FEASIBILITY OF PRODUCING

LUNAR LIQUID OXYGEN

by

Brian S. Teeple

Submitted to the Department of Aeronautics
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Abstract

There is a projected demand for liquid oxygen for future space missions. This paper analyzes the feasibility of producing and delivering liquid oxygen from the lunar surface, referred to as lunar liquid oxygen (LLOX). An analysis of the transportation system shows that LLOX is theoretically feasible for delivery to LEO, low lunar orbit, and for use on the lunar surface.

The transportation system is composed of two vehicles: the orbital transportation vehicle (OTV) and the lunar lander (LL). These vehicles are used to deliver the initial mass of the facility, deliver LLOX from the lunar surface, and resupply the LLOX plant. The vehicles are chosen to be roughly the size of the shuttle orbiter so they can be delivered using a Space Transportation System (STS) derived launch system. Analysis of the transportation system gives the break-even production costs.

The production cost is determined as a function of production rate. This analysis indicates that LLOX is not feasible for use in LEO, but is extremely attractive for demands on the lunar surface. A demand greater than 35 metric tons (MT) on the lunar surface would make LLOX a cost effective alternative to delivery from Earth. Mars missions could be met feasibly, but the savings are marginal for the projected demand.

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I'd also like to thank Jeanette Kao for putting up with me while I struggled through this paper. She also helped proofread the thesis and goad me on to completion. I'd also like to thank all my past roommates, particularly Paul Dans and Rob Mitra, for helping me relax and enjoy myself when possible.

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Symbol	Description	Units
R	Propellant to Fuel Ratio	None
M_{O_2}	Mass of Oxygen	MT
M_{H_2}	Mass of Hydrogen	MT
Isp	Specific Impulse	s
c	Characteristic Velocity	m/s
g	Acceleration due to Gravity	m/s^2
d	Net lunar oxygen delivered to LEO	None
MPR	Mass Payback Ratio	None
PR	Production Ratio	None
$C_{Production}$	Cost of Production	\$/MT
C_{Earth}	Cost of Delivery from Earth	\$/MT
M_S	Structural Mass	MT
M_{pl}	Payload Mass	MT
M_o	Gross Mass	MT
M_p	Propellant Mass	MT
C_D	Transportation R/D Costs	\$
C_{LL}	Cost of Lunar Landers	\$
C_{OTV}	Cost of OTV	\$
CC	Capital Costs	\$
OC	Operations Costs	\$
r	Discount Rate	None
n	Amortization Period	yrs
π	Production rate	MT/yr
C	Capacity	MT/yr
λ	Duty cycle	None
P	Power	MW
E	Energy	MJ
Y	Time in a year	s
P_p	Power for processing plant	MW
c_p	Cost of power	\$/MW
n_t	Types of lunar base modules	None
c_n	Cost of modifying space station modules	\$
n_m	Lunar base modules	None
c_u	Cost of each module	\$
c_f	Processing/storage facility cost	\$
c_t	Earth-to-Moon transportation cost	\$/MT
$N_{flights}$	Number of flights in lifetime of vehicle	None
m_p	Specific mass of power system	MT/MW

m_m	Mass of lunar base modules	MT
m_f	Mass of processing/storage facility	MT
V	Volume	m^3
S	Surface Area	m^2
n_r	Number of resupply missions	yr^{-1}
n_b	Ground support manpower	None
n_f	Ground support overhead factor	None
C_{LEO}	Earth-to-LEO transportation cost	$/MT
M_R	Resupply Mass	MT/yr
N_{OTV}	Number of OTV Replaced Yearly	yr^{-1}
N_{LL}	Number of LL Replaced Yearly	None
η_s	Purity of soil relative to highlands	None
m_{ps}	Specific mass of solar cells	MT/MW
m_{pb}	Specific mass of storage devices	MT/MW
c_s	Unit cost of solar cells	$/MW
τ	Curve fitting constant	MT/yr
β	Efficiency of using LLOX for Mar missions	None
α	Ratio of delivery costs	None

Chapter 1

Introduction

1.1 Motivation

As man turns his attention to "the final frontier," he wonders where to begin. The logical stepping stone to the stars is the moon. Here, he will learn how to survive in an alien environment. The moon will serve as a testing ground for further colonization of the solar system.

There are many parallels to the colonization of the American frontier. Europeans sailed across the ocean to settle on the Eastern seaboard. The moon will be our foothold in the solar system. The pilgrims brought supplies with them, but had to learn how to live off the land to survive. We will also have to learn how to use the resources at hand.

Oxygen will be the most important resource for two reasons. It is an absolute necessity for manned space missions. Oxygen is necessary to support respiration and can be used together with hydrogen to synthesize water. Liquid oxygen (LOX) is also important as a propellant. If LOX can be produced from the lunar soil and transferred to the delivery site at a savings compared to transporting it from the earth, it would empower us with a greater flexibility in further space missions.

1.2 Previous Studies

An analysis of the costs associated with lunar liquid oxygen (LLOX) production was performed by Michael C. Simon entitled *A Parametric Analysis of Lunar Oxygen Production*. In it, a linear cost model is derived:

$$\text{Capital Cost}=(P^*c_p)+(n_t^*c_n)+(n_m^*c_u)+c_f+c_t^*[(P^*m_p)+(n_m^*m_m)+m_f]$$

$$\text{Operations Cost}=c_t^*[(n_r^*m_m)+(1-d)^*(125000)]+(n_b^*n_f^*\$100{,}000)$$

"The operations costs are the annual costs of manufacturing 1000 MT of LO_2 per year and delivering to LEO as much of this LO_2 as possible." [6, pg. 532] Baseline values were assigned with the ground rule that lowest risk technologies be used. A sensitivity analysis was done to decide which technologies will have the greatest impact on LLOX production.

This study has several deficiencies that need to be addressed if the feasibility of producing LLOX is to be determined. Simon's model does not fully address the transportation system necessary to transfer the LLOX. The model has also been restricted to only one production rate (1000 MT/yr). Unit costs are likely to be dependent on the production rate. Simon's model is also restricted to delivery to low earth orbit (LEO). It does not address the feasibility of using LLOX on the lunar surface or in lunar orbit.

1.3 Proposed Study

This paper will start with an analysis of the transportation system. Dimensionless parameters are chosen to evaluate the efficiency of the system. The goal is to decouple the optimization of the transportation system and the production system. Once the transportation system is optimized, unit costs of production are determined for the various delivery sites. Finally, the cost of production is determined as a function of the production rate and feasible costs of production are translated into feasible production rates.

Using the model proposed by Simon as a base, the production costs are determined as a function of LLOX production rates. The dependence of each variable on the production rate is determined and incorporated into a modified model to determine the economies of scale. The modified model also includes the costs of the transportation system costs. Using this model, it becomes possible to compare different chemical processes and power sources. The model was also used to investigate how differences in ore type and sunlight availability affect feasibility.

This study also explores how the location of the demand affects the feasibility of using LLOX. Possible future projects such as the space station, commercial needs, Mars Missions, and lunar bases are investigated to estimate amounts and locations of future demands. Several test cases will be analyzed to see how close projected demands come to being met feasibly by a LLOX plant. Possible benefits from by-products will be discussed, but are not considered when deciding feasibility.

1.4 Design Philosophy

Since this is a feasibility study, cost was the main driver in all decisions; however, proven technologies were chosen over possibly cheaper alternatives that are still in the design stage. These technologies under development may lower costs, but are only briefly mentioned in this paper. The framework will be able to analyze the impact of these developmental technologies once their merits are proven, but the goal is to establish feasibility for existing technologies.

In an age of project overruns and overestimated performance, estimations were made to err on the conservative side. Where possible, actual designs were chosen over scaled parameters if they were near optimal points. Since any analysis will tend to have many estimates, at best, the break-even point can be determined as a bandwidth of production rate. The goal of this paper is to determine the upper end of this bandwidth, that is, the point above which production is feasible regardless of estimation error. It may actually be feasible below this point. Simon addresses the variational effects for a single production rate in his paper. This paper does not address the effect of estimation on the break-even point, but provides the framework by which this can be done.

Chapter 2

Transportation System

2.1 Dimensionless Analysis

The first goal in optimization is to choose a figure of merit to be optimized. For the transportation system, there are several characteristics to consider: the amount of LLOX consumed by the transportation system, the amount of fuel that has to be resupplied from earth, and the capital and operational costs. The following sections describe how these characteristics can be described using dimensionless parameters, which can be combined to form an overall figure of merit. Dimensionless parameters allow comparison between transportation systems of various sizes and strategies. The same overall figure of merit is related to the production costs necessary for feasibility.

2.1.1 Propellant to Fuel Ratio (R)

The propellant to fuel ratio is a measure of the relative amounts of fuel and oxidizer. A high ratio indicates a greater demand on the LLOX while a low ratio indicates a greater demand on fuel (H_2) which has to be supplied from the earth.

$$R = \frac{M_{O_2} + M_{H_2}}{M_{H_2}} = 1 + \frac{M_{O_2}}{M_{H_2}}$$

The propellant to fuel ratio also affects the efficiency of the propulsion system. Propulsive efficiency is commonly expressed as specific impulse (Isp). Propulsive efficiency can also be expressed by the characteristic velocity (c).

$$c = gIsp \qquad (g = 9.81\ \mathrm{m/s^2})$$

Figure 2-1 shows the relationship between the propellant to fuel ratio and Isp for a specific engine.

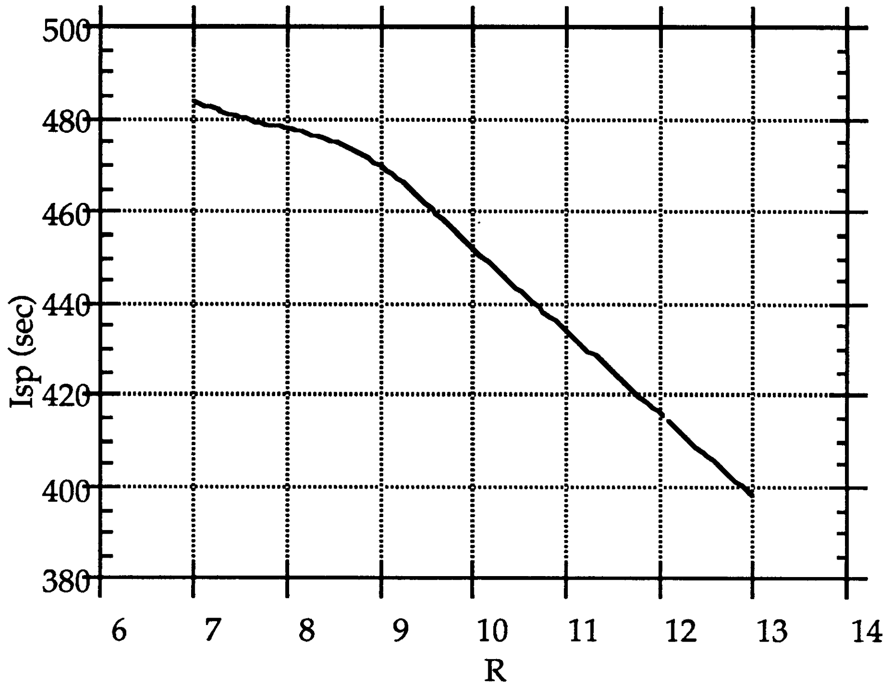

Figure 2-1. Isp vs. R [7, pg. 63]

The maximum Isp (484 s) occurs at a fuel rich mixture of R=7. The stoichiometric ratio is R=9. Below the stoichiometric ratio, the Isp gradually increases to the maximum at R=7. Above the stoichiometric ratio, the Isp drops off a greater slope.

2.1.2 Delivery Ratio (d)

Another dimensionless parameter of interest is the ratio of the delivered mass to the total mass produced. This is known as the delivery ratio and is commonly denoted by the symbol d. It ranges between zero and one, with higher delivery ratios being preferable.

The delivery ratio is a measure of how much of the manufactured product is consumed by the transportation system. For non-propellants, the delivery ratio would be one. For a LLOX plant, part of the manufactured product will be used as propellant by the transportation system. Therefore, the delivery ratio will be less than one. This also illustrates that the delivery ratio is not a good overall figure of merit. If it were, it would indicate that LLOX should not be used as propellant in the transportation system. The transportation system would require all of the propellant to be supplied from earth, a more expensive option.

A high delivery ratio is desirable; however, there is a fundamental limit to the delivery ratio. The limit depends on the energy (Δv) needed to transport the LLOX from the lunar surface to the delivery site and the propellant to fuel ratio (R). From the rocket equation, it is easy to show that:

$$d < \frac{1}{1 + \left(e^{\Delta v / c} - 1\right)\left(1 - \frac{1}{R}\right)}$$

It can also be shown that if delivery is broken down into several legs that the effective delivery ratio is the product of the delivery ratios of each leg.

$$d = \prod_i d_i$$

This paper will consider three delivery sites: LEO, LLO, and the lunar surface. The Δv requirements for each are given in Table 2-1.

Delivery Site	Δv (m/s)
Lunar Surface	0
LLO	2000
LEO	3650

Table 2-1. Δv Requirements for Various Delivery Sites

Using the data of Figure 2-1 and Table 2-1, the limits on delivery ratio are shown in Figure 2-2 as a function of the propellant to fuel ratio.

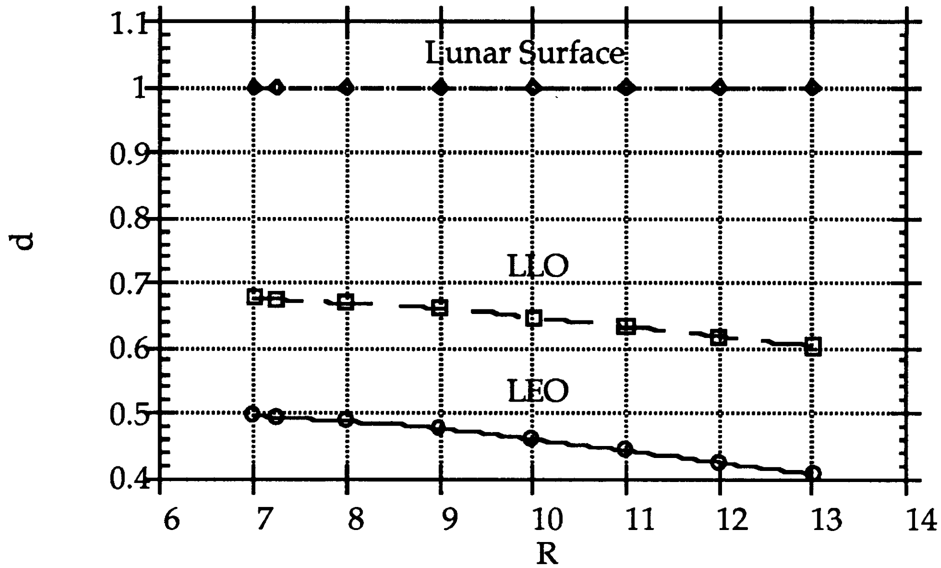

Figure 2-2. Bounds on Delivery Ratio

The maximum delivery ratios occur at the lowest propellant to fuel ratios, but the slope is fairly flat in this region.

2.1.3 Mass Payback Ratio (MPR)

Another dimensionless parameter important for the evaluation of the transportation system is the ratio of the mass that has to be brought up from the earth's surface to the delivered mass. This is known as the mass payback ratio and is denoted as MPR. It ranges between zero and one for feasible systems. (Values of greater than one are possible, but indicate that the process is unfeasible regardless of the cost of manufacturing.) Only the mass consumed by the transportation system (H_2) is considered in the optimization. Other masses are accounted for in the production cost model.

A low MPR is desirable. As with the delivery ratio, a theoretical bound for MPR can be derived from the rocket equation:

$$\mathrm{MPR} > \frac{1}{R}\left(e^{\Delta v/c} - 1\right)$$

If the delivery is broken down into L legs, the effective MPR can be expressed as a series formula.

$$MPR = \frac{\sum_{L}\left[MPR_i \prod_{i} d_i\right]}{\prod_{L} d_i}$$

Figure 2-3 shows the limits on MPR for the various delivery sites as a function of the propellant to fuel ratio.

