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FINAL REPORT:

LUNAR LANDER GROUND SUPPORT SYSTEM

by

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Abstract

This paper involves the design of the Lunar Lander Ground Support System (LLGSS). The basic design time line is around 2010-2030 and is referred to as a second generation system, as lunar bases and equipment would have been present.

Present plans for lunar colonization call for a phased return of personnel and materials to the moon's surface. During the first phase, a basic base will be set up with power supplies, basic supplies, and various necessary equipment. The original base personnel would then land at the base(s) and stay for a period of roughly 30 days and return to earth. Shortly after that, a second group would land at the base(s) and stay for periods of 60 to 180 days and would perform long term experiments in an effort to prepare for future explorations.

During these times, the lunar lander is stationary in a very hostile environment and would have to be in a state of readiness for use in case of an emergency. Cargo and personnel would have to be removed from the lander and transported to a safe environment at the lunar base. An integrated system is required to perform these functions.

This project addresses these needs centering around the design of a lunar lander servicing system. The servicing system could perform several servicing functions to the lander in addition to cargo servicing. The following were considered: 1) reliquify hydrogen boil-off, 2) supply power, and 3) remove or add heat as necessary.

The final design incorporates both original designs and existing vehicles and equipment on the surface of the moon at the time considered. The importance of commonality is foremost in the design of any lunar machinery, due to the extreme expense of placing such machinery on the surface of the moon.

The FAMU/FSU Senior Aerospace Design was funded through USRA (University Space Research Association) as a two semester research project involving ten Senior Aerospace engineering students. The group consisted of seven members of the mechanical discipline and three of the electrical discipline.

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1. Introduction

1.1 Project Background Description

This year's project, like the previous Aerospace Groups's project, involves a lunar transportation system. The basic time line will be in the neighborhood of the years 2010-2030 and will be referred to as a second generation system because lunar bases will be present. The project design completed this year is referred to as the Lunar Lander Ground Support System (LLGSS). Not many projects have been attempted in this area due to the fact that it involves the design of a system that is not nearly as glamorous as it is necessary.

Present plans for lunar colonization call for a phased return of personnel and materials to the moon's surface [1]. During the first phase, a base will be set up with power supplies, basic supplies, and various necessary equipment. The first base personnel would land at the base(s) and stay for a period of roughly 30 days and return to Earth. Shortly after that, a second group would land at the base(s) and stay for periods of 60 to 180 days and would perform long term experiments in an effort to prepare for further explorations in the future. At these times, the lunar lander will be stationary in a very hostile environment and will have to be in a state of readiness for use in a contingency plan in the case of an emergency. Cargo and personnel will have to be removed from the lander and transported to a safe environment at the lunar base. This project addresses these systems and the problems encountered.

The interaction of the following three types of vehicles will have to be analyzed:

- A reusable lunar lander
- A servicing vehicle
- A transportation vehicle

The basic operational scenario is as follows: A lunar lander descends from a transportation node in LLO (Low Lunar Orbit) to the surface and lands at a prepared area close to the base. The transportation node will be a stopover point for the lander delivering vehicle traveling from Earth to the moon. A transportation vehicle will then bring out a servicing vehicle to be attached through umbilicals to the lander. The cargo or personnel would be removed and transported the distance from the landing pad to the lunar base. Some of the transferral operations will be performed through the use of remotely operated cranes or robots, referred to as teleoperations. Once the personnel or cargo items have returned to the base, a "servicer" vehicle will keep the lander in a state of readiness.

The lander will be of a similar design to lunar landers of the Apollo era. It has been stated in various reports that a reusable lander that burns liquid hydrogen and liquid oxygen would be necessary for such missions. The lander will be able to perform standard docking functions with an orbital lunar node.

The servicer will serve several servicing functions to the lander, the following are suggested: 1) reliquify hydrogen and oxygen boil-off, 2) supply power, and 3) remove or add heat as necessary. A drive system would be unnecessary for a "vehicle" that would be immobile a majority of the time.

The transport vehicle will be made to operate manually or through the use of teleoperations and robotics. It will serve the dual purpose of carrying the servicer out to the landing pad and transporting cargo or personnel back to the base. The basic configuration will be similar to that of a lunar roving type vehicle.

A great deal of practical engineering was applied to the various systems and interactions of the project. Several NASA and contractor personnel showed interest in particular areas of our proposed project. None of the personnel contacted had seen a detailed design and analysis of a project involving many systems interacting between three vehicles such as ours. Specific areas of interest of industry personnel included: 1) vehicle interactions; 2) vehicle interfaces and associated procedures; 3) heat rejection while on surface; 4) fuel storage and reliquification procedures; 5) radiation protection; and 6) dust problems in all systems. Practical knowledge of materials selection, heat transfer, electrical engineering, and structure design can be applied to most aspects of the project.

1.2 Emphasis of Design

The area chosen for analysis encompasses a great number of vehicles and personnel. The design of certain elements of the overall lunar mission is a complete project in itself. The fundamental design of bases, lunar landers, heavy moving vehicles, lunar nodes, and earth to moon transportation systems are extensive projects in their own right. It is for this reason that the project chosen for the Senior Aerospace Design is the design of specific servicing vehicles and additions or modifications to existing vehicles for the area of concern involving servicing and maintenance of the Lunar Lander while on the surface.

The design of certain vehicles and structures not directly related to but interfacing with the servicing system was assumed. Examples of the vehicles and structures that were considered as outside design parameters that the Lander servicing system depended on are as follows: Lunar Lander, Lunar Orbiting Node, Space Transportation System, central lunar base systems, and lunar heavy moving vehicles.

The most plausible design for a lunar lander for the years 2010-2030 was found in a conceptual lunar lander report by Eagle Engineering [2]. The report describes a second generation single stage lunar lander and the related physical parameters. The lander is taken as a reusable vehicle powered by liquid hydrogen and liquid oxygen chemical rockets. The basic design is shown in Fig. 1.1 . All other significant parameters are assumed as given from this and similar reports. A large number of reports on base designs exist. The general layout varies a great deal from one report to the next, but the content and ideas are the same. The lunar bases previously designed exist across a broad spectrum ranging from exposed rectangular structures to completely buried cylindrical structures. A typical design used for the Aero group's servicing system project is one found a second Eagle report [3]. The basic structure is one of a series of cylindrical pressurized structures attached together (see Fig. 1.2) and partially buried under regolith.

The current examples of Space Transportation Systems (STS's), or Space Shuttles, were extrapolated or used "as is" for design parameters to consider during the synthesis of a lunar

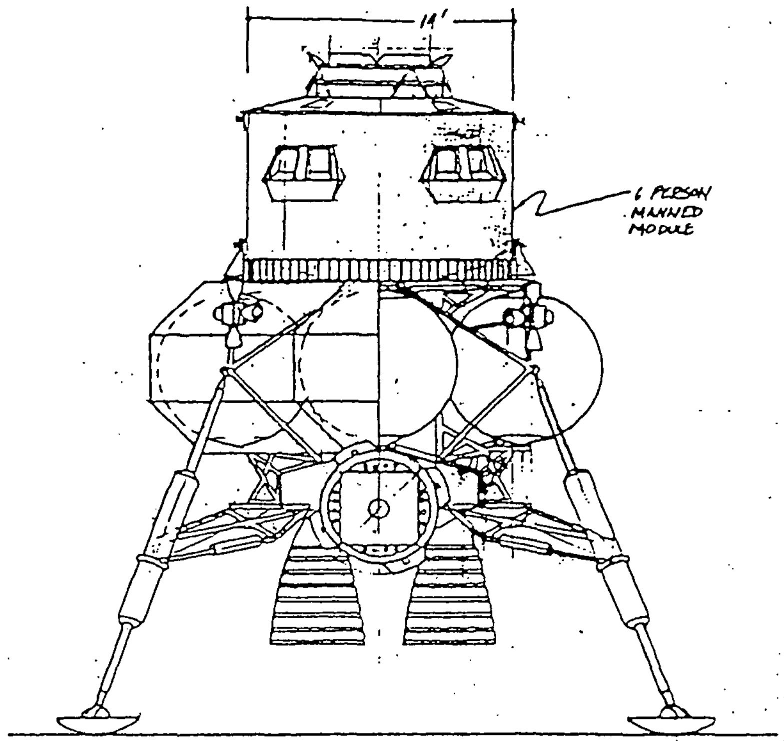

Figure 1.1. Basic Configuration of a Lunar Landing Vehicle.

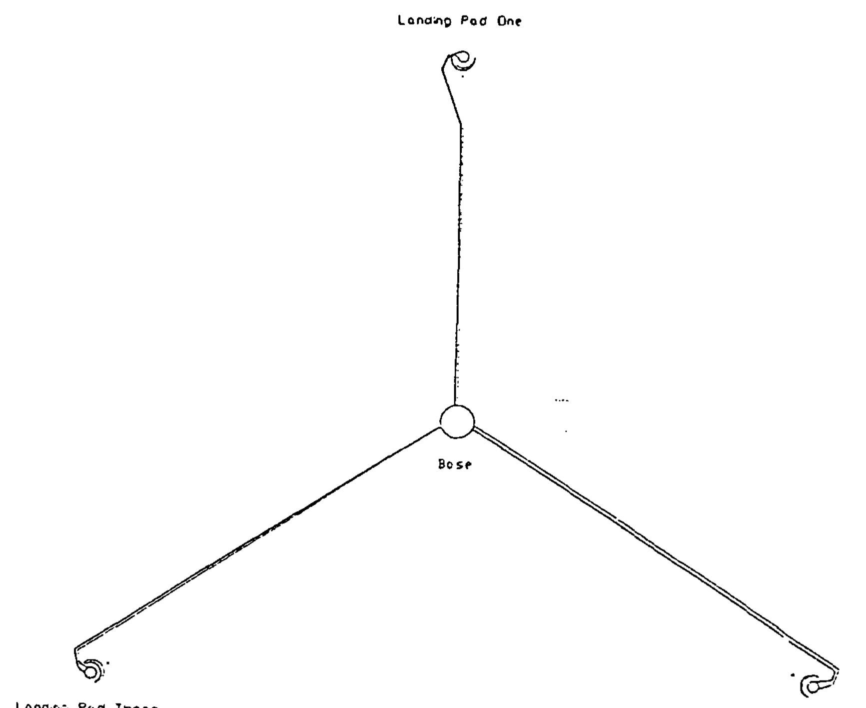

Figure 1.2. Configuration of a Lunar Base.

lander servicing system. The particular STS configuration considered dictates the aspects of any lunar surface design through limitations on geometry, mass, and safety issues.

Many examples of predicted designs for second generation lunar surface vehicles exist. The basic operations are modified for operations pertaining to lunar lander servicing. The designed vehicles in this project take advantage of existing vehicles when possible and involve heavy modification where necessary. A primary concern in the design of any lunar system is the commonality of both components and complete vehicles when necessary.

1.2.1 Design Project Components

The Senior Aero Group project concentrates on the specific design of the servicing vehicle, various transport options for travel from the lunar base to the lunar lander, and the landing sites. The following vehicle and component designs are discussed in this report:

- Servicer
- Heavy Mover (Tractor)
- Large Cargo/Servicer Trailer (including teleoperated crane)
- Small Servicer Trailer
- Interface connections
- Various modifications to existing equipment

1.2.1.1 Landing Sites

The landing sites are designated to be at a distance of 1000 m from the main central base structures - *a restriction based on the blast damage caused by descending Lunar Landers* [4]. The lunar base has three separate landing pads that can have operations going on in any order. The landing sites are triangular with the main base structures in the center (see Fig. 1.3). Graded regolith walls protect the lunar base and other landing pads from debris ejected upon landing. Graded travelways also connect the landing sites with the central base. A system of both electronic and visual landing aids would be present for aid in navigation to landing site out of orbit.

1.2.1.2 Servicer

The Servicer (see Fig. 1.4) is a stationary support vehicle placed next to the lunar lander while on the surface to provide basic functions of hydrogen reliquification, power supply, status communication, daytime cooling, and nighttime heating. The structure is one of a frame enclosing fuel cells, electronics, and spherical tanks. The exterior is covered with a rigid stand-off shield/thermal tile protection system. The upper surface consists of a fluid loop radiator for heat rejection of both fuel cell waste heat and waste heat from the lander during the day. Strip antennas are used for all communication with the base.

The loss of hydrogen fuel through boil-off due to heat conduction is generally not a large concern for short surface stay times. The boil-off becomes a real concern when the stay times are extended to 180 days, as is expected for third phase lunar inhabitation. Simple

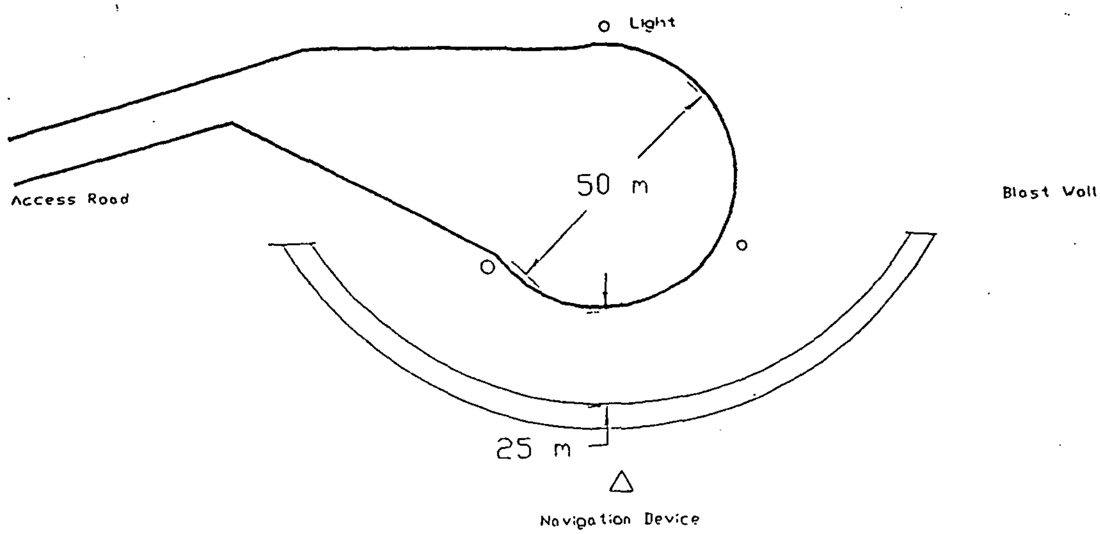

Figure 1.3. Lunar Base Landing Site Layout.

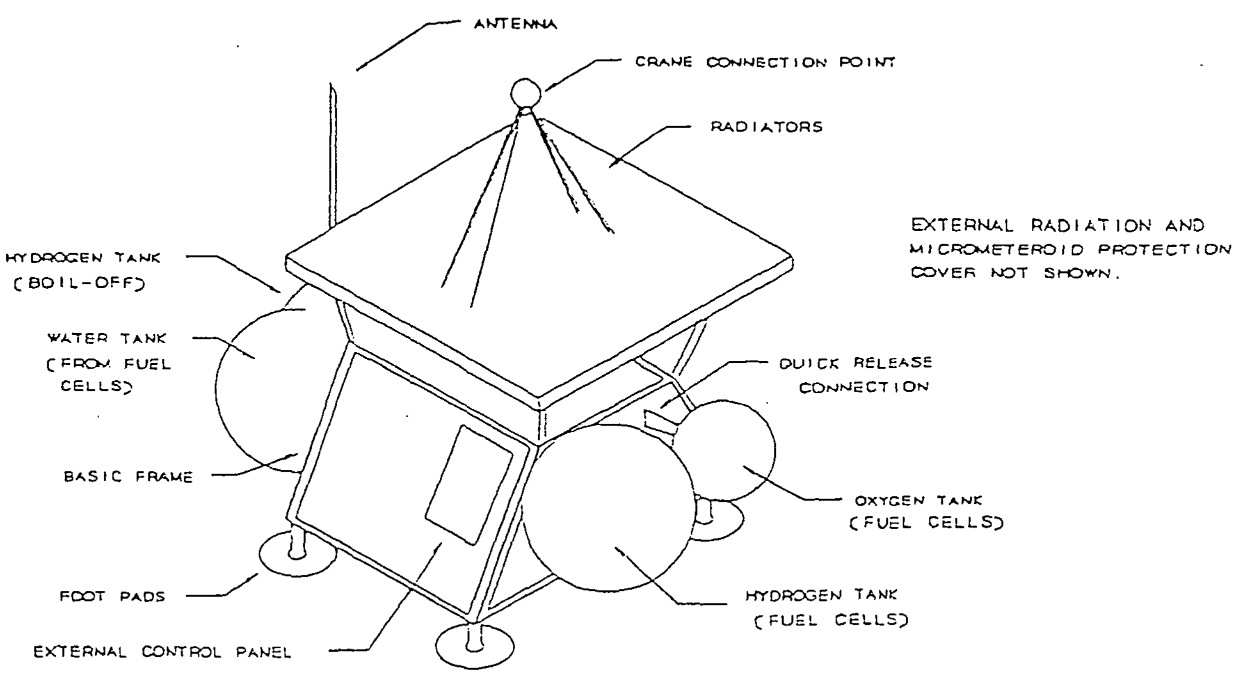

Figure 1.4. Servicing Vehicle for Lunar Lander.

calculations indicate that up to 50% of a lunar lander's liquid hydrogen can be lost to boil-off during a 180 day stay[5]. This necessitates a system for storing H2 boil-off and reliquifing it to pump back into the lander's tanks. The basic idea is to store the gaseous hydrogen that boils off during the hot lunar day in a large tank on the servicer. At nightfall the temperatures would drop to near absolute zero, which would allow an extremely efficient helium cycle to be used to reliquify the hydrogen. The reliquified hydrogen would then be pumped back into the landers tanks. Oxygen production is assumed during this time, making oxygen boil-off irrelevant due to the availability of O_2 on the surface [6]. Oxygen also is less susceptible to boil-off than hydrogen due to its higher boiling point and lower heat conduction coefficient.

The power system for the lander would be designed for relatively short flight times, typically on the order of one hour. The additional system weight would be unwarranted for the 180 day stay times on the lunar surface. The servicer provides the additional power while the lander is on the surface, which allows the lander power supply to be reserved for flight time only. The electronics and cooling system components would need power during the 180 day stay, along with controls testing to avoid seizing of components. The servicer fuel cell power supply would support the lander during the surface stay.

Communications would be provided through the servicer to the base concerning the status of both the servicer and the lander. This data would include all electrical and thermal aspects of both vehicles.

The additional daytime lander heat input due to reflection from the lunar surface would require active heat rejection from the lander. The servicer would provide relief from this additional lander heat load by removing it through fluid loops to the servicer.

During the 14 day lunar night, the heat problem is completely reversed. The electronics and fluid loops of the lander must be kept in a state of readiness through heat input, due to the fact that the lander would cool down to the temperature of the lunar night otherwise. The electronics and fluid loops have to be kept warm in order to function at all, much less properly. The heat input from the servicer to the lander is through both electrical and fluid means.

A stand off shield is necessary as a result of the increased probability of a meteorite impact due to the 180 day stay time. Multi-layer thermal tiles attached to the stand-off shields also provide protection from the intense solar radiation impinging on the servicer during the lunar day.

1.2.1.3 Transport Options

The lander can deliver either cargo or personnel. When in cargo mode, the lander is assumed to be unmanned and remotely piloted or computer guided. When in personnel delivery mode, the lander is expected to be piloted by a crew member present. Each of these two cases dictate different transport techniques upon landing. The simplified net result is that the lander is brought out to the landing site and cargo or personnel are retrieved to the base. Potentially confusing detailed descriptions of post-landing activities are described through diagrams and description in the following section. Alternative operations for some of the vehicles incorporated are also mentioned.

1.3 Landing Scenario

The scope of the Senior Aerospace Group design project can be demonstrated best through "cartoons" demonstrating the course of activities that are to take place at a lunar base after a Lander descent. As mentioned above, there are two distinct scenarios for a lunar landing: cargo or personnel. Much of the equipment used in one case may be used in the other with an emphasis on commonality and duality when possible.

Fig. 1.5 depicts the course of operations at the base after an unmanned cargo lander has descended. Step 1 shows the Prime Mover vehicle towing the Large Cargo/Servicer Trailer out to the landing pad where the cargo Lunar Lander waits. The Servicer can be seen on the trailer along with the cargo moving teleoperated crane. Step 2 shows the Servicer having been removed and the crane readying the removal of the cargo atop the Lander. In Step 3, the cargo has been placed on the trailer by the teleoperated crane. Step 4 shows the Prime Mover vehicle towing the trailer with cargo back to the lunar base. It is possible that several trips may have to be made, depending on the size and weight of the cargo.

Fig. 1.6 depicts operations after a human piloted personnel lander has descended. Step 1 shows the Prime Mover towing a pressurized Crew Trailer out to the lander. The Servicer can be seen atop a small trailer attached to the rear of the Crew Trailer. The trailer with the Servicer is detached next to the Lander, as shown in Step 2. Step 3 shows the Crew Trailer being attached to the crew hatch of the lander with personnel being evacuated. Step 4 shows the Prime Mover traveling back to the base. The Crew Trailer used in the personnel landing scenario is part of a pressurized long distance exploration vehicle that would be present during a third phase lunar inhabitation.

The Crew Trailer used in the personnel landing scenario would have its own drive system and steering, but would be controlled by the Heavy Mover as it would be controlled by the power cart while on an exploration. Modifications or attachments would have to be made to the hatch on the rear of the Crew Trailer in order to attach directly to the Lander. The Crew Trailer would already have been designed to mate with the base hatch system. This demonstrates a use of existing lunar equipment in the servicing project when possible.

In both of the landing scenarios, a crew in EVA suits will return to the landing site to finish connecting the Servicer to the Lander and to begin operations. A single or double EVA crew would return periodically to the site to perform maintenance activities and basic physical checks.

In both cases, the Prime Mover is piloted by a single EVA person. The existence of a teleoperated crane and EVA person on the same mission allows for parallel processing techniques to be used. While at the base, the crane is operated through IVA activities while the EVA person performs other activities. The scheduling mentioned in following sections details this technique.

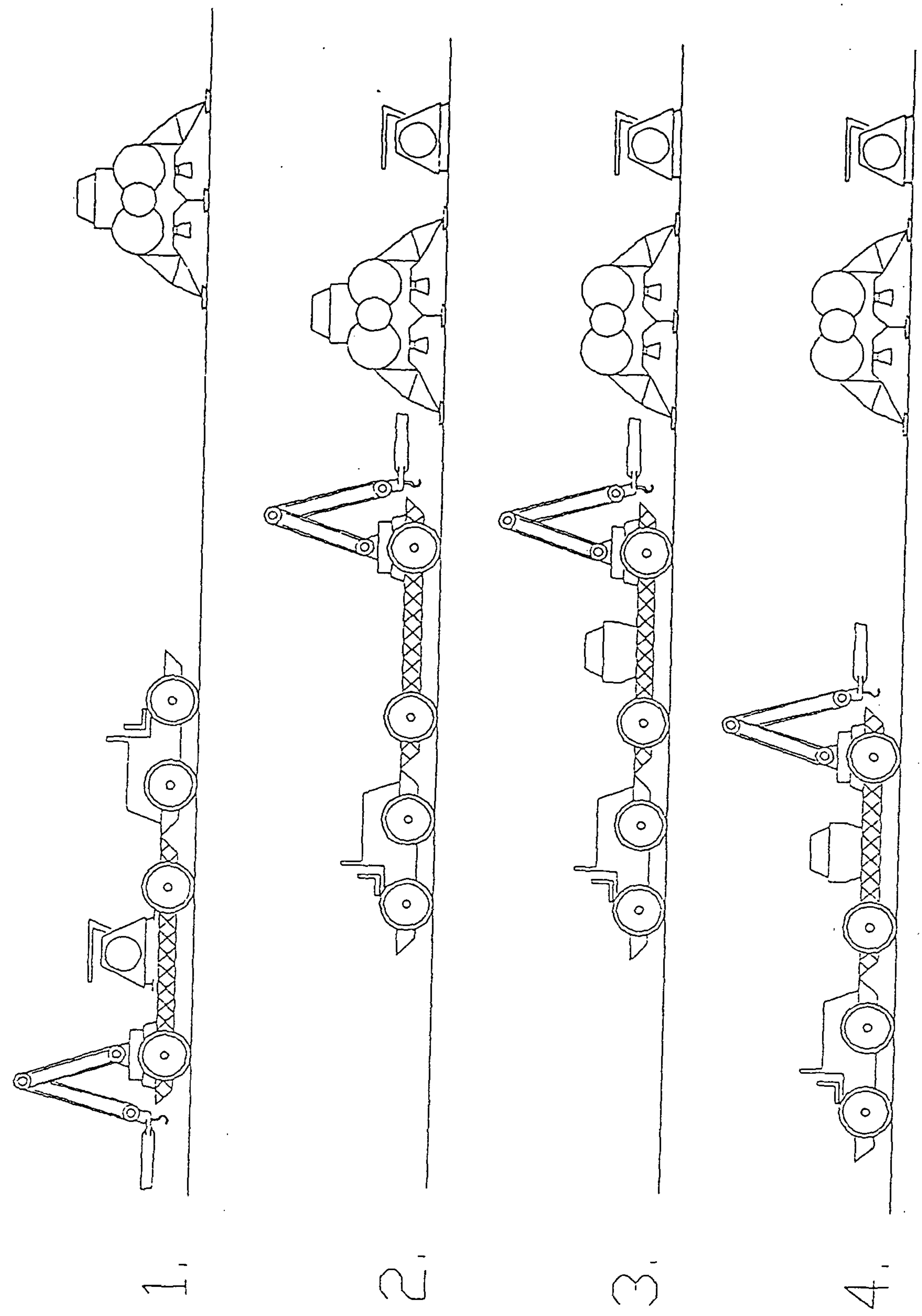

Figure 1.5. Cargo Mode Post Landing Operations.

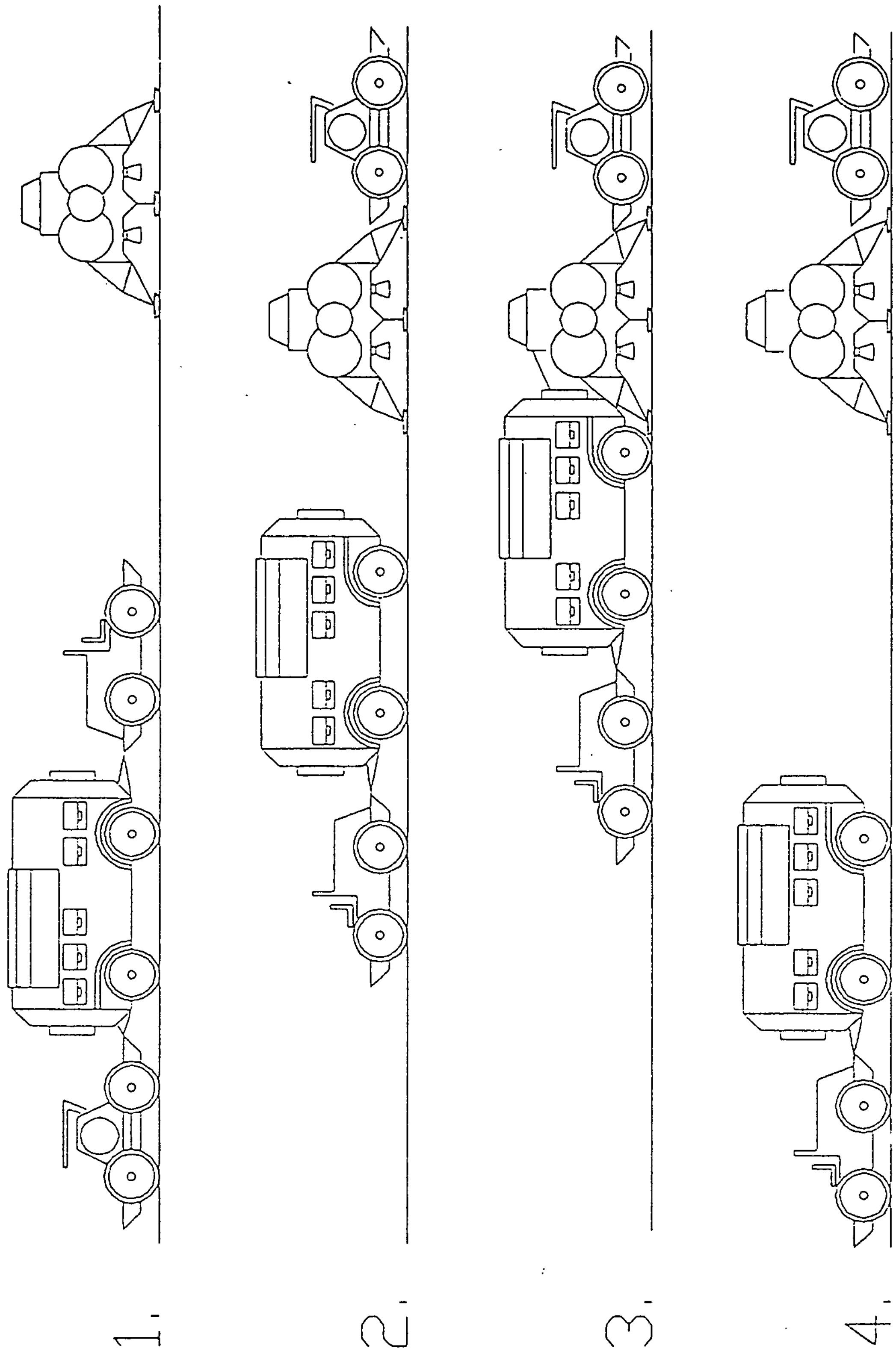

Figure 1.6. Crew Mode Post Landing Operations.

There are several alternative operations for the Prime Mover vehicle used in the servicing activities. These include lunar soil mining and moving for oxygen production plants, base site construction, landing site construction and maintenance, and roadway grading. The large Servicer/Cargo Trailer could also be used in some these activities.

1.4 Project Group Structure and Management

The Aerospace Group consisted of ten Seniors working as a design team with a project manager. The project organization chart can be found in Fig. 1.7 , which depicts the project management function and component design elements of the group structure. Ten main component areas were assigned to each of the group individuals. Five project management areas were also assigned to five of the same individuals. The five management functions were as follows:

Systems Engineer (Dan Litwhiler):

Responsible for integration of various components of project and internal project documentation. Must be aware of changes "across the board" and how they affect other systems through the entire design process. This includes interfaces (data, electrical, and physical) and compatibility concerns.

Documents Librarian (John Kynoch):

Responsible for upkeep of Aerospace library and computer catalog database. This will be the person to get assistance from on checking out and returning aerospace library documents. There will be sign out sheets for documents in order to keep library organized. Any outside information gathered (reports, articles, etc) should be given to the librarian for addition to the library.

TEX Editor (Matt Pickett):

Responsible for general knowledge of TEX commands and macro's. Individual segments of reports should be cleared through this individual for consistency in style and macro use. The editor is also responsible for the integration of individual efforts into reports due throughout the project (interim, final, etc).

Program Control (Jeanne O'Konek):

Responsible for scheduling and project software. Will update and distribute Gantt charts, PERT charts, etc pertaining to schedule control. Also responsible for accounting of aerospace group funds and the associated requests to Dr. Chandra.

Display Design (Todd LaSalle):

Responsible for display setup at conference (including graphs, posters, etc). Also responsible for any scaled models produced, whether contracted out or produced "in-house".

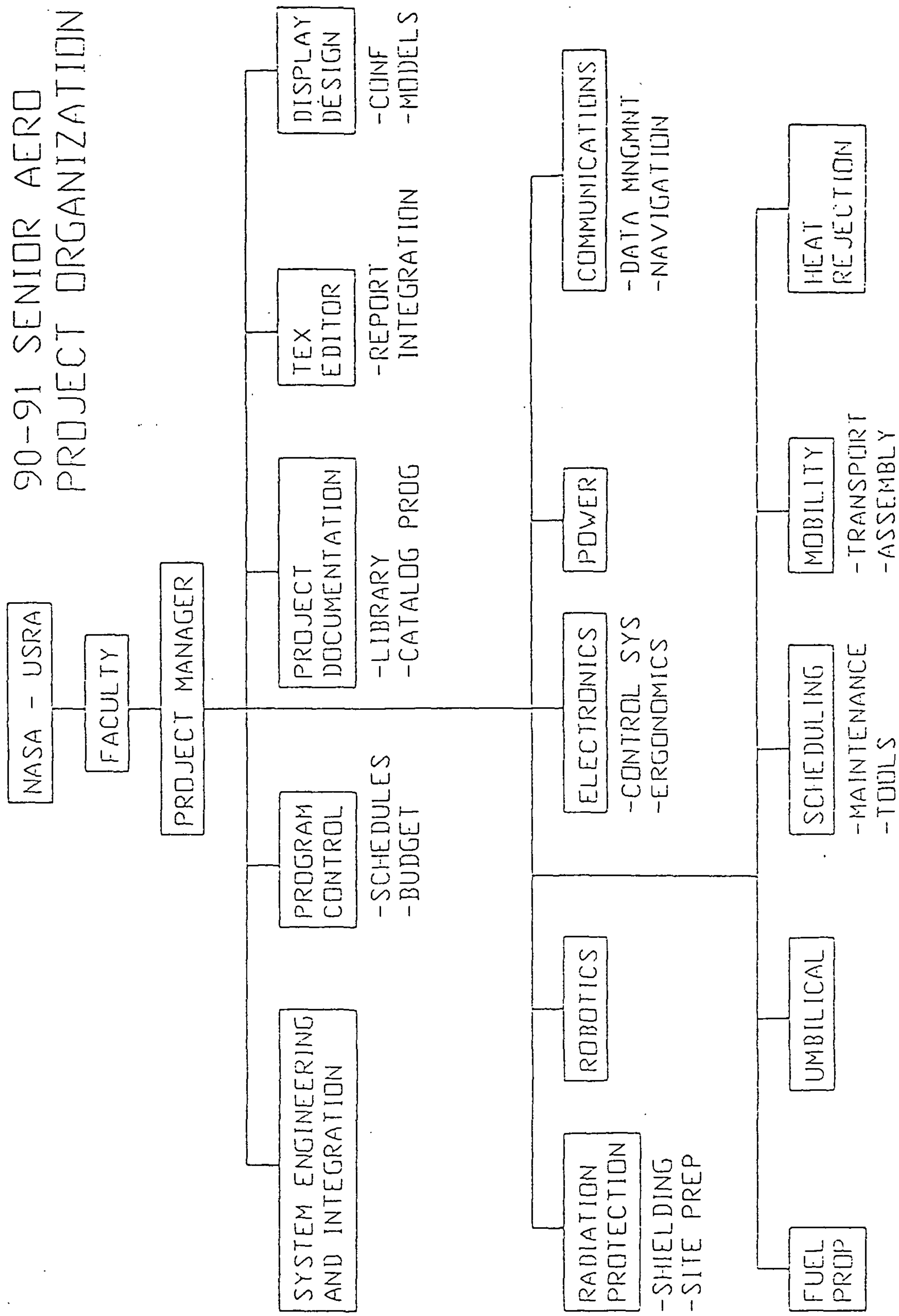

Figure 1.7. Project Organization Chart.

One of the first activities in the project initiation was to assign a work breakdown for the design project. The following is a basic work breakdown (a detailed work breakdown can be found in Appendix A):

Dan
- Radiation protection
- Shielding
- Site preparation

Robert
- Robotics

Marcus
- Electronics
- Control systems
- Ergonomics

Jeanne S.
- Power

Jeanne O.
- Communications
- Navigation

Doug
- Fuel system

Todd
- Umbilical

Chris
- Maintenance/tools

Matt
- Mobility
- Transport and assembly
- Overall structure/frames

John
- Heat rejection

A schedule was developed in the form of a Gantt chart, with both primary and secondary detailed charts done throughout the year. The primary Project Schedule can be found in Fig. 1.8 . The secondary detailed charts can be found in Appendix B.

One of the primary concerns in this project was the area of interface issues. A project involving many multi-purpose vehicles operating together on prescribed schedules necessitates careful consideration of all interactions. A seemingly subtle change in one vehicle or component during the design synthesis can create catastrophic design problems for future work of other component managers. It is for this reason that very careful attention was paid to interface issues through weekly component meetings and periodical design reviews.

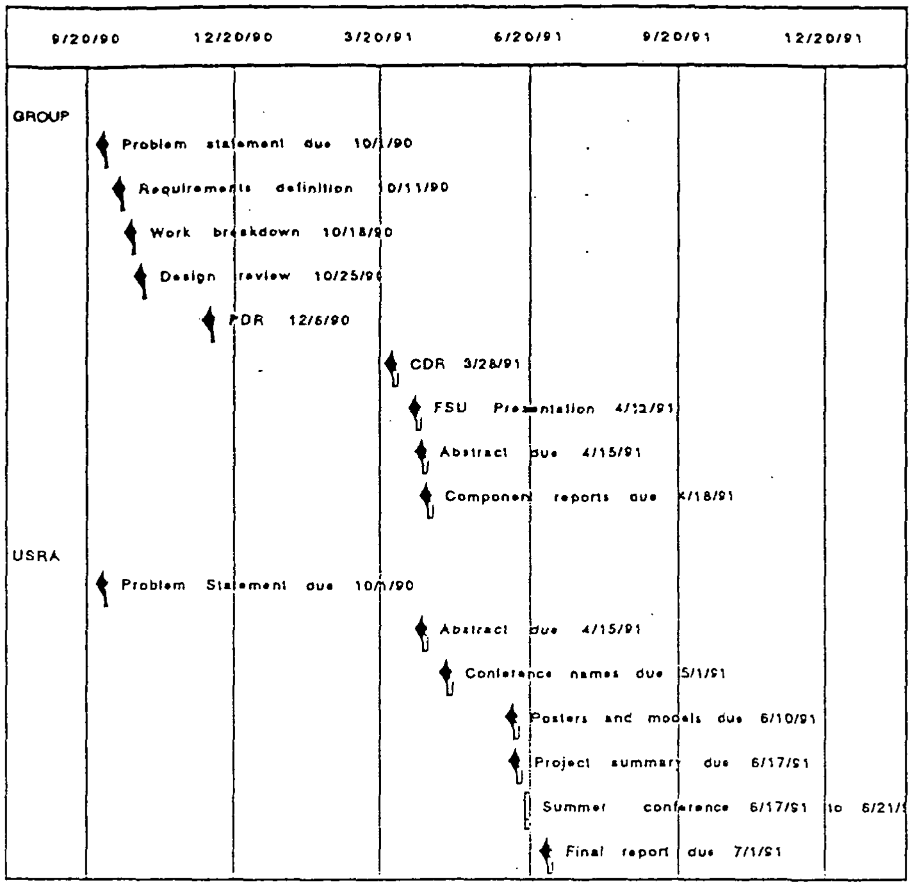

Figure 1.8. Overall Project Schedule for LLGSS.

2. Overall Mission Requirements/Considerations

Servicer

- Spacecraft power (standard) 28 V DC to be supplied, power conditioning needed.
- Weighs approximately 4000 kg.
- Suggested power supply for the servicer: 7 kW Space Shuttle fuel cell design.
- Approximate weight of power supply: 1300 kg (7kW).
- Load of above system (reliquify, cooling, heating) OF 3-8 kW (nominal to peak).
- Space power system can be assumed on line.
- Inspection and test equipment needed.
- Use teleoperations/telerobotics when possible.
- Provide lighting for area around lander during lunar night.
- Must perform hydrogen, oxygen reliquification.
- Heat rejection for lander during lunar day.
- Heating for lander during night.
- Radiation protection must be provided.
- Power to be supplied to the lander when on surface.

Transport

- Cargo work would be best done by teleoperations.
- Power to payloads during exchange?
- Data/video/fluids interface for payload/module during exchange.
- May have to transverse unprepared area.
- Vehicle may operate in manned or unmanned mode.
- Own heat rejection.
- Own power system.
- Own drive system.
- Suspension system, including wheels.
- Control system.
- Cargo/servicer cargo manipulation capacity (robotic arm with end effectors).
- Teleoperations used extensively.
- Have capacity for contingency manual operation if teleoperations fail due to solar flares, etc.

- Reach of teleoperations must be capable of exchange under all conditions.
- Teleoperated crane and transport capable of 3-5 mt servicer and 5 mt cargo-personnel module carrying, 1 mt cargo in addition to cargo baseline.
- Transport has alternative site preparation, mining, etc considerations built into it.

