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MOONPORT:
TRANSPORTATION NODE IN LUNAR ORBIT

SUBMITTED TO:

DR. WALLACE FOWLER
DEPARTMENT OF AEROSPACE ENGINEERING AND ENGINEERING MECHANICS
THE UNIVERSITY OF TEXAS AT AUSTIN

PRESENTED BY:

SPACE PORT SYSTEMS
THE UNIVERSITY OF TEXAS AT AUSTIN

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INTRODUCTION

This document outlines the work done by SPACE PORT SYSTEMS to design an orbital transportation system between the Earth and the Moon. The design work focused on the requirements and configuration of an orbiting lunar base. Design utilized current Space Station technologies, but also focused on the specific requirements involved with a permanently manned, orbiting lunar station. A model of the recommended configuration was constructed. In order to analyze Moonport activity and requirements, a traffic model was designed, defining traffic between the lunar port, or Moonport and low Earth orbit. Also, a lunar base model was used to estimate requirements of the surface base on Moonport traffic and operations. A study was conducted to compare Moonport operations based in low lunar orbit and the L_2 equilibrium point, behind the Moon. The study compared delta-V requirements to each location and possible payload deliveries to low Earth orbit from each location. Products of the Moonport location study included number of flights annually to Moonport, net payload delivery to low Earth orbit, and Moonport storage requirements.

LIST OF ACRONYMS

ASE	Aerospace Engineering
DEC	Direct Energy Conversion
DOD	Department of Defense
DSN	Deep Space Network
ECLSS	Environmental Control/Life Support System
EM	Earth-Moon
EMF	Electromotive Force
EVA	Extravehicular Activity
GCR	Galactic Cosmic Rays
GPS	Global Positioning System
GRO	Gamma Ray Observatory
HLLV	Heavy-Lift Launch Vehicle
IOC	Initial Operation Configuration
ISEE-3	International Solar Earth Exploration
I_{sp}	Specific Impulse
IVA	Intravehicular Activity
JPL	Jet Propulsion Laboratory
KSC	Kennedy Space Center
LEO	Low-Earth Orbit
LH2	Liquid Hydrogen
LLO	Low Lunar Orbit
LV	Lunar Vehicle
LO 2	Liquid Oxygen
LSPI	Large Scale Programs Institute
LTV	Low Thrust Vehicle
MMU	Manned Manuevering Unit
MPOM	Manned Proximity Operations Module
MRMS	Mobile Remote Manipulator System
MTV	Maneuverable Television
MWRL	Morale, Welfare, Recreation, and Logistics
NASA	National Aeronautics and Space Administration
OMV	Orbital Transfer Vehicle
ORS	Orbital Refueling System
OSCRS	Orbital Spacecraft Consumables Refueling System
OTV	Orbital Transfer Vehicle
PL	Payload
POM	Proximity Operations Vehicle
PP	Propellant
REM	Roentgen Equivalent Man

RFP	Request for Proposal
RMS	Remote Manipulator System
SE	Sun-Earth
SEP	Solar Energetic Particles
SPS	Space Port Systems
SRB	Solid Rocket Booster
SSME	Space Shuttle Main Engines
STBE	Space Transportation Booster Engine
TDAS	Tracking and Data Acquisition System
TDRS	Tracking and Data Relay Satellite System
TMS	Teleoperator Manuevering System
TPS	Thermal Protection System
UPOM	Unmanned Proximity Operations Module
WBCV	Winged Booster Cargo Vehicle

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1.0 PROJECT OVERVIEW

In accordance with RFP# ASE274, Space Port Systems (SPS) has conducted a preliminary design study of a transportation node in close lunar proximity. This lunar space station, or Moonport will be located in low lunar orbit (LLO).

After construction of an Earth-orbiting space station, the next focus of space development should be the construction of a base on the lunar surface. This permanently manned lunar base would process the lunar soil into useful products, including oxygen and silicon. These products could then be delivered into low Earth orbit (LEO), to be used for vehicle fueling and construction of space structures. In order to construct and operate this lunar base, transportation between the Earth and the Moon must become safe and efficient.

One possible means of addressing to this transportation problem is the creation of Moonport, a vehicle transportation node near the Moon. Vehicles from the Earth can dock with this station and deliver payloads and personnel bound for the lunar surface. Also, when the lunar base starts to produce products, lunar vehicles can deposit payloads at Moonport, to be stored until an Earth-bound ship is ready to deliver them to low Earth orbit (LEO). Moonport will have vehicle servicing and refueling facilities and storage facilities for vehicle payloads. Also, Moonport will have habitation facilities to house crewmen. Moonport can be man-tended for short periods of time early in lunar base development. Later, as traffic to Moonport increases, the port can become permanently manned.

As an overview to the design project, several topics will be discussed. Section 1.1 presents a description of the tasks which SPS has performed during the contract period. Section 1.2 specifies the work that has been done by the Traffic Analysis Division. An overview of Moonport design is reviewed in section 1.3.

1.1 DESIGN TASKS

In order to accomplish design goals within the contract period, the original RFP tasks have been reduced. As described in the contract proposal (SPS DOC#1), the SPS design effort was divided into three main tasks:

1) definition of traffic model for Moonport operations
2) definition of preliminary requirements of Moonport subsystems
3) a design of a preliminary Moonport configuration

1.2 TRAFFIC MODEL DEFINITION

A preliminary traffic model analysis has been conducted to compare LLO and L_2 as Moonport locations during steady state traffic, and to provide estimates of fuel and payload sizes for Moonport subsystem design. The traffic model produces the required number of flights and net payload delivered at LEO using a transportation system based entirely on lunar LO2. In addition, key factors effecting the net payload return to LEO were studied. The Large Scale Programs Institute (LSPI) Lunar Base Model was used to provide an estimate for Lunar Base resupply mass and vehicle masses.

1.3 MOONPORT DESIGN

In order to complete design tasks, the expected functions of Moonport were determined. The Moonport will serve as:

1) a transportation node, handling personnel and cargo

2) a platform to support lunar base construction and expansion
3) a storage depot for materials going to and from the Moon
4) a vehicle servicing facility
5) a foundation for future missions, including interplanetary missions and expanded lunar exploration

The final products of the SPS port design include a list of requirements for subsystems associated with port functions and a preliminary Moonport configuration.

This document also describes and defines several Moonport subsystems. The areas that have been developed include:

1) port configuration
2) cargo storage facilities
3) vehicle servicing facilities
4) habitation
5) radiation protection
6) electrical power supply (including heat rejection)
7) low thrust vehicle (LTV) requirements

A study of subsystem requirements has been conducted, and a preliminary Moonport configuration has been designed.

SECTION 2

DESIGN GUIDELINES AND ASSUMPTIONS

2.1 REFERENCE MISSION SCENARIO

2.2 VEHICLES

2.3 THE LUNAR BASE MODEL

2.0 DESIGN GUIDELINES AND ASSUMPTIONS

In order to accomplish the required tasks in a timely fashion, design guidelines and assumptions have been defined. These guidelines have been used during the design of Moonport subsystems and operations, and have helped coordinate the port development and traffic analysis studies. Section 2.1 describes the reference mission scenario for port construction and deployment. In Section 2.2 a description of the vehicles that have been used for the design study is presented. The assumed lunar base and its vehicle traffic requirements are addressed in Section 2.3.

2.1 REFERENCE MISSION SCENARIO

To coordinate the design efforts of the port development and trajectory analysis divisions, a reference mission scenario has been constructed. This scenario describes the assumptions used to design the general Moonport construction and deployment mission. Key items of the mission scenario are:

1) construction of Moonport in low-Earth orbit
2) delivery of Moonport into low lunar orbit via low-thrust vehicle (LTV)
3) final placement of Moonport

2.1.1 CONSTRUCTION OF MOONPORT IN LEO

A variety of mission considerations makes construction in LEO a desirable option. First, the radiation protection afforded by the Earth's atmosphere and radiation belt will greatly increase the safety of crewmen during extravehicular activity (EVA). Due to the large amounts of radiation, any construction in low lunar orbit would be accomplished in short shifts, and at great risk to human life. Second, it is assumed that at the time of Moonport construction, vehicles will be available to transfer materials and crews to the construction site. In addition, an Earth-orbiting transportation node, Earthport, is assumed to exist, and will be used as a base to house the crew and store materials. Third, the close proximity to Earth will allow the possibility of rescue by the space shuttle or some equivalent system in case of emergency. If a large amount of early construction were attempted in lunar orbit, the difficulty of rescue would be significantly increased.

2.1.2 DELIVERY OF MOONPORT TO LUNAR ORBIT VIA LTV

After port construction in LEO, an LTV will be used to transport Moonport on a long, spiral trajectory to the desired lunar orbit. Options for integration of Moonport with the LTV are discussed in Chapters 3 and 6.

2.1.3 FINAL MOONPORT LOCATION

As described previously, the final location for Moonport was chosen to be LLO, not the L_2 point. Each location offered advantages and disadvantages to steady-state port operations. For a comparison study, trajectory data to both LLO and L_2 were computed. A study of perturbations in LLO has also been done.

2.2 VEHICLES

Due to the time constraints associated with this project, no new vehicle configurations have been designed. Current and projected vehicle configurations have been used for all aspects of the mission. Some vehicle specifications have been modified to accommodate design requirements. This section presents general vehicle configurations. A breakdown of payloads for the orbital transfer vehicles and the lunar vehicles is given in Section 4.

2.2.1 LAUNCH VEHICLES

The transfer of small payloads and personnel into LEO will be accomplished using the current configuration of the space shuttle, assuming a fleet of five orbiters during the construction phase. The baseline mission is five days with a five person crew.

Because of space shuttle constraints, a method of lifting larger and heavier payloads into LEO is desired. A current joint NASA/DOD Space Transportation Architecture Study indicates that a partially reusable vehicle, the Winged Booster Cargo Vehicle, with a payload capacity of 45.4 metric tons (100 kips) to 68.0 metric tons (150 kips) is economically effective when compared to expendable vehicles and fully reusable vehicles. Such a vehicle, currently under study at Marshall Space Flight Center,[1] will be used for delivery of materials during Moonport construction in LEO.

2.2.2 ORBITAL TRANSFER VEHICLES (OTV)

In order to efficiently transfer cargo and personnel between Earthport and Moonport, a reliable, reusable, cost effective Orbital Transfer Vehicle (OTV) must be used. Since minimizing fuel consumption is a primary concern in any vehicle design, SPS has decided to use an aerobraking design rather than an all-propulsive design. The aerobraking concept utilizes the Earth's atmosphere to slow the OTV and modify its trajectory.

2.2.2.1 OTV CONFIGURATION

The basic design of the OTV is shown in Figure 2.1[2]. The OTV is made of two major sub-assemblies, the aerobraking surface and the Propulsion/Avionics unit. Both units can be carried into orbit by the current space shuttle and assembled at Earthport. In its shuttle-loaded configuration, the aerobraking surface is folded to save space, and unfolded when assembled to the propulsion unit. Once unfolded, the aerobraking surface is never folded back to its original position.

A single OTV is approximately 13.4 m (44 feet) in diameter (at the aerobraking surface), and 10.7 m (35 feet) long (Figure 2.1). The flexibility of the design allows the OTV to be staged for larger payloads. Taking this into consideration, the total maximum length of a two-staged OTV, with a maximum payload (whose length is equal to 18.3 m [60 feet]) , is just under 41.1 m (135 feet). This maximum length must be considered when designing a hangar for the OTV at both Earthport and Moonport.

The configuration of the fuel tanks allows refueling of the OTV in one of two ways. The primary refueling technique will be to fill the fuel tanks while they are still attached to the OTV; however, since the fuel tanks are readily accessible, they can be removed and replaced by full fuel tanks.

2.2.2.2 SPACE-BASED OTV MISSION

Initially, the OTV is sent to Earthport as major sub-assemblies that can be delivered into orbit by the space shuttle. The OTV is hangared at Earthport, assembled , and loaded with its payload. An Orbital Maneuvering Vehicle (OMV) is used to transfer the OTV (with or without a payload) to and from the hangar at either Earthport or Moonport. Once clear of the port, the OTV begins its flight to its intended destination.

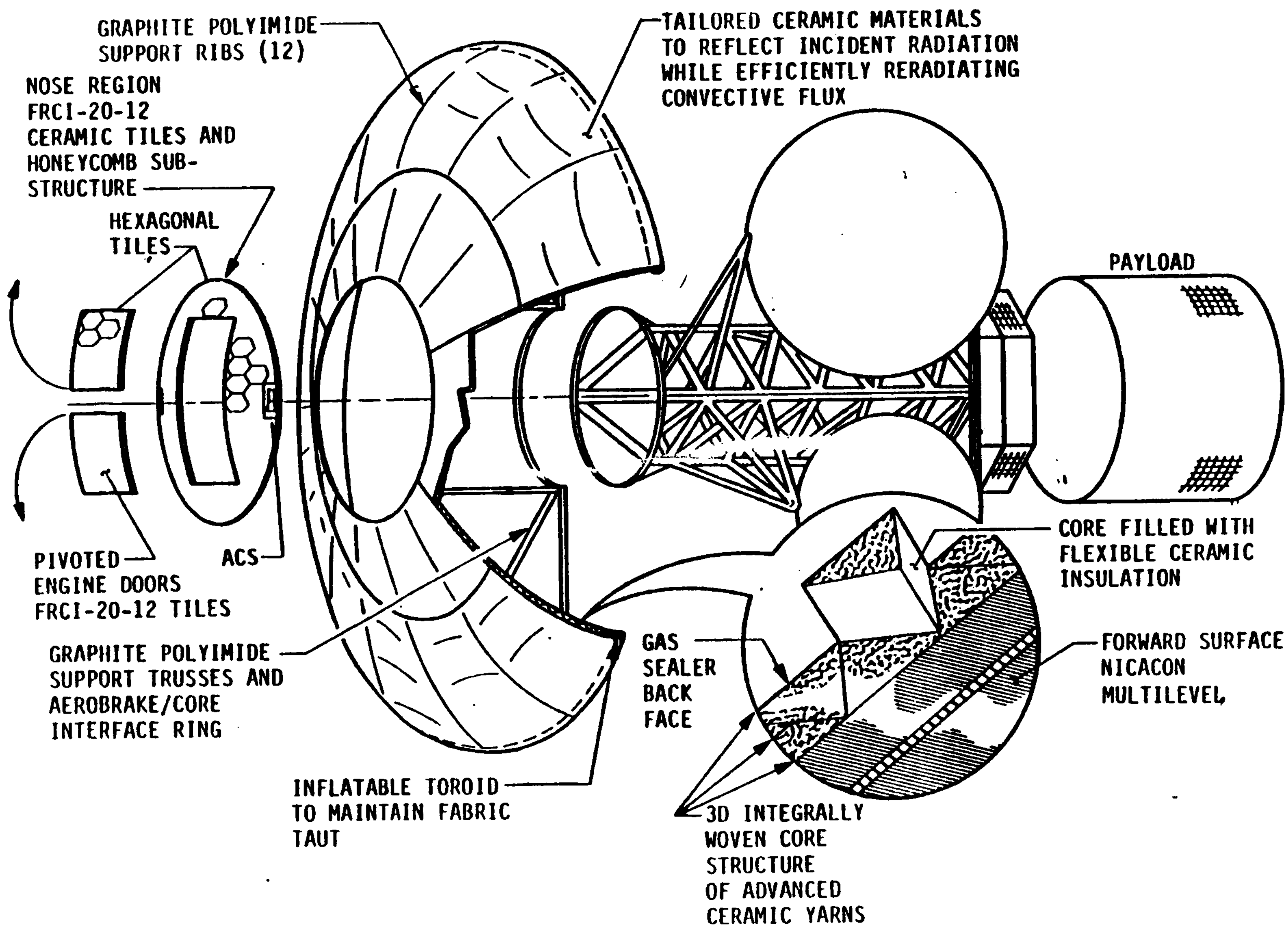

FIGURE 2.1 - MARTIN MARIETTA RIGID-FLEX OTV [2]

If the mission is to originate at Moonport, the OTV and payload are transferred from Moonport by means of an OMV. The OTV performs a burn to a transfer orbit which will bring it to a suitable altitude for completing the aerobraking maneuver. After the aerobraking maneuver is completed , it performs a circularization burn into an OMV compatible orbit. Once docked with the OMV, the OMV / OTV 'stack' is stored at Earthport for refueling and maintenance.

2.2.3 ORBITAL MANEUVERING VEHICLE

The Orbital Maneuvering Vehicle, is a high-thrust, limited-range vehicle which is based at Earthport and Moonport. Its main duty is to retrieve the OTV or other payload from an Earth-Moon transfer orbit, and guide it back to a port hangar.

2.2.4 LUNAR VEHICLES (LV)

Two types of LV's are be used for transportation between LLO, L_2, and the lunar surface. All the vehicles have a baseline LO2-LH2 engine with an LO2/LH2 ratio of 7. The first type of LV is a manned reusable vehicle for crew transportation. The manned LV is used to transport a crew of 4 to 6 people from LLO or L_2 to the lunar surface. It consists of a pressurized cylindrical vessel. This vehicle is shown in Figure 2.2[3].

The second lunar vehicle is a larger scale vehicle designed to accommodate both cargo and crew vehicle. The vehicle's primary cargo is LO_2, which can be stored either in integral tanks or attachable modules. The habitation module to accommodate a crew of 6 to 8 people is attached above the propulsion system. The cargo can be placed next to the crew module[4]. This vehicle's configuration is shown in Figure 2.3[5].

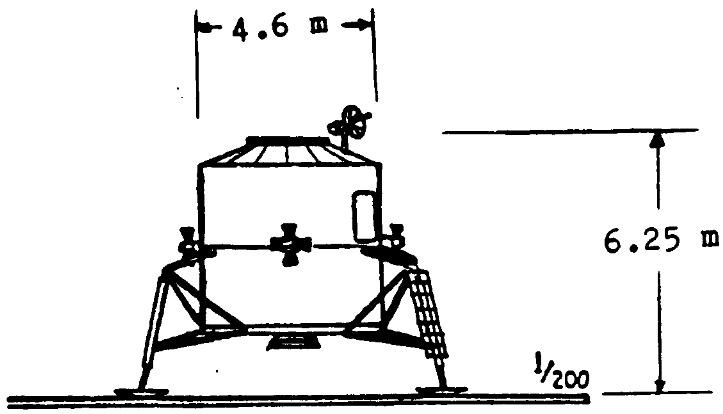

FIGURE 2.2 - REUSABLE MANNED LUNAR VEHICLE[3]

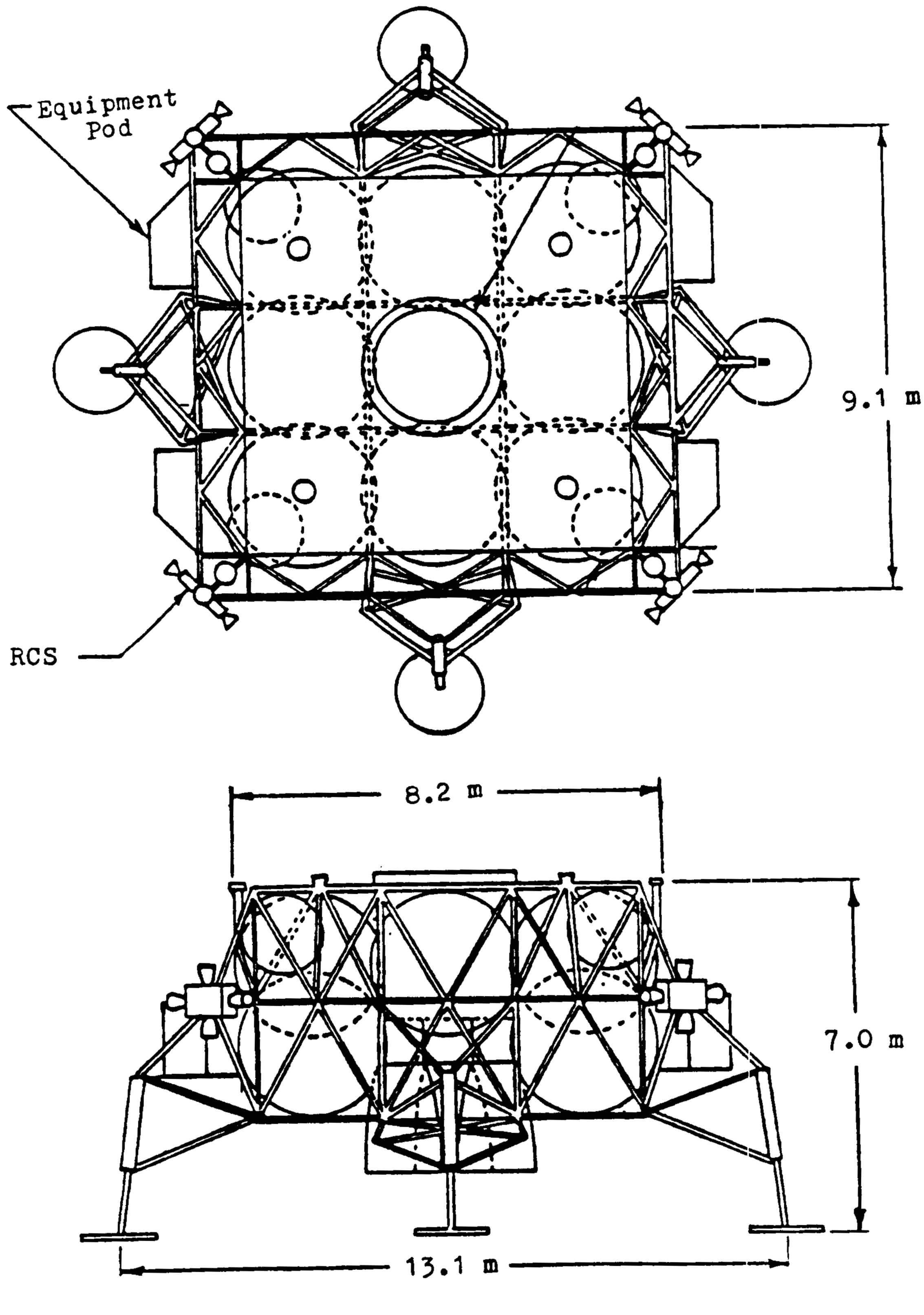

FIGURE 2.3 - CARGO/CREW LUNAR VEHICLE [5]

2.3 THE LUNAR BASE MODEL

Since the primary purpose of Moonport is to support lunar base construction and operations, a lunar base model had to be assumed. The model that has been used comes from the LSPI. This model uses the Lotus Symphony™ database management system as a foundation for a program which determines the base requirements from a set of user inputs. These inputs include the amount of resource exports from the lunar base per month and the size of the scientific team , if any, at the lunar base. The model produces a detailed list of base requirements. Requirement data used for design analysis include base subsystem masses and construction data.

2.3.1 LUNAR BASE INPUTS

These data are used by the lunar base model to calculate the specific type of base to be modeled, along with its associated production and requirements. There are more choices for resource exports from the model. Some of these choices include aluminum, iron , and steel. The size and composition of the scientific crew and resource exports are listed in Tables 2.1a and 2.1b.

TABLE 2.1a SCIENTIFIC CREW

Scientific Personnel	Persons
Astronomy	2
Physics	1
Surface Science	2
Other	2
TOTAL	7

TABLE 2.1b RESOURCE EXPORTS

Resource	Exports (MT/mo)
Oxygen	100.0
Silicon	20.0
Glasses	50.0
Shielding	150.0
TOTAL	320.0

2.3.2 LUNAR BASE OUTPUTS

The mature base subsystem mass breakdown is contained in Table 2.2. Base housing, mission habitats, and resource processing equipment contribute more than two-thirds of the total base mass of 1503 MT (3314 kilo-lbm). The majority of the resupply for the base is for the purpose of replenishing human and mission consumables at quantities of 62 and 277 MT/yr (137 and 611 kilo-lbm/yr), respectively. The remaining 26 MT/yr (57 kilo-lbm/yr) is to resupply lunar base hardware.

In addition to the mass and resupply breakdown, the model determines the total number of personnel for the mature lunar base. Twenty-three additional crew members are needed as support personnel to maintain the lunar base, to mine and process lunar resources, and to aid scientific personnel. The base crew (30 persons total) is housed in 17 habitation modules. The total electrical power output needed to support this lunar base model is 1.2 MW.

According to model assumptions, lunar base construction will last 19.2 months. Prior to the arrival of the base construction crew, construction equipment and approximately 25% of the lunar base material will be pre-placed on the Moon. The construction equipment will be carried along with Moonport on the LTV. It has a mass of 51 MT (112 kilo-lbm). A construction crew of fourteen persons is used to build the base, consisting of two construction engineers, four riggers and mechanical technicians, two electricians, two pipe/instrument fitters, and four operating engineers. Each crew member needs 3.184 MT (7.019 kilo-lbm) of consumables per month, yielding a total of 44.6 MT/mo (98.3 kilo-lbm/mo) of resupply materials.

TABLE 2.2 MATURE LUNAR BASE MASS ESTIMATES

Area	Mass (MT)	Resupply (MT/yr)
Housing and Mission Habitats	536.6	8.0
Central Powerplant, 1.25 MW	41.6	0.6
Power Control System, 1.25MW	106.0	1.6
Central Radiator, 11.43 MW peak	127.0	1.9
Thermal Control System, 11.43 MW	45.7	not avail.
Science Equipment	80.0	9.0
Resources Equipment	522.5	3.7
Maintenance Equipment at 3.00%	43.8	1.3
Human Needs		62.0
Mission Needs		215.0
TOTAL	1503	303.1

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SECTION 2 REFERENCES: GROUND RULES AND ASSUMPTIONS

1. Billy Shelton, "Advanced Heavy Lift Launch Vehicles," AAS Preprint of Paper for AAS Annual Meeting October 26-29, 1986. San Diego: AAS Publications Office, 1986.

2. Martin Marietta, "Orbital Transfer Vehicle: Concept Definition and Analysis Study, August 1985". Denver, Colorado.

3. University of Michigan Design Group, " Lustar Report, 1982," pp. 2.

4. Large Scale Programs Institute, " A Demonstrative Model of a Lunar Base Simulation on a Personal Computer, " 1986. p. 76.

5. University of Illinois Design Group, " Development of a Lunar Oxygen Transportation System Utilizing an Orbital Transfer Vehicle, Lunar Transfer Vehicle, and Lunar Service Station, 1986," p. 229.

SECTION 3

EARTHPORT/MOONPORT LOCATION

3.0 EARTHPORT/MOONPORT LOCATION

The Earthport altitude and inclination was established to determine a regression rate for the LEO. This rate was used in conjunction with nodal precession rates about the Moon to establish launch opportunities between the Earth and the Moon. In addition both low lunar orbits and the halo oribt about L_2 were studied to characterize the advantages and disadvantages of the choice for port locations.

3.1 EARTHPORT LOCATION

The primary task in establishing a location for the Earthport is to determine the rotation rate of Earth's orbit. Regression rate of LEO is a function of altitude and inclination.

Altitude is the primary factor that determines port lifetime and propulsion requirements for drag compensation. These requirements, in turn, depend on the atmospheric density, the Earthport's velocity, mass, aerodynamic and geometric characteristics, and drag compensation propulsion system characteristics.

The Van Allen radiation belts are one of the major orbit determining factors. High-energy charged particles such as protons and electrons are trapped by an electromagnetic field and form the radiation belts in space. Earthport orbit has to be located below approximately 560 km (300 nmi) to avoid radiation-induced injury to both personnel and damage to spacecraft equipment.

Space shuttle launch performance is dependent upon launch mode (either Nominal or Direct Ascent), as well as target orbit altitude and inclination. The space shuttle orbiter cannot reach orbits above approximately 370 km (200 nmi) without a Direct Insertion Ascent. Even though other vehicles such as heavy-lift launch vehicles can be used, launch energy requirements grow as the orbit is shifted to higher altitudes.

The decay of a spacecraft from a circular orbit depends on the spacecraft ballistic coefficient B, defined by

$$B = m/(C_D A)$$

where m is the spacecraft mass, C_D is the averaged drag coefficient, and A is a reference area for the drag coefficient. The drag coefficient depends on many variables - atmospheric composition, mean free path, density, material, and the shape of the spacecraft. Its value depends not only on the spacecraft's altitude but on its physical design characteristics. C_D is varied between 2.0 and 3.0, and is estimated as 2.2 + .3 This value was found to be nearly constant for altitudes below 400 km (low solar activity) to 600 km (high solar activity). Recent proposals for the space station have ballistic coefficients in the 31 to 61 kg/m^2 (0.2 to 0.4 $slug/ft^2$) range, which is small compared with 182 kg/m^2 (1.2 $slug/ft^2$) value of the Skylab. Figure 9.1 shows the yearly propellant required for a space station to maintain orbit[1].

A safety standard that has been considered for the space station is a minimum lifetime requirement of 90 days, assuming a complete failure of the drag compensation propulsion system. Below 330 km (178 nmi), the station can never attain a 90-day minimum lifetime. It is notable from the figures that the amount of propellant required to provide drag-compensation for the space station is small. Higher altitude orbits require even less. Thus, it is not the drag-compensation propulsion fuel requirements that have the greatest impact on Earthport orbit selection, but rather the 90-day minimum lifetime.

Figure 3.2 shows overall constraints in determining the Earthport orbit. From this figure, orbits between 460 km (250 nmi) and 540 km (290 nmi) are considered to be suitable for the Earthport orbit altitude. A 28.5^{o} inclination 486 km (260 nm) altitude orbit can be achieved into once a day from KSC. It has a phasing time of 11.9 hours.

