DESIGN OF A THERMAL AND MICROMETEORITE PROTECTION SYSTEM FOR AN UNMANNED LUNAR CARGO LANDER

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ABSTRACT

DESIGN OF A THERMAL AND MICROMETEORITE PROTECTION SYSTEM FOR AN UNMANNED LUNAR CARGO LANDER

The first vehicles to land on the lunar surface during the establishment phase of a lunar base will be unmanned lunar cargo landers. These landers will need to be protected against the hostile lunar environment for six to twelve months until the next manned mission arrives. The lunar environment is characterized by large temperature changes and periodic micrometeorite impacts.

The design team designed an automatically deployable and reconfigurable thermal and micrometeorite protection system for an unmanned lunar cargo lander. The protection system is a lightweight multilayered material consisting of alternating layers of thermal and micrometeorite protection material. The protection system is packaged and stored above the lander common module. After landing, the system is deployed to cover the lander using a system of inflatable struts that are inflated using residual fuel (liquid oxygen) from the fuel tanks.

Once the lander is unloaded and the protection system is no longer needed, the protection system is reconfigured as a regolith support blanket for the purpose of burying and protecting the common module, or as a lunar surface garage that can be used to sort and store lunar surface vehicles and equipment. The design team also constructed a model showing deployment and reconfiguration of the protection system.

Key Words: Inflatable Structures, Spacecraft Protection, Thermal Protection, Micrometeorite, Lunar Lander

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INTRODUCTION

This report presents the design of a thermal and micrometeorite protection system for an unmanned lunar cargo lander. The first section of this report discusses the project background and methodology. The second section discusses the alternative designs considered by the design team followed by a third section which discusses the design concept. Lastly, the report presents conclusions and recommendations.

The United States' National Aeronautics and Space Administration (NASA), with major facilities in Houston, Cape Canaveral, Pasadena and elsewhere, was founded in 1958 for the purpose of coordinating the nation's space development efforts. One of NASA's long term goals is the manned exploration of space. The research necessary for such exploration is being accomplished through the coordinated efforts of the United States government, the private aerospace industry, and the universities.

The Universities Space Research Association (USRA), headquartered in Houston, Texas, was established in 1969 by the National Academy of Sciences. It is a consortium of universities dedicated to the exploration and development of space. Through the USRA, NASA currently sponsors space related projects at universities throughout the country. As one of the participating members of the USRA, The University of Texas at Austin is currently conducting space research. More specifically, the Mechanical Engineering Department is conducting preliminary conceptual design of various projects essential to the establishment of a lunar base, which is an intermediate step toward the manned exploration of space. The design team designed a thermal and micrometeorite protection system for a lunar landing vehicle.

BACKGROUND

Presently, the primary benefit of a lunar base is the possibility of extracting oxygen from the lunar soil (LUNOX) and using such oxygen as fuel for further exploration of space. In the present scenario, the first few missions to the moon will alternate between manned and unmanned cargo missions. The first mission will consist of unmanned lunar cargo landers carrying supplies essential for the establishment of the lunar base (see Figure 1). These landers will have to remain unattended on the lunar surface for a period of six to twelve months until the next manned mission arrives. During the unattended phase, the cargo landers must be protected against the hostile lunar environment.

The lunar environment is characterized by temperatures ranging from -171 to +111 degrees Celsius (-276 to +232 degrees Fahrenheit) and periodic impacts by micrometeorites with velocities averaging 20 kilometers per second (12.5 miles per second). [6] Thermal effects can cause internal stresses and damage to equipment in the lander while micrometeorites can cause surface damage. Therefore, a system which can protect the lander against the lunar environment is needed.

Transporting material to the moon is expensive—approximately one million dollars per pound.[2] In order to minimize waste of material, the protection system will be reconfigurable into an alternate use.

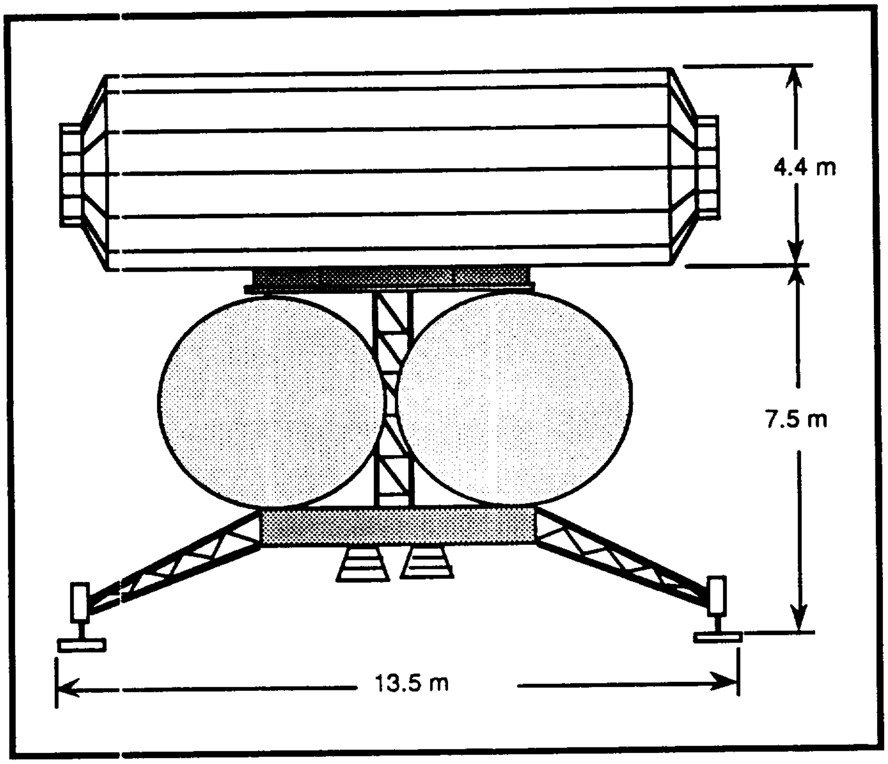

Figure 1: BASELINE LUNAR LANDING VEHICLE

PROJECT REQUIREMENTS

The requirements of the project are as follows:

1. To design a thermal and micrometeorite protection system for an unmanned lunar cargo lander.

2. To construct a demonstration model of the protection system showing deployment and reconfiguration into an alternate use.

PROJECT CRITERIA

The design criteria for the protection system are listed below:

1. The protection system must protect the lunar cargo lander from temperature extremes caused by thermal radiation.

2. The protection system must protect the lunar cargo lander from impact damage caused by micrometeorites.

3. The size, volume, and weight of the system must be minimized.

4. The protection system should automatically deploy after landing.

5. In order to minimize the waste of materials, the system should be reconfigurable into an alternate use.

PROJECT METHODOLOGY

The design team accomplished the project requirements in five stages: research, synthesis, evaluation, analysis, and construction. Since the synthesis stage was an iterative process, some of these stages inevitably occurred simultaneously.

The research stage included literature research of general background information, micrometeorite protection, thermal protection, and deployment systems. In addition, the design team periodically discussed the project's progress, goals, and merits with Dr. Wallace Fowler of the Department of Aerospace Engineering at The University of Texas at Austin.

The synthesis stage began with a brainstorming session. The concepts generated by the brainstorming session were evaluated in order to arrive at viable alternative designs.

After preliminary analysis, the design team evaluated the alternative designs for advantages and disadvantages. The evaluation considered factors such as weight, volume, reliability, and reconfiguration utility.

In the analysis stage, the design team performed a detailed analysis of the final design solution. The goal of the analysis stage was to ensure feasibility, safety, and reliability of the design, and to make recommendations leading to an optimal final design solution.

In order to meet the project requirements and demonstrate the operating principle, the design team constructed a demonstration model showing deployment and reconfiguration of the protection system. This model was integrated with other lunar base models constructed at the University of Texas at Austin.

The next section of this report discusses the alternate designs considered by the design team.

ALTERNATE DESIGNS

The design team conducted a series of brainstorming sessions and came up with a number of alternate designs for the deployment and reconfiguration of a thermal and micrometeorite protection system for a lunar cargo lander. The team also performed a literature search on alternative materials for micrometeorite impact protection and thermal protection. The alternative designs presented in this chapter were selected after a preliminary evaluation to ensure that the designs are feasible. The base line configuration, below, was used to picture the areas of the lander that need to be covered (see Figure 2). [13]

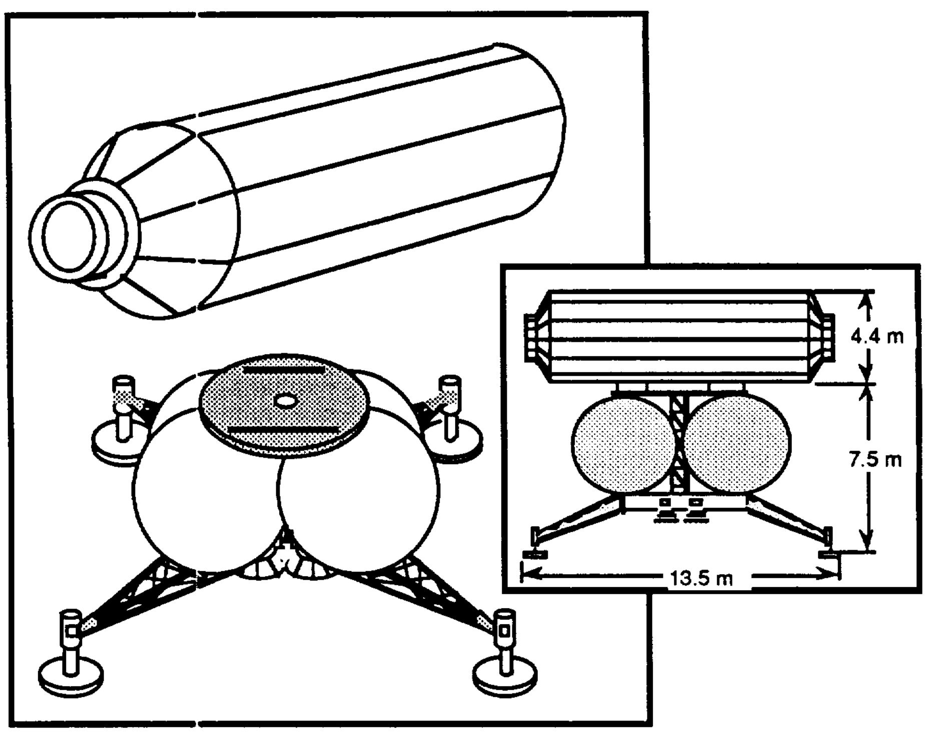

Figure 2: SEPARATED VIEW OF HABITAT MODULE AND LUNAR CARGO LANDER.

PROTECTION PRIOR TO LANDING

One of the alternative designs considered by the design team involves protecting the lunar cargo lander in low-Earth orbit (LEO). In this design, there would be no need for a deployment system which covers the whole lander. The only portions of the lander that are left uncovered are parts such as visual/navigational instruments which will need to be exposed during the journey, and areas such as the undercarriage of the lunar cargo lander. Once on the moon, an independent and simple deployment system could be used to protect uncovered portions. The covering for this protection system may be metallic or non-metallic.

The major advantage of such a protection system is low weight and volume; since the protection system is "tailored" to perfectly fit the lander, a minimum amount of material is used. Any critical parts which need covering after landing can be protected with smaller sub-systems. A major disadvantage of this system is that it does not easily reconfigure into an alternate use since the protective system is specifically tailored for the lunar cargo lander. Also, if the protection system is damaged during flight, it is virtually impossible to repair.