Reusable Lander

- 3-6 flights per year have been baselined.
- Lander must be able to abort to LLO (Low Lunar Orbit) within 6 hours at any given time.
- Lander life span is approximately 5 years.
- Lunar lander uses LOX and LH2 for propulsion.
- Lander carries six persons up and down in personnel mode.
- Radiation of heat from lander necessary, approx 2 kW additional during day due to reflected surface radiation.
- Heat exchanger in lander required to avoid system sensitivity when servicer is cooling (or heating).
- Landing gear pad circle diameter on lander: 12 to 20 meters.
- 20 mt maximum cargo when in cargo mode, unloaded in segments of 5000 kg maximum.
- 1 kW for lander electronics while on the surface.
- 7.5 m height from the surface to top of the crew module.

Crew Module

- 4500 kg total for 6 people and 3 days.
- 3 kW for crew module life support and maintenance.
- Dimensions: 2.67 m h x 4.3 m diameter.
- 1000 kg of additional cargo possible.

Landing Site

- Landing site distances from base:
  - 50 m - severe damage.
  - 400 m - minimal protection (one used).
  - 1-2 km - effects small.
- Landing site dimensions: 50 m diameter, prep or unprep, no obstacles within 100 m of edge of site.
- Landing site: Taurus-Littrow, only one with sufficient data to certify (Apollo 17 site).
- Other areas have 60-65 meter resolution at best.
- Flight navigation systems will require meter accuracy, transponders close to the base could be used for triangulation.

- Pad loading: 10 psi - static, 120 psi - dynamic (Apollo info)
- Landings to take place early in the lunar night to take advantage of thermal conditions.
- Lighting system needed for pad.

General

- Use teleoperations/telerobotics when applicable.
- Need visualization of processes, critical to teleoperations.
- Micro-meteorite protection necessary, also blast protection from other landings necessary.
- Lunar base population of 4-8 during stays to work with for performing functions.
- Allow for possibility of off-pad landings.
- Lander can be on surface for as long as 180 days for permanent base scenario.
- Redundant and backup systems should be available where possible (double and triple redundancy if critical component).
- Communications should be possible at all times between all system components or personnel.
- Space Power System (SP-100: nuclear 100kW) can be assumed on-line during permanent base inhabitation phase considered.
- Inspection and test equipment needed for all components.
- Decontamination and detection equipment will be needed to detect and remove fuel, coolants, hydraulics, etc prior to exchange or repairs.
- All interfaces should be totally compatible.
- Components should be as interchangeable as possible (commonality) to reduce repair time and complexity.
- Components should have easy accessibility.
- Dust protection should be of utmost importance (especially on physical interfaces).
- All physical connections should be quick disconnect for contingency plans.
- Man/machine interfaces to be worked out.
- Maintenance schedules approximated.
- Consider Space Transportation System craft limitations when designing lunar equipment.
- Vast importance on weight reduction of delivered equipment (on orbit cost per lb:$3500 current system, as low as 300 projected system, $10000/lb to deliver to lunar surface).
- Assume basic lunar base materials and equipment.
- All fueling is done in LLO.

- Extensive landing aids such as lighting and communications will be available to the lander when ascending or descending.
- Line of sight communications necessary at all times.
- Extensive micrometeoriod protection (multi layer prot, etc) on all vehicles and systems.
- Materials used that will not degrade under the presence of atomic oxygen, atomic hydrogen, or caustic fuels left over from Apollo.
- Consider thermal vacuum environment conditions, extensive use of heat shields is necessary: beta cloth, etc.
- Heads up display technology available at the time.
- Electronics to withstand solar flare intensity radiation.
- EVA not used unless absolutely necessary.
- 6 hr EVA time limitation.
- Need visualization of processes, critical to teleoperations.

3. Life Support

A majority of the EVA (Extra Vehicular Activity) and life support system design was derived from conceptual reports coupled with projections of EVA technology. This project centered around the servicing vehicles themselves and not EVA concerns, except in cases that directly related to the project.

The considerations of modifications or additions to be made to the EVA suits to specialize them for use on the servicing missions are necessary. The following will have to be incorporated into the suit designs:

- low mass regenerative closed heat control, no venting
- thicker/better shell for radiation protection
- disposable dust 'over-suit' covering entire EVA suit
- lighting integrated into suit for night-time use

Radiation Dosage Concerns

- space radiation protection standards:
  - males: Career=200+7.5(age-30) rem
  - females: Career=200+7.5(age-30) rem
- GCR (Galactic Cosmic Rays)
- during solar flare worst case:
  - 94 rem/hr unshielded, 22 rem/hr 30 cm shielded Al.
  - 500 rem behind 2 g/cm^2 Al, 11 rem behind 20 g/cm^2 Al.
- GCR (2 g/cm^2 Al).
  - solar min: 45 rem/hr, solar max: 18 rem/hr.

The requirement of a closed loop heat rejection system avoids the water (or other fluid) vapor production in the immediate vicinity of the lunar base and lunar equipment. The landing aids and visual controls will incorporate optics that will be susceptible to the presence of fluid vapors in the immediate atmosphere that possibly adhere to lens surfaces and affect vision or controls.

The advances in radiation protection will allow for longer safe EVA activities, such as those involved with setting up the servicing system and the maintenance that goes along with it. The same crew would be present at the lunar bases for extended periods. During the longest stay of 180 days, a crew member would log many hours on an EVA. This makes improved radiation protection a primary concern.

A dust over-suit design will incorporate a disposable radiation resistant polymer suit that could be zipped up over the EVA suit. This outer suit will be worn one time during an EVA and would then be discarded, which would greatly reduce the serious problem of dust contamination of the components and seals of the EVA suit. This is a well documented and serious problem seen on all missions to the lunar surface [7].

Integrated lighting systems are necessary during both the lunar day and the lunar night. Any maintenance work could be done at night with a lighting system built into the EVA suit to compliment any exterior lighting system that any other vehicles would have. The lighting is also necessary during the lunar days due to the fact that the lack of atmosphere results in extreme contrasts on the lunar surface in rough terrain. It is nearly impossible to judge the depth of holes or craters when they appear black, regardless if they are 1 cm or 1 m deep.

4. Maintenance

Maintenance concerns are paramount to the success of any mission on the lunar surface. The problem of contamination has been extensively documented in past excursions to the lunar surface.

The environment that the equipment will be exposed to is one of the harshest experienced by equipment to date. The following tools and equipment will be necessary for operations concerning the vehicle servicing activities:

- hand held nitrogen (helium) blow-off for dust.
- brushes for dust removal.
- fuel/leak detection hardware, gas analysis.
- dosimeters for radiation warnings.
- tools for LRU (Lunar Replaceable Units):
  - battery (backup) 24x8x14 cm 70 kg.
  - fuel cell core 150 kg.
  - fuel cell tank replacement:
  - hydrogen: 1 m dia 30 kg.
  - oxygen: 0.5 m dia 150 kg.
  - data interface unit 15x16x7 cm 15 kg.
  - electronics/power module 17x9x8 cm 15 kg.
  - pump assemblies.

5. Operations Scheduling

Scheduling is a problem that affects all aspects of a mission. The limited human and machine resources require very efficient scheduling techniques. There are many parameters that affect a mission schedule. The extent of scheduling in the present project involves only concerns of maintenance and Lander support operations. The concerns of overall mission operations are assumed from reports of projected lunar base operations.

The landing schedule would involve preparing lunar base equipment for a lunar landing and the actual operations of setting up the Lander support equipment upon landing. The scheduling is complicated by the fact that in one landing situation (cargo landing), both personnel and teleoperated cranes will be operated at the same time. The cargo landing scenario necessitates parallel processing scheduling.

The techniques of parallel processing have not been addressed in many reports or books to date concerning a lunar landing situation. When more than one individual or teleoperated machine is involved in an activity, the technique of linear scheduling is not practical compared to parallel scheduling.

The times allowed for a given task are limited by personnel EVA radiation and machinery setup concerns. The time limit for a person during an EVA on the moon during a non-solar event activity is generally allowed as 6 hrs . Advances in EVA suits and medicine technology would allow EVA times of up to 10 hrs. This time estimate is used for all scheduling considerations. The machinery will have to be set up fairly rapidly due to the effects of additional heat input to the Lander and components in excess of the heat input from the propulsion system. The Lander would be assumed to have extensive passive heat rejection or storage systems for the time shortly after landing. The lander would still need the excess heat rejected within a short time of landing by the Servicer.

At least two EVA crew members are needed if a schedule of 10 hrs is to be met for Servicer delivery and cargo/crew removal. The Servicer setup and startup is initiated after initial cargo and delivery to base has been completed.

A single EVA person will complete Servicer startup using a small personal transport that would be used around the base for this time frame. The same one will be used for maintenance activities throughout the stay of the Lander.

The following represent approximate activity times for operations at the lunar base pertaining to the servicing and preparation for Lander vehicles, either cargo or crew.

Approximate Activity Times:

Activity Title	Time(hrs) IVA	EVA
LANDING		
Landing site inspection.	0.25	
Landing sys activation and checkout.	3.0	

Track Lander decent.	3.0	
Monitor landing.	1.0	
Monitor system status.	0.0	
SERVICER SETUP		
Transport to landing site.		1.0
Connect to communications/power.	0.25	1.0
Connect/verify H2 connection.	1.25	2.5
Connect/verify LOX vent.		1.0
CARGO MODULE REMOVAL		
Disconnect interfaces of lander, cover.	0.5	1.0
Cargo removed to Transport.	1.0	2.0
Transport from landing site.		1.0
Interface connections.		1.0
PRELAUNCH ACTIVITIES		
Power up and verify lander controls.	1.0	
Automatic Lander systems check.	1.25	
Closeout activities.	1.0	
Configuration video check.	1.0	

5.1 Schedule for Lander in Cargo Mode

The following is an example of parallel processing scheduling techniques for the case of a Lander in cargo mode. This schedule was created by considering published estimated times for activities [8]. Series scheduling is used up to the time that the Lunar lander arrives at the lunar base. After the lander has arrived, the Servicer is brought out to the landing pad on the Prime Mover trailer. At this point, two base personnel are "present" and parallel processing techniques are used. One base crew member is physically present in an EVA suit while the other is present as an IVA (Intra Vehicular Activity) crew member operating the teleoperated crane remotely from the base.

Parallel Processing Schedule (Cargo Landing Mode)

Schedule

T-24h	Transport conversion
T-4h	Pre-landing maintenance
T-2h	Transport ready
T+0h	Post-landing safety check
	- Engine shutdown
	- No fuel present
T+1.0h	Transport travels to landing site
	Servicer setup/cargo work

	Teleoperated	EVA
T+1.0h	- placement of servicer on pad and begin warm-up	- detachment of cargo mods hold-downs, connections - attach Trans Mod umb.
T+2.5h		- attachment of Servicer lines to lander
T+4.0h	- place Cargo Mod on Transport	- detach Trans warm-up power connection to Servicer
T+6.0h	Transport to base	
T+6.5h	Dock to base	

5.2 Schedule for Lander in Personnel Mode

The following is an example of how parallel processing scheduling techniques are used for the case of a Lander in personnel mode. The scheduling operation is similar to the scheduling technique used in the case of cargo landing shown previously. The exception is the fact that there is no remotely operated crane, but there is a second EVA person on the mission out to the landing pad. The parallel processing techniques are used in the operation and interaction of the two EVA personnel.

Parallel Processing Schedule (Personnel Landing Mode)

Schedule

T-24h	Transport conversion.
T-4h	Pre-landing maintenance.
T-2h	Transport ready.
T+0h	Post-landing safety check. - Engine shutdown. - No fuel present.
T+1.0h	Transport travels to landing site. Servicer setup/personnel setup.

	EVA 1	**EVA 2**
T+1.0h	- detach servicer trailer on pad and start warm-up.	- detachment of cargo mod hold-downs, connections.
T+2.5h	- relocate Trans trailer for crew pickup.	- attachment of Servicer lines to lander.
T+4.0h	-Trans crew to Transport.	- detach Trans warm-up power connection to Servicer.
T+6.0h	Transport to base.	
T+6.5h	Dock to base.	

6. Servicer Frame Design

In this section, the frame of the Servicer is modeled and analyzed by using PAL2 finite element programs. A FORTRAN program is then used to size each of the individual members of the Servicer frame. The importance of going through this analysis rests in the fact that the weight must be reduced in any structure that will be transported to the lunar surface. Extreme importance is put on reduction of mass carried to the lunar surface due to the fact that it costs roughly $ 10000 per kilogram to deliver said mass to a lunar base. The frame member sizes are optimized to add strength where it is needed and remove weight where it is unnecessary. Locations of component attachments can also be altered.

The loading conditions will be of several different types, each to be modelled in succession; static loading of components while the Servicer is at rest and loading of the upper members when weight of Servicer is suspended by the upper supports by a crane during moving operations.

It could be argued that the various sized aluminum tubing would be expensive to manufacture in that it would have to be extruded at each of the required diameters. However, the primary concern in any structure design going to the lunar surface is the reduction of weight at all costs.

6.1 Servicer PAL2 Frame Model

The basic frame is modelled as a pin connected simple beam structure, actual frames would be complex welded or bolted frames that would be difficult to analyze. The actual frame of such equipment would be more adequately described as a structure, in that it would have tension, compression, *and* bending on its members. This would require much more complex software than is currently available. The modelling technique used in this analysis assumes a truss design; a design where members are subjected only to tensile and compressive forces. The present analysis is a fair approximation, and is adequate for a conceptual design of this type.

The Servicer frame PAL2 text model can be found in Appendix G. A picture of the Servicer frame can be found in Fig. 6.1 with node numbers included. The basic frame was drawn in AutoCad, modified and material properties added using the MSCMOD software available, and analyzed in PAL2. The model chosen used tubular beams of a constant cross section and material properties of aluminum. All material properties can be found in the PAL2 text model SERV2.TXT in Appendix *.

The Servicer frame can be modeled in two ways, either as loading experienced while stationary on the surface or as loading experienced while the Servicer is being lifted. The two loading cases result in very different forces and deflection in the members of the frame. The two loading cases are covered in following sections.

6.1.1 Surface Loading of Frame

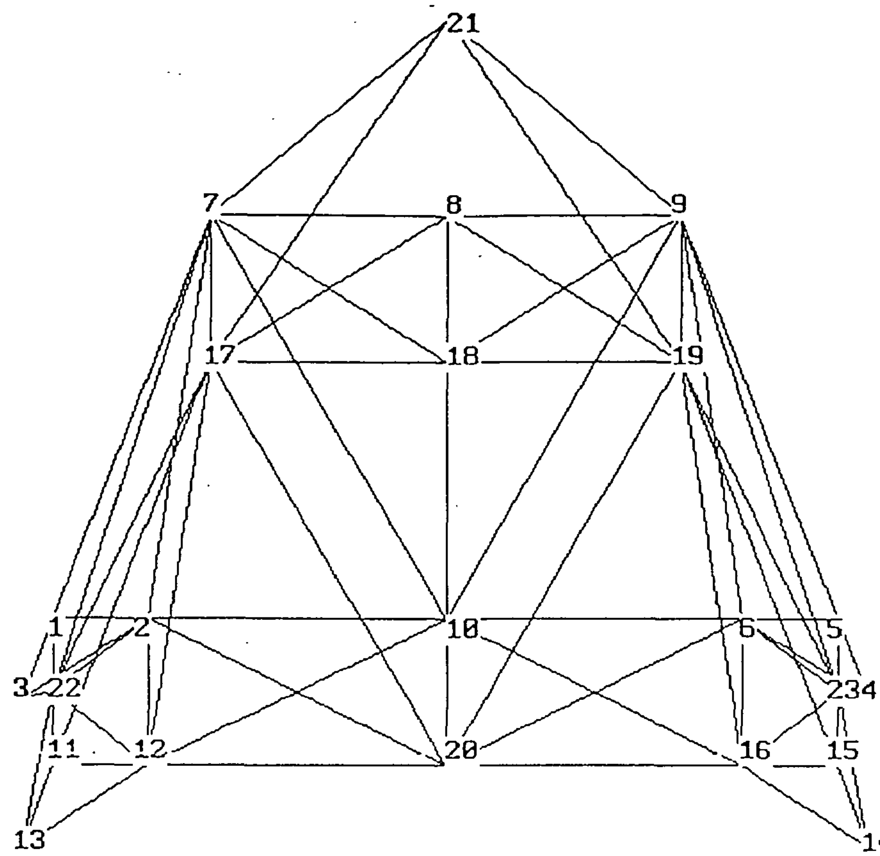

Figure 6.1. PAL2 Servicer Frame Model.

The frame is loaded by downward forces resulting from the weight of the different components of the Servicer. These downward forces are divided and applied at the ends of the members as a fair approximation. The beam elements selected for use in the model do not allow application of a force in the center section, as this would result in a bending loading of the member. The loading file STAT2.TXT can be found in Appendix C, showing the application of the loading due to Servicer component weights. The file also shows the restriction of movement at the support pads set equal to zero, completing the static model requirements for stationary loading of the Servicer due to lunar gravity.

The resultant deflections of Servicer frame members can be seen in Fig. 6.2 . The lower members are deflected considerably as would be expected, where as the upper members are unaffected. The deflection is not to scale.

6.1.2 Lifting Loading of Frame

In the case of lifting loading, the crane is connected to the upper support and lifts the entire servicer by this four pronged support. The Servicer component weights are modeled as in the stationary case, with the top section of the Servicer frame displacement being set equal to zero to simulate the lifting by the upper support. The loading file STAT3.TXT can be found in Appendix C.

The deflections that are caused by this loading can be seen in Fig. 6.3 . The deflections are seen to occur predominately in the upper section of the frame, as would be expected. The Servicer pads carry no load and corresponding members are seen with no deflection.

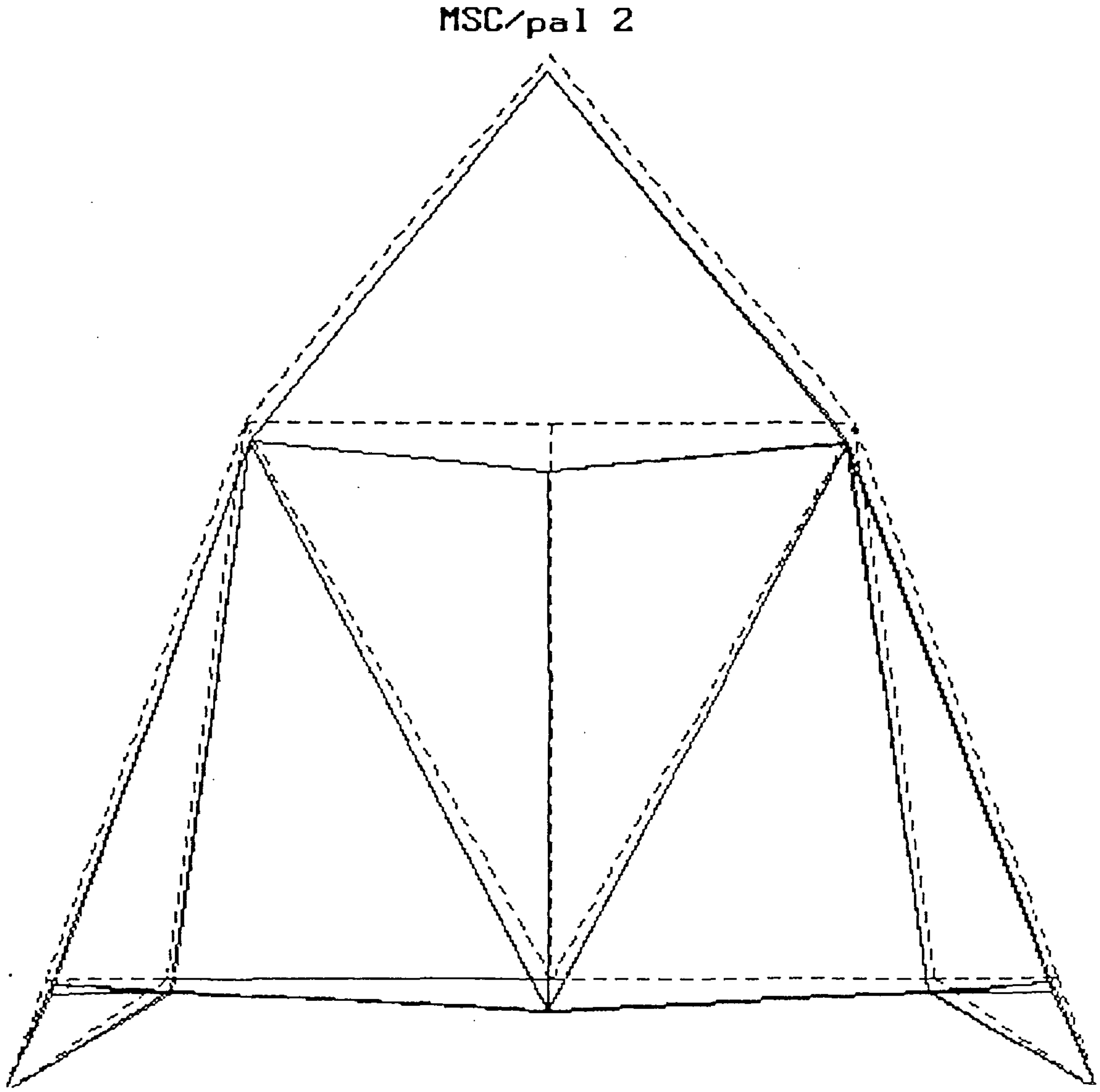

Figure 6.2. Deflection of Servicer Frame Due to Static Loading.

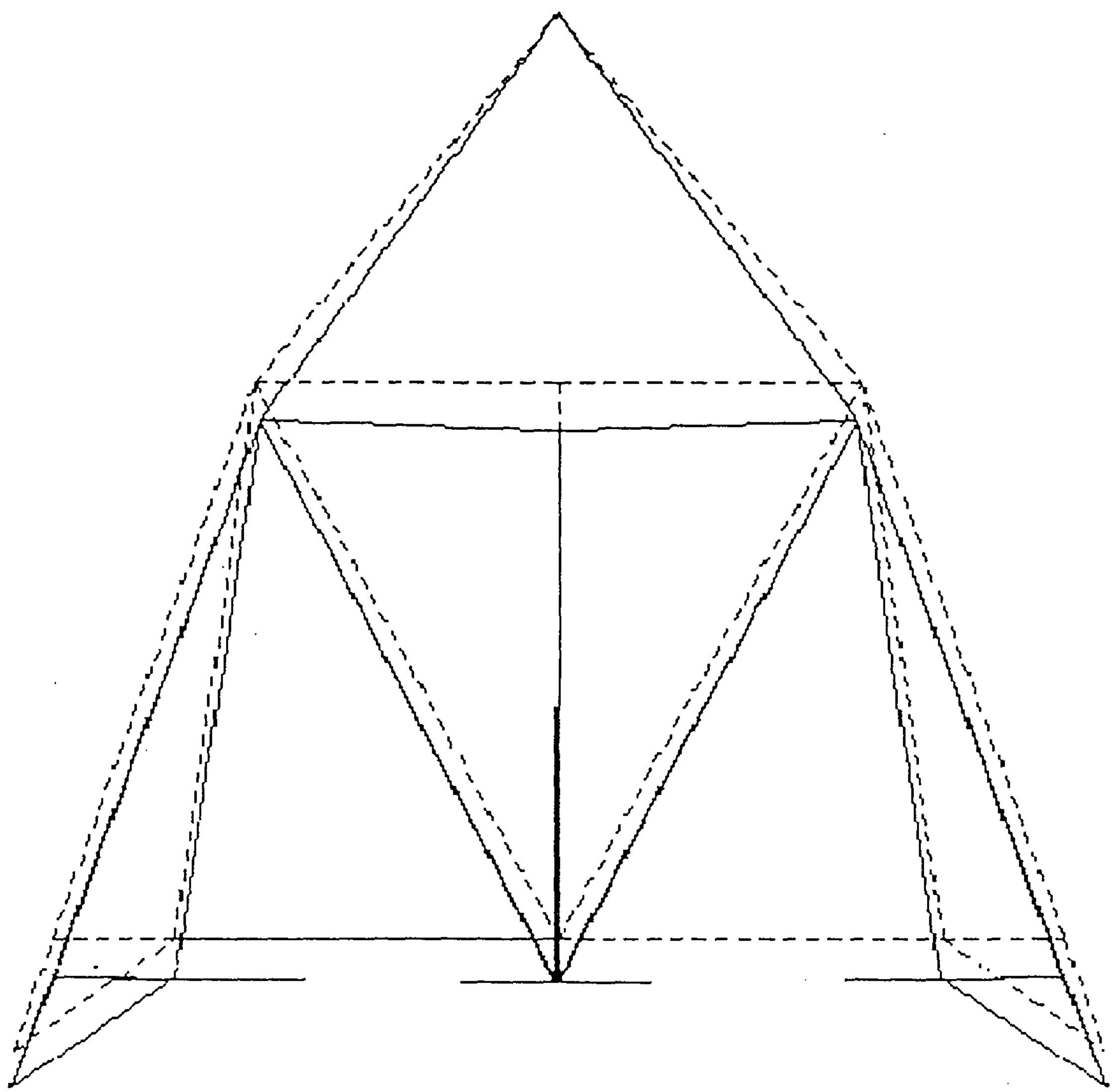

Figure 6.3. Deflection of Servicer Frame Due to Lifting Loading.

6.2 Frame Sizing Program

A static analysis for the static surface scenario was completed and the results of the analysis, including forces, were printed to the file FORCE2.PRT. The program FINDWORD.EXE was written to extract the element numbers and forces (either positive or negative) from the long PAL2 data file, and write them to FORCE2.OUT.

The FORTRAN program FORCE.FOR (see Appendix C) was written for solving for diameters of the aluminum tubing of a given thickness (2 mm) making up the frame of the Servicer. The conditions of maximum allowable axial stress and maximum loading before buckling were considered in the program. A bisect root solver subroutine was used to solve for the diameter in the case of buckling (negative force).

The check on maximum tensile force was performed in the program using the following formula [9]:

$$P_{cr} = \theta_{max} A = \theta_{max} \left[\frac{\pi}{4} \left(D^2 - d^2 \right) \right]$$

The thickness of the wall of the tubular aluminum beams is set at a constant 2 mm. A maximum allowable stress of 296 MPa was used for aluminum. With the thickness held constant, the equation was written explicitly as a function of outside diameter (D) only. The equation was as follows:

$$D = 125 \left[\frac{P_{cr} 4}{296 \times 10^6 \pi} + 0.000016 \right]$$

In the case of buckling, the equation can not be solved explicitly as in the case of tensile loading. The formula for the critical load that a pin supported simple column element can withstand for given section properties was as follows [9]:

$$P_{cr} = \frac{\pi^2 EI}{l^2}$$

The length was set at 2 meters for a worst case situation for column loading in the structure. A Young's modulus of 71.0 GPa was used for aluminum in these calculations.

After substitution of material and section properties, the equation can be written as follows:

$$P_{cr} = \frac{\pi^3}{4 \cdot 64} (71.0 \times 10^9) \left[D^4 - (D - 0.004)^4 \right]$$

The values of diameter (D) were solved for using the bisect root solver subroutine of the program. The results printed to the file DIAMETER.OUT, giving the outside diameter of each of the 68 members (see Appendix C), listing element numbers, forces, and required beam diameters.

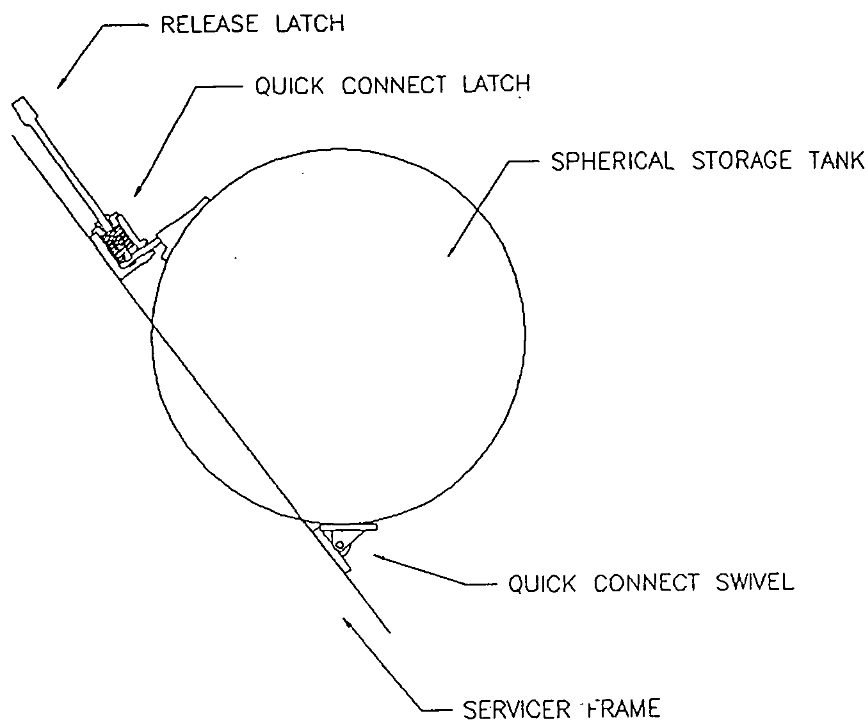

Figure 7.1. Servicer Tank Quick Release Design.

7. Quick Release Design for Servicer Tanks

A quick release attachment was designed for use on the hydrogen, oxygen, and water tanks of the Servicer. The quick release is to be used to exchange empty tanks with filled ones during refueling of the Servicer. The advantage of this procedure lies in the fact that pressure relief concerns are alleviated, although cryogenic connections and disconnections will still have to made by the EVA person. In the case of tank exchanges, the concerns of dangerous hydrogen and oxygen venting in the vicinity of the crew member are avoided.

The release design consists of a top mechanical release and a lower pivoting section (see Fig. 7.1) The large handle release is activated and the tank is pivoted out and away from the Servicer during the operation. The Servicer external radiation and micrometeoroid shielding doors would have been opened for this operation. The large handle allows easy operation of the release by the EVA person. This feature can be seen in a detailed release drawing found in Appendix D.

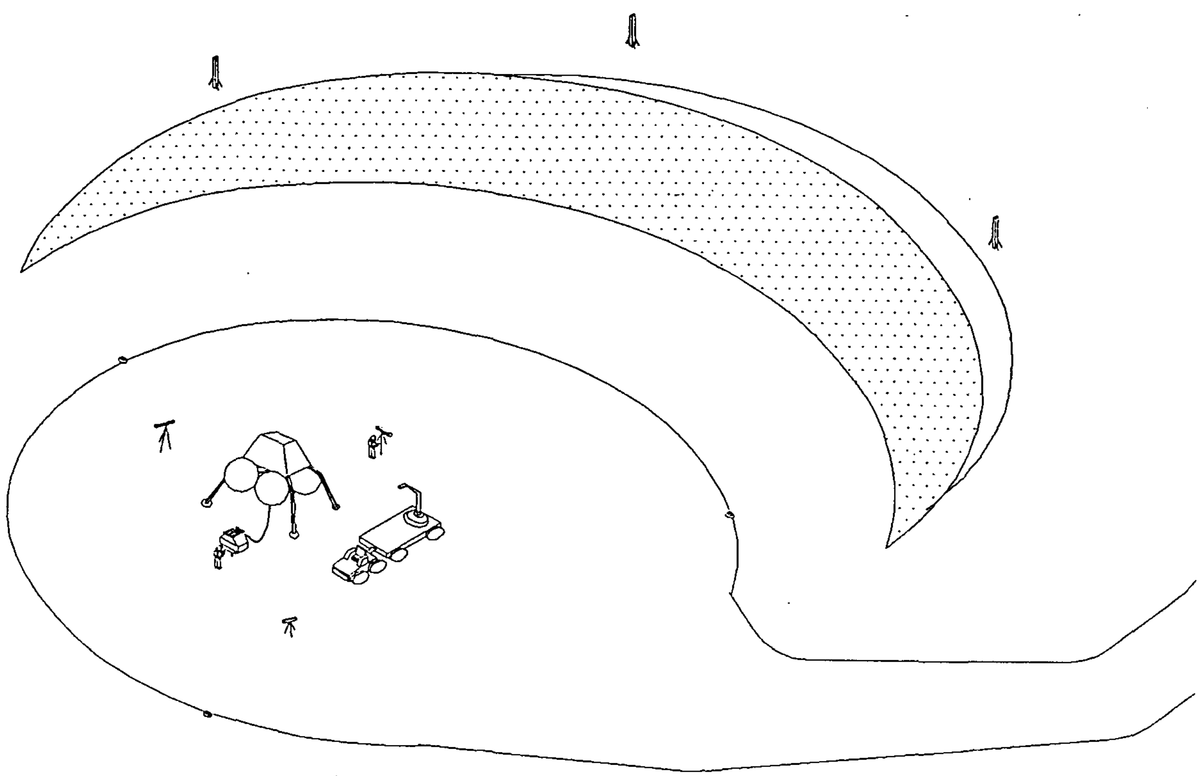

Figure 8.1. Landing Pad Conceptual Design.

8. Landing Pad Sites

8.1 Introduction

The site preparation for the landing pads will be based on location, safety, and accessibility. The pads will be used for Lunar Lander landings and takeoffs for material and personnel supply for the lunar base. Most, if not all, of the lunar base will be located underneath the lunar surface for radiation and micrometeorite protection. Outer air locks, antennae, surface vehicles, and observatories will be the only exposed parts of the base. The lunar landing paths will be perpendicular to the base location to reduce the possibility of danger from lunar landing navigation errors. A landing pad scenario can be seen in Fig. 8.1 , showing a typical operation during night operations.

The landing pads are going to be set up for ease of location and navigation for the lunar lander. The alternative setup was to have the three pads in one location in a line for accessibility for the transport vehicles. In this setup the transport would only have to move to one location. However, there were problems in the blast protection for the neighboring pads, there was a need for a complicated navigation system, and there was the danger of landing accidents due to possible navigation errors. The solution that was chosen was to have the three pads in a triangular formation with the lunar base in the center. This allowed for minimum protection for each landing pad from its neighboring pads, the use of a simple

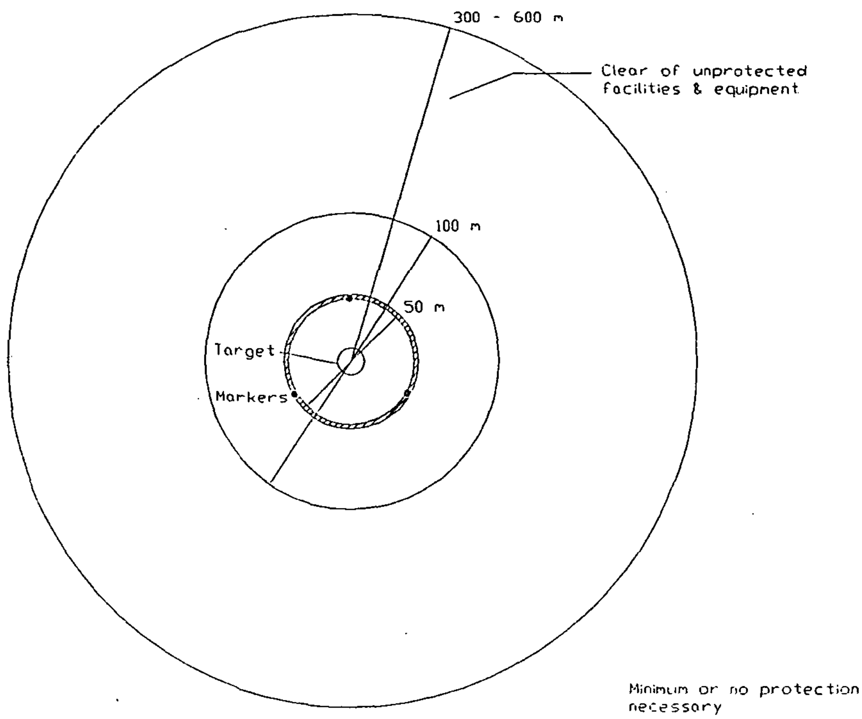

Figure 8.2. Landing Pad Zones.

triangulation navigation system for the lunar lander with respect to the lunar base, and providing a safety margin for navigation landing errors.

8.2 Landing Pad Set-up

The landing pads should be comparatively flat with no slope and free of any depressions. All rocks and boulders should be removed to prevent any damage that could occur from the landing to the lunar lander. The landing pad itself will be 50 m in diameter with a target area of about 25 m in the middle of the pad. An circular area around the landing pad of 100 m diameter will be cleared of obstacles for any emergency or navigation error landings. An area of between 300-600 m radius from the center of the pad should be cleared of any open vehicles or devices to prevent any blast damage from the lunar landings. A figure of this set-up can be seen in Fig. 8.2 .

The landing pads will be located at a sufficient distance away from the lunar base so as to reduce the effects of the blast from the lunar lander landings and takeoffs. This will negate the necessity for any large amount of extra shielding for the base for protection. Studies have shown that minimal safe distance for a landing pad from the base is 400 m[10]A distance of 1000 m was chosen because the base will experience negligible amounts of blasting from the lunar landings and will be comparatively safe from any possible landing errors[11]

8.3 Pad Preparation

The landing pads will be prepared by marking out prospective sites, then clearing the large boulders and rocks, grading the surface flat, and then compacting the surface to make a safe landing pad. The landing pad should have less than a 6° slope with no humps or depressions. The 100 m diameter safety zone should have less than a 12° slope with no humps or depressions[10]. All of this will be done by the Prime Mover that will be located already at the lunar base from previous landings. The diameter of the landing pads will be 50 m and there will be a total of three landing pads. The landing pads will in a triangular formation so that navigation to the base and the pads are less complicated. In the later stages of the lunar base development, the compacted pad surface will be replaced by a gravel and paved tile surface[10]. This surface will reduce the dust problem present in the plain compaction, and will be a longer lasting surface with minor maintenance requirements.

8.4 Blast Protection

The blast wall will be located 25 m away from the pad and will be formed using the Prime Mover. The walls will be 2 m in height with a wall base width of 8.6 m and a repose angle 35 degrees. This will allow sufficient protection for the navigation devices and the base by shadowing the navigation device at each pad and directing the dust away from the base. The semicircular formation of the walls around the pad will provide the shadowing and protection for the base and the other two landing pads. The triangular formation of the pads around the base gives them enough distance in between to have minimal protection necessary for the equipment located on any neighbor pads.

8.5 Pad Access

There will be a prepared roadway from the base to the three pads consisting of a graded surface free of boulders. This provides a low-risk travel for the maintenance equipment on repeated travels to the landing pads and to increase their life, in effect reducing later costs. This is not to rule out the possibility of the transport traversing unprepared surfaces. This compaction reduces the number of bumps and hills it has to encounter so that less power will be required, a safer trip will be made, and less time will be required for travel between pads. In the early stages, the roadways will be setup like the landing pads with the compacted surface. However, as time goes on, in the later stages of the lunar base development, a gravel and paved tile surface will be used for road travel. This will make the road even smoother and greatly reduce the wear and tear on the transport vehicles[10].

8.6 Pad Lighting and Markers

Each landing pad will contain three permanent markers protected by placement in the lunar regolith[11]. They will be seen only from above for visual verification and adjustments. They will be located on the perimeter of the 50 m diameter landing zone. Because they only need to operate approximately ten minutes every two to three months, they can be battery operated and activated in pre-landing maintenance or radio activated[11]. The batteries can be solar charged batteries. The lights for lunar night operations can be portable and carried out with the servicer on the transport flat bed. Basically they are flood lights that can be set up in the area of operation.

Each pad will also contain a navigation device located on the other side of the regolith blast wall. The device used for this pad set-up is described further in the Navigation section of this report. Because of the size and location of the regolith wall, the device can be located just about anywhere behind the wall. The device, then, will be located 5 m behind the wall in line from the center of the pad to the base.

8.7 Pad Maintenance

The maintenance of the pads will consist basically of cleaning out the landing markers if necessary, and compacting the landing pads. The roadway may also need compacting. In the later stages, due to the tiles, the only maintenance necessary is the cleaning of the markers and minor dusting of the pad and road surfaces.