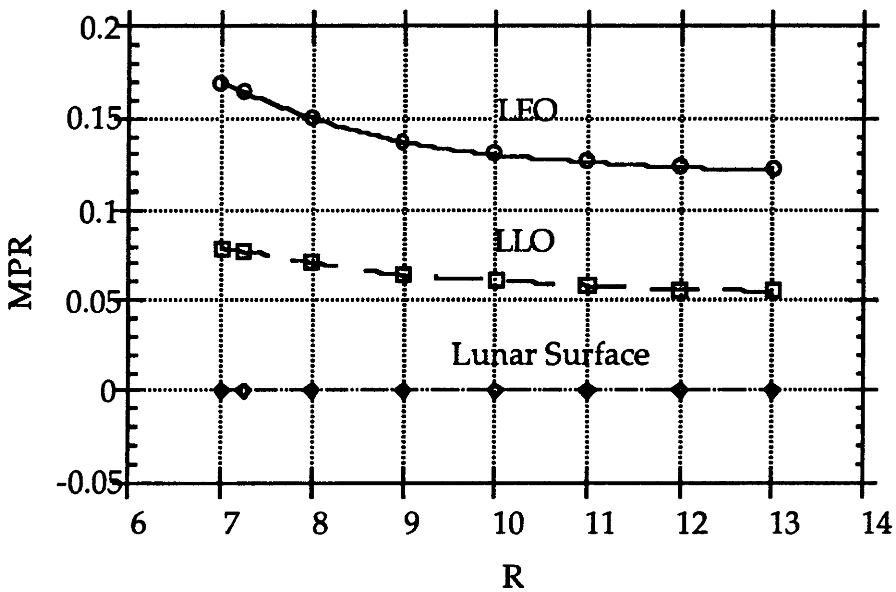

Figure 2-3. Bounds on Mass Payback Ratio

The lowest MPR occurs at the maximum propellant to fuel ratio. This is in direct contrast with the optimum delivery ratio.

2.1.4 Production Ratio (PR)

Neither the delivery nor the mass payback ratio is the proper figure of merit for the transportation system. As R is decreased, d is optimized; as R is increased, MPR is optimized. In order to find the proper combination of these dimensionless parameters, their effects on feasibility most be investigated.

The dimensionless parameter for the manufacturing plant is the ratio of the unit cost of production to the unit cost of delivering from earth. It is designated as the production ratio (PR). (The cost of transporting fuel for the transportation system is not included in this definition of unit cost of production.) The PR ranges between zero and an upper limit for feasible systems. The upper limit cannot exceed one and will be determined by the transportation system.

The total costs include the unit cost of production and the cost of transporting fuel for the transportation system. This must be less than the cost of delivering an equal amount of LOX from the earth for LLOX to be economically feasible. The criterion for feasibility can be expressed as:

$$C_{Production} + C_{Earth} MPRd \leq C_{Earth} d$$

Rearranged, this gives us an upper limit to the production ratio if the process is to be feasible.

$$PR \leq d(1 - MPR)$$

Therefore, the combination of d and MPR that maximizes the expression d(1-MPR) is the optimal transportation system, because it allows for the highest production ratio. The cost of delivery from earth is independent of the transportation system used to deliver from the LLOX plant. Figure 2-4 shows the limits on the production ratio for the various sites as a function of the mixture ratios.

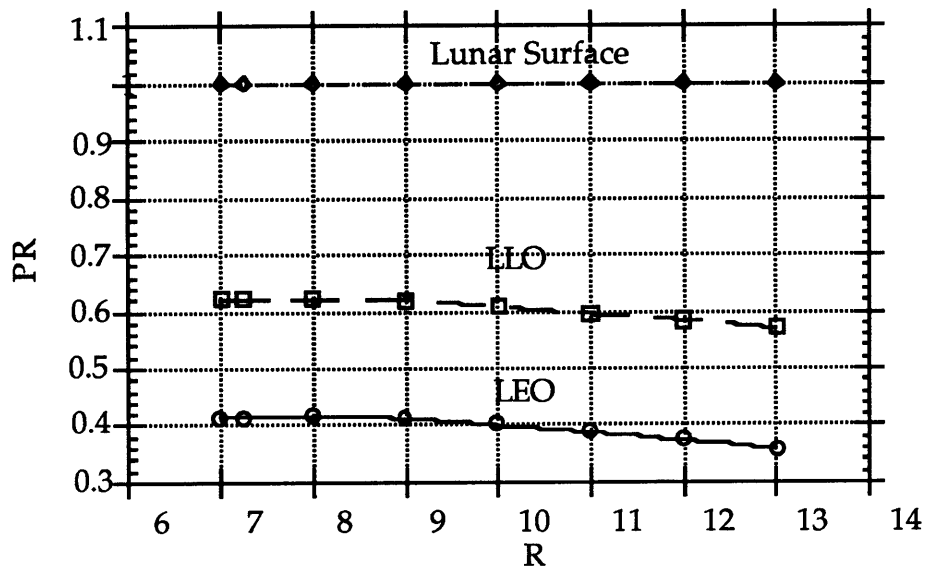

Figure 2-4. Bounds on Production Ratio

For example, assuming an ideal transportation system to LEO, a lunar plant must be able to produce LOX for less than 41% of the cost of transporting it from the earth.

At this point, it should be emphasized that the theoretical values are only limits. They can be used to decide if a process is not feasible. They are not sufficient to determine feasibility. For that, a complete design is necessary. It is also important to note that the theoretical limits are not dependent on the mass, or increments of mass, delivered, while the real values will.

2.2 General Overview

The decision to use two dedicated vehicles was made for the following reasons:

1. The LS-LEO transfer naturally breaks down into two legs.

2. Energy will not be used to transfer equipment dedicated to only one leg thereby increasing d and decreasing MPR.

The two vehicles are designated as the orbital transfer vehicle (OTV) and the lunar lander (LL). The OTV will transport payloads between LEO and LLO, while the LL will transport payloads between LLO and the lunar surface.

The decision was also made to go with as large of vehicles as possible because they are more efficient. If the structural to payload mass ratio is known, d and MPR can be calculated exactly:

$$d = \frac{1}{1 + \left(1 + \frac{M_S}{M_{PL}}\right)\left(e^{\Delta v/c} - 1\right)\left(1 - \frac{1}{R}\right)}$$

$$MPR = \frac{1}{R}\left(e^{\Delta v/c} - 1\right)\left(1 + \frac{M_S}{M_{PL}}\right)$$

By inspection, it can be seen that these values approach the theoretical limits as the structure to payload ratio approaches zero.

It is logical when scaling the inert mass of a vehicle that part of the inert mass will be proportional to the gross mass (engines, landing gear, etc.), another part will be proportional to the propellant mass (tanks, etc.), and some of the equipment mass will be the same regardless (computers, GNC, etc.).

$$M_S = AM_o + BM_p + C$$

The gross mass is the sum of the inert mass, the propellant mass, and the payload mass.

$$M_o = M_S + M_p + M_{pl}$$

Combining the above two relationships with the rocket equation, it can be shown that as the payload size approaches infinity, the structure to payload ratio approaches a minimum at:

$$\frac{M_S}{M_{PL}} \rightarrow \frac{(A+B)e^{\Delta v/c} - B}{1+B-(A+B)e^{\Delta v/c}}$$

This does more than confirm the intuitive result that larger vehicles are more efficient. It gives us a more realistic limit to compare with-one that takes structural limitations into account. Using the above limit for the structure to payload ratio, the limits for d, MPR, and subsequently PR, can be modified accordingly.

Finally, the decision was made to require reusable spacecraft. Although this increases capital costs and decreases the efficiency of the transportation system, the savings in operations costs more than compensate. Current goals are to design vehicles capable of 58 operational missions [7, pg. 125]. For purposes of this paper, a previous, and more conservative, estimate of 30 flights per vehicle is used [2, pg. 34].

2.3 Velocity Requirements

2.3.1 Orbital Transfer Vehicle (OTV)

The orbital transfer vehicles (OTV) will be used to transfer the LLOX from low lunar orbit (LLO) to LEO. The OTV will require multiple burns. The first burn will allow it to escape from earth orbit and head towards the moon. Another burn will be required to circularize into LLO. After the LLOX has been transferred to the OTV, a burn will be performed to leave LLO and head back towards the earth. A final burn will be performed to enter LEO.

Studies show that aerobraking improves the efficiency of the transportation system. The decrease in the velocity requirement more than compensates for the additional mass of the aerobrake. The most conservative estimation of the benefits of aerobraking was chosen. Allowances are also made for mid-course corrections. Table 2-2 shows the velocity requirements for each leg of the OTV journey.

Mission Leg	Δv (m/s)
Earth Escape	3150
Mid-Course Correction	50
Lunar Circular Orbit at 100km	850
Lunar Escape	850
Mid-Course Correction	50
Enter LEO with Aerobraking	700

Table 2-2. Δv Requirements for OTV

In addition to transporting the LLOX from LLO to LEO, the OTV will carry excess H_2 for the lunar lander from LEO to LLO. In sizing the OTV, allowances must be made for both payloads. The fact that the sizes of the payloads are interdependent further complicates the calculations.

2.3.2 *Lunar Lander (LL)*

The lunar lander is designed to transport payload between the lunar surface and low lunar orbit (LLO) at 100 km. Payload from the lunar surface will be the LLOX. Payload from LLO to the lunar surface will include the excess H_2 for the next mission. Table 2-3 shows the velocity requirements will be used throughout this paper.

Mission Leg	Δv (m/s)
Ascent to Lunar Circular Orbit at 100 km	2000
Descent to Lunar Surface	2100

Table 2-3. Δv Requirements for Lunar Lander [3, pg. 111]

The descent leg has a higher velocity requirement due to increased gravity losses.

2.3.3 Summary of Transportation System

The OTV will leave LEO and arrive at LLO with enough H_2 for the return trip and for refueling the lunar landers. The lunar landers will leave the lunar surface and arrive at LLO with only O_2 for the return trip and for refueling the OTV. Figure 2-5 illustrates this arrangement and the Δv requirements for each stage.

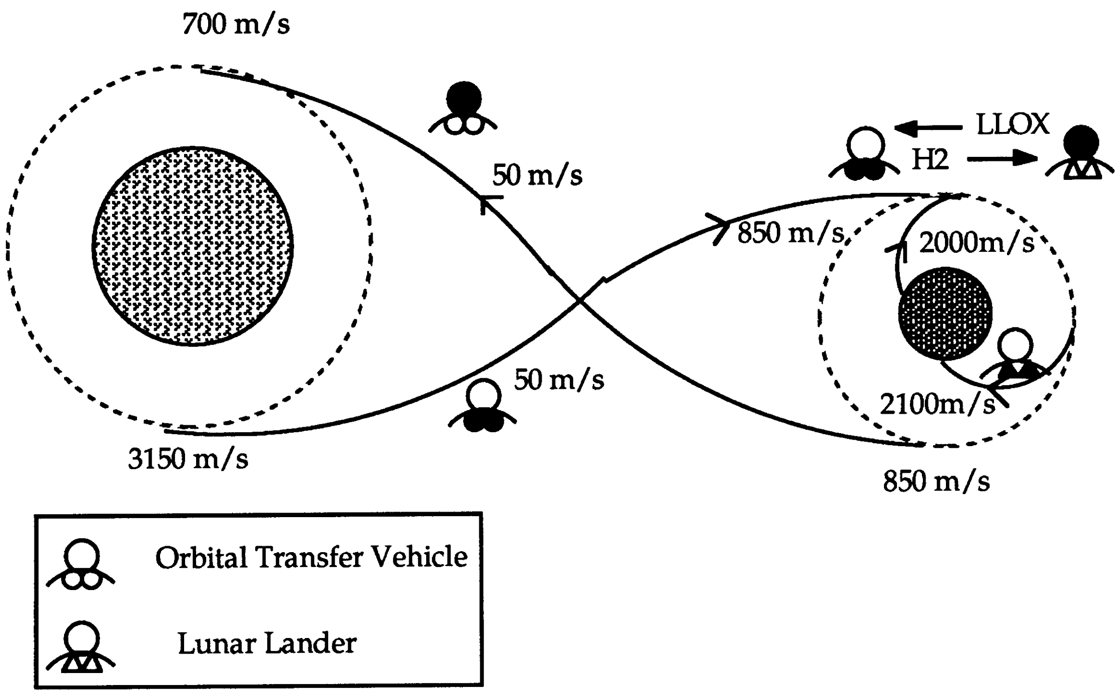

Figure 2-5. Overview of Transportation System

2.4 Orbital Transfer Vehicle (OTV)

The design listed here is the largest configuration possible by General Dynamics' S-4C modular tank concept. Little gain in production ratio is possible by increasing the size; therefore, this design was kept because it employs proven technologies and avoids scaling inaccuracies.

System	Mass (lbs)
Structure	5,078
Tanks	3,559
Propulsion system	3,128
Thermal control system	533
GNC	150
Electrical systems	555
Aerobrake (reusable)	2,298
Propellant	285,901
Residual propellant	3,401
Pressurant	63
Dry mass	15,301
Wet mass	301,202

Table 2-4. Mass Summary of OTV [7, pg. 6]

The OTV is designed with modular tanksets to allow for easy exchange of propellants. It is also only slightly larger than the space shuttle and can be placed in orbit using a Space Transportation System (STS) derived launch system. If this is not possible, it can be delivered piecemeal by the SPS launch system because of its modular design.

2.5 Lunar Lander (LL)

The reference OTV derived lunar lander was found to be too small. A significant increase in production ratio can be achieved by developing a larger, higher thrust vehicle. A scaling equation was derived from current designs and it determines the inert mass as a linear function of the gross mass at takeoff (M_o) and the propellant mass (M_p).

$$M_S = AM_o + BM_p + C$$

The following scaling law was applied to the OTV (without aerobrake) and the constants were calculated.

$$M_{SOTV} = .02724M_o + .01245M_p + .562$$

The approximation that an additional 5% of the maximum landing weight (this was taken as the gross weight in case of an aborted takeoff) is needed for landing gear, etc. was used [2, pg. 32]. This gave a scaling equation for the lunar lander.

$$M_{SLL}=.07724M_o+.01245M_p+.562$$

The gross mass can be broken down and expressed as:

$$M_o=M_S+M_p+M_{pl}$$

The inert mass can then be expressed as:

$$M_S=\frac{A}{1-A}M_{pl}+\frac{A+B}{1-A}M_p+\frac{C}{1-A}$$

The M_{pl} will be O_2 used to fill the OTV tanksets. The LL will also have to return with enough H_2 (supplied from the OTV) for the next mission.

An iterative program was used to match the LL design to the capacity of the OTV (See Appendix I). An integral number of LL was assumed to completely fill the OTV tanksets. The number of LL/OTV was chosen as two for several reasons:

1. The size of a single LL capable of refilling an OTV was prohibitively large (twice the size of an OTV).
2. The dimensionless parameters are practically independent of the number of vehicles.
3. The structural mass of this LL was only slightly larger than the OTV, so an OTV derived LL is possible.
4. The gross mass of the LL was slightly less then the gross mass of the OTV, so the same launch system could be used.
5. The LL/OTV ratio should be kept as low as possible to minimize recurring costs.

2.6 Selection of Mixture Ratios

An iterative program was used to match the payloads of two lunar landers to the propellant capability of the OTV. The figure of merit was the production ratio. Table 2-5 shows how the production ratio varies for different combinations of R_{OTV} and R_{LL}. The optimum point will vary with different size vehicles and scaling laws.

R_{ll} \ R_{otv}	7	7.25	8	9	10	11	12	13
7	.154	.154	.154	.152	.138	.122	.104	.083
7.25	.158	.158	.159	.156	.143	.127	.109	.089
8	.171	.171	.172	.169	.156	.141	.124	.105
9	.182	.182	.183	.180	.168	.154	.138	.120
10	.182	.182	.183	.181	.169	.156	.140	.123
11	.180	.180	.180	.178	.167	.154	.140	.123
12	.175	.175	.176	.174	.163	.151	.137	.121
13	.169	.169	.169	.168	.157	.146	.132	.117

Table 2-5. PR as a Function of R_{LL} and R_{OTV}

The maximum PR occurs with R_{LL}=10 and R_{OTV}=8. With this combination, the cost per kilogram of LLOX produced must be 18.3% of the cost per kilogram of LOX delivered from the earth.

2.7 Comparison with Theoretical Limits

The absolute theoretical limit to the production ratio is 41%. Essentially, this assumes a massless spacecraft. A better limit is obtained using the OTV scaling law. This limit is 35.1%, but assumes infinitely large vehicles. Both of these limits assume an expendable fleet of vehicles. Although this increases the possible PR, the recurring costs of the transportation system, which must be included in the PR calculation, increase drastically. The planned transportation system is "large" according to the scaling law, but pays a penalty for being reusable. A theoretical limit for reusable spacecraft can be calculated using the scaling laws. This is the theoretical limit that will be referred to in the future. Table 2-6 illustrates how the actual transportation system compares to these theoretical limits.

