOD 1	327	3.8	60°	4.4	-
APOLLO 15	2	2.03	30°	2.78	0.94
APOLLO 16 (STATION 4)	1	2.03	30°	2.78	0.94
	3	1.28	30°	2.29	0.94
APOLLO 16 (STATION 10)	1	2.03	30°	2.78	0.94
	3	1.28	30°	2.29	0.94

Table 2.1.1-1 Slope Data for Lunar Surface as a Function of Length of Segment Measured (from Ref. 70)

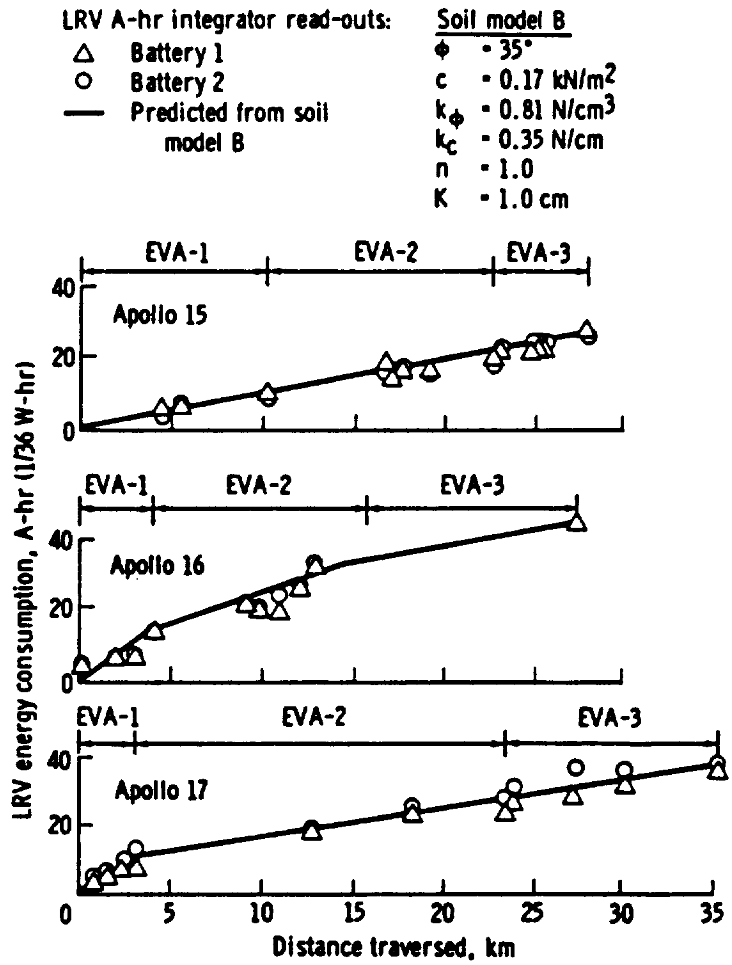
Terrain type	*Algebraic standard deviation, deg*						*Mean absolute slope, deg*					
	ΔL 25 m	*ΔL 50 m*	*ΔL 100 m*	*ΔL 200 m*	*ΔL 500 m*	*ΔL 1000 m*	*ΔL 25 m*	*ΔL 50 m*	*ΔL 100 m*	*ΔL 200 m*	*ΔL 500 m*	*ΔL 1000 m*
Mare Serenitatis	5.8	4.8	3.4	2.3	1.4	1.2	4.7	3.7	2.5	1.7	1.3	1.2
Mare Serenitatis	6.1	4.9	3.6	2.5	1.3	.8	4.8	3.8	2.8	1.8	1.0	.7
Cayley Plain	6.2	6.3	5.6	4.9	3.4	1.5	5.4	4.5	3.8	3.2	2.3	2.0
Uplands, near Proclus	6.4	6.0	5.0	4.3	3.2	2.2	5.4	4.6	3.9	3.3	2.6	2.1
Uplands, north of Vitruvius	7.8	6.6	5.9	5.5	5.2	4.7	6.2	5.5	5.1	4.9	4.7	4.3
Uplands, near Glaisher	9.9	8.8	8.2	7.8	7.3	6.7	8.4	7.9	7.5	7.3	6.9	6.4
Mare Fecunditatis	–	–	–	–	3.6	2.6	–	–	–	–	3.1	2.0
Uplands, near Censorinus	–	–	–	–	10.5	9.2	–	–	–	–	7.8	6.3
Littrow, landing site	4.5	–	–	–	–	–	3.8	–	–	–	–	–
Littrow, west of landing site	4.6	–	–	–	–	–	3.9	–	–	–	–	–
Hadley, landing site	6.8	–	–	–	–	–	5.7	–	–	–	–	–

Table 2.1.3-1 Average Material Properties of Surficial Lunar Soil at Apollo 14-17 and Luna Landing Sites (from Ref. 46)

Soil consistency	G,[a] N/cm^3	*Porosity, percent*	*Void ratio, e*	D_r,[b] *percent*	ϕ_{TR},[c] *deg*	ϕ_{PL},[d] *deg*
Soft	0.15	47	0.89	30	38	36
Firm	0.76 to 1.35	39 to 43	0.64 to 0.75	48 to 63	39.5 to 42	37 to 38.5

[a] G = penetration resistance gradient.
[b] D_r = relative density = $(e_{max} - e)/(e_{max} - e_{min})$, based on standard American Society for Testing Materials methods.
[c] ϕ_{TR} = angle of internal friction, based on triaxial compression tests.
[d] ϕ_{PL} = angle of internal friction, based on in-place plate shear tests.

Figure 2.1.3-2 Measured Energy Consumption of the Apollo LRV as Compared to Predicted Values Based on the Soil Properties Indicated (from Ref. 45)



2.2 Study Baseline Mission Guidelines

In order to develop conceptual designs of lunar surface transportation vehicles, guidelines are required which baseline the functional vehicle performance required to accomplish the anticipated missions. This section defines a generic baseline for the mission objectives to be achieved during lunar traverse missions. Activity and equipment requirements necessary to implement the objectives are described in order to derive payload and vehicle definition parameters. Several baseline traverse missions are defined that accomplish the majority of the mission objectives. Finally, transportation vehicle functional performance requirements are specified as a baseline for guiding the conceptual designs of the vehicles.

2.2.1 Non-Base Surface Mission Objectives

Surface traverses away from the base will attempt to accomplish many objectives. Primary among these are to study the structure, tectonism, cratering history, petrology, mineralogy, stratigraphy, age, developmental history, resources, and morphology of the lunar surface and crust. These studies will provide a better understanding of planetary geological evolution and solar system development. In addition, geological data that is necessary to effectively utilize lunar resources for such activities as oxygen production, construction, and manufacturing will be developed.

Success of the mission objectives will depend on the ability to perform experiments at geographically diverse locations. Some activities will occur over contiguous surface features, while others will concentrate on a single feature. Some experiments will require activities to be performed at specific locations remote from the lunar base, while others can be performed near the base. Features that will be of interest include craters, rim deposits, ejecta blankets, rills, fault scarps, volcanic complexes, mare regions, highland regions, and mountains.

2.2.2 Payload Equipment Requirements

It is assumed that, for local traverses within kilometers of the lunar base, samples and data will be collected during the traverse and returned to the base for analysis. For longer traverses (hundreds of kilometers and several weeks or more), it may be more effective to perform the analysis at the collection site and leave most of the samples behind. A list of potential tools and equipment required to perform three categories of surface activities has been compiled [reference 3]. The data are summarized in Table 2.2.2-1 and discussed in the following paragraphs.

Surface sample collection will require such tools as rock hammers, tongs, rakes, scoops, shallow drills, core tubes, sample collection bags, and sample storage boxes. These tools occupy approximately 0.3 cubic meters (10.6 ft^3), have a mass of approximately 80 kilograms (176 lbs), and require about 0.5 kilowatts of power when used.

Selenophysical experiments will assist in mapping the seismic, magnetic, and electrical properties of the subsurface and its density variations. Equipment for these experiments could include profiling active seismic arrays, thumpers, explosive packages, a magnetometer, a gravimeter, and an electrical properties experiment package. This equipment occupies approximately 0.4 cubic meters (14.2 ft^3), has a mass of approximately 650 kilograms (1,433 lbs), and requires about 0.1 kilowatts of power when used.

Equipment for selenology exploration could include cameras, film, a stadiametric range finder, a sun compass/azimuth indicator, an inclinometer, and a trenching tool. This equipment occupies approximately 0.3 cubic meters (10.6 ft^3), has a mass of approximately 150 kilograms (330 lbs), and requires about 0.5 kilowatts of power when used.

2.2.3 Mission Definitions

Three baseline mission types illustrate most of the scenarios that a lunar surface transportation vehicle will encounter. These are a local traverse, a long-range surface applications mission, and a sortie to a remote location to accomplish a localized mission.

2.2.3.1 Local Transportation Mission

This mission would use an unpressurized vehicle for deploying experiments, collecting samples, surveying, and transportation near the lunar base. As many as four personnel would be transported. Teleoperation of the vehicle would allow completion of simple errands without requiring crew EVA. Its operating range would be constrained by the distance it could travel out from the base and back in one work day. Total EVA time per day per crewman is assumed to be about eight hours. Assuming a minimum desired productive mission work time of one hour, maximum driving time would be seven hours per trip. The vehicle for this mission is designated the Local Transportation Vehicle (LOTRAN).

2.2.3.2 Long-Range Surface Applications Traverse Mission

Trips to conduct lunar surface science and utilization applications require travel at long ranges from the lunar base. This type of mission would last from several days to many weeks and, thus, would require a pressurized vehicle. Activities performed during this mission would include surface and deep drill sample collection, prospecting, surveying, and the deployment of geophysical experiments over one or more geographical features. The mission would be constrained by the size of the feature or features to be explored, and could range for hundreds of kilometers. Such a long duration would require the vehicle to combine the features of a habitation module and a laboratory in the form of a mobile transportation vehicle.

Many of the surface activities, such as sample collection and drilling, could be performed in a teleoperated mode from inside the vehicle. Other activities, such as equipment deployment, surveying, and collection of hard-to-access samples, would require EVA.

Four crewmen are planned for the long-range surface applications missions. Using rotating crew shifts, the vehicle would be driven for up to 12 hours per Earth day. The vehicle for this mission is designated the Mobile Surface Applications Traverse Vehicle (MOSAP).

2.2.3.3 Remote Site Ballistic Flight Mission

During this mission, a team of astronauts would fly from the base to a remote location, and perform surface applications activities within five to ten kilometers (3.1 to 6.2 miles) of the landing site. This would require a vehicle capable of ballistic flight, soft landing, and return. A rover type vehicle could be attached to the ballistic vehicle, and deployed at the landing site.

A contingency mission for this vehicle would be to rescue crewman from the MOSAP if required. Potential events which could require MOSAP crew pickup are MOSAP failure or a solar flare event.

The mission duration would be on the order of days. Crew size would be up to five astronauts. Unlike the Apollo missions, it would be desirable to return all tools and equipment to the point of origin, in this case the lunar base. The vehicle for this mission is designated the Ballistic Transportation Vehicle (BALTRAN).

2.2.4 Vehicle Functional System Requirements

Based on the development of the baseline mission guidelines, the vehicle functional performance requirements have been identified and documented in Table 2.2.4-1. For the vehicle to be used in each of the three types of missions, functional performance requirements are tabulated as "Required" or "Desired".

Table 2.2.2-1 Potential Mobile Surface Applications Payload Equipment

PAYLOAD TYPE	PAYLOAD EQUIPMENT TYPE	MASS (kg)	VOLUME (cm^3)
Surface Sample Collection (LSE-001)	Tongs	1.8	3,181
	Rock Hammer	1.3	1,984
	Rake	1.5	6,000
	Scoop	0.4	141
	Drive Tools	0.9	1,852
	Shallow Drill	22.7	25,017
	Core Tubes	10.8	16,380
	Sample Bags	1.4	182,400
	Sample Boxes	23.6	55,680
	Rock Drill	18.3	3,393
	TOTAL	82.7	296,028
Selenophysical Experiments (LSE-003)	Deep Seismic Arry	25.0	18,750
	Explosive Pkg	584.1	296,000
	Shallow Seis Arry	3.0	1,904
	Thumper	6.0	2,825
	Magnetometer	4.6	11,760
	Gravimeter	5.0	18,000
	Electrical Prop.	10.0	39,167
	Hi Freq Magnetmtr	10.0	18,000
	Solar Wind Exp	6.0	30,380
	TOTAL	653.7	436,786
Selenology Exploration (LSE-006)	Sampling Equip.	63.8	289,186
	Film Cameras	75.0	4,500
	Film	10.0	3,704
	Surveying Equip.	1.4	2,000
	Trenching Tool	1.3	1,125
	Inclinometer	0.3	19,154
	Sun/Gyro-compass	0.5	125
	TOTAL	152.3	319,794

Table 2.2.4-1 Lunar Surface Transportation Vehicle Functional System Requirements

FUNCTIONAL PERFORMANCE DESCRIPTION	VEHICLE REQUIREMENTS					
	LOTRAN		MOSAP		BALTRAN	
	Req.	Desired	Req.	Desired	Req.	Desired
Crew Size	4		4		3	5
Max Range from Base (km)	50		500	1500	1500	5464
Max. Total Travel Dist (km)	100		1000	3000	3000	10928
Max. Mission Duration (hrs)	8		336	1000	200	
Gross Payload (kg)	850		2000		1000	
Max. Velocity (km/hr)	15		10	15		
Night Operations Limitations	Within sight of base		None, except drive slower		None	
Pressurized (psia)	No		8-10		8-10	
EVA Events	N/A		12	24	12	
Communications	Contin. Voice & Data		Contin. Voice & Data		Contin. Voice & Data	
Teleoperation/ Automatic Mode		Yes		Yes	Yes	
Remote Manipulator Sys.		1	1		N/A	

3.0 Lunar Surface Transportation Systems Survey

3.1 References

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What you can not find in Lunar Geoscience you can find in this lunar handbook. If it is not in either, it probably has not been done yet.

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Good discussion of crater distribution and visibility while driving. Old document.

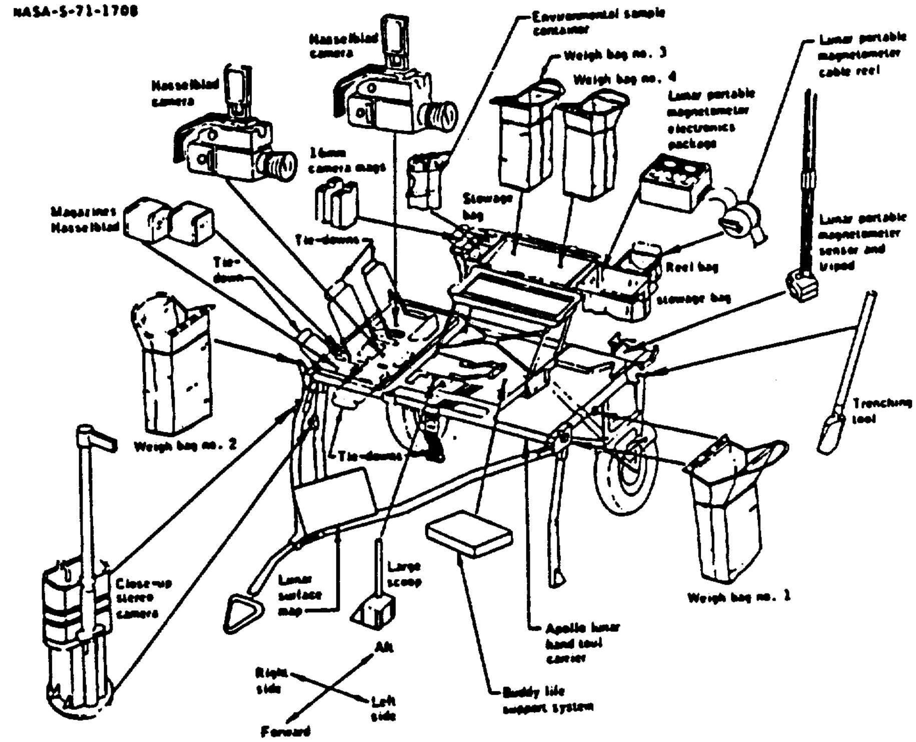
3.2 Reference Matrix

REPORT SECTION	REFERENCE ITEM NO. DIRECTREFERENCE	GENERALREFERENCE
2.1	45, 46, 48, 70	15, 29, 34, 47
2.2		3-6, 8-12, 16, 18, 31, 38, 41, 50, 60, 64
4.2	15	2, 7, 14, 21, 23, 24, 27, 28, 32, 33, 35, 37 39, 51, 52, 54, 55, 56, 59, 63, 66, 69, 70
4.3	13	
4.4		40, 53, 58
5.2	17, 40, 61	
5.3	64	

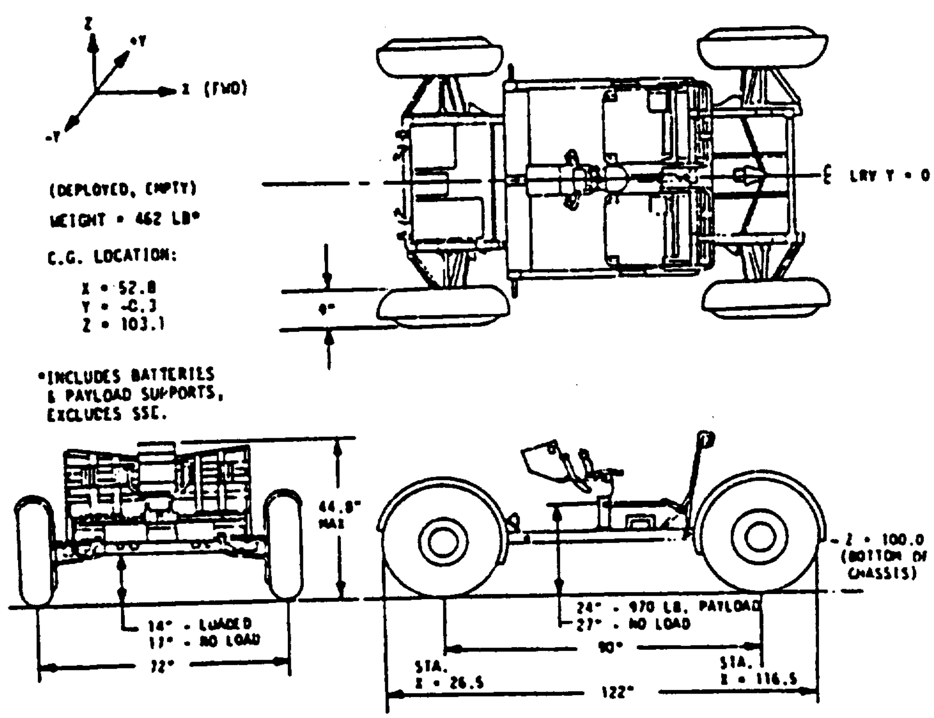
3.3 Pictorial Summary

A good method for describing mobility concepts briefly is to provide illustrations and pictures for easy review. The graphic material in Figure 3.3-1 is presented as a sample of previous planetary surface locomotion studies and designs. The data are presented in no preferential order. In addition, the illustrations from all previous work are not necessarily included in the samples presented.

Figure 3.3-1 Samples of Previous Planetary Surface Locomotion Design Illustrations



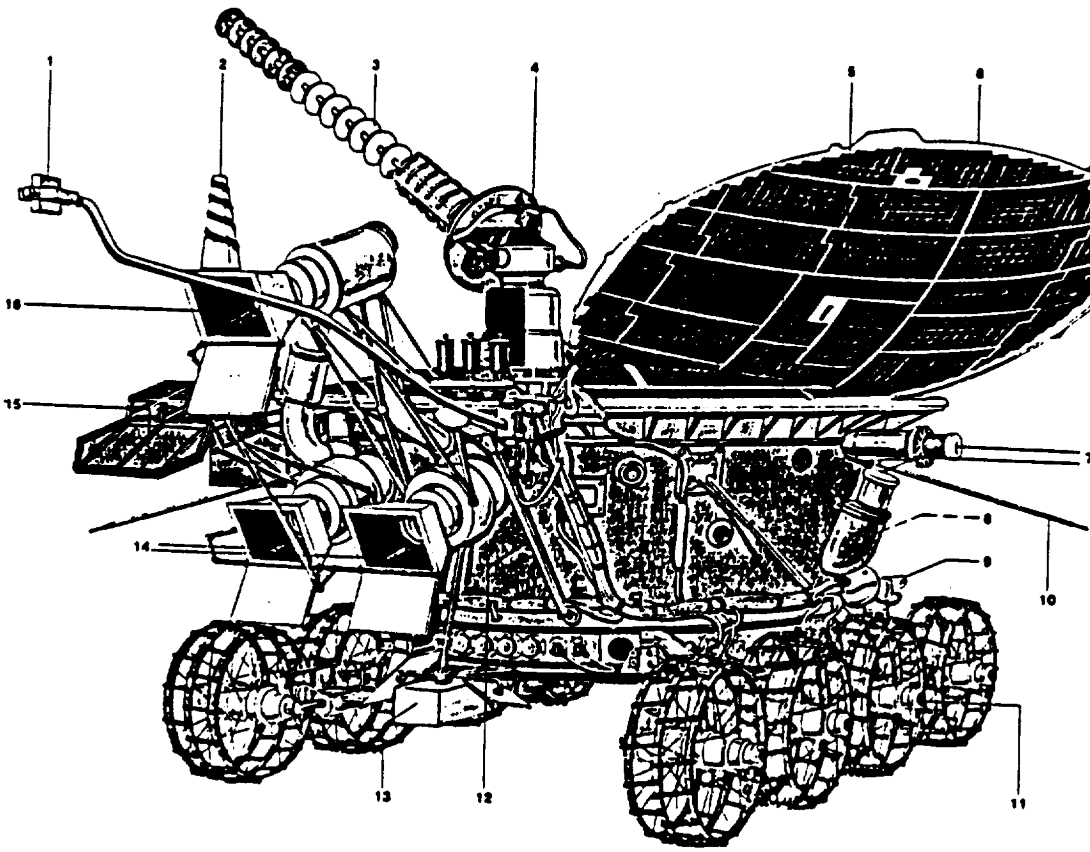
Modular equipment transporter and equipment.



b. MOBILITY SUBSYSTEM DIMENSIONS

Lunar Roving Vehicle.

Figure 3.3-1 Samples of Previous Planetary Surface Locomotion Design Illustrations (Continued)

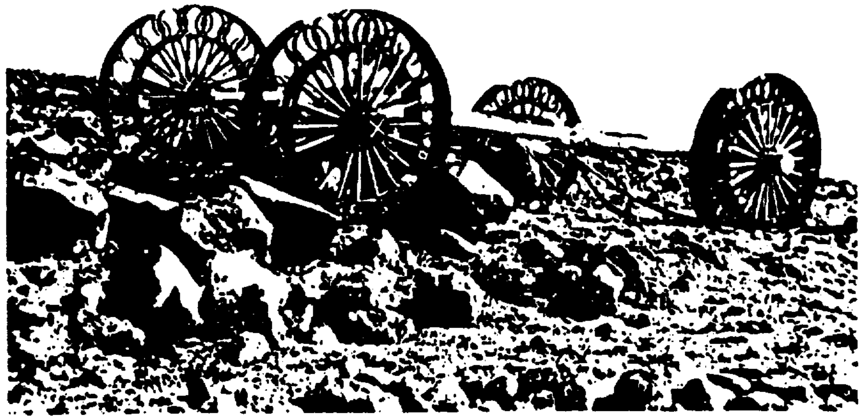


Lunokhod 2 (Luna 21)
1 Magnetometer.
2 Omni-directional antenna.
3 Narrow-beam directional antenna.
4 Antenna pointing mechanism.
5 Solar cells (generate electricity from sunlight to recharge chemical batteries).
6 Hinged lid (closed during transit and when "parked" during the lunar night).
7 Horizontal and vertical scan panoramic cameras.
8 Nuclear heater with reflector shield; also 9th wheel for distance measurement (obscured at rear.)
9 Soil probe (retracted).
10 Telescopic antenna.
11 Wheel unit.
12 Pressurised compartment.
13 Rifma-M chemical soil analyser (X-ray spectrometer) in retracted position.
14 Stereoscopic pair of television cameras with lens hoods and dust covers.
15 French-built laser reflector.
16 Television camera with lens hood and dust cover.

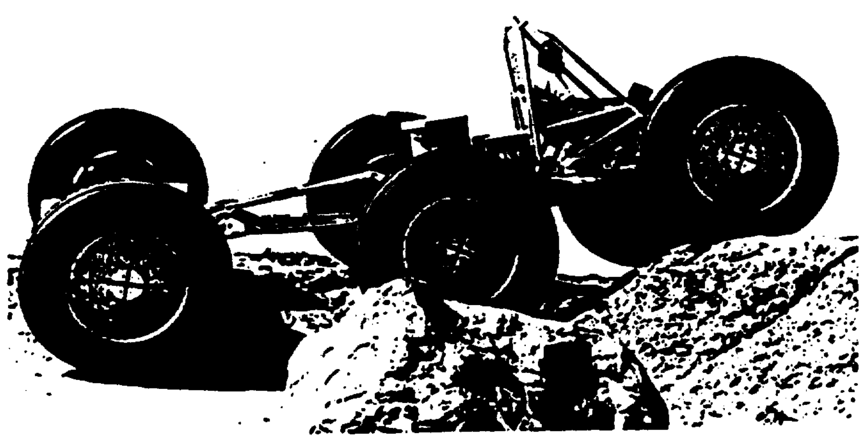
Luna 21 soft-landed inside Le Monnier crater near the eastern rim of the Sea of Serenity at 0135 Moscow Time on 16 January 1973. The first period of lunar exploration began on 17-18 January when Lunokhod 2 moved off from the landing site in a south-easterly direction over basalt lava, negotiating craters and boulders. Panoramic pictures received on Earth clearly showed the surrounding scene, including mountains bordering the Sea of Serenity.

Technical Data
Dimension over four wheels: 87in (221cm).
Wheel track: 63in (160cm).
Wheel diameter: 20in (51cm).
Weight: 1,852lb (840kg) at launch, about 220lb (100kg) heavier than Lunokhod 1 which operated on the Sea of Rains for 10½ months from 17 November 1970.

Figure 3.3-1 Samples of Previous Planetary Surface Locomotion Design Illustrations (Continued)



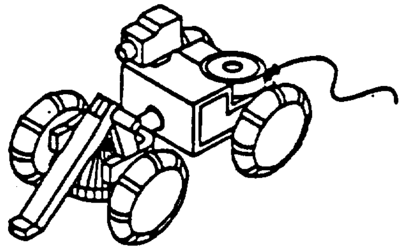
Chassis of Bendix lunar roving vehicle (MOLAB) developed for the National Aeronautics and Space Administration.



Chassis of the General Motors lunar roving vehicle (MOLAB) developed for the National Aeronautics and Space Administration.

Figure 3.3-1 Samples of Previous Planetary Surface Locomotion Design Illustrations (Continued)

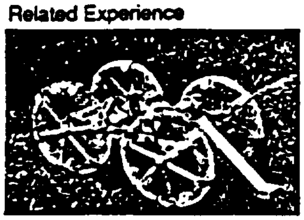
(A) LOCAL ROVER EXTENDS SCIENCE CAPABILITY WITH MINIMUM RISK



Physical Parameters

Length	.8 m
Width	.5 m
Height	.4 m
Mass	60 kg
Stowed Volume	.16 m 3

Related Experience



Viking '79 Study Program Prototype

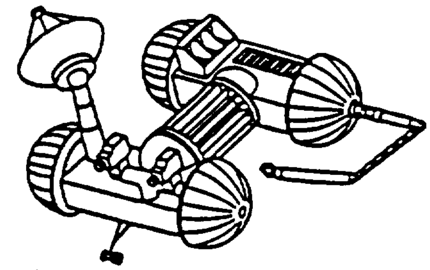
Features

- Inexpensive Contingency Sample Collector Augments Ascent Vehicle's Sampling Arm
- Lander Provides Power, Commands & Data Processing for the Rover

Developments Required

- Martian Sample Collection & Transfer with Small Rovers
- Cable Deployment & Retrieval in Congested Terrain

(B) MINIMUM ROVER MAINTAINS SIMPLICITY AND EXPANDS RANGE



Physical Parameters

Length	1.6 m
Width	1.2 m
Height	1.2 m
Mass	400 kg
Stowed Volume	1.5 m 3

Related Experience

Roving Technology IR&D Prototype

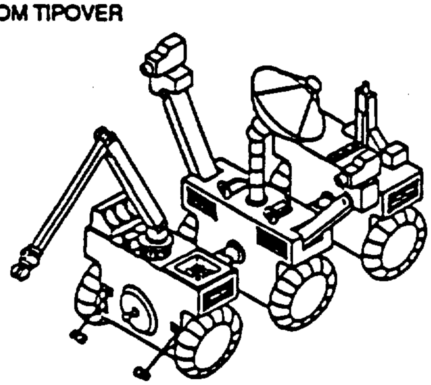
Features

- Articulated Body, Compact Stowage, Easy Deployment, & Simple Design
- Thermally Isolated RTG Four Wheel Drive

Developments Required

- Large Obstacle Negotiation with Smaller Rovers
- Electronic Component Packaging in Smaller Rovers

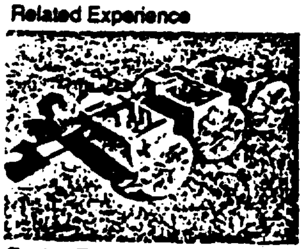
(C) MODERATE ROVER IS CAPABLE OF SELF RECOVERY FROM TIPOVER



Physical Parameters

Length	2.8 m
Width	1.4 m
Height	1.5 m
Mass	700 kg
Stowed Volume	4.4 m 3

Related Experience



Roving Technology IR&D Prototype

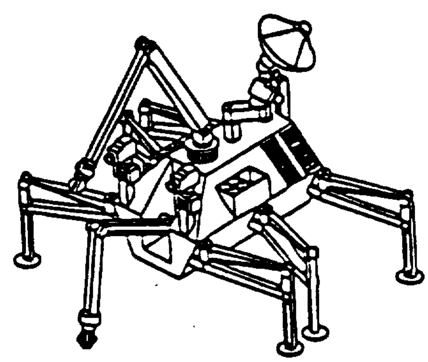
Features

- Self Recoverable with Active Articulation & Assistance from Sampling Arms.
- Excellent Traction with Independent Drive Wheels & Adjustable Vehicle CG

Developments Required

- Simplify Control & Complexity of Interbody Joints
- Power System Design for Multi-SegmentVehicles with Multiple RTS's

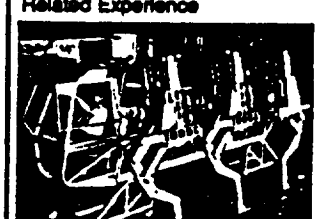
(D) MAXIMUM ROVER IS CAPABLE OF TRAVERSING VERY DIFFICULT TERRAIN



Physical Parameters

Length	3.3 m
Width	3.5 m
Height	2.2 m
Mass	1500 kg
Stowed Volume	10.3 m 3

Related Experience



OSU Advanced Suspension Vehicle

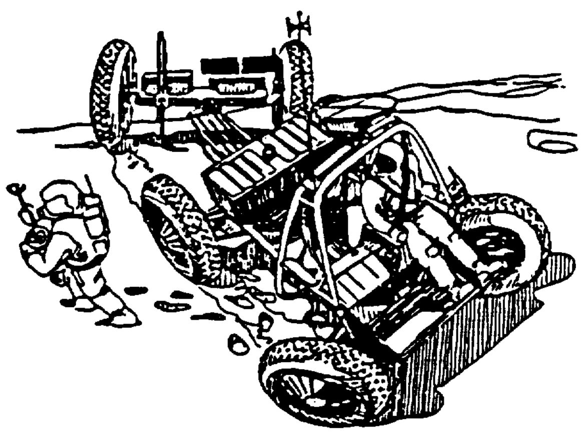
Features

- Very Stable Vehicle with an 11.5 m 2 Footprint
- Good Mobility Over all Types of Terrain with 6 Legs Assisted by 2 Sampling Arms

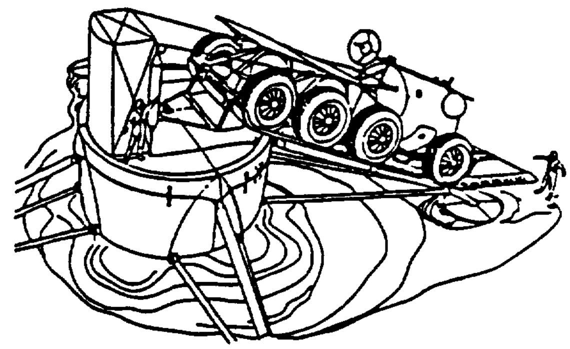
Developments Required

- Computational Requirements, Control, & Surface Pressure of Walking Vehicles
- Traverse Velocity & Reliability of Waking Vehicles

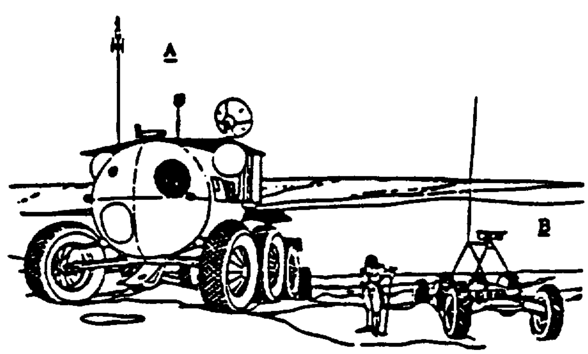
Figure 3.3-1 Samples of Previous Planetary Surface Locomotion Design Illustrations (Continued)



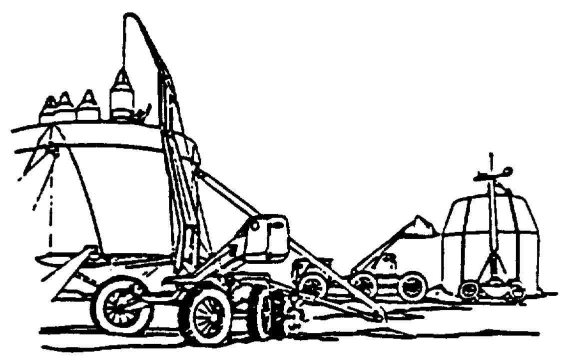
Local survey vehicle for lunar site operations