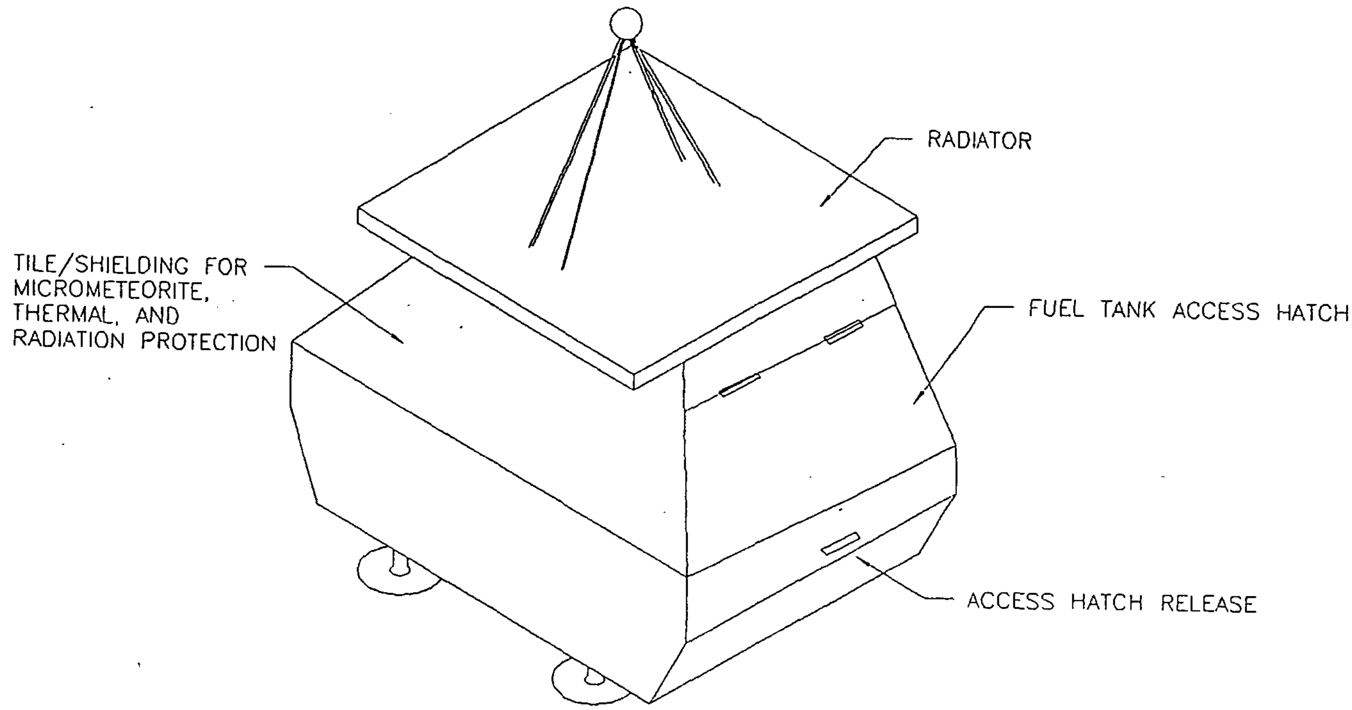

Figure 9.1. Lander Servicer (With Thermal Protection.

9. Servicer and Transport Protection

9.1 Introduction

The considerations for a base and its launch and landing facility will be radiation and micrometeorite protection. Radiation protection as well as micrometeorite protection is a concern for humans as well as electronic equipment. The main concern for this study is the protection for the servicer and the transport. The Servicer design with the radiation and micrometeorite protection can be seen in Fig. 9.1 . The outer covering of the servicer is a stand-off shield design consisting of a few walls for its protection. The cut-away view view of the stand-off shield can be seen in Fig. 9.2 .

9.2 Considerations for Shielding

9.2.1 Radiation

9.2.1.1 Radiation Hazards

The radiation components that will be encountered on the moon consist of galactic cosmic rays (GCR), and solar proton events (SPE). GCR is a highly penetrative dose of radiation, which induces the need for unique shielding requirements. SPE comes from the solar flare

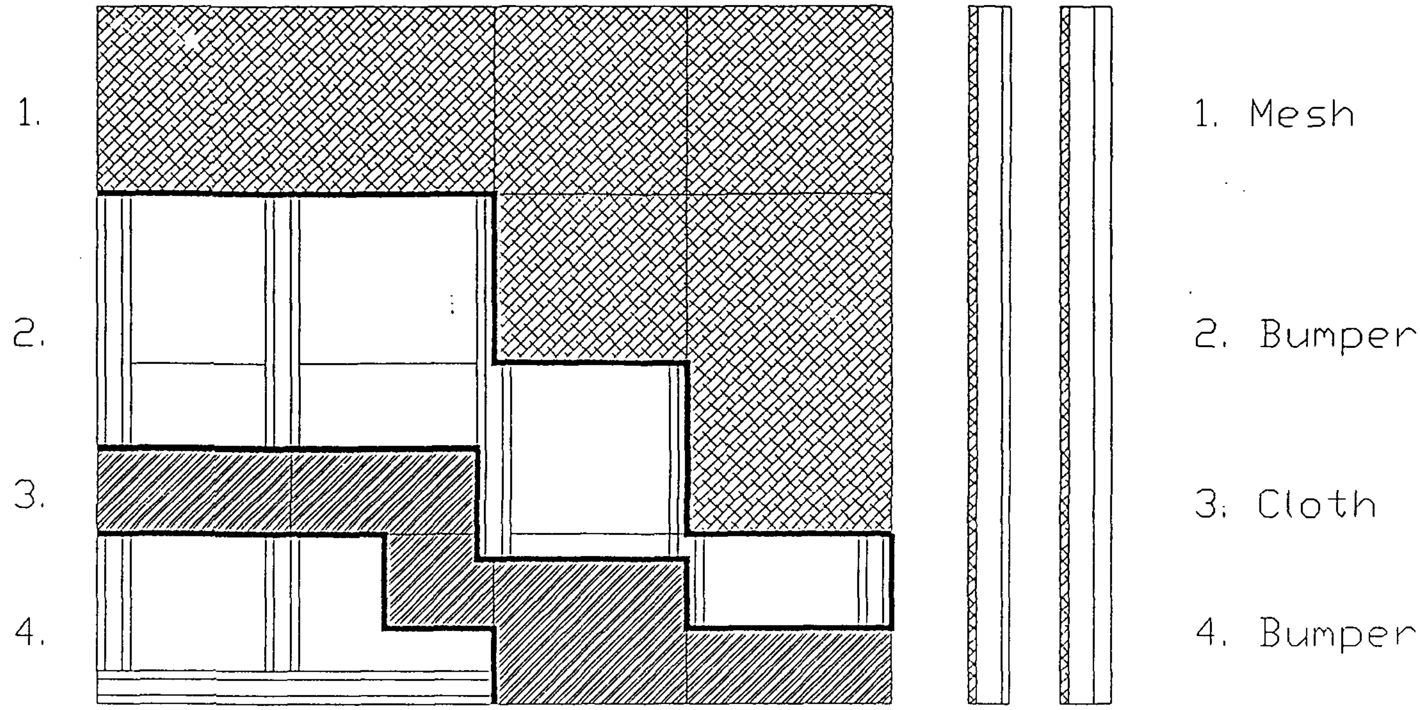

Figure 9.2. Shielding Set-up for Servicer and Transport.

activity from the sun. The flares deliver deliver highly concentrated doses of radiation over short periods of time ranging between a couple of hours to a few days. SPE's are relatively unpredictable, however, their occurrence corresponds to the sun spot cycle[12].

The intensity and potential danger to human life due to SPE dosage would lead to the requirement of a radiation shelter strong enough to reduce the effects of the radiation to minimum levels. In most studies, the base is located underneath the lunar surface to provide the maximum amount of protection against radiation effects. Lunar regolith is the best medium for radiation protection. All of the compound will be under at least 4 m of regolith, which will be sufficient to reduce the dose to less than 5 rem during the flare events[11]. All activity will be suspended during the time these events are to occur to protect the crew and the equipment. With this in mind, the only protection necessary is for the GCR doses.

9.2.1.2 Radiation Shielding

GCR is a continuous source of highly penetrating particles consisting of protons and heavy nuclei[12]. The maximum flux expected from GCR occurs at solar minimum, when there is little sunspot activity. GCR Flux is at its minimum intensity during the solar maximum[12].

In John R. Letaw's report on the radiation effects, a study was made on aluminum shielding between 0 and 70 g/cm2. This report found that at solar minimum, with maximum GCR dose at 45 rem/year, 20 g/cm2 (7.5 cm) aluminum shielding reduce the dose equivalent by a factor of two[12]. The report stated that the reduction level decreased at higher levels of thickness, and it was unnecessary to increase the thickness beyond 7.5 cm. Further reductions could be accomplished by selection of different materials[12].

The selection of the radiation protection is based on this study. A 7.5 cm wall will be used to protect the Transport and the servicer against the dose from GCR.

9.2.1.3 *Future Considerations for Radiation Shielding*

Aluminum was used in this project because of the studies made on the use of this material. Aluminum is also commonly used structural material in aircraft today[12].

The use of water in aluminum as a shielding alternative could be a possibility. The water shielding was introduced as a better medium against the GCR and SPE doses[12]. With regolith being the best protection against radiation doses, a study of a shield made out of lunar materials could be done. Other examples of possible good shielding against GCR are methane, hydrogen, plastic, and boron and lithium hydrides with aluminum[12].

9.2.2 *Micrometeorites*

9.2.2.1 *Micrometeorite Hazards*

Another hazard that the servicer and transport will encounter is that of micrometeorites. Both the servicer and the transport require some kind of shielding to protect the vehicle and equipment from meteoroids and blast ejecta from lunar landers. The task is to create a shield that is low in weight and in cost that will protect the equipment from most of the particles. The shield that has been used in the past years for Aerospace vehicles is the "Whipple" shield. This shield consists of an aluminum sheet spaced in front of a larger aluminum wall. This form has been successful in the past. However, increased studies in the hypervelocity particle threat have called for a new form of shield to be sought[13]. This new style of shielding consists of "stand-off" formation with lighter materials and more walls[14].

Another hazard that can be eliminated by using more walls and better materials is the ejecta cloud. The ejecta cloud is formed when the hypervelocity particle hits the front wall and explodes. The resultant particles shoot out in all directions, including the direction the the particle originally came from. These particles can damage other parts of the equipment that was missed in its original path. This problem is reduced by the ability of using a mesh layer to catch the particles. The Whipple wall is unable to utilize this layer because it needs two strong walls to stop the particles. The figure of the whipple wall and the ejecta formation can be seen in Fig. 9.3 .

9.2.2.2 *Double Bumper Consideration*

The proposal for our design was the use of a mesh double bumper system. The system consisted of four different layers. The first layer was a mesh layer, the second layer was an aluminum bumper wall, the third was an intermediate cloth, and the fourth was the final

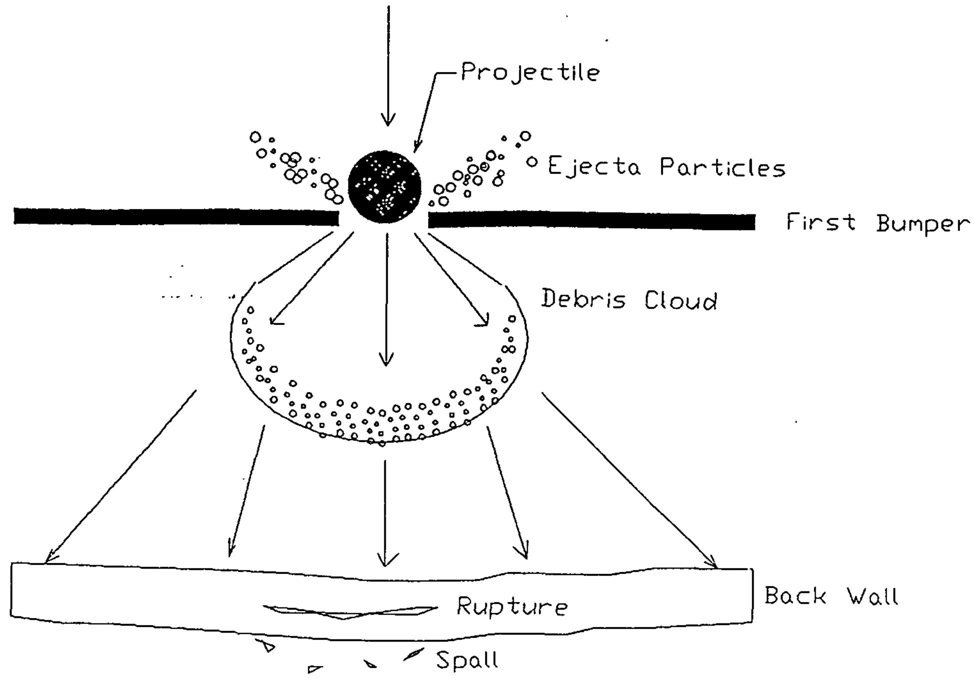

Figure 9.3. Whipple Shield.

aluminum bumper wall. The Aluminum bumper walls are aluminum alloy plates (A2024-T4), basically because of its high tensile strength[15]. The mesh was made out of aluminum 5056 wires with a gap of 0.5 mm between the wires. The intermediate cloth is Spectra cloth because of its high strength to weight ratios.

The way the shielding works is by having the particle hit the front mesh layer. When the particle hits, it breaks up the particle to be shocked by the second bumper. The mesh also spreads the debris over a wider area than a conventional wall and thus reduces the impact force on the second wall. The second bumper is used to shock the fragments and melt or vaporize them or further pulverize them[13]. The third layer, the Spectra cloth, is used to slow the advance of the debris cloud from the second wall and reduce the momentum loading on the back bumper. The purpose of the back plate is to resist the impulsive loading from the debris cloud[13].

The spacing between the walls of the shield is based primarily on the diameter of the particles hitting it. Optimum spacing between the first two bumpers should be at least 3 times the diameter of the particle. Optimum spacing between the last two walls should also be around 3 times the diameter of the particle. The optimum overall spacing between the front wall and the rear wall should be at least 30 times the diameter of the particle. This arrangement can be seen in Fig. 9.4 .

The sizes and speeds of the hypervelocity particles can vary from the very small to the very large. The best thing for design is to base it on the average sizes and speeds of the

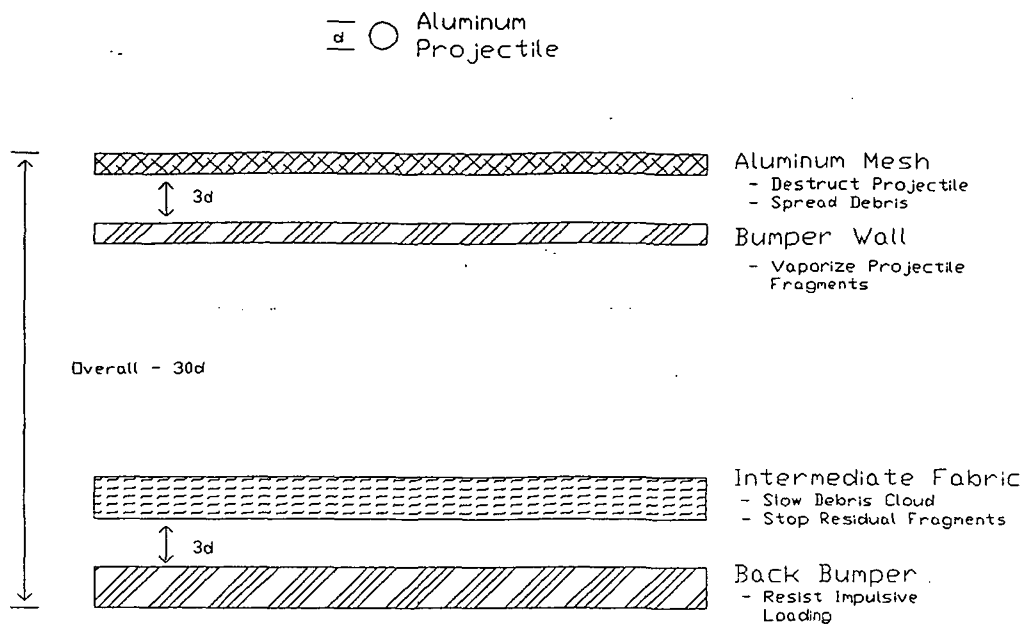

Figure 9.4. Aluminum Mesh Double Bumper System..

particles that can be encountered. The mass of the meteorites can range from 0 to over 1000 kg. As can be imagined, to design a shield to protect against a particle of up to a couple of thousand pounds is almost impossible with today's technology. Most of the lunar dust is less than 5 mm in diameter[11], so the shield will be designed with this in mind.

9.3 Consideration of Tile Shielding

There is a need to have a shielding system that will be easy to replace or repair. The lunar environment does not allow for completion of complex tasks, so the system has to be easy for the astronaut in the cumbersome suit. The proposal for this project is to use a tile system for both the servicer and the transport. This will allow the astronaut to replace the shielding with ease since it will be close to a "modular" design.

This design allows for a simple outer shell to cover the whole lander. The tiles can be attached in a fashion similar to that of the space shuttle. The tile design also allows for easier transport because it can be stored easier than a whole shell design. This will help in the reduction of weight and space requirements. These tiles can also be compatible between the servicer and the transport. Only one shield style will be necessary, which will help to reduce costs.

The tile contains the micrometeorite protection and the radiation protection. The rear aluminum wall is used as the radiation protection and the four walls serve as the micrometeorite protection, as described above. The tiles will have to be replaced only when damaged by large meteorites or after the tiles radiation life time, where it would have to be replaced with a fresh tile because of the built up secondary particles.

9.4 Future Considerations on Shielding

The intermediate and rear bumpers of the shielding can be studied further in the area of lighter and stronger materials and composites. A study in the feasibility of the manufacturing of shielding materials for radiation and micrometeorite protection from the lunar surface materials can also be made.

10. Transportation System

The transportation system for this year's design project is actually a "bookend" type of consideration and not the main focus of the design work. Because the transportation system is an adjunct project component to the servicer, most of the design work for the transportation system is conceptual. Much of the design of the transportation system is concerned with modifying lunar transportation systems that are already in existence or have already been designed.

One of the most important considerations for the transportation system is that it must be multipurpose. Strictly speaking, the transport itself does not have to be multipurpose for it to function properly, but it must be multipurpose in order to justify the time and expense that will go into the design and manufacture of the transportation system. As mentioned previously, the transport system is simply a secondary part of the LLGSS. The main purpose for the transport system for the LLGSS is to provide the servicer with a mode of transportation to and from the base and to take any cargo and/or crew to or from the lander. During one period of operation, the servicer may function continuously for a period of up to 180 days. During that same period of operation, the transportation system will be used for 6.5 hours at the beginning of the cycle to deliver the servicer and pickup crew or cargo; 52 hours during the operation of the servicer for weekly maintenance of the servicer and tank change-out assuming a two hour maintenance period; and 6.5 more hours at the end of the working period to take crew or cargo to the lander and retrieve the servicer. Therefore, during a working cycle of 180 days for the servicer, the transportation system will be used only 1.5% of the time. If the transportation system were not multipurpose, it would be sitting idle for over 98% of a working cycle, which is not very efficient. From this perspective, a transportation system that can perform other tasks while it is not needed for the LLGSS is much more practical.

9.4.1 Operating Conditions

Another major concern for designing the transportation system is the lunar environment itself. Unfortunately, no prepared roads or highways already exist on the moon, so the transport will have to be able to move in the lunar soil and dust. The lower force of gravity the moon exerts makes traction difficult. In addition to the poor driving surface, the transport may also encounter rocks, craters, and other obstacles while it is operating on missions that require it to travel away from any prepared surfaces near the base. Therefore, it is necessary to find a design for the transportation system that will be able to encounter the rough lunar terrain.

10.1 Prime Mover

The prime mover, as seen in Fig. 10.1 , will be the basis for transportation on the moon, and is simply an unpressurized heavy duty lunar rover. In addition to using the Prime Mover for the transportation system of the LLGSS, the astronauts may use it for simple

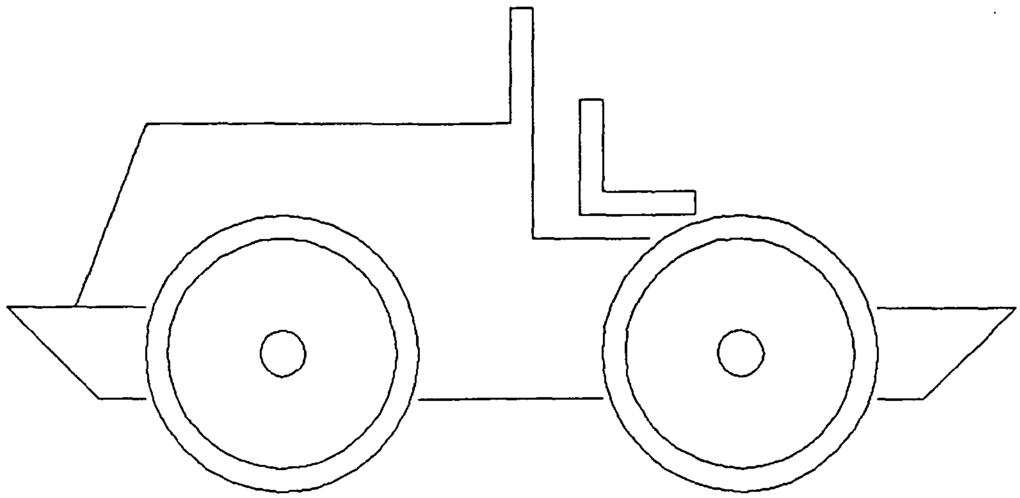

Figure 10.1. Prime Mover.

transportation around the base. Eagle Engineering has produced a conceptual design for their own lunar rover that has been outlined in [16]. The rover that Eagle Engineering designed has a mass of 650 kg and a payload capacity of 1000 kg. The rover that will be used as the Prime Mover for this project will have to be a bit more rugged and stronger than the one considered by Eagle Engineering , and will have an estimated mass of 3000 kg. Eagle Engineering designed their rover primarily for simple transportation and did not consider it for other purposes. The Prime Mover as outlined for this project will serve more purposes than the rover designed by Eagle Engineering .

The Prime Mover for this project will essentially be of the same form as the rover designed by Eagle Engineering and the lunar rover used for the Apollo moon exploration missions. The Prime Mover will be unpressurized and open to the lunar environment, which will necessitate the use of EVA suits by the astronauts. For the LLGSS, the Prime Mover will be the "field command center" of the operation. The astronauts will be seated in the Prime Mover when they are driving the transportation system to and from the landing pad in order to control the transportation system . All of the wheels on the carts that are attached to the Prime Mover will be independently motorized, but the power for the motors will come from the PEM fuel cells that will be on the back of the Prime Mover . The fuel cells used on the Prime Mover are discussed in the Power section of this report. The carts that are attached to the Prime Mover also have no means of independent means of steering and depend upon the lead of the Prime Mover for steering.

10.1.1 *Power Requirements*

The power requirement of the transportation system was determined by using the simple physical equations relating power to force, velocity, and time. According to Soil Man, a vehicle will require 0.08 $\frac{W}{kg\ km/hr}$ of power due to rolling resistance [17]. With a total combined mass of 7,000 kg, the entire transportation system will require 2.5 kW of power to travel at a constant speed of 4 km/hr along a level path to and from the landing pad. When the transportation system is carrying the servicer or the pressurized cart with astronauts inside, the transportation system should go no faster than 4 km/hr in order to avoid severe jostling of the servicer and the astronauts inside the pressurized cart .

While the Prime Mover is on a mission by itself, it may travel at speeds of up to 10 km/hr. Speeds faster than 10 km/hr are too fast for the rough terrain of the lunar surface. The Prime Mover may encounter slopes of 30° while it is on a mission away from the base. Assuming the worst case of a 30° slope that the Prime Mover will encounter while travelling at the maximum speed of 10 km/hr, the Prime Mover will need 6.8 kW of power just to go up the slope plus an additional 2.4 kW of power to compensate for the rolling resistance. With a factor of safety of 2, the Prime Mover will need 18.4 kW of power.

Probably the worst case scenario would involve the Prime Mover starting from a dead stop and accelerating up a 30° slope at 0.278 m/s^2 to a velocity of 10 km/hr. For this case, the Prime Mover will require 9.2 kW of power for the acceleration up the slope and an additional 2.4 kW of power to overcome the rolling resistance of the lunar soil. Again, using a factor of safety of 2, the Prime Mover will require a maximum of 23.2 kW of power.

The fuel cells that will be used for the Prime Mover are PEM fuel cells manufactured by Ergenics Power Systems Inc, and have a peak output of 1.6 kW of power. Combining 15 of these fuel cells will provide a peak power output of 24 kW. The 15 fuel cells will be placed on the back of the Prime Mover , and some may be removed or added, depending upon the mission requirements. According to Ergenics Power Systems Inc, their PEM fuel cells consume 320 g/kW hr of oxygen and 40 g/kW hr of hydrogen. Thus, for the worst case of needing 24 kW of power, the fuel cell will consume 7.7 kg/hr of oxygen, 1 kg/hr of hydrogen, and will produce 8.7 kg/hr of water. For the more reasonable mission of taking the servicer back and forth to the landing pad, the fuel cells on the Prime Mover will only consume 1.6 kg/hr of oxygen, 0.2 kg/hr of hydrogen, and will produce 1.8 kg/hr of water. The details and calculations for the power requirement of the Prime Mover may be seen in Appendix F.

10.1.2 *Prime Mover Attachments*

In addition to providing basic transportation around the base, several attachments have been designed for the Prime Mover so that it may perform other tasks around the base. These attachments will mainly be used for construction work and surface preparation. A couple of the attachments for the Prime Mover may be seen below in Fig. 10.2 . The roller attachment as seen in Fig. 10.2(a) is similar to a steam roller on earth because it is used to compact the lunar soil and dust. The roller will be constructed so that it is hollow. This will reduce the weight of the roller attachment significantly, for a solid roller to compact the soil will have to

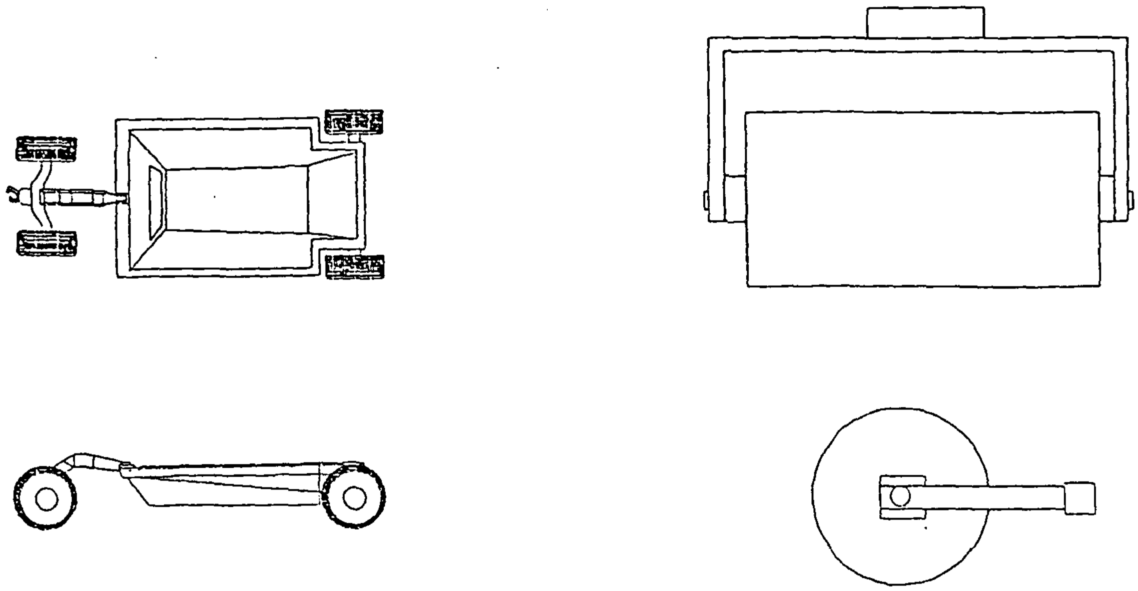

Figure 10.2. Attachments for the Prime Mover (a) roller attachment, (b) scraper attachment.

be quite heavy. Because the cost of sending 1 kg of cargo to the moon costs nearly $10,000, weight conservation is a big issue. Once the roller is on the moon it may be filled with soil, dust, and rocks from the lunar surface to give it the weight necessary for compacting the lunar soil. The scraper attachment shown in Fig. 10.2(b) is used in a fashion similar to a road grader on earth. The scraper will essentially scoop up any rocks and soil on the surface to create a level area. Once the surface has been levelled by the scraper, the roller may be used to finish preparing the area for a road, landing pad, or even as a foundation for some structure.

10.2 Trailer

The trailer of the transportation system is a simple flatbed trailer. The trailer involves a little more actual design for this project because the trailer will work in conjunction with the robotic arm that will be used to lift the servicer and cargo. The trailer as shown in Fig. 10.3 is 8 meters long, 3 meters wide, and is 1.5 meters in height. This does not include the height of the robotic arm. The estimated mass of the trailer is 2000 kg unloaded and without the robotic arm. The robotic arm has a mass of 450 kg and is located at the rear of the trailer. The trailer will use six independently motorized wheels. The rear wheels are placed next to each other to support the weight of the robotic arm and the additional forces that will be placed on the robotic arm while it is lifting the servicer and/or cargo. The servicer is estimated to have a mass of 3000 kg. Assuming that 3000 kg is the largest amount of mass

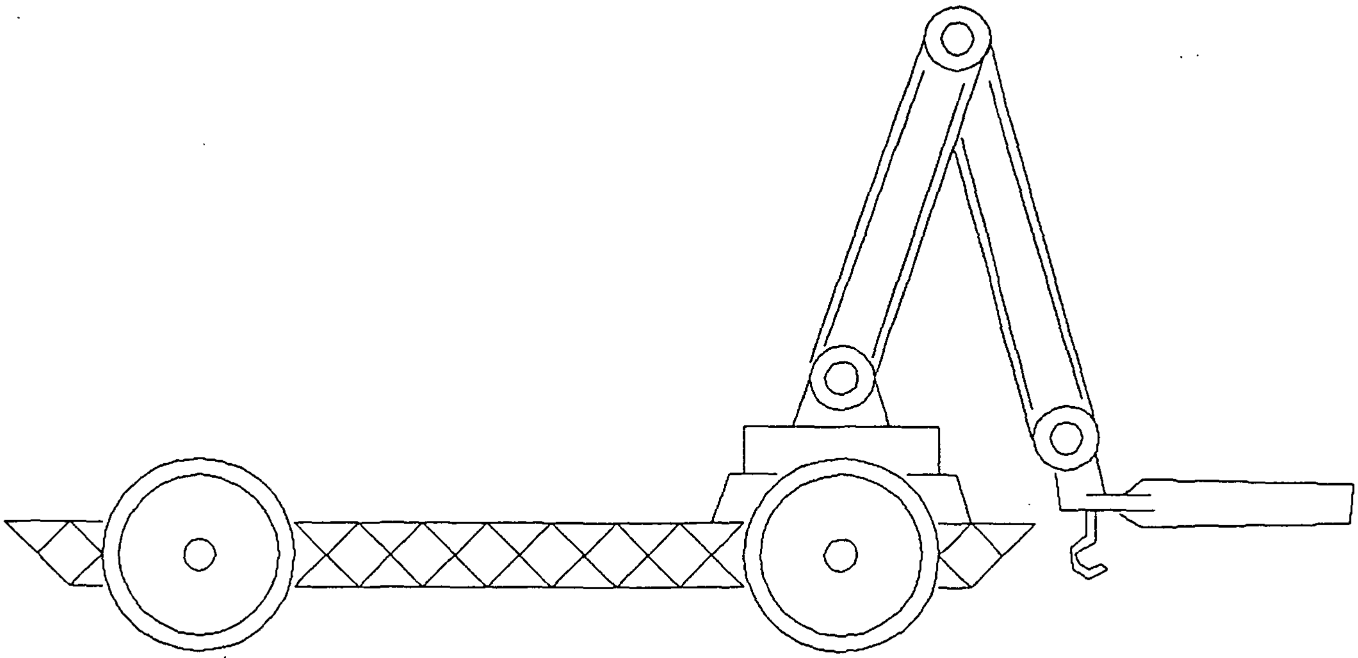

Figure 10.3. Trailer With Robotic Arm.

that the robotic arm will lift, this will create a moment large enough to tip the trailer over when it is lifting 3000 kg. Therefore, it is necessary to place extendable foot pads that will extend 1.5 meters from the trailer and provide the necessary support for the trailer to remain stable while the robotic arm is lifting. This is especially needed when the load is off to the side of the trailer.

10.2.1 Servicer Tie-Down

Another important feature of the trailer is some means to enable the servicer and other cargo to remain on the trailer while it is in motion. Obviously, the transport will not be breaking any land speed records as far as acceleration is concerned, so the servicer or cargo will not slide around solely due to acceleration of the transport, which is only 0.278 m/s^2. For a parcel of cargo that has a mass of 1000 kg, the coefficient of friction between the bottom surface of the cargo parcel and the trailer surface only needs to be 0.176, which is quite small, to prevent motion of the cargo. The details may be seen in Appendix E. The road out to the landing pad will be rough even though it will be prepared. The rough ride will cause an unrestrained cargo parcel or the servicer to move around and slide off the trailer which will result in severe damage. Therefore, some type of restraining device is required in order to keep objects on the trailer from falling off.

A very simple restraining system for the trailer is shown in Fig. 10.4 . The restraining system consists of small tracks that are permanently affixed to the top of the trailer surface along each side. A cross-section of the tracks may be seen in Fig. 10.4(a). The tracks will

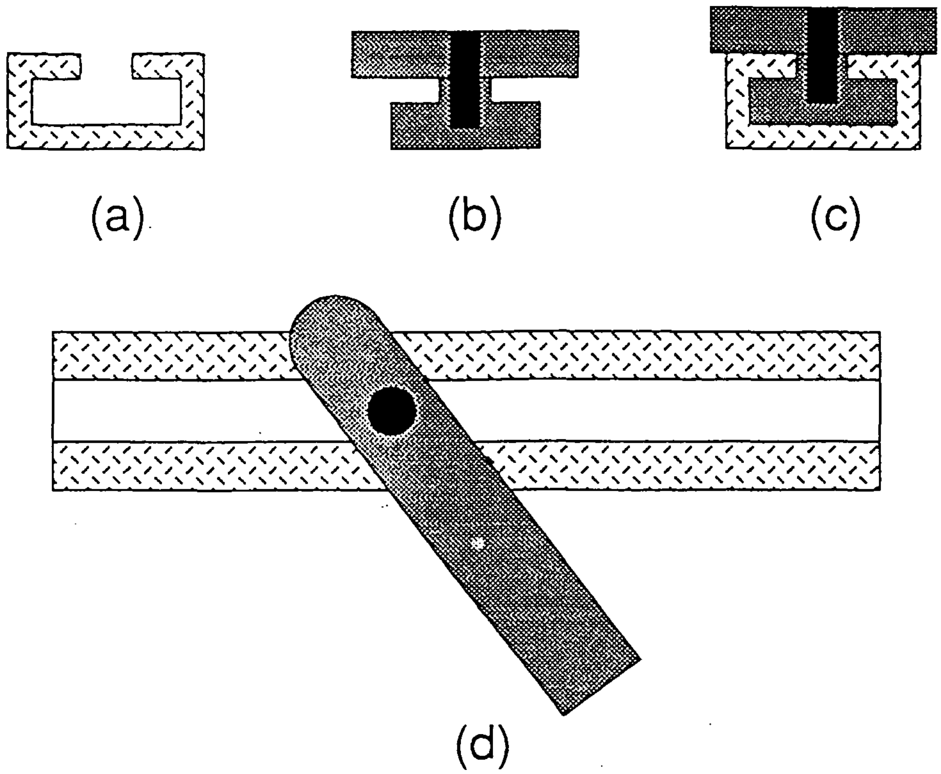

Figure 10.4. Cargo/Servicer Restraint for the Trailer.

be manufactured with open ends that will have plastic endcaps. This will ease the cleaning process of the tracks that will surely accumulate lunar dust because they are open to the environment at the top. Cleaning the tracks will be a simple matter of removing the endcaps, sliding the guides out through the open ends, and blowing a gaseous jet along the insides of the tracks to remove any dust particles that have accumulated.

Fig. 10.4(b) shows a cross-section of the slide assembly that slides along the inside of the tracks. The bottom piece of the slide assembly is the guide that actually slides inside of the tracks. The top piece of the slide assembly is a rail that is connected to another guide on the other side of the trailer. The rail is the device that holds the cargo and/or servicer in place, and is attached to the guides by a pin joint that is sealed from the lunar environment. The rail is attached to the guides with pin joints so that the rails may be rotated to any angle to compensate for any orientation that the cargo or servicer may be placed onto the servicer. Fig. 10.4(c) shows how the components fit together. As mentioned previously, the tracks are free to rotate to any angle as is shown by the top view in Fig. 10.4(d). The servicer will have special foot pads designed to work in conjunction with the restraining system. Once the servicer is placed onto the trailer, the rails may be slid into place to sit just above the servicer's footpads and to rest against the side of the servicer. Once the rails are in place, they may be "locked down" in order to prevent them from sliding, thus keeping the servicer and cargo in place on the trailer.

10.3 Pressurized Crew Transport

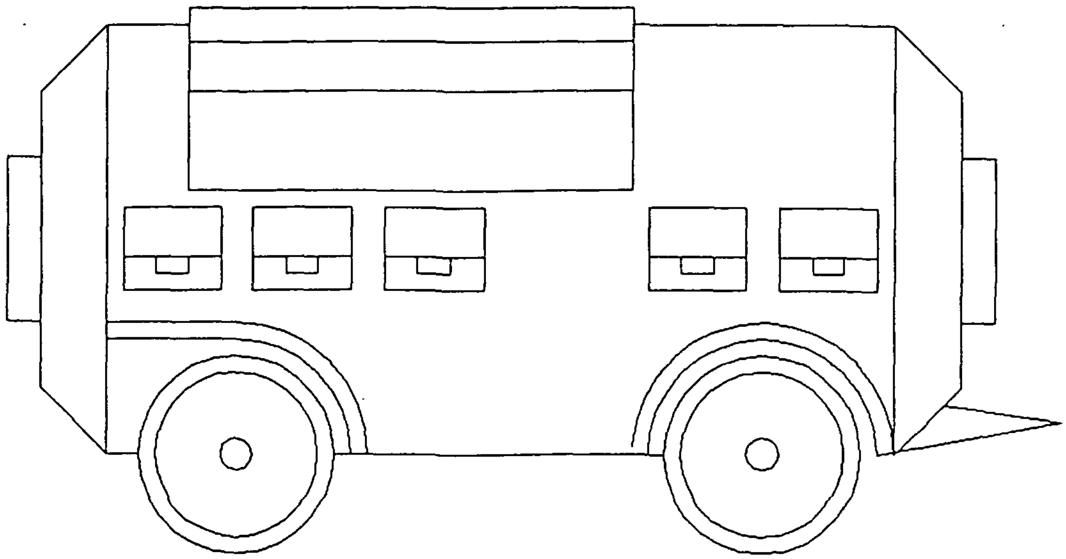

Figure 10.5. Pressurized Crew Cart.

For the scope of this project, the pressurized crew transport, as seen in Fig. 10.5 , will be used only when the lunar lander is in the crew mode. The pressurized crew transport is simply a pressurized trailer that will be used to take the crew from the lander to the base, or from the base to the lander. Using the pressurized cart to transport the crew eliminates the use of cumbersome EVA hardware and the need for a man-rated crane to remove the crew module from the lander with the crew still inside for transport to the base.

The pressurized cart to transport the crew members around, designed by Eagle Engineering [16], is shown below in Fig. 10.5. The pressurized cart was not actually designed for transporting crew from the lunar lander to the base. Eagle Engineering originally designed the pressurized cart as a habitation unit for scientists on extended exploration missions during the early years of lunar operations. The pressurized cart , called the Habitation Trailer Unit by Eagle Engineering, has been designed to comfortably house several scientists for periods of up to three months. While the trip from the landing pad to the base will only take several hours at the most, the pressurized cart makes an excellent way for astronauts to travel without the use of cumbersome EVA suits. Another advantage of using the pressurized cart is that it will already be built and in use by the time its services will be needed.

