NASA CR-172051

Lunar Lander Conceptual Design

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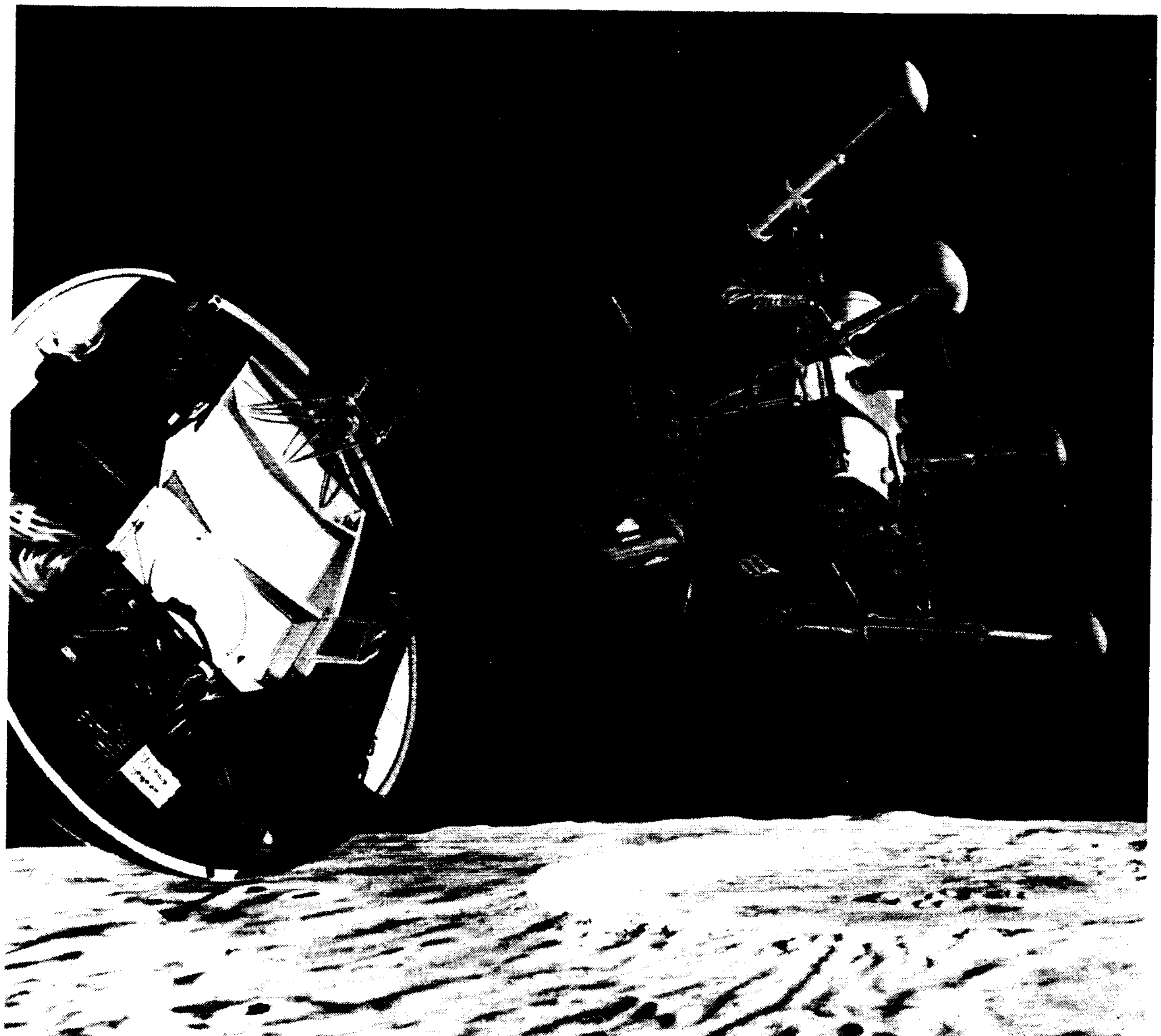

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Lunar Lander Conceptual Design

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National Aeronautics and Space Administration
Lyndon B. Johnson Space Center
Advanced Projects Office

Lunar Base Systems Study Task 2.2

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Foreword

This report was prepared during Dec. - March, 1987/88. It is an attempt to start the process of designing a new lunar lander.

Dr. John Alred was the NASA JSC technical monitor for this contract. The NASA task manager was Ms. Jonette Stecklein. Mr. Andy Petro provided valuable technical advice.

Mr. W.B. Evans was the Eagle Project Manager for the ASTS contract. Mr. Bill Stump was the Eagle Task Manager for this study. Other participants included Dr. Alex Adorjan, Mr. Tom Chambers, Mr. Mike D'Onofrio, Mr. John Hirasaki, Mr. Owen Morris, Mr. Greg Nudd, Mr. Pat Rawlings, Mr. Chris Varner, Mr. Charley Yodzis, and Mr. Scott Zimprich.

LUNAR LANDER CONCEPTUAL DESIGN

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1.0 Executive Summary

This study is a first look at the problem of building a lunar lander to support a small lunar surface base. One lander, which can land 25 metric tons, one way, or take a 6 metric ton crew capsule up and down is desired. The initial idea was to build a re-usable lander, suitable for minimizing the transportation cost to a permanent base, and use it from the first manned mission on, taking some penalty and perhaps expending expensive vehicles early in the program in order to avoid building multiple types of landers and focusing the effort on a space maintainable, single-stage, reusable vehicle. Given a long term, permanent base to support, and the general conclusion that it is possible to build such a vehicle, advocates of other lander approaches must show this approach will not work.

A single stage lander is feasible from low lunar orbit. Initial calculations do not show large weight penalties (15-30%) over two-stage vehicles. A lander capable of multiple roles, such as landing cargo one way or taking crew modules round trip is possible with some penalty (5 to 10%) over dedicated designs. The size of payload delivered to lunar orbit may vary by a factor of two however.

A single type of engine usable for several different size landers appears to be possible. Different size landers and radically different payloads may require multiple trips with the OTV delivery vehicle(s) and storage of the first payload in lunar orbit, or a performance penalty due to additional tankage mass carried for small payload missions.

A four engine design for a multi-purpose vehicle, with total thrust in the range of 35-40,000 lbf (12 to 13,000 lbf per engine) and a throttling ratio in the 13:1 to 20:1 range is proposed. Initial work indicates a regeneratively cooled, pump-fed engine will be required due to difficulties with regenerative cooling over wide throttling ranges with pressure-fed systems. The engine is the single most important technical development item. Reuse and space maintainability requirements make it near or beyond the current state of the art. Study and simulation work should continue until this engine is defined well enough such that long lead development can start.

Initial calculations indicate low lunar orbit offers the lowest LEO stack mass. Low altitude lunar orbits are unstable for long periods of time. The instability limit may set the parking orbit altitude.

LEO basing for the lander appears possible, with some penalty in LEO stack mass (10-25%) over a scheme that bases the lander in low lunar orbit (LLO) or expends it. The lander will require a special OTV to aerobrake it into LEO however. Loading all propellants from Earth on the lunar surface does not appear to be practical because of the additional propellant needed to land this propellant on the lunar surface. An additional mission is needed.

The lander must be designed from the start for ease of maintenance, and simplicity. Design features, such as special pressurized volumes will be needed to make the vehicle maintainable in space. Space maintainability and reusability must be made a priority.

Liquid oxygen/liquid hydrogen propellants show the best performance, but hydrogen may be difficult to store for long periods of time in the lander on the surface. Earth storable and space storable propellants are not ruled out. Liquid hydrogen storage over a 180 day period on the lunar surface at the equator needs study. A point design of a liquid oxygen/liq. hydrogen lander needs to be done in order to have a good inert mass data point that shows the performance gain is real.

2.0 Introduction

A series of trade studies are used to narrow the choices and provide some general guidelines. Given a rough baseline, the systems are then reviewed. A conceptual design is then produced. The process has only been carried through one iteration. Many more iterations are needed.

A transportation system using reusable, aerobraked Orbital Transfer Vehicles (OTVs) is assumed. These vehicles are assumed to be based and maintained at a low Earth orbit Space Station, optimized for transportation functions. Single and two-stage OTV stacks are considered. The OTVs make the Translunar Injection (TLI), Lunar Orbit Insertion (LOI), and Trans-Earth Insertion (TEI) burns, as well as mid-course and perigee raise maneuvers.

3.0 Assumptions and Groundrules

1) The lander is assumed to be one of the key elements in a three phase return to the Moon. The first phase involves unmanned exploration with lunar orbiters and unmanned surface rovers, and perhaps sample returns. The second phase involves the return of humans to the surface and ends approximately when permanent habitation of a base begins. The third phase begins with permanent habitation. This effort focuses on the second phase, base building, and man-tended operations.

2) The study will focus on a single stage reusable lander, to be used from the first manned landing on.

3) Options for propellant loading to be considered include: a) All propellants are brought from Earth in a tank and transferred to the lander in lunar orbit. b) All propellants are landed on the lunar surface in tanks and transferred to the lander on the lunar surface. c) All propellants are loaded in low Earth orbit at the Space Station. The lander is returned to the Space Station after every flight.

4) Two design criteria will be considered to size the lander. In the first case, the cargo landing mode, the lander will use all the tankage capacity for descent to land a 25,000 kg cargo and not return to orbit. The round trip payload will then be calculated based on this tank size. In the second case, the required round trip capacity to deliver crew to and from the surface will be determined and the tanks will be sized to do this. The max. cargo capacity will then be what the tanks can land, ending up empty on the surface. When carrying the manned capsule up and down, no major payload capacity is required. 1,000 kg payload capacity up and down, in addition to the crew capsule, consumables, etc. is baselined.

5) All propellant for the landers comes from Earth in Phase II of the base. Any lunar oxygen produced on the surface will be used in test bed experiments and not in the lander/launchers.

6) The crew capsule can support a crew of four for three days. The crew capsule must be able to land a maximum crew of six and support them for a minimum of one day.

7) The crew compartment can be detached from the lander/launcher and handled as a piece of cargo if required.

8) For options in which the lander is returned to the LEO Space Station after every mission, only one crew capsule on the lander will be used for the round trip. For options in which the lander is based in low lunar orbit or on the lunar surface, two crew capsules will be used, one carried by the OTV in LLO and the other on the lander.

9) The manned reusable lander/launcher is expected to fly 3 to 6 times per year.

10) The lander/launcher must be able to sit on the lunar surface, with propellant on board for 200 days. Power will be provided on the surface during this period and some thermal protection in the form of a tentlike structure may also be available.

11) The ability to abort the mission and ascend from the lunar surface to the return stage at any time is desired.

12) The baseline landing site is Lacus Verus (87.5 W, 13 S). Other sites to be considered include the Apollo 17 site, the South Pole, and Mare Nubium.

4.0 Delta Vs

One of the most important aspects of lunar mission analysis is the determination of the velocity change (delta V) requirements. From these requirements, it is possible to do trade studies for different engines, propellants, and payloads. Since, the delta V requirements are the most basic description of the mission, they are the first step in the process of spacecraft design.

Several general locations in Earth-Moon space are of special interest. Nine will be discussed in the following paragraphs. Figure 4-1 shows the locations graphically.

A transportation node in low Earth orbit (LEO) will be used as a base for the stacking and maintenance of lunar spacecraft. The Space Station is expected to be in a circular Low Earth Orbit (LEO) which has an inclination of 28.5°. The orbital altitude is assumed to be 463 kilometers (km) (250 nautical miles (nm)).

Geosynchronous Orbit (GEO) is 35,780 km above the surface of the Earth. All satellites in equatorial GEO have the unique ability to remain stationary over a fixed point on the equator.

The Earth-Moon Lagrangian point is the point between the Earth and Moon where the Earth's gravitational force is exactly offset by the Moon's gravitational force. This point, hereafter referred to as "L2" and located on the Earth-Moon line, is 320,000 km from the surface of the Earth, and 56,600 km from the Moon's surface on the average.

The equilateral Lagrangian points known as "L4" and "L5" are in the same orbit as the Moon and are approximately 384,400 km from the Earth and the Moon. The L4 and L5 points are theoretically stable; objects placed at these points will not move away from the point, and if displaced will tend to return.

The Low Lunar Orbit (LLO) altitude is defined to range between 93 km (50 nm) and 111 km (65 nm). These were typical altitudes for Apollo lunar orbit rendezvous.

The Lunar Far Side Lagrangian point (L1) is the point at which the centrifugal force of travelling around the Earth at the Moon's orbital speed is exactly offset by the Moon's and Earth's gravitational force. L1 is an unstable Lagrangian point which is located beyond the Moon on the Earth-Moon line, 62,700 km above its surface, or 442,500 km from the surface of the Earth.

Earth-Moon Escape occurs when a spacecraft gains enough energy to travel away from the Earth and the Moon on a hyperbolic orbit. Earth-Moon escape is necessary for any interplanetary flights.

In Table 4-1 these nine locations have been arranged to form a delta V Chart. The numbers in the delta V chart have units of <km/sec>. To travel from any location along the left hand side of the chart to any location listed at the top of the chart requires a velocity change equal to the number located at the intersection of the two (read horizontally across from the "FROM" location and vertically down from the "TO" location).

The delta V chart assumes that reentry or aerobraking is possible when travelling to the Earth's surface or to LEO, but aerobraking to any other location is less efficient than a Hohmann impulsive burn. If reentry to the Earth's surface or aerobraking to LEO is not desired then read the chart in reverse, as if the "FROM" location was the "TO" location and visa-versa.

LEO is assumed to be the only location accessible from the Earth's surface, LLO is assumed to be the only location accessible from the Moon's surface, and the Moon's surface is not accessible from any location except LLO. Launches direct from the surface to high orbits and descents from high orbits to the surface are possible, but difficult, and will not result in total delta Vs significantly different from the cumulative delta Vs of ascent/descent to/from low orbit and a Hohmann transfer. In addition, landers will almost certainly be constrained to land at a specific point on the lunar surface, and LLO can provide the best trajectory accuracy required for a "pin-point" surface landing.

The delta V required to get from the Earth's surface to LEO is affected by launch site location, insertion orbit inclination, and launch vehicle configuration and performance. 9.1 km/sec represents the median delta V for the Saturn V, the Titan, and the Conestoga launch vehicles. It may vary by as much as 1 km/sec.

This chart assumes Hohmann orbit transfers and does not take into account the effects of plane changes or flight time limitations. The values listed are minimums and represent optimum flight paths.