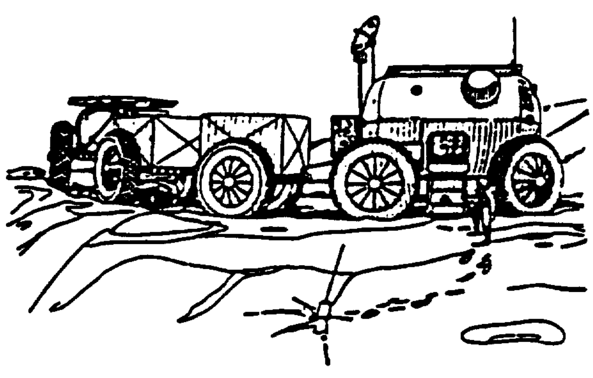
Base reconnaissance vehicle deployed from spacecraft



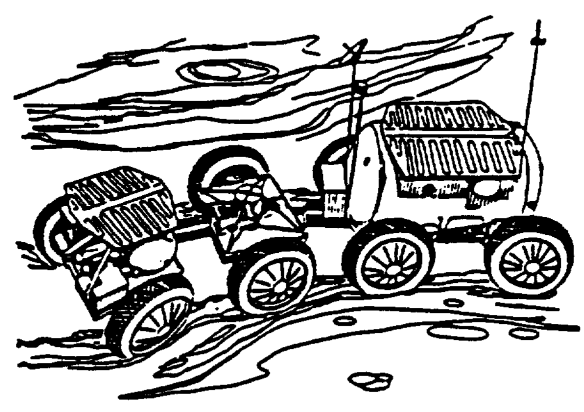
a. Long-range reconnaissance vehicle
b. Local utility vehicle



Construction and material handling vehicles

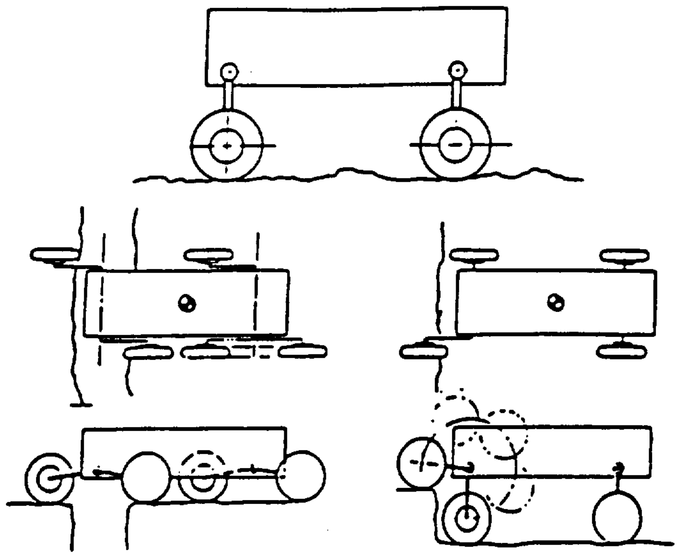


Heavy logistics vehicle for level 4 operations

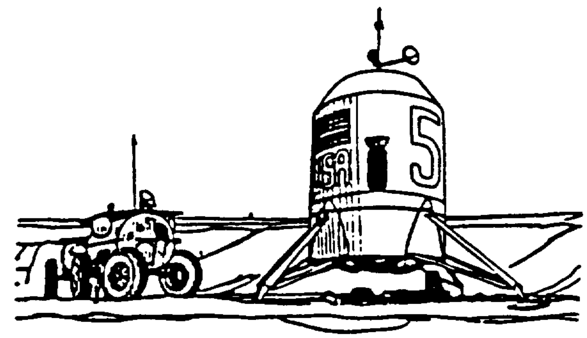


Logistics vehicle for level 3

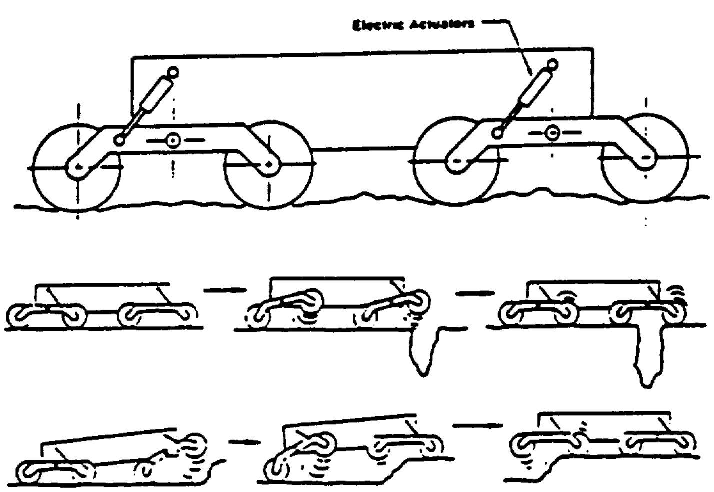
Figure 3.3-1 Samples of Previous Planetary Surface Locomotion Design Illustrations (Continued)



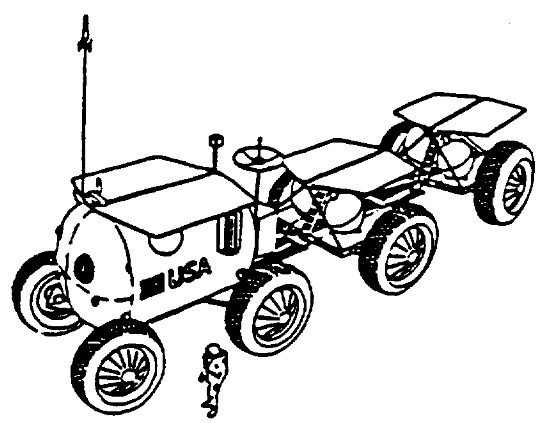
The Crank Mounted Wheeled vehicle has each wheel mounted on the end of a crank arm.



Scientific exploration vehicle for exploration

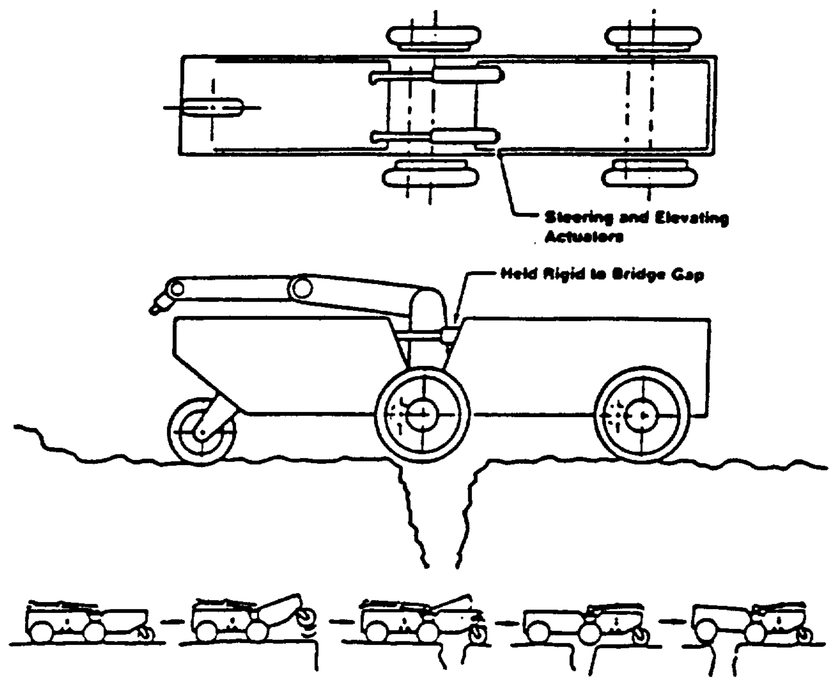


The single-bodied Dual Truck Suspension vehicle carries four wheeled bogies at either end. The bogies are moved about a central axis by mechanical actuators.

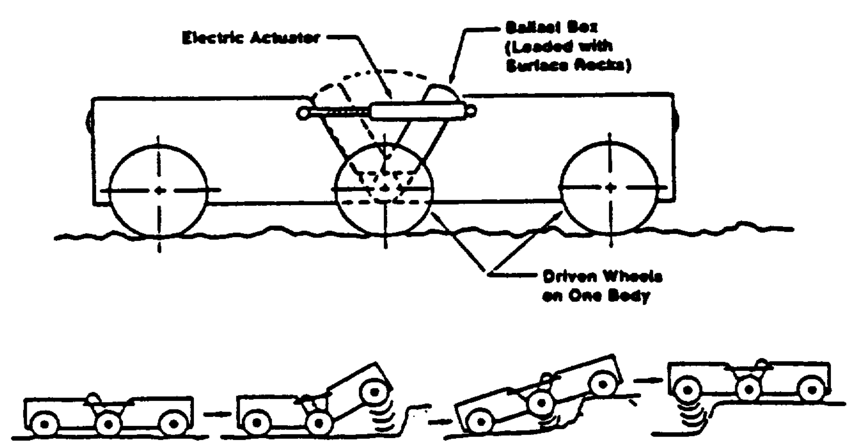


Long-range reconnaissance vehicle for lunar base operations

Figure 3.3-1 Samples of Previous Planetary Surface Locomotion Design Illustrations (Continued)

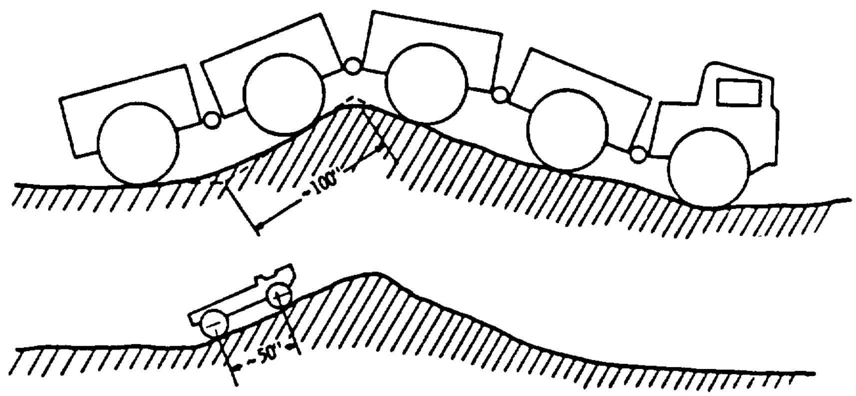


On the 5-Wheeled Articulator concept, the manipulator arm can be swung from one end to the other to move the C.G. of the vehicle. Opposite action of the actuators is used to steer.

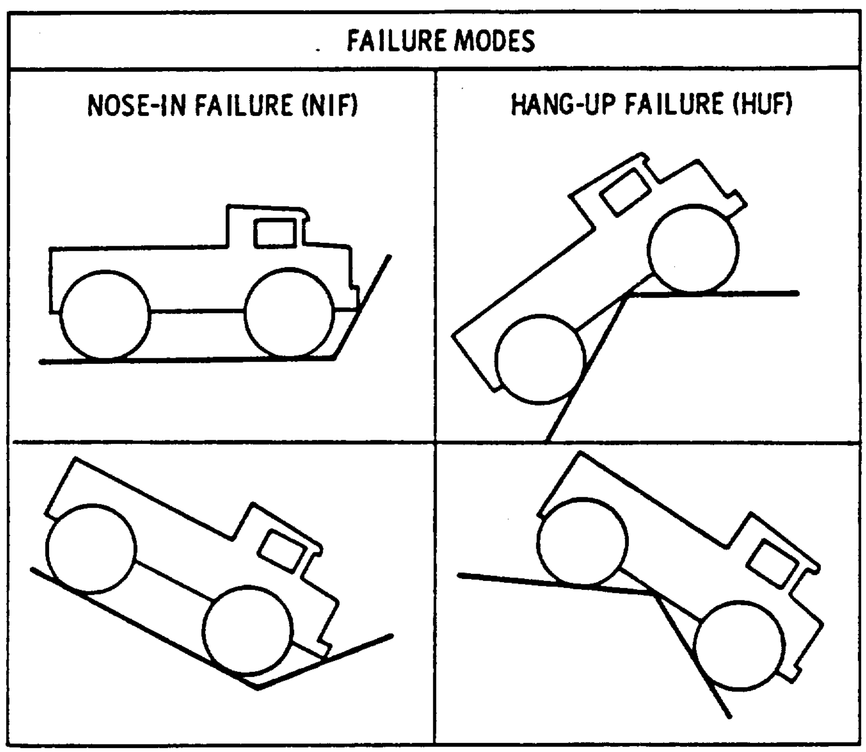


For the Externally Ballasted Articulator concept, either body may be raised by actuators depending on the location of the C.G. which is varied by a ballast box.

Figure 3.3-1 Samples of Previous Planetary Surface Locomotion Design Illustrations (Continued)



Surface roughness and vehicle size.



Two types of vehicle failure on four types of obstacles, each formed by two intersecting planes.

4.0 Vehicle Systems Comparison Analyses

There are various alternatives for accomplishing the functions required in the vehicles to be designed for lunar surface transportation. The key vehicle systems functions are reviewed here (e.g., power, thermal control, life support systems, locomotion method, etc.) and alternate methods for implementation are identified. In a generic study, it is difficult to conclude which system implementation options are the optimum choice since the conclusion requires specific knowledge of the surface transportation mission. Obviously, the best implementation approach for a particular function could be different for missions having diverse objectives and requirements. Therefore, in the vehicle systems comparison analyses which follows, the range of systems implementation options are explored with emphasis given to recording the likely performance variation in varying mission conditions. In fact, the baseline mission conditions defined in section 2.0 are used as the guidelines for the potential lunar surface transportation requirements.

The analyses of this section are intended to review the important systems implementation parameters and how they interact with varying missions. That is, the review is independent of specific vehicle design. In the next report section, a specific conceptual design is developed for each of the baseline missions. These conceptual design efforts utilized the information developed in this section. However, the systems comparison data should remain useful for other conceptual design efforts beyond this task study for new missions with different definitions.

4.1 Power Systems

Batteries, solar photovoltaic cells, and fuel cells are common space power sources that have been applied in numerous previous spacecraft. Planning parameters for these space power systems have been compiled and documented in Table 4.1-1. The various planning parameters are applied in the following analysis to explore potential technology for lunar surface transportation vehicle power systems.

The considerations involved in selection of a power system are significantly different for each of the three baseline missions. Therefore, the discussion of power supply is provided separately by baseline mission.

4.1.1 Local Transportation

For power sizing purposes, assume that the LOTRAN will have a loaded mass of 2000 kg (three times the mass of the LRV which consumed energy at the rate of 100 wh/km). As an approximation, the LRV required 100 wh/km for a load of 708 kg or 0.1412 wh/km/kg. Based on this approximation, the LOTRAN could require the product of (0.1412 wh/km/kg)(50 km)(2000 kg) which equals 14.12 kwh of energy. If the power is required at a constant average level for the eight-hour mission, the power source output would need to be 1.765 kilowatts. As a result, secondary or regenerative batteries can be used to meet these demands. Recharging of the batteries would be done upon return to base using the base power supply.

Considerable development is underway on both nickel-hydrogen and lithium-metal sulphide high energy density rechargeable batteries (HEDRB). A reasonable target for the lithium technology is approximately 110.2 wh/kg (50 wh/lb) and the nickel-hydrogen specific energy could be about 37.5 wh/kg (17 wh/lb). Based on these projected specific energies and an eight-hour mission at continuous power, the lithium battery would weigh about 73 kg (160 lbs) per kilowatt and the nickel-hydrogen batteries would weigh approximately 214 kg (470 lbs) per kilowatt.

If the mission is constrained to daylight, an alternative would be the use of a solar photovoltaic (PV) array using gallium arsenide cells. At a power conversion efficiency of 18 percent, the required area would be approximately 4.12 m^2 (44.3 sq ft) per kilowatt.

This PV array should be gimbled with increased area for contingencies and secondary batteries added to allow for short term disorientation of the array; however, the array would be relatively insensitive to misalignments of as much as 10 degrees. Without considering energy storage ramifications, the photovoltaic system mass would be about 36.4 kg (80.2 lbs) per kilowatt. In addition to offering higher efficiency than silicon cells, gallium arsenide cells are more forgiving at the elevated temperatures on the lunar surface.

4.1.2 Long Range Surface Applications Traverse

Nighttime operation at the kilowatt levels expected of this 42 day mission presents a significant challenge in selecting a power system. Battery weights are prohibitive. Fuel cell system weight and volume are dictated by the size of the reactant supply. Other chemical systems are less efficient than the fuel cell and would require a greater weight of reactants. Solar energy systems would only be applicable in the daylight periods. The accuracy of alignment required on solar dynamic systems most likely rules these out as potential systems and they are not considered further in this analysis.

Radioisotopes, as a thermal source for power generation, constitute a serious earth launch hazard and extensive environmental protection systems and planning are required before use. However, many unmanned interplanetary probes as well as the ALSEP (Apollo Lunar Surface Experiment Package) have used such systems. Further, the short half-life of the more readily available PO-210 creates a significant logistics problem for a multikilowatt system.

A nuclear reactor system reduces the earth launch environmental risk as compared to radioisotopes. However, shielding will be required during surface operations. Due to the surface to volume aspects of a fully enclosed reactor, a considerable amount of mass is required in the 10 kilowatt region. The exact shielding mass required depends on many variables including safety policies and specific design configuration. Shadow shielding would be more suitable except that the reactor remains a residual hazard after use.

A spacecraft nuclear power system on the order of 1 to 5 kilowatts is currently under development by the Department of Energy, the Air Force and the Strategic Defense Initiative Office. The system is known as the Dynamic Isotope Power System or DIPS and current mass estimates are in the range of 200 to 500 kg (reference 3). The life of these systems can be measured in years. This system, adapted for lunar application, would clearly outperform other systems considered. However, nuclear systems involve some difficult political and safety issues. An adapted DIPS type system will not be considered here. When improved data are available and the mission is better defined, nuclear power deserves additional study, particularly if very long range and long duration are desired.

4.1.2.1 Comparison of Candidate System Options

Considerations towards selecting power systems dictate that the logical candidates are either chemical (fuel cells) or solar/chemical.

1) Fuel Cells:

Hydrogen/oxygen fuel cells using a conservative specific reactant operational consumption rate of 0.4 kg (0.9 lbs) per kilowatt-hour have been assumed throughout this section. In the long range mission, the fuel cells would require approximately 457 kg (1,008 lbs) of hydrogen and 3,658 kg (8,064 lbs) of oxygen for a 10 kilowatt average load over 1008 hours (10,080 kw/hr). Both reactants would be stored subcritically as a liquid with gaseous reactant in the emptied volume. The approximately 7.7 cubic meter (272 ft^3) hydrogen tank of 2.45 m (8.04 ft) diameter would have a mass of 1454 kg (3200 lbs) and the 5.3 cubic meter (187 ft^3) oxygen tank of 1.1 m (3.63 ft) diameter would have a mass of 682 kg (1,500 lbs). The total mass of a 10 kw system would be approximately 6,440 kg (14,200 lbs) or about 644 kg (1,420 lbs) per average kilowatt over a 1008-hour mission. This ratio would be linear down to a few kilowatts.

2) Photovoltaics/Batteries:

During sunlight hours, a 10 kw load could be supported by a gallium arsenide array of an area of approximately 41.2 square meters (443 ft^2) of active cells. Orientation could be maintained by remote manual or automatic control but precision orientation is not required.

The lithium-metal sulphide battery system would use the high energy density rechargeable battery which would be recharged after returning to the main base.

With this system selection, a premium is placed on planning the mission for a sequence of lunar day-night-day operation. Even so, the single night at 10 kw would require 30,909 kg (68,000 lbs) of batteries.

3) Photovoltaics/Fuel Cells:

The photovoltaic array for this system would be identical to the array used for the Photovoltaic/Battery system. Dark side power would be furnished from non-regenerative hydrogen/oxygen fuel cells.

As in the case of the Photovoltaic/Battery system, a premium is on planning around a day-night-day mission. During the lunar day the heat leak into the storage tanks would result in the release of about 0.5 or 0.6 percent of reactants per day. These reactants would be passed through the fuel cells and result in the contribution of about one kilowatt from the fuel cells during each lunar day. The total energy generated over 1,008 hours is 4,032 kwh. In this case, a 10 kw system would have a mass of 2,940 kg (6,481 lb) or approximately 294 kg/kw (648 lb/kw) for the 1,008-hour mission.

If a night-day-night mission is conducted, the residual power available from the fuel cells during the daylight period would be about 1.7 kilowatts. The total energy generated by the fuel cells is 7,291 kwh. The 10 kw mass of this system would be approximately 5,022 kg (11,071 lb) or 502 kg/kw (1,107 lb/kw) for the 1,008-hour mission.

4.1.2.2 Evaluation Comments

It is assumed that the gallium arsenide cell reliability will approach that of the present silicon cell by the time this mission enters the active buildup stage. The sheer number of cells to be used would allow the loss of some cells without significant array output degradation with care in wiring circuitry design.

Fuel cells rate behind secondary batteries from a reliability standpoint; however, nickel-hydrogen or lithium secondary batteries are still in development. The fuel cells have rotating components that are required for thermal and moisture control. Yet, a high level of reliability of this fuel cell technology has been amply demonstrated in the Apollo and Shuttle program.

It is clear that solar energy is the most efficient power source whenever the vehicle is exposed to daylight. The most attractive power system for operation in darkness is hydrogen-oxygen fuel cells. The additional thermal burden imposed by the fuel cell operation will occur when the radiator surfaces are operating at maximum capability. Hydrogen and oxygen could be available since they may be required for propulsion purposes. Any potential shortcomings in the reliability of the fuel cell recirculation system will be more than offset by its significant weight advantages over batteries.

4.1.3 Remote Sight Ballistic Flight

Both the flight vehicle, which is required to serve as a base of operations for eight days, and the local rover operating for eight hours at a time require electrical power.

1) Flight Vehicle:

As in the long range surface mission, electrical power system selection is greatly dependent upon the mission constraints that planners are willing to accept.

For this analysis, it was assumed that the total duration of the power level required for the flight modes was six hours. It was also assumed that the average level required during the stay period would be 60 percent of that required during the flight mode. For all systems considered, the weight of each would be essentially a direct function of power for levels of power contemplated in the application.

If this mission was constrained to daylight only, a photovoltaic array could be used to furnish power. A gallium arsenide cell array would have a mass of 36.4 kg (80.2 lbs) per kilowatt. Secondary batteries could be used to support the flight operations and would have a mass of 54.6 kg (120 lbs) per kilowatt for the six hours of operations. With an arbitrary load of 10 kw during flight and 6 kw during the stay period, the system mass would be 764 kg (1,684 lbs). The batteries would be recharged after returning to base. Conceivably, the batteries could also be recharged from the solar array after landing; however, the full energy is required in the batteries on departure from the base in order to accommodate a potential abort.

Fuel cells could be used in lieu of both the array and batteries and this system would not be constrained to daylight hours. To furnish 1,176 kwh, the mass of this system, including separate tankage from the propulsion system, would be approximately 890 kg (1,962 lbs) for the power profile used above. The fuel cell system mass breakdown is:

Fuel cell:	190 kg	419 lb
Hydrogen:	52 kg	115 lb
Oxygen:	418 kg	921 lb
Hydrogen Tank:	125 kg	276 lb
Oxygen Tank:	105 kg	231 lb

If the propulsion system uses hydrogen-oxygen, fuel cells are more favorable since the mass of separate tanks for the reactants would not be required.

2) Remote Location Lunar Rover:

This vehicle poses essentially the same challenges as the LOTRAN but would probably be smaller and lighter. As a result, it is more likely that this vehicle would use secondary batteries even if the mission was constrained to daylight. The batteries would be recharged from the flight vehicle power system between uses.

Table 4.1-1 Space Power Systems Planning Parameters

POWER SYSTEM	PARAMETER	
High Energy Density Rechargeable Batteries:		
Nickel-Hydrogen:	26.7 kg/kwh	58.8 lb/kwh
Lithium-Metal Sulphide:	9.1 kg/kwh	20 lb/kwh
Gallium Arsenide Photovoltaic (PV) Cells:		
Available Solar Flux:	1.350 kw/m^2	0.1254 kw/ft^2
Power Conversion Efficiency:	18 Percent	
PV Array Unit Weight:	8.84 kg/m^2	1.81 lb/ft^2
Power Output per Unit Area:	0.243 kw/m^2	0.0226 kw/ft^2
PV Array Area per Unit Power:	4.12 m^2/kw	44.3 ft^2/kw
PV Array Mass per Unit Power:	36.4 kg/kw	80.2 lb/kw
Hydrogen/Oxygen Fuel Cells (Non-regenerative):		
Reactant Consumption Rate:	0.36 kg/kwh	0.8 lb/kwh
O/H Consumption Weight Ratio:	8:1	
Liquid Hydrogen Packaging Density:	59.3 kg/m^3	3.7 lb/ft^3
Liquid Oxygen Packaging Density:	692 kg/m^3	43.2 lb/ft^3
Liquid Hydrogen Tank Mass Ratio for Hydrogen Mass in 50 kg (110 lb) Range:	2.4 kg Tank/kg H	2.4 lb Tank/lb H
Liquid Oxygen Tank Mass Ratio for Oxygen Mass in 400 kg (882 lb) Range:	0.25 kg Tank/kg O	0.25 lb Tank/lb O

Table 4.1.2.1-1 Long Range Traverse Power Options Mass Requirements

COMPONENT	10 KILOWATT POWER SYSTEMS MASS (kg)				
	FUEL CELLS	PHOTOVOLTAIC / BATTERIES		PHOTOVOLTAIC / FUEL CELLS	
		DAY-NIGHT-DAY	NIGHT-DAY-NIGHT	DAY-NIGHT-DAY	NIGHT-DAY-NIGHT
Liquid Hydrogen	457	-	-	183	330
Liquid Oxygen	3,658	-	-	1,463	2,696
Hydrogen Tank	1,454	-	-	504	1,016
Oxygen Tank	682	-	-	236	476
Fuel Cells	190	-	-	190	190
Batteries	-	30,909	61,818	-	-
PV Cells	-	364	364	364	364
TOTAL MASS	6,441	31,273	62,182	2,940	5,022

4.2 Locomotion Methods

The design of a mobility system is based on many factors including terrain, soil physics, mobility objective, and vehicle load parameters. In reference 15, M.G. Bekker presents a flow chart [reference 15, Figure 4.2-1] that describes the interaction between design requirements and vehicle characteristics. The vehicle environment is described in section 2.1 and the vehicle missions are described in section 2.2 of this report.

4.2.1 Locomotion Design Options

Vehicle locomotion concepts and attributes are developed in the following paragraphs. When a mobility system was developed for the Apollo program, the range of environmental and mission requirements was fairly narrow. With very specific mission requirements, the design engineers quickly converged on an LRV design featuring four wire mesh wheels, independent four wheel drive, double Ackerman steering, and a pair of parallel triangular suspension arms for each wheel. Mobility requirements for a lunar base are far more diverse. Mission requirements vary greatly and the environment of locomotion may be literally anywhere on the moon. To best meet these diverse needs, all practical mobility concepts must be re-examined with lunar base mobility in mind.

4.2.1.1 Screw Driven Buoyant Vehicles

Screw driven buoyant vehicles were originally considered for the Apollo program (by GM in 1963) when lunar soil properties were unknown. This mobility concept works well in very weak soil conditions (such as snowy or marshy terrain on earth). Two characteristics of screw driven vehicles deserve special note: power consumption and drawbar pull. This concept is capable of providing a lot of drawbar pull, but it uses a lot of power in the process. An equation has been developed for unit-power consumption of a rotor in sand. This equation does not consider sink slippage, and assumes pretapped grooves. It underestimates power consumption and is good only for a first order analysis. The data indicate that drawbar pull characteristics are very good. A screw driven vehicle has been optimized for sandy conditions on earth. The findings are:

o Length / Diameter ratio of rotor should be about 6.
o The rotor should displace 50% of the carried load.
o Height / Diameter ratio of blade should be about 0.2

o Helix angle should be about 50 degrees
o To size power needed: (2) * (load) * (rotor radius) = max torque

Screw driven vehicles have few lunar base applications. These vehicles tend to be heavy, slow, and power hungry. Their only advantage is that they work well in very weak, very soft soil. During Apollo, it was determined that most of the soil on the moon could support ground contact pressures between 7 and 10 kPa. Screw driven vehicles are not needed for daily locomotion. The one potential application may be off-loading landing vehicles. Loads may be great, distances will probably be short, and power use will not be as important. Screw driven vehicles should be considered when addressing that application.

4.2.1.2 Tracked Vehicles

Tracked vehicles outperform wheeled vehicles in soft soil and with large payloads. The performance characteristics of tracks are determined by the large track contact area. Large contact area means excellent flotation characteristics, large drawbar pull values, and a high degree of motion resistance (which translates to energy loss and power use). Some good analytical expressions have been derived for tracked vehicles; compaction, static track sinkage, maximum soil thrust, and drawbar pull are all important.

Tracks are used on earth when their large footprint area is needed (when soil is soft). Considering the low lunar gravity, however, such soil strength would have to be very low by terrestrial standards. Tracks used for earth applications have very poor wear characteristics. There is a high frequency of breakdown and tracks are only made practical by making them very big, heavy, and sturdy. In addition, large military tracked vehicles must normally be transported on wheeled trailers to move long distances.

Some points must be made in favor of tracks. First, a "spaced link" track has real promise. A spaced link track is a track that features widely spaced angular cleats. Performance depends on soil conditions, but spaced links provide adequate flotation for lunar gravity conditions, and have tremendous drawbar pull characteristics at a reduced weight. In sandy loam on Earth, a conventional tracked vehicle weighing 10,000 lb will pull 9,900 lb. A spaced link track vehicle weighing 10,000 lb will pull 17,800 lb. Many tractor applications such as bulldozing or towing non-propelled vehicles are strongly linked to gravity conditions and are very difficult on the moon. If these tractor applications are required, spaced link tracks may prove to be the best suited for the job.

4.2.1.3 Walkers

Walkers are the subject of much active research, and undoubtedly many advances will be made in the future. At the present, however, walkers are very complicated and inefficient vehicles. Walkers are plagued by large dynamic loads, non-uniform motion, and a vehicle geometry that must follow the random geometry of the terrain. Walkers are inefficient in their use of energy. A walker taking short steps spends much of its energy compressing soil. A walker that has long strides spends a lot of its energy moving the cg of the vehicle up and down.

The Defense Advanced Research Projects Agency is interested in walkers because they have tremendous obstacle crossing potential. Obstacle crossing capability is in its infancy, but already a walking vehicle has been built that can climb a three foot vertical step.

A potential mission for a walker might be a traverse into Tycho crater, a 56 mile wide bowl. The floor of Tycho is a mass of contorted hillocks, ragged flows and jagged knobs. Such a surface would have cracks and fissures, rubble, steep peaks, and sharp valleys.

Today, walkers are not yet a viable mobility concept. In the future, they may realize their potential for crossing very uneven terrains. They will always be inefficient from an energy standpoint, but mission planners may be willing to pay that penalty to get obstacle crossing ability.

4.2.1.4 Rocket Propulsion Systems

Rocket propulsion systems are the preferred mode of transport for long range missions where the journey end location is the objective and the environment traversed is of no interest. The circumference of the moon is about 11,000 km. Transportation systems traveling along the lunar surface have maximum cruise velocities that are determined by the terrain and the vehicle. A surface vehicle that has a cruising velocity of 10 km/hr would require 46 Earth-days of continuous driving to travel from one pole to the other and back. Electrical power and life support consumables for 46 days of travel are significant. This leads to a large and complicated vehicle. An alternative to ground transport is a ballistic rocket propulsion system. The travel time and associated supplies are significantly reduced; however, the propellant mass required may be large. Lunar surface transportation by ballistic rocket propulsion is considered further in sections 4.3 and 5.3.

4.2.1.5 Wheels

Wheels have proved to be an excellent mobility choice for past lunar mobility systems. The MET (Modular Equipment Transport) used in Apollo 14, the LRV used in Apollo 15-17, and the Soviet Lunokhod have all demonstrated the successful use of wheeled vehicles on the moon. These three vehicles also demonstrated the wide range of wheeled vehicle options. The MET was a two wheeled rickshaw type vehicle with pressurized (4psi) tires (40 cm diameter, 10 cm wide). Its mass was 75 kg fully loaded. The LRV was a four wheeled vehicle driven by astronauts. The mass of the vehicle was 218 kg, (708 kg fully loaded). Its wheels were flexible wire mesh with chevron shaped treads (82 cm diameter, 23 cm wide). The Lunokhod had a mass of about 600 kg and had eight rigid wheels (46 cm diameter, 18 cm wide).

Wheels will be the preferred mobility option for many missions. Wheels are mechanically efficient, can be designed into lightweight systems, and can be built with excellent reliability. One problem with wheels in terrestrial all-terrain applications is they tend to have a small footprint. In the reduced gravity field of the moon, having a large ground contact area is not required. Bootprints were deeper that LRV tracks during Apollo missions and the LRV had chevron treads covering only 50% of the wheel.

Wheels have tremendous versatility. There is a large range of wheel types, sizes, numbers and configurations. While rigid wheels and pneumatic tires are not well suited for many lunar applications, wheel options that deserve further study include: wire mesh tires, metal-elastic tires, elliptical wheels, hemispherical tires, and cone wheels.

A mature lunar base may have a fleet of vehicles designed to meet a wide range of missions. While specialty vehicles have their place in this fleet, the bulk of the missions will be best performed by wheeled vehicles.

All wheeled mobility power requirements have been scaled at the baseline rate of 0.08 wh/km/kg. Values in this range were derived using models developed by Becker, published by Pavlics, and confirmed by Apollo 15 results. Batteries work well for short duration missions and fuel cells are best used for longer missions requiring a larger store of energy. Solar voltaic cells are particular effective for power required over long periods of time (many hundreds of hours) since the energy is stored in the Sun. However, the sun is also a limitation since the solar cells do not work during the lunar night.

4.2.2 Mobility Factors

There are some mobility factors relating to locomotion systems that deserve further discussion. These factors do not necessarily relate to one vehicle concept, and are listed below in no preferential order.

