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Lunar Lander Conceptual Design

A Design Project

in AE 442

Presented To

The Faculty of the School of Engineering

and Applied Science

University of Virginia

and

Universities Space Research Association

by

Joo Ahn Lee
John Carini
Andrew Choi
Robert Dillman
Sean J. Griffin
Susan Hanneman
Caesar Mamplata
Edward Stanton

May 17, 1989

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ACRONYM LIST

ATP	Authority to Proceed
C&DM	Communication and Data Management
DSN	Deep Space Network
ECLSS	Environmental Control and Life Support System
EVA	Extravehicular Activity
I_{sp}	Specific Impulse
LEM	Lunar Excursion Module
LEO	Low Earth Orbit
LH2	Liquid Hydrogen
LLO	Low Lunar Orbit
LOX	Liquid Oxygen
LSS	Lunar Space Station
MLI	Multi-Layer Insulation
OTV	Orbital Transfer Vehicle
PDR	Preliminary Design Review
P/L	Payload
PLSS	Primary Life Support System
SPE	Solar Particle Event
SS	Space Station
STS	Shuttle Transportation System
VCS	Vapor Cooled Shield

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PROJECT RATIONALE

As a new millennium awaits, the exploration of the moon and utilization of its resources pose challenges and rewards for America's space agenda. The scientific and commercial benefits of returning to the moon are alluring incentives for such a program. The wealth of scientific knowledge which could be gained from research on the moon span many disciplines, from geology to astrophysics to space medicine. The manufacture of technological products from lunar raw materials opens up the realm of lunar manufacturing for commercial application. Aside from these uses, a lunar base could also provide logistics and equipment support as a transportation node for expeditions traveling out to the further reaches of the solar system.

In the long run, the advantages of a manned presence on the moon outweigh those of an unmanned lunar program. The full potential of the benefits mentioned above cannot be realized with unmanned probes. A manned presence allows flexibility to modify experiments as a data from previous ones come in and thereby speeds up the pace of research. Future colonization of distant planets, such as Mars, should be prefaced by colonization of the moon in order to understand the requirements of more ambitious ventures. Certainly the greater cost and complexity of manned space programs over unmanned ones require careful consideration, but a dedication to advancing scientific research, space colonization, and space manufacturing with ambitious strides can best be served by adding the human dimension to a return to the moon. The excitement and challenge of a lunar venture lies not only with the possibilities mentioned already but also with the unimagined possibilities which await our curiosity.

The success of such a program rests with the design of the elements of a lunar infrastructure and with logistical planning to maximize efficiency. Two elements essential to the success of a lunar venture will be a Lunar Lander and a Lunar Space Station. The Lunar Space Station will be dedicated to fiberglass and semiconductor manufacturing from lunar materials and to serving as a transportation node in the lunar infrastructure. The focus of this report, however, is the conceptual design of a Lunar Lander, which will be the primary vehicle to transport the equipment necessary to establish a surface (LUNAR) base, the crew that will man the base, and the raw materials which the Lunar Station will process.

SPECIFICATIONS SUMMARY

In a Lunar Base Systems Study, NASA outlined three phases for development of a manned lunar base [1]. During Phase I, three years would be set aside for unmanned exploration of the moon using orbiting satellites and surface probes. Phase II would span approximately 10 years and would be devoted to establishing a surface base ready for a permanently manned presence, at which time Phase III operations would commence. The study identified that a lunar lander capable of shuttling cargo and crew to and from a lunar space station and a surface base would be essential to the establishment and support of the lunar base. Consequently, Phase II requirements were used to derive the following specifications for the Lander:

<u>Payload Capability</u>

(1) transport 15 metric tons to the surface and return unloaded
(2) transport up to 5 metric tons of lunar raw materials a year to the Lunar Space Station
(3) ferry a minimum of four astronauts to and from low lunar orbit to the surface base

<u>Operational Requirements</u>

(4) have twice the needed lifespan
(5) must be designed for ease of maintenance

<u>Operating Range</u>

(6) operate between a 200 kilometer orbit and the lunar surface
(7) reach a surface base located within 20° of the lunar equator

<u>Landing Performance</u>

(8) land on slopes of up to 10°
(9) negotiate 5 foot high obstacles
(10) land within 2 meters of the destination

Safety

(11) provide emergency life support for a crew of four for 24 hours

The largest payload expected for a reusable Lander would be a LOX pilot plant of 15 metric tons. In addition, the Lunar Station will require 5 metric tons of raw materials a year to manufacture fiberglass and semiconductors. The Lander will provide for crew rotation of the surface base. In an emergency, the Lander should be able to evacute the entire crew of the Lunar Station to the surface base.

The cost and logistics of replacing equipment on the Lander necessitate a long enough lifespan to assure reliable performance. However when maintenance is necessary, an astronaut should be able to gain easy access to all systems.

To minimize orbital perturbations of the Station and propellant requirements of the Lander, the Station will be placed in a 200 kilometer orbit. The majority of potential landing sites surveyed lie within 20° of the lunar equator.

In order to land upon an unprepared surface, the Lander must be able to negotiate 10° slopes and 5 foot high obstacles. The Lander will therefore be able to land on the majority of the lunar surface excluding the Highlands. The guidance, navigation, and control system should allow no more than 2 meters of range error during landings.

In case an aborted mission requires an immediate landing, the crew module should provide 24 hours of life support until rescued. The redundancies must be specified for all systems such as propulsion, guidance, and power.

EXECUTIVE SUMMARY

A Lunar Lander will be needed to operate in the regime between the lunar surface and low lunar orbit (LLO), up to 200 kilometers. This Lander is intended for the establishment and operation of a manned surface base on the moon and for support of the Lunar Space Station.

The lander will be able to fulfill the requirements of three basic missions:

1. a mission dedicated to delivering maximum payload for setting up the initial lunar base
2. multiple missions between LLO and lunar surface dedicated to crew rotation
3. multiple missions dedicated to cargo shipments within the regime of lunar surface and LLO.

The structural mass of the Lander will be 13.5 metric tons, and the propellant mass will be 35 metric tons. The payload mass will be approximately 39 metric tons for maximum payload missions and 15 metric tons for the cargo delivery missions. The lander will be approximately 10 meters (33 feet) in height and 9 meters (30 feet) in diameter.

The lander will be supported by four aluminum alloy landing legs which are attached to a rocket platform. The platform supports four regeneratively cooled rocket nozzles and the necessary engine components. The engines will be fed from four tanks storing liquid hydrogen (LH_2) and liquid oxygen (LOX) which are mounted between the rocket platform and an upper cargo platform. The avionics equipment will be located on the propellant level and will be capable of autonomous operation or human control. The cargo platform will have a common attachment mechanism for the crew module and the cargo pellets.

A modified space shuttle fuel cell system will supply electrical power for up to two weeks for a total output of 720 kilowatt-hours at an average draw of 2 kilowatts. The environmental control and life support system (ECLSS) will be open-loop and non-pressurized which will require the crew to remain in spacesuits. The ECLSS will be capable of providing a crew of 5 people with air, food, and water for up to three days.

CHAPTER 1: SYSTEMS OVERVIEW

A systematic approach to design and verification of the Lunar Lander system will ensure that it achieves performance goals and is developed within cost and schedule constraints.

The system engineering and integration approach is defined in Figure 1.1 with the results of activities during three major phases of the program -- Authority to Proceed (ATP) to Preliminary Design Review (PDR).

> ATP-PDR -- This phase of the program provides a selected design concept prior to proceeding with detail design and fabrication of hardware.

As illustrated, operations analyses are performed to identify operational and functional requirements of the flight system. The flight vehicle is then synthesized, and subsystem design requirements are established. Lunar ground support operations necessary to support the flight system are also identified. The system requirements will reflect various trade-off functions based on the need to minimize cost per vehicle/flight and maximize system performance. The requirements will be incorporated into system specifications which form the basis for design, integration and control of the Lunar Lander system.

For example, one major element in operations is the lunar lander payload (P/L) capability. Estimated payload capability, through system analysis, begins with program event timeline requirements (Figure 1.2). After timeline requirements are identified a mission profile is initially established (Figure 1.3). The mission profile also reflects the estimated P/L capacity per flight of the lander under specific mission constraints. This estimated P/L

Figure 1.1:
SYSTEM REQUIREMENTS DEFINITION AND PRELIMINARY DESIGN APPROACH

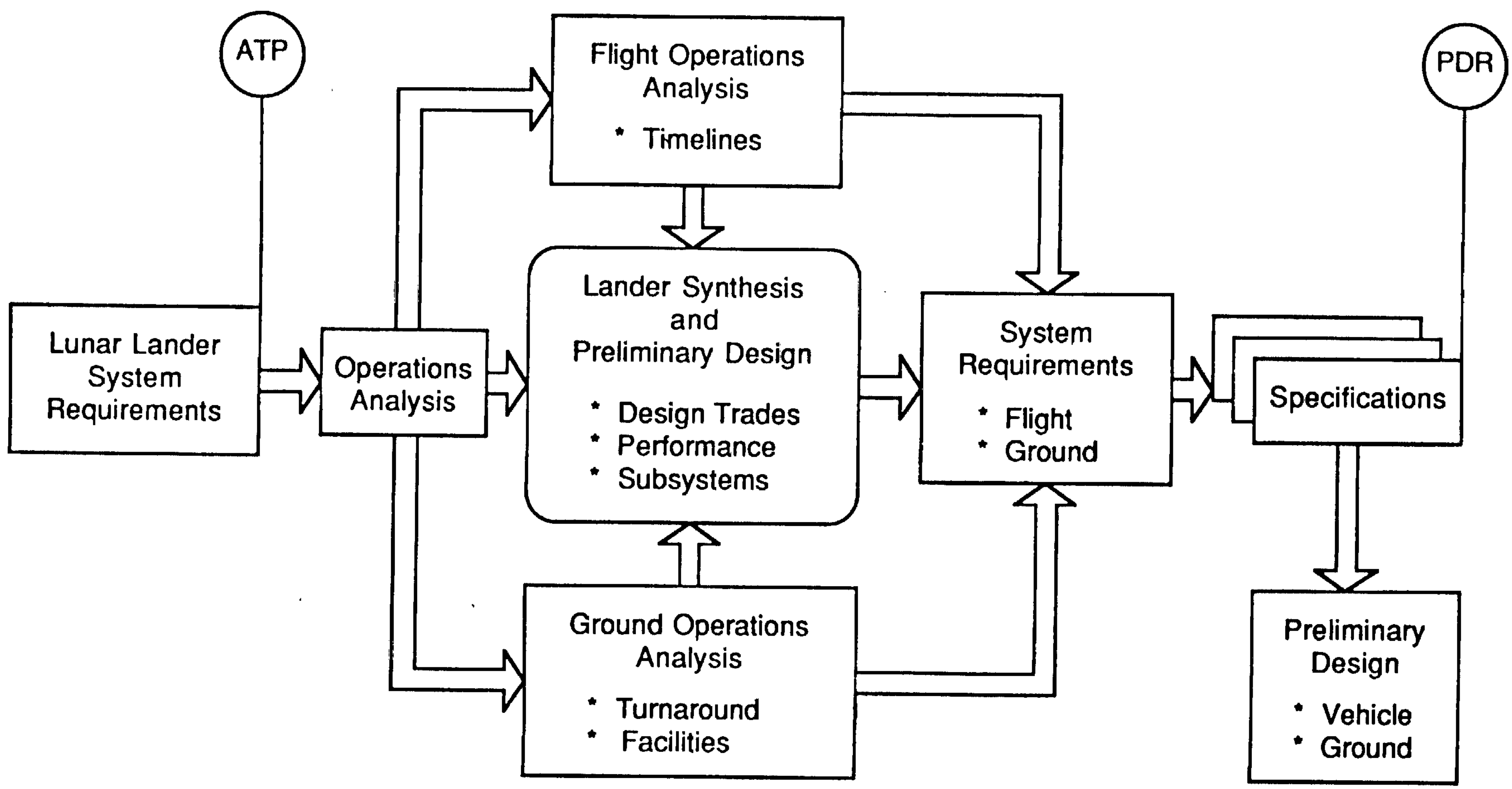


Fig. 1.2: Lunar Infrastructure Evolution

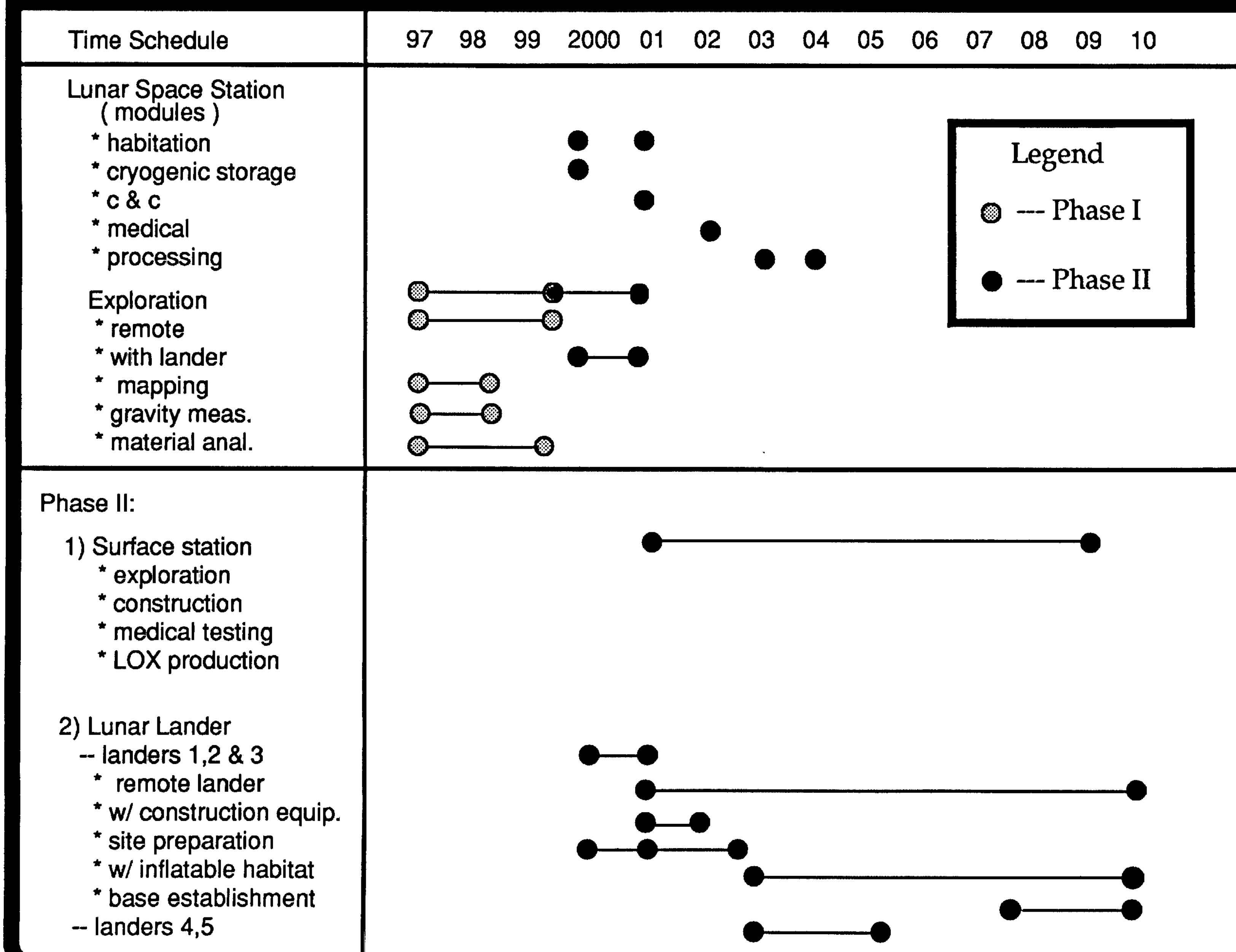


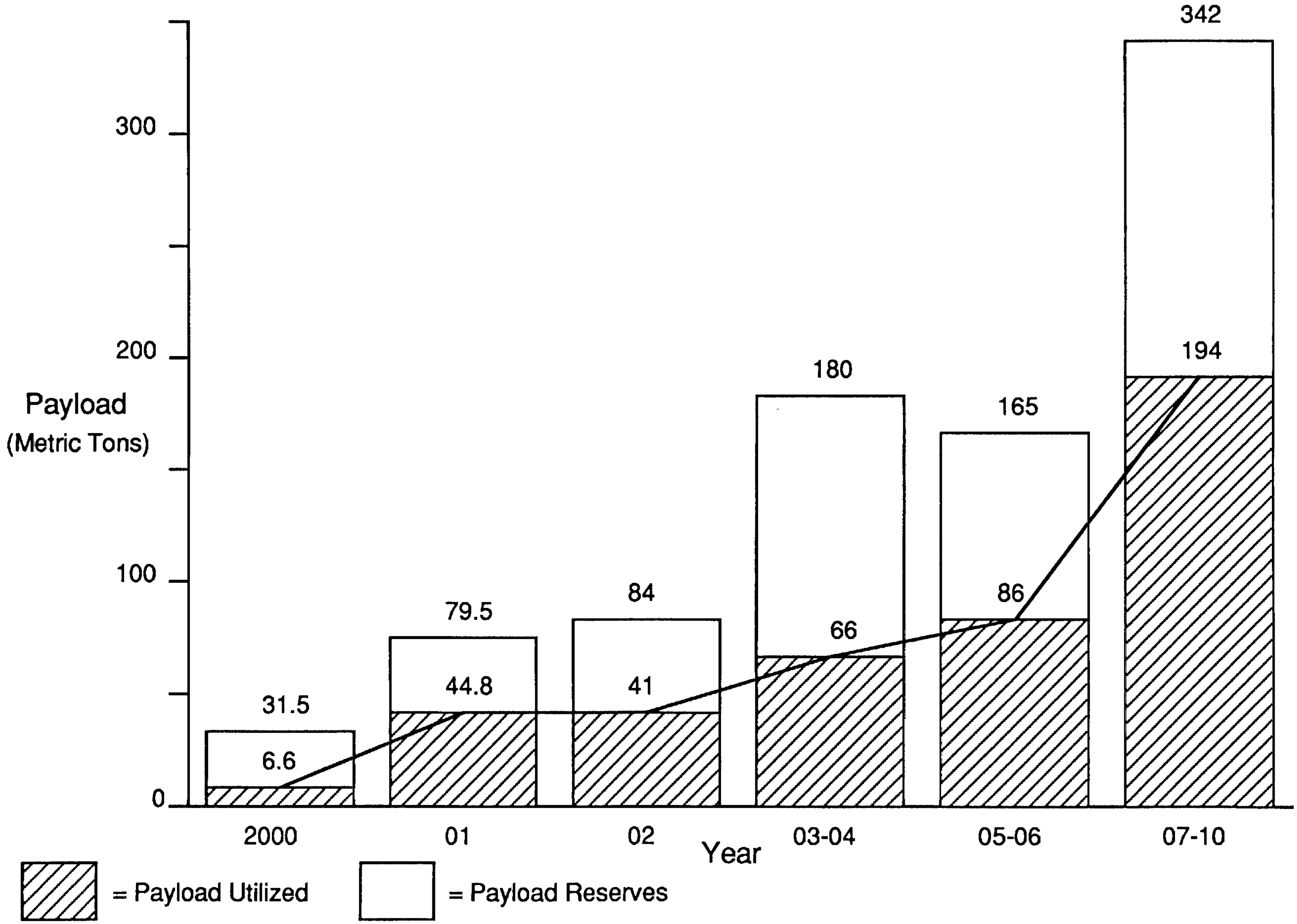
Fig. 1.3: Mission Profile

Mission \ Time	2000	01	02	03	04	05	06	07	08	09	10
Site Exploration (manned)	3										
Construction Shack		4									
Equipment			4								
Site Preparation			4								
Base Habitation											
30 Days											
- Crew				6	6						
- Cargo				2	2						
60 Days											
- Crew						4	4				
- Cargo						3	3				
90 Days											
- Crew								3	3	3	3
- Cargo								4	4	4	4

capacity per flight was reached through an iterative design analysis taking into account propellant tank sizing, propulsive thrust maximization, optimum trajectories, etc.. The final result allows the establishment of a lunar lander payload profile (Figure 1.4). The P/L profile gives mission planners a crosscut view of system payload capacity versus utilization. Mission scheduling and P/L assignment are now allowed flexibility in meeting program requirements.

Reusability, maintenance and repair requirements are to be given continuous high priority throughout the life cycle of the lander. The cost of supporting maintenance facilities in space will be enormous, and performance must be carefully weighed versus maintainability. While some maintenance may be performed in Low Earth Orbit (LEO), constantly returning the lander to LEO will seriously affect lunar resupply capability. Initially, a lunar station capable of carrying out all but the most major repairs is considered a necessity. Transfer of this capability to the lunar surface is initiated upon extended habitation/duration missions. The lander must be designed in such a way that almost all systems are easily accessible. To insure that the lander's systems are easily maintainable, provide a long service life, and achieve high reliability, proven technology should be utilized whenever possible while still meeting performance requirements. For example, the high performance rocket engines with life expectancies of the order needed to satisfy lander requirements are not yet available. For the purpose of reliability, the design of the lander propulsion system should provide redundancy where possible to reduce individual component replacement demands. In addition, the major segments of the propulsion system should be placed on a common pallet for ease of removal and resulting in ease of repairability. Other systems of the lander needing advances in state-of-the-art

Fig. 1.4: Payload Profile



and, therefore requiring redundancy, are electro-mechanical energy absorption systems, low maintenance actuation systems, and multi-layer insulation.

To facilitate lunar station and surface based maintenance a design improvement will have the lander's sub-systems (e.g., avionics, ECLSS, etc.) installed in removable racks and canisters similar to those on the station. Repair procedures inside the pressurized station will be enhanced since certain repairs require shirt-sleeve access. In addition, the lander's design should be made compatible with tele-robotic servicing in order to reduce extravehicular activity (EVA) and increase vehicle turnaround timing.

In conjunction with baseline system definition and insuring compatibility of subsystem interfaces, activities during ATP-PDR include:

a) Definition of system and subsystem requirements and interface constraints.

b) Evaluation of design and operational alternatives to select best flight/ground systems, recognizing mission requirements and launch costs.

c) Providing specifications for the procurement of operational hardware.

It should be noted that with this aerospace systems engineering approach a baseline is developed which will result in detailed designs, future fabrication of hardware and verification that the hardware and software will satisfy lunar lander program requirements [1].

CHAPTER 2: OPERATIONS

Phase I of lunar development will begin in 1997 and will entail exploration of potential Lunar Base sites using unmanned probes and orbiting satellites. These spacecraft will generate maps of the lunar surface and obtain data on mass concentrations and lunar resources.

Phase II will begin with delivery of the first Lunar Lander to lunar orbit in 2000, along with the initial modules of the Lunar Space Station. During the first year of Phase II, the Lander will be used for manned exploration of sites selected during Phase I (Figure 1.2). With the introduction of a second Lander in 2001, four missions will be scheduled for delivery of construction equipment, astronauts, and materials to the lunar surface. Certain massive payloads, such as the initial habitation module, will require that the Lander used remain on the surface until it can be refueled. See Table 2.1 for a detailed list of equipment and estimated masses. Table 2.2 lists the payload capability of the Lander for four payload delivery scenarios.

The initial delivery mission in 2001 will be cargo-intensive, and the Lander will not have sufficient fuel to reach orbit again. It will be landed remotely and will remain on the lunar surface until it can be refueled. One Lander will deliver a 20 metric ton construction "shack" derived from the basic Space Station module design and some of the materials necessary for preparation of the inflatable habitat.

The habitat will be a sphere, 16 meters in diameter and capable of supporting a crew of twelve [2]. Once inflated the sphere's inner truss structure will be installed and the habitat will be covered with sandbagged lunar soil. The mass of the inner structure will

Table 2.1: Mission Manifest

Equipment	Mass (kg)
Inflatable Habitat	2 200
Supporting Structure for Inflatable	16 300
Mobile Crane	1 000
Truck	1 400
Excavator	1 900
Rover	550
Bucket Wheel Excavator	300
Conveyor	250
Angle Dozer	200
Drill Core	250
Augers	150
Bagger	250
Module Trusses	4 800
Soil Constraints	600
Explosives	250
Landing Aids	40
Nav. Aids	100
Block & Tackle	40
Inflatable Bags	950
TOTAL	31 740 kg

Base Resupply - estimate for 60 day/six crew:

Payload	Mass (kg)
Crew Module	1 500
Food, Water, Atmosphere	4 450
EVA	400
Science	320
Equipment + Subsystems	1 800
TOTAL	8 470 kg

Table 2.2: Payload Mission Profiles

Lander Mission	Lander Payload Down (metric tons)	Lander Payload Up (metric tons)
Expendable	39	N/A
Down with Maximum, Up without payload	15	0
Down without payload, Up with Maximum	0	11
Down with payload, Up with Crew Module	9.5	4

require several Lander missions for complete delivery.

The second Lander will be used to transport the construction crew of four between the lunar surface and the LSS, and will deliver the remaining equipment and materials. A mobile crane will be included in the first construction mission for removal of heavy payloads from the other Lander. The first task will be to prepare the pressurized module to support the crew while they build more permanent facilities. During three separate day-long missions, the crew will move the module to the lunar surface, build a truss structure around it, and cover it with lunar soil for radiation protection. Other tasks include landing site smoothing and site preparation for habitat and lab modules, and the deployment of the inflatable habitat. The larger habitation module will also need to be covered with lunar soil. These preparations will require four missions each lasting about ten days.

The main power equipment will be delivered once the site has been prepared, and containers with liquid oxygen (LOX) will also be brought down to the lunar surface. The LOX will serve primarily as a backup for the habitat module's ECLSS, but may be used to refuel Landers on the surface once base operations are expanded. A vehicle will then be necessary to move LOX between the Lander and the storage tanks.

Following the establishment of a manned presence at the Base in 2003, the Lander will be primarily used for crew rotation, ECLSS resupply, and delivery of additional payloads such as a LOX pilot plant. Starting in 2004, 5 metric tons of lunar raw materials, two missions with 2.5 metric tons, will be transported to the Lunar Station for processing. Crews of four are expected to man the surface base for thirty day tours; there will be a thirty day unmanned period between tours. In 2005, after expansion of base operations, rotations will consist of six people in a sixty day "on" - thirty day "off" scenario. This will eventually be extended to ninety day "on" - thirty day "off" for the final four years of Phase II (2007-2010). (A mission timeline is given in Figure 1.2 along with a mission profile given in Figure 1.3.)

The design of the Lander was tailored to meet the requirements of three missions:

* Cargo Delivery
* Personnel Delivery, using attached module
* Cargo Delivery without return to LSS.

For the first case, the Lander can deliver 15 metric tons of cargo to the surface and will have sufficient fuel to return to lunar orbit. It will be remotely piloted from the LSS Control Module. For the second case an non-pressurized crew module will be attached to the Lander cargo pallet. It provides seating for six space-suited astronauts, and the option of piloting the Lander from either the crew module or the LSS. Additional crew may travel in the module; there will be straps with which they can secure themselves while standing. If payloads over the 15 metric ton limit are desired, the Lander may be remotely landed with up to 39 metric tons of cargo. It would reach the surface with its minimum design fuel reserve of 15% and would remain there until it could be refueled.

Transportation Nodes

The Lunar Lander will be taken into low earth orbit (LEO) by the Advanced Launch System (ALS). Two designs are currently being considered by NASA and the Department of Defense. In scenario I, the ALS will have a payload envelope of 33 ft. in diameter and 80 ft. in length, including the dome, and the launch vehicle will be capable of transporting 80,000-120,000 lbs. (36-54 metric tons) of payload. In scenario II, the ALS would have an envelope of 43 ft. in diameter and 120 ft. in length, again including the dome, and the vehicle will be able to carry up 120,000-220,000 lbs. (54-100 metric tons) of payload. The ALS of scenario I will be used to transport the Lander to LEO; the Lander will be taken up without propellant and then fueled at LEO in order to maximize launch safety and minimize cryogenic propellant bleed-off. The ALS will be operational in 1998. A total of five ALS missions will be required during the ten year period of Phase II lunar development. Payloads for the Lander will be taken up by the Space Shuttle or the Shuttle-C derivative.

A departure window for a minimum delta V transfer from the LEO space station to lunar orbit will be available every nine days [3]. One OTV mission will be required for the delivery of each Lander to lunar orbit, as well as additional missions for the delivery of cryogenic fuel.

CHAPTER 3: ORBITAL MECHANICS

Low Earth Orbit to Low Lunar Orbit Transfers

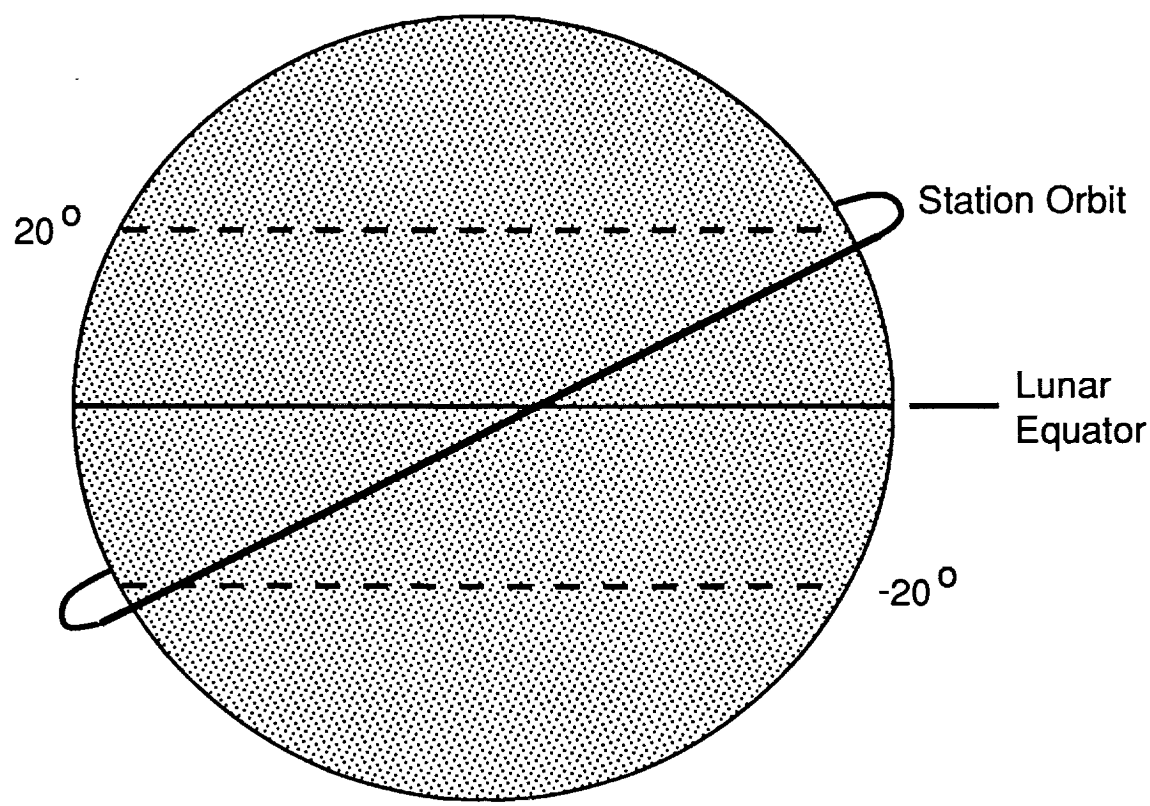
It is assumed that an Orbital Transfer Vehicle based at the LEO Space Station will be used to transfer cargo and personnel to the LLO Space Station; the Lunar Lander will operate only between LLO and the lunar surface.