Table 4-1, Delta Vs for Earth-Moon Flight

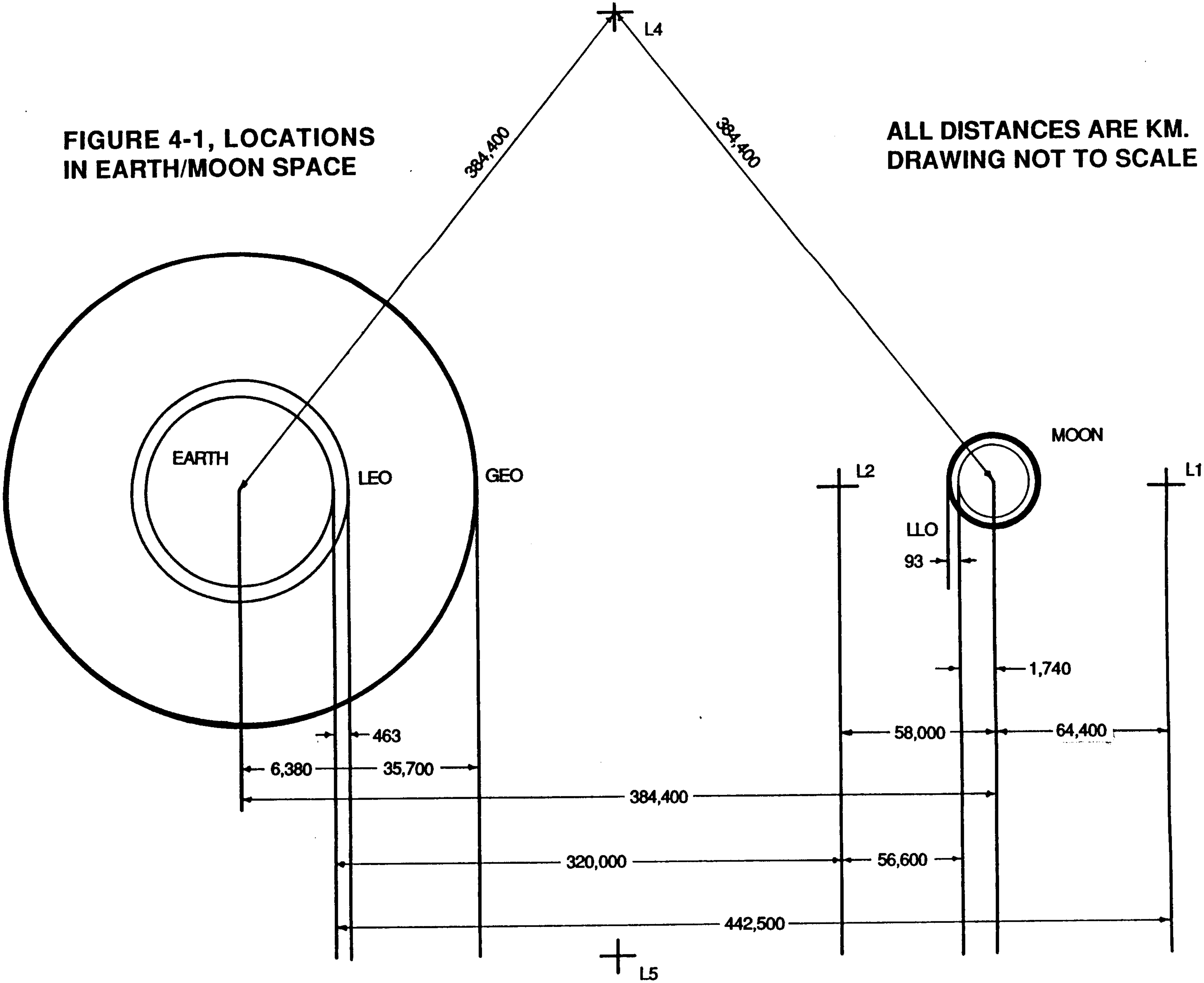
From \ To	Earth's Surface	LEO Station (463km)	GEO (35,780km)	L2-middle (320,000km)	L4, L5 (378,000km)	LLO (111km)	Lunar Surface	L1-Moon (442,500km)	Earth-Moon Escape
Earth's Surface	*	9.1	-	-	-	-	-	-	-
LEO Station (463km)	0.2_{AB}	*	3.8	3.9	3.9	4.1	-	3.9	3.2
GEO (35,780km)	1.5_{AB}	1.6_{AB}	*	1.6	1.6	1.9	-	1.6	1.3
L2-middle (320,000km)	0.9_{AB}	1.0_{AB}	1.6	*	0.3	0.7	-	0.2	0.5
L4, L5 (378,000km)	0.9_{AB}	1.0_{AB}	1.6	0.3	*	0.7	-	0.3	0.4
LLO (111km)	0.9_{AB}	1.0_{AB}	1.9	0.7	0.7	*	2.1	0.8	0.7
Lunar Surface	-	-	-	-	-	1.9	*	-	-
L1-Moon (442,500km)	0.9_{AB}	1.0_{AB}	1.6	0.2	0.3	0.8	-	*	0.5
Earth-Moon Escape	0_{AB}	0.1_{AB}	1.3	0.5	0.4	0.7	-	0.5	*

All the numbers in the hatched area assume an aerobraking maneuver

Units are in km/s unless otherwise specified

FIGURE 4-1, LOCATIONS IN EARTH/MOON SPACE

ALL DISTANCES ARE KM.
DRAWING NOT TO SCALE



5.0 Mission Multipliers for Earth - Moon Flight

Table 5-1 shows the ratios between masses at various stages in the transportation system, assuming a given type of system. These ratios provide an approximate means for estimating the effect of additional mass at one point in the transportation system on the mass at another point. Table 5-1 shows these ratios based on several different transportation systems using 450 sec. Isp OTVs and landers with Earth aerobraking to return the stages. Staging is accounted for.

Table 5-2 shows the mass ratios based on delta Vs from Table 4-1. These ratios are exact, but require single stage through the series of burns specified, which becomes increasingly unrealistic as more and more burns are summed up.

Adding mass to hardware that travels round trip, such as a crew module, will have considerable impact in terms of the added propellant and stage mass required, since the module must be boosted through five phases. Thus the cumulative impact of adding mass to the crew module is much greater than increasing the mass of other elements which travel through fewer phases.

Table 5-2 uses the rocket equation. The velocity change requirements for each phase, and the specific impulse properties of each engine/propellant combination are used to calculate the ratios of the initial mass over the mass at burnout.

$$\frac{M_i}{Mbo} = e\,\{\text{Delta V}/(\text{Isp}^* g_e)\}$$

where Mbo = Mass at burnout
M_i = Initial mass (at the start of the burn)
Delta V = Change in velocity
I_{sp} = Specific impulse
g_e = Gravitational constant

The ratio of the initial mass over the mass at burnout provides a measure of the additional propellant and stage mass required to boost additional mass across each phase.

Phase	Description	Delta V
1	Earth Surface to Low Earth Orbit (Space Station)	9.1
2	Low Earth Orbit (Space Station) to Low Lunar Orbit	4.1
3	Low Lunar Orbit to Lunar Surface	2.1
4	Lunar Surface to Low Lunar Orbit	1.9
5	Low Lunar Orbit to Low Earth Orbit	1.0*

* assumes aerobraking

Mass multipliers are dimensionless, so the units of the added mass become the units of the added propellant/stage mass.

Table 5-1, Mission Multipliers (Ratios based on vehicle conceptual designs)

Each of the numbers below relate the mass at one point in the flight to another point. For example (*), an extra ton carried back up to LLO after landing, increases the initial stack mass in LEO by approx. 11 tons. These ratios hold only for small weight increases.

Each of the three cases uses aerobraked OTVs and all LO2/LH2 propulsion.

Flight Phase	2 Stage OTV, Reuse. Lander returned to Space Station***	2 Stage OTV, Expended Single Stage Lander**	Single Stage OTV, Expended Single Stage Lander**
LEO-LLO (LEO mass/post LOI mass)	2.7	2.6	2.4
LLO-LS (Lander in LLO/on surf.)	1.6	1.6	1.6
LS-LLO (Lander on surf./in LLO)	1.5	--	--
LLO-LEO (LLO stack/returned mass)	1.3	1.3	1.3
LEO-LS (LEO mass/landed mass)	7.0	5.2	5.7
LEO-LS-LLO * (LEO mass/lander in LLO)	11.0	--	--
LEO-LEO (LEO mass/returned mass)	6.0	24.6	12.5
LEO Stack mass, metric tons****	127.0	172.0	188.0

** Lands 25 m tons on lunar surface

*** 6 m ton crew module payload carried from LEO to lunar surface and back.

**** This is the initial stack mass in LEO of the vehicles used in these calculations. It is included for gross comparison purposes.

Table 5-2, Mass Ratios (based on the rocket equation, exp(delta V/g*Isp))

Ratios = mass before burn/mass after burn.
A single stage is assumed for all the flight phases included, with no staging. For example (*), an extra ton carried to the lunar surface by a single stage vehicle would add approx. 4 tons to the LEO mass.

		---Specific Impulse (Isp, sec.)---			
Flight Phase/Burn	Delta V km/sec	330 (storable)	350 (LO_2/ hydrocarbon)	455 (LO_2/LH_2)	480 (LO_2/LH_2) High Perf.
ES-LEO	9.1	16.7	14.2	----	----
TLI	3.2	2.7	2.5	2.1	2.0
LOI	0.9	1.3	1.3	1.2	1.2
LLO-LS (Descent)	2.1	1.9	1.8	1.6	1.6
LS-LLO (Ascent)	1.9	1.8	1.7	1.5	1.5
LLO-LEO (TEI)	1.0	1.4	1.3	1.3	1.2
LEO-LLO (TLI & LOI)	4.1	3.6	3.3	2.5	2.4
LEO-LS (TLI,LOI, and descent)	6.2	6.8	6.1	4.0 *	3.7
LEO-LS-LLO (TLI,LOI, descent and ascent)	8.1	12.2	10.6	6.2	5.6
LEO-LEO (TLI,LOI,des. & ascent, TEI)	9.1	16.7	14.2	7.7	6.9
LEO-LEO (TLI,LOI,TEI)	5.1	4.8	4.4	3.1	3.0

6.0 Trade Studies and other Design Issues

A series of trades studies must be performed to further define the lander. The initial trades concern choosing number of stages, payload mass, parking orbit altitude, and propellant type. To do these trades requires a set of equations relating the quantities of interest. These equations are well known. The problems come in defining the inert masses of the vehicles and their relationships to propellant types. Section 6.1 addresses this problem.

Other important trades and issues include plane change capability, propellant loading and maintenance location, and reusability considerations.

6.1 Scaling Equations

A set of equations are defined in this section to scale the lander such that it matches the Apollo Lunar Module (LM) at one point and accounts for different payloads and propellants in the inert mass as well as the propellant mass.

List of Variables:
(all masses are in kilograms unless otherwise specified)

Db	=	Bulk Density of the Propellant <kg/m^3>
Df	=	Density of the Fuel <kg/m^3>
Do	=	Density of the Oxidizer <kg/m^3>
Go	=	Gravity at the Surface of the Earth (0.0098 <km/s^2>)
Isp	=	Propellant Specific Impulse <s>
Mboa	=	Ascent Burnout Mass
Mbo	=	Descent Burnout Mass
Mc	=	Invariant (constant) Mass
Me	=	Engine System Mass
Mf	=	Total propellant for flight performance reserve
Mi	=	Inert Mass
Mg	=	Gross Mass
Mgu	=	Gross (unloaded) Mass
Mla	=	Mass of Ascended Load (Ascent Payload)
Mld	=	Mass of Load Down (Payload Descended) -- includes Ascent Payload
Mn	=	Total unusable propellant
Mp	=	Descent Propellant Mass (total)
Mpa	=	Ascent Propellant Mass
Mpd	=	Descent Propellant Mass
Mpn	=	Mass of unusable propellant
Mpf	=	Mass of flight perf. reserve propellant
Mps	=	Mass of the usable propellant
Mr	=	Mass of Reaction Control System (RCS) -- excludes Propellant
Mrp	=	Mass of RCS Propellant
Mrpn	=	Mass of Unusable RCS Propellant
Mrps	=	Mass of Usable RCS Propellant
Mrpf	=	Mass of Flight Performance Reserve RCS Propellant
Ms	=	Structural Mass
Mtps	=	Mass of the Thermal Protection System
Mt	=	Propellant Tank System Mass
ΔVd	=	Velocity Change Required for Descent <km/s>
ΔVa	=	Velocity Change Required for Ascent <km/s>

Lunkhod, Surveyor, and the Apollo Lunar Module (LM) have all soft-landed on the Moon. This study is focusing on a lunar lander sized larger than the LM. The LM therefore provides the best historical data point from which scaling equations can be formulated.

The last LM to land on the moon was LM 11, which flew on Apollo 17. A high level mass breakdown of LM 11 is shown in Table 6-1. Appendix A shows a more detailed mass breakdowns. All masses are in units of kilograms <kg>. The mass data was obtained from reference 1.

Table 6-1, Apollo Lunar Module 11 (Apollo 17 Mission) Weight Statement

	Ascent Stage, kg	Descent Stage, kg
Structure:	459	471
Engine (Main) Systems:	106	224
RCS (Dry) System:	119	0
Docking/Landing System:	23	220
Thermal Protection:	170	179
Tank Systems:	108	268
Power, Control, & Data:	516	390
Propellant Usable:	2,232	8,260
Propellant Unusable:	121	566
RCS Propellant Usable:	231	0
RCS Propellant Unusable:	56	0
Environmental Systems:	288	195
Gov't. Furnished Equipment:	284	480
Other Liquids and Gasses:	60	235
Explosive Equipment	12	12
Total Mass:	4,785	11,500

The ultimate objective is to create a scaling equation which will predict the gross mass (Mg) of the vehicle. On the highest level the gross mass is the sum of the propellant mass (Mp), the inert mass (Mi), and the payload mass (Ml -- Mld or Mla).

6.1-1 $$Mg = Mi + Mp + Ml$$

The propellant mass is a combination of the main propellant (both usable and unusable) and the Reaction Control System (RCS) propellant (usable and unusable). The inert mass is normally a function of the propellant mass. Some systems are dependent on the mass of the propellant. Other systems are considered to be independent of propellant or other vehicle mass, such as the data processing system. A simple equation, found in reference 2 describes this mathematically.

6.1-2 $$Mi = B * Mp + A$$

This equation has been used for Orbital Transfer Vehicle (OTV) trade studies, but it requires some elaboration for use with a lunar lander. On a lunar lander numerous systems are dependent on the gross mass. If equation 6.1-2 is rewritten to include systems which vary based on the gross mass, it would look like:

6.1-3 $$Mi = C * Mg + B * Mp + A$$

When it becomes necessary to compare vehicles using cryogenic propellant systems with vehicles using storable propellant systems, the equation needs even further modification. Due to the typically high volume associated with cryogenic propellants, it is expected that the tank systems and the thermal protection systems will be larger than for storable propellants of the same mass. Equation 6.1-3 does not take such effects into account.

One solution to this problem is to provide a table which relates the coefficients of propellant mass (B) to different types of systems. This means that the value of "B" would be different depending on the type of propellants being used. This solution may prove most satisfactory in the end, but no data is available for a LO_2/LH_2 lander, therefore some other solution must be used for the moment. The next step in this effort may be a low level design of a LO_2/LH_2 lander, to provide a good point for this scaling equation, if nothing else.

Another solution is to make the second term of the equation a function of the propellant Bulk Density (Db). The bulk density is the total mass of propellant divided by the total volume of propellant. The tank inert mass is inversely related to the bulk density, therefore the equation should be rewritten as:

6.1-4 $\quad Mi = C * Mg + B * Mp / Db + A \quad$ (Linear Law)

Mp/Db is the total volume of propellant. This equation is a linear scaling function. It assumes that those systems which are dependent on the propellant, or bulk density are scaled linearly with propellant mass or volume. Other scaling laws based on tank surface area are possible, and some efforts with them were made. Derived rigorously for multiple spherical tanks, they become complex. A simplified version may be written:

6.1-5 $\quad Mi = C * Mg + (B * Mp/Db)\string^(2/3) + A$

This equation was compared to the linear law for a few cases without large differences occurring in the results.