Figure of Merit	Limit	Actual
$(M_S/M_{PL})_{LL}$	0.147	0.189
$(M_S/M_{PL})_{OTV}$	0.089	0.112
D_{LL}	0.573	0.547
D_{OTV}	0.650	0.500
D	0.372	0.272
MPR_{LL}	0.083	0.092
MPR_{OTV}	0.077	0.144
MPR	0.205	0.329
PR	0.296	0.183

Table 2-6. Comparison with Theoretical Transportation Limits

From examination of Table 2-6, it can be seen that the size of the vehicles is acceptable. Some gain can be achieved by increasing the size of the vehicles, but it would cause launch and design problems that would be costly. All of the other differences can be associated with the requirement of reusable spacecraft.

2.8 Costs Associated with Transportation System

The following sections discuss the costs of the transportation system. These costs will be used to modify the capital and operations cost model. The discussion of how the transportation system costs are included in the modified model is reserved for Chapter 3.

All recurring costs are first unit prices. No learning curve is assumed for later units. The presence of a learning curve would lower the average price. It is suggested that a 95% learning curve be used for less than ten units [11, pg. 681]. This effect could lower average production costs by about 10%. In keeping with the conservative philosophy, this was considered as margin for unforeseen costs.

2.8.1 *Cost of Design (C_D)*

Design costs for a new lunar lander have been derived using a cost estimating relationship (CER) model. The total design and development costs are estimated to be $1539 million [7, pg. 131]. Since the assumption is that the lunar lander and OTV are derived vehicles, one would expect the same or lower design costs for the OTV. The OTV is significantly simpler in design than the lunar lander because it does not require landing gear, but the complexity of LLOX transfer has not been addressed. It is assumed that this cost will make up the difference and make the cost of design for the OTV the same as the lunar lander. The total cost of design for the two vehicles is then $3078 million. This value will be used in the capital costs calculation.

2.8.2 *Cost of Lunar Lander (C_{LL})*

The recurring production cost of the lunar lander has also been estimated using a CER model. Each vehicle is calculated to cost $759 M [7, pg. 131]. This does not include the launch costs.

2.8.3 *Cost of OTV (C_{OTV})*

Since the OTV and LL are derived vehicles, one would expect similar costs. If anything, one would expect lower costs for the OTV because it has a lower structural mass. Therefore, the cost of an OTV was also assumed to be $759 M, not including launch costs, in keeping with the philosophy of conservative estimates.

2.8.4 *Launch Costs*

It is assumed that an STS derived launch system can be used to deliver the OTV and LL to LEO. Currently, the shuttle launch system can deliver 230,000 lbs to LEO [7, pg. 21] at a cost of $190 M [11, pg. 671]. The OTV, the larger of the two vehicles, is heavier than the shuttle but may be able to use a derived launch system. If this is not feasible, the vehicles could be delivered piecemeal using the current STS launch system. The costs are scaled by the weight of the OTV. This gives a launch cost of $250 M per launch.

Chapter 3

Production Costs

3.1 Cost Model

3.1.1 Capital Costs (CC)

Capital costs include all the costs prior to operation of the LLOX plant. The number of years that this cost would be distributed over would depend more on funding considerations then technological ones. In addition to the capital costs of Simon's model, the modified model requires the addition of the R&D costs of the transportation system.

$$CC=(P*c_p)+(n_t*c_n)+(n_m*c_u)+c_f+c_t*[(P*m_p)+(n_m*m_m)+m_f]+C_D$$

3.1.2 Operations Costs (OC)

Operations costs are the recurring costs that are necessary to maintain the LLOX plant each year excluding the cost of transporting H_2 for use by the transportation system. This cost was accounted for in the dimensionless analysis when production cost was defined to exclude this expense. It was also decided to handle resupply masses in the same manner. They are not included in the operations cost model, but are accounted for in a modified calculation of MPR. The transportation system was modified accordingly. In addition to the operations costs of Simon's model, the recurring cost of the OTV and LL must also be accounted for.

$$OC=(n_b*n_f*\$100{,}000)+N_{OTV}C_{OTV}+N_{LL}C_{LL}$$

3.1.3 Production Ratio (PR)

The production ratio is defined as the ratio of the unit cost of production for LLOX to the unit cost of delivery from earth. The next two sections detail how these terms are calculated.

3.1.3.1 Unit Cost of Production ($C_{Production}$)

In order to determine the unit cost of production, the lifetime, or the number of years over which the capital costs are to be amortized, needs to be known. This variable (n) has the unit of years. The total costs are normalized be the total number of units produced. The production rate (π) is the number of units produced per year. The unit cost of production is (without discounting):

$$C_{Production}=(CC+nOC)/n\pi$$

If discounting is taken into account, the expression maintains the same form:

$$C_{Production}=(CC+xOC)/x\pi$$

$$x=\frac{\left[(1+r)^n-1\right]}{r(1+r)^n}$$

Here, r is the discount factor. It is important to note that discounting has exactly the same effect as decreasing the amortization period.

3.1.3.2 Unit Cost of Delivery from Earth (C_{Earth})

The unit cost of delivery from earth includes both the costs of production on earth and launch costs; however, the launch costs tend to dominate. The following values were used:

Delivery Site	C_{Earth}($/kg)
LEO	3,400
LLO	8,000
Lunar Surface	10,000

Table 3-1. Cost of Delivery from Earth [6, pg. 534]

The cost to LLO was extrapolated from the cost to LEO and the velocity requirements.

3.2 Definition of Basic Terms

For purposes of this report, production rate (π) is the amount of LLOX produced per year. Capacity (C) is the rate at which LLOX would be produced if the plant ran nonstop. Duty cycle (λ) is the fraction of the time that the plant is running. These three variables are related by the following equation:

$$C = \frac{\pi}{\lambda}$$

The goal is to parameterize the unit cost of production in terms of the production rate. The unit cost of production also depends on the capacity; but, as will be shown later, the duty cycle will be decided by the selection of a power source. This determines the capacity as a multiple of the production rate.

3.3 Power (P)

3.3.1 Power for Ground Support

The power necessary to maintain the support personnel has been estimated to be 8.5 kW/person [6, pg. 629]. This includes the energy necessary for climate control and other life support functions. The lunar environment temperatures vary between -240°F to 250°F [9, pg. 5], so temperature maintenance becomes an important consideration. This requirement may be lowered by strategic location of the plant, but the mentioned value is considered a conservative estimate.

3.3.2 Power for Processing Plant

The primary demand for power will come from the processing plant itself. The amount of energy per MT of LOX produced is determined by the chemical processes involved. The energy used by the plant in a year can be determined as a function of the production rate.

$$E = \varepsilon\pi$$

The necessary power is obtained by dividing by the time of operation in a year, which is the duty cycle multiplied by time in a year (Y). This determines the power as a multiple of the capacity.

$$P_p=aC$$

The units of the constant (a) are MW-yr/MT. When multiplied by the capacity, with units of MT/yr, it gives the required units of MW.

Therefore, the total power can be expressed as the sum of the power for the plant and the power for life support:

$$P=aC+.0085N_b$$

In Section 3.17, it is shown that the number of support personnel can be expressed as a function of the production rate. This allows us to calculate the necessary power from the production rate and the duty cycle. Later, it will be shown that the duty cycle is determined by the selection of a power source. The power can be determined solely as a function of the production rate.

3.3.2.1 Ilmenite Reduction

Ilmenite is the name given to the ore richest in oxygen ($FeTiO_3$). The reduction process involves reagents, substances that are necessary for the reaction, but are not consumed by the process. Two processes have been suggested: hydrogen and carbothermal reduction.

3.3.2.1.1 Hydrogen

The hydrogen reduction process uses hydrogen as a reagent. The result is water, which is then electrolyzed to recover the oxygen and the hydrogen is recycled for further use.

$$FeTiO_3+H_2\rightarrow Fe+TiO_2+H_2O$$

$$H_2O+e^-\rightarrow H_2+1/2\ O_2$$

Studies of the hydrogen reduction process show that $a=5.7\times10^{-3}$ MW-yr/MT [6, pg. 556].

3.3.2.1.2 Carbothermal

The carbothermal process is similar to hydrogen reduction, but uses carbon as a reagent. The overall chemical process is:

$$FeTiO_3 \rightarrow Fe + TiO_2 + 1/2 O_2$$

The steps in the process are (not including parial reduction of TiO_2):

$$FeTiO_3 + (1+x)C \rightarrow FeC_x + CO + TiO_2 \quad \text{(slag-metal bath reduction)}$$

$$FeC_x + x/2\, O_2 \rightarrow Fe + xCO \quad \text{(iron decarburization)}$$

$$yCO + (2y+1)H_2 \rightarrow yH_2O + C_yH_{2y+2} \quad \text{(reforming)}$$

$$C_yH_{2y+2} \rightarrow C_y + (y+1)H_2 \quad \text{(hydrocarbon cracking)}$$

$$H_2O \rightarrow H_2 + 1/2\, O_2 \quad \text{(electrolysis)}$$

Table 3-2 details the energy necessary for each step of producing one MT of LLOX using the carbothermal process.

Process Step	Energy (GJ)
Reduce 3.68 tons of iron (75% efficiency)	12.1
Heat to melt 3.68 tons of iron (500kWh/MT)	6.6
Heat content of 4.25 tons of slag 470 kWh/MT)	7.2
Heat content of off gas (1350 C^o effective heating)	8.0
Energy to electrolyze water (60% efficiency)	28.9
Energy to liquefy oxygen	5.4
Total energy consumption, carbothermal process	68.2

Table 3-2. Energy Requirements of Carbothermal Process [6, pg. 564]

This can be used to derive the power requirements for the plant.

$$E = 68.2 \times 10^3 \pi$$

$$Pp = \frac{E}{\lambda Y}$$

$$P_p = 2.1611 \times 10^{-3} C$$

Calculations show that a=2.1611 x 10^{-3} MW-yr/MT.

3.3.2.2 Electrolysis

Electrolysis uses electric current to recover the oxygen. Oxygen is collected at the anode, while metallic components collect at the cathode. The process may or may not use a fluxing agent to facilitate the reaction.

3.3.2.2.1 Magma Partial Oxidation

The major processing steps for magma partial oxidation are [3, pg. 69]:

1. Melting magnetically cleaned, degassed iron rich (mare) soil in the presence of oxygen.
2. Controlled devitrification.
3. Cooling, pulverizing, and magnetic extraction.
4. Dissolving magnetic spinel phases in electrochemically stable aqueous mineral acid.
5. Electrolysis of the above solution(s) to recover iron and oxygen.

The power requirements are listed in Table 3-2 for a plant with a throughput of .4403 MT/hr [3, pg. 73].

Process Step	Power (kW)
Input heating	46.5
Grinding	20
Electrolysis	234.6
Liquification	660 [6, pg. 564]
Margin	50
Total	1,011

Table 3-3. Power Requirements of Magma Partial Oxidation Process

Normalizing these power requirements by the throughput gives a=2.62×10^{-4} MW-yr/MT.

3.3.2.2.2 Fused Salt

In the fused salt approach, the lunar soil is dissolved in an electrolyte of molten fluorides and electrolyzed. The process occurs at 1000°C. This process requires 14.3 kWh/kg O_2 [8, pg.344]. An additional 1.5 kWh/kg O2 is required for liquification. Calculations show a=1.80×10^{-3} MW-yr/MT.

3.3.2.2.3 Molten Silicate

The molten silicate process involves electrolysis of molten lunar soil. The electrolysis occurs at 1425°C. This process requires 26.0 kWh/kg O_2 [8, pg.344]. An additional 1.5 kWh/kg O_2 is required for liquification. Calculations show a=3.14×10^{-3} MW-yr/MT.

3.3.2.3 Vapor Phase Pyrolysis

Vapor phase pyrolysis takes advantage of thermophysical processes instead of electrochemical ones. The lunar soil is ground into a fine powder and vaporized. The vapor is heated until it reaches the temperature where the oxygen and sub-oxides dissociate. A process involving rapid cooling of the dissociated vapors causes the sub-oxides to liquefy and the oxygen is collected.

The interesting aspect of vapor phase pyrolysis is that it uses direct thermal energy. This allows for the possibility of using direct heating by a solar or nuclear source. For a production rate of 100 MT/yr (duty cycle of .43), the power requirements are:

Requirement	Type	Power (kW)
Heating	Thermal	658
Mechanical	Electrical	258
Liquification	Electrical	40
Total		956

Table 3-4. Power Requirements of Vapor Pyrolysis Process

Calculations show a=4.11×10^{-3} MW-yr/MT [5, pg. 129]. If this process is to be considered, the mass of the power system should be adapted to account for a purely thermal power source (68.8% of the total demand.) An alternative approach would be to use a value of a=1.28×10^{-3} MW-yr/MT, which accounts for the electrical energy, and adapt the mass of the processing facility to account for the thermal source. A further discussion can be found in Section 3.11.

3.4 Cost of Power (C_p)

The cost of power will be a function of the required peak power. The function will be different depending on the type of power selected. The total cost of power includes the cost of design and the cost per unit of power produced. For simplicity, it was assumed that both vary linearly with the peak power capability of the power system. Since C_p is the total cost normalized by the peak power, this results in C_p being a constant (dependent on the power type).

$$C_p = b$$

3.4.1 Solar Arrays

It is estimated that the amount of solar radiation incident on the lunar surface is 1.33 kW/m^2 [9, pg. 37]. For a \$7 cell, η=.17, and an area of 1.8 cm^2 [1, pg. 395], the cost of power is \$172 M/MW. These are very low efficiency cells (1964). Since then, the efficiency has gone up and the price has come down. The baseline scenario assumed that the cost of solar power is \$100 M/MW and that was considered a conservative estimate.

3.4.2 Nuclear Plants

At this time, the most appropriate research into nuclear power sources is in the SP-100 program. The nuclear core produces nearly 2.4 GW of thermal power and with a Stirling cycle added to the standard thermionic device would provide at least 350 kW of electrical power. The cost is very difficult to estimate at this time. For radioactive power sources, the primary cost is for the radioactive material. Table 3-5 lists characteristics and costs of some radioactive isotopes.

Isotope	Compound	Power Density	Half-life	$/thermal kilowatt
Ce^{144}	CeO_2	1,953	285d	14,000
Pm^{147}	$Pm2O_3$	167	2.6y	1,630,000
Sr^{90}	$SrTiO_3$	111	27.7y	23,000
Cm^{242}	Cm30Ni	14,770	162d	45,000
Pu^{238}	PuC	552	86.4y	45,000
Pu^{238}	Pu	581	86.4	1,600,000

Table 3-5. Cost and Characteristics of Isotopes [1, pg. 384]

For an output of 2.4 GW of thermal power and 350 kW of electrical power, the cost ranges between $96 M/MW and $188 M/MW except for the two most expensive isotopes. These are relatively crude estimates based on materials other than the ones actually used in the SP-100. Therefore, the worst case scenario of $200 M/MW was assumed.

3.4.3 Fusion Plant

Fusion power is not technically feasible at this time, but may be possible in the near future. The presence of He^3 on the moon would make fusion an attractive option especially for very large power requirements. Studies indicate that fusion remains attractive down to powers of 100 MW [7, pg. 80]. Even at a cost of $1 billion/MT for He^3, the cost of power is on the order of $1 M/MW. Even if facility costs are an order of magnitude greater than the fuel costs, the total costs would be an order of magnitude less than alternative sources.

3.4.4 Storage Devices

The most likely energy storage device is an H_2-O_2 fuel cell because of its high power density. Because of the simplicity of such fuel cells, the unit cost is considered negligible compared to the other costs of the power system. Storage devices will only be necessary if solar power is used. The lunar night lasts for 14 nights and at least a minimal power reserve will be necessary for life support.

3.5 Number of types of lunar base modules (N_t)

The effect of this variable is to determine the cost of modifying space station modules to house the ground support. In order to minimize cost, it was assumed that modularity would be the top priority in modifying the modules for lunar use. The total cost of redesign ($N_t x C_n$) was assumed directly. From Simon's baseline scenario, this quantity was assumed to be $300 million.

$$N_t x C_n = \$3x10^8$$

3.6 Cost of Modifying Space Station Modules (C_n)

See Section 3.5.