Since the lander will be traveling through space, and thus will be exposed to damaging thermal radiation, why is the lander not protected at the onset of its mission? To answer this question, one must first consider the duration of the transitory phase between LEO and the moon and the unattended phase on the moon. Recall that the lander must remain unattended on the lunar surface for up to a year while travel time to the moon will take only a few days. Secondly, current methods of spacecraft temperature control involve rotating the spacecraft so as to alternately expose different parts of the vehicle to the sun and outer space. This concept can keep the temperature within allowable ranges. Once on the lunar surface, however, spacecraft rotation is not a viable alternative. [2]

THE "ACCORDION" DESIGN

The need for a thermal and micrometeorite protection system for a lunar lander was identified by engineers from McDonnell-Douglas as part of their ongoing lunar base study. A design suggested by these engineers involves the erection of a tent–like structure by astronauts. [13]

The design team modified this design to make the system deploy automatically. The modified design uses guy wires running from the top of the cargo module to the base of the legs (see Figure 3-A). These wires are fixed in position in LEO and could be used to stabilize the cargo module on the lunar lander. The protective material itself is folded much like curtain drapes or an accordion. Small guide rings (through which the guy wires pass) are attached to the folds (see Figure 3-B). Upon landing on the moon, the protective material is released from its folded position and drawn down the guywires. The covering would be drawn by wires connected to a small motor. The material in this configuration will act as a thermal shield and micrometeorite bumper. The materials considered for this system were metallic and non-metallic materials. The metallic version of this design would use hinged plates rather than a flexible fabric.

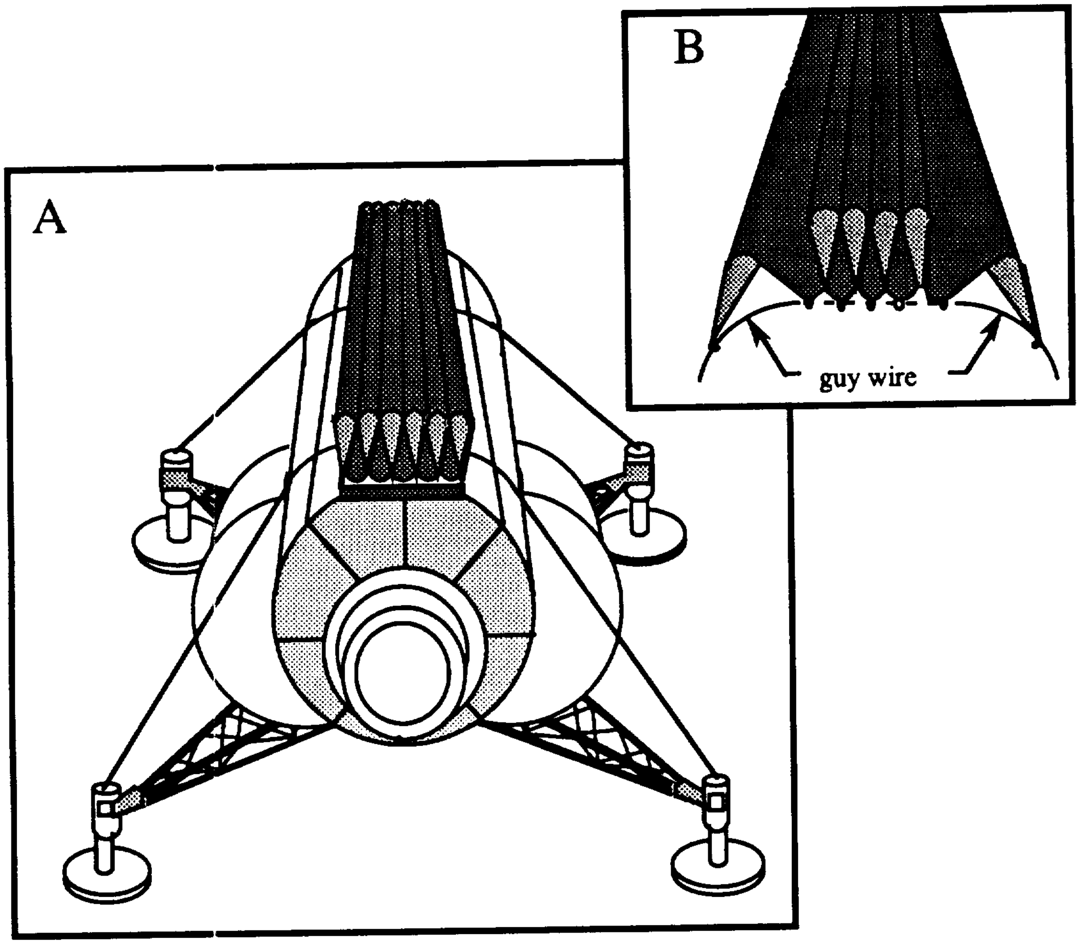

Figure 3. (A) ACCORDION PROTECTION CONCEPT WITH (B) DETAILED VIEW OF GUIDE WIRE AND GUIDE RINGS.

The major advantages of the system are that it is lightweight (for a non-metallic material) and can be easily reconfigured, for example, into a large tent. By providing insertion seams in the protection system and inserting light support rods (e.g. graphite rods) , the system could easily be reconfigured into a protective tent or garage for lunar surface vehicles.

One disadvantage of this design is that it does not cover all sides of the lander—two sides are left exposed. With a north to south orientation of the common module, however, these sides will never be subject to any direct solar radiation. In addition, since most micrometeorites impact at angles close to normal, the probability of a side impact at the exposed sides might be sufficiently low.

NON-METALLIC INFLATABLE STRUCTURES

Since the weight of the protection system is of major concern, light non–metallic inflatable structures seem to be a natural way to deploy large structures which must be packed in small volumes. The design team has spent much effort on researching the possible use of such materials. Several design concepts implementing different material combinations and deployment methods have been examined. Examples of non-metallic materials that can be used for micrometeorite shielding are Kevlar, Mylar, fiberglass, and Nylon. Foam fillers can be used as rigidizing agents and as micrometeorite shielding to further reduce weight. [16]

Inflatable structures are relatively easy to deploy, lightweight, and reliable. The energy necessary for deployment is readily available without requiring any complex apparatus. Lunar lander propellant is used to deploy the protection system. High pressure oxygen forces the material to unroll according to the same physical principle as that of a Chinese whistle (see Figure 4). Boiling-off and expansion of the liquid oxygen fuel may be accomplished with a simple expansion valve. Moreover, inflatable designs may later be rigidized with a hardenable foam or resin.

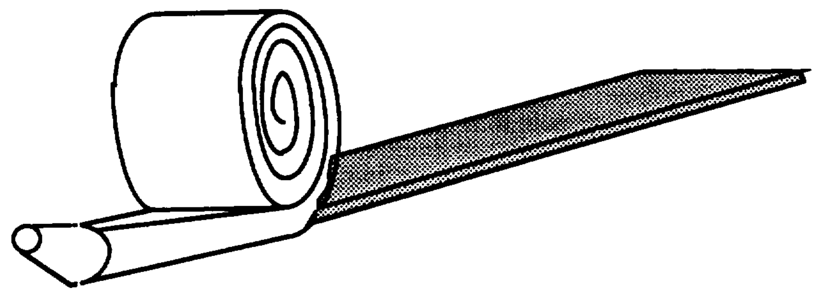

Figure 4: THE INFLATABLE SYSTEM DEPLOYS MUCH LIKE A CHINESE WHISTLE.

The design team is considering two different inflatable material structures for the protection system (see Figure 5). One structure, (A), consists simply of two layers of fabric materials coated with a gas-impermeable substance. This structure would be inflated much like an ordinary balloon. For improved impact resistance, using a multilayered structure, such as the bladder structure developed by the Goodyear Aerospace Corporation, is also possible. [16] The second inflatable structure, (B), is one in which inflatable support tubes, or struts, integrated with a fabric, are used to deploy the flexible fabric.

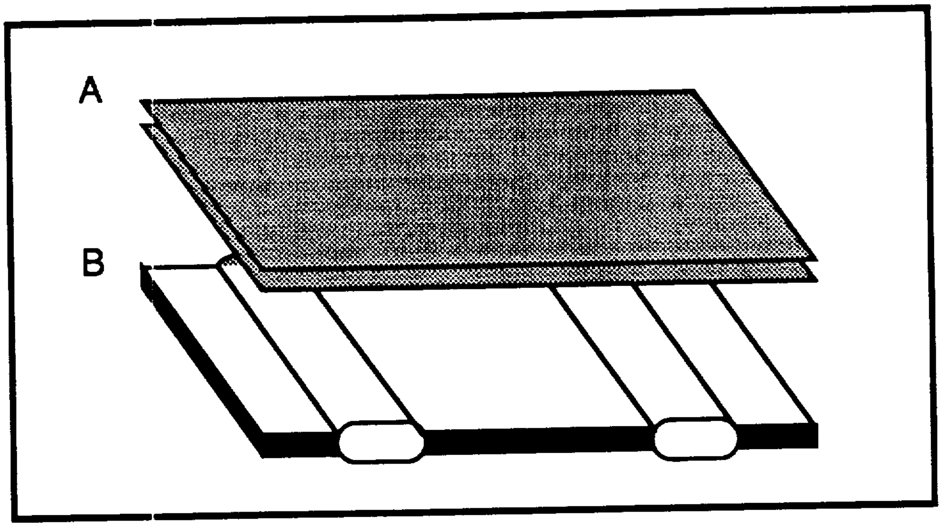

Figure 5: INFLATABLE MATERIAL STRUCTURES.
A. LAYERED STRUCTURE
B. STRUT STRUCTURE

The next section discusses two alternate deployment configurations for inflatable structures. These alternatives are the single inflatable protection system and the subsystems inflatable protection system..

<u>Single Inflatable Protection System</u>. This inflatable design is rolled and folded to fit on top of the lander module much like the "accordion" design. When deployed, it covers the entire lander vehicle (see Figure 6). This system can be reconfigured into a single garage for the storage and protection of surface vehicles and equipment. The disadvantage of the single protective system is that it has to cover a large area and is therefore relatively heavy and occupies a large storage volume. To reduce weight and volume, the protection system can be split into several subsystems.

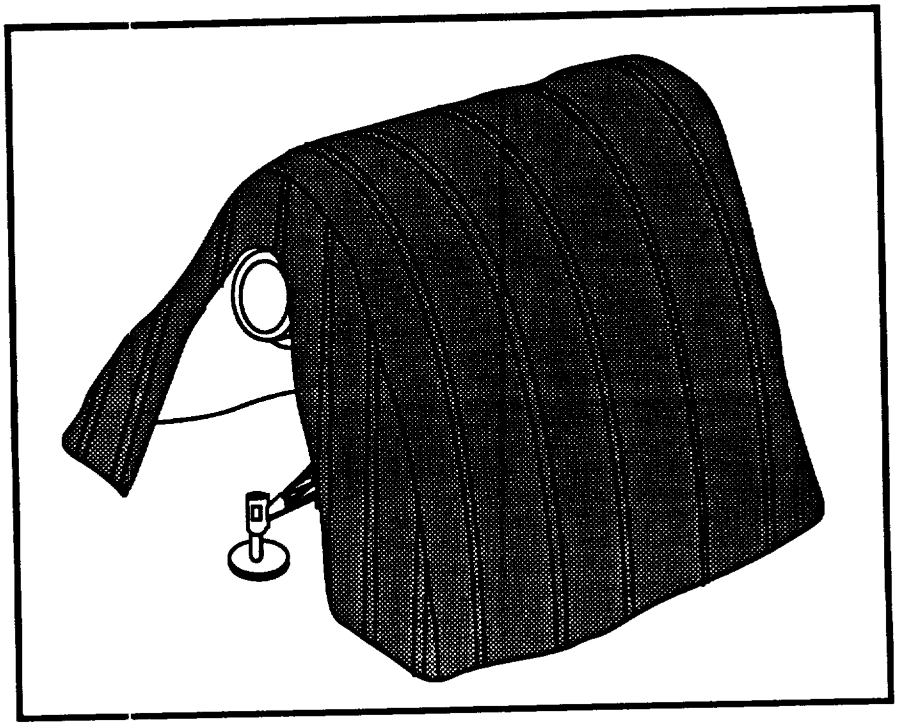

Figure 6: SINGLE PROTECTIVE SYSTEM COVERS BOTH THE LANDER AND MODULE.