Plane Changes

Operating from the Lunar Station's inclined orbit, the Lunar Lander will pass over or nearly over all sites within 20° of the equator once every two weeks (Appendix I). Proper scheduling should thus allow the Lander to reach any desired point on the surface without consuming fuel in a plane change maneuver, as shown in Figure 3.1. This will restrict the landing schedule as a function of station position, but will yield significant savings of fuel, and thus of operating costs and the required number of OTV resupply missions.

For scientific or exploratory missions beyond the ±20° band, the Lander will be capable of round trips to sites within 58° of the equator while carrying the four metric ton crew module. In case of emergency, the Lander can reach sites within 41° of the current orbit and still return to the Lunar Station if stripped to the basic structure and fully fueled. Emergency missions beyond this inclination will require that the Lander either carry extra fuel tanks or that it be refueled by another vehicle after its initial plane change.

Fig. 3.1: Landing Zone



No Plane Change

Landing Trajectory

The Lander will make an initial in-plane burn to enter a Hohmann transfer ellipse from the Lunar Station altitude of 200 kilometers down to a lower orbit 93 kilometers high; this is illustrated in Figure 3.2. Any plane-change burn would be combined with the initial burn for minimum fuel consumption. The burn is made 90° ahead of the desired landing site, in order to position the perigee of the ellipse directly above the site.

Slightly before reaching perigee the Lander thrusts a second time to circularize its orbit, and then begins a minimum fuel maneuver to bring it to the lunar surface [1]. The engines fire in the direction of travel to reduce velocity, and the trajectory begins curving downward. When the Lander reaches a position directly above the desired landing site, the horizontal velocity will have been reduced to zero and the Lander will perform a pitch-over maneuver to orient itself vertically. The main engines continue firing to slow the rate of descent to 1.6 meters per second, and then throttle back to maintain this rate until shut-down two meters above the lunar surface. The Lander drops the remaining distance. The touch-down sequence is illustrated in Figure 3.3.

The time after pitch-over at which the final descent rate is reached is dependent on the desired duration of the vertical descent. If a large hover time is desired, as on an exploratory mission where careful selection of the exact landing site will be required, the engines continue to provide maximum thrust until the final rate of descent is reached. In the case of a landing at the Lunar Base, the site is assumed to be clear of obstructions and a faster descent pattern may be used, with the rate of 1.6 meters per second being reached only slightly before main engine cut-off.

Fig. 3.2: Launch/Landing Pattern

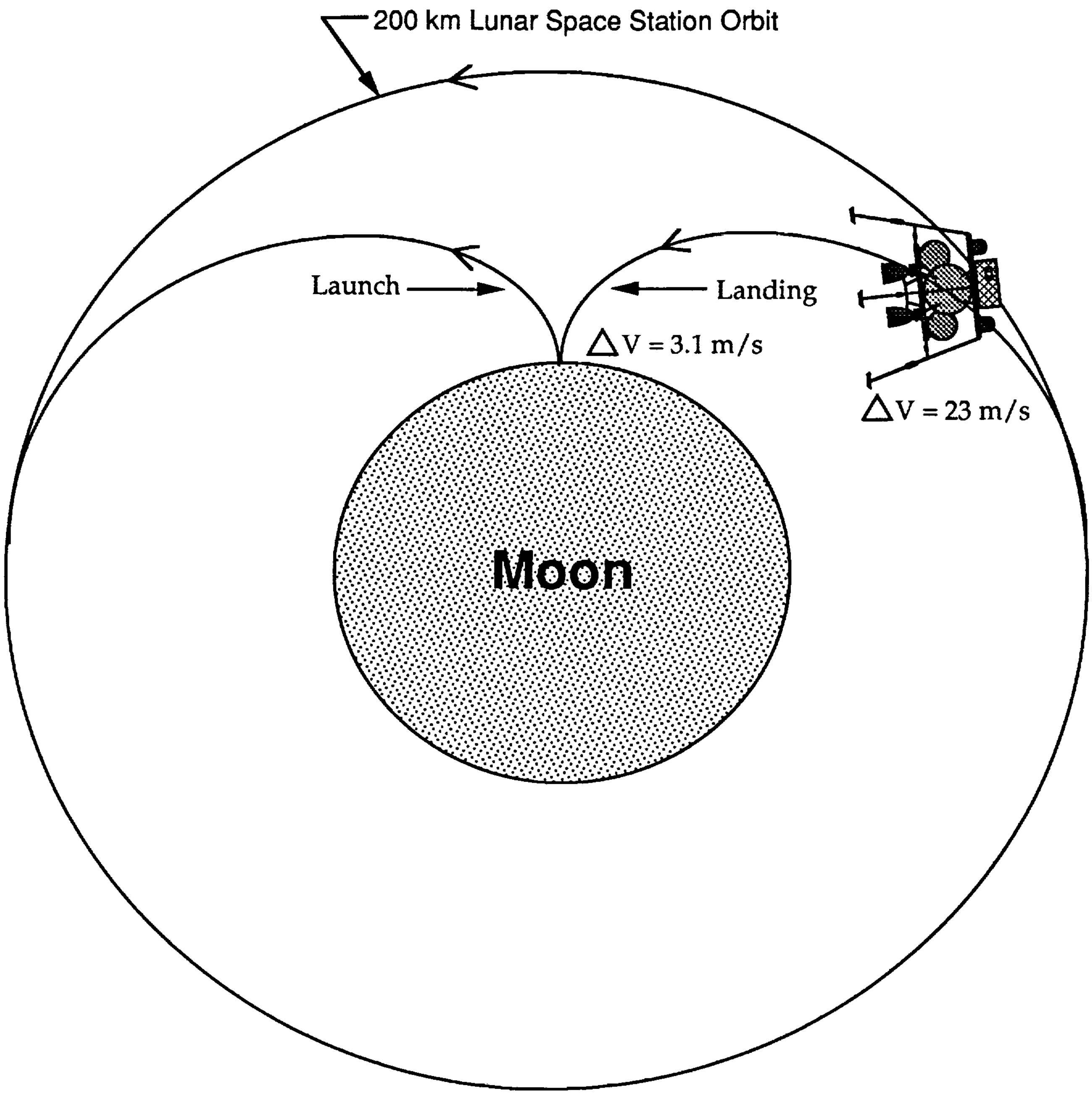
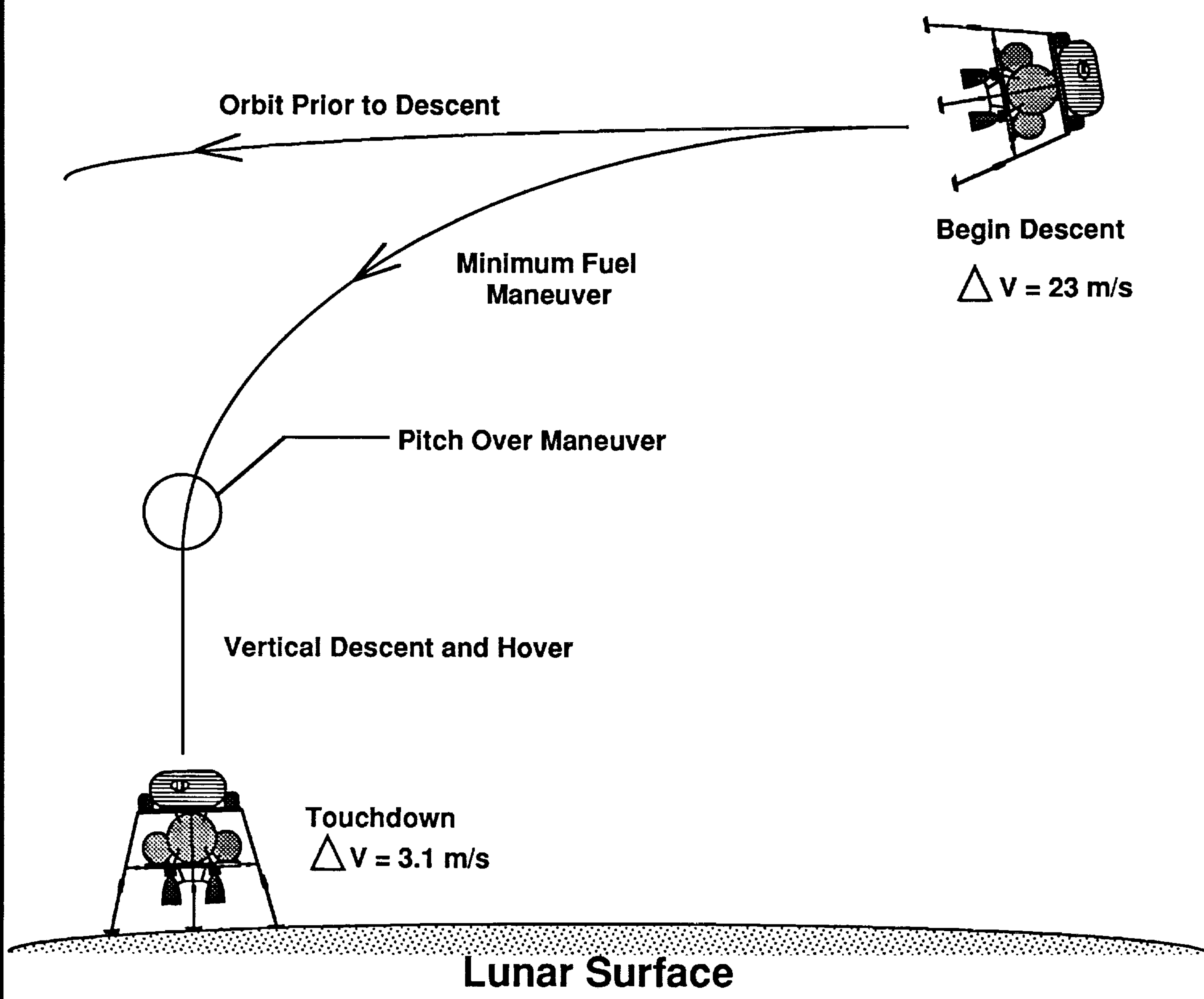


Fig. 3.3: Touchdown Sequence



Launch Trajectory

The Lander lifts off at maximum thrust, rising vertically at first and then executing a minimum fuel maneuver to return to the elliptical transfer orbit used for landing. Upon reaching the velocity required for transfer out to the orbit of the Lunar Station, the Lander main engines are shut off and the vehicle coasts to intercept with the Station. At Station altitude the Lander makes a circularization burn and any required plane change.

CHAPTER 4: STRUCTURES

All aspects pertaining to the Lander structure are discussed in this chapter. Firstly, the structural options of each of the three mission scenarios are examined. Next, a discussion of the structural requirements as determined by mission sequence, long term operation, and human safety factors is undertaken. Here, the types of analyses used in the design are discussed. The supporting framework of the structure, propellant tank arrangements and attachments, and the gimballing structure of the engine are addressed in the following section. Finally, an explanation of the cargo pallet structure and docking and cargo transfer is given. Also, a weight statement for the Lander's structural members is given.

4.1 Introduction & Design Options
Mission Requirements

For the second phase of lunar base development, a Lunar Lander has been designed meeting the following criteria:

1. 39,000 kilogram descent payload capacity with no ascent capability -- an expendable Lander.
2. 15,000 kilogram descent payload with ascent capability carrying no payload.
3. 4,000 kilogram descent/ascent payload capability.

The Lander will be responsible for bringing construction equipment as well as building and living supplies to the lunar base personnel. In addition, it will deliver raw materials to the space station's processing factory in LLO from the lunar surface.

The Lander will also serve one other function, that is, as a personnel shuttle. The distinction between the cargo transportation option and the shuttle option is a simple one. In the cargo transportation option, large or small payloads can be attached to the cargo pallet located above the propellant tanks. A manned module is attached to the top of the cargo pallet in the shuttle option. The vehicle is remotely piloted from this module.

Required Components

The following components of the Lander structure are discussed in this report: the structural frame, the propellant tanks and attachments, the landing gear and landing legs, the footpads, the turbomachinery and fuel lines, the engine, the gimbal hinge and its supporting frame, the manned module, and the cargo pallet. The location of the avionics, communication radar, and power systems is also discussed as is the sizing of the members of the frame.

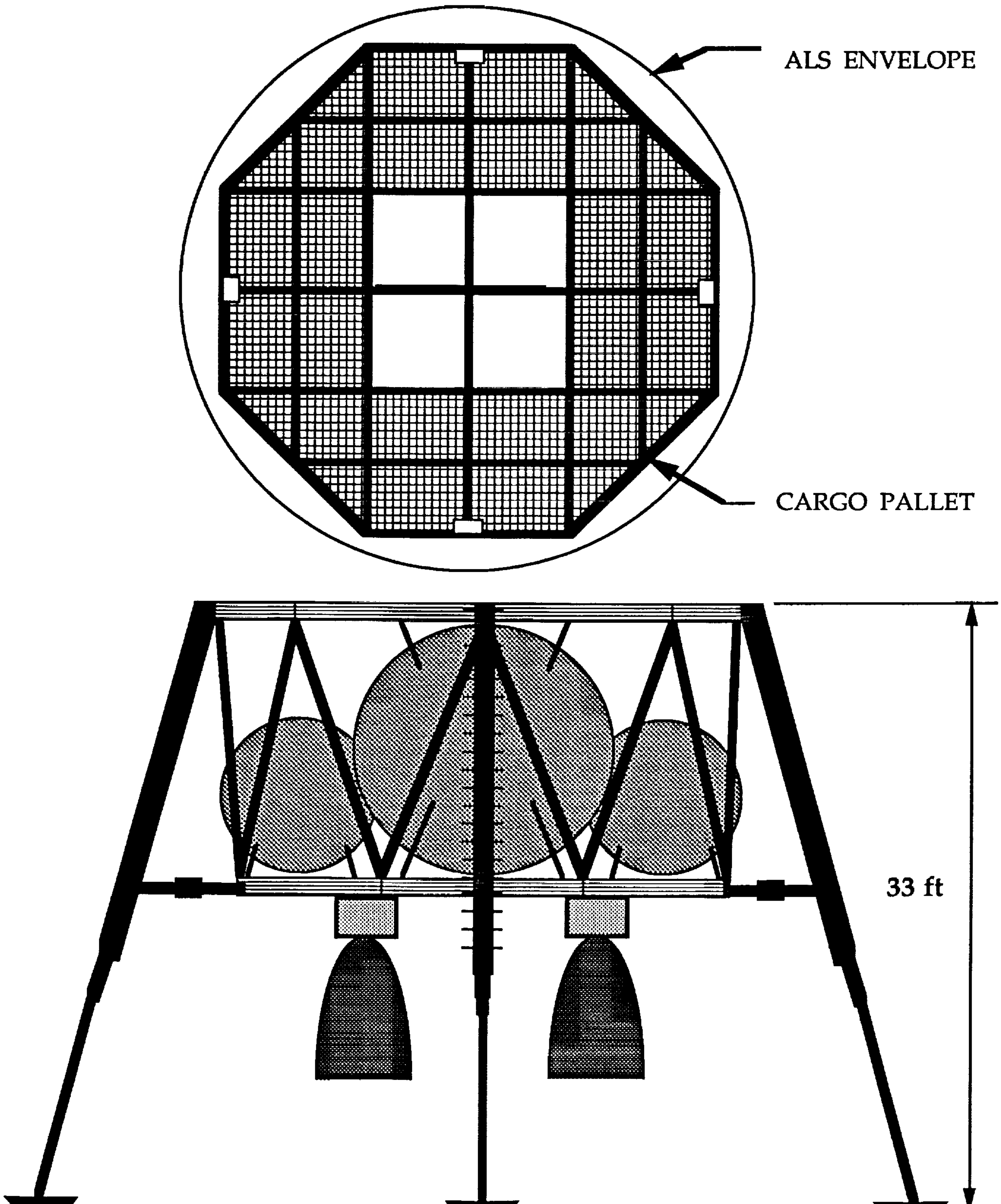
In this chapter of the report, the structural frame, propellant tanks, gimballing, cargo pallet and cargo transfer are detailed. Also, a weight statement is given.

Mission Structural Requirements

Each of the above components has been designed to endure all phases of the Lander's lifetime beginning with its transport to LLO in the OTV. The major constraint here is that the Lander must fit into the ALS. The firing of the Lander's engines in LLO and during liftoff, subjecting the engine and structure to vibration, loads, and high thermal gradients, was also considered. A low center of gravity and an effective attitude control system were incorporated into the Lander for good inclined impact landing characteristics. In addition, certain important long run factors were targeted for the design including: (1) low weight to minimize fuel consumption and maximize payload, (2) ease of maintenance, and (3) minimization of structural fatigue and micrometeoroid damage. Occupant safety was considered of paramount importance in both the long and short run.

The Lander design proposed to meet the stated mission requirements is shown in Figure 4.1. It stands just over 34 feet tall and is just over 30 feet in overall diameter with the legs in their collapsed position. The individual structural components of the Lander are discussed below.

Fig. 4.1: Overall View



4.2 The Structural Frame

There are two types of frame options which have been used for past legged Landers -- that of the Apollo and Viking Landers. These frames have been tested for axial, shear, and torsional loading, as well as applied bending moments in various landing scenarios. Also, resonant frequency characteristics have been determined for these frames [3:9]. Our Lander's structure resembles of the Apollo LEM, having an octagonal frame.

The propellant tank arrangement shown in Figure 4.2 rests upon the octagonal framework shown in Figure 4.3. This octagonal frame provides support for the payload and cargo pallet via truss members connected around the circumference of the propellant tanks. The frame also provides support for the rocket engines by way of their gimbal hinges, and for the propellant tanks by way of their supports. The four rocket engines will hang below the tank level, allowing plenty of access from below for maintenance. In order to avoid the threat of endangering the landing legs with the engine exhaust, the engines were placed at the four corners of the octagon not having legs. The four landing legs have three attachment points per leg -- two on the lower supporting frame and one on the cargo pallet.

As illustrated in Figure 4.4, the upper octagonal frame is oriented a 22.5 degree axial rotation from a lower octagonal frame of equal dimensions, with truss members supporting the pair. Truss members connect the corners of the upper and lower octagons.

I-beams were chosen as the members of the octagonal platforms because they will primarily experience bending stress and shear loads due to engine and propellant tank support (see Figure 4.5). The truss members, however, are tubular so as to withstand the compressive stress of supporting the cargo pallet without buckling. The structural members will be made of an aluminum lithium alloy, such as Weldolite, because of its high strength-to-weight ratio. Also, the propellant tanks, discussed next, will be made of this alloy.

Fig. 4.2: Propellant Tank Arrangement

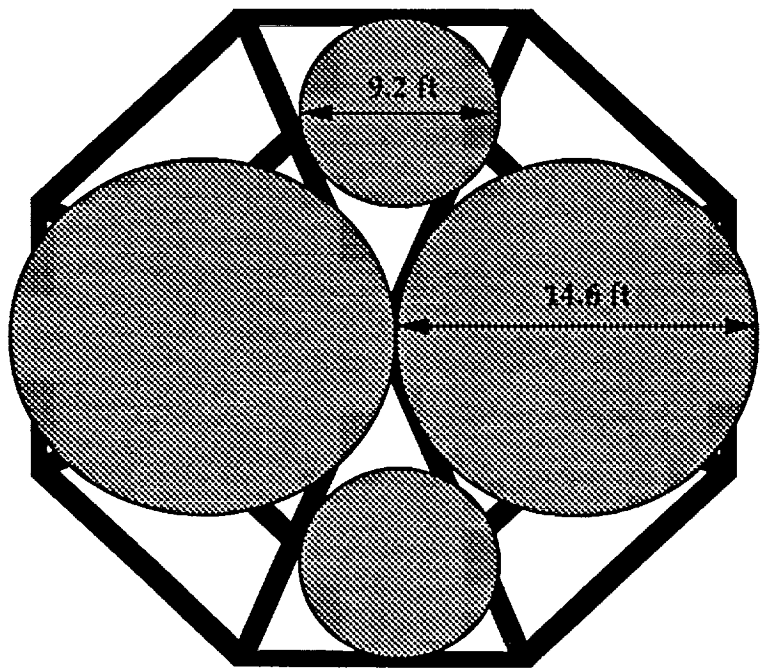
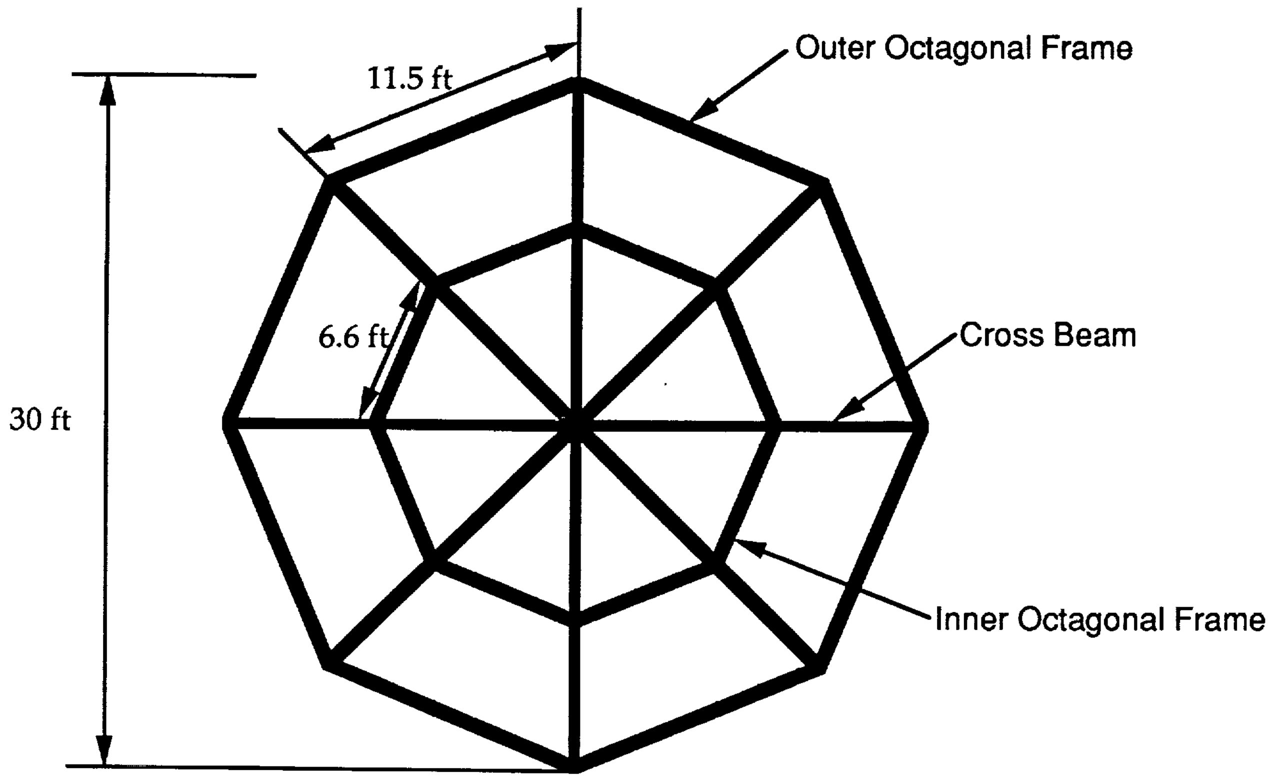


Fig. 4.3: Engine Pallet Configuration



Rocket Pallet Rotated with respect to the Cargo Pallet

Fig. 4.4: Overview

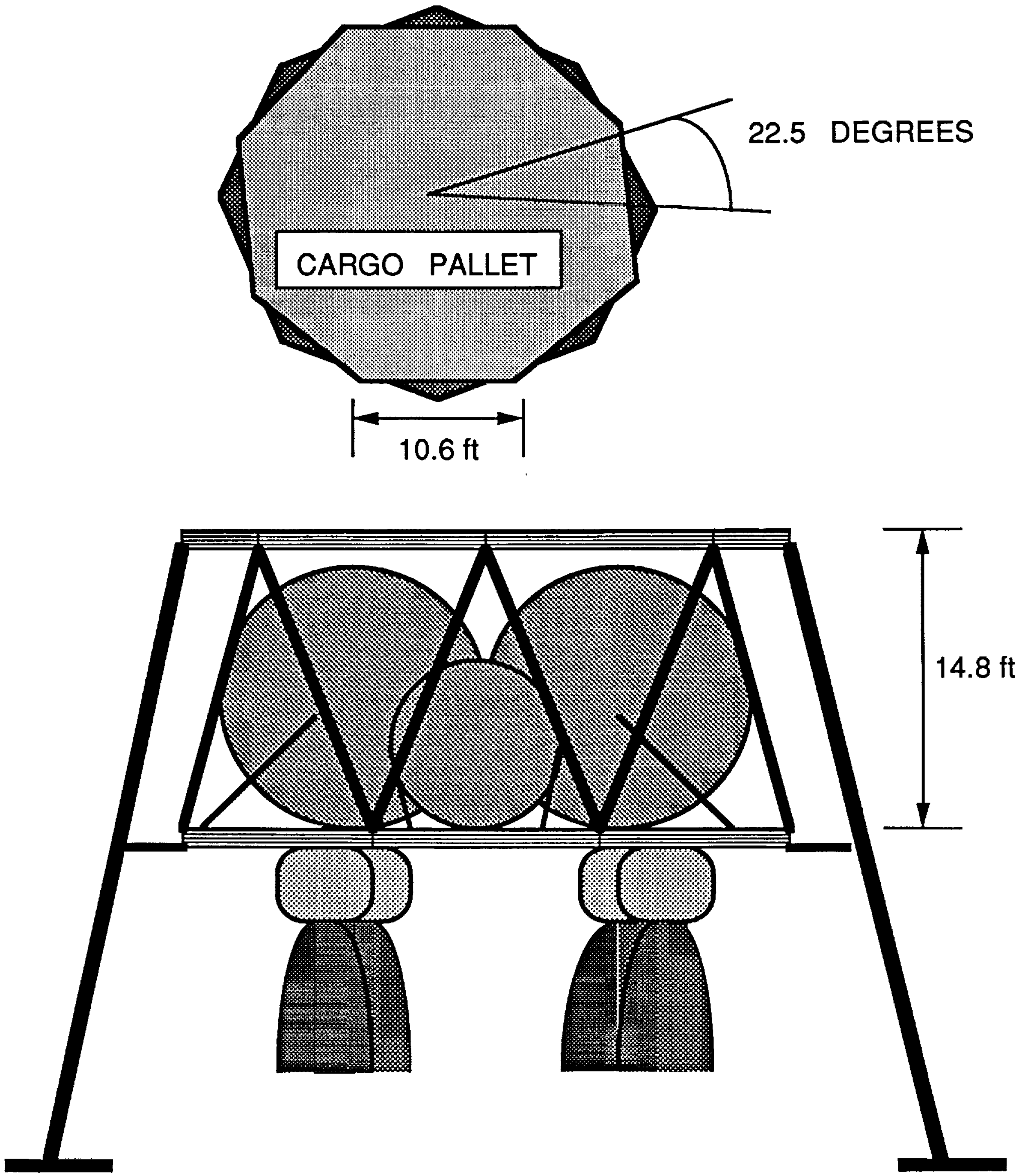
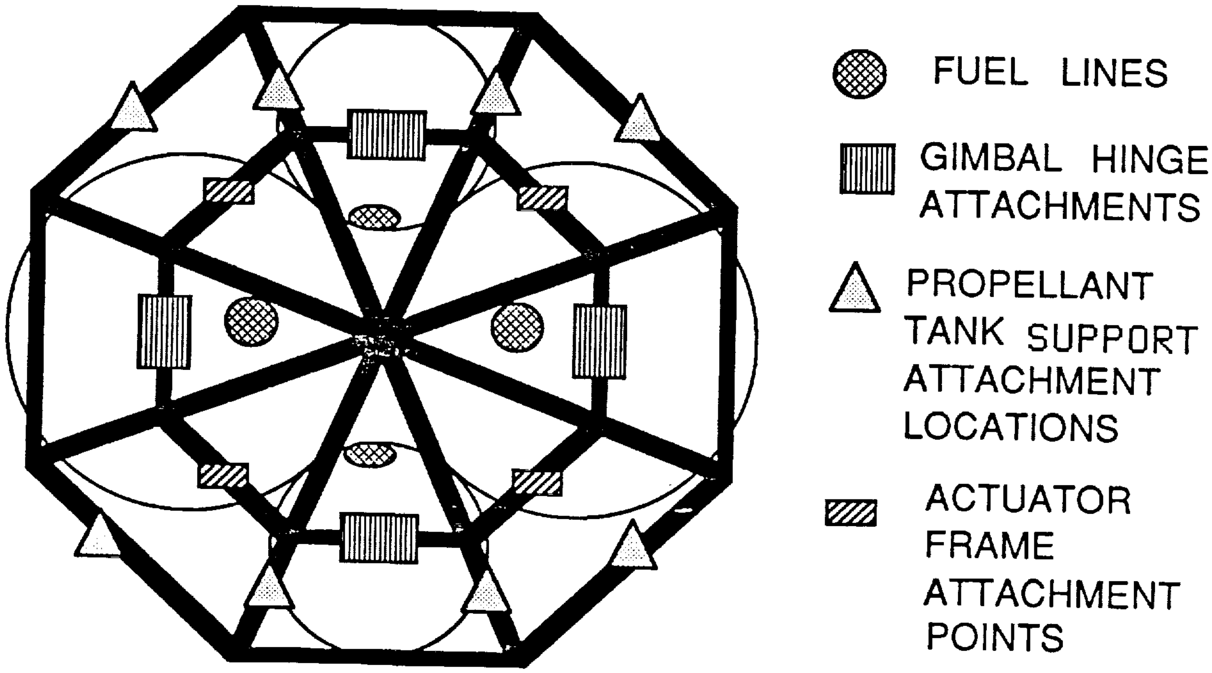
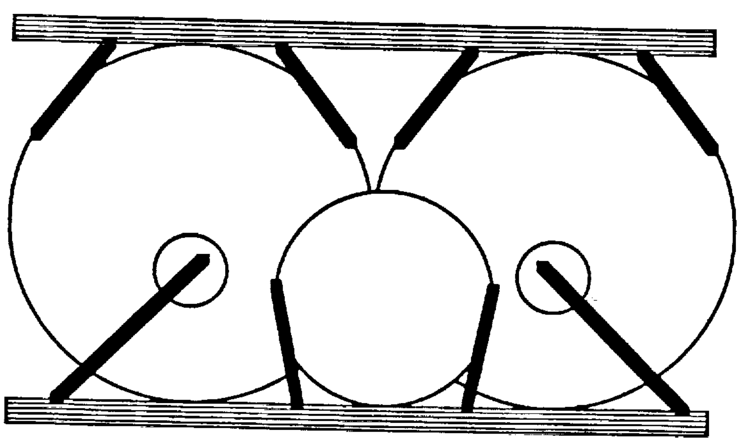


FIG. 4.5: PROPELLANT TANK ATTACHMENTS



SIDE VIEW OF PROPELLANT TANK ATTACHMENTS SUPPORTS



TANK SUPPORT ATTACHMENTS ARE INTEGRATED INTO SKIN

TRUSS MEMBERS ARE REMOVED TO ALLOW A BETTER VIEW OF SUPPORTS

4.3 Propellant Tank Arrangements and Attachments

The arrangement for the propellant tanks shown in Figure 4.2 was driven by the following: (1) a need to fit the required mass of propellant into the ALS, (2) propellant tank weight minimization, and (3) Lander moment of inertia minimization. This tank arrangement choice was an important one since the tanks are a significant fraction of the total Lander mass. Other important aspects of this arrangement are its symmetry (for stability) and the easy service access to the turbomachinery of the engine. It is noteworthy to mention that the number and size of the propellant tanks that can be fit within the assumed ALS inner diameter of 33 feet without being stacked are limited. Again, Figure 4.2 shows the best choice.