3.7 Number of Lunar Base Modules (N_m)

There will have to be lunar base modules to house the support personnel. Each module will be capable of housing a predetermined number of personnel. Therefore, N_m can be expressed as a function of the number of support personnel.

$$N_m = N_b / (\text{number of personnel per module})$$

From the baseline scenario, it was estimated that each module could house a crew of 20.

$$N_m = N_b / 20$$

It will be shown that the number of personnel can be expressed as a function of the production rate; therefore, the number of lunar base modules can be determined as a similar function.

3.8 Unit Cost of Lunar Base Modules (C_u)

From the baseline scenario, the unit cost for each lunar base module is assumed to be $200 million [6, pg. 535]. The production costs are very insensitive to any variations in this variable. (It was ranked last in sensitivity in Simon's study [6, pg. 536].)

3.9 Processing/ Storage Facility Cost (C_f)

There are two components to the facility cost. The first part is the design and development costs. The other part is the actual cost of parts and assembly. It was assumed that the total cost is a linear function of the mass of the plant. Three points were available from Simon's model. These are listed in Table 3-6.

Case	M_f (MT)	C_f ($M)
Baseline	30	500
Best	15	300
Worst	50	1000

Table 3-6. Costs and Masses of Processing/Storage Facility

If a line is fit to these points, crossover occurs at a negative cost! The decision was made to go with a more conservative estimation, one that passes through the origin. This gives the smallest economy of scale.

$$C_f = 19.2\ M_f$$

The mass of the facility can be determined as a function of the capacity (See Section 3.13). Therefore, the cost of the facility can be determined as a function of the capacity.

3.10 Earth to Moon Transportation Cost (C_t)

This term is designed to include costs to place the plant and habitat modules on the lunar surface. This term should reflect the complete cost of transporting the plant, modules, and the vehicles to LEO. This includes both launch and vehicle costs. The simplest plan is to use an LL to transport payload from LEO directly to the lunar surface. This would allow 15 MT to be delivered for each LL (See Appendix I). A more efficient plan is to use the LL to soft land the equipment using an OTV as a booster stage. An iteration was done to optimize the payload to be landed with that constraint (See Appendix I).

Using this configuration, an LL can deliver 44 MT to the lunar surface (see Appendix I); therefore, the costs include the number of lunar landers, the fraction of the OTV lifetime consumed by the transfers, and the cost of refueling the OTV for each trip. Each vehicle requires a launch costing \$250 M and each LL has a unit cost of \$759 M. An additional \$431 M is needed to transport the fuel consumed by the OTV each trip (at \$3400/kg).

$$c_t = \frac{1}{44\mathrm{MT}}\left[\left(1+\frac{1}{N_{\mathrm{Flights}}}\right)\$1009\mathrm{M} + \$431\mathrm{M}\right]$$

After normalizing the cost by the mass delivered, c_t is calculated to be \$33,490/kg or \$33.5 M/MT. This is much greater than the baseline scenario of \$10,000/kg. However, the lunar landers used to deliver the production equipment will also be used for transporting the LLOX, a cost not accounted for in the Simon model.

3.11 Specific Mass of Power System (M_p)

The total mass of the power system is assumed to vary linearly with the peak power. Therefore, the power density is a constant.

$$M_p = c$$

The constant (c) is dependent on the type of power source selected.

3.11.1 Solar Arrays

Solar power systems for lunar use have been estimated in the past to have a specific mass of 8 kg/kW [5, pg. 129], while more recent projections are around 5.3 kg/kW [6, pg. 565]. It was decided to use the average of the two estimates to be conservative. This gives Mp=6.65 MT/MW. The disadvantage of using solar power is the 14 day long lunar nights. Although the plant does not have to be running during this time, life support would have to be powered by storage devices.

3.11.2 Nuclear Plants

The SP-100 design performance requirements specify 100 kW for less than 3000 kg. An additional 250-350 kW could be produced by using the rejected heat in a Stirling or Brayton system, while 500-700 kg could be saved by eliminating the shadow shield. Instead of transporting shielding material from earth, lunar materials can be used, as well as taking advantage of natural craters.

The additional engine system mass can be approximated. A Rankine cycle capable of 250 kW that operates between similar temperatures and uses potassium as the working fluid has a specific power of 10 lbs/kW or 4.54 kg/kW. This would give the modified SP-100 design an overall specific power of 10 kg/kW.

The lifetime of the SP-100 can be extended to 7 yrs with very little change. Since the lunar plant is to be amortized over 10 yrs, the specific power must be scaled accordingly.

$$M_p = 14.84 \text{ MT/MW}$$

Once again, this is a conservative estimate using proven technologies.

3.11.3 Fusion Plant

Although highly speculative, fusion power would provide the highest power density possible (with the exception of matter-antimatter annihilation). The mass of the reactor is the primary driver. As with nuclear reactors, the mass of the core remains fairly constant, regardless of power output. This can cause small power reactors to be very heavy, but large reactors become attractive. Shielding mass tends to vary linearly with the power output.

The most promising fusion possibility involves isotopes of deuterium (D) and helium-3 (He^3).

$$D+He^3 \rightarrow p(14.7\ \text{MeV})+He^4(3.7\ \text{MeV})+18.4\ \text{MeV}$$

This reaction in theory does not involve neutrons or radioactive species that cause severe damage to surrounding reactor components. In practice, some side DD reactions cause up to 1% of the energy to be released in the form of neutrons. This means that 99% of the energy is released in the form of charged particles. Estimations are that 70-80% of the energy can be converted to electricity using electrostatic means [12, pg. 460]. Since there is less damaging radiation, less shielding is required.

It is hard to estimate the specific mass of the power system. The fuel mass is negligible. At 60% efficiency, one ton of He^3 would provide 12,000 MW-yr of electrical energy [12 pg. 470]. As mentioned in Section 3.4.3, studies indicate that fusion would be feasible only for power sources of greater than 100 MW, a demand larger than any considered in this report.

3.11.4 Storage Devices

For storage devices, there are several choices including batteries and fuel cells. Fuel cells provide the highest power densities by almost an order of magnitude. An H_2-O_2 fuel cell contains 1600 W-hrs/lb. The following equations [1, pg. 400] were used with t=336 hrs:

$$t = \frac{14(2v-1)}{(1-v)^2}$$

$$\eta = .67v$$

Calculations show that the efficiency of the fuel cell is 55% and the specific mass would be 170.8 MT/MW. If the fuel cell is used in a regenerative loop with a converter of 70% efficiency and a specific mass of 70 lbs/kW, the specific mass of the entire system is 275.7 MT/MW. Even the best projections for secondary batteries are more than five times heavier [11, pg. 362]. It is obvious that the transportation cost of the storage devices will dominate their effect on the overall capital costs.

3.12 Mass of Lunar Base Modules (M_m)

The mass of the modules is determined by optimizing the mass per personnel constrained by the launch system. The lunar base module is assumed to be a modified space station module. In the baseline scenario, this mass was assumed to be 20,000 kg.

3.13 Mass of processing/storage facility (M_f)

The mass of the plant will be a function of the capacity and will depend on the chemical process. The mining and beneficiation mass is assumed to vary linearly with the capacity while the plant mass is assumed to follow a 2/3 relation, unless otherwise specified.

In general, the mass is proportional to the surface area, while the capacity is proportional to the volume. Surface area is related to a length squared and the volume is related to a length cubed. For example, the volume of a spherical tank is related to the radius cubed, while the surface area is proportional to the radius squared.

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

$$S = 4\pi r^2$$

Therefore, the surface area (mass) varies proportionally to the volume (capacity) to the 2/3 power.

3.13.1 Mining and Beneficiation

It is estimated that for a 1000 MT plant operating at 100% duty cycle, the mining and beneficiation equipment will have a mass of 10.8 MT [6, pg. 565]. The plant mass can be expressed as:

$$M_f = .0108C + dC^{2/3}$$

The power of the mining and beneficiation equipment is assumed to be the same regardless of the chemical process being used. Slight variations may occur due to differences in process yield or the amount of beneficiation necessary. These differences tend to counteract each other; more beneficiation usually leads to higher yields and vice-versa.

3.13.2 Processing Plant

In addition to the power needed to mine and process the lunar soil, energy is needed to separate, collect, and liquefy the oxygen. These chemical processes are well understood, but some uncertainties remain. The major unknown is the effect of a low g environment.

3.13.2.1 Hydrogen Reduction

Table 3-7 shows the mass budget for an experimental apparatus utilizing the hydrogen reduction process.

Component	Mass (kg)
Pump	3.5
Furnace	2.7
Vessel (empty)	0.8
Tubing, miscellaneous	1.5
Electrolysis cell	2.0
Condenser	1.1

Table 3-7. Mass Summary for Hydrogen Reduction Process

The capacity of the test apparatus was .8 mg/min. Calculations give a value of $d=1.5x10^{-2}$ $(MTyr^{2})^{1/3}$, but this does not include increases that would result in designing for the lunar environment. One would expect at least an order of magnitude difference between a design for earth use and one for lunar production. Therefore, it was decided to use the same mass as the one used for the carbothermal process, because numbers were available for a lunar design and the two reduction processes are very similar.

3.13.2.2 Carbothermal Process

Studies estimate that a processing plant utilizing the carbothermal process and capable of producing 1000 MT/yr would have a mass of 30.4 MT [6, pg.565]. The processing plant mass is assumed by the authors to vary with the 2/3 power. Therefore, for the carbothermal process, the plant mass can be expressed as:

$$M_f=.0108C+.304C^{2/3}$$

3.13.2.3 Vapor Phase Pyrolysis

Table 3-8 shows the mass summary of a 100 MT/yr vapor pyrolysis plant.

Facility	Mass (kg)
5 Processing Columns	4,500
5 Solar Concentrators	1,500
Photovoltaic Power Plant	2,100
Power Conditioning	500
Cooling System	2,600
Oxygen Collection System	1,400
Solar Shields	300
All Facilities	12,900
Contingency (10%)	1,300
Plant Total	14,200

Table 3-8. Mass Summary of Vapor Pyrolysis Process [5, pg. 130]

According to the study, a column with a capacity of 20 MT/yr is the optimal size for production; therefore, the processing plant mass scales linearly. The mass of the solar concentrators is included, but the mass of the photovoltaic cells is accounted for as part of the power system. The solar concentrators are included because they are able to take advantage of thermal energy without converting to electrical energy. The mass of the facility can be expressed as a scalar function of the capacity.

$$M_f=.1186C$$

3.14 Cost of Design (C_D)

As discussed in Section 2.8.1, the cost of design for the transportation system is taken as twice the estimated R&D cost of the lunar lander. Therefore, the total cost of design is $3.078 B. This cost is assumed to cover the OTV, the derived lunar lander, and the equipment necessary to transfer payloads between the two vehicles.

3.15 Number of Lunar Base Resupply Missions per Year (N_R)

This factor was manipulated to give the proper amount of mass needed to resupply the LLOX plant. The support crew will need life support supplies, and reagents, used for the chemical process, will need to be replaced periodically. Life support needs are estimated in Table 3-9.

Supply	Demand (lbs/man-day)
Oxygen	4.19
Food	1.36
Water in Food	1.10
Food Prep Water	1.58
Drink	4.09
Hand/Face Wash Water	4.00
Shower Water	8.00
Clothing (disposable)	2.50
Clothes Wash Water	27.50

Table 3-9. Life Support Needs [12, pg. 507]

The amount of oxygen was changed to the calculated value in Section 6.1.1. If clothes are washed instead of replaced daily, the total input to the life support system is 51.82 lbs/man-day. It is estimated that 90% of the water can be recovered (except for the water in the food [12, pg. 507]). This reduces the amount that has to be supplied from the earth to 11.167 lbs/man-day. If LLOX is used for the respiration needs and 90% of the water requirement, only 2.91 lbs/man-day are necessary. This translates to 483 kg/man-yr, or .5 MT/man-yr to be conservative.

Other ideas to reduce life support resupply needs have included using plants such as wheat and potatoes to recycle the air and reduce the amount of food that has to be transported from earth. The life support demands may be further reduced from the calculated value.

In addition to the resupply needs of the life support, reagents involved in the chemical process may have to be replaced. The mass of the regents is directly proportional to the production rate and the constant of proportionality would depend on the chemical process involved. The number of resupply missions that would yield the correct resupply mass is:

$$N_r=(.5N_b+f\pi)/M_m$$

Since it is assumed to that LLOX will be used for life support purposes, the production rate must be replaced by a modified production rate in the calculation of the unit cost of production. This subtracts "off the top" the fraction (2.81%) of the LLOX consumed by the support personnel.

$$C_{Production}=(CC+nOC)/n\pi'$$

$$\pi'=\pi-.0281\pi=.9719\pi$$

The modified production rate will also be used for calculating the number of vehicles necessary to transport the LLOX.

3.16 Net Lunar Oxygen Delivered (d)

This will be determined by the amount of LLOX to be delivered to LEO, Lunar orbit, and to the moon base itself.

$$d=(N_{LEO}d_{LEO}+N_{LLO}d_{LLO}+N_{LB}d_{LS})/(N_{LEO}+N_{LLO}+N_{LS})$$

From the analysis of the transportation system (See Appendix I), the following values of delivery ratios were used:

Location	d
Lunar Surface	0.947
LLO	0.518
LEO	0.256

Table 3-10. Delivery Ratio for Different Delivery Sites

These values take even the LLOX used to transport resupplies into account.

3.17 Ground Support Manpower (N_b)

The number of people needed to operate and maintain the facility will increase as the size of the plant increases. Simon used twenty as the ground support personnel for a production rate of 1000 MT. Assuming that the number of personnel scales linearly with the production rate, the following relationship holds:

$$N_b = .02\pi$$

This will tend to slightly overestimate the ground support personnel for production rates over 1000 MT, since one would expect an economy of scale for the ground support.

3.18 Ground Support Overhead Factor (N_f)

The overhead factor is a multiplier that accounts for any costs associated with the ground support personnel in terms of man-year salaries. In Simon's original model, the baseline value was 25, with the range being between 5 (best case) and 50 (worst case). For the purposes of this report, the overhead factor of 25 was considered sufficiently conservative.

3.19 Number of Lunar Landers (N_{LL})

The total number of lunar landers is a scalar of the production rate. The number of lunar landers per year is more complicated. A LL can be used for solely resupply missions, transportation of the LLOX, or a combination of resupply and delivery. The optimal usage of the LL will depend on the delivery site of the LLOX.

A number of lunar landers will have been already used to deliver the production equipment and are accounted for in the capital costs. It is assumed that the remaining lunar landers to be delivered are spread uniformly over the amortization period (n). If more lunar landers are used to deliver the initial mass than are needed to transport the LLOX and yearly supplies, N_{LL} is set to zero.

3.19.1 LEO

For delivery to LEO, LL missions will both resupply and deliver. Two LL are needed to service each OTV. Therefore, the total number of LL will be twice the number of OTV and the number of LL that have to be replaced yearly is:

$$N_{LL} = 2N_{OTV} - \frac{Pm_p + n_m m_m + m_f}{n44MT}$$

3.19.2 LLO

For delivery to LLO, LL will serve two purposes. One will be to transfer LLOX to the delivery site. Each LL mission can deliver 60 MT to LLO. The other purpose will be to refuel and assist the OTV for resupply missions. Three LL are needed to refuel the OTV and transport supplies (See Appendix I) to the lunar surface. Therefore, the number of lunar landers that are needed each year is:

$$N_{LL} = \frac{d\pi'}{N_{flights} 60MT} + 3N_{OTV} - \frac{Pm_p + n_m m_m + m_f}{n44MT}$$

3.19.3 Lunar Surface

This is the location that obtains the maximum benefit from a LLOX plant. No propellant is necessary to deliver the LLOX to the lunar base. Lunar landers will still be necessary for purely resupply missions. Therefore the number of lunar landers that have to be supplied yearly is:

$$N_{LL} = 3N_{OTV} - \frac{Pm_p + n_m m_m + m_f}{n44MT}$$

3.20 Cost of Lunar Landers (C_{LL})

The total cost of procuring and delivering a lunar lander to LEO is \$1.009 B. See Sections 2.8.2 and 2.8.4.

3.21 Number of OTV (N_{OTV})

The number of OTV is easier to calculate. Only the fraction of the lifetime used in transporting the initial mass was accounted for in the capital costs. OTV can also be used in three manners: to deliver LLOX, resupply missions, and a combination of resupply and delivery.

