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A LUNAR TRANSPORTATION SYSTEM

A design project completed by the senior students in the Department of Aerospace Engineering at Auburn University, Auburn, Alabama, under the sponsorship of USRA Advanced Missions Space Design Program.

Auburn University

Auburn, Alabama

June, 1986

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ABSTRACT

Due to large amounts of oxygen required for space travel, a method of mining, transporting, and storing this oxygen in space would facilitate further space exploration. The following project deals specifically with the methods for transporting liquid oxygen from the lunar surface to the Lunar Orbit (LO) space station, and then to Lower Earth Orbit (LEO) space station.

Two vehicles have been designed for operation between LEO space station and LO space station. The first of these vehicles is an aerobraked design vehicle. The aerobraked Orbital Transfer Vehicle (OTV) is capable of transporting 5000 lbm of payload to LO while returning to LEO with 60,000 lbm of liquid oxygen, and thus meeting mission requirements. The second vehicle can deliver 18,000 lbm of

payload to LO and is capable of bringing 60,000 lbm of liquid oxygen back to LEO.

A lunar landing vehicle has also been designed for operation between LO and the established moon base. This vehicle is capable of delivering some 20,000 lbm of payload to LO space station. This payload can be composed of all liquid oxygen or it may be a combination of liquid oxygen and other materials and equipment.

The use of an electromagnetic railgun as a method for launching the lunar lander has also been investigated. The feasibility of the railgun is doubtful at this time; however, future developments may make it a viable choice.

A system of spheres has also been designed for proper storing and transporting of the liquid oxygen. The system deals with spheres to be used primarily in returning the oxygen from the lunar surface to the LO space station, and then to LEO space station. The system assumes a safe means for transferring the liquid oxygen from tank to tank is operational.

A sophisticated life support system has also been developed for both the OTV and the lunar lander. This system focuses on such factors as the vehicle environment, waste management, water requirements, food requirements, and oxygen requirements.

FOREWORD

The Lunar Surface Return Project, which is currently being investigated by a number of people, involves establishing a permanent manned lunar base for producing oxygen from oxygen rich lunar rocks and conducting scientific experiments. Using oxygen from the moon will save millions of dollars when launching deeper space missions from Earth orbit. This report from Auburn University examines a part of that project—the design of an orbital transfer vehicle (OTV) to operate between low Earth and lunar orbits. The primary function of the OTV will be to return a payload of 60,000 lbm of lunar oxygen to low Earth orbit (LEO). A lunar lander (LL) is proposed to bring the oxygen from the surface to lunar orbit (LO). The OTV will also be capable of carrying supplies to lunar orbit as well as transporting personnel for crew rotations.

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INTRODUCTION

This report is the result of a grant awarded to the Aerospace Engineering Department of Auburn University by the USRA (Universities Space Research Association). The project, carried out by the senior design class consisting of approximately forty students, was directed by Dr. James O. Nichols who is an Associate Professor in the Aerospace Engineering Department at Auburn. Dr. Nichols has taught the design class, which is a three quarter series, for several years.

During the first quarter of the design series, the students organized themselves into specialized groups to investigate the various aspects of the space project. The groups then began an extensive literature search in order to gain a basic understanding of their chosen topic and to gather current, technical information. With this data, basic design ideas were developed and conceptual configurations were formed.

The second quarter was the primary design stage in which students began the actual design and redesign process. Size, performance, requirements, etc. were determined by each group and then coordinated between the groups toward a final configuration. During this stage technical support was provided by NASA Marshall Space Flight Center, Huntsville, Alabama through meetings with NASA personnel.

During the final stages, or third quarter, the "finishing touches" were put on the design as each group organized their material into a report. These reports were in turn combined into this final project report.

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ASSUMPTIONS

Before beginning the design, several assumptions were made concerning support facilities. The operational date for the OTV will be in the mid 1990's or after deployment of the permanently manned low Earth orbit space station which will provide docking, repair, and maintenance facilities for the OTV. A space shuttle or similar vehicle will supply the space station and OTV with any needed materials as well as return personnel to Earth.

The lunar base is also assumed to be fully established and operational. The existence of a lunar space station or platform will aid in transferring oxygen and supplies between the LL and the OTV.

Two configurations for an OTV are discussed in this report—a retro-braked vehicle and an aeroassist braked vehicle. Each design exhibited merits and enough advantages to prove itself feasible. It is difficult to pick a superior design because many of the advantages/ disadvantages are technological, and with technology progressing as rapidly as it is now, the best design today may not be the most optimal design by the operational date. Therefore, both designs are presented along with their advantages and disadvantages.

RETRO-BRAKED CONFIGURATION

OVERVIEW

Perhaps the strongest argument for the retro-braked OTV is its use of current technology, most of which has already been proven in space flight. By using current, proven technology, the time and cost associated with researching and developing new technology can be avoided; thus solutions to design problems can be completed with more certainty. New technology is also possibly the strongest argument against an aero-assist braked OTV. The theory of aeroassist braking is relatively new and uncertain, and irregularities in the atmosphere make predicting results difficult.

Because of these facts, the retro-braked OTV will be safer for transporting personnel. The OTV never enters the atmosphere and therefore is never in danger from the high heat loads caused by friction. The engines chosen in this design are very reliable, and are capable of executing the required velocity changes, even if one engine should fail.

Due to the choice of engines, accelerations/decelerations never exceed one g in the retro-braked OTV. The deceleration during aero-assist braking would be three to four times this value. These higher g-loads would increase structural stresses and passenger discomfort. The major disadvantage of the retro-braked OTV, however, occurs during return to Earth orbit when the aeroassist design would provide the most benefit. It is estimated that almost 180,000 lbm of propellant used in slowing the retro-braked OTV could be saved with the aeroassist design. Due to the other advantages, however, this design deserves consideration.

CONFIGURATION

The retro-braked design consists of four primary parts: the propulsion unit, the command module, the payload unit, and the frontal maneuvering unit (Figure 1.1). The payload unit depicted in the figure is the oxygen sphere/basket unit. As will be seen, each of the major components is small enough to fit into the payload bay of the present day space shuttle. This fact will allow easy transportation to space where the OTV can be assembled.

Command Module

Since the operational date is after the development of the space station, some cost can be saved by using a modified space station module for the command module. The proposed modules for the space station are basically the same, differing mainly in the type of components with which they are equipped. For example, one of the modules for the space station is to be used for sleeping quarters for a crew of eight to twelve. However, the shell (outer-most structure) of all the modules will remain somewhat standardized.

The intended crew of the OTV will have no more than three or four members. This number allows enough crew to operate the OTV and also allow crew rotations from the moon. By shortening the length of one space station module to 23 ft, it can be adapted to function as a command and living module. This module will contain all necessary navigation, communications, and piloting equipment for the proper operation of the transfer vehicle as well as provide ample space for sleeping quarters and other necessary life support systems. This

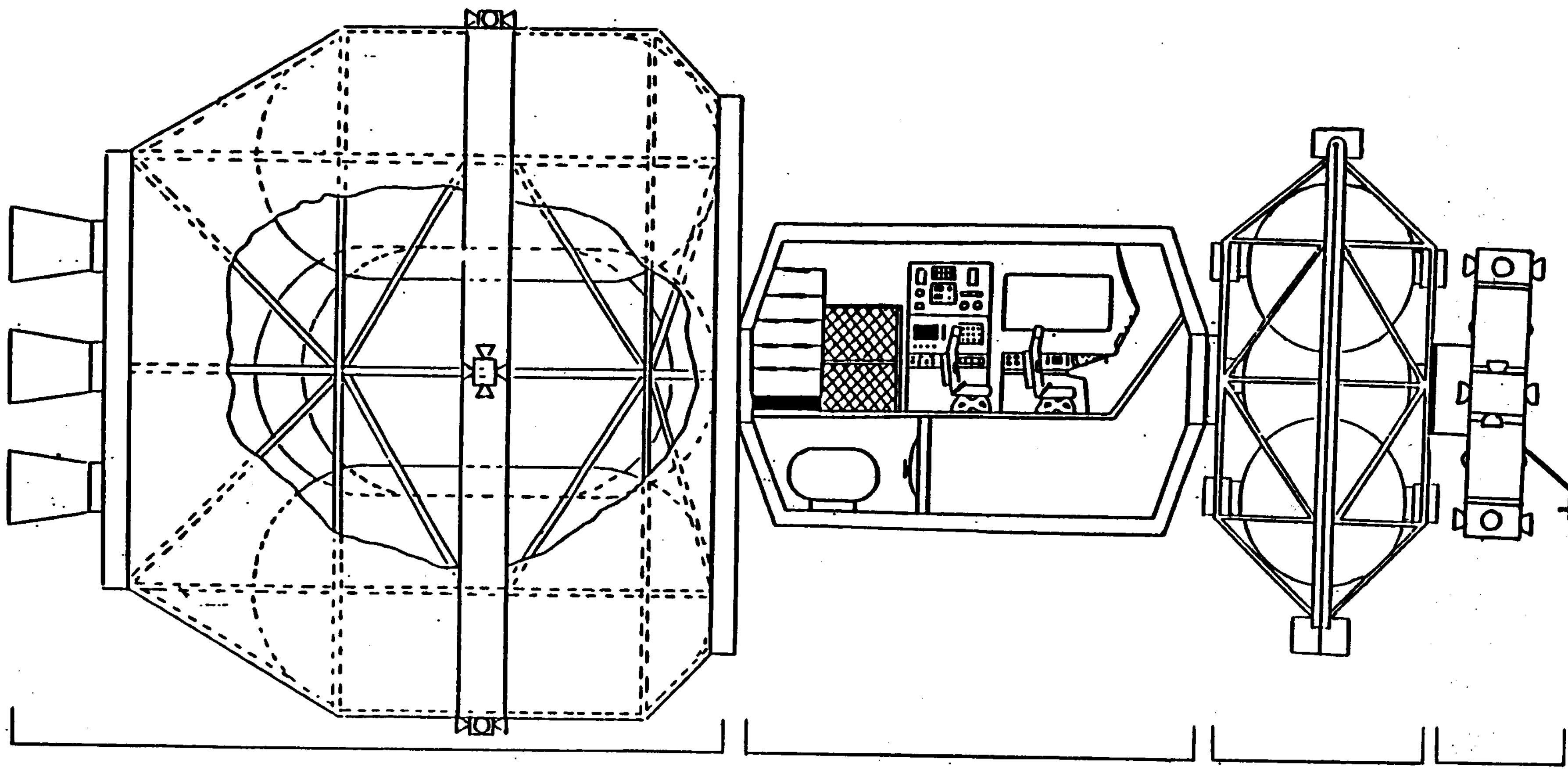


Figure 1.1. Retro-braked OTV Configuration

command module, fitted with the previously mentioned equipment, will have a total mass of approximately 8000 lbm and is shown in Figure 1.2.

The command module will be attached immediately in front of the propulsion unit via a standardized docking collar. The propulsion unit will be quipped with a reinforced docking mechanism capable of withstanding the forces generated by the firing of the main engines. Assuming more than one vehicle is built, the docking mechanism on the propulsion unit will allow different combinations of propulsion units and command modules to be joined. Thus, if one unit is down due to repair or maintenance, the rest of the vehicle will remain operational.

<u>Frontal Maneuvering Unit</u>

The most aft portion of the transfer vehicle is the propulsion unit. Directly ahead of this is the command module which is followed by the payload units. Since there is only a simple skeletal structure used to encase the fuel spheres, there is no place ahead of the command module suitable for the positioning of the forward maneuvering thrusters. Therefore, a modified space station orbital maneuvering vehicle is used. The slightly smaller version will be attached directly to the cargo unit, allowing for directional control of the vehicle. This unit is shown in Figure 1.3. The lack of external bracing will require that the FMU be remotely controlled from within the command module. Furthermore, video cameras will be mounted on the FMU to provide the necessary viewing otherwise inhibited by the payload.

The frontal maneuvering unit will not only be used to maintain directional control during orbital transfer, but also to position the transfer vehicle so that the propulsion unit may be used for retro-braking during orbital insertion. This unit, consisting of avionics,