4.2.2.1 Maximum Speed

Maximum speed on the lunar surface is affected by terrain, gravity, and vehicle characteristics. Reduced lunar gravity makes it easy for a vehicle to become airborne. At times during the Apollo missions, all four wheels of the LRV left the ground. The terrain at the Apollo sites was very hummocky. Driving speed during Apollo 15 was not determined by the maximum performance of the vehicle, but by the very uneven terrain. Future vehicles with better performance may be similarly limited by terrain conditions. LRV maximum speed was approximately 13 km (7 miles) per hour.

4.2.2.2 Suspension Systems

Suspension systems become critically important whenever vehicle speeds exceed 5 km/hr. Dynamic loads can be passively absorbed by a component of the vehicle or they may be absorbed by an active suspension system. The LRV absorbed shocks with the wire mesh wheel and an active suspension system. The suspension system bottomed out three times during the Apollo 15 surface activities. Rigid wheels were ruled out in the design of the LRV because they could not absorb dynamic loads. This characteristic should be noted when designing future systems, especially those designed for higher speeds. Cone wheels can be designed to have the ability to passively absorb dynamic loads.

4.2.2.3 Configuration Assembly and Deployment

Weight, packing, and vehicle deployment were prime design drivers in the past studies. 40% of the engineering effort of the Grumman MOLAB design was committed to packing and deployment. The LRV was not permitted to have six wheels because a six wheel configuration could not be packed. Weight packing and deployment should still be important considerations, but they can't be allowed to drive the design. On a lunar base, a vehicle can be truly assembled. A vehicle does not have to be deployed in less than an hour. Wear and performance become more important than weight and packing. The wire mesh wheel on the LRV only weighed 12 lbs on Earth and performed well, but the Apollo 15 vehicle only traversed 28 km.

4.2.2.4 Crater Accommodation

Craters deserve special note as the most critical environmental obstacle in the design of a mobility system. Craters can bottom out a suspension, cause a hang-up failure, cause excessive sink, or cause a vehicle to roll. Craters are hard to distinguish in high sun angles and easy to drive into. All Apollo landings occurred during times of low sun angle (for better visibility) and astronauts left the lunar surface before the sun angle was greater than 45 degrees. Even during these times of good lighting, crater detection was a problem. Apollo 15 commander Dave Scott reports, "In general, 1-meter craters were not detectable until the front wheels had approached to within 2 to 3 meters." The lip of a crater poses the worst driving conditions; the geometry involves a steep dropoff and the soil is often soft. Normally, LRV tracks were 3 to 4 cm deep, but tracks were as much as twice as deep at the lip of craters. The Lunokhod 1 almost got stuck in the loose soil at the lip of one crater, and a later Lunokhod rover is supposed to have rolled over as it crossed the lip of a crater. The only LRV failure occurred during Apollo 15 when two wheels were hung up because they were both in deep craters. The vehicle was picked up, carried out of the craters and driven off. Therefore, a mobility system must be designed for the craters it might encounter.

4.2.2.5 Wheel Configuration

The design of wheeled vehicle systems is a complicated science. While required ground contact area can be calculated fairly easily, there is an almost infinite combination of wheel sizes, geometries, numbers, and configurations that can meet a contact area specification. More smaller wheels have more redundancy and better reliability. Fewer larger wheels tend to be mechanically simpler and weigh less. Tall skinny wheels have less bulldozing resistance and better clearance, but they are harder to pack. Wheeled vehicles with the same contact area but different wheel configurations can have very different performance characteristics.

Wheels can be sized for different applications based on an understanding of the LRV wheel performance. The LRV wheel was a wire mesh with metal chevrons covering 50 percent of the contact area. The wheel had a diameter of approximately 80 cm and was about 23 cm wide. Fully loaded, the vehicle had a mass of 708 kg. Under equal loading, each wheel carried a 287 N load.

According to the Apollo 15 mission report, wheel sink was approximately uniform across all soil conditions, provided the terrain was flat and all wheels loaded equally. On flat lunar terrain, the wheels sunk about 3 cm in the softest soil. Boot prints were 5 to 8 cm deep for the same conditions. Chevron markings were very clearly marked in the surface, demonstrating little wheel slip.

With 3 cm sink, the wheel displaces an area 30 cm long and 23 cm wide. The contact region subtends an angle of 44.7 degrees. Carrier reports in the Lunar Sourcebook that the Moon's regolith can support a ground contact pressure of 7 to 10 kPa. On the LRV, a 287 N load was carried by a 0.069 m^2 area. Therefore, the LRV contact pressure was 4.2 kPa.

The ratio of wheel sink to wheel diameter is important. It determines slip conditions and power use. Manned vehicles should probably have ratios similar to the LRV. Large trucks and excavators can probably have greater ratios.

Carrier reports vehicle mileage to be 35-56 W-hr/km and mass mileage to be 0.050-0.080 W-hr/km/kg. The range of values is mostly due to slope distribution, changing soil values, and errors in measuring equipment. Measuring errors may be 20 percent. Changes in slopes and soils can change power requirement values 10 to 15 percent.

4.2.2.6 Vehicle Articulation

Articulated vehicles are the most reliable vehicles that also have good obstacle crossing capability and are presently available. Articulation allows the wheels of the vehicle to follow the ground contour. Bekker states that frame articulation adds more obstacle mobility to a vehicle than any other structural feature. This capability does not come without a cost. Rigid framed vehicles are better suited to large heavy payloads, and the extra joints must be protected from the lunar dust.

4.2.2.7 Tribology

Despite more than twenty-five years of successful spacecraft missions by a variety of manned and unmanned vehicles, there is still a need for more progress to be made in the field of space tribology. The space environment is varied and systems able to survive must experience conditions which are harsh in the extreme. Pressures range from the space vacuum to thousands of pounds per square inch. Chemical environments range from pure oxygen and other oxyidizing substances to the reducing environment of hydrogen, hydrazine, and some metals. Comtamination is typically minimal, but on the lunar surface, it constitutes gross dust levels of an abrasive nature. Radiation levels in space are damaging for exposed surfaces. In addition, nuclear power plants, used minimally in the past, but with more potential merit on the lunar surface, are much more damaging to lubricants than solar radiation.

Vehicles and machinery clearly require specialized lubrication considering the total environment. The hard vacuum eliminates lubrication techniques common on Earth. Volitiles quickly escape from a petroleum grease leaving a calcium or sodium soap residue. The low temperatures raise viscosity to the point where a semi-fluid grease becomes as hard as asphalt. This same cold condition removes the flexibility needed for a rubber boot designed to keep lubricant in and dust out. The reduced gravity level may also influence tribology designs, but the net effect may be beneficial. The difficulties of performing a major overhaul while on the Moon implies a need for many trips, each free from failures, such as a frozen wheel bearing or a gear in a power train. The cold temperature and hard vacuum eliminates the use of liquid lubricants. Dry film lubricants are too short-lived even without considering the abrasive dust aggravated by static electricity. Composites, such as filled polytetrafluoroethylene (PTFE) products, may provide the most promise for bearings of the electric drive motor, steering and shock absorber linkages, and wheel bearings. However, these are also lacking in the long life requirements for such vehicles. Therefore, further study and research are required to develop lubricant technologies and/or design approaches to lubricate material-to-material moving interfaces.

4.3 Rocket Propulsion

The overall requirement for the rocket propulsion system of the BALTRAN is that it enable the vehicle to initiate and gently complete ballistic trajectories. Surface distances ranging from several hundred kilometers to several thousand kilometers in any azimuth must be accommodated. Quick departure for emergency rescue may be required. Because of these requirements, propulsion systems for the ballistic transportation vehicle will be traditional chemical rocket systems. These systems consist of two major subsystems, the propulsive power subsystem and the propellant subsystem. The characteristics of each subsystem are strongly coupled to the selection of propellants. The following discussion will indicate why a liquid oxygen/hydrogen chemical propulsion system appears to be the most appropriate.

Logistics for the support of the BALTRAN will also depend on the selection of propellants. Consequently, the vehicle and the base will have significant interactions and the selection of propellants will be linked with the lunar base design. It is assumed that the lunar base will include a lunar oxygen production plant. Cryogenic systems tend to be the overall best selection since the system capitalizes on a lunar resource. The logistics required for the rocket propulsion system based on the lunar surface will be greatly reduced if the propellants can be produced on the lunar surface.

4.3.1 Design Options

The ballistic nature of the BALTRAN trajectory indicates a requirement for the propulsion system to provide thrust in as nearly an impulsive manner as possible. Traditionally, this has been accomplished with chemical rocket systems. Chemical propulsion systems are well developed and currently very reliable in Earth launch applications. Two important improvements which must be developed, but which will not be considered further in this report, are space maintainability and lifetime reliability.

The options available involve the selection of propellants. There are many propellant combinations from which to chose. Fuels and oxidizers which make up the propellant combinations can be either cryogenic or storable. The cryogenic propellants are a liquid oxygen oxidizer and a liquid hydrogen fuel. Storable propellants include oxidizers such as nitrogen tetroxide and fuels such as monomethyl hydrazine, propane, or other hydrocarbons. For the purposes of this discussion, chemical systems using cryogenic oxygen and hydrogen, a hybrid with cryogenic oxidizer and storable fuel, and all-storable propellants will be examined. The characteristics of these propellants with respect to how they are stored, where they are produced, and how they perform in the rocket are all of interest to this discussion.

For this study, only pump-fed systems are examined. The differences between pump-fed and pressure-fed systems do not change the overall results significantly. In the final analysis, reusability and throttling requirements will probably eliminate pressure fed systems as a viable option. Solid propellant rocket motors are not considered here since they tend to have limited flexibility in thrust and impulse levels. For the variety of mission trajectories to be handled by the BALTRAN, throttling is essential.

4.3.2 Comparison Factors

The propulsion system consists of the rocket engines and the propellant subsystems. The propellant subsystem is the propellant, tankage, and piping to the engines. In general, the subsystem having the largest effect on the overall design of the vehicle is the propellant and associated storage requirements.

The mass of the rocket engine tends to be small compared to the mass of propellants and tanks. The primary design drivers in determining engine mass are chamber pressure, nozzle expansion ratio, and thrust level. Specific Impulse (Isp) is a measure of the engine's fuel efficiency and, thus, is a prime factor in the amount of propellant required. There is some effect of Isp on engine mass. The Apollo Lunar Module engine had about 45,000 Newtons (10,116 lb) thrust with 300 seconds Isp and a 180 (396 lbs) kilogram mass.

Propellant storage varies with choice of propellants. Factors affecting this include propellant mass requirements, propellant densities, and thermal considerations, important for cryogenic propellants. The mass of these systems can be expressed as a percentage of actual propellant mass for conceptual design studies. These storage subsystem masses can vary from 2 to 10 percent of propellant requirements.

For comparison of chemical systems, the following factors must be considered:

1) Masses - Vehicle dry mass dry and vehicle mass with propellants.

2) Propellant Volumes - Propellant volumes are indications of the relative sizes of the vehicles for comparisons.

3) Isp - Because specific impulse will affect the amount of propellant, it plays a major role in the selection of a propellant option.

4) Tankage Fraction - The fraction of the propellant that is tankage. This has a strong influence on the overall size and mass of the vehicle.

5) Storability - Some types of propellants are easily stored without significant equipment. Some propellants such as cryogenics, require significant amounts of equipment.

6) Availability - For some selections, the moon may provide a local source of propellants. In this case, the propellants do not need to be transported from Earth. This is a major savings although it depends very heavily on the type of lunar base and mission model under consideration.

7) Safety - Safety hazards can pose major drawbacks for certain propellants. The toxicity of the propellants as well as their corrosiveness are indications of how difficult they will be to handle and what care must be taken for crew safety.

8) Mixture Ratio - This factor is coupled with propellant availability when only one of either fuel or oxidizer must be imported.

9) Toxicity - Toxic materials increase handling complexity and procedures due to increased crew hazards and corrosion control actions. A vehicle that must be maintained on the lunar surface will do well to use non-toxic propellants.

4.3.3 Comparison Analysis

Table 4.3.3-1 is a comparison of various types of propellant selections. For the purposes of direct comparison, a baseline mission has been selected. For this mission, a vehicle mass of about 7,000 kilograms (15,400 lbs) excluding tanks, engines, and landing gear was selected. This is about equivalent to the Apollo Lunar Module dry mass including both ascent and descent stages. The selection of propellants is fairly independent of this mass as long as reasonable ranges are considered. Landing gear are assumed to be 2 percent of the landed mass which is also the loaded mass in this case. The LM ascent engine thrust level was about 2 times the gravity forces on the ascent stage loaded mass. The factor of 2 was also used for this analysis. Velocity change requirements of 6,720 meters per second (22,046 ft/sec) are used as the estimated maximum round trip requirement for a 180 degree BALTRAN mission (1,680 meters per second each for base departure, site arrival, site departure, and base arrival). This would allow the BALTRAN access to the entire Moon from one base site.

For each candidate system, the table shows the velocity requirements, Isp, vehicle mass with and without propellants, propellant masses and volumes, engine thrust levels and minimum throttle settings, tankage fractions, oxidizer to fuel mixture ratios, and comments on storability, availability and safety.

4.3.4 Evaluation Comments

The cryogenic system requires only half the mass of propellants needed by the all-storable system. Although the volume of the all-cryogenic system is highest and it is difficult to store, the all-cryogenic system is recommended because of its performance, lack of toxicity, and possible local availability.

In reviewing Table 4.3.3-1, it is apparent that effects of different factor combinations on loaded vehicle system masses is large. The cryogenic system vehicle dry mass is lighter than the others by only 5 to 10 percent. When propellants are considered, however, the cryogenic system is 50 percent the mass of storable systems. This mission is a high energy mission, requiring more velocity change than a single stage lunar landing and ascent. As a result, propellant performance (Isp) effects the propellant selection very forcefully. Some further design trades will be needed regarding numbers of engines and throttling capabilities. Throttling ranges in Table 4.3.3-1 are fairly broad, varying from

9:1 to 14:1. The Apollo LM system had a 10:1 range. Multiple engines may be indicated by this, although maintenance requirements will then be increased. Since the engines will be reused, active thrust chamber cooling will be required. This fact, in combination with the high throttling range, eliminates pressure-fed engines. The variety of feed pressures and flow rates tend to be incompatible with active thrust chamber cooling systems. Other design issues such as serviceability and reliability must be considered when engine selection or development is undertaken.

The qualitative factors such as propellant storability and availability also indicate a wide variation between systems. The weights applied to these factors are coupled with the lunar base itself but tend to favor cryogenic systems. Trades between systems must consider not only the base but the state of base development when the BALTRAN is introduced. If some propellants are available on the surface from insitu processes, the trades will be weighted in favor of use of these propellants. If the base is in position to support and fuel lunar descent and ascent vehicles then the selection of propellants will be weighted on the side of the same system for the BALTRAN.

Cryogenic propellants are not easily stored without boiloff losses. Liquid hydrogen temperatures are very low (-253 °C) and heat transfer is high. Propellant boiloff will be a problem during the mission. These propellants are, however, not toxic and will tend to vaporize and dissipate in a spill. Depending on the particular lunar base emphasis and the timing, the propellants may be available locally. Currently, schemes for producing hydrogen on the Moon are not considered promising. Hydrogen deliveries from Earth are the lightest solution. On the other hand, almost all permanent lunar base schemes include some type of oxygen production. Overall, the all-cryogenic systems seem to be the safest and offer the best possibility of supply independent from Earth.

The hybrid systems tend to be better than all-cryogenic systems from a storage loss standpoint since the fuel can be stored at room temperatures (-18 to 93 °C). Liquid oxygen is still cryogenic though, and will require some special provisions. The fuels will tend to be toxic and crew safety issues will begin to become complex. Crews returning to the habitable volume will have to be monitored to ensure that no residual fuel is present on their suits. Hybrid systems can take advantage of the local oxidizer supplies but since mixture ratios are smaller than for the all cryogenic systems, the advantages are not as great. The hybrid system will result in the import of 3.5 times the imported cryogenic propellants.

Systems using all storable propellants allow storage without significant boiloff losses; in fact freezing may become a problem. However, storage ease is the extent of their advantages. Both the oxidizer and fuels are toxic and corrosive which would make maintenance very difficult. Safety procedures will be rigorous; especially if the propellants are hypergolic. In addition, none of the propellants are currently thought to be available locally. As much as 15 times more propellants must be imported to support a storable propellant BALTRAN than a cryogenic BALTRAN with surface oxygen production. Because the storable propellant performance is inferior to cryogenic, supplies will be costly and crew hazards will be high. Storables are generally unsatisfactory.

Table 4.3.3-1 Rocket Propulsion Transportation Factors and Options

FACTOR	UNITS	CRYOGEN LOX/LH2	HYBRID LOX/ STORABLE	PUMPED STORABLE
Delta V	m/s	6,720	6,720	6,720
Isp	sec	460	370	340
Mass of Vehicle (Dry)	kg lbm	9,600 21,100	9,800 21,600	10,600 23,300
Mass of Vehicle (Wet)	kg lbm	42,600 93,720	62,500 137,500	79,700 175,340
Mass Ratio		4.4	6.4	7.5
Propellant Mass	kg	33,000	52,700	69,100
Fuel	kg	4,700	17,600	26,600
Oxidizer	kg	28,300	35,100	42,500
Propellant Vol.	m^3	90	50	60
Fuel	m^3	70	20	30
Oxidizer	m^3	20	30	30
Engine Thrust	newtons lbf	139,200 31,300	204,200 45,900	260,400 58,500
Minimum Throttle Setting		11%	8%	7%
Tankage Fraction [Ref 13]	N/A	0.04545	0.0228	0.0228
Mixture Ratio		6	2	1.6
Storability	N/A			
Fuel		Poor	Good	Good
Oxidizer		Fair	Fair	Good
Availability	N/A			
Fuel		Import/Loc	Import	Import
Oxidizer		Local	Local	Import
Safety/Toxicity				
Fuel		Fair	Poor	Poor
Oxidizer		Good	Good	Poor

4.4 Thermal Control

The thermal control system is responsible for maintaining temperatures of interior and exterior subsystems. The thermal control system consists of a passive and active system. The passive control consists of external insulation, shielding, and thermal isolation systems. The active thermal control system is made up of heat sinks, cooling loops, and heat exchangers. Any subsystem requiring a heater should be designed with its own heating system. The thermal control system must be capable of maintaining a shirt sleeve atmosphere while being exposed to the lunar night and day temperature fluctuations.

4.4.1 Design Options

The passive thermal control system may consist of three major components; insulation blankets, thermal coatings and thermal isolators. Heat sinks and heat generators can also be effectively utilized on the lunar surface transportation systems. The thermal blankets can be constructed of fibrous bulk and multi-layer materials (weighing approximately 2 pounds per cubic foot, see Table 4.4.1-1) similar to the Shuttle insulation. The Space Station module insulation consists of 20 layers of organically coated aluminized film with dacron mesh separators weighing approximately 0.25 pounds per square foot. Thermal isolation techniques such as vacuum isolation and utilization of materials with minimal heat transfer characteristics can be used to thermally isolate the cabin structure from its exterior support structure. Many new plastics (ie. polycarbonates, polyurethane, and phenylene oxide) and advanced composite materials (carbon/graphite, carbon/epoxy and carbon/glass fibers) exhibit such heat transfer characteristics. Another technique for shielding the vehicle from solar heat is through the use of movable sun screens (baby buggy covers). With proper coatings these shields could either reflect or absorb heat depending on the need of the vehicle. Several coatings which have been used on various spacecraft in the past are compared in Table 4.4.1-2.

The active thermal control system will be required to reject heat buildup from the propulsion/power, payload and avionics systems as well as solar radiation. The active thermal control system utilized in the Space Shuttle is an excellent example of the options available for the lunar vehicles. This system consists of three methods of heat rejection: radiators, flash evaporators (water vaporizer) and ammonia boilers. The radiators and associated cooling loops would be a closed system. This closed system can be either a single or double phase design. The flash evaporator and ammonia boilers are both open systems. Another form of transporting heat is through advanced heat pipe designs. Heat pipes can be used in place of a fluid circulation lines and in radiator designs similar to the proposed Space Station type.

The MOSAP and BALTRAN will both require similar passive and active control systems since they both are pressurized vehicles. Thermal loads will be significantly different between these two vehicles which will require separate trade-offs on the specific design criteria. The LOTRAN thermal control system needs to maintain a nominal temperature for electronics equipment using heaters and remove any heat build-up from the on board sub-systems.

4.4.2 Comparison Factors

The primary comparison factors for the thermal control system are safety, reliability, efficiency (performance) and weight. Safety is the primary concern of any system and appropriate safety factors must be factored into the design of both the passive and active systems. Many insulation and heat transfer materials can be toxic to humans especially when exposed to the space environment. Adequate pressure relief/control must be designed into fluid loops where heat build-up may occur.

The passive systems selection will be based on its ability to obtain a heat balance that will not result in local internal cold spots of pressurized cabins or excessive heat leaks. The weight of the isolation techniques is determined by the material strength and the design features of the vacuum barrier material. Surface coating techniques are relatively light when compared to the other two methods.

The active system major design criteria should be its reliability, efficiency, and maintainability. The crew's survival is dependent on the operation of this system. When determining whether a single or two phase flow should be used, the comparison of these factors is very important. The weight of the system will be a secondary factor in selecting which system or combination of systems (radiators, flash evaporators or boilers) is utilized.

4.4.3 Comparison Analysis

The lunar vehicles will be exposed to widely varying temperatures from -233 to 127^{o}C (40-400^{o}K) during their respective missions and therefore must be designed to isolate the pressurized cabin from its exterior environment. The vehicle and its propulsion system can be viewed as a heat source which will require some sort of heat rejection capability. Much of this heat can be used to keep the vehicle warm during cold soaking periods (night time). However, during hot soaking periods (day time) the thermal control system must be designed to reject excess heat.

Rejecting heat from the vehicle is limited by the surface area of the radiators and the amount of water and ammonia which can be carried on the vehicle. Therefore the passive system should be designed to isolate the vehicle from varying temperatures and minimize the solar heat absorption. A vacuum barrier between the inner liner of the cabin and the exterior shell would create a thermos type of isolation from the changing temperatures. Thermal conduction from the cold or hot exterior shell to the inner shell will be decreased if materials with low thermal conductivity coefficients, such as plastics or carbon graphite type materials, are used to support the vacuum structure. A honeycomb design for the vacuum shell would provide a support structure for the outer shell layer. Thermal isolation between any heated subsystem and the cabin can be enhanced by minimizing the attach points of these subsystems to the cabin shell.

The weight of extra thermal blankets and thermal isolation materials is significantly less than that of extra radiators, water, and/or ammonia used in the active systems. These insulation materials currently exist and are used on the Space Shuttles exterior surfaces. A detailed analysis of the lunar heat transfer (radiation) characteristics is necessary to determine how much insulation versus the number of radiators will be required on each vehicle. Any radiators will require wiper blades or at least be accessible to the crew to prevent dust build-up.

The active thermal control system used in the Shuttle is a proven system which is also being looked at for use in the Space Station. The system consists of single phase flow freon loops which circulate through a combination of coldplate networks, heat exchangers, and three heat sinks (radiators, flash evaporators, and an ammonia boiler). Reliability of the system design will increase if moving parts can be eliminated wherever possible such as in the pump design. Generally speaking, pumps used in single phase flow system have less moving parts than that of a compressor used in a two phase flow system. Current technology for fluid pumps has eliminated many reliability/maintainability problems associated with mechanical shaft seals (required in gas compressors) by utilizing a "canned" pump design. A two phase flow system would also add complexity due to the need for a condenser and a cool condensing medium. By designing the radiators as the primary means of radiating heat to space, the amount of water and ammonia needed for the flash evaporators will be minimized. Water produced from the fuel cells could be used for the flash evaporators, but ammonia used in the boiler would have to be carried as a weight penalty. The three possible heat rejection options are compared in Table 4.4.3-1. Water changing from liquid to vapor has a cooling capacity of about 1,000 Btu per pound while ammonia has approximately 500 Btu per pound. Another method of transferring heat is through advanced heat pipes. These devices usually do not have any moving parts so reliability can be relatively good. The three heat transportation methods described above are characterized in Table 4.4.3-2.

Current technology for the Shuttle radiators utilizes aluminum panels (2) with tubes bonded to the internal sides of the facesheets (reflectors). New coatings should be investigated for more efficient radiation of heat. Based on heat rejection loads of the Shuttle radiators, the MOSAP vehicle could be expected to handle loads of 15,000 Btu's an hour. Radiators utilizing heat pipes are being considered for Space Station. The rational for using this type of radiator is primarily based on weight savings.

For safety concerns a toxic cooling fluid should not be circulated inside the pressurized cabin in case of a leak. Therefore, such a system requires that an extra heat exchanger be located outside the cabin. Several types of heat transfer mediums other than freon should be investigated for their toxicity levels so that this extra heat exchanger could be eliminated. Efficiency is decreased every time an extra heat exchanger is placed in the system. As depicted in Table 4.4.3-3, the use of cold plates versus an air-to-air heat removal method may require less space but is generally more complex. Realistically, the equipment heat rejection subsystem will probably require a combination of coldplates and airducts.

Table 4.4.1-1 Properties of Various Insulating Materials

MATERIALS	k (Btu/hr-ft^2-^{o}F/in.)	DENSITY (lb/ft^3)	PRESSURE ON INSULATION
Silica Aerogel	0.12	8.5	1 ATM
Polyurethane Foam	0.14	2.0	1 ATM
Teflon	0.60	130.0	1ATM
Multilayer (S-I) Insulations			
S-I 10	0.00078	2.5	10^{-4} mm Hg
S-I 12	0.00108	2.0	10^{-4} mm Hg
S-I 44	0.00024	4.7	10^{-4} mm Hg
S-I 91	0.00012	7.5	10^{-4} mm Hg

Table 4.4.1-2 Control Coating Comparisons

COATINGS vs CRITERIA	ZINC OXIDE	BLACK ANODIZE	SILVER TEFLON	CHROMIC ACID ANODIZE
Absorbance/ Emittance Ratio	0.2/0.8	0.8/0.8	0.2/0.8	0.3/0.6
Heat Rejection Rate (BTU/hr ft^2)	20	--	25-30	--
Potential Uses	Radiator Coating Any Heat Generating Body	Heat Sinks: External Water Lines	Radiator Coating Any Heat Generating Body	External Chassis or Structural Components

Table 4.4.3-1 Active Thermal Control Heat Rejection Options

CRITERIA	RADIATOR	WATER VAPORIZER	AMMONIA BOILER
Safety	Very Good	Fair	Fair-Poor
Consumables Required	NO	YES	YES
Performance: Heat Rejection Rate (Capacity)	25-30 $BTU/HR\text{-}FT^2$	(1,000) BTU/lb of H_2O	(600) BTU/lb of NH_3
Maintainability:			
-Dust Susceptability	HIGH	LOW	LOW
-Failure Rate	LOW	MEDIUM	MEDIUM
Mass (kg/m^3)	167 *	998 H_2O @ 15^oC	596 NH_3 @ 30^oC
(lb/ft^3)	10.4	62.3 H_2O @ 60^oF	37.2 NH_3 @ 86^oF

* Estimated density of Shuttle radiators.

Table 4.4.3-2 Passive Thermal Control Heat Transportation Methods

CHARACTERISTICS	SINGLE PHASE FLUID	TWO PHASE FLUID	HEAT PIPE
Safety			
-Operating Pressure	LOW	HIGH	LOW-MEDIUM Fluid Dependant
-Toxicity	********Fluid Dependant********		
-Flammability	********Fluid Dependant********		
Reliability	HIGH	FAIR	FAIR
Performance			
-Pumping Cost	FAIR	HIGH	LOW
Complexity			
-Controls	Simple	Nominal	Complex
-Manufacturing Difficulty	Simple	Nominal	Complex

Table 4.4.3-3 Options For Removing Heat From Equipment/Hardware

CHARACTERISTICS	COLDPLATE	AIR DUCT
Weight (kg/watt) (lb/watt)	 1.6×10^{-3} 3.5×10^{-3}	 2.2×10^{-2} 4.8×10^{-2}
Volume (cm^3/watt) ($in.^3$/watt)	 6.23 0.38	 62.3 3.8
Power (watt/watt)	6.3×10^{-4}	0.16
Electronic Load (watts/m^2) (watts/ ft^2)	 3,100 288	 2,583 240
Manufacturing Difficulty	HIGH	LOW
Integration Difficulty	HIGH	LOW

Reference: Lunar Base Synthesis Study, Final Report, Vol. III Space Division, North American Rockwell.

4.5 Pressure Vessels

The overall configuration of the pressurized vessel (cabin and airlock) is partially constrained by the Earth to Moon transfer vehicle capabilities. The most critical constraints are size and weight. To minimize the required pressure vessel weight a spherical (highly impractical for fabrication of internal structures) or cylindrical shaped vessel should be used. An outer shell will serve as a passive insulator and as structural reinforcement. The total weight of the pressure vessel (inner and outer shells) can be minimized by utilizing new carbon/graphite composite materials for structural reinforcement.

4.5.1 Design Options

4.5.1.1 Shell Materials

The conventional material used for the inner liner of pressurized cabins (Space Shuttle and Apollo) is 2219 aluminum alloy with integral stiffening stringers and internal framing welded together. Several options exist for the outer shell material which will provide meteoroid protection, structural strength and passive insulation. These include:

1) aluminum inner shell with stainless outer shell
2) aluminum inner shell with carbon/graphite composite outer shell
3) aluminum inner shell with titanium outer shell
4) aluminum inner shell with aluminum outer shell

4.5.1.2 Shell Configuration

There are four basic design options for the shell configuration:

1) cylindrical with spherical end caps (rounded)
2) cylindrical with flat end caps
3) non-cylindrical (boxed shaped or other polygon)
4) elliptical

4.5.2 Comparison Factors

The primary comparison factors for the pressure vessel are safety, the materials strength to weight ratio, manufacturing capability, and overall configuration. Safety is obviously the most critical factor for the pressure shell and therefore appropriate safety factors must be added to whichever option is chosen. Typically a pressure vessel will be designed to contain four times the maximum working pressure which in this case is estimated at 69 kN/m^2 (10 psi). Other factors which must be considered in the vessel design are the weight of normal contents, impact loads applied from sudden pressure increases or externally applied shock/vibration loads, and the effects of temperature gradients. These additional loads will contribute to the amount of stiffeners and structural reinforcements required for the final design. Minimizing weight is critical for launching vehicles and surface propulsion systems. The carbon/graphite materials appear to have the highest strength to weight ratio (1.15) followed by titanium (0.58), aluminum (0.5), and stainless (.17). The maintainability and machining properties of aluminum make it the best suited for welding, machining and forming. The strength and hardness of titanium make it much more difficult to work with than aluminum. The stainless steel weight penalty makes it very undesirable for a shell material. The technology for forming and shaping carbon graphite materials is rapidly making it more viable to manufacture complex shapes. None of these materials are significantly effected by radiation. See Table 4.5.2-1 for a materials comparison.

4.5.3 Comparison Analysis

The pressurized cabin must be supported by the vehicle suspension system to minimize shock and vibration loads from travel over rough terrain. The strength to weight ratio favors the combination of an aluminum inner shell and a carbon/graphite composite outer shell. The aluminum inner shell is a proven material for lining pressurized habitable enclosures due to its weight to strength ratio and its manufacturing capability. Depending on the actual graphite material used, a weight savings can range from 15% to 40% over conventional metals. The yield strength of these composites can reach as high 1,516,847 kN/m^2 (220 ksi) as compared to aluminum of 503,317 kN/m^2 (73 ksi). Current uses for these carbon/graphite materials include the canister for the MX missile, the Navy's F/A-18A jet fighter structures, the Shuttle orbital maneuvering subsystem reaction control system skin cover, and the payload bay doors. Titanium, possessing yield strengths from

482,633 to 1,103,162 kN/m^2 (70 to 160 ksi), offers some weight savings over aluminum. However, it is also known for being difficult to weld and machine. Stainless steel offers no weight savings.

A cylindrical shaped pressure vessel requires less wall thickness than a boxed shaped vessel to contain the same internal pressure. A flat sided vessel requires many stiffeners and reinforcements to counter the resulting stresses. The end caps should also take on a rounded configuration rather than a flat end to reduce weight and simplify the manufacturing process. A spherical shape requires less wall thickness than a cylinder, however, it is more difficult to interface hardware to structure surface within a sphere or ellipse. An elliptical shaped vessel would offer more floor space for working, but manufacturing an ellipse is difficult and additional stiffening is needed.

The internal volume of the vessel is dependant on the estimated volumes of the following:

Personal hygiene station	1.7 m^3	(60 ^{f}t3)
Airlock	5.7 m^3	(200 ft^3)
Galley	1.7 m^3	(60 ft^3)
Emergency equipment	0.8 m^3	(30 ft^3)
Cockpit	2.5 m^3	(90 ft^3)
Experiments	0.8 m^3	(30 ft^3)
Avionics	1.7 m^3	(60 ft^3)
ECLSS	2.5 m^3	(90 ft^3)
Workstation	1.7 m^3	(60 ft^3)
Sleeping area	6.8 m^3	(240 ft^3)
Working area	16.3 m^3	(576 ft^3)
Miscellaneous equipment	5.7 m^3	(200 ft^3)

This totals approximately 57-71 m^3 (2,000-2,500 ft^3) for the estimated volume of the MOSAP. Based on a safety factor of 4 and an inner and outer shell constructed of aluminum the approximate weight of the shells alone (no structural reinforcements) would be 2,045 kg (4,500 lbs). The following assumptions were made. The inner shell diameter equals 4.3 m (14 ft) with a length of 4.9 m (16 ft). The outer shell has a diameter 4.34 m (14.25 ft) by 5.0 m (16.5 ft) in length and both have a thickness of 0.4 cm (0.16 in). The end cones, both inner and outer, are aluminum half spheres with a thickness of 0.2 cm (.080 in) and a diameter of 4.3 m (14 ft).

View ports will be incorporated throughout the cabin for various operational and experimental reasons. The frame/support structure for the view port should be designed for changing out either the inner or outer panel without violating the pressurized cabin. It is anticipated that the outer view port material will eventually become marred from debris or sunlight and require periodic maintenance.

Table 4.5.2-1 Material Properties

MATERIAL	YIELD STRENGTH (ksi)	DENSITY (lb/ft3)	MACHINING CAPABILITY	STRENGTH/WEIGHT RATIO
Aluminum 7075-T6	73	166	GOOD	0.5
Carbon Graphite	120-220	104	FAIR	1.15
Titanium Ti-3Al-8V-Cr-4Zr-4Mo	160	276	FAIR-GOOD	0.58
Stainless 8Cr-2Mo	80	466	FAIR-GOOD	0.17

4.6 Airlocks

4.6.1 Design Options

The function of an airlock is to allow the crew and equipment inside the pressurized part of the vehicle to go outside without depressurizing the entire vehicle.