One concept for transferring the crew of the lunar lander from the pressurized lunar module to the pressurized crew transport is through the use of a pressurized tunnel. The pressurized tunnel, also designed by Eagle Engineering [18], is shown in Fig. 10.6 . The pressurized tunnel is similar to mobile stairways used by airlines. According to Eagle Engineering, the tunnel is basically a trailer with a special pressurized tunnel and universal

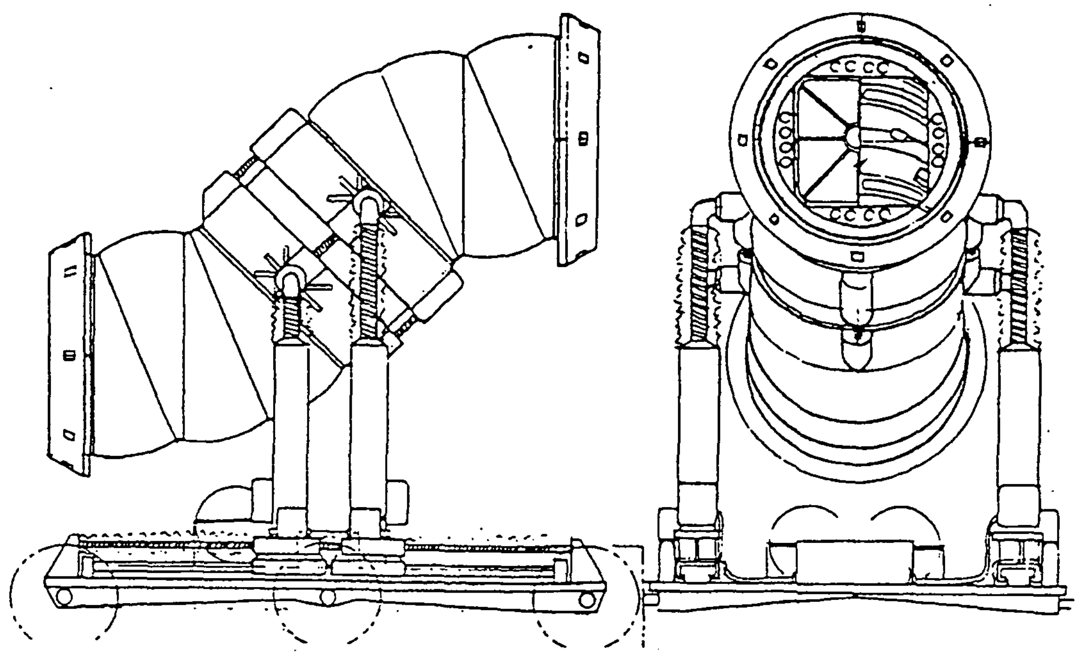

Figure 10.6. Pressuized Crew Transfer Tunnel.

docking adapters/hatches at both ends of the tunnel, and will have a mass of approximately 3 metric tons. Like all of the components in the transportation system , the wheels of the pressurized cart will be independently powered so that it may be moved easily. The ends of the tunnel are flexible so that it will be able to mate with the unlevel hatches of the lander and the pressurized cart . The ends of the pressurized tunnel will be raised up and down using power screws and motors as shown in Fig. 10.6.

10.4 Modularity and Compatibility Considerations

In order to make repair and manufacture of the transportation system easier and more economical, the common parts of the Prime Mover , trailer, and pressurized cart should be standardized. Examples of some of the common parts are wheels, motors, cables, and hitches. If the common parts are the same, one or two spares of the same part could be taken along on a mission to repair a broken part on any vehicle. If the common parts are different, then a spare of each type must be taken along on a mission for backup. Three or four motors of different sizes would certainly take up more room than one or two motors of the same size. Identical parts would also make operations for a stockroom and manufacturing plant much easier also. Manufacturing several different parts would require adjustments to the machines that produce the parts, and possibly even several distinct machines. If the parts are the same for each vehicle, manufacturing plants would only require one set-up. Stockrooms would not need to know which part went to what vehicle. If someone came in and wanted a tire, the stockroom clerk would simply go over to the tire bin and pull out any of the tires in the bin to give to the person, because one size fits all. Manufacturing parts common to each vehicle to be identical makes more sense than manufacturing different parts to perform the same basic function.

11. Control Systems

The control systems described in the following report are designed to monitor the three umbilical units which connect the lander to the servicer, and the mobile robotic arm. All of the control systems designed for this project are based on microprocessor technology, which allows for easier "task-specific" manipulation. The control system design was based on a series of criteria which included the ability to withstand the extreme fluctuations in radiation, temperature, and other factors unique to the lunar environment. All controls are based on a closed-loop system wherein sensors and encoders are used to feedback signals for control purposes. These components are also used for the identification of an unknown or changing process[19].

11.1 Hydrogen Umbilical

The first of the two umbilicals connecting the servicer to the lander will be used for the transportation of cryogenic liquid hydrogen. The same umbilical will be used to transport the hydrogen boil-off from the lander to the servicer, and return the reliquified hydrogen to the lander. The pressure, temperature, and flow rate inside the umbilical must be closely and accurately monitored. The control system designed for this purpose consists of a central control unit located on board the servicer, and sensors distributed within the umbilical.

11.1.1 Control Unit

The central control unit is based upon the Model SP3000 Control Unit produced by Sponsler Inc[20]. This unit is self-diagnostic, self-contained (modular), and at .22 M x .31 M x .19 M it is relatively compact. The unit is capable of monitoring and adjusting the temperature, pressure and flow rate within the umbilical, and can itself withstand temperatures ranging from 0 to 120 degrees Celsius. An automatic "shut-off" feature is provided in the event of a punctured or torn umbilical which would be exhibiting an unusual decrease in pressure and increase in flow rate.

11.1.2 Sensors

There are three types of sensors involved in the hydrogen umbilical control system. The first of these, the Sponsler series GR-200 Germanium Resistance Temperature Sensor, will be employed in the measurement of the temperature of the hydrogen in its liquid state[21]. It is capable of accurately sensing temperatures ranging between .05 K to 100 K, and is particularly sensitive and extremely low temperatures. The second type of sensor in this system, the Sponsler Series RF-800 Rhodium-Iron Resistance Temperature sensor, will be used to measure the temperature of the hydrogen in its gaseous state[21]. Its operational sensing range is between 1 K and 500 K and gives a nearly linear response at temperatures above 100 K. The last of the three hydrogen sensors is the Sponsler Precision Turbine Flow Meter, which will be used to measure the flow rate of the hydrogen in the umbilical[22]. The unit has unlimited pressure capabilities, and can withstand temperatures ranging from 16 K

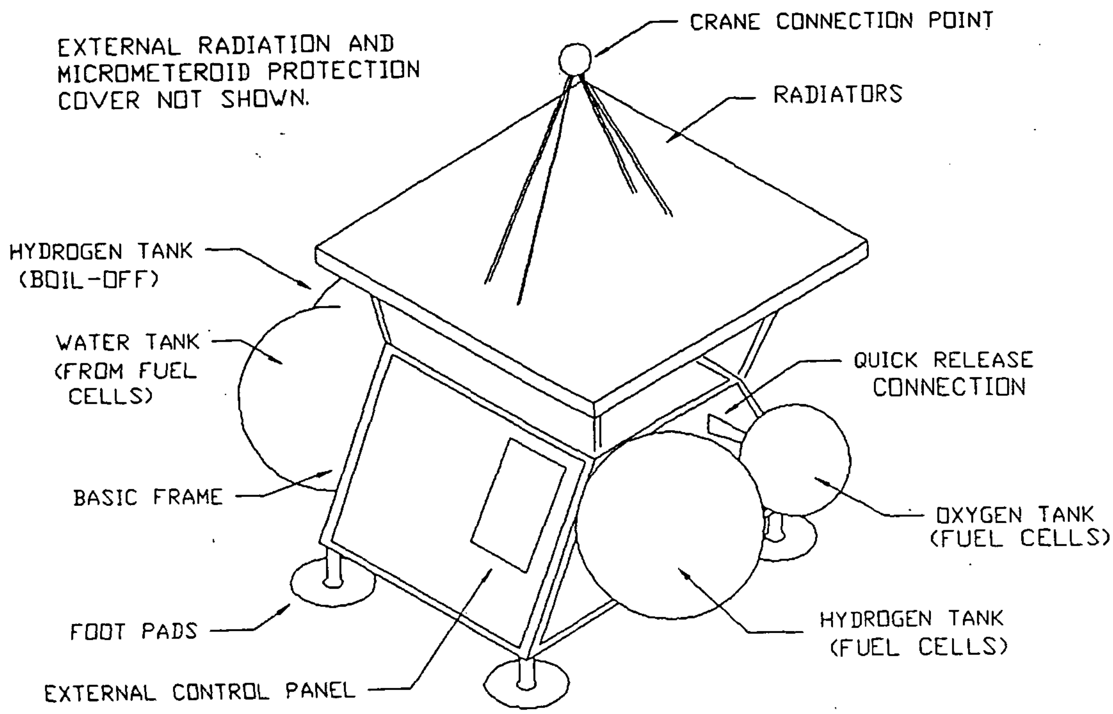

Figure 11.1. Servicer.

to 811K. All of the sensors chosen for this system are highly stable and operate with a high percent of accuracy.

11.2 Power Umbilical

The second umbilical connecting the servicer to the lander is used to transmit power from the servicer to the lander. A control system is required to monitor and adjust power levels as the requirements of the lander change periodically. The central control unit is operated by a TTL Microprocessor- Based System. The sensors, which are located on the umbilical and the servicer will consist of two volt meters. These meters will monitor the amount of power that is remaining in the lander's power supply, the amount of power remaining in the servicer's power supply, and the amount of charge that is exchanged between the two. The sensors will relay this information to the microprocessor/computer which will adjust the amount of power exchanged between the lander and servicer based on changing requirements. See Fig. 11.1 .

11.3 Robotic Arm

The control system for the robotic arm will monitor the angle of each arm piece, the strain on each joint and arm, the object-arm incident pressure, and the environment surrounding the arm. The control system is also able to interrupt the activity of the arm in order to prevent damage to the arm itself or harm to personnel and equipment. The arm is primarily operated by a teleoperational system, however the process of aligning the end-effector with a target object will be carried out by an automated control system. See Fig. 11.2 .

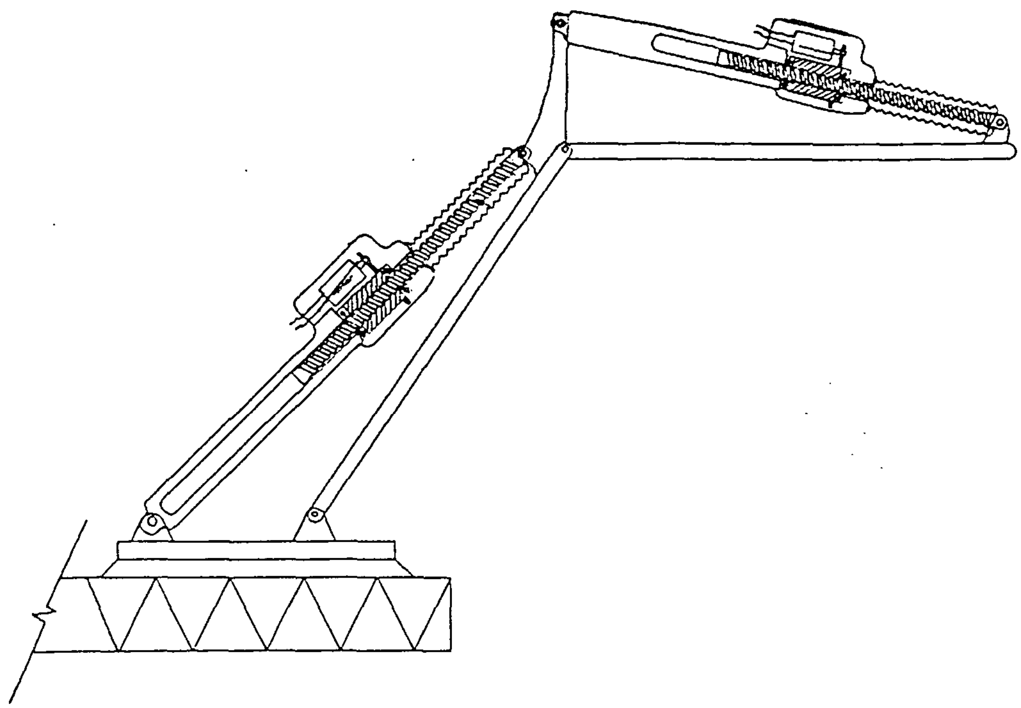

Figure 11.2. Robotic Arm.

11.3.1 Lifting and Rotation

The angle sensors will monitor the amount of angle with respect to each arm piece[23]. Each arm piece is connected to a joint which will be equipped with an optic sensor to record this data. The angle of each arm piece will be relayed to a microprocessor/computer which will be able to determine the exact location of each arm piece relative to the arm base. The strain sensors will measure the amount of linear and torque strain on each joint and arm. The Strain-Gauge Load Cell uses a resistance gauge to monitor the amount of strain being exerted on each piece[24]. If excessive strain is measured during an attempt by the arm to lift an object, the central control unit will not initiate the lift. In the event of a lift, the unit will monitor the angular velocity strain exerted on the arm and will adjust the speed of rotation should this strain prove excessive. The actions of the control unit will be visibly relayed to a human operator via a series of colored lights. The Load Cells will be sealed within the arm to help increase longevity of the sensor.

11.3.2 Object-Arm Incident Pressure

The pressure gauge will measure the amount of object-arm incident pressure. Initially, a spring loaded parallel plate capacitor was considered. This capacitor would not give an accurate measurement, however, because the capacitance is a function of both the area of the plates and the distance of separation which were not great in this case. A better choice, therefore, was a dual resistive sensor which would measure a change in resistance when it moves closer to an object. Target objects, which are themselves designed to be handled by the arm, will be equipped with resistive strips which are detected by these sensors. The sensors on the arm will enable it to automatically align its end-effector with respect to the grip on the object. At this point in the lifting process, the arm reverts to its teleoperational mode.

11.3.3 Environment Monitoring

The immediate area surrounding the robotic arm (ie. its environment) will be closely monitored by a system of pressure pad sensors. These sensors will be capable of detecting the presence of any body foreign to the environment and will automatically interrupt the activity of the arm as a result. This feature was incorporated into the design of the arm primarily as a safety measure. The interruption of activity will prevent harm to personnel and/or equipment which find themselves in the path of the robotic arm, and will allow the area to be cleared before resuming any activity. Although the interruption of activity will occur immediately upon the application of pressure to the sensor pad, the arm will resume its task only after a timed delay following the release of the pressure. This delay should allow sufficient time for a complete clearing of the area surrounding the operating arm.

11.3.3.1 Pressure Pads

The pressure pads used in the environment monitoring system will be modelled after the CVP3032 Safety Mat produced by Tapeswitch Corporation[25]. These pads are based upon pliable "ribbon switch" technology which will allow them to be stored and transported on spools. These spools can easily be designed as an integral part of the whole robotic arm transport and operation vehicle. These pads are equipped with Fail-Safe Monitoring Circuits which respond to power failures, open or shorted leads, or internal breakage. The system automatically runs a series of self diagnostic tests upon start-up and shut-down to ensure the proper working condition of each zone in the system. See Fig. 11.3 .

11.3.3.2 Conclusion

The controls systems play an integral role in the operation of any automated system, and are generally relied upon to accurately monitor and maintain safe and optimal working conditions. Advances are being made in control system technology on a regular basis, and eventually an entire integrated system may be available on a single chip. This project was designed on the basis of available technology, with the understanding that a future developments and discoveries will enhance and change it considerably.

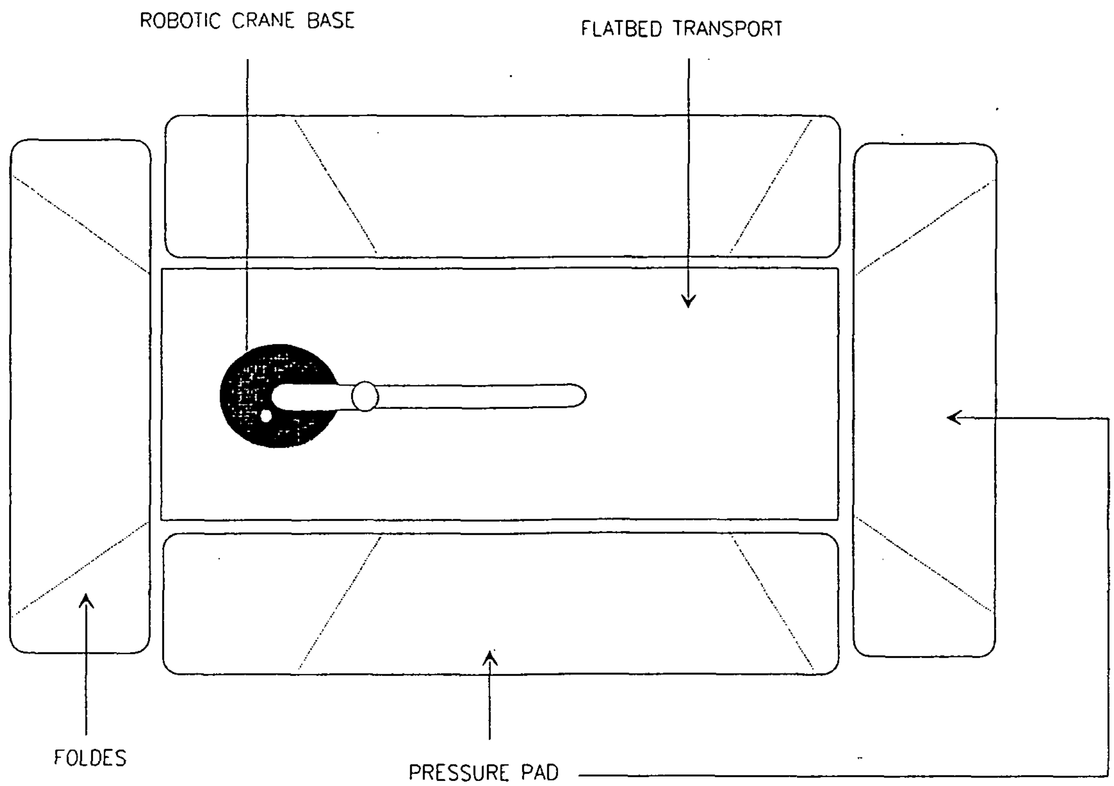

Figure 11.3. Robotic Transport Setup.

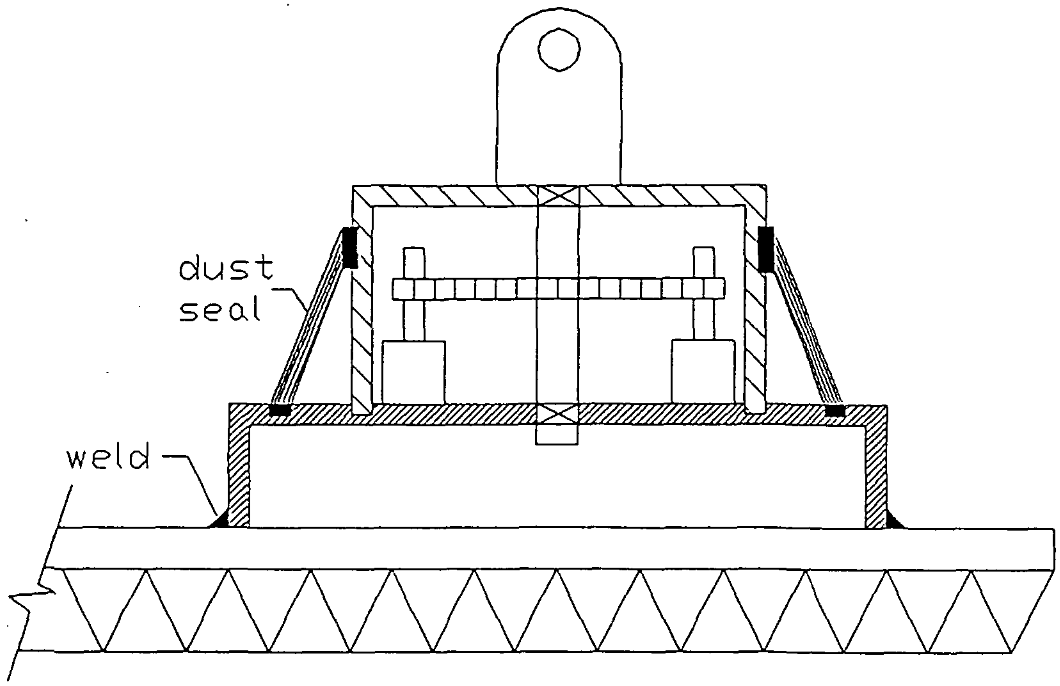

Figure 12.1. Illustration of Robotic Arm Connection to Cart.

12. Telerobotics/Teleoperations

In lunar base operations, the ability to load and unload supplies and cargo as well as well as other heavy equipment will be essential to sustaining lunar base activity. Also, moving of servicing equipment to support lunar lander operations will be required. To facilitate these requirements, a robotics crane will be used that has maximum maneuverability and lifting capacity.

For the loading and unloading of cargo and equipment, which will equal approximately 20 Mt, a robotic arm with the a lifting capability of 3 Mt will be used. The robotic arm will be transported using a motorized tractor; referred to as the prime mover (PM) which will pull a cart which contains the robotic arm and servicer. Upon reaching the landing pad, the robotic arm will be used to unload the servicer from the cart and place it near the lunar lander. In its second mode, the arm will remove the 3 Mt segmented cargo and place it on the cart for transport to the lunar base.

12.1 Mounting and Degrees of Freedom

The robotic arm will consist of a permanent mounting to the cart and will have a revolving base capable of rotating 360° as shown in Fig. 12.1 . The arm will utilize two sections which are connected using revolute joints. Shown in Fig. 12.2 , are exploded views of the arm segments showing the planes of rotation for each segment and the corresponding coordinate

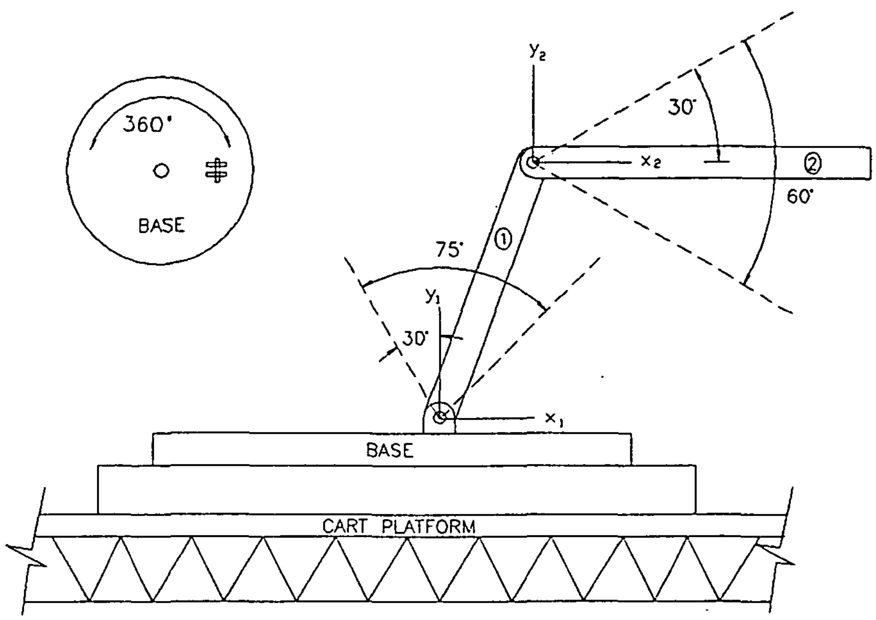

Figure 12.2. Planer Rotation and Coordinate System for Each Arm Section.

system used. The base mounted section, subsequently referred to as section (1), will be mounted to the revolving base using a revolute joint. Section (1) will have only one degree of freedom in the x-y plane and a total rotation of 75°. This will include both positive and negative rotation from the vertical. Section (2) which is connected to section (1) by means of a revolute joint, will be limited to a total rotation of 60°, and for this section rotation will consist of plus and minus angular displacements from the horizontal. Connected to this second section, will be the end effector which will be interchangeable for different robotic arm requirements. Three such different end effectors that will be utilized for these specific applications include a hook, a clamping jaw, and a ball and socket type end effector.

For adequate cargo removal, it is required that the arm have a height capacity of 9.0 meters (29.50 ft.) and a minimal reach in the x-direction of 10 meters (32.80 ft.). To determine the appropriate lengths of the arm sections, it is first required to know the dimension of the robotic arm cart and the positioning of the cart from the lunar lander.

For the cart, the dimensions will be 3.0 meters by 8.0 meters with a platform elevation of 1.5 meters from the lunar surface. For the positioning of the cart next to the lunar lander, it was estimated that a maximum distance of 3.0 meters was adequate. At this distance, it was determined that interference of the cart or robotic arm would not be a problem and that in the event of a mishap such as accidental dropping of the cargo or arm failure the lunar lander would be far enough away that damage to the lander would not result. Also, just in case an accident does occur, the cart containing the robotic arm will be positioned on the opposite side of the lander away from the servicer, umbilicals and the EVA personnel. Using

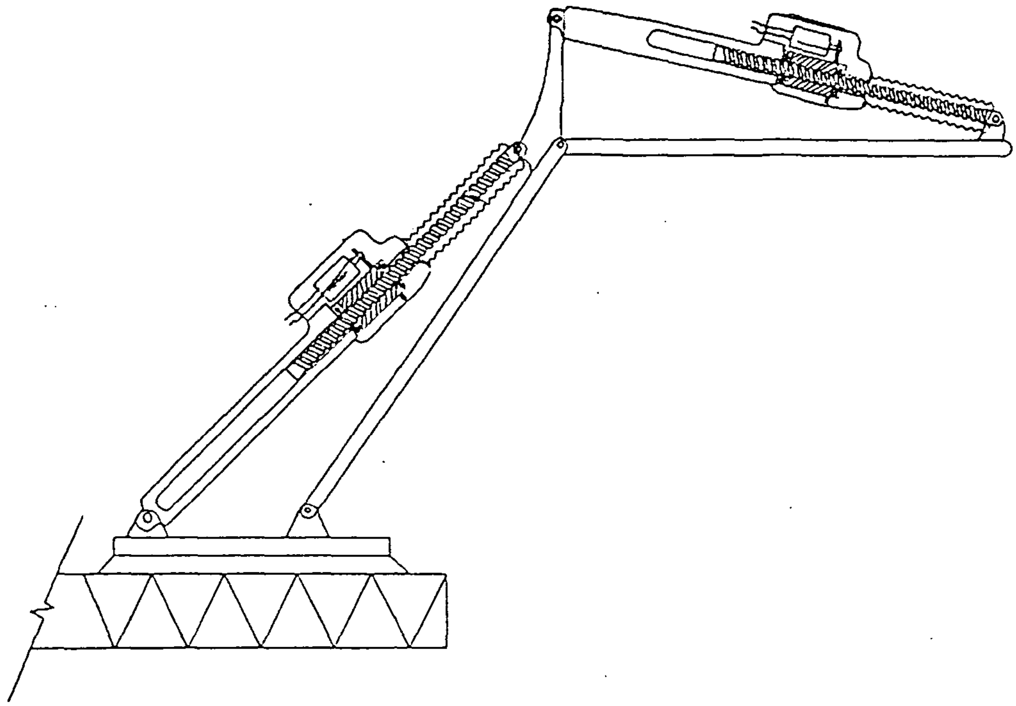

Figure 12.3. Simplified View of Robotic Arm Assembly.

the specification listed above for the cart and the positioning, It was determined that the two sections for the arm must have a maximum length of 6.0 meters (19.70 ft.). Utilizing these section lengths and the angular rotations previously mentioned, the maximum horizontal reach in the x-direction will be 10.0 meters. This maximum reach occurs when section (1) is at a full negative rotation (assuming counter clockwise is positive) of 45° and section (2) is at horizontal. Note, when section (1) is at full rotation, section (2) will have to be rotated to maintain horizontal positioning. For the y-direction, maximum reach is obtained when section (1) remains vertical and section (2) is rotated to a full positive angular displacement of 30° from horizontal. At these positions, a maximum y-direction reach of 9.0 meters (29.50 ft) is obtained. Shown in Fig. 12.3 , is a simplified illustration of the robotic arm showing its sections and the drive system used.

12.2 Structural Evaluation

The structural criteria for the arm is crucial in that the arm will be subjected not only to heavy load requirements, but it will also be exposed to environmental hazards that could also cause failure to the entire arm or one of its components. Also, due to the high cost of sending cargo into space; a cost which exceeds $10,000 per kg, weight is a critical aspect that can not be ignored in the structural design.

12.2.1 Arm Loading

Having the capability to carry increased amounts of weight makes the robotic arm more feasible for lunar operation. This is due to the environmental conditions that are constantly changing on the lunar surface, it is mandatory that expedient loading and unloading take place.

For the robotic arm, determination of the maximum load carrying capability was determine by using the servicer weight and cart carrying capacity as a guideline. The cart carrying capacity was derived using a program ;shown in the appendix, that calculates the maximum weight that the soil can withstand and still maintain adequate cart traction and movement. Utilizing this data gives a maximum arm carrying mass of 3000 kg (3Mt). Shown in the appendix, are free-body diagrams that illustrate the location of the loading and the reactions that occur. As shown in these drawings, the major carrying weight of the arm is concentrated at the end of section (2). Due to the long length of section (1), this weight tends to cause large moments about the revolute joint in section (1) and the base. Because of these large moments, the sizing of the arm segments is critical so as to prevent failure or large deflections from occurring. To accommodate for these high loadings, moments, and NASA requirements for large safety margins, a factor of safety (F.S.) of five will be used for the arm sections.

12.2.2 Material Selection

As mentioned previously, weight is a critical aspect of designs for lunar applications, which is why most space application type designs use aluminum as their selected material. For the robotic arm, aluminum will also be employed as the major material. For the two arm sections, base, pins, and power screws heat treated aluminum 2024- T6 will be used. It has a specific weight of approximately $2800 kg/m^3$, coefficient of thermal expansion of $23.0x10^{-6}/^{\circ}C$ and a yield strength of 441 MPa in tension and 246 MPA in shear. For the other non-load carrying components such as the drive system case 6061-T6 heat treated aluminum will be utilized. 6061-T6 has a smaller specific weight and approximately the same thermal expansion therefore helping to reduce the overall weight of the structure [26].

12.2.3 Bending and Stresses

As stated previously, accuracy is critical to the overall arm performance. To help maintain the accuracy within the 5 cm desired, it is crucial that deflection due to bending of the arm be minimized or prevented. As shown in the free-body diagram mentioned previously, the arm is subjected to two external forces. They are the one that comes from the load being lifted and the one from the power screw drive system. Note, the load from the power screw can act in both directions depending on which way the arm section is being moved. The largest load results when the power screw is pushing against the arm moving it in a clockwise position. Therefore, determination of the arm sizes is done using these values and the bending moment equation (1) [27] which is

$$\sigma = \frac{Mc}{I} \tag{1}$$

where

σ_y = yield stress
M = maximum bending moment
c = point of maximum stress from neutral axis
I = moment of inertia

and the deflection which is determined from [27]

$$y_{max} = \frac{M_{max} l^2}{2EI} \tag{2}$$

Using equation (1), preliminary values for the arm section are determined and then these values are substituted into equation (2) and the corresponding deflection values determined. As stated earlier, a (F.S) of five is being used and therefore minimal deflections that result (below 5 cm) in the analysis can be neglected.

The shape of the cross-section for the arm was determined using four criteria consider the most critical. The criteria listed in order of most important to least is,

1) weight
2) strength
3) cost of production
4) usefulness

Based on these demands, four different cross-sections were examined and they are solid-round, hollow-round, solid-rectangular and hollow-rectangular. Applying the above criteria the hollow round section was chosen. Compared to the solid sections, the hollow sections can be designed to carry the same loading by increasing their size. Even with an increase in their size the hollow section still maintains less weight than the solid section.

From the loadings and knowledge of the cross-section area, it is possible to rearrange equation (1) to determine the appropriate size of the two sections,

$$\sigma_y = \frac{4Mc}{\pi(R^4 - r^4)} \tag{3}$$

Replacing c by R and setting the section thickness to 37.5 mm (1.5 in.) in equation (3) you get

$$\sigma_y = \frac{4MR}{\pi(R^4 - (R - 1.5)^4} \tag{4}$$

Solving for R in equation (4), results in the sections being .20 meters (8 in) in diameter. Utilizing these dimensions and inserting into the deflection equation (2) for a cantilever beam which for a hollow round section is written as

$$y_{max} = \frac{2Ml^2}{\pi(R^4 - r^4)} \tag{5}$$

allows the deflection of the sections to be determined. For the arm structure, a maximum deflection of 2.65 cm was obtained and it occurs at the loaded end of section (2).

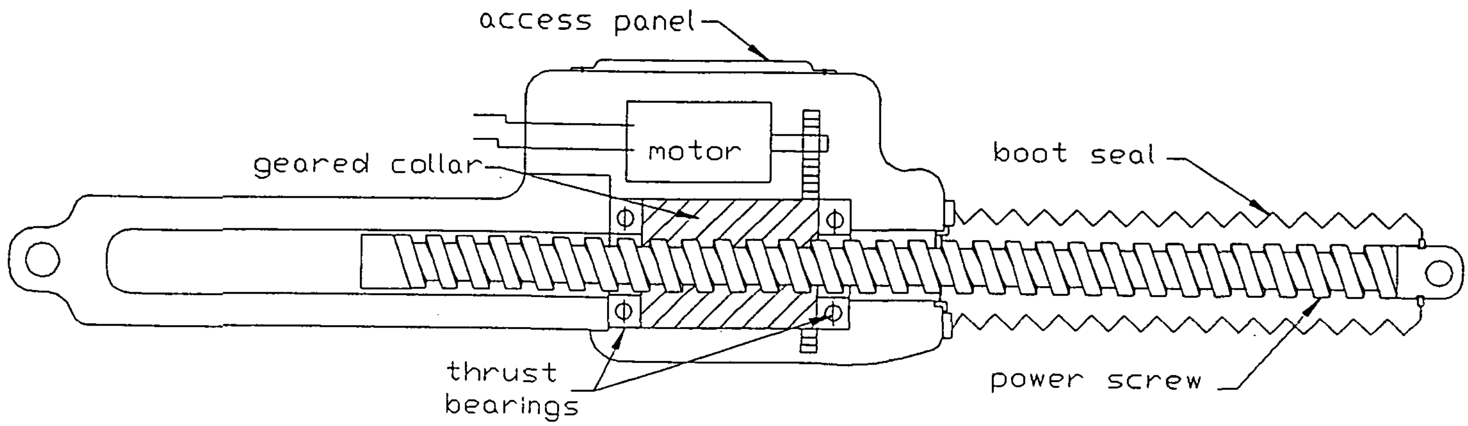

Figure 12.4. Exploded View of Power Screw Drive System.

12.3 Drive System

To facilitate movement of the individual sections, a drive system must be used that can maintain steady progressive movement at the constant speed desired. This system must be capable of operation under severe lunar conditions without having to have frequent maintenance and repairs. It should be self contained so that all major components are enclosed or protected by some means. And finally should be designed simple enough so that repairs can be achieved if breakdown was to occur during operation.

Shown in Fig. 12.4 , is an illustration of the drive system to be used. As seen in the drawing, the movement of each arm section is accomplished by using a power screw. The power screw is driven by a motor that is linked to the power screw by a gear and threaded collar. This collar is geared in such a way as to transfer the torque to the power screw which in turn provides the necessary force required to move the arm to the desire angular rotation. To counteract the thrust force that will be exerted on the power screw, two thrust bearing will be placed on the power screw assembly. As shown in the drawing, these bearings will be placed on the front and back side of the geared collar.

Protection of the power screw is necessary because lunar dust can get into the power screw and then into the drive system case to cause damage to the bearing, motor, or power screw itself. To prevent this from occurring, an accordion type boot seal will be placed over the power screw and attached to both the case and the end of the power screw. At the case end, the boot will have a fixed attachment that can be removed if boot replacement is necessary. At the other end, the boot will have a collar attached that will rotate in a groove placed in the power screw. For secondary protection, a seal will be placed at the end of the case to stop any smaller particles that might slip under the boot. For further protection, the entire power screw drive system will be wrapped in several layers of aluminum coated mylar. These layers will not only help protect against dust particles but will prevent the drive system components from being subject to the adverse thermal conditions that the arm could see during operation. To maintain easy movement of the drive system parts, lubrication will be required to prevent lockup from occurring during arm operation. Lubrication will consist of coating all moving parts with PTFE resins. PTFE resins have been used in the shuttle program over the past decade and have proven to have superior lubricating capabilities in space operation [28]

12.3.1 Power Screw Sizing

Since the responsibility of the power screw is to transmit the torque from the motors to the arm sections, it is critical that they be strong enough as to prevent failure under loaded conditions. As a safe guard, a factor of safety of ten for the power screws will be used for their sizing. The power screw will encounter both torsion and tensile loading during its operation and therefore must be sized to withstand both of these conditions. Determination of the size under torsion is determined from [29]

$$\tau_{max} = \frac{Tc}{J} \tag{6}$$

and for axial load by

$$\sigma_y = \frac{P}{A} \tag{7}$$

Since the resulting stress caused by the tensile force is so small compare to the torsional stress, it can be neglected. Therefore, equation (6) for the torsional stress can be used to determine the power screw size. Substituting the appropriate values into equation (6) results in a power screw size of 0.04 meters (1.50 in.).

The torque required to raise the cargo is determined from the load required to balance the cargo weight as shown in the free-body diagram in the appendix. This torque can then be calculated using the equation for power screws in reference [27]

$$T = \frac{Fd_m}{2}\frac{(l + \pi u d_m)}{(\pi d_m - ul)} + \frac{Fu_c d}{2} \tag{8}$$

where

d_m = mean diameter of power screw
u = coefficient of friction
l = lead, which equals the pitch times thread type

12.4 Power

Power to the robotic arm will be obtained from the prime mover which will have is own independent power source. Power will be transfer to the arm by utilizing power cables connected between the two systems. In the event of main power failure, a backup system will take over and continue to supply power to the robotic arm. The backup system will consist of nickel-hydrogen batteries which will be located on the prime mover. [30] Detailed discussion of the power systems will not be included in this section but will be explained in Chapter nine. Maximum power for the robotic arm will be in the range of 1.5 kW, but for most situations will be approximately 1.0 kW.

12.5 Thermal and Environmental Protection

Due to the extreme temperatures, solar radiation and micro- meteorite particles present on the lunar surface, it will be required that the robotic arm contain some type of protection to prevent thermal stresses, deterioration, and freeze up of the components. As mentioned earlier, lock up can be controlled by use of a lubricant on all the moving parts. But, this lubricant is not sufficient if the arm is left completely exposed to the environment. To protect against these conditions, several layers of a aluminum coated mylar will be wrapped around the arm structure [31]. The reflectivity of the aluminum coating will help stop the radiation waves from penetrating the arm and the mylar will help to maintain the arm material at a constant temperature [32], [33].

12.6 Future Generation Applications

The use of robotics will play a significant role in the exploration of the lunar surface. The robotics used in the early stages of the lunar exploration and base setup will be primarily be responsible for the handling of heavy cargo. In later generations, robotics will be completely responsible for lunar surface exploration without the need for EVA personnel The information gathered on the performance of the early moon based robotics will be critical in the advancement of superior self sustaining robotics that require less human interaction and support. Therefore, this second generation equipment will be used as a test bed for the development of future robotics.

13. Communications

13.1 Introduction

This communication system was designed to accurately provide various links between the lunar lander, servicer unit, robotic arm, EVA persons, and the lunar base. Since all of the aforementioned equipment will be located in the designated landing area, a line of sight method of transmission will be used. Through the use of video, voice and data transmission, an effective design was achieved.

13.2 Constraints

The communications fulfilled the necessary requirements that met the needs of the various types of transmission. In order for the teleoperator of the robotic arm to be able to see the environment in which he is operating, a video link was established between the robotic arm and the base. Another important link was the voice transmission, which allowed open communications between the EVA persons and the base. The data transmission provided many vital links across the landing area. It connected the servicer unit to the base, EVA persons to the base, and the robotic arm to the base.

13.3 Alternative Designs

Throughout the total design, the communication links were consistent; the initial frequencies of transmission were changed to better meet the requirements of the desired design. Initially, the video transmission was to be transmitted at a frequency of 5.5 MHz and voice transmission was accomplished by transmitting a 259.7 MHz frequency on a very high frequency (VHF) carrier. After speaking with experts in the field of communications, these frequencies were changed to produce a optimum communication system. In particular, they were changed to meet the requirements of the new antennas.