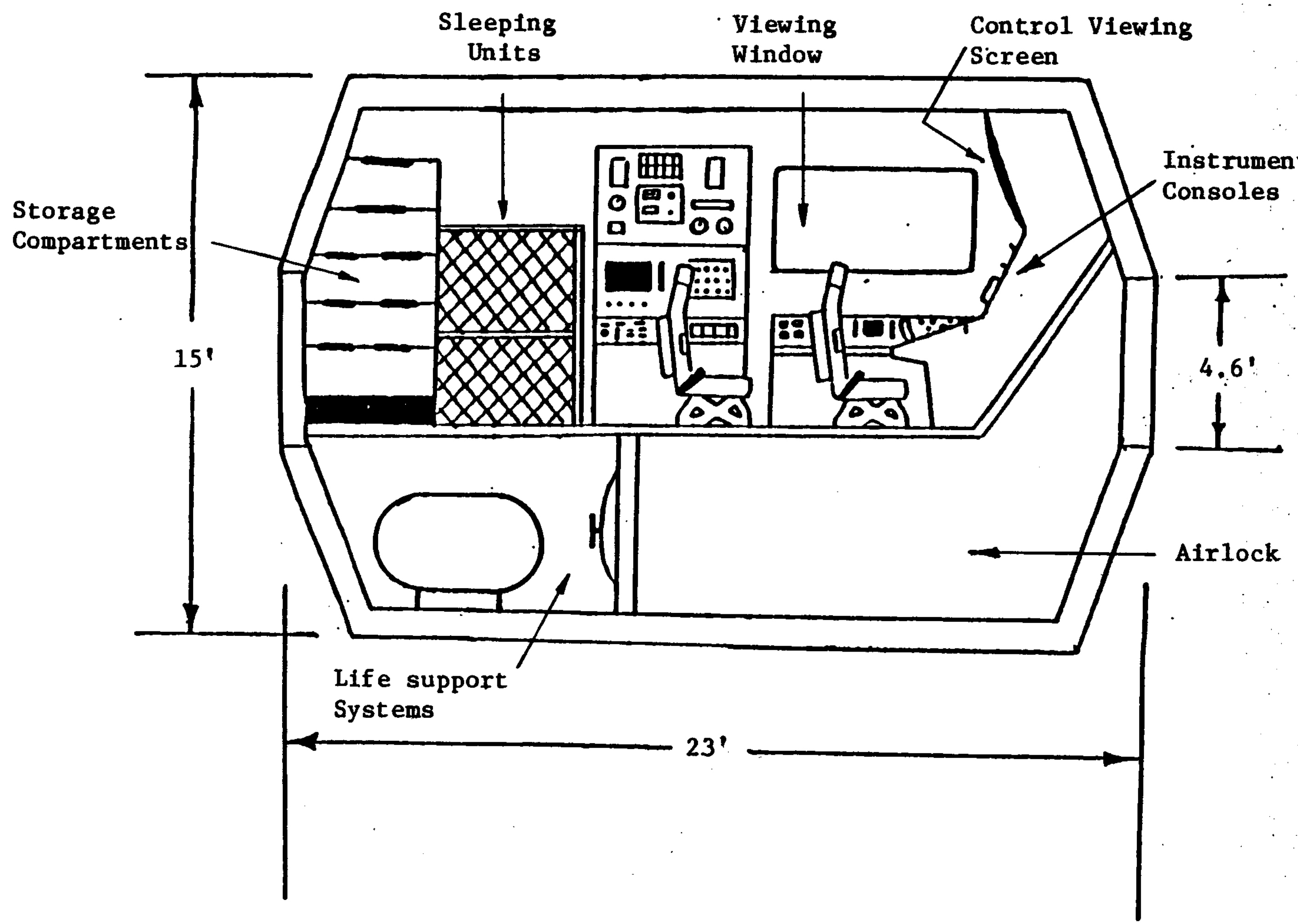


Figure 1.2. Command Module Cut-away

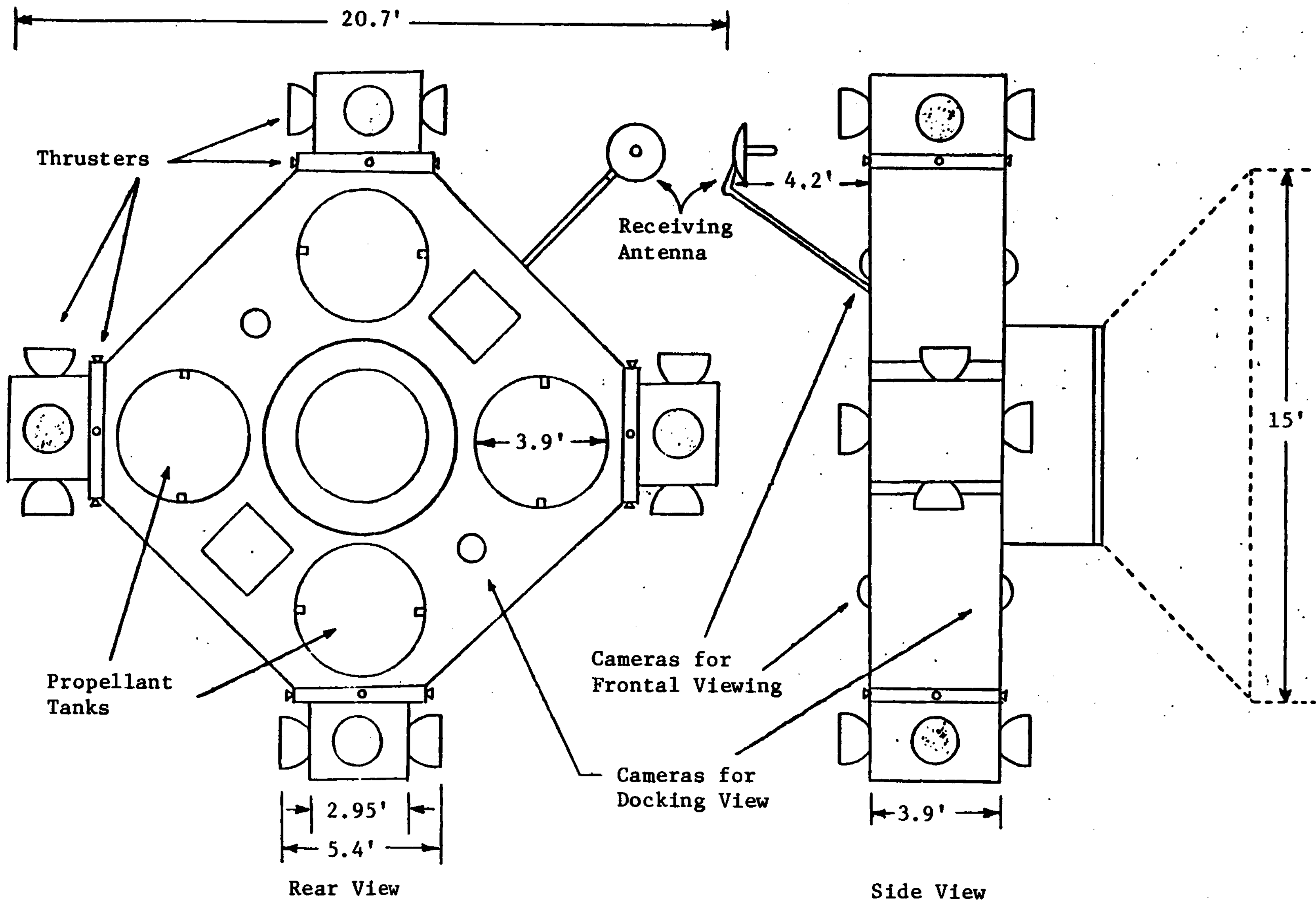


Figure 1.3. Frontal Maneuvering Unit

RCS, and casing, will have a total mass of approximately 1800 lbm. As can be seen from the figure, the unit contains the propellant tanks, pressure system, and thrusters necessary for a self-contained unit.

Propulsion Unit

A detachable propulsion unit has been designed to allow for ease of replacement and use on different command modules. The production of liquid oxygen (LOX) on the lunar surface facilitated the choice of hydrogen/oxygen engines. The decision was made to use five Pratt and Whitney RL-100 engines. The engine choice and mission requirements are discussed in the Propulsion and Mission Requirements section.

It is obvious that a large amount of propellant will be needed to transport payload between the space stations. The tankage for the propellant presents a design problem of its own in that sufficient volume is needed to carry a round trip supply of hydrogen. The LOX is assumed to be resupplied at the lunar orbit. In order to avoid mass distribution problems it was necessary to place the LOX tank along the OTV's centerline, with three hydrogen tanks placed around the LOX tank. The mission requirement calculations revealed that almost 56,000 lbm of hydrogen are needed for the round trip. Three cylindrical tanks 29.07 ft in length with spherical ends 7.5 ft in radius are required to carry the needed hydrogen. The size of the oxygen tank was governed by the amount of oxygen needed for the fully loaded return trip. A single cylindrical tank 23.2 ft in length with spherical ends 7.5 ft in radius is sufficient to carry the maximum required oxygen.

An additional function of the propulsion unit is as a location for the aft reaction control system (RCS). This RCS will function independently of the forward RCS in that the propellant storage and feed systems are separate. The location of the RCS thrusters as well as the entire propulsion unit are shown in Figure 1.4. The propulsion unit will be covered with a layer of insulation of low absorptance/emittance of about 0.2 and the oxygen tank will be separately wrapped to help keep the propellants in liquid form. A total dry mass of the propulsion unit is approximately 10,200 lbm.

Payload Unit

In order to meet mission requirements a suitable method of returning large amounts of LOX to the Earth was needed. To ease the handling and transport of the LOX a "basket system" has been developed. This system works on the same principal as the payload system for the aeroassist braked design, and that section should be seen for detailed operation description. The retro-braked OTV system contains five spherical tanks 9.5 ft in diameter locked into a cage-like frame (Figure 1.5). This frame can in turn be docked in line to the OTV aft of the FMU. A crane mechanism will be required at both space stations to assist in attaching or removing the basket from the OTV. The cage system fully loaded will allow enough LOX to be brought back to supply the next mission plus an additional 60,000 lbm to be added to a stockpile.

Docking/Collar Mechanisms

At this point nothing has been said to indicate how the combination of the primary units are to be docked together to provide structural stability and integrity. It is proposed to link all major units axially

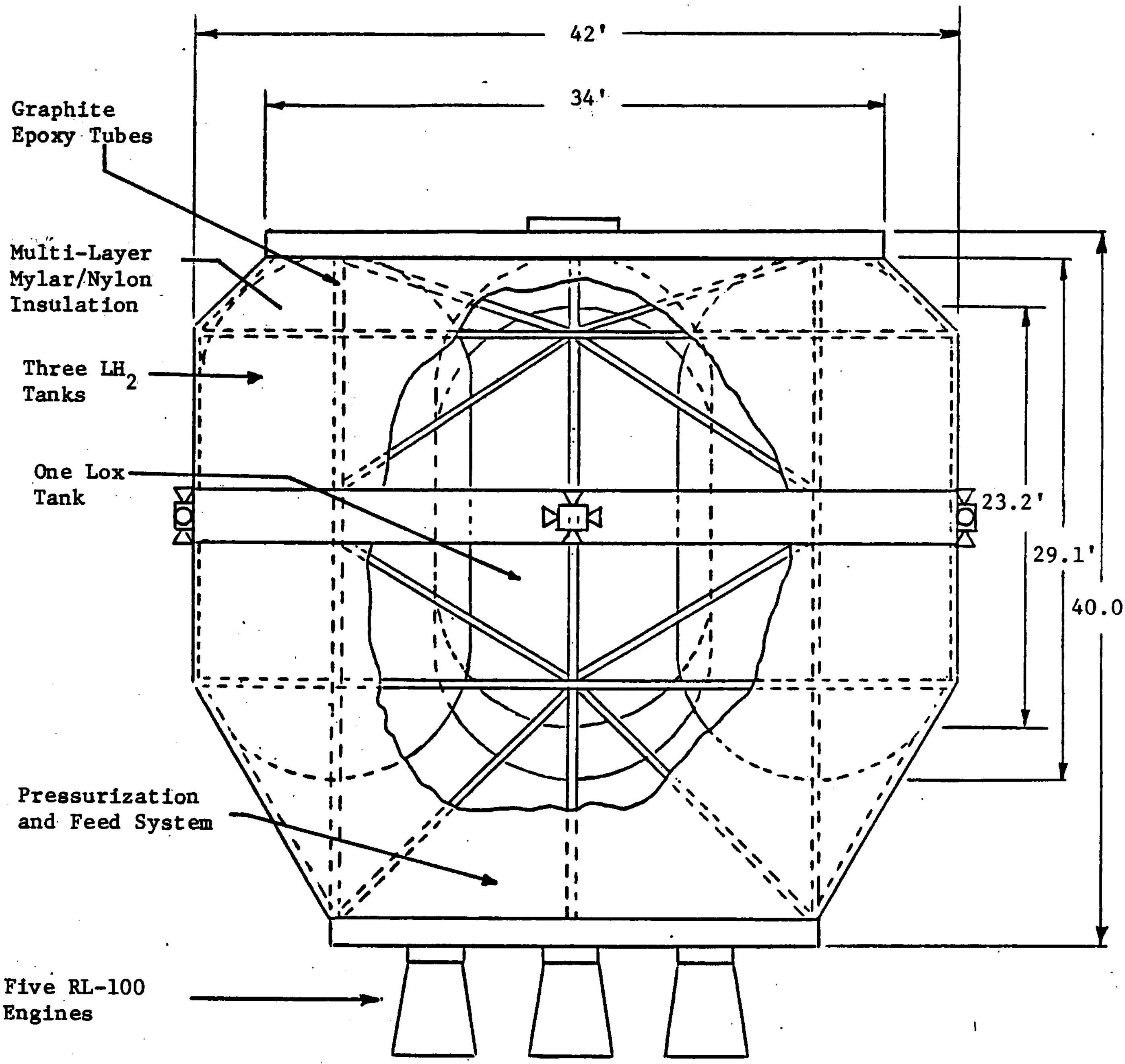


Figure 1.4. Propulsion Unit

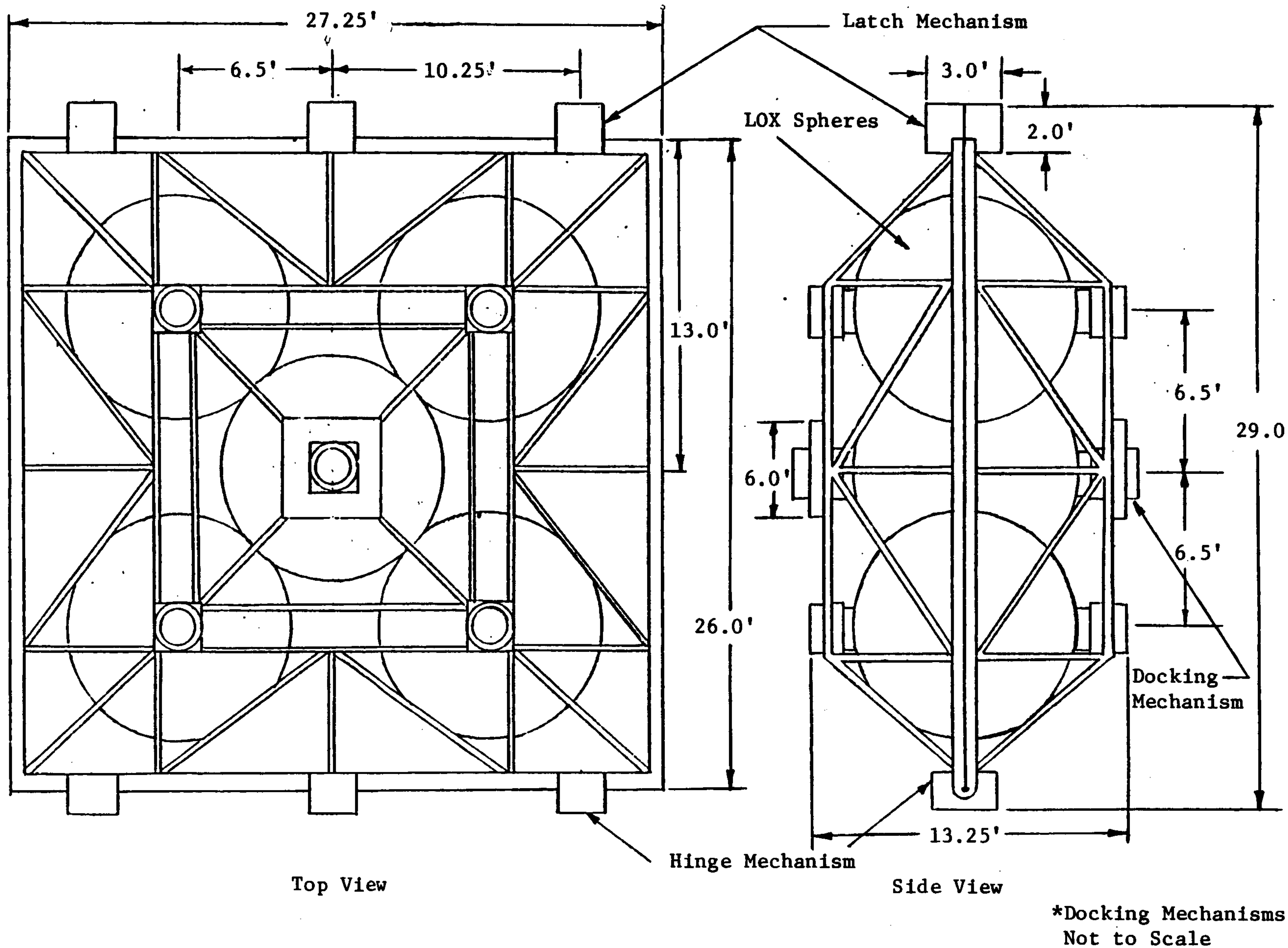


Figure 1.5. LOX Transport Basket System

to ensure that the propulsive thrust passes through the center of mass of the entire system, preventing any undesirable bending moments.

It is also suggested that if the docking mechanisms are not strong enough to provide adequate rigidity, then a collar mechanism is to be placed around the entire joint, resulting in an increase in rigidity. This collar will not have to be attached to the elements themselves, but merely fit snugly between them to prevent excessive deflections.

PROPULSION & MISSION REQUIREMENTS

Propulsion

The main propulsion system will be called upon to make three primary velocity changes (four if an orbit inclination change is required) each half of the trip. The largest change will be to leave or reenter low Earth orbit. An estimated change of velocity of 10,350 ft/s is required to accomplish this task. The next largest velocity change, approximately 2690 ft/s, will be to establish or leave lunar orbit. The third velocity change is about 180 ft/s allotted for a mid-course correction, should it be necessary.