Airlocks will be required on both the Mobile Surface Applications Vehicle (MOSAP) and the Lunar Ballistic Sortie Vehicle (BALTRAN). Only two crewmen are generally required to leave the vehicle at any one time. Depressurizing the entire vehicle for each Extravehicular Activity (EVA) would require that all crew don pressure suits. In the situation where there are not sufficient operating pressure suits for all crewmen, no further EVA would be possible. Therefore, the vehicle would have to carry enough pressure suits for all crewman plus spares. If an airlock is present, only the two EVA crewman need suits. Two EMUs plus critical spare parts provide fault restoration sufficient to continue the nominal mission after most failures. For reference, each EMU weighs 136-227 kg (300-500 lbs.) and requires a minimum of 0.5 m^3 (18 ft^3) of storage.

The key vehicle airlock related issues are related to the design of the airlock and operational issues. Major airlock related options include: 1) atmospheric pressure in vehicle and EMU, 2) volume and type of airlock, 3) extent of gas recovery during airlock depressurization, and 4) extent of EMU servicing and repair in the vehicle.

4.6.1.1 Atmospheric Pressure

The recommended pressure of the vehicle and EMU will affect the amount of prebreathing required of the crew doing an EVA. Airlock pressures can range from about 69 kN/m^2 (10 psia) to normal sea level atmospheric pressure of 101 kN/m^2 (14.7 psia). Figure 4.6.1.1-1 shows the relationship between cabin pressure and suit pressure at different levels of prebreathing.

4.6.1.2 Airlock Volume and Type

Several options are available for the airlock. These options include:

1) Rear Entry EMU/Airlock Concept:

The pressure suits themselves are used as an airlock by developing a closure for a rear entry type hard upper torso (HUT) that fits to a hatch. In such a scheme the crew climbs into the suit, the life support unit is closed and sealed over the entry port. Then an inner hatch attached to the vehicle is closed over the entry, the EMU is undocked from the vehicle, and the pressure connection broken. The effective volume of such an airlock is less than 0.06 m^3 (2 ft^3). The interior volume of the EMU is about 0.14 m^3 (5 ft^3), but the crew displaces about 0.11 m^3 (4 ft^3).

2) Man-lock Concept:

The second option is a tightly fitting container that conforms to the shape of a single crewman in an EMU. In such a design the astronaut in an EMU climbs into the tightly fitting box by opening the pressurized side. The box is then closed, the pressure is relieved, and the unpressurized side is opened. The effective volume, estimated to be about 0.3 m^3 (10 ft^3), of such an airlock is the difference between the volume of the pressure suit and the airlock.

3) Conventional Airlock:

In a conventional airlock, both crewman enter a cylindrical or spherical chamber with a volume of 7-9 m^3 (250-300 ft^3). The gas in the airlock is pumped out and into the main vehicle cabin or a dedicated tank or inflatable bladder. After about 94% of the gas is pumped out the remainder is vented and the crew opens the outer hatch.

4.6.1.3 Extent of Gas Recovery

Gas recovery on the Space Station is done using a four stage turbopump with four intercoolers. The unit is estimated to weigh about 45 kg (100 lbs). The pump takes the airlock pressure from 101 kN/m^2 (14.7 psia) to about 5 kN/m^2 (0.7 psia) in 5 minutes, and recovers about 94% of the gas, assuming that the pumpdown is nearly isothermal rather than adiabatic.

4.6.1.4 Extent of EMU Servicing

After each EVA, the EMU's consumables will have to be replaced, and a minimum cleaning and disinfecting be done to maintain crew health. These functions are considered to be mandatory requirements for the vehicle and include:

1) cleaning, drying, and wiping down the interior of the EMU with biocide
2) cleaning, drying, and applying biocide to the liquid cooling and ventilation garment (LCVG)
3) recharging batteries
4) replacement of CO_2 absorbing media (LiOH) or breaking down the absorbing media if a system such as a metal oxide is chosen which decomposes at modest temperatures
5) refilling of O_2 tanks
6) servicing the cooling system which may involve chilling a phase change material such as wax, and/or refilling water for the sublimater
7) servicing the devices collecting urine, feces and vomitus
8) refilling food and drink containers
9) updating cuff checklists and/or display screens for Heads Up Displays also called Helmet Mounted Displays in the EMU

The equipment to support all of these functions is called the Checkout and Service Equipment (CASE) on the Space Station. It is estimated to weigh about 477 kg (1,050 lbs). A substantial weight reduction program may be instituted that possibly could reduce the weight of the system by 50%. This technology should be well established long before the next generation of lunar surface transportation systems are designed.

Additional functions that may be required include fault detection and isolation, replacement of failed parts, and resizing the suits to fit more than one crewman.

4.6.2 Comparison Factors

The key comparison factors of the different airlock concepts are: 1) weight of airlock equipment, 2) consumption of gases when depressurizing the airlock, 3) convenience of operation, and 4) ability to continue function in event of failure of part of airlock or EVA system.

4.6.3 Comparison Analysis

The design concepts have been analyzed in terms of the stated comparison factors. The observations developed in this review are provided in the following paragraphs.

4.6.3.1 Atmospheric Pressure

The cabin atmosphere in the vehicle can be run at any pressure between about 38 kN/m^2 (5.5 psi) and 101 kN/m^2 (14.7 psi).

1) Weight Issues:

The cabin atmospheric pressure affects system weight in three ways: 1) increased cabin pressure results in a thicker cabin pressure vessel wall and hatches, 2) increased cabin pressure indirectly requires an increase in EMU pressure resulting in increased EMU weight and 3) higher cabin pressure results in a greater loss of gas during each depressurization cycle of the airlock. Thus, many factors drive the design to the lowest practical cabin pressure. One reason to maintain a high cabin pressure in the vehicles is for application of a pressure stabilized structural approach to supporting dynamic loads. Other reasons have to do with flammability and partial O_2 effects on the human body. The structural trade between increased pressure vessel wall thickness and vehicle rigidity is beyond the scope of this investigation.

2) Gas Consumption Issues:

No matter which approach to an airlock is selected, the amount of gas vented overboard at the beginning of each EVA is roughly proportional to cabin pressure. Thus the cabin should be run at the lowest possible cabin pressure.

3) Operational Convenience Issues:

The lower the pressure that the cabin is run, the lower the pressure that the EMU can be run and not require prebreathing by the crewman before entering the EMU. The function of the prebreathing is to reduce the nitrogen level in the crewman's blood so that he or she will not experience bends when exposed to the lower pressure of the EMU. In general, the lower the pressure of the EMU the greater its flexibility resulting in increased mobility. The Space Station baselined a cabin pressure of 101 kN/m^2 (14.7 psi) because of concern with interpretation of results of biological experiments. This relatively high cabin pressure required that the EMU operate at a pressure of 57 kN/m^2 (8.3 psi) or greater to eliminate prebreathing (Figure 4.6.1.1-1 from NASA STD 3000). Assuming that a pressure of 69 kN/m^2 (10 psi) is suggested for the vehicles then the EMU pressure can be as low as 26 kN/m^2 (4 psi) without the requirement for prebreathing.

4) Failure Tolerance:

The cabin pressure has little relation to the failure tolerance of the system.

4.6.3.2 Airlock Volume and Type

The Space Station airlock is a large chamber, 7-9 m^3 (250-300 ft3), because it must allow transfer of many modest size Orbital Replaceable Units (ORUs) and one of the airlocks also serves as a hyperbaric chamber.

1) Weight Issues:

The weight of adding a hatch to mate with the EMU entry plane is about 45 kg (100 lbs).

The weight of the structure of a 6 m^3 (200 ft^3) airlock will be at least 273 kg (600 lbs). The depress pump will add another 45 kg (100 lbs) and additional valves, plumbing, fittings and electronics such as an extra radio antenna will add at least another 68 kg (150 lbs) resulting in a minimum of 386 kg (850 lbs).

The weight of a man-lock roughly 1x1x2 m (3x3x7 ft) is 113 kg (250 lbs) including structure, hinges, and valves. The depressurized volume is small enough that adding a depress pump is not warranted.

2) Gas Consumption Issues:

The entry into the rear of an EMU effectively vents no air.

The density of air at 21°C (70°F) and 69 kN/m^2 (10 psi) is 0.8281 kg/m^3 (0.0517 lbs/ft^3). A 6 cubic meter (200 ft^3) airlock contains about 5 kg (11 lbs) of gas at 69 kN/m^2 (10 psi). Thus in 6 EVAs as much as 30 kg (62 lbs) of air could be vented.

A man-lock will vent about 0.3 m^3 (10 ft^3) of gas of mass 0.235 kg (0.517 lbs) each time it is depressurized. For a two man EVA it will be depressurized twice during egress and once during ingress. It remains depressurized during the EVA. Thus over a 6 EVA cycle, about 5 cubic meters (180 ft^3) of gas will be vented or 4.3 kg (9.3 lbs).

3) Operational Convenience Issues:

Use of the rear plane of the EMU as the entry into the airlock will not allow replacement of any of the EMU's pressure seals, such as the couplings at the wrist, elbow, or foot. These couplings serve to connect items which receive considerable wear and tear during an EVA including the sizing elements which allow the EMUs to be adjusted to fit different size crewman, and the gloves which are specific to individual astronauts. Thus the rear entry method is not practical as the sole airlock.

The conventional airlock is the most convenient, allowing both crewman to egress or ingress simultaneously. This concept also allows large equipment to be taken in and out with the crew.

The man-lock only allows one crewman to egress or ingress at a time which will add several minutes to the total time of each EVA; however, with proper scheduling of activities, no significant operational impact will be encountered. The principal function that will be lost is the ability to pass tools and equipment into and out of the pressurized volume with the crewman. This function is not anticipated to be significant since most of the tools and equipment should be stored outside the pressurized volume. Most samples will also have to be stored outside the pressurized volume. In the event that an extraordinary event requires that something be passed into or out of the pressurized volume, it can still placed in the airlock without a crewman. Two man-locks could be implemented to allow simultaneous 2-man passage while also providing redundant systems.

4) Failure Tolerance Issues:

The rear entry method as the sole airlock approach does not allow for a repair in the event of failure of any of the EMU pressure seals. Therefore, the concept is not acceptable.

The conventional airlock can be designed to meet normal fault tolerance requirements as it has on the Space Station.

The man-lock can be provided with redundant manual valves equipped with screw on pressure sealing caps. Such a system will allow full functionality in the event of either a failed open or failed closed valve. Double O-ring seals will probably be required to insure adequate sealing in the event that one of the O-rings is cut or otherwise improperly sealing. Replacement of leaking O-rings should be possible using a combination of IVA or EVA activity depending on which seal is leaking.

4.6.3.3 Extent of Gas Recovery

The airlocks can be depressurized by simply opening them to space, as has been done on Gemini, Apollo, Skylab and Space Shuttle. The Space Station will be the first U.S. program to recover gas during depressurization.

1) Weight Issues:

Extensive studies during Space Station Phase B by Work Package 2 contractors has determined that several pump types are available that can recover at least 94% of the gas in an airlock. The baseline Space Station pump evacuates an 8 cubic meter (280 ft^3) airlock and draws about 5 kw for 5 minutes. The pump weighs about 45 kg (100 lbs). Thus, for the conventional airlock, mass payback occurs in 6-EVA missions. About ten full 6-EVA missions would be required to achieve similar mass payback for the man-lock.

2) Gas Consumption issues:

The practical limit to gas recovery is about 95%; the Space Station will achieve about 94%. Assuming 94% gas recovery, gas loss can be reduced to about 1.8 kg (4 lbs) for a 6-EVA mission using the 6 cubic meter airlock..

3) Operational Convenience Issues:

Operating a depressurization system adds five to ten minutes to each EVA. Although some time can be used to check out the EMU pressure integrity a portion of the time is effectively lost.

4) Failure Tolerance Issues:

A gas recovery system requires a high speed pump. In the event of pump failure, a spare pump could be installed or the vehicle could carry enough extra air to allow gas venting during each EVA. If the crew is not put in danger by the loss of extensive EVA capability, it may be deemed acceptable to carry only enough extra air to support one or two EVAs without gas recovery.

4.6.3.4 Extent of Vehicle Servicing of the EMU

The baseline vehicle design will have to support a concept similar to the Checkout and Service Equipment (CASE) located in each Space Station Airlock. Additional repair of failed EMU components will require approximately one cubic meter (30-40 ft^3) of storage for spares and sizing elements and extensive software for fault detection and isolation.

1) Weight Issues:

   The exact volume of spares that must be taken on a vehicle is not well established. However, the maximum volume is approximately equal to the volume of the most complicated EMU parts--the helmet and the back pack or Life Support System. These units, when broken down to components, take less than 1 cubic meter (20 ft^3) of gross storage assuming a 50% packing factor. Suit sizing elements and gloves compress to small volumes and may be considered to fit in that volume.

2) Gas Consumption:

   No comparisons applicable.

3) Operational Convenience:

   The technology and software for EMU fault detection and isolation is being developed for the Space Station EMU. The technology should be mature in time for the next generation lunar surface EMU. Ideally a computerized system will identify and recommend to the crew the corrective action. Proper EMU design will allow ready replacement of virtually any component.

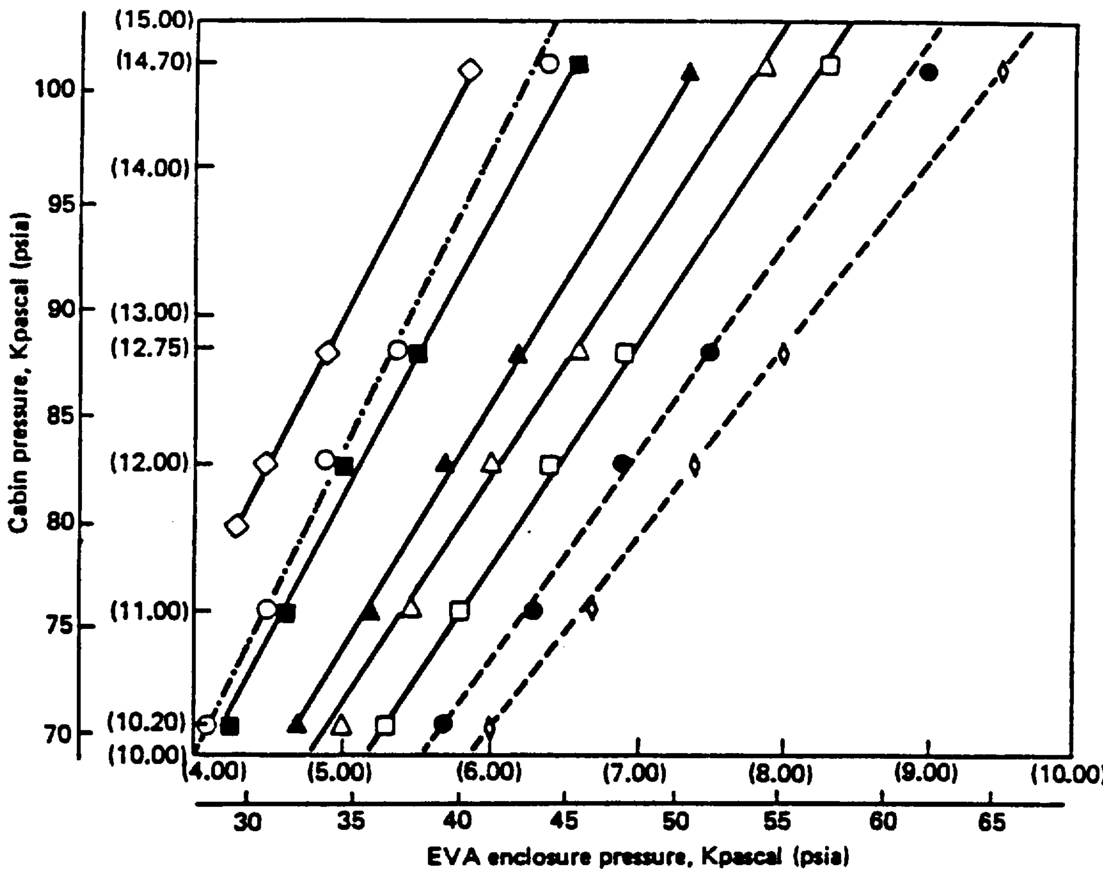
4) Failure Tolerance:

   The ability to repair the EMU's remotely from the lunar base will certainly increase the ability to carry out the preplanned vehicle/EVA missions with a minimum deviation. Failure to provide for vehicle based servicing will certainly decrease vehicle scientific and resource exploration effectiveness and increase the hazard to the crew.

4.6.4 Evaluation Comments

The recommended airlock concept is a man-lock which can hold one crewman in an EMU at a time. The man-lock concept is the minimum weight per individual traverse mission. The vehicle should operate at a pressure low enough to not require the EVA crew to pre-breath before entering a low pressure EMU. Nominally a pressure of 69 kN/m^2 (10 psia) is recommended for the vehicle. Recovery of depressurization air might be desirable on the MOSAP which is assumed to be less weight sensitive than the BALTRAN. However, even for the MOSAP, several traverses are required for enough air to be saved to make up for the weight of the pump and thus a weight break even. No air recovery should be baselined for the BALTRAN, since the fuel required to move the pump around will certainly exceed the mass of any diminished gas that must be carried. The man-lock without air recovery can provide full functionality and fault tolerance at a minimum weight. The vehicle should be able to supply the EMU consumables, and provide all cleaning necessary for reasons of health and sanitation. The vehicle should also carry adequate spares to replace the most failure prone EMU components so that the traverse mission can be successfully completed even if part of an EMU breaks down.

Figure 4.6.1.1-1 Prebreathing Relationship Between Cabin Pressure and Suit Pressure (from Ref. 42)



- ◊ R = 1.22, No O_2 prebreathe
- ● R = 1.22, 30 minutes O_2 prebreathe
- □ R = 1.40, No O_2 prebreathe
- △ R = 1.40, 30 minutes O_2 prebreathe
- ▲ R = 1.40, 1 hr O_2 prebreathe
- ■ R = 1.40, 2 hr O_2 prebreathe
- ◇ R = 1.40, 3 hr O_2 prebreathe
- ○ R = 1.80, For purge operation suit pressure, not to exceed 30 minutes exposure

	R = 1.22	R = 1.40
% Symptoms	0	5
% Bubbles	20	35
Maximum grade of bubbles *	2	4

* Bubble grades

0 - None

1 - Bubbles in less than 1/2 heart sounds

2 - Bubbles in more than 1/2 heart sounds

3 - Bubbles in each heart sound

4 - Bubbles obscuring heart sounds

4.7 Environmental Control and Life Support System (ECLSS)

4.7.1 Design Options

4.7.1.1 LOTRAN Options

The basic LOTRAN ECLSS design option is whether the vehicle has an ECLSS or whether the ECLSS is provided by crew operation in an independent EVA suit. In one case the crew person travels in the LOTRAN and works at the LOTRAN site in a Lunar Surface EMU, which is a self-contained EVA suit identical to the one used at the lunar base. In the case of a pressurized cab with an ECLSS, the crew member can open his suit visor, but must still wear the EVA suit. An airlock is not consistent with the lightweight design strategy for the LOTRAN. The helmet is not normally a removable unit in a rear entry EMU which reflects the current US and USSR state of the art. In both cases, the crew returns to the lunar base to reservice the suit including drying, cleaning, regenerating the CO_2 absorbing media, servicing the cooling system, changing the batteries, and removing urine, feces, and vomitus.

4.7.1.2 MOSAP Options

The MOSAP must have a dedicated ECLSS because the crew will be living in and working from the MOSAP for up to 42 days. The life support system must be reliable and be a derivative of the lunar lander vehicle ECLSS. As a derivative, the MOSAP ECLSS will minimize development costs, minimize crew maintenance training time, maximize access to spares on the lunar surface, and minimize launch weight requirements. The major consideration is the extent to which the system is open compared to a system which is regenerated. Other studies [Ref. 64] have indicated that a partially closed regenerative ECLSS does not compete on a mass basis with open systems until time periods exceed a month.

4.7.1.3 BALTRAN Options

The BALTRAN must have a dedicated life support system because the crew will be living and working from the BALTRAN for up to 8 days at a time. The trade between a totally closed regenerative system and partially closed regenerative system has the same options as the MOSAP. However, the BALTRAN is more sensitive to the weight and power requirements of a closed regenerative system. Therefore, by using the same ECLSS concept as the lunar lander vehicle, the BALTRAN will minimize ECLS weight requirements. The ECLSS must be reliable, a derivative of the lunar lander system to minimize development costs and crew maintenance training time, and inexpensive to deliver to the moon from earth.

Because of the maximum duration of 8 days and the remoteness of the BALTRAN locations, the crew will do critical repairs on the life support system, and the system will require substantial redundancy and failure tolerance.

4.7.2 Comparison Factors

4.7.2.1 LOTRAN Design Comparison Factors

The main comparison factors are vehicle weight increments, power requirements, atmosphere consumables, operations considerations, and cost.

4.7.2.2 MOSAP and LOTRAN Design Comparison Factors

Three design areas for comparison of types of ECLSS design options are:

1) Absorbtion of CO_2 from the cabin air,
2) Water removal from the cabin air, and
3) Method of heat rejection.

The CO_2 removal system can either remove the CO_2 in a non-regenative media such as LiOH which is dumped, or remove the CO_2 and store it for regeneration in the vehicle or back at the base. Water can be removed from the air and then decomposed to H_2 and O_2 for reuse, or it can be dumped, possibly through a sublimator. Heat rejection for the vehicle can be done using water sublimator or radiators.

The performance comparison factors of interest are :

1) Operations Duration
2) Maintenance Requirements
3) Mobility
4) Power Required
5) Hardware Weight
6) Consumables Weight
7) Total Weight

4.7.3 Comparison Analysis

4.7.3.1 LOTRAN Comparisons

The comparison of the crew wearing an EMU was made against the crew wearing an EMU and sitting in a LOTRAN cab with a life support system. The LOTRAN cab ECLSS equipment adds weight to the vehicle (over 1,300 lbs.), requires additional power, and fuel to generate the power and wastes approximately 6 lbs. of atmosphere for every cab entry or exit. It also requires additional development cost for the LOTRAN, adds to the mass to be delivered to the lunar surface, requires additional crew training time for maintenance and operations, and requires maintenance and spares. The only advantages are that the crew can lift up their EMU visors when in the cab putting slightly less demand on the EMU consumables and, if the EMU develops a leak too large to make it back to base but small enough to return to the LOTRAN, the redundant LOTRAN cab allows safe crew return to base. The crew must continue to wear the EMU while in the LOTRAN cab.

Since the crew will always be within convenient returning distance from the lunar base, the crew uses their EMU's for life support. As explained above, there are no significant advantages to the LOTRAN carrying a life support system and there are several disadvantages. The objective of the LOTRAN is to provide inexpensive, local transportation around the immediate area of the lunar base. This is a circle with an approximately 50 km radius.

To remain within the constraint to make the LOTRAN as low weight as possible, no vehicle life support is baselined. The crew wears an EMU evolved from the proven

Apollo and Shuttle suit designs which will have further evolved through the Space Station to the Lunar Base design. No additional development or production funds will be required to meet the LOTRAN objectives of lunar base maintenance, operations, and exploration.

For a contingency where one crewperson's suit fan may fail, an umbilical kit will be carried on the LOTRAN which allows one EVA suit to provide consumables and cooling to two crewpersons. The lowest weight solution will be a moderately light (about 35 lbs.) BSLSS (Buddy System Life Support used on Apollo). A LOTRAN cabin life support system does not add to the crew's performance or safety.

4.7.3.2 MOSAP Comparisons

The ECLSS options are summarized by design factor for the MOSAP in Table 4.7.3.2-1. The added weight from the loss of consumables in an open system must be traded against the extra weight of an ECLS system which completely recycles CO_2 and H_2O on the vehicle or at the base. The two advantages of a closed system are the reduced environmental pollution with H_2O and the reduced fluid load that must be delivered from the earth. Option 1 has a disadvantage because of the large O_2 and H_2O losses and venting pollution which will impact scientific instruments over time. Only Option 2, in which the spent consumables are returned to the base for recycling and 3, in which the spent consumables are completely recycled by the vehicle, will be compared. Because of the maximum duration of 42 days, the crew will do regular maintenance on the life support system.

The MOSAP partially closed, non-regenerative system is compared to the totally closed regenerative system in Table 4.7.3.2-2. The closed loop system is a more efficient use of consumables but an unnecessarily costly addition to the design of the MOSAP. By using the partially closed system, 1,000 pounds are saved not including the reduction in power requirements. The waste products from metabolism, CO_2 and H_2O, can be collected and returned to the lunar base for inclusion in the lunar base consumables recycling process. The extra expense, crew distraction, and increased maintenance time for a totally closed design do not add to the safety or capability of the MOSAP.

The disadvantage of loosing the H_2O to the lunar atmosphere is avoided by returning the gases to the lunar base. The crew servicing and maintenance time is simplified with the partially closed loop system. Water, which is 1,445 lbs. of the partially closed loop consumable expense, is available as a product from fuel cell power generation. When this 1,445 lbs. is available as a by-product of power generation, the advantage of the partially closed loop system is greater. This water is also available to use in a sublimator to provide thermal control for the crew and the MOSAP.

4.7.3.3 BALTRAN Comparisons

The ECLSS options are summarized by design factor for the BALTRAN in Table 4.7.3.2-1. The comparison comments are basically the same as for the MOSAP.

The BALTRAN partially closed loop, non-regenerative system is compared to the totally closed loop system in Table 4.7.3.3-1. The major consideration is the weight of the BALTRAN ECLS system. Since the BALTRAN will be launching and landing frequently on the lunar surface, the vehicle must minimize weight. The weight from the loss of consumables in an open system must be traded against the extra weight of an ECLSS which completely recycles CO_2 and H_2O. Because of the 8 day duration of the mission, only 76 pounds of nitrogen and oxygen is lost per 8 day mission in a partially closed system. The additional weight of CO_2 bed regeneration and water electrolysis units needed to recycle consumables is 2,721 pounds. This exceeds the weight of the additional O_2 which can be recovered from the metabolic H_2O and CO_2. In addition, more propellant is needed to lift this closed loop weight off the lunar surface. Although 600 pounds of water is needed with the partially closed loop system, this is available from the fuel cell power generation. An open loop ECLSS will be the lightest. The two advantages of a closed consumables system, the environmental pollution protection and the conservation of consumables, appear to be less important criteria.

4.7.4 Evaluation Comments

4.7.4.1 LOTRAN

No life support system is needed because the crew is in self-contained suits and returns to the lunar base nightly to doff the suits. The LOTRAN carries an emergency umbilical kit which allows two crew persons to breath from the same suit.

4.7.4.2 MOSAP

A partially closed loop ECLSS is used. In addition to saving 1000 lbs. of weight, waste products of metabolism can be collected and returned to the lunar base for recycling to minimize net H_2O and CO_2 losses from the lunar base supplies. Weight is not a critical criteria. Crew time and simplicity of operations and reliability are.

4.7.4.3 BALTRAN

A partially closed loop ECLSS is used since 2,700 lbs. are saved. Weight is the most important criteria. The BALTRAN must be kept as light as possible which saves propellant for launches.

Table 4.7.3.2-1 ECLSS Design Options for the MOSAP and BALTRAN

DESIGN FACTOR	OPTION 1	OPTION 2	OPTION 3
Recovery	Open	Partially closed	Totally closed
Consumables	Non-regenerate	Base regenerate	Vehicle regenerate
O_2	Carry all	Carry all	Carry part ; recover part from CO_2 & H_2O
CO_2	Absorb, dump	Absorb, carry back to base	Regenerate in vehicle
H_2O (metabolic)	Absorb, dump	Condense and carry back or sublimate	Electrolysis in vehicle
Cooling	Sublimator	Sublimator	Radiator
Losses	O_2 and H_2O	Only lose water for cooling by sublimator O_2 is recovered at base from H_2O and CO_2	Nothing

Table 4.7.3.2-2 Performance Comparison of MOSAP Life Support Options

PERFORMANCE FACTOR	OPTION 1		OPTION 2	
	Partially Closed Loop: Non-regenerative CO_2 and H_2O sublimator cooler		Totally Closed Loop: CO_2 bed regenerating, H_20 electrolysis, radiator cooler	
Duration (days)	42		42	
Maintenance	High		Higher	
Mobility	Attached		Attached	
Power Required (watts)	1,050		4,000	
Hardware Mass (kg) (lb)	807	1,780	2,177	4,800
Consumables (42 days)				
Nitrogen, oxygen	248	546*	95	210
LiOH	381	840*	0	
Water-metabolic loss	655	1,445++	0	
Water-System	91	200	363	800
Fuel for power gen.	4 times more			
Total Mass (kg) (lb)	2.182	4,811	2,635	5,810

* Return CO_2 and H_2O to the base for regeneration

++ Return H_2O to the base for regeneration or use in the sublimator for vehicle cooling.

Table 4.7.3.3-1 Performance Comparison of BALTRAN Life Support Options

FACTOR	OPEN LOOP: NON-REGENERATIVE		CLOSE LOOP: REGENERATIVE	
Duration (days)	8		8	
Maintenance	High		Higher	
Mobility	Attached		Attached	
Power required (watts)	1,050		4,000	
Hardware Mass (kg) (lb)	807	1,780	2,132	4,700
Consumables (8 days)				
Nitrogen, Oxygen	47	104*	18	40
LiOH	73	160*	0	
Water-Metabolic loss	125	275++	0	
Water-System	91	200	227	500
Fuel for power gen.	-		4 times more	
Total Mass (kg) (lb)	1,143	2,519	2,377	5,240

* Return CO_2 and H_2O to the base for regeneration.

++ Return H_2O to the base for regeneration or use in the sublimator for vehicle cooling.

4.8 EVA Systems

4.8.1 Design

4.8.1.1 LOTRAN

While the crew person is traveling in the LOTRAN and working at the LOTRAN site, he will be in an EMU identical to the one used at the lunar base. The most economical method is to use this suit. The suit is reliable since it has evolved from the Gemini, Apollo, Shuttle and Space Station designs. It is available on the moon since it will be the standard suit used at the lunar base. The crew is familiar with its operation which minimizes the risk of mistakes from using unfamiliar equipment. The crew is trained in emergency and maintenance procedures which removes the need for additional training time.

4.8.1.2 MOSAP and BALTRAN

The crew needs EMU suits identical to the lunar base suits for the same reasons discussed above for the LOTRAN. The crew will perform regular maintenance on the suits and resupply consumables. The MOSAP and BALTRAN will each carry EMU servicing racks, similar to that planned for the Space Station and available at the lunar base.

4.8.2 Evaluation Comments

The factors determining the costs of the EVA suit include costs for development, manufacturing, training, transportation to the lunar surface, maintenance, and spares. In all of the above respects, it is most cost effective to develop one EVA suit and use it at the lunar base and when traveling with the LOTRAN, MOSAP and BALTRAN.

4.9 Lighting for Surface Transportation

Light will be required for normal and contingency operations on the lunar surface, whether they are crew operated or teleoperated. Extended lunar surface traverses to processing plants or science sites will require continuous operations through all phases of the lunar lighting cycle. The lunar surfaces traversed and exterior vehicle maintenance/service areas must be properly illuminated to enable normal operations and repair of the vehicle itself. A summary of the operational techiques and vehicle requirements related to surface lighting for lunar traverse vehicles are provided in Section 4.9.3.

4.9.1 Lighting Options

Two options are available to satisfy these light requirements, natural and artificial. In either case, certain factors effect the way objects are illuminated and perceived by our eyes. The albedo of the viewed surface and the angles of light incidence and viewing both effect the amount of light seen. These factors are both taken into account by the photometric function.

4.9.1.1 Natural Lighting

Natural lighting on the moon is very dependent on location. Any spot on the moon goes through the same light cycle every 28 days. Every part of the moon (except possibly small remote polar regions) experiences 14 days of sunlight at angles proceeding from sunrise to sunset. The incident light angle also depends on the latitude of the lunar surface position.

In addition to this sunlight, some useful natural light is available in the form of earthshine (sunlight refected from the earth). The Earth is bigger than the Moon (Moon radius = 1,738 km, Earth radius = 6,738 km) and the Earth global average albedo (0.39) is greater than the Moon average albedo (0.09). Brightness is a function of albedo times radius squared; therefore, the brightness of full earthshine as seen from the moon is 58 times greater than the brightness of a full moon as seen from the earth. This earthshine light is site dependent because the same spot (the geographic center at longitude and latitude zero) on the moon always faces the earth. Sites on the far side of the moon never see earthshine. Sites on the near side see the earth go through all phases, but with varying efficiencies.

Efficiency is a function of when the sunshine overlaps the earthshine during a site's lighting cycle. Earthshine overlapped by sunshine has no benefit.

4.9.1.2 Artificial Lighting

Natural lighting will not be available for entire lunar cycles at most sites. Lacus Veras is near the earthshine terminator. Therefore, it receives only minimal earthshine when not in sunlight. Artificial light or redirected sunlight (see below) will be required to continue external lunar surface vehicle operations during the dark of the lunar night.

Artificial lights are available in the form of floodlights and narrow beam lights. Both will be required for the local base operations. Traverse vehicles will require light at a level of 0.5-1.0 ft-L (foot-Lamberts) at the worst case stopping distance for each specific vehicle. A level of 0.112 has been found to allow for rough and smooth terrain definition in emergency landing operations.

One scheme to project sunlight from lunar orbit to a 26 km (16.1 mile) diameter spot on the lunar surface uses mirrors. The projected diameter of these mirrors is about 8.7km (5.4 miles) and four of these would be required for continuous one ninth power sunlight. This level of light is calculated to be enough to run the base described in the scheme on solar power. This is far more than necessary for visual purposes. The mass for this system is considerable (218 metric tons per mirror). Trade studies should be run to determine whether the cost of this system will be less than using ground level artifical lights over the life of the base.

4.9.2 Lunar Vehicle Operation Factors

4.9.2.1 Albedo

Albedo is the fraction of light or electromagnetic radiation reflected by a body or particular surface. As previously stated, the average albedo of the moon is very low, about 0.09 (earth global is about 0.39). That means that only 9% of the light received by the moon is reflected back to space or to objects on the surface. Shadows of different lengths are cast depending on the angle of incidence. These shadows and the reflected light is what we see that shows the outline of craters, boulders, rilles, and other surface features.