Originally, a whip antenna was located at the base for the communication link between EVA persons. And a beam antenna was the main antenna for the entire communication system. These antennas where found to be fragile, heavy and expensive in comparison to the chosen antenna type.

13.4 Final Design and Specifications

The final design of the communication system consists of video, voice and data transmission. The video transmission was achieved by using two video channels. The optimum frequencies were found to be 296.8 and 264.2 MHz. A bandwidth of 32.6 MHz, which is 11% of the maximum frequency is required for the chosen antennas. [34] These frequencies were chosen so that the highest possible resolution could be achieved. In order to improve the broadcasting capability, a carrier signal was added to aid in the detection and demodulation of the signals. Also knowing that video signals contain a large amount of low-frequency information, amplitude modulation was necessary for broadcasting. Two video cameras were located on the robotic arm to provide the teleoperator with a view of the environment of the arm. One wide angle camera was located high on the robotic arm and another camera with a telescopic lens was mounted at the end effector. Through these two cameras, the operator was able to see the entire area in which the robotic arm was operating in, plus a close up view of the location of the end effector.

The communication system for voice transmission was designed to operate much like a home intercom system, where one person is able to speak and everyone listens. Voice transmission was designed to operate from a sideband of the video frequency. The minimum operating frequency is 4 kHz which meets the designs requirements. When transmitting at VHF signal levels greater than -105 dBm, the best voice quality was relayed [7].

Data transmission will transmit using analog frequency modulated signals and will also be operating on a sideband of the video frequency of 296.8 MHz. The desired baud rate for the data signals was 56 Kbaud. A voltage to frequency (V/F) converter was required in order for the transmitter to send the signals to the receiver. Once the signals are received it will necessary to convert them back through a F/V converter. Also, digital signals are necessary for computer processing, therefore the signals must be converted by using an analog to digital (A/D) card. These digital signals will be used at both the lunar base and in the EVA suits. The heads-up display (HUD) will provide the EVA persons with critical information about human aspects, the servicer, the robotic arm, and prime mover. See Fig. 13.1

The antennas are an integral part of the communication design, not only must they be capable of withstanding the lunar environment, they must also be able to minimize the intermodulation distortion. The selected antennas are called microstrip antennas and are available through Antennas America, Inc. Their advantages over parabolic and whip antennas are: they are proven to work in space, they are pliable and can conform to any shape, they are capable of withstanding moderate impacts, they are lightweight, and are relatively inexpensive. In order to operate at the highest frequencies of 296.8 MHz, two microstrip antennas were located at each landing pad. Their dimensions are 0.61 m X 0.61 m X 0.03 m and weight less than ten pounds. Other operating parameters which apply to these antennas are: bandwidth of 32.6 MHz and gain of 40 dBm [34].

For the EVA person communication system, separate antennas where needed. Each EVA person needed a small whip antenna attached to the EVA backpack. Plus, a 2.44 m omnidirectional antenna was required at the lunar base to complete the EVA communication link [7].

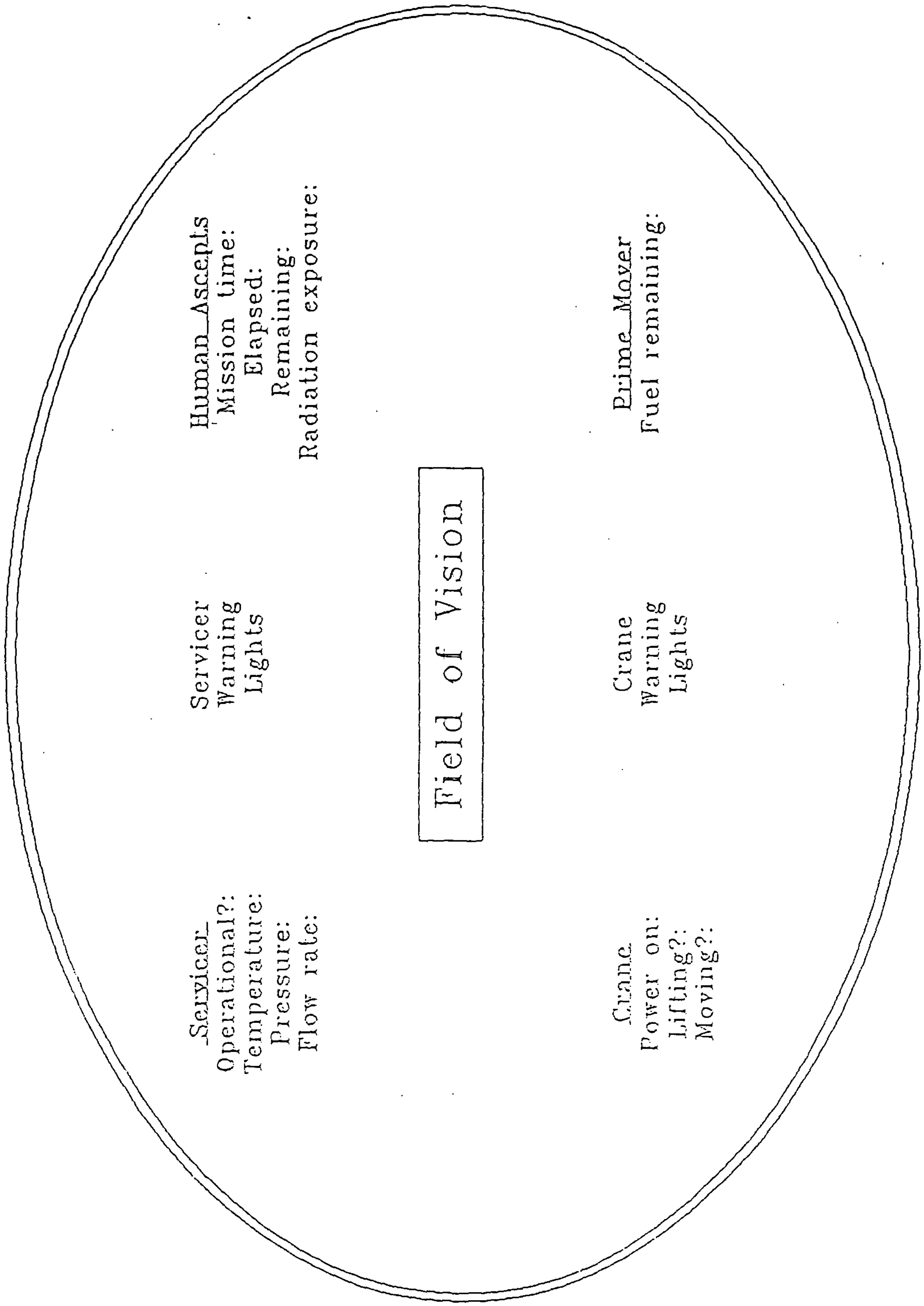

Figure 13.1. Heads Up Display Screen.

14. Navigation

14.1 Introduction

The automated navigation system was designed to accurately carry the lunar lander vehicle from lower lunar orbit to one of three possible landing pads on the lunar surface. This system was designed to relay directional information to the pilot via a digitally displayed coordinate system. Upon entering lower lunar orbit, the navigation system will guide the lander, using triangulation, to a predetermined landing pad on the lunar surface.

14.2 Constraints

The landing area was an integral part of the entire navigation system and was designed to accommodate triangulation. The base itself will be located in center of the landing area surrounded by three landing pads, each 1000 m from the base. See Fig. 14.1 . Triangulation navigation utilizes three beacon signals that are sent at periodic intervals from known locations on the landing area to the lander. These signals track the distance and direction of the orbiting lander, which allows the pilot to accurately determine his location with respect to the landing pads.

Personnel safety was also a primary factor in the design of the landing area. The triangular shape of the area allows the lander to approach any one of the landing pads without orbiting directly over the lunar base. In the case of an accident during a landing, the hazard of impact with the occupied base is therefore minimized.

14.3 Alternative Designs

The original navigation system was capable of determining and tracking the location, and measuring the attitude and acceleration of the lander. The initial design consisted of Very High Frequency Omni-Range (VOR) and Distance Measuring Equipment (DME)).

The VOR, which determines distance and direction, measures man-made radiation emitted by space-based transmitters. It operates in the 108 to 118 MHz band with the channels spaced 50 kHz apart. In particular, the Collins VHF Omnidirectional Range-Marker Beacon Receiver maximizes the bearing accuracy by minimizing the effects of temperature variation and aging. It implements 30 Hz band pass filters to achieve this accuracy. Also, it tracks ground frequency modulation. The electrical characteristics of this receiver are: power 115 V ac at 400 Hz, 30 VA, and a maximum power factor of 0.9. Its physical area is 3 MCU and has a weight of 4.4 kg [35].

The DME determines the distance by a standardized pulse-ranging system and operates in the 960 to 1215 MHz range. Specifically, the Collins DME-700/ARINC-709 was chosen. It has two power output devices to achieve 700 W nominal power output. When operating at L-band transmit channel frequency, it achieves its highest accuracy. The transmitter transmits at a frequency band of 1025 MHz and has a minimum power of 25 dB W. The receiver operates at a frequency band from 962 to 1213 MHz with a capability of up to 252 channels[36].

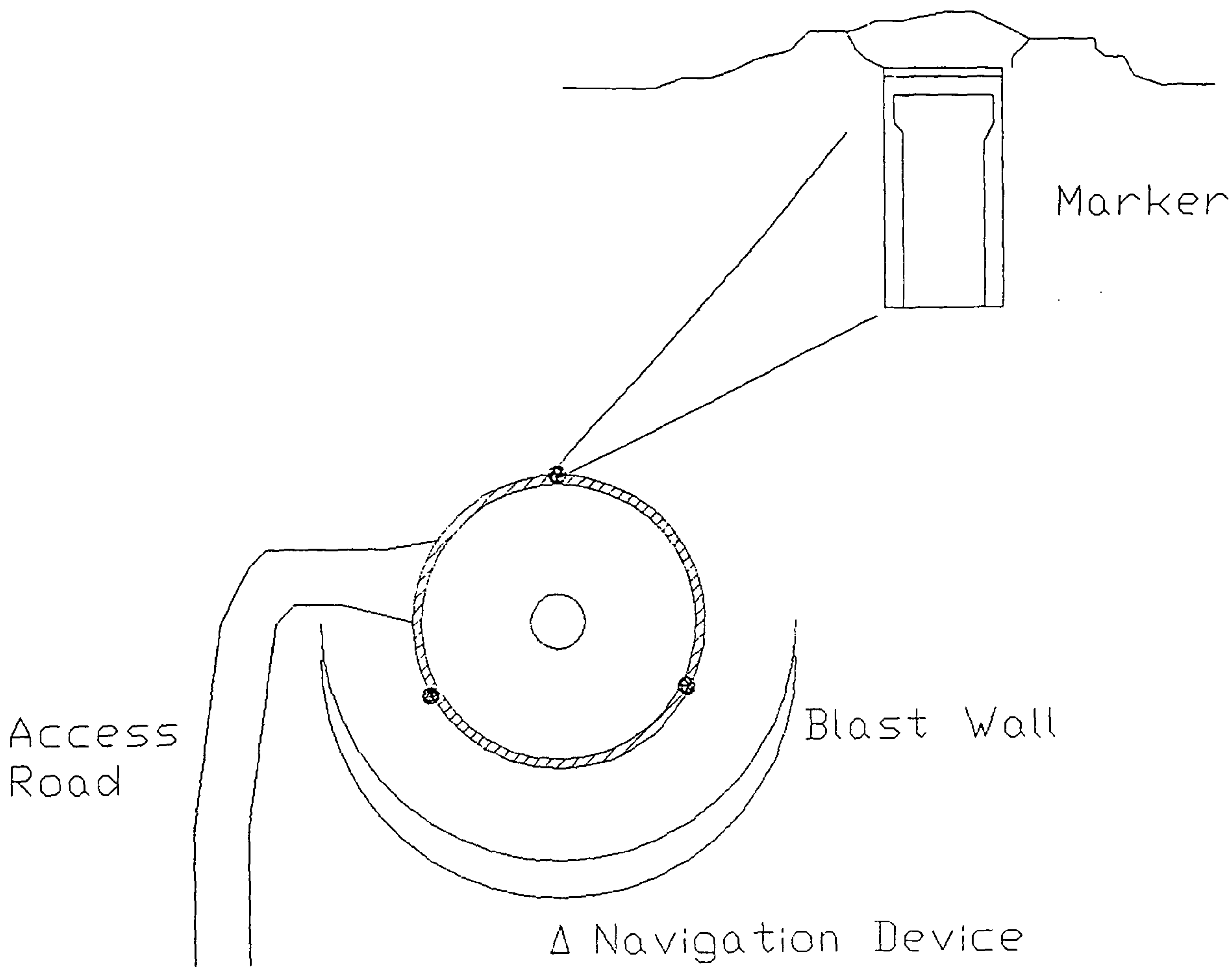

Figure 14.1. Landing Pad Layout.

Although this initial design was certainly feasible, it restricted the pilot to only one approach pattern per landing pad and more flexibility was needed.

14.4 Final Design and Specifications

The final design of the navigation system consists of both DME and MLS (Microwave Landing System). DME is capable of determining the distance from the lunar lander to the landing sites. It is a standardized pulse-ranging system that operates between the frequencies of 960 to 1215 MHz, with its channels spaced at a minimum of 1 MHz apart. Since the landing site consists of three separate landing pads, the DME will operate at three separate frequencies. Channel 1 (landing pad #1) will operate at 1030MHz, channel 2 (landing pad #2) will operate at 1085MHz, and channel 3 (landing pad #3) will operate at 1140 MHz. The DME responds to the periodic signals emitted from the ground beacons located at each landing pad.

The MLS determines the angular displacement and the attitude between the lander and the pads by transmitting two signals from the lunar surface to the lander. One signal determines the angular displacement of the lander with respect to the centerline of the landing pad. The other signal indicates the attitude of the lander. The most obvious advantages of an MLS over VOR/ILS, are the increased number of possible landing approaches, and its higher level of accuracy. In order to incorporate an MLS into this design, three discrete systems are required. There must be one MLS located at each landing pad, and each must operate at separate frequencies [35].

By using a DME in conjunction with three discrete MLS, this design maximizes accuracy, efficiency, and safety during the vehicle landing procedure.

15. Umbilical Cables

15.1 Introduction

To connect the lunar lander to the servicer, two main types of servicing umbilicals are required. The first umbilical is used to transfer that hydrogen which boils off in the lander's propellant tanks, and return said hydrogen to the lander after the servicer has reliquified it. The second cable is used for DC power transmission to the lander to maintain the onboard electronic systems. Both cables have similar features that differ little, such as the cable end connectors' external functions i.e. their ability to handle applied loads, their external shape, and their securing mechanism.

The restrictions on the internal operation of the hydrogen cable are much more diverse than those on the power cable. Thus, the design approach used on the hydrogen cable comprises the majority of this section's content.

15.2 Hydrogen Cable

The hydrogen cable(HC) is fundamentally composed of two components: the end-caps, and the cable in between, or main segment. There are several restrictions on the conceptual design of these components which will be discussed in detail in the following sections. Many of the restrictions and limitations were arrived by group discussions concerning interface issues. Since the umbilical cables are, in fact, the means of interfacing the servicer's and lander's systems, its overall design limitations are dominated by these issues.

15.2.1 Hydrogen Cable Main Segment: Vital Parameters

15.2.1.1 Physical Dimensions and Geometry

The maximum length of the cable was set at 10 meters. This value was influenced heavily by three criteria: The ability to maintain the hydrogen in a liquid form as it was transferred back to the lander during the refueling operation; the ability of the EVA person to handle the bulk of the cable; and the physical distance of the servicer from the lander.

To determine the external diameter of the main cable a simplified heat transfer model (HTM) was developed. From this model it was determined that the diameter should fall in the range of 7 cm to 10 cm. For diameters outside this range, the model indicates that it will be near impossible to avoid excessive heating of both the hydrogen gas and liquid in the cable. Also, if the cable was too small, there would not be enough room for the electronic sensors, projectile shielding, and thermal insulation to be contained inside the cable.

From the previously mentioned HTM, the internal dimensions of the cable were iteratively optimized according to the schematic shown in Fig. 15.1 .

The range of the liquid H_2 cable diameter was found to be between 0.5 cm and 1.5 cm. The gaseous H_2 conduit was more difficult to optimize because of its important role as a heat

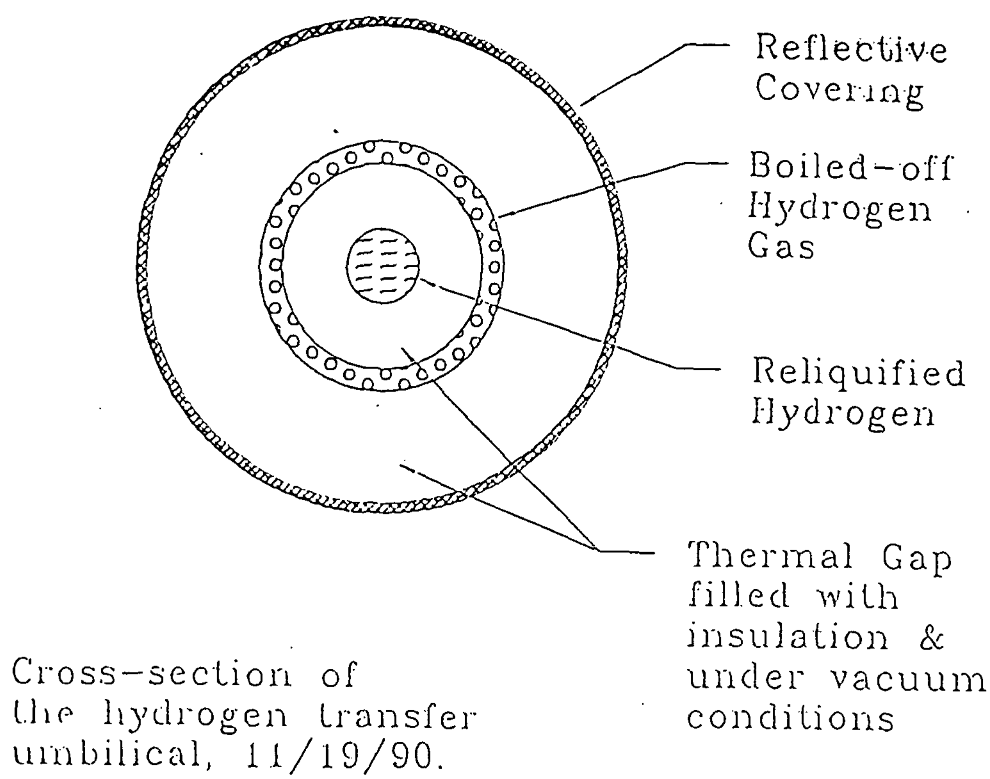

Figure 15.1. Hydrogen umbilical cross-section conceptual design used in the heat transfer analysis

sink for any energy that might contribute to a phase change in the liquid line. According to the results from the HTM, the thin ring annular gas duct should be located as close to the surface of the cable as possible, and be as thin as possible. This result was correct as far as the simplified model was concerned. However, in reality, the viscous flow losses would be dominant in such a thin duct. After personal consultation with an expert in the field, the range of the inner radius of the annular ring was decided to be between 2.0 cm to 2.75 cm. For the thickness, a range of .3 cm to .6 cm was decided upon.

15.2.1.2 Mass

For various and sundry reasons the maximum mass per length of the HC was decided at 17 kg/m. The main reason for this limitation was that of weight consideration involving physical effort of the crewman on the EVA. At this mass, the total weight experienced by the crewman would be approximately 280 N, or 63 lb_f.

15.2.1.3 Heat Transfer

The HTM used to analyze the cable indicates that the temperature rise of the gaseous H_2 would be approximately 30° K which results in an exit temperature of 50° K. The factors that contribute to this heating are accounted for in the HTM by the following methods.

Direct solar radiation was assumed to have an intensity of 1360 W/m^2 This radiation was assumed incident over the projected area of one half of the cylindrical surface of the cable. This is equivalent to the area of a rectangle of dimensions 0.07m x 10m.

Conduction from the surface regolith to the cable was assumed over a transfer area equivalent to one half of the cylindrical surface of the cable. This condition would exist if the cable were buried half way into the regolith and the sun was directly over head. This model was intended to represent a worst case scenario.

Internal convection of heat to the hydrogen flows was based on a constant heat flux model for fully developed, mixed laminar-turbulent pipe flow. Orifice regulators will be used in the annular duct to throttle the H_2 gas. In doing so the gas' pressure and temperature will be reduced with little change in the mean flow velocity. The reasoning behind this additional internal complexity, was based on the idea of minimizing the temperature gradient between the liquid and the gaseous hydrogen, and hence, minimizing the internal heat transfer to the liquid hydrogen. The placement of the orifices was not determined at the time of this writing. The required effect of the orifices was the only parameter considered in the HT calculations.

15.2.1.4 *Materials*

The selection of materials that could be used for this application were limited. Nearly all restrictions and limitations applied to the different materials could be grouped into two categories: mechanical and thermodynamic.

One of the most stringent mechanical restrictions applied to the HC materials was the ability to remain flexible at near-lunar-surface temperatures. The liquid H_2 section of the HC must satisfy this restriction at the normal boiling point of hydrogen (20° K). A second important limitation was the HC's ability to handle high internal pressures i.e. 1.4 MPa, or 200 psi. Basically, the cable shall have to be a flexible pressure vessel; difficult, but not impossible. The last major mechanical restriction is linked to the previous two. Fatigue resistance should be high enough so that a fatigue failure induced either by pressure variation or by physical flexing of the cable would be minimized. The external casing material of the cable should have a high impact resistance to projectiles. This resistance must be large enough to withstand the impact of 90% of the particles that have a reasonable chance of hitting the cable.

The main thermodynamic restriction on the materials that would be used for insulation was the physical property of a low thermal conductivity. The value used in the HTM for the cable is that of evacuated, layered aluminum foil and glass paper (75 layers), which is $1.7 \cdot 10^{-5}$ $W/m\,K$. The external casing of the HC should have a high reflectivity coefficient, (.7 or greater) so as to reduce the overall absorbtion of the solar radiation incident on the cable surface. This was the value used in the previously mentioned HTM.

15.2.1.5 *Flow and Flow Rates*

The hydrogen gas flow rate per day was decided, after group consultations, to be equivalent to two liters of liquid boiled to a gas state at saturation conditions. This was in part based on the boil off values for equivalent H_2 storage tanks produced commercially. Line pressure of the gas, at the cable exit, was set to be no more than 1.4 MPa, or 200 psi. This value was the result of calculations on the storage tank pressure of the servicer after 14 lunar days of continuous H_2 boil off from the lander.

The reliquified hydrogen has both a lower and an upper flow rate limit. At the upper end, turbulent friction losses and turbulent heat transfer dominate the calculations. At the lower boundry, the liquid spends too much time in the cable; it absorbes too much heat. Thus, flow rates outside this range will result in the liquid H_2 vaporizing before it reaches the lander.

15.2.1.6 Projectile Shielding

As stated before, the cable's main section should be able to withstand the impact of 90% of the cosmic particles that have a reasonable chance of hitting it. Thus, this restriction mainly limits the materials that may be used for the cable's casing.

15.2.1.7 Miscellaneous

The flexibility of the cable was defined by determining the smallest circular loop that the cable could be formed into without permanent damage. This "wrapping"of the cable would have to take place under operating conditions i.e. with liquid and gaseous H_2 flowing inside and the outside casing at lunar surface conditions. The diameter of this arc was decided to be not more than 2.0 m.

15.2.2 Hydrogen Cable End Caps: Vital Parameters and Their Restrictions

15.2.2.1 Physical Dimensions and Geometry

The dimesions that were necessary to the design of the hydrogen cable end cap (HCEC) had fewer restrictions than those of the main segment. However, the overall geometry of the HCEC had many restrictions that have to mentioned. Basically, the removal of, and sealing against intrusion by the lunar dust was the main consideration when analysing a geometrical configuration. An external configuration that was the result of discussions with experts in the commercial industry, and information contained in the lunar Apollo reports is shown in Fig. 15.2 .

15.2.2.2 External Operations

The external geometry of the HCEC has several functions to perform. First, it must correctly align the cable for proper interfacing to the lander and servicer. As shown in Fig. 15.2, this operation is performed by use of vertically elevated alignment fins. Second, it must not be bulky to handle, i.e. the tendancy to drop the HCEC should be minimized. Third, the geometry must be such as to minimize weight, while maximizing strength.

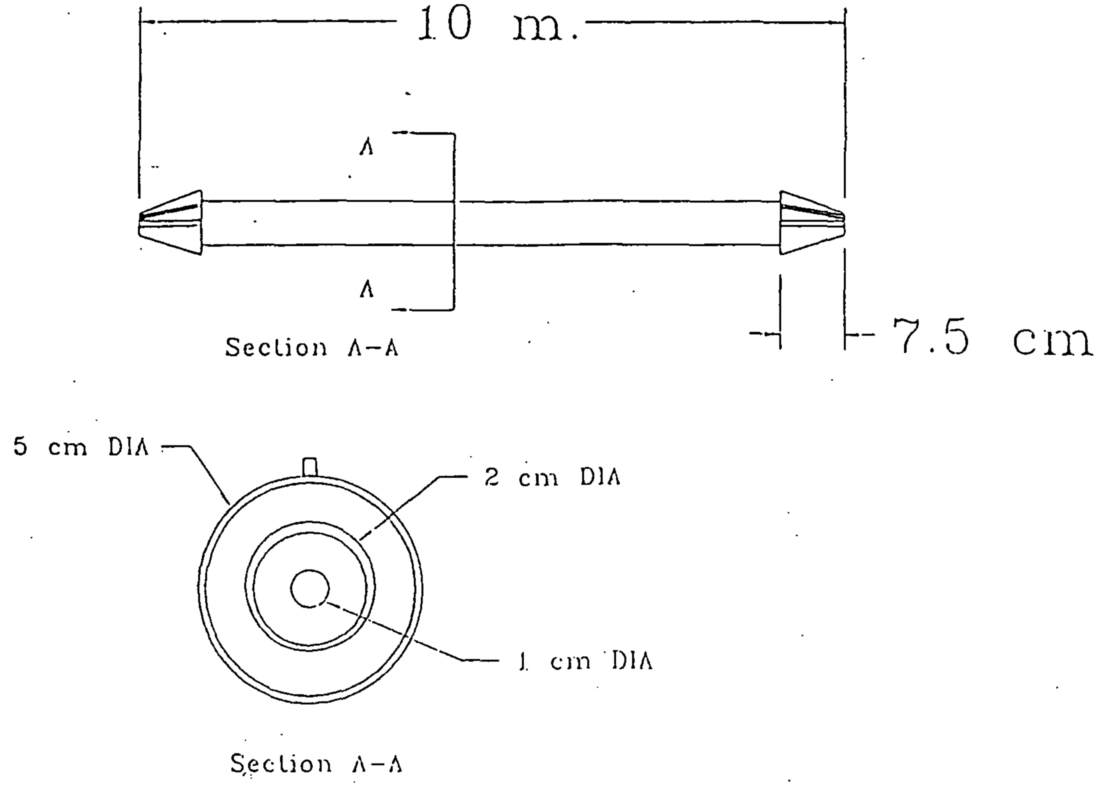

Figure 15.2. Hydrogen umbilical end cap conceptual design dimensions and geometry

15.2.2.3 Internal Operations

Three main internal operations must be performed by the HCEC. First, it must maintain a perfect seal against the entry of lunar dust particles into the hydrogen connectors. Second, the gaseous and liquid hydrogen

16. Power Systems

16.1 Introduction

Deciding on he type of power system to support the servicer and the prime mover was a critical step in the design process. The system should meet the necessary requirements, while accounting for other factors such as weight optimization, efficiency, reliability and cost. It is assumed that by the time that NASA has established a lunar base, the technology will be more advanced, more efficient, lighter, and will have a higher energy density and lower consumption rates. However, for this project, the design decisions and calculations were based on existing technology.

16.2 Constraints

The projected stay of the lunar lander on the moon's surface is a maximum of 180 days. The power system would, therefore, have to be long lasting because the servicer must power the lander during its stay at the base. The servicer will need to be lifted by a robotic arm and placed on the prime mover, which leads to limits on the weight requirements. These requirements are also limited for the prime mover; lightness is necessary because it has large power requirements and it has to be mobile. The weight should also be minimized due to the very high cost of transporting each kilogram of payload to the lunar surface. Because of the need for easy replacement by EVA personnel, the system should be modular as well as light weight.

16.2.1 Power Requirements

In choosing the power system, the power requirements for the servicer and prime mover were taken into consideration. The peak power requirement for the Hydrogen reliquification system of the servicer was 1.5 kiloWatts (kW) [37]. The lunar lander was provided with 1.0 kW of power. The electronics and the cooling pump on the servicer were 220 kW and 1kW, respectively [20]. The total power required by the servicer was assumed to be 3 kW, which would not be exceeded even at worst case performance.

The prime mover's power requirements were calculated for two different modes of operation due to its multifunctional uses. Calculations were based on the prime mover being used for surface grading, or for transporting the servicer and cargo to and from the base. For the prime mover, with no trailer connected, in grading mode, the peak power needed for acceleration on an incline was 23.2 kW, for incline only 18.4 kW, and it had an average power requirement of 10.0 kW. With the prime mover and the trailer travelling to the pad, the requirements were 10.4 KW and 5.0 kW for the peak acceleration and average power, respectively. The robotic arm which was mounted on the trailer will use the prime mover's power source since the prime mover will be stationary when the arm is in operation. The main motor driving the arm needs 1.0 kW [27]of power, with the controls needing 220 W [20]and pressure pads 120W [25].

16.3 Alternate Designs

Before deciding on a specific power system to meet the requirements, many options were researched. The power system selected for the Lunar Lander Ground Support System consists of Proton Exchange Membrane (PEM) fuel cells. These cells are also called Solid Polymer Electrolyte (SPE) fuel cells by United Technologies Hamilton Standard. This type of fuel cell was chosen because of its many advantages over alkaline fuel cells, batteries and photovoltaic cells. The primary disadvantage of alkaline fuel cells is the lack of long term catalyst stability which leads to performance degradation over time.[38] The PEM fuel cell has a long stable useful lifetime with almost no degradation of performance. The cell consists of a proton exchange membrane used as the sole, solid electrolyte. This simplifies the problem of electrolyte containment in alkaline cells, while keeping a competitive efficiency level.

In comparison to batteries, the fuel cells have a much smaller, and therefore more practical, weight requirement. To have batteries as the primary power source would be much more expensive[39] and they would be much harder to handle by someone in an EVA suit. The SPE fuel cell has also been proven successful in many space power applications, for example, the Gemini missions used a SPE fuel cell which was developed by General Electric Co[38]. See Fig. 16.1 [40]. The use of photovoltaics as the main power system was also discounted; to generate the power required for each system, the sheet of photo cells would have to be too large to fit on either the servicer or prime mover.

The backup system was originally going to be dual redundant fuel cells, with a main system and a backup system. Each system would have had its own plumbing to guard against failure. The main and backup systems would share the cryogenic storage tanks, the heat exchanger and radiator. For prevention of prolonged disuse and to provide maintenance time, the two systems would have been used alternately. Initial start up of the fuel cells was to be provided by a small Nickel-Hydrogen battery, with the total start up sequence taking less than one minute.[40]After further review, it was found that a battery was not needed for start up, instead it would be used in place of dual redundant fuel cells as the backup system.

16.4 PEM Fuel Cells

The construction of the PEM fuel cell is simplified and leads to low cost, high yield with consistent performance, and high reliability. The mean time between failure (MTBF) for the cell is over 5000 hours. The system has proven to be greater than 99.9% successful for a 300 hour mission. The lifetime of each cell has been shown to be greater than 60,000 hours. The slight performance degradation is less than one microvolt per hour.[40]

16.4.1 Operational Features

PEM fuel cells were selected for their many advantageous features. They have a long, stable, useful lifetime with almost no degradation of performance over long periods of time. The reactants used in the fuel cell stacks are hydrogen and oxygen which will be stored as cryogenic liquids. Storing the reactants as cryogenic fluids rather than as pressurized gases

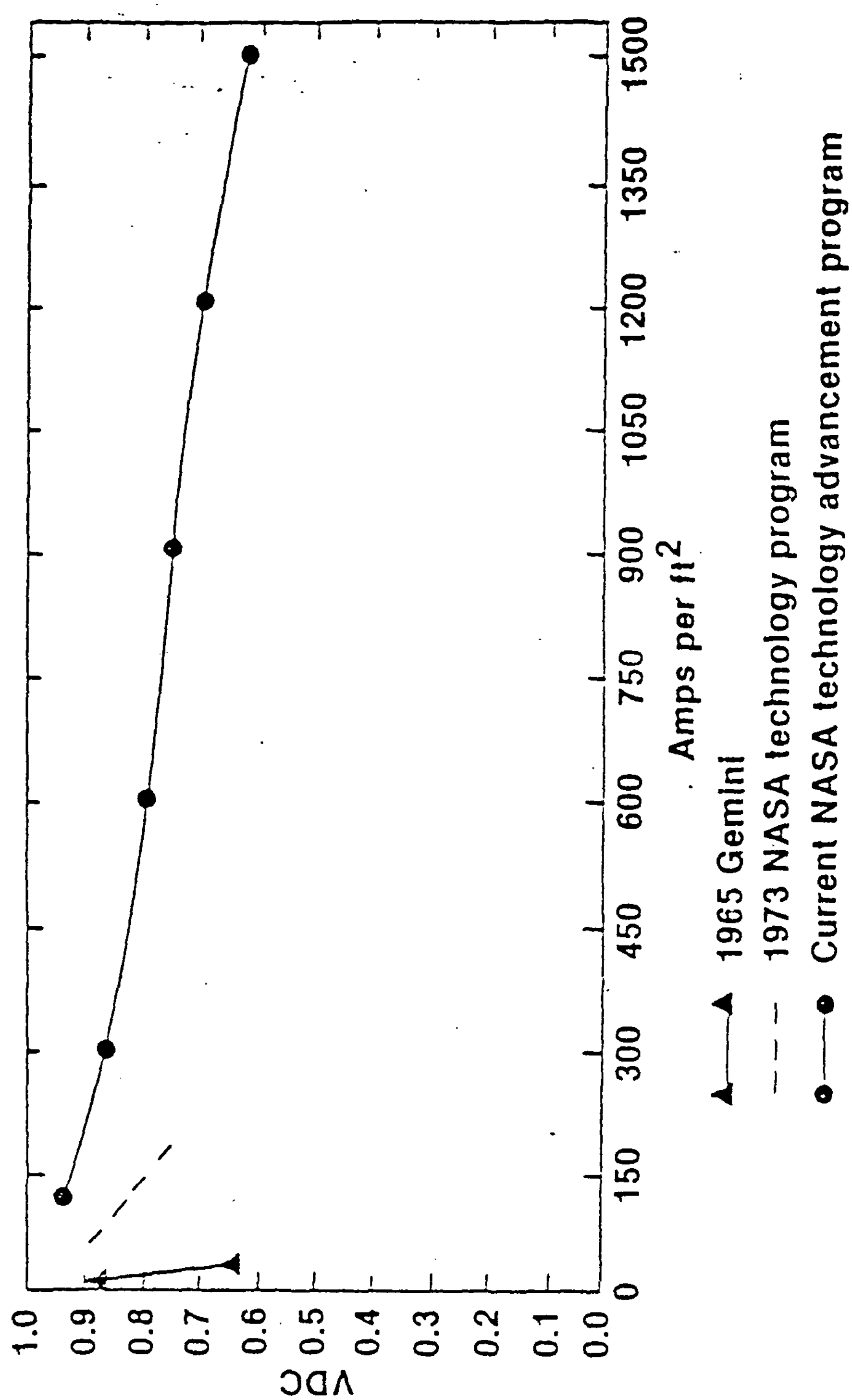

Figure 16.1. Performance Growth in H_2/O_2 Fuel Cells.

will reduce the size, weight and meteoroid vulnerability of the storage tanks.[41] We have assumed that the lunar base will provide a supply of liquid hydrogen and oxygen for the fuel cells. The reactants will be heated when they leave the tanks and vaporized before entering the stacks. Water will be a byproduct of the reactants; it is cooled and then stored as a liquid in a separate tank.

16.4.2 Elecrtolyte Features

Nafion, which is made by DuPont, will be the electrolyte used in the PEM fuel cells; it is a solid sheet of sulfonated fluoropolymer which is similar to teflon. The membrane is approximately 10mm thick, and when saturated with water, it serves as an excellent ionic conductor[42]. This solid membrane acts as a barrier separating the two reactants which increases the safety factor. See Fig. 16.2 . The membrane has been tested to 5000 psid without electrolyte blow-through[42]. The PEM fuel cells are capable of reaching instantaneous full load and can be cycled from open circuit to any electrical load and back as rapidly and as often as necessary[40]. This is due to demand fed reactants, passive product water control, and no delay time.

16.5 Ergenics Power Systems, Inc.

The specific fuel cells used for the power system of the servicer and the prime mover are from Ergenics Power Systems, Inc. These are standardized fuel cells which can be ordered directly from the company. The advantage of using existing fuel cells is that actual numbers can be obtained for the necessary calculations. These fuel cell units are modular and are placed in an enclosed container. These individual units include everything needed for operation except the cryogenic reactants for which there are hook up valves. The units have been successfully tested for the zero gravity environment. The small systems are air cooled, which would not be successful on the lunar surface due to the lack of atmosphere. Therefore, we will assume that we will be using a larger system which will be water cooled. [43] See Fig. 16.3 .

16.5.1 Operating Parameters

The specific parameters for the 1kW standardized fuel cell available from Ergenics were used for the project. The fuel cell operates at a rated power of 1000 W and a peak power of 1600 W. The peak power required for the servicer is 3 kW with 24 kW for the prime mover. The servicer will need three units to reach its requirement, while the prime mover will require 15 of the fuel cells. The fuel cells will run at a projected efficiency of 80%. The cell's rated voltage is greater than or equal to 24 volts DC. It can therefore operate at the standard space voltage of 28 V DC. The operating temperatures are from -40 to 48.8 C with a maximum stack temperature of 70 C. [44] The reactant consumption rate is 40 and 320 grams per kilowatt-hour for hydrogen and oxygen, respectively. Product water is produced from the reactants at a rate of 360 grams per kilowatt-hour. These consumption and production rates were used to determine the tank sizes for the liquid hydrogen, liquid oxygen, and product water.

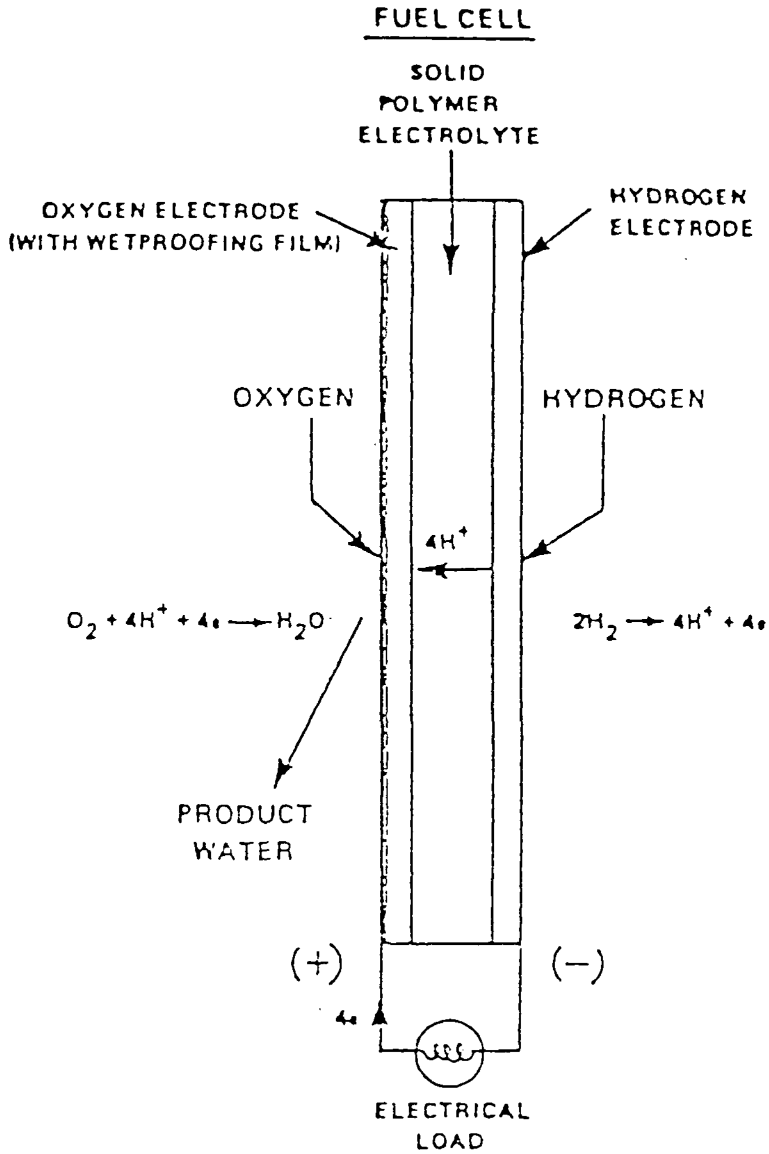

Figure 16.2. Electrochemical process of a single cell.