Various propulsion systems were studied to find the most efficient one available that would meet the mission requirements of the OTV. One of the most attractive propulsion systems available is ion propulsion. Ion propulsion systems use a magnetic field to accelerate charged gas particles. The acceleration of these particles causes a thrust to be exerted on the craft. The advantage of ion propulsion systems is that they have very large values of specific impulse. The major drawback of the ion system is low thrust production. The low thrust is mainly due to the low mass flow rates of the ion engines. The specific impulse of some engines reach as high as 6000 sec., yet the strongest thrust is found to be 0.5 newtons (Ref. 7).

There are many methods for improving the performance of ion propulsion engines. However, none of these methods yield an engine anywhere close to the minimum design requirements for the OTV. Research in the field of ion propulsion is continuing and it may someday be a *feasible power source for an OTV, but that time appears too far away to be realistically considered for this design.*

Having eliminated ion propulsion, chemical systems seemed to be the best choice. Due to its abundance on the moon and many desirable qualities, oxygen was chosen as the oxidizer. Hydrogen was chosen as the fuel since the combination yields a high specific impulse and low combustion temperature. Although hydrogen and oxygen have a high optimum weight ratio, they have a low molecular weight.

A LOX-hydrogen system would combine to give a specific impulse possibly as high as 473 seconds. "The 444 seconds of specific impulse presently available with today's RL-10a3 engine can be improved by 25 to 30 seconds with application of advanced technology. This will be an important consideration in the definition of an OTV configuration and the related cost-performance trade-offs (Ref. 3)."

The hydrogen and oxygen combination is a cryogenic bipropellant which must be stored at extremely low temperatures (Ref. 2); keeping the propellants in liquid form will be possible through the use of pressure tanks, insulation, and the cold of space. A boiloff of about one per cent per day is expected.

Problems introduced with the use of pressurized tanks are sloshing, keeping a constant pressure, and maintaining a constant fuel flow. Three ways to deal with these problems are: 1) a piston or diaphragm which can be moved through use of an inert gas or hydraulics, 2) a capillary barrier which will use the fluid's own surface tension to keep gas bubbles from getting into the feed lines, or 3) the use of small jets to give the rocket a forward momentum which in turn will force the fuel toward the pump. These methods are designed to keep down sloshing of fuel and also insure a proper flow to the pumps (Ref. 2).

Should cooling be needed for the rocket nozzle there are several effective methods for achieving this: the methods include regenerative cooling, film cooling, insulation cooling, and oblative cooling. Of the methods listed above, regenerative cooling seems to be the most effective. Regenerative cooling works by taking fuel from the storage tank and passing it through tubes positioned length-wise of the nozzle (Ref. 3). This process serves two purposes: 1) cooling the inner walls of the nozzle and 2) preheating the fuel before entering the combustion chamber.

Even though LOX-hydrogen has a relatively low combustion temperature, it is still high enough to cause dissociation in the products. This dissociation takes useful energy out of the flow, even if recombination occurs in the nozzle. However, LOX-hydrogen has a relatively small amount of dissociation compared to other fuel and oxidizer mixtures.

The propulsion system consists of five RL-100 engines, each of which produces 15,000 lbs of thrust and has a mass flow rate of 31.1203 lbm/s. This system will produce a total thrust of 75,000 lbs. and have a total mass flow rate of 155.6 lbm/s (Ref. 14). The RL-100, which should be operational by the time OTV production begins, is regeneratively cooled and will have a life expectancy of 10 hours or 300 firings. The selection was based on choosing the minimum number of engines to keep burn times below engine specifications and still provide enough thrust to perform the mission. The minimum number of engines was desired to keep the dry weight low.

Mission Requirements

Table 1.1 shows a mass breakdown of the OTV. The mass of the OTV (dry mass) was determined to be about 25,650 lbm. With the chosen propulsion system, 19,500 lbm of payload can be delivered to the moon with the OTV still able to return a surplus of 60,000 lbm of oxygen to low Earth orbit. (Note: Of this 19,500 lbm, approximately 1,800 lbm is the payload unit itself.)

The following represents a typical mission for the OTV. The total mass of the OTV leaving LEO for the trip to lunar orbit is 210,000 lbm. Of this mass 25,650 lbm is the dry mass of the craft itself and 60,000 lbm is the cargo carried to lunar orbit. Note that of this 60,000 lbm, cargo, 40,500 lbm is hydrogen needed for the return trip to LEO. It is assumed that all of the oxygen necessary for the return trip will be supplied by the lunar system. Of the total OTV mass leaving Earth orbit, 124,350 lbm is propellant used for propulsion to lunar orbit— 17,200 lbm hydrogen, 103,202 lbm oxygen, and 3948 lbm boiloff. The mass change and time required for each firing is shown in Table 1.2.

Upon reaching lunar orbit the main oxygen tanks will be filled with 227,000 lbm of oxygen that will be burned in the flight from lunar orbit to low Earth orbit. The payload spheres will be filled with 163,202 lbm. of oxygen to be transferred back to LEO. Of this 163,202 lbm, 60,000 is the required cargo and 103,200 lbm is the amount necessary for the next trip to lunar orbit. The total mass leaving lunar orbit is 460,000 lbm. Table 1.3 shows a mass breakdown of the lunar to Earth transfer.

The total burn time for the entire mission is approximately 41 min. This value gives the propulsion unit a life expectancy of 14 missions

PROPULSION UNIT	MASS (lbm)
Oxygen Tank (1)	1,200.
Hydrogen Tank (3)	4,800.
RL-100 Engines (5)	2,175.
Pressurization and Feed	770.
External Insulation	300.
Reaction Control Systems	1,295.
Engine Truss	110.
Struts	500.
Subtotal	11,150.
COMMAND MODULE	MASS (lbm)
Case	8,000.
External Insulation	603.
Electrical Power (Fuel Cells)	662.
Avionics	2,000.
Life Support	1,000.
Subtotal	12,265.
FRONTAL MANEUVERING UNIT	MASS (lbm)
Reaction Control Systems	1,295.
Avionics	200.
Casing	300.
Subtotal	1,795.
MISC.	MASS (lbm)
Docking Provisions	440.
TOTAL DRY WEIGHT	25,650 lbm

Table 1.1. Estimated Mass Breakdown

Maneuver Performed	Velocity Change (ft/s)	Propellant Required (lbm)	Time Required (min)
Exit LEO	10,350	102,203	10.94
Mid-Course Correction	180	1,243	.133
Establish lunar orbit	2,690	16,956	1.82

Total O_2 consumption	103,202 lbm
Total H_2 consumption	17,200 lbm
Total burn time	12.89 min

Table 1.2. Mission Requirements for Lunar Transfer

Maneuver Performed	Velocity Change (ft/s)	Propellant Required (lbm)	Time Required (min)
Exit Lunar Orbit	2,690	73,200	7.84
Mid-Course Correction	180	4,460	.477
Establish LEO	10,350	186,079	19.93

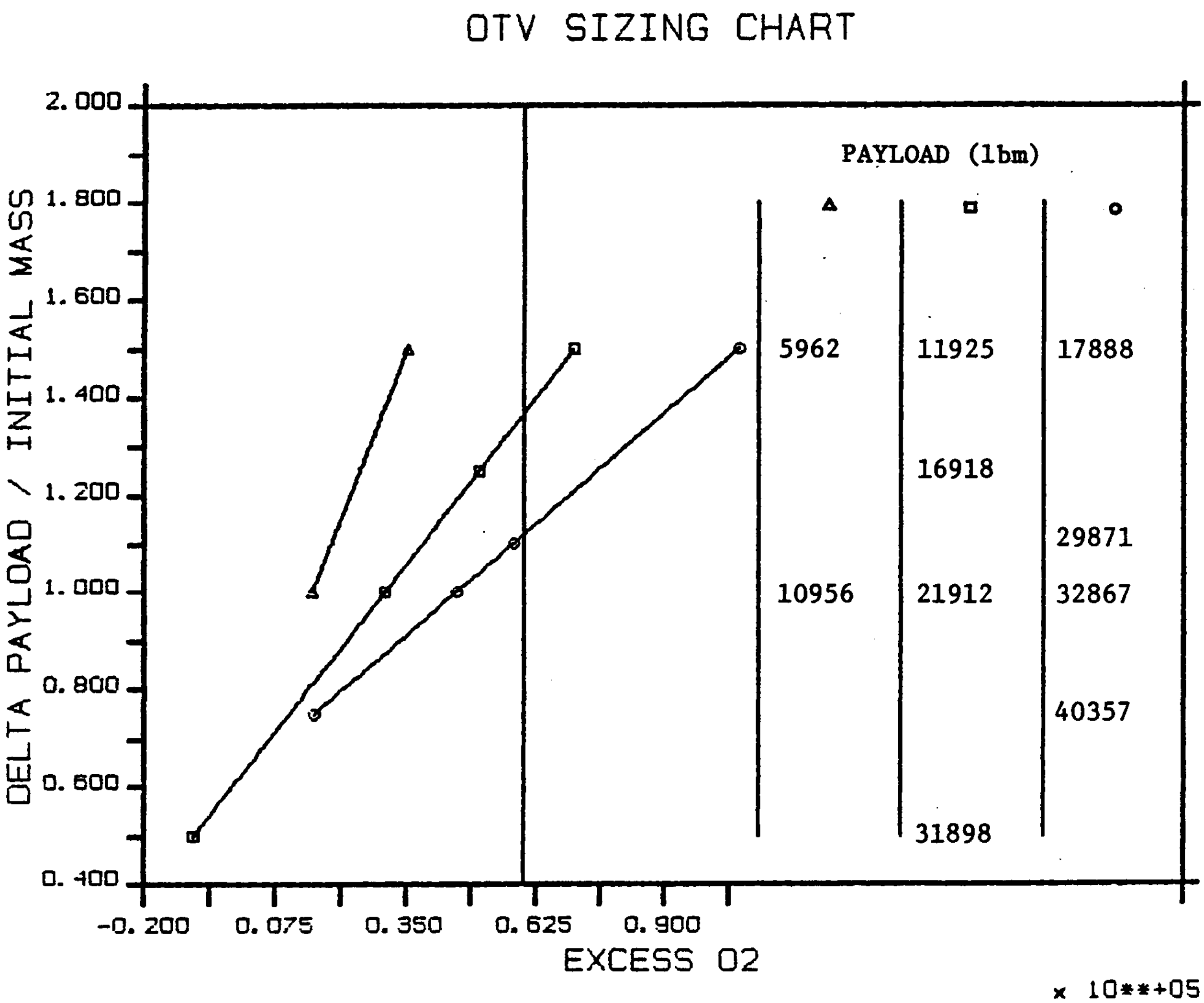
Total O_2 consumption	226,062 lbm
Total H_2 consumption	37,677 lbm
Total burn time	28.25 min

Table 1.3. Mission Requirements for Lunar to Earth Transfer

between overhauls. Also, the largest g load experienced by the structure has a value of only 0.837 and is experienced during entry into lunar orbit where the mass is the least.

OTV Sizing

This section discusses a quick and easy method to determine the approximate OTV size required for various payloads. The method uses a chart to illustrate the OTV performance limits in terms of payload transfer for the retro-braked vehicle. The curves for the chart (Figure 1.6) were generated by assuming that the dry mass of the vehicle is twenty percent of the initial propellant mass. Ten to fifteen percent is normally used, but twenty percent is allowed here to account for the mass of the extra tankage necessary to transport the LOX from lunar orbit. After returning to LEO, enough oxygen to get back to lunar orbit is subtracted from the payload delivered to LEO. The remaining payload is called "Excess O_2" and is available for other missions. Note that for a given initial mass, as the delta payload is increased, the excess oxygen increases, but the payload delivered to lunar orbit decreases. In order to increase the lunar payload, a larger initial mass would be required.



▲ 100000 lbm Vehicle configuration.
◘ 200000 lbm Vehicle configuration.
○ 300000 lbm Vehicle configuration.

Figure 1.6. OTV Sizing Chart

How to use the chart

To use the OTV sizing chart follow the instructions below.

A. Mission defined by oxygen returned to LEO.

1. Choose the amount of oxygen to be delivered to LEO.
2. Find this value on the horizontal axis of the plot.
3. Move vertically upward until intersection with the curve corresponding to the initial vehicle mass.
4. Move horizontally to the right until the column corresponding to the correct initial vehicle mass is reached. The value in this column indicates the amount of payload which can be delivered to lunar orbit. Some interpolation may be necessary.

B. Mission defined by payload to be delivered to lunar orbit.

1. Choose the amount of payload to be delivered to lunar orbit.
2. Using the column which corresponds to the correct initial vehicle mass, locate the vertical position of this payload mass.
3. Move horizontally to the left until intersection with the curve for the corresponding initial vehicle mass is reached.
4. Move vertically downward to the horizontal axis to read the amount of excess oxygen which can be delivered to LEO.

SUMMARY

Although the design requires large amounts of propellant, it uses technology that is available today and is of an uncomplicated nature. One possible solution is to use this design until aeroassist braking becomes more thoroughly developed at which time the more economical aeroassist design could be implemented. A parametric cost analysis (based on the various system weights of the OTV) yielded a design and development engineering cost of 1316.4 million dollars (Table 1.4). This cost is slightly less than the cost predicted for the aeroassist braked OTV. It is possible, though, that the savings could be even greater since part of the technology for the aeroassist OTV would have to be developed.

COMPONENT FORMULATION:

Structural	$C_{D1} = 3.084\ (\text{wt.})^{0.38}$	$= 103.09$
Avionics	$C_{D2} = 1.95\ (\text{wt.})^{0.5}$	$= 121.18$
Propulsion	$C_{D3} = 4.584\ (\text{wt.})^{0.5}$	$= 498.69$
System Level	$C_{D4} = (C_{D1} + C_{D2} + C_{D3})$	$= 593.39$
TOTAL COST	$C_T = C_{D1} + C_{D2} + C_{D3} + C_{D4}$	$= 1316.4$

SYSTEM FORMULATION:

TOTAL COST $\quad C_T = 8.785\ (\text{DRY WEIGHT.})^{0.5} = 1406.97$

* All cost are in millions of dollars.

Table 1.4. Cost Analysis

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AEROASSIST BRAKED CONFIGURATION

OVERVIEW

Aeroassisted braking is the process by which a space vehicle uses the upper atmosphere to slow down. This process is performed by one or more passes of the vehicle through the atmosphere transforming kinetic energy (velocity) into thermal energy (heat). For this reason a heat shield must be used in order to protect the vehicle from the intense heat generated.

Using this maneuver the main engines need provide only a small velocity change (approximately 330 ft/sec) to insert the OTV into LEO.This reduction in engine use requires significantly less propellant for the mission. The savings in propellant represents the greatest advantage in aeroassisted braking. The major drawback to aeroassisted braking is a lack of experience. This area is currently being studied and could be thoroughly understood by the early 1990's, thus fitting the needs of the aeroassist braked OTV.

CONFIGURATION

The main components of the aerobraked OTV, as shown in Figure 2.1, are four propellant tanks (two hydrogen and two LOX), an aerobrake, three rocket motors, the supporting framework, and a payload. The OTV is designed to be fully automated requiring no crew. Provisions are made, however, for humans to be transported via a manned payload module so that crew rotations will be possible.