The coefficients of the linear scaling law (Equation 6.1-4) are determined by matching the masses calculated from the law with those of the Apollo LM for its various subsystems.

The payloads of interest for a single stage crew lander are in the range of 5,000 kg. This is the approximate mass of the Apollo LM ascent stage. The LM ascent stage and other equipment, as shown in Table 6-2, are therefore considered payload and the remaining LM mass is considered a "model" for the scaling equations.

The payload mass for this LM model is 5,300 kg (see Table 6-2). The idea is to use the descent stage of the LM and systems from the ascent stage to model a single stage, stand-alone lander with four tanks and four legs.

It is assumed that the RCS system of the ascent stage is part of the descent stage for the LM model. This is because the system is required for flight stability during descent. The environmental control systems, explosives, and Government Furnished Equipment (GFE) -- consisting primarily of the Lunar Rover and other scientific packages -- located on the Descent stage of the LM are assumed to be part of the payload for the LM model, since they are not required for descent operations.

Table 6-2, Lander Payload Approximation from the LM Masses

Ascent Gross Mass	4,785 kg
Less (RCS) Dry system	(119)kg
Less RCS Propellant Usable	(231)kg
Less RCS Propellant Unusable	(56)kg
Descent Environmental Systems	194 kg
Descent Explosives	12 kg
Descent Gov. Furn. Equipment	480 kg
Descent Liquids and Gasses	235 kg
Lander Payload	5,300 kg

The descent liquids and gasses are, for the most part, required for cooling and thermal control on the LM vehicle. However, most of the thermal control for which these liquids and gasses were used, was provided after landing, during the three days that the LM spent on the lunar surface. The descent liquids and gasses are therefore unnecessary for descent operations, and considered as a payload item. This means that the "LM" model does not have any thermal control. Since thermal control is undoubtedly necessary, it must be included with the payload, thereby reducing the actual payload capacity. This is not a bad way of handling the problem, since the sizing of the thermal control system is dictated more by lunar stay time and sun angle than by lander ascent/descent performance.

The total velocity change assumed for descent from a 93 km (50 nm) circular orbit is 2.10 km/s. During ascent the propulsion system is assumed to provide a velocity change of 1.85 km/s. These delta Vs were back-calculated from a detailed Apollo 17 weight statement in order to match Apollo 17 theoretical performance. Other published Apollo delta Vs are similar. For example, Apollo 11 published post-mission delta Vs were 1.85 km/sec for ascent, and 2.14 km/sec during descent, both for a 50 nm orbit (Ref. 4). The ascent delta Vs do not include an allowance for rendezvous, which was handled by the RCS in Apollo. An ascent/descent simulation is needed to further refine these numbers with new vehicle designs.

The LM used nitrogen tetroxide oxidizer and Aerozine-50 fuel as propellants. These propellants have a specific impulse and bulk density as shown below for the given mixtures. The "LM" model makes use of these same values. Nitrogen tetroxide/monomethylhydrazine and hydrogen oxygen propellant values are also shown below.

Propellant	Bulk Density		Mixture	Isp
	lbm/ft^3	kg/m3	Ratio	lbf-sec/lbm or kgf-sec/kg
N_2O_4/Aer 50	72.83	1,168	1.6:1	300
N_2O_4/MMH	73.17	1,170	1.9:1	330
LO_2/LH_2	22.54	361	6:1	450

Using the previous information, the coefficients of the scaling equation can be found and equation 6.1-4 becomes:

$$Mi = 0.0640 * Mg + 0.0506 * (1{,}168 / Db) * Mp + 390 \langle kg \rangle \quad (6.1\text{-}6)$$

Db is in units of kg/m^3. Mg and Mi are kg.

On the LM, there are four major subsystems which are assumed to be scaled by the gross mass (Mg). They total 1,034 kg. 45% of this 1,034 kg is required for the structure. 22% is Engines and related systems. 11% is required for the RCS (dry). Landing systems make up the remaining 21%.

447 kg is the total mass of the two LM subsystems that are assumed to be scaled by propellant mass (Mp). 40% of the 447 kg is used for passive thermal protection systems; and the remaining 60% is required for propellant tanks and plumbing.

The invariant (constant) mass of 390 kg is related to the power, control, and data subsystems.

The RCS propellant (Mrp) is the total of the usable, unusable and FPR RCS Propellant. The usable RCS propellant (Mrps) is calculated using the following scaling equation:

$$Mrps = 0.0068 * Mg * \Delta V \quad (6.1\text{-}7)$$

Where ΔV is in units of <km/s>

This equation is derived by matching the LM RCS requirements during ascent and descent. The flight performance reserve (FPR) RCS propellant (Mrpf) is calculated to be 20% of the usable propellant. The unusable RCS propellant (Mrpn) is to be 5% of the usable. These numbers are conservative, based on the Apollo weight statement. The RCS provided attitude control during LM powered ascent and descent and its propellant is therefore related to the delta V. The LM engines did not gimbal to control attitude, the RCS provided this function.

The unusable propellant (Mpn) in the main propulsion system is estimated to be 3% of the usable propellant (Mp). Again this number is conservative, and may be reduced with design effort.

The flight performance reserve propellant (Mpf) required for descent is calculated to be 4% of the usable propellant, based on the Apollo weight statement. This allowed roughly 30 seconds of hover in Apollo. Another 20 seconds of hover was part of the baseline Apollo propellant load and is assumed to be included in the delta V. The FPR reserve propellant is not included in ascent. It is for descent and is assumed to be used during descent or during ascent to lift itself. Additional propellant is not needed to lift it.

Therefore if:

$$Mn = Mpn + Mrpn = \text{Total unusable propellant} \quad (6.1\text{-}8)$$

$$Mf = Mpf + Mrpf = \text{Total FPR propellant} \quad (6.1\text{-}9)$$

Then the mass of the ascent propellant (Mpa) is calculated from Tsiolkosky's Equation.

6.1-10 $$Mpa = (Mi + Mla + Mn) * (e^{\{\Delta Va/(Isp*Go)\}} - 1)$$

Where:
- Mi = Inert Mass <kg>
- Mla = Ascent Payload Mass <kg>
- ΔVa = Ascent Velocity Change (1.85 <km/s>)
- Isp = Propellant Specific Impulse <lbf*s/lbm> or <kgf*s/kg>
- Go = Earth's Surface Gravity (0.0098 <km/s>)

The mass of the descent propellant (Mpd) is:

6.1-11 $$Mpd = (Mi + Mld + Mpa + Mn + Mf + Mrps/2) * (e^{\{Vd/(Isp*Go)\}} - 1)$$

Where:
- Mld = Descent Payload <kg> (includes "Mla")
- Vd = Descent Velocity Change (2.1 <km/s>)
- Mrps = Usable RCS propellant

The total usable propellant (Mps) is the sum of the Ascent and Descent propellants.

6.1-12 $$Mps = Mpa + Mpd$$

And the total propellant (Mp) is the sum of the usable and unusable propellant, the FPR propellant, and the RCS propellant (usable, unusable and FPR).

6.1-13 $$Mp = Mps + Mpn + Mpf + Mrps + Mrpn + Mrpf$$

The Gross (unloaded) Mass (Mgu) is the sum of the inert mass, the total propellant mass.

6.1-14 $$Mgu = Mi + Mp$$

The Gross (Total) Mass (Mg) is the sum of the gross (unloaded) mass, and the Descent payload.

6.1-15 $$Mg = Mgu + Mld$$

Table 6-3 is a listing of the output obtained from the "LM" model with a mass breakdown for each of the subsystems previously discussed.

Table 6-3, "LM Model" Weight Statement

Delta-V (Ascent)		0
Ascent Payload		0
Delta-V (Descent)		2.10 km/sec
Descent Payload		5,300.0 kg
Inert Mass (Mi):		1,902.0
Mass Scaled by Gross		1,048.0
Structure	(45%)	472.0
Engines	(22%)	231.0
RCS Dry	(11%)	115.0
Landing	(21%)	220.0
Mass Scaled by Propellant Tank Size		464.0
Protection	(40%)	186.0
Tanks	(60%)	278.0
Constant Mass (390 kg):		390.0
Propellant Mass (Mp):		9,171.0
Usable Propellant (Mps)		8,298.0
FPR Propellant (4% of "Mps")		332.0
Unusable Propellant (3% of "Mps")		248.0
RCS Propellant (Mrp)		292.0
Usable RCS Propellant (Mrps)		234.0
FPR RCS Propellant (20% of "Mrps")		47.0
Unusable RCS Propellant (5% of "Mrp")		12.0
Gross (Unloaded) Mass:		11,073.0
Gross (Total) Mass (Mo):		16,373.0

6.2 Single or Two Stage Lander?

Many spacecraft operate on the principle of staging. By separating the propellant into independent stages the payload efficiency of the vehicle can be increased. Each stage powers the craft until it exhausts its supply of propellant, then separates from the vehicle. The total inert mass is reduced each time a stage separates, and the next stage does not have to provide propulsion for the "dead" or "burnout" mass of the stage proceeding it. However, for every stage that is added to a spacecraft, an additional level of complexity is also introduced. Extra parts are added; plumbing is rerouted; separation equipment is installed; and the mass fraction (the ratio of propellant mass to total mass) is reduced. As the mass fraction falls, the payload efficiency of the vehicle is reduced. There comes a point in the design of the spacecraft where the payload increase (due to the addition of an extra stage) is exactly offset or is less than the payload loss (due to the reduction in the mass fraction). At this point, the addition of an extra stage will not change the amount of payload that can be delivered to a specific destination. This point in the design process determines the maximum number of stages that a vehicle will have.

Other factors, such as maintainability, simplicity, development cost, and operational complexity are best served by as few stages as possible. All these factors may drive a lunar lander to sacrifice some performance to maintain a single stage configuration.

For conventional propellant spacecraft, the number of stages is heavily dependent on the total velocity change (ΔV) that must be imparted to the payload. Experience in Earth launch vehicles shows that one stage is required for every three (3) <km/s> of velocity change. For instance, to get from Earth to low Earth orbit (LEO) requires approximately nine (9) <km/s> of velocity change; and therefore, normally requires a 3 stage vehicle. While there are other criteria for staging such as complexity, reusability, delayed circularization, and "g" loads, the total velocity change is the dominate factor, especially in high gravity environments like the Earth and Moon.

The total ΔV required to descend to and then ascend from the Moon is approximately four (4) <km/s>. From the discussion above, it is expected that a two stage vehicle would be slightly more efficient than a single stage vehicle. To prove this, the scaling equations presented in Chapter 6.1 are used to model both a single and a two stage lander.

In order to use the scaling equations in their most accurate region, an Apollo size payload is derived (Table 6-4). This also allows comparison with the LM mass itself. To compare single and two stage on a the same basis, the same scaling equations are used for both. Direct comparison with the LM may not be entirely appropriate.

The payload for these vehicles is a crew module. The crew module is similar to that used by the LM, and is capable of supporting two occupants for approximately three (3) days on the lunar surface. We can obtain the mass of this crew module by removing from the lander payload mass (Table 6-4), the mass of all of those systems that are not used for crew life support or module separation. Table 6-4 shows that the approximate mass of the crew module is 2,068 kg.

Table 6-4, Crew Module Approximate Mass

Lander Payload (Table 6-2)	5,300	kg
Less Ascent Propellants Usable	(2,232)	kg
Less Ascent Propellants Unusable	(121)	kg
Less Ascent Engines	(106)	kg
Less Ascent Tanks (Propellant)	(108)	kg
Less Ascent GFE	(184)	kg
Less Descent GFE	(481)	kg
Crew Module Mass	2,068	kg

The single stage vehicle transporting 2,068 kg to and from the lunar surface must have a gross mass in orbit, prior to descent, of 21,824 kg. By separating the vehicle into two stages (Ascent and Descent), applying the derived scaling equations, and assuming that the descent payload is equal to the ascent gross mass, it is found that the total gross mass of the two stage lander prior to descent from orbit is 18,903 kg. The real LM, which is not an entirely equivalent case, had a mass of 16,285 kg. As is expected, the mass of the single stage lander is greater, but not significantly greater, than the mass of the two stage lander carrying the same payload. Tables 6-5 and 6-6 show the mass breakdown of each subsystem for the two landers considered.

Table 6-5, Single Stage Crew Lander

Delta-V (Ascent)	1.85
Ascent Payload	2,068
Delta-V (Descent)	2.10
Descent Payload	2,068
Inert Mass (Mi):	2,652.0
Mass Scaled by Gross (6.40% of "Mo"):	1,397.0
Structure (45%)	629.0
Engines (22%)	307.0
RCS Dry (11%)	154.0
Landing (21%)	293.0
Mass Scaled by Propellant Tank Size (5.06% of prop. vol.)	865.0
Protection (40%)	346.0
Tanks (60%)	519.0
Constant Mass (390 kg):	390
Propellant Mass (Mp):	1,710.4
Usable Propellant (Mps)	15,621.0
FPR Propellant (4% of "Mps")	625.0
Unusable Propellant (7% of "Mps")	469.0
RCS Propellant (Mrp)	390.0
Usable RCS Propellant (Mrps)	312.0
FPR RCS Propellant (20% of "Mrps")	62.0
Unusable RCS Propellant (25% of "Mrps")	16.0
Gross (Unloaded) Mass:	19,756.0
Gross (Total) Mass (Mo):	21,824.0

Table 6-6, Two Stage Crew Lander

	Descent	Ascent
Delta-V (Ascent)	0	1.85
Ascent Payload	0	2,068.0
Delta-V (Descent)	2.10	0
Descent Payload	*6,179.0	0
Inert Mass (Mi):	2,136.0	946.0
Mass Scaled by Gross (6.40% of "Mo"):	1,210.0	395.0
Structure (45%)	544.0	177.0
Engines (22%)	266.0	87.0
RCS Dry (11%)	133.0	44.0
Landing (21%)	254.0	0
Mass Scaled by Prop. Tank Size (5.06% of prop vol)	536.0	160.0
Protection (40%)	214.0	64.0
Tanks (60%)	321.0	96.0
Constant Mass (390 kg):	390	390
Propellant Mass (Mp)	10,588.0	3,165.0
Usable Propellant (Mps)	9,580.0	2,868.0
FPR Propellant (4% of "Mps")	383.0	115.0
Unusable Propellant (7% of "Mpd")	287.0	86.0
RCS Propellant (Mrp)	337.0	97.0
Usable RCS Propellant (Mrps)	270.0	77.0
FPR RCS (20% of "Mps")	54.0	16.0
Unusable RCS Propellant (25% of "Mrp)	13.0	4.0
Gross (Unloaded) Mass:	12,724.0	4,111.0
Gross (Total) Mass (Mo):	18,903.0	*6,179.0

The results of this analysis show that the two stage lander can operate with more payload or will have a smaller mass than the single stage lander. From a performance point of view, the two stage vehicle is definitely the better vehicle. However, the total mass difference between these vehicles is only about 15 percent of the gross mass. When considering problems such as reusability, complexity, and "g" loads; the single stage lander is preferable to the two stage option.