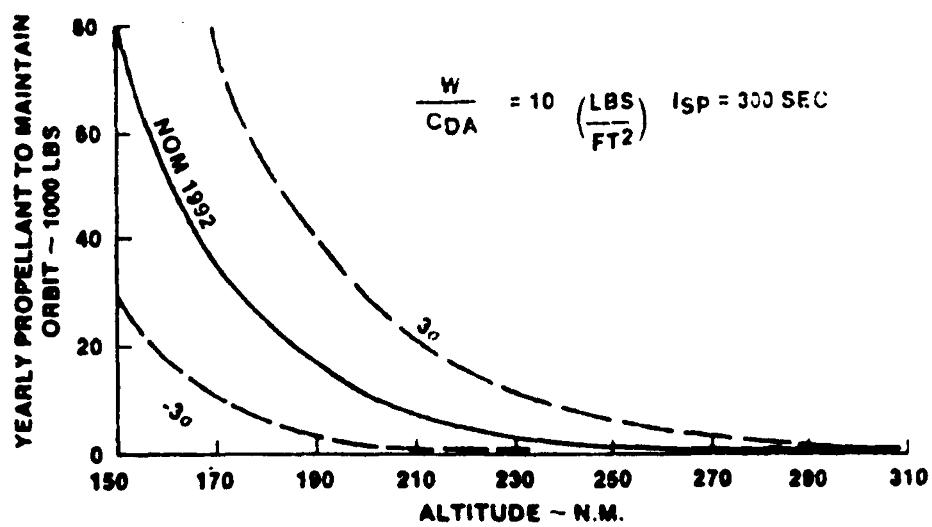

Figure 3.1 Propellant Required to maintain orbits[1]

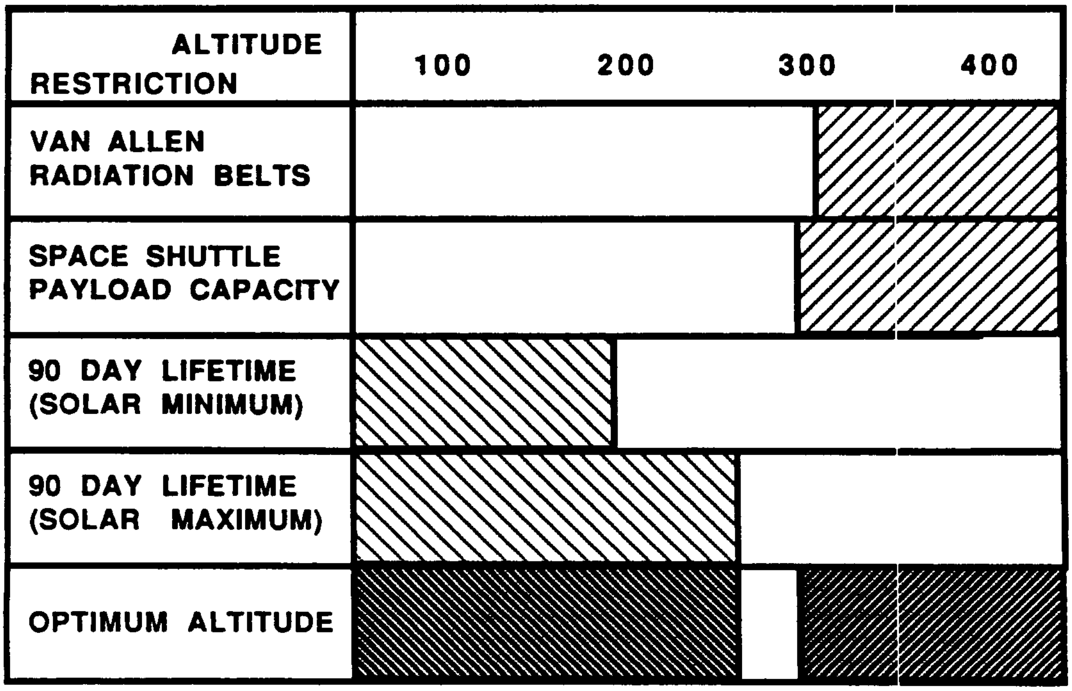

Figure 3.2 Earthport Altitude Constraints

The Moon's orbital plane variation and the possible landing site place restrictions on the Earthport orbit inclination. The Moon's orbit is inclined to the ecliptic by approximately 5.15°. The line of Earth-Moon node, which is the intersection of the Moon's orbital plane with the ecliptic, rotates westward, making one complete revolution in 18.6 years. As depicted in Figure 3.3, the inclination of the Moon's orbit relative to the equator varies between 18.15 and 28.75°. Therefore, the Earthport orbit has to be in this range of inclinations to minimize the plane change propellant requirement. A LEO inclination of 23° would only require a maximum plane change of 5° instead of the 10° required by 28.5° inclined orbit.

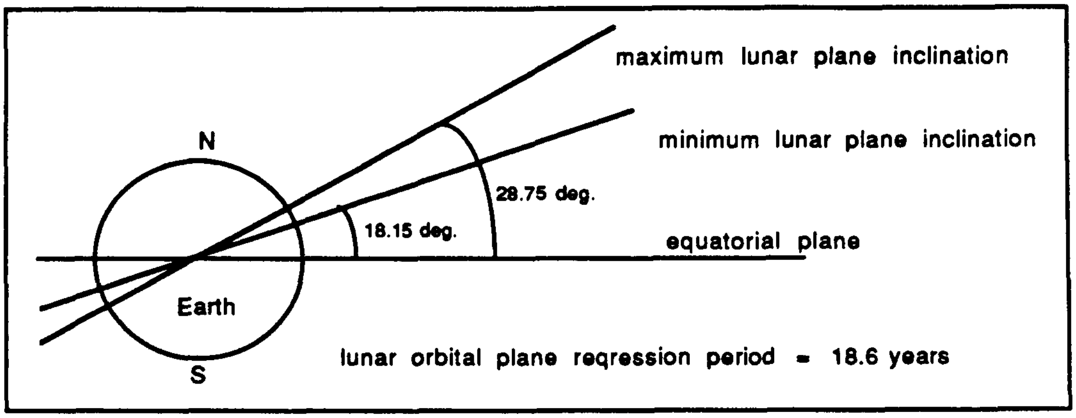

Figure 3.3 Moon's Orbital Plane Change

The space shuttle has crossrange capability of approximately 750 nautical miles or about 12° in latitude. If Edwards Air Force Base (34.9° N, 117.8° W) is considered as a possible landing site, the Earthport orbit inclination has to be higher than 23°. From the Kennedy Space Center (28.5° N, 80.5° W), placing Earthport in an orbit lower than 28.5° will impose severe launch vehicle plane-change penalties. If low latitude places such as Johnston Island (17° N) and Hawaii (19° N) are available as possible launch sites, then a 23° inclination orbit will be optimum considering Earth-Moon orbital plane relations. A orbit with an altitude of 260 nautical miles and a 28.5° inclination was used to study the synodic period of LEO and LLO orbits.

3.2 MOONPORT LOCATION

The Traffic Analysis Division evaluated two locations for the Moonport. A low lunar orbit will be used until the close of the lunar base construction phase of development. Once all necessary materials have been transported to the surface and the construction phase of the lunar base is complete, the port may be moved to a halo orbit centered on the L_2 Lagrangian point, located at a mean distance of 64,500 kilometers from the Moon on the side opposite the Earth. The detailed characteristics of the L_2 point are outlined in Section 3.3.

3.3 LOW LUNAR ORBIT

The primary advantage of a low lunar orbit is the low ΔV required to travel to and from the lunar surface. This only applies to surface destinations that are below the orbit. A LLO port location also provides the option for a return trajectory in which the OTV is capable of returning to LEO without additional burns in the vicinity of the Moon. A free-return trajectory will fly around the moon at an altitude of approximately 100 km; therefore, an LLO at this altitude is preferred.

During the initial phases of lunar base construction, Moonport will be situated in a low lunar orbit to act as a transportation hub for the transfer of construction materials, equipment, and personnel to and from the lunar surface. Several types of lunar orbits were considered for the initial location of the lunar port. These included retrograde equatorial, polar, and retrograde low inclination orbits with high nodal progression rates. The primary consideration in selecting an LLO orbit was accessibility to trajectories to and from LEO.

An accurate model may include the perturbing effects of the Earth, Sun, and planets; solar radiation pressure; and the perturbing effects due to the asymmetry of the Moon's gravitational field. Past analyses of this problem have demonstrated that the effects of the planets can be neglected. The Earth and Sun's perturbing influence were included in this model. The effects of solar radiation pressure were not included in this analysis in order to simplify the model.

The mathematical formulation of this problem was based on an n-body approximation which included the lunar gravitational potential function as the main perturbing force. The Earth and Sun were modeled as point masses and the lunar gravitational potential function was modeled out to its second order term. The formulation also included all the necessary transformations between geocentric lunar coordinates to selenocentric inertial coordinates, heliocentric Earth-Moon barycenter coordinates to selenocentric inertial coordinates, and from satellite selenocentric inertial coordinates to selenocentric body-fixed coordinates.[2] Numerical integration of the six first-order ordinary differential equations characterizing the lunar orbits was done using a Runge-Kutta-Hull fourth-order method with an embedded second-order method for automatic stepsize selection. The results of this analysis produced nodal progression rates for various inclinations and altitudes. These rates were a key element in determining the launch opportunities between low lunar orbit and low Earth orbit.

TABLE 3.1 - NODAL PROGRESSION OF LLO (DEG/DAY)

	INCLINATION (DEG.)		
ALTITUDE (KM)	179.5	150	90.5
50	1.761	1.196	0.00876
100	1.607	1.089	0.00864
200	1.287	0.912	0.00834
400	0.511	0.661	0.00773

Using the nodal regression rate for the selected LEO orbit and the precession rate for an LLO, a synchronous orbit was determined using a TK! Solver model. A synchronized ratio of 3:2 was possible if the lunar orbit regressed approximately -0.661°/day. The 3:2 ratio requires that the LEO ascending node completes 3 revolutions for every 2 revolutions of the LLO. The 3:2 ratio of the nodal rotation rates is with respect to the rotating Earth-Moon line, which has an angular velocity of 13°/day. In order to transform inertial nodal rotation rates to rates with respect to the Earth-Moon line, 13°/day must be added to the nodal rotation rate of both LEO and LLO orbits. The 3:2 ratio provided for an alignment of the orbital nodes every 52 days. The long synodic period of the two orbital nodes and the requirement that the LLO be regressing restricted the LLO to an equatorial orientation. The regression of the LLO node requires the orbit to be posigrade, an orientation not achievable by a free-return trajectory. An equatorial LLO is readily accessible from LEO since it lies in the Moon's orbital plane and no additional ΔV is needed for LLO plane changes[3]. An LLO equatorial orbit with an altitude of 100 km was recommended and is used in determining the traffic model.

3.4 L_2 HALO ORBIT

There are several factors to consider in determining the best permanent location for Moonport. An ideal location would be a spot that can be reached from LEO at every launch opportunity, that requires little or no fuel to stay in place, and that has an flexible launch window. The regression of line of nodes makes possible to launch to the Moon every 9 days. Since L_2 maintains a fixed position to the Moon it is also accessible every 9 days.

Because the Earth and Moon are not fixed in space, there are five equilibrium points (Lagrangian or libration points) where the gravitational attraction of Earth is balanced by the attraction of the Moon. In the Earth-Moon system, the L_2 Lagrangian point is the most favorable location for a transportation node. The L_1 Lagrangian point is another candidate, but since the ability to use a lunar-gravity-assist from LEO is lost and there is no significant difference in ΔVs to reach either point, the L_2 location was preferred over L_1[4]. Figure 3.4 shows the geometry of an L_2 halo orbit. The mean distance between the Earth and the Moon is 384,400 km (238,855 miles), and between the Moon and L_2 is 64,500 km (20,728 miles). The radius of the halo orbit is 3500 km (2175 miles), and the orbit period is approximately 15 days[5].

The slow velocity of a spacecraft in a halo orbit ($\approx$ 20 m/sec) allows inexpensive plane changes to be made at L_2. This allows the Moon to be approached at any geometry, thereby providing easy access to the Moon's surface. An L_2 halo orbit can be maintained with almost negligible propellant requirements. This has been verified experimentally at the analogous Sun-Earth (SE) L_1 point with the International Sun Earth Explorer (ISEE-3) satellite launched in 1978. The ISEE-3 was maintained in a halo orbit about L_1(SE) for four years with a station-keeping ΔV expenditure of 10 m/s per year (32.8 ft/s per year).[6]

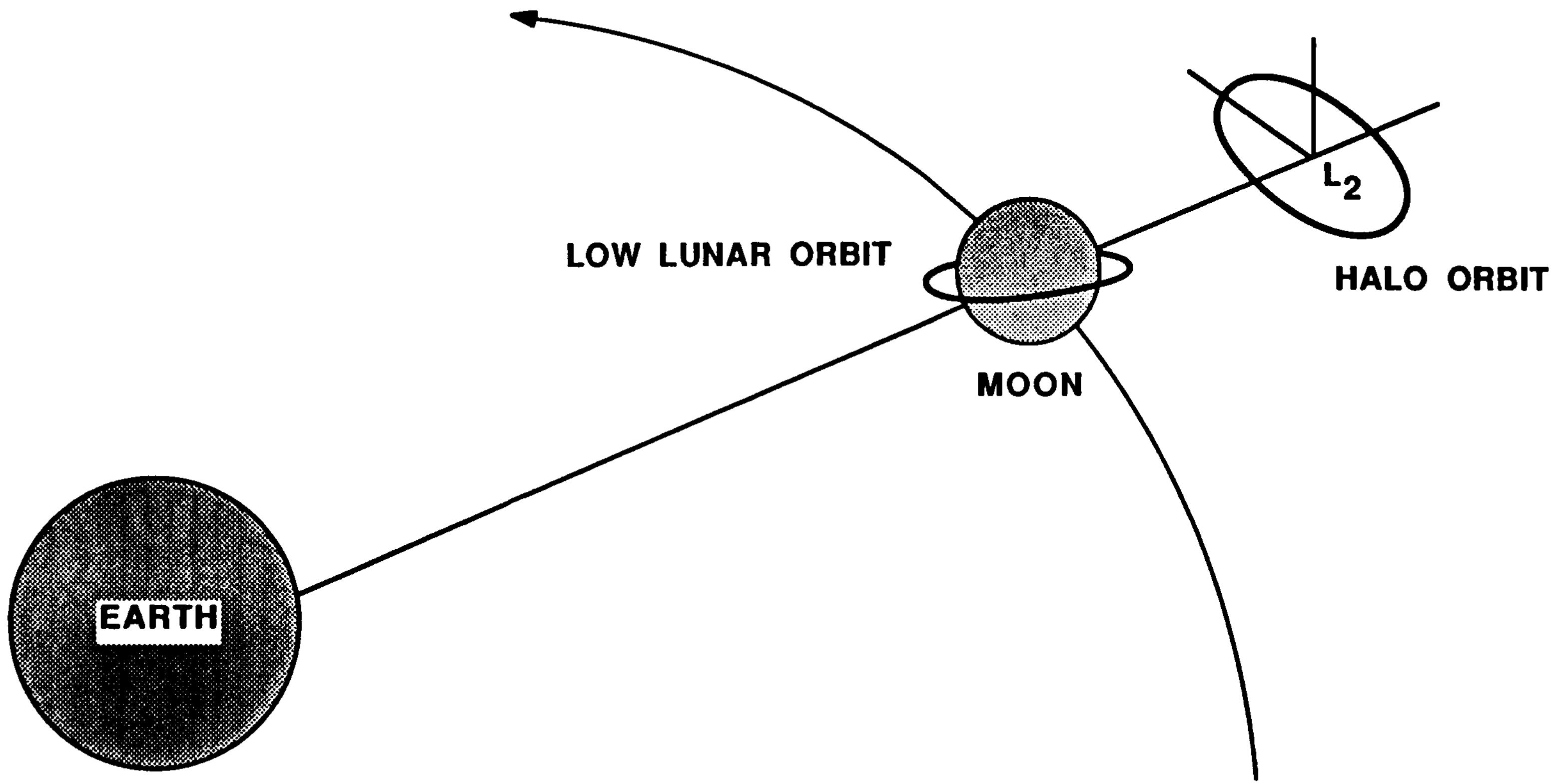

FIGURE 3.4 POSSIBLE MOONPORT LOCATIONS

A L_2 halo orbit also offers continuous line-of-sight contact with the Earth and the Moon's far side. Placement of a single relay satellite in a halo oribt about the Earth-Moon L_1 would allow the port in halo orbit to have continuous communication with almost every point on the Moon or in orbit about it. This type of communications network offers the additional advantage of being stationary with respect to the lunar surface. Earth stations currently in operations can also cover the near-side of the lunar surface.

The station-keeping ΔV costs for the two orbits are almost equal - approximately 122 m/s per year (400 ft/s per year), but the halo orbit has an additional advantage. Moonport could remain in the vicinity of the L_2 point at a cost of only 31 m/s per year (100 ft/s per year).[7]

The final advantage of the L_2 halo orbit has already been addressed - the ease of initiating interplanetary missions from this orbit. This is a long-term advantage, since the immediate responsibility of the port is to support lunar surface operations. It should not be ignored, however, considering the possibilities it offers for future space travel.[8]

SECTION 3 REFERENCE:
EARTHPORT/MOONPORT LOCATION

1. International Cooperation and Space Missions. New York: AIAA, 1984

2. Hawkins, David.Characteristics of Near Circular Lunar Satellite Orbits, Austin: Thesis, 1966.

3. Marshall Space Flight Center,Lunar Flight Handbook, Part 3, 1963.

4. Advanced Space Transportation Systems (ASTS) Study - Final Presentation on Contract No. NAS9-17335, ADL Ref. 54104, Arthur D. Little, Inc., Cambridge, Massachussetts, 1987.

5. Farquhar, Robert W., "A Halo-Orbit Lunar Station," Astronautics & Aeronautics, June 1972, pp. 59-63.

6. Ibid.

7. bid.

8. Ibid.

SECTION 4

TRAFFIC MODEL ANALYSIS

4.0 TRAFFIC MODEL ANALYSIS

The primary function of the traffic model analysis was to compare locations for the Moonport and to determine the transportation support requirement the Moonport would have to fulfil. The transportation system consists of OTV's that transport cargo and personnel between LEO and the Moonport and LV's that operate between the Moonport and the lunar surface. The traffic analysis was used to estimate the net lunar resource delivered to LEO and the Earth launched mass required to support the transportation scenario. The goal of the transportation system is to be able to deliver more lunar resource to LEO than the required Earth launched mass to support such a transportation system. The payload delivered to LEO is comprised primarily of lunar-derived liquid oxygen and other lunar minerals. The oxygen can be utilized to fuel other planetary missions.

4.1 VEHICLE ASSUMPTIONS

The initial mass of all the vehicles is based on the values used in the LSPI Lunar Base model. These values were used because they are based on previous transportation studies[1]. The Δ V's used are based on a transportation study conducted by the Arther D. Little (ADL) Corporation[2]. Using the ΔV's and the initial vehicle mass the propellant mass was determined using the ideal rocket equation, with oxygen/hydrogen ratio (LO2/LH2) was assumed to be 7 and an Isp of 460 sec. The structure mass was taken to be 20% of the empty mass, and the payload support structure was assumed to be 15% of the empty mass[3]. The structure mass includes the propulsion system, excluding the propellant, and the associated support structure. The empty mass is initial wet mass (m_i) minus the mass of the propellant (m_p). Using the equations below the payload is 68 % of the empty mass.

$$R = m_p/m_i = 1 - \exp(-\Delta v/v_e)$$

$$m_p = R * m_i$$

$$m_{empty} = m_i - m_p$$

$$m_{cargo} = 0.8 * m_{empty}$$

$$m_{payload} = 0.85\ m_{cargo}$$

where

m_i = initial wet mass

m_p = mass of propellant

The Δv's and the associated payloads are shown in Figures 4.1 and 4.2 for Moonport locations at LLO and L_2 respectively. The ΔV's to LLO assumed the Moon to be in a circular orbit about the Earth. The ΔV's to LLO were based on a patched conic model with the time of flight approximating a free-return trajectory. The ΔV's to L_2 were based on a elliptical transfer orbit and the ADL report included an additional 2% ΔV for losses and midcourse corrections for all trajectories[4]. A vehicle lifetime of 25 fully loaded flights was assumed.

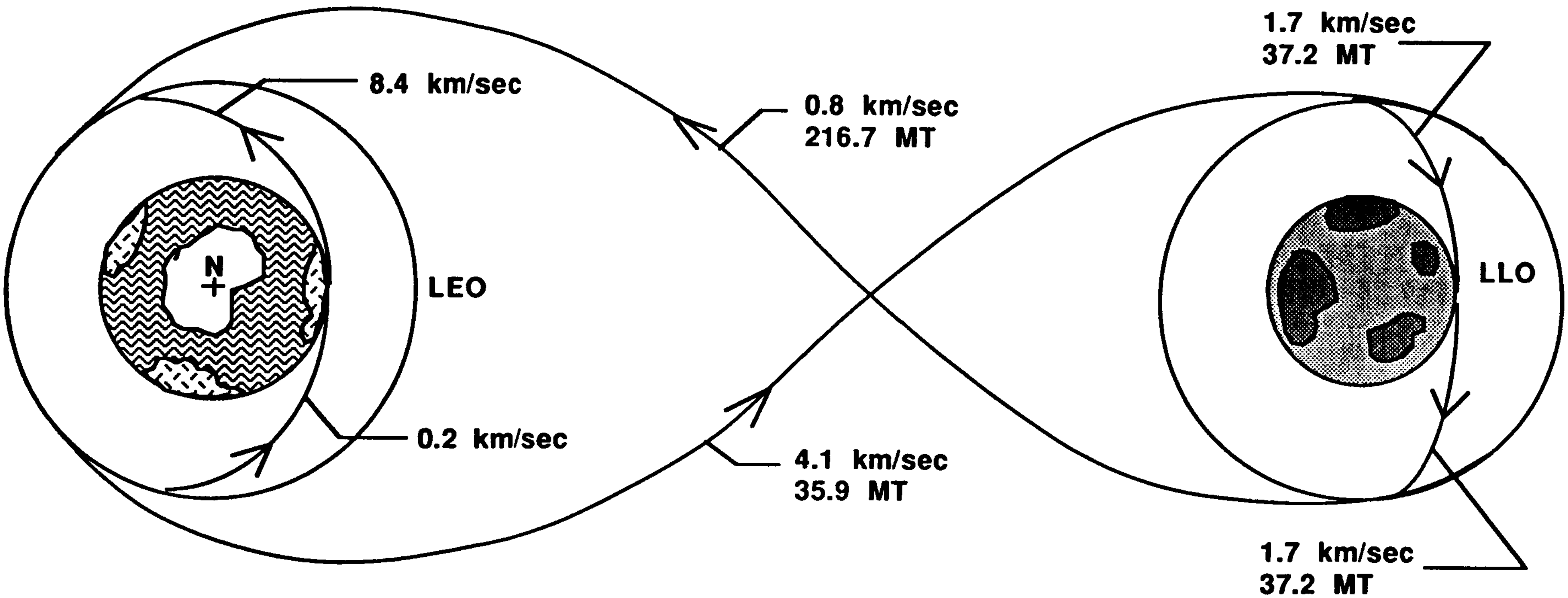

FIGURE 4.1 LLO ΔV's AND PAYLOAD CAPACITY

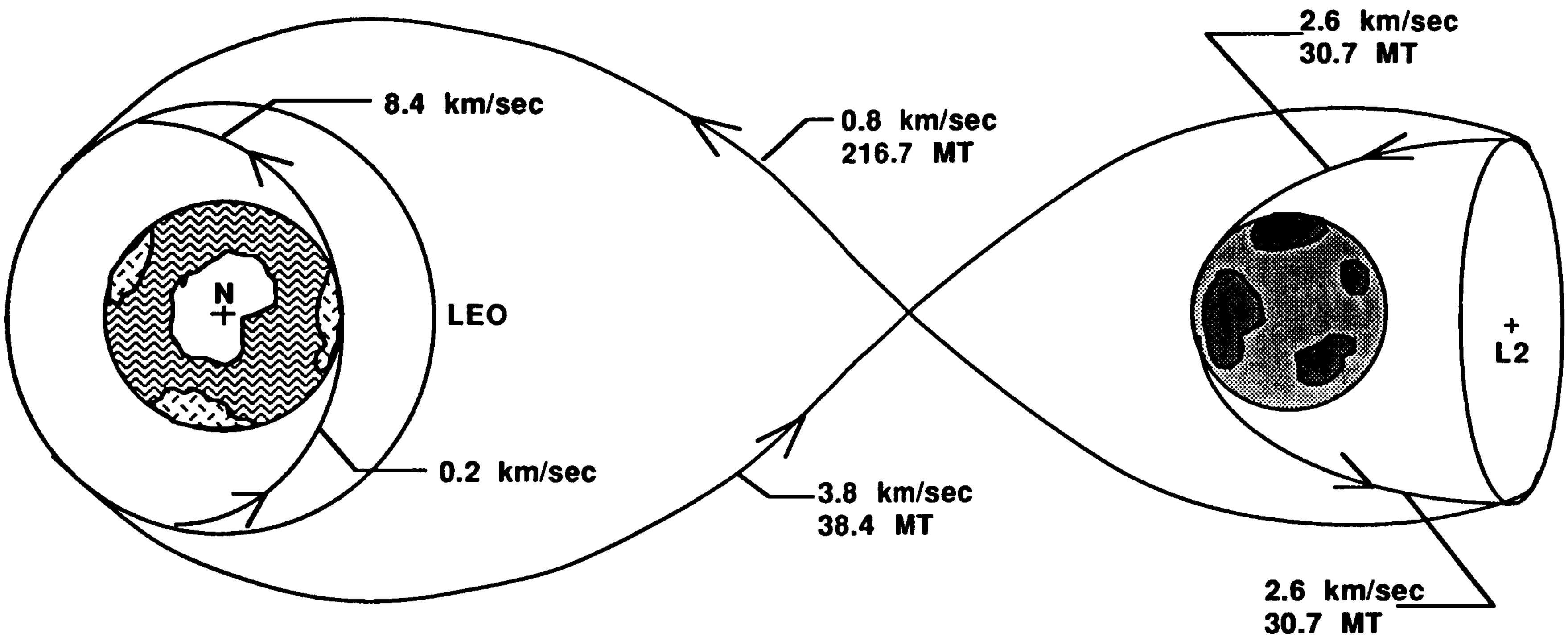

FIGURE 4.2 L2 ΔV's AND PAYLOAD CAPACITY

4.2 TRAFFIC MODEL ASSUMPTIONS

The LSPI Lunar Base Model on Symphony was used to provide an annual resupply requirement for the Lunar Base. The initial value of resupply was based on the model discussed in Section 2.3. Moonport resupply and crew transportation were not included in the traffic model study. The traffic model used a transportation system based entirely on lunar LO2 production. For steady state operations no Earth launched LO2 was required. Mass of LH2 and LO2 consumption was included for both LV and OTV flights. All LO2 consumption was subtracted from lunar base production. The LO2 required for LV ascents was subtracted from the total lunar LO2 production to come up with the lunar export mass. The export mass is the amount that is delivered to the Moonport to support LV descent and OTV flights and provide the payload delivered to LEO.

All the LH2 consumption was supplied from Earth to the Moonport in addition to the Lunar Base resupply mass. The total mass delivered to the lunar surface is the resupply mass plus the LH2 mass for LV ascents. Using the mass to be delivered on each leg and the payload of the vehicle for that leg the number of flights was determined. This procedure is outlined in Figure 4.3 for each leg. The differences in the number of ascent/descent flights and resupply/delivery flights were used to determine the number of empty flights. The LO2 and LH2 consumption for these flights was also taken into account in this traffic model.

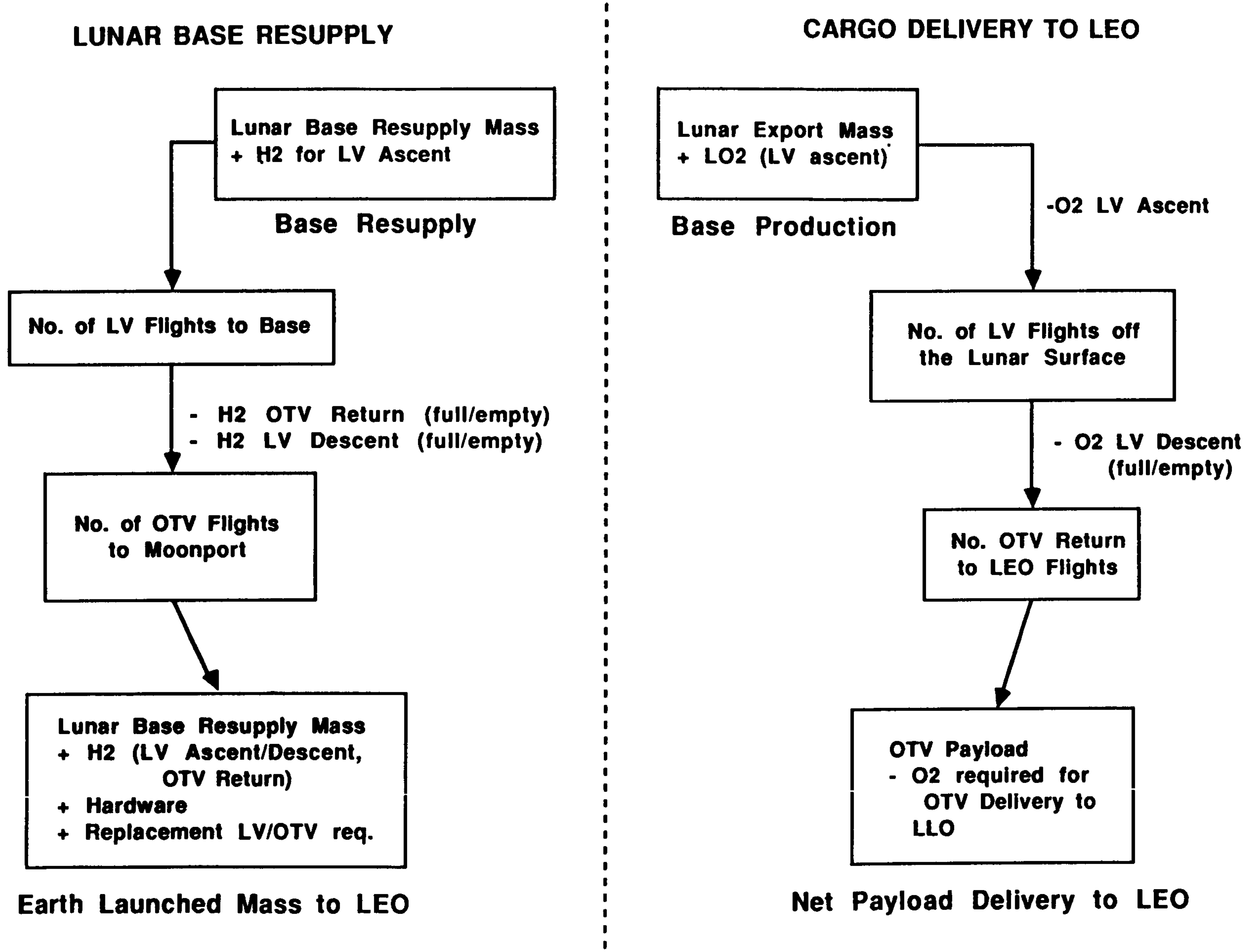

FIGURE 4.3 TRAFFIC MODEL FLOW CHART

4.3 TRAFFIC MODEL RESULTS

A TK! Solver program was used to construct the traffic model. This program provided flexibility in defining inputs and outputs. It also allowed constraints to be placed to on certain variables. The first analysis was performed comparing net payload delivery to LEO using LLO and L_2 for Moonport location. For both port locations the lunar export mass was fixed at 3820 MT/yr and a resupply mass of 303 MT/yr. The net payload delivered to LEO for each were compared. These results are shown in Table 4.1. Also shown in the Table is the total LO2 production at the Lunar Base and the required Earth launched mass for resupply. It can be seen that LLO port location produces a larger net delivery to LEO than via port placed at L_2. Although the transportation system provided a net delivery to LEO, it is considerably less than the Earth launched mass that is needed to the support the transportation system. From these results it is clear, that with conic trajectories to LLO and L_2, LLO is a more favorable location for the Moonport.