4.9.2.2 Incident and Viewing Angle (Photometric Function)

When more detailed study is required, albedo does not contain enough information. The next stop is called the photometric function. The photometric function describes how much light reflects off of a surface to be received by the viewer. The simplest photometric function is that of an ideal reflective surface. The photometric function for the moon is like that of a field of unmowed grass. It takes into account the fact that the intensity of reflected light varies depending on the angle of incidence and the angle of viewing. The lunar photometric function is a function of the phase angle (the angle between incident and viewing lines) and the angle between the surface normal and viewing positions. This number is multiplied in the following equation to give the true luminance or brightness:

$$B = (E \alpha \phi) / \pi$$

where:

where:	B	is the brightness
	E	is the amount of luminous flux incident on the surface
	α	is the albedo
	ϕ	is the lunar photometric function
	π	is 3.1416.

4.9.2.3 Light Scattering

Light travels from its source to the eye in interesting and complicated ways, especially on the moon. On earth the dust, water molecules, and air molecules all attenuate light by redirecting some of the radiation to new directions. This action brings a portion of the light to places not in line of sight with the source, making objects in the shadows visible. With no atmosphere to scatter the light, the ambient light level of the moon drops to only what is reflected by objects and the ground. Astronauts from Apollo found that their suits could be used to reflect light into the shadows when a little extra was needed. There are three situations where the direction of the available light causes visibility problems that are attributed to the lack of ambient light: low light angles, high light angles, and back lighting.

4.9.2.3.1 Low Light Angles (Shadowing)

Low light source angles (less than thirty degrees elevation) cause extremely long shadows which can hide obstacles and craters.

4.9.2.3.2 High Light Angles (Washout)

On the moon, a phenomenon known as washout occurs when the light source is within thirty degrees of normal. This high angle does not allow reflected light to reach the eye. The object simply does not appear until you are right next to it. There is also a lack of shadows on most of the lunar landscape, (some steep sided objects may have shadows) which causes objects to appear less defined.

4.9.2.3.3 Back Lighting (Moving Down-Light)

In this situation the sun or light is directly behind the observer. The result is similar to that of washout, but only in one direction. The surface appears featureless when looking about ten degrees either side of the down sun or down light direction.

4.9.3 Proposed Operating Standards

Artificial lights will be necessary at least for close proximity operations in shaded areas. Small lights, such as those on the Shuttle EMU may provide satisfactory light for this purpose. But, much larger systems will be required for lighting the path of the vehicles described in this and the construction tasks. There are large (50 watt) flood lights being developed for use on the Space Telescope. These could be used on the lunar surface transportation vehicles.

4.9.3.1 Minimum Illumination

The lunar surface transportation vehicles will need a minimum illumination of the lunar surface to allow time for detection and avoidance maneuvers or stopping before running into various obstacles and craters. A level between 0.5 and 1.0 ft-L has been recommended for the determined viewing distance. The nominal speed of the LRV (~10 km/hr) used during Apollo was limited by the hummocky surface of the moon, rather than power.

The rover became airborne at higher speeds. Changes in future vehicle designs could increase this speed limit and, also, increase the time and distance needed to avoid problems.

4.9.3.2 Power Limits

Lights use a lot of power, especially incandescent lights. Therefore, use of artificial lights should be carefully considered when designing surface vehicles and when planning vehicle traverses. Natural lighting should be used whenever possible. Nominally, on board sensor systems should shut off artificial lights when the natural light level is brighter. A manual override must be provided for servicing and repair operations.

4.9.3.3 Driving with Shadows

When natural light is available, but is not in a direction where needed, a reflective surface should be used to redirect and/or concentrate the light. If this is not possible, artificial light can be used.

4.9.3.4 Driving in High Angle Light

To avoid the washout problem associated with high Sun, the use of neutral density filters is recommended. These can be incorporated into the astronauts suit visor or sun glasses when traveling intravehicular.

When using artificial lighting, washout can be avoided by designing the vehicles with lights that are below the drivers line of sight. Using two headlights about 4 feet on each side of the driver(s) should provide good light for surface traverses.

4.9.3.5 Driving with Back Lighting

When in sunlight, the best way to avoid the back lighting problem is to avoid traversing in the direction that causes the problem. When this is impossible, polarized filters can help, when worn by the crew. When using artificial lights, the back lighting effect can occur, but is correctable with the use of polarized filters on the headlight itself.

4.10 Emergency Breakdown Recovery

Two classes of emergency breakdown will be discussed. The first class includes emergencies in which the crew can perform its own recovery operations. The second includes emergencies that require outside assistance from the base.

4.10.1 Crew Recoverable Breakdowns

Crew recoverable breakdowns include equipment failure, such as failure of batteries, steering, brakes, or lighting; and some vehicle accidents, such as an overturned LOTRAN or a stuck LOTRAN or MOSAP. Generally, equipment failure can be rectified by switching to redundant equipment. Where there is no redundancy, or the redundant equipment also fails, on-site repairs could be attempted. This would require that a tool kit and some spare parts (wire, nuts, bolts, etc.) be carried on the vehicle. If equipment repair is not possible, the crew could attempt to switch to a different mode of operation. Brake failure could require the crew simply to reduce traverse speed so the vehicle can coast to a stop whenever it needs to halt. A LOTRAN that is stuck in a depression or that becomes bogged down in deep surface material may be able to be pulled free manually by the crew. It may be necessary first to unload equipment from the vehicle to reduce its weight.

A stuck MOSAP vehicle, however, will be too heavy for a crew to free manually. An alternative is to equip the MOSAP with a winch that can be placed at either the front or rear of the vehicle. The free end of the cable can be secured to stakes that are driven into the Lunar surface or that are placed into holes drilled into the surface.

4.10.2 Breakdowns Requiring Assistance

Irreparable emergency breakdowns may require that rescue and repair missions be sent from the base. LOTRAN breakdowns and near-base MOSAP breakdowns could be addressed by sending out a MOSAP rescue vehicle. The rescue vehicle would retrieve the crew and attach to the failed vehicle for towing back to the base. Minimum rescue crew size would be two, so the MOSAP emergency vehicle would need the capacity to carry a maximum of six crew back to the base.

Remote MOSAP breakdowns and BALTRAN breakdowns could be addressed by sending out a BALTRAN emergency vehicle to retrieve the crew. The failed vehicle would be left behind if it were irreparable. Minimum rescue crew size would be one, so the BALTRAN emergency vehicle would need the capacity to carry a maximum of five crew on the return trip.

Accidents that cause breakdowns may also involve injured crew members. The emergency vehicles would need to be outfitted in such a way that immobilized crew members could be loaded into the vehicle. Alternatives for the MOSAP include manually lifting a crew member into the vehicle man-lock; or using a hydraulic or electric lift to raise the crew member to the man-lock. BALTRAN crew loading mechanisms could include the lift described for the MOSAP, or block and tackle arrangements.

All these scenarios presume the disabled vehicle is able to communicate its distress to the base. Should the vehicle transmitter be damaged, however, another means of contacting the base is required. One method resembles that employed by ships in distress on the Earth's oceans: the use of a rocket-propelled "flare" that would transmit an emergency beacon signal.

4.11 Communications

Lunar surface traverse communications requirements encompass voice, data, and video signal processing to and from the traverse vehicles. The vehicles must be able to communicate with the lunar base, with Earth-bound stations, and with astronauts performing EVA's from the vehicles. The following sections describe alternative communication approaches for near-base and remote traverses.

4.11.1 Near-Base Traverse Communications

Local traverses with the LOTRAN vehicle will require continuous voice communications between the lunar base, possibly the Earth, and up to four astronauts performing EVA's from the LOTRAN. In addition, intermittant live video transmissions to the base and to Earth are expected. Several approaches to these requirements are described below.

4.11.1.1 Direct Line Communications To The Lunar Base

This approach requires that the vehicle remain "visible" to the base. For distances greater than several kilometers, a tower antenna would be needed at the base, the vehicle, or both. Table 4.11.1.1-1 shows several combinations of tower heights that would allow direct line communications for up to 50 kilometers (31 miles). A drawback to this approach is that a smooth surface is presumed, with no geographic features to come between the vehicle and the base. Should the vehicle drive behind a mountain or into a crater, communications could be lost.

4.11.1.2 Communications Through Relay Stations

Relay stations placed at strategic locations on the surface could eliminate the need for large tower antennas, and could address the problem of blocked signals. If a region is to be visited frequently, permanent placement of the relays would be effective. Otherwise, temporary relays could be placed by the LOTRAN on the journey out. Retrieval of these relays would require that the vehicle return along the same route that it took out. Unless redundancy were introduced, failure of a single relay would interrupt all communications.

4.11.1.3 Communications Through Relay Satellites

One or more lunar orbiting relay satellites would also allow communications between a LOTRAN and the base. Use of one low lunar orbit satellite would limit communication periods to those times when the satellite is visible to both the base and the vehicle. Continuous communications would require that several low lunar orbit satellites be spaced appropriately in lunar orbit. Alternatively, a single satellite parked at the L1 libration point would also allow continuous communications from the near side.

Parking a satellite at a fixed location would eliminate the need for complex satellite tracking systems at the base and on the vehicle. If the satellite antenna were omni-directional and the base antenna direction fixed, the vehicle antenna could be manually steerable. A solar powered satellite would need to operate on low power. Therefore its ability to amplify signals would be low. This means, in turn, that the base and vehicles would require higher power transmitters and higher gain receivers.

4.11.1.4 Use Of Earth Bound Relay Stations

If lunar orbiting satellites were not available, continuous communications between a LOTRAN vehicle and the base could be accomplished using relay stations on Earth. A minimum of two stations, 180 degrees apart around the circumference of the Earth, would be required. The inconvenience of a three to five second transmission/response delay during the signal's round trip would be a drawback to this approach.

4.11.1.5 Broadcasting Frequencies

Three broadcasting frequencies were considered for near-base communications. The VHF spectrum (30 - 300 megaHertz) is effective for direct line base communications, communications with EVA astronauts, or communications through lunar-based relays. The technology is well established and has been proven on the moon. Bouncing VHF signals off L1- or Earth-based relays is not as effective, however, because of overcrowding of this bandwidth in the vicinity or direction of the Earth.

The S-Band (1850 - 2300 megaHertz) would be a better alternative for use with L1- or Earth-based relays because it is not as crowded and again the technology has been proven on the moon. The Ka-Band (30 - 50 gigaHertz), whose technology is still under development, is uncrowded and would be an even better alternative.

4.11.2 Remote Traverse Communications

Remote traverses with the MOSAP and BALTRAN vehicles will require continously available intermittant voice, data, and video communications with the lunar base and with the Earth; and continuous voice communications with up to five astronauts performing EVA's. Traverses to the near and far side of the moon must be considered. The lunar base is presumed to be on the near side. Several approaches to these requirements are described below.

4.11.2.1 Communications Through Relay Stations

As with near-base traverses, temporary or permanent relay stations could be established to facilitate communications for remote traverses. Telescoping antennae that extended to 100 meters would allow placement of the relays every 75 kilometers (46.6 miles).

4.11.2.2 Communications Through Relay Satellites

A single satellite parked at the L1 libration point would suffice for communications between the base and a vehicle that is located anywhere on the near side of the Moon. Communications with a vehicle at the far side of the Moon would also require a satellite in a halo or hummingbird orbit in the vicinity of the L2 libration point. In this case, transmissions would originate at the vehicle, be broadcast to L2, relayed to L1, then relayed again down to the base. A satellite based at L2 would also allow communications with the Earth from the far side of the Moon.

4.11.2.3 Use of Terrestrial Relay Stations

If a satellite were located at L2, but not at L1, an Earth bound relay station could be substituted for the L1 relay, allowing communications to the base from the far side. Total signal delay for a transmission and a response would be six seconds.

4.11.2.4 Trans-Lunar Transmissions

It may be possible to transmit signals at very low frequencies directly through the moon from the vehicle to the base. However, the range of such a signal through the lunar material and the effects of encountering changes in subsurface material are unknown at this time. Technology for trans-lunar communication transmissions would have to be further developed before it is considered practical for remote lunar communications applications.

4.11.2.5 Broadcasting Frequencies

VHF would also not be practical for remote communications requiring satellite relays, because of the problem with overcrowding in that spectrum. As with near-base communications requiring lunar orbiting or terrestrial relays, the S-Band and Ka-Band spectrums appear to offer the best features.

VHF would suffice for vehicle communications with EVA astronauts.

4.11.3 Conclusions

The following approaches appear to best meet the stated communications requirements:

REMOTE TRAVERSES

Vehicle/EVA Communications:	Direct-line, VHF spectrum.
Vehicle/Earth Communications:	Direct-line or use of L2 relay satellite, Ka-Band spectrum.
Vehicle/Base Communications:	Use of L1 or L2 relay satellite, Ka-Band spectrum.

NEAR-BASE TRAVERSES

Vehicle/EVA Communications:	Direct-line, VHF spectrum.
Vehicle/Earth Communications:	Direct-line, Ka-Band spectrum.
Vehicle/Base Communications:	Early stage, use of Earth-based relay station. Later, use of L1. Ka-Band spectrum.

Table 4.11.1.1-1 Vehicle/Base Antenna Heights Required For Direct-Line Lunar Communications To 50 Km (31 miles).

VEHICLE ANTENNA HEIGHT		BASE ANTENNA HEIGHT	
meters	(feet)	meters	(feet)
1	(3.3)	670	(2198)
5	(16.4)	605	(1985)
10	(33)	560	(1837)
15	(49.2)	530	(1739)
20	(65.6)	500	(1640)
30	(98.4)	460	(1509)
50	(164)	390	(1280)
75	(246)	330	(1083)
100	(328)	285	(935)

4.12 Radiation Protection

4.12.1 Radiation Environment and Effects

Earth orbital operations at low altitudes and low inclinations are protected from solar proton events by the earth's magnetic field. The chances of encountering a solar proton event during the short duration Apollo missions was small and no major event was encountered. For extended operations on the lunar surface, neither of these protective conditions are present. There is no magnetic field around the moon and near-continuous occupancy of the lunar surface is planned. Major solar flares can be expected in the period 1999 to 2004. Thus more stringent protection from such events must be incorporated into lunar surface transportation mission planning.

The stay-times on the lunar surface are planned to gradually increase until they are 180 days in duration. This prolonged period under reduced gravity conditions will cause physiological changes which currently are not completely or well understood. To date reduction in bone calcium and muscle density and changes in the red blood cells have been observed. Table 4.12.1-1 shows the threshold for acute radiation effects. These effects are caused by high radiation doses delivered in a brief period of time (1-4 days or less). The symptoms shown in this table are derived from data obtained under one-g conditions and it is anticipated that they will occur at lower levels for crew members who have been in reduced gravity for an extended period. In Table 4.12.1-2 it can be seen that these acute radiation effects may be delayed for periods of from three to four weeks. Recovery from radiation damage is not well understood. The National Council On Radiation Protection and Measurements reported in NCRP Report No. 29, January 1962, that 10% of all radiation produced permanent damage and that recovery from the balance of damage occurred at a rate of 2.5% per day. This data was considered applicable only to the acute effects of radiation and admitted that "... the whole question of time-intensity variation is so complex that each situation will undoubtedly require its own interpretation".

4.12.2 Exposure Limits

The allowed doses of radiation under current NASA flight rules are shown in Figure 4.12.1-1. These rules have been approved by the National Council On Radiation Protection and Measurements (NCRP). They have been applied to the Apollo and Space Lab missions. These rules are designed to minimize carcinogenic effects later in life, but there are considered to be little or no acute radiation damage effects. During the above missions no exposure to a large solar proton event occurred, and radiation doses received were much less than that allowed by the flight rules as can be seen in Table 4.12.2-1. For lunar base missions a more stringent, lower set of dose limits may be expected to be promulgated.

Thus, for lunar operations of 180-day duration, the revision of present flight rules appears likely, resulting in reduction of allowable radiation doses for lunar surface operations crews. Both the 30-day and 1-year allowed levels under current flight rules thus appear too high for application to the lunar surface transportation operations.

4.12.3 Protection Strategy

The strategy which should apply to radiation protection on the lunar surface is to avoid radiation exposure by moving to a completely safe shelter when necessary. If exposure cannot be avoided, then minimize the delivered dose. It is assumed that a permanent lunar radiation safe haven habitat is buried at the Lunar Base beneath the lunar surface to a depth where the radiation levels from galactic cosmic rays or solar protons is near zero. This depth is estimated to be about 3 meters (10 ft). Therefore, any sudden radiation problems will be because crews are away from the base in one of the three classes of lunar surface transportation vehicles. When considering the long eleven-year cycling of solar storm activity and the small number of significant events and the relatively small amount of travel time away from the base, the likelihood of an away-from-base solar event is very low. Based on this strategy, the radiation protection design goal is to minimize transportation vehicle shielding penalties and plan to return to base for protection from significant solar flares.

4.12.4 Protection Design Options

Two classes of radiation protection devices are required to implement the protection strategy. The first is a partial protection garment that is donned by an individual to protect the most vital areas of the body for a limited time in a contingency situation. The other device is a cylindrical capsule with walls providing shielding sufficient to prevent a lethal radiation dose over 36 hours for expected solar flare intensities. The cylindrical device is referred to as a radiation storm cellar.

It is assumed that either each of the lunar surface transportation vehicles is equipped and continuously using radiation detection rate meters or that it is in continuous communication with the lunar base. If the onset of a solar proton event is detected and if return to the lunar base can be made in three hours or less, then little radiation shielding is required. Examination of Figure 4.12.4-1 shows a typical rise time to dangerous intensity levels requires around three (3) hours. Problems are caused by the current state of solar flare knowledge and the unpredictable buildup characteristics. First, the three hour rise time is not dependable. Second, the unpredictable rise time may cause many false alarms when a "sensed" solar storm does not occur. A policy will be required on whether many actually safe missions should be abandoned.

For the LOTRAN, complete radiation protection is not practical. Scheduled use of the vehicle should avoid periods when solar flares appear imminent, or alternately, should restrict its range of operation to return times of three hours or less. If because of an emergency or some other serious extenuating circumstance extension beyond these limits is needed then some additional protection can be provided with the partial protection garment.

For the MOSAP, where immediate return to the lunar base by the vehicle is not possible, the use of the BALTRAN to return personnel to the base should be the first defense against radiation exposure. If all the personnel cannot be returned in the three hour time limit, then the partial protection garment can reduce the dose until later flights can be affected. If no means of rapid return to base is available and onboard protection is planned, then a radiation storm cellar would be required.

The BALTRAN will plan to return to the lunar base for protection from a solar flare. Since the vehicle may require a reasonable period of time for flight preparation, it should be equipped to carry the partial protection garment. The installation of the radiation storm cellar does not appear to be practical.

4.12.5 Shielding Materials

In earlier studies for the Apollo and Space Lab missions, aluminum (atomic no. 13) was almost universally studied as the shielding material since it was the structural material of the spacecraft. On a weight basis there is an advantage in using materials of a lower atomic number. A weight saving of around 20 percent can be expected by using materials of a lower atomic number. Space considerations require that the material selected have as high a density as possible, but many higher density materials such as iron, lead, etc. have the disadvantage of producing secondary radiations. Some of the very low atomic number plastics are not UV resistant and may have poor wear resistance. As far as possible, existing vehicle materials should be used to minimize shield mass penalty requirements. The materials which are candidate shield materials are:

1) Water from existing vehicle supplies
2) Carbon fiber or graphite epoxy composites
3) Carbon fiber cloth
4) Mylar, kelvar or other low atomic number epoxy composite or cloth Lexan

4.12.6 Partial Protection Garment

This garment is designed to shield the head and torso at all times. The arms are shielded when not in use. To allow mobility the legs below the knees are not shielded. The garment reduces the delivered dose to allowed limits to the blood forming organs for a period of around three hours and provides a reasonable degree of mobility. As can be seen from Figure 4.12.6-1, shielding dose reduction per unit shield thickness drops rapidly beyond 10 gm. per sq. cm. (0.14 lb/in^2) in aluminum or about 8 gm. per sq. cm. (0.11 lb/in^2) of lower atomic number material. This is a thickness of eight cm (3 in) of carbon fiber cloth and would appear reasonable for a partial protection garment. This amount of shielding should reduce the delivered dose by a factor of 2 to 4 times, depending upon the spectrum of solar event, and provides a shield thickness basis for the partial protection garment. The free space dose for two large solar flares versus shield thickness is shown in Figure 4.12.6-2. Since the lunar surface provides protection over half the total stereradians, the delivered dose is about half of the amount shown in Figure 4.12.6-2. If it is assumed that dose limits are reduced by a factor of two from present flight rules, the allowed 30-day dose of radiation would be received in about three hours inside the protective garment.

The garment is conceived as a sleeveless cloak extending to the knees. A helmet and protection for the back of the neck are also included. A lexan visor provides eye protection. The cloak could be constructed of carbon fiber cloth weighing 0.68 kg/m^2 (20 oz/sq yd) to provide 8 gm. per sq. cm (0.11 lb/in^2) consisting of 118 layers and about eight cm (3 in) of shielding. For a crewman in the upper 95 percent of height, 191 cm (75 in), and wearing a space suit, the garment is estimated to have an Earth weight around 170 kg (375 lbs) which is the equivalent weight of 28 kg (62 lbs) on the lunar surface. Storage volume for the garment is estimated to require 0.1 m^3 (5 ft^3). Using an estimated cost of $47 per square meter ($40/sq yd) of 0.68 kg/m^2 (20 oz./sq.yd), the cloth in the garment costs about $12,000. A dose rate monitor beneath the garment permits measurement of exposure dose. A fabricated garment cost is estimated at $30,000. Alternate materials, which contain a larger percentage of hydrogen, should be investigated for further weight and cost reductions. The design is very preliminary, but indicates a method of implementing the partial protection garment approach.

4.12.7 Solar Storm Cellar

4.12.7.1 Configuration

This device is designed to maintain the dose of a crewman to below the allowed 30-day exposure limit. The device provides from 20 to 30 gm./sq.cm. over 4 pi stereradians, and consists of a cylinder flattened on one side to be occupied by the crewman with an internal volume of 1.7 m^3 (60 ft^3). This level of shielding is estimated to limit the total dose to 25 REM or less. As can be seen from Figure 4.12.4-1 the stay time in this volume would be about 24 hours for the initial portion of this particular event. Other events may require longer stay times, possibly up to 36 hours. For stay periods of this duration, there is a need for life support and vehicle control functions which should be performed by the crew while remaining in the shielded volume. The completely shielded cylinder, not taking advantage of the lunar surface or spacecraft shielding, is estimated to have a mass of 1,500 to 2,300 kg (3,600 to 5,060 lbs). An alternate two man shelter was also studied and appears to have some weight per crewman advantage. The following sections describe the equipment requirements needed to provide effective operation during a solar flare.

4.12.7.2 Shielding Material

4.12.7.2.1 Liquid

Since lower atomic number material is more effective than higher atomic number material and fluids are more easily stored in a minimum volume than solids, water is a highly effective shielding material. If available, petroleum fuels might be used, but this type material can introduce toxicity and flammability problems. To insure shielding of the proper thickness is maintained, some rigid solid structural material may be required and these structural elements should also be of a low atomic number material. Sizing of these structural elements will depend upon the amount of gravity or acceleration that the shield will experience. The type of construction used in inflatable boats would appear to apply in this case. The shielding volume should be compartmented to prevent complete loss of shielding in the event of a leak. The structural elements must be configured to minimize their space requirements when not in use. The source of liquid shielding material is materials aboard the vehicle in other systems. If they are not available then this system appears to hold little advantage over a system with all solid components.

4.12.7.2.2 Solid

An alternate to liquid filled type shielding is carbon fiber composite type structure. This would also provide a low atomic number material shield with mass and thickness properties similar to water. Detailed studies of the spacecraft configuration may allow dual use of this material such as bunks and deck plating. An alternate would be to use carbon fiber cloth in conjunction with existing structure. This approach may have advantages over an entirely rigid structure by minimizing storage.

4.12.7.2.3 Hybrid

A final possibility is a hybrid structure using a carbon fiber composite structure which can be filled with water. Permanent spacecraft structure can provide some shielding and the amount of portable shielding can be reduced by locating the erectable structure to take maximum advantage of the spacecraft structure.

4.12.7.3 Access Hatch

The access hatch is located near the head of the crewman for ingress and egress to the shielded volume. For this study it is assumed that entrance is made from a pressurized environment. If subsequent studies indicate that this is not the case, an airlock will have to be added to the design.

4.12.7.4 Ventilation

The atmosphere in the volume should be replaced at a rate comparable to that in a space suit. The penetration of the shielding volume by intake and exhaust fittings must be made in a manner that will not violate the shielding integrity of the shield, or at least exposes a portion of the body that is relatively insensitive to radiation. To accomplish this end, intake and exhaust fittings penetrating the shield are located near the feet of the crewman. Two plenum chambers are located outside the end shield in this area. A number of small ducts, which change directions while penetrating the shield, are used to eliminate or minimize radiation leakage. For the supply side the discharge of the ducts into the shielded volume are discharged into a third plenum. The outlet for this last plenum is a duct which discharges near the head of the crewman. No exhaust plenum is required. This arrangement should insure that there is no build up of carbon dioxide in the shielded volume. The system supplying atmospheric gas is not part of the shelter equipment.

4.12.7.5 Visibility

It is desirable to provide a window which will allow clear vision from within the shielded volume. A water filled volume between two parallel lexan plates is installed in the vicinity of the crewman's head. The window allows limited visual inspection of the spacecraft interior.

4.12.7.6 Operations

There may be shielded volumes for each crewman in use simultaneously. Voice communications between the shielded volumes and between the shielded volumes and mission control should be incorporated into each shielded volume. A miniature portable computer keyboard and display are part of the equipment taken into the shield to allow inspection and operation of vital spacecraft systems. A slim keyboard and a flat screen display system minimizes the volume required by such a device. A flat lightning fixture is installed above the crewman's head. The use of a flat cable within the shield will minimize any impact on the available shielded volume. To get rations and water or discharge body wastes, a shield hatch which permits hands to be used outside the shielding volume is incorporated. For emergency cases which cannot be handled without greater mobility, the partial protection garment may be required.

4.12.7.7 Waste Disposal

Since occupancy time may be as much as 36 hours, provision must be made for handling body wastes. The types of urine and fecal devices used in the Gemini and Apollo Programs with some modification for female crew should meet this requirement. A miscellaneous trash bag for other waste should be provided. Sealed waste containers can be removed from the shielded volume through the shield lock described above.

4.12.7.8 Food and Water

C Rations and bottled water are supplied, and are estimated at 3 gallons and 5 meals per man.

4.12.7.9 Spacecraft Interfaces

1) An umbilical and gas circulation pump supplying air from the ECS system provides flow rates comparable to that used in space suits.

2) Cabling and connectors for communications, computer terminal output and lighting.

3) Storage for portable shielding if employed and when not in use.

4) If liquid shielding is used charging/discharging pumps which can transfer required water or petroleum fuel to the shielding volumes. Charging time should be as short as possible, but in no case greater than 1.5 hours.

5) Provision for cleaning and restoring shielding devices.

6) Space designation for location of shields during solar proton events.

7) The incorporation of electronics which allow the operation of spacecraft systems from the computer terminal inside the shielded volume.

4.12.8 Mass and Cost Estimates

All support to the shielded volume is supplied by the spacecraft in which the shelter is installed. The mass of water or fuel would be approximately the same as for composite materials. The thickness of the shielding will increase inversely in proportion to the density. Table 4.12.8-1 shows approximate dimensions for composite, water, and fuel for single occupancy. Volume considerations may preclude the use of liquids. Two designs are considered:

1) Single crewman occupancy
2) Two crewman occupancy

A sketch of the single occupant shelter is shown in Figure 4.12.8-1. Weight and cost figures for this configuration are given in Table 4.12.8-2. Double occupancy was studied by stretching the center section of the single occupant shelter by 0.6 m (2 ft). Weight and cost estimates for this configuration are shown in Table 4.12.8-3. If protection of two crewmen in a vehicle is required then this configuration offers some weight and cost saving.

Table 4.12.1-1 Early Effects of Acute Radiation Exposure

PHYSIOLOGICAL EFFECT	EFFECTIVE DOSE IN REM ABSORBED IN 1 DAY OR LESS FOR 10, 50, 0R 90 % OF A POPULATION OF NORMAL PEOPLE TO HAVE THE INDICATED EFFECT		
	10 % EFFECTED	50 % EFFECTED	90 % EFFECTED
Anorexia	40	100	240
Nausea	50	170	320
Vomiting	60	215	380
Diarrhea	90	240	390
Death (20-60 days)	220	285	350

* Exposure for a duration of 1 day or less to blood forming organs (greater than or or equal to 5 cm tissue depth)

** Table was created from SCC 86-02 from Severn Communications Corporation

Table 4.12.1-2 Summary of Clinical Symptoms of Radiation Sickness

Time after exposure	Lethal dose (600 r)	Median lethal dose (400 r)	Moderate dose (300–100 r)
First week	Nausea and vomiting after 1–2 hours. No definite symptoms. Diarrhoea. Vomiting. Inflammation of mouth and throat.	Nausea and vomiting after 1–2 hours. No definite symptoms.	No definite symptoms.
Second week	Fever. Rapid emaciation. Death. (Mortality probably 100 percent.)	Beginning epilation.	
Third week		Loss of appetite and general malaise. Fever. Severe inflammation of mouth and throat.	Epilation. Loss of appetite and general malaise. Sore throat.
Fourth week		Pallor. Petechiae, diarrhoea, and nosebleeds. Rapid emaciation. Death. (Mortality probably 50 percent.)	Pallor. Petechiae. Diarrhoea. Moderate emaciation. (Recovery likely unless complicated by poor previous health or superimposed injuries or infections.)

Figure 4.12.1-1 NASA Flight Rules for Crew Radiation Exposure Limits

R	RULE	
		MANAGEMENT
	14-10	CREW RADIATION EXPOSURE LIMITS THE FOLLOWING OPERATIONAL CREW IONIZING RADIATION EXPOSURE LIMITS WILL BE ADHERED TO: EXPOSURE LIMITS (REM)

CONSTRAINT	DEPTH (5 CM)	EYE (0.3 MM)	SKIN (0.01 CM)
30 DAY	25	100	150
ANNUAL	50	200	300
CAREER	100 - 400*	400	600

MALE - *200 + 7.5 (AGE - 30) REM, UP TO 400 REM MAXIMUM
FEMALE - *200 + 7.5 (AGE - 38) REM, UP TO 400 REM MAXIMUM

STS crew radiation exposure limits were recommended to NASA by the National Council on Radiation Protection and Measurements in 1987 and are expected to be legally adopted as the Agency's Supplementary Standard for compliance with 29 CFR 1960.18. STS flights are nominally constrained to the 30-day exposure limits, which are conservatively set to preclude any mission impact. (Rule 14-6 reference)

14-11 UNCONFIRMED ARTIFICIAL EVENT

FOR ALL FLIGHT PHASES AND PRELAUNCH, IF AN ARTIFICIAL EVENT IS UNCONFIRMED, THE FLIGHT DIRECTOR WILL BE NOTIFIED AND CONFIRMATION WILL BE PURSUED FROM ALL DATA SOURCES.

No action is required other than notification of the Flight Director since the predicted or reported event may not be real.

ALL	BASELINE	9/1/87	SPACE ENVIRONMENT	14-3
MISSION	REV	DATE	SECTION	PAGE NO.

Table 4.12.2-1 Dosimetry Data From U.S. Manned Spaceflights

Flight	Duration (hrs/days)	Inclination (deg)	Apogee-Perigee (km)	Average dose (mrad)	Average dose rate (mrad/day)
Gemini 4	97.3 hrs	32.5	296 - 166	46	11
Gemini 6	25.3 hrs	28.9	311 - 283	25	23
Apollo 7*	260.1 hrs			160	15
Apollo 8	147.0 hrs		lunar orbital flight	160	26
Apollo 9	241.0 hrs			200	20
Apollo 10	192.0 hrs		lunar orbital flight	480	60
Apollo 11	194.0 hrs		lunar orbital flight	180	22
Apollo 12	244.5 hrs		lunar orbital flight	580	57
Apollo 13	142.9 hrs		lunar orbital flight	240	40
Apollo 14	216.0 hrs		lunar orbital flight	1140	127
Apollo 15	295.0 hrs		lunar orbital flight	300	24
Apollo 16	265.8 hrs		lunar orbital flight	510	46
Apollo 17	301.8 hrs		lunar orbital flight	550	44
Skylab 2**	28 days	50	altitude ≈ 435	1596	57 ± 3
Skylab 3	59 days	50	altitude ≈ 435	3835	65 ± 5
Skylab 4	90 days	50	altitude ≈ 435	7740	86 ± 9
Apollo-Soyuz Test Project	9 days	50	altitude ≈ 220	106	12
STS-1***	34 hrs	38	altitude ≈ 140	12.6	8.9
STS-2	57.5 hrs	38	altitude ≈ 240	12.5 ± 1.8	5.2
STS-3	194.5 hrs	38	altitude ≈ 240	52.5 ± 1.8	6.5
STS-4	169.1 hrs	28.5	altitude ≈ 297	44.6 ± 1.1	6.3
STS-5	120.1 hrs	28.5	altitude ≈ 297	27.8 ± 2.5	5.6
STS-6	120.0 hrs	28.5	altitude ≈ 284	27.3 ± 0.9	5.5
STS-7	143.0 hrs	28.5	altitude ≈ 297	34.8 ± 2.3	5.8
STS-8	70/75 hrs	28.5	altitude ≈297/222	35.7 ± 1.5	5.9
STS-9†	240.0 hrs	57	altitude ≈ 241	103.2 ± 3.1	10.3
STS-41B	191.0 hrs	28.5	altitude ≈ 297	43.6 ± 1.8	5.5
STS-41C	168.0 hrs	28.5	altitude ≈ 519	403.0 ±12.0	57.6

*Doses for the Apollo flights are skin TLD doses. The doses to the blood-forming organs are approximately 40 percent lower than the values measured at the body surface.
**Mean TLD dose rates from crew dosimeters.
***STS-1 data are from an active dosimeter; all other STS data are averages of USF TLD-700 (^7LiF) readings from the Area Passive Dosimeter.
†Spacelab (SL-1).