Subsystems Inflatable Protection System. This alternative design separately protects the cargo module and the lander fuel tanks and engines (see Figure 7 and 8). The major advantage of separate protection is reduced weight and volume. For reconfiguration, this design may be rigidized and employed as a lunar surface garage or as permanent protection for manned modules.

The common module protection system is rolled along the axis of the module and stored on top of the common module. The lander base protection system is rolled into a toroid and fixed just below the cargo attachment platform. Pressurized air will extend the skirt over the tanks and engines. If needed, the legs of the lander may be protected either with a hardened foam, with separate inflatable shields, or with a larger protective skirt that extends over the legs.

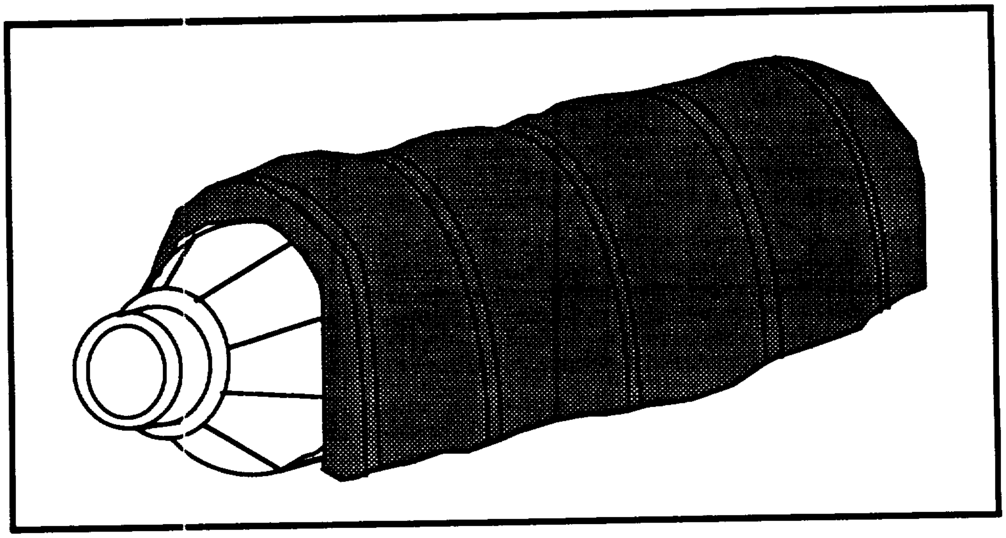

Figure 7: SUBSYSTEMS PROTECTION—MODULE.

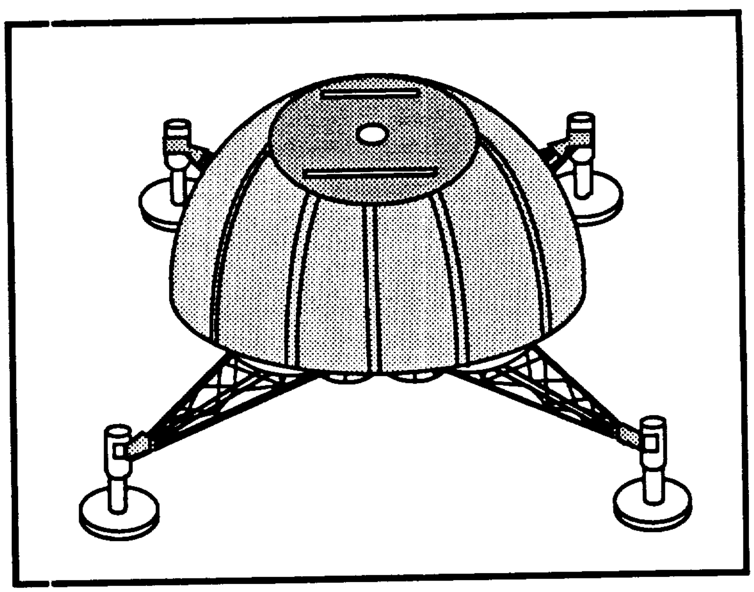

Figure 8: SUBSYSTEMS PROTECTION—LANDER.

MATERIAL ALTERNATIVES

Thermal Protection. Thermal radiation protective materials should be highly reflective, durable, and stable in the vacuum and the ultraviolet environment of space. Alternative materials for thermal protection considered by the design team were aluminized Kapton, Mylar, and Tedlar films. [18] These films are secured to the surface of the micrometeorite protection structure.

Micrometeorite Protection. Three types of materials were considered by the design team for protection against micrometeorite impacts. These are aluminum, non-metals such as Kevlar, Mylar, Fiberglass, Nylon, Nomex, and foam materials such as polyurethane foam. All of these materials, with the exception of polyurethane foam, have high strength to weight ratios. Some types of flexible foams have been proven in laboratory tests to be highly protective against hypervelocity impacts (most of the energy is dissipated into heat). [16]

The next section of this report discusses the final selection of the deployment system, materials selected, and the reconfiguration of the protection system into an alternate use.

THE FINAL DESIGN SOLUTION

The design team selected the final thermal and micrometeorite protection system after carefully weighing the advantages and disadvantages associated with each alternative design. Using several decision matrices, each member of the design team independently evaluated the merits of each design alternative (see Appendix A). The weighing factors of the decision matrix were calculated by using the method of pairs. These weighing factors were established as the averaged values assigned to each design parameter by individual team members. The final design uses the subsystem inflatable strut concept to deploy a protective covering made of several layers of aluminized Mylar spaced by a Dacron mesh. This covering serves for both thermal protection and micrometeorite impact protection.

THE SUBSYSTEMS INFLATABLE DESIGN

The design team chose the subsystems inflatable concept for the final design solution. There are several advantages gained by using inflatable structures on the moon (or in space for that matter). The argument for using subsystems protection is that a one-piece inflatable structure would be both heavier, and more cumbersome to package and deploy than several smaller inflatable systems. The design team recognizes that the final design, described in more detail latter in this report, is a single system inflatable structure. This apparent contradiction occurred because the base line lander configuration changed late into the project. The new base line configuration was proposed by another University of Texas at Austin team designing the lunar cargo lander (see Figure 9). [10] The design team concluded that a large "sub-system" is better suited to the new geometry. However, the design team recommends that the subsystems design philosophy be adopted wherever possible.

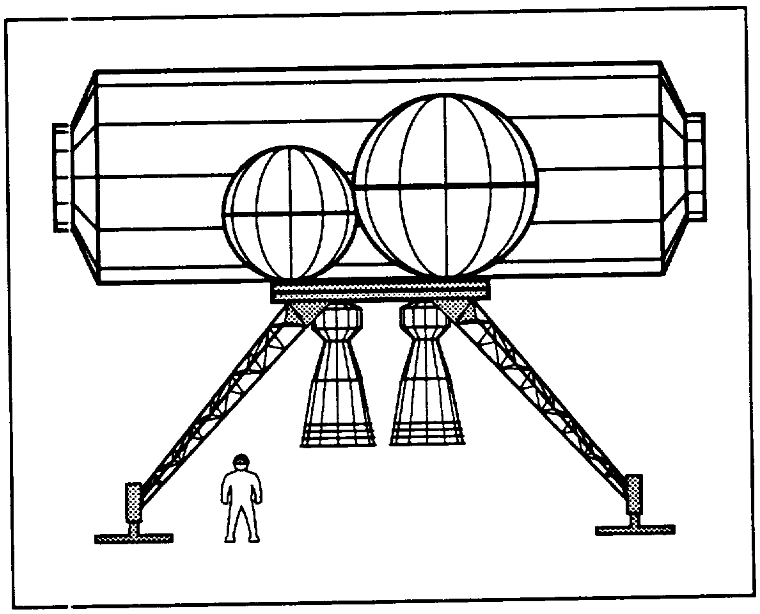

Figure 9: NEW BASE LINE CONFIGURATION.

The foremost advantage of an inflatable design is that it is extremely packageable. The volume of gas necessary to inflate the structure can be easily stored in compact cryogenic liquid tanks. Specifically, a small amount of surplus fuel from the spacecraft can be used to inflate the structure. Thus this design concept uses a readily available gas source. In addition, inflatable designs offer a very simple deployment mechanism which cannot be equalled by most contemporary mechanical devices. The merits of inflatable designs were expressed by Chow and Lin in "Structural Engineers Concept of Lunar Structures"; they concluded that self-supporting fabric membrane structures are "the optimum solution for the moon."[1]

INFLATABLE STRUTS

Inflatable struts further reduce the weight of an inflatable structure by reducing the volume which must be pressurized. Smaller volumes require less bladder material and less gas to inflate the structure. The struts serve both as structural members that add rigidity to an otherwise flaccid fabric and as a more efficient inflatable deployment device. Why is the rolled strut more efficient than a double-membrane "balloon" design? The answer is that the balloon design would require more gas (and thus exhibit more leakage), more material, and give a slower deployment response than the double-membrane balloon structure.

The strut bladder material selected by the design team was developed by the Goodyear Aerospace Corporation (GAC). The bladder material is made of a Nomex unidirectional cloth structural layer coated with Viton–B50, an elastomer. The Viton–B50 bonds the Nomex fibers and serves as a gas impermeable layer. Several plies of Nomex/Viton–B50 material can be laminated together to achieve the desired strength. This combination of materials was selected by GAC after an intensive eight month study and was certified by NASA for use in the Shuttle orbiter crew cabin and in the Spacelab module [16]

THERMAL PROTECTION

In the absence of an atmosphere, the main mode of heat transfer on the moon is radiation. The temperature which a body on the moon will attain is a strong function of its reflectivity. Therefore, it is desirable to cover the lunar cargo lander with a highly reflective surface. The design team chose to use a passive thermal control system. This means that the lander is protected only against the heat of the lunar day. The cold temperature of the lunar night was considered acceptable for design purposes. The material chosen for thermal protection was a multilayered thermal insulation (MLI) blanket (see Appendix B). [20]

The design team determined that the protection system only needs to cover the lander components located above the lander platform. The rocket engines, located underneath the platform, are shielded against micrometeorite impact. Also, the engines are designed to withstand large temperature extremes and do not need to be thermally protected.

The unprotected lander legs should pose no problem. Since vacuum occupies the volume between the loosely packed lunar regolith, the regolith is a very poor thermal conductor. Thus conduction from the lunar regolith through the unprotected legs may be assumed to be negligible. Also, reflected radiation from the lunar surface may be assumed to be negligible due to the low diffuse reflectivity (albedo) of the regolith. [19] However, if future studies suggest otherwise, the base of the legs can be coated with Teflon and the entire legs pre-coated with thermal insulation.

The thermal analysis predicts that the steady-state temperature for one layer of aluminized film insulation is -39 degrees Celsius. With multilayered insulation (12 layers), thermal protection is even better. Thus, the highest predicted temperature for the lander is -39 C during the lunar day (see Appendix C).

IMPACT PROTECTION

Since the moon is devoid of an atmosphere, any meteorites that impact on the moon do so without the retarding atmospheric effects to which meteorites impacting the Earth are subject. Consequently, there is a larger meteorite problem on the moon than on the Earth. However, the problem is not as serious as was first anticipated by the design team. Since the probability of any sizable micrometeorite impact is negligible, the design team concluded that impact protection plays a secondary role to the primary concern of thermal protection (see Appendix D). [14]

Protection against micrometeorite impact is also provided by the MLI covering. The impact energy of a micrometeorite is dissipated by the many layers of Mylar which act as a multi–wall sacrificial impact bumper (see Appendix E). The idea of using sacrificial bumper plates for meteorite protection was first suggested by Dr. Fred Whipple in the 1940s. By such a method, most of the impact energy of a micrometeorite would be dissipated after first impacting against a sacrificial bumper. The sacrificial bumper would greatly deplete the impact energy of the micrometeorite and would allow only milder impact debris to strike the underlying lander superstructure. [17]

The impact analysis was performed for an aluminum bumper which was scaled down by a factor of 7.5 to arrive at an equivalent MLI bumper thickness of 3 millimeters (12 layers of MLI) with a mass per unit area of 0.03 gram per square centimeters (see Appendix F).