TABLE 4.1 LLO AND L_2 COMPARISON

	LLO	L_2
LV FLIGHTS	124 (84)*	158 (94)
OTV FLIGHTS	40 (18)	46 (28)
LUNAR LO2 PRODUCTION	6214 MT	7625 MT
DELIVERY TO LEO	+ 376 MT (6%)**	-516 MT (-6.8%)
EARTH LAUNCHED MASS	1426 MT	1862 MT

* - No. of Fully Loaded Flights (No. of Empty Flights)
** - % of Total Lunar LO2 Production

It was discovered that previous lunar transportation studies have achieved a return a to LEO of approximately 20-30% of total LO2 production at the Lunar Base[5]. In an attempt to determine the low percentage return produced by our model, the payload capacity used by previous models was used[6]. These newer values of payload were based on Apollo 17 ΔV's and using an Isp of 480 sec with an LO2/LH2 ratio of 7. The payloads represented a 20-25% increase from the values initially used and the propellant mass increased 5-10%. The values for the upgraded payloads is shown in Table 4.2 compared to the initial payloads used. The traffic model was run using initial export and resupply mass. Results from this run are shown in the first column in Table 4.3. The percentage return jumped to 22.6% and the net delivery to LEO is actually greater than the Earth launched mass. Figure 4.4 shows the lunar LO2 consumption using original payload capacities and Figure 4.5 shows the consumption using the upgraded payloads.

TABLE 4.2 COMPARISON OF UPGRADED PAYLOAD CAPACITIES

	ORIGINAL PAYLOAD (MT)	UPGRADED PAYLOAD (MT)
OTV DELIVERY TO LEO	216.7	277.5
OTV RESUPPLY TO LLO	38.4	47.8
LV ASCENT	30.7	51.2
LV DESCENT	30.7	47.6

TABLE 4.3 COMPARISON OF UPDATED RESUPPLY MASS

	LLO 303 MT Resupply	LLO 680 MT Resupply
LV FLIGHTS	89 (61) *	187 (129)
OTV FLIGHTS	29 (9)	61 (21)
LUNAR LO2 PRODUCTION	5670 MT	11941 MT
DELIVERY TO LEO	1282 MT (22.6%) **	2437 MT (22.6%)
EARTH LAUNCHED MASS	1175 MT	2437 MT

* - No. of Fully Loaded Flights (No. of Empty Flights)
** - % of Total Lunar LO2 Production

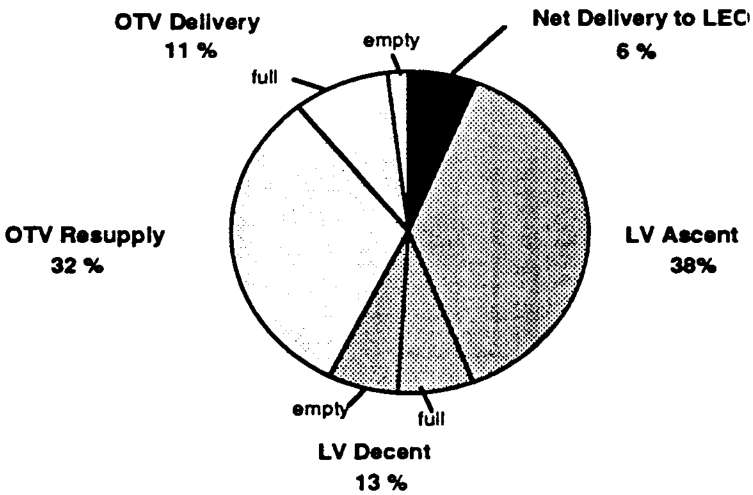

FIGURE 4.4 LUNAR LO2 UTILIZATION
(ORIGINAL PAYLOAD CAPACITIES)

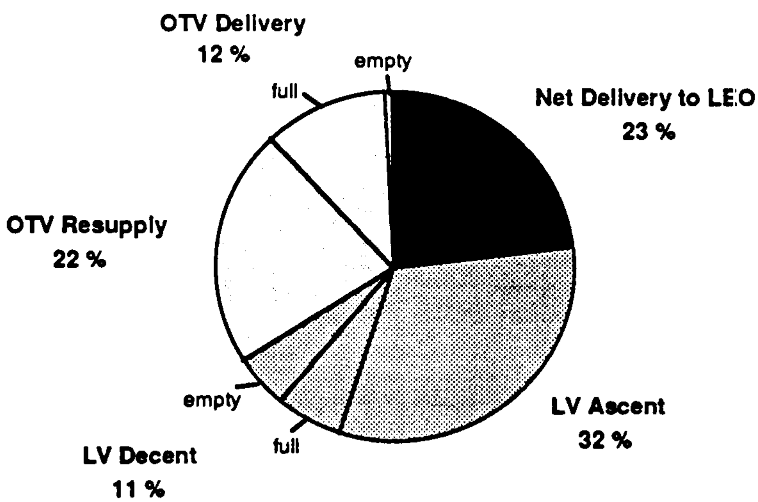

FIGURE 4.5 LUNAR LO2 UTILIZATION
(UPGRADED PAYLOAD CAPACITIES)

This large shift in the net delivery indicates the transportation system modeled is fairly sensitivity to ΔV's and Isp. This study was carried a step further by adjusting the resupply mass using the updated required lunar oxygen production. With a LO2 production of 5736 MT/yr the resupply mass was calculated to be 683 MT/yr using the LSPI Lunar Base Model. To facilitate comparison of new results, the percentage of net delivery was constrained to 22.6%. As a result of this restriction both lunar oxygen production and the required earth launched mass were doubled. If the resupply mass was again updated in the traffic model it results in a still greater lunar LO2 production and Earth launched mass, since the resupply mass is greatly increased. In the LSPI Lunar Base Model the resupply mass for the lunar base is determined by linearly scaling the requirements for 1000 MT/yr LO2 producing base[7]. This analysis indicates that for large lunar LO2 production (>1000 Mt/yr) a better resupply model must be developed. As it was seen, reducing the ΔV's and increasing the Isp greatly improves the efficiency of the lunar transportation system. A increase in payload capacity of the transportation vehicles increase the percentage of total lunar LO2 delivered to LEO. In contrast the number of vehicle flights are reduced. A reduction in the number of vehicle flights in turn reduces the amount of support required from the Moonport. Further studies are required to improve the resupply model for the LSPI Lunar Base Model and to see if further reductions in transfer ΔV's are feasible to increase payload capacities. In addition the traffic model can be improved by taking into account transfer losses in LO2 and LH2 and by including Moonport and Earthport resupply requirements.

SECTION 4 REFERENCES: TRAFFIC MODEL ANALYSIS

1. Large Scale Programs Institute, " A Demonstrative Model of a Lunar Base Simulation on a Personal Computer, " 1986, p85.

2. Advanced Space Transportation Systems (ASTS) Study - Final Presentation on Contract No. NAS9-17335, ADL Ref. 54104, Arthur D. Little, Inc., Cambridge, Massachussetts, 1987.

3. Personal communication with Curt Bilby at University of Texas at Austin.

4. Advanced Space Transportation Systems (ASTS) Study - Final Presentation on Contract No. NAS9-17335, ADL Ref. 54104, Arthur D. Little, Inc., Cambridge, Massachussetts, 1987.

5. Personal communication with Curt Bilby at University of Texas at Austin.

6. Davis, Hubert P., "Lunar Oxygen Impact Upon STS Effectiveness," Report No. EEI -8363, Eagle Engineering, Inc., Houston, TX, May 1983, Appendix A.

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SECTION 5

MOONPORT CONFIGURATION

5.1 CONFIGURATION DESIGN REQUIREMENTS

5.2 TRUSS SYSTEM REQUIREMENTS

5.3 CHARACTERISTICS OF TRUSS STRUCTURE DESIGNS

5.4 PRELIMINARY MOONPORT CONFIGURATION

5.0 MOONPORT CONFIGURATION

The configuration of Moonport integrates various subsystems so that mission objectives can be accomplished safely and efficiently. The Moonport configuration presented in this document is based on preliminary design requirements defined for each subsystem. Specific subsystems are discussed in detail in subsequent sections of this document.

The following sections describe the preliminary Moonport configuration. Included in the discussion are a brief comparison of Moonport and Space Station configuration requirements, a list of Moonport configuration design requirements, and a summary of the specific requirements of the supporting truss system. The characteristics of current Space Station truss configurations are examined. These truss configurations are judged on their potential application to Moonport design. Finally, a specific Moonport configuration is presented.

5.1 CONFIGURATION DESIGN REQUIREMENTS

Extensive research has been done to support current designs of an Earth-orbiting Space Station. While this research will prove useful for Moonport design, the mission objectives, and therefore the configuration requirements, will differ substantially from Space Station design work. Placement in LEO provides Space Station with protection from radiation and places it in close proximity to Earth-based facilities. These advantages will not be available to Moonport. Additionally, Space Station is being designed as a relatively isolated space structure, with numerous experiments requiring specific attitude pointing and stable gravity environments. The primary function of Moonport will be the processing and servicing of various vehicles and their payloads. Therefore, the environment of Moonport will be less isolated and will require greater stiffness and structural strength than Space Station. By comparing the requirements of Moonport and Space Station, SPS designers decided what current technologies could be used for Moonport design, and what technologies should be developed further for successful Moonport operations.

The following paragraphs describe specific requirements that impact the design of an integrated Moonport configuration:

1) Moonport will establish a secure environment to support manned operations in close lunar proximity. As currently designed, Moonport will have a permanent crew of five people. This design assumes that, at the time of steady state operations, there will be sufficient traffic at Moonport to require constant manned supervision and activity. Therefore, the Moonport configuration must provide a secure, protected environment from which the crew members can conduct mission operations. Shielding must be provided for protection from radiation caused by cosmic rays and solar events, and redundant life support systems must be available and easily accessible. In addition, critical subsystems must be protected from meteorite impact.

2) Moonport will establish a stable, secure transportation node which will be able to support vehicle docking and operations. Moonport configuration must provide for easy access and good clearance for proximity operations of vehicles. Also, a reliable vehicle retrieval system must be designed to assist the docking of vehicles with limited proximity operations capability, such as OTVs. The support structure of the port must be of sufficient strength to withstand the impacts of vehicle docking, and the structure must be stiff to accommodate attitude control operations. The lowest natural frequency of the structure must be sufficiently large to

minimize disturbances caused by vehicle docking and operations. These frequency modes must also be compatible with manned operations. Additionally, the configuration must provide a base for vehicle support services, such as refueling and maintenance.

3) Moonport will establish a facility for handling and storage of a variety of materials. A key function of Moonport will be the handling and storage of vehicle payloads. Initially, vehicles will deliver materials from the earth to be taken to the lunar surface. Later, when the lunar processing facility is established, lunar products will be delivered and stored at Moonport. In both cases, these materials will be stored until vehicles arrive to transport them to the appropriate destination. The configuration design must provide for material storage and handling, including mobile remote manipulator system (MRMS) operations.

4) The construction of Moonport must be as safe and efficient as possible. The number of shuttle flights required for construction will be minimized to avoid scheduling problems. Other methods of transportation, including heavy-lift launch vehicles, will be emphasized. The construction should be as automated as possible, minimizing the requirement of EVA support.

5.2 TRUSS SYSTEM REQUIREMENTS

A key element of Moonport design will be the supporting truss structure. The truss must be configured to provide the most advantageous integration of Moonport subsystems, and to serve as the foundation from which mission objectives can be achieved.

Typical requirements for truss structures in space [1] include the ability to provide:

1) a structural foundation for construction operations
2) surface area for attachment of payloads and utilities
3) structural stiffness to minimize control problems and provide elevated first mode frequency
4) a road bed for the Mobile Remote Manipulator System (MRMS) to aid in construction and transportation of payloads
5) a redundant structure that will offer alternate load paths if a member of the truss is damaged
6) to provide structural repair capability without the loss of structural integrity.

These requirements were used to choose an adequate truss system for Moonport.

5.3 CHARACTERISTICS OF TRUSS STRUCTURE DESIGNS

As mentioned earlier, significant research has been conducted to design configurations of an Earth-orbiting Space Station. While the design requirements of Space Station differ from Moonport, a review of the current truss structure designs was conducted to utilize past research and to evaluate current truss design characteristics consistent with Moonport requirements. Two basic types of truss systems were examined relative to Moonport design requirements: the boom truss and the delta truss.

5.3.1 BOOM TRUSS CHARACTERISTICS

The boom truss is the primary support structure for Space Station configurations such as the Power Tower (Figure 5.1[2]) and the Dual Keel (Figure 5.2). The Power Tower design has few characteristics that can be used for Moonport applications. The limited surface area of the

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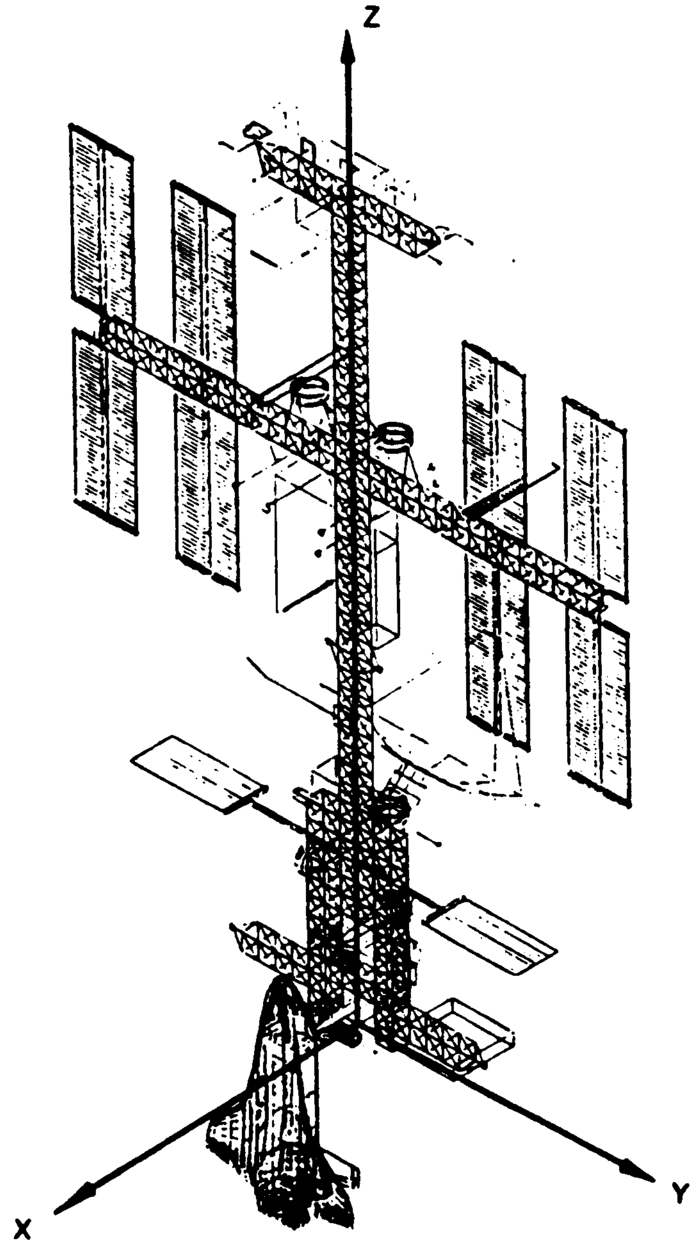

FIGURE 5.1 - POWER TOWER CONFIGURATION[2]

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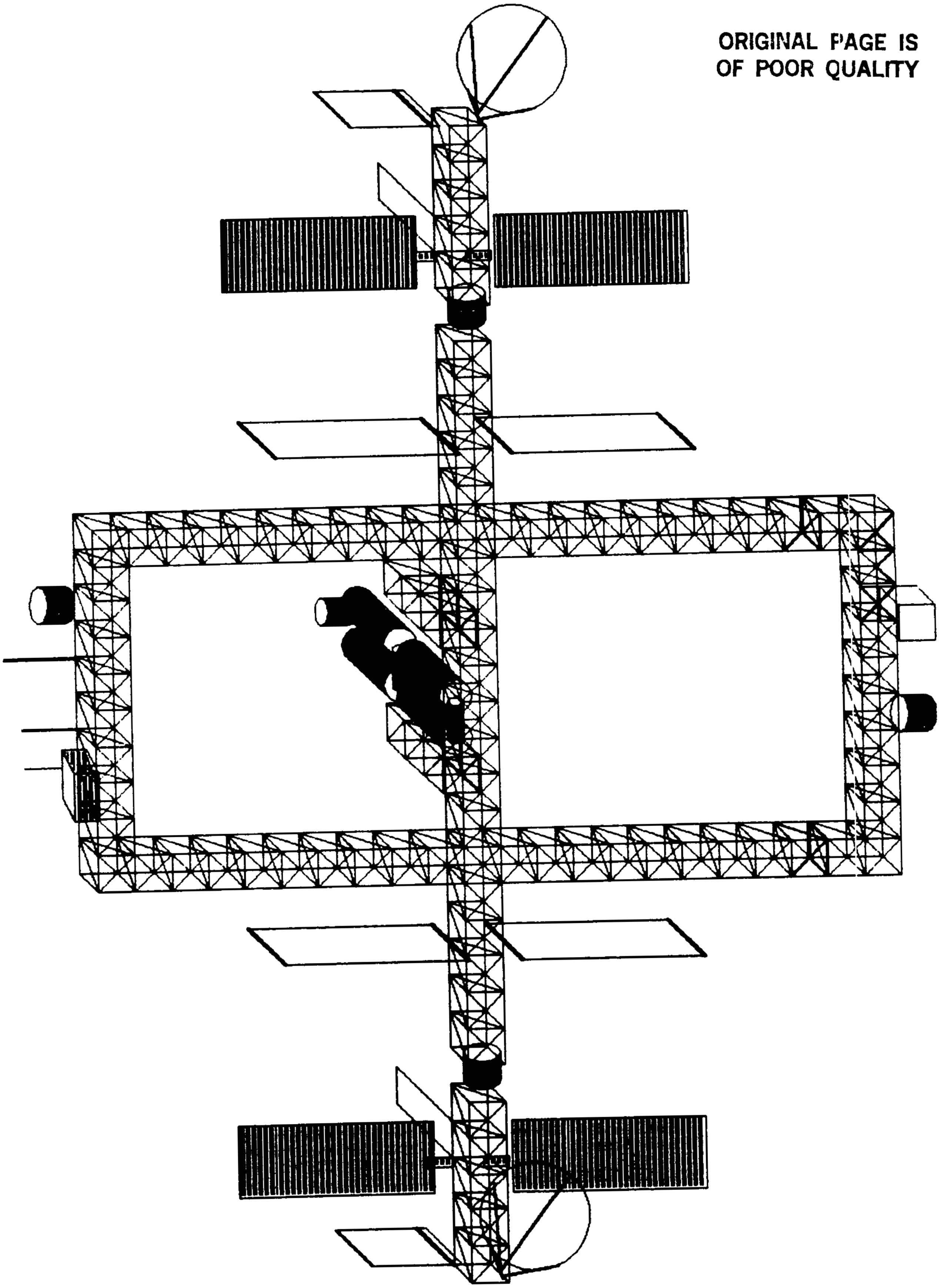


FIGURE 5.2 - DUAL KEEL CONFIGURATION

configuration reduces the possible locations where payloads can be attached or housed. Also, because of its long, thin shape, the beam truss is substantially flexible, impairing the ability of the structure to accommodate frequent vehicle docking. Also, the natural frequency of the station was found to be unsettling to crew members. A vehicle servicing facility would be possible with this configuration , but the beam truss would offer little support for the facility. To provide a facility large enough to handle vehicle servicing and refueling, the structural integrity of the facility would have to be independently strengthened, adding weight and size to the design. Because of its long, slender design, the Power Tower configuration takes advantage of the gravity gradient forces in orbit to maintain stability. However, these stability advantages would be greatly reduced in lunar orbit, and totally non-existant at L_2.

Because of the box-like keel design, the Dual Keel configuration is stiffer than the Power Tower. Consequently, the Dual Keel is better suited to accommodate vehicle docking. However, surface area is still limited, which hinders payload storage and garage integration. Also, there are alternative truss designs with better stability.

5.3.2 DELTA TRUSS CHARACTERISTICS

A configuration which offers many significant advantages for Moonport design is the delta truss configuration. The name 'delta' comes from the triangular orientation of three planar trusses (Figure 5.3[3]). The dimensions of one planar truss are shown in Figure 5.4[4]. These three trusses are joined at their ends to form an triangular, equilateral cross-section. The characteristics of the delta truss can be successfully applied to a variety of Moonport operations. The triangular orientation of

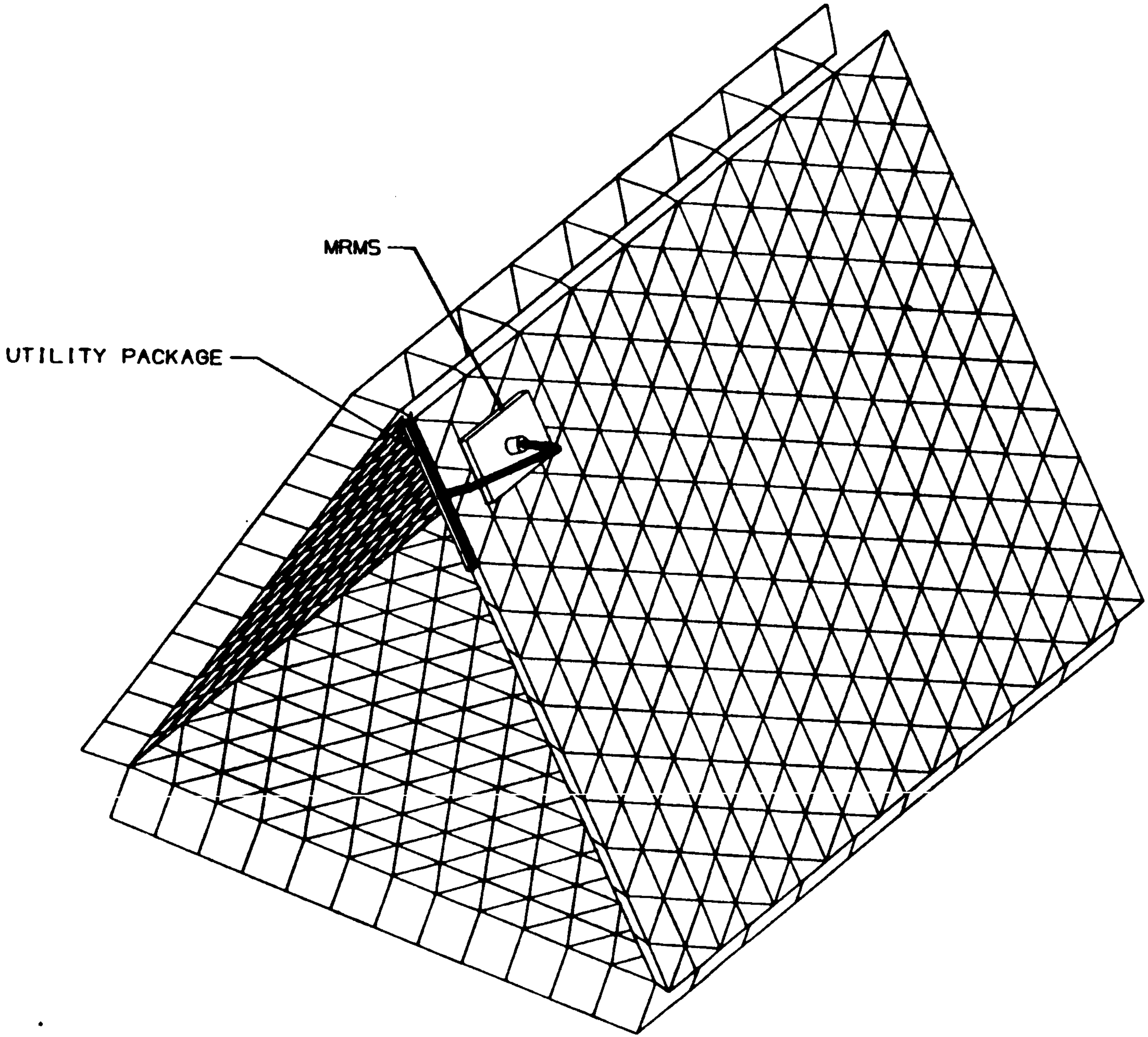


FIGURE 5.3 - DELTA TRUSS CONFIGURATION[3]

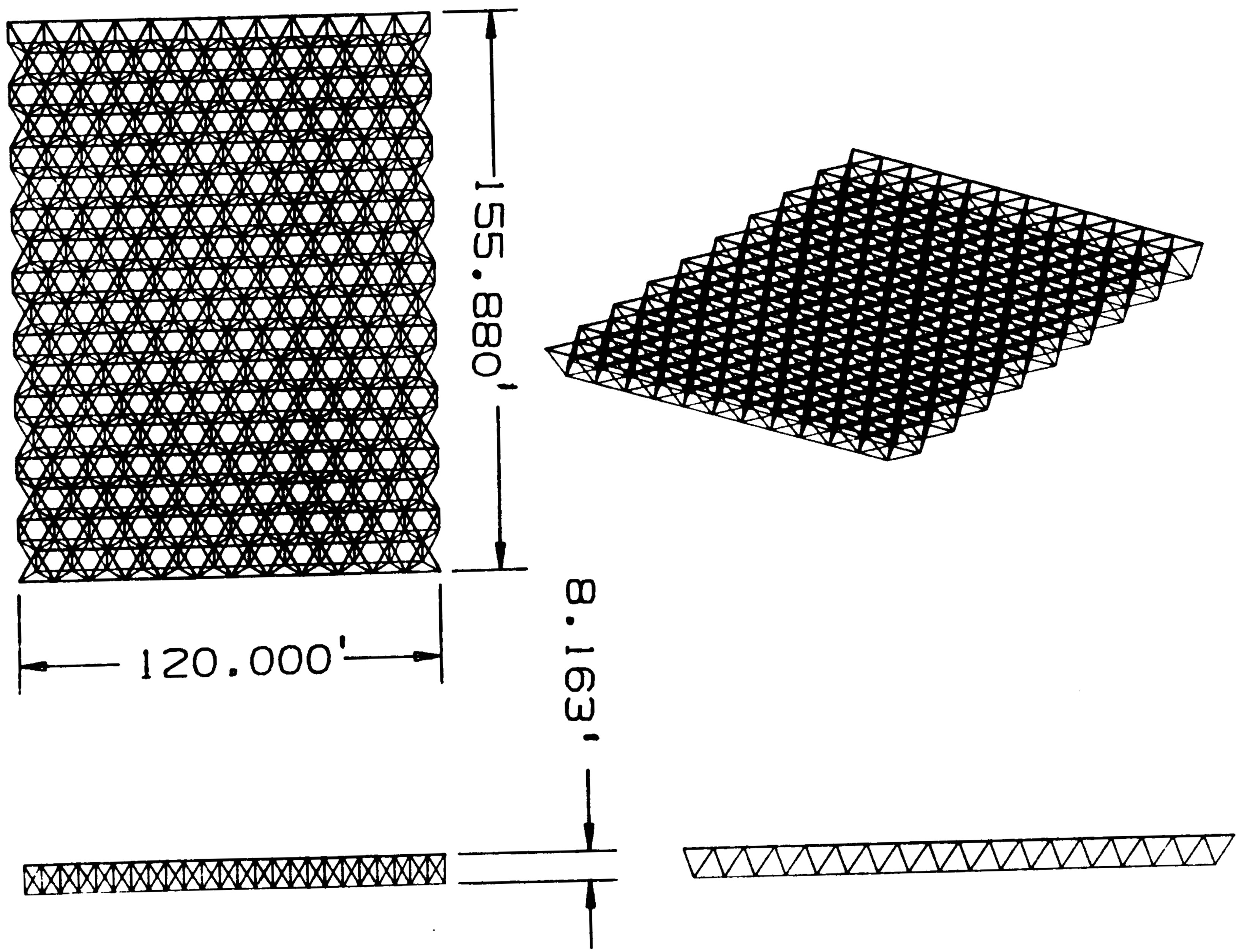


FIGURE 5.4 - PLANAR TRUSS DIMENSIONS[4]

the delta gives it greater stiffness and strength than the beam truss system. The configuration has better stability and is more resistant to disturbances. The lowest mode of vibration is orders of magnitude higher than any other Space Station configuration.[5] The substantial strength and stiffness of the delta truss will provide a firm support structure that will be able to handle the various mission disturbances due to vehicle docking and operations. Likewise, the improved stiffness will make the delta truss structure easier to control.