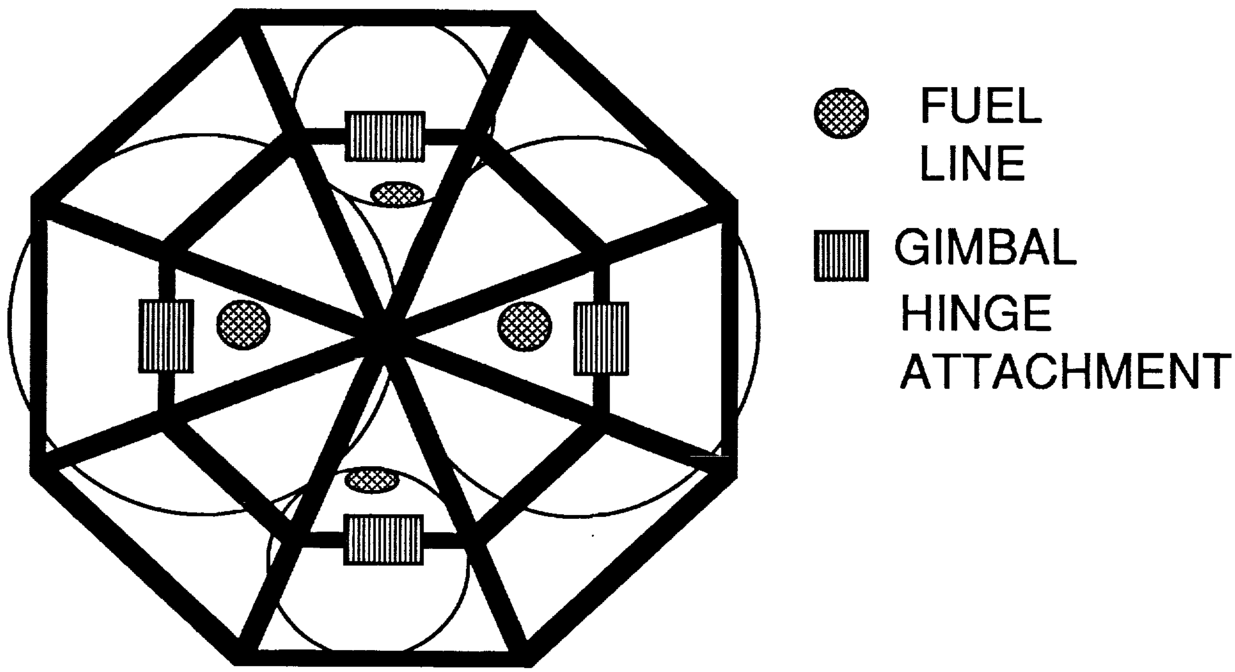
A 4 to 1 mixture ratio of fuel to oxidizer was chosen, since this ratio of LOX to LH2 yields the greatest specific impulse [8:192]. Using this ratio and an estimate of the total required propellant, the tank size was calculated using the Propellant Mass Program (see Appendix VI). The liquid oxygen and hydrogen tank sizes were calculated to be 9.2 feet and 14.6 feet in diameter, respectively. As previously stated, an aluminum alloy will be used for the tank skin.

Once the tank size was known, various arrangements were examined. A study of tradeoffs of various arrangements is given in Appendix II. After much consideration, the tank arrangement in Figure 4.2 was finally chosen.

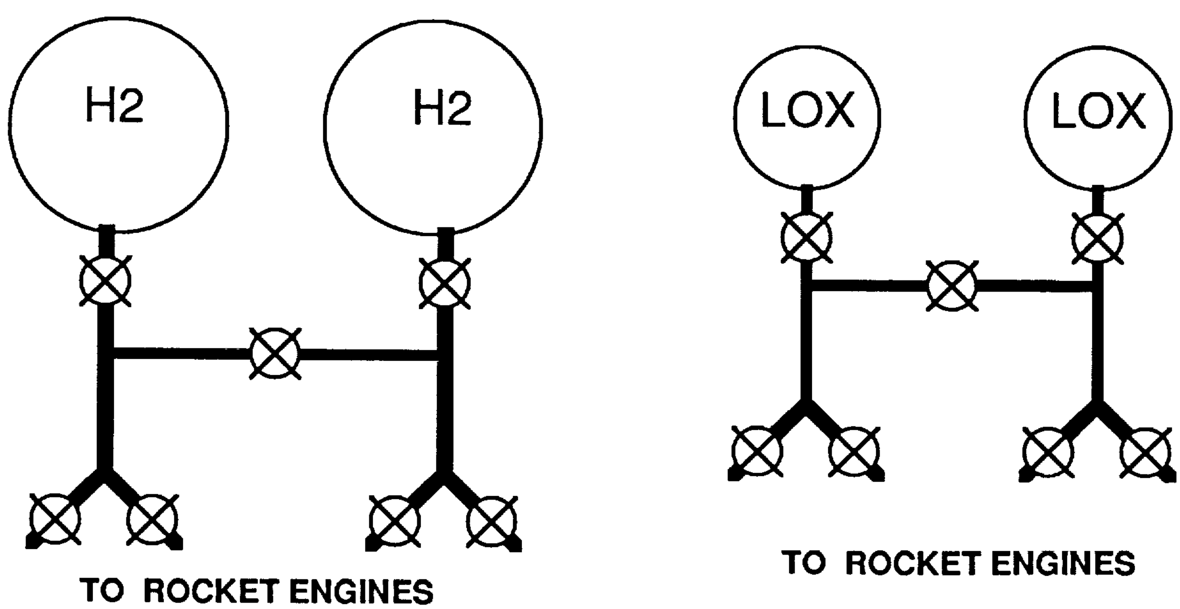
For the design of the propellant tank attachments, shown in Figure 4.5, it was assumed that connections to the tank would be made integral with the tank skin in the manufacturing process. There are two connections to each of the four tanks from the lower octagon, and two connections to the hydrogen tanks from the cargo pallet. There is one connection to the top of each oxygen tank from the truss members. Thermal conduction to the tanks from the engines beneath the lower frame and configuration stability in the event of a rough landing were the primary considerations for designing the attachments.

The fuel lines to the tanks are shown in Figure 4.6. The oxygen tanks and hydrogen tanks have a connecting line between them to provide for the diversion of fuel in the event of tank or engine failure. Valves are located at various locations along the lines, and fuel lines are spaced so as not to interfere with the turbomachinery of the rocket engines or the gimballing of the rocket, which is discussed next.

Fig. 4.6: Fuel Lines



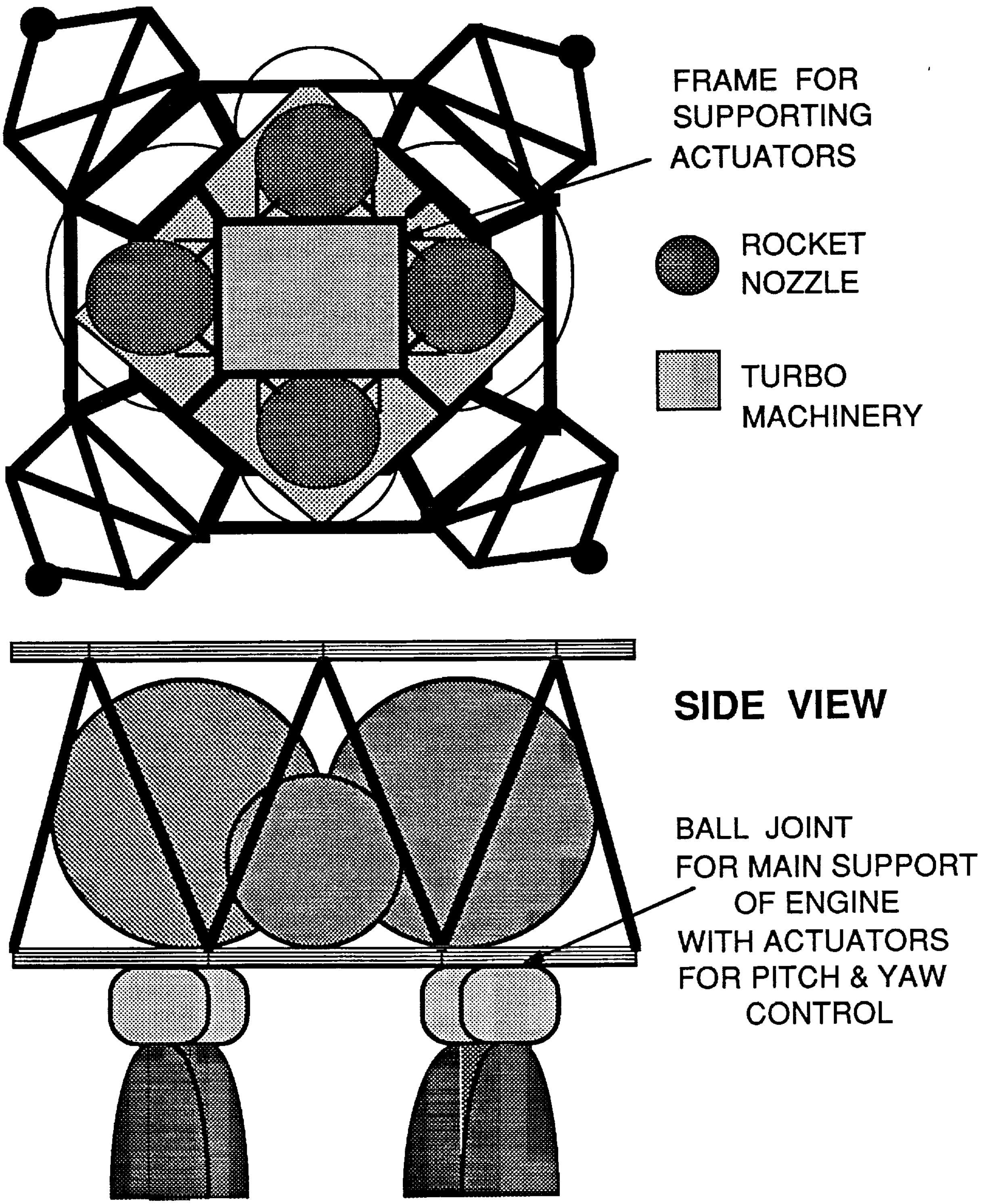
SCHEMATIC (DOUBLE REDUNDANCY)



4.4 Gimbal Hinge Actuator and Support Structure

The gimbal hinge consists of a ball joint connected to an inner cross member of the octagonal framework. Figure 4.7 illustrates the frame used to support the actuators which are, in turn, used to gimbal the engines. There are two actuators per engine connected between the rocket nozzle and the supporting frame, thus allowing two rotational degrees of freedom for each engine. This design allows each engine to gimbal as much as 10 degrees in any direction without damaging any of the other engines or the legs.

FIG. 4.7: ACTUATORS



4.5 Cargo Pallet

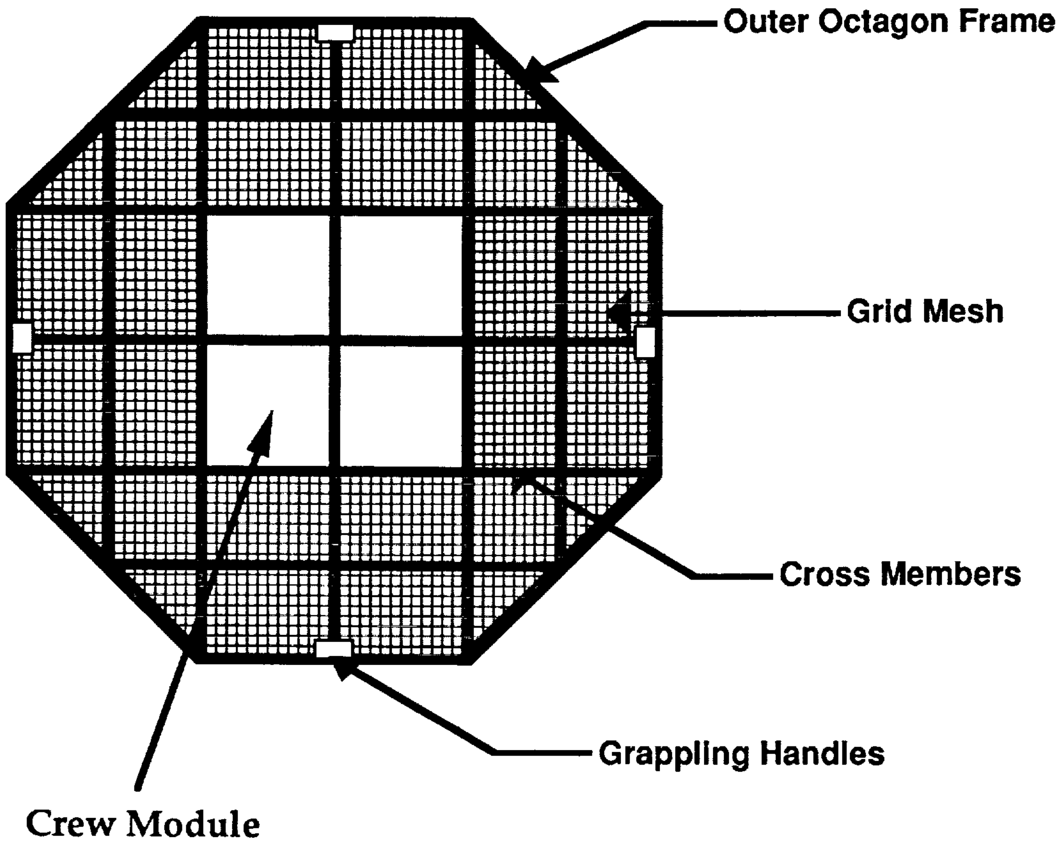
The cargo pallet shown in Figure 4.8 consists of a grid of supporting beams with a finer mesh placed on top. The center section is open to allow for the placement of a manned module. This design provides (1) for the carrying of large or small cargo and (2) a platform for the astronauts to stand on when leaving or entering the manned module.

The grid design allows for the most convenient placement of cargo. As in a C-5 airplane, the grid allows the cargo to be positioned and locked into place in such a way so as to best balance the craft with respect to weight without being constrained to aligning the cargo with certain attachment points. Since cargo will be loaded onto and unloaded from the pallet by a crane on the lunar surface and a mechanical arm on the station, the grid will save cargo manipulation time and may allow for more bulky items to be brought down on the Lander.

Soil Containers

The soil containers used to carry up the soil needed for fiberglass and silicon production on board the station will be relatively small. Due to the lunar soil density and the Lander's payload return limit of 4,000 kilograms, the size of a cubic container needed to hold the 900 kilograms of soil required for one batch process of silicon production is only about 3 feet (0.91 meters) per side. Therefore, a maximum of four containers of this size could be brought up at one time. The fiberglass production, requiring three cubic meters of soil per batch, would require two containers of the lunar soil. The containers will have a grab point, allowing for easy crane manipulation.

FIG. 4.8: Cargo Pallet Configuration



Manned Crew Module

The manned crew module shown in Figure 4.9 is capable of accommodating six people and is placed on the cargo pallet. Its overall dimensions are 12 X 13 X 7 feet. A remote control system will be located on board the module, as stated earlier. The crew module will be non-pressurized due to the short flight time (approximately thirty minutes from LLO to the lunar surface). Thus, the need for airlocks is eliminated. This choice also reduces the required structural weight of the module. The walls will be constructed with a layer of corrugated aluminum and several layers of kevlar (for micrometeoroid protection) sandwiched between two aluminum sheets.

The astronauts will wear suits, and an extra day's supply of oxygen will be located in the storage areas. While on the surface, it is assumed that the astronauts will be using the oxygen from the pressurized safety shack at the base instead of the oxygen stored in the manned module. Additional storage space for food, water, and emergency supplies is located under the seats. A ladder attached to one of the Lander legs will provide the astronauts with an easy access to and return from the lunar surface.

Also, a hatch is located on the top of the module for docking with the space station. This docking hatch will be discussed later in the text.

Cargo/Module Attachments

The manned module and cargo will be held in place with locking mechanisms such as those shown in Figure 4.10. These mechanisms are located on the underside of the manned module and the soil containers at the four corners and will grip the beams or the grid beneath them. The catches on the manned module may be released from within the module. This feature will simplify docking procedures, as described below.

FIG 4.9: MANNED CREW MODULE
Non-pressurized

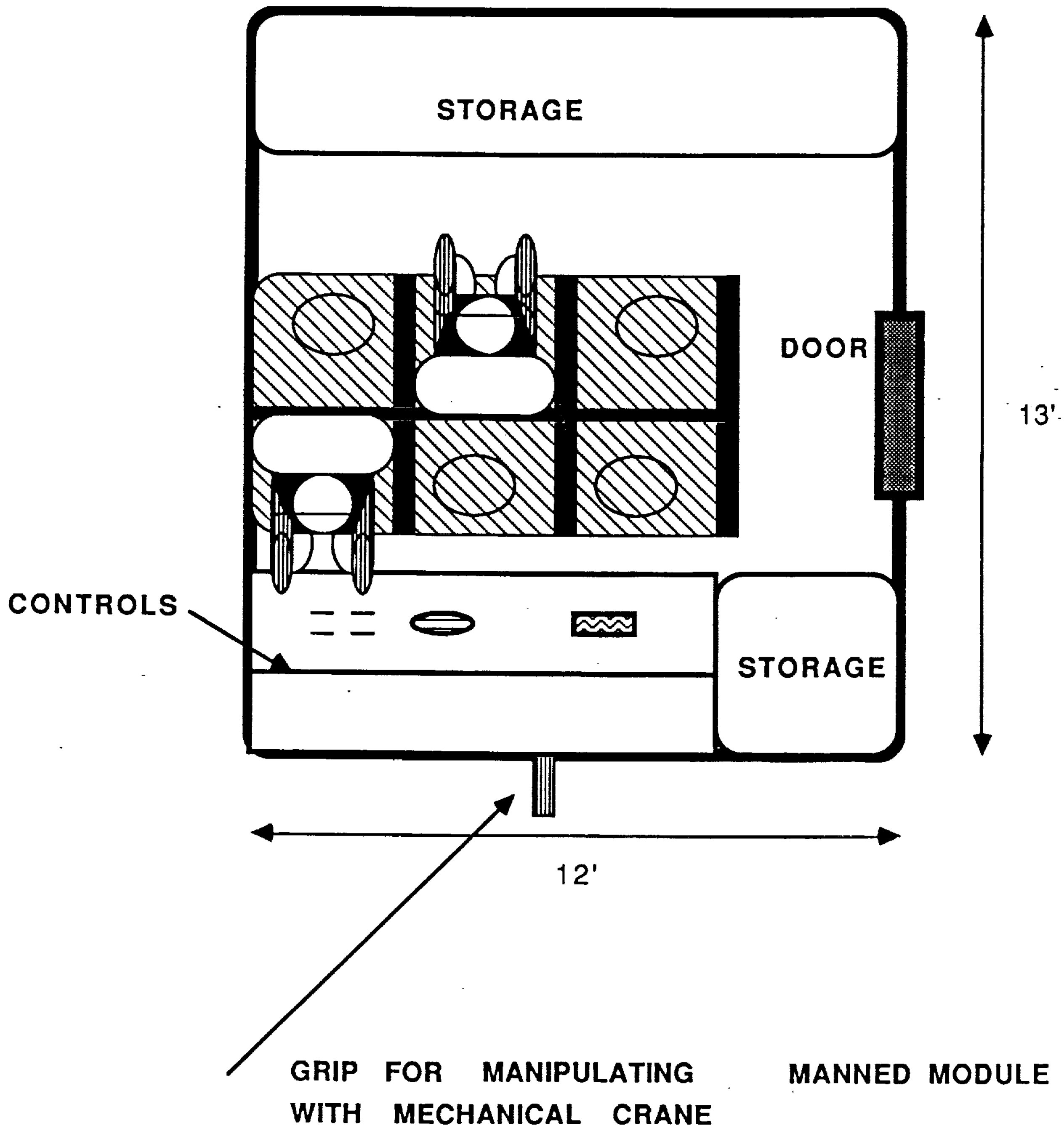
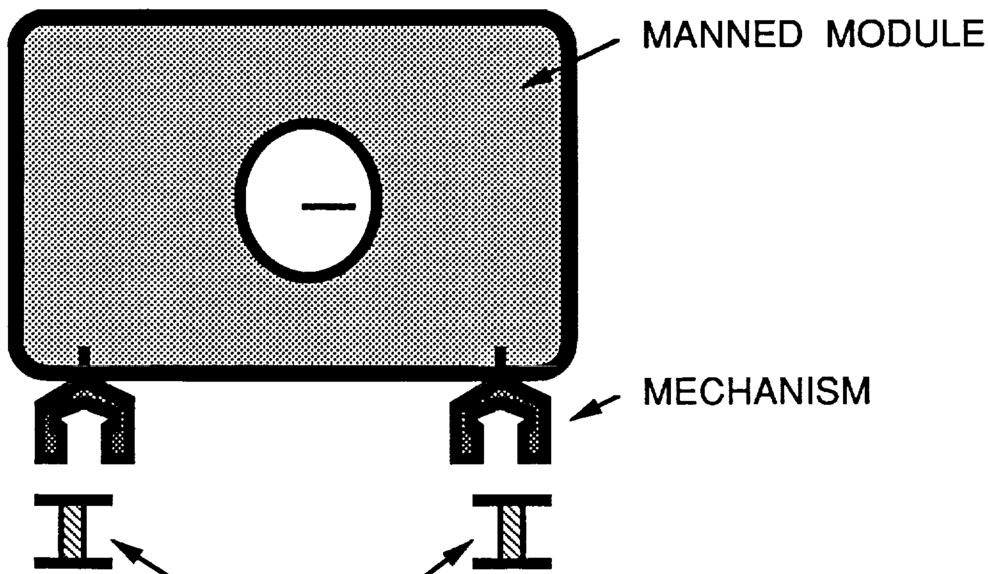
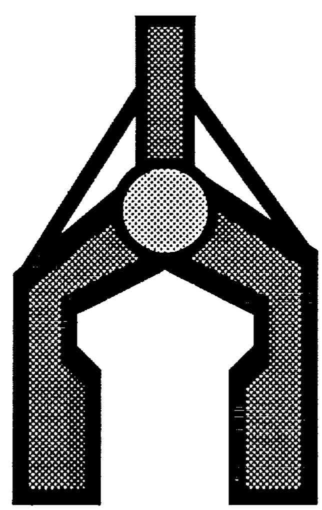


Fig. 4.10: Module Attachments



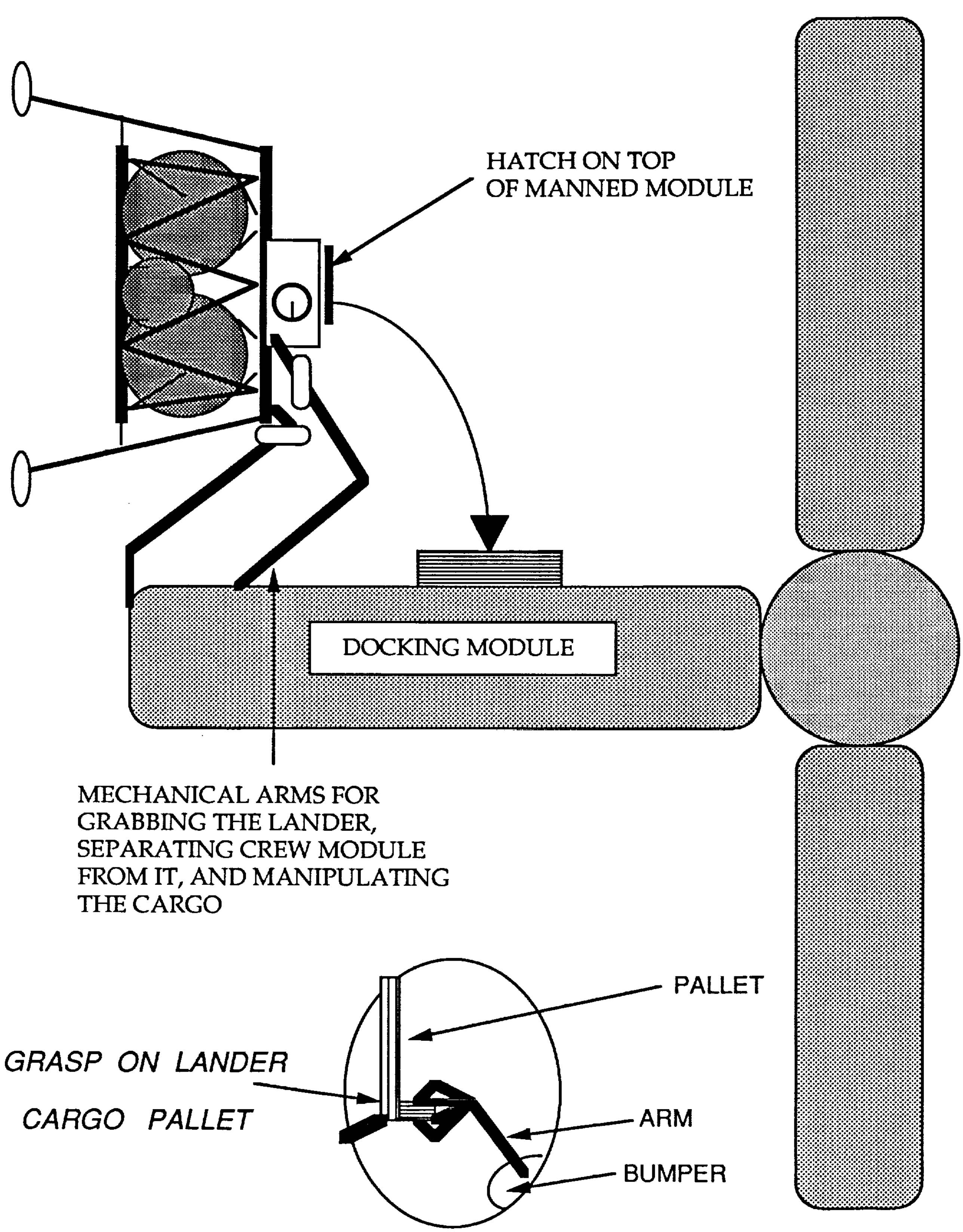


CLOSE UP OF ATTACHMENT MECHANISM

Docking

The docking system shown in Figure 4.11 utilizes the Lander's attitude control system for Lander position control and the use of two "grabbing" arms with bumpers for Lander capture. After the Lander has been maneuvered into position by its attitude control system, one of the arms will grasp a grab point on one of the legs, and the other will grasp a grab point on the manned module. Then the module's attachment mechanisms will be released, and the arms will separate the manned module from the rest of the Lander, bringing it into the docking bay and airlock. If the Lander is on a cargo dedicated mission, then the arm can be used, with the aid of a suited astronaut, to remove additional cargo from the top of the Lander.