FINAL GEOMETRY AND DEPLOYMENT

The protection system is rolled along a single axis of symmetry and forms two adjacent cylindrical tubes (see Figure 10). The radius of the cylinders is 28.5 centimeters. The overall weight of the protection system is approximately 320 kilograms (see Appendix G). During the voyage to the moon the protection system is stored in a protective container (see Appendix H).

After landing on the lunar surface, the protection system is released from its stored position on top of the cargo lander. Surplus liquid oxygen fuel is expanded through a valve and used to inflate the struts. Approximately 1.5 kilograms of liquid oxygen fuel will be needed for deployment (see Appendix G). Two pressure supply lines are extended from the oxygen fuel tank to both ends (a redundant path is provided for safety) of the center strut. [19] The center strut supplies gas to the remaining struts (see Figure 11).

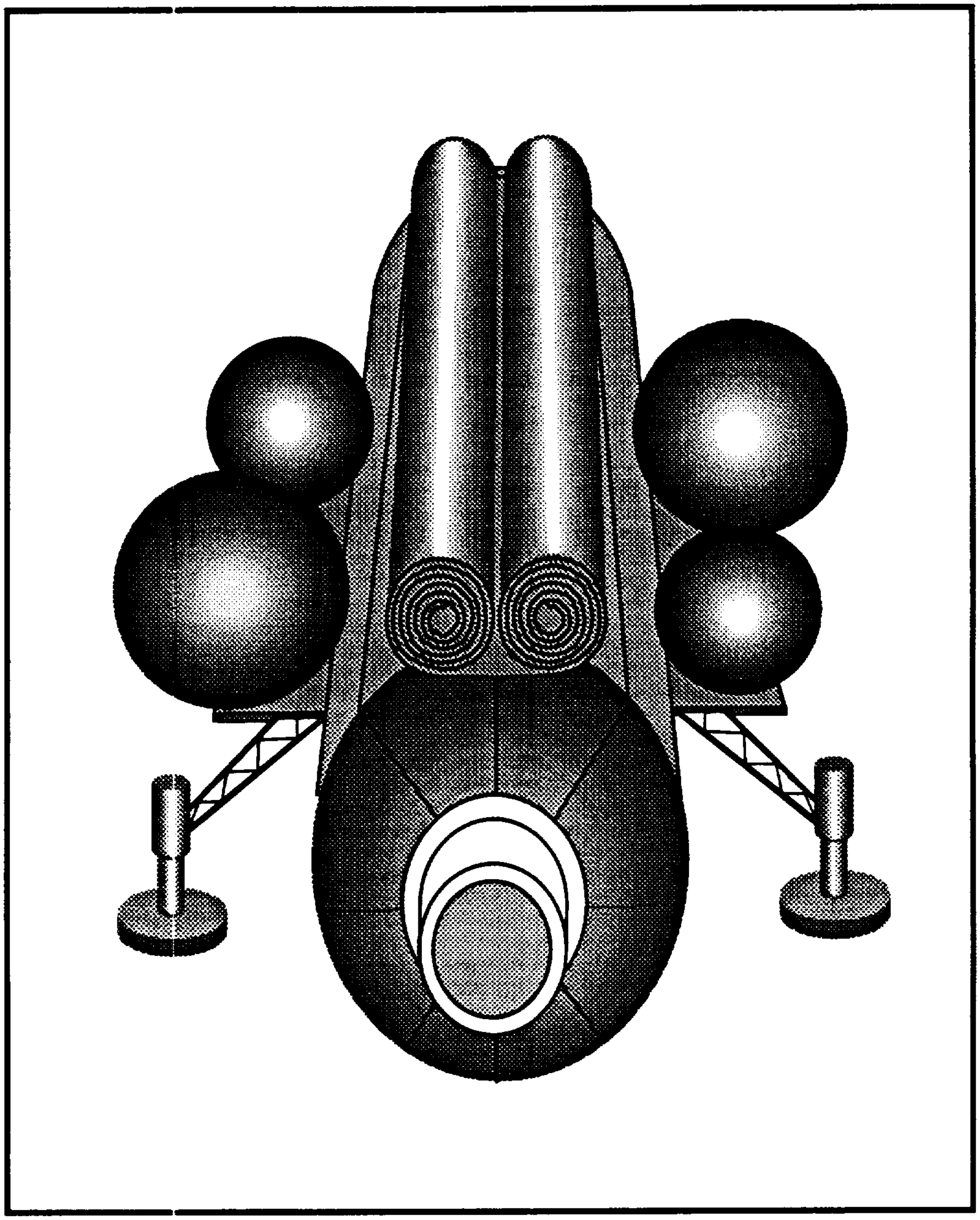

Figure 10: STORED CONFIGURATION OF THE PROTECTION SYSTEM

The protective cover is then extended over the common module and fuel tanks (see Figure 12). A rough approximation of the pressure for deployment is 0.1 bar (1.5 psi). This pressure is sufficient to deploy the protection system while, at the same time, allowing the struts to remain non–rigid. A low pressure is necessary to minimize the membrane stresses in the struts which may become brittle during the extreme cold of the lunar night.

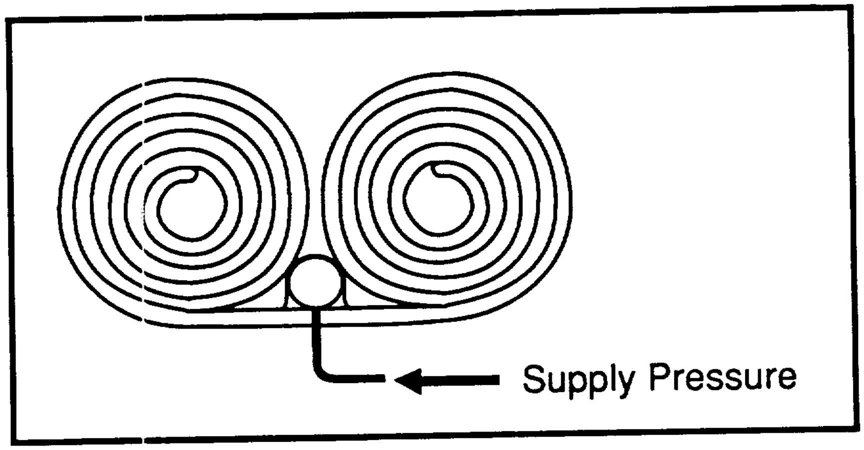

Figure 11: GAS SUPPLIED TO THE CENTER STRUT BY THE OXYGEN FUEL TANK.

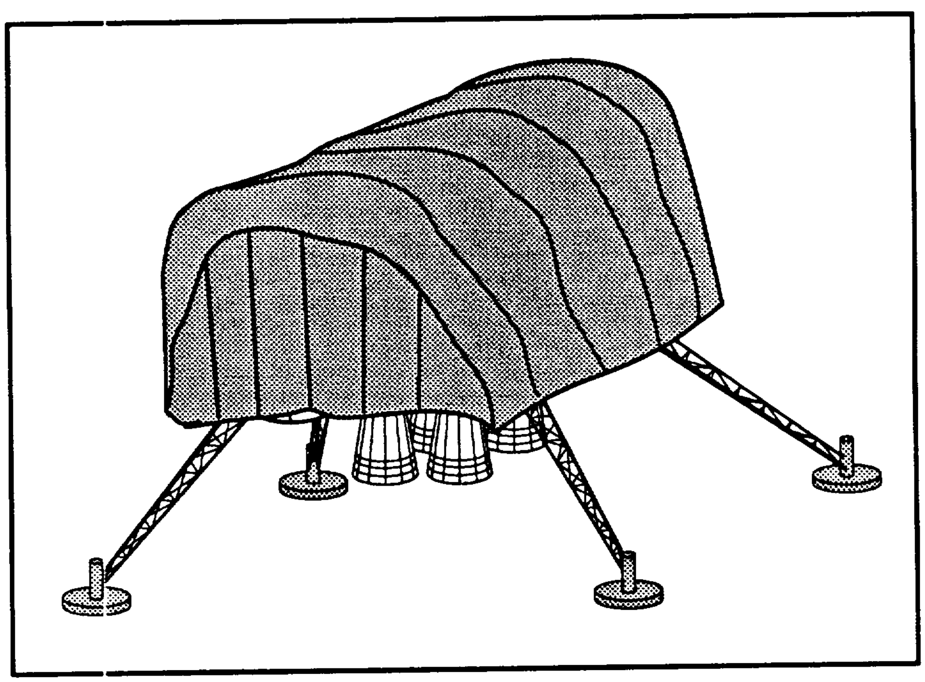

Figure 12: FULLY DEPLOYED CONFIGURATION OF THE PROTECTION SYSTEM

RECONFIGURATION

Once the thermal and micrometeorite protection system has served its initial purpose of protecting the lunar cargo lander, the system must be reconfigured into an alternate use. The protection system can be reconfigured into two alternative uses:

1. a regolith support blanket for the purpose of burying and protecting the common module (see Figure 13), or
2. a surface garage for the purpose of storing lunar vehicles and other surface equipment.

The design team felt that both of these “second uses” warranted enough importance to justify designing the protection system for either task. The regolith support blanket is simple and requires only that the material not tear under the weight of the regolith. [10] The surface garage, however, requires more critical design.

The surface garage will be necessary to store cargo and surface vehicles. The geometry of the garage will be a simple drive-through "twin tunnel" that will allow easy access (see Figure 14). If necessary, the ends of the garages can be covered with side blankets. Eight semi–circular struts and five horizontal struts provide sufficient load carrying capacity and stability to the garages. Both garages are connected by the center gas supply strut.

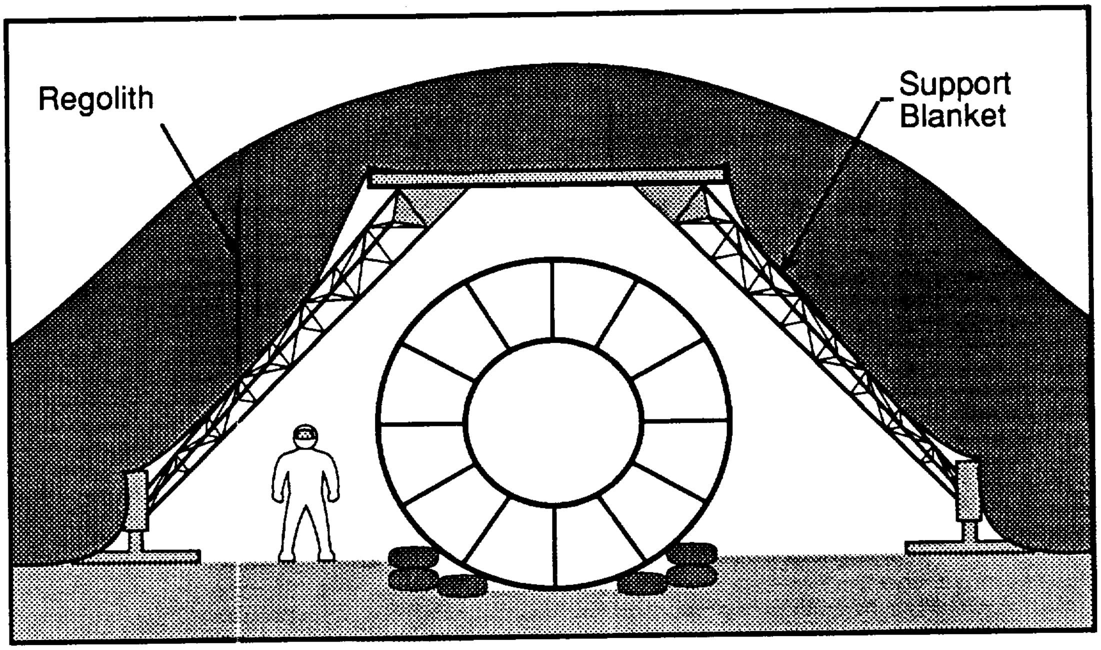

Figure 13: THE PROTECTION SYSTEM SHOWN AS A REGOLITH SUPPORT BLANKET.