6.3 Single Stage Performance Plots: Payload, Parking Orbit, Propellant Type

There are three cases of interest when studying the single stage lunar lander. The scenario where the lunar lander is used only to place a payload on the surface is called the "Cargo Down" case. In the "Cargo Down" case, the lander does not have propellant to ascend to orbit after delivering its payload. It, therefore, must stay on the lunar surface until refueled. The case in which the lander places a payload on the surface, and has enough propellant remaining to return its inert mass to orbit, is called the "Inert Returned" case. There is also a scenario in which the lunar lander carries a crew module down to the surface and then back to orbit. This case is called "Crew Module Round Trip".

The following plots show the relationship of total mass to payload mass for each of the three cases discussed above.

The first 3 plots (Figures 6-1, 6-2, & 6-3) show the lander performance to and from a 93 km orbit using different propellants. The first plot is for a lander using a 1.6 mixture ratio of nitrogen tetroxide and Aerozine-50 propellants. These are the propellants that were used on the Apollo LM. The second plot is for a 6:1 mixture ratio of liquid oxygen and liquid hydrogen. This plot represents a lander using cryogenic propellants. The third plot is for an advanced storable propellant lander using a 1.9 mixture ratio of nitrogen tetroxide and monomethylhydrazine. For a 6 m ton crew module going round trip from 93 km, the loaded masses are:

Propellant	N_2O_4/Aer 50	N_2O_4/MMH	LO_2/LH_2
Isp, sec.	300	330	450
Gross (Deorbit) Mass, Metric Tons	56	43	33

The cryogenic vehicle shows better performance, but not as much as expected. The low density of hydrogen drives the propellant mass multiplier up in the scaling equation (6.1-6). The equations may be biased against a pump-fed cryogenic system because they are scaled from a pressure-fed storable system. A good data point for a pump-fed cryogenic system of this nature is unavailable. A more detailed investigation of the inert mass of a cryogenic pump-fed lander is needed to determine a more realistic performance gain. Even though the pressure-fed derived equations bias against the pump fed systems somewhat, more detailed work may easily reduce the performance advantage of the cryogenic vehicle.

The next four plots (Figures 6-4 thru 6-7) show the lander performance to and from various orbits using advanced storable propellants. The performance plots are shown for 200, 400, and 1,000 km orbits. The last plot within this group is for a lander which is traveling to and from the second Lagrangian point (L2), located between the Earth and the Moon, Moon - Libration Point - Earth (M-LP-E). The delta V used for this last plot was approximated by assuming that the lander was flying to and from a 35,000 km circular lunar orbit.

The next four plots (Figures 6-8 thru 6-11) show the lander performance to and from the same orbits using cryogenic propellants.

The last plot (Figure 6-12) shows the gross (unloaded) mass of the lunar lander using advanced storable propellants as a function of the payload mass for the three performance scenarios that are being considered. The gross (unloaded) mass is the total mass of the lander without the payload that it is to carry. This plot is useful in determining the payload capability of a specific lander under all three scenarios. For instance, if it is specified that the lander be able to deliver a 25,000 kg payload to the lunar surface without returning, it is easily determined that the lander must have a gross mass of 41,000 kg without its payload. That same 41,000 kg lander can place 15,000 kg of payload on the surface and then return to orbit; or it could take a 7,000 kg crew module round trip. In a like manner, a vehicle capable of carrying 6 m tons round trip has an unloaded mass of 37,000 kg. This same unloaded mass could in theory land as much as 22,000 kg on the surface if all the propellant was expended. This sizing is only approximate, as can be seen in Section 8.0 where an attempt was made to combine the round trip and 25 m ton one way down requirements in a detailed weight statement. Combining two functions will result in an increased inert mass and some penalty over a dedicated lander.

Figure 6-1

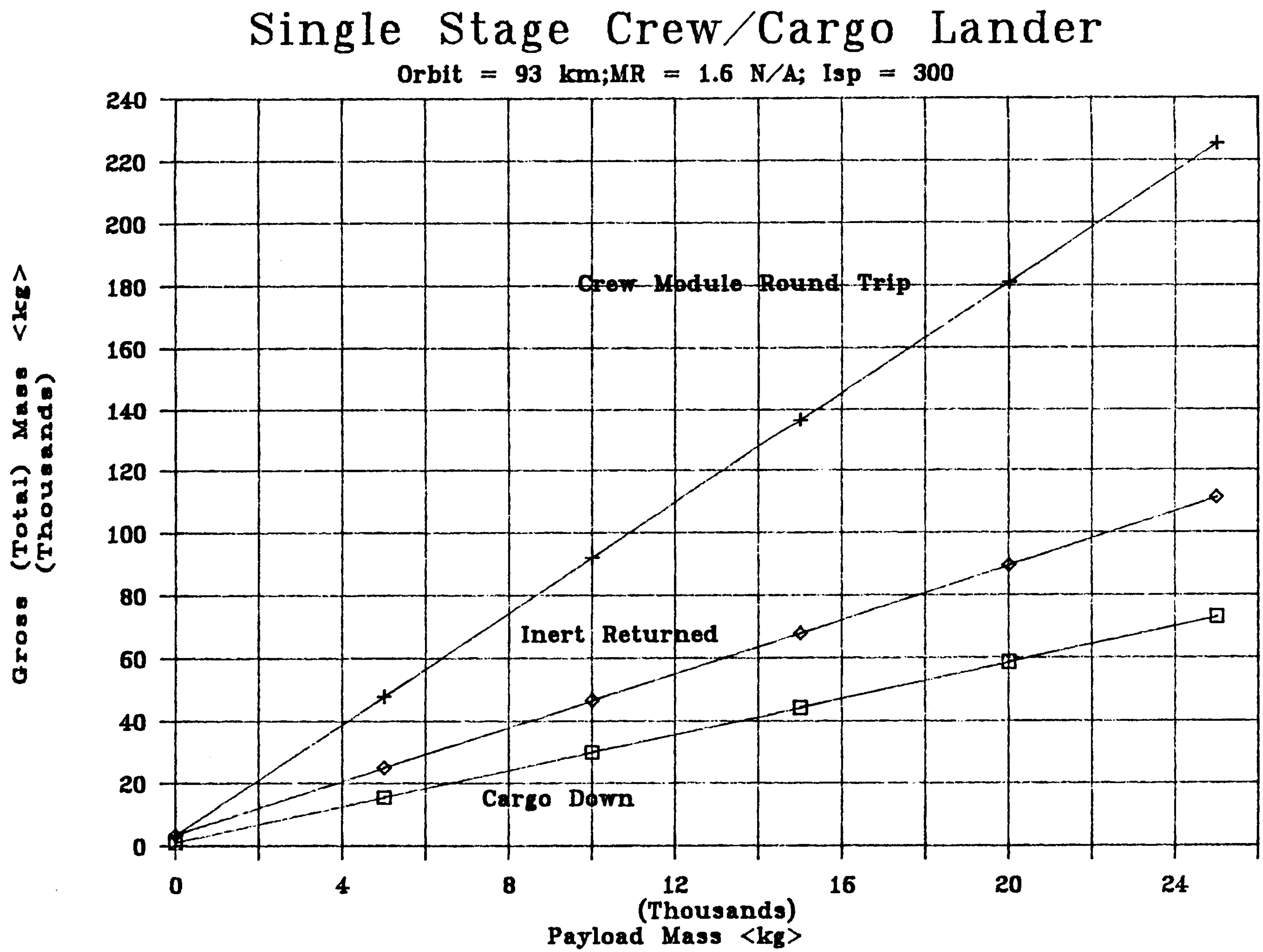

Single Stage Crew/Cargo Lander
Orbit = 93 km;MR = 1.6 N/A; Isp = 300
Gross (Total) Mass <kg>
(Thousands)
Crew Module Round Trip
Inert Returned
Cargo Down
0
20
40
60
80
100
120
140
160
180
200
220
240
0
4
8
12
16
20
24
(Thousands)
Payload Mass <kg>

Figure 6-2

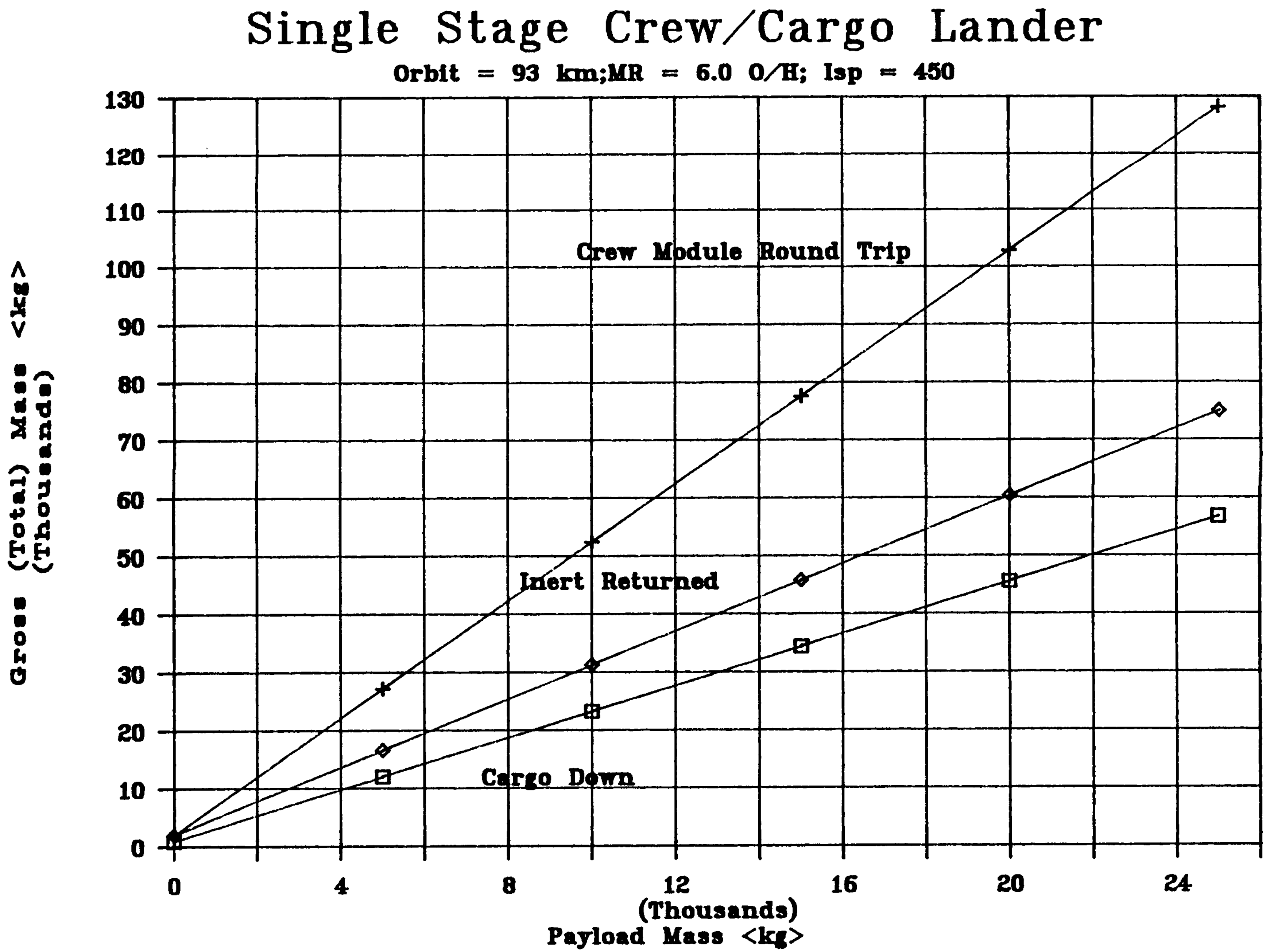
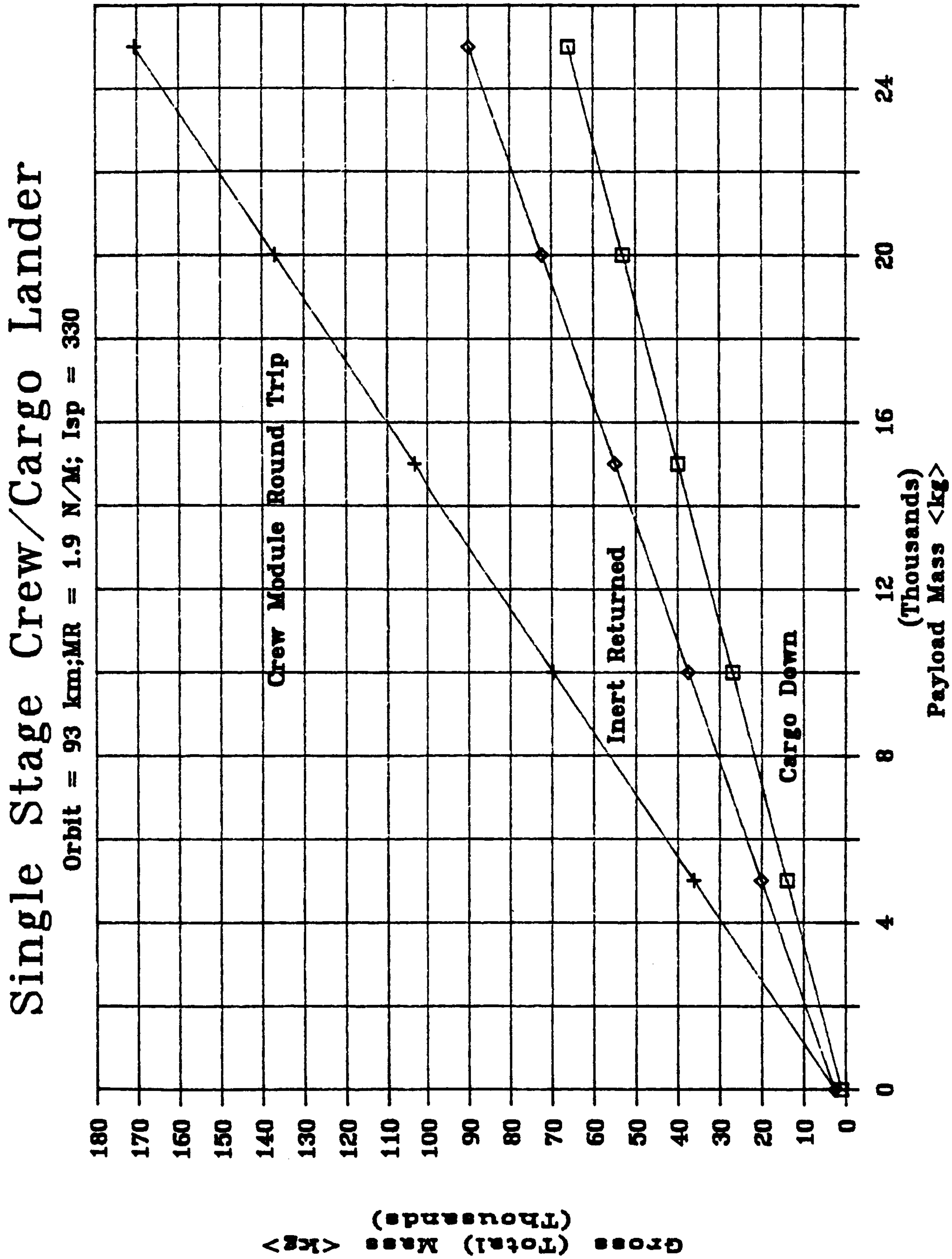