Propellant Tanks

The main consideration limiting the propellant tank diameter was that the tanks must fit into the shuttle cargo bay (15 ft. diameter) for transport to LEO. Calculations showed that the desired mission could be accomplished with four propellant tanks--two oxygen and two hydrogen. The LOX tanks are 9 ft in diameter while the hydrogen requires 12.5 ft diameter tanks. The tanks will be constructed of aluminum, wound with a composite shell. The shell will be tailored to withstand the increased pressure caused by deceleration during aerobraking. The composite shell structure will also provide some protection from micro-meteoroid impact. Hard points will be incorporated into the shell at the maximum diameter to facilitate attachment to the OTV framework. The hard points must be designed to radiate stresses over large portions of the tank through the composite shell. The tanks will be spherical, or very nearly so, to keep the vehicle as short as possible because the aerobrake diameter is directly related to the vehicle's length (Ref. 5). The LOX tanks, in addition to providing oxygen for combustion, will serve to transport

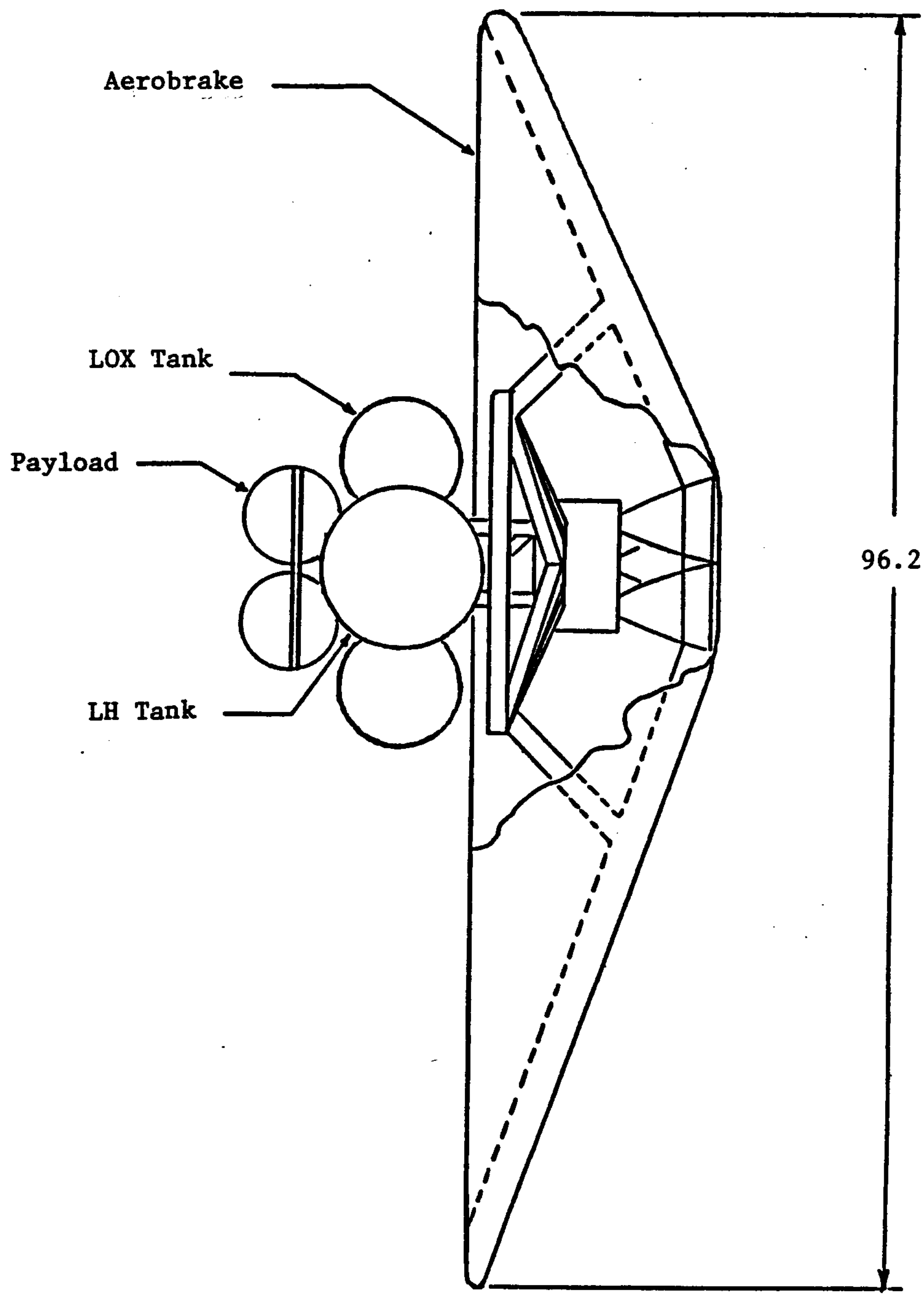


Figure 2.1. Aerobraked Orbital Transfer Vehicle

oxygen payload from LO to LEO. Four tanks were chosen to give enough flexibility to carry large loads to and from the moon.

Aerobrake

For this configuration to accomplish orbital transfers from lunar orbit to low Earth orbit an aeroassist maneuver will be used. An aeroassist maneuver involves reducing the speed of a space vehicle by using the aerodynamic forces generated during a pass through the Earth's atmosphere which dissipates kinetic energy. Aerobraking is initiated by using a small propulsive maneuver to lower the trajectory to a level where the spacecraft makes a brief pass through the atmosphere. This pass must be a precise maneuver so as to prevent re-entry too far into the atmosphere and burning up the craft, and also to prevent losing too little velocity and coasting into the Van Allen radiation belts.

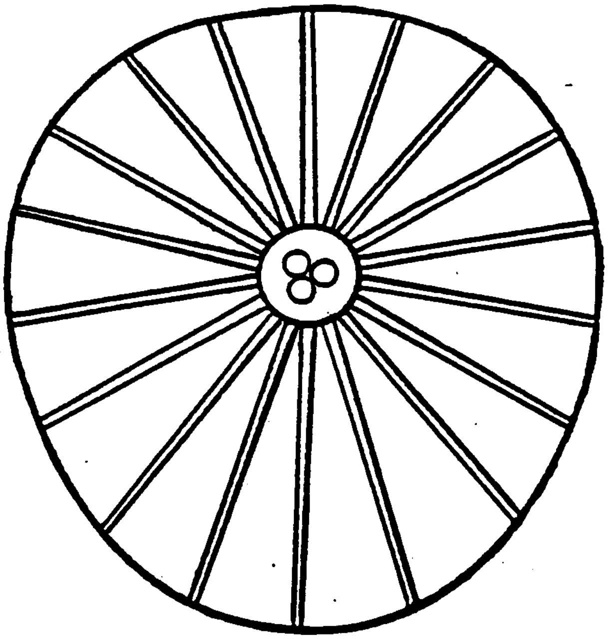
This maneuver reduces the expenditure of propellant as compared to an all-propulsive braking maneuver. This savings simply means that aerodynamic drag from the upper atmosphere is substituted for part of the propulsive braking maneuver, and therefore provides a more efficient procedure for braking the vehicle and providing the total velocity decrement necessary to transfer from lunar return trajectory to the low Earth orbit. This maneuver is also complicated by atmospheric density variations for which the vehicle must compensate. This compensation can be accomplished using an aerobraking concept which varies vehicle drag directly to correct for density variations by using changes in engine thrust to vary the shape of the surrounding field of flow (Ref. 1).

Multiple passes through the atmosphere were rejected because of the difficulty in hitting the atmospheric entry window each pass.

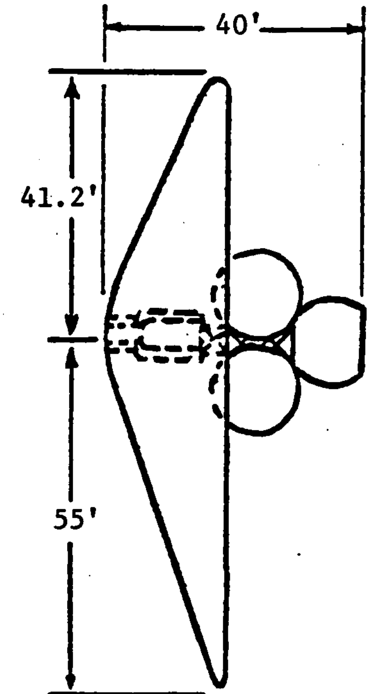
Attempting to do so would not be cost effective, would greatly increase the time required, and would endanger the craft, any passengers, and cargo.

The drag device chosen for the aerobrake design is the conical lifting brake with an area of 7872.75 sq ft . Since the vehicle will remain in orbit outside the atmosphere between missions, the brake will be continuously deployed until it is no longer useful. Therefore the brake will only be folded to fit in the shuttle cargo bay when it is initially taken into orbit. The drag brake was designed as a 70 degree spherical cone with a lift coefficient of 0.0487 and a drag coefficient of 1.6472 or a L/D of 0.0295. See Figure 2.2. These values were obtained by using Newtonian flow theory. This geometry allows for reduced surface heating and favorable aerodynamic stability. The engines will also be fired at a highly throttled condition through an opening in the center of the brake to provide cooler gas on the shield's surface and alleviate the high heating rate at the stagnation point. The ratio of brake diameter to vehicle length should be approximately 2.06 to prevent wake impingment on the vehicle surface which will cause high local heat fluxes (Ref. 5). The brake was designed to revolve about a hinge point connected to the main structure in order to achieve attitude control.

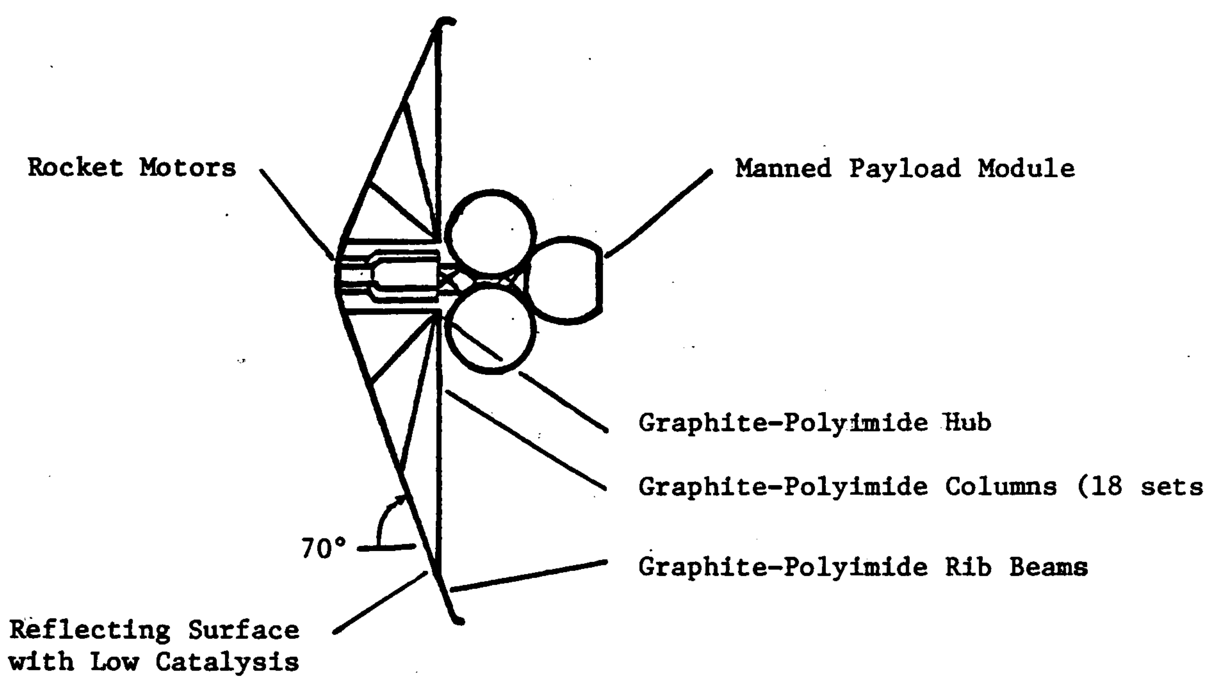
Although this design will effectively obtain the rapid decelerations and large velocity decrements required for orbital change maneuvers from high altitudes, the design will not be capable of making the large plane-inclination changes characteristic of low Earth orbit rendezvous without some assistance. With only one pass through the atmosphere there is simply not enough time to conduct the change. The



Front View



Side View



Cut-Away View

Figure 2.2. Aerobraked OTV with Schematic

vehicle will therefore implement propulsive thrust as an addition to its aeroassist capability. The LEO orbital change maneuvers are accomplished by propulsive thrust maneuvers involving three burns: the first burn achieves elliptical transfer orbit; the second burn produces the required plane inclination change at apogee; and the third burn recircularizes the vehicle into the target LEO after the aeroassist maneuver (Ref. 5).

The brake design shown in Figure 2.2 will consist of five major components: 1) the surface fabric, 2) rib beams supporting the fabric, 3) the insulator between the surface fabric and the support beams, 4) the columns supporting the rib beams, and 5) the mechanism supporting the entire apparatus. The surface fabric will be a reflective material such as silica or Nextal. These materials will be woven into a cloth on the order of 0.25 mm thick. The support structure, consisting of the ribs, columns, and hub, will be made of a lightweight carbon composite such as graphite polyimide. Thermal control paints will cover all exposed surfaces of these components (Ref. 5).

Stability is a main concern for the conical lifting brake. Since the brake is asymmetrical, it can become unstable; the negative static margin resulting from the front pressure surface and rear cargo location. Longitudinal stability may be provided by inclining the brake backward at an angle which ensures that the center of pressure is aft of the center of gravity. Roll stability may also be obtained, due to asymmetry of the overall geometry (Ref. 4).

Rocket Motors

The OTV main propulsion system will consist of three liquid-fuel rocket engines. The engine type will be a derivative of the Pratt & Whitney RL-10 family known as the 34.5K. This engine type uses liquid hydrogen as its fuel with liquid oxygen as the oxidizer. This engine type was chosen (over the others of the RL-10 family) because it is reusable and has a higher thrust rating. It is, however, still in the development stage at this time. Table 2.1 displays the engine specifications.

Design Structure

The tanks, engines and payload are all attached to a supporting structure. The arms for tank attachment will be a composite material with an I-beam type cross section. These arms are attached to a central box truss framework. The truss members are also of composite construction. These materials must be able to exist in the space environment (very high vacuum, extreme high and low temperatures, bombardment by protons and neutrons, and ultraviolet radiation) without losing their desirable qualities (high stiffness to weight, high strength to weight, and low coefficient of thermal expansion). They must also withstand the large compressive forces experienced during both acceleration and deceleration as well as torsional and bending loads experienced during orbital maneuvering. To the respective ends of the truss are attached the rocket motors and payload. The necessary avionics and the propellant for reaction controls will be arranged within the "carry-through" truss.