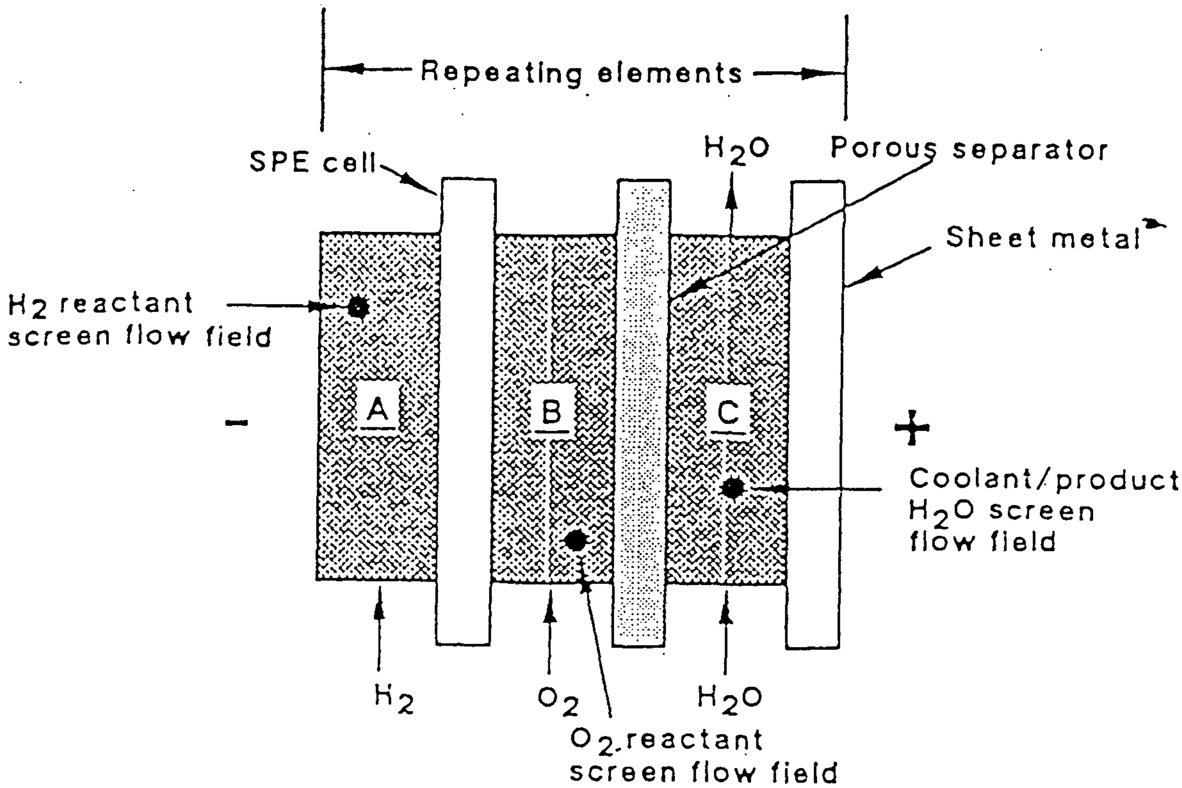

Figure 16.3. Cut away view of the cooling process.

16.5.2 Physical Parameters

Each fuel cell unit is portable and can be easily carried by EVA personnel. All components necessary for the cell other than the reactants are contained in the unit and enclosed in a suitcase-type outer covering. See Fig. 16.4 for a general schematic of the enclosed components of the fuel cell. The weight of the the individual units is 20 kg. Its dimensions are 79 cm in length, 33 cm in height, and 41 cm in width. See Fig. 16.5 to see pictures of the actual units.[44]

16.5.3 Features

Many operational and safety features are built in to the fuel cell units. It has an automatic, turn key start up (i.e. on/off switch). The emergency shutdown features include over temperature shutdown, high/low reactant pressure shutdown, and emergency shutdown annunciator lights. The cell voltage is monitored, reactant pressure is regulated, and there is an over pressure relief valve. The units operate with passive product water removal, as well as passive, internal reactant prehumidification.[44]

16.6 Back Up System

It was necessary to have a back up system for the servicer and the prime mover in case of an emergency situation or failure of the main system. This system could also be used as supplemental power for peak loads. Nickel- Hydrogen batteries were chosen to power the

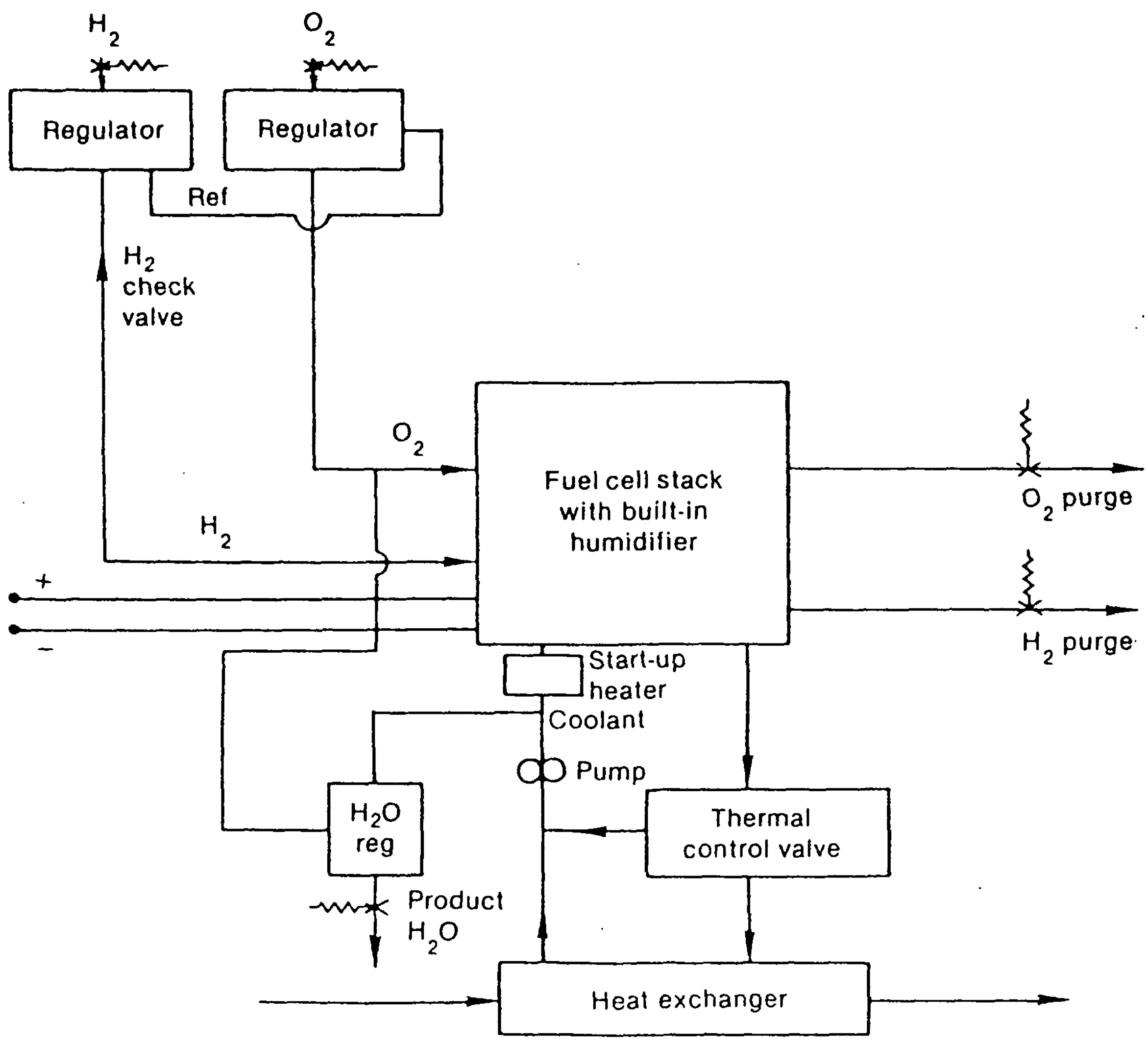

Figure 16.4. Generic SPE (or PEM) Fuel Cell Schematic.

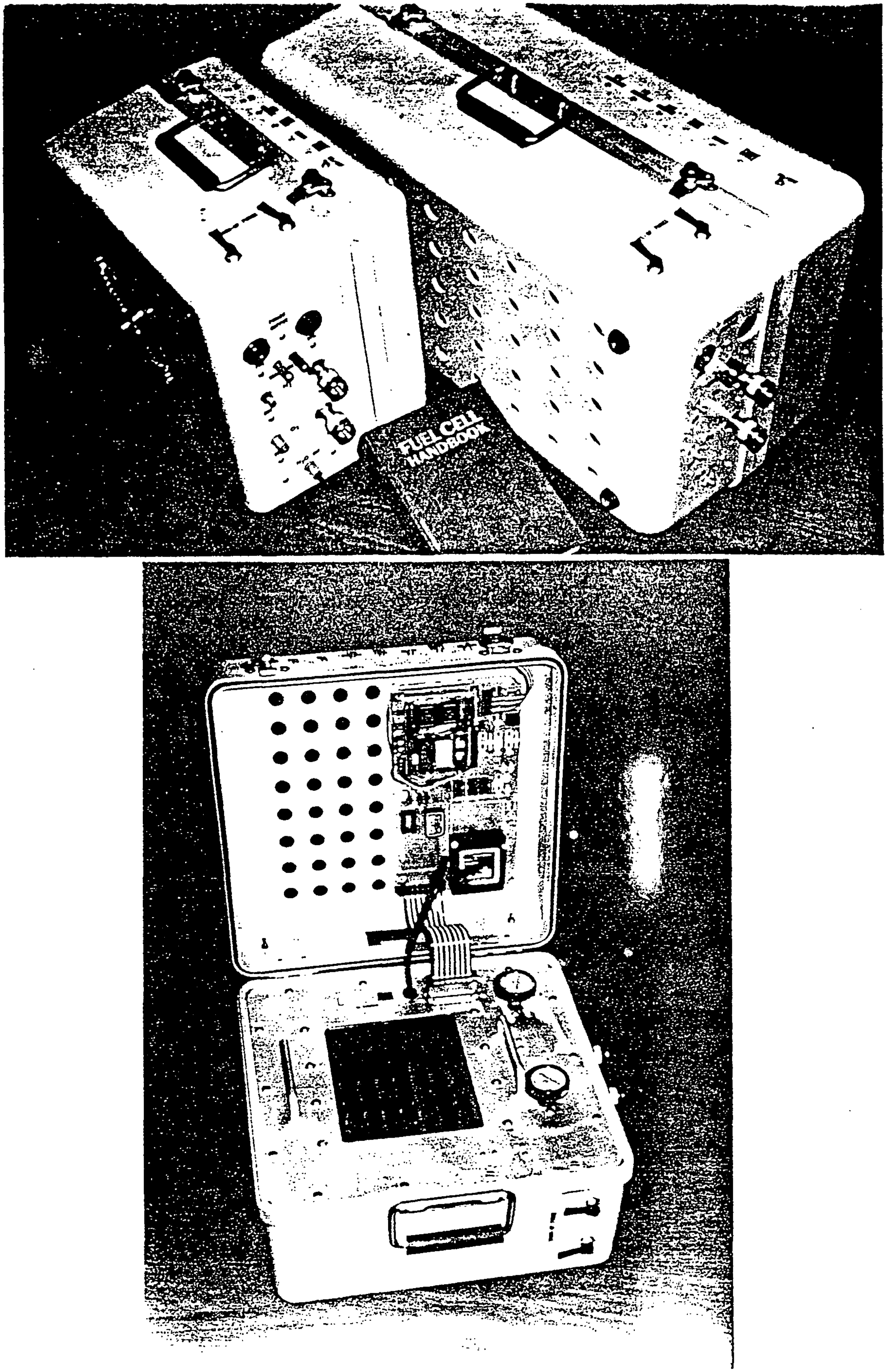

Figure 16.5. Ergenics Power Systems, Inc PEM Fuel Cell Units.

backup system over other types of batteries due to its fast recharge/discharge capability. This battery has a specific density of 50 Whr/kg with an efficiency of 82% [30].

16.7 Conclusion

The power system designed for the servicer and the prime mover was based on the power requirements and constraints established by the original project abstract. PEM fuel cells were the optimal design choice due to the advantages in weight, reliability, and cost. Ergenics Power Systems, Inc. provided the information on specific fuel cells which represent today's most advanced portable power equipment with high energy density and silent operation.[43]Ergenics Power Systems, Inc. has been working on new hydride hydrogen storage technology to replace liquid or gaseous hydrogen as a reactant for the fuel cells. Currently, this new development is only considered useful for small portable devices with maximum power ratings of 2 kW. However, for many applications in the future, this technology should prove to be far superior to storing hydrogen as a pressurized gas or as a cryogenic liquid. Lower volume, faster recharge and significantly longer projected cycle life are among its expected advantages. [45]

17. Propulsion System

One of the main functions of the lunar lander servicer being designed will be to support the propulsion system on board the lunar lander. The servicing vehicle will be necessary to maintain the needed amount of fuel on board the lunar lander during the 180 day stay.

17.1 Lander Tank Design

The design of the lunar lander propulsion system is taken directly from an already existing lander design. The system is based on an Eagle designed lunar lander [2]. A complete description of the Eagle lunar lander is given in the introduction of this report. The system will consist of two large propellant storage tanks. The first tank will house the liquid hydrogen reactant and the second tank will house the liquid oxygen. Both of these reactants are stored as cryogenic propellants. The actual design of the lander tanks will not be considered. Only the size of the tanks will be used. The tank material and insulation will be taken as already existing for optimum use.

17.2 Reliquification System

The main function of the servicer is to enable the lunar lander to rest on the lunar surface for a period of up to 180 days. After 180 days the lunar lander must be capable of launching itself back into lower lunar orbit. This will require maintenance of a certain amount of propellant reactants to power the lander off the lunar surface. Since the propellants are stored as cryogenic fluids and they are existing in an environment with an extremely high daily temperature, the propellant tanks will experience a certain amount of propellant boil-off per day. The boiled-off fluids can be reliquified and pumped back onto the tanks on board the lander. The servicer will provide this necessary service using a simple refrigeration system and a large storage tank.

17.2.1 Liquid Hydrogen Boil-off

An equation was found in an Eagle engineering report that will calculate the percent rate of propellant boil-off. The equation is as follows:

$$R\% = \frac{F}{\rho H r}$$

Where F is the heat flux encountered, ρ is the density of the propellant, H is the heat of vaporization of the propellant, and r is the tank radius [46].

In order to calculate the heat flux encountered by the stored propellant, the insulation thickness and insulation thermal conduction coefficient must be known. These values were not obtained for the Eagle designed lander tanks. Therefore this equation was not used to calculate the exact boil-off rate of the liquid propellants.

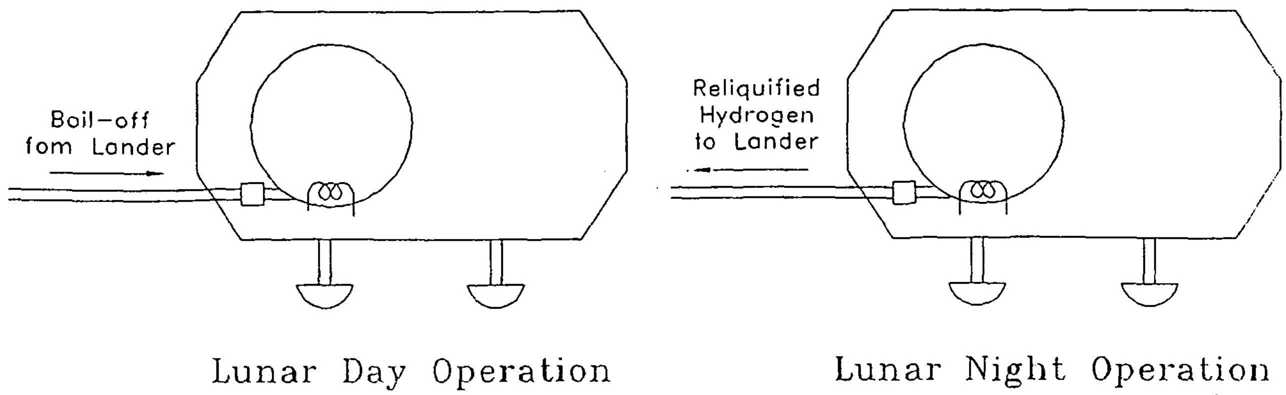

Figure 17.1. Basic Boil-off Scenario.

However, a company that manufactures cryogenic storage tanks was contacted to obtain a reasonable rate of boil-off. A Cryofab specification list provided a boil-off rate of 1 liter per day for liquid hydrogen storage tanks [47]. Using a factor of safety of 2, 2 liters per day will be boiled-off.

Due to the very high density of the liquid oxygen, approximately 15 times greater than liquid hydrogen, the boiled-off oxygen will not present a problem in the 180 day stay. The lander would only boil-off 24 liters of liquid oxygen in 180 days as compared to 360 liters of liquid hydrogen. Therefore, the liquid hydrogen will require the reliquification system.

The hydrogen will only boil-off during the period known as the lunar day. The lunar day is equivalent to approximately 14 earth days. During this 14 day period, the hydrogen will be allowed to freely boil-off and flow into a storage tank on board the servicer. Once the lunar night sets in, creating very low temperatures, the gaseous hydrogen can be reliquified by a simple refrigeration system. With the help of a small pump, the reliquified hydrogen will be pumped back into the lander tanks. The same umbilical line will be used to collect the hydrogen boil-off and to refuel the lander tanks. This process of reliquification and feeding back on to the lander will take place during the lunar night. Due to the extremely low temperatures occurring during the lunar night, the liquid hydrogen can be pumped into the lander tanks without evaporating inside the umbilical line. The basic hydrogen boil-off scenario is shown in Fig. 17.1 .

17.2.2 System Design

The evaporated hydrogen gas will be collected from the lander tank using a single umbilical line attached directly to the pressure release valve on the lander tank. The 2 liters per day of fluid will be able to flow directly into the storage tank on board the servicer. As the days go by the pressure inside the servicer storage tank will increase and become greater than the assumed boil-off pressure of 0.0101MPa. This pressure rise will be due to the increase in temperature of the hydrogen gas. The first increase in temperature will occur as the gas passes through the umbilical into the storage tank. The temperature will rise from 20K to 50K at the entrance of the storage tank. The calculation of these temperatures is found in the umbilical chapter. A further increase in temperature will occur as the gas is allowed to freely expand in the storage tank. A small compressor must be used to pump the boil-off into the storage tank when this increase in pressure occurs. The maximum pressure encountered after the 14 days of storage will be 0.848MPa. The storage tank will be evacuated at first and fill continuously for the 14 day period. The small compressor used will require less than 1W of power to pump the boil-off into the storage tank.

After the gas has entered the storage tank, it will heat up and expand due to the solar flux acting on the surface of the storage tank. The temperature inside the vessel will increase to 63K and the pressure will rise to 0.730MPa. These values were obtained using the computer program found in Appendix I. Using the above computer program an optimum storage tank diameter of 1.0m was decided on. This tank diameter will easily store the needed amount of boil-off while minimizing the pressure and temperature increase. The tank will be attached to the servicer and will be made of two concentric vessels. The spherical shaped vessels will each be made of aluminum [48]. The outside vessel will be aluminum 7079-T6 and the inner vessel will be aluminum 2024-T3. The reason for this material selection is later discussed in the fuel system tank design section. The hot gas is now ready to be reliquified using a refrigerator.

The refrigerator must be able to reject the heat necessary to completely reliquify the gas. This heat will include the energy required to drop the gas temperature from 63K to 20K and the energy required to condense the gas into a liquid. The solar flux will not contribute to the internal energy of the system because the reliquification will take place during the lunar night. These values add up to a total of 5.0W of necessary rejected heat. This value is also obtained using the computer program found in Appendix I.

The equation used to calculate the total energy required for reliquification is

$$Q = mH_{vap} + mC_p\Delta T$$

The total mass of hydrogen gas after 14 days is 1.984kg, the heat of vaporization of hydrogen is 445.87KJ/kg, the specific heat is 11.00 KJ/kgK, and the initial and final temperatures are 63K and 20K, respectively [49].

The two refrigeration systems studied were the the Brayton refrigeration cycle and a two-stage, closed-loop helium refrigeration system.

The first system, the Brayton refrigeration cycle, employs a working fluid circulating in a closed-loop cycle containing two compressors, two heat exchangers, and a turbine [50]. The only adequate working fluid would be helium. The low critical temperature of helium(5K) allows the hydrogen to be cooled to the necessary temperature of 20K. This system would work in the lunar environment. The cooling capability of the Brayton cycle is related to the pressure levels inside the close-looped system. The design of such a system is not necessary when an equivalent refrigerating system can be purchased off the shelf.

The final option is the "off-the-shelf" refrigeration system that has already been designed for a number of uses. This is the option selected for the servicer reliquification system. The system selected is a CTI- Cryogenic Cryodyne Refrigeration System. This system will provide an adequate cooling capacity without a high level of input power. The exact model selected is the Cryodyne Model 350 CP. The refrigeration capacity is 5 watts at 20K. This cooling capacity is more than adequate for the 0.8W required. The refrigerator unit weighs 10.5kg and the compressor unit weighs 58kg. A typical cool-down rate to 20K is 50 minutes. For each additional .45kg mass the cool-down time increases by 25 minutes. Each compressor in the compressing unit will require 1.5kW of power [51]. The entire reliquification process will be completed in approximately 96 hours.

Once the boil-off has been completely reliquified, the liquid hydrogen will be pumped back into the lander tanks for reuse. This process will use the same pump installed for the boil-off. The refueling process will take 8 hours to complete. The flow rate required for this 8 hour period is 0.00417 liters/min.

The entire reliquification and pumping system will require approximately 1.5kW for an elapsed time of 104 hours. This entire process will take place during the lunar night. A simple conceptual representation of the boil-off storage tank design is shown in Fig. 17.2 .

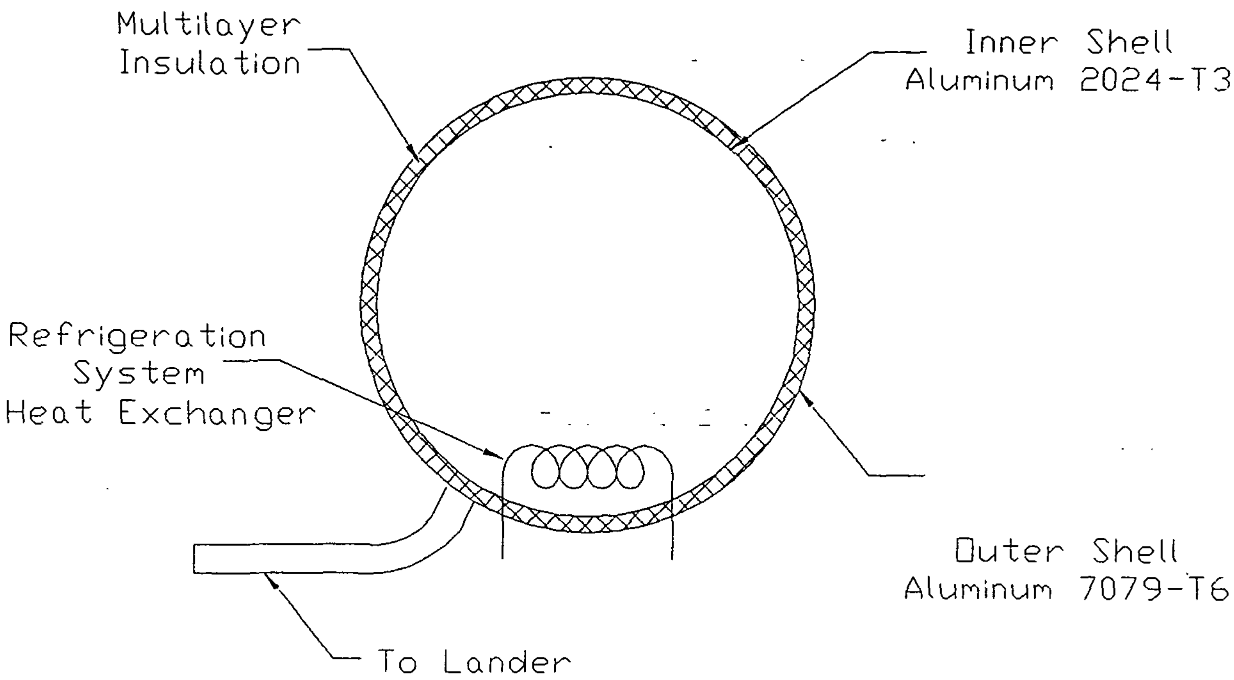

Figure 17.2. Hydrogen Boil-off Storage Tank.

18. Fuel Systems

The two different fuel systems designed include the fuel system on board the prime-mover and the fuel system on board the lander servicer. Each system will require separate design of fuel tank sizes and insulation. The fuel systems in each case are liquid hydrogen/liquid oxygen fuel cells [44].

18.1 Servicer Fuel System

The design of the fuel system on board the servicer involves the tank dimensions and tank insulation for the cryogenic fuels. These fuels will be used to run the fuel cells that supply the power needed by the servicer and the lander. A complete description of the fuel cells selected is found in the chapter on Power Systems. The design of a third tank is required to store the water product supplied by the fuel cells.

18.1.1 Cryogenic Fuel Tanks

The two cryogenic fuel tanks will store liquid hydrogen and liquid oxygen to be used by the power supplying fuel cells. Each tank will be housed inside the micrometeorite/radiation protection shields surrounding the servicer. This protection will greatly reduce the lost fuel due to boil-off. The fuel tanks will be simply attached to the servicer to allow for easy access and exchange. The attachment design is discussed in the servicer maintenance chapter.

18.1.1.1 Tank Material

The material selected for for use in the design of the fuel tanks was aluminum. Aluminum has very high strength with a very low density. Titanium was also looked into but the density of titanium is 1.6 times as great as the density of aluminum[50]. The greatest constraint in the tank design is the weight of the necessary material. Since these tanks will be shipped via the shuttle to the lunar surface, they must be manufactured as light as possible. The cost to send 1kg of cargo to the moon is $10000.00. The cost to manufacture the tanks in aluminum or titanium is small in comparison to this cost.

Three different types of aluminum were studied: aluminum 6061-T6, aluminum 2024-T3, and aluminum 7079-T6 [50]. Selected physical properties of these three aluminum alloys are found in Table 18.1 .

For the inner vessel, aluminum 2024-T3 was selected. The selection was made due to the low thermal expansion coefficient of this aluminum type. The inner vessel will house the cryogenic fuel at the extremely low temperature of 20K. Aluminum 2024-T3 has the lowest thermal expansion coefficient of the three types studied. The inner vessel will encounter pressures up to 0.73MPa. Using the following equation, the tank thickness was calculated as being 0.942cm (Appendix H).

$$\sigma_{y,max} = \frac{p(D_i + t)}{2t}$$

Table 18.1. Selected Properties of Aluminum Alloys.

Aluminum Type	Density (g/cm3)	Coef. of Thermal Expansion.	Thermal Conduct.	Yield Stress Kpsi	Ult. Stress Kpsi
2024-T3	2.77	12.6	110	50	70
6061-T6	2.70	13.5	90	40	45
7079-T6	2.74	13.7	70	68	78

In this equation, $\sigma_{y,max}$ is the maximum yield stress, p is the interior pressure, D_i is the inside diameter, and t is the tank thickness. The tank thickness was calculated using a factor of safety of 10 [52].

The outer vessel will be made from aluminum 7079-T6. Since the outer vessel will be subjected to the highest temperatures, a low thermal conductivity constant is desired. The storage tank will be subjected to heat transfer by both conduction and radiation. The radiation protection will be studied in the heat transfer chapter and will be minimized by the use of a special goldized coating. Heat will transfer through conduction through the outside vessel until the insulation is reached. Aluminum 7079-T6 has the lowest thermal conductivity along with the highest yield and ultimate stress. The high strength of the outer vessel will help protect against micrometeorites. The storage tanks on board the servicer will not require the high tensile strength but the tanks on board the prime mover will not be protected by a surrounding capsule.

18.1.1.2 Tank Dimensions

The tank dimensions for each liquid fuel is greatly dependent on the time interval allowed between tank replacements. The fuel cells require 0.040 kg/kWhr of liquid hydrogen and 0.320 kg/kWhr of liquid oxygen [44]. In order to handle the peak power surge of 3kW, the fuel cells will require a diameter of 1.0m for the liquid hydrogen tank and 0.75m for the liquid oxygen tank. These values correspond to tank replacements every 7 days. The dimension

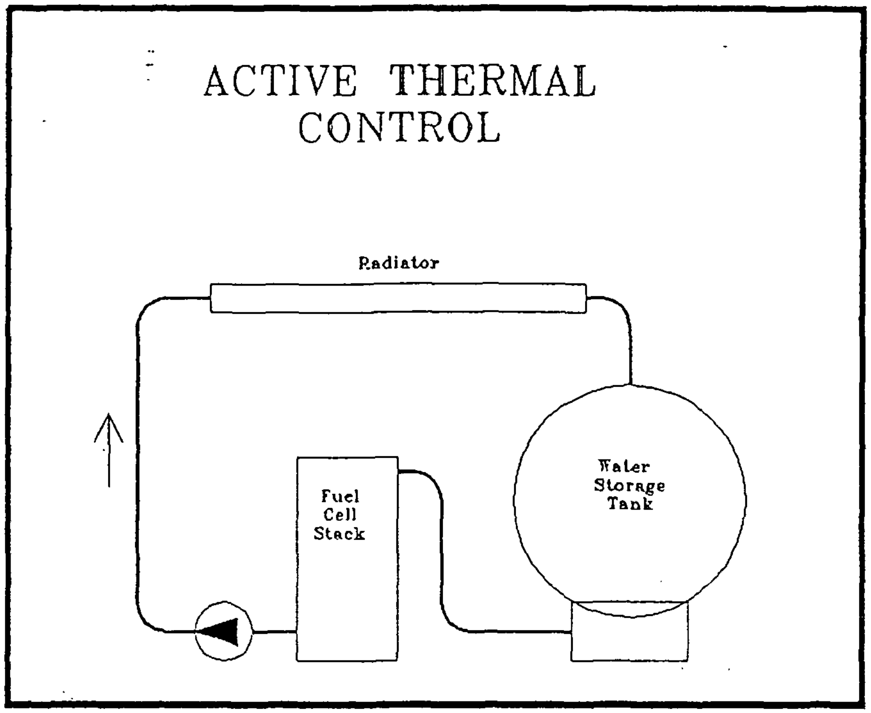

Figure 18.1. Fuel Cell Water Production Network.

calculations are found in Appendix H.

18.1.1.3 Fuel Cell Water Production

The fuel cells, described completely in the fuel cell chapter, on board the servicer will produce 0.360 kg/kWhr of water[44]. This water will be stored and later used in the heat rejection system to cool certain components. The water produced will be stored in a separate tank located on the servicer. The tank will have a spherical shape with a diameter of 0.9m. A tank of this size is able to store the produced water for 7 days. After these 7 days, the water must be either redistributed through the radiator or the tank must be replaced with an empty storage tank. The water storage tank will also be made from the two types of aluminum discussed previously, Al 2024-T3 and Al 7079-T6. Fig. 18.1 shows the network that uses the water produced by the fuel cells to transfer produced heat.

18.1.2 Tank Insulation

Each storage tank must be heavily insulated to reduce the effects of the heat flux acting on the tank surface. Multi-layer insulation was selected to insulate the storage tanks. A full description of the insulation and its properties is located in the heat transfer chapter. The multi-layer insulation is comprised of 450 layers of thin nylon mesh and goldized kapton. The goldized kapton is used to reduce the effects of radiation.[53]. the layers of kapton reduces the effects of conductive heat transfer. The tank will not experience any heat transfer due to convection because the vessel is contained in evacuated conditions.

Insulation of the propellant storage tanks is very important because the propellants must remain in liquid form. The storage tanks must not produce any propellant boil-off in the seven day period of use.

18.2 Prime Mover

The fuel system on board the prime mover will be similar to the system on the servicer. Due to time constraints and emphasis on the complete design of the lander servicer, an actual design containing numbers and materials was not performed for the prime mover.

19. Thermal Control

19.1 Introduction

19.1.1 Purpose

Placed in the harsh environment on the lunar surface, the servicer will require control systems to regulate the temperature of its components. This control is accomplished through both active and passive heat transfer systems.

19.1.2 Lunar Environment

A major hindrance to lunar operations are the inhospitable conditions in which personnel and machinery must work. The lunar day and night each last for a period of 14 days, during which the temperature changes from 108C (day) to -269C (night). The absence of an atmosphere results in a solar flux of 1360 W/m^2 [11], and the limitation of heat transfer to radiation and conduction. In addition, no protection from UV radiation exists. The soil can be considered a black body with an emissivity of 1 and an absorptivity of 0.9. The loose soil on the surface is powdered pumice stone which is used for cryogenic insulation. Thus, the soil can be considered a nearly perfect insulator where high temperature gradients exist across regions receiving different [11]solar fluxes (ie. shadows, or craters). The powder like soil tends to form a difficult to remove dust over components such as radiators causing a change of the optical properties.

19.1.2.1 Day Time Operations

During the lunar day the thermal control system will involve protecting the components from the large solar flux. The hydrogen storage tank receives a flow of hydrogen gas from the lander at 50K. Until the gas can be reliquified during the lunar night, its temperature and pressure continuously rise. These increases must be minimized to lower reliquification power requirements, and to reduce risks of vessel failure. In addition, storage tanks containing LOX and LH which power the fuel cells, as well as the tank holding the fuel cell's product water are at risk from temperature and pressure increases. A final thermal concern during day time operations is the regulation of the fuel cell temperature. The solid polymer electrolyte (SPE) fuel cell must remain saturated with water to operate. If the system is allowed to reach conditions where the product water vaporizes, failure occurs [54]. Maximum efficiency of the fuel cell occurs when the unit operates at a temperature of 5°0C [44].

19.1.2.2 Night Time Operations

Once the sun vanishes over the horizon, the adverse effects of the solar flux can be corrected. After fourteen 24 hour periods of sun light, a total of 2 Kg of hydrogen gas has been stored in the tank reaches a final temperature of 63K and a pressure of 0.98 MPa. A refrigeration system removes 1842 kJ of heat from the hydrogen gas in a 96 hour period using 1.5 KW of electricity [51]. The heat generated by the compressor is directed to a suitable location. Fuel cell operation at an ambient temperature of 4K requires thermal control of several components. As stated previously, the SPE fuel cell must remain saturated with water. At low temperature, freezing of thc working fluid dries out the electrolyte, and the system fails [55]. Once this product water leaves the fuel cell, care is taken to prevent freezing which could result in piping damage. Lastly, a conversion of the LOX and LH into gas is needed for use by the fuel cell.

19.2 Passive Systems

19.2.1 Purpose

Passive thermal control effectively slows the effects of heat transfer. Multilayer insulations, and proper optical coatings help to reduce thermal expansion of structures power requirements and sizes and weights of active heating and cooling systems.

19.2.2 Theory

The absence of an atmosphere limits the path of heat transfer between separate entities to radiation. Since radiative heat transfer is a surface phenomenon, controlling the absorption, emission, and reflective properties of object determines the amount of heat flow to or from it. Multilayer insulation consists of thin layers of material with a low emissivity and a high reflectivity separated by a layer of nonconducting material. The entire assembly is stapled together to form a flexible blanket. The first layer experiences a heat flux, most of which is reflected. The temperature it reaches gives off a heat flux of which the next layer absorbs only a fraction dependant on its absorptivity. This gradual degradation of the heat flux between the inner and outer layers is related to the number of layers (N) by Equation (1).

$$G_{i,o}(0) = \frac{1}{N+1} G_{i,o}(N) \tag{1}$$

where $G_{i,o}(0)$ is the heat flux between the inner and outer layer with 0 layers between. [56] The effectiveness of MLI is limited by the number of layers added and the conductance of heat through the staples. As shown in the equation above, adding additional layers has a reduced effect with each layer added.

A second method of passive thermal control involves the use of thermal coatings (paints) whose optical properties determine the amount of heat absorbed or given off. Organic coatings experience degradation due to the intense UV bombardment, and cracking resulting from thermal cycling. Inorganic coatings such as potassium silicate bonded zinc oxide, offer low absorbance, high emissivity, and resist UV damage. These qualities make this paint an ideal coating for the radiator [57].

19.2.3 Thermal Model of Servicer

For the benefit of analysis, a thermal model of the servicer was conceived. The assumptions made in creating the model were intended to simplify the calculation of wall temperature and to offer a worse case. For all such calculations, heat flux from the sun originates directly overhead of the vehicle. Walls are presumed to be made up of the aluminum plates used as meteorite protection. The plates have an emissivity of 0.11 [58] and a thermal conductivity of 177 W/m.K [56]. Neglecting transfer by conduction in the legs, the sole mechanism for heat flow is radiation. The direction of the transfer is from the hotter object to the cooler one regardless of the surface properties. Therefore, since the top and bottom of the servicer are exposed only to the high temperature sources of the radiator (323K) and lunar surface (383 K), heat flow is into the servicer. Once equilibrium has been established, the top and bottom of the servicer are at the temperature of the radiator and lunar surface, respectively. Where as the sides of the servicer are exposed to both the lunar surface and deep space (4 K) [11]. Thus, the sides act as fins for the top and bottom. The calculation of the temperature distribution of the servicer's sides is complicated by the fact that the heat flux out to deep space is not constant along the length of the side. The resulting differential equation was solved using the Runge Kutta algorithm shown in the appendix. [59] This algorithm determined the temperature drop from the bottom to the center of the side to be less than 10 degrees. From this data, all subsequent calculations of temperatures inside of the servicer assumed a worst case wall temperature of 383 K.

19.2.4 Hydrogen Boil Off Storage Tank

As previously stated, the regulation of the temperature and pressure rise within the hydrogen storage tank is important to the safety and energy requirement of the mission. To limit the amount of heat transfer into the tank, a multi layer insulation is wrapped around the tank. The insulation consists of thin layers of kapton coated on both sides with gold. With an emissivity of .003, the gold restricts radiative heat transfer. In between each layer is a nylon net intended to prevent conduction between layers [60]. The entire thermal blanket contains 450 layers at a thickness of 16.5 cm. [58]After 14 days of exposure to the hot elements, the hydrogen's temperature rose 4 K to 54 K which was accompanied by a pressure rise to 0.8 MPa.

19.2.5 Fuel Cell Storage Tanks

Due to space limitations, the hydrogen and oxygen used for the fuel cell are stored as cryogenic liquids. Their temperature must also be controlled to prevent excessive boil off. A blanket of 25 layers of goldized kapton surrounds each tank. Less protection is needed since the liquids must change state into a gas before entering the fuel cell. The tank which stores the product water and all related piping is also surrounded by a 25 layer blanket. The active cooling system uses this water to remove excess heat from the fuel cell. The blanket prevents the water from heating to a temperature where its cooling ability is affected. Conversely, during the lunar night the water must not be allowed to freeze.