3.21.1 LEO

The number of OTVs needed will depend on the total mass to be delivered ($d\pi$) and the payload mass of each OTV (59 MT).

$$N_{OTV} = \frac{d\pi'}{N_{flights}59MT}$$

For delivery to LEO, (d=.224, M_{PLOTV}=59 MT, and $N_{flights}$=30), the number of OTV is simply a scalar of the production rate:

$$N_{OTV}=1.266 \times 10^{-4}\pi'$$

3.21.2 Low Lunar Orbit

The only foreseeable demand for LLOX in LLO would be for Mars missions. OTV will be necessary only for resupply missions. Each OTV can deliver 60 MT of payload to LLO. The OTV will have to deliver the normal resupply demands, $(.01+f)\pi$, and the H_2 used by the LL to deliver the LLOX, $.048\pi'$,or $.046\pi$, (See Appendix I). The H_2 consumed by the LL during the resupply missions is already accounted for in the transportation calculations (See Appendix I).

$$N_{OTV} = \frac{(.056+f)\pi}{N_{flights}46.8MT}$$

3.21.3 Lunar Surface

OTV will be necessary only for resupply missions. The H_2 consumed by the LL is already accounted for in the transportation calculations (See Appendix I).

$$N_{OTV} = \frac{(.01+f)\pi}{N_{flights}46.8MT}$$

3.22 Cost of OTV (C_{OTV})

The total cost of procuring and delivering a lunar lander to LEO is $1.009 B. See Sections 2.8.3 and 2.8.4.

3.23 Summary of Functional Dependence

This section summarizes the functional dependence of all the variables that affect the capital and operations costs. Table 3-11 demonstrates the interdependence of the variables. The equations of Table 3-11 can be rearranged to reveal that all the variable can be reduced to functions of the production rate and the capacity. The resulting equations are listed in Table 3-12.

Variable	Units	Dependence on Other Variables
P	MW	$aC+.0085n_b$
C_p	$/MW	b
$N_t x C_n$	$	$3x10^8$
N_m	Number of modules	$.05n_b$
C_u	$/module	$2x10^8$
C_f	$	$19.2x10^6 m_f$
C_t	$/MT	35,000
M_p	MT/MW	c
M_m	MT/module	20
M_f	MT	$.0108C+dC^e$
CD	$	$3.078x10^9$
N_R	Number of modules	$(.5n_b+f\pi)/M_m$
d	Fraction	See Section 3.16
N_b	Number of heads	$.02\pi$
N_f	Multiplier	25
N_{LL}	Number of LL	See Section 3.19
C_{LL}	$/LL	$1.009x10^9$
N_{OTV}	Number of OTV	See Section 3.21
C_{OTV}	$/OTV	$1.009x10^9$

Table 3-11. Interdependence of Cost Variables

Variable	Units	Dependence on Production/Capacity
P	MW	$aC+1.7x10^{-4}\pi$
N_p	\$/MW	b
$N_t x C_n$	\$	$3x10^8$
N_m	Number of modules	$.01\pi$
C_u	\$/module	$2x10^8$
C_f	\$	$.2074C+19.2dC^e$
C_t	\$/MT	35,000
M_p	MT/MW	c
M_m	MT/module	20
M_f	MT	$.0108C+dC^e$
C_D	\$	$1.539x10^9$
N_r	Number of modules	$(.01+f)\pi/20MT$
d	Fraction	See Section 3.16
N_b	Number of heads	$.02\pi$
N_f	Multiplier	25
N_{LL}	Number of LL	See Section 3.19
C_{LL}	\$/LL	$1.009x10^9$
N_{OTV}	Number of OTV	See Section 3.21
C_{OTV}	\$/OTV	$1.009x10^9$

Table 3-12. Cost Variables Expressed as Functions of Production Rate and Capacity

Chapter 4

Implications of Model

4.1 Ore Type

The percentage of oxygen varies slightly with ore type. It ranges between 44.6% by mass in the highland soils and 39.7% in the mare soil [2, pg. 14]. If the soil is lower in oxygen content, more of it has to be mined and processed to achieve the same production rate. Both the mass and power required by the mining and beneficiation equipment will increase.

The highlands soil will be defined as the standard soil. Therefore, the plant mass will equal:

$$M_f = \frac{.0108}{\eta_S} C + aC^{2/3}$$

$$\eta_S = \frac{\text{Mass}\%O_2}{44.6\%}$$

The increase in power is negligible when compared to the total power. Figure 4-1 illustrates the difference caused by mining an oxygen poor ore.

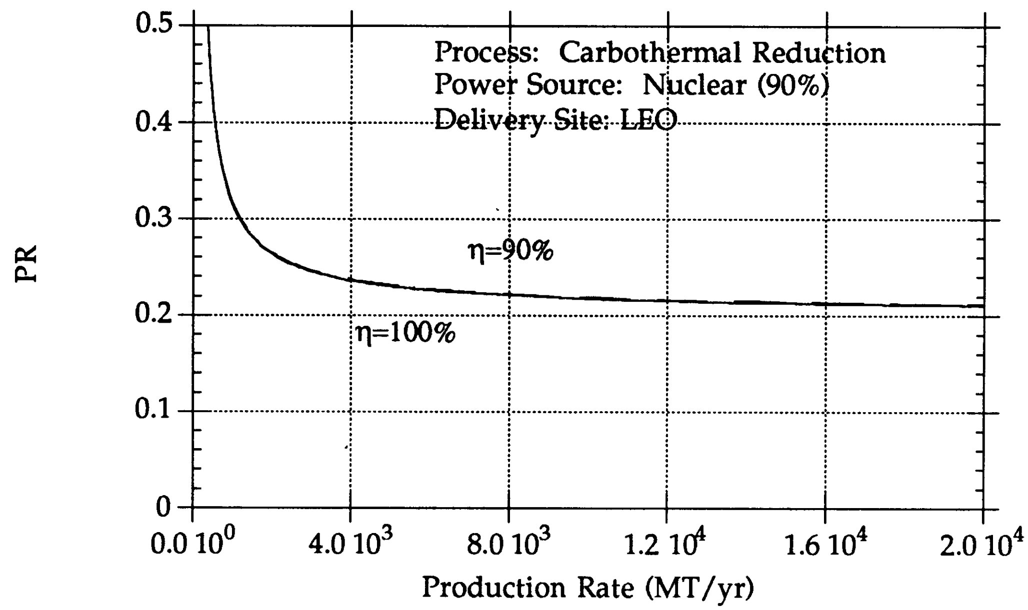

Figure 4-1. Effect of Oxygen Poor Soil on PR

It is obvious that the mass percentage of the soil has very little effect on the production ratio (at least over the expected range). Further calculations assume that highlands soil is available. However, other considerations may place the plant in a less oxygen rich area.

4.2 Location of Processing Plant

Two factors important in choosing the location of the processing plant are considered: the availability of oxygen rich soil and the availability of sunlight. The implications of ore type have already been discussed. The location of the plant at a lunar pole may enable continuous sunlight to reach the plant. This would eliminate the need for storage devices and allow for longer duty cycles for solar arrays. Figure 4-2 illustrates the impact this would have on the production ratio.

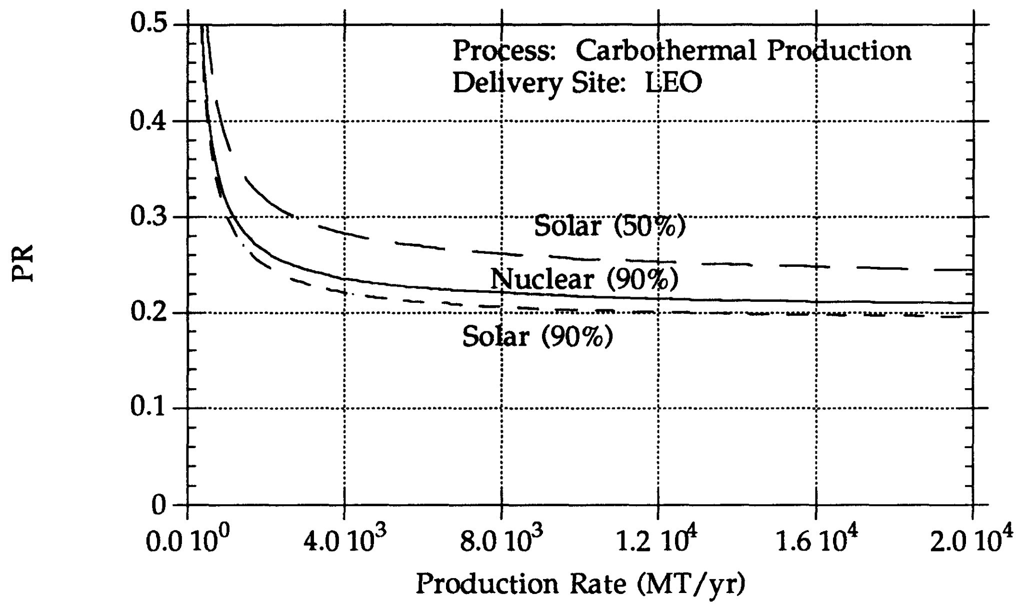

Figure 4-2. Effect of Continuous Sunlight

If continuous sunlight were available, solar arrays would become the most desirable power source. The benefit is greater at higher production rates. To achieve continuous sunlight at the poles, the solar arrays would have to be mounted on surrounding hills or on platforms. This would become more difficult as the size of the power system increased. At this time, a LLOX plant located at the lunar pole is not considered a viable option. There is not enough area for solar arrays of considerable size where continuous sunlight is available.

4.3 Use of Fuel Cells

This section explores the use of fuel cells in the LLOX power system. On one hand, using fuel cells to extend the duty cycle increases both the mass and cost of the power system. On the other hand, by increasing the duty cycle, which decreases the capacity, the mass of the plant is decreased.

The only terms that are affected by the use of fuel cells are those containing the capacity and the constants b and c. All these terms are associated with the capital costs. The next step is to explore how the use of fuel cells affects the constants b and c. Using a solar array and assuming sunlight is available 50% of the time, the mass and cost, if the cost of the fuel cells is considered negligible (See Section 3.4.4), of the array-battery system are:

$$Pm_P = m_{PS}\left(1+\frac{1.7\text{x}10^{-4}}{1.7\text{x}10^{-4}+\frac{a}{\lambda}}\right)\frac{\lambda P}{.5}+P(\lambda-.5)m_{PB}+P\left(\frac{1.7\text{x}10^{-4}}{1.7\text{x}10^{-4}+\frac{a}{\lambda}}\right)m_{PB}$$

$$Pc_P = c_S\frac{\lambda P}{.5}\left(1+\frac{1.7\text{x}10^{-4}}{1.7\text{x}10^{-4}+\frac{a}{\lambda}}\right)$$

Therefore, the constants can be expressed in terms of the duty cycle:

$$c = m_{PS}\left(1+\frac{1.7\text{x}10^{-4}}{1.7\text{x}10^{-4}+\frac{a}{\lambda}}\right)\frac{\lambda}{.5}+(\lambda-.5)m_{PB}+\left(\frac{1.7\text{x}10^{-4}}{1.7\text{x}10^{-4}+\frac{a}{\lambda}}\right)m_{PB}$$

$$b = c_S\frac{\lambda}{.5}\left(1+\frac{1.7\text{x}10^{-4}}{1.7\text{x}10^{-4}+\frac{a}{\lambda}}\right)$$

Substituting these into the capital costs yields a function of production rate and duty cycle. In theory, it is possible to analytically solve for the duty cycle that minimizes the capital costs in terms of production rate. For this model, a simpler method is to graph the PR curves for several duty cycles. Figure 4-3 shows that the costs increase for $\lambda>.5$.

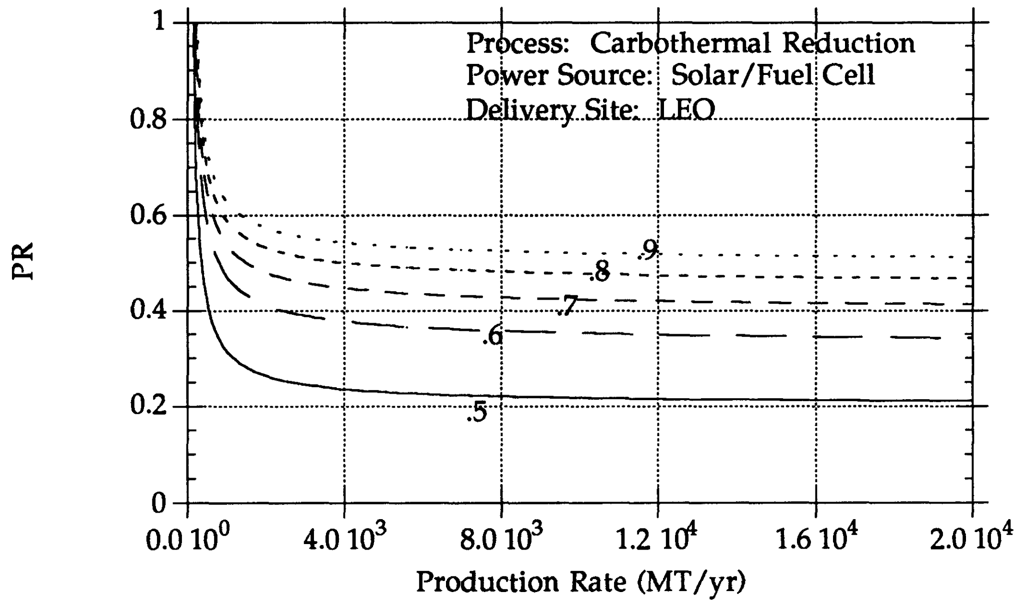

Figure 4-3. Effect of Using Fuel Cells to Increase Duty Cycle

In summary, to minimize the overall cost, fuel cells should be use only to provide continual life support. The mass of the batteries increases faster than the mass of the plant decreases due to an increase in the duty cycle.

4.4 Choice of Power Type

The power type affects the production costs in three ways: direct cost of power, cost to transport mass of the power, and the duty cycle. As shown in Section 4-3, the power system affects only the capital costs.

$$CC=(P^*c_p)+(n_t^*c_n)+(n_m^*c_u)+c_f+c_t^*[(P^*m_p)+(n_m^*m_m)+m_f]+C_D$$

As shown in Section 4-3, the maximum duty cycle for solar power is 0.5. It is assumed that the plant will be down 10% of the time for servicing. Therefore, the maximum duty cycle for nuclear power is 0.9. Figure 4-4 illustrates the difference between the PR curves for solar and nuclear power.

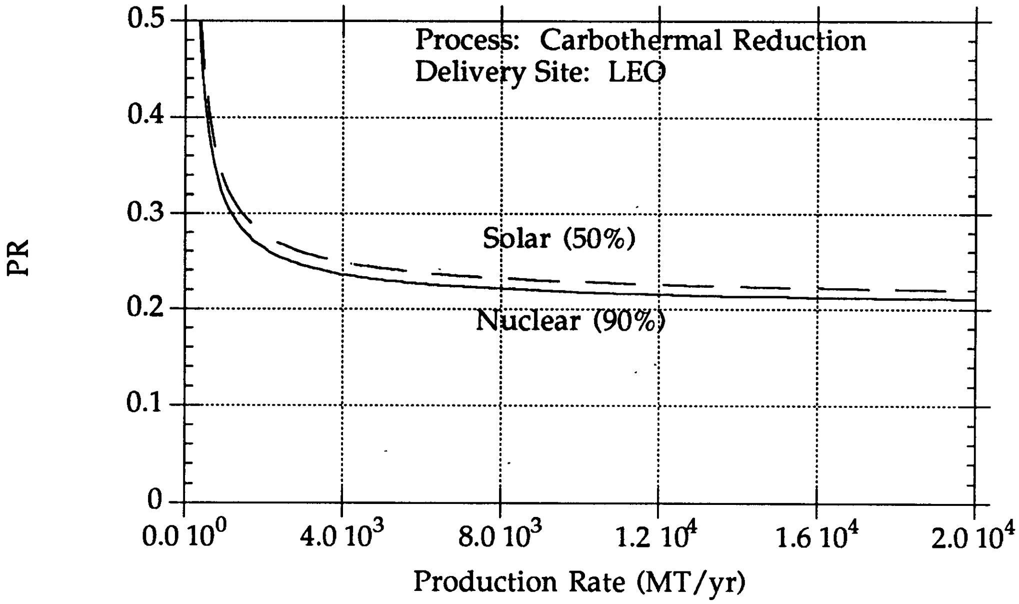

Figure 4-4. Solar vs. Nuclear Power

For comparison purposes, the calculations for the solar curve did not include fuel cells for life support. This makes it possible to examine the affects due only to C_p, M_p, and the duty cycle.

At low production rates (<1000 MT/yr), solar power is competitive because of its lower cost and higher power density. At high production rates, nuclear power provides a significant cost savings because it allows for a higher duty cycle, which decreases the mass of the plant.