Single Stage Crew/Cargo Lander
Orbit = 93 km;MR = 6.0 O/H; Isp = 450
130
120
110
100
90
80
70
60
50
40
30
20
10
0
Gross (Total) Mass <kg>
(Thousands)
Crew Module Round Trip
Inert Returned
Cargo Down
0
4
8
12
16
20
24
(Thousands)
Payload Mass <kg>

Figure 6-3



Single Stage Crew/Cargo Lander
Orbit = 93 km;MR = 1.9 N/M; Isp = 330
Crew Module Round Trip
Inert Returned
Cargo Down
0
4
8
12
16
20
24
(Thousands)
Payload Mass <kg>
0
10
20
30
40
50
60
70
80
90
100
110
120
130
140
150
160
170
180
Gross (Total) Mass <kg>
(Thousands)

Figure 6-4

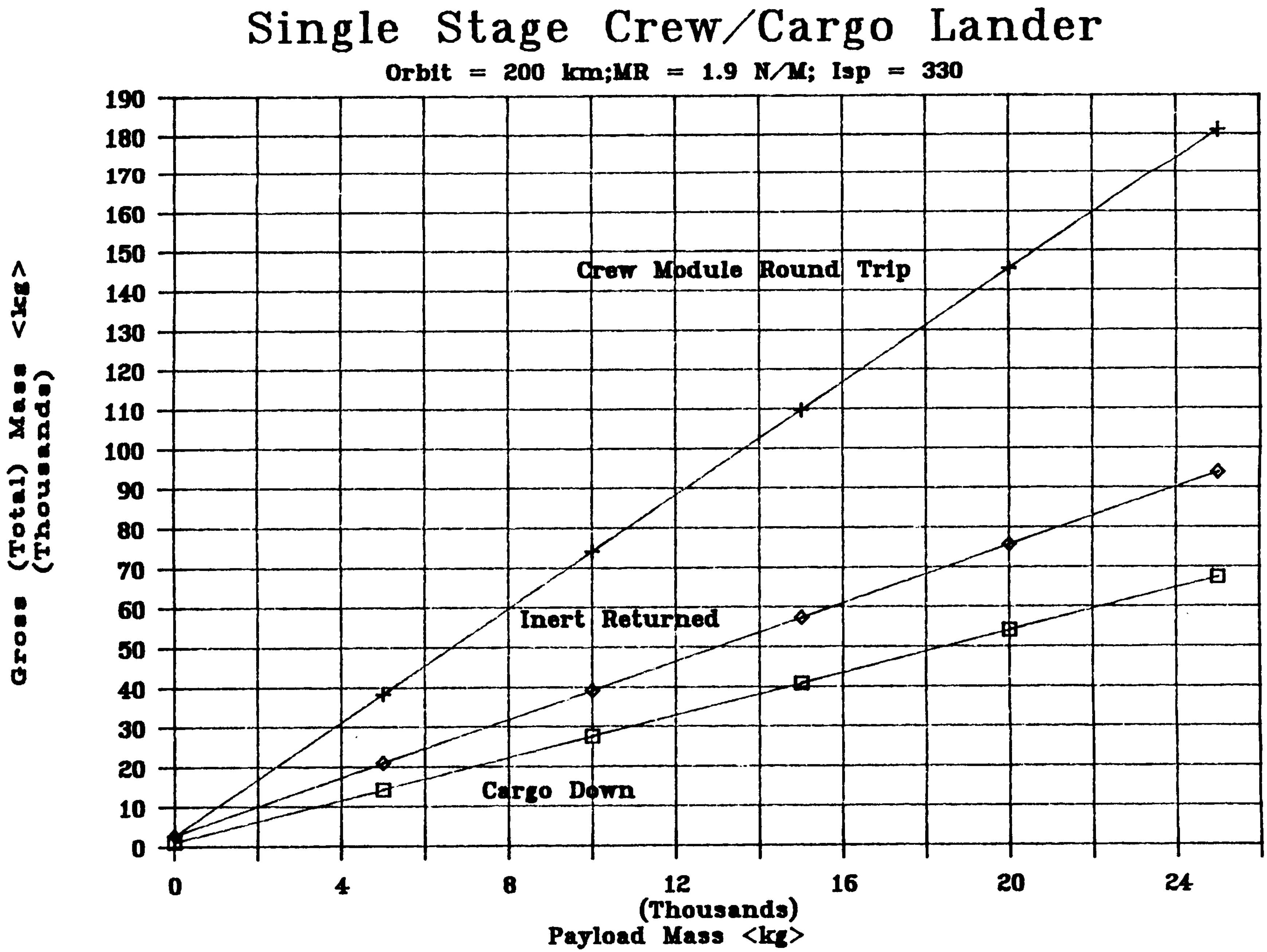

Single Stage Crew/Cargo Lander
Orbit = 200 km;MR = 1.9 N/M; Isp = 330
Gross (Total) Mass <kg>
(Thousands)
190
180
170
160
150
140
130
120
110
100
90
80
70
60
50
40
30
20
10
0
Crew Module Round Trip
Inert Returned
Cargo Down
0
4
8
12
16
20
24
(Thousands)
Payload Mass <kg>

Figure 6-5

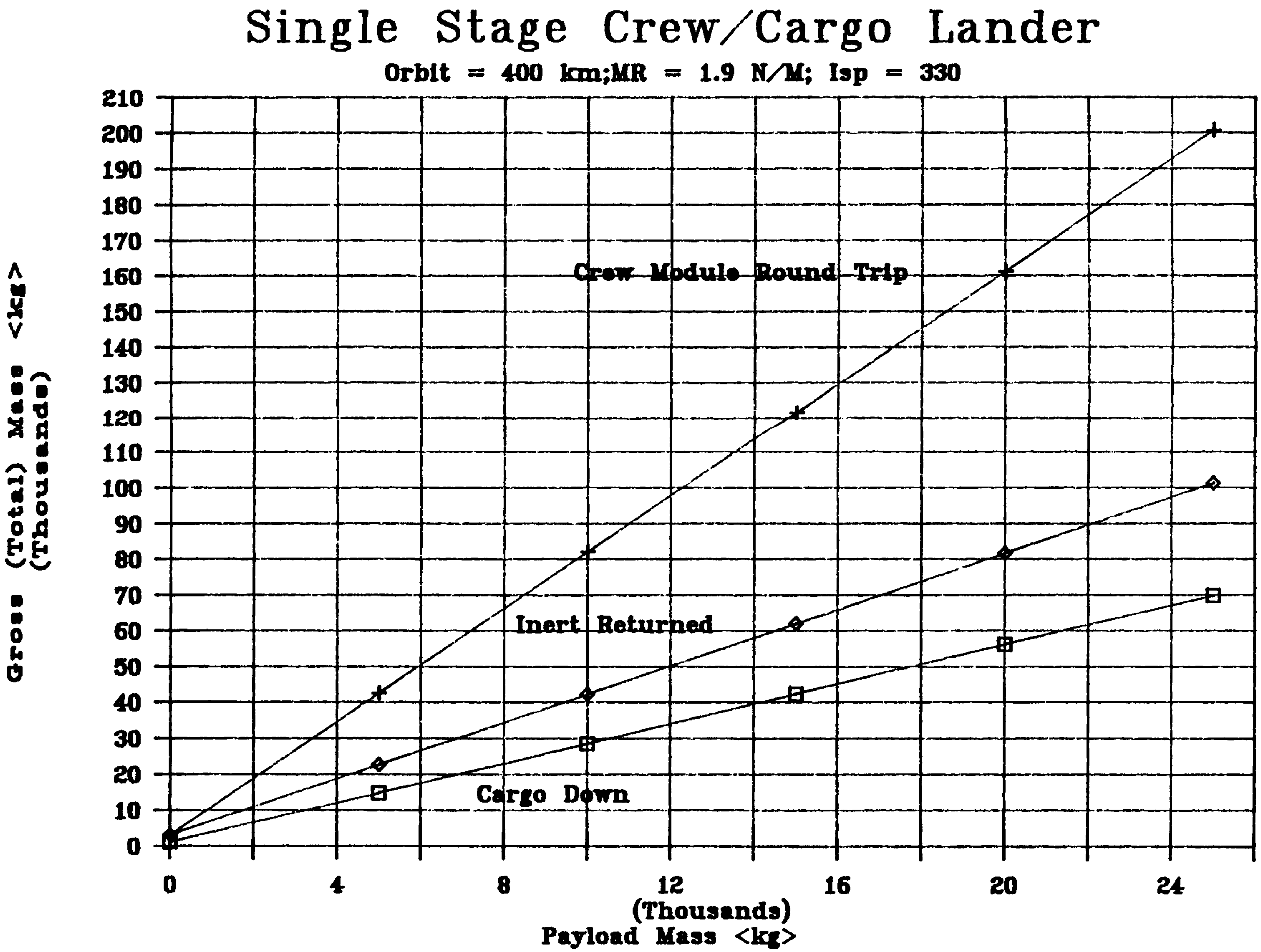
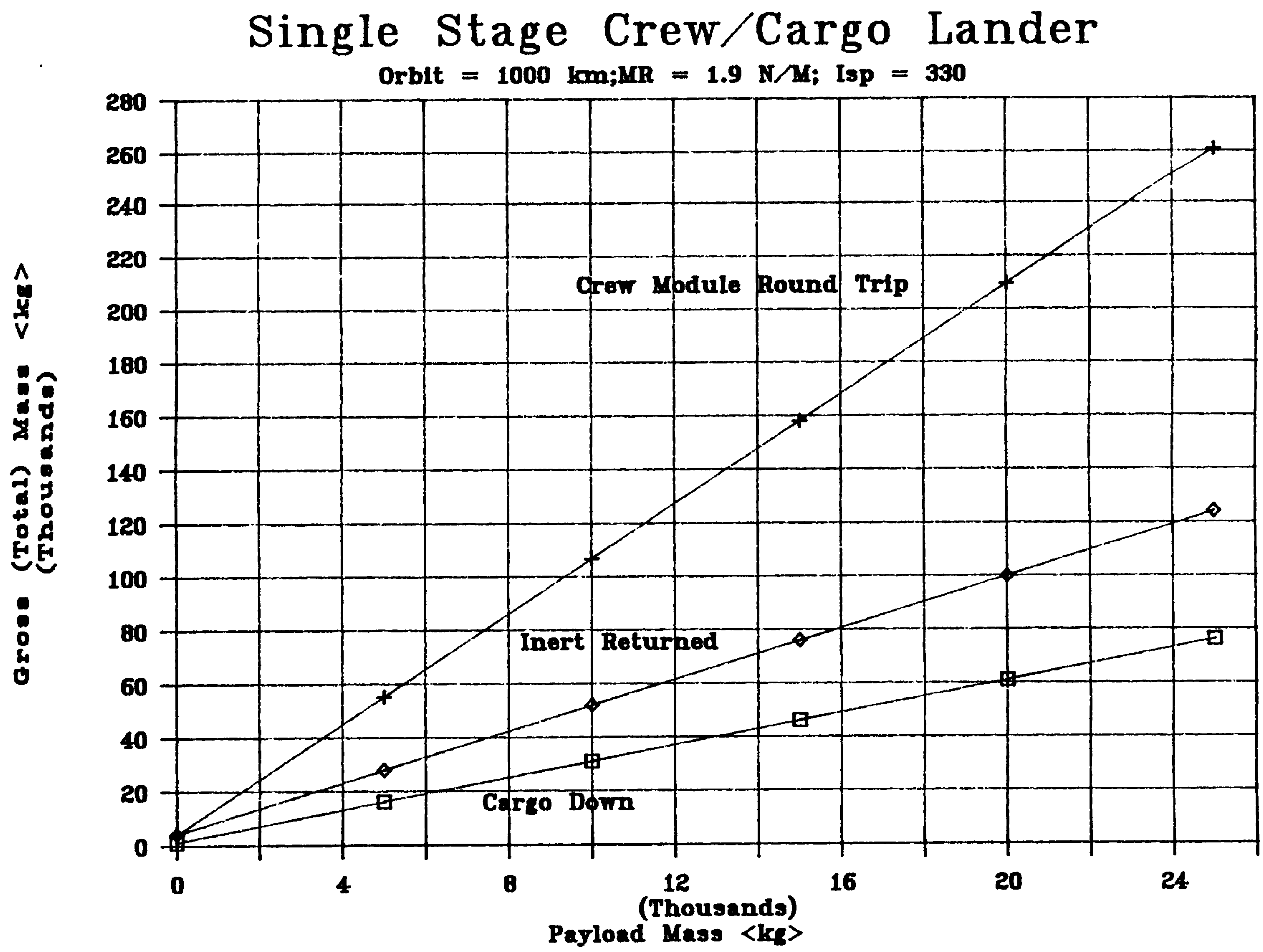

Single Stage Crew/Cargo Lander
Orbit = 400 km;MR = 1.9 N/M; Isp = 330
Gross (Total) Mass <kg>
(Thousands)
210
200
190
180
170
160
150
140
130
120
110
100
90
80
70
60
50
40
30
20
10
0
Crew Module Round Trip
Inert Returned
Cargo Down
0
4
8
12
16
20
24
(Thousands)
Payload Mass <kg>

Figure 6-6



Single Stage Crew/Cargo Lander
Orbit = 1000 km;MR = 1.9 N/M; Isp = 330
Crew Module Round Trip
Inert Returned
Cargo Down
Gross (Total) Mass <kg>
(Thousands)
Payload Mass <kg>
(Thousands)
0
4
8
12
16
20
24
20
40
60
80
100
120
140
160
180
200
220
240
260
280