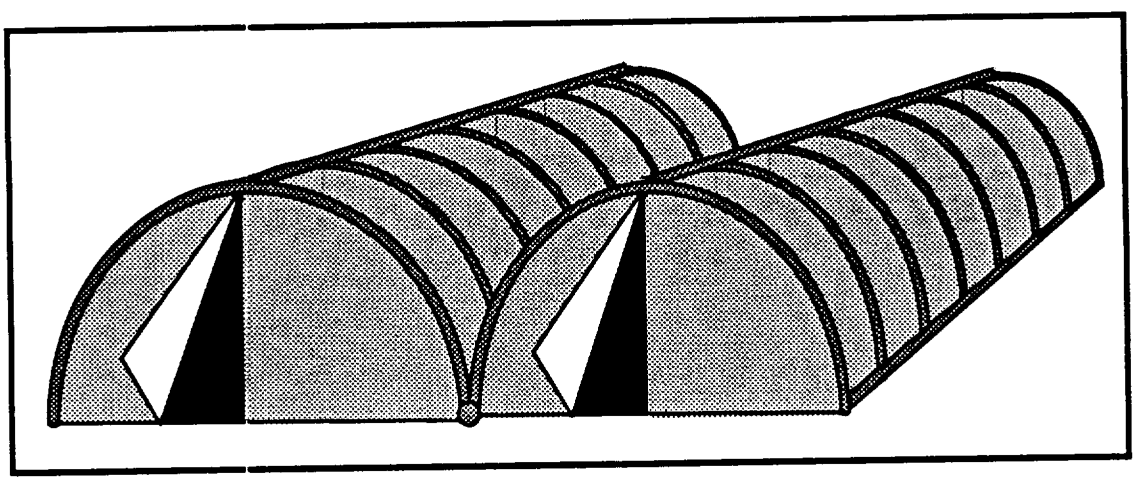

Figure 14: THE PROTECTION SYSTEM SHOWN RECONFIGURED INTO THE SURFACE "TWIN TUNNEL" GARAGE.

Since the garage requires that the inflatable strut acquire an arch shape, draw chords will be incorporated into the design. These draw chords are placed on the underside of the inflatable struts and run through either small guide rings or through a fabric sheath (see Figure 15). When the draw chords are pulled taut, the underside of the strut is compressed and the strut assumes an arch shape.

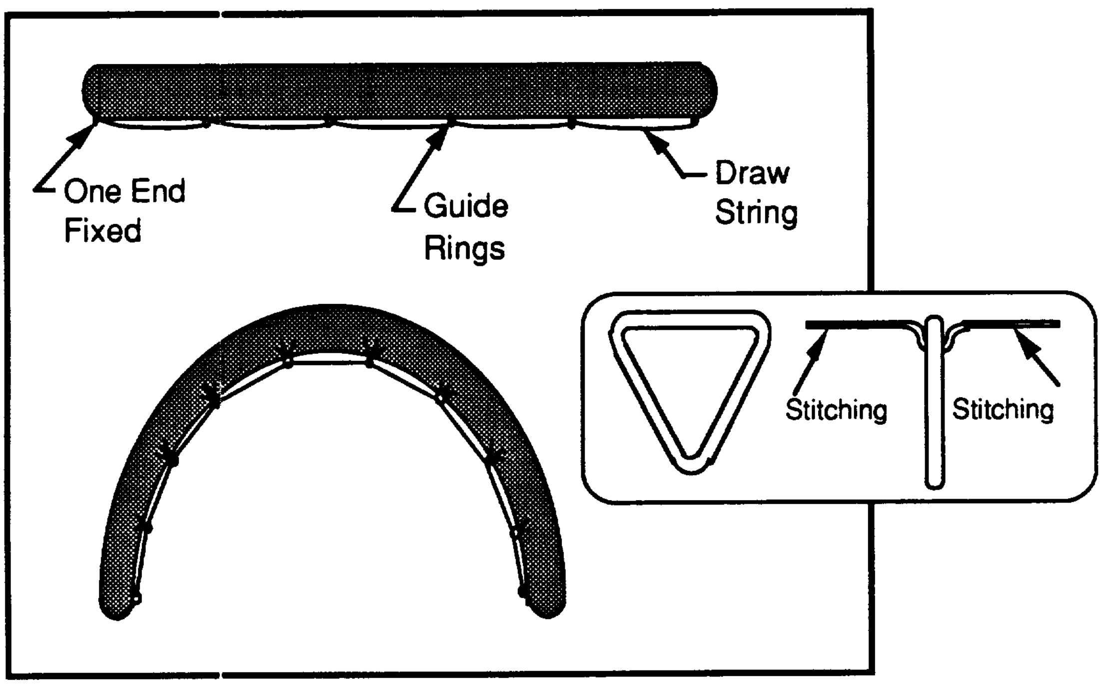

Figure 15: INFLATED STRUT SHOWING SHAPING INTO ARCH CONFIGURATION.

If future studies reveal elastomer (Viton-B50) embrittlement to be a problem, a small heating strip element may have to be incorporated into the strut design to keep the elastomer relatively warm during the lunar night. Added insulation (additional layers of MLI) around the strut will insure that the power consumption will be low. Preliminary calculations showed that a minimal amount of heating power (and thus a small battery), would be sufficient to maintain a warm temperature (-40 degrees Celsius).

ANALYSIS OF INFLATABLE STRUTS

The maximum load on the system will occur when used as a self supporting garage. The struts have to carry the full weight of the protection blanket as well as their own weight. Two pressures must be determined; the pressure that is required to prevent the struts from collapsing due to the weight of the protection blanket, and the pressure that causes rupture of the strut material.

The struts are modeled as simple pressure vessels shaped as semi–circular beams with circular cross sections (see Appendix J). Inflated structures on earth that are used for human occupation have a safety factor of four. [5] In space, a safety factor of five for manned modules is common. But since there are no human factors in the proposed protection system, a factor of four was used to determine the permissible pressure in the struts.

It was initially planned to have the protection system reconfigure into a large single garage. However, the pressure needed in the struts for this configuration was too high. Therefore, the design team decided to use the "twin tunnel" garage configuration. A total of eight struts are used. The end struts have a cross sectional radius 0.20 meters while the middle struts have a radius of 0.14 meters. The strut curvature radius is 3.25

meters and the strut material thickness is 1.5 millimeters. The Nomex/Viton-B50 strut material, which has a tensile strength of 185 mega Pascals, gives a permissible pressure range of 2.3 to 6.7 bar.

In order to minimize leakage of oxygen, the pressure in the struts should be kept low and as close to 2.3 bar as possible. According to the Goodyear Aerospace Corporation, leakage of gas from pressurized space modules at 2.0 bar can be limited to 0.5 psi (0.036 bar) per month. [16] If the pressure in the struts is 3.2 bar, the system needs refilling of oxygen approximately every two years. Alternately, a constant pressure may be maintained by using a pressure regulating device (see Appendix H)

DESIGN SUMMARY

The following table summarizes the final design of a thermal and micrometeorite protection system for an unmanned lunar cargo lander (see Table I).

Table I.
Design Summary

Deployment System Concept	Sub-Systems Inflatable Protection Design
Thermal Protection Material	12 layers of MLI
Impact Protection Material	Protection provided by the 12 layers of MLI
Reconfiguration	Regolith Support Cloth or Surface Garage
Strut Bladder Material	Nomex Structural Layer and Viton-B50 Elastomer
Overall Weight	320 kilograms
Surplus Fuel Required	1.5 kg (for deployment) 22 kg (for reconfiguration)
Inflating Pressure	0.1 bar (for deployment) 2.3 bar (for surface garage)
Packaged Volume	Two adjacent cylinders of radius:28.5 cm Length:14.5m

CONCLUSIONS

The design team chose the subsystems inflatable protection system. This deployment system is lightweight and reliable. Deployment of the protection system is provided by means of inflatable struts which, when inflated, cause the protection system to unroll and deploy from its stored position on top of the lander.

Gas for inflation of the struts is provided by surplus liquid oxygen in the fuel tanks. The inflatable struts use less gas to inflate, are more lightweight, and leak less gas than a balloon-like dual membrane. The inflatable struts are made of a Nomex structural layer covered by an elastomer layer of Viton-B50. This material has been certified by NASA for use on the Shuttle Orbiter and in the Spacelab.

The multilayered insulation material, which provides both thermal and micrometeorite impact protection, has also been certified by NASA and has proven its merits on the Spacelab. This material is comprised of an outer skin of thermally protective Kapton followed by layers of aluminized Mylar which are separated by layers of a Dacron mesh. The protection system is light weight (320 kg) and occupies a compact cylindrical volume (two adjacent cylinders of radius 28.5 cm and length of 14.5 m.) on top of the lander.

The design team designed the protection system to reconfigure into either a lunar regolith support blanket or as a "twin tunnel" surface garage. The future lunar base scenario will inevitably need protective garages for sorting and storing lunar surface vehicles and equipment. Finally, the design team constructed a model of the protection system showing its deployment and subsequent reconfiguration into an alternate use.

RECOMMENDATIONS

The design team makes the following recommendations:

1. Rigidizing foams may be used to provide greater strength and eliminate the leakage problems associated with inflatable structures.
2. The required MLI material thickness of 3 millimeters was calculated from empirical studies which need to be verified at 20 kilometers per second.
3. The design team recommends that the entire lander be coated with a "cold coating" (high emissivity and low absorptivity, for example white paint) for added thermal protection.
4. The subsystems design philosophy should be used for space systems whenever possible. Such a concept provides increased versatility.
5. The basic concept of an inflatable multilayered non-metallic thermal/micrometeorite protection system can easily be adapted to other space structures such as the space station Freedom.
6. Deployment of light space structures, by using inflatable struts, should be considered in future space systems design.

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APPENDICES

APPENDIX A

THE DECISION MATRIX

APPENDIX A

THE DECISION MATRIX

The deployment system selected was the sub-systems inflatable protection design. The deployment system concepts were evaluated on the basis of reconfiguration utility, overall weight of the system, volume and packagability, extent of protection, reliability, and other special considerations (see Table A-I). Weight was given the largest weighing factor (0.286 out of 1.0). The necessity of minimizing the weight of spacecrafts becomes evident when one considers the high transportation cost--about one million dollars per pound. The reconfiguration weighing factor was assigned the next highest value (0.238). This high value was based on the design team's philosophy of accepting moderate weight penalties in return for improved reconfiguration utility. The combined weight and reconfiguration weighing factors (0.524) accurately reflects the the designs team's commitment to making "every pound count".

The "extent of protection" parameter considers the surface area which is protected by the system. This design parameter was distinguished from the overall reliability so that the design team could consider if the entire lander needed to be protected. The reliability parameter was based on the "confidence level" the design team felt each system merited without regard to any detailed analysis.

Finally, the special considerations parameter was used so that any noncritical peculiarities of the system could be lumped into one category. For example, the accordion design required a preferred landing orientation while the inflatable alternatives required a small amount of surplus oxygen in the fuel tank.

Table A-I.

Decision Matrix for the deployment system.