Performance Parameter	Expected Value
Vacuum Thrust, lbs.	34,500
Mixture Ratio	6:1
Chamber Pressure, psia	560
Area Ratio	26:1
Vacuum Specific Impulse, sec	431
Operation	Full Thrust and Cruise Thrust
Weight, lbs.	325
Installed Length, in	76
Propellant Flow Rate, lbm/sec	80.1
Diameter, in	35
Life (with time between overhauls)	7 missions
Year Available (potential)	1990
R & D Cost (million $, 1984)	140

Table 2.1. Engine 34.5K Specifications

Payload

The OTV manned payload will be cradled between the four propellant tanks and attached to the "carry-through" truss. The size of the payload returned to LEO through aerobraking must be constrained such that the total vehicle does not exceed approximately forty feet so that flow impingment will not occur. A basket system of four spheres was chosen to transport the LOX. It also will be attached at the front end of the "carry-through" truss. This configuration allows the OTV to maneuver near the space stations, which are equipped with a similar basket,and transfer the oxygen tanks all at once,and produce only compressive loads on the sphere system through the use of baskets which are permanently attached to the space structures involved in the transfer operation. This system was chosen because of its simplicity of construction and maintenance, low weight, and low operating cost.

Figures 2.3 and 2.4 show top and side views of the basket system respectively. There are four connectors in the basket, one for each oxygen tank. The connectors are mechanical devices which extend into the tanks, lock and pull the tanks firmly into position in the basket. Located in the center of the basket is a transmission which may operate each of the connectors collectively or independently, thus allowing the tanks to be loaded one after the other, then transferred together.

Assuming a pressure of 300 psia in the tanks gives a specific volume of oxygen of 0.01845 ft^3/lbm. The mission requirement of 60,000 lbm of oxygen per mission set the diameter of the spheres to 8' 6-1/2" the width of the basket at 17' 7", and the diagonal width of the basket at 25 ft.

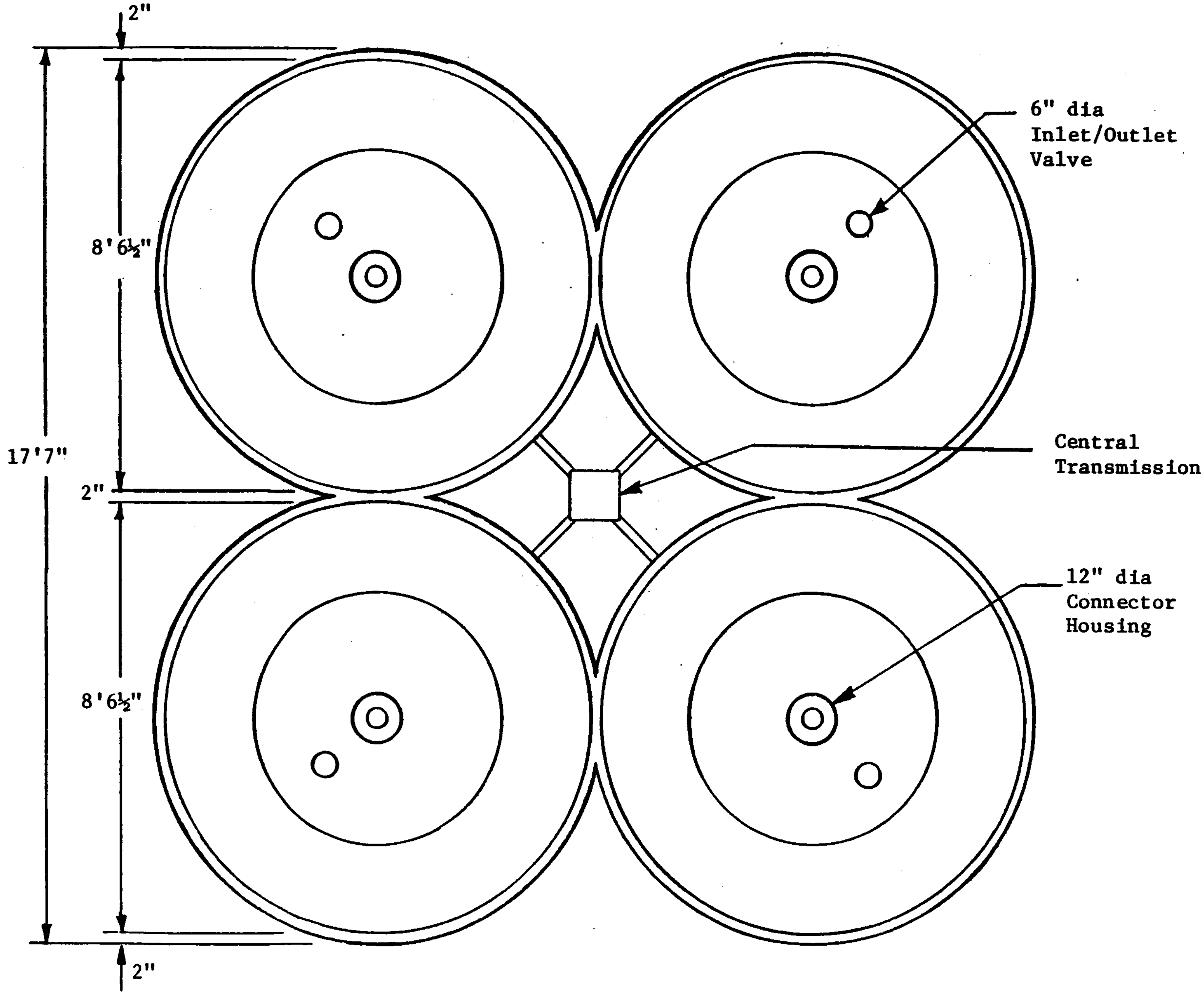


Figure 2.3 Top View of LOX Basket System

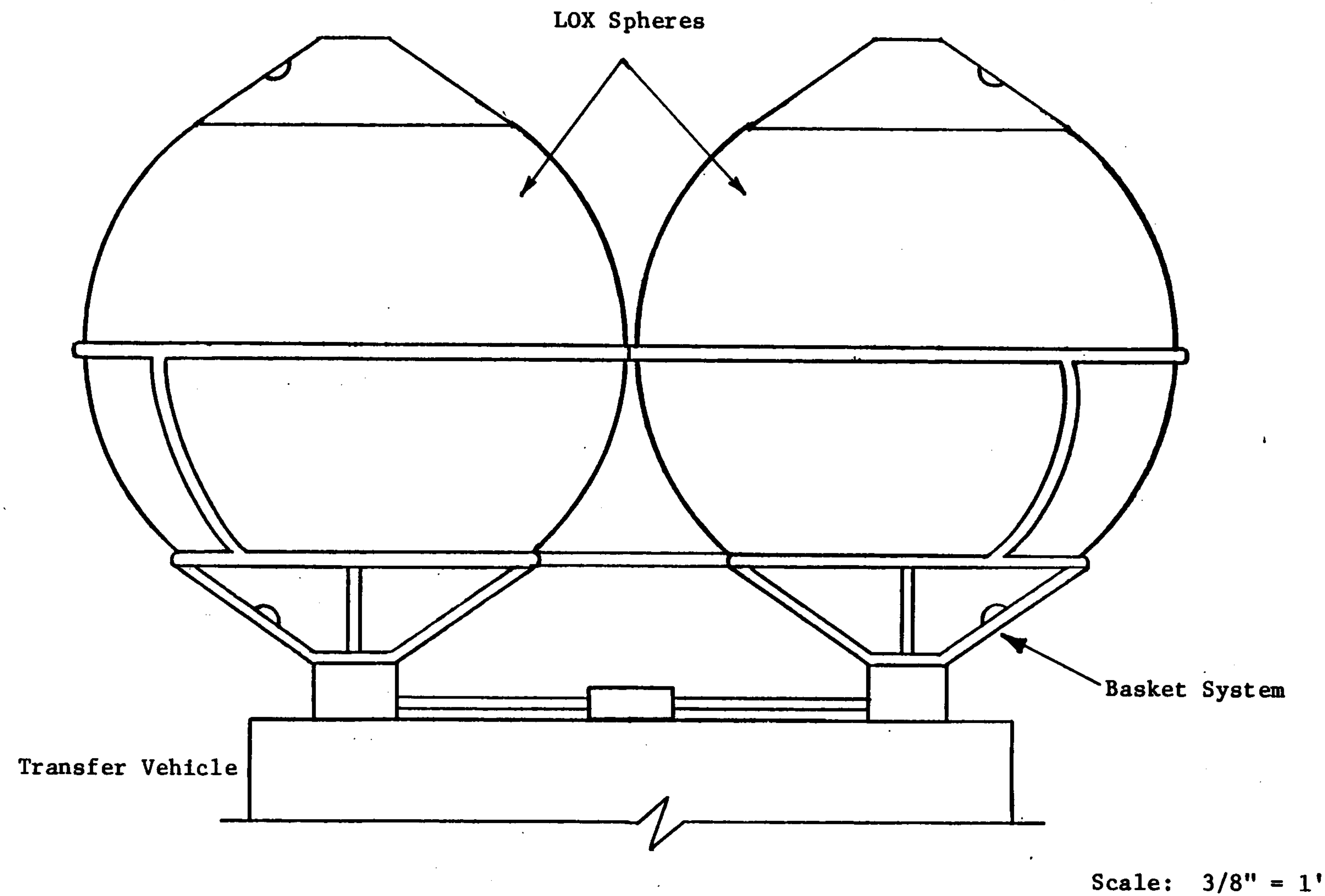


Figure 2.4. Side View of LOX Basket System

Figure 2.5 shows the male/female connector system and illustrates how the plunger, which is activated from the transmission through a set of worm and ring gears, is retracted forcing the four cams outward and into the recesses in the female connector of the oxygen sphere. The rotating action of the shaft turns a worm gear which runs on a ring gear that rotates about the plunger. This rotation, through the threads cut into the plunger causes the plunger to move in either direction depending on the input shaft rotation direction.

It was decided to use 7075 T6 aluminum alloy for the bulk of the structure. However, all bearing surfaces such as gears, plunger, and cams are to be made of steel. The basket will be 2" dia. 7075 T6 tubing with 1/4" wall thickness while the connector housings will be forged 7075 T6. The material selection was based on such factors as strength, weight, and cost. The maximum loading experienced by the structure should be approximately 3 g's. The stress in the connector housing and the basket, however, even at 8 g's is only 791 psi and 66 psi respectively—well within the material limitations.

As seen in Figure 2.3 and 2.4, a fueling inlet/outlet is located on the side of the sphere to facilitate filling the spheres on the lunar surface. With this device, it would also be possible to use the oxygen tanks as fuel cells for the return trip from the moon if they were needed as such.

The analysis performed on the basket-connector system shows that the configuration is feasible and can withstand any and all foreseeable loads imposed upon it during the prescribed mission. These loads can be handled when the basket system is used in a single layer. If stacking of the baskets is attempted to increase payload, the loading becomes a

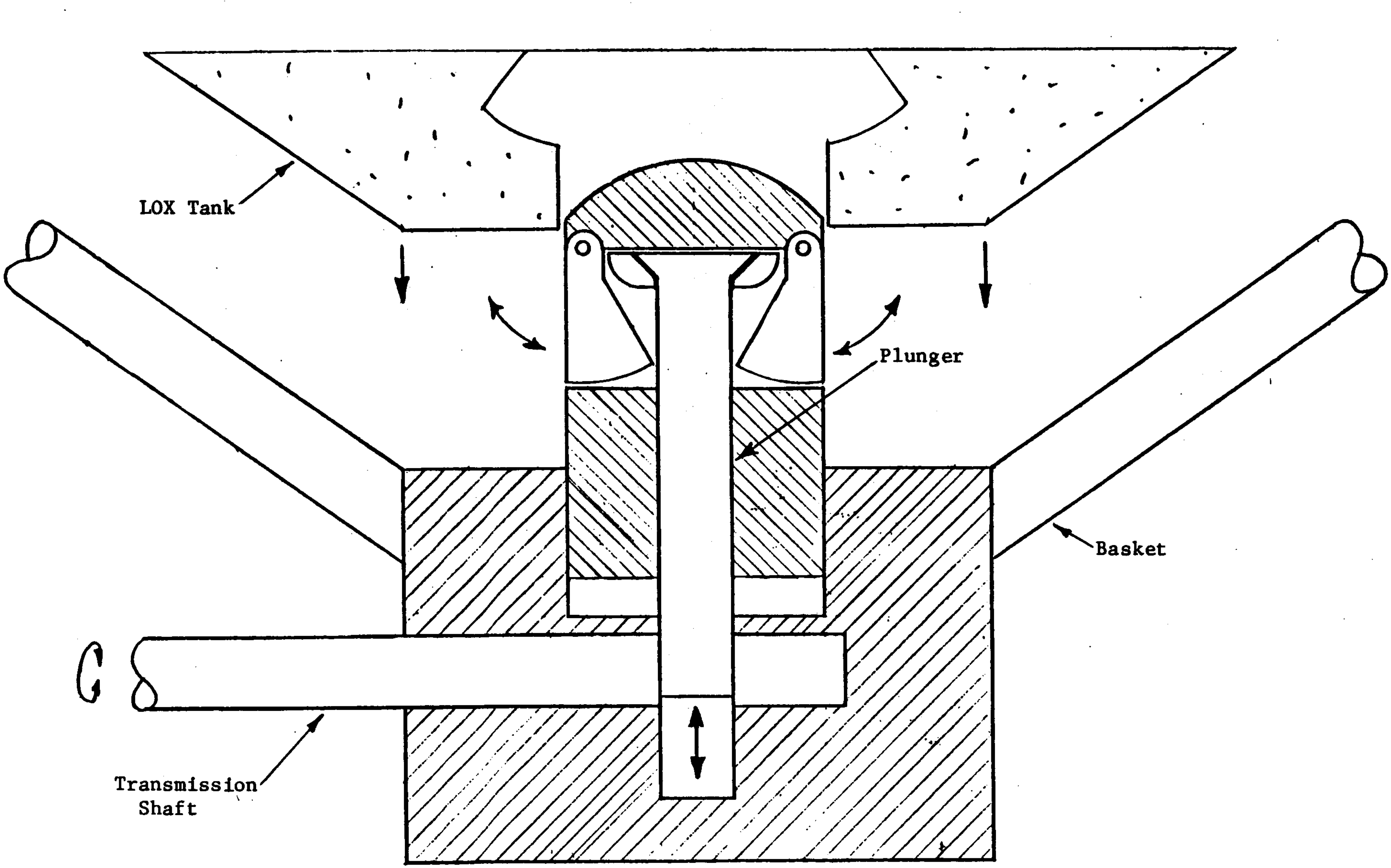


Figure 2.5. Connector Detail

complex problem. Due to the manner in which the baskets are attacked, stacking is not recommended. If a larger payload is required, the spheres should be resized to hold the increased payload.

MISSION REQUIREMENTS

The propulsion requirements (thrust required, accelerations, burn time, and propellant required) were calculated by breaking the trip into two different segments. These segments were then subdivided as follows:

1) Outbound to Moon
- *a) leave low Earth orbit*
- *b) mid-course correction*
- *c) enter lunar orbit*

2) Return to Earth
- *a) leave lunar orbit*
- *b) mid-course correction*
- *c) return to low Earth orbit*

The outbound leg of the trip required a velocity change approximately 10,500 ft/s to leave Earth orbit. The mid-course correction had a velocity change of 150 ft/s. The velocity change for lunar entry was approximately 2200 ft/s. The mass ratios were 2.055, 1.011, and 1.155 respectively.

The velocity changes and mass ratios are the same for the return trip. The aerobraking system will be used to slow the OTV during the return to low Earth orbit. The engines will then be used to place the OTV in the proper circular orbit. The structure was taken to have an acceleration limit of 3 g's (96.6 ft/s). The overall empty mass of the OTV is approximately 25,600 lbm as seen from the mass breakdown in Table 2.2. Table 2.3 displays the calculated propulsion system parameters for each stage of the trip.