The three planar trusses of the delta configuration can provide ample workspace for the storage of a variety of modules. The truss structure provides excellent attachment versatility with its numerous nodes, much like a "pegboard." [6] After vehicle docking procedures have been completed, cargo modules can be attached to the truss itself via the MRMS.

The triangular area enclosed by the three planar trusses provides a viable location for a vehicle servicing facility. The interior walls of the delta truss itself can provide structural support for the facility. The capturing and docking of vehicles with the facility will be assisted by the large triangular opening at the front of the truss , which provides ample space for maneuvering into the facility.

Because the Space Station will use solar power as its main power source, a primary design consideration for Space Station configurations like Power Tower and Dual Keel was the accommodation of articulated solar arrays to permit a fixed orientation along the local radius vector for stability and payloads placement. In the original delta truss configuration, the solar arrays were placed on one of the faces of the planar truss, requiring constant attitude correction of the station to point the arrays toward the sun. This constraint lead to the selection of Dual Keel over delta as the current Space Station configuration. However, the main source of power for Moonport will be nuclear, so there is no need for constant pointing for power. Thus, this pointing restriction no longer applies.

5.3.3 SUMMARY OF TRUSS CONFIGURATION STUDY

The characteristics of the three truss configurations (Power Tower, Dual Keel, and delta) were judged based on their potential application to Moonport design. The truss requirements (Section 5.2) were used to define the specific design areas to be judged , and each configuration was ranked (Table 5.1).

After examining the requirements, the delta truss was chosen as the best configuration overall. The delta truss was determined to be the easiest to construct since the entire delta truss can be compressed into one orbiter payload bay, and deployed to orbit in one shuttle flight. Power Tower and Dual Keel are built by erecting beam trusses. While the deployment method used by the delta seems to be the easiest and quickest, there are questions concerning how much effort would be required to strengthen the truss after deployment. The deployed truss may require tightening, which would involve extended crew extravehicular activity (EVA). Because of these uncertainties, construction methodology will not be a driving requirement. However, the delta does have significantly greater surface area and structural stiffness than the other configurations. Consequently, the delta truss was chosen as the foundation for the Moonport configuration.

TABLE 5.1

DECISION MATRIX FOR TRUSS STRUCTURE DESIGN

	DELTA TRUSS	POWER TOWER	DUAL KEEL
EASE OF CONSTRUCTION	1	2	2
SURFACE AREA	1	3	2
STABILITY	1	3	2
ROAD BED FOR MRMS	1	1	1
REDUNDANT STRUCTURE	1	3	2
REPAIR CAPABILITY	1	1	1

1-BEST OPTION 3-WORST OPTION

5.4 PRELIMINARY MOONPORT CONFIGURATION

The preliminary Moonport configuration (Figures 5.5-5.7) consists of two basic structural elements. The main delta truss will house the manned systems involved with Moonport operations. The refueling facility, the nuclear reactor, and the LTV thrusters will be positioned on a tripod boom structure extending away from the delta truss, to isolate these potentially hazardous environments from the habitated sections of Moonport. A breakdown of the masses for Moonport is constrained in Table 5.2. The final mass of Moonport is 802.5 MT (1769.0 kips). To place this amount of mass in LEO requires 36 shuttle launches (22.7 MT, 50.0 kips per flight) or

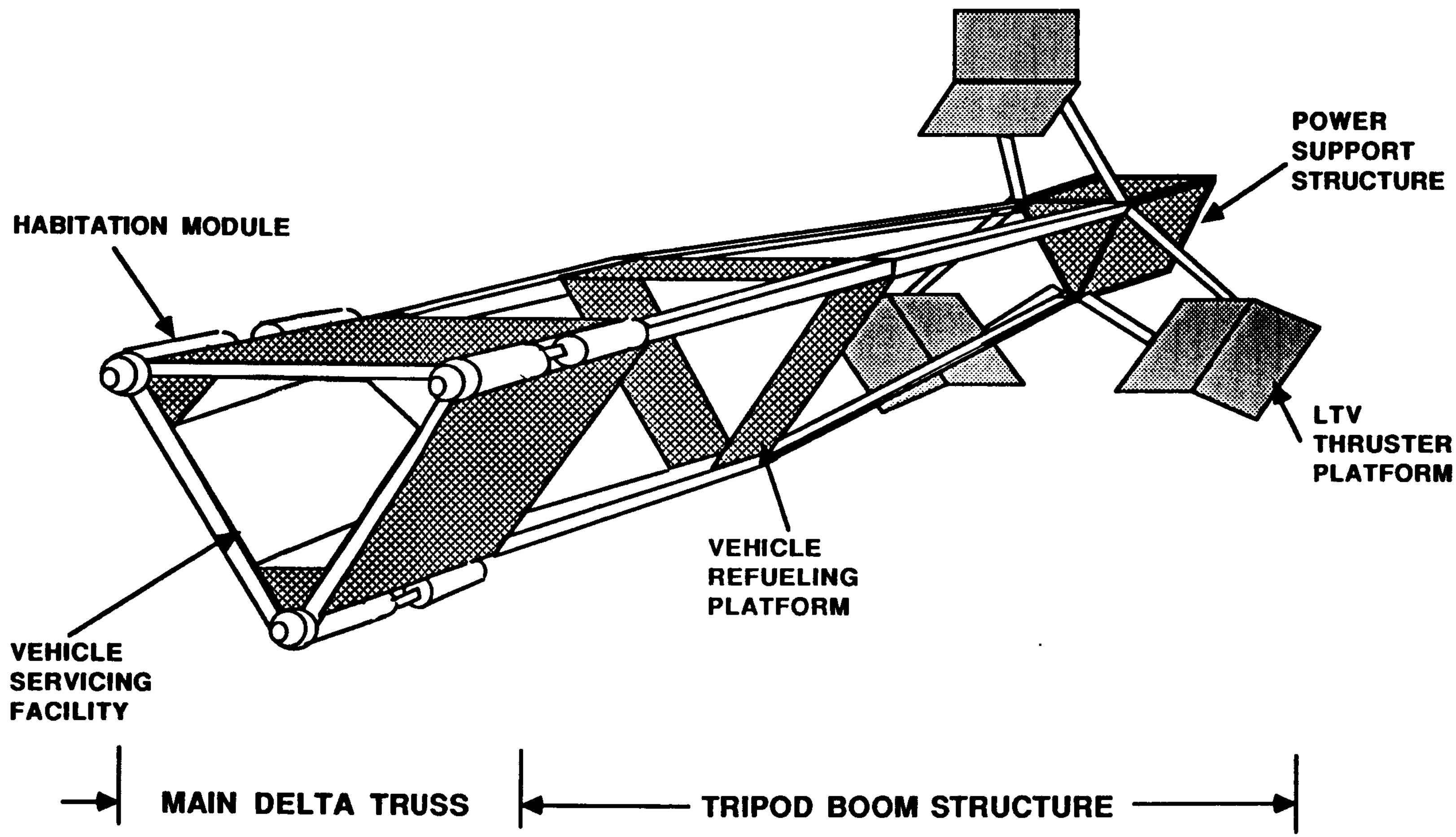


FIGURE 5.5 - PRELIMINARY MOONPORT CONFIGURATION DESIGN

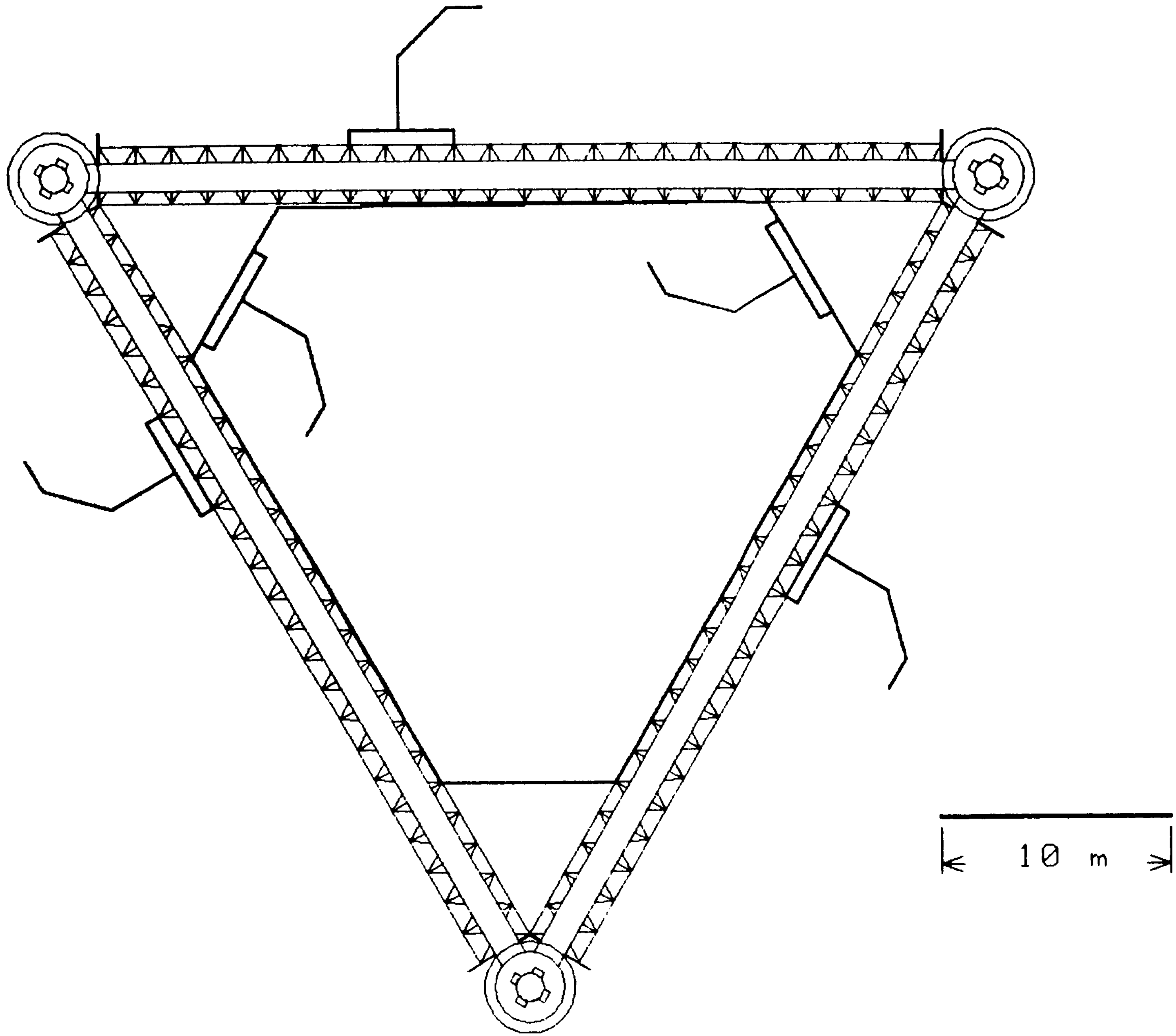


FIGURE 5.6 - MOONPORT FRONT VIEW

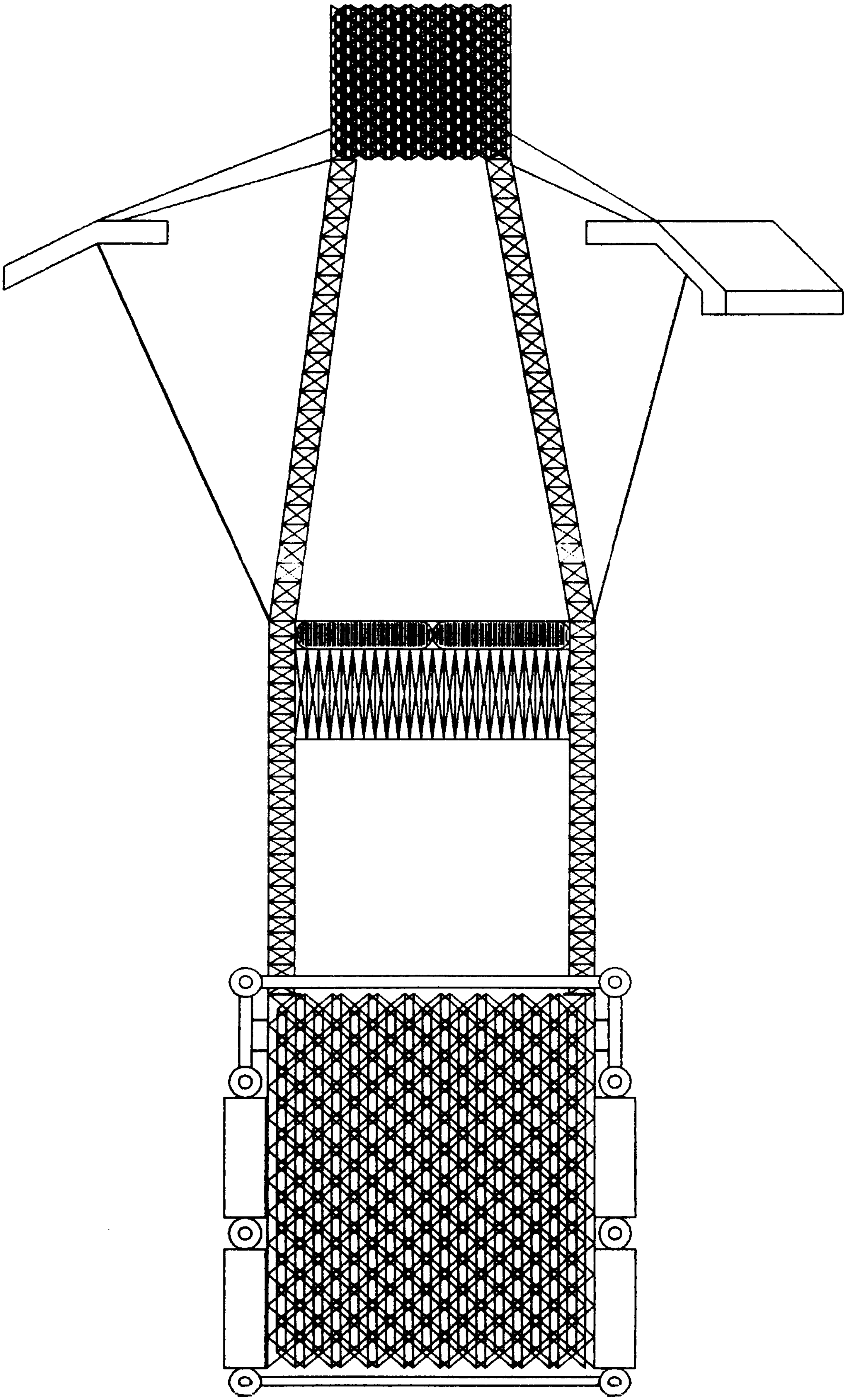


FIGURE 5.7 - MOONPORT:SIDE VIEW

TABLE 5.2 MOONPORT MASS BREAKDOWN

COMPONENT	MT	KIPS
MAIN DELTA TRUSS	12.0	26.5
SUPPORT TRUSS	1.0	2.2
SERVICING GARAGE	65.0	143.3
REFUELING PLATFORM	75.0	165.3
HABITATION MODULES (6 TOTAL)	108.0	238.1
HABITATION MODULE SHIELDING (ALL)	148.2	326.7
COMMUNICATIONS	1.1	2.4
THRUSTERS W/ AUXILLARY SYSTEMS	7.8	17.2
PRIMARY POWER PLANT	82.4	181.6
REACTOR SHIELDING	113.0	249.1
EMERGENCY FUEL CELLS	1.0	2.2
SUBTOTAL	614.5	1354.6
10% ERROR ESTIMATE	61.5	135.5
PROPELLANT (INCLUDING TANKAGE)	126.5	278.9
TOTAL	802.5	1769.0

12 HLLV launches (68.0 MT, 150.0 kips per flight), exclusively.

5.4.1 DELTA TRUSS STRUCTURE

The main elements of Moonport will be supported on the delta truss structure itself. The delta truss system will integrate various Moonport subsystems, including the habitation and logistics modules, cargo handling facilities, and a vehicle servicing garage. Also, the truss can support the transportation and deployment of initial lunar base payloads.

5.4.1.1 DELTA TRUSS CONSTRUCTION

The entire delta truss structure can be carried to orbit and deployed in one shuttle mission. Once deployed, the truss system can be used as a foundation to assist in further construction of Moonport. The delivery of other components of Moonport will emphasize use of alternate vehicles, such as heavy lift launch vehicles. The use of orbiters for delivery of construction materials is minimized. It is assumed that an Earth-orbiting space station, Earthport, exists to assist in Moonport construction.

5.4.1.2 HABITATION AND LOGISTICS MODULES

For Moonport design, the habitation modules are placed on the apexes of the delta truss . Habitation modules are positioned with two on each apex. The modules are connected by a long tunnel running from one apex to another. In order to provide escape from any module, tunnels will be located at both triangular faces of the delta. By placing them on each apex, the modules are relatively isolated from one another. This situation provides a safer overall environment; the problems of one apex can be easily quarantined. There is a considerable amount of pressurized volume in the interconnecting tunnels, but the power supplied by the nuclear reactor is sufficient to allow for this.

5.4.1.3 CARGO HANDLING

The large surface area of the planar trusses can be utilized for the storage of cargo modules (Figure 5.8). By designing special attachments for use with the nodes of the planar truss, cargo modules can be attached to the face of the truss itself (Figure 5.9[7]). The delta truss configuration allows for MRMS access to all areas of a particular truss face. Each face will have its own MRMS.

5.4.1.4 VEHICLE SERVICING FACILITY

The space enclosed by the walls of the delta truss provides a suitable location for the vehicle servicing facility (Figure 5.10). This facility will be used to support the maintenance of disabled vehicles in lunar orbit. The facility will be shielded from micrometeorites by a thin aluminum shell to provide a safe environment for suited crew members. The walls of the delta truss also help support and protect the facility. The servicing facility extends along the entire length of the delta truss. The habitation modules are located nearby, thus providing quick sanctuary in case of emergency. Two MRMS are provided for movement of vehicles and materials within the facility. A more detailed discussion on vehicle servicing facilities is presented in Section 6.

5.4.2 TRIPOD BOOM CONFIGURATION

Another major element of Moonport configuration is the tripod boom structure (Figure 5.5). Each leg of the tripod is connected to the main delta truss, in the face opposite the opening of the vehicle servicing

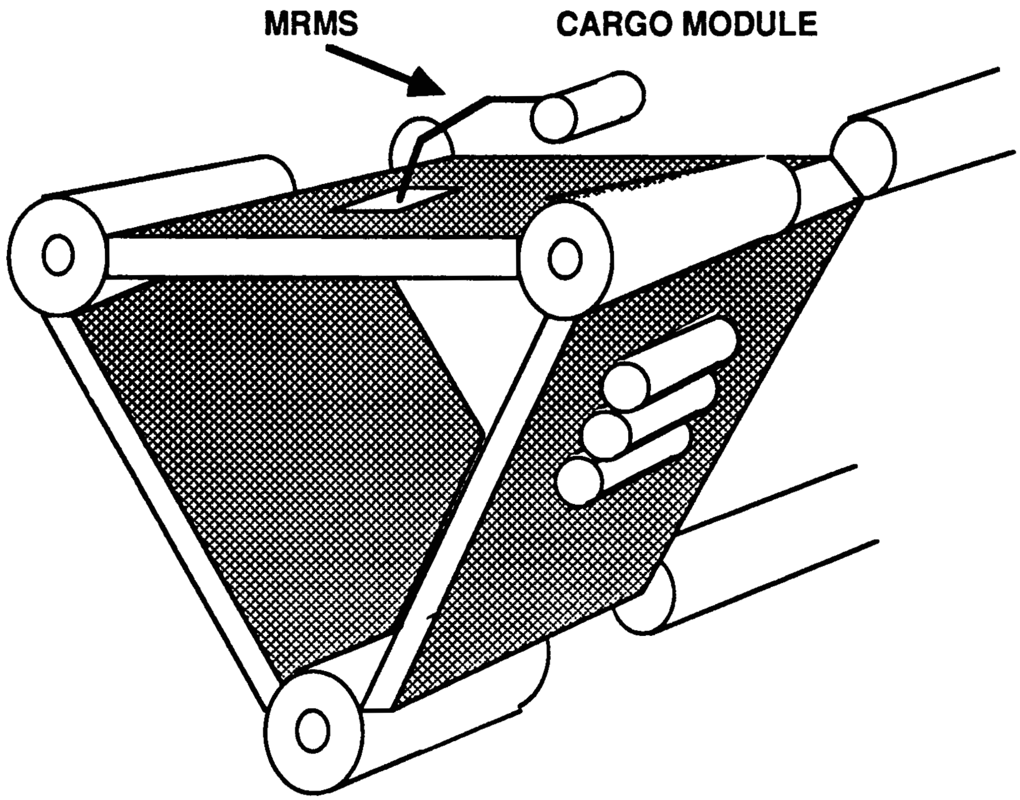
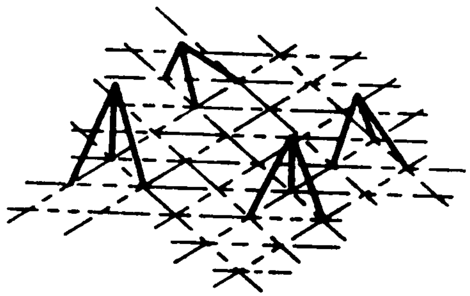
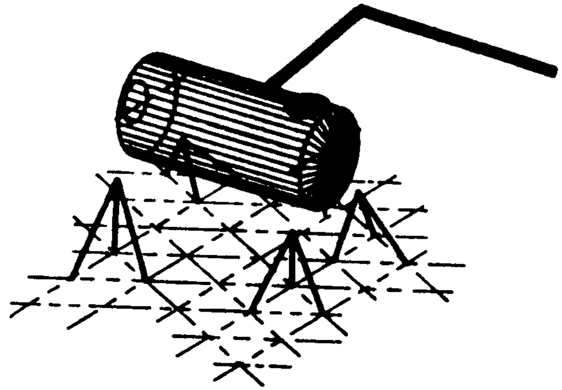


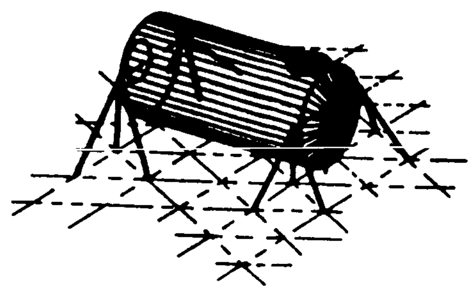
FIGURE 5.8 - DELTA TRUSS CONFIGURATION WITH MRMS AND CARGO MODULE



1. INSTALL TRIPODS IN PLACE



2. POSITION MODULE IN PLACE



3. FINAL INSTALLATION

FIGURE 5.10- ATTACHMENT OF MODULES TO DELTA TRUSS[7]

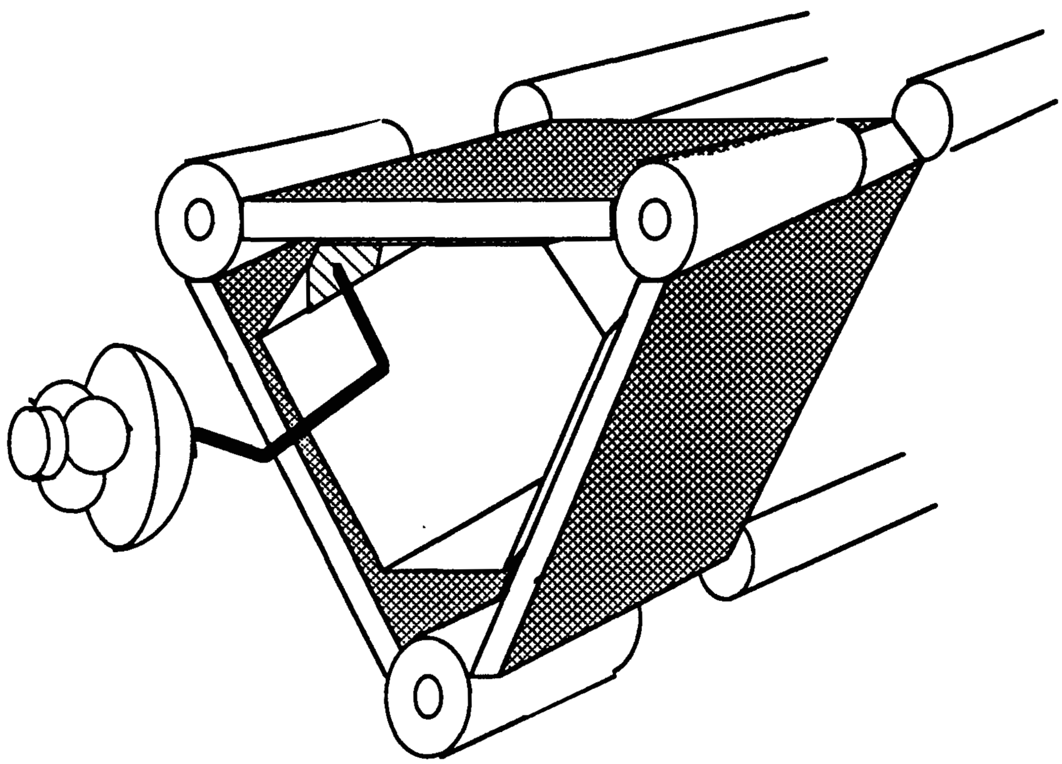


FIGURE 5.10 - DELTA TRUSS CONFIGURATION WITH VEHICLE SERVICING FACILITY

facility. The legs of the first section of the tripod extend out , perpendicular to the delta, for 40 meters (131.234 feet). The outer section of the tripod extends out another 60 meters (196.8 feet), converging at the power support structure. The tripod boom structure is used to isolate the vehicle refueling facility, the nuclear reactor, and the LTV thrusters from the rest of Moonport.

5.4.2.1 VEHICLE REFUELING FACILITY

The vehicle refueling facility is located on the first part of the tripod boom. The front edge of the facility is 25 meters (82 feet) from the main delta truss. This facility is comprised of a vehicle landing platform, fuel tanks, and electrolysis equipment. The refueling facility is separated from the servicing garage to minimize contamination of habitated areas due to vehicle exhaust, fuel leaks,etc. Also, proximity operations can be carried out easily and safely in an open environment, away from the habitation modules.

5.4.2.2 POWER SUPPORT STRUCTURE

Beyond the refueling facility, the tripod legs will extend 60 meters (196.8 feet), converging at the power support structure. This truss houses the nuclear reactor. Also, the LTV thrusters will be attached to the truss. For complete details of LTV integration, see Section 8.

Further study will be required to design the actual method of construction and attachment of the tripod boom structure, taking into account boom stability, strength, and reaction to forces caused by vehicle acceleration. The legs of the tripod will probably be erected, and attached to the delta truss by means of a plate. Additionally, the tripod will have to be designed to permit the thrust vector of the low thrust vehicle to move as closely through the center of mass of the system as possible. This will aid in the control and stability of Moonport as it is transported to the Moon.

SECTION 5.0 REFERENCES: MOONPORT CONFIGURATION

1. NASA LBJ Space Center. Engineering and Configuration of Space Station Platforms. Park Ridge, N.J.: Noyes Publication, 1985, pp.531.

2. Ibid.

3. Delta Space Station JSC-20414. Houston: NASA LBJ Space Center, March 31,1985.

4. Ibid., p.290.

5. Ibid., p.1.

6. Ibid., p.1.

7. Ibid., p.316.

SECTION 6

VEHICLE SUPPORT SERVICES

6.0 VEHICLE SUPPORT SERVICES

Since the spaceports will serve primarily as transportation nodes, vehicle support services are an integral component of spaceport operations, and in the case of Moonport, a necessity.due to its relative isolation from Earth. These services should include vehicle retrieval and deployment; maintenance and repair of both the vehicles and the spaceport itself; transportation, storage, and transfer of propellants and other consumables; and special facilities designated for servicing and refueling.

Port traffic will include orbital transfer vehicles (OTV), lunar vehicles (LV), port-local vehicles such as the orbital maneuvering vehicle (OMV), and possibly lunar satellites.

6.1 VEHICLE RETRIEVAL AND DEPLOYMENT

Retrieval and deployment (R/D) vehicles should be able to pick up an incoming spacecraft at its rendezvous orbit (up to 1 km away from the port), transport the spacecraft safely to the spaceport, and either dock it or hand it over to an MRMS for berthing or servicing. Once the payload transfer, fueling, servicing, and other port operations have been concluded, the R/D vehicle should be able to take the spacecraft from its dock or from the MRMS, transport it safely back to the rendezvous orbit, and return to port.

6.1.1 RATIONALE

Methods for vehicle retrieval and deployment are needed because of limits on proximity operations. For example, the OTV does not have the capability to use its own propulsion system within 300-1000 m (984-3281 ft) of the space station. Primary concerns dictating this limitation are environmental contamination due to propulsion effluents, plume impingement on the port structure or subsystems, and crew and structure safety. A secondary consideration is the potential difficulty of achieving a zero-momentum, precision docking into a relatively small hangar or berthing area, particularly with an unmanned vehicle. Accurate pre-programming for close proximity operations is currently infeasible because of constantly changing conditions, and the OTV is not currently designed to be operated from the spaceport. Consequently, some form of "tugboat" vehicle is required for close proximity retrieval and deployment.

Contamination is a concern for several reasons. First, a tenuous atmosphere is created around the vehicle or structure (either by close-proximity powered flight or by propellant transfer leakage or venting). This atmosphere can either impair the field of view of the vehicle, spaceport, or other optical instruments, or it could actually damage optical instruments or cryogenically-cooled surfaces;. Finally, a return from EVA in such an atmosphere could bring dangerous materials into habitable areas.

Propellants such as hydrazine (N_2H_4), nitrogen tetroxide/monomethyl hydrazine (NTO/MMH), or liquid hydrogen/liquid oxygen (LH2/LO2) are all potential hazards due to the nature of these fluids. Cold-gas propulsion systems such as gaseous nitrogen (GN_2), however, are considered 'safe.'