	Design Parameters of Deployment Systems						
Design Considerations / Weighing Factor / Design Concept	Reconfigur-ation	Weight	Volume	Extent of Protection	Reliability	Special Considera-tion	Sum of Products
	.238	.286	.143	.048	.190	.095	1.000
Pre-Protection	1 / 0.24	9 / 2.57	10 / 1.43	6 / 0.29	7 / 1.33	3 / 0.29	6.15
"Accordion" Design	6 / 1.43	7 / 2.00	7 / 1.00	5 / 0.24	6 / 1.14	5 / 0.48	6.29
Single Inflatable Protection	9 / 2.14	6 / 1.72	5 / 0.72	10 / 0.48	7 / 1.33	7 / 0.67	7.06
Sub-systems Inflatable Protection	9 / 2.14	7 / 2.00	8 / 1.14	8 / 0.38	7 / 1.33	7 / 0.67	7.66

APPENDIX B

MULTILAYERED PROTECTION

APPENDIX B

MULTILAYERED PROTECTION

The current technology in spacecraft thermal/micrometeorite protection uses many layers of alternating thermal and impact protection materials. This approach has been successfully used in the Spacelab and has been found to be very light weight and effective. [20]

The design team decided to use a Multilayered Insulation (MLI) which consists of several layers of thermal protection material separated by layers of low a thermal conductivity impact protection material. The outer skin of the MLI is typically 25 microns aluminized Kapton while the remaining layers are made of aluminized Mylar separated by a Dacron mesh.[20] The aluminizing process is done by vapor-depositing aluminum on the film surface.The layers of aluminized Mylar provide redundant thermal insulation while the Dacron mesh of low thermal conductivity reduces conductive heat transfer. Micrometeorite protection is provided by the Dacron mesh layers and the aluminized Mylar layers. Multilayered micrometeorite protection has the same effect as using many closely spaced micrometeorite bumpers.

Empirical studies on MLI have shown that it can provide the same impact protection as an aluminum bumper, but at 13 percent of the aluminum mass per unit area. For example, the impact protection provided by 0.13 cm of aluminum (mass per unit area of 0.364 gm /cm^2) can be provided by 0.5 cm of MLI (19 layers of MLI) with a mass per unit area of only 0.05 gm/cm^2. [20] Thus, using MLI conserves weight.

APPENDIX C

THERMAL PROTECTION ANALYSIS

APPENDIX C

THERMAL PROTECTION ANALYSIS

The assumptions of the thermal analysis are: the only mode of heat transfer is direct solar radiation, only one layer of thermal insulation (actually 12 layers) is used, and the deployed geometry of the protection system is a half-cylinder.

The data needed for thermal analysis are:

Stefan Boltzmann's constant (σ) = 5.67×10^{-8} W / m^2 / K,
absorptivity of thermal coating (α) = 0.2,
emissivity of thermal coating (ε) = 0.7,
direct solar flux on lunar surface (S) = 1350 W / m^2,
albedo flux on the Moon (S_a) = assumed negligible,
protection system radius (R) = 6.5 m,
protection system length (L) = 14.5 m.

With both sides covered, the area of the protective covering on the lander ($A = \pi RL + \pi R^2$) is 428 m^2. The projected area of the covered lander on the lunar surface ($A_p = 2RL$) is 188.5 m^2. At steady state the absorbed solar radiation equals the emitted radiation:

$$\alpha S A_p = A \varepsilon \sigma T^4 \quad \Rightarrow \quad T = \left(\frac{\alpha S A_p}{\varepsilon \sigma A}\right)^{\frac{1}{4}} = 234 \text{ K} = -39 \text{ C}$$

The calculated steady-state temperature of the protected system is -39 degrees Celsius and the daytime lander temperature therefore remains lower than -39 degrees Celsius. This temperature is allowable.

APPENDIX D

MICROMETEORITE FLUX ANALYSIS

APPENDIX D

MICROMETEORITE FLUX ANALYSIS

An assumption was made that the flux equations below are valid for micrometeorites impacting at all angles (worst case). According to available data from the Apollo missions, the average cumulative flux meteoroid model can be described by the following equations:

$$10^{-12} \leq m \leq 10^{-6} \text{ grams:} \quad \text{Log}N_t = -14.566 - 1.584 \log m - 0.063 (\log m)^2,$$
$$10^{-6} \leq m \leq 1 \text{ grams:} \quad \text{Log}N_t = -14.597 - 1.213 \log m, \qquad (1)$$

where m is the mass of the micrometeorite and N_t is the micrometeorite flux distribution. [19] The overall width (R') and length (L) of the lander are 13.0 and 13.5 meters, respectively. The surface area of a semi-cylindrical covering around the lander is $A_{land} = \pi R'L + \pi R'^2 = 1082.28$ m^2. The useful life (t) of the protection system, including reconfigurable use, is 20 years (6.3×10^8 seconds).

The probability of impact with a meteorite of mass greater than **m** grams is $Pr = N_t\, t\, A$. The design team chose a 99 % confidence level. Therefore, the mass for Pr=0.01 needs to be calculated. Given

$$Pr = 0.01,\ A_{land} = 1082.28\ m^2,\ \text{and } t = 6.307\times10^8\ s,$$

$$N_t = Pr / (A_{land} t) = 1.46\times10^{-14}\ \text{impacts} / m^2 / \text{second}$$

From equation (1), **m** is then found to be 0.2450 grams. Therefore, using a factor of safety of four (normally used by NASA for unmanned missions), the design team chose to design for impact of a micrometeorite of mass 1 gram with a velocity of 20 kilometers per second.

APPENDIX E

SACRIFICIAL BUMPER THEORY

APPENDIX E

SACRIFICIAL BUMPER THEORY

Bumper walls can be designed to provide optimum protection against a micrometeorite of known mass. An optimum bumper thickness would vaporize both the micrometeorite and the local bumper wall. However, one problem associated with such an approach is that impacts from micrometeorites smaller than the anticipated micrometeorite would cause only partial shock loading of the bumper wall and lead to spalling of the bumper back wall. In such a case, the lander superstructure would be damaged not by micrometeorite impact, but by subsequent hypervelocity impact of bumper particles. A solution to this problem is to adequately space the bumper wall from the underlying lander superstructure and to provide additional layers of bumpers as may be necessary. [15]

Extensive research in the concept of meteorite bumpers has been done by Dr. Cour-Palais of NASA-Johnson Space Center in Houston. [4] Unfortunately, most of the data available is for metallic bumpers. The analysis of non–metallic micrometeorite bumpers is still in a developmental stage, and needs to be experimentally supported.

Problems associated with the inherent shock transmission capabilities of metals and their high density make metals a poor choice for bumper material. [3] It has been suggested to the design team that there exist other undisclosed state-of-the-art impact protection materials for use in combat tank armor. These materials are highly classified and were thus unavailable for consideration.

APPENDIX F

MICROMETEORITE IMPACT ANALYSIS

APPENDIX F

MICROMETEORITE IMPACT ANALYSIS*

Thickness of the Micrometeorite Protection Blanket

Since the moon is devoid of an atmosphere, any meteorites that impact the moon are neither decelerated nor vaporized by an atmosphere. Consequently, there is a larger meteorite problem on the moon than on the Earth. Because the probability of any sizable micrometeorite impact is negligible, the design team concluded that impact protection plays a secondary role to the primary concern of thermal protection.

The protection blanket being used is multilayered insulation (MLI). The lunar lander must be protected against a micrometeorite with a density (ρ_m) of 0.5 gm/cm^3, maximum mass (m) of one gram, and velocity (v) of 20 kilometers per second (see Appendix D).

The average diameter of a one gram micrometeorite is

$$D_m = 2\sqrt[3]{\frac{3m}{4\pi\rho}} = 1.56 \text{ cm}$$

The aluminum bumper thickness needed to protect against a meteorite of 1.56 centimeters diameter can be calculated from empirical data as $0.04D_m$ = 0.0625 centimeters. This gives a mass per unit area of 0.175 gm/cm^2. An MLI bumper gives the same micrometeorite protection as aluminum, but at only 13.3 percent the mass per unit area. Therefore, the mass of

* All equations in this appendix are from reference [4]

MLI bumper needed is 0.023 gm/cm^2. The design team chose to use an MLI thickness of three millimeters, or a mass per unit area of MLI of 0.03 gm/cm^2. Hence, sufficient impact protection is provided.

Minimum Thickness of the Spacecraft Wall

Spallation damage inside the lander can occur from hyper velocity impact on the lander outer wall. The spallation damage increases with decreasing distance (s) between the lander and the protection system.

The inflatable struts provide a bumper spacing between the protection system and the lunar lander. When fully inflated, the thinnest struts have a cross sectional diameter of 28 centimeters. Since the struts are not fully inflated, the design team assumed a bumper spacing of only 5 centimeters. Minimum lander wall thickness to prevent spallation is then given by

$$t_b = \frac{.075 m^{1/3} v}{\sqrt{s}} = 0.67 \text{ cm}$$

Hence, the minimum lander wall thickness should be 0.67 centimeters. The actual wall thickness of the lander is greater; therefore the protection system provides sufficient protection.

APPENDIX G

WEIGHT AND VOLUME CALCULATIONS

APPENDIX G

WEIGHT AND VOLUME CALCULATIONS

The calculations in this appendix are only approximations of the mass and volume of the protection system.

Packaging Volume

The protection system is thickest at the end–struts since the side blanket is folded near the end. When rolled up, the end–struts determine the minimal packing radius. The material thicknesses at the struts are:

strut material = 2 x 1.5 millimeters,
top blanket = 3 millimeters,
side blanket = 3 millimeters,
strut back cover = 3 millimeters.

Therefore, the overall thickness at the end–struts is 0.012 meters. The curvature radius (R) of the protection system is 6.5 meters, thus, the length ($l = \frac{\pi R}{2}$) of the protection system material from the middle to either side is 10.2 meters. The volume per unit length (V = lt) of the material at the end–struts is then 0.122 m^2.

The protection system is rolled to form two cylinders, each of length (L) 14.5 m. If it is assumed that the material is packed solid, the radius of the packed cylinder is ($R_{solid} = \sqrt{\frac{V}{\pi}}$) 0.197 meters. For good deployment, the system must be well–packed. Even if we assume the packed volume to be twice the solid packed volume, the packing radius is ($\sqrt{2}R_{solid}$) 0.285 meters. Thus for the latter case, the system is packed into two adjacent cylinders, each of radius 0.285 meters and length 14.5 meters.

Required Surplus Oxygen for deployment

The following are the inflated volumes of the struts:

two end–struts = 2.6 m^3,
six middle–struts = 3.8 m^3,
five stabilizing struts = 4.5 m^3.

The total volume of the struts and thus the volume that will be filled with oxygen, is 10.9 m^3. At a deployment pressure of 0.1 bar and a temperature of 0 degrees Celsius, oxygen has a density of 0.14 kg/m^3. For 10.9 m^3 of oxygen at this state, 1.5 kilograms will be needed to deploy the protection system. At a surface garage pressure of 2.3 bar and a temperature of 0 degrees Celsius (assuming active heating of oxygen in the struts), 22 kilograms of oxygen is needed for the surface garage.

Total Mass of the Protection System

There are three components that contribute to the total mass of the protection system. These are listed below with their approximate mass:

the struts and protection blanket : 314 kg
the oxygen needed for deployment: 1.5 kg
Pressure lines, expansion valve, etc: 4.5 kg (estimated)

The estimated total mass is 320 kilograms, which is 1.3 percent of the cargo capacity of the lunar lander.

APPENDIX H

A CONCEPTUAL CONTAINER FOR THE PROTECTION SYSTEM

APPENDIX H

A CONCEPTUAL CONTAINER FOR THE PROTECTION SYSTEM

The protective system is stored above the common module in its own strap-on container (see Figure H-1). This container is made of a 1 mm aluminized Mylar covering. Mylar, in a film form, exhibits excellent tear resistance. However, once a tear is initiated, the tear tends to propagate with little resistance. The protection system is then released by initiating a small tear which quickly runs the entire length of the fabric container. The tear could be initiated by chemical degradation of the Mylar covering. Another alternative for a releasing mechanism is to use several wrap–around straps. However, NASA's design philosophy is to minimize single point failure—if one strap failed to release, the deployment of the protection system could be jeopardized. [19]

Three Kevlar straps are wrapped around the common module to firmly fix the protection system container above the module. Storing the system at the highest elevation allows a gravity-assisted deployment. The protection system is rolled on either side and thus possesses a single axis of symmetry.