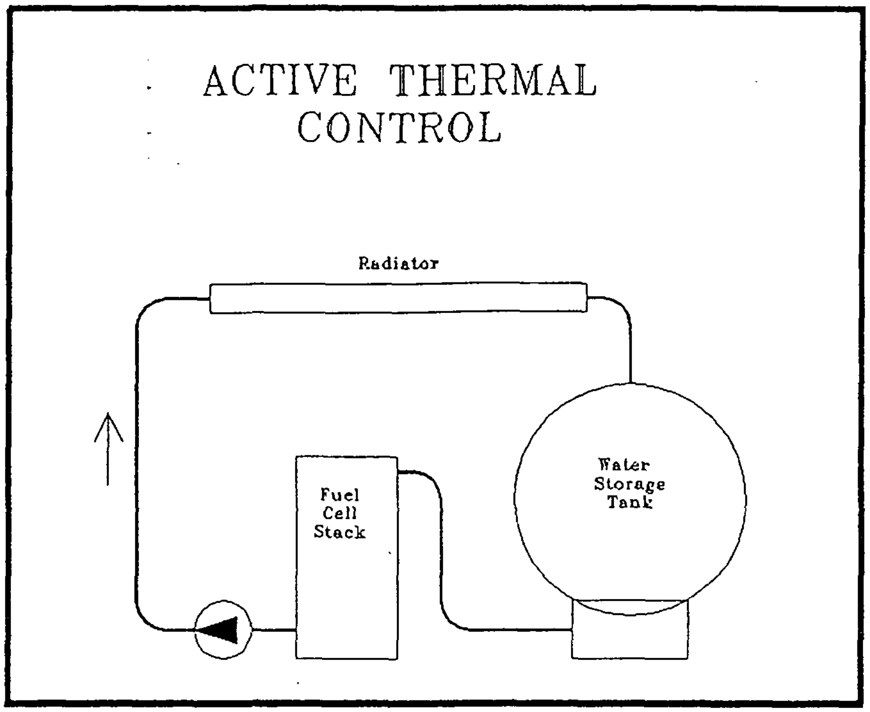

Figure 19.1. Cooling Loop for Fuel Cell.

19.3 Active Thermal Control

Since operation of the fuel cells involves a chemical reaction, the power they produce is dependant upon their temperature. However, an increase in temperature decreases the life of the fuel cell. It has been determined that the optimum temperature is 50°C (323 K) [55]. To maintain this temperature, the product water is used to remove the heat, as shown in Fig. 19.1 Excess water stored in the tank is pumped through the fuel cell and up to a radiator located on top of the servicer. The water's temperature is lowered in the radiator and it then returns to the insulated storage tank. This exchanger loop transfers 600 W for a 3 kW fuel cell whose efficiency is assumed at 80%.

19.3.1 Fluid Pump

The pump used to move the water is a centrifugal pump using 35 W. The water flows through a 3 cm pipe at a mass flow rate of 0.5 kg/s. The associated pipe friction and 3 m rise to the radiator results in a pump head of 5.7 m. A feed back loop measuring fuel cell temperature controls the mass flow rate of the pump to maintain the proper temperature. To ensure a sufficient amount of water is present at start up, 30 L of water should exist in the tank when the servicer is brought out to the landing pad.

19.3.2 Radiator

The radiator is the most critical component of the active thermal control system. High degrees of reliability and thermal performance, as well as low weight are compulsory qualities

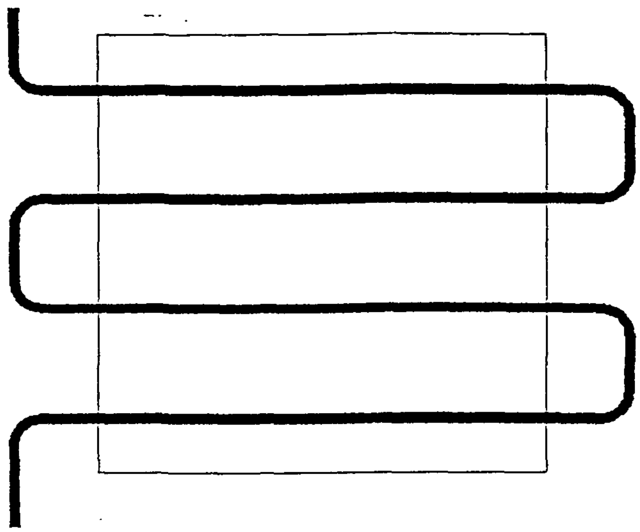

Figure 19.2. Conventional Radiator.

of the required radiator. Since the radiator is directly exposed to micro meteoroid bombardment, the mechanism by which the heat is spread over the radiating area must not be prone to single point failure. Using the zinc oxide coating mentioned earlier, the radiator must reject 600 W of heat carried by the moving water in an area of 4 sq. m. To accomplish this task, four designs were considered.

19.3.2.1 Radiator Design Alternatives

The first radiator was a conventional arrangement. The heat is transferred to the radiating surface by conduction from a pipe carrying the hot water (see Fig. 19.2).

A second radiator consisted of six modular units, each composed of an exchanger, a heat pipe, and a radiator. Each segment contains a radiator 1 m x .67 m, a heat pipe 10 cm in diameter and an exchanger .5 m long. The fluid loop described above passes through each exchanger in a series formation. The exchanger directs the water flow into many small tubes which surround the heat pipe. To hold the heat pipe inside of the exchanger, pressurized gas applies a force on a flexible liner enveloping the small tubes. This allows for the removal of a heat pipe without loosing water from the loop or the pressurized gas. If micro-meteorite damage occurs on a section of the radiator, the unit can be removed and replaced without shutting down the cooling system (see Fig. 19.3).

A third radiator design was similar to one developed in reference [61]. The water passes through a central duct and transfers heat to heat pipes brazed onto the top and bottom. Four

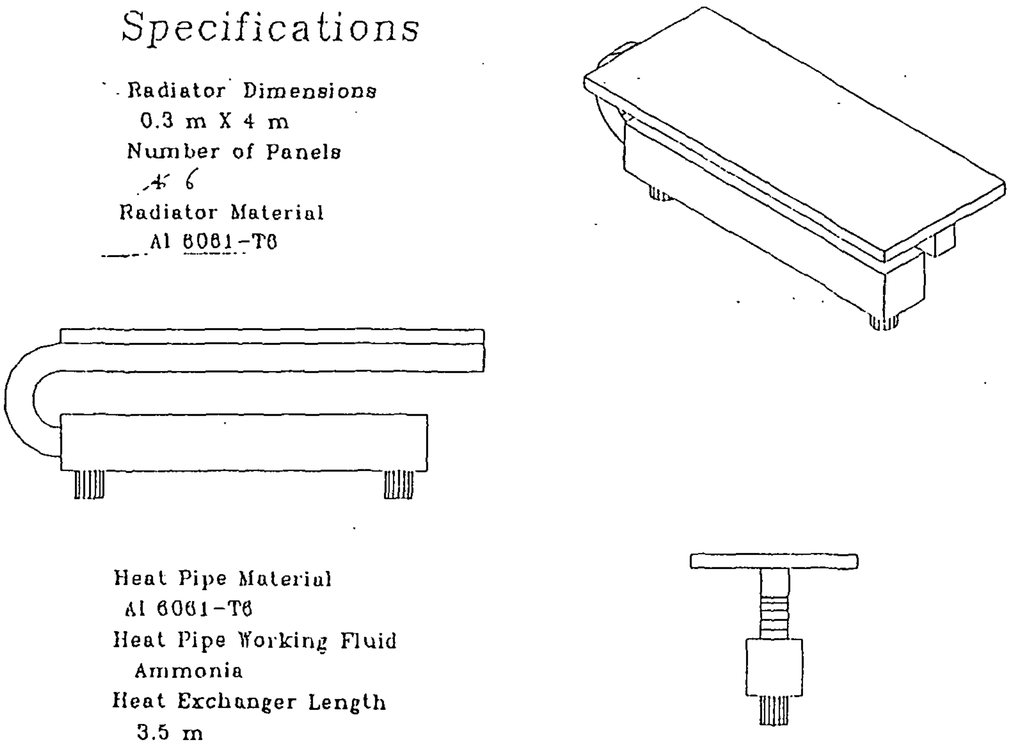

Figure 19.3. Heat Pipe Radiator - Design No .1.

hundred heat pipes 1 cm in diameter are used as the final radiating surface. The use of many small heat pipes allows for the loss of some due to micro meteorite bombardment without a system wide failure occurring (see Fig. 19.4).

The final heat radiator design is an adaptation of the third design. To increase the transfer of heat between the water, heat pipe interface, the pipes were solidly mounted inside of the channel containing the flowing water. In addition, since the diameter of heat pipe required to transfer the heat contained a surface area too small to radiate this amount of heat away, a fin of 2 cm was placed on each side of the heat pipe. The allowed for a reduction in the number of heat pipes to 160 without sacrificing the reliability of the system,as shown in Fig. 19.5

19.3.2.2 Radiator Armor Requirements

To quantitatively determine the danger to a radiator from micro-meteorites, a statistical analysis was performed. A model was developed by NASA predicts the flux of micrometeorites mass M or greater. This model is then used along with a Poisson's distribution and an armor penetration formula to determine the chance of damage for a chosen armor thickness [62].

Detailed equations are located in the appendices. It was found that a 5 mm plate of titanium was needed on crucial components (ie. those that would fail if hit once). For this thickness, the likelihood of penetration remained less than 1/10 of 1%. On less crucial areas, a thickness of 2.5 mm yields a probability of 1%. These thickness requirements were used in determining the weight for each system.

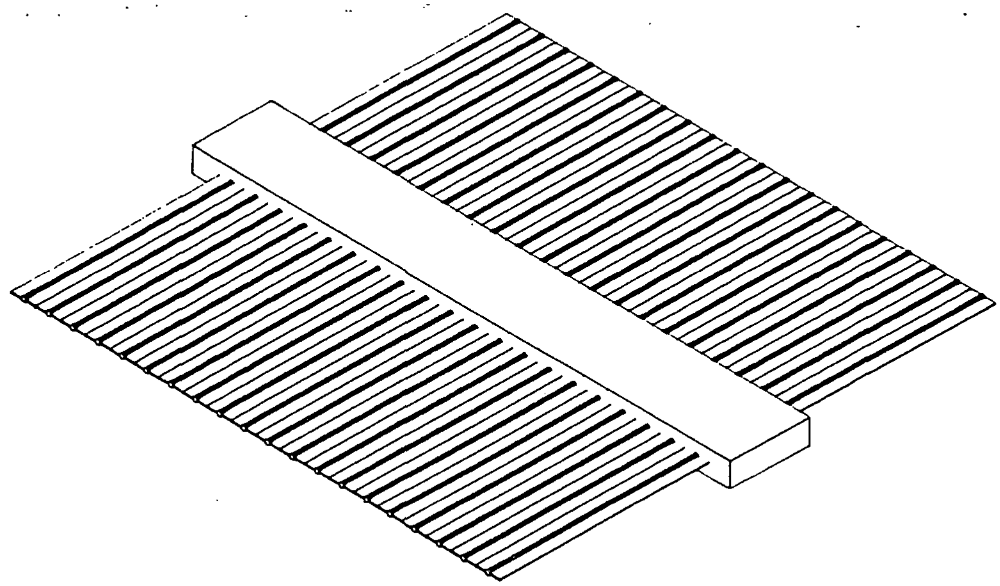

Figure 19.4. Heat Pipe Radiator - Design No. 2.

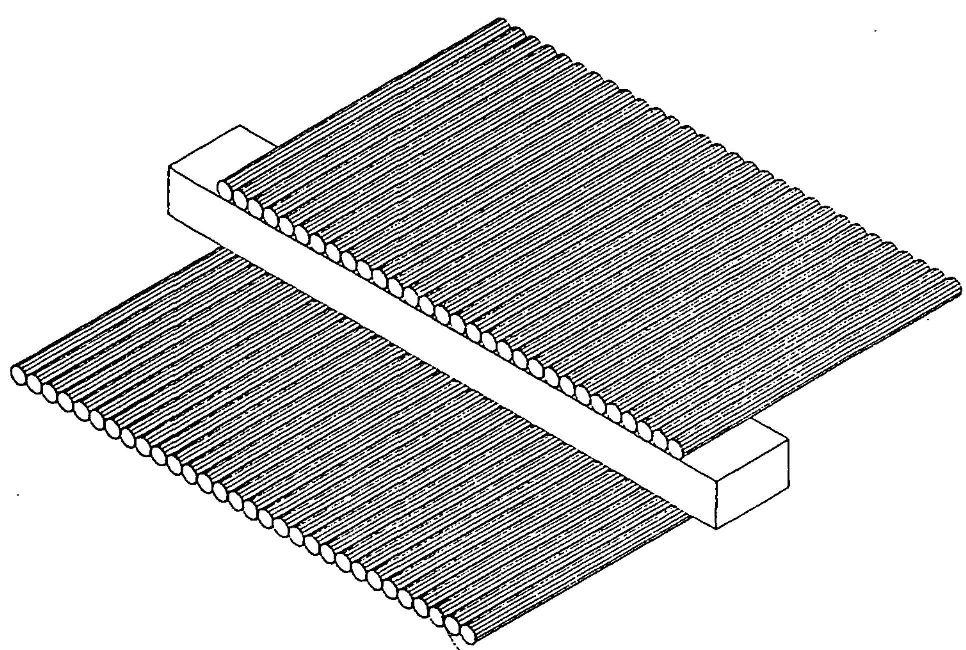

Figure 19.5. Heat Pipe Radiator - Design No. 3.

Conventional Radiator	210 kg
#1 Heat Pipe	240 kg.
#2 Heat Pipe	228 kg.
#3 Heat Pipe	132 kg.

19.3.2.3 Basis for Radiator Choice

The conventional radiator required an armor thickness of 5 mm covering the entire top and sides. This radiator experiences the lowest grade of reliability of the four radiators looked at. The reason for this is the large area covered by the piping below the armor. If the armor is penetrated in this area the entire system is prone to shut down. This size of the area covered by piping is driven by the ability of the titanium to conduct the heat over the radiating surface. The closer the pipes are together, the better the thermal performance.

Heat pipe radiator #1 requires a 5 mm covering over any of the heat pipes. Although the system can operate with one heat pipe lost, the thermal performance is reduced by a factor of 1/6. The use of a small number of large heat pipes also results in a poor temperature distribution along the radiating surface.

Heat pipe radiator #2 attempts to reduce the problem of failure due to meteor impact. Each heat pipe has the ability to transfer 10 W of heat [63]. Thus, only 60 heat pipes are needed to transfer the heat, but more are used to fulfil the radiating area requirement. The placement of the heat pipes on the top and bottom of the exchanger protects the fluid loop from meteor damage, but reduces the transfer surface area between the heat pipe and the hot water.

The third heat pipe radiator attempts to reduce the weight of heat pipe radiator by replacing some of the heat pipes with fins. Also, the problem of the small contact area by placing the evaporator section of the heat pipe within the flow channel. The top of the channel is then thickened to 7 mm for additional protection.

The radiator characteristics discussed above were quantitatively judged, and the results were placed within a weighted property index table (see Appendix I). This table assigns scaling factors to each quality (Reliability, Thermal Performance and Weight) and allows a score to be attained for each design. Based upon the final score, heat pipe radiator #3 was chosen for use in the active control loop. This table is also located in Appendix I.

19.3.2.4 Radiator Dust control

A serious problem arises in the placement of the servicer on the landing pad shortly after touch down. The dust kicked up during landing can seriously degrade the optical qualities of the radiator surface. This dust problem is most severe in the first few days, while the dust is still settling from the rocket blast. At this time, the hydrogen has yet to begin boiling off of the lander at a rate high enough where a high load is placed upon the fuel cells. This results in a low load upon the radiator. The determination of the radiator area allows for a degradation of the absorptivity from .16 to .31 . During the maintenance trips out to the servicer, a gas jet is used to remove the dust build up.

Appendix A. Senior Aerspace Design Group Work Breakdown

1. Dan
   - -Radiation protection
     - -Servicer and Transport
   - -Shielding
     - -stand-off shielding for Servicer and Transport.
     - -micrometeorite aspect of any 'tent' design.
   - -Site preparation
     - -roadway to landing sites.
     - -overall landing site plan.
     - -light setup for night landing.

2. Robert
   - -Robotics
     - -structure of robotic crane itself.
     - -selection/design of end effectors.
     - -end effector brackets on cargo/modules.
     - -base design and interaction with Transport structure and center of gravity.

1. Marcus
     - -Electronics
     - -radiation resistant components.
   - -Control systems
     - -robotic arm control system and sensors.
     - -Servicer control system, sensors.
     - -basic Transport control system.
     - -life support sensors.

4. Jeanne S.
   - -Power
     - -fuel cell design, calculations.
     - -battery backup.
     - -motors for robotic arm, Transport.

5. Jeanne O.
   - -Communications
     - -link between Servicer/Base, Crew Module/Transport, and Transport/Base.
       - -sensor communications.
       - -voice communications.
       - -visual communications.
   - -Navigation
     - -nav system for Transport.
     - -radio beacon setup around landing site.

6. Doug

- -Fuel system
  - -insulation design for lander tanks.
    - -heat flow calculations (day/night).
    - -related boil-off calculations.
  - -pumps and calculations for reliquification.

7. Todd

- -Umbilical
  - -electrical/fluid between Servicer/Lander.
  - -electrical between Servicer/Transport.
  - -electrical between Crew Module/Transport.
  - -dust removal system prior to mating.

8. Chris

- -Ergonomics
- -Maintenance/Scheduling
  - -crew module life support considerations.
  - -EVA suit considerations for activities.
  - -parallel scheduling of activities after landing and prior to takeoff.
  - -scheduling and definition of intermittent maintenance.
  - -alternative operations of Transport.

9. Matt

- -Mobility
- -Transport and assembly
- -Overall structure/frames
  - -frame design of Servicer.
  - -frame design of Transport.
    - -wheels, steering.

10. John

- -Heat rejection
- -radiator designs
  - -fuel cells.
  - -Transport.
  - -Servicer reliquification heat.
- -insulation aspect of any 'tent' design.
- -heat rejection methods for electronics.
- -heat exchanger for lander in (day/night).
- -heat storage designs.

Appendix B. Gantt Chart Project Schedules

Fig. B.1 shows the detailed schedule for the first quarter of the project year.

Fig. B.2 shows the detailed schedule for the second quarter of the project year.

Fig. B.3 shows the detailed schedule for the second half of the project year.

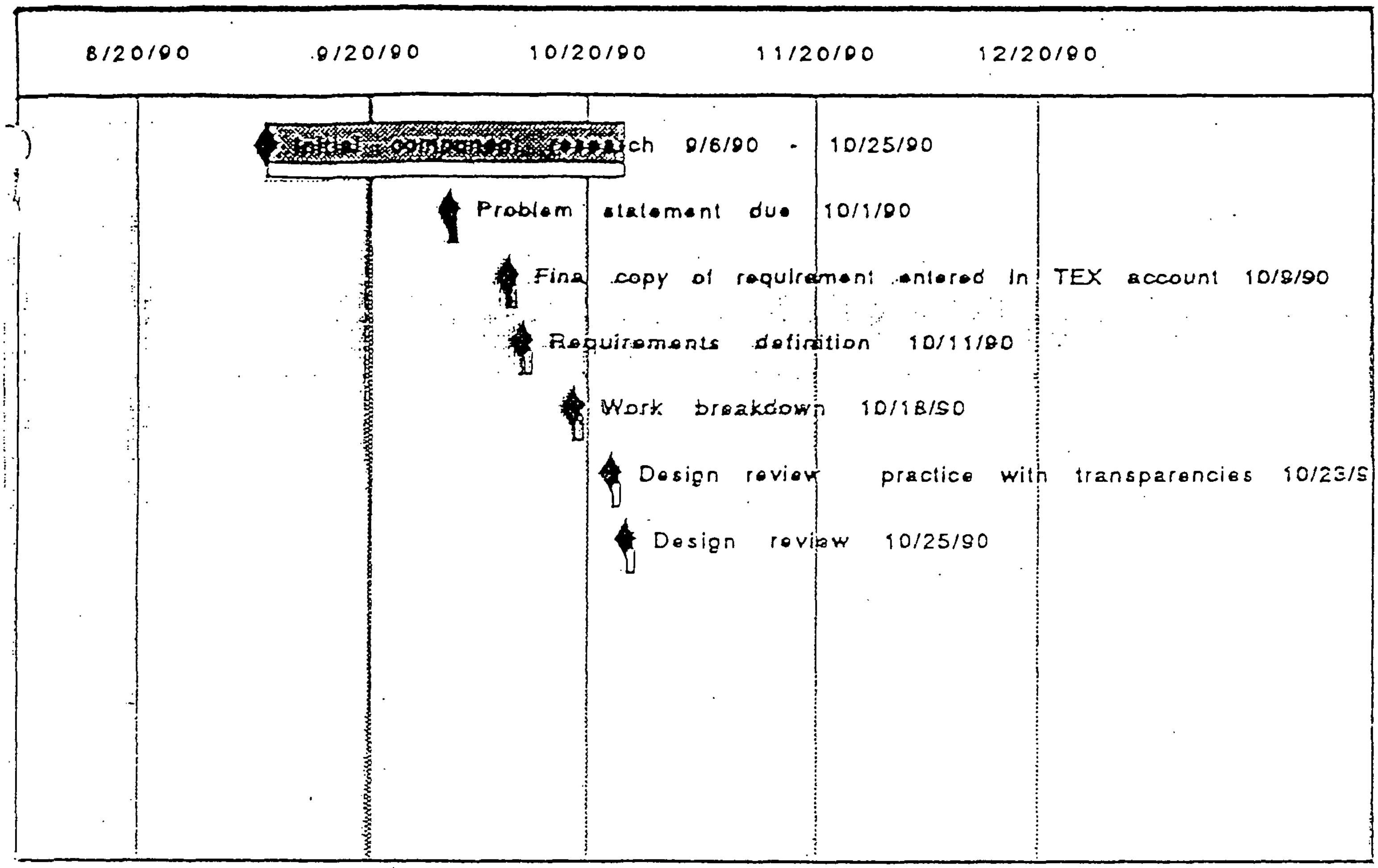

Figure B.1. First Quarter Detailed Schedule.

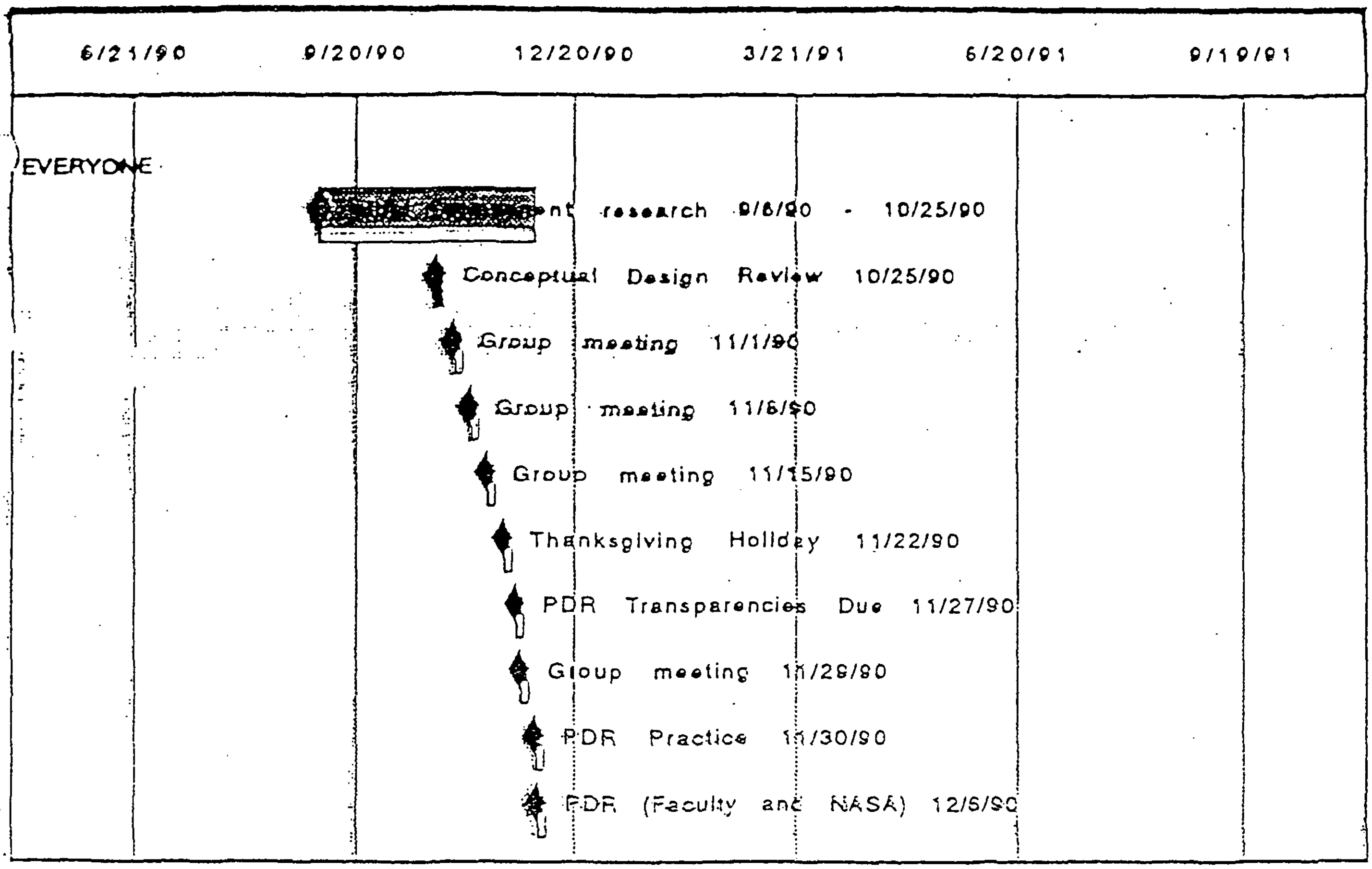

Figure B.2. Second Quarter Detailed Schedule.

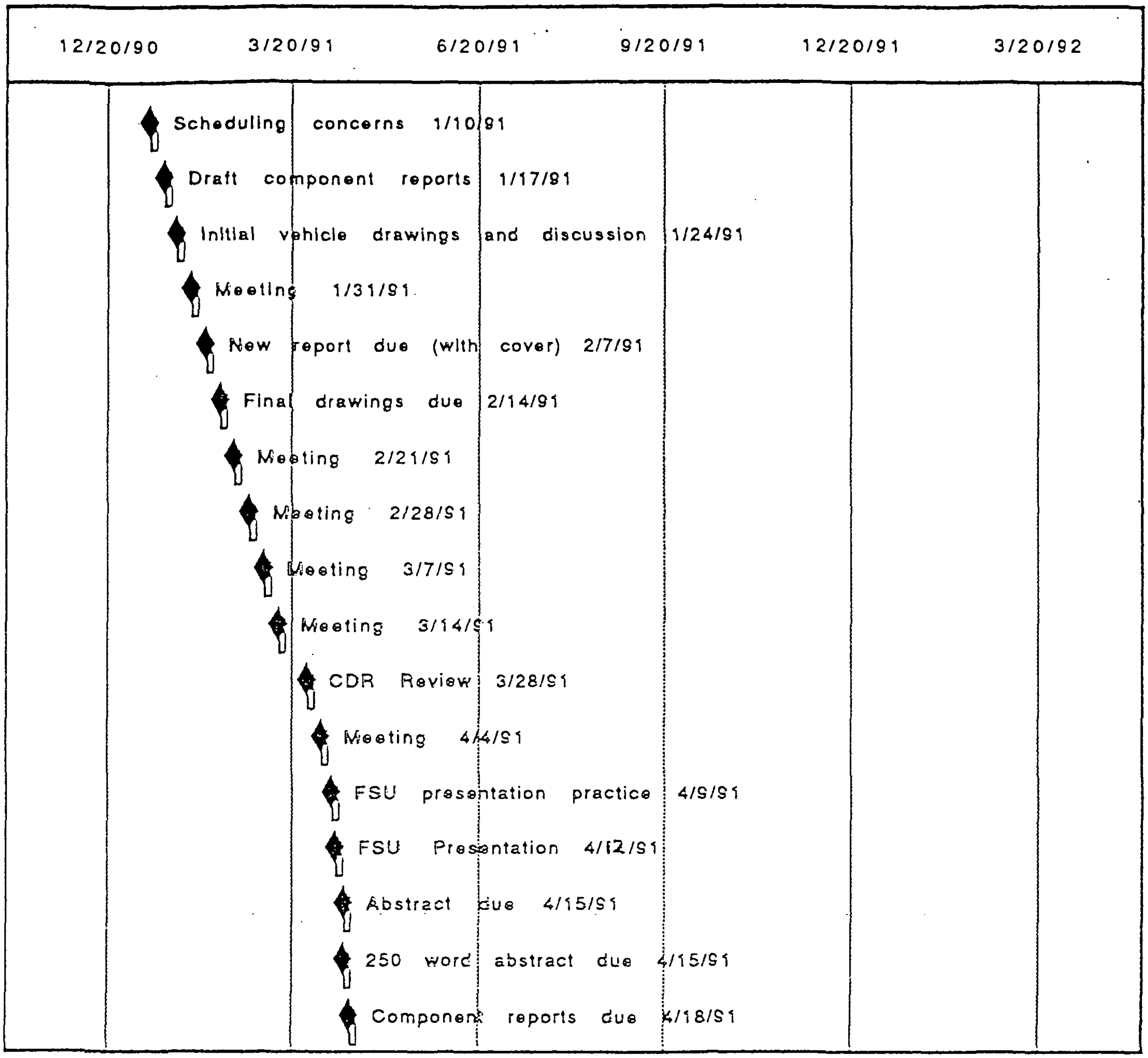

Figure B.3. Second Half Detailed Schedule.

Appendix C. Servicer Frame Model Files

C.1 Pal2 Frame Model File SERV2.TXT

```
TITLE                    SERVICER FRAME MODEL
NODAL POINT LOCATIONS 1
    1 1.5      1.5      0.0
    2 1.799999 1.5      0.0
    3 1.399999 1.2      0.0
    4 4.099998 1.2      0.0
    5 4.       1.5      0.0
    6 3.7      1.5      0.0
    7 2.       3.       0.0
    8 2.75     3.       0.0
    9 3.5      3.       0.0
   10 2.75     1.5      0.0
   11 1.5      1.5      1.5
   12 1.799999 1.5      1.5
   13 1.399999 1.2      1.5
   14 4.099998 1.2      1.5
   15 4.       1.5      1.5
   16 3.7      1.5      1.5
   17 2.       3.       1.5
   18 2.75     3.       1.5
   19 3.5      3.       1.5
   20 2.75     1.5      1.5
   21 2.75     4.       0.75
   22 1.5      1.5      0.75
   23 4.       1.5      0.75
--BLANK LINE--
MATERIAL PROPERTIES 71.0E9,26.2E9,2710.0,0.334,296.0E6
BEAM TYPE 3,0.07,0.063
CONNECT     7 TO    17
CONNECT    17 TO    18
CONNECT    18 TO     8
CONNECT    18 TO    19
CONNECT    19 TO     9
CONNECT     7 TO    18
CONNECT     8 TO    17
CONNECT     8 TO    19
CONNECT    18 TO     9
CONNECT    19 TO    15
```

```
CONNECT    15  TO    14
CONNECT    14  TO    16
CONNECT    16  TO    15
CONNECT    16  TO    19
CONNECT    17  TO    11
CONNECT    11  TO    13
CONNECT    13  TO    12
CONNECT    12  TO    11
CONNECT    12  TO    17
CONNECT    12  TO    20
CONNECT    20  TO    18
CONNECT    17  TO    20
CONNECT    20  TO    19
CONNECT    20  TO    16
CONNECT    21  TO     7
CONNECT    21  TO     9
CONNECT    19  TO    21
CONNECT    21  TO    17
CONNECT    13  TO    22
CONNECT    22  TO     3
CONNECT     1  TO    22
CONNECT    22  TO    11
CONNECT    12  TO    22
CONNECT    22  TO     2
CONNECT     2  TO    12
CONNECT    10  TO    20
CONNECT    20  TO     2
CONNECT    12  TO    10
CONNECT    10  TO    16
CONNECT     6  TO    20
CONNECT     6  TO    16
CONNECT    16  TO    23
CONNECT    23  TO     6
CONNECT     5  TO    23
CONNECT    23  TO    15
CONNECT    23  TO     4
CONNECT    23  TO    14
CONNECT     7  TO    22
CONNECT    22  TO    17
CONNECT    19  TO    23
CONNECT    23  TO     9
CONNECT     3  TO     1
```

```
 CONNECT    1 TO    2
 CONNECT    2 TO    3
 CONNECT    1 TO    7
 CONNECT    7 TO    2
 CONNECT    1 TO   10
 CONNECT   10 TO    7
 CONNECT    7 TO    8
 CONNECT    8 TO   10
 CONNECT   10 TO    6
 CONNECT    6 TO    4
 CONNECT    4 TO    5
 CONNECT    5 TO    6
 CONNECT    5 TO    9
 CONNECT    9 TO    6
 CONNECT    8 TO    9
 CONNECT   10 TO    9
ZERO 1
RA OF ALL
-- BLANK LINE --
END DEFINITION
```

C.2 Pal2 Stationary Loading File STAT2.TXT

```
DISPLACEMENTS APPLIED 1
TX 0 3 4 13 14
TY 0 3 4 13 14
TZ 0 3 4 13 14
-- BLANK LINE --
FORCES AND MOMENTS APPLIED 1
FY -91 7 17 9 19 8 18
FY -182 2 10 6 12 20 16
-- BLANK LINE --
SOLVE
QUIT

```

C.3 Pal2 Lifting Loading File STAT3.TXT

```
DISPLACEMENTS APPLIED 1
TX 0 21
TY 0 21
TZ 0 21
-- BLANK LINE --
```

```
FORCES AND MOMENTS APPLIED 1
FY -91 7 17 9 19 8 18
FY -182 2 10 6 12 20 16
-- BLANK LINE --
SOLVE
QUIT
```

C.4 Pal2 Model Analysis Output File FORCE2.PRT

```

    03-06-91 23:11              MSC/pal 2                          Page    1



                                 MSC/MOD


       STATIC   ANALYSIS   SUBCASE NO.    1    ELEMENT RECOVERY



ELEMENT     1  BEAM     FORCE RESULTS     ( NODE      7  )      ( NODE     17  )
AXIAL   -2.219E+02 V SHEAR  3.370E-03 V MOMENT -4.630E-03 V MOMENT  4.630E-03
TORSION 0.000E-01 W SHEAR  6.173E-03 W MOMENT -2.528E-03 W MOMENT  2.528E-03

        MAXIMUM STRESSES FOR BEAM          VON MISES CRITERION
ELEMENT MAJOR      MINOR       SHEAR       STRESS  % YIELD  @NODE CONNECTIVITY
    1  4.974E-12 -3.039E+05  1.519E+05  3.039E+05    0.1     7       7   17


ELEMENT     2  BEAM     FORCE RESULTS     ( NODE     17  )      ( NODE     18  )
AXIAL   -2.906E+02 V SHEAR -7.411E+00 V MOMENT -1.573E+00 V MOMENT  1.573E+00
TORSION 0.000E-01 W SHEAR  4.195E+00 W MOMENT  2.779E+00 W MOMENT -2.779E+00

        MAXIMUM STRESSES FOR BEAM          VON MISES CRITERION
ELEMENT MAJOR      MINOR       SHEAR       STRESS  % YIELD  @NODE CONNECTIVITY
   2  6.992E-12 -6.631E+05  3.316E+05  6.631E+05    0.2    17      17   18


ELEMENT     3  BEAM     FORCE RESULTS     ( NODE     18  )      ( NODE      8  )
AXIAL    1.480E+02 V SHEAR  5.568E-04 V MOMENT -4.703E-03 V MOMENT  4.703E-03
TORSION 0.000E-01 W SHEAR  6.271E-03 W MOMENT -4.176E-04 W MOMENT  4.176E-04
```

```
      MAXIMUM STRESSES FOR BEAM        VON MISES CRITERION
ELEMENT MAJOR      MINOR      SHEAR       STRESS  % YIELD  @NODE CONNECTIVITY
    3  2.028E+05  0.000E-01  1.014E+05  2.028E+05    0.1    18      18    8


ELEMENT    4  BEAM     FORCE RESULTS    ( NODE    18  )      ( NODE    19  )
AXIAL   -2.904E+02 V SHEAR  7.361E+00 V MOMENT  1.588E+00 V MOMENT -1.588E+00
TORSION 0.000E-01 W SHEAR -4.235E+00 W MOMENT -2.760E+00 W MOMENT  2.760E+00

      MAXIMUM STRESSES FOR BEAM        VON MISES CRITERION
ELEMENT MAJOR      MINOR      SHEAR       STRESS  % YIELD  @NODE CONNECTIVITY
    4  1.054E-11 -6.627E+05  3.313E+05  6.627E+05    0.2    19      18   19
```

C.5 QuickBasic file FINDWORD.BAS

```
        DIM i, j, k, start AS INTEGER
        DIM in$, word$
        DIM z AS SINGLE

        CLS
        INPUT "Enter input file name >> ", infile$
        INPUT "Enter output file name >> ", outfile$
        OPEN infile$ FOR INPUT AS #1
        OPEN outfile$ FOR OUTPUT AS #2

        INPUT "enter word to look for >> ", word$
        word$ = UCASE$(LTRIM$(RTRIM$(word$)))
        i = 0
        WHILE NOT (EOF(1))
           LINE INPUT #1, in$
           in$ = UCASE$(in$)
           start = INSTR(in$, word$)
           IF start = 0 THEN GOTO 10
           start = start + LEN(word$)
           z = VAL(MID$(in$, start))
           i = i + 1
           PRINT "Element #"; i; "   Value = "; z
           PRINT #2, i, z
10 :
        WEND
```

```
CLOSE #1
CLOSE #2
PRINT "Program Done"
END
```

C.6 QuickBasic Extracted Force Data File FORCE2.OUT

```
1          -221.9
2          -290.6
3           148
4          -290.4
5          -221.9
6          -87.51
7          -87.84
8          -87.52
9          -87.84
10          59.13
11          36.4
12         -39.69
13          6.366
14          126.5
15          59.37
16          36.53
17         -39.92
18          6.309
19          126.3
20         -33.4
21         -74.87
22          141.5
23          141.2
24         -34.46
25          593.7
26          594.2
27          593.8
28          594.2
29          2.052
30          1.641
31         -1.016
32         -.4654
33          .8511
34          1.919
35          3.616
```

```
36          10.39
37         -7.571
38         -7.516
39         -5.176
40         -5.301
41          2.309
42          .6664
43          1.264
44         -1.084
45         -.7334
46          1.711
47          1.863
48          2.715
49          2.741
50          2.795
51          2.718
52          34.31
53          36.75
54         -37.02
55          58.99
56          126.4
57         -32.38
58          141.4
59         -290.6
60         -74.85
61         -33.79
62         -39.51
63          36.31
64          6.487
65          58.97
66          126.7
67         -290.8
68          141.7
```

C.7 FORTRAN Program FORCE.FOR for Sizing Servicer Frame Members

```



      PROGRAM FORCE

CCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCC
C                                                                    C
```

```
C  Program written for structural analysis of Senior Aero       C
C  project LLGSS Servicer. Inputs file FORCE1.OUT containing    C
C  frame element numbers and forces, then calculates diameters  C
C  of given wall thickness aluminum tubing for both maximum     C
C  axial stress and buckling possibilities.                     C
C                                                               C
CCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCCC

      REAL XL,XU,XR,ES,EA,LOAD,D,D1,D2
      INTEGER MAXIT,ITER,ELENUM

      OPEN(UNIT=1,FILE='FORCE1.OUT',STATUS='OLD')
      OPEN(UNIT=2,FILE='DIAMETER.OUT',STATUS='NEW')

      WRITE(2,5) 'ELEMENT','FORCE(N)','DIAMETER(M)'
      WRITE(2,*)
    5 FORMAT(1X,A7,3X,A8,3X,A11)
   10 READ(1,*,END=100) ELENUM,LOAD
        D2=0
        D1=125*((100*LOAD*4)/(296E6*3.1415)+0.000016)
        IF (LOAD.LT.0) THEN
          XR=0.0
          EA=0.0
          ITER=0
          XL=0.0
          XU=1.0
          ES=0.001
          MAXIT=1000
          CALL BISEC(XL,XU,ES,XR,EA,MAXIT,ITER,LOAD)
          D2=XR
        ENDIF
        IF (D1.GE.D2) THEN
          D=D1
        ELSE
          D=D2
        ENDIF
        IF (D.LT.0.01) THEN
          D=0.01
        ENDIF
          IF (ITER.GE.MAXIT.and.load.le.0) THEN
            WRITE(2,15) ELENUM,'--- NO CONVERGANCE'
   15       FORMAT(1X,I3,3X,A30)
```

```
          ELSE
            WRITE(2,20) ELENUM,load, D
 20         FORMAT(1X,I3,6X,2(F10.5))
          ENDIF
      GOTO 10
100   END


      FUNCTION F(X,LOAD)
      REAL F,X,LOAD
      F=34.4E9/4*(X**4-(X-0.004)**4)-50*abs(LOAD)
      RETURN
      END


      SUBROUTINE BISEC(XL,XU,ES,XR,EA,MAXIT,ITER,LOAD)
      REAL F
      ITER=0
      EA=1.1*ES
 10   IF (EA.GT.ES.AND.ITER.LT.MAXIT) THEN
        XR=(XL+XU)/2
        ITER=ITER+1
        IF (XL+XU.NE.0.0) THEN
          EA=ABS((XU-XL)/(XL+XU))*100.0
        ENDIF
        TEST=F(XL,LOAD)*F(XR,LOAD)
        IF (TEST.EQ.0.0) THEN
          EA=0.0
        ELSE
          IF (TEST.LT.0.0) THEN
            XU=XR
          ELSE
            XL=XR
          ENDIF
        ENDIF
        GOTO 10
      ENDIF
      RETURN
      END
```

C.8 FORTRAN Sizing Program Diameter Output DIAMETER.OUT

ELEMENT	FORCE(N)	DIAMETER(M)
1	29.75000	0.01000
2	-130.50000	0.03816
3	15.01000	0.01000
4	-129.80000	0.03809
5	29.61000	0.01000
6	-7.52100	0.01589
7	-8.94200	0.01672
8	-7.60300	0.01594
9	-8.95600	0.01673
10	-170.60001	0.04154
11	-217.20000	0.04486
12	-225.30000	0.04539
13	11.93000	0.01000
14	-16.60000	0.02013
15	-171.30000	0.04160
16	-217.70000	0.04490
17	-229.89999	0.04568
18	10.79000	0.01000
19	-18.51000	0.02081
20	-139.70000	0.03899
21	-76.81000	0.03229
22	142.30000	0.01000
23	140.20000	0.01000
24	-141.80000	0.03918
25	-0.49240	0.01000
26	0.18930	0.01000
27	-0.51270	0.01000
28	0.11410	0.01000
29	-111.30000	0.03629
30	-109.30000	0.03608
31	0.77030	0.01000
32	2.34600	0.01000
33	10.12000	0.01000
34	10.96000	0.01000
35	12.88000	0.01000
36	41.98000	0.01000
37	-28.17000	0.02365

38	-30.61000	0.02426
39	-19.45000	0.02112
40	-22.49000	0.02208
41	8.20100	0.01000
42	7.25000	0.01000
43	10.60000	0.01000
44	0.55300	0.01000
45	1.41800	0.01000
46	-111.40000	0.03630
47	-108.50000	0.03600
48	-53.99000	0.02892
49	-51.89000	0.02857
50	-53.17000	0.02878
51	-51.95000	0.02858
52	-227.70000	0.04554
53	130.60001	0.01000
54	-204.80000	0.04403
55	-174.89999	0.04187
56	-11.73000	0.01813
57	-122.40000	0.03739
58	140.10001	0.01000
59	-130.80000	0.03819
60	-76.70000	0.03227
61	-129.80000	0.03809
62	-218.20000	0.04493
63	-222.20000	0.04519
64	14.64000	0.01000
65	-174.89999	0.04187
66	-15.25000	0.01962
67	-131.50000	0.03825
68	142.89999	0.01000

Appendix D. Quick Release Drawing

Fig. D.1 on the following page shows the detailed quick release design.