The following data represents a typical OTV mission. The overall mass of the vehicle upon exiting Earth orbit (including propellant for

Component	Mass (lbm)
Structure	8,960
Thermal	760
Aerobrake	9,017
Engines	975
Avionics	3,840
Miscellaneous	2,048
Total Mass	25,600 lbm

Table 2.2. Estimated Mass Breakdown

Stage	Acceleration, ft/sec^2 (initial, burnout)	Propellant Used, lbm	Thrust, lbf	Burn Time, sec.
I	45.43, 93.36	37,611.3	103,500	231.08
II	31.12, 31.46	383.8	34,500	4.82
III	31.46, 36.33	4,793.1	34,500	69.93
IV	31.86, 36.79	14,039.1	103,500	69.05
V	12.26, 12.40	985.5	34,500	12.23
VI	24,80, 29.08	2,099.8	69,000	13.23

I - Earth Exit and Escape

II - Outbound Mid-Course Correction

III - Lunar Entry

IV - Lunar Exit

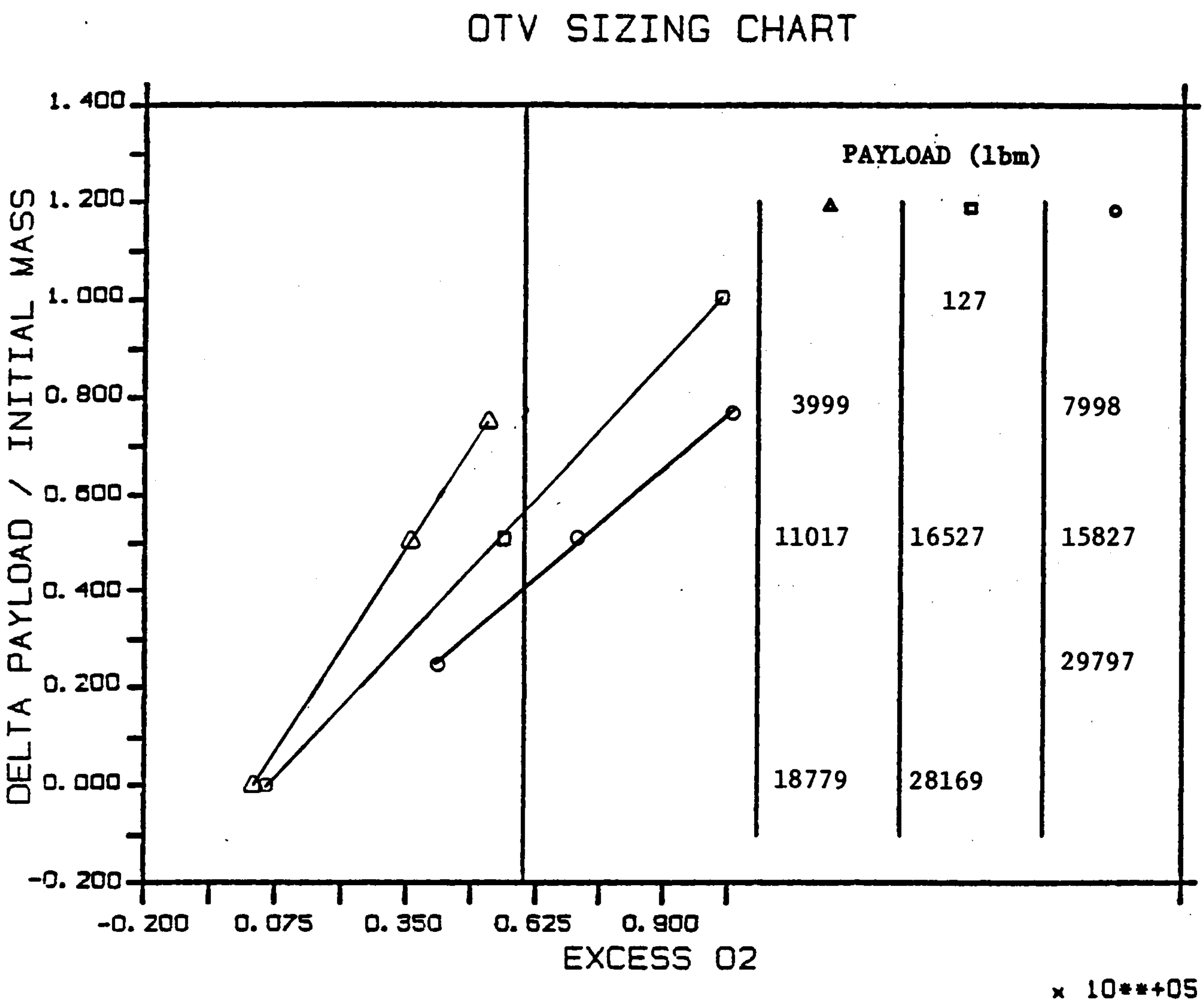
V - Return Mid-Course Correction

VI - Earth Orbit Injection (after aerobraking)

Table 2.3. Propulsion Requirements and Associated Parameters

earth exit, mid-course correction, and lunar entry) is 73,359.2 lbm. Burn times were calculated by dividing the velocity change at each stage by the initial acceleration at that stage. The acceleration will, of course, not be constant so that the value of burn time is not correct. However, the burn time value represents an upper bound as the acceleration will increase throughout the burn thus resulting in a smaller burn time (in actuality). The total mass of the vehicle upon lunar exit is approximately 104,614.1 lbm. This includes a payload and propellants required for lunar exit, mid-course correction, and final orbit adjustment. This particular scenario provides only approximately 5,600 lbm of payload to lunar orbit and 25,000 lbm to LEO. Due to the extra volume provided by the propellant tanks, some mission flexibility is provided without changing the basic design. For the OTV to bring back approximately 75,000 lbm, the initial mass would have to be increased to 100,000 lbm. This mission would increase the total required propellant to 94,000 lbm including 13,500 lbm of hydrogen, and allows close to 5,600 lbm of cargo to be taken to lunar orbit. Figure 2.6 is a sizing chart for the aerobraked OTV. The chart was constructed using the same procedure and assumptions as for the retro-braked OTV plus a heat shield mass of fifteen percent of the initial mass.

To stay within the stipulated acceleration limit, the engines are fired at a throttled level except during Earth and lunar orbit exits. The three engine design will also provide a safety margin in case of an engine failure. The engines are expected to have a life of seven missions given the expected mission requirements.



▲ 100,000 lbm Vehicle Configuration

◘ 150,000 lbm Vehicle Configuration

○ 200,000 lbm Vehicle Configuration

Figure 2.6. Aerobraked OTV Sizing Chart

SUMMARY

A parametric cost analysis showed that the aerobraked OTV would have a design and development engineering cost of approximately 1405.6 million dollars. This cost, while being roughly 90 million dollars more than the retro-braked design, would quickly be repaid in propellant savings. The aeroassist OTV can return a surplus of 60,000 lbm of oxygen and use little more total propellant than the amount of hydrogen alone required for the retro-braked OTV to do the same. This savings would more than make up for the extra cost of the aeroassist braked OTV after a few missions.

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LUNAR LANDER

OVERVIEW

This section presents the design of a system which enables a Lunar Lander (LL) to operate between the lunar surface and the lunar space station. The manned vehicle is designed to deliver supplies to the moon base and bring liquid oxygen back from the lunar surface to the lunar space station. An extensive study of the use of an electromagnetic railgun to launch the LL results in the conclusion that at the current time this would not be the most ideal launching system despite the savings in amount of propellant used in conventional launching. A more feasible approach to launching the LL is shown along with descent trajectory procedures.

CONFIGURATION

The lunar lander has to be capable of launching and landing from the lunar surface with a maximum payload of 20,000 lbm. The payload will consist of liquid oxygen contained in spherical pressure vessels, power equipment, and food supplies capable of sustaining life on the moon.

The proposed lunar lander can be seen in Figure 3.1. This figure shows a cargo bay which will be used to carry LOX spheres. The sphere sizes used in the basket concept are 9.5 ft and 8 ft 6.5 inches in diameter for the retro-braked and the aerobraked configurations, respectively; therefore, either of the two can be easily stored in the LL cargo bay, which is approximately 20 ft in length. A crane stationed on the lunar space station will transfer the LOX spheres from the LL cargo bay to the OTV's payload baskets. Some of the LOX will be used to refuel the OTV and the LL can then carry the empty spheres back to the lunar space station. Since the spheres are made of polished aluminum, they have the ability to reflect heat and insure safe transportation of the LOX.

The main cabin of the LL shown in Figure 3.1 allows the pilot and co-pilot to have a visual reference on takeoff from the surface and at the time of docking. The flight deck will house the pilot, co-pilot, and all the controls for lift-off,landing, and docking procedures. Two extra seats are also provided behind the cockpit to handle the exchange of crewmen in the event of an emergency or at the time of normal crew rotation. Thus, the total seating capacity is four and the length of the flight deck is approximately 10.5 feet. The area under the flight

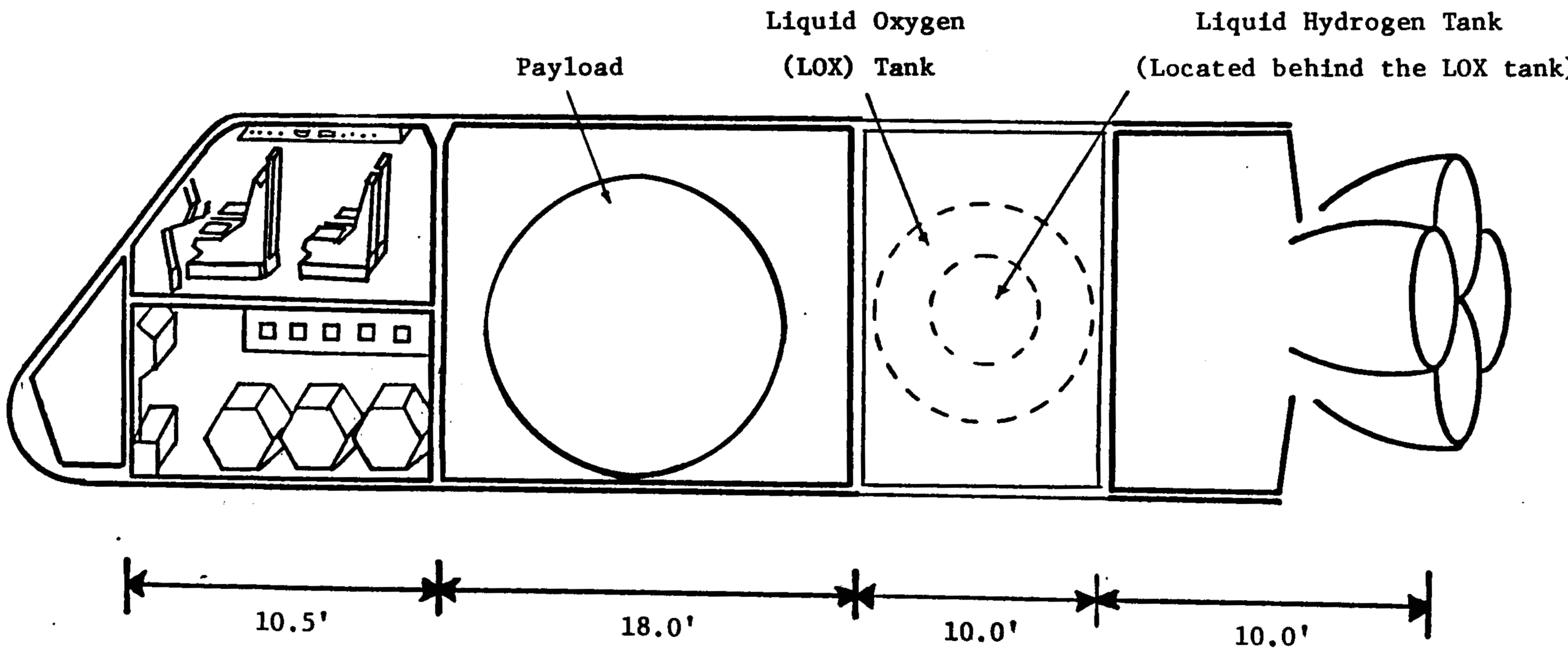

Figure 3.1. Detailed Lunar Lander

deck will house the guidance systems, attitude control systems, and the life support systems for the vehicle.

The cargo bay located behind the flight deck will house the oxygen tank as well as food and hydrogen to replenish supplies on the surface. This compartment can be loaded through cargo doors which operate similar to the doors on the shuttle. Machinery needed for mining processes will also be housed in the cargo compartment on the return trip to the lunar surface.

The propulsion unit is located behind the flight deck and cargo compartments. The LL will be propelled by 4 RL-10II-B step throttle engines. Each engine delivers 15,000 lbf thrust, has an Isp of 460 seconds, and a proposed life of 10 missions. During the landing process, the engines will provide the reverse thrust necessary to brake the vehicle as it approaches the surface. The vehicle will land on four retractable legs. It is assumed that an advanced guidance system will enable the craft to land without a visual fix of the surface by the pilot. Reaction rockets will be used to guide the LL during docking procedures.

The overall length of the LL vehicle is 42 feet. The flight deck and successive compartments will be 10.5 feet high and the entire vehicle will be 17 feet in width. The LL will have a center of gravity located 12.5 feet as measured back from the nose. The expected C.G. shift from a loaded LL to an empty LL can be considered negligible.

ELECTROMAGNETIC RAIL GUN LAUNCHER

One device considered for the launching of the LL to the lunar space station from the lunar surface was an electromagnetic rail gun. The rail gun is a simple concept. The rail contains capacitors placed at different positions along the length of the rail. The armature is made of a magnetic material and serves as the "mass driver" for the rail gun. A switch is also connected between the main power source and the rail.

The rail gun is basically a large circuit which is left open until the actual lift-off process begins. At this time, the circuit is closed by throwing the switch between the main power source and the rail. When the switch is thrown, a current passes through the rail and causes the capacitor located closest to the armature to discharge. This creates a magnetic effect and causes the armature to accelerate down the rail. Each capacitor, when reached by the armature, will discharge into the rail and continue to accelerate the LL. It is for this reason that the capacitors must be placed at precise intervals in order that the specified maximum acceleration will not be exceeded.

The proposed rail gun has one track positioned on top of the other. Upon examination, it should be noted that the bottom rail is shorter than the top rail. The difference in rail lengths corresponds to the difference between the armature "arms" on the OTV. Once the OTV has been accelerated down the rail, both the front and rear arms must leave the track at the same time. This will eliminate the possibility of creating a pitching moment on the LL due to an additional force on the rear of the vehicle. The difference in height of the arms is linked to the design of one rail on top of the other. The starting position of the rail is slightly elevated. This elevation causes the LL to move slightly once it has been released for launch. The reason for an initial velocity is to keep "hot-spots" from forming in the rail once the switch has been thrown. Hot-spots will cause a breakdown of the rail at that particular point or in other points and would prove to be costly in repairs to the rail.

Once the LL has been prepared for flight to the lunar space station, it will be placed on the rail by the crane. When it is released, it will be accelerated down the rail.

To employ the rail gun, one limiting factor has to be taken into consideration, the amount of energy produced by the gun creates extremely large g-forces. Therefore, the amount of acceleration supplied at launch is limited by the number of g-forces that a man can withstand. Because of this stipulation, the rail has a maximum acceleration of 8 g's. This value will have to be used to determine the total length of the rail and the separation distance between the capacitors. Another problem which arises from use of the rail gun is the amount of power needed to make the rail work properly.