4.5 Chemical Process Selection

The selection of the chemical process affects the model in two ways: the mass of the processing plant and the necessary power. These, in turn, affect other variables such as the cost of the power plant. Table 4-1 lists the important characteristics of the chemical processes.

Chemical Process	a (kW-yr/MT)	M_f	Resupply
Hydrogen Reduction	5.7	$.0108C+.304C^{2/3}$	Hydrogen
Carbothermal Reduction	2.16	$.0108C+.304C^{2/3}$	Carbon
Magma Electrolysis	2.62	NA	None
Fused Salt Electrolysis	1.8	NA	Electrolytes
Molten Silicate Electrolysis	3.14	NA	None
Vapor Pyrolysis	1.28	$.1186C$	None

Table 4-1. Summary of Chemical Processes and Important Characteristics

The decision was made to go with the carbothermal process. The electrolysis techniques involve high temperatures, so one would expect high plant masses. Vapor pyrolysis seems attractive on the surface, but it is limited to daylight operation since it takes advantage of solar concentrators. Also vapor pyrolysis scales linearly instead of following a 2/3 relationship. Figure 4-5 illustrates how these two factors influence the PR curve.

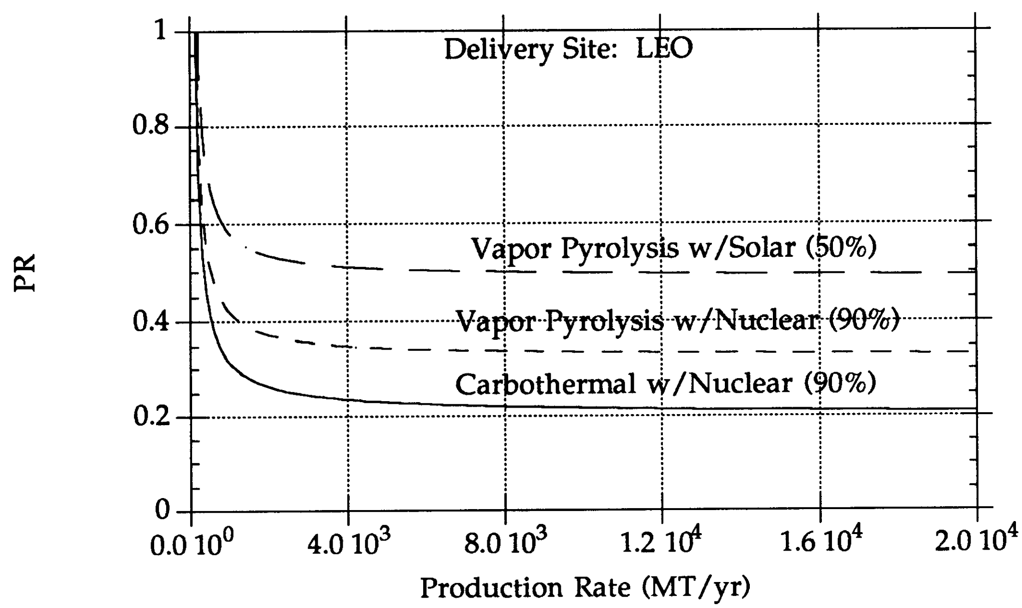

Figure 4-5. Comparison Between Vapor Pyrolysis and Carbothermal Reduction

Vapor pyrolysis is competitive at very low capacities. For comparison purposes, the calculations for the solar curve did not include the mass of the fuel cells. Even if a nuclear source could extend the duty cycle to 90%, the scaling law for the plant mass makes carbothermal reduction the preferred process production rates over 27 MT/yr.

Between the two reduction processes, the one using hydrogen as a reagent consumes more power and would require H_2 to be resupplied since a small fraction of the reagent is lost each time it is used. The carbothermal process uses less power and the carbon could be provided from CO_2 or scrap metal.

4.6 Economies of Scale

It is to be expected that as the production rate increases the cost per unit should decrease. The model shows this quite clearly. A good approximation to the PR curves would be:

$$PR \approx PR_{\infty} e^{\tau / \mathrm{Rate}}$$

As the production rate increases, the production ratio asymptotically approaches PR_{∞}. The incremental savings achieved by increasing the production rate decreases and approaches zero as the production rate approaches infinity. Figure 4-6 shows how the approximation compares to the actual costs.

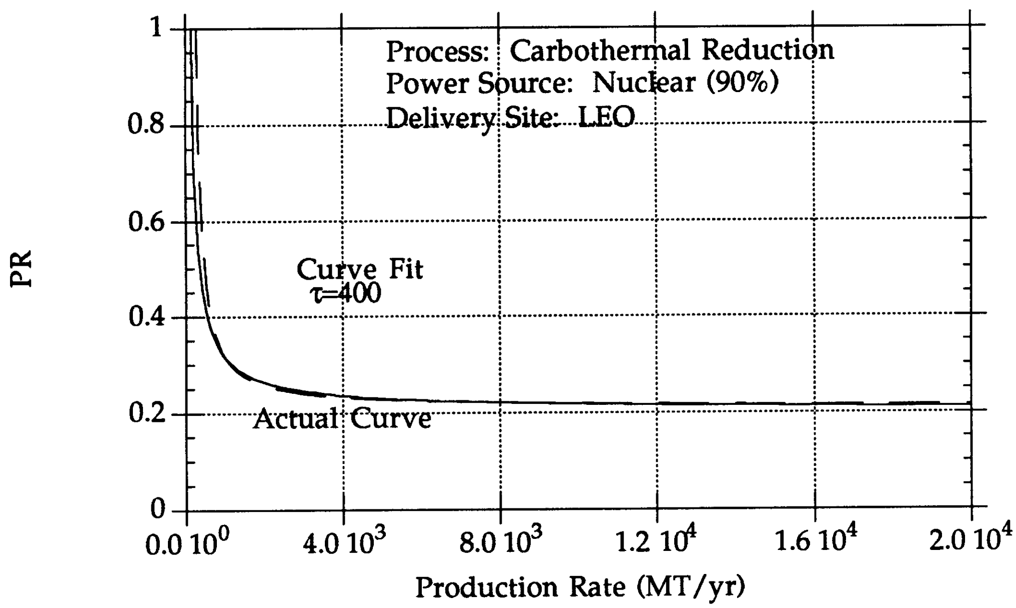

Figure 4-6. Curve Fit Approximation

The curve was chosen to match at a production rate of 1000 MT/yr; this gives excellent agreement for rates greater than 1000 MT/yr and conservative estimates for lower production rates.

Chapter 5

Sources of Demand

5.1 Oxygen Requirements

5.1.1 Respiration

According to an introductory text, the average amount of O_2 used by a resting man is 240 ml/minute (13, pg. 25). This measurement was taken at standard temperature (T=300 K) and pressure (P=101325 N/m^2). The number of O_2 molecules can be found by the ideal gas law.

$$PV=nRT \quad (R=8314\ J/(kmole\ K)$$

$$\text{or}\quad n=PV/(RT)$$

Therefore, a man at rest uses $9.75x10^{-6}$ kmole/minute. This corresponds to $3.1x10^{-4}$ kg/minute, or .0187 kg/hr.

The next step is to examine how the oxygen consumption varies with exercise. Studies show that ventilation can increase 5, 10, or 15 fold with increasing workload (13, pg. 192). One study indicates that up to 28 fold occurs at maximal performance. However, this is equivalent to a sprint and can only be maintained for about 22 s. For estimation purposes, workload has been broken down into three categories: light, medium, and heavy. These correspond to the 5, 10, or 15 fold figures.

The typical work day, either on the space station or at a lunar base, is projected in Table 5-1.

Activity	Workload	Hrs/Day
sleep/relaxation	none	12
technical/recreation	light	8
manual	medium	2
exercise	heavy	2

Table 5-1. Respiration Requirements

The necessary mass of O_2 is estimated to be 1.9 kg/man-day, or 697 kg/man-yr. The mass of O_2 currently budgeted for manned space flight is .806 kg/man-day, or 295 kg/man-yr. This number does not account for the increased physical activity demanded on a lunar base, both for construction and for exercise necessary for extended low-g environments.

This demand seems small compared to other projected demands (a lunar crew of 6 would require only 4.2 MT/yr), but is actually quite significant. The large savings over transporting from Earth ($10,000/kg), make this portion of the LLOX production extremely cost efficient.

5.1.2 Water

It is estimated that 4.627 lb H_2O/man-day, or 2.1 kg H_2O/man-day, is required for life support. This corresponds to 767 kg H_2O/man-yr for each man. Since water is composed of 88.8% oxygen by mass, each man requires 680 kg O_2/man-year in the form of water. Only the hydrogen, which is already necessary for the transportation system, needs to be transported from the earth's surface.

5.1.3 Propulsion

As evident by the transportation system, the greatest demand on LLOX is for propulsion. Demand for propulsive purposes could come from commercial satellite transfers from LEO to GEO, delivery missions for a lunar base, and direct Mars missions.

5.2 Project Demands

5.2.1 Commercial

Commercial satellites designed for operation in geosynchronous orbit (GEO) can first be delivered to LEO and then transferred to GEO along a transfer trajectory. The baseline scenario is designed to provide projected commercial needs. It delivers 492 MT/yr of LLOX to LEO. The projected demand for commercial satellites is assumed to be 500 MT/yr.

5.2.2 Space Station

For a crew of ten the space station would require 13.8 MT/yr of oxygen for life support, 7.0 MT/yr for respiration and 6.8 MT/yr for water synthesis.

5.2.3 Direct Mars Missions

A Mars mission may use LLOX as propellant. Two options are possible: LLOX could be delivered to the vehicle in LEO, or the vehicle could first be moved to LLO where it would be supplied with the LLOX for the transfer to Mars. Some have placed this demand at approximately 28,000 MT over a ten year period [12, pg. 456]. Initial analysis shows that only 28.8% (806.4 MT/yr) of this demand could possibly be met with LLOX.

5.3.4 Lunar Base Demands

A lunar base would require oxygen for life support and for transportation needs. Depending on the delivery scheme for the lunar base mass, the LLOX may have to be delivered to LEO, lunar surface, or some combination of the two. Table 6-1 shows the possible development of a lunar colony.

Stage	Population
Automated Surface Exploration/Site Preparation	2-5
Initial Lunar Base	6-12
Early Lunar Settlements	100-1000
Mature Lunar Settlement	1000-10,000
Autonomous Lunar Civilization	10,000-100,000

Table 5-2. Stages of Lunar Development [6, pg. 86]

It is extremely difficult to estimate the LLOX demand for lunar base delivery. The life support demands are small but very easy to estimate. It shall turn out that even small demands on the lunar surface can make LLOX a cost efficient alternative.

Chapter 6

Test Scenarios

6.1 Demand only in LEO

The first scenario is a demand for LLOX only in LEO. This location has a delivery ratio of 0.256 and a MPR of 0.397. This means that a production cost ratio of 0.168 ($C_{Production}$= \$524/kg) is required for feasibility. From Figure 6-1, it is obvious that LLOX will not be feasible for delivery to LEO only. The capital costs are too high to be recovered by delivering to LEO (the least profitable delivery site considered).

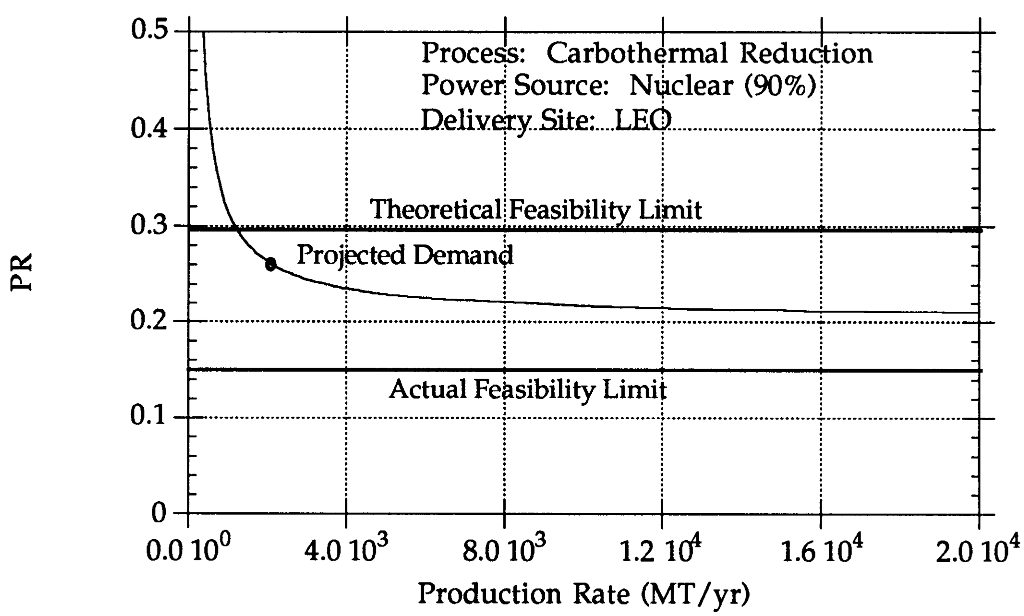

Figure 6-1. Production Ratios for LLOX Delivered to LEO

To meet projected demands, a production rate of approximately 2000 MT/yr would be necessary. At this rate, the production ratio is 0.263. Substituting this into the feasibility equation, the total cost of the LLOX delivered to LEO would be \$4840/kg, 36% more than the cost of delivering from earth (\$3400/kg).

To meet the theoretical limit (assuming infinitely large reusable vehicles), it would be necessary to have a production rate exceeding 1200 MT/yr. Such a plant would be necessary to meet projected demands, but larger vehicles would present a whole host of problems in delivering them to space.

6.2 Demand for Mars Missions Only

For direct Mars missions, there are two strategies. The vehicle could be fueled using LOX from the earth and depart from LEO. Another strategy would be to transfer the vehicle to LLO, fuel it with LLOX, and depart from LLO. Since LLOX is competing with earth LOX delivered to LEO, the cost of delivery will be the cost of delivery to LEO (\$3400/kg).

A difference in supplying Mars missions is that the quantities for feasibility are not the same. For instance, it may require 2 MT of LLOX to be delivered to LLO to save from having to transport 1 MT from earth to LEO. The ratio of the amount of earth LOX that has to be delivered to LEO to the amount of LLOX that has to be delivered to LLO will be denoted β. The PR criterion then becomes:

$$PR \leq d(\beta - MPR)$$

If $\beta=1$, the original criterion is recovered.

For this delivery site, d=0.518 and MPR=0.108 (See Appendix I). From examination of Figure 6-2, it is discovered that there are feasible production rates for $\beta \geq 0.6$.

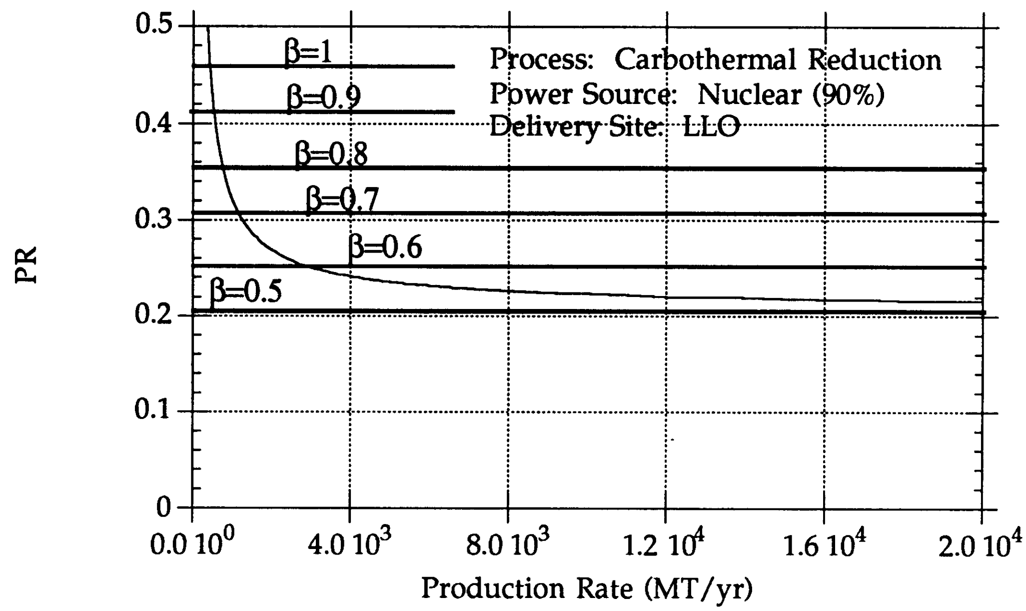

Figure 6-2. PR Curve for Delivery to LLO

Table 6-1 gives a breakdown of the values of β, the corresponding part of the projected demand (2800 MT/yr), the cumulative production rate necessary to meet demands for that β and above, and the average β for the cumulative production rate.