Several of the satellite retrieval vehicles considered in this study for OTV and unmanned LV retrieval employ a cold-gas reaction control system for close-proximity operations, and other propellants for primary propulsion. Other candidates use only cold-gas propulsion. All of the candidate vehicles, however, are remotely-controlled/teleoperated within a respective range. Short descriptions of the vehicles considered and a decision matrix used to choose the best candidate appear in section 6.1.3.

6.1.2 PROXIMITY OPERATIONS REQUIREMENTS

Specific requirements are that the R/D vehicle should:

1) minimize the contamination hazard by ultilizing a 'safe' propulsion system for close proximity operations
2) require minimal manpower -- preprogrammed/remotely-controlled is better than remotely-controlled, which is better than EVA
3) have as large a range as feasible
4) be as versatile as possible -- not only should it be able to retrieve and deploy, but also aid in servicing, refueling, contingency operations, and self-maintenance
5) have the beneficial characteristics of television monitoring, radio communication, lighting fixtures, large propellant reserves, and small size
6) be able to handle up to 380 MT (838 kips) since the primary (and largest planned) retrieval target is the OTV -- this mass allows for a single stage OTV with maximum payload and full return propellant

6.1.3 CANDIDATES

The manned and unmanned proximity operations modules (MPOM,UPOM) are free-flying, cold-gas (GN_2)-propelled vehicles currently conceived for satellite retrieval and servicing. The UPOM is remotely flown with an approximate range of 1 km (3281 ft) and a payload capacity of 11.0 MT (24.3 kips). The MPOM uses the Manned Maneuvering Unit (MMU) for propulsion, has a range of 305 m (1000 ft), and a capacity of 2.5 MT (5.5 kips). The MMU can be used with, or without the MPOM as previously mentioned, or without. It uses cold-gas propulsion and has an approximate range of 1 km (0.6 mi). The core size of these proximity operations modules (POMs) is 1.12 x 0.74 x 0.89 m (44"x29"x35"), and allows attachment of various retrieval and servicing kits. Figure 6.1[1] shows sample POM add-on kits for servicing.

The Orbital Maneuvering Vehicle (OMV) is currently designed to operate as an OTV space tug. It uses hydrazine propulsion (or NTO/MMH) and is approximately 3.96 m (13 ft) in diameter and 0.94 m (37 in) thick. It is unmanned, preprogrammed, and with the addition of "snap-on kits," has limited capability to service vehicles.

The OMV is actually an upgrade of the earlier-proposed Teleoperator Maneuvering System (TMS), which is a remotely-controlled free-flying "mini-tug" originally envisioned for satellite retrieval and servicing . The upgrade was necessary to allow the vehicle to service spacecraft and large space structures. The TMS is roughly the same size as the OMV, and its flight can be either pre-programmed or remotely-controlled. The vehicle can also be remotely programmed. The propellant used for large ΔV maneuvers is either hydrazine or NTO/MMH, while GN_2 is used for close-in operations and reaction control. Other benefits include lighting and television monitoring, as well as the add-on servicing kits which are also available with the OMV. In LEO the TMS can retrieve up to 23.0 MT (50.7 kips).

Both the OMV and TMS are mentioned here because the OMV is not currently designed to have all of the capabilities originally suggested for the TMS. As it is designed, the TMS seems to be more applicable to Moonport operations.

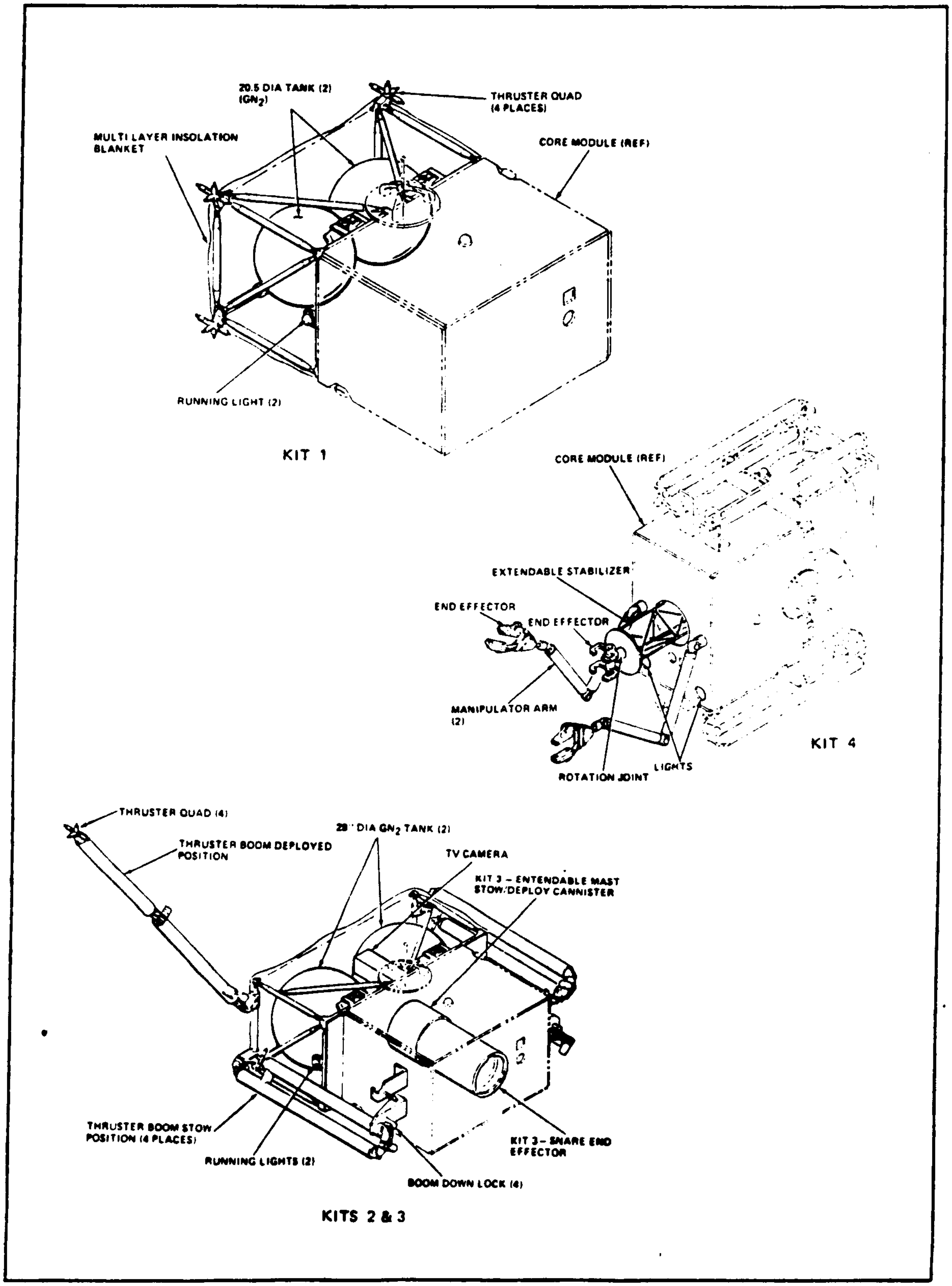


FIGURE 6.1 POM ADD-ON KITS FOR SERVICING[1]

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The Remote Manipulator System (RMS) consists of a manipulator arm mounted onto the spaceport structure or a mobile RMS (MRMS) mounted onto a track. It also has a wrist-mounted closed-circuit television camera and lighting fixtures. Various end effectors enhance servicing capability. Payload capacity is 14.5 MT (32.0 kips) baseline, 29.5 MT (65.0 kips) contingency. Its range depends on its intended use. The RMS can be manned or unmanned, and is either remotely-operated or operated by the an EVA crewmember.

The Maneuverable Television (MTV) is a small remotely-controlled spacecraft with video and telemetry transmission capabilities, and is the basis for the OMV/TMS design. The MTV utilizes a cold gas propulsion system, has a range of 4.8 km (3 mi), and is used for vehicle or spaceport inspection.

6.1.4 RECOMMENDATIONS

As seen from Table 6.1, the OMV/TMS is the best choice for vehicle retrieval and deployment. However, as previously discussed, more features of the conceptual TMS should be incorporated into the Moonport OMV design, particularly the cold-gas reaction control system.

Recommendations are:

1) an upgraded OMV should be used for spacecraft retrieval, deployment, and some remote servicing of both vehicles and the spaceport.
2) because of its smaller size, the UPOM should be used for in-house servicing, satellite R/D, some remote servicing, and as an aid in port-based refueling.
3) the MRMS should be used for actual hangar entry and exit and also for port-based work.

TABLE 6.1 VEHICLE RETRIEVAL CANDIDATE COMPARISONS

VEHICLE	PROP	CAPAC.	MANPWR	RANGE	SERVICING	TOTAL
OMV/TMS	2	1	1	1	1	6
UPOM	1	3	1	1	1	7
MMU/UPOM	1	4	3	2	2	12

NOTE: LOW NUMBERS ARE BEST

4) the MTV should be used for remote inspection and observation of spacecraft, storage facilities, and the spaceport itself. Use of the MTV in this capacity, particularly for pre-retrieval and post-deployment, would allow more efficient use (in time and propellant) of the R/D vehicles.

5) finally, EVA (with the MMU, MPOM, RMS, or tethers) should be used only for backup, special, or contingency operations. Minimizing EVA reduces manpower requirements, time, and expense, and is safer for the crew.

6.2 SERVICING

Vehicle servicing is a vitally important function of the spaceport. Maintenance and repair will be required both on various spacecraft and on the spaceport itself, and will include both scheduled and unscheduled servicing.

6.2.1 GENERAL DEFINITION

Scheduled servicing is defined as that work which is performed to ensure on-going operation of the vehicle. It includes vehicle inspection, component testing, replenishment of depleted resources, preventive maintenance, changing out of equipment, and mission-specific reconfigurations.

Unscheduled servicing is that which is needed to restore the vehicle to an acceptable level of operation following an off-nominal occurence. It includes any repair or maintenance necessary to effect this. Scenarios requiring both of these types of servicing can be predicted and trained for. Most of the LV and Moonport R/D vehicles scheduled servicing is done at Moonport; however, OTV scheduled servicing is done in LEO.

6.2.2 METHODS AND EQUIPMENT

Many space-based servicing techniques currently require some amount of EVA or Internal Vehicular Activity (IVA) interaction; however, there is a widely recognized push towards fuller automation. Factors include the small crew size planned for the space station and spaceports, the amount of time required for servicing, and the time and cost of EVA training.

Although servicing is targeted to be as automated as possible, retention of humans in the control loop through teleoperation or other remote control systems has many advantages. Services should therefore be either completely automated, automated with remote control backup, or completely remotely-controlled. EVA and IVA should only be required as backup or as contingency operations; minimizing required EVA will also minimize the crews' exposure to radiation.

It is not within the scope of this report to detail specific servicing tools or methods, since the equipment needed will depend on the specific vehicles chosen and on the types of repair and maintenance predicted at that time. In addition, some work still needs to be done to develop appropriate servicing equipment or techniques, and to automate the equipment as much as possible.

Many of the vehicle retrieval candidates, as previously discussed, have limited remote servicing capabilities with the pre-flight attachment of specialized repair kits. The majority of servicing will most likely be done in a hangar or garage facility (Section 6.2.3) with the MRMS and attached tools, or with variations of the retrieval vehicle repair kits. Attachments for the MRMS currently include the Manipulator Foot Restraint and Work Restraint Unit, which provide a stable platform and a stable work restraint for manned activity; a tilt table for properly orienting the vehicle being reserviced; and MRMS special purpose end effectors, or tools, such as appear in Figure 6.2[2].

6.2.3 HANGAR FACILITY

A hangar or garage is recommended for vehicle servicing, refueling, and storage. Such a facility could provide radiation, thermal, and micrometeoroid protection in the form of a lighted, contained environment. This would allow safe storage and servicing of vehicles, equipment, tools, and spare parts, as well as provide for safer EVA activity.

6.2.3.1 OPTIONS

Many shapes and locations for the hangar are possible. Because of the projected heavy use of the facility, it is recommended that it be attached to the spaceport structure to avoid control problems that would arise if it

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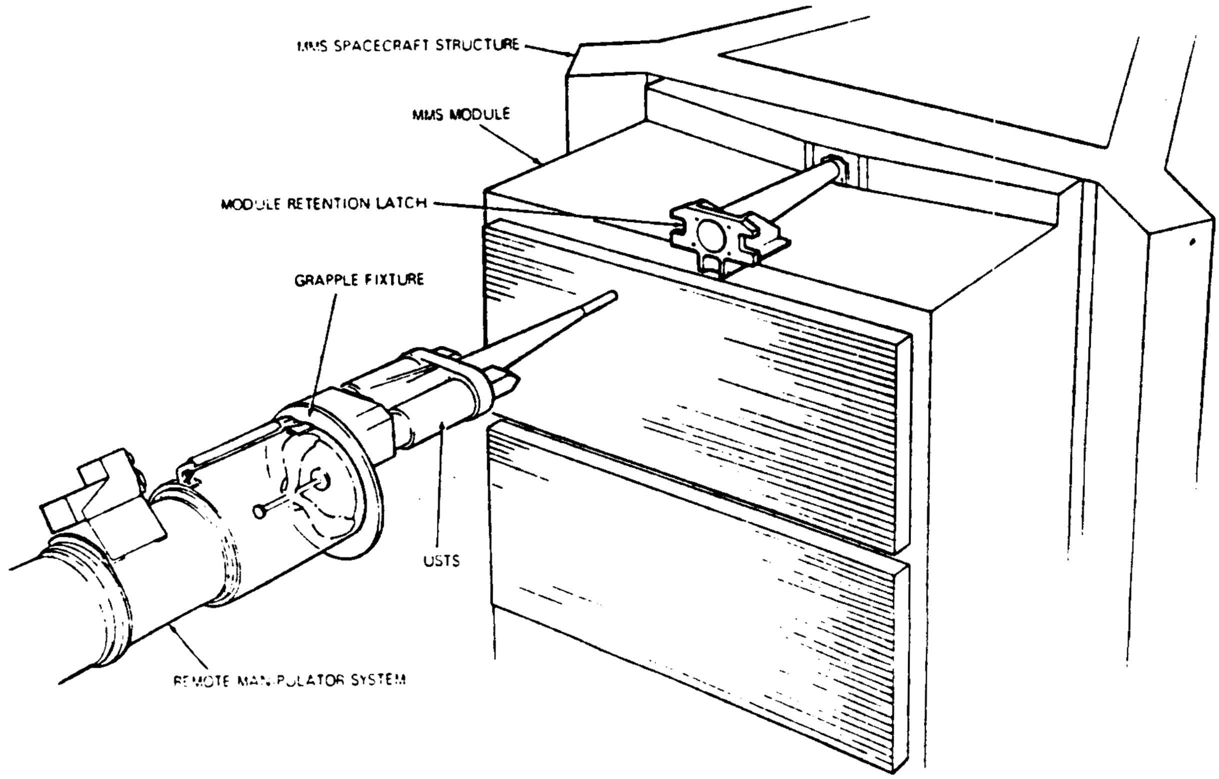
UNIVERSAL SERVICE TOOL SYSTEM (USTS) CONCEPT



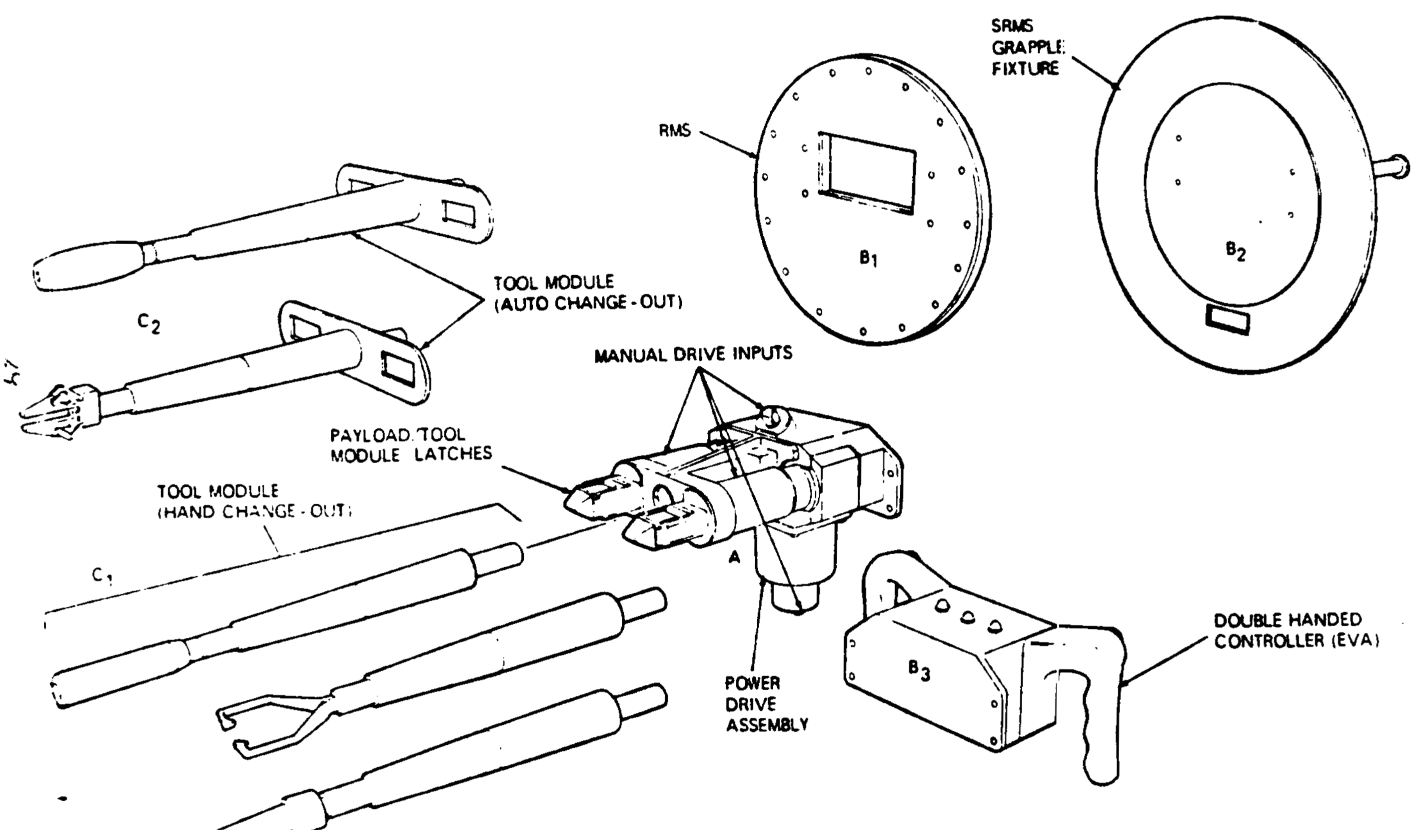


FIGURE 6.2 - TYPICAL RMS END EFFECTORS[2]

were tethered or free-floating. An example of a 41m x 23m (135x75 ft) unpressurized hangar, designed by Martin Marietta and capable of berthing a two-stage OTV with PL, appears in Figures 6.3[3] and 6.4[4]; this design was used as the basis for the Moonport servicing hangar.

Other considerations are whether the hangar should be pressurized, enclosed, or heavily shielded, and whether there should be separate facilities for servicing and refueling. If enclosed, the question of doors arises. Advantages and disadvantages for each option are listed in Table 6.2.

6.2.3.2 RECOMMENDATION

The recommended servicing hangar configuration is illustrated in Figure 5.11. The servicing and refueling facilities will be separate to minimize possible contamination from fueling leaks and to make efficient use of work space and scheduling. The fueling facility is described in Section 6.3. The hangar is enclosed (aluminum shell) to provide a lighted, contained environment with micrometeoroid protection and some thermal control, but it is not fully shielded against cosmic radiation--at least initially. Some provision will be made to allow later addition of further shielding (most likely comprised of extra layers of aluminum or lunar regolith), if so desired.

Doors are recommended to avoid full-sun or deep-space exposure of delicate components uncovered during servicing, and to provide more uniform thermal control. The hangar opens at both ends with two-part, electrically-operated doors, a manual override capability is included. Having doors at both ends of the hangar provides redundancy and will minimize scheduling conflicts.

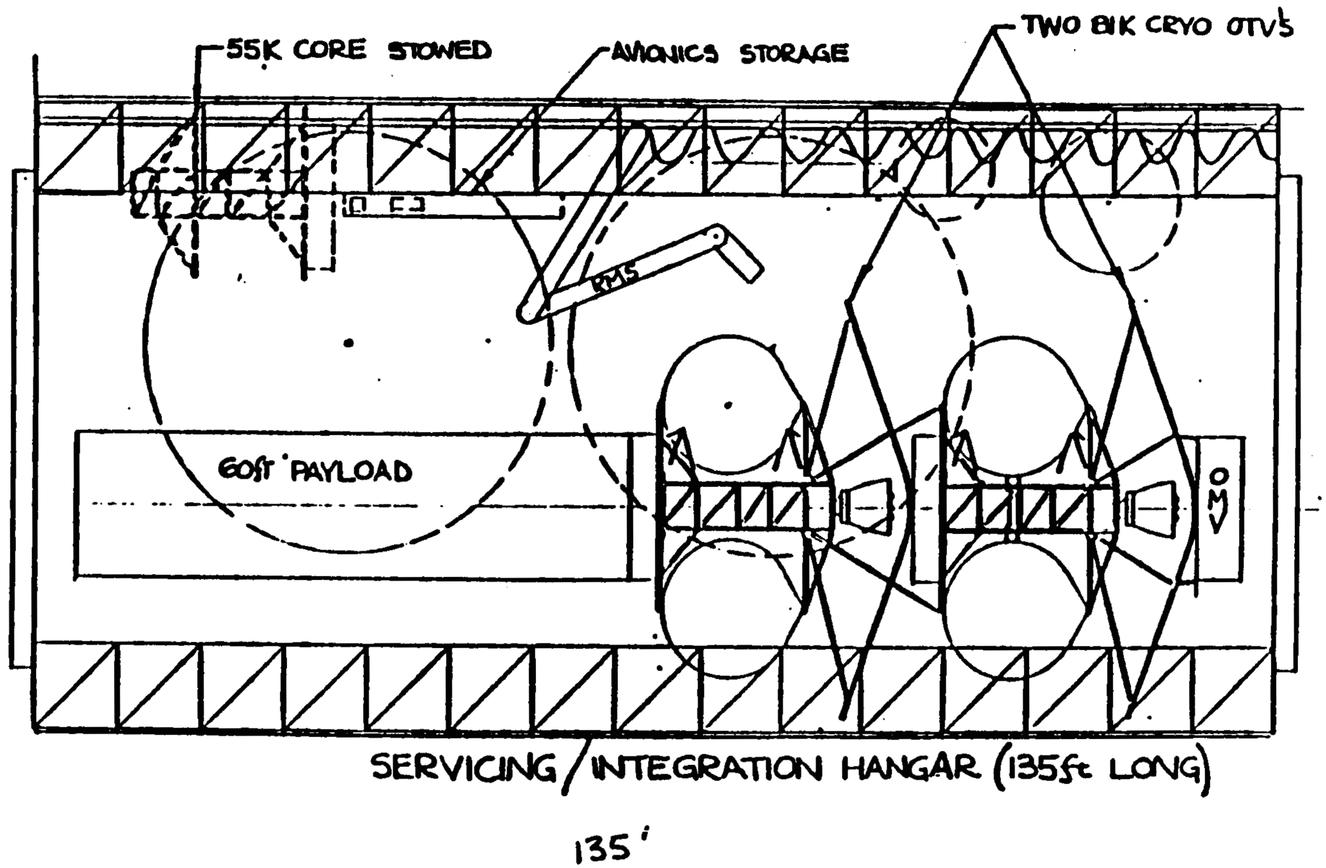


FIGURE 6.3 - VEHICLE SERVICING HANGER[3]

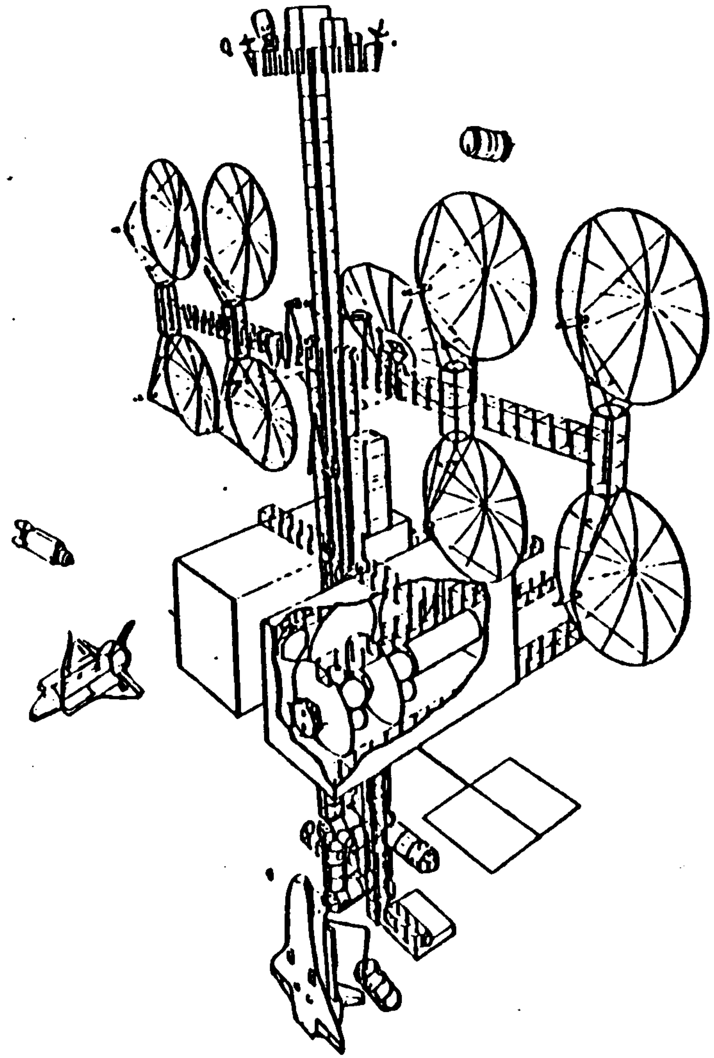


FIGURE 6.4 - PERSPECTIVE VIEW OF SERVICING HANGER[4]

TABLE 6.2 · EVALUATION OF HANGAR OPTIONS

OPTION	ADVANTAGES	DISADVANTAGES
ENCLOSED	* better protection; no worry about orientation * accident containment	* greater mass and less drag * could be perturbation by door opening/closing
PRESSURIZED	* shirtsleeve environ.; easier EVA	* lose atmosphere or use more power recycling * extra equip., mass, & complexity * less fail-safe (i.e. sudden decompression)
SHIELDED	* better radiation protection	* substantially more mass * if minimize EVA, shielding = wasted effort
SEPARATE FUEL/SERV. HANGARS	* minimize contaminat. * allow simultaneous fueling & servicing on two vehicles * provides redundancy * enlarges storage area	* increases system mass

The hangar configuration recommended consists of a vertically-symmetric, polygon-shaped hangar, enclosed and unpressurized, located along the length of the inside of the delta structure. This location was chosen in order to have efficient use of the area inside the delta, to maximize storage space on the outside of the truss faces, and to use the truss structure for partial support of the hangar shell.

In addition to the large doors, several manually-operated escape hatches will be located along the flat bottom of the hangar either for (1) emergency evacuation of EVA personnel, or (2) as an alternative EVA exit under nominal conditions conserve door power.

Pressurization is not recommended for the hangar because of the large volume of air involved (25944 m^3; 916203 ft^3) and the complexity of the machinery required to operate recycling on such a large scale. In addition, greater wall thickness (i.e. greater mass) is required for a pressurized environment, and failure of a pressurized environment is more catastrophic, particularly for people in shirtsleeves whose only barrier against the vacuum of space may have just ruptured.

Two MRMSs, mounted on tracks running the entire length of the upper inclined walls, will be used for in-hangar spacecraft movement, stabilization, and servicing. These MRMS's will be shorter and sturdier than the others on the spaceport, since they will need to operate in a limited-room environment.

This servicing hangar configuration provides berthing space for two single-stage OTVs without payload, plus a small number of manned and unmanned lunar vehicles. The specific number depends on the amount of space taken up by parts/tool storage, by the MRMSs, and by other servicing equipment, and on the number of berthing mechanisms inside the facility. If much servicing is not required overall, the area can be used for temporary and long-term storage of vehicles. This allows the spacecraft to be berthed in a more protected environment The space below the hangar within the delta structure can be used to store payloads or other supplies, and can also function as an escape route from the hangar "floor" hatches.

6.3 REFUELING

Replenishment of fluids and other consumables, particularly of propellants and pressurants, is another major capability needed at the spaceport. This function would involve replenishment or replacement of primary propellants, secondary propellants, pressurants, and reactants for electrical power or other subsystems, as well as fluid transportation and storage. Primary fluids include LH2/LO2, MMH, NTO (N_2O_4), GN_2, N2H4, and gaseous helium. Other fluids include freon, ammonia, methanol, superfluid helium, and water. Table 6.3 below shows propellant types and approximate amounts for some of the planned vehicles.

It is assumed that the initial 73,500 kg (162,000 lbm) of OTV propellant includes the amount needed for return, so Moonport refueling of the OTV is not planned at this point.

6.3.1 PROPELLANT TRANSPORTATION AND STORAGE

The primary focus of this portion of the study is on LH2/LO2 for two reasons. First, LH2 is one of the most difficult fluids to handle in space since it is very toxic, corrosive, and explosive. In addition, it has a low density and a high boiloff rate. Second, LH2/LO2 comprises the bulk of

TABLE 6.3 CURRENT-DESIGN VEHICLE PROPELLANT AMOUNTS

VEHICLE	PROPELLANT	KG	LBM
OTV/TMS	LH2/LO2	36700	81000
OMV	NTO/MMH;GN2	3175	7000
UMLV	LH2/LO2	19500	43000
MLV	LH2/LO2	5900	13000
SLV	LH2/LO2	47627	105000
POM	GN2	130-160	300-350
MMU	GN2	10	22.4

propellant required at Moonport, both before and during steady state lunar LO2 production. In general, the transportation, storage, and refueling of other fluids such as hydrazine and GN2 is either simpler or on a much smaller scale.