Figure 4.12.4-1 Solar Proton Flux - August 2-12, 1972

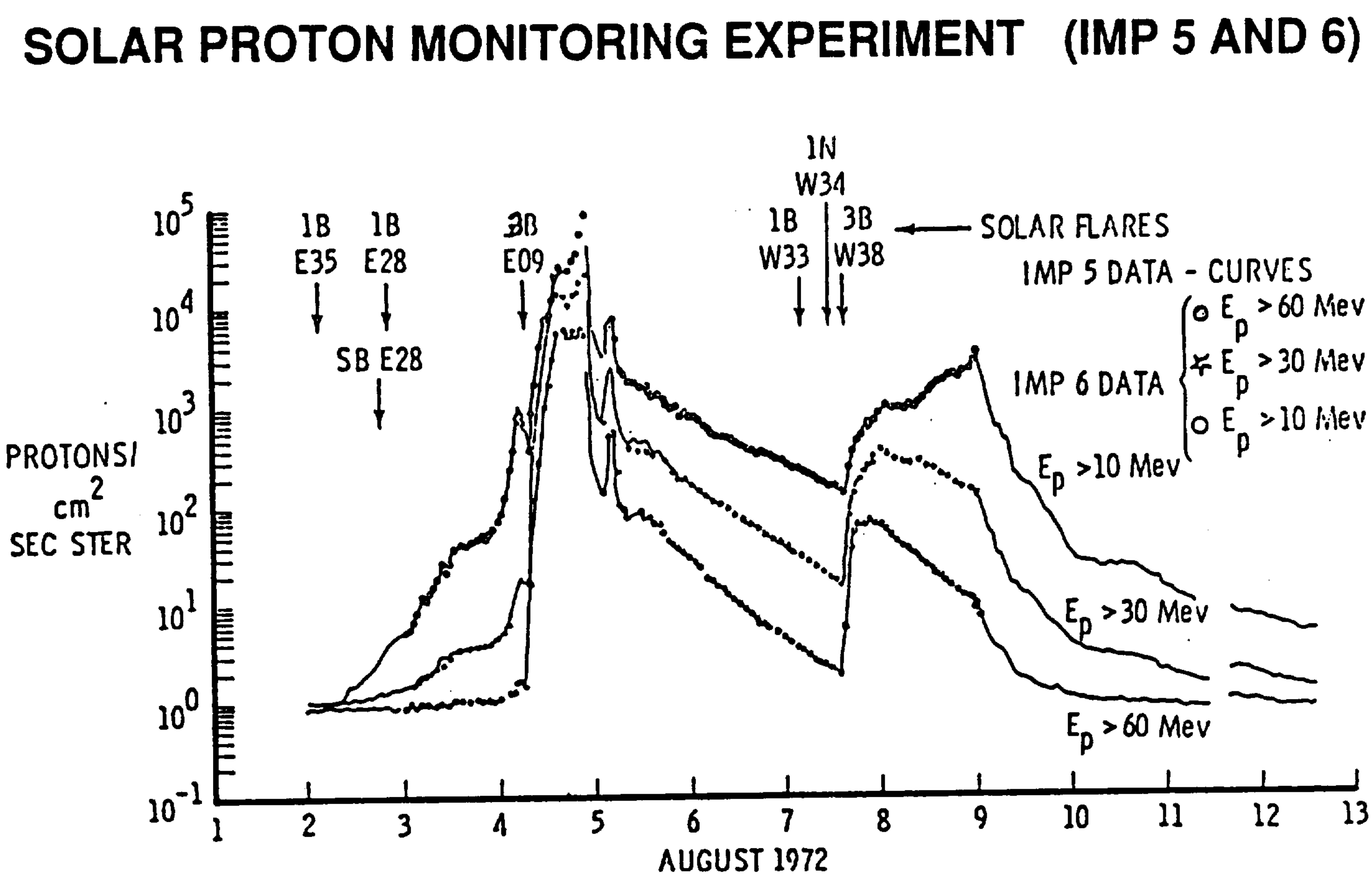
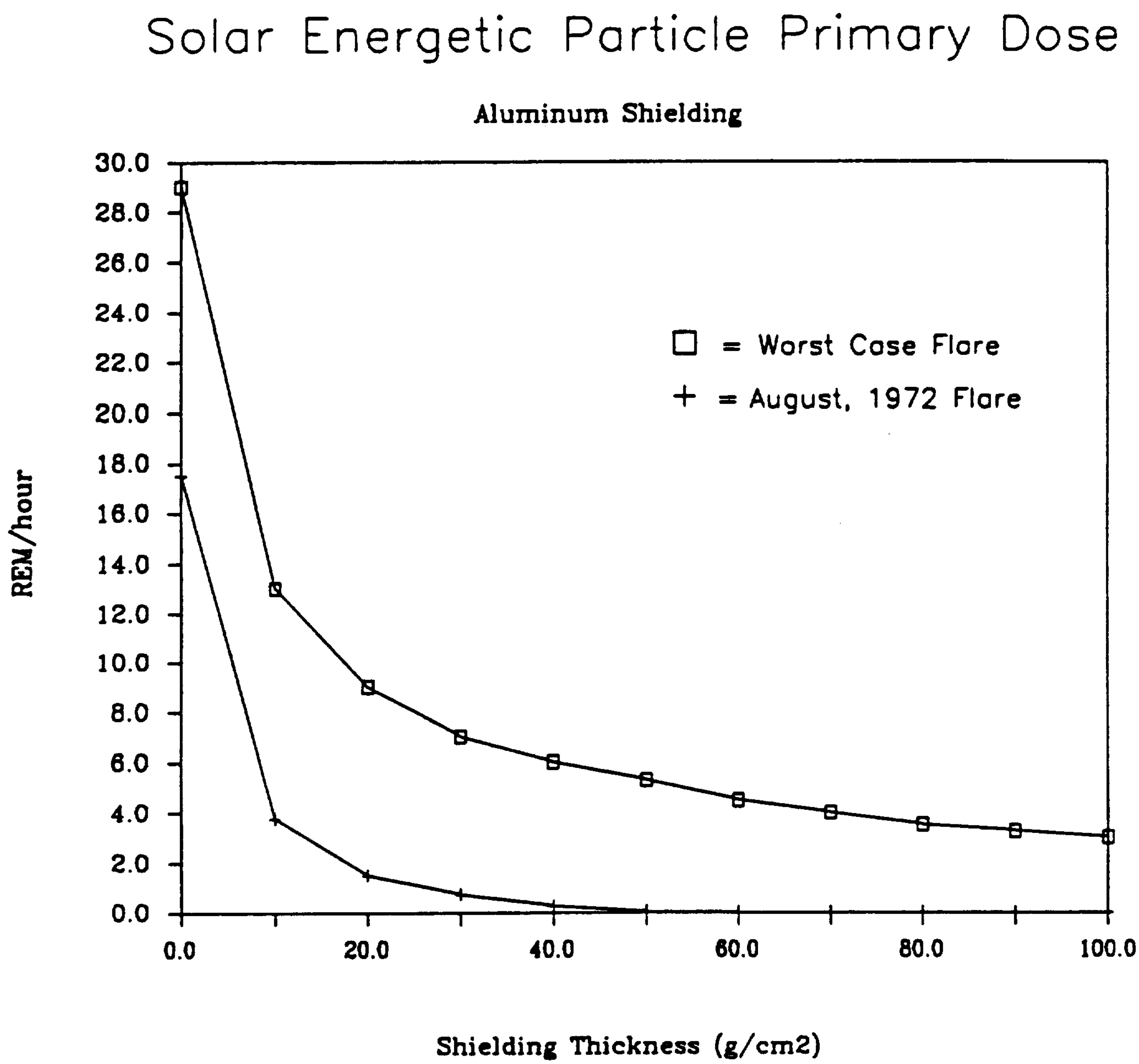


Figure 4.12.6-1 Dose Equivalent Versus Shielding Depth for Solar Energetic Particles



*NOTE: Graph was recreated from that by Severn Communications Corp.

Figure 4.12.6-2 Solar Particle Dose vs Spherical Shield Aluminum Thickness

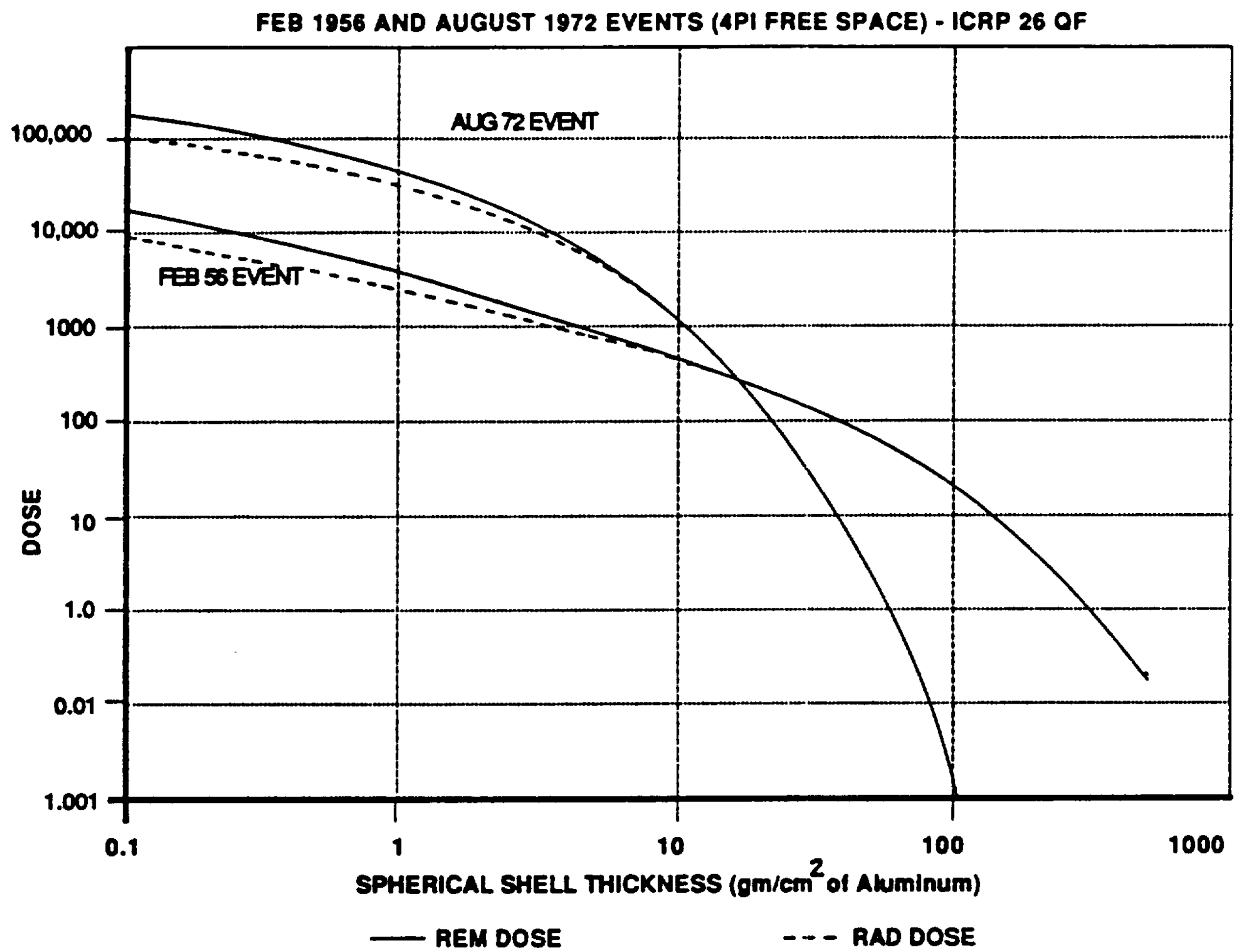


Table 4.12.8-1 Shielded Volume Dimensions

MATERIAL	DIMENSIONS FOR ALL SURFACES SHIELDED (Meters)					
	INSIDE DIA.	INSIDE LEN.	MIN. OUTSIDE DIA.	MAX. OUTSIDE DIA.	MIN. OUTSIDE LEN.	MAX. OUTSIDE LEN.
Composite	1.07	2.13	1.33	1.47	2.4	2.53
Water	1.07	2.13	1.47	1.67	2.53	2.73
Fuel	1.07	2.13	1.64	1.82	2.7	2.88

MATERIAL	STORAGE VOLUMES (Cubic Meters)					
	NOT IN USE		ERECTED		STORED+ERECTED	
	MIN.	MAX.	MIN.	MAX.	MIN.	MAX.
Composite	3.35	4.28	3.35	4.28	3.35 *	4.28 *
Water	1.67	2.49	4.28	4.61	5.95	7.1
Fuel	2.08	3.08	5.69	7.47	7.77	10.55

* Single Storage

Figure 4.12.8-1 Single Occupant Solar Flare Shelter

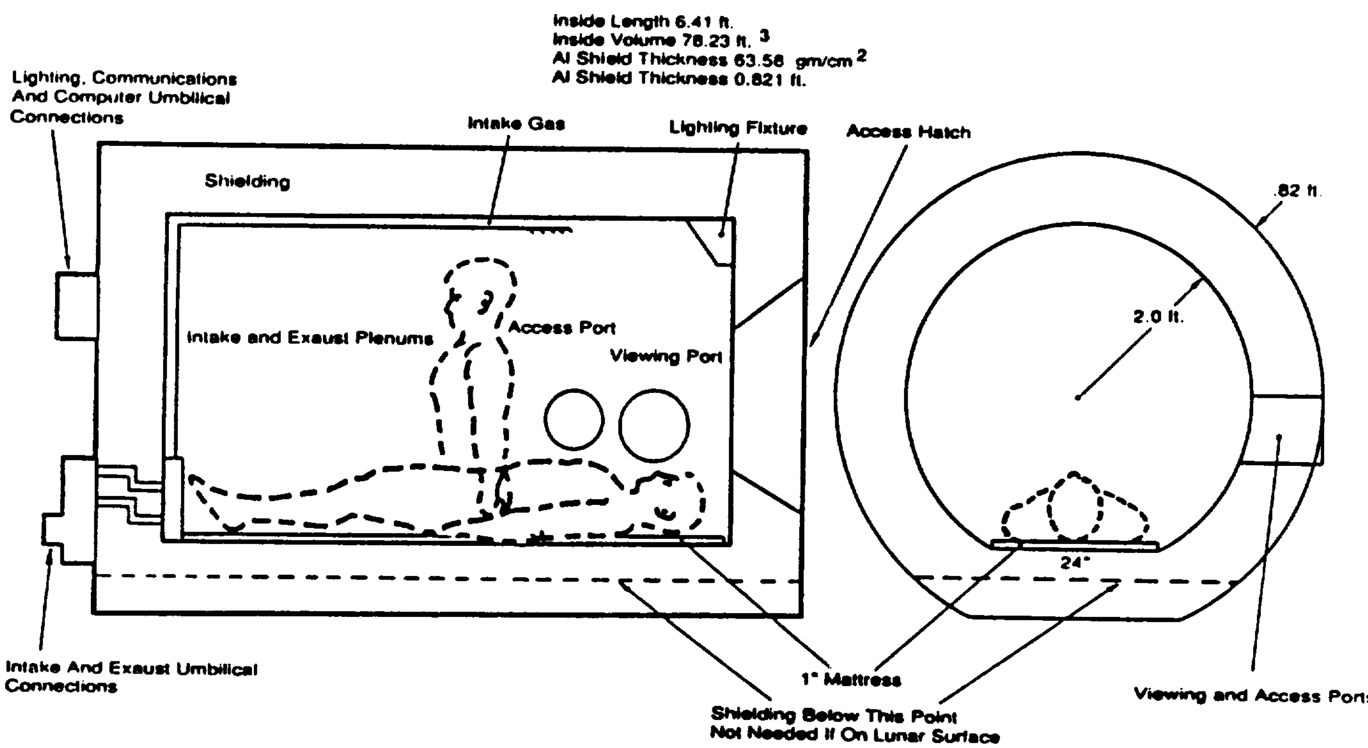


Table 4.12.8-2 Weight Estimate for Single Occupancy Solar Flare Shelter

The shelter is a closed volume. Construction-composite epoxy/carbon fiber. The shelter is a cylinder with a segment of the circular cross section at the bottom removed to allow the crew to rest flat on a surface. The bottom may or may not require shielding.

VOLUME	=	V	=	60.00 cu. ft.	(Habitable Volume)
LENGTH	=	L	=	7.00 ft.	(Habitable Cylinder Length)
SECTION AREA	=	A	=	8.57 sq. ft.	(Crossec. of Habitable Vol.)
CHORD LENGTH	=	W	=	24.00 Inches	(Bottom Section)
RADIUS	=	R	=	1.71 ft.	(Habitable Volume)

X is angle subtended by radii to center of chord & one end

X = Arcsin	(W/2R)	=	35.80 Degrees
	(W/2R)	=	.58

ASSUMED SHIELDING THICKNESS FROM 20 TO 30 GM./SQ. CM.

DENSITY = 1.5 gm/cu. cm. or	=	93.6 lb./cu. ft.
SHIELD THICKNESS	=	.437 to .656 ft.
MEAN	=	.5465 ft.
RADIUS MIDDLE OF SHIELD (MEAN)	=	2.26 ft.
MASS CYLINDER SURFACE	=	2399.0 lb.
MASS BOTH ENDS	=	876.0 lb.
MASS SHIELDED BOTTOM	=	716.0

TOTAL MASS ALL SURFACES SHIELDED: 3,992 lb (Mean wt)
3,192 lb (Lowest wt)
4,792 lb (Highest wt)

MASS BOTTOM UNSHIELDED = 27.30 lb assumes .25 in. thickness

TOTAL MASS NO BOTTOM SHIELD: 3,303 lb (Mean)
2,646 lb (Lowest wt)
3,959 lb (Highest wt)

COST RANGES FROM 20 TO 260 $/LB. FABRICATED*

*"Aerospace Materials" Aerospace America/June 1987/Anon. p52 (60/100 $/lb.)

*"Pace of structural materials slows for commercial transports"
Alan S. Brown, Aerospace America/June 1987 pp. 18-28 (240/260 $/lb.)

*"Materials pace ATF design" Aerospace America/April 1987, Alan S. Brown
pp. 17-22 ("20/40 $/lb. prepregs, 65/75 $/lb. bismaleimide, 80/100 $/lb. thermoplastic prepregs)

Assume low cost fabrication at 20 to 50 $/lb.

COST	ALL SURFACE SHIELDED		BOTTOM NOT SHIELDED	
	@ $20/lb.	@ $50/lb.	@ $20/lb.	@ $50/lb.
Mean	$79,840	199,602	$66,064	165,160
Lowest	$63,843	159,608	$52,936	132,341
Highest	$95,838	239,595	$79,191	197,979

Table 4.12.8-3 Weight Estimate for Double Occupancy Solar Flare Shelter

This shelter is made by adding a 2 ft. wide rectangular section to the center of single occupancy shelter.

VOLUME INCREASE	=Delta V	=	70.00 cu. ft.
TOTAL VOLUME	=V	=	130.00 cu. ft.
DELTA AREA TOTAL	=Delta A	=	28.00 sq. ft.
DELTA MASS TOTAL ALL SURFACES SHIELDED			
MEAN	=Delta M	=	1430 lb.
LOWEST WEIGHT	=Delta L	=	1145 lb.
HIGHEST WEIGHT	=Delta H	=	1719 lb.

TOTAL MASS ALL SURFACES SHIELDED

5,423 lb. (Mean Weight)
4,337 lb. (Lowest Weight)
6,511 lb. (Highest Weight)

MASS OF BOTTOM UNSHIELDED	27.30 lb. (.25 in. thickness)
TOTAL MASS NO BOTTOM SHIELD	
MEAN =	3,603 lb. (Mean Weight)
LOWESTWEIGHT=	2,949 lb. (Lowest Weight)
HIGHESTWEIGHT=	4,263 lb. (Highest Weight)

DELTA COST	ALL SURFACES SHIELDED		BOTTOM NOT SHIELDED	
	@ $20/lb.	@ $50/lb.	@ $20/lb.	@ $50/lb.
	28,619	71,547	14,855	37,138
	22,905	57,264	11,998	29,997
	34,384	85,962	17,378	44,346

TOTAL COST	ALL SURFACES SHIELDED		BOTTOM NOT SHIELDED	
	@ $20/lb.	@ $50/lb.	@ $20/lb.	@ $50/lb.
	108,460	271,150	72,070	180,176
	86,749	216,873	58,990	147,476
	130,223	325,558	85,270	213,175

5.0 Conceptual Designs

5.1 Local Transportation Vehicle, Unpressurized (LOTRAN)

5.1.1 Design Requirements

Several aspects of lunar mobility systems must be considered in detail before credible surface transportation vehicles can be designed. These aspects include; locomotion methods, articulation, suspension, vehicle mass, power systems, and maximum speed. A brief evaluation and baseline selection for the LOTRAN is developed in a conceptual vehicle configuration in this section.

Weight, packing, and vehicle deployment were the prime design drivers for the Apollo Lunar Roving Vehicle (LRV). The LRV had to be deployed in less than an hour and had severe packing constraints. Six wheels could not be packed and, while wire mesh wheels had poor durability; they only weighed 5.4 kg (12 lb) each. New evaluation criteria will be used when future lunar surface transportation systems are designed.

In future designs, mass, packing, and deployment will still be important considerations, but they cannot be allowed to drive the design. At a lunar base, a vehicle can be truly assembled. Long mission durations make wear and performance more important than mass and packing. Table 2.2.4-1 documents the LOTRAN functional system requirements.

5.1.2 Conceptual Design Definition

5.1.2.1 Locomotion

As on Earth, wheeled systems provide the best locomotion systems for almost every lunar application. Three very different types of wheels have already been successfully used on the Moon. The Apollo 14 Modularized Equipment Transporter (MET) had pressurized (4 psi) tires, the Soviet Lunokhod had eight almost rigid wheels, and the LRV used four flexible wire mesh wheels. Wheels are mechanically efficient, can be designed into lightweight systems, and can be built to have excellent reliability. One problem with wheels in terrestrial all-terrain applications is they tend to have a small footprint. In the reduced gravity field of the Moon, this is usually not a problem.

Wheels have tremendous versatility. There is a wide range of wheel types, sizes, numbers, and configurations. Metal-elastic wheels and cone wheels deserve special attention. They both have the potential to support large loads, provide passive suspension, have excellent wear resistance, and accommodate uneven terrain.

Other mobility concepts were studied and dismissed. Tracks tend to be heavy, mechanically complicated, and unreliable. Screw driven buoyant vehicles are heavy and power hungry. Walkers are complicated, unreliable, and require extensive control systems to operate. Free flyers have more unsafe failure modes and require more operator training.

Cone wheels, metal-elastic wheels, and space-linked tracks are the three locomotion methods that show the most promise. Cone wheels can support large loads. Metal elastic wheels are reliable and can provide passive suspension. Space-linked tracks can move heavy loads and may be appropriate for lunar surface vehicles supporting construction and assembly operations.

The LOTRAN conceptual design utilizes six metal elastic wheels because this vehicle is a workhorse, utilitarian vehicle that will be used extensively. The reliability and simplicity of metal elastic wheels are well suited to the needs of the LOTRAN. The wheels are tall and narrow having a diameter of 1.35 m and width of 20 cm.

5.1.2.2 Chassis Articulation

Chassis articulation is the key to obstacle crossing capability. However, articulation places constraints on the size and placement of payloads and articulation adds weight and complexity to the vehicle. If the LOTRAN is to be the personnel, all-terrain vehicle of the Moon, the vehicle frame should be articulated. In this conceptual design, an articulation joint has been located between each of the three sets of metal elastic wheels. Plus or minus 30° of movement is allowed about each of the pitch, roll, and yaw axes. The LOTRAN articulated chassis is illustrated in Figure 5.1.2.2-1.

The three sections of the LOTRAN are referred to as the cab (over the front wheels), the bed (over the center wheels), and the trailer (over the rear wheels). The trailer articulation joint can be disconnected converting the LOTRAN into a 4x4 vehicle. The wheelbase between each set of wheels is 1.85 meters. The tread width is 1.8 meters and the overall vehicle width is 2 meters. The total LOTRAN length is 5 meters. Without the trailer section, the LOTRAN length is 3.2 meters.

5.1.2.3 Suspension

The LRV had active and passive elements in its suspension system. For a new lunar surface transportation vehicle moving at low speeds, dynamic loads can be passively absorbed by a component of the vehicle such as the metal-elastic wheel. When vehicle speeds exceed 8 km/hr (5 mph), dynamic loads become important. Although the simplicity of a passive suspension system is highly desired, vehicles carrying crew should incorporate an active suspension system element to meet the larger dynamic forces caused by greater vehicle speeds. Therefore, the LOTRAN is planned to have an active suspension element. The specific implementation design will require more operations requirement information and detailed design analysis.

5.1.2.4 Vehicle Weight

Conceptual designs of various surface vehicles including a truck, an excavator, and an unpressurized crew transporter have been investigated. The weight estimates for these vehicles give a vehicle weight-to-payload ratio of approximately 0.65. This is higher than the LRV ratio of approximately 0.44. This weight penalty is needed to provide the durability and mechanical reliability needed for these new long life vehicles. Since the design payload mass is 850 kg, the LOTRAN mass is calculated as 0.65 times 850 or a result of 550 kg. The total mass of the loaded LOTRAN is 1,400 kg.

5.1.2.5 Energy and Power Requirements

All locomotion power requirements for mobility have been scaled at a baseline rate of 0.08 wh/kg/km. The minimum locomotion energy required is 11.2 kwh (0.08 wh/kg/km * 1,400 kg * 100 km). A contingency factor of 50 percent has been used to establish the operational locomotion energy requirement as 16 kwh. Other energy requirements are tabulated in Table 5.1.2.5-1. The result is a power requirement while moving of 2.15 kw and 1.3 kw while parked on station. The total energy that must be provided by the LOTRAN is 21 kwh.

Batteries work well and are very reliable for the short duration missions of the LOTRAN. Four lithium-metal sulphide batteries have been selected to provide the necessary operations energy. Each 36-volt battery weighs 48 kg and is rated at 146 ampere-hours.

5.1.2.6 Crew Stations

The permanent crew station from which the LOTRAN is operated is located on the cab section. Two crewmembers sit side-by-side after entering from the front of the vehicle. The front guard rail drops down to form an entry step. A center console provides control displays allowing either crewmember to drive.

The bed section of the LOTRAN is a utility bed for multiple uses. In one configuration, seats for two additional crewmembers can be installed. These two crewmembers are passengers.

5.1.2.7 Vehicle Control

The vehicle has a 6x6 drive train to provide the best obstacle performance. Six electric motors are mounted near the wheel hub and drive the wheels directly. LOTRAN steering is accomplished by controlling differential wheel speed which is consistent with a design approach of building a simple, rugged, durable vehicle.

5.1.3 LOTRAN Configuration Description

The LOTRAN is designed to provide mobility for all lunar base EVA tasks. Payload capacity and maximum range from the base are sized to meet EVA requirements; four crew with 130 kg payload or two crew with 490 kg payload and 50 km from the base. The maximum velocity for wheeled surface vehicles is established by the lunar terrain and will range from 10 to 15 km/hr. Driving time will be set by EVA duration at approximately eight hours per mission. Although the LOTRAN is only used away from the base in lunar daylight, two headlights are provided just below the wheel axle level for night operations around the base and for operation in shadows on trips away from the base. Table 5.1.3-1 lists pertinent items providing a summary definition of the LOTRAN.

The LOTRAN as configured in this conceptual design is a six-wheeled, articulated vehicle. Tall, narrow, metal-elastic wheels provide sufficient ground contact area and are tall enough to provide a high degree of passive suspension. Cone wheels might also meet performance requirements but tend to be too rigid for this lightweight vehicle. Active suspension is provided to permit speeds of 15 km/hr over uneven terrain. Articulation of up to plus or minus 30 degrees about the pitch, roll, and yaw axes facilitates steering and improves obstacle crossing capabilities.

If desired, the last set of wheels and frame can be detached by removing a pin at the yaw joint. The vehicle can then be driven as a 4x4 vehicle with limited payload capacity.

Figure 5.1.2.2-1 LOTRAN Articulated Chassis

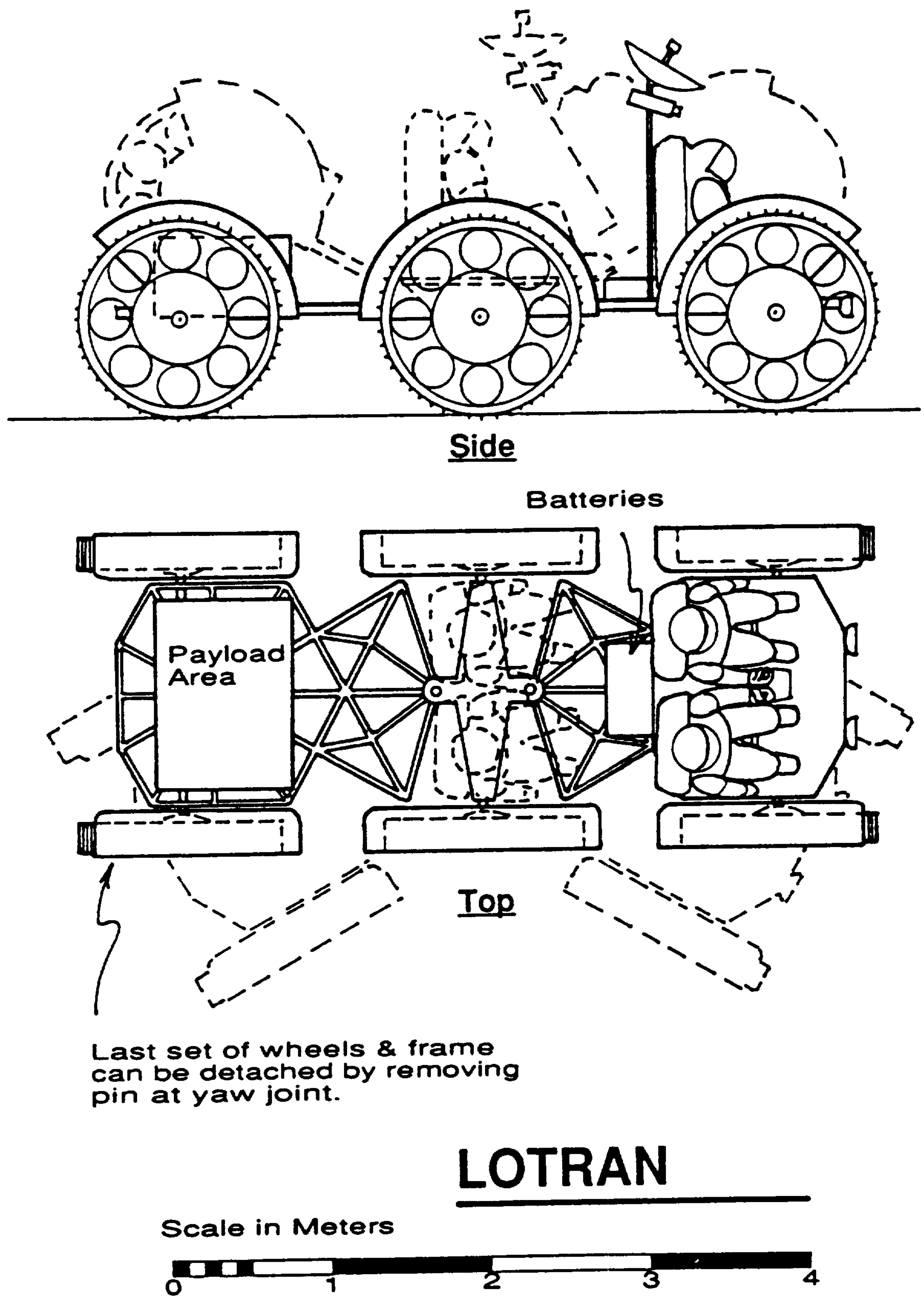


Table 5.1.2.5-1 LOTRAN Electrical Energy Specification

LOTRAN FUNCTION	POWER (kw)		TIME USED (hours)	ENERGY (kwh)
	MOVING	PARKED		
Locomotion	1.600	0	7	16.0
Control Electronics	0.08	0	7	0.6
Navigation	0.02	0	7	0.2
Lights	0.05	0.05	2	0.1
Thermal	0.1	0.1	8	0.8
Communications	0.15	0.15	8	1.2
Science and Applications	0.15		7	1.1
		1.0	1	1.0
TOTAL	2.15	1.3		21.0

Table 5.1.3-1 LOTRAN Configuration Definition

Three Articulated Sections:	Cab, Bed, and Trailer
Number of Wheels:	6
Wheel Type:	Metal-Elastic
Wheel Diameter:	1.35 m
Treadwidth:	1.8 m
Overall Width:	2.0 m
Suspension:	Passive + Active
Cab-to-Bed Wheelbase:	1.85 m
Bed-to-Trailer Wheelbase:	1.85 m
Total Length:	5.05 m
Detachable Section:	Trailer
Length Without Trailer:	3.2 m
Permanent Crew Stations:	Two on Cab
Installable Passenger Stations:	Two on Bed
Cargo Areas:	Bed and Trailer
Power Source:	Four Lithium-Metal Sulphide Batteries
Battery Specification (each):	36 volt, 146 Amp-hr, 48 kg
Total Energy Stored:	21 kwh
Locomotion Power Requirement:	1.6 kw
Maximum Power Requirement:	2.15 kw
Gross Payload Mass:	850 kg (2 Crew + 490 kg or 4 Crew + 130 kg)
Vehicle Mass:	550 kg
Total LOTRAN Mass (Loaded):	1,400 kg
Range:	100 km
Operational Speed:	15 km/hr
Maximum Driving Duration:	7 hrs

5.2 Mobile Surface Applications Traverse Vehicle (MOSAP)

The mobile surface applications (MOSAP) is a pressurized vehicle intended to carry out a variety of transportation missions on the lunar surface. These missions can be short duration local trips and transfers, medium duration sortie missions, or long duration traverses. To accommodate these missions, the MOSAP is conceived as a system of modular elements.