Figure 6-7

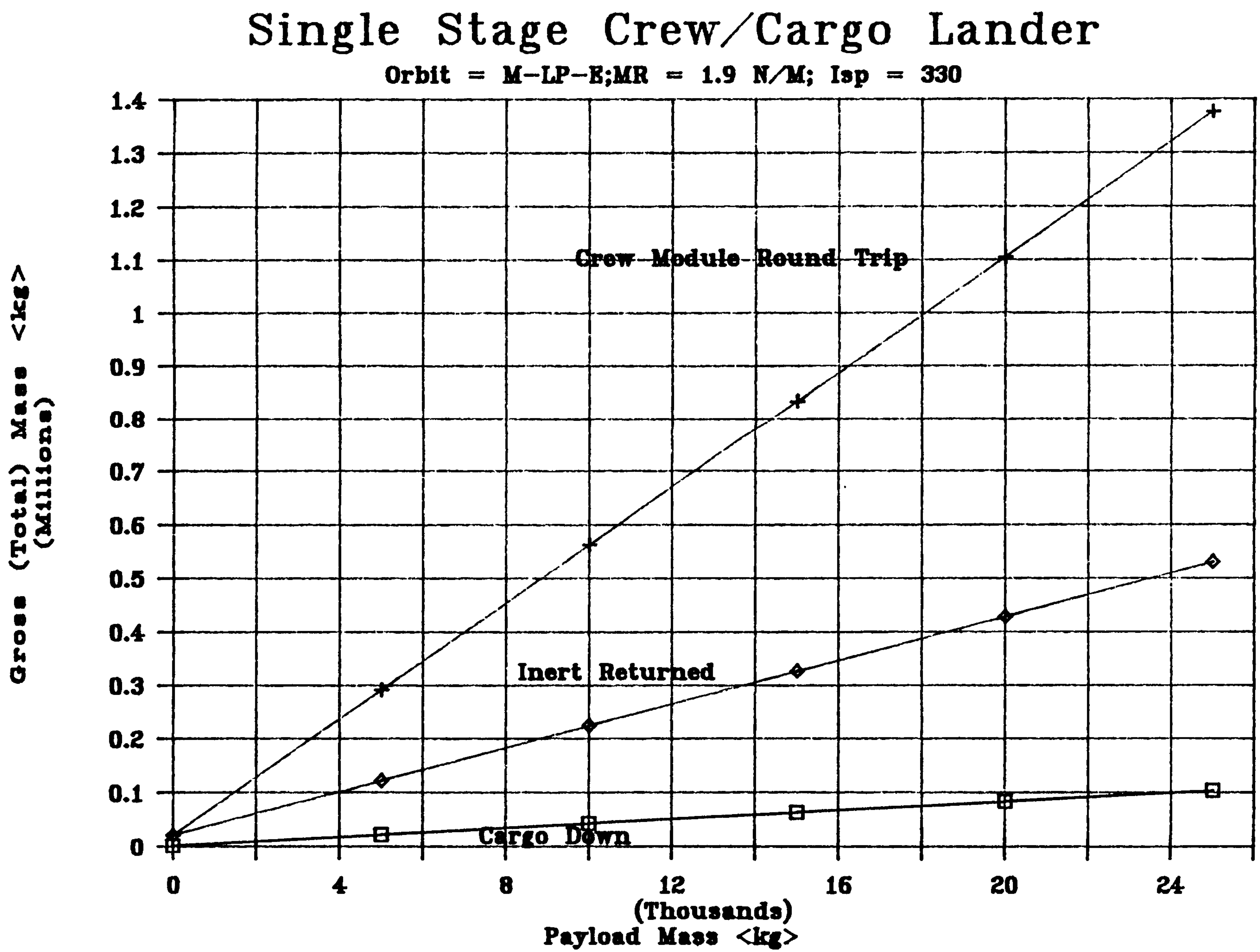

Single Stage Crew/Cargo Lander
Orbit = M-LP-E;MR = 1.9 N/M; Isp = 330
Crew Module Round Trip
Inert Returned
Cargo Down
Gross (Total) Mass <kg>
(Millions)
1.4
1.3
1.2
1.1
1
0.9
0.8
0.7
0.6
0.5
0.4
0.3
0.2
0.1
0
0
4
8
12
16
20
24
(Thousands)
Payload Mass <kg>

Figure 6-8

Single Stage Crew/Cargo Lander

Orbit = 200 km;MR = 6.0 O/H; Isp = 450

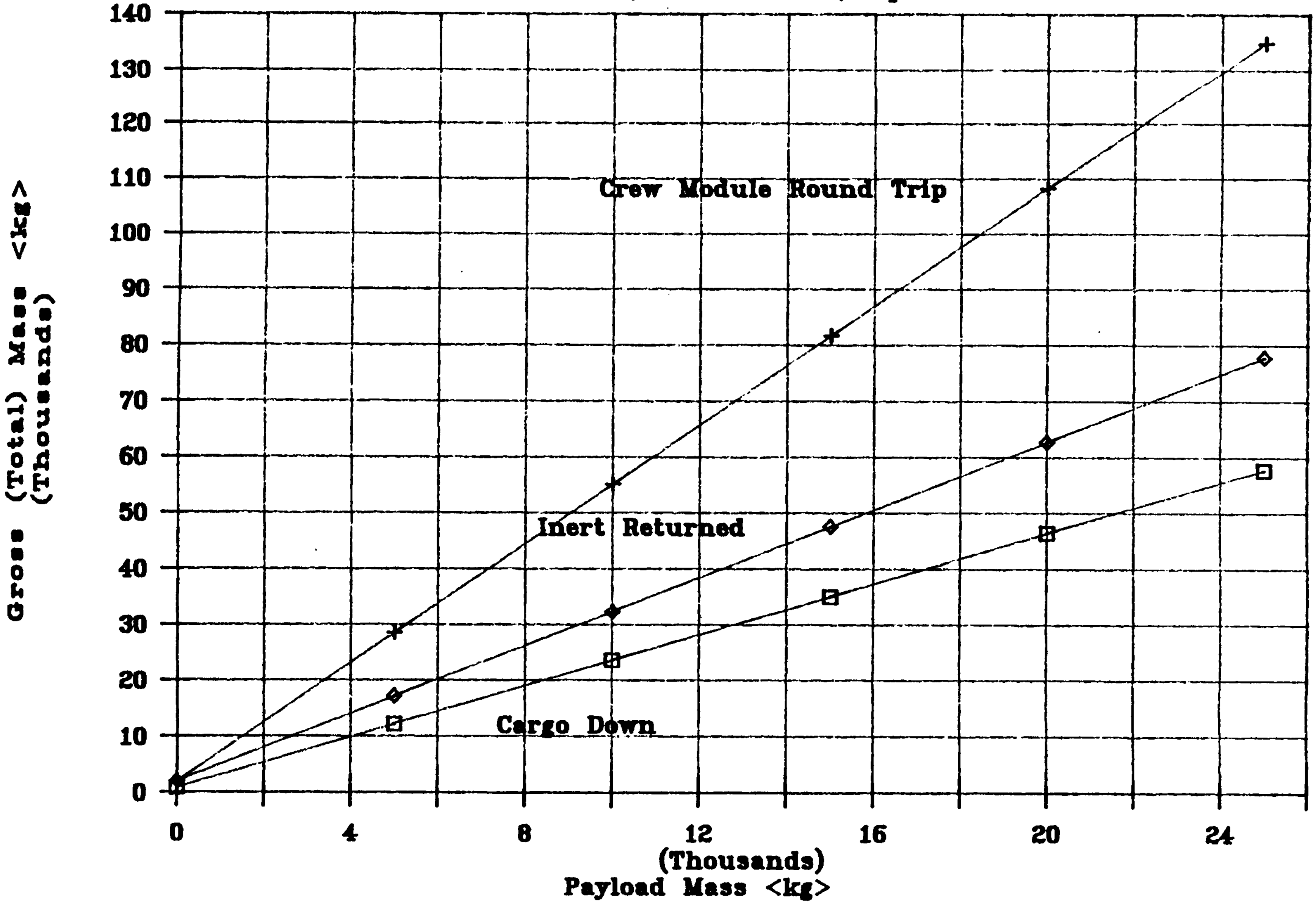


Figure 6-9

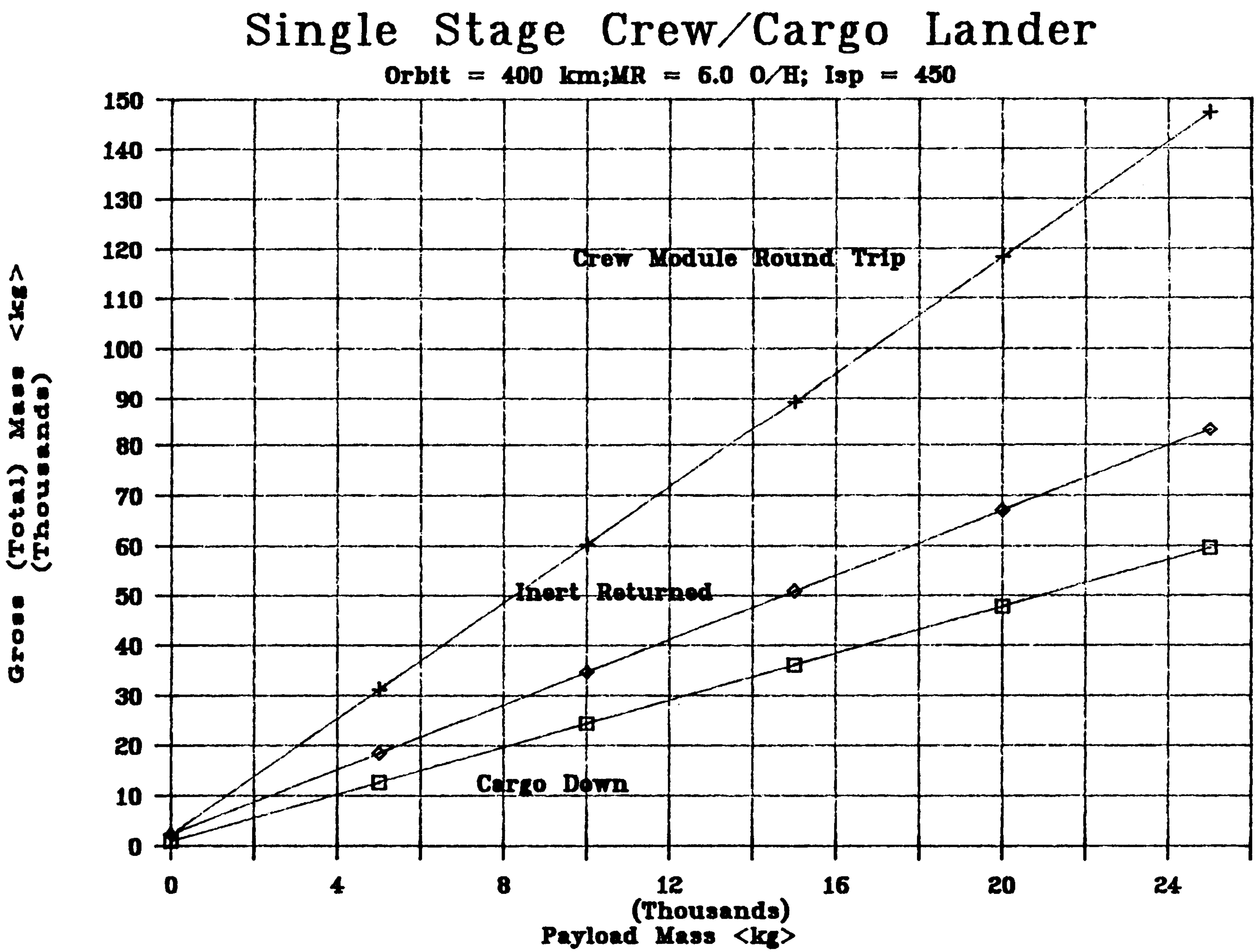

Single Stage Crew/Cargo Lander
Orbit = 400 km;MR = 6.0 O/H; Isp = 450
Crew Module Round Trip
Inert Returned
Cargo Down
Gross (Total) Mass <kg> (Thousands)
Payload Mass <kg> (Thousands)
0
10
20
30
40
50
60
70
80
90
100
110
120
130
140
150
0
4
8
12
16
20
24

Figure 6-10

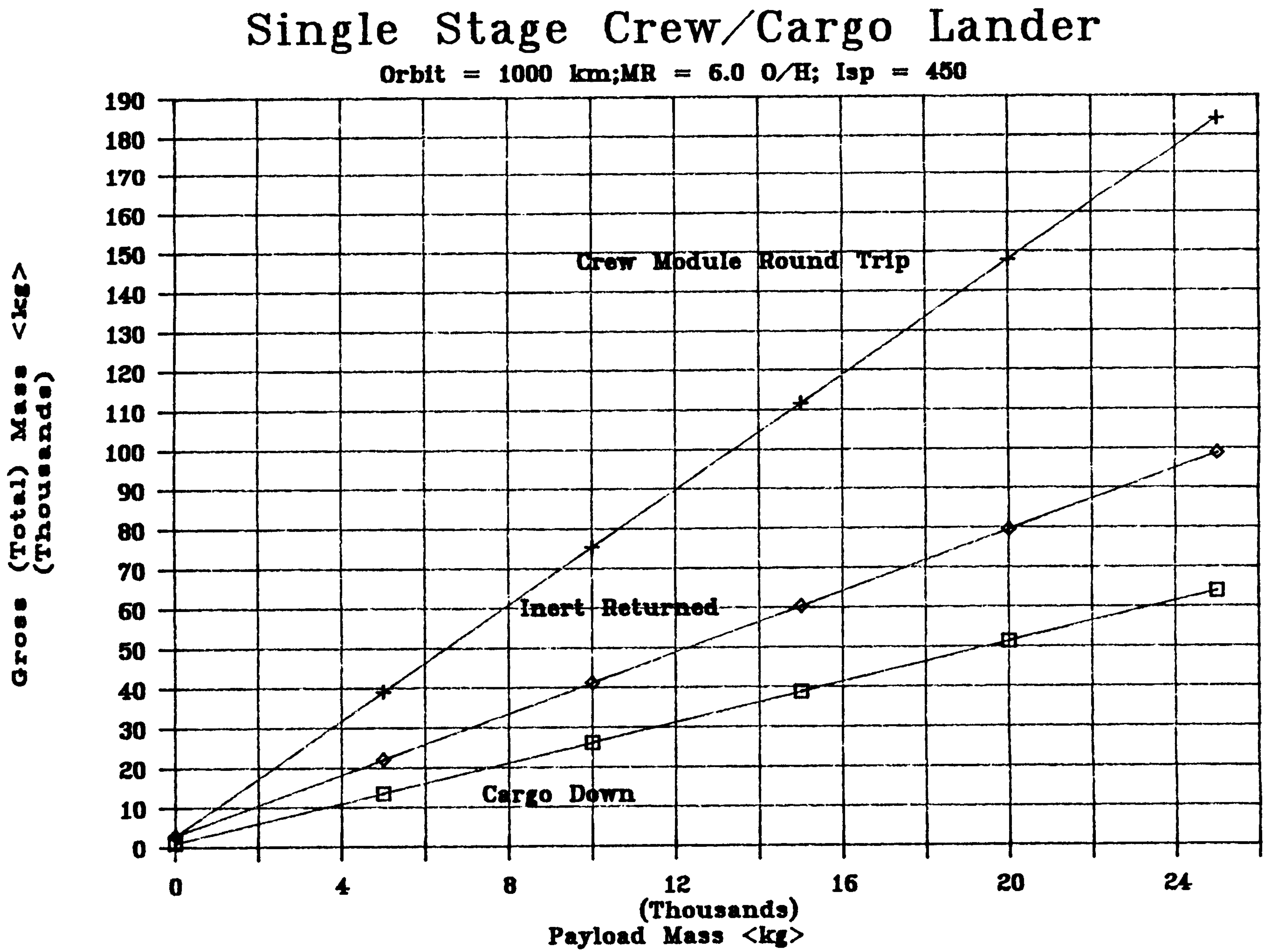

Single Stage Crew/Cargo Lander
Orbit = 1000 km;MR = 6.0 O/H; Isp = 450
Crew Module Round Trip
Inert Returned
Cargo Down
Gross (Total) Mass <kg>
(Thousands)
190
180
170
160
150
140
130
120
110
100
90
80
70
60
50
40
30
20
10
0
0
4
8
12
16
20
24
(Thousands)
Payload Mass <kg>