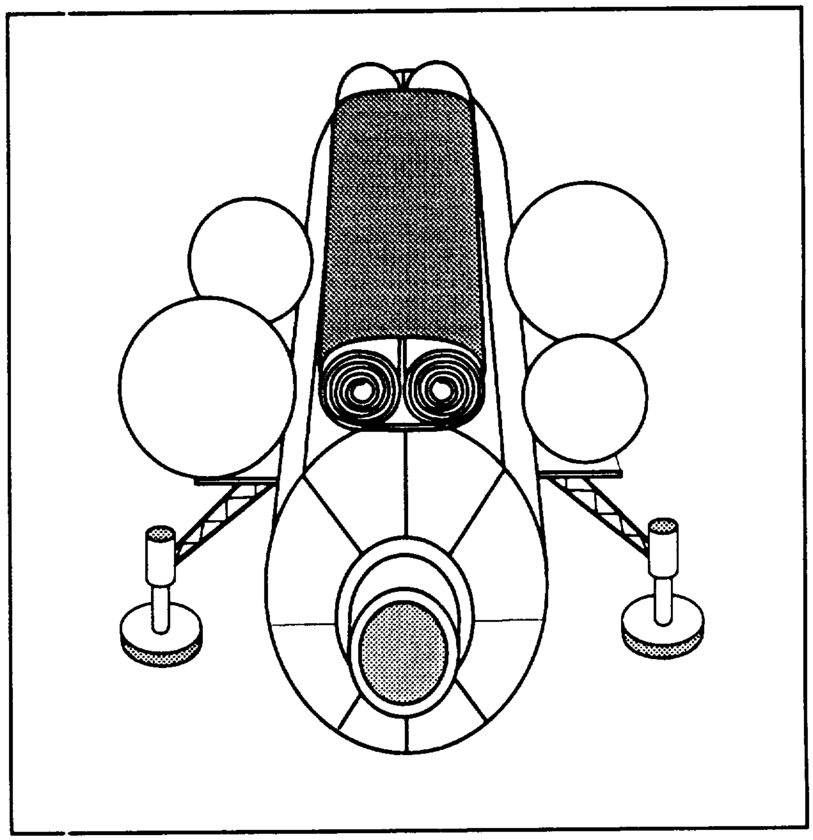

Figure H-1: LANDER SHOWING PROTECTION SYSTEM IN CONCEPTUAL CONTAINER

APPENDIX I

GAS LEAKAGE CONSIDERATIONS

APPENDIX I

GAS LEAKAGE CONSIDERATIONS

All pressurized vessels are subject to pressure loss due to gas leakage. This is especially true of nonmetallic pressurized vessels. Preliminary studies conducted by the Goodyear Aerospace Corporation (GAC) concluded that the leakage rate for a large deployed volume in space was a problem. [16] The design team deduced that the leakage rates for the inflatable struts, which use a much smaller volume than the large volume used by GAC, would pose no serious problem. Also, GAC predicted that by using new technology, the leakage rate could be reduced by a factor of ten. In any case, the design team proposes two alternative ways to regulate the pressure of any inflatable structure if the need should arise. First, the pressure can be kept constant by incorporating out-gasing solids which would replenish the gases that escape due to the permeability of the pressure vessel membrane. These out-gasing solids would sublimate at a predetermined rate equal to the predicted rates of gas leakage. A second alternative would be to supply more gas to the inflatable structures from a small reservoir. In this concept, the temperature of a cryogenic gas reservoir tank would be controlled by either actively using an electric heating element or passively by altering the view factor of the storage tank in relation to the sun (see Figure I-1).

Self sealing inflatable structures could be used to protect these structures against pressure loss due to either micrometeorite puncture or accidental man/equipment punctures. The self sealing concept is similar to

Earth applications on automotive tires Also, if the inflated structure can be made rigid with hardenable foams, puncture and permeability leakage would not pose a problem.

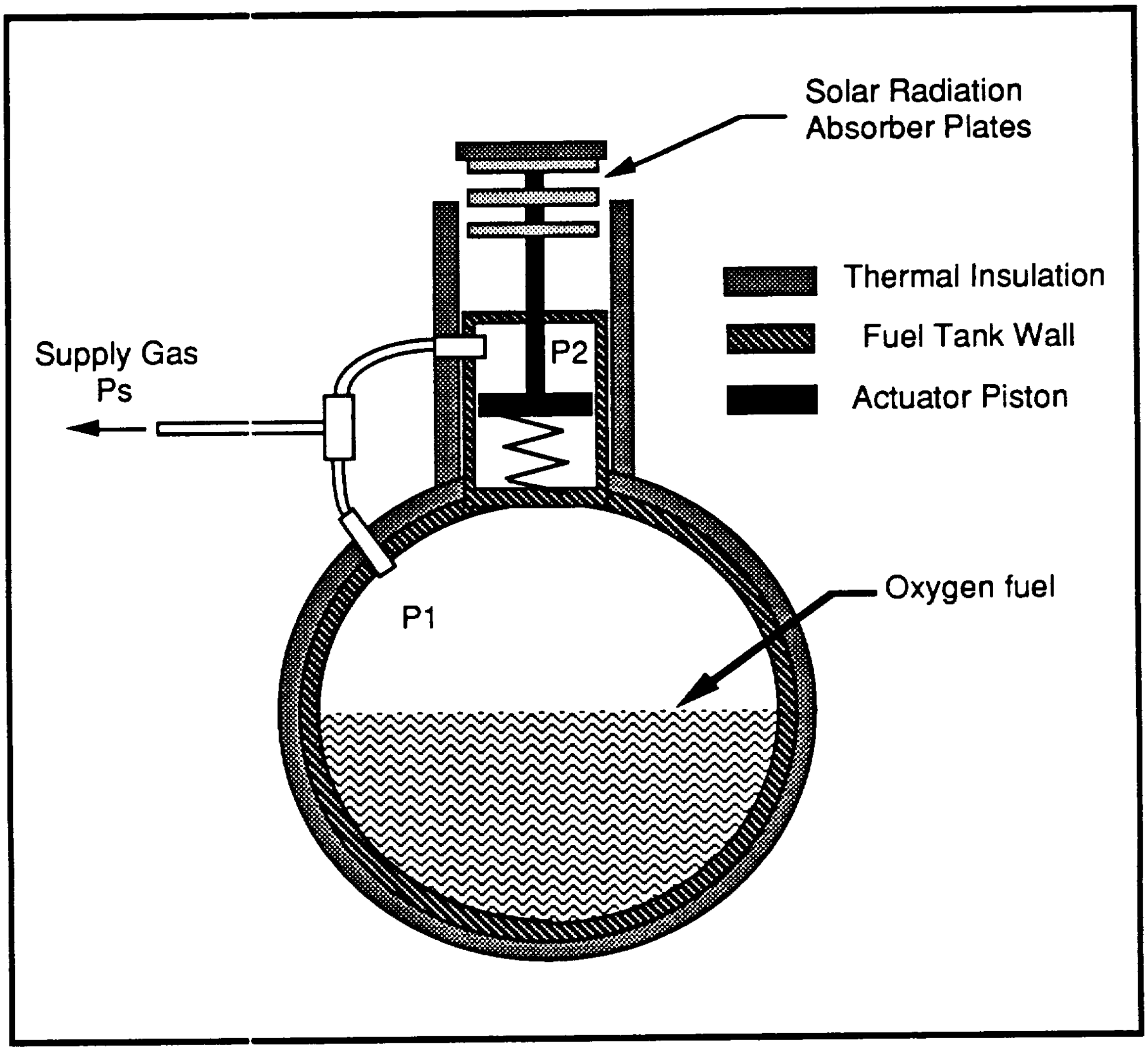

Figure I-1: CONCEPTUAL DRAWING OF A SELF REGULATING PRESSURE SUPPLY.

The self regulating pressure supply can keep constant pressure by using a feedback loop. When the supply pressure Ps (Ps=P1=P2 at equilibrium), becomes sufficiently low, the spring force overcomes the pressure force applied on top of the piston. The solar radiation absorber plates are then moved out of the thermal shield and exposed to the sun's heating radiation. Conduction through the top cylinder heats the liquid oxygen which then boils off until equilibrium pressure (the design pressure to maintain strut rigidity) is reached. This apparatus should be designed so that the absorber plates are fully shielded from the solar radiation when equilibrium pressure is maintained.

APPENDIX J

STRUCTURAL ANALYSIS OF INFLATABLE STRUTS

APPENDIX J

STRUCTURAL ANALYSIS OF INFLATABLE STRUTS

The pressures obtained by this analysis can only be used as an approximation since several important assumptions have been made about the analytical model. The major assumptions are that the struts have a semi–circular geometry, there is a zero moment at the end points of the struts, and that the strut is a shell (a membrane will experience creep). A FORTRAN computer program was developed to perform the structural analysis of inflatable struts (see Appendix K).

Maximum loading occurs on the end–struts since these struts have to carry the weight of the side blanket and a part of the top blanket (assuming the garages are covered at the ends). The struts are modeled as semi–circular pressure beams with a circular cross section. Internal pressure in the struts causes tension in the strut material. Consider a section of the end–strut (see Figure J-1). There are tangential tensile stresses ($\delta_{tt} = \frac{PR}{t}$) and axial tensile stresses ($\delta_{ta} = \frac{PR}{2t}$), where P is the internal pressure of the strut.

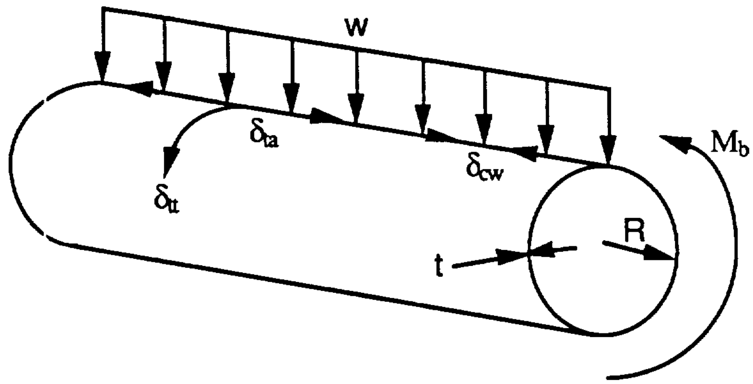

Figure J–1: SECTION OF STRUT SHOWING STRESSES

For a given uniformly distributed load (w) the resulting bending moment (M_b) causes compressive stresses (δ_{cw}) in the strut (the load also causes compression, but its effects are negligible). Shear stresses are not critical since the load is uniformly distributed and the strut curvature is large compared to the cross section of the strut. If the compressive stresses are greater than the tensile stresses, local wrinkles will start developing in the upper fibers of the strut. For the beam to collapse, a wrinkle must extend through the cross section of the strut; it can be shown that the compression stress needed for collapse is approximately twice the stress necessary for wrinkling. [5] Thus we introduce a factor of safety of two by designing for wrinkling and not collapse. The axial tensile stress is half the tangential stress, therefore

$$\delta_{ta} \geq \delta_{cw} \quad \Rightarrow \quad \frac{PR}{2t} \geq \frac{M_b}{J},$$

where R and J are the radius and the area moment of inertia of the circular cross section. Solving for the internal pressure needed to avoid wrinkling gives

$$P \geq \frac{2M_b}{J} \quad . \tag{1}$$

For the strut not to rupture, the total tensile stresses must not exceed the yield strength (δ_y) of the strut material:

$$\delta_y \geq \delta_{tw} + \delta_{ta} \qquad \delta_y \geq \delta_{tt},$$

where δ_{tw} is the tensile stress due to the bending moment. Solving for the internal pressure gives

$$P \leq \frac{2\delta_y}{R} t - \frac{2M_b}{J} t \quad \text{and} \tag{2}$$

$$P \leq \frac{\delta_y}{R} t \quad . \tag{3}$$

The permissible pressure range in the struts in defined by equations (1), (2), and (3). To calculate these pressures, the maximum bending moment in the beam must be determined.