To suit the needs of a particular mission, the MOSAP is outfitted with the appropriate elements. Among the elements are a primary control research vehicle (PCRV), a supplemental auxiliary power cart (APC), a habitation trailer unit (HTU), and an experiment and sample trailer (EST). For the purposes of this study, these four elements will be considered. There are certain to be other elements and some experiments may be mounted on their own dedicated element. The PCRV is essentially a stand alone vehicle capable of accomplishing short and medium range missions by itself. The APC provides energy reserves needed to accomplish longer duration traverses and missions with power and energy intensive experiments. The HTU is carried on long duration missions to provide increased living space and more work room for maintenance and servicing of EVA equipment. Finally, the EST is used to carry experiments out to their destinations and to carry samples back to the base. The EST will be outfitted with a teleoperated manipulator to allow cargo removal and loading without EVA.

The fully outfitted MOSAP resembles a train; each element is connected to the other with the PCRV leading. The elements are provided with individually powered wheels and in some cases may be driven separately by teleoperation. The MOSAP modular design provides a means for satisfying a broad range of applications. By itself, the MOSAP PCRV is capable of short missions. Each of the other elements may be used by themselves although not necessarily for transportation operations. The HTU can be used for a temporary outpost for remote site projects. The APC has numerous applications at the base such as providing power to flight vehicles while they are on the surface.

5.2.1 Design Requirements

The requirements of the MOSAP must be defined with respect to the mission it must accomplish. Table 2.2.4-1 presents the desired and required capabilities of the MOSAP system. To provide a basis for the design of the MOSAP, vehicle requirements are developed based on three design reference missions (DRM's). The first, DRM 1, is a short mission lasting two days or less; the second, DRM 2, is a medium range mission lasting 12 days and covering about 1000 kilometers; and the third, DRM 3, is a long traverse which represents the MOSAP limits of about 42 days and 3000 kilometers. The required capabilities listed in Table 2.2.4-1 are represented in DRM 2 while DRM 3 covers the desired capabilities.

DRM 1 is a short traverse within 50 km of the base. The vehicle would be pressurized and capable of carrying experimental equipment, 2 crewmen and their supplies. The total mission duration would not exceed 3 days. Total drive time for the 100 km round trip is approximately 10 hours if an average speed of 10 km/hour is assumed. The purpose of DRM 1 is to perform experiments local to the base but at distances large enough to preclude the use of unpressurized transport. The PCRV has the capability to "dock" with another pressurized structure. The crew would transfer from the base to the vehicle, travel to the experiment site, and begin EVA operations from the PCRV. This allows the use of EVA time in actual work performance instead of in transportation. The crew returns to the vehicle, spends the night, and can perform another EVA the next day if needed, before returning to the base. Pressurized transfer can also be used to get crewmembers to and from the base and other pressurized areas such as flight vehicles.

The experiments which would be of interest within close proximity of the lunar base include sample collection, light drilling and general mapping of the surface and subsurfaces. Equipment needed for this type of work ranges from small hand tools to seismographic types of tools. Samples collected on the surface will require shovels, scoops, rakes, small shallow drills and a variety of hammers. Sample containers must be provided on the vehicle since many items will be transported back to the base for further analysis. Since EVA is required by this mission, the vehicle must incorporate an airlock. The cabin will generally not be designed to be evacuated since this would require all crewmembers to be suited.

DRM 2, the second mission, is a mid-range traverse of up to 500 km or a round trip of 1,000 km. The vehicle carries a crew of 4 and their supplies for 12 days. A wide range of scientific experiments will be performed on this mission. The travel time needed to cover the 1000 km is 100 hours at an average speed of 10 km/hour. If driving activity takes place 12 hours per day, eight of the 12 days will be taken up by driving. Extensive EVA's will be required and the vehicle must have an airlock and consumables to support approximately 12 EVA events (two crew performing EVA on six occasions).

It is conceivable that the crew could drive for 2, 12-hour shifts to add working days to the mission, but sleeping during a bumpy traverse may not be acceptable. In addition, many of the experiments would require at least 3 crew members, two EVA and one IVA.

The type of experiments which would be carried out could range from sample collection to deep drilling. The objectives of the trip would be to accurately map the region travelled, to collect as much surface and subsurface data as possible, and to locate any useful mineral deposits within close proximity to the lunar base. Tools required for this work include many hand tools identified in the first scenario and such devices as profiling active seismic arrays, thumpers, explosive devices, a magnetometer, a gravimeter and surveying equipment.

A deep drilling device could also be included as an experiment. No definition of this device is available, but it is assumed that it will be large and probably carried as a separate wheeled vehicle behind the MOSAP.

The final mission, DRM 3, a long range traverse of approximately 1,500 km or a 3,000 km round trip. This mission represents the maximum for all MOSAP capabilities. The vehicle will carry 4 crewmembers and their supplies for a 42 day mission. Experiments for this mission will generally be the same as for DRM 2, but will obviously cover a larger lunar area. The travel time for a trip of this length at a speed of 10 km/hour is 300 hours. At 12 driving hours per day, this amounts to 25 days driving and 17 working days. Extensive EVA is anticipated and consumables for up to 36 EVA events would be carried.

The primary difference between DRM 2 and 3 is the distance and duration. Because the duration is longer, more volume will be needed for food, and other consumables and for crew free space. In addition, more extensive personal hygiene facilities will be needed for DRM 3 since the duration is so long. Reference 40 provides a graph of the free space needed for varying mission durations. In addition, because both the duration and distance are longer, the energy storage requirements for DRM 3 will be higher than for

DRM 2. Table 5.2.1-1 shows the overall mission requirements and estimated subsystem and consumable volumes. In addition, an estimated length is shown for a 3 meter diameter cylindrical shell.

5.2.2 Configuration Options

There are two approaches to configuring the MOSAP system. First, the MOSAP can be one vehicle with sufficient space and consumables storage capability to accommodate DRM 3. Second, the MOSAP can be a modular system of elements with one of the elements designed to handle one of the shorter design reference missions. In examining the merits of these two approaches, the system adaptability must be considered.

Obviously, the single vehicle can be used for the two shorter duration missions. If the system is designed to handle a 42 day mission, it will certainly have enough room to handle the lower free space and consumables volumes of the shorter missions. In addition, if it has enough energy storage for the long traverse it will have ample storage for DRM's 1 and 2. However, the size of the single vehicle is very large. As shown in Table 5.2.1-1, the total length will be about 12 meters assuming a 3 meter diameter. This is nearly the length of a Space Station module and will make the use of the MOSAP cumbersome. Because of its length, it will be difficult to maneuver. The entire vehicle must be prepared for each mission regardless of whether all the capabilities are needed. Further, while the vehicle is on any mission, no servicing can be performed on individual items. In general, the single MOSAP vehicle size makes it suitable for long duration missions but somewhat over designed for shorter missions.

The multiple vehicle MOSAP will be configured as a system of vehicles. The primary vehicle would be designed to accommodate one of the shorter missions and additional vehicles would be set up to add extra energy and power systems, habitation and consumables as needed. The MOSAP will have to be designed to function as a train-like articulated vehicle no matter what configuration option is chosen since large experiments must be handled as separate elements. This system will allow much better maneuvering capabilities than the single vehicle for short duration missions. Long duration missions will have some degradation due to the control of a long train of MOSAP elements. The multi-vehicle system answers the servicing and preparation problems of the larger vehicle since only the elements necessary for a mission are prepared and the others can be used elsewhere or serviced while a PCRV is out on a mission.

The multi-vehicle configuration approach is chosen as a baseline for the MOSAP in this study. The primary vehicle is designed with appropriate interior volume for DRM 2. The power system has adequate energy storage for DRM 1. The DRM 1 configuration would use only the primary vehicle. For DRM 2, the primary vehicle, a flatbed trailer type device and a supplemental power system would be needed. DRM 3 would require the primary vehicle, the trailer, the power supply and an additional habitation element. This configuration provides an adaptable system to handle capability requirements and allows expansion with additional elements to achieve the desired capabilities.

5.2.3 Conceptual Design Definition

The MOSAP system can consist of a number of different elements. The primary element will be designed to handle DRM 1, the short range mission with no other elements. This element can act as a stand alone vehicle. Called the Primary Control Research Vehicle (PCRV), the element would be delivered as the first element of the MOSAP. The remaining elements could be delivered at the same time or later as mission planning dictates. A supplemental power system will be designed to provide the increased energy requirements of the medium missions such as DRM 2. The power supply will be called the APC for Auxiliary Power Cart. The power will be supplied by a fuel cell and supplemented by a solar panel. Energy storage will be by liquid oxygen and hydrogen for the fuel cell. A supplemental habitation element called the Habitation Trailer Unit (HTU) will provide increased volume for the long missions. The HTU gives crewmembers the increased free space they will need for longer periods and provides some redundant subsystems, extra consumables, and better personal hygiene capabilities. The HTU is designed so that it can operate connected to the PCRV or separate from the PCRV. Finally, an external payload carrying element, the Experiment and Sample Trailer (EST), will be used for transporting scientific instruments and emplacing them. In addition, the EST can be used to bring samples back from missions. The EST is outfitted with a remote manipulator to allow emplacement of instruments and other cargo handling without the need for EVA.

5.2.3.1 Primary Control Research Vehicle

The PCRV is a four-wheeled vehicle, 8 meters long, about 4.5 meters tall, and 4 meters wide. Figure 5.2.3.1-1 is a drawing of the vehicle exterior. The pressurized shell consists of a 6-meter long, 3-meter diameter cylinder with 1-meter long spherical ends. The vehicle has two driving stations. The primary driving station is located at the top of the vehicle near the forward end of the cylinder. The driver sits beneath a bubble with a movable sunshield. At this level the driver has a broad view of the terrain for navigation and a good view of the surface starting about five meters ahead for driving. A secondary drive station is located lower and at the front to allow better views of rocks when the vehicle must negotiate boulder fields. At the rear of the cylinder is a phased array antenna for vehicle-to-Earth and vehicle-to-base communications. Between the primary drive station and the antenna is the thermal control system radiator. The choice of wheel type has not been made in detail although the vehicle shown uses 2-meter diameter individually powered and suspended cone-wheels for locomotion. The fuel-cell power system is located outside the pressurized volume on one side of the vehicle between the wheels while the active thermal control system is located on the other side. Fuel cell water is piped to the potable water tanks inside the vehicle. The rear of the pressurized cabin is fitted with a standard docking adapter so the PCRV can be docked to other pressurized areas of the lunar base and to allow it to be mated with the HTU. Overall, the PCRV masses about 5,000 kilograms. These masses are broken down in Table 5.2.3.1-1.

The interior of the vehicle provides a total of 50 cubic meters of pressurized volume. Of this volume, 14 cubic meters of free space needed for DRM 2 are provided for the crew. This volume represents about 3.5 cubic meters per crewmember. Figure 5.2.3.1-2 is a drawing of the interior layout of the PCRV. Two man-lock style airlocks are located at the front of the vehicle on either side. The locks open beneath the vehicle so the astronauts do not have to climb down off the vehicle when exiting. Because the cabin is 1.5 meters above the surface, EVA astronauts will have to stoop to get out from under the vehicle. Using a hoist in the man-lock, this configuration also provides an easy means of lifting samples into the man-lock if needed or for lifting a disabled astronaut. Clear space 1.5 meters wide by 2.2 meters high begins at the primary drive station, runs down the center of the cabin, and ends at the rear of the vehicle. Along this corridor are the teleoperation station, the vehicle avionics and communications subsystems, the galley and waste disposal stations, the personal hygiene station, and crew sleeping and living areas. The power distribution wiring and the ECLSS hardware are located below the floor of the clear space.

5.2.3.2 Habitation Trailer Unit

The HTU is a four-wheeled vehicle which is the same basic size as the PCRV. Its prime purpose is to provide additional space for crew operations, additional storage volume, and additional EVA servicing area for the long duration missions such as DRM 3. The HTU is a four-wheeled vehicle 8 meters long, about 4.5 meters tall and 4 meters wide. Figure 5.2.3.2-1 is a drawing of the vehicle exterior. The pressurized shell consists of a 3-meter diameter cylinder with 1-meter long spherical ends. In general, the exterior of the HTU resembles a PCRV without a driving station on top. The radiator is located in the same area and the antenna is required to provide communications when the HTU is used separately from the PCRV. The thermal control and power systems are located between the wheels on either side of the pressurized cylinder. As with the PCRV, 2-meter diameter powered cone-wheels are shown for this vehicle. In this case both ends of the cylinder are fitted with standard docking adapters to provide pressurized access on both ends. This allows the HTU and PCRV to be prepared for a mission with connection to only one base location. In addition, if mission requirements dictate, more than one HTU may be used at one time. Table 5.2.3.2-1 lists the mass of the HTU which totals about 5,000 kilograms.

Figure 5.2.3.2-2 shows the interior of the HTU. The vehicle has the same general free space corridor and the same overall layout as the PCRV. The 27 cubic meters of free space provided bring the total free space to over 10 cubic meters per crewmember. Reference 40 indicates that this volume is good for missions of 40 days and over. The HTU is not typically fitted with an avionics or teleoperation station, and does not usually contain crew sleeping facilities or a galley. These facilities are provided in the PCRV and redundancy is probably not needed. The vehicle has two man-locks at the rear of the pressurized volume. These are configured the same as the man-locks in the PCRV, and are included to provide redundancy for the many EVA's needed for the long DRM 3. An additional hygiene station with a shower is provided to enhance the comfort of the crew during the long mission. A full shower is not included in the PCRV since it is used for short missions and sponge baths by the crew would be acceptable. More space is provided for experiments in the HTU so additional work can be done inside. The ECLSS and power distribution wiring is housed under the floor as in the PCRV. Finally, the HTU has additional space for consumables and waste storage.

5.2.3.3 Auxiliary Power Cart

The auxiliary power cart is intended to be an auxiliary power and energy system which provides additional power to the overall MOSAP on long and medium duration missions.
The MOSAP power requirements are developed in section 5.2.4.1. The system provides 14 kilowatts of electrical power and about 1,500 kilowatt-hours of energy for the 1,000 kilometer DRM 2. For DRM 3, the energy required would be around 7,000 kwh and the necessary power level would be 25 kw.

There are several options available for the supplemental power system. Among them, are fuel cells, photovoltaic arrays (solar cells), and batteries. Batteries will not be examined for this system since the storage requirement of 1,500 kilowatt-hours will result in a massive system. Masses as low the 9 kg/kwh of lithium-metal sulphide batteries would result in a 13.5 metric ton system. In addition, solar cells will not be considered as a primary power supply, since the MOSAP may be operated during the lunar night and solar cells could not be used for continuous power. In addition, the 14 kilowatts would require 60 square meters of solar cells with a mass of 500 kilograms. However, solar cells can be used as a supplemental source to the primary power generation system.

Fuel cell technology is well developed and application to the Space Shuttle and previous programs has proven it to be an operational technology. As a result, a fuel cell system is proposed as the primary power supply for the MOSAP and is used in the APC or "power cart".

The APC consists of cryogenic hydrogen and oxygen tanks, liquid water tanks, and a fuel cell system mounted on a four-wheeled cart as shown in Figure 5.2.3.3-1. When the MOSAP is prepared for DRM 2, the APC is connected to the end of the vehicle system and provides power for the entire MOSAP system. Upon return to the base, the APC can be recharged by connection to a regeneration station. The power cart can be used for other applications at the lunar base such as supplemental power for landers as described in the landing facility report of the Lunar Base Systems Study [Reference 26].

The estimated mass of the auxiliary power cart is 1,380 kilograms. Table 5.2.3.3-1 provides a mass breakdown and dimensional data for this system. The fuel cell system is sized as two 7-kilowatt Space Shuttle systems [reference 61]. This system is currently operational and can accommodate the power requirements easily as well as provide additional service for peak loads. Fuel cell reactant consumption of 0.36 kg/kwh results in storage requirements of 540 kilograms of fuel for DRM 2. Of this 60 kilograms is hydrogen and the remaining mass is oxygen. The hydrogen is assumed to boil off at a rate of around 1% per day, increasing the required amount by nearly 20 kilograms. No attempt has been made to analyze the thermal requirements or to perform trades of active and passive systems for this study. The boil-off is vented and not collected. The reactants required are slightly more than those contained in one Shuttle cryogenic tank set which is 40 kilograms of hydrogen and 354 kilograms of oxygen. As a result the sizes and masses are only slightly larger. Shuttle tank masses are around 2.4 kg/kg of hydrogen, 0.25 kg/kg of oxygen, and 0.25 kg/kg of water [reference 61]. Since the capacities are similar these factors were used to scale the power cart tanks. A 2.5 meter by 3 meter square solar panel is mounted above the tanks to provide additional power to off-load the fuel cells when solar energy is available. This gallium arsenide panel provides a maximum of about 1 kilowatt when the sun is at an appropriate angle and masses about 40 kilograms. The cart itself is assumed to be simple as shown in Figure 5.2.3.3-1 and has a mass of about 150 kilograms. The same methods of locomotion are used as were used with the HTU and PCRV. The fuel cell is not regenerative.

To accommodate the long DRM 3 which requires 7,000 kwh, five of these APC elements would be needed. If another APC is designed to meet the energy requirements of DRM 3, it will be substantially larger. Reactants would mass around 2,500 kilograms and overall vehicle mass would be about 5,000 kilograms. In appearance, the 7 megawatt-hour vehicle will be very similar to the smaller 1.5 megawatt-hour APC described above. Table 5.2.3.3-1 also presents the mass breakdown of the larger system. Depending on the needs of the MOSAP, there may actually be two types of APC, a 1.5 megawatt-hour cart and a 7 megawatt-hour cart.

5.2.3.4 Experiment and Sample Trailer

The EST is a utility element intended to be used to carry samples and instruments that will be used outside the vehicle. The EST can handle the 2,000 kilograms of payload required for the MOSAP system. Figure 5.2.3.4-1 is a drawing of the EST.

The mass of the EST is about 390 kilograms without payload. Table 5.2.3.4-1 lists the masses for various components. The bed is 4 meters wide by 4 meters long to provide 16 square meters of surface. The area is somewhat arbitrary and may be altered with only small impact on the vehicle mass since the bed should be light-weight. At the size shown, the mass should be less than 130 kilograms. The EST is provided with a remote manipulator system for loading and unloading equipment and samples. The mass of the RMS is assumed to be the same as that of the Space Shuttle RMS at about 40 kilograms. This may prove to be inappropriate because the Shuttle RMS and the EST RMS operate in such different environments, but even doubling the mass would not be significant to the EST with payload. The remainder of the EST is a base cart with four cone-wheels which is basically the same as the cart of the APC. A small fuel cell is provided to provide some power for the system while it is being used without the other MOSAP elements.

5.2.4 Subsystems

The subsystems of the MOSAP have been described in some detail in previous sections of this report. Additional description of the sizing and configuration are in order for some of the major subsystems.

5.2.4.1 Power

Power systems for the MOSAP elements are sized generally by the methods described for the APC above. The power and energy requirements for the PCRV and the HTU are developed using the subsystem powers described in Table 5.2.4.1-1. Locomotion power and energy is calculated based on the maximum 0.08 Wh/km/kg presented under the locomotion section of this report. Energy storage is found by the estimated distance traveled and an estimated average 1 kilowatt of power usage by the remaining systems. Some of the subsystems not typically used in the HTU are indicated since, at times, the HTU may be operated by itself.

The energy storage requirements for the APC are established for DRM 2 using the PCRV, the EST and the APC. Total expected mass will be around 8,800 kilograms and the expected distance is 1,000 kilometers for a total of 8.8 million kilogram-kilometers. At 0.08 Wh/kg-km about 700 kilowatt-hours are needed for locomotion. At an average of 1 kilowatt of subsystem power for 12 days, 300 more kwh are needed. The total APC energy needed for DRM 2 then will be 1,000 kwh. The APC described above has 1,500 kwh to provide 500 kwh for contingency. As for the power level, the 8,800 kilograms at the maximum 15 kph will need about 11 kilowatts. The 14 kw provided by the two Shuttle power plants provides an additional 3 kw for subsystem use.

For the larger APC designed for DRM 3, which has the PCRV, the HTU, the large APC, and the EST, the total estimated MOSAP is 17,600 kilograms for the 3,000 kilometer journey. This amounts to the need for about 4,200 kwh. The 2 kw average subsystem and experiment power for 42 days is another 2,000 kwh for a total requirement of 6,200 kwh. The large APC develops 7,000 kwh and provides for 800 kwh of contingency energy. The locomotion power level needed for the 17,600 kg system at 15 kph is about 21 kilowatts. The 25 kilowatts provides 5 kw for subsystem use.

The power requirement for the PCRV is shown to be 7 kilowatts maximum continuous and 12 kilowatts peak. Continuous power requirements are based on the diversity of use of the subsystems needing electrical power. The two general modes of operation are driving and stationary. While driving, the locomotion system will consume most of the power and only vehicle and crew survival subsystems such as thermal control, ECLSS, and some avionics will be active. While stationary, the locomotion subsystem will not draw any power and more onboard systems will be working, including experiments. The power requirements are the same as capabilities of one of the Space Shuttle fuel cell power plants. These fuel cells are each capable of providing 7 kilowatts continuous and 12 kilowatts peak power for a 15 minute period. The Space Shuttle fuel cell is therefore the baseline for the PCRV. The onboard storage system for the PCRV must be sized to handle at least half of the distance of DRM 2 to provide return capability upon failure of the APC. This results in storage for 200 kilowatt-hours of energy or 75 kilograms of reactants. Storage of these reactants will require 0.9 meter diameter hydrogen and waste water tanks, a 0.7 meter diameter oxygen tank.

The HTU will generally be supplied power from a APC. As such, it is conceivable that the HTU would not even need a power supply. However, since the HTU may be used by itself for some period, some type of power is desirable. The size of the system is chosen to be the same as for the PCRV more for purposes of commonality than for specific power level and subsystem power needs. Therefore, the HTU uses the same power system as the PCRV.

The EST will have only minimal power needs when it is operating without supplemental power. At full load it should not require more than 3 to 3.5 kilowatts for locomotion. The vehicle will only travel short distances so 40 kilowatt-hours or 15 kilograms of reactants are provided. This amount is enough for about 200 kilometers of travel at full load and requires three 0.5 meter tanks for reactants and waste water.

Power distribution will be accomplished by a bus system that will run through the length of each element. When the elements are connected, the buses from each element are also connected. Power can be distributed to individual element subsystems from this bus. The bus must be sized to handle at least the 25 kilowatt load needed by the DRM 3 configuration. At the standard spacecraft voltage of 28, the bus must handle a current of almost 900 amperes and the bus conductor must be 7 centimeters in diameter. This would mass about 700 kilograms for each element. It is evident that the system must run at a higher voltage to decrease the cable size. A system running at 400 volts will only have to handle less than 100 amperes and conductor size could be reduced to 0.7 centimeters with a mass per element of about 10 kilograms. The power distribution systems for individual elements require much more study.

5.2.4.2 Thermal Control

The thermal control system consists of both a passive and an active system. The passive thermal control system is mainly the insulation between the two shells of the pressure vessel. This system basically is used to isolate the exterior of the vehicle from the interior. The exterior surface preparation of the vehicle also is a part of the passive thermal control system. The coating will be white paint with a solar absorptivity no greater than 0.3 and an infrared emissivity no less than 0.8. Multilayer insulation is used between the two shells with the space between evacuated. This insulation will be about 1.5 to 2.5 centimeters thick and will have a conductivity of about 0.002 watts per meter-Kelvin. This combination effectively isolates the interior of the pressure vessel from the exterior. The loads on the interior from the exterior can require from about 50 watts cooling to about 150 watts of heating depending on day or night operation. Vehicle skin temperatures may vary as widely as 365 degrees Kelvin under direct sunlight to as low 200 degrees Kelvin at night.

Because the passive system isolates the interior of the vehicle, the loads on the active control system are relatively insensitive to the time of operation. This load is estimated to be about 1.4 kilowatts. Of this load, about 500 watts is from metabolic heat of the four astronauts and the remaining 900 watts is from the operation of equipment inside the vehicle.

The active thermal control system described here is one concept for removing internal loads. It is roughly analogous to the Space Shuttle ATCS. Detailed trade analyses must be performed before selection of a final system can be made. The system is a two loop single-phase, pumped fluid system. The external loop consists mainly of a radiator mounted on top of the vehicle, a water flash evaporator attached to the outlet of the radiator, a water chilling heat exchanger, and circulating pump. The working fluid is Freon as in the Shuttle system or another high performance low freezing point refrigerant. The Freon loop is maintained completely outside the pressurized volume because of safety and toxicity issues. The interior loop circulates chilled water through equipment heat exchanger and the ECLSS to cool the interior air, and rejects this heat to the exterior loop in the water chilling heat exchanger.

The external radiator provides about 14 square meters of surface area and can reject about 100 watts per square meter at about 290 degrees Kelvin while in direct sunlight. If loads exceed the radiator capability, the flash evaporator is activated. The evaporator uses water from the potable water tanks which also receive water from the fuel cells. The evaporator uses water at about 1.5 to 2 kilograms per hour for every extra kilowatt of load.

The thermal environment of the MOSAP has not been investigated in detail. The effects on the vehicle loads will be small because of the passive control system but the effects on the radiator can be profound. The vehicle can operate with the Sun at all angles depending on the time in the lunar day and the orientation of the vehicle. Lunar surface features such as mountains, crater walls, and large boulders may be "seen" by the radiator and thus may affect its efficiency. In addition, the loads in the vehicle will depend on a variety of factors including the level of activity by the crew and the equipment they are using. The nature of these loads and how they vary has not been investigated in this study.

5.2.4.3 Locomotion

The locomotion subsystem for the MOSAP has been examined in detail during this study. Wheels have been selected for a variety of reasons described previously in this report, but the size and configuration of the wheels has not been studied. The 2-meter diameter cone wheels shown are one concept for accomplishing wheeled locomotion. Each wheel is assumed to be individually powered for all the elements of the MOSAP. The low draw bar pull of wheeled vehicles and the high drag of unpowered wheels bulldozing through the lunar soil would preclude unpowered trailers. The MOSAP is semi-articulated in this configuration with each element having a rigid frame. The connections between vehicles provide the articulation. The diameter of the wheels shown, which provides good ground clearance, and the size of the vehicle is compatible with a four wheel configuration.

The issues surrounding the selection, sizing, number, and articulation of wheels has not been addressed in this study. The energy consumption baselined here is consistent with the Apollo data presented earlier, but a great deal of further study is needed on locomotion systems for the MOSAP.

5.2.4.4 Pressure vessel

The pressure vessel for the MOSAP PCRV and HTU are both double walled aluminum cylindrical vessels with semi-spherical ends. Aluminum is selected for the outer shell as a baseline over composites to provide a more conservative mass estimate. A cylindrical shape is selected because it provides good structural efficiency.

The walls are each about 0.3 centimeters thick to provide a factor of safety of 4 at one atmosphere interior pressure. The vehicle will nominally operate at lower pressures during missions requiring EVA. The base may operate at the higher pressure though and the vehicle will be mated with the base during servicing and therefore must be designed to the higher pressure.

5.2.4.5 Man-locks

The MOSAP uses man-lock type airlock systems as described previously in this report. The PCRV and the HTU each have two locks primarily so two crewmembers can exit at once and EVA astronauts are not left alone on the surface. In addition, two locks provide total system redundancy while adding less than 120 kilograms.

5.2.5 Conclusions

The conceptual design of the MOSAP system indicates that such a system is feasible. The MOSAP elements can be configured to accomplish the required capabilities as well as the desired capabilities.

The modular design allows it to be adapted to a wide variety of other missions. The system can be phased so that its capability grows. The PCRV can be brought to the Moon first to accomplish short missions such as DRM 1. Next, the APC and the EST can be supplied to allow for medium range missions such as DRM 2. Finally the large APC and the HTU can be brought to the Moon for the long duration missions such as DRM 3.

Based on the established feasibility, significant detailed design is in order as the next step of MOSAP system development. The overall configuration must be reexamined and the crew free space, and equipment and subsystem volumes reevaluated to verify the vessel size. Of the subsystems, the locomotion for this vehicle needs the most detail. Virtually no specific detail was developed for MOSAP locomotion except that wheels will be used. Detailed trade studies of the configuration, number and size of the wheels and the type of wheels must be done. The power distribution and generation systems also should be reviewed in detail. The distribution of power from the vehicle bus to the individual points of use and the control of and interfaces with the bus itself are among the items needing work. The particulars of the thermal control system must be defined. The exterior environment and its variation along with better estimations of the internal cooling loads and their diversity or duty cycle must be performed. The chilled water distribution along with the ECLSS must be detailed. The configuration and design of the pressure vessel must be confirmed and the rest of the structure of the vehicle must be designed. Dynamic as well as static structural loads require analysis.

Table 5.2.1-1 Dedicated Design Volume Requirements

COMPONENT	PCRV for DRM 1 (m^3)	PCRV for DRM 2 (m^3)	PCRV for DRM 3 (m^3)	HTU (m^3)
Crew Free Space	5.7	13.6	40.8	27.2
Man-locks	4.0	4.0	4.0	4.0
Drive Station	4.0	4.0	4.0	
Interior Items				
Avionics	1.7	1.7	1.7	
Teleoperation Sta.	1.7	1.7	1.7	
ECLSS	2.6	2.6	2.6	2.6
Experiments	1.7	1.7	1.7	1.7
Safety Equipment	1.7	1.7	1.7	1.7
Sleeping	3.4	6.8	6.8	
Personal Hygiene	1.7	1.7	1.7	1.7
Galley	1.7	1.7	1.7	
Shower			1.7	1.7
Consumables	0.0	0.3	1.5	1.2
Contingency and Wasted Space (20%)	6.0	8.3	14.3	8.3
TOTAL	35.9	49.8	85.9	50.1

PRESSURIZED CABIN	(meters)	(meters)	(meters)	(meters)
Diameter	3.0	3.0	3.0	3.0
Cylinder Length	4.0	6.0	11.0	6.0
End Length (each end)	1.0	1.0	1.0	1.0
Total Length	6.0	8.0	13.0	8.0
Total Volume	36.5	50.0	86.2	50.0

Figure 5.2.3.1-1 **Layout Drawing for the Primary Control Research Vehicle**

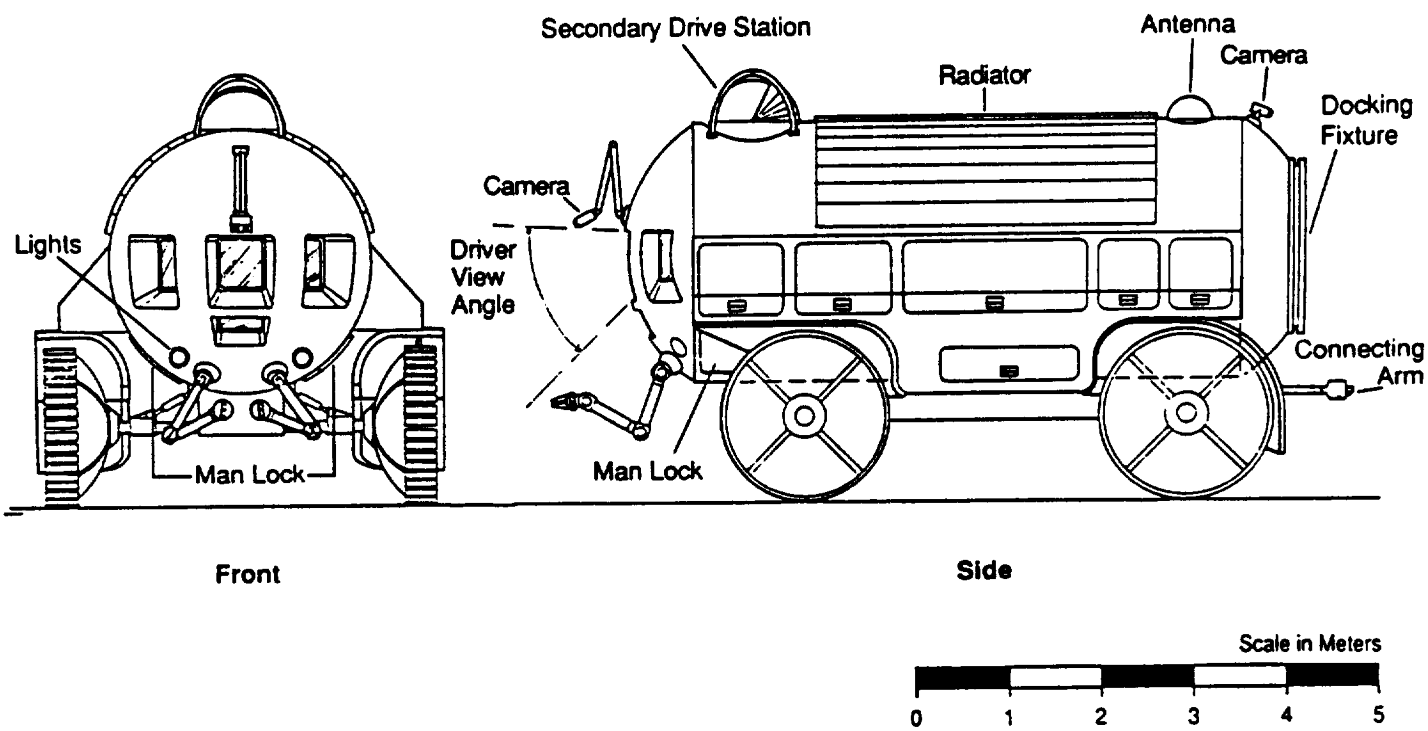


Table 5.2.3.1-1 **Primary Control Research Vehicle Weight Statement**

Structure and Pressure Vessel	
Inner Shell	490 kg
Outer Shell	500 kg
Other Structure	200 kg
Insulation	130 kg
Active Thermal System	
Radiator	160 kg
Pump	20 kg
Heat Exchanger	50 kg
Piping	100 kg
Refrigerant	300 kg
Power System	
Hydrogen Tanks	20 kg
Oxygen Tanks	15 kg
Water Tanks (incl. potable)	40 kg
Reactants	75 kg
Fuel Cell	90 kg
Power Distribution	100 kg
Wheels and Locomotion	300 kg
Man-locks	230 kg
Galley	70 kg
Personal Hygiene	90 kg
Emergency Equipment	30 kg
Avionics	90 kg
ECLSS	200 kg
Drive Stations	80 kg
Workstation	40 kg
Sleep Quarters	60 kg
Crew	360 kg
EMU's (3)	680 kg
Experiments and Payload	500 kg
TOTAL	5,020 kg