Figure 6-11

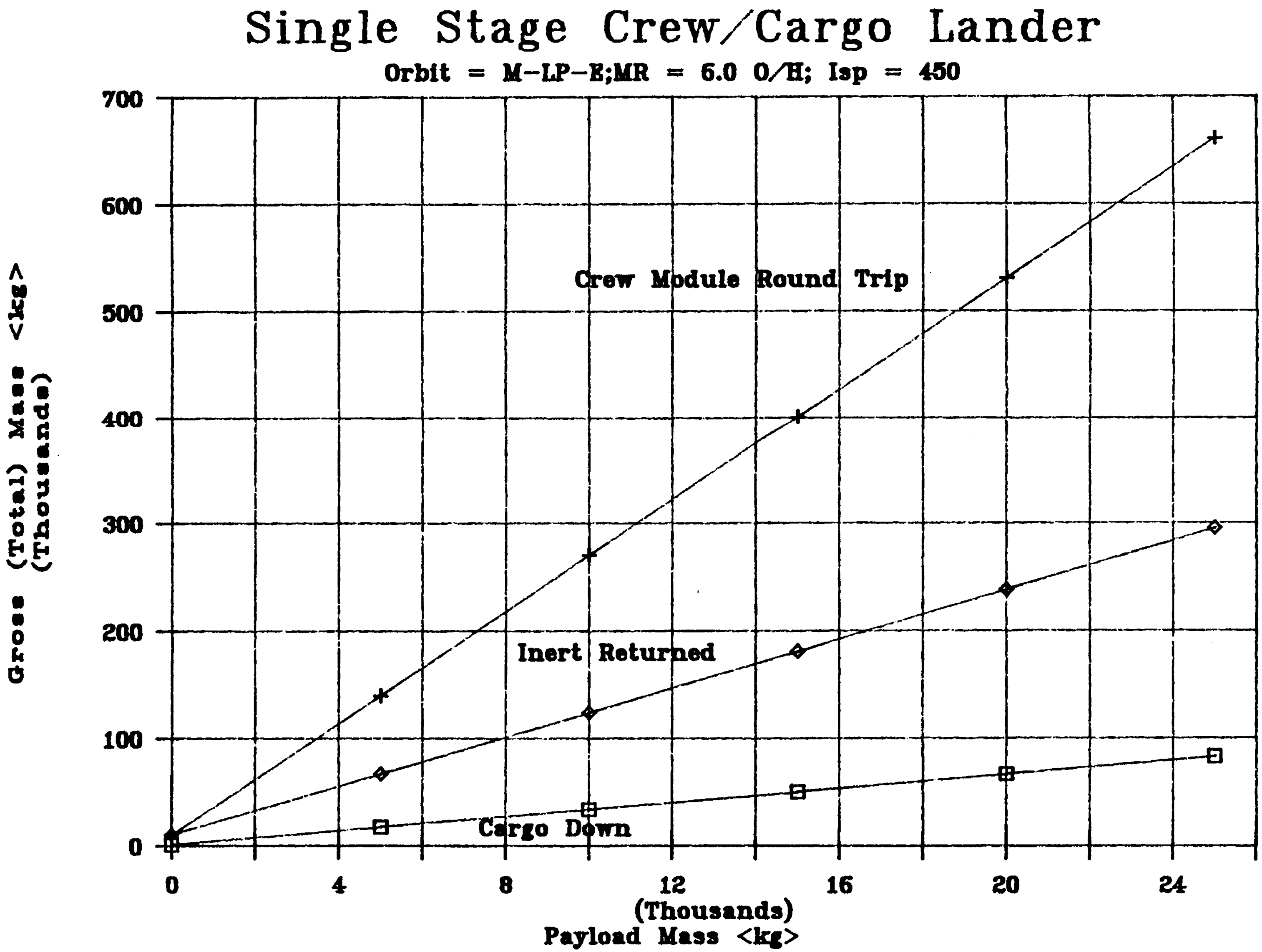
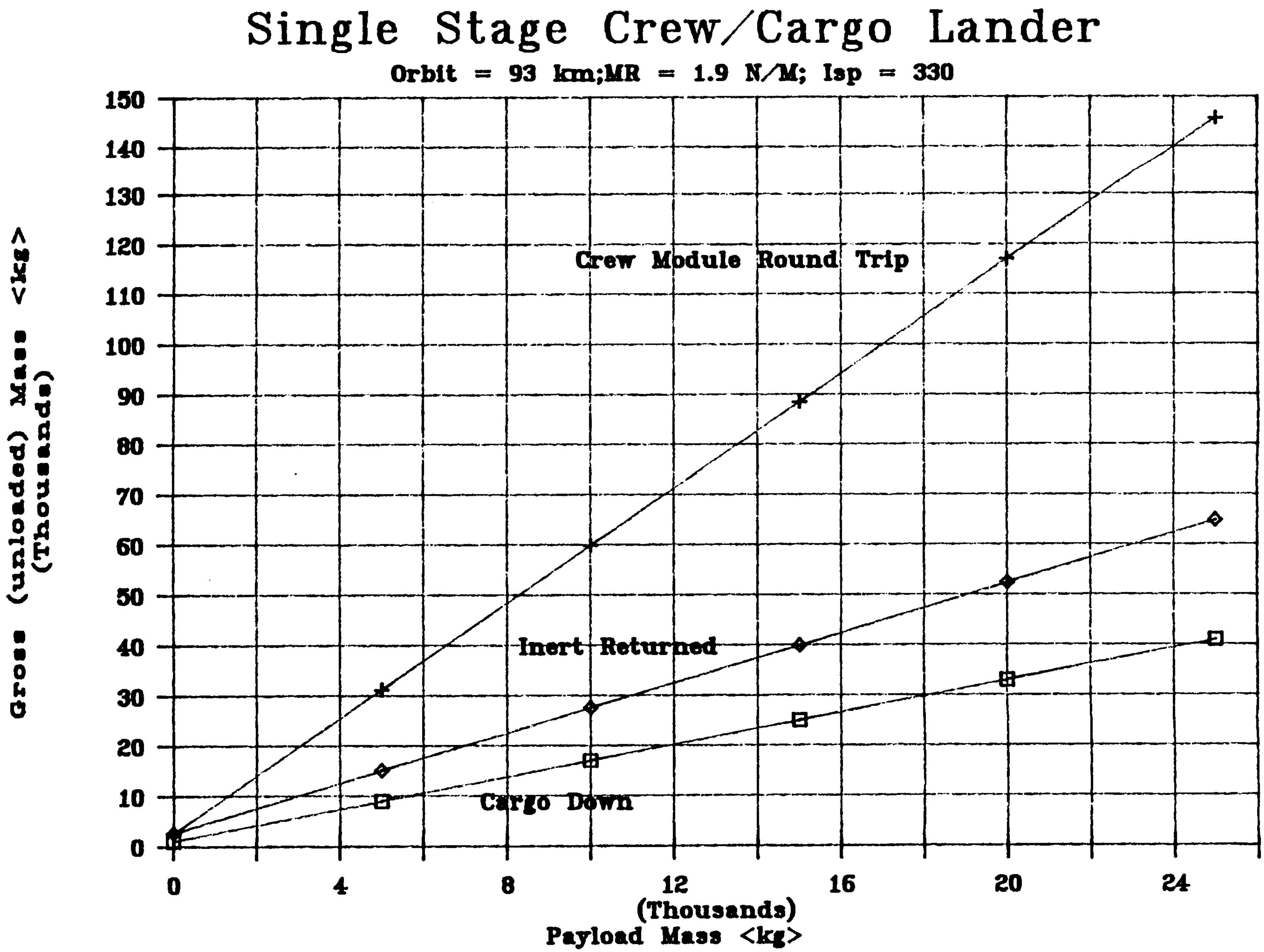

Single Stage Crew/Cargo Lander
Orbit = M-LP-E;MR = 6.0 O/H; Isp = 450
Gross (Total) Mass <kg>
(Thousands)
700
600
500
400
300
200
100
0
Crew Module Round Trip
Inert Returned
Cargo Down
0
4
8
12
16
20
24
(Thousands)
Payload Mass <kg>

Figure 6-12



Single Stage Crew/Cargo Lander
Orbit = 93 km;MR = 1.9 N/M; Isp = 330
Gross (unloaded) Mass <kg>
(Thousands)
150
140
130
120
110
100
90
80
70
60
50
40
30
20
10
0
Crew Module Round Trip
Inert Returned
Cargo Down
0
4
8
12
16
20
24
(Thousands)
Payload Mass <kg>

6.4 Parking Orbit Altitude

Tables 6-7 and 6-8 show how lander mass increases steadily as lunar orbital altitude goes up. Table 6-9 shows how LEO stack mass also goes up with lunar orbit altitude. The LEO stack mass does not rise dramatically until orbits of 1,000 km or over are used. From a performance standpoint, the lowest orbits are therefore preferable. Apollo experience has indicated that very low orbits, on the order of 100 km may be unstable over time periods of months. The best altitude will therefore be the lowest altitude which is stable for the time period required. Early Apollo work (Ref. 12) came to the same conclusion. They found a lower limit related to abort concerns of roughly 50 nm (93 km) for short stay times.

Table 6-7, Lander Mass versus Alt., Crew Transfer Case (6 motor round trip)

Circ. Orbit Altitude, km	Isp-450 sec. Deorbit Mass	Isp-450 sec. Inert Mass	Isp-450 sec. Propellant Mass	Isp=330 sec. Deorbit Mass	Isp=330 sec. Inert Mass	Isp=330 sec. Propellant Mass
93	3	6	20	43	5	32
200	34	6	22	46	5	35
400	37	7	24	50	6	38
1,000	46	9	31	66	7	53
L2 (M-LP-E)	166	13	147	344	38	300

Table 6-8, Lander Mass Vs. Altitude, 25 m ton Cargo Down Case

Circ. Orbit Altitude, km	Isp-450 sec. Deorbit Mass	Isp-450 sec. Inert Mass	Isp-450 sec. Propellant Mass	Isp=330 sec. Deorbit Mass	Isp=330 sec. Inert Mass	Isp=330 sec. Propellant Mass
93	57	8	24	66	6	35
200	58	8	25	68	7	36
400	60	8	27	70	7	36
1,000	64	9	30	76	7	44
L2 (M-LP-E)	84	13	46	100	11	64

Table 6-9, LEO Stack Mass as a function of Lunar Orbit Altitude

All masses are metric tons
All OTVs are LO_2/LH_2, 455 sec Isp.
Space Station Orbit altitude - 450 km
Delta Vs as given in *
All LEO-LLO trajectories are 75 hour transfers
No plane changes are accounted for
OTVs are "rubber" and optimized to the given payload

OTVs assume:
15% of entry mass is aerobrake
5% of prop. is tankage, etc.
2.3% of prop. is FPR and
Other OTV inerts = 2.5 m tons
for 2 stage, 4.5 m tons, for 1 stage

*LLO Altitude km	Lander Deorbit Mass	-------- LEO Stack Mass -------------------			
		1 stage OTV-----		2 Stage OTV-----	
		Load lander propellants in: LLO	LEO	LLO	LEO
- 6 m ton crew capsule round trip, LLO-LS-LLO, 450 sec. Isp Lander					
93	32	111	136	101	127
200	34	120	142	107	133
400	37	121	150	112	140
1,000	46	142	174	131	165
36,000 (L2)	170	500	535	471	506
- 25 m ton cargo one way, 450 sec. Isp expended lander					
93	57	190	190	174	174
200	58	192	192	176	176
400	60	195	195	180	180
1,000	64	202	202	187	187
36,000 (L2)	84	268	268	246	246
- 6 m ton crew capsule round trip, LLO-LS-LLO,330 sec Isp lander					
93	44	148	169	137	159
200	46	155	172	144	162
400	50	162	184	152	173
1,000	66	205	226	191	214
36,000 (L2)	344	963	1,115	904	1,039
- 25 m ton cargo one way, 330 sec. Isp expended lander					
93	66	217	217	199	199
200	68	221	221	204	204
400	70	229	229	208	208
1,000	75	238	238	219	219
36,000 (L2)	100	314	314	290	290

*Delta V Table

Lunar Orbit	TLI	**LOI/TEI	Total
93	3.101	0.846	3.947
200	3.101	0.832	3.933
400	3.102	0.809	3.910
1,000	3.102	0.759	3.861
35,000 (L2,M-LP-E)	3.084	0.863	3.947

**LOI and TEI are assumed to be the same.

450 km SS Orbit
Flight Time = 75 hr.

6.5 Plane Change Capability

The lander will require a small plane change capability to have reasonable launch windows from the surface.

Most plane changes will be circular orbit, constant velocity plane changes, where the orbital velocity remains unchanged. In this type of plane change the velocity change (ΔV) that is required to change the plane by θ radians can be calculated from the following equation.

$$\text{6.5-1} \qquad \Delta V = 2 * \{MU / (Ro + Alt)\}^{\wedge 0.5} * \text{SIN}(\theta / 2)$$

Where:

ΔV	=	Required Velocity Change <km/s>
MU	=	Gravity Constant <km^3/s^2>
Ro	=	Planet radius <km>
Alt	=	Orbital Altitude <km>
θ	=	Angle of the Plane Change <rad>

For the lunar case Mu = 4,900 km^3/sec^2, Ro = 1,740 km.

In Table 6-10, equation 6.5-1 is used to calculate delta Vs for various circular orbits. The delta Vs can then be used to calculate an approximate increase in vehicle mass.

Table 6-10 shows that one time plane changes on the order of 15 degrees can be built in for modest lander mass increases on the order of 10%. This will also result in a LEO stack mass increase of at least 10%. Table 6-10 also shows that the plane change delta V and vehicle mass increase does not vary much with lunar orbit altitudes below 1,000 km for a given angle of plane change. As the orbit altitude increases above 1,000 km, plane change delta V goes down drastically but the lander mass goes up drastically due to increased ascent and descent delta V.

The ability to change planes widens the launch window the vehicle has to reach high inclination lunar orbit. For a landing site such as Lacus Verus at 13° South latitude it might allow a lander to ascend to an OTV or LLO Space Station in lunar equatorial orbit at any time. This is a highly desired feature. For a high latitude base and parking orbit, polar for instance, a 15 degree plane change capability would allow launch on roughly 4.5 days out of 27 days in a lunar month.