β	Demand (MT/yr)	Cum. Prod. Rate (MT/yr)	Average β
1.1	60	120	1.1
1.0	0	120	1.1
0.9	120	360	0.97
0.8	0	360	0.97
0.7	260	875	0.81
0.6	220	1310	0.74
0.5	70	1450	0.72
0.4	290	2035	0.63
0.3	160	2345	0.58

Table 6-1. β Values for Proposed Mars Missions [12, pgs. 456-7]

As more of the demand is included, the production rate increases, driving down the production ratio. However, the average value of β decreases, causing the feasibility limit to increase. Table 6-2 shows how the necessary production rate increases as the average β decreases.

β	Feasible PR	Necessary Production Rate (MT/yr)
1.1	0.514	328
0.97	0.447	429
0.81	0.364	683
0.74	0.327	916
0.72	0.317	1010
0.63	0.270	1920
0.58	0.244	3600

Table 6-2. Necessary Production Rates to Meet Mars Mission Demand

Comparing Table 6-1 and Table 6-2, one can see that LLOX is a cost efficient alternative for all demands with $\beta \geq 0.4$, $\beta \geq 0.5$, $\beta \geq 0.6$, and $\beta \geq 0.7$. In other words, using LLOX for any of these parts of the demand is less expensive than supplying all of the demand from earth. Calculations were done to see which alternative saved the most money. Table 6-3 compares these possible strategies.

β≥	Demand (Production Rate) (MT/yr)	PR	Cost ($/kg)
0.7	440 (875)	0.333	3310
0.6	660 (1310)	0.295	3190
0.5	730 (1450)	0.288	3120
0.4	1020 (2035)	0.267	3290

Table 6-3. Comparison of LLOX Supply Strategies

The most efficient strategy is to use LLOX only for $\beta \geq 0.5$. At higher values of β, the lower production rate causes the PR to be higher, raising the cost. At lower values, the average β decreases, requiring a greater amount of LLOX to be produced to substitute for the same amount of earth LOX. This raises the average cost.

6.3 Demands for Lunar Surface Only

The resupplies have to be delivered to LEO, where the transportation system ferries them to the lunar surface. Since the resupplies are delivered to a site other than the LLOX delivery site, the feasibility criterion has to be modified.

$$PR \le d(1-\alpha MPR)$$
$$\alpha = \frac{C_{LEO}}{C_{earth}}$$

For LLOX demand on the lunar surface, d=0.947, MPR=0.018, and α=0.34. This means that a production cost ratio of 0.941 ($9,410/kg) is required for feasibility. This can be accomplished with production rates of 35 MT/yr or greater. Figure 6-3 shows that large discount rates are possible for delivery to the lunar surface.

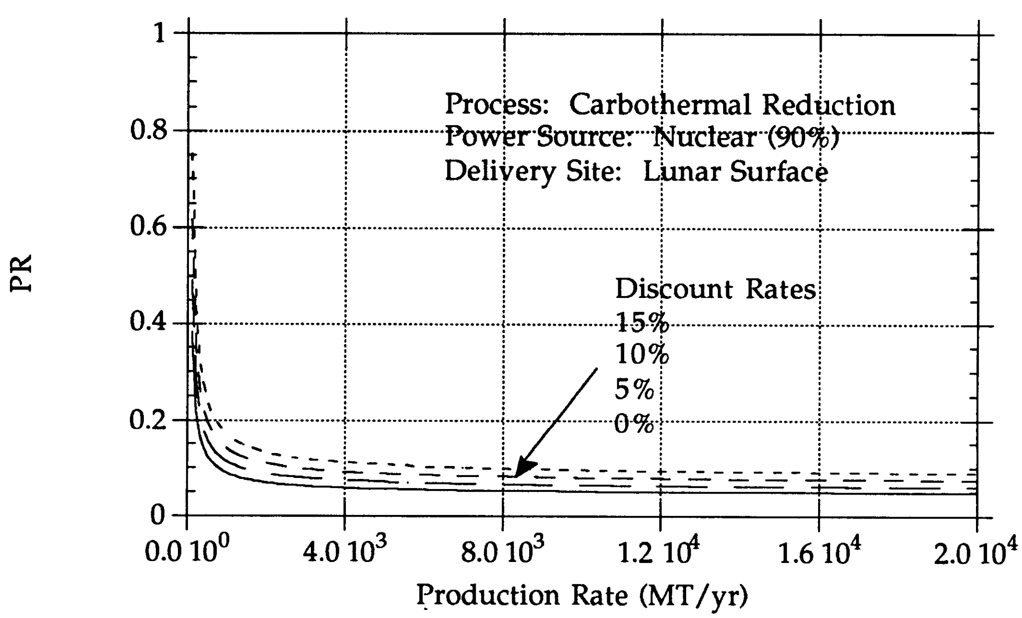

Figure 6-3. Effect of Discount Rate on LLOX Delivered to Lunar Surface

For production rates of over 80 MT/yr, over a 15% return on the capital investment can be expected. In the presence of a lunar base, LLOX becomes attractive even to private investment.

Chapter 7

Conclusions and Recommendations

7.1 Conclusions

7.1.1 LEO

At this time, LLOX is not a feasible way of supplying LEO demands, regardless of the amount of demand. If the size of the transportation system were vastly increased, LLOX could effectively meet the projected demand. This would require the ability to assemble the vehicles in orbit, an ability not available at this time. Other advances may make LLOX feasible, so this option should be reexamined when any new technologies become available. The most likely technologies are discussed later this chapter.

7.1.2 Mars Missions

LLOX is not feasible for single missions, but is marginally feasible for possible project scenarios (such as the one in Reference 12). The total unit cost would be \$3,120/kg if LLOX were used for projected demand with $b \geq 0.4$, as compared to \$3,400/kg for earth LOX. This would translate to a total savings of \$180 M, or 8.2%. A greater demand would result in a larger percentage savings.

7.1.3 Lunar Base

A manned lunar base provides the most likely use for LLOX. If the demand exceeds 35 MT/yr, LLOX becomes cost effective. Thirty-five MT corresponds to the life support demands of a crew of greater than 23. If LLOX was implemented to deliver the mass of the lunar base, even greater savings would be realized. For later stages of lunar base development, LLOX almost becomes a necessity.

7.2 Recommendations for Future Study

7.2.1 Refinements of Model

7.2.1.1 Nuclear Power

SP-100 is designed primarily for space flight missions. Significant increases would probably result from a nuclear plant designed specifically for the lunar surface in the megawatt range. Such a plant would serve as a prototype for future colonization power sources.

7.2.1.2 Lunar Lander

The structural mass of the lunar lander was calculated assuming an OTV derived scaling law. A complete design should be done to assure that these numbers are reasonable. Such an analysis should also investigate additional complexities such as propellant transfer, adaptability to cargo delivery, and serviceability on the lunar surface.

7.2.2 Synergism with other Products

Lunar complexes could arise that are capable of manufacturing many products useful for space exploration and colonization. Many products would compliment each other. The proper strategy is to study the benefits of multi-product systems in order to plan the proper complement of products.

7.2.2.1 By-Products

In the production of the LLOX, many by-products are possible. Metals such as Fe, Al, Si, and Ti could be used for lunar construction. Even the slag could be useful as shielding or pavement.

7.2.2.2 Hydrogen

Hydrogen is most useful when used in conjunction with a LLOX plant. Not only can it be used a fuel, but it can also be combined with LLOX to form water, an essential part of the life support resupply demand. There are several possible sources of hydrogen on the lunar surface.

One source is solar protons. These could be collected directly or separated from the lunar soil where they collect. The amount that could be collected would be a function of latitude and phase [9, pg. 8]. Measurements would be required to determine the efficiency of such a method.

Other possible sources include water, hydrocarbons, and volcanic gases [9, pg. 8]. The ability to separate hydrogen from these compounds is as well understood as the chemical processes involved in the LLOX production, but the availability of these compounds on the lunar surface is not known very accurately.

7.2.1.3 Helium-3

Fusion shall continue to be the goal in power sources. Not only could significant savings result from a lunar fusion facility, financial opportunities for mining helium-3 could arise for terrestrial fusion facilities. Helium from solar winds has been deposited in the lunar surface. Helium exists at about 30 ppm in the lunar soil. Helium-3 occurs at about 300 ppm of this helium [12, pg. 477]. Therefore, helium-3 only accounts fro 9×10^{-9} of the lunar soil. However, helium-3 could be worth as much as $1 B/MT, not accounting for the many ecological benefits of fusion power.

7.2.1.4 Solar Cells

It has been proposed that large solar arrays placed in orbit could be used to transmit power to the earth and moon. Due to inefficiencies in transmitting the power, the arrays would have to be much larger than an array on the surface of the moon or the earth. Because of the difference in delivery costs, solar arrays in earth orbit may be more cost efficient than solar power on the moon.

Such a project would also create another opportunity for lunar resources. One study shows that up to 98% of the solar array material could be produced using lunar materials [5, pg. 13].

7.2.3 Mass Drivers

Mass drivers have been suggested for transporting the LLOX from the lunar surface to LLO. Capsules of LLOX would be accelerated to escape velocity on electric tracks. This would involve large startup masses, but may be more efficient than other propulsion systems. The technology is too uncertain to be included in this model, but future progress in this field should be monitored.

It has also been suggested that tethers could be used in conjunction with mass drivers to lower the velocity requirements. Tethers could be used to absorb the momentum of the "smart" capsules. This momentum could then be dissipated in the same manner that reaction wheels are despun, by using such devices as gravity gradient booms, magnetic torquers, or thrusters.

7.2.4 Funding

Probably the most difficult consideration is how to fund such a project. Although this study was designed to meet demands at the same cost, or even at a savings, a large capital investment is necessary, an investment that does not yield the returns characteristic of high risk projects such as this one. Therefore, it is assumed that such a project would have to be publicly funded, since it is not likely to be funded by private investors. Also, the best returns occur in conjunction with a lunar base, which is most likely to be a publicly funded project.

Appendix I Transportation Spreadsheets

Appendix I contains all the spreadsheets used in the transportation calculations. All the values that were used to determine d and MPR for the various delivery sites can be found in these tables. Microsoft Excel was used to perform the necessary computations and iterations.

OTV	MT				LL/OTV	2
Mi	6.94	dV1	1600	m/s	Match	1.776E-15
Mo1	136.62	dV2	4050	m/s	MPR	0.328897
Mf1	97.12	PL1	62.1330875	MT	d	0.272318
Mp1	39.50	PL2	11.4911157	MT	PR	0.182753
Mo2	55.43	Isp	478	s		
Mf2	23.37	c	4689.18	m/s	Data	
Mp2	32.06	R	8		R	Isp
					7	484
Lunar Lander	MT				7.25	482
Mi	11.81	dV1	2000	m/s	8	478
Mo1	130.64	dV2	2100	m/s	9	470
Mf1	83.21	PL1	62.37	MT	10	452
Mp1	47.43	PL2	0	MT	11	434
Mo2	26.58	Isp	452	s	12	416
Mf2	16.56	c	4434.12	m/s	13	398
Mp2	10.03	R	10			

Table A.1-1. Matched System for LEO Delivery

Lunar Lander	MT			
Mi	11.81	dV1	6150	m/s
Mo1	131.64	dV2	0	m/s
Mf1	32.89	PL1	16.33	MT
Mp1	98.75	PL2	0	MT
Mo2	16.56	Isp	452	s
Mf2	16.56	c	4434.12	m/s
Mp2	0	R	10	

Table A.1-2. Lunar Lander Used to Soft Land Equipment

OTV	MT			
Mi	6.94	dV1	2724	m/s
Mo1	268.26	dV2	2926	m/s
Mf1	150.08	PL1	131.64	MT
Mp1	118.18	PL2	0	MT
Mo2	18.44	Isp	478	s
Mf2	9.88	c	4689.18	m/s
Mp2	8.56	R	8	
Lunar Lander	MT			
Mi	11.81	dV1	3426	m/s
Mo1	131.64	dV2	0	m/s
Mf1	60.78	PL1	44.2273173	MT
Mp1	70.86	PL2	0	MT
Mo2	16.56	Isp	452	s
Mf2	16.56	c	4434.12	m/s
Mp2	0.00	R	10	

Table A.1-3. OTV Use as a Stage to Soft Land Equipment

OTV	MT				LL/OTV	2
Mi	6.94	dV1	1600	m/s	Match	0.000
Mo1	136.62	dV2	4050	m/s	MPR	0.596
Mf1	97.12	PL1	59.06	MT	d	0.256
Mp1	39.50	PL2	14.05	MT		0.104
Mo2	61.50	Isp	478	s		
Mf2	25.93	c	4689.18	m/s	Data	
Mp2	35.57	R	8		R	Isp
					7	484
Lunar Lander	MT				7.25	482
Mi	11.94	dV1	2000	m/s	8	478
Mo1	131.97	dV2	2100	m/s	9	470
Mf1	84.06	PL1	62.37	MT	10	452
Mp1	47.91	PL2	1.15	MT	11	434
Mo2	28.71	Isp	452	s	12	416
Mf2	17.88	c	4434.12	m/s	13	398
Mp2	10.83	R	10			

Table A.1-4. Matched System for LEO Delivery with Resupply

OTV	MT				LL/OTV	3
Mi	6.94	dV1	1600.00	m/s		
Mo1	107.02	dV2	4050.00	m/s		
Mf1	76.08	PL1	0.00	MT		
Mp1	30.94	PL2	46.79	MT		
Mo2	136.62	Isp	478.00	s		
Mf2	57.60	c	4689.18	m/s	Data	
Mp2	79.02	R	8.00		R	Isp
					7	484
Lunar Lander	MT				7.25	482
Mi	11.88	dV1	2000.00	m/s	8	478
Mo1	102.98	dV2	2100.00	m/s	9	470
Mf1	65.59	PL1	36.70	MT	10	452
Mp1	37.39	PL2	15.60	MT	11	434
Mo2	50.13	Isp	452.00	s	12	416
Mf2	31.22	c	4434.12	m/s	13	398
Mp2	18.91	R	10.00			

Table A.1-5. Matched System for Purely Resupply Missions

Appendix II Production Cost Spreadsheet

Appendix II is a sample spreadsheet of the the production cost model. The sample below is the breakeven scenario for delivery to the lunar surface.

Power required	P	MW	0.0918890		Production	35.7545	a	2.16E-03
Cost of power	Cp	$/MW	2.00E+08		Duty Cycle	0.9	b	2.00E+08
Number of types of lunar base modules	Nt	#	1		Capacity	39.7272	c	1.48E+01
Cost of modifying space station modules	Cn	$	3.00E+08		Mlunar	6.04720	d	3.04E-01
Number of lunar base modules	Nm	#	0.0357544		Mresupply	0.35754	e	6.67E-01
Unit cost of lunar base mondules	Cu	$	2.00E+08				f	0
Processing/storage facility cost	Cf	$	7.62E+07					
Power system mass	Mp	MT/MW	1.48E+01					
Earth-to-Moon transportation cost	Ct	$/MT	3.35E+07					
Mass of lunar base modules	Mm	MT	2.00E+01					
Mass of processing/storage facility	Mf	MT	3.97E+00					
Number of lunar base resupply missions/year	Nr	#	0.0178772					
Net lunar oxygen delivered to LEO	D	#	0.947					
Ground support manpower	Nb	#	0.7150896					
Ground support overhead factor	Nf	#	25					
Cost of Design	Cd		3.08E+09					
Number of OTV	Notv	#	2.55E-04					
Cost of OTV	Cotv	$	1.01E+09					
Number of LL	Nll	#	0.00E+00					
Cost of LL	Cll	$	1.01E+09					
Number of Flights	Nflight	#	30					
Number of Years	n		10					
Discount Rate	r		1.00E-08					
Equivalent Number of Years	x		9.9999993					
Capital cost			3.25E+09					
Operations cost			2.04E+06					
Amoritorized Cost ($/kg)			9412.12					
Cearth			10000.00					
PR			0.9412					

Demand(MT/yr)	
LEO	
LLO	
Moon Base	
Production Rate	3.58E+01
Enter Duty Cycle	0.9

Process	
1. Hydrogen Reduction	
2. Carbothermal Reduction	
3. Magma Electrolysis	
4. Fused Salt Electrolysis	
5. Molten Silicate Electrolysis	
6. Vapor Pyrolosis	
Enter Process Number	2

Power	
1. Solar	
2. Nuclear	
Enter Power Type	2

Appendix III Data Used for Figures

Appendix III lists all the data used for the figures in the text. Table A.3-1 is a key that describes the data is stored in each column. The first column lists the sampled production rates (MT/yr). Explanation of numbers under process and power can be found in the production cost model in Appendix II.