Figure 5.2.3.1-2 Primary Control Research Vehicle Interior Layout Drawing

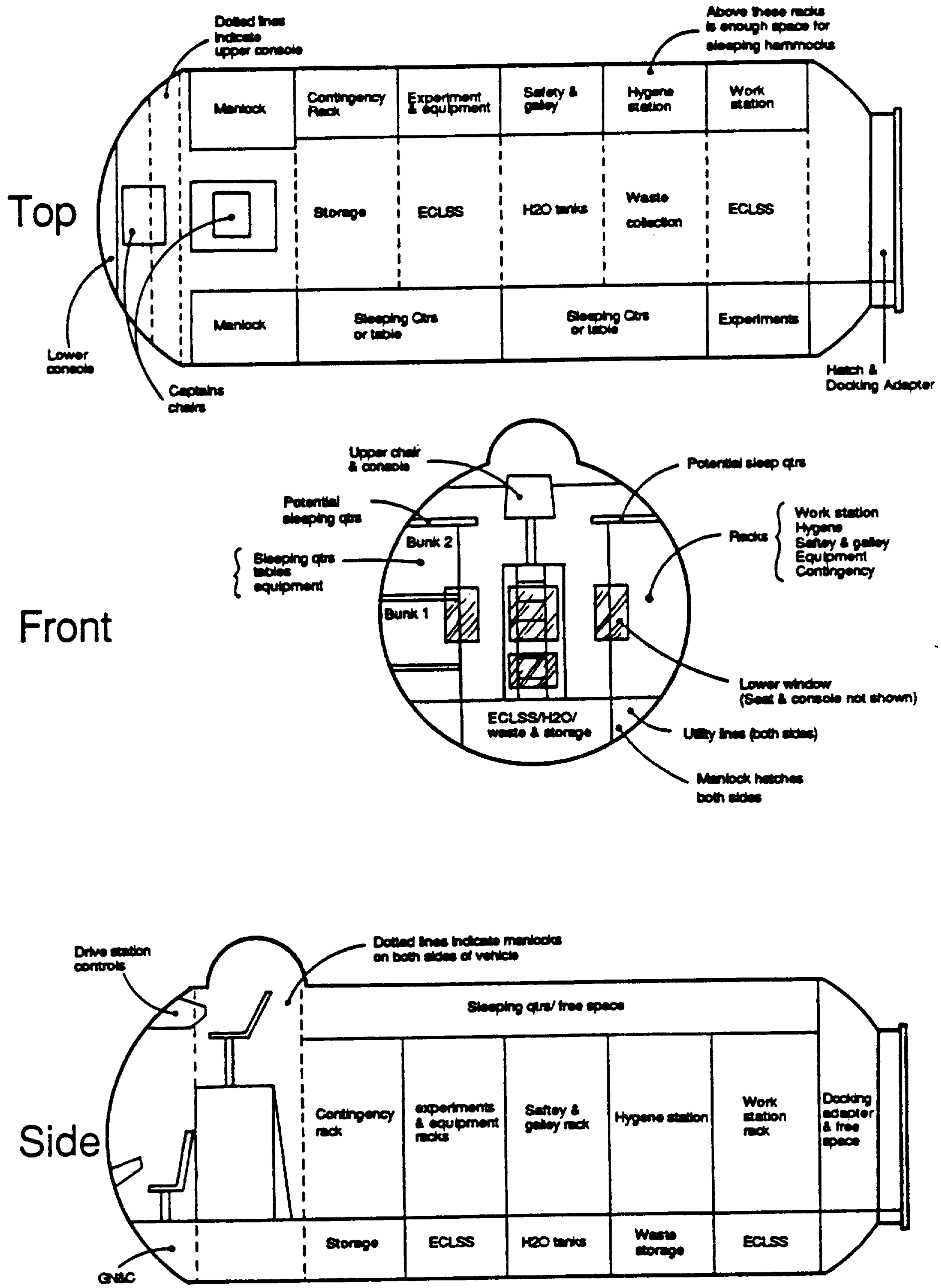


Figure 5.2.3.2-1 Layout Drawing for the Habitation Trailer Unit

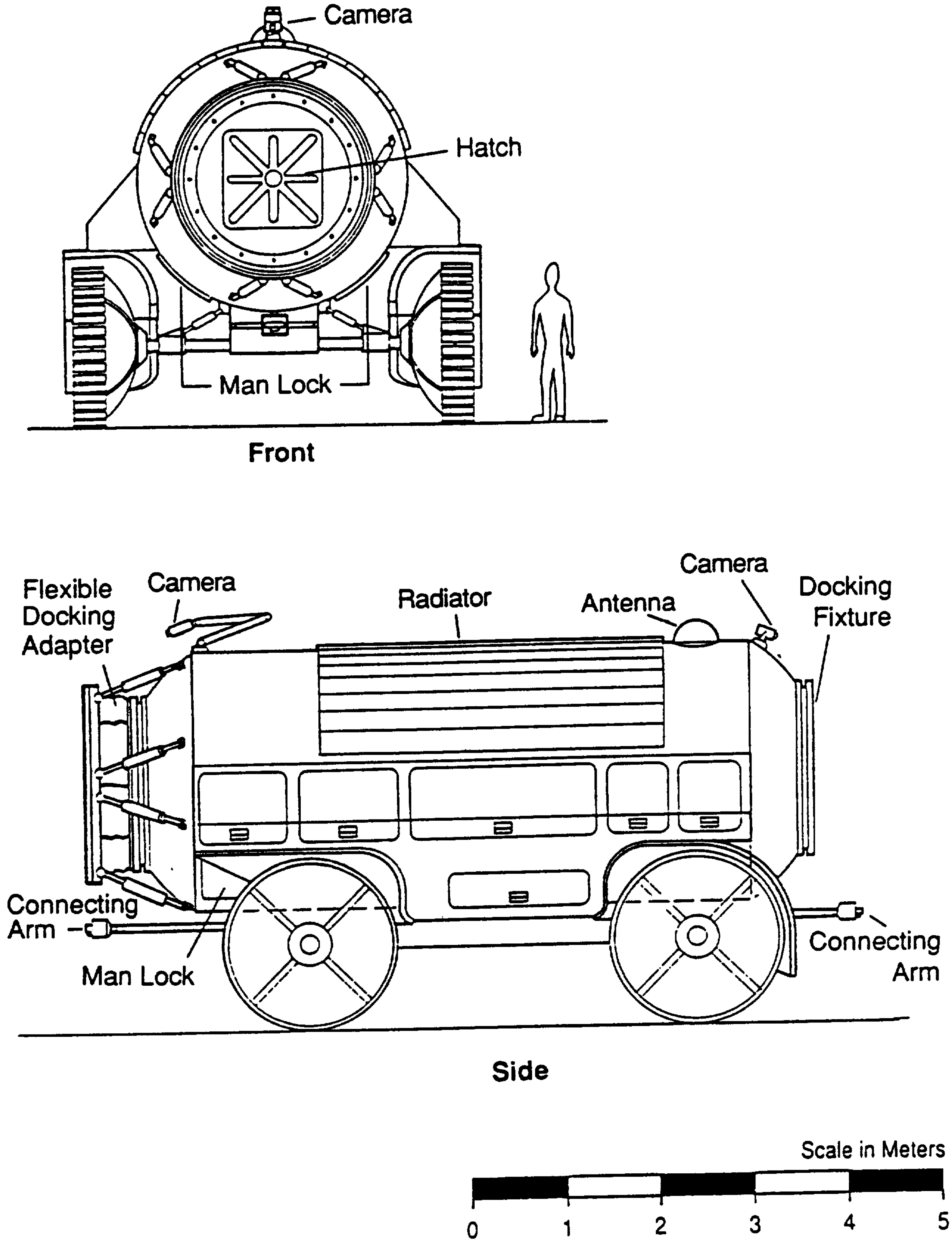


Table 5.2.3.2-1 Habitation Trailer Unit Weight Statement

Structure and Pressure Vessel	
Inner Shell	490 kg
Outer Shell	500 kg
Other Structure	200 kg
Insulation	130 kg
Active Thermal System	
Radiator	160 kg
Pump	20 kg
Heat Exchanger	50 kg
Piping	100 kg
Refrigerant	300 kg
Power System	
Hydrogen Tanks	20 kg
Oxygen Tanks	15 kg
Water Tanks (incl. potable)	40 kg
Reactants	75 kg
Fuel Cell	90 kg
Power Distribution	100 kg
Wheels and Locomotion	300 kg
Man-locks	230 kg
Galley	70 kg
Personal Hygiene	90 kg
Shower	80 kg
Emergency Equipment	30 kg
Avionics	90 kg
ECLSS	200 kg
Workstation	40 kg
EMU's (3)	680 kg
Experiments and Payload	900 kg
TOTAL	5,000 kg

Figure 5.2.3.2-2 Habitation Trailer Unit Interior Layout Drawing

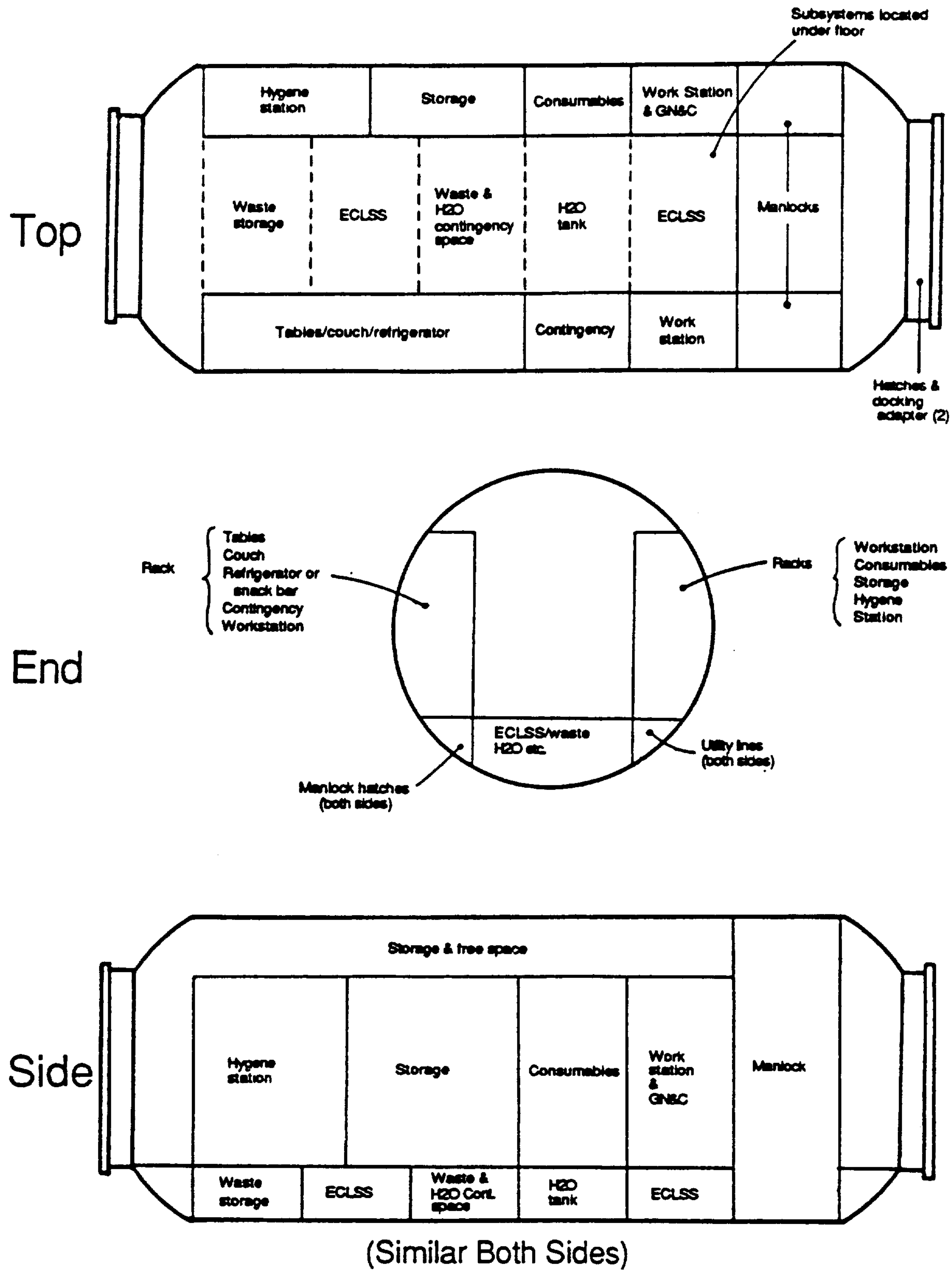


Figure 5.2.3.3-1 **Layout Drawing for the Auxiliary Power Cart**

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OF POOR QUALITY

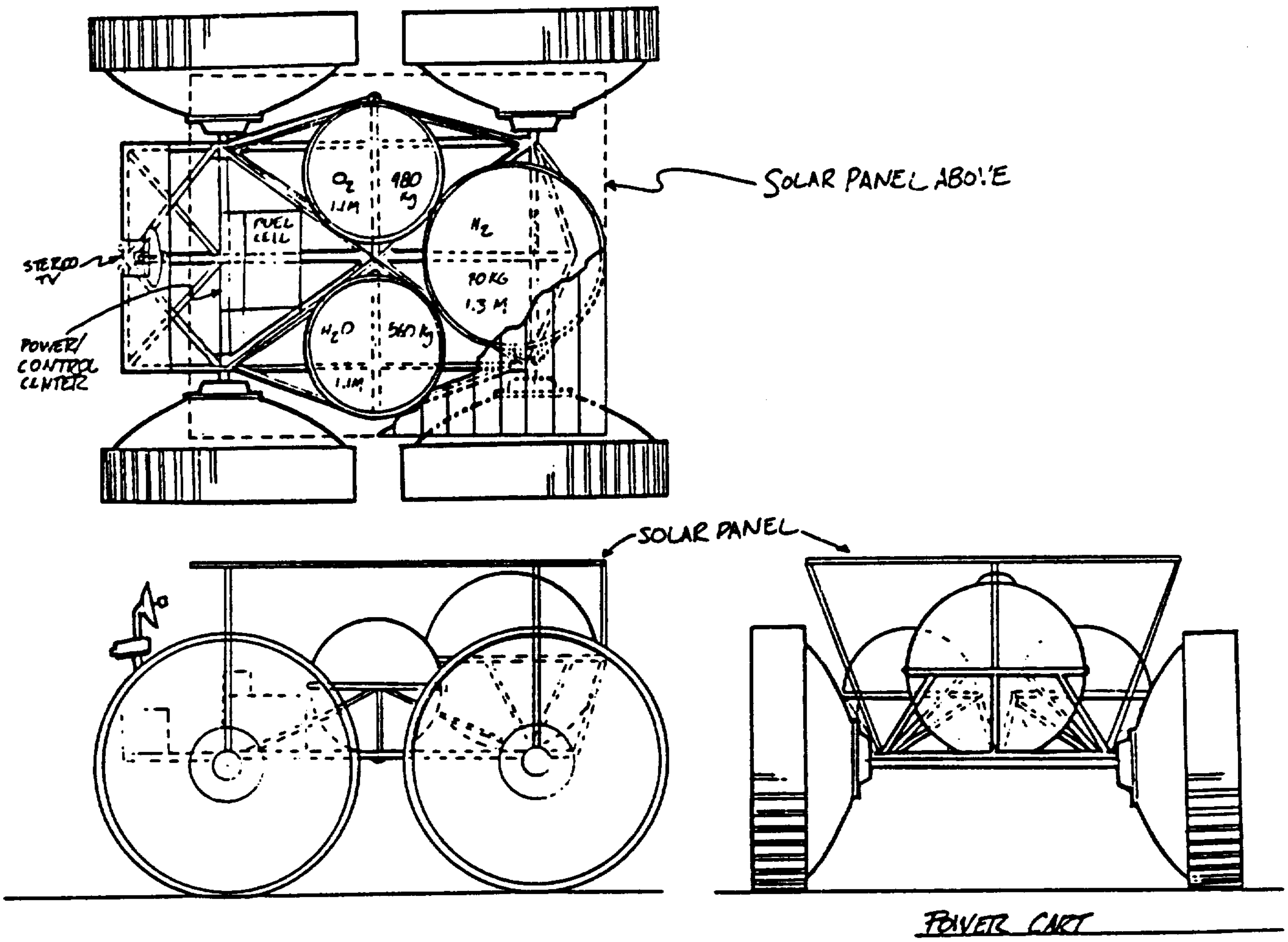

Table 5.2.3.3-1 Auxiliary Power Cart Weight Statement

Cart with 14 kilowatts, 1500 kilowatt-hours Capacity

Tanks	
Hydrogen	190 kg
Oxygen	130 kg
Water	130 kg
Fuel Cell	180 kg
Solar Panel (1 kw)	40 kg
Cart	150 kg
DRY MASS	730 kg
Reactants	560 kg
TOTAL	1,380 kg
Tanks	
Hydrogen	1.3 m Diameter
Oxygen	1.1 m Diameter
Water	1.1 m Diameter

Cart with 25 kilowatts, 7000 kilowatt-hours Capacity

Tanks	
Hydrogen	670 kg
Oxygen	570 kg
Water	570 kg
Fuel Cell	360 kg
Solar Panel (4 kw)	160 kg
Cart	300 kg
DRY MASS	2,630 kg
Reactants	2,520 kg
TOTAL	5,150 kg
Tanks	
Hydrogen	2.3 m Diameter
Oxygen	1.7 m Diameter
Water	1.7 m Diameter

Figure 5.2.3.4-1 Layout Drawing for the Experiment and Sample Trailer

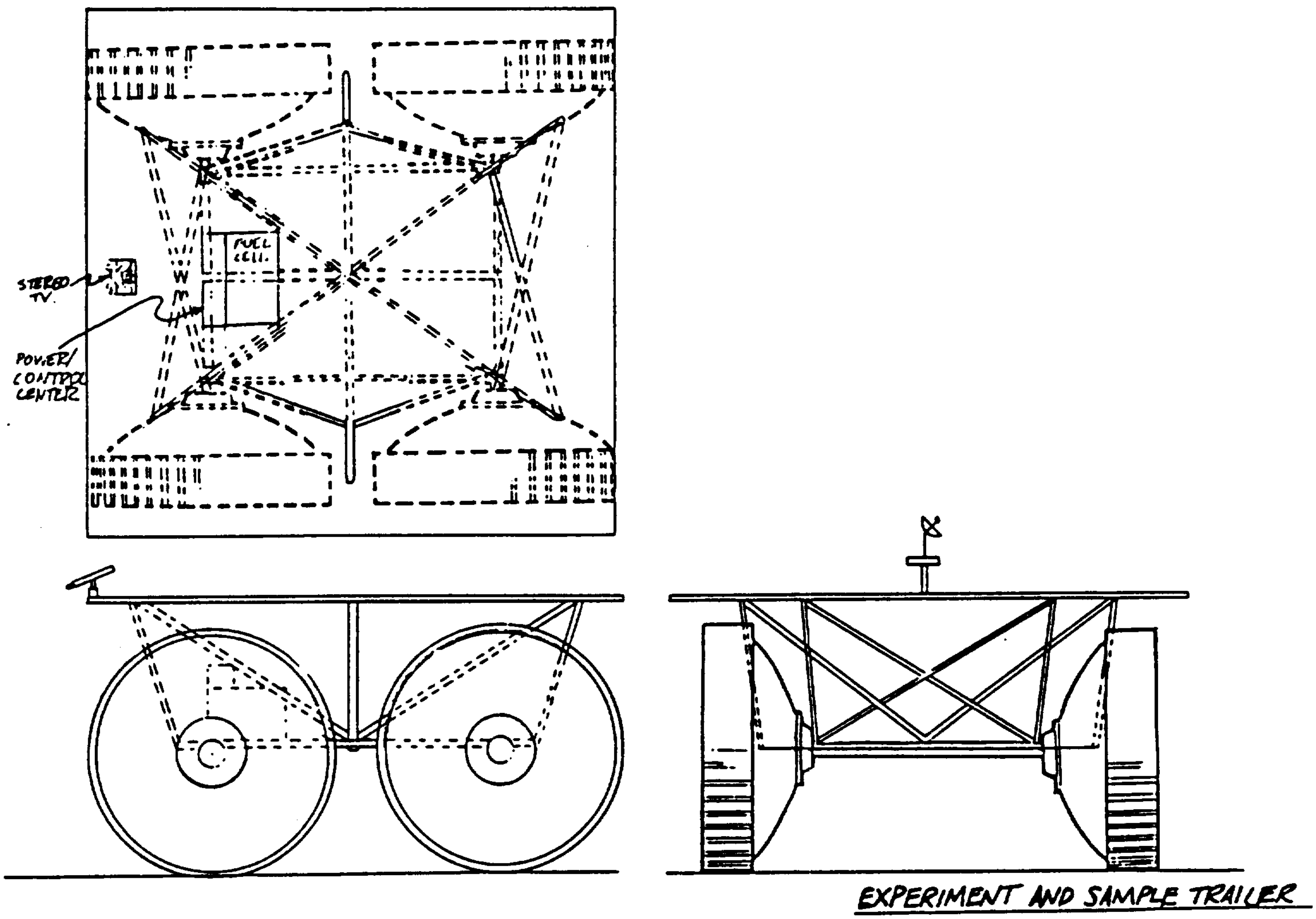

Table 5.2.3.4-1 Experiment and Sample Trailer Weight Statement

Bed	130 kg	
RMS	40 kg	
Fuel Cell	40 kg	
Tanks		
Hydrogen	5 kg	0.5 m OD
Oxygen	5 kg	0.5 m OD
Water	5 kg	0.5 m OD
Reactants	15 kg	
Cart	150 kg	
TOTAL	390 kg	
Payload	2,000 kg	
TOTAL with PAYLOAD	2,390 kg	

Table 5.2.4.1-1 Power Requirements

SUBSYSTEM	PCRV Watts	HTU Watts
Operation and Drive Station		
Computers	200	100
Communications	140	70
GN&C	200	100
ECLSS		
Air Revitalization	10	10
Water Coolant Loop Pump (1)	60	60
Air Pressure Control	10	10
Active Thermal Control		
Cold Plate Pump (1)	60	60
Circulation Fans (10)	140	140
Freon Coolant Loop Pumps (2)	700	700
Hygiene Station		
H_2O Pump	50	50
Commode	50	50
Galley		
Oven, Microwave (700 W @ Peak)	700	700
Refrigerator	400	400
H_2O Heater	70	400
Lighting		
Interior 15,16.5 Fluorescent	250	250
Exterior	1,000	1,000
Experiments	1,000	1,000
Drive Motors	6,000	6,000
Miscellaneous	1,000	1,000
SUM (No duty cycles)	12,040	12,100

5.3 Ballistic Transportation Vehicle (BALTRAN)

5.3.1 Design Requirements

The rationale for the BALTRAN is to satisfy requirements for point-to-point travel over distances greater than 100 km. In these cases, all interest is in the destination point and there is essentially no interest in the surface along the travel path. The specific transportation vehicle functional requirements for the BALTRAN are listed in Table 2.2.4-1. One great advantage in ballistic travel over the lunar surface is the extremely short travel time compared to surface contact vehicles. A review of Table 5.3.1-1 confirms these short travel times; one hour to the opposite side of the Moon in the BALTRAN versus approximately 540 hours of continuous driving in a surface vehicle.

As conceptual planning of the BALTRAN started, it was quickly realized that the velocity change design requirements for the BALTRAN would be approximately twice that of a reuseable lunar lander. The delta velocity required to initiate ballistic flight to the opposite side of the Moon (1.68 km/sec) is approximately the same as to depart for low lunar orbit. Then the BALTRAN would have to remove the velocity at the destination and repeat the whole cycle to return to the original point. The BALTRAN is required to carry propellant for the four major velocity changes since the remote destination is not a transportation node. The reuseable lunar lander is only required to start and stop one time before refueling.

Basic calculations were performed to derive the approximate propellant mass and vehicle inert mass for a BALTRAN designed to travel varying distances. The BALTRAN vehicle is based on a vehicle designed in reference 64. These results are provided in Table 5.3.1-2 for a BALTRAN based on the inert mass of the Multi-purpose Lander described in Table 8-2 of reference 64. The Multi-purpose Lander has been designed for a combination of the most demanding requirements for the intended missions. Therefore, although it is a realistic or practical operational vehicle design approach, the vehicle has excess capability/mass in some applications. The gross or net, loaded mass for the Multi-purpose Lander is 48.2 mt. Referring to Table 5.3.1-2, the BALTRAN gross mass would be 101.8 mt. to reach the opposite side of one moon. Even if a design based on less inert mass is considered, the BALTRAN would still have a mass 77.5 mt. See Table 5.3.1-3 which is based on the lightest of the Dedicated Landers from Table 8-3 of reference 64.

5.3.2 Lunar Surface Ballistic Transportation Requirements Implementation

Based on the above observations, the BALTRAN as a new dedicated vehicle appears to be impractical. The reuseable lunar lander is a mandatory vehicle in the lunar base plans and will be the transportation link between lunar orbit and the lunar surface. Although transportation from point-to-point on the lunar surface is important, it is not reasonable to consider developing a new, dedicated vehicle that performs ballistic flight and landings like the lunar lander but is more than twice as large. This is particularly true since distant points can be visited with the original lunar lander from orbit. It is not recommended that the BALTRAN be developed. No further BALTRAN conceptual design is provided.

It is suggested that the reuseable lunar lander, as found in the lunar base inventory of standard vehicles, perform the lunar surface ballistic transportation. This point-to-point lunar surface transportation capability should be added as one of the standard lunar lander design reference missions. Using the Multi-purpose Lander and referring to Table 5.3.1-2, a ballistic roundtrip mission could be flown to a lunar surface point approximately 950 km from the base. A review of Table 5.3.1-3 indicates that the lightweight Dedicated Lander could support a roundtrip to a point 732 km from the lunar base.

There are other possibilities for use of the lunar lander in transportation between various points on the lunar surface. A conceptual evaluation and comparison of these lunar lander applications in lunar surface transportation are included in section 6.0, Concluding Comments.

Table 5.3.1-1 Lunar Ballistic Flight Parameters

Approximate Distance
to Opposite Side of Moon = 5400 km
Multi-purpose Lander Delta V = 4.38 km/sec*
Dedicated Lander Delta V = 3.95 km/sec**

TRAVEL DISTANCE (km)	TOTAL++ DELTA V (km/sec)	LUNAR+ ALTITUDE (km)	ONE-WAY TIME (minutes)
50	1.131	12	4
100	1.589	25	6
200	2.216	49	9
300	2.677	72	11
400	3.049	94	13
500	3.364	116	14
600	3.638	136	16
700	3.879	156	17
732	3.951	162	18
800	4.095	175	19
900	4.290	194	20
950	4.38	202	21
1,000	4.467	211	22
1,100	4.629	227	23
1,200	4.778	243	24
1,500	5.160	284	28
2,000	5.640	333	34
2,500	5.988	357	39
3,000	6.243	356	44
3,500	6.430	330	48
4,000	6.564	279	50
4,500	6.653	204	53
5,000	6.704	107	54
5,400	6.718	15	54

* Descent, ascent, and 15° plane change from/to 93 km circular orbit.
** Descent and ascent from/to a 93 km circular orbit.
+ Maximum altitude in trajectory.
++ Four burns, two ascent, two landing all equal in magnitude.

Table 5.3.1-2 BALTRAN Mass Parameters Based on Multi-purpose Lander Design

Lunar Lander Delta V = 4.38 km/sec
Crew Module (Payload) Mass = 6.0 mt
Systems Inert Mass = 9.823 mt
Reserved Propellant Mass = 0.07 * PROP. MASS
RCS Propellant Mass = (1.073/4.38) * Delta V
GROSS MASS is Total of Above Numbers

TRAVEL DISTANCE (km)	TOTAL DELTA V (km/sec)	PROP. MASS (mt)	INERT MASS (mt)	GROSS MASS (mt)
50	1.131	4.8	16.3	21.1
100	1.589	7.2	16.5	23.7
200	2.216	11.0	16.9	27.9
300	2.677	14.3	17.2	31.5
400	3.049	17.3	17.4	34.7
500	3.364	20.2	17.6	37.8
600	3.638	22.9	17.9	40.8
700	3.879	25.5	18.1	43.6
732	3.951			
800	4.095	28.0	18.3	46.3
900	4.290	30.4	18.5	48.9
950	4.38	31.5	18.6	50.1
1,000	4.467	32.7	18.7	51.4
1,100	4.629	34.9	18.8	53.7
1,200	4.778	37.1	19.0	56.1
1,500	5.160	43.2	19.5	62.7
2,000	5.640	52.2	20.2	72.4
2,500	5.988	59.8	20.7	80.5
3,000	6.243	66.1	21.2	87.3
3,500	6.430	71.1	21.6	92.7
4,000	6.564	74.9	21.9	96.8
4,500	6.653	77.6	22.1	99.7
5,000	6.704	79.2	22.2	101.4
5,400	6.718	79.6	22.2	101.8

Table 5.3.1-3 BALTRAN Mass Parameters Based on Dedicated Lander Design

Lunar Lander Delta V = 3.95 km/sec
Crew Module (Payload) Mass = 6.0 mt
Systems Inert Mass = 5.802 mt
Reserved Propellant Mass = 0.07 * PROP. MASS
RCS Propellant Mass = (1.013/3.95) * Delta V
GROSS MASS is Total of Above Numbers

TRAVEL DISTANCE (km)	TOTAL DELTA V (km/sec)	PROP. MASS (mt)	INERT MASS (mt)	GROSS MASS (mt)
50	1.131	3.6	12.2	15.8
100	1.589	5.4	12.4	17.8
200	2.216	8.3	12.7	21.0
300	2.677	10.8	12.9	23.7
400	3.049	13.0	13.1	26.1
500	3.364	15.2	13.3	28.5
600	3.638	17.3	13.5	30.8
700	3.879	19.2	13.6	32.8
732	3.951	19.8	13.7	33.5
800	4.095	21.1	13.8	34.9
900	4.290	22.9	14.0	36.9
950	4.38			
1,000	4.467	24.7	14.1	38.8
1,100	4.629			
1,200	4.778			
1,500	5.160	32.8	14.8	47.6
2,000	5.640			
2,500	5.988	45.4	15.7	61.1
3,000	6.243			
3,500	6.430			
4,000	6.564			
4,500	6.653	59.0	16.8	75.8
5,000	6.704			
5,400	6.718	60.6	16.9	77.5

6.0 Concluding Comments

Conceptual designs for three categories of lunar surface transportation have been described. The level of understanding for the capabilities and design approach varies between the vehicles representing these categories. A summary of the vehicle categories and current state of conceptual design is provided in the following section. Finally, a brief evaluation and discussion is provided for a systematic comparison of transportation categories and effectiveness in supporting specified transportation objectives.

6.1 Summary of Vehicle Categories

A vehicle of the LOTRAN class is certain to be required for lunar base operations. The extended range provided to an EVA astronaut is indispensible. The LOTRAN is conceived to be a workhorse which is rugged, simple, and adaptable. There is certainly more detail design required for the LOTRAN, but no major questions about the mission purpose or capabilities.

A MOSAP is also a necessary transportation element if remote terrain away from the lunar base is to be explored and developed. One-way travel time from the lunar base of more than a few Earth-days is a radical increment over the LOTRAN in vehicle design impact and uncertainty in an appropriate maximum design range. Some planners have suggested that the MOSAP function is comparable to one of the recreational camping motor vehicles on Earth today. The MOSAP is not like any camping vehicle in past experience. There is no refueling along the travel route, no restocking of supplies, no hitchhiking back to base if vehicles fail, and the logistics difficulties increase directly with destination distance. There is no question a mobile surface applications traverse vehicle is needed to enable crew stays on the lunar terrain away from the base; the question is how far from the base can such a vehicle be effective. More study is required to develop the appropriate range for the MOSAP in view of the long trip times, the increasing mass required as the range increases, and alternative modes for exploring remote regions of the Moon.

Ballistic flight from one lunar surface point to another is much quicker than surface vehicles moving across the terrain. However, a much higher traveling speed is involved and the result of errors can be more catastrophic. Ballistic flights of less range than

50 km probably are not warranted. The growth in propellant mass required with increasing range limits the practical ballistic range to about 950 km. Ballistic flight is extremely efficient in transportation time which can save much astronaut time resources, a very valuable asset. The major disadvantage of the ballistic flight transportation approach is that the terrain between the base and final destination cannot be sampled and surveyed in detail. Another disadvantage is that large quantities of propellant (10-20 mt) must be provided on the surface for each mission. The ballistic flight vehicle should be the lunar base reuseable lunar lander. The conceptual design of this vehicle is well understood.

6.2 Transportation Effectiveness Comparison

Table 6.2.-1 includes all the vehicles discussed in this report. The last column in Table 6.2-1 shows the propellants and consumables delivered to low lunar orbit per mission for each vehicle. With no lunar produced propellants, the BALTRAN requires almost unreasonable amounts of propellant to be delivered. Based on this comparison, every effort should be made to extend the range of the MOSAP, and the multi-purpose lander, descending from orbit, should be used for longer sorties.

Downsizing of the crew capsule of the ballistic transportation vehicle from 6 to 2 metric tons (an Apollo lunar module type capsule) will increase its range, perhaps beyond that of the MOSAP, but the doubling of delta V will always make it inefficient when compared on an equal basis with descent from orbit.

It may be reasonable to pay the penalty for keeping a ballistic flight transportation system on the surface for rescue purposes, but this first look indicates it will not be competitive with the MOSAP for regular transport until lunar produced propellant is available.

Table 6.2-1 Vehicle Transportation Effectiveness Comparison

Vehicle	Range From Base, (km)	Mass at Start of Mission, Loaded (kgm)	Payload (kgm)	Landed Consumables Req-Per Mission (kgm)	Propellants and Consumables Delivered to LLO Per Mission (kgm)
LOTRAN	0-50	1,400	850	~0	-
MOSAP	10-1,500	17,600	3,400	+300-3,000	-
*BALTRAN (no lunar surf. prop.)	50-950	48,000	1,000	30,000+	~60,000+
*BALTRAN (with lunar O_2)	50-950	48,000	1,000	4,300	~14,300
*BALTRAN (with lunar O_2 & H_2)	50-950	48,000	1,000	~0	-
*Multi-purp. (orbit based -All prop. from Earth)	50-5,400	48,000	1,000	-	33,000

*Depends on base regeneration capability

*Same basic vehicle assumed

**Assumes LLO based lander and includes propellants needed to land consumables.