Fig. 4.11: Docking



4.6 WEIGHT STATEMENTS

Target weights for the various aspects of the Lander are given in Table 4.1. These target weights were estimated from Eagle Engineering's Lunar Lander Conceptual Design [5:87] and from the sizing analysis of frame members performed in Appendix III. The individual sizing and weight estimates for each of these elements is also given in Appendix III. Finite element analysis using SIMPAL was preformed to determine the sizing of the members of the cargo pallet and the octagonal frames. The pallets were divided into quarters and analyzed so as to simplify and speed the analysis process. Table 4.2 gives the masses of some specific components of the structure.

TABLE 4.1: WEIGHT STATEMENTS

(For 15,000 kilograms down, with no return payload)

Expected Payload	15 000 kg
Total Inert Mass	13 500
Structure	2 000
Engines	1 000
Attitude Control System	1 000
Avionics and Power System	1 000
Propellant Tanks	7 000
Turbomachinery	1 500
Propellant Mass	35 000
Usable Propellant	33 250
Unusable Propellant (3%)	1 750
Attitude Control Propellant	200
TOTAL MASS	**63 500 kg**

TABLE 4.2: COMPONENT MASSES

Cargo Pallet	260 kg
Octagonal Frame	290
Actuator Frame	50
Tank Attachments	45
Truss Attachments	130
Landing Legs	820
TOTAL MASS	**1 595**

4.7 Landing Leg Assembly

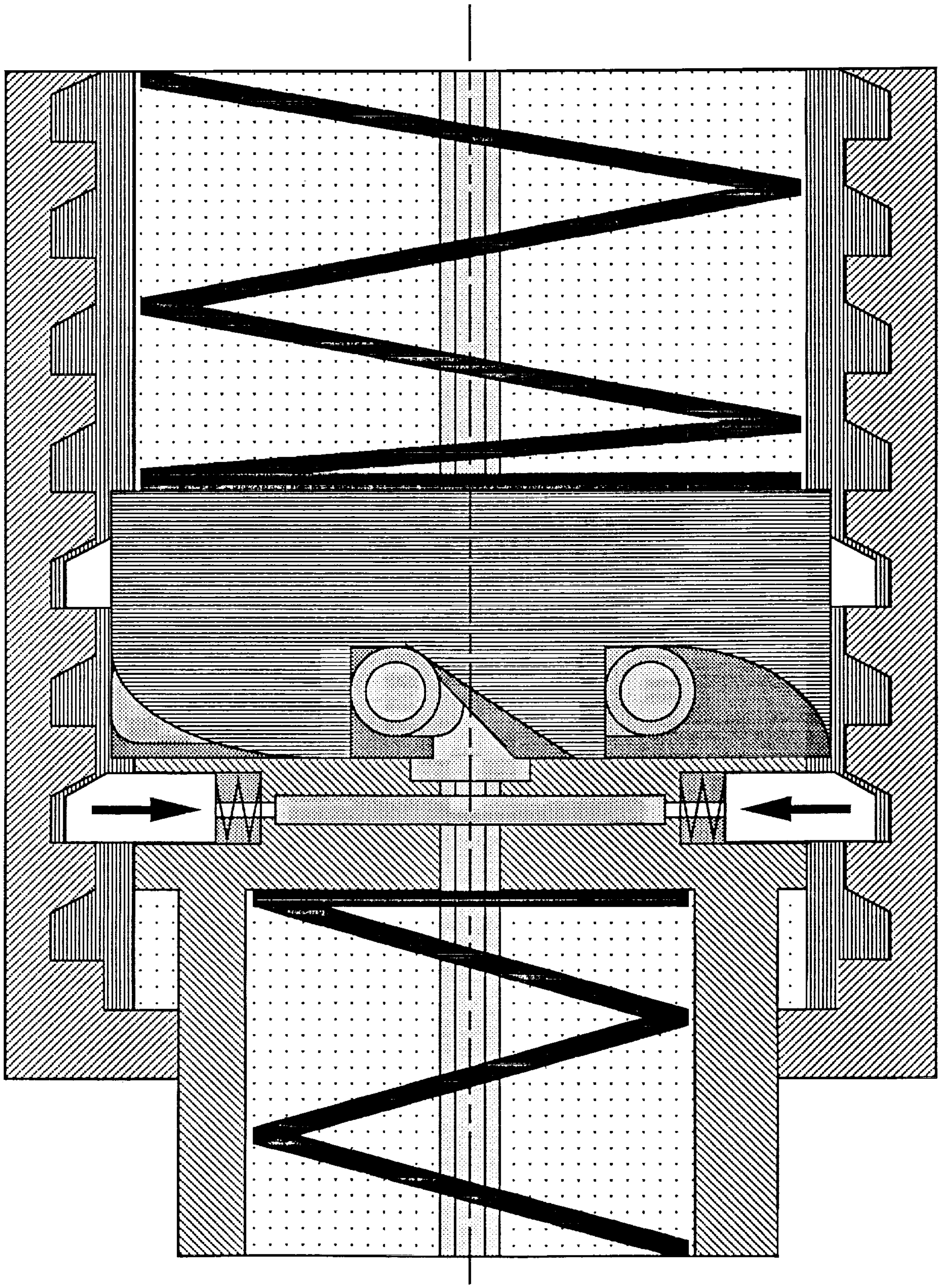
In order to provide the most predictable, reliable, and maintenance-free operation, a purely mechanical shock absorption system was devised. Hydraulic or pneudralic systems normally used in comparable industrial settings were considered too susceptible to the risk of outgassing in the high vacuum environment of space. Alternatively, hybrid electro-mechanical systems were considered too complex and susceptible to electrical power failures. In addition, electro-magnetic shock absorption systems were also found to be prohibitively massive, besides suffering from the same weaknesses as electro-mechanical systems. A mechanical shock absorption system, however, has the advantage of being simple and self-contained, with no environmental hazards to consider. Two types of mechanical shock absorption systems were analyzed: a friction brake and a spring-and-ratchet assembly. The friction brake was considered inferior since it would result in increased wear and would generate significant amounts of heat. On the other hand, the spring-and-ratchet mehanism was found to be extremely reliable and safe, with minimal wear and maximum longevity inherent in the design.

The shock absorber assembly designed for the Lunar Lander legs is shown in Figures 4.12a,b,c. It should be able to absorb high energy impacts well beyond the expected number of cycles in the service life of the Lander. The assembly utilizes a spring-type mechanical shock absorbers coupled with a ratchet-type retrieval mechanism. Two such mechanisms are used to absorb the energy of impact in each leg while the ratchet mechanisms are used to hold and sequentially release the stored energy in small, controlled increments. A combination of both light and heavy-duty springs provides two levels of shock absorption to accomodate both "feather-soft" and one-legged, angled hard landings.

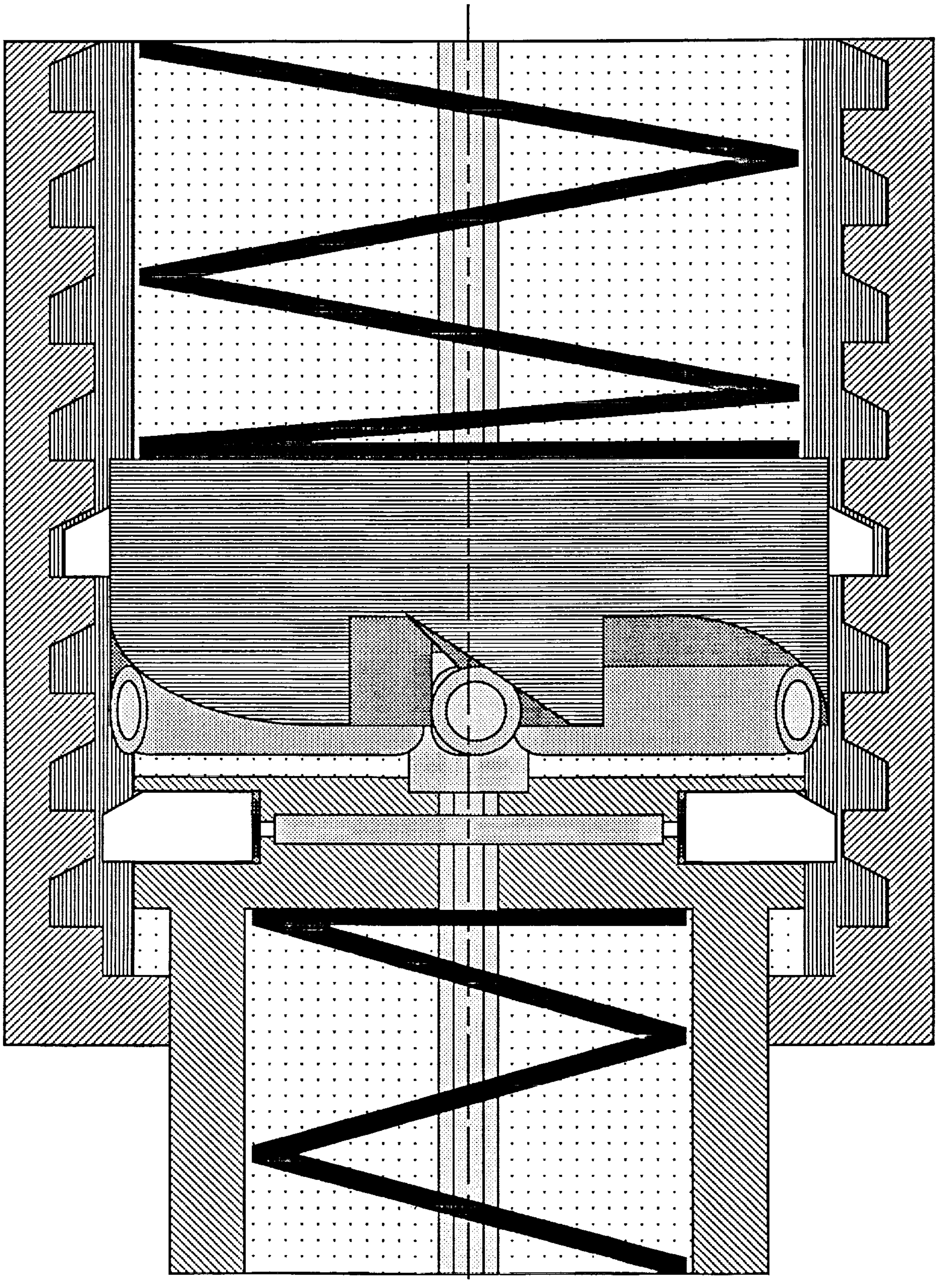
The release of the springs occurs in three steps, each activated by separate motors in each leg. Figure 4.12a shows the first step of the process. Upon landing, the spring-loaded latches slide into place once the lower portion of the leg has compressed the upper spring. At this point, the crew activates the leg retrieval motor at the top of the leg which continuously turns the central shaft, spindle, and cam linkages in the same direction. The cam action first pulls in the lower latches while the upper latches take up the load. In step 2 (Fig. 4.12b), rollers on the turning spindle allow the arms to roll up the inclined surfaces of the central piston, thus pushing the lower portion of the leg down, extending the leg. In step 3 (Fig. 4.12c), the extension of the lower portion of the leg allows the lower latches to reach the next ratchet slot, whereupon another camming action engages the lower latches. Then the upper latches are similarly disengaged and the arms of the spindle fall back into their original position. The upper latches are engaged and the entire process is repeated until all of the energy in the spring is dissipated and the leg has been fully retrieved.

Leg Dampening Assembly

Step 1

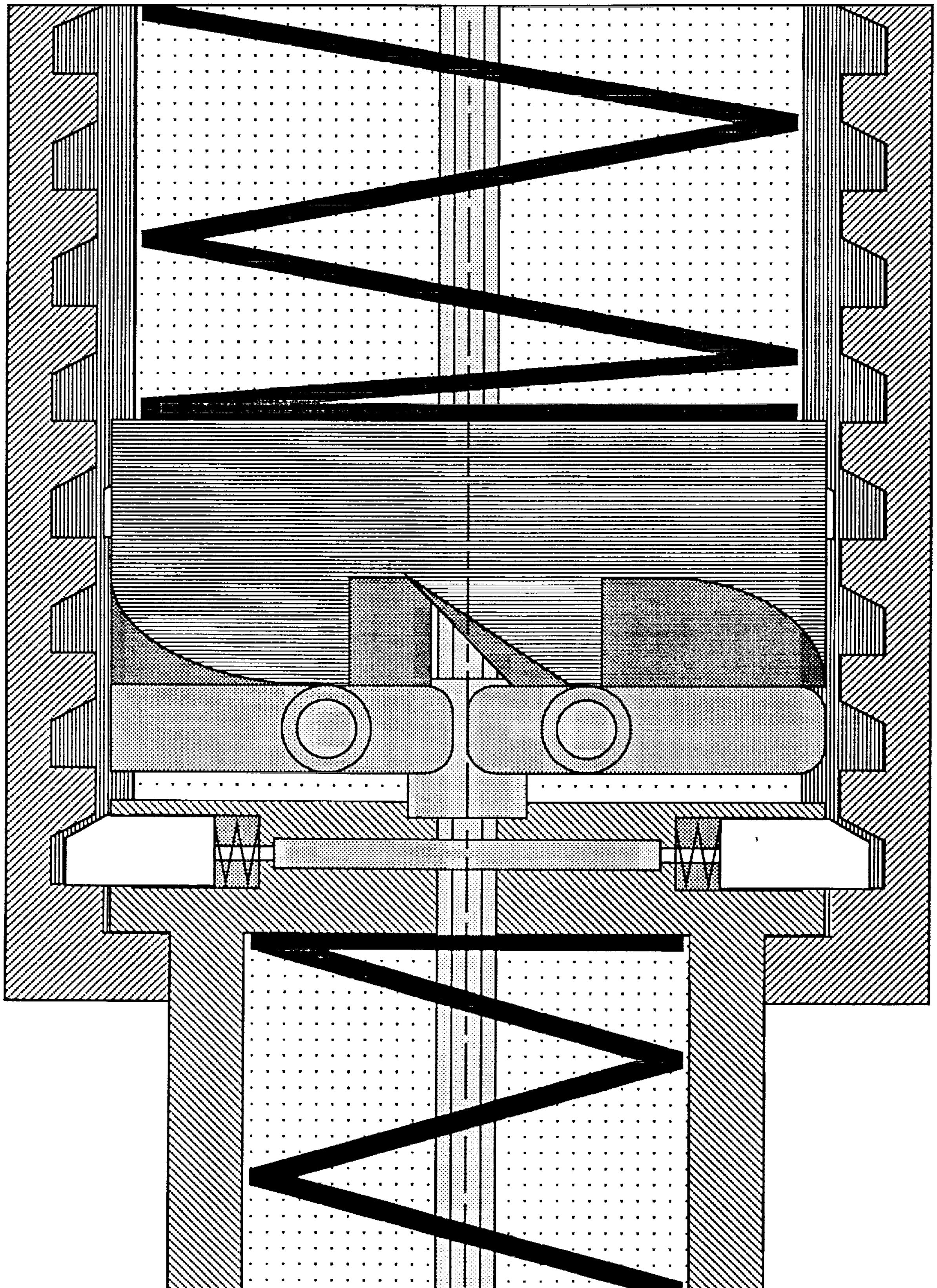


Step 2



Leg Dampening Assembly

Step 3



CHAPTER 5: ROCKET ENGINES

Each Lunar Lander, capable of thirty missions before replacement, will perform around fifteen missions involving payloads of varying mass and size as well as the transport of personnel for future lunar base operations. Since the cargo pallet of the Lunar Lander can accommodate a manned module and several payload combinations for ascent and descent, the ratio of the maximum thrust available to the minimum thrust available and the throttling ratio of the rocket, must be high in order to execute the wide range of maneuvers required to properly land or boost the Lander. In addition, the Lander should be designed for redundancy, allowing single or multiple engine failure without loss of the Lander or human life [2]. Lunar Lander reusability requires modularity, maintenance accessibility and easy engine replacement to keep Lunar Landers in safe and continuous operation, and engine cooling techniques to prevent rocket engine nozzle burn up and thermal damage [2]. Finally, high rocket engine specific impulse maximizes the payload mass deliverable to the lunar surface, minimizes the mass of the propellants required, and maximizes propellant properties desirable for a Lunar Lander with stack mass configurations up to ninety metric tons in some cases. Hence Lunar Lander mission profiles require the following rocket engine parameters: high I_{sp}, high throttling ratio, redundancy and modularity, and good engine cooling techniques.

The type of engine satisfying the engine parameters given would be a regeneratively cooled, liquid hydrogen fuel and liquid oxygen oxidizer engine in a four engine, individual turbo-pump feed configuration for redundancy. The decision matrices used for selection of the propellant type, engine cooling technique, and engine configuration (Figures 5.1 to 5.3) show why the choice was made.

Fig. 5.1: PROPELLANT SELECTION

PROPELLANT OPTIONS	ADVANTAGES	DISADVANTAGES	RANK
LOX / LH_2 (Liquid Hydrogen / Liquid Oxygen)	* ECLSS support (Oxygen and Water) * Power source * OTV compatible * High Isp (>400)	* Expensive * Cryogenic and high volume storage * Losses due to vaporization	5
Hydrazine (Mono-Propellant)	* Single Feed System * Easy and long duration storage	* Corrosive and toxic * Low Isp (300-350)	3
Hydrocarbon Fuel / Cryogenic Oxidizer Combinations	* Easy to handle * Readily available * Low cost	* Cryogenic storage * Low Isp (300-350)	2 1 = worst 5 = best

Fig. 5.2: ENGINE COOLING

COOLING TECHNIQUES	ADVANTAGES	DISADVANTAGES	RANK
Regenerative Cooling	* Best Heat Transfer * Reusable / Restartable * Increases Exit Velocity	* High Power Requirement * Large Weight Penalty (Pump-Feed System)	5
Ablative Cooling	* Low Cost * Lightweight * No Power Required	* Effectiveness diminished by multiple firings * Eventually deteriorates	2
Cooling Jackets	* Good Heat Transfer * Reusable / Restartable	* High Power Requirement * Great Weight Penalty (Pump-Feed System)	4 1 = worst 5 = best

Fig. 5.3: ENGINE CONFIGURATION

ENGINE CONFIGURATION	ADVANTAGES	DISADVANTAGES	RANK
4 - Engine, Single Pump-Feed System	* Modularity * Full redundancy with single engine failure * Thrust Vectoring	* High Throttling Ratio	5
Single Engine, Multiple Pump-Feed System	* Low Throttling Ratio * Pump Redundancy	* No Engine Failure * Long Maintenance Time	3
2 - Engine, Single Pump-Feed System	* Some Thrust Vectoring	* High Throttling Ratio * No Engine Failure allowed	2 1 = worst 5 = best

A liquid oxygen/liquid hydrogen propellant system satisfies the high specific impulse requirements required to maximize mission payload. Keeping all other items constant, higher I_{sp} allows for higher Lander burnout mass, and hence higher payload mass [3]. An I_{sp} for LOX/LH_2 propellants can approach 450 seconds while I_{sp} for organic, storable monopropellants such as hydrazine, the only suitable chemical rocket propellant alternative for the proposed Lunar Lander mission, usually cannot exceed 350 seconds [3]. In addition, propellant mass flowrate for a given thrust maneuver is inversely proportional to I_{sp}, meaning less propellant required for higher I_{sp} propellants [3]. In addition to high I_{sp}, the liquid hydrogen fuel and liquid oxygen oxidizer (LH_2/LOX) propellant combination provides a source of water through the mixing of hydrogen and oxygen, a source of pure oxygen for breathing, and a source of power for fuel cells.

The question of redundancy for the sake of safety is a choice between multiple main propulsion propellant feed systems serving one huge, main thruster turbine pump or pump systems distributing propellants to several rocket engines. The reliability of the rocket engine remains the most important performance criteria in the case of manned missions, since maximum thrust often saves the crew and mission in a worst case scenario. The likelihood of rocket engine failure, much higher than pump system failure, requires constant engine maintenance to guarantee engine start-ups and shutdowns as needed. As an alternative to fear of engine failure and to the cost and meticulous maintenance program for a single thruster, a set of four rocket engines with any three capable of meeting the maximum thrust requirement, allows for single engine failure, and symmetry for better attitude control and thrust vectoring. In addition, the modularity of a four engine configuration with individual feed systems permits scavenging operations and the exchange of rocket engine parts or whole rocket engines in order to keep most of the Lunar Landers operating and to provide a use for inactive Landers.

Regenerative cooling provides the best heat transfer properties, thus ensuring better thermal protection of the rocket engine nozzle and increasing exhaust exit velocity by dumping the heat absorbed from the nozzle to the propellants [3]. Regenerative cooling systems require the propellant tanks to feed propellant through turbine pump systems rather than through pre-pressurization of the propellant tanks since the regenerative cooling system usually uses fuel as a coolant flowing through an extensive system of tubes to cool the inner lining of the exit nozzle [3].

Despite the weight and size penalty inherent to the system, the increased payload mass, ECLSS and power system support, and the safety of the design overshadow its disadvantages.

The options available, given the constraints on the engine type, include a scaled down version of the Space Shuttle Main Engines (SSME's), a reusable version of the Pratt & Whitney RL-10, expander cycle OTV engines, and a totally new engine design. The SSME's have been designed for reusability and are integrated into a multiple engine system. Rocketdyne would be able to scale down SSME's to the size and thrust requirements needed for a Lunar Lander; this would consume a large amount of time and money for the initial prototype, but eventually Lunar Lander rocket engines could be manufactured on a production line basis. The Pratt & Whitney RL-10 has characteristics very close to the final engine configuration, as shown in Figure 5.4 [4]. Some companies have, however, developed concepts for OTV propulsion which appear to be compatible with the Lunar Lander rocket engine requirements [5]. The expander cycle achieves high I_{sp} by efficient

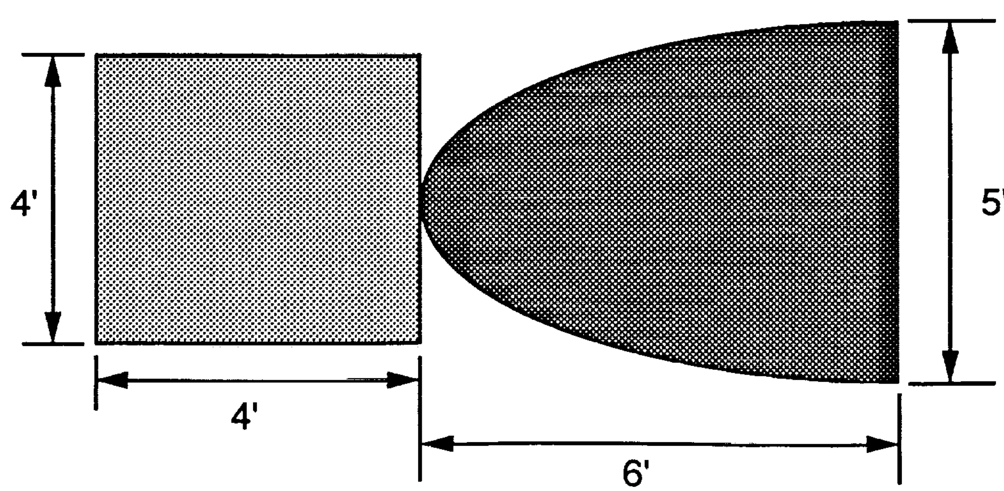
Fig. 5.4: Engine Specifications

Thrust Parameters
- Specific Impulse = 450 seconds
- Maximum Thrust = 79.0 kN (18,000 lbs)
- Throttling Ratio = 36:1

Propellant
- Turbopump Fed LOX/LH2
- Mixture Ratio = 4:1

Cooling
- Regenerative Cooling Jackets
- LH2 Is Coolant

Dry Weight
- 250 kg



expansion at the exit nozzle and full propellant burn [3]. However, these conditions require very high combustion chamber pressures and higher turbine power requirements [3]. In fact, the hydrogen turbopump speeds required in most expander cycle engines are beyond state-of-the-art [1]. Hence, a final, not very attractive option, involves a totally new engine design to satisfy all Lunar Lander rocket engine requirements without worrying about engine design conversions and compatibility.

The final rocket engine configuration includes assumptions previously made: a specific impulse of 450 seconds, a four to one mixture ratio, regenerative cooling with hydrogen fuel as the coolant, and a fuel/oxidizer feed system for each engine. The exceptionally large throttling ratio assumes a maximum thrust maneuver to deorbit a ninety ton stack mass with one engine out and a minimum thrust maneuver to land an almost empty Lunar Lander with all engines on. In general, the higher the throttling ratio, the greater the complexity of the rocket engine system. Despite this fact, a dry engine mass of 250 kg seems reasonable in comparison to weights of rocket engines used at similar thrust levels [4].

PROPELLANT TANK SIZING

Propellant tank sizing involves computing the radius and thickness of LOX and LH_2 tanks, to determine the propellant capacity of the Lunar Lander and the mass of the propellant tank arrangement.

The ALS diameter, calculations concerning the center of gravity location of the Lander, and a mixture ratio of four to one set the maximum allowable LOX and LH_2 tank radii, respectively, at 1.5 m and 2.5 m. Then, according to the program FIRE.BAS in Appendix VI, the given tank sizes limit the maximum propellant capacity of the Lunar Lander to 35 metric tons. Assuming maximum tank pressures of 700 psia, the required LOX and LH_2 tank thicknesses, are 10 mm and 16 mm. The resulting propellant tank arrangement has a dry mass of 7.5 MT which includes a 12 cm layer of multilayer insulation (MLI). The 35 MT propellant capacity and the 7.5 MT propellant tanks limit the payload capacity of the Lunar Lander.

PAYLOAD MASS CALCULATIONS

Propellant capacity and properties, Lunar Lander inert mass, and Lander burn times and trajectories determine the payload deliverable for each of the following missions:

1) Expendable Lander descent, no ascent.

2) Descent with maximum payload, ascent empty.

3) Descent with no payload, ascent with maximum payload.

4) Descent with maximum payload, ascent with crew module.

The mission profiles, propellant properties, and Lander inert mass were used to generate the payload schedule given in Table 2.2. The major assumptions employed in the propellant mass calculations appear in Appendix VII. According to tank sizing computations, the propellant capacity of the Lunar Lander, based on FIRE.BAS calculations, is 35 MT, the initial propellant mass used in all missions examined. For the sake of safety, the breakdown of propellant usage includes 85% usable for main propulsion, 5% for emergencies, 5% for boil-off to vapor, and 5% for ullage. The LOX/LH_2 combination has an mixture ratio of four to one and an specific impulse of 450 seconds, a maximum value consistent with most current LOX/LH_2 systems. These assumptions characterize the nature, division, and amount of propellant used in the payload calculations.

The mass statement of the Lunar Lander used in the mission payload calculations, as shown in Appendix VIII, overestimates some essential Lander inert components by as much as 25 percent for the sake of safety. This explains the unusually high inert Lander mass of 13.5 metric tons for all missions.

A general Lunar Lander mission includes an in initial deorbit burn, a descent phase, payload changes, an ascent phase, and an orbit insertion burn. The burn required for deorbit consumes a very small fraction of propellant, which can be incorporated into the 5% of propellant associated with emergencies. The descent stage consists of a constant thrust braking maneuver to execute a minimum fuel maneuver for the main descent; a gradual thrust reduction to hover thrust to prepare for vertical descent; and an almost constant thrust hover to land to achieve a touchdown landing velocity of less than 1.6 m/sec [2]. The ascent stage consists of a constant thrust burn and a heading change from a launch angle of 90 degrees to an orbit angle of 0 degrees for insertion into LLO [2]. The propellant required for the orbit insertion is sufficiently small enough to include in the 5% of propellant devoted to emergencies.

For the constant thrust maneuvers, the propellant mass required equals thrust times time divided by the product of the specific impulse and acceleration due to gravity, referenced to earth at sea-level. The thrust for each maneuver equals the product of Lunar Lander stack earth weight and a thrust to weight ratio corresponding to the different accelerations or velocities associated with the maneuver.

To ensure mission success during the entire Lunar Lander mission profile, the thrust to weight ratios are all based on an Apollo 11 mission profile [2]. The minimum fuel maneuver thrust to weight ratio of 0.275 limits the Lunar Lander maximum deceleration to 9 ft/sec^2; likewise, the ascent thrust to weight ratio of 0.321 limits the Lunar Lander maximum acceleration to 6 ft/sec^2 [2]. In addition, the hover to land thrust to weight ratio of 0.0875 corresponds to a touchdown velocity of 1.6 m/sec or less [2].

CHAPTER 6: ATTITUDE CONTROL

The attitude control system of the Lunar Lander will be responsible for performing two primary functions. First, it must be able to perform all docking maneuvers required for the Lander to dock with the space station in LLO. Second, the attitude control system will be required to provide attitude control while in flight during launch and landing. While the Lander is docked with the Lunar Station, the station will provide all orbit-keeping operations.

The attitude control system of this Lander will use the space tested systems of the Apollo LEM and the Space Shuttle. Both of these vehicles are designed to perform certain functions that our Lander will also have to perform. For example, the Space Shuttle is capable of docking maneuvers, and the Apollo Lander obviously was capable of landing on the moon. These capabilities compare well with our Lander's requirements.

PROPELLANTS

Both of these vehicles have used a hypergolic propellant system. This type of bipropellant system eliminates the need for an igniter, and is therefore considered highly reliable. Cold gas systems such those used in many small satellites offer extreme simplicity, high reliability, and low weight. However, such systems, even when using a heavier gas such as argon, have a relatively low performance compared with that of hot gas systems. NASA experts agree that a hot gas or hydrazine-type attitude control system is necessary for a vehicle of the size of our Lander [1].

Since performance, high reliability, and simplicity must guide and have guided our design decisions, a hypergolic, bipropellant attitude control system was chosen for our Lander. The Lander will use monomethylhydrazine (MMH) as a fuel and nitrogen tetroxide (N_2O_4) as an oxidizer. The propellant mixture ratio is 2 to 1, fuel to oxidizer, by weight. Also, the propellant has a specific impulse of 280 seconds, steady state.

THRUSTERS

The attitude control system is capable of complete control about all three axes and translation along each axis. Accomplishing this are a total of sixteen thrusters grouped in four clusters of four thrusters each. All thruster clusters are spaced equally apart (90 degrees) about the circumference of the Lander, and each cluster is specifically located 45 degrees from each leg. Figure 6.1 illustrates this placement with a top view of the Lander. Also, all clusters are located vertically at the approximate height of the c.g. of the loaded Lander, as shown by a side view of the Lander in Figure 6.2. Each thruster is capable of producing 125 pounds of thrust and has a mass flow rate of 0.203 kilograms per second.