Column	Process	Power	Site	Duty Cycle	r	Special
A	2	2	LS	0.9	0	
B	2	2	LLO	0.9	0	
C	2	2	LEO	0.9	0	
D	2	2	LS	0.9	0.05	
E	2	2	LS	0.9	0.1	
F	2	2	LS	0.9	0.15	
G	2	1	LEO	0.5	0	
H	2	2	LEO	0.9	0	O_2 Poor
I	2	1	LEO	0.9	0	Continual Sun
J	2	1	LEO	0.6	0	
K	2	1	LEO	0.7	0	
L	2	1	LEO	0.8	0	
M	2	1	LEO	0.9	0	
N	2	1	LEO	0.5	0	No Fuel Cells
O	2	2	LEO	0.9	0	Curve Fit (τ=400)
P	6	2	LEO	0.9	0	
Q	6	1	LEO	0.5	0	No Fuel Cells

Table A.3-1. Key to Appendix III

	A	B	C	D	E	F	G	H	I	J
100	0.381	1.155	1.161	0.492	0.616	0.753	1.269	1.163	1.135	1.350
200	0.222	0.696	0.692	0.285	0.357	0.436	0.789	0.695	0.676	0.874
300	0.167	0.541	0.536	0.215	0.268	0.327	0.624	0.537	0.521	0.711
400	0.140	0.463	0.458	0.179	0.223	0.272	0.539	0.459	0.443	0.627
500	0.123	0.416	0.410	0.157	0.196	0.239	0.487	0.411	0.396	0.576
600	0.111	0.384	0.378	0.142	0.177	0.216	0.452	0.379	0.364	0.541
700	0.103	0.361	0.355	0.132	0.164	0.199	0.426	0.357	0.341	0.516
800	0.097	0.343	0.338	0.124	0.154	0.187	0.406	0.339	0.323	0.497
900	0.092	0.330	0.324	0.117	0.146	0.177	0.391	0.326	0.310	0.482
1000	0.088	0.319	0.313	0.112	0.139	0.169	0.378	0.315	0.299	0.469
1100	0.084	0.310	0.304	0.108	0.134	0.162	0.367	0.306	0.290	0.459
1200	0.082	0.302	0.297	0.104	0.129	0.157	0.359	0.298	0.282	0.451
1300	0.079	0.296	0.290	0.101	0.125	0.152	0.351	0.292	0.276	0.443
1400	0.077	0.290	0.285	0.098	0.122	0.148	0.344	0.286	0.270	0.437
1500	0.075	0.285	0.280	0.096	0.119	0.144	0.338	0.281	0.265	0.431
1600	0.074	0.281	0.276	0.094	0.116	0.141	0.333	0.277	0.261	0.426
1700	0.072	0.277	0.272	0.092	0.114	0.138	0.329	0.273	0.257	0.422
1800	0.071	0.274	0.269	0.090	0.112	0.136	0.325	0.270	0.254	0.418
1900	0.070	0.271	0.266	0.089	0.110	0.134	0.321	0.267	0.251	0.414
2000	0.069	0.268	0.263	0.088	0.109	0.132	0.317	0.264	0.248	0.411
2100	0.068	0.266	0.260	0.086	0.107	0.130	0.314	0.261	0.246	0.408
2200	0.067	0.263	0.258	0.085	0.106	0.128	0.312	0.259	0.243	0.405
2300	0.066	0.261	0.256	0.084	0.104	0.126	0.309	0.257	0.241	0.402
2400	0.066	0.259	0.254	0.083	0.103	0.125	0.306	0.255	0.239	0.400
2500	0.065	0.258	0.252	0.082	0.102	0.124	0.304	0.253	0.238	0.398
2600	0.064	0.256	0.251	0.082	0.101	0.122	0.302	0.252	0.236	0.396
2700	0.064	0.254	0.249	0.081	0.100	0.121	0.300	0.250	0.234	0.394
2800	0.063	0.253	0.248	0.080	0.099	0.120	0.298	0.249	0.233	0.392
2900	0.063	0.252	0.246	0.079	0.098	0.119	0.297	0.247	0.232	0.391
3000	0.062	0.250	0.245	0.079	0.097	0.118	0.295	0.246	0.230	0.389
3100	0.062	0.249	0.244	0.078	0.097	0.117	0.293	0.245	0.229	0.388
3200	0.061	0.248	0.243	0.078	0.096	0.116	0.292	0.244	0.228	0.386
3300	0.061	0.247	0.242	0.077	0.095	0.115	0.291	0.243	0.227	0.385
3400	0.060	0.246	0.241	0.077	0.095	0.115	0.289	0.242	0.226	0.384
3500	0.060	0.245	0.240	0.076	0.094	0.114	0.288	0.241	0.225	0.382
3600	0.060	0.244	0.239	0.076	0.094	0.113	0.287	0.240	0.224	0.381
3700	0.059	0.243	0.238	0.075	0.093	0.113	0.286	0.239	0.223	0.380
3800	0.059	0.243	0.237	0.075	0.092	0.112	0.285	0.238	0.223	0.379
3900	0.059	0.242	0.236	0.074	0.092	0.111	0.284	0.238	0.222	0.378
4000	0.058	0.241	0.236	0.074	0.091	0.111	0.283	0.237	0.221	0.377
4100	0.058	0.240	0.235	0.074	0.091	0.110	0.282	0.236	0.220	0.376
4200	0.058	0.240	0.234	0.073	0.091	0.110	0.281	0.236	0.220	0.376

	A	B	C	D	E	F	G	H	I	J
4300	0.058	0.239	0.234	0.073	0.090	0.109	0.280	0.235	0.219	0.375
4400	0.057	0.238	0.233	0.073	0.090	0.109	0.279	0.234	0.219	0.374
4500	0.057	0.238	0.232	0.072	0.089	0.108	0.278	0.234	0.218	0.373
4600	0.057	0.237	0.232	0.072	0.089	0.108	0.277	0.233	0.217	0.372
4700	0.057	0.237	0.231	0.072	0.089	0.107	0.277	0.233	0.217	0.372
4800	0.057	0.236	0.231	0.072	0.088	0.107	0.276	0.232	0.216	0.371
4900	0.056	0.236	0.230	0.071	0.088	0.106	0.275	0.232	0.216	0.370
5000	0.056	0.235	0.230	0.071	0.088	0.106	0.275	0.231	0.215	0.370
5100	0.056	0.235	0.229	0.071	0.087	0.106	0.274	0.231	0.215	0.369
5200	0.056	0.234	0.229	0.071	0.087	0.105	0.273	0.230	0.214	0.369
5300	0.056	0.234	0.229	0.070	0.087	0.105	0.273	0.230	0.214	0.368
5400	0.055	0.233	0.228	0.070	0.087	0.105	0.272	0.229	0.214	0.367
5500	0.055	0.233	0.228	0.070	0.086	0.104	0.272	0.229	0.213	0.367
5600	0.055	0.233	0.227	0.070	0.086	0.104	0.271	0.229	0.213	0.366
5700	0.055	0.232	0.227	0.069	0.086	0.104	0.271	0.228	0.212	0.366
5800	0.055	0.232	0.227	0.069	0.086	0.103	0.270	0.228	0.212	0.365
5900	0.055	0.232	0.226	0.069	0.085	0.103	0.270	0.227	0.212	0.365
6000	0.055	0.231	0.226	0.069	0.085	0.103	0.269	0.227	0.211	0.364
6100	0.054	0.231	0.226	0.069	0.085	0.103	0.269	0.227	0.211	0.364
6200	0.054	0.231	0.225	0.069	0.085	0.102	0.268	0.226	0.211	0.364
6300	0.054	0.230	0.225	0.068	0.084	0.102	0.268	0.226	0.210	0.363
6400	0.054	0.230	0.225	0.068	0.084	0.102	0.267	0.226	0.210	0.363
6500	0.054	0.230	0.224	0.068	0.084	0.102	0.267	0.226	0.210	0.362
6600	0.054	0.229	0.224	0.068	0.084	0.101	0.266	0.225	0.210	0.362
6700	0.054	0.229	0.224	0.068	0.084	0.101	0.266	0.225	0.209	0.362
6800	0.054	0.229	0.223	0.068	0.083	0.101	0.266	0.225	0.209	0.361
6900	0.053	0.229	0.223	0.067	0.083	0.101	0.265	0.224	0.209	0.361
7000	0.053	0.228	0.223	0.067	0.083	0.100	0.265	0.224	0.208	0.360
8000	0.052	0.226	0.221	0.066	0.082	0.099	0.261	0.222	0.206	0.357
9000	0.052	0.224	0.219	0.065	0.080	0.097	0.259	0.220	0.204	0.355
10000	0.051	0.223	0.217	0.064	0.079	0.096	0.256	0.219	0.203	0.353
11000	0.050	0.222	0.216	0.064	0.078	0.095	0.255	0.217	0.202	0.351
12000	0.050	0.220	0.215	0.063	0.078	0.094	0.253	0.216	0.201	0.349
13000	0.050	0.220	0.214	0.063	0.077	0.093	0.251	0.215	0.200	0.348
14000	0.049	0.219	0.213	0.062	0.076	0.092	0.250	0.215	0.199	0.347
15000	0.049	0.218	0.213	0.062	0.076	0.092	0.249	0.214	0.198	0.346
16000	0.049	0.217	0.212	0.061	0.076	0.091	0.248	0.213	0.198	0.345
17000	0.048	0.217	0.212	0.061	0.075	0.091	0.247	0.213	0.197	0.344
18000	0.048	0.216	0.211	0.061	0.075	0.090	0.246	0.212	0.197	0.343
19000	0.048	0.216	0.211	0.060	0.074	0.090	0.245	0.212	0.196	0.342
20000	0.048	0.216	0.210	0.060	0.074	0.089	0.245	0.211	0.196	0.342

	K	L	M	N	O	P	Q
100	1.409	1.454	1.490	1.220	11.466	1.381	1.220
200	0.936	0.983	1.021	0.740	1.552	0.935	0.774
300	0.774	0.823	0.861	0.575	0.797	0.786	0.625
400	0.692	0.741	0.780	0.490	0.571	0.712	0.551
500	0.641	0.691	0.730	0.438	0.467	0.668	0.506
600	0.607	0.657	0.697	0.402	0.409	0.638	0.477
700	0.582	0.632	0.672	0.377	0.372	0.617	0.455
800	0.563	0.614	0.654	0.357	0.346	0.601	0.440
900	0.548	0.599	0.639	0.343	0.328	0.588	0.427
1000	0.536	0.587	0.628	0.332	0.313	0.578	0.417
1100	0.526	0.577	0.618	0.322	0.302	0.570	0.409
1200	0.517	0.569	0.610	0.314	0.293	0.564	0.402
1300	0.510	0.562	0.603	0.307	0.286	0.558	0.397
1400	0.504	0.555	0.597	0.301	0.279	0.553	0.392
1500	0.498	0.550	0.591	0.296	0.274	0.549	0.388
1600	0.494	0.545	0.587	0.292	0.270	0.545	0.384
1700	0.489	0.541	0.582	0.288	0.266	0.542	0.381
1800	0.485	0.537	0.579	0.284	0.262	0.539	0.378
1900	0.482	0.534	0.575	0.281	0.259	0.536	0.375
2000	0.479	0.531	0.572	0.278	0.256	0.534	0.373
2100	0.476	0.528	0.569	0.275	0.254	0.532	0.371
2200	0.473	0.525	0.567	0.273	0.252	0.530	0.369
2300	0.471	0.523	0.565	0.271	0.250	0.528	0.367
2400	0.468	0.521	0.562	0.269	0.248	0.526	0.365
2500	0.466	0.519	0.560	0.267	0.246	0.525	0.364
2600	0.464	0.517	0.559	0.265	0.245	0.524	0.362
2700	0.462	0.515	0.557	0.263	0.244	0.522	0.361
2800	0.461	0.513	0.555	0.262	0.242	0.521	0.360
2900	0.459	0.512	0.554	0.260	0.241	0.520	0.359
3000	0.458	0.510	0.552	0.259	0.240	0.519	0.358
3100	0.456	0.509	0.551	0.257	0.239	0.518	0.357
3200	0.455	0.508	0.550	0.256	0.238	0.517	0.356
3300	0.454	0.506	0.548	0.255	0.237	0.516	0.355
3400	0.452	0.505	0.547	0.254	0.236	0.515	0.354
3500	0.451	0.504	0.546	0.253	0.235	0.515	0.354
3600	0.450	0.503	0.545	0.252	0.235	0.514	0.353
3700	0.449	0.502	0.544	0.251	0.234	0.513	0.352
3800	0.448	0.501	0.543	0.250	0.233	0.513	0.352
3900	0.447	0.500	0.542	0.249	0.233	0.512	0.351
4000	0.446	0.499	0.541	0.249	0.232	0.512	0.350
4100	0.445	0.498	0.541	0.248	0.232	0.511	0.350
4200	0.445	0.498	0.540	0.247	0.231	0.510	0.349

	K	L	M	N	O	P	Q
4300	0.444	0.497	0.539	0.246	0.230	0.510	0.349
4400	0.443	0.496	0.538	0.246	0.230	0.510	0.348
4500	0.442	0.495	0.538	0.245	0.230	0.509	0.348
4600	0.442	0.495	0.537	0.244	0.229	0.509	0.347
4700	0.441	0.494	0.536	0.244	0.229	0.508	0.347
4800	0.440	0.493	0.536	0.243	0.228	0.508	0.347
4900	0.440	0.493	0.535	0.243	0.228	0.507	0.346
5000	0.439	0.492	0.535	0.242	0.227	0.507	0.346
5100	0.438	0.492	0.534	0.242	0.227	0.507	0.346
5200	0.438	0.491	0.534	0.241	0.227	0.506	0.345
5300	0.437	0.491	0.533	0.241	0.226	0.506	0.345
5400	0.437	0.490	0.533	0.240	0.226	0.506	0.345
5500	0.436	0.490	0.532	0.240	0.226	0.505	0.344
5600	0.436	0.489	0.532	0.239	0.226	0.505	0.344
5700	0.435	0.489	0.531	0.239	0.225	0.505	0.344
5800	0.435	0.488	0.531	0.238	0.225	0.505	0.343
5900	0.434	0.488	0.530	0.238	0.225	0.504	0.343
6000	0.434	0.487	0.530	0.238	0.224	0.504	0.343
6100	0.434	0.487	0.529	0.237	0.224	0.504	0.343
6200	0.433	0.486	0.529	0.237	0.224	0.504	0.342
6300	0.433	0.486	0.529	0.237	0.224	0.503	0.342
6400	0.432	0.486	0.528	0.236	0.224	0.503	0.342
6500	0.432	0.485	0.528	0.236	0.223	0.503	0.342
6600	0.432	0.485	0.528	0.236	0.223	0.503	0.342
6700	0.431	0.485	0.527	0.235	0.223	0.503	0.341
6800	0.431	0.484	0.527	0.235	0.223	0.502	0.341
6900	0.430	0.484	0.527	0.235	0.223	0.502	0.341
7000	0.430	0.484	0.526	0.234	0.222	0.502	0.341
8000	0.427	0.481	0.524	0.232	0.221	0.500	0.339
9000	0.425	0.478	0.521	0.230	0.220	0.499	0.338
10000	0.423	0.476	0.519	0.228	0.219	0.498	0.337
11000	0.421	0.475	0.518	0.227	0.218	0.497	0.336
12000	0.420	0.473	0.517	0.225	0.217	0.497	0.336
13000	0.418	0.472	0.515	0.224	0.217	0.496	0.335
14000	0.417	0.471	0.514	0.223	0.216	0.496	0.334
15000	0.416	0.470	0.513	0.222	0.216	0.495	0.334
16000	0.415	0.469	0.513	0.222	0.215	0.495	0.334
17000	0.414	0.469	0.512	0.221	0.215	0.494	0.333
18000	0.414	0.468	0.511	0.220	0.215	0.494	0.333
19000	0.413	0.467	0.511	0.220	0.214	0.494	0.333
20000	0.412	0.467	0.510	0.219	0.214	0.494	0.333

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