The following model is a simplified method for finding the maximum bending moment on a strut due to the weight of the protection system. The physical model of the strut is a three–pin arch beam (see Figure J–2).

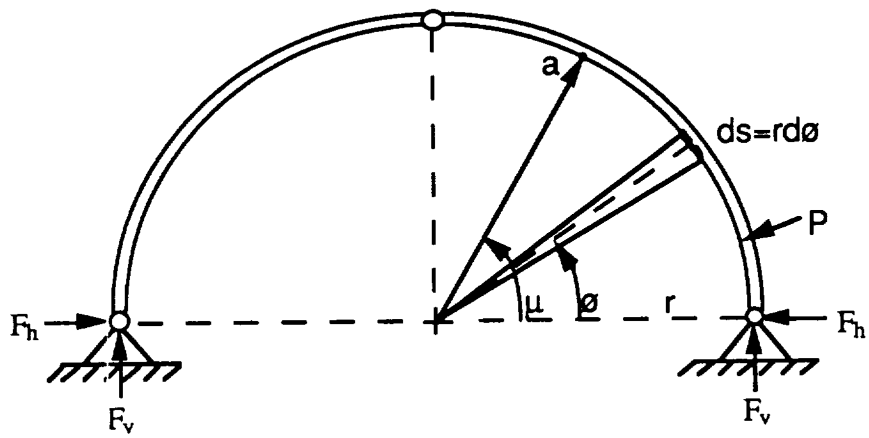

Figure J–2: ANALYTICAL MODEL OF END–STRUT

The loads causing the bending moment on the beam are the weight of the top blanket, side blanket, and strut. The weight of the strut and the contributing weight of the top blanket can be expressed as load per unit arch length (w_t)

With a moment arm of $(\cos(ø)-\cos(\mu))r$, the moment of the load on the arch length **ds** about point **a** due to w_t is given as (clockwise positive)

$$dM_t = w_t ds(\cos(ø)-\cos(\mu))r$$

By integrating from $ø=0$ to $ø=\mu$, we obtain the moment about **a** due to w_t :

$$M_t = w_t r^2(\sin(\mu) - \mu\cos(\mu)).$$

The end–strut carries the additional weight of the side blanket. This weight can be expressed as weight per unit area (W_s) as follows: the area under the arch length **ds** is **ds**sin(ø)rsin(ø), and therefore

$$dM_s = w_s \mathbf{ds} \sin(ø) r \sin(ø)(\cos(ø) - \cos(\mu))r \ .$$

Integrating from ø=0 to ø=μ, we obtain the moment about point a due to w_S:

$$M_s = w_s r^3 (\frac{\sin(\mu)^3}{3} - (\frac{\mu}{2} - \frac{\sin(2\mu)}{4})\cos(\mu)) \ .$$

The combined moment about point **a** is therefore the sum of w_t and w_S.

APPENDIX K

FORTRAN PROGRAM TO ANALYZE INFLATABLE STRUTS

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      PROGRAM STRUT
*BAARD VESTGAARD 110389
*THIS PROGRAM FINDS THE MAXIMUM MOMENT OCCURING IN A SEMI-CIRCULAR
*PRESSURIZED STRUT, PERMISSIBLE PRESSURE RANGE, AND VOLUME OF EITHER
*END-STRUTS OR MIDDLE-STRUTS.
*DEFINITION OF VARIABLES:
* FS,FT,WS,WT=TOTAL AND SPECIFIC WEIGHTS ** VOLUME=VOLUME OF STRUTS
* MS,MT=MOMENTS DUE TO WEIGHT ON STRUT ** FV,FH=REACTION FORCES **
* MOMENT=TOTAL MOMENT ABOUT POINT ON STRUT ** P=INTERNAL PRESSURE IN
* STRUT ** SIGMAY=TENSILE STRENGTH OF STRUT MATERIAL ** RAD=STRUT CROSS
* SECTION RADIUS ** R=STRUT RADIUS ** J=AREA MOMENT OF INERTIA OF
* STRUT CROSS SECTION ** SFACT=SAFETY FACTOR ** L=LENGTH OF PROTECTION
* SYSTEM ** NO=NUMBER OF STRUTS ** DMLI=DENSITY OF MLI (KG/M2) **
* DNOM=DENSITY OF NOMEX/VITON-B50 (KG/M3)
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      REAL WT,WS MS,MT,FV,FH,FS,FT,MOMENT,PI,MAX,MAXMOM,ANGLE,VOLUM
     +SIGMAY,RAD,J,PLOW,PHIGH,PII,PIII,SFACT,L,NO,DMLI,DNOM,DUMMY
      DATA SFACT,DMLI,DNOM,NO,L,R,PI,SIGMAY/4.,.3,860.,8.,14.5,3.25,
       +3.1416,185000000/
      WRITE(6,10)
10    FORMAT(10X,'ENTER CROSS SECTION RADIUS, STRUT THICKNESS,',
     +' AND STEPSIZE')
      READ(6,*) RAD,T,C
      WRITE(6,15)
15    FORMAT(10X,'ARE YOU ANALYSING END STRUTS OR MIDDLE STRUTS? 1/2')
      READ(6,*) I
      DUMMY=PI*R*L*DMLI/(NO-1)
      FT=(PI*R*2.*PI*RAD*T*DNOM+DUMMY)*1.635
      WT=FT/(PI*R)
      IF(I.EQ.1) THEN
          FT=FT-DUMMY/2.*1.635
         WT=FT/(PI*R)
           VOLUME=2.*PI*R*PI*RAD**2.
            FS=PI/2.*R**2.*DMLI*1.635
        WS=DMLI
      ELSE
           VOLUME=(NO-2.)*PI*R*PI*RAD**2
        WS=0.
      END IF
      J=PI/4.*((RAD+T)**4.-RAD**4.)
      C=2.*PI*C/360.
      MAX=PI/2.
      FV=(FS+FT)/2.
      FH=(FV*R-MT(R,WT,MAX)-MS(R,WS,MAX))/R
      WRITE(6,20)
      WRITE(1,20)
20    FORMAT(////,'                    ****************RESULTS****************')
      WRITE(6,25) RAD,T
      WRITE(1,25) RAD,T
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25      FORMAT(10X,'FOR CROSS SECTION RADIUS OF ',F3.2,' M AND',
       +/,10X,'STRUT MATERIAL THICKNESS OF ',F5.4,' M:')
        WRITE(6,30) FS,FT,FS+FT,FV,FH
        WRITE(1,30) FS,FT,FS+FT,FV,FH
30      FORMAT(/,10X,'SIDE BLANKET WEIGHT: ',F5.1,' N',/,10X,
       +'TOP BLANKET AND STRUT WEIGHT: ',F5.1,' N',/,10X,
       +'TOTAL WEIGTH: ',F5.1,' N',/,10X,'VERTICAL REACTION FORCE: ',
       +F5.1,' N',/,10X,'HORIZONTAL REACTION FORCE: ',F5.1,' N',/)
        WRITE(6,40)
        WRITE(1,40)
40      FORMAT(10X,'POSITION (DEG)',10X,'MOMENT (Nm)')
*****************************************************************
*FINDS MOMENTS THROUGHOUT STRUT BEAM DUE TO WEIGHT LOAD
        MAXMOM=0.
        I=10.
        DO 60 MAX=PI/2.,-C,-C
            MOMENT=-FV*R*(1.-COS(MAX))+FH*R*SIN(MAX)+
       +    MT(R,WT,MAX)+MS(R,WS,MAX)
          IF(ABS(MAXMOM).LT.ABS(MOMENT)) THEN
          MAXMOM=MOMENT
          ANGLE=MAX
          ELSE
          END IF
          IF(I.EQ.10.) THEN
                WRITE(6,50) MAX/2./PI*360.,MOMENT
                WRITE(1,50) MAX/2./PI*360.,MOMENT
50              FORMAT(15X,F3.0,17X,F6.2)
              I=0.
          ELSE
          END IF
           I=I+1
60      CONTINUE
        WRITE(6,70) MAXMOM,ANGLE/(2.*PI)*360.
        WRITE(1,70) MAXMOM,ANGLE/(2.*PI)*360.
70      FORMAT(/,10X,'MAXIMUM MOMENT OF',F5.2,' Nm AT ',F3.0,' DEGREES')
*****************************************************************
*FINDS THE LOWEST AND HIGHEST ALLOWED PRESSURE IN THE STRUT
        PLOW=2.*ABS(MAXMOM)*RAD/J
        PII=SIGMAY*T/RAD
        PIII=2.*PII-PLOW
        IF(PIII.GT.PII) THEN
          PHIGH=PIII
        ELSE
          PHIGH=PII
        ENDIF
        PLOW=PLOW*(SFACT-1.)/100000.
        PHIGH=PHIGH/SFACT/100000.
        WRITE(6,80) PLOW,PHIGH
        WRITE(1,80) PLOW,PHIGH
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80      FORMAT(10X 'PRESSURE RANGE: ',F5.2,' TO ',F5.2,' BARS')
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        WRITE(6,90) VOLUME
        WRITE(1,90) VOLUME
90      FORMAT(10X 'VOLUME OF STRUTS: ',F4.2,' M3')
        WRITE(6,*)'                   ****************************************'
        WRITE(1,*)'                   ****************************************'
        END
************************************************************
*FUNCTIONS TO CALCULATE MOMENTS DUE TO WEIGHT LOADS
        FUNCTION MT(R,WT,MAX)
        REAL MT,R,WS,MAX
        MT=WT*R**2*(SIN(MAX)-MAX*COS(MAX))
        END
        FUNCTION MS(R,WS,MAX)
        REAL MS,R,WS,MAX
        MS=WS*R**3*(.333*SIN(MAX)**3-(MAX/2.-SIN(2*MAX)/4.)*COS(MAX))
        END
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SAMPLE OUTPUT

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***************RESULTS**************

FOR CROSS SECTION RADIUS OF .14 M AND
STRUT MATERIAL THICKNESS OF .0015 M:

SIDE BLANKET WEIGHT: 0.0 N
TOP BLANKET AND STRUT WEIGHT: 29.3 N
TOTAL WEIGHT: 29.3 N
VERTICAL REACTION FORCE: 14.7 N
HORIZONTAL REACTION FORCE: 5.3 N
   POSITION (DEG)        MOMENT (Nm)
         90.                 0.00
         80.                 0.20
         70.                 0.75
         60.                 1.56
         50.                 2.46
         40.                 3.26
         30.                 3.68
         20.                 3.47
         10.                 2.34
          0.                 0.00
MAXIMUM MOMENT OF 3.70 Nm AT 28. DEGREES
PRESSURE RANGE: 2.37 TO 9.71 BARS
VOLUME OF STRUTS: 3.77 M3
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`***************RESULTS**************`

FOR CROSS SECTION RADIUS OF .20 M AND
STRUT MATERIAL THICKNESS OF .0015 M:

SIDE BLANKET WEIGHT: 8.1 N
TOP BLANKET AND STRUT WEIGHT: 32.2 N
TOTAL WEIGHT: 40.4 N
VERTICAL REACTION FORCE: 20.2 N
HORIZONTAL REACTION FORCE: 8.9 N

POSITION (DEG)	MOMENT (Nm)
90.	0.00
80.	1.11
70.	2.59
60.	4.23
50.	5.77
40.	6.88
30.	7.20
20.	6.38
10.	4.07
0.	0.00

MAXIMUM MOMENT OF 7.22 Nm AT 32. DEGREES
PRESSURE RANGE: 2.27 TO 6.75 BARS
VOLUME OF STRUTS: 2.57 M3

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