PROPELLANT FEED/PRESSURIZATION

The propellant feed/pressurization system for the attitude control system of our Lander is almost identical to that used by the Apollo Lander and is shown in Figure 6.3. It consists of two cylindrical propellant tanks -- one for the fuel, one for the oxidizer. A positive expansion bladder (Figure 6.4) in each propellant tank will be used to expel the propellants. Helium, stored in a spherical tank at 3000 psi and used at 179 psi, will be used as the expulsion gas.

Fig. 6.1: Attitude Control System

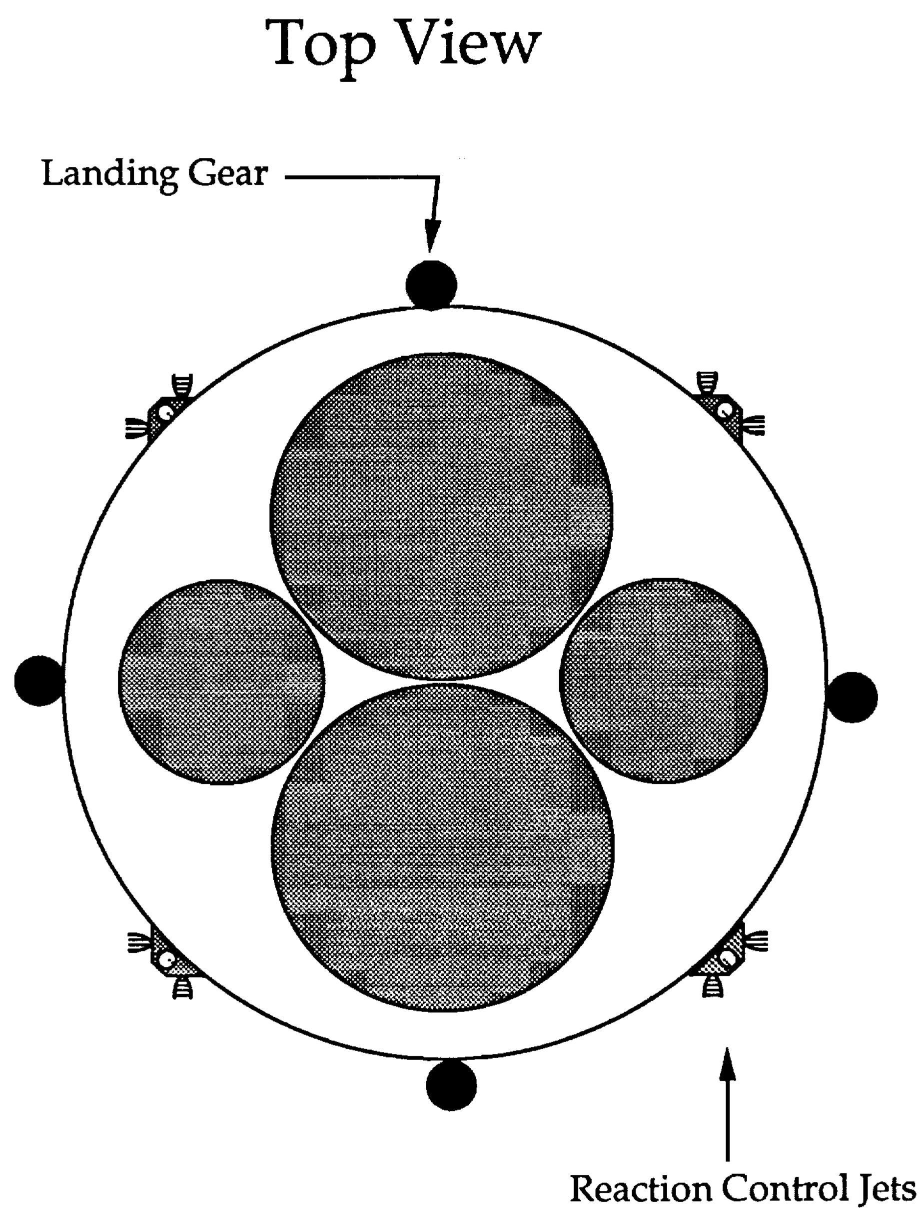


Fig. 6.2: Attitude Control System

Side View

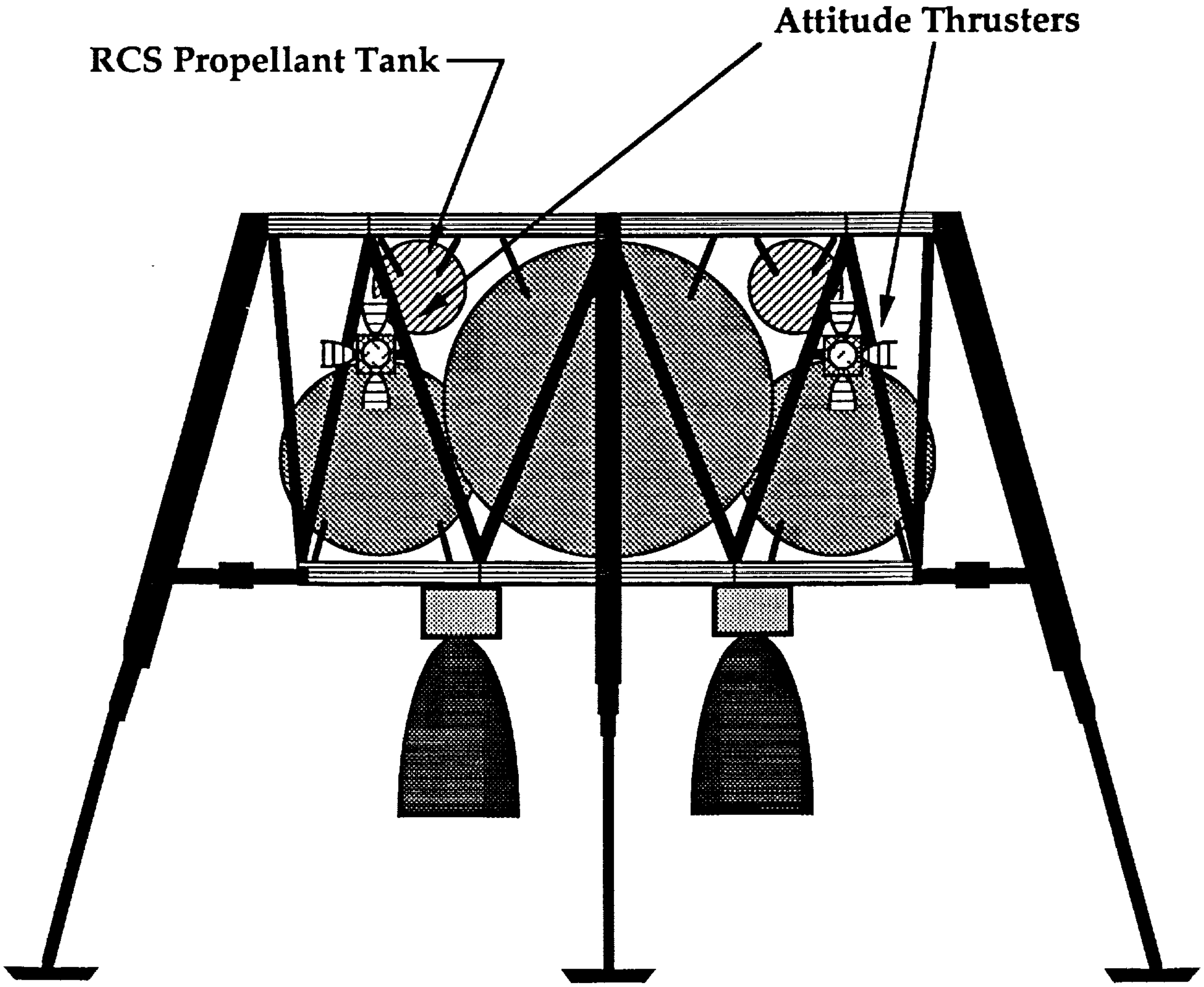


Fig. 6.3: Propellant Pressurization System

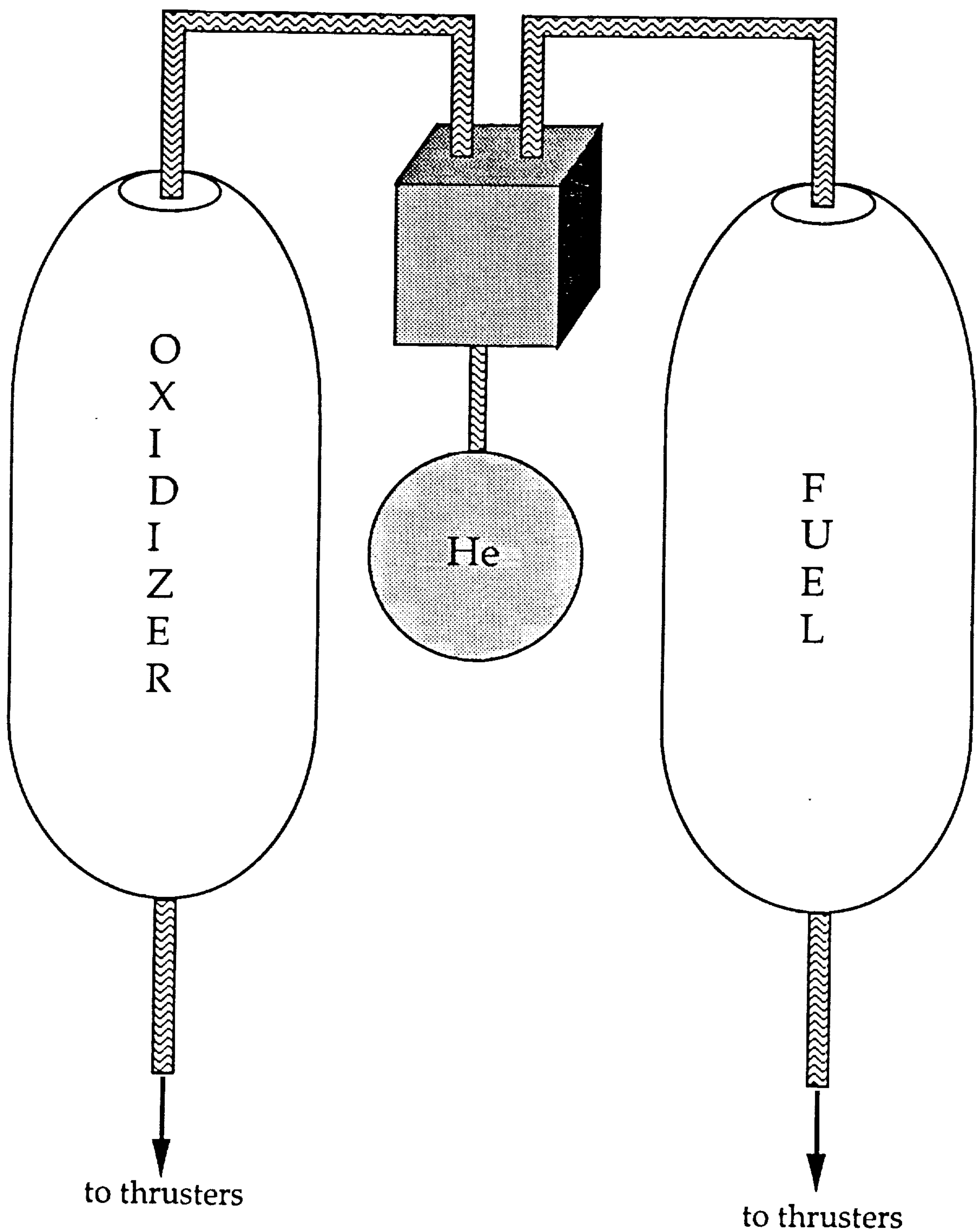
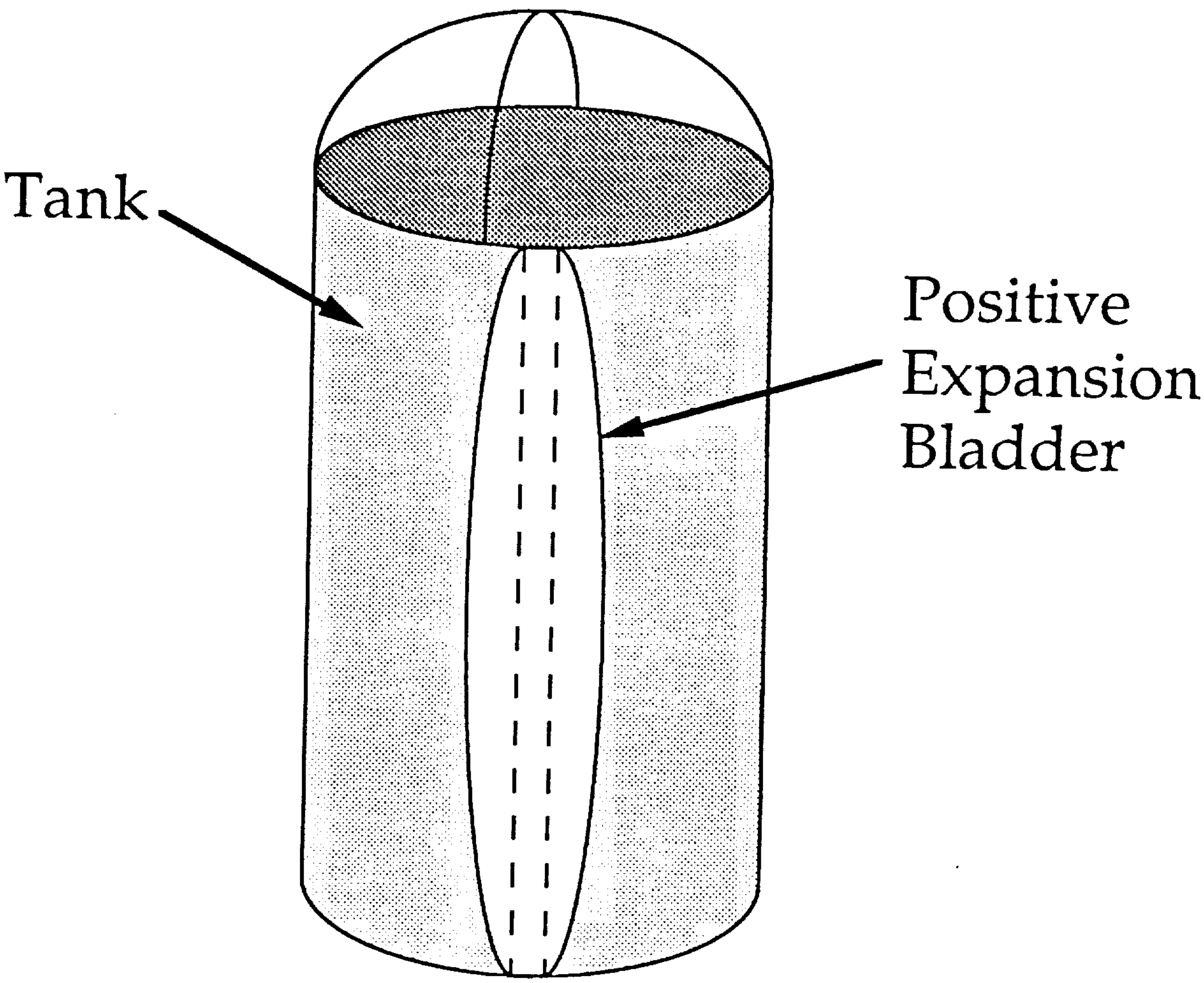


Fig. 6.4: Propellant Expulsion



WEIGHT ESTIMATION

An estimate of the total burn time for each thruster allowed calculation of total propellant mass to be 16 kilograms including a safety factor of 1.5 and allowing for 5 percent ullage. This calculation was based on an estimated average total burn time for each thruster of: 1 second for docking in short bursts of 0.01 to 0.03 seconds and 2 seconds for landing.

Since each thruster has a mass of only 4.5 kilograms, all sixteen thrusters have a total mass of only 73 kilograms. Therefore, since both thruster mass and propellant mass is small, and the mass of tanks, piping, etc. is estimated to be small relative to the Lander mass, it is reasonable to say that the mass of the total attitude control system will be small compared with that of the Lander -- under 200 kilograms.

CHAPTER 7: CRYOGENIC FUEL STORAGE

On Lander

Liquid oxygen and hydrogen will be stored on the Lunar Lander in four spherical tanks, containing a total of 28 metric tons of oxygen and 7 metric tons of hydrogen. The oxygen tanks will each be 3.0 meters in diameter and will be maintained at a temperature of 88 Kelvin. The hydrogen tanks will be 4.9 meters in diameter and will be kept at 22 Kelvin. A typical cross section is shown in Figure 7.1. The tank walls will be made of Weldolite 049, an aluminum-lithium compound; they walls are designed to withstand 700 psia. The walls will be covered with 12 centimeters of Multi-Layer Insulation (MLI), made of aluminized Mylar. A vapor-cooled shield (VCS) will be located approximately halfway through the MLI; it will be cooled by boiloff from the hydrogen tank. Boiloff vapor will be routed through small tubes on the inner wall of the aluminum VCS shell, and then vented to the ambient pressure. Hydrogen vapor will cool both sets of tanks; the system eliminates oxygen boiloff. Hydrogen losses will amount to tens of kilograms per month.

At LLO Station

The cryogenic fuel storage depot at the LLO Station is modeled after a proposed LEO storage depot for the OTV [1]. Two cylindrical hydrogen tanks and one of oxygen will contain enough propellant to fully fuel the Lander twice. The system will maintain a tank pressure of 14 psia, and will require 2.7 kilowatts of electrical power; tank dimensions are given on Figure 7.2. Hydrogen boil-off will once again be used to cool the vapor-cooled shields located in the midst of the MLI, but will then be re-liquified and pumped back into the hydrogen tanks. The MLI will be 10 centimeters thick. This system totally eliminates

Fig. 7.1: Cryogenic Propellant Storage

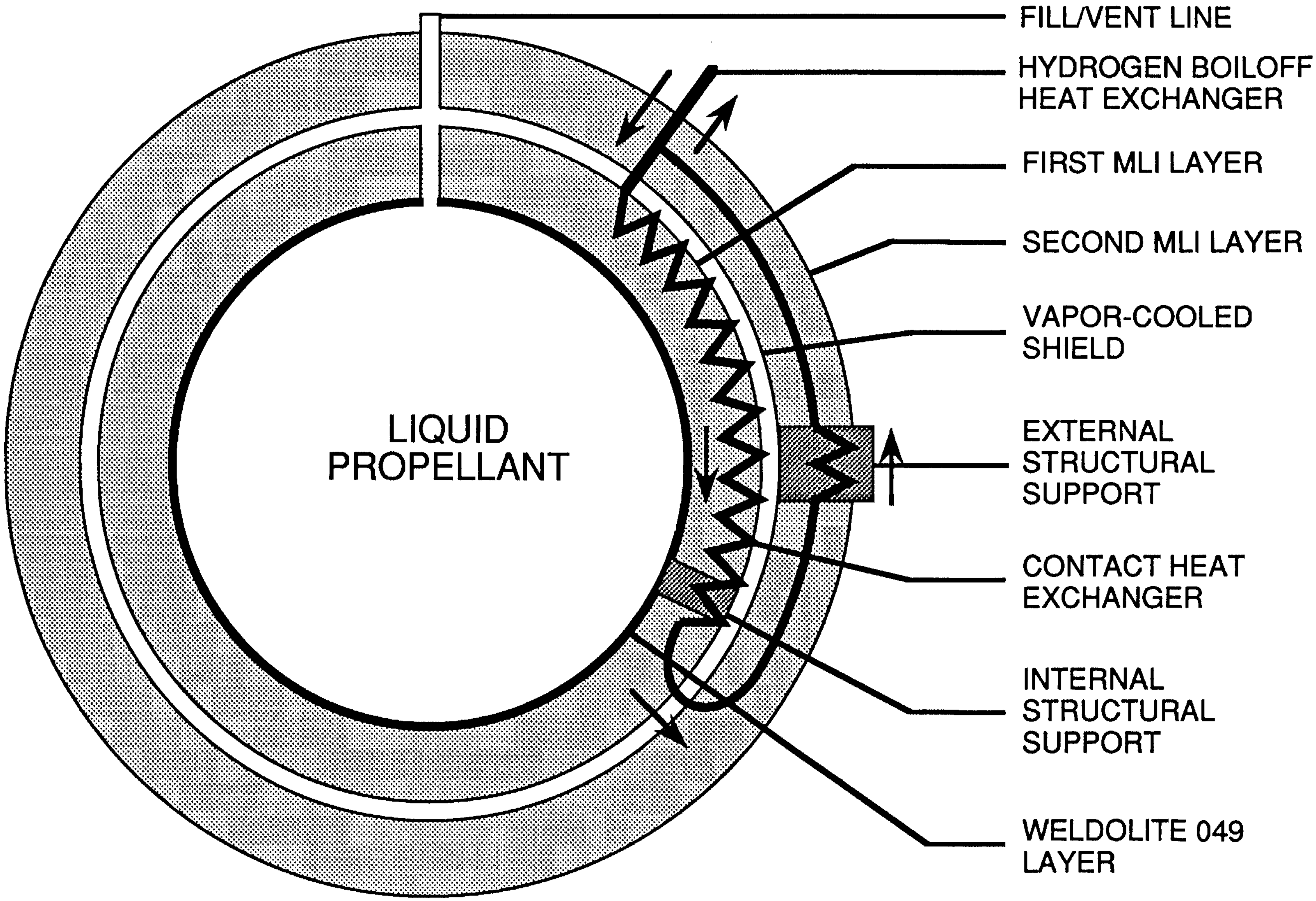


Fig. 7.2: Lunar Lander

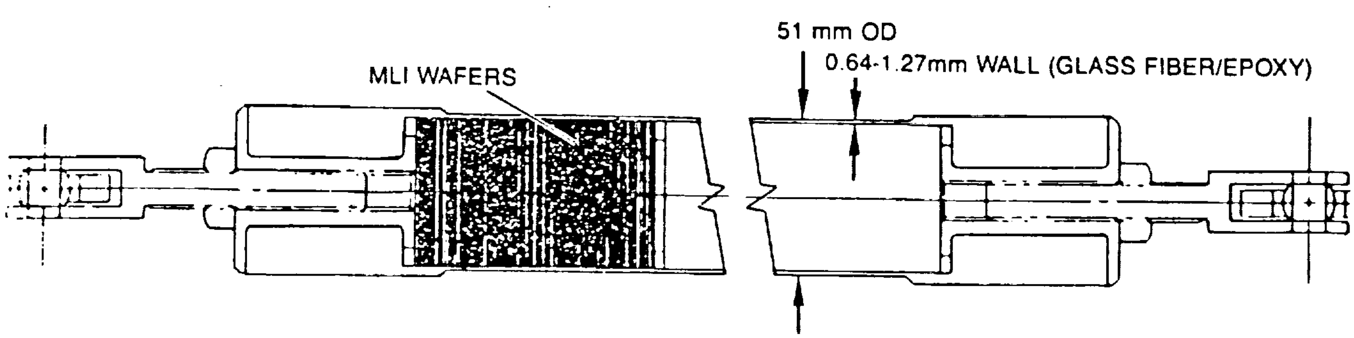
Cryogenic Fuel Storage

On Lander:

28 metric tons of Oxygen, in two spherical tanks (D=3.0m).
7 metric tons of Hydrogen, in two spherical tanks (D=4.9m).
Tanks are at 700 psia.
Tanks are insulated with 10 centimeters of Multi-Layer Insulation.
Vapor-Cooled Shield is located within MLI; it is cooled by Hydrogen boil-off.
Hydrogen vapor cools shield and is then vented to space.
Zero Oxygen loss.
Hydrogen loss rate on order of tens of kilograms per month.

On Station:

56 metric tons of Oxygen, in cylindrical tank (D=4.3m, L=4.15m).
14 metric tons of Hydrogen, in two cylindrical tanks (D=4.3m, L=8.2m).
Tanks are at 14 psia.
Tanks are insulated with 10 centimeters of MLI.
Hydrogen-cooled VCS is located within MLI.
Hydrogen boil-off cools shields on all three tanks and is then recompressed and put back in tank.
No fuel loss.
System requires 2.7 Kilowatts of electrical power to run compressors.



fuel loss, but was thought too complex and heavy for use on board the Lander.

Structural supports for tanks in both locations are designed with integral MLI wafers to reduce heat conduction to the supercooled tanks [1]. The strut ends attached to the fuel tanks will be conductively cooled by tubes from the VCS system. An illustration of the insulated strut is provided in Figure 7.2. Only four of these insulated struts will be in contact with each Lander fuel tank, providing structural support but minimal heat transfer. The Station storage tanks will also be supported by the minimum number of struts necessary for sufficient strength; the actual configuration has been left to the Station design team.

The fuel lines will also be conductively cooled at the tank attachment point by proper routing of the VCS tubes. The remainder of the fuel line, from the Lander tanks to the engines or from the Station tanks to the refueling attachments at the docking node, will be insulated with several centimeters of MLI, but will not be kept at cryogenic temperatures in between periods of use. Prior to refueling the Lander from the Station tanks, fuel will be vented into the lines and will cool them as it vaporizes. Once the fuel lines are brought to the proper temperature, full fuel flow will begin; when the Lander tanks have been filled, the valves at both the Lander and the Station tanks will be closed. The fuel remaining in the lines will be allowed to vaporize as the fuel lines slowly heat up.

Refueling on the lunar surface, once a supply of cryogenic fuels has been established, will require a similar procedure. EVA crews will attach the fuel lines to the Lander tanks, cool the lines, and then open the Lander tank valves to refill the tanks. Much of the cooling of the lines can be avoided by transferring the cryogenic fuels during the lunar night when exposed fuel lines will already be chilled.

The tanks will also have interior emergency heaters; they will be used to maintain tank pressure and thus fuel flow when little fuel remains. In the maximum cargo delivery case the Lander will reach the surface with only the 15 percent of total fuel allocated for boiloff, ullage, and a safety factor remaining. The emergency heaters will increase the amount of fuel deliverable to the fuel cells, thus increasing the available electrical power, both in quantity and in duration.

CHAPTER 8: ENVIRONMENTAL CONTROL AND LIFE SUPPORT SYSTEM

The Environmental Control and Life Support System (ECLSS) is normally defined by the categories of atmospheric revitalization, life support and thermal control. Each category contains subsystem elements ranging from humidity control to waste management. Specifically, the Lunar Lander ECLSS design entails unique trade-off considerations affected by mission requirements. The conceptual ECLSS is based upon supporting a crew of from one to six from the lunar surface to low lunar orbit and return. Individual considerations and resulting design trades as to open vs. closed loop systems, pressurized vs non-pressurized, expendable capacities, etc. are based on information from the NASA Shuttle Transportation System (STS), Eagle Engineering and Space Station (SS) data (see Figure 8.1) [3].

Assumptions made during system analysis are as follows:

- Personal hygiene accommodations will be similar to the extravehicular activity (EVA) suit design.
- All critical subsystems will be redundant.
- Crew stay time will be 1 day maximum.
- Mission timelines can be extended up to 3 days by adding expendables in the payload.
- EVA suit design is an open loop system.
- No airlock but docking/subsystem interface connections available.

Final decisions concerning closed vs. open loop systems are based upon Lunar Lander mission durations. Assent and decent profiles show the Lander functioning over a period

Fig. 8.1: ECLSS DESIGN TRADE TABLE

AREA	ELEMENT	ADVANTAGES	DISADVANTAGES	TECHNOLOGY	RANK
<u>ECLSS</u>	Closed loop	* Self-sustaining * Reusability of Subsystems	* Increased Cost * Technology TBD * Increased Power and Hardware Mass	Available	1
	Open loop	* Maintenance * Cost * Mass * Safety	* Mission Constraint * Increased Resupply	Available	5
<u>ATMOSPHERIC REVITALIZATION</u>	Pressurized	* Shirt-sleeved Environment * Mission duration	* Safety / Shielding * Suit Operations * Increased Hardware	Available	2
	Non-Pressurized	* Access / Egress * Shielding * Safety * Cost	* Mobility * Increased Storage and Recharge	To Be Developed	5
<u>LIFE SUPPORT</u>	Food Management	* Rehydratable * Extended EVA Support	* Preparation and Storage	Available	5
	Waste Management -Suit vs. On-board	* EVA Operations * Storage	* Mission Constraint * Hardware Requirements	Available	5
<u>THERMAL CONTROL</u>	Subsystem Heat Transport	* Avionics Cold-plating * Radiators * Cost * Improved Heat Rejection Ratio	* Added Water	Available	5

of minutes or a few hours. Under extreme conditions the Lander has a capability to operate for several days given sufficient stored expendables. Overwhelming evidence (e.g. cost, additional mass, etc.) shows that a closed loop system is only required for long range missions (e.g. several days) and demands extensive investment. Therefore the optimum system selected is the open loop configuration [3:68-73].

Pressurization requires much of the same elements (e.g. investment, maintenance, etc.) as a closed loop system. An non-pressurized environment allows simplicity for crew operations and enhanced radiation shielding with personnel inside EVA suits. Each suit will be an advanced version of current suit development where current EVA suit development lies between the AX-5 and the ZPS-Mark 3. For example, the ZPS-Mark 3 space suit (Figure 8.2) [4], operating at 8.3 psi, is a combination of both hard and soft elements. The ZPS-Mark 3 utilizes soft suit jointing claiming it provides optimum comfort and improved motion range during pressurized operations. Hard suit elements of aluminum are employed in areas requiring higher pressure loads (e.g. upper torso) and all bearings are made of stainless steel [1,4:3,37]. The suit design allows for a 13-inch diameter helmet [6:9] and entry/egress from the suit is by a rear hatch. Soft fabric joint elements are used in the elbow, arm, knee and ankle from the current Shuttle suit. It is felt that the usage of fabric will allow greater flexation vs. metallic components. Attached to the AX-5 hatch is the Primary Life Support Subsystem (PLSS) whose details are exhibited in Figure 8.3 [5:27]. Newer versions must allow longer mission duration, greater radiation shielding, material flexibility and zero prebreathe (Figures 8.3) [2:8,16].