6.3.1.1 LH2/LO2 TARGET RESUPPLY AMOUNT

From the perspective of Moonport propellant-handling, two distinct phases of operation are assumed. The first is pre-lunar-LO2-production, and the second is steady-state lunar-LO2-production.

During the first phase, OTVs will be carrying construction materials and other supplies from Earth to LLO. Part of this payload includes expendable LV's which arrive in LLO fully fueled for the descent to the lunar surface. Because they are expendable, they will not need refueling. In addition to these vehicles, approximately six reusable, manned LVs based on the Symphony Model lunar vehicle will be used per year during construction. This amounts to a storage requirement of approximately 50 mt (144 kips) of LH2/LO2 every two months. In this case, all propellant is transported from Earth and stored on Moonport.

During the second phase, LH2 (and, as before, secondary fluids) will be shipped from Earth to be stored on Moonport for refueling, while LO2 will be delivered from the lunar surface to Moonport, where it will be stored both for refueling and as Earth-bound payload. Since LO2 comprises 7/8 of the total propellant mass required per vehicle, lunar LO2 production will considerably reduce supply requirements from Earth.

Refueling will be necessary during this steady-state phase for both manned and unmanned LV's. For more detail on the preliminary traffic model on which the storage requirements are based, see section 4. Refueling requirements plus predicted LO2 payload production amounts results in Moonport storage requirements of at least 50 mt (803 kips) LH2 every 20 days and 250 mt (247.5 kips) LO2 every 30 days. Provisions for propellant storage should include a ten percent reserve and allowances for one missed resupply or pickup. Most current LH2/LO2 storage facility concepts baseline a 45.360-90.720 MT (100.000-200.000 kips) total propellant capacity.

6.3.1.2 LH2/LO2 vs. WATER TRANSPORTATION AND STORAGE

Before production of lunar LO2, propellant will need to be supplied entirely from Earth. It can be transported as either LH2/LO2 or as water (with later electrolysis to obtain the LH2 and LO2), and can be either sent out originally with the Moonport/LTV or supplied later to the established Moonport via regularly scheduled OTV shipments. Transporting and storing the propellant as water has many advantages over transporting and storing LH2/LO2. Advantages and disadvantages of these two methods are listed in Table 6.4. Various transportation and storage scenarios for use before production of lunar LO2 are listed and evaluated in Table 6.5.

Further study needs to be done to compare power requirements of the thermal control system chosen for each of the two methods (water or LH2/LO2), to calculate electrolysis power and system requirements, and to compare transportation costs. In addition, it must be kept in mind that the need for electrolysis in the steady-state phase is greatly reduced, as water will be used then only as backup or as reserve.

An illustration of one of the Long Term Cryogenic Storage Facility System (OTV Cryopropellant Depot) concepts by General Dynamics appears in Figure 6.5[5].

An LO2/LH2 storage mass ratio of 4.3:1 was recommended by Martin Marietta in space station studies for the 45.360 mt (100.000 kips) capacity Tethered Orbital Refueling Facility. This value would allow for long-term storage of the propellant as LH2/LO2 (therefore subject to greater boiloff losses), and an OTV resupply ratio of 6:1.

TABLE 6.4 LH2/LO2 VS. WATER STORAGE

	ADVANTAGES	DISADVANTAGES
LH2/LO2	• already in prop. form	• losses due to boiloff • extra structure wt. used to separate tanks, TCS, and fluid management systems • larger storage volume • tanks require high pressure or cryogenic cooling
WATER	• non-flammable, non-toxic, non-explosive, non-corrosive • no boiloff problem • compact volume (1/3 LH22/LO2 volume) • variable uses, incl. shielding • relativley simple storage	• takes power to crack • electrolysis equip. needed • propellant not avail. 'on call'

(*) Water could be transported from Earth to Earth-orbit via Heavy Lift Launch Vehicle, and from Earth-orbit to Moonport via OTV or LTV.

TABLE 6.5 TRANSPORTATION AND STORAGE SCENARIOS

TRANSPORT - STORE (PRE-LUNAR-LO2)	MIN. STRUCT. MASS	MIN. FLUID MASS	MIN. PROP. LOSSES	MIN. VOLUME	XPORT SAFETY	STORAGE SAFETY	TOTAL
LH2/LO2 - LH2/LO2	4	1	5	4	3	4	21
LH2/LO2 - TANKS *	–	–	–	–	–	–	–
WATER - LH2/LO2	2	1	3	3	1	4	14
WATER - WATER	1	2	1	1	1	1	7
WATER - BOTH **	3	3	2	2	1	2	13

* variable depending on number and size of tanks
** comparison figures will change with different proportions of LH2/LO2 to other storage

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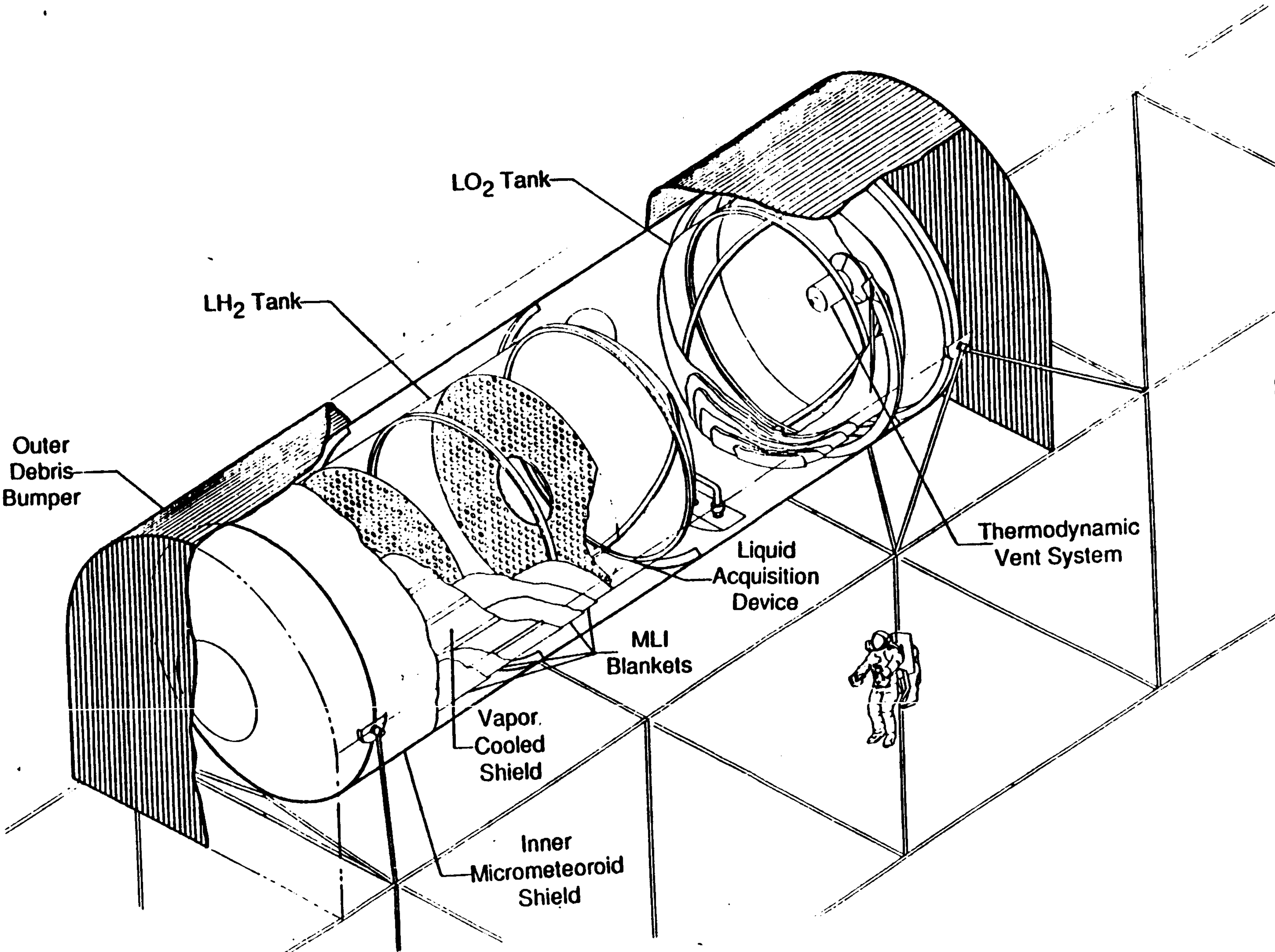

FIGURE 6-5 - LONG TERM CRYOGENIC STORAGE FACILITY [5]

However, given the possibility of long-term storage as water and an LV resupply ratio requirement of 7:1, a 7:1 short-term storage ratio was deemed to be more appropriate for the Moonport case. Comparisons of needed masses and volumes for the two given storage ratios are shown in Table 6.6. The total mass of 45.360 mt (100.00 kips) was chosen both for comparison purposes, and because most current studies target that capacity.

A note of interest is that the volume for 45.360 mt of water (488 m^3; 1600 ft^3) is approximately one-third the volume of the same mass of 7:1 LH2/LO2 (1496 m^3; 4909 ft^3). The densities of water, LO2, and LH2 are 1000 (62.43), 1010 (63.04), and 62.2 (3.88) kg/m^3 (lbm/ft^3), respectively. The amounts of LH2 and LO2 and of water were also calculated for 156,760 kg (345,600 lbm) of propellant (7:1), (this is the amount required for three lunar vehicle flights plus reserve). These numbers appear in Table 6.7.

TABLE 6.6 MASSES AND VOLUMES FOR MASS RATIOS OF 4.3 AND 7

MASS RATIO	PARAMETER	LH2	LO2	TOTAL
4.3:1	mass(kg;lbm) vol. (m3;ft3)	8620(19000) 91 (4900)	36470(81000) 36 (1285)	45360(100000) 176 (6185)
7:1	mass(kg;lbm) vol. (m3;ft3)	5670(12500) 91 (3221)	39690(87500) 39 (1388)	45360(100000) 131 (4609)

TABLE 6.7 PROPELLANT MASSES AND VOLUMES FOR THREE LV FLIGHTS PLUS RESERVE

	LH2	LO2	TOTAL	WATER
mass(kg;lbm)	19600 (43200)	137170 (302400)	156760 (345600)	176360 (388800)
vol. (m3;ft3)	315 (11134)	136 (4797)	451 (15931)	176 (6228)

* yields 156760 kg LO2 (155 m3) & LH2 shown in left hand column

In order to obtain 19.600 MT of LH2 from electrolysis, 176.360 MT (388.800 kips; 176 ft^3) of water is required (less water is needed to produce the required amount of LO2). This amount yields the values indicated on the arrow in Table 6.7. Uses will need to be found for the excess LO2. More research needs to be done on spaceport electrolysis methods, but it is assumed that the water would be "cracked" (electrolyzed) during refueling, and then either transferred directly to the receiving vehicle or stored in small short-term storage tanks enroute to the vehicle. A working number for electrolysis power requirements is 4.41 kW/kg (2 kW/lbm) water.

6.3.1.3 FUTURE OPTIONS

Future options in propellant transportation, storage, and refueling are:

1) water storage, electrolysis, and LV refueling on the lunar surface
2) development of all-lunar-material propulsion systems (sulfur/LO2, phosphorus/LO2, magnesium/LO2, aluminum/LO2, aluminum/helium/LO2) to minimize both required propellant mass and reliance on lunar port or earth supplies.
3) finding a use for excess LO2 caused by electrolysis

4) development of a safe and efficient method for transporting LH2 from Earth to LLO

6.3.1.4 RECOMMENDATIONS

Prior to lunar LO2 production, it is recommended that LH2/LO2 be transported from the Earth as water, and that the bulk of it be stored as water to minimize required volume, boil-off losses, and system complexity. A relatively small amount of the propellant should be kept available in LH2/LO2 form. In addition, rather than upgrading a current storage depot concept to handle a large amount of propellant per month; or designing a new larger-capacity depot; several depots are baselined in this study, primarily for redundancy, safety, and modularity.

During lunar LO2 production, it is recommended that LH2 be transported from Earth for refueling, along with some water for reserve or non-propellant uses. Meanwhile, LO2 should be transported from the moon. A driving technology is the development of a safe and efficient method for long-term LH2 transportation and storage. These recommendations are summarized in Table 6.8 below.

TABLE 6.8 RECOMMMENDED TRANSPORTATION AND STORAGE

	PRE-LUNAR-LO2	LUNAR LO2 PRODUCTION
FORM FOR TRANSPORT	WATER (EARTH)	LH2(EARTH), LO2(MOON)
FORM FOR STORAGE	WATER (MOST), LH2/LO2 (SOME)	LH2 (FUEL), LO2(FUEL, PAYLOAD), H20(BACKUP)
AMOUNTS NEEDED	50 MT LH2/LO2 PER 60 DAYS	LH2: 50 MT / 20 DAYS LO2: 250 MT / 30 DAYS

6.3.2 PROPELLANT TRANSFER

Many studies have been done on fluid transfer in low-gravity and vacuum environments. The specific method of transfer depends on the types of fluids transferred and on the types of supplying and receiving tanks. Briefly, the three major methods are ullage recompression, ullage vent, and ullage exchange, where ullage is the gas pressurizing the tank. Each has its advantages and disadvantages, but an evaluation should be done in context with the other specific design characteristics of the refueling system, and this is beyond the scope of this study.

6.3.2.1 REFUELING CONCEPT CANDIDATES

Refueling methods are mostly conceptual at this point. The Orbital Refueling System, a small-scale hydrazine transfer facility (0.9x1.22x1.52 m; 3x4x5 ft) requiring EVA, is one of the only designs to have been flown and tested on the space shuttle.

The Orbital Spacecraft Consumables Resupply System (OSCRS) has received considerable study by Rockwell International. The initial configuration was designed to resupply the Gamma Ray Observatory with hydrazine; growth is projected to either 9000 lbm or 18,000 lbm of bipropellant (NTO/MMH) and pressurants. Its size and shape are currently configured for the space shuttle cargo bay. A new structure utilizing the OSCRS technology could be designed, but this depends on the retrieval vehicle propulsion system and requirements. The OSCRS initial design requires some EVA interaction, but the OSCRS study targets automatic/remote operations as an eventual goal.

Another method studied is the Orbital Resupply Module Concept, which would be carried by an OMV or OTV. Capacity would include 20,400 kg (45,000 lbm) of propellant and almost 544 kg (1200 lbm) of helium.

6.3.2.2 RECOMMENDATIONS

Because these designs are concepts only, a decision matrix cannot yet be made. A system such as OSCRS, however, will most likely be used for MMH/NTO, pressurant, and secondary fluid resupply, since that design already incorporates bipropellant capability and a push for automation. Moonport OSCRS capacity depends on the retrieval vehicle selection and frequency of flights, and on bipropellant (NTO/MMH) storage characteristics. It is recommended that these fluids not be resupplied from Earth more often than the LH2/LO2 or water. A fleet of two or three Moonport OSCRS (upgraded) vehicles is necessary. In addition, tanks such as in Figure 6.5 should be used for LH2/LO2 storage. For steady-state operations, payload LO2 should be kept in its own storage container (or module), while that used for refueling should be kept in another. This aids Earth-bound transport and provides redundancy.

6.3.3 REFUELING FACILITIES

Safety is a major requirement for any system. As previously discussed in the servicing hangar section , the servicing and refueling areas will be separate to achieve minimum contamination and scheduling problems, and maximum safety. In addition, The refueling facility should provide berthing area for vehicles being refueled, along with storage area for propellants, pressurants, secondary fluids, electrolysis equipment and refueling equipment. The area should also be isolated from habitation modules in case of a rupture, leak, or other hazardous event. Finally, the refueling facility must be able to store the required propellant quantities .

6.3.3.1 OPTIONS

Possible locations for the refueling facility include: the inside of the delta structure--either as part of the servicing hangar or separate from it; the outside of the delta faces; or somewhere along the boom structure. And as with the servicing hangar, the facility can be enclosed, pressurized, or shielded.

6.3.3.1 RECOMMENDATION

The Moonport refueling facility consists of three 9x30 m truss sections for vehicle berthing, and three 5x30 m tank storage areas (or six 5x15 m areas), all joined together in a miniature delta on the boom structure. These tank areas can be used to hold storage tanks for LH2/LO2, water, and other consumables; to store equipment for electrolysis or other refueling; and to store OSCRS-type refueling vehicles.

The refueling facility is located 25 m down the boom from the main delta truss to provide both isolation for the delta and clearance for vehicles using the "back door" of the servicing facility. The berthing trusses are 9 m wide to provide secure berthing for the large-size SLV, and should include many berthing or docking fixtures. This would allow relatively out-of-the-way short-term storage of vehicles they are being neither serviced nor refueled.

Two MRMSs are baselined for this structure, to run on tracks on either side of the berthing platform and around the boom components, as in the illustrations. These MRMSs will be used to dock vehicles and assist in refueling operations, including deployment of the OSCRS vehicles.

The size of the storage tank areas allows for more than enough propellant storage as well as for expandability and versatility.

The refueling area is not enclosed, as enclosing adds more mass, and an enclosed triangular shape would not be efficient in the amount of useful working room. It was determined that micrometeoroid protection and evenness of thermal control are not as crucial for the refueling facility as for the servicing hangar for primarily two reasons. First, the propellant tanks will have their own protection and thermal control systems, and second, the vehicles should be designed such that no delicate components will be exposed during fueling operations. In addition, any contamination from leakages would either be quickly dispersed in the vacuum of space, or contained by a waste scavenging system designed for just such a purpose.

SECTION 6 REFERENCES: VEHICLE SERVICING

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3. Martin Marietta. Orbital Transfer Vehicle Concept Definition and System Analysis Study. Denver, Colorado: August, 1985, pp,107.

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SECTION 7

HABITATION AND RADIATION

7.0 HABITATION AND RADIATION

The efficiency of Moonport greatly depends on the efficiency of its major elements. This chapter will address the human element. The habitation analysis group is composed of two design engineers. One engineer focused on radiation and the other focused on habitable environments.

7.1 HABITATION

The habitation analysis group was tasked with developing a preliminary design for the crew accommodations on the Moonport facility. The habitation scenarios addressed are:

-support of a 5-6 member steady-state crew for 6 months with a resupply interval of 3 months
-support of a lunar base construction crew until the base is habitable (maximum of 14 people)
-support of lunar base personnel in the event of an emergency evacuation (maximum of 30 people)

The requirements for human sustenance in a lunar environment include: hygiene facilities; a health maintenance facility; radiation protection; and a partially closed Environmental Control/Life Support System (ECLSS) based on closing the metabolic oxygen and water cycles (space station technology).

A "black box" approach is being taken in designing the modules to meet these basic requirements, although specific equipment deemed necessary within these "boxes" will be denoted. The specifics of radiation shielding are addressed in the radiation section of this document; however, the radiation protection concept has been incorporated in the following habitation analysis. The ultimate goals of this analysis were to determine the volume requirements of the crew accommodations, broken into subsystems, and to choose a general module design.

7.1.1 VOLUME SIZING

The primary focus of the habitation design engineers has been to propose an environment which optimizes human productivity. To this end, the following design guidelines were established:

1) congestion avoidance (i.e., optimum free-space allocation)

2) promotion of crew interrelationship

3) good accessibility to facilities and equipment

4) design facilities to meet the health problems of a zero-gravity environment (e.g., resistance exercises and a strong vertical orientation)

The first question to address in volume sizing was to determine the optimum free-space, or personal living space, to be allocated to each crew member. The curve in Figure 7.1 was taken from a bioastronautics study[1], and was used to determine this free space allocation. For steady-state operation, it was determined that the optimum volume per member is 7.1 cubic meters (250 cu ft), resulting in a crew quarters requirement of 48 cubic meters (1700 cu ft).

For the emergency and construction scenarios, the group has baselined the free-space volume requirement at the emergency minimum denoted on the curve. This value is 3 cubic meters (100 cu ft) per member. Since these scenarios would allow volume usage in shifts, the total volume requirement for them is approximately equal to the steady-state scenario. Thus, the subsequent analysis is based on steady-state operation with the

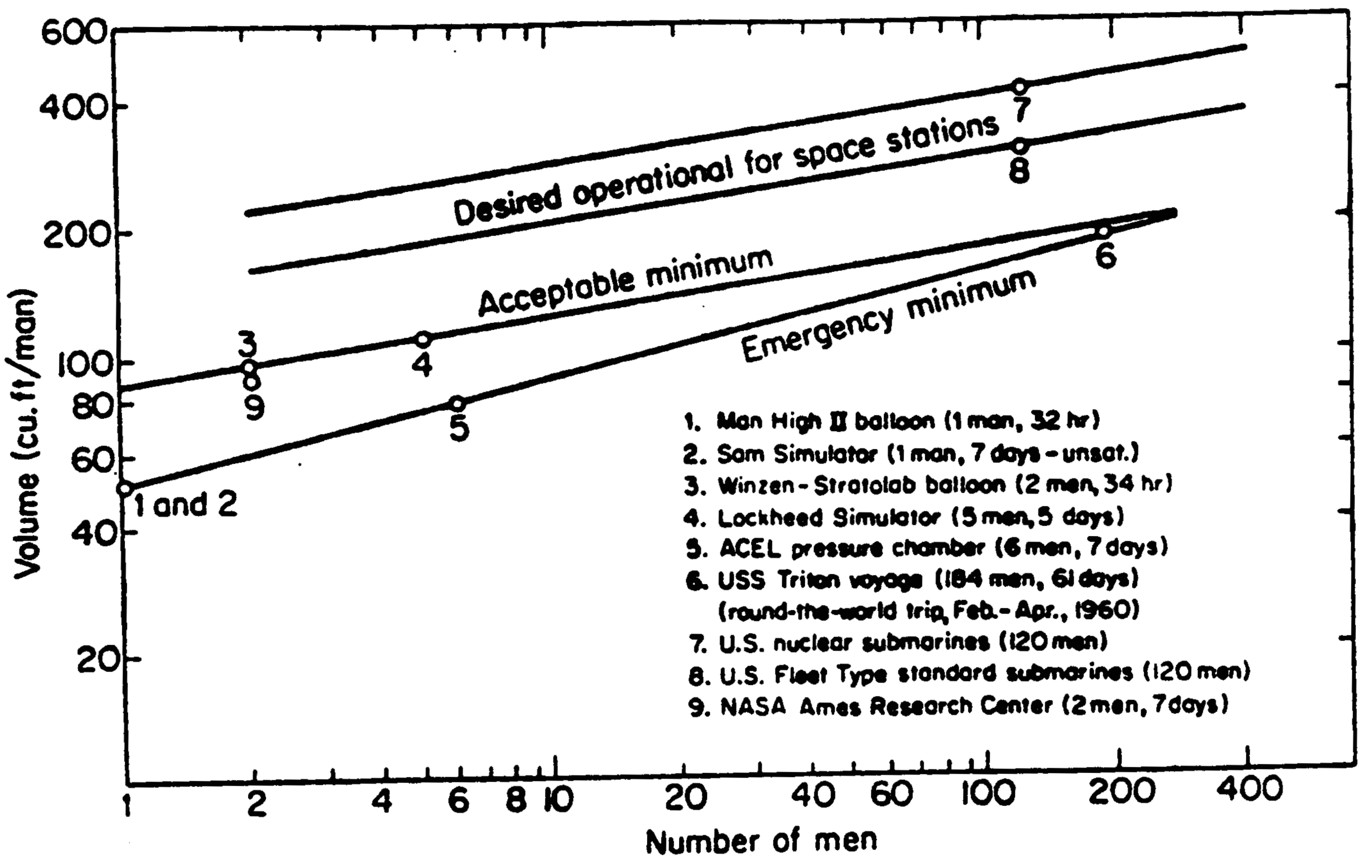

FIGURE 7.1 - LIVING SPACE PER MAN FOR SEVERAL[1]
SEALED CABIN EXPERIMENTS

assumption that the habitation modules allocated for the construction and emergency scenarios will be modified as necessary.

To address additional volume sizing of the systems, the analysis group chose to use the 1985 Space Station module configuration[2] as a baseline. Adjustments to the equipment and access volumes were then made to meet this project's specific guidelines. Tables 7.1, 7.2, and 7.3 contain the volume allocations for the following subsystems.

7.1.1.1 CREW QUARTERS

The private quarters for each crew member will be located in Habitat Module 2, and will provide approximately 7 cubic meters (250 cu ft) to each member, as opposed to the space station allocation of 4.7 cubic meters (150 cu ft) . The following equipment is deemed necessary within that volume:

1) sleep station
2) IVA communications
3) desk
4) portable command/control center
5) storage volume of 0.6 cubic meters (20 cu ft)
6) audio/video entertainment
7) library (actual books) volume of 0.6 cubic meters (20 cu ft)

7.1.1.2 GALLEY/WARDROOM

To promote crew interaction, the galley/wardroom, which is essentially the meeting/game/dining room and kitchen, is allocated 47 cubic meters

(1650 cu ft). The Space Station allocation was 34 cubic meters (1195 cu ft) for the galley/wardroom, resulting in less than a square meter (2 sq ft) per crew member. Located in Habitation Module 1, the galley/wardroom will require the following:

1) equipment and supplies necessary for the storage and preparation of food and drink (Note: Storage will be for 14 days with the remaining 90-day supply stored in the MWRL module)
2) dining area
3) audio/video entertainment equipment
4) game kits
5) windows
6) IVA communications

7.1.1.3 HEALTH MAINTENANCE FACILITY/EXERCISE AREA

In both habitation modules, 6.5 cubic meters (229 cu ft) is allocated for medical supplies and monitors. To promote crew interrelations and good zero-g health, there will be an exercise/miniature gym area in the Morale, Welfare, Recreation, and Logistics (MWRL) module along with the 90-day stores. Table 7.3 outlines the volume allocations for the MWRL module. Forty cubic meters (1420 cu ft) has been allocated for the exercise facility, which will contain two bicycle ergometers, two treadmills, some resistance exercise machines, and some type of zero-g competitive game equipment.

TABLE 7.1 MOONPORT HABITAT MODULE 1 VOLUME ALLOCATION (cubic meters)

Inside Ceiling and Floor	Equipment	Access
Maintenance Work Station	4.25	6.92
Command/Control Station	2.83	2.50
Medical	2.50	4.00
Hygiene	6.46	3.45
Galley/Wardroom (inc. equiop)	25.5	18.4
Solar Activity Monitor	1.42	1.70
Storage Area	5.66	2.83
Secondary Structure	3.40	COM
Subtotal	52.00	35.80
Total	87.8	
Outside Ceiling and Floor	Equipment	Access
ECLSS	4.59	COM
Lighting	1.10	COM
Utilities	2.07	COM
Storage (spare)	9.37	1.10
Subtotal	17.1	1.10
Total	18.2	
Connecting End	35.6	
GRAND TOTAL	**141.6**	

COM: Common Access Area

TABLE 7.2 - MOONPORT HABITAT MODULE 2 VOLUME ALLOCATION (cubic meters)

Inside Ceiling and Floor	Equipment	Access
Laundry Facility	1.10	1.90
Solar Activity Analysis Station	2.83	4.25
Secondary Control Station	1.42	1.70
Medical	2.50	3.99
Crew Quarters	42.48	5.66
Hygiene	6.48	3.45
Secondary Structure	10.02	COM
Subtotal	66.8	21.0
Total	87.8	
Outside Ceiling and Floor	**Equipment**	**Access**
ECLSS	4.59	COM
Lighting	0.82	COM
Utitlities	2.07	COM
Storage	9.66	1.05
Subtotal	17.13	1.05
Total	18.2	
Connecting End	35.6	
GRAND TOTAL	**141.6**	

COM: Common Access Area

TABLE 7.3 MOONPORT MWRL MODULE VOLUME ALLOCATIONS (cubic meters)

Inside Ceiling and Floor	Equipment	Access
Freezer	1.70	1.56
Refrigerator	0.57	0.57
Storage Area	15.0	13.30
Excersize Facility	28.04	27.10
Subtotal	45.3	42.5
Total	87.8	
Outside Ceiling and Floor	**Equipment**	**Access**
ECLSS	4.59	COM
Lighting	0.82	COM
Utitlities	2.07	COM
Storage	9.66	1.05
Subtotal	17.1	1.05
Total	18.2	
Connecting End	35.6	
GRAND TOTAL	**141.6**	

COM: Common Access Area

7.1.1.4 PERSONAL HYGIENE AREAS

Located in both habitation modules, these areas provide facilities for body waste collection and disposal, personal cleanliness, and bathing. Although space station technology is assumed for the equipment, the access volume was increased by 0.1 cubic meters (4 cu ft) to increase the "free space" in the hygiene areas. This results in a total volume of 10 cubic meters (350 cu ft).

7.1.1.5 WORKSTATIONS

Habitation Module 1 will contain the major Command/Control workstation, an equipment maintenance workstation, and a solar activity monitor station. The volumes allocated for these workstations may be found in Table 7.1.

The laundry facility, a secondary control station, and the Solar Activity Analysis station will be located in Habitat Module 2 (see Table 7.2). The Solar Activity workstations are there primarily to warn the crew of solar storms. Thus, an alarm system, routed to every module,will also be included in the final design. An additional objective of the Solar Activity Analysis station is to monitor, record, and analyze solar activity data. This data is to be used in radiation protection research, since solar radiation is a key problem in prolonged space habitation.

7.1.1.6 SECONDARY STRUCTURE

Secondary structure in the modules consists of walls, floors, tunnels, and other frameworks. Tables 7.1, 7.2, and 7.3 denote the volumes allocated to secondary structure for each of the modules. As Table 7.2 indicates, the secondary structure volume allocated in Habitat Module 2 is much greater than in the other two modules. This is because the walls surrounding the crew's quarters are to be additionally shielded against radiation, resulting in a solar storm shelter. These walls include the inner walls of the crew quarters, so that 360° of protection will be provided. This will not only provide protection against solar storms, but will also decrease the amount of overall radiation the crew absorbs .