On the surface of the moon, the LL will be given an initial velocity by the rail gun. The exact specifications for the rail gun, such as length and launch angle, are dependent on the location of the moon base.

TRAJECTORY ANALYSIS

Ascent Trajectory

It was found that using the electromagnetic railgun to launch the LL, the optimal launch angle will be about 30 degrees. This launch angle will give a launch velocity of 3878.84 ft/sec and a track ground distance of 51,914.9 ft. These values were calculated using a 8 g acceleration factor. Lowering the g-load would increase the track distance and launch angle drastically. Because of the high acceleration factor along with the lengthy track necessary, the electromagnetic gun would not be a feasible system of launching at this time. Therefore, the conventional vertical launch will be used.

Descent Trajectory

After the LL has delivered or received a payload at the lunar space station, it must return to the lunar base. The technique used for this maneuver is the same procedure used during the Apollo Mission. The LL will deorbit using retro-thrust to acquire a velocity impulse of about 328 ft/sec the LL will then coast in a ballistic elliptical trajectory to an altitude of approximately 15.5 miles. At this altitude, the final braking phase will begin and will reduce the LL velocity to less than 16.4 ft/sec at an altitude of about 700 ft. The terminal braking phase will reduce the lateral velocity to zero and will pitch the thrust vector to give a hovering altitude. At this time, corrections in landing position can be performed with negligible fuel consumption. For an orbit of 100 nautical miles a 90 degree range angle (the angle from lunar deorbit to initiation of the final braking phase) gives the most fuel efficient braking techniques.

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REACTION CONTROL SYSTEM

The reaction control system must supply the thrust necessary to perform attitude adjustments which allow the OTV to be controlled. The system must be reliable, powerful, and at the same time be efficient enough to prevent excessive mass. The criteria involved in the RCS selection are high performance propellant, proper storage and supply techniques, and fuel availability. Development costs are also a consideration since there would be a budget for the OTV.

Fuels

The use of monopropellants as the fuel for the reaction control system was eliminated as an alternative after brief research. The first problem encountered with monopropellants is their low performance qualities as a group: the specific impulse (Isp) for most monopropellants is much lower than those for bipropellants. Also, those monopropellants having reasonable performance characteristics are inherently unstable. This instability is crucial because monopropellants must decompose completely and immediately once ignited. Furthermore, the unstable nature of monopropellants presents a problem in safe storage; high performance monopropellants can be both explosive and corrosive.

After some preliminary investigation, bipropellants, specifically oxygen/hydrogen systems, were chosen as the final propellant type for the reaction control system. The first design decision was to use oxygen as the oxidizer in the bipropellant system. The main factor in this decision was the availability of oxygen. In addition, bipropellant *systems usually have an oxidizer/fuel ratio (r) greater than two. This ratio means that a smaller amount of fuel weight, compared to the available oxidizer, would need to be transported from Earth. This decrease in the weight of the supplies needed to be brought from the surface of the earth will save a considerable amount of money over an extended period of time. Finally, using oxygen as the oxidizer is preferable because many different fuels may be used with it to obtain high performance. Because most bipropellant systems are similar, minor modifications to the system could allow for different fuels to be used. Three of the most acceptable fuel alternatives are hydrogen, kerosene, ethanol and methanol (Ref. 5).*

The storage/supply system for the propellants will be of the pressurization type. Currently, the main propellant tanks will be maintained at 300 psi using helium as the inert pressurization gas. Pump systems were considered as a supply method, but they are usually mechanically complicated. This complexity can lead to early system failure and, without proper sealing, fuel leakage. Simpler gas pressurization systems are limited to low thrust durations (30 seconds); which is precisely what this reaction control system requires.

RCS Accumulators

The RCS accumulators for the gaseous oxygen and hydrogen will be initially charged to allow immediate use of the RCS thrusters. They will be maintained at or above a minimum operation pressure by the tapping of the boiloff from the main liquid oxygen and liquid hydrogen storage tanks, which supply the main thrusters. It may be assumed that a small amount of helium will be encountered in the boiloff gases, but no literature was uncovered which predicted any problems with the mixed propellant performance. If the propellant boiloff is greater than the capacity of the RCS accumulator system, the remainder will be dumped overboard or used in some other area.

Configuration

The thrusters themselves will consist of fuel and oxidizer valves which will mix and inject the oxygen and hydrogen into the thrust chamber. A standard spark igniter will be used for ignition. The nozzle will be constructed of titanium. Titanium has a melting point of 3020°F and the hydrogen-oxygen flame temperature is 4500°F but since the firing times are one second or less, the ablation will be minimal and the nozzle should outlast the rest of the RCS system. If the ablation should become a problem, a ceramic insert could be used to raise the melting temperature to well beyond the flame temperature.

The LOX and liquid hydrogen storage tanks for the main thrusters will be maintained at a minimum pressure of 300 psi. With a 25 psi loss of pressure due to piping and valves, a pressure drop across the nozzle of 50 psi can be used. The pressure in the thrust chamber can thus be held at 200 psi by a regulator in the line coming off of the accumulators. This will allow for changes in the RCS accumulator pressure while maintaining a constant pressure in the combustion chamber. This will give a constant thrust which will simplify control design and operation. The parameters for the RCS are shown in Table 4.1.

The thrusters will be doubled at each location to provide the redundancy necessary for a failsafe system. If one system has a

Performance Parameter	Expected or Assumed Value
Thrust Required	20 lbf/thrust
Mixture Ratio	4.0
Specific Impulse	345 sec
Chamber Pressure	200 psi
Storage Pressure	300 psi
Pressure Loss in Trans.	50 psi
Pressure Across Nozzle	50 psi
Molecular Weight	10
Burn Temperature	4500° F
Ratio of Specific Heats	1.3
Throat Area	0.053 sq.in.
Characteristic Velocity	5898.4 ft/s
Exit Velocity	11,109.0 ft/s
Fuel Flow Rate	0.05795 lbm/s
Thrust Coefficient	1.88

Table 4.1. Summation of Performance Parameters for RCS System

complete failure, this doubling will ensure there will be no loss of reaction control.

The life expectancy of the single RCS system is approximately 10 hours of firing time. With firing of one second or less, each motor can be fired a minimum of 36,000 times before replacement is necessary. With the double system, the life is doubled to 72,000 firings since the use of each motor will be cut in half.

Development

There are two approaches to the development of this RCS system. The first of these two approaches is to develop the system from scratch. The system will have the most efficiency since state of the art techniques can be tailored to this specific application and all excess weight can be eliminated. The drawbacks to this approach are the development time, cost, and that the system will be untested in actual use. The estimated cost of this development approach is 100 million dollars.

The alternative approach is to modify the existing RCS system currently used on NASA's Space Shuttle. This RCS system uses nitrogen tetroxide as the oxidizer and monomethel hydrazine as the fuel. The Shuttle system has a much higher thrust output per nozzle than the OTV system, because a significant amount of reaction control is necessary during reentry through the atmosphere. The Shuttle system was used as a model primarily due to the fact that the total number of nozzles and fuel supply materials per RCS (the Shuttle has a forward as well as an aft) is comparable with the needs of the proposed OTV system (Ref. 1). The estimated cost to convert this system to the desired specifications is 50 million dollars. The advantages in this method are the lower development cost and that parts of the system would have been used previously on the Shuttle. This prior use of some parts could decrease unforseen problems that often occur in new systems.

The cost for developing this system from scratch or from modifing the Shuttle's RCS system may seem large, but considering that this system can be used many years in the future and that it uses the propellants that will probably be used for the main thrusters of the OTV, the cost is easily justified. In fact, much of the propellant used by this RCS system is what would normally be lost in boiloff; propellant that is often wasted will be put to good use by this system. The development cost could easily be exceeded in several years by the transportation cost of the non-oxygen oxidizers from the surface of the Earth to both Earth and lunar orbits. Another advantage of the proposed system is that it can be converted to utilize hydrocarbon fuels such as methane and ethanol with minimum modifications. This system would be used for many years to come in either its oxygen/hydrogen state or using either oxygen/methane or oxygen/ethanol. In fact, this system should be useful until the technology for electric or ion thrusters is available which is still many years in the future.

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TRAJECTORY ANALYSIS

The fuel required to accomplish a transfer from a low Earth orbit to a lunar orbit is a function of the trajectories chosen to accomplish this transfer. In order to design an OTV it is first necessary to know the impulse requirements it must meet. First, the impulses required for a coplanar free return trajectory are established. And second, the impulses required to make out-of-plane transfers (when coplanar transfers are not possible) are established.

Coplanar Transfers

The transfer from LEO to LO will be accomplished by launching from LEO into a free return trajectory which will return the OTV to Earth if for some reason the mission must be aborted.

A free return trajectory is one in which the transfer vehicle is launched toward some point in the orbital path of the moon so that the vehicle's motion is influenced by the gravitational pull of the moon in such a fashion that it performs a hyperbolic passage and is returned to the Earth if the vehicle's engines do not fire to place it in LO. A mid-course correction can be applied in order to leave the free return path and match the lunar orbit entry parameters more closely (Ref. 2).

An approximation of the impulse requirements necessary to transfer a vehicle from LEO to LO may be obtained from a patched conic approximation technique, a velocity impulse is applied to the transfer vehicle to place it in what is essentially an escape trajectory with respect to the Earth. The vehicle is assumed to travel under only the influence of the Earth's gravitational field until it leaves the sphere *of influence of the Earth and enters the sphere of influence of the moon, at which time the Earth's gravity is "turned off" and the moon's gravity is "turned on" (Ref. 1). The vehicle then performs a hyperbolic passage around the moon and returns to the Earth if an impulse is not applied to place it in lunar orbit (Ref. 2). The method of patched conics yields good approximations of the required velocity impulses to transfer a vehicle from the Earth to the moon, but does not yield good approximations of the necessary impulses for the return trip because of errors in the entry of the moon's sphere of influence. The hyperbolic orbit altitude above the moon at the point of closest approach and the lunar trajectory orientation will also be in error, but the outbound impulse approximations are fairly accurate (Ref. 1). The impulse requirements for LEO-LO free return trajectories are summarized in Table 5.1.*

Orbital Characteristics

Due to the characteristics of the LEO at an altitude of 270 nautical miles, a simple coplanar lunar transfer can only be accomplished six times per year. At other times, an out-of-plane transfer must be accomplished. An out-of-plane transfer will require more fuel than a coplanar transfer; therefore, a launch window must be determined on the basis of fuel requirements for out-of-plane transfers.

Characteristics of the Moon's Orbit

The orbit of the moon about the earth is nearly circular (the eccentricity is 0.0549) with a semimajor axis length of 238,855 miles. The orbit is inclined at a mean angle of 5.133 degrees to the ecliptic

Maneuver	Impulse Requirement
Trans-Lunar Injection	10,276 - 10,500 ft/s
Mid-Course Correction	150 - 180 ft/s
Lunar Orbit Injection	2,200 - 2,919 ft/s
Total Impulse Requirements	12,626 - 13,599 ft/s

Table 5.1, Impulse Requirements for Coplanar Free Return Trajectories

plane (Ref. 1). The orbital inclination varies between a maximum value of 28.583 degrees and a minimum value of 18.317 degrees with a period of 18.6 years; for short periods of time this inclination may be assumed constant (Ref. 1). In the early part of the twenty-first century (2006 AD) the orbital inclination will be at its maximum value of 28.583 degrees; this value will be used for calculations in this part (Ref. 1).

Low Earth Orbit

The orbit of the LEO space station will be at an altitude of 270 nautical miles above the Earth's surface, and will be inclined at an angle of 28.583 degrees to the Earth's equatorial plane in order to insure minimum energy requirements for the lunar transfer missions. This orbit will have a period of four hours twenty-seven minutes, and the velocity of the station will be 24,988 feet per second.

The Earth is not perfectly spherical in shape; it has a "bulge" of mass in the lower hemisphere near the equator. This non-spherical mass distribution causes a phenomenon known as nodal regression of an orbiting body. When a body is orbiting the Earth at some angle of inclination, it is subjected to a torque by the gravitational pull of the distributed mass of the Earth. This torque causes the body's orbit to precess about the Earth's axis of rotation in a westerly direction (for direct orbits), much like a gyroscope under the influence of a similar torque (Ref. 1).

The nodal regression of the LEO at an altitude of 270 nautical miles will be about -6.7 degrees per day (a positive sign would indicate an easterly rotation of the orbit). As a result there will only be six opportunities for a coplanar lunar transfer during a year.

Non-Coplanar Transfers

More than six lunar missions must be accomplished during the year; therefore an out-of-plane transfer could be achieved as follows: a plane change from the orbital plane of LEO to the plane of the moon's orbit by applying an impulse as the OTV separates from the LEO station at the line of nodes (the line of intersection of the low Earth orbit and the moon's orbital plane); this impulse would rotate the OTV into a 270 nautical miles altitude orbit in the moon's orbital plane (Ref. 2); when the OTV reaches the proper point in its orbit, the proper impulse for injection into the free return trajectory would be applied just as in a normal coplanar transfer.

The 270 nautical mile low Earth orbit will experience a nodal regression of 6.7 degrees per day in a westerly direction about the Earth's polar axis. As a result, the angle of inclination (deli) between LEO and the moon's orbital plane changes at an average rate of 2.128 degrees per day. The velocity impulse that is required to accomplish this pure plane rotation is delv. This velocity change must be applied vectorally at an angle of 90 + delv/2 degrees to the direction of motion of the OTV at the node point when the impulse is applied. Impulse requirements for the plane rotation required at various times in the 54-day precession of the 270 nautical mile orbit are summarized in Table 5.2.

delt(days)	*deli(degrees)*	*delv(ft/sec)*	*delvtot(ft/sec)*
±1	2.128	927.96	14272.96
±3	6.384	2782.6	16127.60
±5	10.639	4633.4	17978.40
±10	21.279	9226.9	22571.89
±15	31.918	13740.9	27085.90
±20	42.557	18136.54	31481.54
±26	55.325	23201.84	36546.84

Table 5.2. Impulse Requirements for Out-of-Plane Transfers

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LIFE SUPPORT SYSTEMS

OVERVIEW

The overall life support system involves many components which must work together to provide the environment necessary for the survival of the crew. Establishing the parameters to make this environment as productive and cost effective as possible is a difficult and important job. In many cases the criterion can not be established on a direct scientific basis but must be studied and tested to determine average values for guide lines. Just as no two individuals are exactly alike, the life support system cannot be exactly established but requires detailed studies to evaluate the optimum levels for the life support system parameters.

Every man-made transportation system, whether in the air, on land, or sea, goes through three basic development phases. These three phases are also seen in the development of space vehicles (Ref. 6). The first phase basically deals with demonstrating if a system is feasible or practical. If a need exists a solution is then designed, built and tested in which the emphasis is on functional capacity. The second phase is improving on the first phase as far as reliability and safety goes. Once the second phase is satisfactory, achieving certain standards of comfort and habitability important to maintain optimum performance with minimum fatigue becomes the next objective, hence the third phase begins. The work habitability has been defined as "the qualities of an environment for man" (Ref. 6). This wide range of acceptability for man is what makes habitability lack a really ultimate standard. Thus the third phase of development for optimum performance can be a very complex process when the environment of the transportation vehicle is completely dependent upon a life support system.