Should food requirements be necessary, rehydratables are stored and prepared by injecting hot or cold water and mixing. Waste management will be the primary responsibility of the EVA suit. In the event of pressurization, fire detection and control

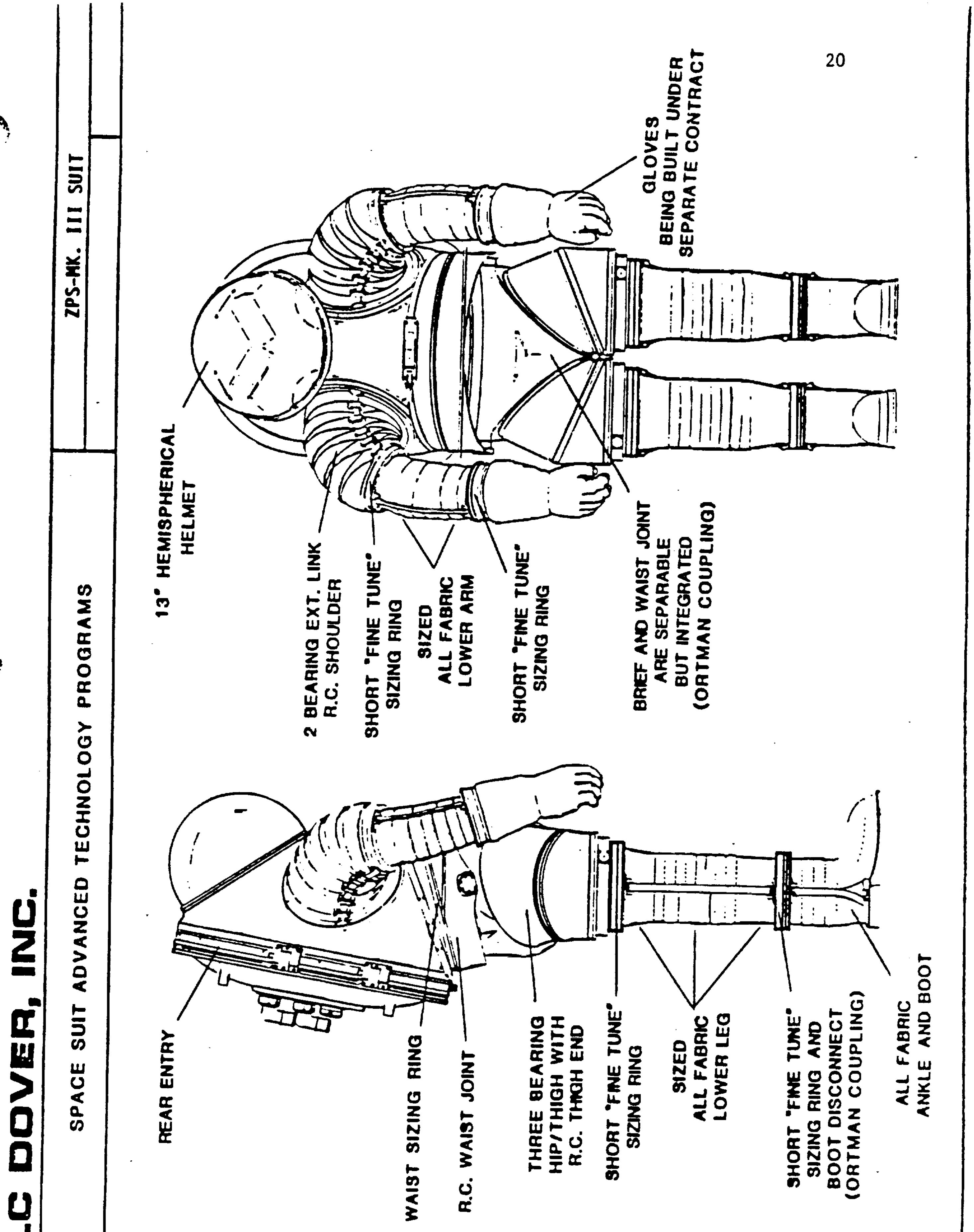

ILC DOVER, INC.
SPACE SUIT ADVANCED TECHNOLOGY PROGRAMS
ZPS-MK. III SUIT
20
REAR ENTRY
WAIST SIZING RING
R.C. WAIST JOINT
THREE BEARING HIP/THIGH WITH R.C. THIGH END
SHORT "FINE TUNE" SIZING RING
SIZED ALL FABRIC LOWER LEG
SHORT "FINE TUNE" SIZING RING AND BOOT DISCONNECT (ORTMAN COUPLING)
ALL FABRIC ANKLE AND BOOT
13" HEMISPHERICAL HELMET
2 BEARING EXT. LINK R.C. SHOULDER
SHORT "FINE TUNE" SIZING RING
SIZED ALL FABRIC LOWER ARM
SHORT "FINE TUNE" SIZING RING
BRIEF AND WAIST JOINT ARE SEPARABLE BUT INTEGRATED (ORTMAN COUPLING)
GLOVES BEING BUILT UNDER SEPARATE CONTRACT

Figure 8.2

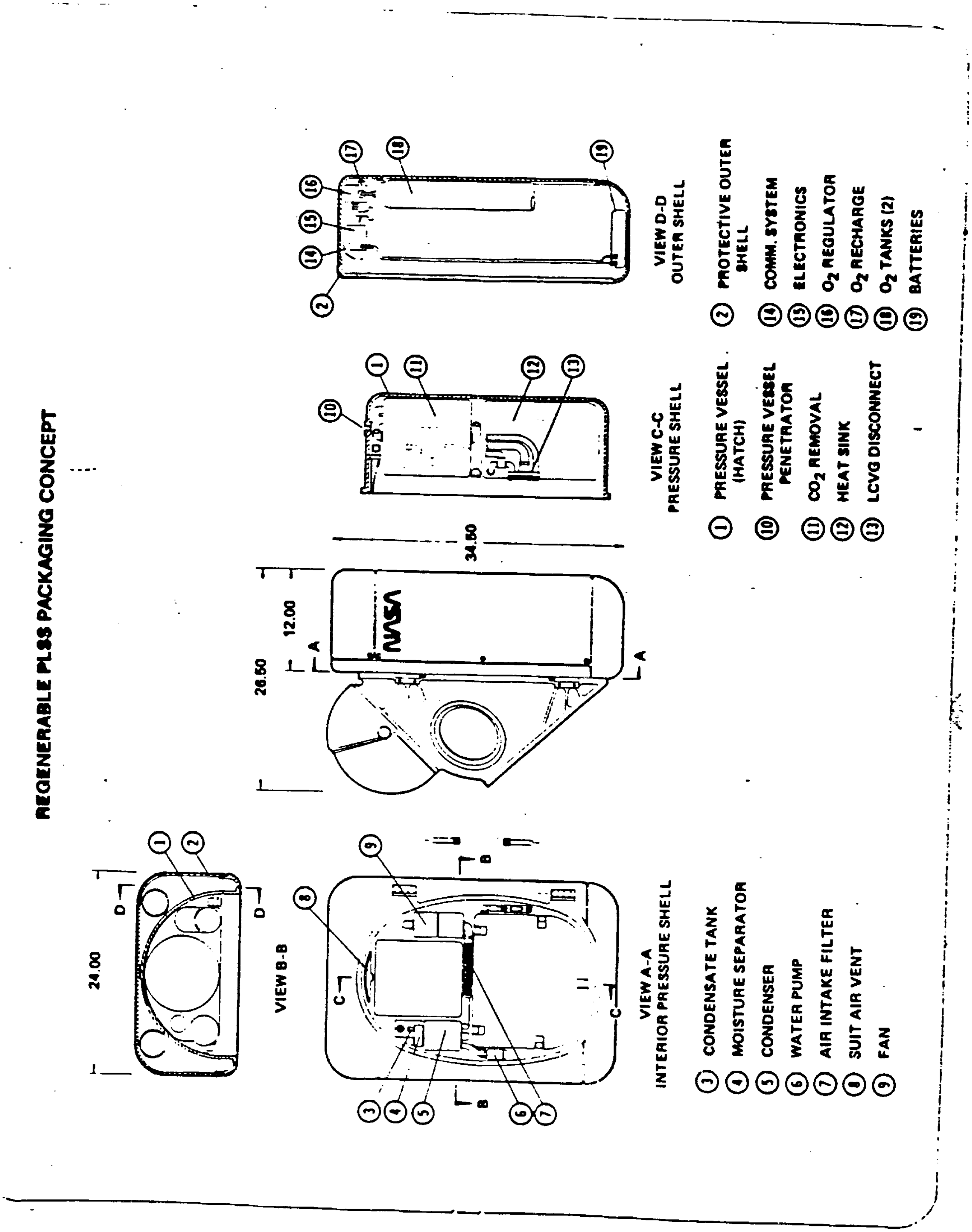

REGENERABLE PLSS PACKAGING CONCEPT
24.00
VIEW B-B
VIEW A-A
INTERIOR PRESSURE SHELL
26.50
12.00
34.50
VIEW C-C
PRESSURE SHELL
VIEW D-D
OUTER SHELL
1 PRESSURE VESSEL (HATCH)
10 PRESSURE VESSEL PENETRATOR
11 CO_2 REMOVAL
12 HEAT SINK
13 LCVG DISCONNECT
2 PROTECTIVE OUTER SHELL
14 COMM. SYSTEM
15 ELECTRONICS
16 O_2 REGULATOR
17 O_2 RECHARGE
18 O_2 TANKS (2)
19 BATTERIES
3 CONDENSATE TANK
4 MOISTURE SEPARATOR
5 CONDENSER
6 WATER PUMP
7 AIR INTAKE FILTER
8 SUIT AIR VENT
9 FAN

Figure 8.3

LUNAR EMU EVOLUTION

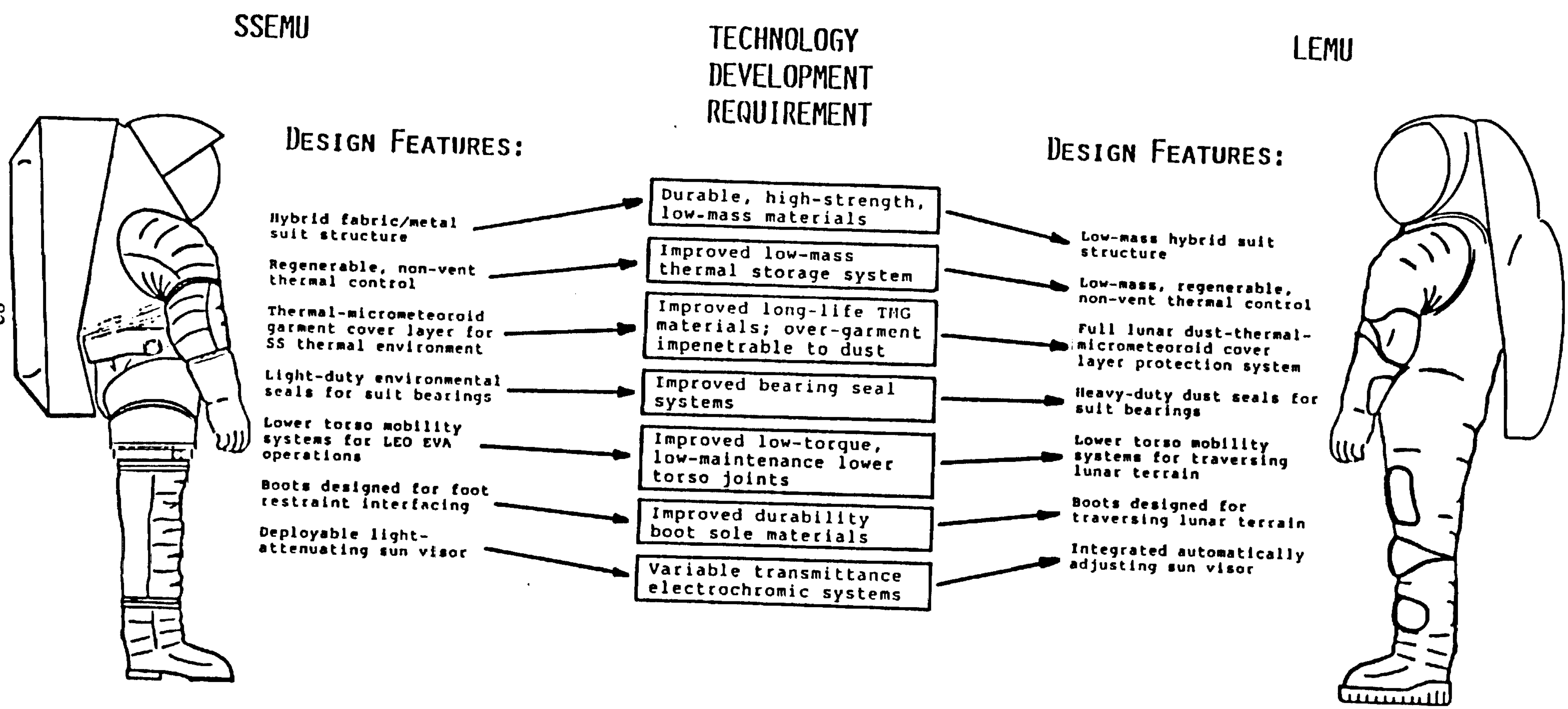


Figure 8.4

REQUIREMENT	ZERO GRAVITY	LUNAR GRAVITY
Foot Restraints	Required for vast majority of tasks.	Not required.
Safety Tether	Mandatory, except for untethered MMU flights.	Not required unless task is high above the surface.
Equipment Tether	Required.	Not required. May be desirable in some cases.
Handholds for Translation	Required.	Required only for ascending or descending.
Handholds for Force Reaction	Required.	Required in some cases.
Rescue Capability	Required.	Required.
Buddy System	Required unless safety of flight.	Required unless safety of base.
Typical Weight of ORU's	Up to 706 lbs.	Up to 240 lbs (60 lunar lbs)
Typical Weight Handled in the Past	Up to 1700 lbs.	Up to 240 lbs (60 lunar lbs)

Comparison Between EVA Requirements in Zero Gravity and Lunar Gravity

Figure 8.5

will be located in crew compartments and avionics bays. The detection system will consist of a light source, gas filter, interferometer and a detection and localization logic. The crew compartments will utilize hand-operated, portable, foam fire extinguishers.

Thermal control of avionics compartments is accomplished by pinfin coldplates. Space radiators (e.g. shunt radiators) are the primary heat sink and can reject the maximum heat load without attitude constraints during all space operations. The radiator design is further enhanced by proven NASA technology coating.

Finally, the water management system will store, distribute and dispose of potable water. Potable water will be stored in separate tanks. For emergencies, the water will be dumped overboard through heated nozzles. Water supply will be integral for EVA suits and thermal control for radiators.

Additionally, the following are recommendations for further lunar EVA development [2:23]:

* Investigate use of new, lower-weight materials
* Develop a solar flare safe-haven
* Study lunar dust impacts on human physiology and hardware systems
* Develop a dust protection over-garment
* Develop dust removal system for tools and suits
* Develop a method of removing dust from optics and sensors
* Develop improved dust protection systems for bearings and sealing surfaces
* Develop improved lower torso mobility systems
* Investigate improved abrasion-resistant materials
* Investigate materials with improved thermal cycling durability

* Develop automatic, quick-response solar visor
* Develop remote communications relay system
* Develop a lunar simulation facility to test all of these developments, including reduced gravity, thermal, and dust environments.

CHAPTER 9: INTERPLANETARY RADIATION AND SHIELDING

Of the many hazards astronauts encounter when they travel in space, one of the most intriguing is that posed by radiation. Normally, human beings on earth are protected from this unseen but deadly poison by the earth's magnetosphere. Beyond this natural protection, however, people must artificially shield themselves from each of several different radiation sources. Included in the list of radiation hazards are the sources that the astronaut will encounter while travelling to and from the moon. The three natural radiation sources of importance during this period are the Van Allen belts, galactic cosmic rays, and energetic solar particles.

The Van Allen radiation belts are the first radiation hazards astronauts will encounter. The inner belt is mainly comprised of protons, while the outer belt consists mostly of electrons. The danger associated with the inner belt is due primarily to doses received from the primary particles. However, secondary radiation is the principal danger in the outer belt; it is produced when low energy electrons are absorbed by shielding materials with high atomic numbers. This secondary radiation consists of X-rays which have a far greater penetrating power than the electrons which produced them. The astronauts will only pass through them for short periods. Readings from the Apollo flights indicate that an astronaut assigned to the Lunar Lander would receive an average dosage of less than 1.14 rems from the Van Allen radiation belts [1:688]. (One rem is defined as the dosage of any ionizing radiation that will cause the same amount of biological injury as a roentgen of X-ray or gamma-ray dosage. A roentgen is the international unit of X radiation or gamma radiation that is the amount of radiation producing ionization in one cubic centimeter of air under ideal conditions of zero degrees Celsius and 760 millimeters mercury pressure.)

Galactic radiation, which arrives at our solar system from all directions, consists of low intensity, extremely high-energy particles. These particles are approximately 85 percent protons, 13 percent alpha particles, and two percent heavier ions [1:690]. Within the solar system the galactic radiation flux level is fairly constant. The only notable fluctuations occur during enhanced solar activity when the galactic radiation flux in interplanetary space decreases. This decrease is due to an increase in the strength of the interplanetary magnetic field. This magnetic field is the solar system's galactic radiation shield.

Since the moon has a negligible magnetic field, the galactic radiation flux level can be assumed to be the same on or near the moon as in interplanetary space, except from shielding due to the moon's blockage of radiation arriving in some directions. The strength of galactic radiation in interplanetary space is approximately 0.165 to 0.265 rems/day [1:696]. For a sixty-day mission, astronauts must be protected from 9.9 to 15.9 rems due to galactic radiation.

The final and most dangerous source of radiation during lunar missions is due to solar particle events (SPE's). There are two types of SPE's which may occur during these missions. Each produces many energetic solar particles which are mainly protons with some smaller numbers of heavy ions (usually less than one to two percent).

The first is the solar flare, which occurs in the solar active region around a sunspot group in the sun's photosphere. A flare is a burst of solar energetic particles travelling outward into the solar system at a speed close to the speed of light. The flux dies off as the inverse of the square of the distance from the sun. A flare is characterized by a high level of radiation flux that may vary over many orders of magnitude and its unpredictable nature; rather, nearly unpredictable nature as we shall soon see.

The second SPE is the erupting filament. A filament is a neutral line dividing regions of oppositely directed large scale magnetic fields in the photosphere. Erupting filaments usually, but not always, produce less energetic radiation.

Any material can be used as a radiation shield. Aluminum is suited to be the radiation shielding material for the Lander. Future material development may produce a composite material may be developed which would offer a better combination of light weight, strength, and protection, than aluminum, but until then aluminum is the best candidate material. Therefore, aluminum will be used for the structure and skin of the Lander. The structure will offer the astronauts some protection. There are two main options for fully protecting the crew members inside: shield the Lander so that the crew members are protected from the maximum amount of radiation flux predictable, or shield the Lander so that the crew members are protected from low-level radiation flux (Van Allen belt and galactic radiation) at all times and provide alternate shelter for high level radiation flux (solar particle events).

The first option would entail a uniform shielding thickness of approximately four to five centimeters. The weight of the shielding that would need to be added to the Lander would be considerable and would detract significantly from the payload capability of the Lander. A storm shelter might still have to be added to protect them against unusually violent solar storms, thereby further diminishing the payload capability. Other complications arise in relation to ECLSS if a prolonged stay inside the storm shelter is required. This option is not deemed to be viable.

The second option, seen as the more feasible, involves a shelter on the moon or in orbit, or both, to which astronauts could retreat during a SPE. In this option no shielding would have to be added to the Lander. This conclusion was arrived at through the following observations: (1) Although solar energetic particles exist at all times, the flux is negligible except during SPE's, (2) Other radiation sources are low enough to be handled in a more efficient way than by vehicle shielding.

The astronauts will be protected from the 10-16 rems due to the Van Allen radiation belts, galactic radiation, and ordinary solar radiation by the LTV structure and their spacesuits. Spacesuits must be worn at all times as the Lander will be non-pressurized.

The only concern with option two is SPE warning time, but this is a minor concern. Reliable (95% accurate) forecasts of SPE's and their size (accurate to one order of magnitude over a range of five possible orders of magnitude) can be made 20-30 minutes before an SPE begins. In addition, it takes another 20-30 minutes before the radiation flux from the SPE rises to a dangerous level. It is also possible to predict one to ten days in advance. Although the one to ten-day-before predictions could not be used to suspend a mission, they would be useful to increase alertness to a possibly developing hazardous situation. A solar telescope that includes a X-ray imager, a hydrogen-alpha scanner, and a solar magnetograph that could always "see" the sun and be located as close to the moon as possible is required in order to reduce transmission delays associated with earth based telescopes [2:674, 680].

CHAPTER 10: GUIDANCE, NAVIGATION AND CONTROL SYSTEMS

The electronics constituting the guidance, navigation, and control (GN&C) system will have the capability to perform autonomous landings and ascents. An autonomous system is technologically feasible, and with such a system, the mission scenarios have been optimized to achieve the following:

* decrease manned time on the Lander
* lessen astronaut exposure to radiation
* increase reliability
* and eliminate the need for a pressurized manned module.

For greater safety and versatility, a manual control override loop will be designed into the system. During unmanned missions, an crew member will pilot the Lander from the Lunar Space Station Docking Control module. During manned missions, such as for crew rotation, an astronaut inside the crew module will pilot the Lander from the avionics panel. For maximum safety, the GN&C system will be programmed for an abort sequence if critical numbers of general purpose computers, star trackers, and inertial measurement units were to malfunction.

The mission performance requirements and capabilities of the GN&C system will be as such:

* to determine instantaneous position and relative velocity
* to control the main engines
* to control the attitude control system
* to deviate only a few meters from landing targets.

To meet this criteria, the GN&C system will require the following equipment:

I. Star Trackers

II. Inertial Measurement Units (IMU's)

III. Video Cameras and UHF Transmission

IV. Landing Radar

V. Rendezvous Radar

VI. Surface Transponders

VII. General Purpose Computers (GPC's).

A detailed list of each of these elements is presented in Tables 10.1 & 10.2. A redundancy factor of three will be required for the star trackers and the IMU's and five for the GPC's. The landing and rendezvous guidance systems will have one active and one stand-by radars; both systems are space rated, but an automatic rendezvous mission has yet to be performed. Each of the four video cameras will rotate 90 degrees and so provide double redundancy in case one fails. The video camera system will aid the astronaut to negotiate away from craters and boulders. Figure 10.1 shows a top view of the video cameras and the landing radar locations, and Figure 10.2 depicts the location of the major elements of the GN&C system.

Stellar trackers are opto-electrical devices which determine vehicle attitude and position from angular measurements of selected stars [1]. The IMU relays data from its laser gyroscopes and accelerometers to the GPC's which calculate the velocity vectors and continuously determine position relative to the moon. The GPC's execute orbital maneuvers by controlling the main engines and the attitude control jets. The Ku-Band rendezvous radar will detect and automatically track the range, velocity, and orientation of the target

Table 10.1: Equip. List & Specifications I

Guidance, Navigation, and Control

I. Fixed Head Star Trackers
- A. Triple Redundancy
- B. BBRC CT401 Specifications
  1. 10 arc-second accuracy with calibration
  2. bright object sensor and protective shutter mechanism
  3. space rated

II. Inertial Measurement Units
- A. Triple Redundancy
- B. Ring Laser Gyroscopes
- C. Pendulum Accelerometers
- D. Horizontal Orientation Gyroscope
  1. landing gear release
  2. cross referenced with vehicle coordinate system

III. Video Camera and UHF Transmission System
- A. Unmanned and Manned Missions
- B. 4 Cameras
  1. Rotates 90 degrees
  2. Double Redundancy

Table 10.2: Equip. List & Specifications II

Guidance, Navigation, and Control

IV. Landing Radar
  A. Dual Redundancy
    1. One Active
    2. One Stand-by
  B. 4 Continuous Wave Beam
    1. Velocity
    2. Altitude
  C. 20 Kilometer to Surface Operational Range

V. Rendezvous Radar
  A. Triple Redundancy
  B. 800+ Kilometer Range
  C. Capable of Autonomous or Manual Guidance

VI. Surface Transponders
  A. 3 Minimum at a Site
  B. One Information Relay Dish at Site

VII. General Purpose Computers (GPC's)

Fig. 10.1: GN&C

Top View

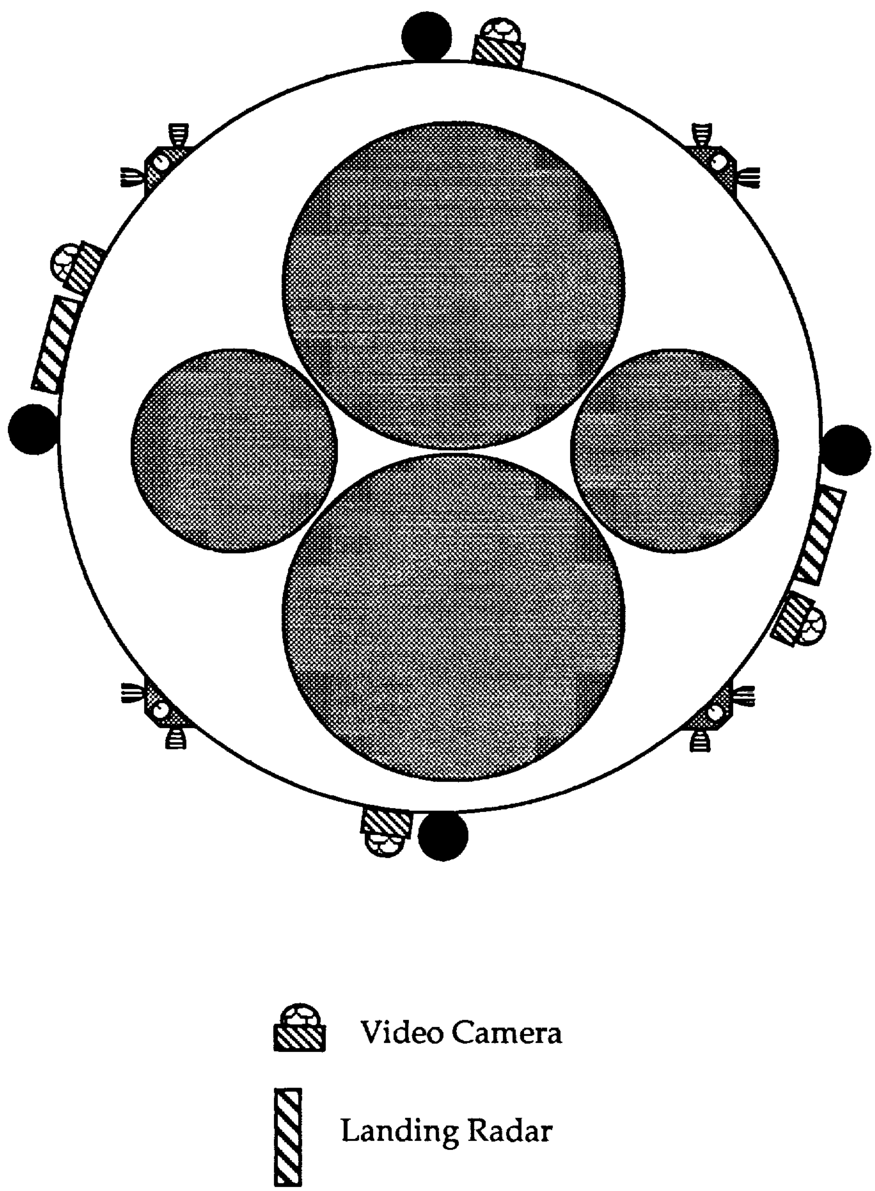
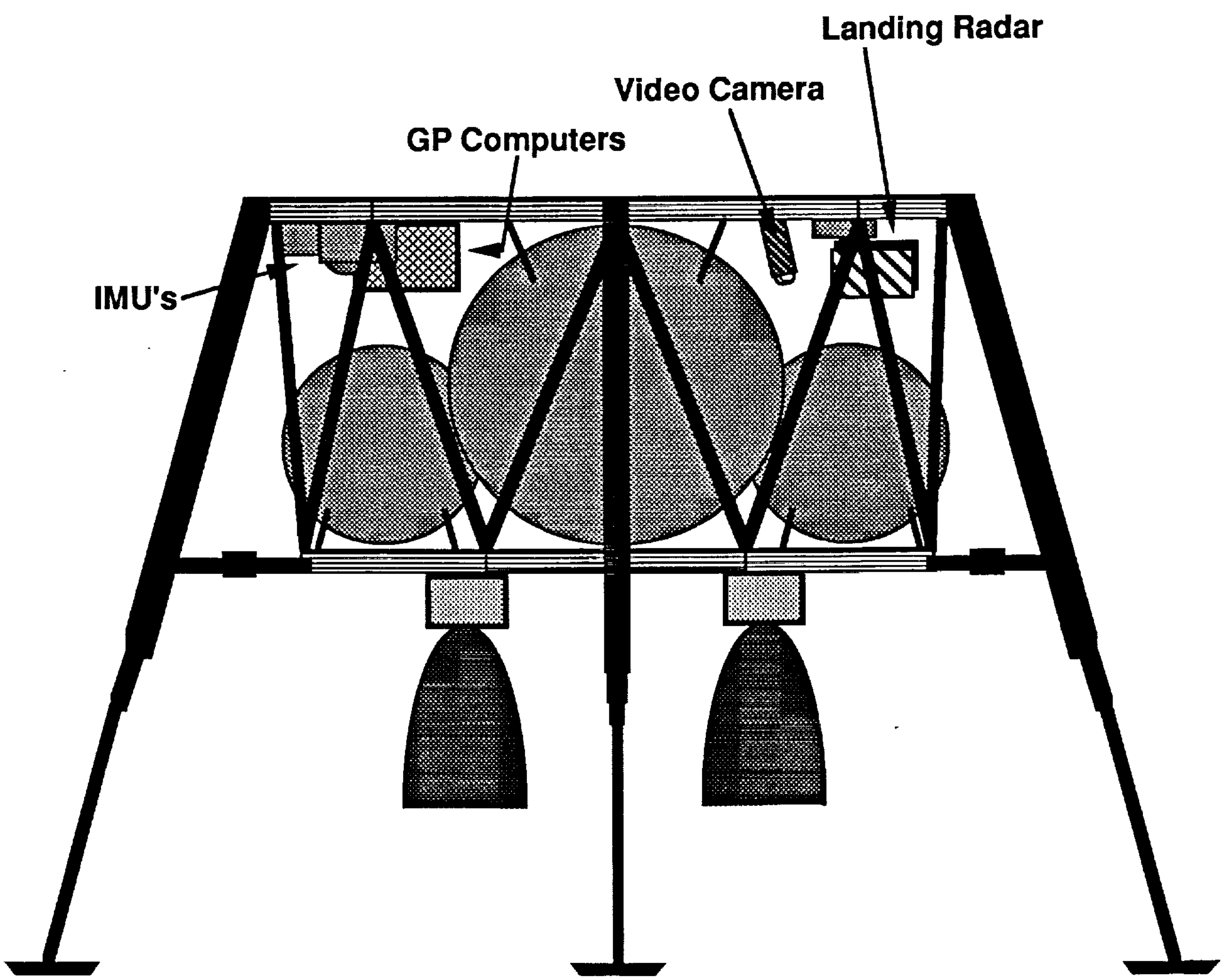


Fig. 10.2: GN&C

Location of Elements



vehicle [2]. Surface transponders will be located at the landing site of the surface base; a total of five will be adequate for proper triangulation. If the surface base were to be located in the highlands, a terrain matching radar would be of benefit in avoiding the mountainous terrain.