7.1.1.7 ADDITIONAL SUBSYSTEMS

The volume allocations for other subsystems, such as ECLSS and lighting, are outlined in Tables 7.1, 7.2, and 7.3. [These allocations are the same as those allotted for the Space Station in Reference 2.]

7.1.2 MODULE CONFIGURATION

The two shapes that were investigated were the sphere and the cylinder. If the sphere had been chosen, a plate would have been placed across the center of the sphere, which would have provided a floor for the top and bottom module sections. The domes of the resulting half-spheres would have been lined with ECLSS equipment.

The chosen cylindrical shape will also use its curved "top" and "bottom" to house ECLSS equipment. Similar to the Space Station concept, two plates will be placed lengthwise in the cylinder. One plate will constitute the ceiling and the other one the floor. There will be a 2.75 meter (9 ft) clearance between the two plates. The cylinder's dimensions will be a 4 meter (13 ft) diameter and an 11.6 meter (38 ft) length.

Table 7.4 contains the decision matrix that was used to determine the shape of the Habitation and MWRL modules. As the matrix indicates, the cylindrical shape was found to be the most satisfactory in meeting analysis criteria.

7.1.3 CONCLUSION

For the steady-state operation, there will be three active modules. These will be Habitat Modules 1 and 2, and the MWRL module. Also located on the Moonport truss will be another set of three modules. These slightly modified modules will be ready to be activated for the construction/emergency scenarios. The major modifications are: (1) The crew quarters volume of Habitat Module 2 will be sectioned so that it can sleep 10 people at a time. (2) The MWRL module will essentially be a logistics module which will contain a 90-day supply of food, clothing, etc., for 30 people.

TABLE 7.4

SHAPE CONFIGURATION DECISION MATRIX

Design Criteria	Sphere	Cylinder
Shielding Mass to Volume Ratio	1	2
Expandability	2	1
Conducive to Strong Vertical Orientation Interior Design	2	1
Surface Area to Mass Ratio	1	2
Minimum Wasted Space	2	1
Ease of Configuration Integration	2	1
TOTAL	10	8

7.2 RADIATION

There are two primary sources of radiation in space near the moon. They are galactic cosmic radiation (GCR) particles and solar energetic particles (SEP). Both exist at levels that combine to create a much more hostile environment than that found in low-Earth orbit.

Galactic cosmic radiation originates outside of our solar system. It exists at levels that would provide a dose of from 20 to 50 REM (Roentgen Equivalent Man) per year to an unshielded astronaut. GCR is far more penetrating than most types of radiation because it composed of by highly energetic particles. Cosmic rays are exceptionally difficult to shield against.

Solar energetic particles originate at the sun, the highest quantity of which are present during solar flares. Although they are not as energetic as cosmic rays, SEP can exist in much greater concentrations during periods of intense solar activity. Although the occurrence of solar flares is completely random and essentially unpredictable, the sun's overall activity, and thus the probability of solar flares, follows a sinusoidal function with a period of 10.9 years.

At first glance it would seem that the highest radiation levels exist during years of the greatest solar activity (solar maxima); however, the energetic plasmas from the sun modulate the cosmic radiation during solar maxima. This results in a lower total radiation from SEP and GCR combined. The years of expected solar maxima are 1991, 2002, 2013, and 2024. Of course, the radiation from SEP will still be extremely dangerous during actual solar flares. Since the GCR is modulated to a lesser degree during years of low solar activity (solar minima), those years are the periods of the highest combined radiation levels. The years of expected solar minima are 1996, 2007, 2018, and 2029.

The exact methods of radiation transportation and inducement to humans are extremely complex; however, an estimation of the energy level from which astronauts need to be protected must be made. Satisfactory results are obtained using the following formula[3].

$$T=(1-5/3\, e^{-1.386\sqrt{RHO}}) \cdot 556 \cdot \ln(1+5.48 \cdot 10^{-6} \cdot E^{1.8}) / RHO$$

where: RHO = density of the material in g/m3,

E = particle energy level in MeV

T = required material thickness in cm..

When the radiation energy level is estimated at 150 MeV, the above equation indicates that 7.56 cm. of aluminum shielding will limit the acquired dose to 50 REM per six months. However, studies[4] indicate that 7.5 cm. of aluminum permit only 50 REM per year. As a conservative requirement, SPS used the 50 REM per six months dose for shielding calculations.

The amount of protection provided by the shielding depends much more on the mass thickness (mass per unit cross sectional area) than the nature of the material used. Thus, dense materials such as lead are not necessarily the optimum choice. This is because the highly energetic cosmic ray particles induce more hazardous secondary radiation in materials made up of heavy elements. In fact, heavy materials such as lead may be substantially worse as shielding materials in space than lighter ones such as aluminum. By the same reasoning, water and concrete are advantageous because of their high hydrogen content.

Table 7.5 contains the thicknesses and mass thicknesses for several materials required to meet the specifications given above. Table 7.6 is a comparison of the total shielding mass of the materials from Table 7.5 required for several possible structural configurations, each of the same interior volume. (Note: the interior volume of 523 m^3 was chosen at random to enable a comparison to be made.) Primary consideration for material selection must be given to mass thickness due to the economics of the huge mass involved. However, other factors such as secondary radiation susceptibility, thermal conductivity, and ease of construction must also be considered. Table 7.7 is a comparison of some of the selection criteria.

TABLE 7.5

RADIATION PARAMETERS OF SELECTED MATERIALS AT E=150 MeV

MATERIAL	DENSITY g/cm^3	THICKNESS cm	MASS THICKNESS g/cm^2
POLYETHYLENE	0.92	14.95	13.75
WATER	1.00	14.35	14.35
REGOLITH	1.20	13.02	15.62
BORON CARBIDE	1.20	13.02	15.62
GRAPHITE COMP.	1.61	10.90	17.54
GRAPHITE	1.75	10.32	18.06
FIBERGLASS	1.94	9.62	18.66
CONCRETE	2.08	9.16	19.05
SILICA GLASS	2.50	8 .01	20.03
ALUMINUM	2.70	7.56	20.41
TITANIUM	4.54	4.95	22.47
LEADED GLASS	6.20	3.76	23.31

TABLE 7.6
TOTAL SHIELDING MASS (METRIC TONS) VS MATERIALS

MATERIAL	SPHERE	CYLINDER		
		75 STA	SPC STA	MOONPORT
	R=5m V=142 m^3	R=1.98m H=9.13 m V=112 m^3	R=2.35m H=15 m V=260 m^3	R=1.98m H=11.56m V=142 m^3
POLYETHYLENE	19.02	20.15	36.84	24.52
WATER	19.81	20.97	38.36	25.51
REGOLITH	21.48	22.72	41.60	27.64
BORON CARBIDE	21.48	22.72	41.60	27.64
GRAPHITE COMP.	23.97	25.31	46.43	30.82
GRAPHITE	24.63	25.99	47.71	31.65
FIBERGLASS	25.39	26.79	49.20	32.62
CONCRETE	25.86	27.29	50.16	33.25
SILICA GLASS	27.11	28.57	52.54	34.80
ALUMINUM	27.60	29.07	53.48	35.42
TITANIUM	30.15	31.68	58.43	38.63
LEADED GLASS	31.16	32.74	60.39	39.80

TABLE 7.7 RADIATION SHIELDING MATERIALS

MATERIAL	MASS THICHNESS	STRUCTURAL CAPABILITY	LIGHT ELEMENT CONTENT	OTHER
POLYETHYLENE	LOW	NONE	VERY HIGH	M OLDABLE
WATER	LOW	NONE	VERY HIGH	
REGOLITH	LOW	BRICKS	MEDIUM	MOON
FIBERGLASS	MED	HIGH	MEDIUM	MOON
CONCRETE	MED	HIGH	HIGH	POOR THERMAL
ALUMINUM	HIGH	HIGH	HIGH	
TITANIUM	HIGH	HIGH	NONE	
LEADED GLASS	HIGH	BRITTLE	MEDIUM	CLEAR

The recommended choice of radiation shielding is polyethylene. Since it is plastic, it will be susceptible to micrometeorite damage. It will therefore require a thin shell of a hard material for protection. A laver of 0.08 cm (.032 in) of titanium is recommended because of its hardness and strength. The resulting shield will then be 14.70 cm (5.79 in) of polyethylene coated by the titanium. this will result in a total mass of 24.02 metric tons of polyethylene plus 0.69 metric tons of titanium, or 24.71 metric tons of shielding per module.

SECTION 7 REFERENCES: HABITATION AND RADIATION

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SECTION 8

POWER AND PROPULSION REQUIREMENTS

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8.0 POWER AND PROPULSION REQUIREMENTS

Two power systems were designed for Moonport. The first was a primary power plant capable of supporting operations under steady-state conditions and the second was an emergency system that provided power in the event of steady-state primary power plant failure. In addition to providing steady-state power, the primary power plant supplies the energy required for the low thrust vehicle (LTV). The design guidelines for the primary power supply were a maximum of 9 to 10 MW and a lifetime of 10 to 15 years. The emergency power supply was designed to provide 75 kW with reliability as a key design requirement.

8.1 THE PRIMARY POWER SUPPLY

As stated above, the Moonport/LTV power requirement was 9 to 10 MW. In addition, the power system must have a minimum lifetime of ten years. Figure 8.1[1] indicates a choice between solar power sources and nuclear reactors.

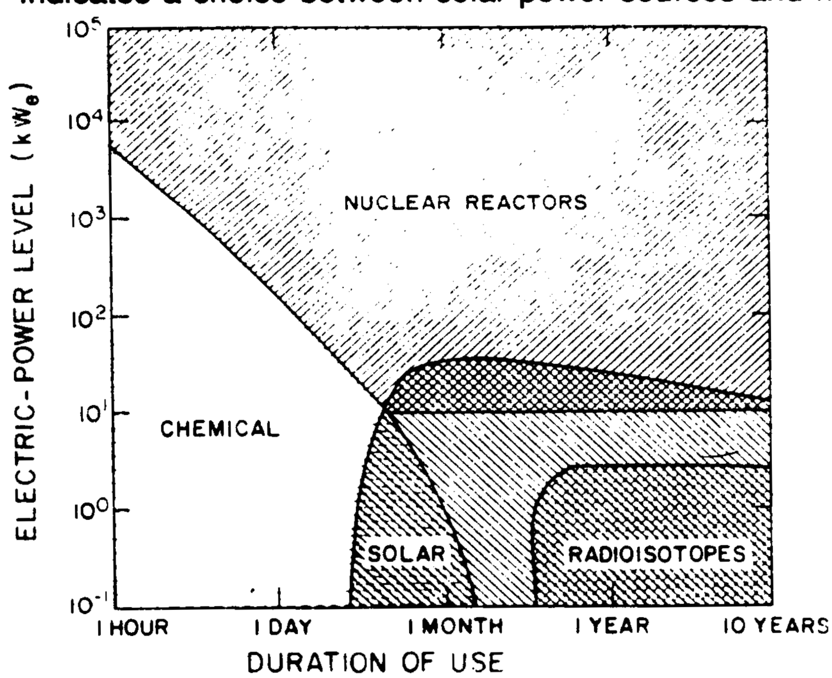

FIGURE 8.1 REGIMES OF POSSIBLE SPACE POWER APPLICABILITY[1]

8.1.1 SOLAR POWER SYSTEMS

There are three main ways of converting solar power to electrical power: photovoltaic, photo-emission, and solar thermal. Photovoltaic and photo-emission systems were the most efficient; however, from a practical point of view they could not be used because the collector area was extremely large, approximately 25,000 m^2 (269,100 ft^2).[2]

Solar thermal power uses the Sun's energy as a heat source for a dynamic power conversion system (i.e. Rankine, Brayton, and Sterling cycles). Heat generation is accomplished with parabolic solar concentrators which focus the Sun's thermal energy onto the working fluid. Structural limitations restrict the diameter of the concentrators to approximately 15 m (49.2 ft). Each collector has a power output capability of 40 kWe, yielding a collector area to power ratio of 4.42 m^2/kWe (47.5 ft^2/kW). Although solar thermal power does not require as large a collection area, the area to power ratio is still considered too large.[3]

8.1.2 NUCLEAR POWER SYSTEMS

There are presently five ways of converting reactor thermal energy to electricity. Two of these, thermoelectric and thermionic, fall under the category of static or direct energy conversion (DEC) systems. Thermoelectric converters use a temperature difference between two metals to create an electromotive force (EMF). Thermionic converters cause electrons to flow from a hot emitter to a cooler collector, thereby creating a current.

The three remaining conversion systems (the Rankine, Brayton, and Sterling cycles) are considered dynamic systems. The Rankine cycle is ideally characterized by isothermal heat addition and rejection, and isentropic expansion and compression of the working fluid. The Brayton cycle is characterized by constant pressure heat addition and rejection, and isentropic expansion and compression. The Sterling cycle has constant volume heat addition and rejection, and isothermal expansion and compression.

Of the five conversion methods discussed above, the Rankine and Brayton cycles were considered further. Thermoelectric and thermionic conversion had high efficiencies as the emitter temperature increases; however, this lead to operating temperatures of up to 2000 K, which were too high due to material melting points. The Sterling cycle offered great potential for space applications with efficiencies of 30%, but this cycle has not been adequately developed for such a large power requirement.[4]

8.1.3 THE RANKINE AND BRAYTON CYCLES

The Rankine and Brayton cycles had many advantages associated with them. To help decide which would be used for Moonport primary power, a decision matrix (Table 8.1) was created.

The efficiency from Table 8.1 was the overall thermal efficiency. The Brayton cycle had a higher thermal efficiency because of the higher operating temperatures and lower heat rejection temperatures. Although the efficiency was higher, the peak temperature puts severe thermal stresses on system components which could reduce the operating lifetime of the Brayton cycle.

TABLE 8.1 RANKINE AND BRAYTON CYCLE DESIGN CRITERIA

Criteria	Rankine	Brayton
Efficiency	2	1
Peak Temperature	1	2
Low Temperature	1	2
Back Work Ratio	1	2
Working Fluid	2	1
Total	7	8

Note: Low numbers are best

In addition, the lower heat rejection temperature for the Brayton cycle was a disadvantage from the standpoint of heat rejection equipment mass. According to the Stefan-Boltzmann Law for a fixed heat output rate, the radiator surface area decreases inversely with temperature raised to the fourth power. As a result, a higher radiating temperature meant a lower radiating area and hence a lower radiator mass.

Another disadvantage of the Brayton cycle was its back work ratio. The back work ratio is the ratio of the compressor or pump power input to the turbine power output. Since the Brayton cycle uses a gas as the working fluid, it must use a compressor to raise the pressure. The Rankine cycle has an advantage because raising the pressure is accomplished by pumping a liquid which requires much less work, and pumps can then be built with higher efficiencies.

Although the gaseous working fluid of the Brayton cycle was a disadvantage to the back work ratio, it was a better choice with respect to corrosion and erosion of system piping. Brayton cycles usually employ inert gases which do not react with the containment materials. Also, they do not erode the turbine blading since they never come close to their saturation domes. On the other hand, the liquid metals that are used in Rankine space power systems can have damaging effects on piping and turbine blading.

The above analysis dictated that the Rankine cycle should be used for Moonport primary power. The driving parameter behind this decision was that the Rankine cycle would have a smaller specific mass (ratio of system mass to net power output) since the heat rejection equipment could operate at a higher temperature.

6.1.4 RANKINE CYCLE COMPONENTS AND THEIR OPERATION

At this point, it is time to cover the Rankine cycle in detail. The components that makeup the cycle will be described in terms of their individual functions. In addition, the operation of the cycle will be discussed.

The process begins at the nuclear reactor, which is the energy source for this system. There are two basic types of nuclear reactors - thermal and fast reactors. A fast reactor is chosen for space applications utilizing the Rankine cycle since it does not require a moderator. A moderator is used to slow down fast neutrons, resulting in more thermal neutrons and hence a thermal reactor. Thermal reactors utilizing the Rankine cycle are not used for space applications, since the presence of a moderator loop complicates the cycle and increases the mass.

Heat from the reactor is removed by the primary loop. This loop consists of a working fluid to transport heat from the reactor to the boiler. If a liquid loop is used (as opposed to heat pipes) a pump must also be provided.

The next loop is the power conversion loop. This loop uses a two-phase working fluid which goes from liquid to vapor (possibly super-heated vapor) as it passes through the boiler. After exiting the boiler it enters the turbine, where the thermal energy of the working fluid is converted to mechanical energy and then to electrical energy in the alternator. As the working fluid passes through the turbine, it condenses. This condensation is removed from the vapor by moisture separators. Next, it passes through a heat exchanger where it undergoes a phase change back to the liquid state. Finally, the working fluid is pumped back into the boiler where the process is repeated.

Another major component is the main heat rejection loop. This loop condenses the working fluid in the power conversion loop by transferring heat from the power conversion loop to the heat rejection loop. The heat rejection loop dissipates the heat out to space using radiation. Standard fin radiators may be used; however, lower radiator masses can be achieved if liquid droplet radiators are employed.

Liquid metals are usually chosen as the working fluids in space nuclear reactors using the Rankine cycle because of the high operating temperatures. They have excellent thermophysical properties, such as high thermal conductivity and low vapor pressures. In addition, liquid metals with low atomic numbers have relatively high specific heats and volumetric heat capacities. It should be noted that they also have corrosive characteristics which requires the use of specialized containment materials.[5]

The last component, radiation shielding, applies to any nuclear power system and does not contribute to the power generation process. For manned missions, the shielding contributes the single largest mass fraction to the power system. The reactor and power system is located as far from the rest of the port as possible to minimize shield mass.

8.1.5 MOONPORT PRIMARY POWER PLANT DESIGN

The Moonport primary power plant design employs a uranium nitride fast reactor with an estimated lifetime of 50,000 hours. The key to this design is its modularity. When operating at full potential the reactor has a thermal power output of 45.8 MW which can generate 9.2 MW of electrical power. Three primary loops circulate through the reactor. Each loop is capable of producing 4.6 MW of electrical power resulting in two loops being on line at full power with the third on stand-by. It was decided to incorporate the stand-by system into the design to help assure mission success during the low thrust journey. The efficiency and unshielded specific mass of this design at 9.2 MW are 17.5% and 9.0 kg/kW (68 lbm/kW), respectively.

The primary power plant is composed of two working fluid loops and heat pipes for the heat rejection loop. The primary loop uses lithium because its high boiling temperature keeps it in the liquid phase throughout the process. The power conversion loop uses potassium because of its low melting point and high thermal conductivity. The main heat rejection loop uses potassium heat pipes.[6]

A mass breakdown can be found in Table 8.2. The values have been extrapolated from the SPR-6 power plant. It should be noted that this is for the complete three-module design. Figure 8.2[7] gives the operating conditions throughout the primary power plant.

It can be seen from the mass breakdown that the shield mass is larger than all other components combined. A 2π shield configuration was chosen for the nuclear reactor. This will protect a hemisphical area projecting outward from the reactor in the direction of Moonport. Although this configuration increases the shield mass, it is used because a shadow shield might not offer sufficient protection for proximity operations. The shielding is designed to reduce the radiation from the reactor to 10 REM/yr with neutron radiation comprising (but not compromising) 10% of this total and gamma rays the remaining 90%. The radiation levels have to be kept low because they will be superimposed on solar and galactic radiation. Lithium hydroxide is used for neutron attenuation because of its large hydrogen content and low density and tungsten is used to attenuate the gamma rays since it offers good protection and can withstand the high temperatures near the reactor. A heat shield is used as a thermal barrier between the reactor and the tungsten shield. The tungsten shield has a thickness of 0.26 m (0.85 ft) and a mass of 81 MT (179 kips). The lithium hydroxide shield surrounds the tungsten layer. It has a thickness of 1.41 m (4.63 ft) and a mass of 32 MT (71 kips).[8]

TABLE 8.2 PRIMARY POWER PLANT MASS BREAKDOWN

Component	MT	kilo-lbm
Reactor	9.4	20.7
Primary Loop	6.0	13.2
Power Conversion Loop	27.2	60.0
Main Heat Rejection Loop	29.8	65.7
Shielding	113.0	249.1
Structure and Misc.	10.0	22.0
Total	195.4	430.7

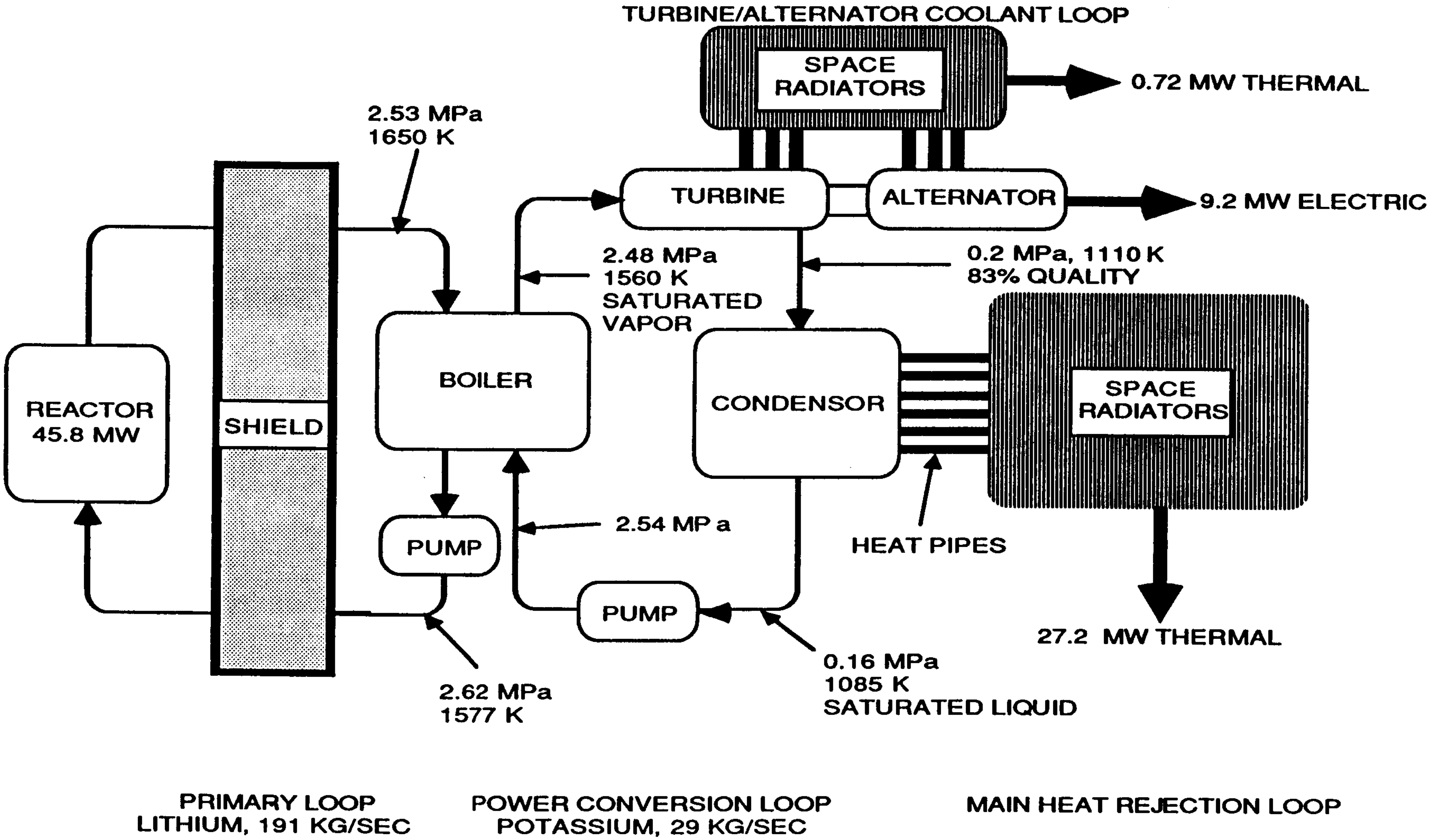

FIGURE 8.3 POWER PLANT OPERATING CONDITIONS[7]

8.2 EMERGENCY POWER SUPPLY

Moonport must have an emergency power supply in the event of primary power plant failure. Alternate nuclear or radioisotope sources are not feasible because they will have to have their own shielding since they are to be removed from the primary power source. This extra shielding increases the specific masses of these systems beyond those of chemical and solar sources.

8.2.1 EMERGENCY POWER CRITERIA

Three types of power sources were considered for the 75 kW emergency power requirement. Batteries and fuel cells derive their power from chemical energy while solar voltaic derives its power from the sun. Table 8.3 shows the decision matrix used to determine the best source.

TABLE 8.3 FUEL CELL DECISION CRITERIA

Criteria	Batteries	Fuel Cells	Solar
Reliability	1	1	1
Peak Power Output	3	1	2
Usable Lifetime	3	2	1
In-use Time Degradation	3	2	1
System Interfacing	1	2	3
Time Restrictions	1	2	3
Specific Mass	2	1	3
Total	14	11	14

Note: Low numbers are best

Reliability was a primary consideration, especially when emergency situations were being considered; therefore, only those which met this criteria were considered. Fuel cells showed excellent potential for peak power output since they could be networked. Solar voltaic had the best usable lifetime since it relied on direct energy conversion without any moving parts or fluids. Another advantage of solar voltaic was the in-use time degradation. Solar voltaic supplied relatively constant power during operation while chemical sources showed a decrease in power due to electrode corrosion. Batteries were the best choice for system interfacing because they were completely contained. Fuel cells required a reactant source and sub-systems for removal of the product(s) from the reaction. Solar cells had the worst system interface rating since solar arrays clutter the configuration and might inhibit vehicle traffic and cargo module storage. Time restrictions dictated when the source may be used. Batteries had a simple start-up procedure and could be put on-line almost instantly. Although solar voltaic had a fast start-up, it could not be used when Moonport was in a shadow. Considered independently, solar voltaic had the lowest specific mass; however, the need for sufficient batteries during the shadow phase of Moonport's orbit drove the overall specific mass beyond fuel cells.

As a result of the above analysis, it was decided to use fuel cells for Moonport emergency power. Their most desirable features were the peak power output and the specific mass. Solar voltaic was not used since shadowing required an additional emergency power source and the solar arrays would inhibit Moonport's operational capabilities. Batteries had many strong points but they did not deliver enough power at a reasonable specific mass.

8.2.2 FUEL CELLS

Two fuel cell reactions were considered for emergency power. One used gaseous hydrogen and oxygen for the reactants and produces either gaseous or liquid water as a product. The other used hydrazine and oxygen (this combination yields a specific impulse of 320 sec) as reactants and had gaseous nitrogen and water as products. Their reactions are represented by the following equations:

$$2H_2(\text{gas}) + O_2(\text{gas}) = 2H_2O(\text{gas or liquid})$$

$$N_2H_4(\text{gas}) + O_2(\text{gas}) = N_2(\text{gas}) + 2H_2O(\text{gas})$$

The optimum efficiency and maximum EMF are two important characteristics of fuel cell reactions. The H_2-O_2 fuel cell has optimum efficiencies of 94.5% and 83.0% for the gas and liquid water, respectively. Their maximum EMFs are 1.184 V and 1.229 V. The N_2H_4-O_2 fuel cell has an optimum efficiency of 99.4% and a maximum EMF of 1.559 V. The efficiencies for each reaction is slightly above average as are the EMFs for the H_2-O_2 reactions; however, the EMF for the hydrazine reaction is exceptionally high compared to other reactions.

These reactants were chosen because there would be plentiful supplies from the vehicle refueling system and the products could be used in the life-support systems. At this point, H_2-O_2 fuel cells are the best choice since there would be more H2-O2 stored on Moonport and they have a proven track record from shuttle data. As an example of what can be achieved from H_2-O_2 fuel cells, space shuttle data is listed in Table 8.4.[9] Extrapolating from this data for a 75 kW power output, 27 MT (60 kips) of H2-O2 propellant (enough for one fully fueled lunar vehicle) would yield power for 332 days.

TABLE 8.4 SPACE SHUTTLE FUEL CELL DATA[9]

Net Powerplant Output, Steady-State	
Min-Max, kW	2-12
Average, kW	7
Voltage, V	27.5-32.5
Restarts Allowed	
Without Maintenance	50
With Maintenance	125
Lifetime	
Without Maintenance, hours	2000
With Maintenance, hours	5000
Mass, kg (lbm)	91.6 (201.9)
Specific Mass, kg/kW (lbm/kW)	13.2 (29.1)
Flow Rate, Average Power	
H2, kg/hr (lbm/hr)	0.032 (0.071)
O2, kg/hr (lbm/hr)	0.284 (0.626)

8.3 PROPULSION

The primary constraint for the propulsion system was that it needed to be able to tow a very large mass from low-Earth orbit to lunar orbit. The system had to satisfy safety and economic considerations. The thruster mission was assumed to be unmanned and the duration would be 270 days.

8.3.1 TYPES OF PROPULSION

Three types of propulsion were investigated: electric, chemical, and nuclear. The following sections provide brief descriptions of the various types of propulsion.

8.3.1.1 ELECTRIC PROPULSION

Electric propulsion is characterized by low-thrust and high efficiency with an I_{sp} in the thousands of seconds. The primary reason electric propulsion was chosen is its high efficiency resulting in a higher payload ratio (ratio of final mass to initial mass, approximately 0.85).

8.3.1.2 CHEMICAL ROCKETS

Chemical rockets are characterized by high thrust and low efficiency with specific impulses in the hundreds of seconds. Chemical rockets were ruled out as a possible option because of their inefficiency. The expected payload ratio would not be much more than 0.05, and the mass of the required payload is too great for chemical rockets to be feasible.

8.3.1.3 NUCLEAR ROCKETS

Nuclear rockets are characterized by high thrust and medium efficiency. A nuclear rocket adds heat to the propellant through a reactor core as opposed to a chemical reaction, hence reducing the amount of propellant. Although nuclear rockets are much more efficient than chemical rockets, they still require that 80% of the initial mass of the spacecraft be taken up by the propulsion system. Since efficiency was the most important constraint in the propulsion selection, nuclear rockets were not chosen. As seen in Table 8.5, electric propulsion best fits the needs of the design. The table shows how each propulsion type ranked according to the given design criteria.