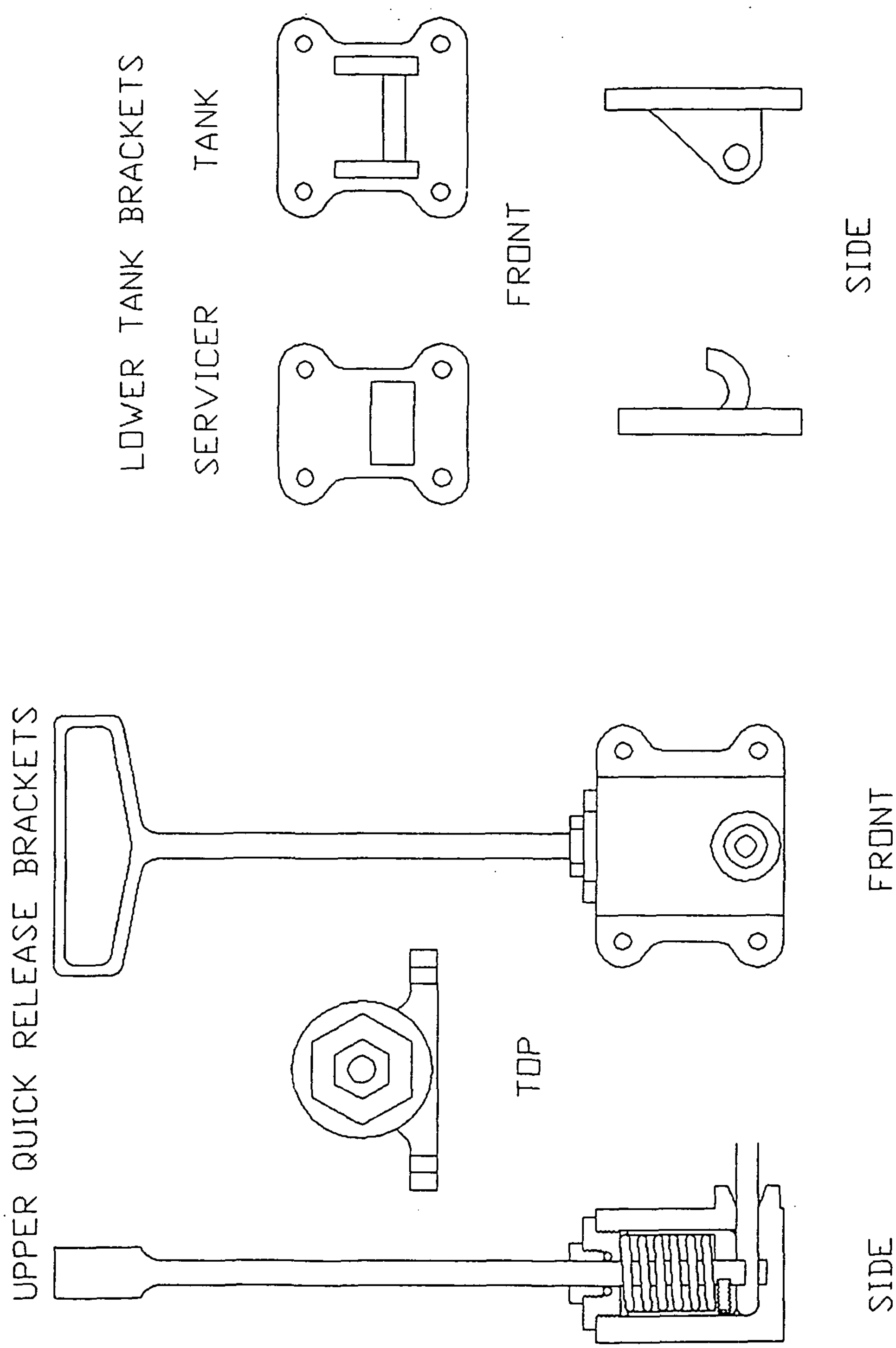

Figure D.1. Detailed Quick Release Drawing.

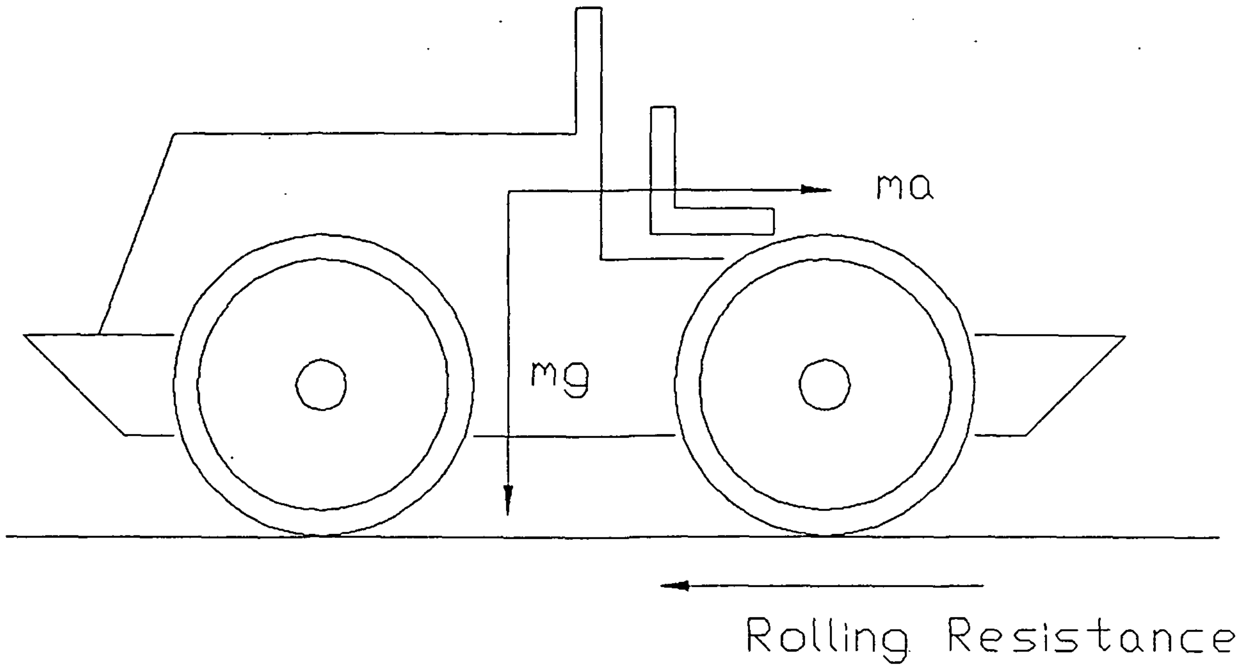

Figure E.1. Free-Body Diagram.

Appendix E. Calculating Forces on a Package on the Trailer

The free-body diagram of a package on the trailer is shown in Fig. E.1 .

Using Newton's Second Law, $F = ma$, as applied to the free-body diagram above:

$$\Sigma F_x = ma_x, \quad \text{and} \quad \Sigma F_y = ma_y$$

These equations result in the following relationships:

$$F_f = ma_x \tag{1}$$

$$N = mg \tag{2}$$

$$F_f = \mu N \tag{3}$$

where F_f is the force due to friction, m is the mass of the cargo parcel, a_x is the acceleration of the trailer, N is the normal force exerted by the package on the trailer, g is the acceleration due to gravity on the moon, and μ is the coefficient of friction between the cargo parcel and the surface of the trailer. By rearranging the above equations, the minimum coefficient of friction to keep the parcel from sliding may be determined.

$$\mu = \frac{ma_x}{mg} = \frac{a_x}{g} = \frac{0.278 \text{ m/s}^2}{1.635 \text{ m/s}^2} = 0.17$$

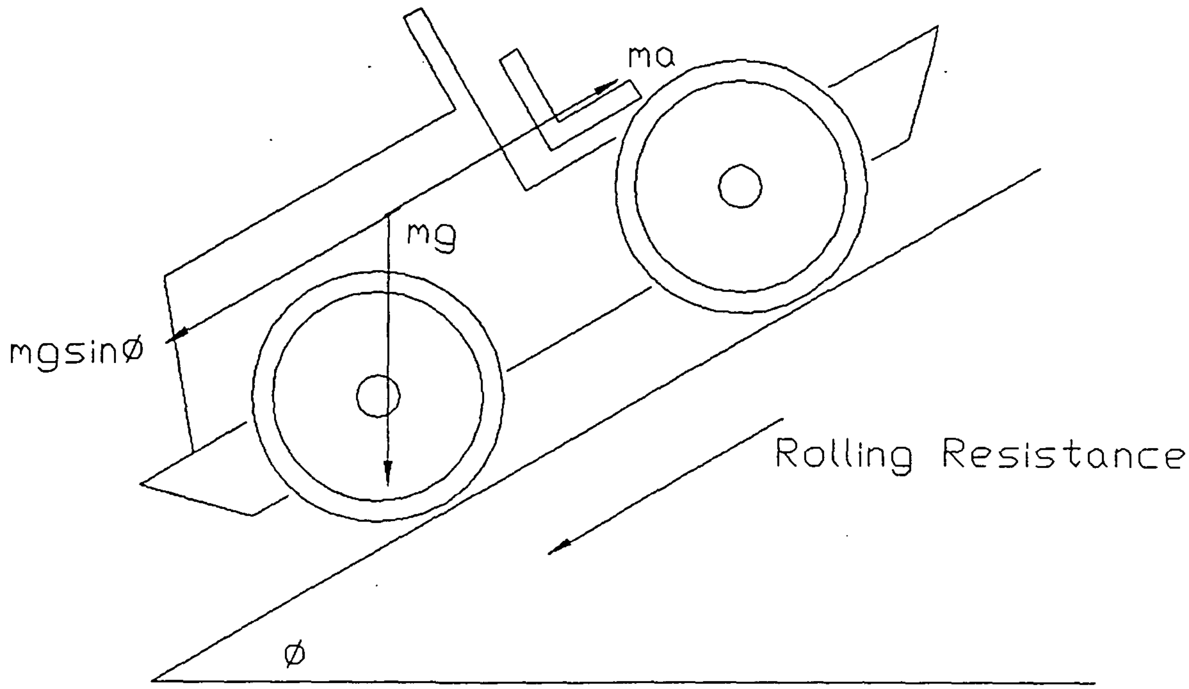

Figure F.1. Free- Body Diagram of the Prime Mover Climbing a 30° Slope.

Appendix F. Power Calculations For the Prime Mover

For the simple case of the transport system with a total mass of 7000 kg travelling out to the pad with a constant velocity of 4 km/hr, the power required my be calculated using the following relation:

$$P = F.S. \times C_R \times m \times V \tag{1}$$

where P is the power, $F.S.$ is the factor of safety C_R is the approximate average rolling resistance factor of 0.08 $\frac{\text{W}}{\text{kg km/hr}}$, m is the mass of the system, and V is the velocity that the transport system is travelling. So, the power required for this case is detemined to be:

$$P = 2 \times 0.08 \frac{\text{W}}{\text{kg km/hr}} \times 7000 \text{ kg} \times 4 \text{ km/hr} = 5.0 \text{ kW}$$

The case involving the Prime Mover accelerating up a 30° incline, or travelling at a constant velocity, the calculation is a little more involved. The free-body diagram for these scenarios ma be seen in Fig. F.1 . While the Prime Mover is travelling at a consant velocity, the acceleration forward will be zero. The Power required will be determined by two different criterion; the power reuired to overcome the rolling resistance (1), and the power requried to propel the Prime Mover up the slope at a velocity of 10 km/hr. These relations are shown below:

$$P_T = F.S. \times (P_R + P_S) \tag{2}$$

where P_T is the total required power, P_R is the power to overcome rolling resistance as defined by (1), and P_S is the power required to climn the slope defined below:

$$P_S = F.S \times F \times V = F.S \times mg \sin\theta \times v \; eqno(3)$$

The power is determined to be:

$$P_T = 2 \times ((3000 \text{ kg})(1.635 \text{ m/s}^2)(\sin 30)(2.78 \text{ m/s}) +$$

$$(0.08 \frac{W}{kg\ km/hr})(3000\ kg)(10\ km/hr) = 18.44\ kW$$

If the Prime Mover is accelerating up the slope at its maximum acceleration of 0.278 m/s^2, the power calculation is essentially the same, but the additional power required to accelerate up the slope also needs to be considered. Thus the power for the rolling resistance, P_R is the same as before. The power required to accelerate the Prime Mover up the slope is shown below:

$$P_S = F.S. \times F \times V = F.S. \times V(ma + mg \sin\theta) \tag{4}$$

Using equation (3), the total power required is:

$$P_T = 2 \times (((3000 \text{ kg})(0.278 \text{ m/s}^2) + (3000 \text{ kg})(1.635 \text{ m/s}^2)(\sin 30)) \times 2.78 \text{ m/s}$$

$$+(0.08 \frac{W}{kg\ km/hr})(3000\ kg)(10\ km/hr)) = 23.2\ kW$$

Appendix G. Propulsion System

Calculation of energy required to reliquify hydrogen gas.

$$Q = mC_p\Delta T + mH_{vap}$$

$$Q = (1.984kg)(11.00\frac{kJ}{kgK})(63K - 20K) + (1.984kg)(445.87)\frac{kJ}{kg}$$

$$Q = 1823.04kJ$$

$$P = \frac{Q}{4(24)(3600)}$$

$$P = 5.0watts$$

H.1 Liquid Hydrogen Tank Dimensions

Calculation of radius of liquid hydrogen tank.

$$V = (\frac{0.040kg/kWh}{70.86kg/m^3})(3kW)(7days)(24h)$$

$$V = .2845m^3$$

$$V = \frac{4}{3}\pi r^3$$

$$r = 0.408m$$

H.2 Liquid Oxygen Tank Dimensions

Calculation of radius of liquid oxygen tank.

$$V = (\frac{0.320kg/kWh}{1141.7kg/m^3})(3kW)(7days)(24h)$$

$$V = 0.1413m^3$$

$$r = 0.323m$$

H.3 Water Storage Tank Dimensions

Calculation of radius of water storage tank.

$$V = (\frac{0.360kg/kWh}{1000kg/m^3})(3kW)(7days)(24h)$$

$$V = 0.181m^3$$

$$r = 0.351m$$

H.4 Tank Thickness

Calculation of tank thickness

$$\sigma_{y,max} = \frac{p(D_i + t)}{2t}$$

$$45kpsi = \frac{1kpsi(33in + t)}{2t}$$

$$t = 0.37in = 0.942cm$$

Appendix I. Thermal Calculations

I.1 Weighted Property Index

Quality	Thermal Performance	Reliability	Weight	TOTAL
Scaling	0.2	0.4	0.4	
Tube	70	50	62	
Radiator	14	20	25	64
#1 Heat	60	90	54	
Pipe	12	36	22	70
#2 Heat	80	100	56	
Pipe	16	40	23	79
#3 Heat	100	100	100	
Pipe	20	40	40	100

I.2 Prevention of Meteorite Threat

The externally located radiators are prone to damage from meteorites bombarding the lunar surface. To protect the radiator from a puncture, the exposed surface is armored. The following calculation determines the thickness of the armor required to protect the radiator from failure. A Poisson distribution using the NASA meteoroid model predicts the probability of impact by n particles of mass m or greater [62]

m	Mass of particle	2.6E-4 g
A	Exposed area	$4m^2$
τ	Exposure Time	180 days
P	Probability of an impact	.99
N_t	Mass flux of particles	1.1E-10$\frac{meteors}{m^2 s}$
p_∞	Infinite penetration thickness	
TPT	Threshold penetration thickness	5.0mm
ρ_p	Density of meteorite	$0.5\frac{g}{cm^3}$
ρ_t	Density of radiator material	$4.85\frac{g}{cm^3}$
u_p	Meteorite velocity	$20\frac{km}{s}$
S_t	Titanium property factor	3.8

Probability of impact by 1 meteoroid of mass m or greater

$$P = e^{N_t A\tau}(N_t A\tau)$$

Where

$$log_{10}(N_t) = -14.37 - 1.213\ log_{10}m$$

Infinite-Penetration Thickness

$$p_\infty = \left(\frac{81}{8\pi}\frac{\rho_p}{\rho_t}m_p u_p^2\right)^{\frac{1}{3}}\frac{1}{S_t^{\frac{1}{3}}}$$

The thickness required to provide protection from penetration. The plate may spall, but will not perforate.

$$TPT = 1.5p_\infty$$

I.3 Final Tank Temperature

The boiled off hydrogen stored in the tank is subjected to a radiative flux from surrounding objects. The final temperature of the hydrogen is calculated by iterating a series of steady state problems. The hydrogen stored in the tank is broken up into two lump masses. One mass is the hot hydrogen already in the tank, and the second is the hydrogen which enters the tank in one day. First, the temperature rise in each mass due to the radiative flux is calculated. Then, the final temperature resulting from the two fluids reaching equilibrium is determined.

Determination of the heat flux passing through the insulation.

$$q_{12} = \frac{1}{N}\frac{\sigma A(T_1^4 - T_2^4)}{\frac{2}{\epsilon} - 1}$$

ϵ	emissivity	.03
T_1	External Tank Temp.	383K
T_2	Internal Tank Temp.	50K
A	Surface Area of Tank	3.14 m^2
N	Number of Layers	450
σ	Stefan-Boltzmann Const.	5.67 E-8 $\frac{W}{m^2K^4}$

Temperature change for each mass of gas due to above heat flux

$$q_{12}t = mc_p\Delta T$$

$$m = \dot{m}t$$

The two masses are then allowed to come to equilibrium and the temperature is found using:

$$m_H c_p(T_H - T_{new}) = m_C c_p(T_{new} - T_C)$$

$$\frac{m_H}{m_C}[T_H - T_{new}] = [T_{new} - T_c]$$

$$T_{new} = \frac{\frac{m_H}{m_C}T_H + T_C}{\frac{m_H}{m_C} + 1}$$

$$\frac{m_H}{m_c} = No.\ of\ days$$

m_H	Mass of Hot Gas	
m_C	Mass of Cold Gas	
T_H	Temperature of Hot Gas	
T_C	Temperature of Cold Gas	50K
T_{new}	Final Gas Temperature	

A BASIC program follows which was written to iterate the temperature increase over the fourteen 24 hour periods that the sun is out.

```
REM                              JOHN KYNOCH
REM                        AEROSPACE DESIGN PROJECT

REM      THIS PROGRAM ITERATES THE TEMPERATURE RISE IN THE HYDROGEN
REM      STORAGE TANK ASSUMING THE HEAT TRANSFER FROM THE
REM      SURROUNDINGS IS CONSTANT AT THE MAXIMUM OF 11,212 J/DAY.
REM      THE HYDROGEN IS THEN CONSIDERED IN TWO LUMP MASSES, THE
REM      HOT HYDROGEN STORED IN THE TANK, AND THE COLD HYDROGEN ENTERING
REM      THE TANK EACH DAY. THE TEMPERATURE RISE DUE TO THE HEAT FLUX
REM      IS FIRST DETERMINED FOR EACH MASS. THEN, THE FINAL TEMPERATURE
REM      RESULTING FROM THE TWO QUANTITIES OF FLUID REACHING
REM      EQULIBRIUM IS THEN FOUND.


cp! = 11000!
told! = 50!
flux! = 11212!
rate! = 2! / 14!
day! = 1!


CLS
PRINT , "OUTPUT TEMPERATURE FOR EACH DAY"
PRINT

DO UNTIL day! = 15!
        oldmass! = rate! * day!
        told! = told! + flux! / (oldmass! * cp!)

        IF day! > 1! THEN
                tnew! = 50! + flux! / (rate! * cp!)
                told! = (day! * told! + tnew!) / (day! + 1)
        END IF

        PRINT , told!
        day! = day! + 1!
LOOP
REM -----------------------------------------------------------------
```

I.4 Tank Pressure Determination

The hydrogen boiled off from the lander continuously builds up in the storage tank where the pressure builds.

$\dot{m}$	mass flow	$.142kg/day$
$\dot{v}$	volume flow	$2\frac{l}{day}$
ρ_g	density of H_2 gas	
ρ_l	density of H_2 liquid	$70.97\frac{kg}{m^3}$
M	Total mass	1.99 kg
r	Radius of tank	0.5 m
t	time tank is filled	14 days
T_f	Final temperature of tank	63K
R	Gas constant for H_2	$4124.2\frac{J}{kgK}$
P	Final gas pressure	0.98 MPa

The boiloff rate from the lander is given as 2 liters every 24 hour period. Converting to mass flow:

$$\dot{m} = \dot{v}\rho_l$$

Total mass after 14 days of boiloff.

$$M = \dot{m}t$$

Volume of storage tank

$$V = \frac{4}{3}\pi r^3$$

Density of gas in tank

$$\rho_g = \frac{M}{V}$$

Pressure of gas in storage tank after 14 days

$$P = \rho_g R T_f$$

I.5 Reliquification Requirements

During the lunar night, the reliquification system removes the heat added by the sun, and the energy necessary for a state change to occur.

m	final mass	1.99 kg
c	specific heat of H_2	$11\frac{kJ}{kgK}$
ΔT	Temp. change to 20K	43 K
h_f	heat of fusion	$454\frac{kJ}{kg}$
q_{out}	energy removed	5.W

$$q_{out} = \frac{mc\Delta T + h_f m}{t}$$

I.6 Pump Power Requirement

The fluid loop used to cool the fuel cells requires a pump to move the water around in the system. The calculation for the pump size follows.

$$P = \frac{h_t g \dot{m}}{e}$$

$$h_t = h_f + \Delta z$$

$$h_f = f \frac{L}{d} \frac{V^2}{2g}$$

$$V = \frac{\dot{m}}{\rho A}$$

$$A = \frac{\pi d^2}{4}$$

P	Power Requirement	34W
h_t	total head	5.73m
g	gravitational constant	$1.6 \frac{m}{s^2}$
$\dot{m}$	Mass flow rate	$0.55 \frac{kg}{s}$
e	Pump Efficiency	15%
h_f	Friction head	2.73m
Δz	Height Change	3m
f	Friction Factor	0.04
L	Pipe Length	11m
d	Pipe Diameter	.03m
V	Velocity of water	$0.78 \frac{m}{s}$
ρ	Density	$998 \frac{kg}{m_3}$

I.7 Radiator Sizing Calculation

The radiator receives 600W of heat from the fluid loop at a temperature of 50 C.

$$\dot{E}_{in} - \dot{E}_{out} = 0$$

$$\frac{1}{Aq_i n + \alpha G_{solar} - \sigma\epsilon(T_R^4 - T_S^4) = 0}$$

$$q_{in} + A\alpha G_{solar} - A\sigma\epsilon(T_R^4 - T_S^4) = 0$$

$$A = \frac{q_{in}}{-\alpha G_{solar} + \sigma\epsilon(T_R^4 - T_S^4)}$$

q_{in}	Heat Rate from Loop	600W
G_{solar}	Solar Flux	1360 $\frac{W}{m^2}$
α	Absorptivity of radiator	.16
ϵ	Emissivity of radiator	.97
σ	Stefan Boltzmann Constant	5.67E-8 $\frac{W}{m^2K^4}$
T_R	Radiator Temperature	323K
T_S	Effective Sky Temperature	4 K
A	Radiating Area	1.57 m^2

I.8 Lotus Program

A lotus program was written which contained calculations for the final pressure and temperature in the hydrogen tank,the reliquification energy required, radiator sizing, armor thickness, temperatures and flow rates of the active system, and pump calculations. Using a spreadsheet to perform the calculations enabled the designer to quickly change the system parameters without having to perform laborious calculations. This proved extremely valuable for this project, due to the number of people working on it, and the frequent changes made by other members. The spread sheet follows.

```
------------------------------     --------------
VARIABLE                           VALUE
------------------------------     --------------
VOL FLOW FRM LNDER [L/DAY]                      2
DENSITY HYDROGEN [KG/M^3]                   70.97
Cp HYDROGEN [KJ/Kg.K]                          11
RADIUS OF TANK [M]                             .5
VOLUME OF TANK [M^3]                          .52
SURFACE AREA OF TANK [M^2]                   3.14
MASS FLOW FROM LANDER [KG/S]           1.6428e-06
GAS CONST R HYDROGEN [J/Kg.K]                4124
TOTAL MASS ACCULUMILATED [Kg]                   2
HEAT OF VAPORIZATION [KJ/Kg]                  454
BOILING TEMP HYDROGEN [K]                      20





------------------------------     --------------
VARIABLE                           VALUE
```

------------------------------	--------------
TIME OF STORAGE [DAYS]	14
EXTERIOR TEMPERATURE [K]	383
INITIAL GAS TEMP [K]	50.00
EMISSIVITY OF MLI	0.030
NUMBER OF LAYERS OF MLI	450
WORK SPACE 'X'	5.916e-06
WORK SPACE	1209600
FINAL GAS TEMP [K]	63
FINAL TANK PRESSURE [Pa]	986581
FINAL TANK PRESSURE [PSI]	143
TIME FOR RELIQUIFICATION [H]	96
ENERGY REQUIRED FOR RELIQ [KJ]	1842
HEAT REMOVED TO RELIQ [W]	5
------------------------------	--------------
VARIABLE	VALUE
------------------------------	--------------
RADIATOR SIZING *********************************	
HEAT RATE 4 REJECT [W]	600
SOLAR FLUX [W/m^2]	1360
ABSORPTIVITY OF RADIATOR	.30
EMISSIVITY OF RADIATOR	.93
STEFAN-BOLTZ CONST[W/m^2 K^4]	5.67e-08
RADIATOR SURFACE TEMP [K]	323
DEEP SPACE TEMP [K]	4
RADIATOR AREA	3.62
METEOR THREAT ***************	***************
PROBABILITY OF IMPACT [%/100]	.99
EXPOSURE TIME [DAYS]	180
DENSIT OF RAD(TITANIUM)[g/cm^3]	4.85
THICKNESS OF ARMOR [mm]	5
MAX PARTICLE MASS [g]	2.60e-04
WORK SPACE	-3.59
METEOR FLUX [#/m^2 s]	1.10e-10

WORK SPACE	1.01	
PROBABILITY OF FAILURE	0.006	

VARIABLE	VALUE	
MORE RADIATOR CALCULATIONS		***************
TEMP ENTERING RAD [K]	323	
MASS FLOW OF WATER [Kg/S]	.55	
Cp of WATER [J/Kg.K]	4100	
SIZE OF FUEL CELL [KW]	3	
TEMP EXITING RAD [K]	322.7	
COOLING PUMP CALCULATIONS	***************	
PUMP EFFICIENCY	0	
HEIGHT CHANGE OF PIPE [m]	3	
PIPE LENGTH [m]	11	
PIPE RADIUS [m]	0.015	
DENSITY OF WATER [Kg/m^3]	998	
DYNAMIC VISCOCITY [m^2/s]	3.27e-07	
GRAVITATIONAL ACCEL [m/s^2]	1.63	
ROUGHNESS	1.50e-04	
FLOW VELOCITY [m/s]	7.80e-01	

```
REYNOLDS #                                          17891
ROUGHNESS FACTOR                                     0.01
FRICTION FACTOR (moody diag.)                        0.04
FRICTION HEAD [m]                                    2.73
TOTAL HEAD [m]                                       5.73


PUMP POWER [W]                                         34

æ
```

I.9 Thermal Model Program

```
From:  EVAX::KYNOCH       19-APR-1991 12:27:11.59
To:  AEROSPACE
CC:
Subj:  runge kutta

c John Kynoch
c Aerospace Senior Project
c
c This program utilizes a Runge Kutta algorithm to solve
c an equation describing the  temperature distribution in
c  the servicer walls. The end conditions are a fixed temp
c and a zero slope. The wall is then cooled by a varying
c radiation along a conducting wall.  The equation is of the
c form :
c (dT^2/dx^2)-AT^4+ B = 0
c Where A is a constant
c A=(Boltz const)*(emiss)/k
c and B is
c B=(Boltz const)*(absorpt)*(383^4)/k



real y(2),c(24),w(2,9)

external fun

open(1,file='tempdis.dat',status='new')
n=2

h=0.25
```

```
imax=1./h

x=0
y(1)=383
y(2)=0

write(1,10)x,y(1)
c write(1,20)y(2)
   10   format(' x =',g15.7,'         y=',g15.7)
   20   format('    ',15x  ,'       dy/dx=',g15.7)


tol=0.001
ind=1
ier=0
nw=2

do 100 i=0,imax-1
xend=x+h
call dverk (n,fun,x,y,xend,tol,ind,c,nw,w,ier)
if (ier.ne.0)print*,'*** ier error code:',ier
if (ind.lt.0)print*,'*** ind error code:',ind
write(1,10)x,y(1)
c write(1,20)y(2)

  100   continue

end
       subroutine fun(n,x,y,f)
       real x,y(n),f(n),emis,abs,k,A,B

emis = .5
abs  = .3
k    = 177

A =  5.67E-8*emis/k
B =  5.67E-8*abs*21.52E9/k

f(1)=y(2)
f(2)= -A*(y(1))**4+B

       return
```

```
end
```

Bibliography

1. *Lunar Surface Operations Study*, Eagle Engineering, Nasa Report Number N88-25415, Dec 1, 1987.

2. Eagle Engineering,*The Design of a Conceptual Lunar Lander*,1988.

3. Eagle Engineering,*Conceptual Lunar Base Design*, 1988.

4. *Lunar Base Launch and Landing Facility Conceptual Design*, Eagle Engineering, Nasa Contract NAS9-17878, Mar 25, 1988.

5. *Conceptual Design of a Lunar Lander*, Eagle Engineering,1990 .

6. Briggs, Randall A., Sacco, Albert Jr, *Oxidation and Reduction of Ilmenite: Application to Oxygen Production on the Moon*, Worcester Polytechnic Institute

7. Unknown, *Lessons Learned and Key Technologies for the Apollo Lunar Lander and Rover*, Eagle Engineering Report 89-248, June 22, 1989.

8. Shaffer, J.R., *Lunar Transportation Facilities and Operations Study: Final Report*, McDonnell Douglas Space Systems Company, April 1990.

9. Shigley, Mitchell,*Mechanical Engineering Design*, McGraw-Hill, New York, 1983.

10. Alred, J.G., et. al.: *Lunar Surface Construction and assembly Equipment Study.* NASA Contract No. NAS9-17878. Eagle Engineering, Inc. Report No. 88-194. September 1, 1988.

11. Philips, P.G., et. al.: *Lunar base Launch and Landing Facility Conceptual Design.* NASA Contract No. NAS9-17878. Eagle Engineering, Inc. Report No. 88-178, March 1, 1988.

12. Letaw, J.R., et. al: *Radiation Shielding Requirements on Long-Duration Space Missions*, NASA Contract No. N00014-85-C-2200, Naval Research Laboratory, Report No 86-02. July 21, 1986.

13. Christiansen, E.L., *Advanced Meteoroid and Debris Shielding Concepts*, NASA's Johnson Space Center, Report No AIAA 90 1336, April 16, 1990.

14. Lawrence, R.J. *Stand-off Shields for Hypervelocity Particles*, Strategic Defense Studies Division 1, Report No DE89-016182.

15. Shiraki,K. et. al.: *Shielding Considerations for the Japanese Experiment Module, National Space Development Agency of Japan*, Report No A90-32034, April 16, 1990.

16. Davidson, Drake, and others, *Surface Transportation*, Eagle Engineering, 1989

17. Mr. Jerry J. Soil, Marshall Space Flight Center, Houston Texas

18. Phillips, Simonds, and Stump, *Lunar Base Launch and Landing Facilities Conceptual Design*, Paper No. LBS-88-196

19. Kuo, Benjamin C., sl Automatic Control Systems, 1987 by Prentice-Hall, Inc.

20. Sponsler Co.,Inc,*Electronic Instrumentation*, 9/89.

21. Sponsler Co.,Inc.,*Cryogenic Sensors*, 6/89.

22. Sponsler Co.,Inc.,*Precision Turbine Flowmeters*, 6/88.

23. cancellieri and Ravaioli, *Measurements of Optical Fibers and Devices*, 1984 by Artech House,Inc.

24. Doebelin, *Measurement Systems: Application and Design*, 1986 by McGraw-Hill,Inc.

25. Tapeswitch Corp., *Safety Mats*, 3/89.

26. Smith, William F.,*Principle of Material Science and Engineering*, Mcgraw Hill, 1986

27. Shigley, John E. and Mischke, Charles R.,*Mechanical Engineering Design 5 th ed.*, McGraw Hill, 1989.

28. Harada and Mell,*Inorganic Thermal Coatings.*, NASA Report A83-16504

29. Beer, Ferdinand P. and Johnston, E. Russell,*Mechanics of Materials*, Mcgraw Hill, 1981.

30. Cataldo, Gonzalea-Sanabria, Ghan, Manzo, and Gemeiner,*Test Results of a 60 Volt Bipolar Nickel-Hydrogen Battery*, 1987.

31. Martin, A., Meyer, M., and Boggiatto, D.,*The Design and Development of A High Performance Insulation for European Spacelab*, Proceeding of Spacecraft Thermal and Environmental Control System Symposium, Munich, October, 1978.

32. *Radiation Shielding Requirements on Long Duration Space Missions*.Severn Communication Corporation, July 21, 1986

33. Stanford, M., and Schleher, J. *Radiation Hazards to Space Construction,: Engineering, Construction and Operations in Space Proceedings of Space 88*, Albuquerque, New Mexico. 1988.

34. Interview with Kevin Shoemaker, Antennas America, Inc., March 3, 1991.

35. Interview with John Godleski, Collins Navigation, March 24, 1991

36. Interview with Bill Burnette, Collins Radio, March 10, 1991

37. CTI-Cryogenics Pamphlet, *Cryonyne Refrigeration Systems: Models 22, 350 CP, 1050 CP, 1020 CP*, 1986.

38. Warshay, Marvin and Paul R. Prokopius, *The Fuel Cell in Space: Yesterday, Today and Tomorrow*, 1989.

39. Beauchamp and Sindorf, Johnson Controls, Inc., *Cost Reductions in Nickel-Hydrogen Battery*, 1987.

40. United Technologies Hamilton Standard *SPE Fuel Cell/Electrolysis Capabilities for NASA Applications*, 1988.

41. Kahout, Lisa L.,*Cryogenic Reactant Storage for Lunar Base Regenerative Fuel Cell*, June 5, 1989.

42. United Technologies Hamilton Standard, *SPE Fuel Cell Power Plant for Hermes*, 1987.

43. Interview with Matthew J. Rosso, Jr., Vice President, Ergenics Power Systems, Inc., February 27, 1991.

44. Ergenics Power Systems, Inc., *PEM Fuel Cell Specifications*, 1989.

45. Rosso, Aldhart, and Marmolejo, *Fuel Cell News Supplement: Proton Exchange Membrane Fuel Cell for Space Station*, Vol. VII, No. 1, March, 1990.

46. Eagle Engineering, *Spacecraft Mass Estimation, Relationships, and Engine Data*, 1988, N88-29836.

47. Cryofab, Inc., *Cryogenic Equipment*,1990.

48. Beechcraft Aircraft Corp., *Long-Term Cryogenic Fluid Storage*,1986.

49. Van Wylen and Sonntag, *Fundamentals of Classical Thermodynamics*, John Wiley & Sons, 1986.

50. Baumeister, Avallone, and Baumeister, *Mark's Standard Handbook for Mechanical Engineers*, McGraw Hill, 1978.

51. CTI-Cryogenics, Refrigeration Systems, 1990.

52. Shigley and Mitchell, *Mechanical Engineering Design*, McGraw Hill, 1983.

53. Martin, Meyer,and Boggiatto, *The Design and Development of a High Performance Thermal Insulation for the European Spacelab*, 1979, N79-31301.

54. Linden, David *Handbook of Batteries and Fuel Cells*,New York: McGraw Hill Co. 1984

55. Ergenics Power System, Inc.," PEM Fuel Cell Specs.". ,1989

56. Incropera, F. P., and Dewitt, D. P. 1985. *Fundamentals of Heat and Mass Transfer*, Ninth Edition, New York: John Wiley and Son's

57. *Inorganic Thermal Control Coatings*, IIT Research Institute, Chicago, Illinois, NASA No A83 16504.

58. Avallone, E.A., Baumeister, T., *Marks Standard Handbook for Mechanical Engineers* ,Ninth Edition, New York: McGraw Hill, 1978.

59. Chapra, S.C., and Canale, R. P. 1988. *Numerical Methods for Engineers* New York: McGraw-Hill.

60. "The Design and Development of a High Performance Insulation for the European Spacelab", Spacecraft Thermal and Environmental Control Symposium, Munich Germany 1978.

61. "Design of a 50,000 watt Heat Pipe Radiator ", Harbough, W. E., and Turner, R. C. ASME No.ST-3621, 1968

62. "Design and Development of a Titanium Heat Pipe Space Radiator", Los Alamos National Laboratory, Los Alamos, New Mexico. 1982.

63. Interview with George Meyer, Product Sales manager, Thermacore Inc., Lancaster, Pa. April 1, 1991.