The Lander can continue with planned missions with a minimum of the following operational equipment: (1) four of five GPC's; (2) two of three star trackers; (3) two of three IMU's; (4) two rendezvous radars; (5) three of four video cameras; (6) one of two landing radars; and (7) three of four rocket engines. A mission abort condition would exist if any of the following number of equipment were to fail: (1) two of five GPC's; (2) two of three star trackers; (3) two of three IMU's; (4) two of four video cameras; (5) two of four rocket engines; or (6) one of two rendezvous radars. If any of the mentioned conditions were to occur, the abort programming would automatically initiate an abort sequence with manual override possible by either a member in the crew module or in the Lunar Station Control Module.

CHAPTER 11: COMMUNICATION AND DATA MANAGEMENT

The communication network system will provide voice and data transmission from the Lander to the Lunar Space Station (LSS), the OTV, the Surface Base, and the Deep Space Network (DSN). The Lander missions will require continuous communication and data links not only with elements in the lunar infrastructure but also with the DSN in emergency situations when the Lander may be out of contact with the LSS and with the Surface Base.

The Lander will use a combination of S-Band and Ku-Band transmitters and receivers. The S-Band unit will transmit and receive data and voice, and the Ku-Band unit will transmit data at a faster rate than the S-Band unit. This S/X-Band system was compared against microwave and laser communication systems (Figure 11.1). Although the Ka-Band microwave system promises much better performance than the present systems, and technological forecasts hint at availability by the turn of the century; non-compatibility with existing equipment does not make this system a viable option. The laser system offers many advantages, but the technological immaturity eliminates the system from consideration.

The designated equipment relating to communications and data management on the Lander are:

* S/X-Band Units
* Space Suits
* General Purpose Computers
* Displays

Fig. 11.1: Communication Systems

Equipment Options	Advantages	Disadvantages	Comments	Ranking
S/X Band	* space tested * transmits & receives data * mature technology * compatible with existing systems	* slow data flow rate a few Mbps		5
KA/MM-Wave (microwave link)	* increased data flow 50-300 Mbps * secure communications link; hard to interupt or jam * good signal accuracy	* immature technology * not compatible with existing systems	* development is needed * need to be compatible with present systems * chosen assuming assuming tech. advancement in the field	4
Optical-laser system	* greatest data flow rate 100+ Mbps * immune to interception and jamming	* technology very immature * signal accuracy difficult * short laser lifetime * not compatible with existing systems	* should be considered if major breakthroughs are achieved	2 5=most desired 1=least desired

Table 11.1 lists the equipment and specifications needed for the communication and data management systems. Although the crew module of the Lander will be non-pressurized, the EVA spacesuits will have communication systems within them so that the crew members will be able to talk to each other. The Lander's C&DM units will amplify the suit's signals and link the crew members with others in the lunar communication network. Three GPC's will be used with one stand-by. Display screens relating Lander GN&C data will be located in the LSS Control module and in the crew module avionics board. Advancements in HUD technology may allow display screens in the crew module to be substituted with the HUD in spacesuits.

Navstar-type satellites, used in the Global Positioning System (GPS), will be necessary during the early stage of Phase III as the lunar infrastructure develops. However, GPS-type satellites will not be used during the Phase II development due to costs and since communication within the lunar infrastructure and with the DSN will be possible 72 percent of the time. For approximately 45 minutes per orbit, the Lander will be in continuous contact with the LSS during descent and ascent missions. In Figure 11.2, scenario I illustrates this condition; scenario II shows that communication with the surface base will have to be relayed to the DSN for 64 minutes per orbit. Figure 11.3 shows that no communication by the Lander and LSS will be possible with the DSN nor with the surface base for 37 minutes per orbit; this constitutes 29 percent of non-communication time out of the total 127 minute orbital period. Such a condition is acceptable due to the frequency of orbits (11.3 orbits per 24 hour period) and due to autonomous capability of the LSS.

Table 11.1: Equip. List & Specifications

Communications & Data Management

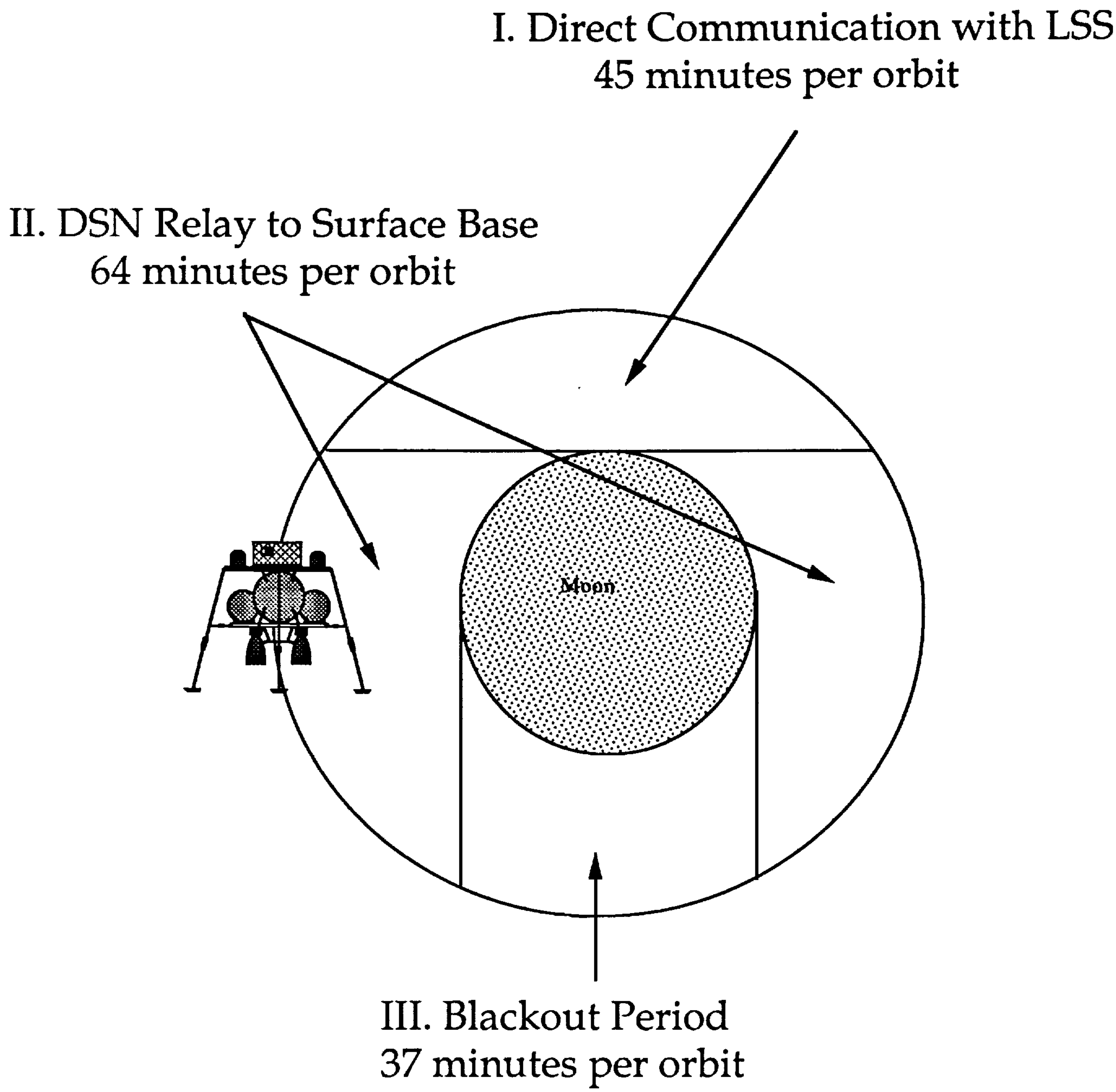
I. S/X Band System
  A. S-Band
    1. Lunar Infrastructure
    2. Deep Space Network Capability
  B. Ku-Band System
    1. Data Transmission
    2. Lunar Space Station
    3. Surface Base

II. Space Suits
  A. Signal Boost
  B. Close Proximity

III. General Purpose Computers
  A. 3 Active
  B. 1 Stand-by

IV. Displays
  A. Plasma Screens
    1. LSS Docking Control Module
    2. Lander Crew Module
  B. Space Suit HUD-Up Display

Fig. 11.2: Communication Contact Periods

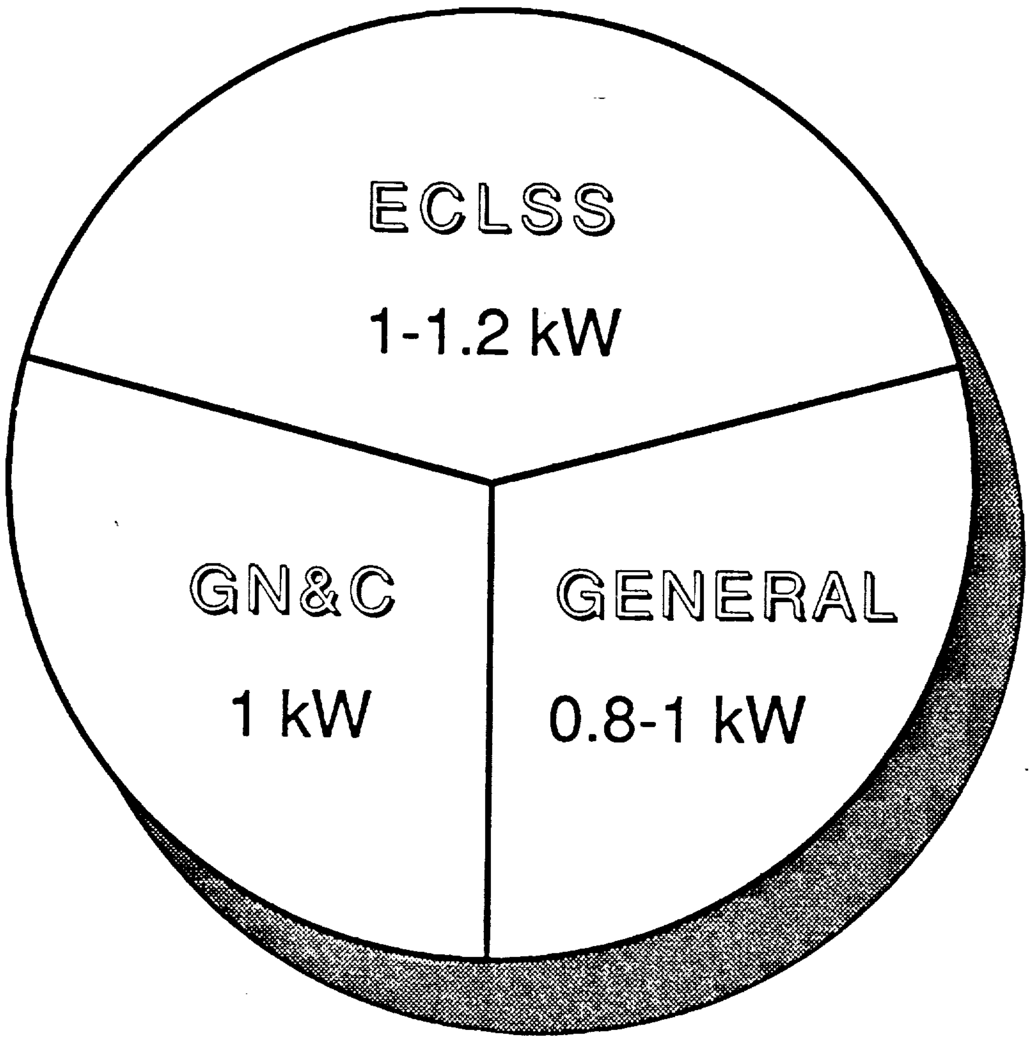


CHAPTER 12: ELECTRICAL POWER SYSTEMS

The power requirements for the Lunar Lander are determined by the worst-case scenario in which the Lander remains trapped on the surface of the moon for the duration of the lunar night. During this time, however, the Lunar Base cannot be relied upon to provide the power to maintain the Lander's electrical systems since it may be operating on temporary power storage facilities which need to be conserved as much as possible until they can be replenished through solar power generation during the lunar day. Consequently, the maximum endurance for the Lander's power supply must match the two-week lunar night. It is estimated that a total of 720 kW of power is needed for the two-week scenario at an average draw of 2-3 kW based on past experience with the Apollo Lander. For the general purpose manned cargo Lander with crew module, the majority of the power loads will come from ECLSS, which is predicted to occupy 40% of the power generated for the Lander (Figure 12.1). This is followed closely by GN&C requirements, which will take up about another 30% of the load. The final 30% will be used for heating and refrigeration of the tanks and lines and for powering various mechanical devices and actuators. For the straight cargo Lander, ECLSS power may be diverted to heavy duty AC motors for loading and unloading operations.

Due to the limitations in size, bulk, and weight for the Lander, as well as harsh mission requirements, it is most desirable to employ a small, rugged, high output power supply. Solar arrays are too bulky and fragile and cannot feasibly handle the load capacity. Nuclear power as well would be far too massive and dangerous. As a result, there are only two major options to consider: batteries or fuel cells. The four most powerful state-of-the-art battery systems were considered in contrast to the latest shuttle-derived fuel cell

Fig. 12.1: Power Distribution



technology (Figure 12.2). For the Lunar Lander, it was decided to go with the shuttle-derived fuel cell for the following reasons:

1. Fuel cells yield the highest energy content and allow for 100% redundancy.

2. The fuel cell has a total system weight (with 100% redundancy) that is only 10% as massive as the lightest battery system.

3. Fuel cells have far more flexibility over battery-powered systems as they can adjust accordingly to the varying demands of the different systems.

4. Fuel cells need only be refuelled, not replaced or recharged as with battery systems.

5. The Shuttle-derived fuel cell can run on the O_2 and H_2 propellant, thus no extra chemicals are required.

6. The fuel cell system can be easily integrated with the main propellant tanks with negligible changes in total fuel requirements.

7. The by-product of fuel cells is water, which can help fulfill ECLSS requirements.

8. Shuttle-derived fuel cell technology is the state-of-the-art and has proven to be highly successful; it requires no further development, whereas the best candidate battery systems are still under development.

Several configurations of each kind of power system were examined by Eagle Engineering. The results are summarized in the following table. Although all indications point towards the use of fuel cells, there is still one major drawback to consider. The biggest disadvantage in using fuel cells lies in the problem of trying to draw the reactants from the tanks when they are nearly empty. This could present serious problems over a two-week period or if leaks should occur in the fuel system.

The preliminary design of the fuel cell system for the proposed Lander is based on the Shuttle-derived fuel cell which is a pre-packaged, self-contained power system which

Fig. 12.2: Comparison of Power Systems

OPTION	ADVANTAGES	DISADVANTAGES	TECHNOLOGY	RANK
BATTERIES	* Rugged, compact	* Must be thrown away		
Ag / Zn	* Highly developed * Inexpensive * Reliable	* Heavy * Short life * Relatively inefficient	Available	2
Ni / H_2		* Complex * Lowest output	Available	1
Li-Al / FeS_2	* Slightly lighter * Newest technology	* Unreliable	Expected Soon	1
Li / TiS_2	* Extremely long life	* Expensive * Needs extensive development	To Be Developed	3
2 SHUTTLE-TYPE FUEL CELLS (100% redundant)	* Highest output * Lightest weight * Long-life * Reusable	* Requires maintenance * Functions poorly at very low fuel levels		
w / Separate Cryogenic Tanks	* Safest configuration	* Large bulk	Available	4
w / Main Fuel Tanks	* Extremely light * Marginal bulk * Easy to integrate		Available	5
Pressurized Tanks	* No cryogenics	* Difficult to install	Available	4

1 = worst
5 = best

can be universally applied with few modifications. The proposed system is a closed-loop system with two stacks of 32 cells each capable of providing 30 Volts DC at all times, which should be sufficient for all normal Lander operations. Each stack is assigned to a different pair of H_2 and O_2 tanks under normal operation, but can switch to the other pair through bypass manifold valves. In case of a leak in the tanks, flow check valves are installed to prevent fuel from the other tanks from flowing back into the tank.

The reactants are drawn from the tanks at a mass flow rate that is directly proportional to the current produced, except during purge operations which will be explained later. The H_2 and O_2 from the tanks flow through heavily insulated lines to the fuel cells where they are mixed and heated and undergo an electrolytic reaction whose product is electricity and water vapor. The water vapor is removed by the hydrogen gas flow which reacts with the oxygen in the stack and carries the resultant water vapor to a condenser before being recirculated to remix with fresh hydrogen from the tanks.

In this manner, 100% of the reactants are consumed in the reaction. However, due to the buildup of various impurities in the fuel cell, it will be necessary to purge the system periodically by simply venting the cell to space and blowing the contaminants overboard. Power generation need not be interrupted, if the flow of reactants is allowed to increase sufficiently. In addition, each system will be purged at alternating times, to insure continuous service.

The electricity produced is channeled through the DC bus distribution system and either passes directly to the loads or is converted to AC power through an AC converter and is then distributed to any AC loads, such as the ramp winch in the cargo bay. The water produced is collected directly in the potable water tanks for use by the ECLSS system as required.

The general characteristics of the proposed electrical power system to be used on the Lunar Lander are given in Table 12.1. These are preliminary design parameters based on past experience with the Apollo and Shuttle Orbiter spacecraft.

Table 12.1: Power System Performance

Components: 2 Fuel Stacks (100% redundant) of 32 cells

Reactant storage: $+31$ kg H_2

$+244$ kg O_2

Size: 0.058 m^3 disp./Stack

Weight: 68 kg/Stack

Total system weight: 685 kg (w/reactants for 15 days)

Operating Temperature: 80 - 95 C

Total Power Production: 720 kWh/Stack at 2kW average

Water production: 260 kg at 3/4 L/hr for 15 days

Operating Range: 0 - 4 kW per stack, 28 - 32.5 max RMS Volts

Maximum Mission Endurance: 15 days.

Appendix I: Orbital Calculations

Lunar Radius = R = 1738 km
Rotation Period = P = 27.3 days = 2.36×10^6s

Station will pass above a point on the equator twice per orbit, neglecting precession - once from north of equator, once from south.
Thus it will pass above that point once every 13.7 days.

Ideal Plane Change Equation:
delta-V = 2 V sin (theta/2),
where V is current velocity, and theta is desired plane change angle plane change.

For a circular orbit, $V_C = (mu / R)^{1/2}$; mu = 4.90287×10^{12} m^3/s^2

Hohmann Transfer Ellipse
tangent to both circular orbits, R_1 = 1938 km, R_2 = 1831 km:

V_{c1} = 1591 m/s
$V_{tran1} = (mu\ (2/R_1 + 1/a))^{1/2}$ = 1568 m/s
a = $(R_1 + R_2) / 2$ = 1884 km
delta-V_1 = V_{c1} - V_{tran1} = 23 m/s

$V_{tran2} = (mu\ (2/R_2 + 1/a))^{1/2}$ = 1660 m/s
V_{c2} = 1637 m/s
delta-V_2 = 23 m/s

period = 2 pi $a^{3/2}$ / $mu^{1/2}$ = 7344 s = 2 hrs, 2 min, 24 s
quarter period = 30.6 min

quarter period of Station orbit = 31.9 min
delta-t = 1.3 min
delta-angle = (1.3 / (4*31.9) * 360°) = 3.7°

Station will be 3.7° ahead of Lander at touchdown, and also at liftoff on missions from lunar surface to Station.

Launch and landing trajectory data computed by FIRE.BAS.

APPENDIX II: PROPELLANT TANK ARRANGEMENTS TRADE STUDY

The following arrangements were analyzed for factors such moment of inertia, total tank weight, geometry, allowance for structural support, simplicity in design. The proposed propellant tank arrangements were analyzed for the following categories.

A: MOBILE TANKS

The propellant tanks are arranged to minimize the moment of inertia of the tanks, since that will minimize the fuel required to control the vehicle during landing. However, in this design, the overhang of the tanks prevents the entire vehicle from fitting into the ALS, the heavy lift launch vehicle which will carry the Lander into orbit. Therefore, if this design were to be implemented, the tanks would have to be shifted in transit and put into place in orbit. It may be fairly difficult or impossible to hook up a cryogenic tank with the turbomachinery while in orbit. That fact coupled with the fact that the arrangement did not allow much room for members to support the upper level, eliminated this option from being chosen (Figure I.1).

B. STACKED TANKS

One way to allow more space on the lower level for supporting members is to stack the propellant tanks (Figure I.2). The problem with this configuration is that the center of gravity of the arrangement up shifts from a single level arrangement by 2 meters. This in itself would not be much of a problem except that the c.g. shifts the level above it up by 2 meters as well. A low center of gravity is desired for landing stability. It is critical that it not tip over, even when landing on a shallow slope. In addition, in this arrangement it may be difficult to remove a hydrogen tank in case it should need to be replaced due to micrometeoroid damage or other problems. Since these vehicles are to be reusable, they must be maintainable. For these reasons this option was not chosen.

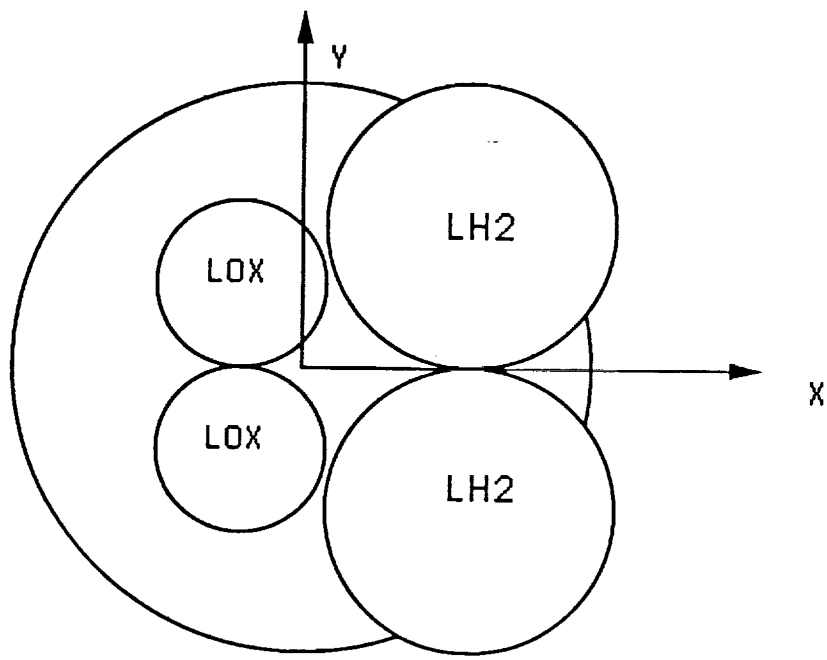
C. CENTERED TANKS

This arrangement is single level, the hydrogen tanks are able to stay within the 33 foot limit imposed by the ALS. The moment of inertia and center of gravity are low. There is a fairly even weight distribution among the four legs, and this arrangement allows for an ample supporting truss between upper and lower levels. This arrangement, shown in Figure I.3, was rated best overall.

D. THREE LOX TANKS

Another way which was sought to keep the tanks within the thirty foot diameter limit was to increase the number of tanks, making them smaller, thus implying less unused space on the lower level. Unfortunately, this idea (Figure I.4) did not work. When the tanks were sized, (from 2-5 tanks of hydrogen and 2-3 tanks of LOX), it was found that increasing the number of tanks at that radius did not significantly decrease the radius, in short, the tanks would still not fit within the 30 foot radius. The only way to make them do so was to stack them and that case has already been examined.

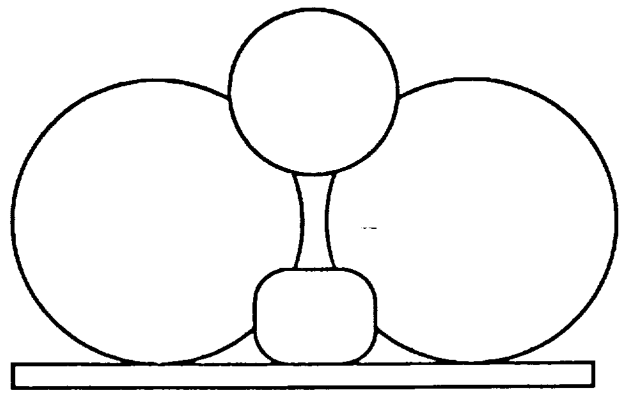
Fig. I.1: Mobile Tanks



Radius O2 Tank = 4.6 ft.

Radius H2 Tank = 7.28 ft.

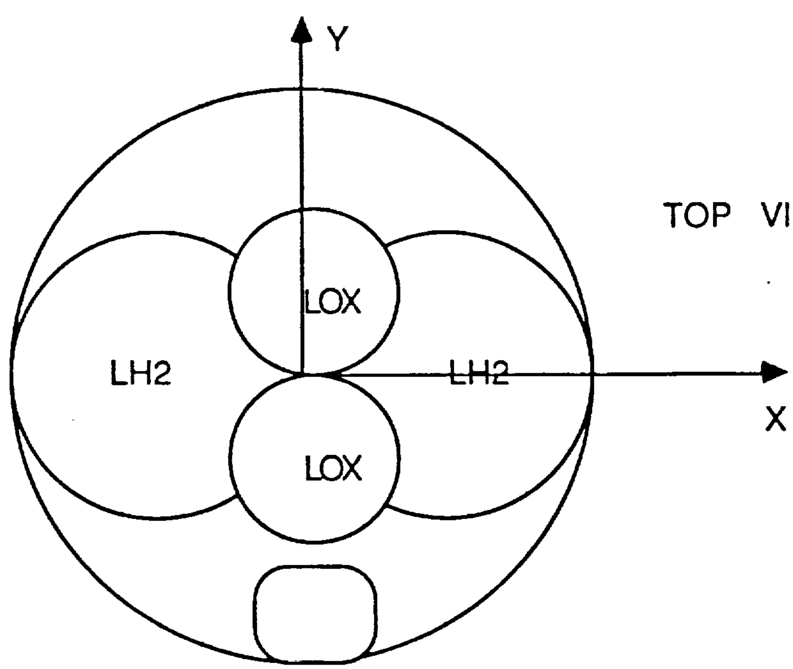
Fig. I.2: Stacked Tanks



SIDE VIEW

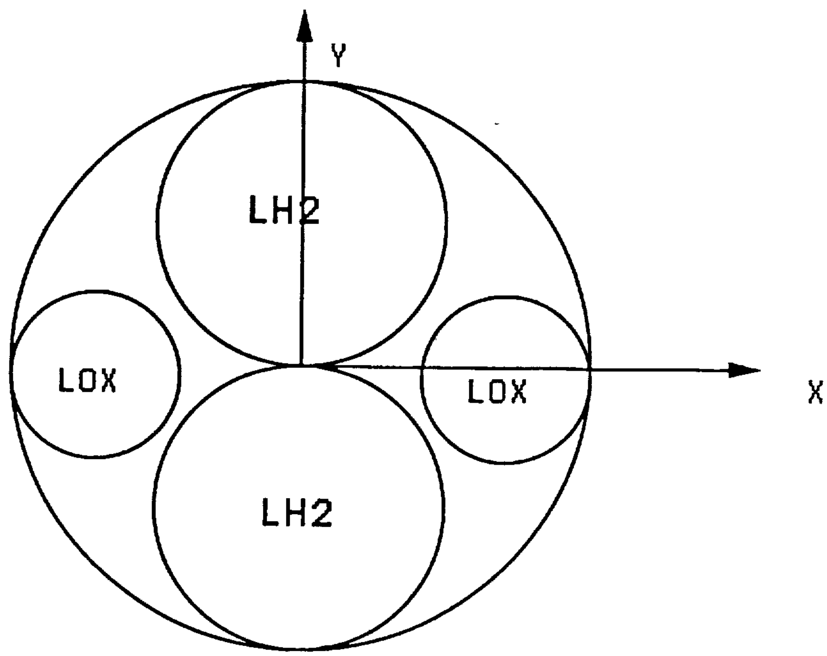
Radius O2 Tank = 4.62 ft.

Radius H2 Tank = 7.09 ft.



TOP VIEW

Fig. I.3: Centered Tanks



Radius O2 Tank = 4.6 ft.
Radius H2 Tank = 7.28 ft.