TABLE 8.5 PROPULSION DECISION CRITERIA

DESIGN CONSTRAINT	ELECTRIC	CHEMICAL	NUCLEAR
HIGH PAYLOAD RATIO	1	3	2
EFFICIENCY (ISP)	1	3	2
PROPELLANT STABILITY	1	3	2
COST OF SIZED SYSTEM	1	3	2
ONGOING RESEARCH AND DEVELOPMENT	2	1	3
TOTAL	6	13	11

8.3.2 THE ELECTRIC PROPULSION SYSTEM

Once electric propulsion was chosen, four different types were investigated. The four types were: electrostatic, arcjets, resistojets, and magnetoplasma dynamic devices.

Electrostatic propulsion was chosen because it yields a higher specific impulse and it has a longer proven lifetime than the other types of propulsion. Since the vehicle will be reusable, a long thruster lifetime was the driving design constraint. Material problems in the other three types of propulsion limit their lifetime to under 1000 hours which falls well below the 6500 hour time of flight for Moonport placement. Electrostatic thrusters also have the ability to optimize the I_{sp} with respect to the payload ratio by having the ability to vary the I_{sp} and the thrust. Basically, I_{sp} increases with increasing input voltage.

Electrostatic propulsion produces thrust by accelerating ions to extremely high velocities by way of a large voltage difference. The fuel is ionized by bombarding the elements with electrons, which leaves the fuel with a positive charge. The fuel is then vaporized and subjected to a voltage difference in the discharge chamber. The voltage difference creates an electric field which accelerates the positively charged fuel ions toward a negatively charged anode grid. A magnetic field is used to control the ion beam as the beam passes through the anode grid and is neutralized by an external electron beam.

Although the mass flow rates through the thruster are small, the velocities imparted to the mass are very large. The electrostatic thrusters yield a thrust as a result of the momentum change. By using less mass (fuel) and greater velocity, the payload ratio is increased. Greater payload can then be transported for a given amount of fuel, thereby reducing the cost of transportation.

In order to determine the thrust requirements for the LTV, SPS needed to find the acceleration required to make the orbit transfer. The relationship between the acceleration and the time of flight for an Earth to Moon spiral trajectory was provided by Shyam Bhaskaran of the University of Texas. In addition, the relationship between the Isp and the mass ratio was supplied for the Earth-Moon trajectory.

A TK! Solver model was used to determine mass estimates for the various components of Moonport. As seen in Figure 8.3, the mass of the LTV is sensitive to the time of flight for the thrust mission. This is caused by an increase in the acceleration as the time of flight decreases. An increased acceleration results in an increase in the power required by the LTV, and hence an increase in the power supply mass (also included in Figure 8.3). In addition, as acceleration increases the propellant mass flow rate increases resulting in a larger total propellant mass despite the decrease in burn time. The range for the time of flight was restricted from 90 to 360 days. At a 90 day time of flight, 47.6 MW of electrical power was needed to drive the thruster. This resulted in a total initial mass of 1321 MT (2912 kips) with 198 MT (437 kips) being allocated as propellant. At a time of flight of 360 days, the total initial mass was 765 MT (1687 kips) with 115 MT (254 kips) for propellant. A time of flight of 270 days was chosen because it resulted in relatively low initial and power supply masses while keeping the trip time at a reasonable level for an unmanned flight.

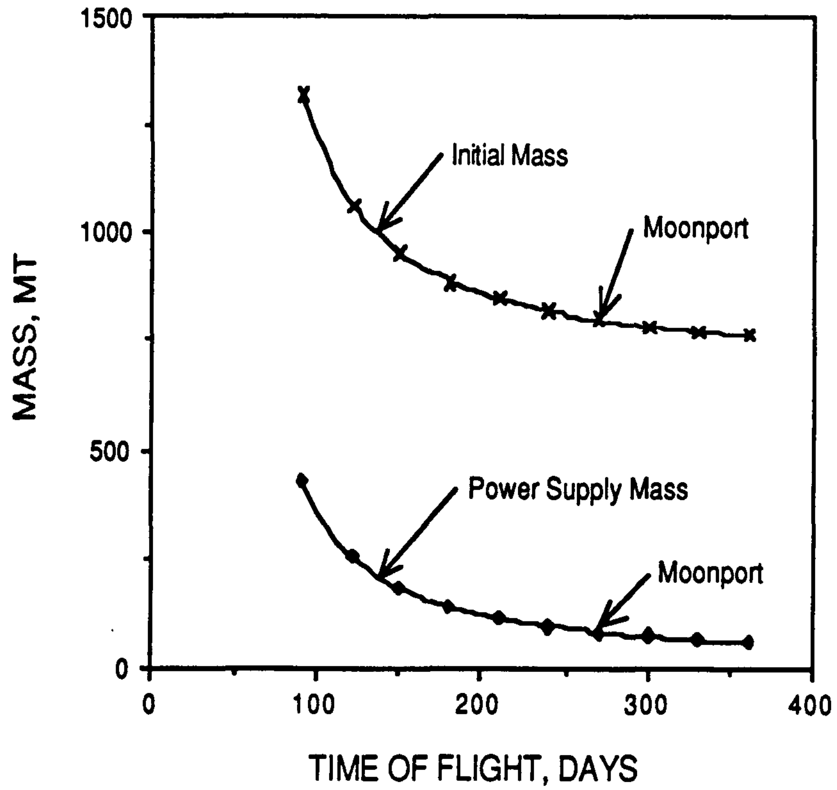

FIGURE 8.3 MOONPORT MASS VS. TIME OF FLIGHT

Moonport thruster specifications are located in Table 8.6.[10] The Isp for the LTV was limited to 6000 sec because the power requirement increases rapidly with increasing Isp. According to data received from an AIAA/JPL journal, thruster efficiencies can be expected to reach 0.85 by the year 2000. Each thruster has a thrust of 3 N (0.67 lbf); therefore, to attain the required acceleration level (3.178×10^{-5} g's), a minimum of 86 thrusters is needed. To assure success during the thrust mission a 50% redundency factor is built into the LTV model thereby increases the number of thrusters to 129. The thrusters are placed on three thrusting platforms with each platform supporting 43 thrusters. The platforms extend out from the support truss and are 120° apart from each other. The extended platforms allow a 20 m (65.6 ft) clearance between the thruster exhaust beam and the main delta truss.

TABLE 8.6 THRUSTER SPECIFICATIONS[10]

Specific Impulse	6000 sec
Thruster Efficiency	0.85
Thrust (each)	3 N (0.67 lbf)
Number Required (w/out redundency)	86
Maximum Acceleration	3.178×10^{-5} g
Propellant	Mercury

SECTION 8 REFERENCES: POWER AND PROPULSION

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SECTION 9

COMMUNICATIONS

9.1 EXTERNAL COMMUNICATIONS

9.2 INTERNAL COMMUNICATIONS

9.0 COMMUNICATIONS

The communication system will be required to provide communications between Moonport and all other elements of the transportation system. The communication system must be capable of transmitting, receiving, and processing a variety of signals including voice, telemetry, commands, wideband data, television, data text and graphics, and private communications. The design effort has focused on Moonport communication links to the Earth, Moonbase, transportation vehicles, EVA crew members, and Mobile Remote Manipulator Systems. Internal Moonport communications were also studied. The 1-2 second time delay for Earth-Moonport communications was not considered a problem in this study. Communication during LTV flights to the moon was not studied.

The use of optical systems for internal and external communications will save on both power and mass. The estimated maximum power needed at any time is 14 kW, and the estimated mass of all equipment is 1.1 MT (2500 lbm). The nuclear power system is expected to generate enough power for any communication system; therefore, mass will be the major factor in system and component selection. SPS has ranked various technology options in Table 9.1 (A through E) against system criteria, [1,2] where a low number is best. Based on the decision tables SPS recommends that further design should focus on:

- Frequency - Optical, Millimeter, Ku-band and S-band
- Antennas - Phased Array and Omni
- Local Comm - Optical
- Internal Comm - Fiber Optics
- Multiplexing - Wavelength Division

TABLE 9.1 TECHNOLOGY DECISION TABLE

A) COMM FREQUENCY

	S-BAND	X-BAND	K-BAND	KU-BAND	KA-BAND	MILLIMETER	OPTICAL
TDAS COMPATABLE	2	N/A	N/A	2	2	1	1
SPACE QUALIFIED	1	2	5	3	4	6	6
BANDWIDTH/ DATA RATE	5	5	4	4	3	2	1
LIFE CYCLE	1	1	4	2	4	3	3
TOTAL	9	8	13	11	13	12	11

B) COMM ANTENNAS

	OMNI	SECTOR	GIMBAL DISH	PHASED ARRAY
SPHERICAL COVERAGE	1	2	4	3
ANT. GAIN TO SATISFY DATA RATE REQUIREMENT	4	3	2	1
TRACKING REQUIREMENTS	1	4	2	3
RELIABILITY	2	3	4	1
LIFE CYCLE COST	2	3	2	1
TOTAL	10	15	14	9

TABLE 7.1 (CONTINUED) - TECHNOLOGY DECISION TABLE

	C) LOCAL COMM		D) INTERNAL COMM TECH	
	OPTICAL	RF	FIBER OPTICS	HARDWARE
SIZE, POWER,WT	N/A	N/A	1	2
COMPLEXITY	1	2	2	1
BANDWIDTH/DATA RATE	1	2	1	2
LIFE CYCLE COST	1	2	1	2
RELIABILITY	2	1	2	1
TOTAL	5	7	7	8

E) MULTIPLEXING

	FREQ DIVISION	TIME DIVISION	WAVELENGTH DIVISION
DATA RATE	2	1	2
COMPLEXITY	1	3	2
RELIABILITY	3	2	1
LIFE CYCLE COST	2	2	1
TOTAL	8	8	6

Moonport's function as a transportation node dictates that the majority of communications will be with vehicles as they approach or depart the Moonport operating area. The Moonport must be capable of tracking many vehicles at one time as well as payload modules and EVA crew members. The current communication scheme being planned for use in the space station relies on the Global Positioning System (GPS) for tracking during rendezvous, proximity operations, and EVA.[3] Since the Moonport scenario does not include a GPS equivalent, a major effort must be undertaken to ensure it can support these operations at a safe level.

9.1 EXTERNAL COMMUNICATIONS

Communication to the external elements will be a complicated task; the large amount of vehicle traffic and payload handling will require a sophisticated and managable system. Figure 9.1 gives an overall view of Moonport's relation to the external elements, and Table 9.2 lists the types of information passed in each external link. Rendezvous and proximity operations require communication with and tracking of multiple OTVs, OMVs and LVs, while EVA operations require simultaneous contact with as many as three crew members during an emergency. Payload handling, fueling and servicing requires control and feedback from the six MRMS units with a minimum of two operating simultaneously.

Communication with Earth can be achieved via the Deep Space Network (DSN). The Earth-based DSN can transmit and receive S-band frequency with 360 degree coverage for continuous Moon contact. The main link to Earth, using Millimeter and Laser communications,[4] will be through the Tracking and Data Acquisition System (TDAS) that will replace the Tracking and Data Relay Satellite System (TDRSS) in the 1990's. The Moonport will experience periodic blackouts of approximately 46.5 minutes every 118 minutes while in a baseline equatorial and circular 100 KM (54 nmi) LLO. Earth communication will be continuous if in an L_2 halo orbit due to the halo orbits large radius.

The link to Earthport will not be direct, but channeled through the Earth-Moonport link. If the lunar base is located on the Earth-facing side of the Moon, Moonbase communications will be completely obstructed by the lunar surface if the Moonport is in an L_2 orbit; therefore, this link will also be relayed through the Earth-ground link. A future objective may be the addition of a satellite relay system around the moon, or a fiber optic line to a ground station on the moon's darkside. Direct communication with Moonbase during LLO will be limited to approximatly 12.5 minutes every orbit as the Moonport passes overhead. For each LLO, an additional 34 minutes of communication to Moonbase will be possible through the Earth-ground link.

9.2 INTERNAL COMMUNICATIONS

The internal communications system must provide video, audio, command, telemetry, data text and graphics between modules, fuel facility and the service facility. Intercom and paging channels should be provided along with duplex voice channels to external elements. Speech recognition and synthesis will be included in the control stations. Distribution of caution and warning tones shall be provided in case of Moonport system failure or solar storm activity.

The crew will use both wireless communicators and wall-mounted speaker/microphone for voice/audio signals. Video play/record capability is provided for recreation/entertainment/leisure and private transmissions. Video cameras will be placed in each module, and external cameras will be used for tracking and assistance in vehicle retrival.

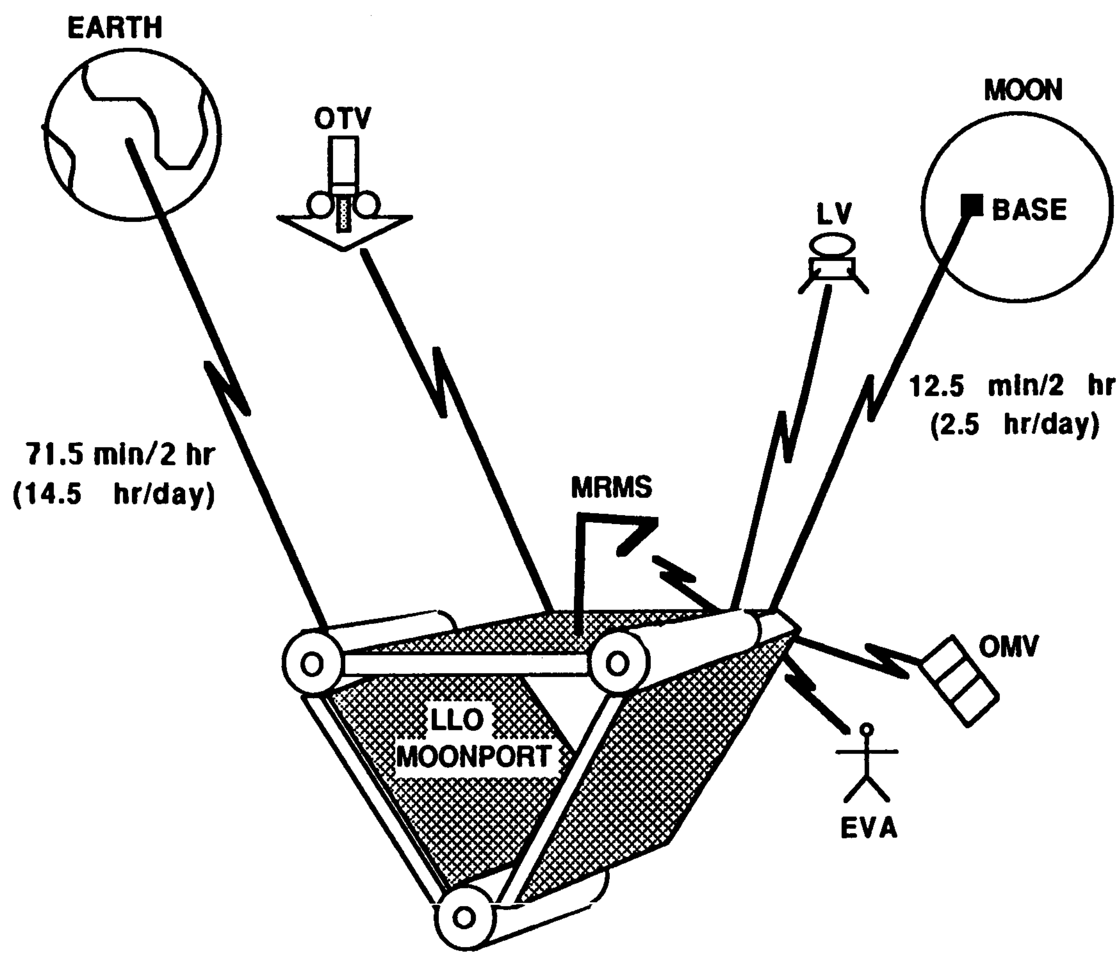

FIGURE 9.1 - EXTERNAL LINK SCHEMATIC

TABLE 9.2 - MOONPORT INFORMATION EXCHANGE

ELEMENT	TRANSMIT	RECEIVE
DEEP SPACE NETWORK	VOICE, TELEMETRY	VOICE, COMMANDS
TRACKING and DATA ACQUISITION SYSTEM	VOICE, TELEMETRY, TV, DATA	VOICE, COMMANDS, TV, DATA TEXT & GRAPHICS
MOONBASE	VOICE, COMMANDS, TV	VOICE, TV
OTV, OMV, LV, MRMS	COMMANDS	TELEMETRY, TV
MANNED OTV & LV	VOICE, COMMANDS	VOICE, TELEMETRY
EVA	VOICE, COMMANDS, FREEZE-FRAME TV	VOICE, TELEMETRY, TV

SECTION 9.0 REFERENCES: COMMUNICATIONS

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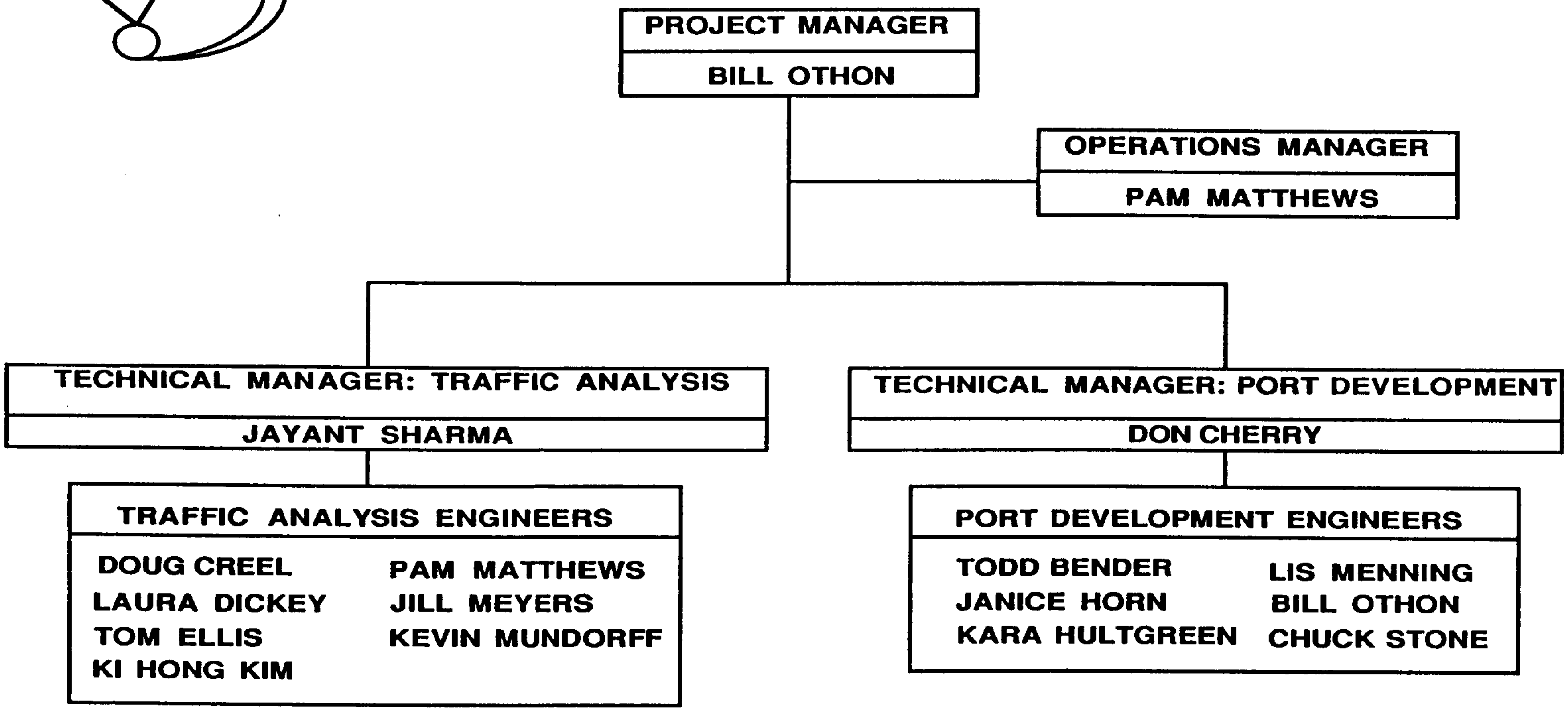
SECTION 10

MANAGEMENT STATUS

10.0 MANAGEMENT STATUS

During work for this preliminary design report, Space Port Systems was headed by an overall Project Manager and a management team of two Technical Managers and one Operations Manager. The Project Manager served as a contact point between the contractor and the contract moniter. Also, the Project Manager coordinated overall design activities between the two technical divisions. The Port Development Technical Manager oversaw Moonport design processes, including subsystem requirements, vehicle servicing, storage facilities, power, and habitation. The Traffic Model Analysis Technical Manager was in charge of developing a model of the transportation system to determine the most suitable location for the Moonport during steady state operations. Also, the traffic model was utilized to determine required Lunar Base LO2 production. The Operations Manager oversees all the administrative functions of the company, including bookkeeping, cost management, and personnel work schedules. The managers also performed engineering duties for the Company. A diagram of Company personnel, including management and design engineers, is presented in Figure 10.1.

The Company structure is designed to accommodate fast, responsive error detection and response. Early problem detection is referred to the particular technical manager. Available help is then diverted to problem solution. More critical design problems are referred to the Project Manager, where a problem management team can be assembled, or a design revision can be drafted. This structure is known as "the collapsing zone."



CONTRACT MONITOR: DR. WALLACE FOWLER
TECHNICAL ADVISOR: DARREL MONROE

FIGURE 10.1 SPACE PORT SYSTEMS PERSONNEL CHART

SECTION 11

COST SUMMARY

11.0 COST SUMMARY

Space Port Systems completed its contracted as stated in the RFP under budget. The total estimate for personnel and materials was $60,736.50. The actual project cost was $59,681.51. Table 11.1 presents a summary of expenditures. A breakdown of expenditures is shown in Table 11.2. The total work hours provided by SPS is shown in Table 11.3.

TABLE 11.1 COSTS SUMMARY

	Projected Cost	Actual Cost	Project Overrun/Underrun
Personnel	57,288.00	57,762.00	474.00
Materials	3,448.50	1,919.51	- 1,528.99
Total	60,736.50	59,681.51	- 1,054.99

TABLE 11.2 BREAKDOWN OF EXPENDITURES

	Projected Cost	Actual Cost	Project Overrun/Underrun
Personnel			
1 project manager	5,250.00	7,625.00	- 2,375.00
2 technical directors	9,240.00	11,935.00	- 2,695.00
1 managing director	4,620.00	4,290.00	330.00
11 engineers	30,537.00	27,720.00	2,817.00
Consulting	5,250.00	3,375.00	1,875.00
Materials			
Macintosh software and peripherals	2,300.00	1,477.00	823.00
IBM software and peripherals	500.00	0.00	500.00
CDC computer time	50.00	67.00	-17.00
Moonport model	200.00	75.51	124.49
Photocopies	35.00	115.00	-80.00
Transparencies	30.00	100.00	-70.00
Miscellaneous	20.00	135.00	-115.00
error estimate	2,204.50		
Total	60,736.50	59,290.99	1,054.99

TABLE 11.3 TOTAL COMPANY HOURS

Position	Total Hours
Project manager (1)	305.0
Technical Directors(2)	542.5
Managing Director(1)	194.0
Engineers(11)	2,035.8
Total	3,077.3

SECTION 12

PROJECT EVALUATION

12.0 PROJECT EVALUATION

The final section of this document examines the design effort conducted by SPS over the past contract period. An effort has been made to identify the strengths of the proposed design, and recommendations are offered for design areas which were not fully covered because of time or resources constraints. The project evaluation is presented in two parts. Section 12.1 examines the traffic model analysis conducted to describe Moonport activities and requirements. The subsystem requirements and overall configuration design of Moonport are critiqued in section 12.2.

12.1 TRAFFIC MODEL ANALYSIS

The traffic model developed for this design effort was undertaken to compare LLO and L2 Moonport location for steady state operations. A LLO orbit was foound to provide a larger payload delivery to LEO with a smaller Earth launched mass and a smaller lunar LO2 production facility. The traffic model in this study took both LH2 and LO2 consumption of all the vehicles into account. Empty vehicle trips were also included in the transportation system. The transportation system used was fully supported by lunar LO2 and the LSPI Lunar Base Model was used to provide an estimate for resupply mass requirements for the Lunar Base. With the given resupply mass estimates, the required lunar LO2 production, for a complete lunar LO2 supported transportation system, and the percentage of Lunar Base production delivered to LEO was determined. It was determined from this study that the net delivery to LEO is sensitive to payload capacity of the vehicles and therefore, to ΔV's and Isp.

Although the traffic model provides estimates of Lunar Base size and the amount of return to LEO, it lacks details, such as boiloff and transfer losses for LH2 and LO2. Crew transportation and Moonport resupply was also not considered in the traffic model. A model for the Moonport resupply needs to be developed to provide resupply data. It was determined that the increase in Lunar Base resupply mass corresponding to an increase in lunar LO2 production results in very large lunar LO2 production facilities (>10,000 MT/yr), if the percentage of lunar payload delivered to LEO is kept constant. This indicates the work is required on Lunar Base resuppy to see if the situation can be improved. Since the traffic model was based on fixed ΔV's, it would be useful to examine transportation between the Lunar Base adn LEO at best case and worst case situations to determine the impact on the traffic model. Further work needs to be conducted on accessibility to LLO and L_2 in an effort to reduce ΔV requirements.

12.2 MOONPORT CONFIGURATION DESIGN

The Moonport configuration design work conducted by SPS represents an initial requirements study. As a preliminary study, the subsystem were examined in general terms. The intent of the design work was to identify general requirements for several subsystems, and to provide a preliminary Moonport configuration based on those requirements. Each subsystem will be examined individually below.

12.2.1 MOONPORT CONFIGURATION

The preliminary Moonport configuration presented in this document is one possible integration of the various subsystems examined in the design study. For a transportation node, the delta truss was the best structural support system of the available truss configurations. The delta has high stiffness, strength, structural redundancy, and adequate surface area for cargo storage. However, the delta truss/tripod configuration requires further study in a variety of areas. An impact study of MRMS activity on Moonport operations must be conducted. A large number of MRMS units have been baselined, but a coordinated MRMS system was not developed. Because of the perturbations MRMS units cause, the number of MRMS units must be optimized.

Another area for future development involves the impacts of vehicle proximity operations on Moonport attitude. The attitude of Moonport must be designed to accommodate incoming vehicle traffic. For example, Moonport could be oriented along its radius vector, to accommodate approached from below. Alternatively, Moonport could be oriented with its front facing the direction of travel in Moonport orbit. In this orientation, incoming vehicles would approach along Moonport's velocity vector. An OTV/OMV stack could wait in a parking orbit higher than Moonport until the port approached (in its faster, lower orbit), then fire down into the appropriate approach orbit.

Because of time constraints and modeling deadlines, the location of the habitation modules was baselined on the apexes of the delta truss. However, there are a variety of alternate designs. The "racetrack" configuration, arranging the interconnected modules together in a square, reduces the amount of pressurized tunnels needed to connect the modules, but if one module has to be shut down, the exits of the adjacent modules are reduced from two to one. Other orientations of the habitation modules are possible, and should be examined in future study.

Overall Moonport attitude control during LTV flight and steady state operations is a major design problem which should be addressed. One possible solution to the control problem involves the use of fluidic momentum controllers, which pump water through tubes placed on the outer edges of the planar truss. The momentum of the pumped water is used to control Moonport attitude. The control problem should not be attempted until subsystem definition is further developed.

12.2.2 VEHICLE SERVICES

Many areas of vehicle servicing require significant research and design work to insure successful Moonport operations. Some of these areas include electrolysis, storage of LH2 and other difficult propellants, and power requirements for cryogenic storage, water storage, and electrolysis.

Technology should be advanced toward total automation of servicing and refueling operations, to avoid the need for EVA and other required manned operations. Since refueling techniques are primarily conceptual at this time, these techniques are not discussed to any great detail in this report. Also, specific servicing tools and equipment are not yet defined; that design awaits a more detailed definition of servicing needs and available equipment. Much of the research done for space station servicing and refueling is applicable to Moonport operations.

The interior design of the hanger facility was not considered, except for the existence of two MRMS's and as much berthing space as possible. The interior design should include instrumentation for vehicle servicing, and some mechanism for berthing the vehicles inside the hanger. More work should be done on the specific hanger design, including specifications such as shell thickness, material , shielding, etc.

12.2.3 HABITATION AND RADIATION

Habitation was designed from the 1985 Space Station Configuration book. This section needs to be brought up to date with more current

Space Station habitation module specifications. Further study is required to determine the amount of shielding that is required for the safe haven. In addition, better techniques need to be developed to determine the effectiveness of sandwich shielding.

12.2.4 POWER AND PROPULSION

The primary power plant for Moonport was determined after studying a wide range of power system. The approach used was to look at the characteristics of various systems in a broad context, then narrow the focus as systems were eliminated. Once a choice of power plants was made, a search for a model plant was conducted. This search resulted in the SPR-6 as a baseline for power plant component mass estimates. A disadvantage of this method was that the SPR-6 was a late 1960's design study and significant developments have probably been made and will continue to be made before Moonport would be established. One of the best outcomes from the power plant design was the use of modularity in the power conversion loops. This was used to provide redundancy in the system. Recommendations for improving the design fall into two main categories. First, the radiation level around the reactor needs to be more accurately determined since a linear relationship between power output and radiation was assumed for the design. Second, liquid droplet radiators need to be studied to determine if the heat rejection mass can be reduced.

Emergency power systems should be further developed. Due to time constraints, this area was not fully developed. Fuel cell characteristics were determined by extrapolation from shuttle technology. There are two problems with this method. One is that the specific mass will usually drop as the power output increases; therefore, the mass estimate is a little high. Also, electrode corrosion could become a significant problem in such a large system.

The sizing of the low thrust vehicle (LTV) was based on estimates of Moonport component masses. As a result, mass errors were carried through when the size of the LTV was calculated. In addition, the data for the time of flight and acceleration was determined by fitting a logrithmic curve to time of flight and acceleration data points. The values used for Moonport had to be extrapolated since they were outside the range of the points used in the curve fitting.

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