In this study of life support system parameters, the purpose is to establish the criterion necessary to maintain a four man crew in not just a survivable environment, but to maintain optimum habitability in the Orbital Transfer Vehicle (OTV) and the Lunar Lander (LL). The OTV mission which calls for a ten day duration is established on the basis of approximately three and a half days to travel from Lower Earth Orbit (LEO) to Lunar Orbit (LO) where the OTV and crew will be supported by the Lunar Orbiting Station (LOS). The mission duration includes the safety supply for a return trip back and three days extra in the event that re-supply is not possible at the LOS. However, a larger safety factor of 14 days supply is used for the atmosphere which is the most important from a survivability standpoint. The short LL mission has a safety factor of two days established for its mission.

ENVIRONMENT

The environment to be maintained in the OTV and LL is probably the most important component of the life support system. It will support the immediate physical needs of the crew, and will be defined by the following parameters: volume, air, composition, pressure, carbon dioxide (CO_2) removal, temperature, humidity, and circulation.

Volume

The volume of free airspace available is of course an important consideration for the floor layout of the OTV. The volume is also important for the morale and productivity of the crew. The cost to employ and maintain an OTV is directly related to its overall volume and weight. Thus minimizing volume and weight puts tight restrictions on the airspace to be provided. The end goal is to maintain an environment for optimal habitability that is not merely survivable or tolerable. An optimum volume is dependent on several factors such as duration, activities to be performed, number of people and even the personalities and tolerance levels of the people. For the OTV under consideration, a volume of approximately 180 cubic feet per man should be provided. This compares well with the ten day duration on the Celentano Optimal Curve in Fig. 6.1 (Ref. 6).

The airspace required for the LL will be much less than that for the OTV. Since transfer from LO to the moon base station will only take about an hour, and since the space suits will be stored in the LL for emergency, the volume needed will be about 65 cubic feet per man.

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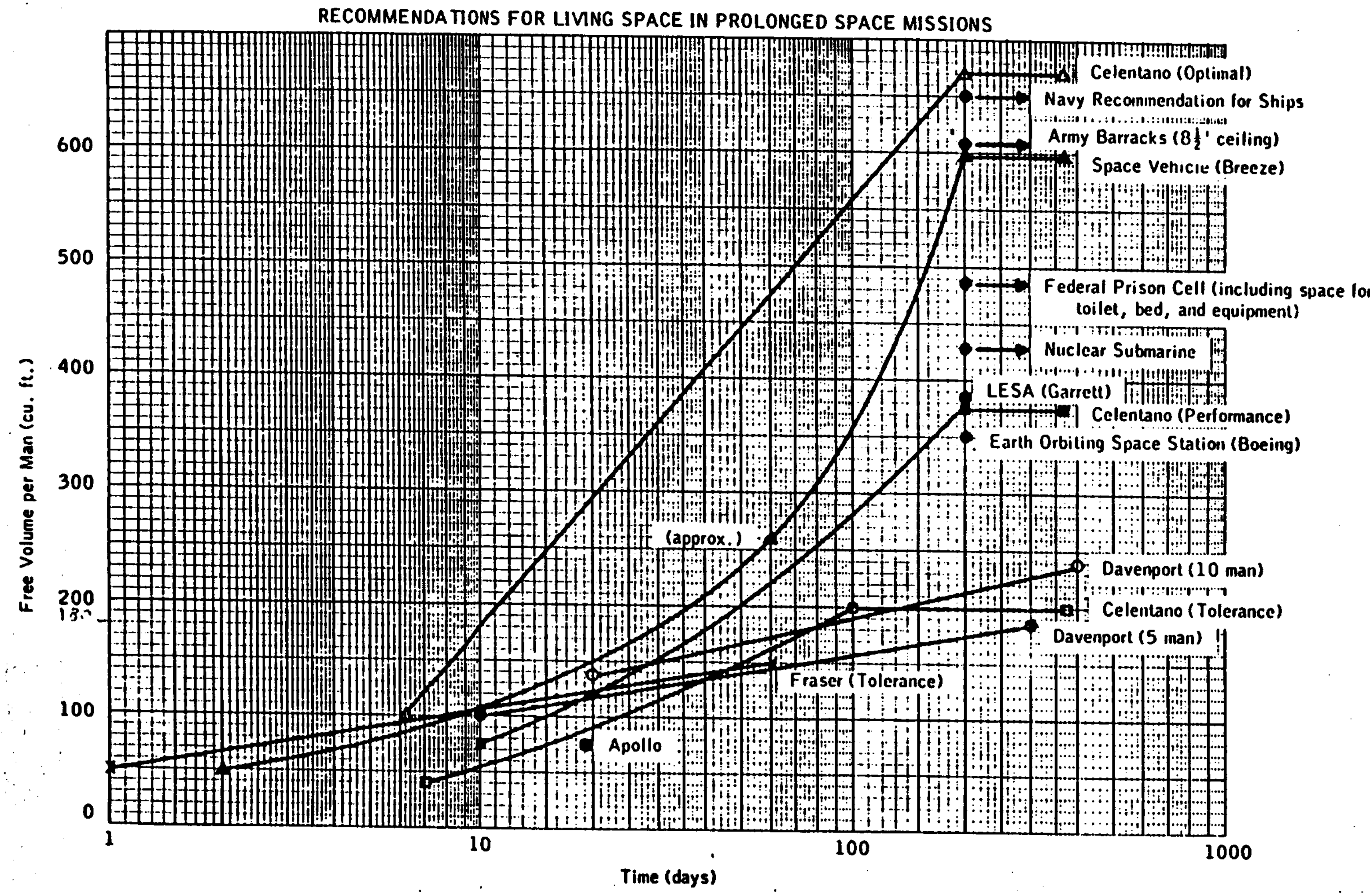

Figure 6.1, Celentano Optimal Curve

Air Composition

The next environmental parameter to be set is the composition and quantity of the elements to be included in the atmosphere. Of course, oxygen is the most important element for support of human life and will need to be continually replenished. It will make up twenty percent of the OTV gas concentration with the remaining eighty percent being nitrogen. A man requires approximately 1.85 lbm of oxygen a day, as seen in Table 6.1

The air composition will be maintained at the sea level value of 14.7 psi. "A minimal atmospheric pressure (about 0.9 psi) is required to keep body fluids in the fluid state, and man's tolerance to extremely high pressures is limited." (Ref. 14) By maintaining a sea level pressure, the comfort of the crew can be maximized by familiarity. Sea level pressure will also eliminate the possibility of evaporation or decomposition of chemical and material products at lower than normal pressure. The 14.7 psi pressure of the eighty percent nitrogen and twenty percent oxygen mixture will also minimize the fire hazard produced by high concentrations of oxygen at lower pressures. This gas concentration and pressure could also make docking and transfer between spacecraft safer and easier if it is maintained between the mating ships.

There will be an airlock between the OTV, Earth Orbiting Station, and LOS. The LL will also need an airlock between the LOS and Lunar Base Station. This will allow the pressure to be kept at 14.7 psi for all the systems. Due to the airlocks, suits will hopefully not be needed, but will be for emergency use only. Current suits utilize a

Item	Person/Day (lbm)	OTV Person/Trip Total (lbm)	LL Person/Trip Total (lbm)
Oxygen	1.85	25.90	3.70
LiOH	2.86	40.04	5.72
Water (potable)	7.70	77.00	-
Water (hygiene)	11.97	119.70	-
Urine	3.30	33.00	-
Feces	.12	1.20	-
Clothing	.88	8.80	-
Spacesuit	-	-	110.0
Personals (OTV)	.88	8.80	-
Personals (LL)	-	-	2.20
Food	3.30	33.00	-

Table 6.1. Weight Requirements Per Person

pressure of 4.3 psi, but a sufficiently mobile suit capable of 8.0 psi is expected to be developed in time for this mission.

CO_2 Removal

Since CO_2 will be the main contaminate being produced by the personnel on board the OTV, and effective method must be employed for its removal. For the short mission duration planned, this can be handled most economically by the catalyst Lithium Hydroxide (LiOH). To safely maintain the CO_2 removal process, there will need to be 2.86 lbm of LiOH per man per day. To make the overlapped safety factor useful, the 14 day supply of air will result in a total weight of 160.16 lbm of LiOH for the four man crew as seen in Table 6.2.

Item	OTV (lbm)	LL (lbm)
Environment	263.76	37.68
Water	786.80	-
Food	132.00	-
Waste	136.80	-
Personal hygiene	70.40	448.80
Crew	720.00	720.00
Equipment	1100.00	220.00
Total	3209.76 lbm	1426.48 lbm

Table 6.2. Total Weight Requirements For a Mission With Crew of Four

WATER

Water is, of course, very necessary for sustaining human life and will be needed not only for drinking, but for washing, and rehydrating foods. The minimum amount of needed water will be available from direct storage tanks with extra water being collected as the by-product of fuel cells. Recycling of hygiene waste water will not be necessary for the mission of the OTV but this water can easily be stored for recycling at the orbiting stations. In order to conserve both weight and space, the OTV will not have a shower facility and most cleaning needs can be handled with disinfectant treated wipes. They will not wash any clothes on board either.

The amount of water needed per man per day is approximately 19.67 lbm as is broken down in Table 6.1. This will result in a total water weight per mission of 786.80 lbm. Since the LL mission will take about an hour, water will not be provided.

WASTE

Most processes in the universe result in some sort of waste by-product being produced, and humans are no exception. "Man generates wastes which, if not properly treated and eliminated from his environment, will eventually threaten his survival." (Ref. 16) When dealing with waste management of the OTV, one must consider all possible waste sources such as waste from left-over food and containers, crumbs, accidental spills, hand wipes, nail clippings, hair stubble, urine, fecal matter, waste wipes, and vomitus. The waste management system must be able to collect, treat if necessary, store, reclaim, or dispose of waste (Ref. 16). In order to collect crumbs, accidental spills, nail clippings, hair stubble, and vomitus, a circulating vacuum hose with replaceable disposable filters will be used. Food containers, leftover food, hand wipes and the above mentioned filters can be taken care of with a trash compactor. No storage space will need to be provided for the trash in the compactor since there is enough room within the compactor to store the waste for the entire mission.

The same waste collection and storage system used by the current space shuttle has been considered, but it has caused problems by the cloud of debris that is produced when the waste is evaporated by exposure to space. A suggested alternative to this system is disposable plastic bags for fecal matter and a urine bag that contains an absorbent tissue material to ensure collection. These bags could then be frozen and stored for disposal, recycling, or testing at a later time.

Sources have shown that the quantity of urine excreted by a man per day is approximately 3.3 lbm, fecal matter is 0.12 lbm and that of vomitus, if any sickness occurs, is 2.11 lbm (Ref. 16). This gives us a total waste weight to provide storeage for in the OTV of 136.80 lbm. The LL will have emergency waste collection bags but no private area for their use.

PERSONAL HYGIENE

The area of personal hygiene within the OTV covers several specific subjects: clothing, hair brushes, combs, toothbrushes, toothpaste, razors, nail clippers, games, books, magazines, etc. The above can be divided into subgroups: clothing and personal hygiene equipment. Clothing will be relatively light since the crew will be provided with optimal environmental conditions. They are allowed 0.88 lbm per person per day. They will be allowed the same amount for their personal belongings (Ref. 10). The total personal hygiene weight is 70.4 lbm and can be seen in Table 6.2.

They will be allowed to bring some of their personal items onto the LL to help keep them company—approximately 2.2 lbm. Their space suits will be stored on the LL and will weight 110.0 lbm each. The total personal hygiene weight is 448.80 lbm and can also be seen in Table 6.2.

Temperature

Temperature is another important parameter for habitability. "Discomfort in the heat or the cold has a deleterious effect on performance. Thermal comfort is, then, a critical factor if optimum performance is to be maintained." (Ref. 14) The temperature aboard the OTV will have a thermostatic control to range between 60 and 90 degrees Fahrenheit. This will allow for quick temperature control for different activities. Such changes might be lowering the temperature for sleeping or exercising or raising the temperature if a crew member is sick. Personal comfort can also be maintained through clothing adjustments.

The temperature aboard the LL will also be regulated around the average value of 71 degrees Fahrenheit.

Humidity

If a comfortable heat balance is maintained, the humidity will not have a strong effect on crew comfort. However, maintaining a relatively low humidity is helpful for effective thermoregulation in the case of overheating or during exercise. High humidity is also more prone to microbial and fungal growth. The optimum for habitability is 10 torr (0.19 psi) (Ref. 14). The OTV water vapor pressure will be maintained between 7 torr and 14 torr (0.12-0.27 psi).

The humidity for the LL will be the same as that of the OTV.

Circulation

To make all these environmental parameters work together, a powerful circulation system must be used. This is important for the cleaning and regulating of the OTV atmosphere. Filters and disinfectants will clean the air while a radiator system will use solar radiation to heat the air. The circulation system will have adjustable as well as fixed vents to allow for adjustment of flow and to help eliminate stagnate pools.

FOOD

When discussing the third parameter, food, we must consider nutrients, calories, and the types of food and their preparation as well as their total weight. The nutrients needed and their percentages that must be provided for are about 17% protein, 32% fat, and 51% carbohydrates (Ref. 15). These nutrients are found in each crew's three meal a day diet. These three meals will provide each with 2800 calories per day based on an average weight of a crew member as being 180 lbm. menus for the meals can include more than 70 food items and 20 beverages (Ref. 11). These nutrients can be provided for in 6 possible categories of food that can be eaten in space. They are listed and described below:

"Intermediate—moisture—pre-cooked, thermally stabilized, or fresh food with the moisture content reduced so that the final moisture content is approximately 10 to 20% (such as dry-roasted peanuts, crackers, and cookies).

Dehydrated—ready-to-eat rehydratable foods with a moisture content reduced to less than 3% (such as cereal, scrambled eggs, green beans, shrimp cocktail, chicken and gravy).

Thermostabilized—pre-cooked, thermally processed food with the temperature reduced below 4.4 degrees C (40 degrees F) prior to launch to increase shelf life (such as catsup, jam, stewed tomatoes, puddings, and bread).

Frozen—pre-cooked food with the temperature reduced below -40 degrees C (-40 degrees F) before launch to retard spoilage, then maintained at -23 degrees C (-10 degrees F) in freezers in the orbital workshop (such as filet mignon, lobster newburg, and the astronauts favorite, vanilla ice cream).

Natural form—fresh breads, cookies, and rolls (used on short missions).

Beverages—rehydratable drinks (such as lemonade, orange drink, coffee, tea, and cocoa)." (Ref. 10)

The need for energy largely determines the weight and volume of the food supply. Sources have stated a range from 3.19 lbm to 4.40 lbm per person per day (Ref. 10). However, each person will be allowed 3.36 lbm per day. This would make the total weight of the food required for the OTV trip equal to 132.0 lbm as seen in Table 6.2. Food could be heated up to 185 degrees F and kept warm at 150 degrees F using heating trays (Ref. 12).

Since the LL mission will be such a short one, no provisions are being made for food.

SUMMARY

In the study, the life support system included the following parameters: environment, water, food, waste, and personal hygiene management (a combination of clothing and personal equipment). For optimum habitability, a detailed study must be made for all the life support system parameters for the specified mission. The OTV mission's life support is based on 10 day period which allows a three day safety factor. An additional four day factor is used for the atmosphere supply in case of extreme emergency. The LL has a life support system based on a safety factor of a possible two day mission.

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