E. CYLINDRICAL TANKS

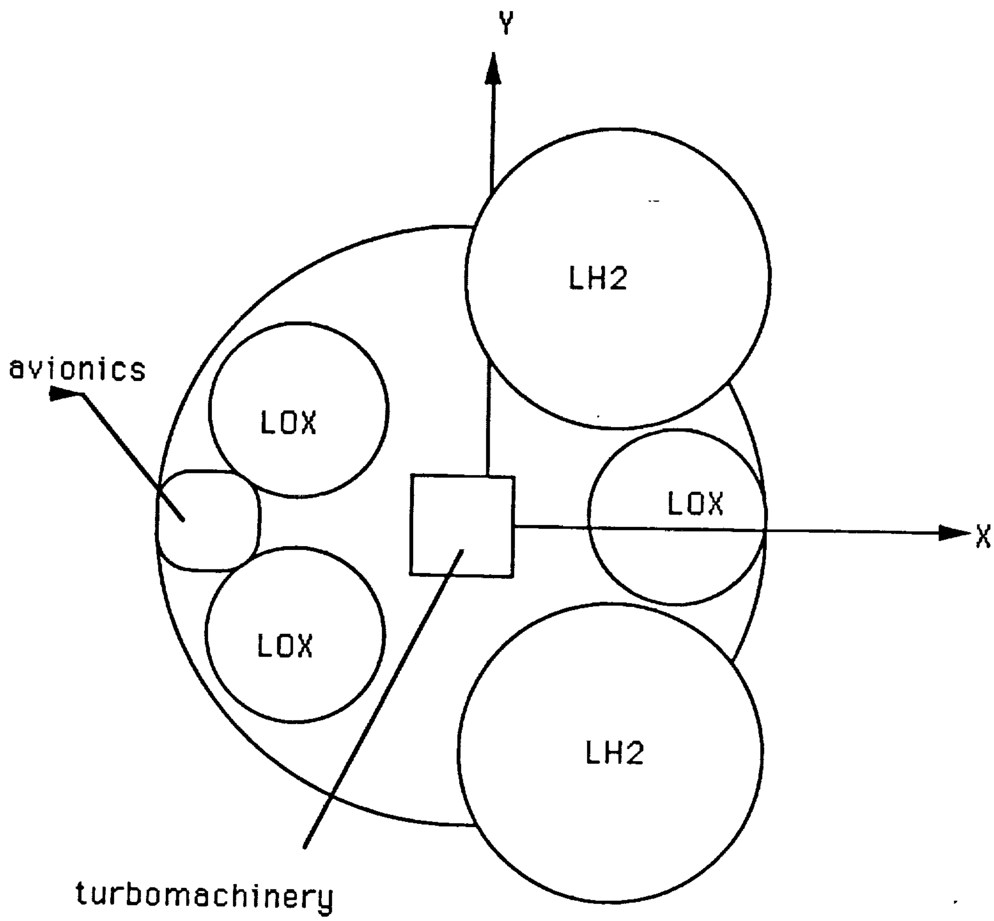
These tanks would allow space for supporting members on the lower level, but were found to be excessively heavy, due to their increased surface area. Weight calculations are given in Appendix III.

F. LOX TANKS UNDER THE LOWER SUPPORT FRAME

This suggestion was also offered in response to allowing for more space on the lower level. Unfortunately, the liquid oxygen tanks, due to their size would be right next to the rocket throat and upper nozzle, from which a great deal of heat is radiated. Since the cryogenic fuel cannot be exposed to that kind of heat source during flight, this option was also considered unfeasible.

An overall ranking of the various options is presented in Table I.1.

Fig. I.4: More Tanks



Radius O2 TANK = 4.02 ft.
Radius H2 TANK = 7.09 ft.

Table I.1: Tank Arrangement Options

RANKING OF ALTERNATIVES

FACTORS	MOBILE TANKS	STACKED TANKS	CENTERED TANKS	CYLINDRICAL TANKS	LOX UNDER
MOMENT OF INERTIA	4	5	4	4	4
EVEN WEIGHT DISTRIBUTION	2	2	2	2	2
EMPTY TANK WEIGHT	5	5	5	1	5
INTEGRATION INTO STRUCTURE	4	4	5	5	5
HEIGHT OF C.G.	3	1	3	3	3
SIMPLICITY	1	3	3	3	1
TOTAL	19	20	22	18	20

RANKING VALUES: 5 = BEST
1 = WORST

APPENDIX III: SIZING OF COMPRESSIVE MEMBERS

A. PALLET MEMBERS

To estimate the weight of the octagonal frame, a finite element code, SIMPAL, was used. The program was used because loads on the structure were primarily those of shear and bending, a situation which could be modelled fairly well with the program, using three dimensional beam members. An arbitrary size I beam was chosen for the structure and using this I beam, a model was built on a finite element code of a quadrant of the octagonal frame. Loads were applied to the model, to simulate landing, and the program was executed. Through an iterative process of examining results, changing the member sizes and rerunning the program, member sizes were found which were structurally adequate. The total weight of the members, which is calculated in the program, was noted and the entire frame weight and mass were calculated. A copy of the input files to the finite element program and results of the finite element analysis are given on the following pages.

B. LANDING LEGS AND THE SUPPORTING TRUSS MEMBERS

Each landing leg was modelled as a series of three straight tubes. The truss members supporting the cargo pallet were also modelled as hollow tubes.

The modes of failure defining the design boundaries of the Lander legs and truss members were: buckling, failure due to compressive stress and fatigue.

Buckling
The buckling analysis provided the following information:
From Euler's Buckling formula:

$$\text{sigma} = \frac{Pcr}{A} = \frac{pi^2(E)}{(L/Rg)^2} = \frac{Rg^2 pi^2(E)}{L^2}$$

where Rg = radius of gyration of the column.
the following can be derived:

(1) $$J = \frac{(Pcr)L^2}{pi^2 E}$$

Landing Legs
Setting Pcr = 44E3 lbs, twice the maximum compressive load that the leg will have to support, L = 240*2 in., the effective length of the upper section of the leg, and E = 1.12E7 psi, the modulus of elasticity of the aluminum lithium alloy, the polar moment of inertia required to prevent buckling of the leg is: J = 91.72 in^4.

Maximum Compressive Stress
The minimum area required to prevent failure due to compressive yield stress of fatigued material:

(2) $$A >= \frac{P}{\text{sigma}} = \frac{44E3}{22E3} = 2 \text{ in}^2$$

Also it is known that: $J = AR^2$. Combining this fact with (1) and (2) implies that $Rg >= 6.77$ in. Choosing a value of $A = 7$ implies $Rg >= 3.62$ This value can be put into the following formula: $Rg^2 = (1/4)(Ro^2 + Ri^2)$ for a hollow tube, where Ro and Ri are the inner and outer radii of the tube. Since $Ri = Ro - t$, where t = the thickness of the tube wall, by choosing an arbitrary wall thickness of 0.25", Ro can be calculated from

$$Rg = 4.79 = [(1/4)(Ro^2 + (Ro - t)^2)]^{1/2}$$

Ro is given by : $2Ro^2 - 0.5Ro - 46.18 = 0$ which yields Ro = 4.99, Ri = 4.74 in, and A = 7.65 in^2 which is fairly close to, but is not less than, the initial guess of A = 7.00 in^2. This procedure could be continued iteratively but for the purposes of a weight estimate it is not necessary.

Continuing on for the middle and bottom sections of this leg, allowing the total weight of the legs to be calculated.
Weight = (num)(rho)(Ac)L where num = number of objects being analyzed, rho = the material density is lb/in^3, Ac = the cross sectional area of the tube, and L = its true length.

Legs, top section	740 lbs.
middle section	120
bottom section	240
est. wt. of landing mechanism	200
est. wt. of landing leg attachments	400
footpads	100
TOTAL	**1 800 lbs = 820 kg.**

Weight of the truss members:
Pcr = 900 lbs/member, L = 198 in. => J = 0.638 in^4
Amin = 0.082 in^2 Let A = 0.8 in^2 => $Rg = 0.89 = [(1/4)(Ro^2 + (Ro - t)^2)]^{0.5}$

yielding: Ro = 1.2 in. Ri = 1.07 in. A = 0.89 in^2

Weight of truss members = 282 lbs = 128 kg

APPENDIX IV: WEIGHT OF CYLINDRICAL TANKS

First on a theoretical basis, the ratio of surface areas of an enclosed cylinder to a sphere will be shown:

rc := cylinder radius
r := spherical radius

$$\text{RATIO} = \frac{2*pi*rc*h + 2(pi*rc^2)}{4(pi*r^2)} = \frac{(h + rc)}{2r}$$

For equal volumes:
4/3*pi*r^3 = h*pi*rc^2
h = 4/3(r^3/rc^2)

RATIO:= (2/3)(r/rc)^2 + rc/2r
Therefore a ratio of r/rc <u>or</u> rc/r which is large will cause the surface area of the cylinder and consequently, the cylindrical shell's mass to be much higher than a spherical shells.

FOR A SPECIFIC CASE:

N = No. of tanks = 2 H2 tanks and 2 LOX tanks
t = thickness of the tank wall 0.013m (0.5 in)
h = cylinder height = 2.966m (9.73ft)
r = cylinder radius rox = 1.1m rh = 4.42m (14.5 ft)
m = mass of the cylinder
V = total tank shell volume
d = density kg/m^3 of the material

V = N (2pi*r*h + 2pi*r^2)*t

for N=2:
V = 4pi*t(r*h + r^2)

for the 2 LOX tanks the tank mass is:
m = [4*3.14159*0.013(1.1*2.966 + (1.1)^2)] * 2643
m = 1931 kg

for the 2 H2 tanks the tank mass is:
m = [4*3.14159*0.013(4.42*2.966 + (4.42)^2)] * 2643
m = 14096 kg

TOTAL MASS: 16026 kg

APPENDIX V: MOMENTS OF INERTIA FOR CYLINDRICAL TANKS

The moment of inertia of a cylinder about a radial axis is:
$I = (1/12)*m(3r^2 + h^2) + md^2$

about its axial axis:
$I = (1/2)*m*r^2 + md^2$

Note: the mass of the oxygen tank + fuel = 13931 kg
The mass of the hydrogen tank + fuel = 17095 kg

For the above case:
$Ixx = [(1/12)*13931*[3*(1.1)^2 + (2.966)^2] + (1/12)*17095*[3*(4.42)^2 + (2.966)^2] + 17095(4.42)^2]*2$

$Ixx = 8.88E5$ kg-m2

$Iyy = 166758 + 96029 = 5.26E5$ kg-m2

$Izz = (1/2)*13931*(1.1)^2 + (13931)*(3.31)^2 + (1/2)*17095(4.42)^2 + (17095)(4.42)^2 = 4.95E5$ kg-m^2

APPENDIX VI: FIRE.BAS ROCKET PERFORMANCE PROGRAM

```
10 REM ********************************************************
11 REM *
13 REM *   FIRE.BAS
15 REM *
20 REM *   CAESAR G. MAMPLATA'S PROPELLANT MASS PROGRAM
25 REM *
30 REM ********************************************************
35 REM
36 CLEAR
37 PI = 4*ATN(1)
39 CLS : PRINT:PRINT
40 REM
45 REM ***** BEGIN INPUT SECTION *****
50 REM FIRST INPUT LANDER MASS COMPOSITION
55 PRINT:PRINT:PRINT "ALL MASS ENTRIES SHOULD BE IN KILOGRAMS"
60 PRINT:PRINT:PRINT "ENTER HYDROGEN MASS GUESS: ";
70 INPUT MLH2:PRINT
80 PRINT "ENTER LUNAR LANDER PAYLOAD DROPPED OFF ON SURFACE: ";
85 INPUT PAYLOAD:PRINT
90 PRINT "ENTER LUNAR LANDER INERT MASS: ";
95 INPUT LANDER:PRINT:PRINT
96 PRINT: PRINT " ENTER '1' TO CONTINUE: ": INPUT ANSWER
97 IF ANSWER = 1 THEN GOTO 120
100 REM **** END INPUT SECTION *****
105 REM
110 REM **** FIRST COMPUTE ACTUAL LANDER STACK MASS ****
115 REM
120 CLS: GOSUB 1000
130 REM MPGUESS = PROPELLANT MASS, TOTAL = LUNAR LANDER STACK MASS
135 REM
140 REM **** COMPUTE PROPELLANT MASS CONSUMED IN DESCENT ****
141 REM
145 CLS: GOSUB 20000
150 REM
155 REM TAKE INTO ACCOUNT ADDITION AND REMOVAL OF PAYLOAD TO
156 REM COMPUTE ASCENT TAKEOFF MASS
160 PRINT:PRINT "ENTER PAYLOAD MASS ADDED TO LUNAR LANDER ON LUNAR SURFACE: ";
165 INPUT MASSADDED
170 REM COMPUTE ASCENT MASS
175 REM ASCENT MASS = LANDING MASS - PAYLOAD DROPPED + PAYLOAD ADDED
177 ASCENT = (HLIMASS-HLPMASS) - PAYLOAD + MASSADDED
180 REM
181 REM **** COMPUTE PROPELLANT MASS CONSUMED IN DESCENT ****
185 GOSUB 30000
900 END
1000 REM ****************************************************
1010 REM ***** SUBROUTINE TO DETERMINE TANK AND MLI MASS *****
1100 REM ***** FOR EACH LOX AND LH2 TANK                *****
1110 REM ****************************************************
1200 REM
1300 REM ***** BEGIN INITIAL PROPELLANT TANK MASS COMPUTATION *****
1310 REM
1350 REM TANK MASS COMPUTATION
1360 PRINT:PRINT:PRINT
1400 PRINT "ENTER OXIDIZER TO FUEL RATIO: ";
1460 INPUT OF
1465 REM
1470 REM *** DIVIDE PROPELLANT MASS INTO FUEL AND OXIDIZER ***
1480 REM
1500 REM MASS OF LIQUID OXYGEN, INDICATED BY "LOX" IN THIS PROGRAM.
```

```
1550 MLOX = OF*MLH2:PRINT:PRINT
1570 REM MASS OF LIQUID HYDROGEN, INDICATED BY "LH" OR "LH2" IN THIS PROGRAM.
1590 REM COMPUTE TOTAL MASS OF PROPELLANTS
1595 REM *** INCORPORATE 5% EACH FOR ULLAGE, BOIL-OFF, AND EMERGENCIES... ***
1600 MPGUESS = 1.15*(MLOX + MLH2)
1610 REM
1620 REM *** COMPUTE LOX TANK VOLUME AND RADIUS ***
1650 REM USER SUPPLIED LOX INPUT VALUES
1700 RHO = 1140: TEMP = 90 : VAPOR = 5
1850 REM INVOKE SUBROUTINE TO COMPUTE VOLUME OF EACH LOX TANK
1900 REM AND RADIUS OF EACH LOX TANK.  THE SAME PROCEDURE IS USED
1950 REM IN DEALING WITH THE LH TANKS.
1960 REM PREPARE MASS FOR SUBROUTINE COMPUTATION.
1980 MASS = MPGUESS*OF/(OF+1): MLOX = MASS
2000 GOSUB 10000
2100 REM SET SUBROUTINE VALUES TO LOX VALUES.
2150 LOXVOL = VOLUME: LOXIRAD = RADIUS
2160 REM
2170 REM *** COMPUTE LH2 TANK VOLUME AND RADIUS ***
2200 REM USER SUPPLIED LH INPUT VALUES
2250 RHO = 71: TEMP = 20 : VAPOR = 5
2300 REM PREPARE MASS FOR SUBROUTINE COMPUTATION.
2350 MASS = MPGUESS/(OF+1): MLH2 = MASS
2400 GOSUB 10000
2410 REM SET SUBROUTINE VALUES TO LH2 VALUES.
2430 LHVOL = VOLUME: LHIRAD = RADIUS
2460 REM
2470 REM *** COMPUTE TANK THICKNESS FOR LOX AND LH2 TANKS ***
2480 REM
2500 REM INPUT VAPOR PRESSURES OF LOX AD LH2
2600 PRINT "ENTER LOX VAPOR PRESSURE IN PSIA: ";
2650 INPUT LOXPRESS
2700 PRINT "ENTER LH2 VAPOR PRESSURE IN PSIA: ";
2750 INPUT LHPRESS
2800 REM INPUT MAXIMUM STRESS
2850 MAXSTRESS = 50000!
2860 REM
2870 REM ** BEGIN ITERATION TO FIND WALL THICKNESSES **
2880 REM
2900 REM INITIALIZE GUESSES AND INITIALIZE LOOP:
2910 LOXORAD = LOXIRAD : LHORAD = LHIRAD : I = 0
2920 REM
2930 REM LOOP START
2950 LOXWT = LOXPRESS*LOXORAD*39.37/MAXSTRESS
3000 LHWT = LHPRESS*LHORAD*39.37/MAXSTRESS
3050 LOXERROR = (LOXIRAD+LOXWT/(2*39.37)) - LOXWT
3100 LHERROR = (LHIRAD+LOXWT/(2*39.37)) - LHWT
3150 LOXORAD = LOXIRAD + LOXWT/(2*39.37)
3200 LHORAD = LHIRAD + LHWT/(2*39.37)
3250 I = I + 1
3300 IF I < 10 GOTO 2930
3350 REM LOOP END
3355 PRINT:PRINT
3360 PRINT "LOX TANK VOLUME IS ";LOXVOL;" CUBIC METERS."
3370 PRINT "LH2 TANK VOLUME IS ";LHVOL;" CUBIC METERS."
3[illegible] PRINT
3400 PRINT "LOX TANK OUTER RADIUS IS ";LOXORAD;" METERS."
3410 PRINT "LH2 TANK OUTER RADIUS IS ";LHORAD;" METERS."
3415 PRINT
3420 PRINT "LOX TANK INNER RADIUS IS ";LOXIRAD;" METERS."
```

```
3430 PRINT "LH2 TANK INNER RADIUS IS ";LHIRAD;" METERS."
3440 PRINT
3450 PRINT "LOX TANK THICKNESS IS ";1000*(LOXORAD-LOXIRAD);" MILLIMETERS."
3460 PRINT "LH2 TANK THICKNESS IS ";1000*(LHORAD-LHIRAD);" MILLIMETERS."
4000 REM
4100 REM *** BEGIN COMPUTATION OF LOX AND LH2 TANK MASSES ***
4150 REM
4200 LOXTANK = (2643*4*PI/3)*(LOXORAD^3-LOXIRAD^3)
4250 LHTANK = (2643*4*PI/3)*(LHORAD^3-LHIRAD^3)
4300 LOXVOLUME = LOXTANK/2643: LHVOLUME = LHTANK/2643
4340 REM
4350 REM *** BEGIN COMPUTATION OF LOX AND LH2 MLI MASSES ***
4360 REM
4400 LOXTMLI = .12: LHTMLI = .12
4450 LOXMLI = (25*4*PI/3)*((LOXORAD+LOXTMLI)^3-(LOXORAD)^3)
4500 LHMLI = (25*4*PI/3)*((LHORAD+LHTMLI)^3-(LHORAD)^3)
4550 REM
4560 REM *** PRINT MASS OF TANKS AND MLI FOR LOX AND LH2 ***
4570 REM
4600 PRINT: PRINT " ENTER '1' TO CONTINUE: ": INPUT ANSWER
4610 IF ANSWER = 1 THEN GOTO 4640
4640 CLS:PRINT "SPECIFICATIONS FOR ONE LOX TANK":PRINT
4650 PRINT        "LOX MASS (KG)    TANK MASS (KG)     MLI MASS (KG)     COMBINED M
SS (KG)"
4700 PRINT USING " ######.###        ######.###         ######.###        ######.##
";MLOX/2,LOXTANK,LOXMLI,LOXTANK+LOXMLI
4740 PRINT:PRINT:PRINT:PRINT "SPECIFICATIONS FOR ONE LH2 TANK":PRINT
4750 PRINT        "LH2 MASS (KG)    TANK MASS (KG)     MLI MASS (KG)     COMBINED M
SS (KG)"
4080 PRINT USING " ######.###        ######.###         ######.###        ######.##
";MLH2/2,LHTANK,LHMLI,LHTANK+LHMLI
4900 PRINT:PRINT:PRINT:PRINT "MASS SPECIFICATIONS FOR PROPELLANT TOTAL":PRINT
4950 PRINT     " PROPELLANT (KG)    TANK MASS (KG)     MLI MASS (KG)     COMBINED M
SS (KG)"
5000 PRINT USING " ######.###        ######.###         ######.###        ######.##
";MPGUESS,2*(LHTANK+LOXTANK),2*(LHMLI+LOXMLI),2*(LHTANK+LHMLI+LOXMLI+LOXTANK)
5100 REM
5102 PRINT: PRINT " ENTER '1' TO CONTINUE: ": INPUT ANSWER
5104 IF ANSWER = 1 THEN GOTO 5150
5110 REM *** COMPUTE TOTAL TANK MASS AND TOTAL MLI MASS FOR ***
5120 REM *** TWO LOX AND TWO LH2 TANKS ************************
5130 REM
5140 REM TOTAL TANK MASS
5150 TANKSTOTAL = 2*(LOXTANK+LHTANK)
5160 MLITOTAL = 2*(LHMLI+LOXMLI)
5190 TOTAL = MPGUESS + TANKSTOTAL + MLITOTAL + PAYLOAD + LANDER
5990 REM
6000 REM *** BEGIN COMPUTATION OF TOTAL LUNAR LANDER MASS ***
6010 REM
6100 CLS:PRINT:PRINT
6150 PRINT:PRINT "TOTAL STACK MASS OF LUNAR LANDER IS ";TOTAL;" KG: ":PRINT
6160 PRINT:PRINT" PAYLOAD = ";PAYLOAD;" KG":PRINT" INERT MASS = ";LANDER;" KG":
     PRINT" PROPELLANT = ";MPGUESS;" KG"
6170 PRINT" TANKS = ";TANKSTOTAL;" KG":PRINT" MLI = ";MLITOTAL;" KG"
6180 PRINT:PRINT:PRINT
6182 PRINT: PRINT " ENTER '1' TO CONTINUE: ": INPUT ANSWER
6184 IF ANSWER = 1 THEN GOTO 8000
6200 REM
6990 REM ***** END OF SUBROUTINE TO DETERMINE TANK AND MLI MASSES *****
7000 REM
```

```
8000 RETURN
10000 REM ***** TANK VOLUME AND RADIUS SUBROUTINE *****
10100 VOLUME = (MASS/RHO)/2
10200 RADIUS = ((3*VOLUME)/(4*PI))^(1/3)
10300 RETURN
20000 REM *******************************************************
20010 REM ***** SUBROUTINE TO DETERMINE THE PROPELLANT MASS *****
20020 REM ***** CONSUMED DURING DESCENT AND ASCENT, TAKING  *****
20030 REM ***** INTO ACCOUNT PREVIOUS REDUCTION OR INCREASE *****
20040 REM ***** IN ASCENT TAKEOFF MASS DUE TO LUNAR OPERATIONS .*
20050 REM *******************************************************
20060 REM
20070 REM **** BEGIN DESCENT PHASE COMPUTATIONS ****
20080 REM
20082 REM
20085 REM *** BEGIN GRAVITY TURN ***
20087 REM
20090 REM IDENTIFY THRUST PARAMETERS FOR FIRST PART OF DESCENT;
20100 REM GRAVITY TURN REQUIRES THE FOLLOWING...
20110 ITWRATIO = .275: FTWRATION = .487: ITIME = 0 : FTIME = 495
20120 ISP = 450
20130 A = (FTWRATIO-ITWRATIO)/(FTIME-ITIME): B = ITWRATIO: C = ISP
20135 P = (-B + SQR(B^2-4*A*C))/(2*A): Q = (-B - SQR(B^2-4*A*C))/(2*A)
20137 TB = FTIME - ITIME
20145 REM PROPELLANT MASS REQUIRED FOR CONSTANT THRUST TO WEIGHT RATIO
20150 GTPMASS = TOTAL*ITWRATIO*TB/ISP
20152 GTMDOT = GTPMASS/TB
20155 PRINT "GRAVITY TURN REQUIRES ";GTMDOT;" KG/SEC OF PROPELLANT."
20160 PRINT "GRAVITY TURN REQUIRES ";GTPMASS;" KG OF PROPELLANT."
20165 REM
20167 REM *** BEGIN REDUCTION TO HOVER ***
20169 REM
20170 REM IDENTIFY THRUST PARAMETERS FOR SECOND PART OF DESCENT;
20180 REM DESCENT TO HOVER REQUIRES THE FOLLOWING...
20190 ITWRATIO = .5: FTWRATIO = .165 : ITIME = 495 : FTIME = 530 : ISP = 450
20200 REM NOW MASS OF LUNAR LANDER IS TOTAL-GTPMASS DUE TO BURN UP OF
20210 REM PROPELLANT DURING GRAVITY TURN...
20220 RHIMASS = TOTAL - GTPMASS
20230 PRINT "LANDER MASS BEFORE REDUCTION TO HOVER IS ";RHIMASS;" KG."
20240 REM REDEFINE INITIAL THRUST TO WEIGHT RATIO AND RATE OF THRUST TO RATIO,
20250 REM RESPECTIVELY, AS 'A' AND 'B'.
20260 B = ITWRATIO:  A = (FTWRATIO-ITWRATIO)/(FTIME-ITIME) : C = ISP
20270 TB = FTIME - ITIME
20280 DET = SQR(B^2 - 4*A*C):  P = (-B+DET)/(2*A) : Q = (-B-DET)/(2*A)
20290 REM
20300 REM PROPELLANT MASS REQUIRED FOR VARYING THRUST TO WEIGHT RATIO...
20310 RHPMASS = RHIMASS*(LOG((A*TB^2+B*TB+C)/C)+B*LOG((TB-P)*Q/((TB-Q)*P)))/(A*(
P-Q))
20312 PRINT "REDUCTION TO HOVER REQUIRES ";RHPMASS;" KG OF PROPELLANT."
20315 REM
20317 REM *** BEGIN HOVER TO LANDING ***
20319 REM
20320 REM IDENTIFY THRUST PARAMETERS FOR THIRD PART OF DESCENT;
20330 REM HOVER TO LANDING REQUIRES THE FOLLOWING...
20340 TWRATIO = .165: ITIME = 0: FTIME = 60 : TB = FTIME - ITIME
20350 REM NOW MASS OF LUNAR LANDER IS TOTAL-PMASS DUE TO BURN UP OF
20360 REM PROPELLANT DURING REDUCTION TO HOVER...
20370 HLIMASS = RHIMASS - RHPMASS
20380 PRINT "LANDER MASS BEFORE HOVER TO LANDING IS ";HLIMASS;" KG."
20390 REM
```

```
20400 REM
20410 REM
20420 REM
20430 REM
20440 REM
20450 REM PROPELLANT MASS REQUIRED FOR VARYING THRUST TO WEIGHT RATIO...
20460 HLPMASS = HLIMASS*TWRATIO*TB/ISP
20470 PRINT "LANDER MASS UPON LANDING IS ";HLIMASS-HLPMASS;" KG."
20480 PRINT "HOVER TO LANDING REQUIRES ";HLPMASS;" KG OF PROPELLANT."
20485 MPDESCENT = GTPMASS + HLPMASS + RHPMASS
20490 PRINT "DESCENT REQUIRES ";MPDESCENT; " KILOGRAMS."
20492 PRINT "DESCENT NEEDS ";MPDESCENT*1.15; " KILOGRAMS."
20494 PRINT "PROPELLANT EXCESS FOR DESCENT PHASE IS ";MPGUESS-MPDESCENT*1.15; "
KILOGRAMS."
20500 RETURN
30000 REM ***** SUBROUTINE COMPUTES THE PROPELLANT MASS *****
30010 REM ***** CONSUMED IN LUNAR LANDER ASCENT PHASE.  *****
30020 REM
30030 REM *** BEGIN ASCENT PHASE ***
30040 REM
30050 REM IDENTIFY THRUST PARAMETERS FOR ASCENT;
30060 REM CONSTANT THRUST TO WEIGHT RATIO FOR ENTIRE DURATION...
30070 ITWRATIO = .321: FTWRATIO = .601 :ITIME = 0 : FTIME = 445 : ISP = 450
30075 SCREEN 0: TB = FTIME - ITIME : C = ISP
30080 A = (FTWRATIO-ITWRATIO)/(FTIME-ITIME): B = ITWRATIO: C = ISP
30085 DET = SQR(-B^2+4*A*C)
30090 TB = FTIME - ITIME
30096 PRINT:PRINT "LANDER MASS BEFORE ASCENT STAGE IS ";ASCENT;" KG."
30098 REM PROPELLANT MASS REQUIRED FOR CONSTANT THRUST TO WEIGHT RATIO...
30100 MPASCENT = ASCENT*ITWRATIO*TB/ISP
30110 PRINT "ASCENT PHASE REQUIRES ";MPASCENT;" KG OF PROPELLANT."
30120 PRINT "FINAL LUNAR LANDER MASS IS ";ASCENT-MPASCENT;" KG."
30140 MPACTUAL = GTPMASS + RHPMASS + HLPMASS + MPASCENT
30143 REM *** INCORPORATE 5% EACH FOR ULLAGE, BOIL-OFF, AND EMERGENCIES... ***
30145 MPNEEDED = MPACTUAL*1.15
30150 PRINT "PROPELLANT MASS USED FOR THE MISSION WAS ";MPACTUAL;" KG."
30155 PRINT "PROPELLANT MASS NEEDED FOR THE MISSION WAS ";MPNEEDED;" KG."
30160 PRINT "PROPELLANT MASS GUESSED FOR THE MISSION WAS ";MPGUESS;" KG."
30180 PRINT "PROPELLANT EXCESS IS ";MPGUESS-MPNEEDED;" KG."
30200 RETURN
```

Appendix VII

Assumptions for Propellant Mass Calculations

1. Mixture Ratio of 4:1

2. Propellant Mass Breakdown for All Mission Profiles:
   - 85 % is usable
   - 5 % for boiloff
   - 5 % for ullage
   - 5 % for contingencies

3. Specific Impulse = 450 seconds

4. Inert Mass = 13.6 Metric Tons

Maximum Lunar Lander Propellant Mass is 35 Metric Tons.

Appendix VIII: Inert Mass Statement

Item	Mass (kg)
Structure	2 000
Engines (4)	1 000
RCS Clusters (4)	1 000
Landing Gears (4)	1 000
Avionics, Radar, & Communications	1 000
Multi-layer Insulation	500
Propellant Tanks	7 000
TOTAL	13 500 kg

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