If circular orbit constant velocity plane changes are not desired then it will be necessary to determine the initial velocity (Vi) and the final velocity (Vf) at which the spacecraft will be traveling, in addition to the plane change angle. When these three values have been found the law of cosines (equation 6.5-2) can be used to determine the velocity change required.

$$\text{6.5-2} \qquad \Delta V = \{Vi^{\wedge 2} + Vf^{\wedge 2} - 2 * Vi * Vf * \text{COS}(\theta)\}^{\wedge 0.5}$$

If the apogee and the perigee of the initial and final orbits is available, then Vi and Vf can be calculated using the "Vis-Viva" Equation.

$$\text{6.5-3} \qquad Vi/f = [MU * \{2 / (R_o + Alt) - 2 / (2 * R_o + Altp + Alta)\}]^{\wedge 0.5}$$

Where:

$V_{i/f}$	=	Speed (initial or final) <km/s>
Altp	=	Altitude of Perigee <km>
Alta	=	Altitude of Apogee <km>
Alt	=	Alt. of initial or final orbit <km>

Table 6-10, Plane Change

Plane Change Required Degrees	Circular Orbit Altitude, Degrees	Delta V Req. for Plane Change km/sec	Approx. Increase in vehicle size over baseline* %
5	93	0.14	3
5	200	0.14	3
5	1,000	0.12	3
10	93	0.28	7
10	200	0.28	6
10	1,000	0.23	5
15	93	0.43	10
15	200	0.41	10
15	1,000	0.35	8
20	93	0.57	14
20	200	0.55	13
20	1,000	0.46	11
25	93	0.71	17
25	200	0.69	17
25	1,000	0.58	14
30	93	0.85	21
30	200	0.82	21
30	1,000	0.69	17
45	93	1.25	33
45	200	1.22	32
45	1,000	1.02	26
90	93	2.31	69
90	200	2.25	66
90	1,000	1.89	54

* This percentage was calculated by comparing the mass ratio e^(Delta V/g* Isp) of a baseline vehicle (450 sec. Isp, 4.1 km/sec total delta V + transfer delta Vs) with the mass ratio of an equivalent vehicle with the additional Delta V required for the plane change. The baseline vehicle delta V is changed as parking orbit altitude changes. 4.1 km/sec corresponds to a 93 km orbit.

6.6 Lander Size

The groundrules of this study require the lander to take down a cargo of 25 m tons and take a crew of up to 6 round trip. This is estimated to result in a mass of 6 m tons round trip. Is it better to build one lander or two to meet these requirements? Two factors of great importance are the engine throttling ratio needed to handle these requirements and the propellant and inert masses needed to perform the different tasks and the penalty doing one task imposes on another. Different masses delivered to LLO also impose penalties on the transportation system carrying the lander from Earth.

6.6.1 Thrust Required

The following groundrules were used to estimate total engine thrust required:

1) Maximum thrust can be defined as the thrust needed to produce the same deorbit deceleration that the Apollo Lunar Module used, roughly 9 ft/sec^2. This is a thrust to Earth weight ratio of about 0.28. A high fidelity descent simulation is expected to show that lower thrust/weights can be used at the expense of more propellant. In the absence of this data, the Apollo number is used. Other trajectory calculations indicate minimum delta V thrust/weight will be at a higher ratio than 0.28. Figure 6-13 from an early Apollo study (Ref. 12), shows how thrust/weight affects descent delta V.

2) Ascent maximum thrust can be defined by the thrust needed to produce the same acceleration off the surface that the ascent stage of the Apollo lunar module had, roughly 6 ft/sec^2 (T/W = .186). An iterative ascent simulation will show that as thrust is reduced and this acceleration goes down, that propellant load must go up to account for increased gravity losses. Figure 6-14, from Ref. 12, shows how ascent delta V was predicted to vary with ascent thrust/weight.

3) Minimum thrust is somewhat less than the thrust needed to hover the vehicle in its least massive condition. For the Apollo Lunar Module the minimum thrust was roughly 40 % of that required to simply hover. Some thrust less than the hover value is required to be able to descend. In the absence of a simulation, this study will simply assume the Apollo LM value (40% of hover thrust) is the minimum.

These groundrules result in the numbers shown in Table 6-11. The table shows various thrusts estimated to be required in different circumstances. The widest range is between deorbiting a 25 m ton payload from a higher low orbit with a low performance propellant (43,000 lbf required) and hovering a crew capsule and the vehicle inert mass just before running out of propellant such as might occur in an abort to the surface or a normal landing requiring propellant loading on the surface (1,760 lbf). The ratio between these two cases is roughly 24 to 1. The Apollo lunar module engine was designed with a 10 to one throttling ratio. If the minimum thrust case is taken as a normal landing for an H_2/O_2 lander with a crew capsule (2,957 lbf), the throttling ratio becomes 13 to 1. Table 6-12 shows a variety of cases and how the throttling ratio might vary.

Reducing the required throttling ratio may have significant advantages. The single, pressure-fed Apollo lunar module engine was cooled by ablation. A reusable engine must be regeneratively cooled. Pressure-fed regenerative cooling over a wide throttling ratio is not possible due to the thrust chamber cooling flow changing a great deal. This leads to a higher chamber pressure pump-fed engine, a much more complicated device, which then leads to two or more engines for redundancy. A single purpose lander, to land only a crew, might function with a pressure-fed single engine. Table 6-12 numbers indicate a throttling ratio of 7 or 8 to one might be enough if one lander was not required to bring down the 25 m ton cargo and the crew capsule as well. The table indicates that a dedicated cargo lander and a dedicated crew lander would each require a throttling ratio of 7 or 8 to one. The crew lander might use one or two engines and the cargo lander four. Other schemes involving shutting off or not using engines are also possible, but result in inert mass penalties.

Figure 6-13, Typical Variation of T/W_o with Characteristic Velocity for Lunar Landing (h_A = 100 Nautical Miles)

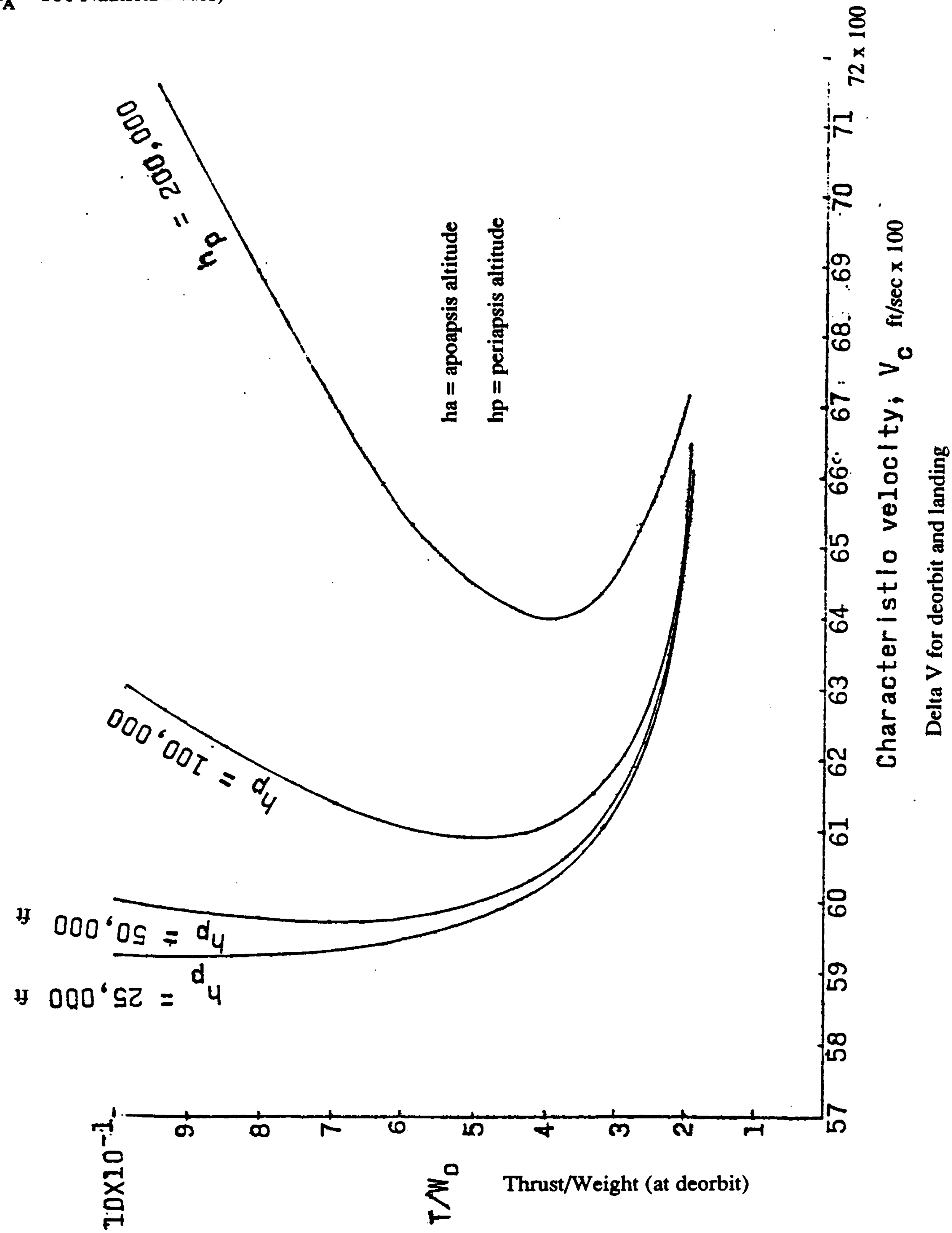
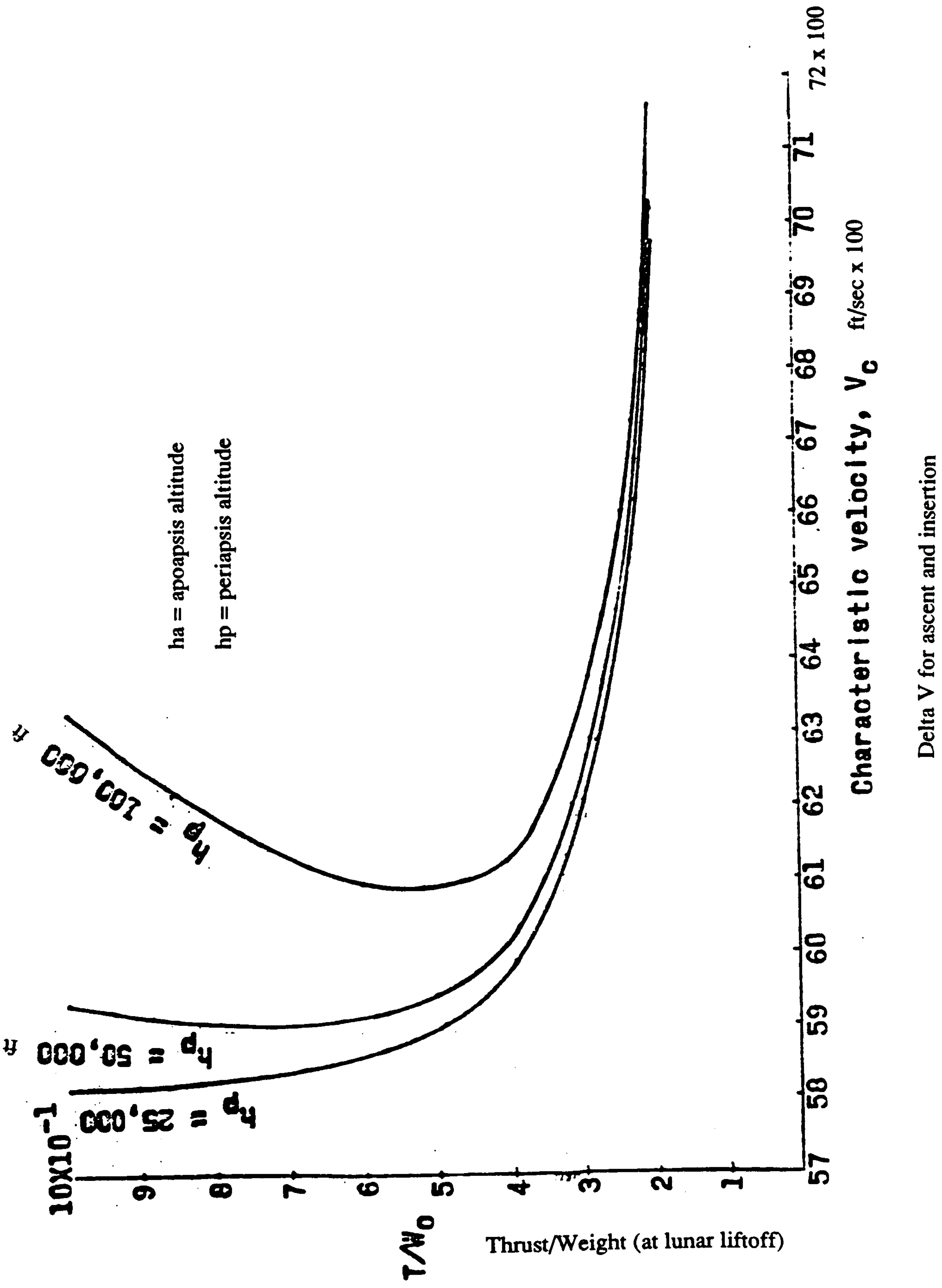


Figure 6-14, Typical Variation of T/W_o With Characteristic Velocity for Lunar Launch (h_A = 100 Nautical Miles)



Another option would be to significantly reduce the minimum acceleration needed for the lander at deorbit. The penalties for doing this should be determined.

On the other hand, pump-fed, cryogenic engines may be able to function well in the 20:1 throttling ratio regime. Some individuals have made this claim. Less work has been done on storable engines with wide throttling ratios. The pump-fed engine may be required even at low throttling ratios because of cooling problems. The relationship between throttling ratio and engine cooling needs to be determined. In particular, the highest throttling ratio, pressure-fed, regeneratively cooled engine, that will work must be determined. If it is below 7 or 8, pressure-fed engines can be eliminated as candidates.

Table 6-11, Req. Thrust for Different Situations

Case	Vehicle Mass, kg	Lunar Weight,kg	Thrust Req. Newtons*	Thrust Req. Lbf*
93 km, 450 sec. Isp, O2/H2 Deorbit with 25 m ton cargo	58,000		159,146	35,665
400 km, 450 sec. Isp, O2/H2 Deorbit with 25 m ton cargo	60,000		164,634	37,000
93 km, 330 sec. Isp Deorbit with 25 m ton cargo	66,000		181,098	40,696
400 km, 330 sec. Isp Deorbit with 25 m ton cargo	70,000		192,073	43,163
93 km, 450 sec. Isp, O2/H2 40% of hover, near empty with 25 m ton cargo	32,000	5,333	20,884	4,693
40 % of hover, near empty with crew capsule only	12,000	2,000	7,832	1,760
Ascent to 93 km, 450 sec. Isp, 6 m ton crew capsule, Abort during descent	27,000	4,500	53,890	12,110
93 km, 450 sec. Isp Deorbit with 6 m ton crew capsule	32,000		87,805	19,731
93 km, 450 sec. Isp 40% of hover before normal landing	20,000	3,360	13,157	2,957
400 km, 330 sec. Isp Deorbit with 6 m ton crew capsule	45,000		123,476	27,747

* The deorbit cases assume an acceleration of 9 ft/sec sq. or 2.74 m/sec sq. is required at the start of the burn. The ascent case assumes an acceleration of 6 ft/sec.^{2} or 1.83 m/sec. sq. is required. The hover case assumes 40% of the lunar weight is the thrust.

Table 6-12, Comparison of Throttling Ratios

Max Thrust, lbf, Orbit alt., Isp, Prop. Situation	Min. Thrust, lbf, Situation	Throttling Ratio
37,000 400 km, 450 sec. Isp, O2/H2 Deorbit with 25 m ton cargo	1,760 40 % of hover, near empty with crew capsule only, abort to surface.	21:1
35,665 93 km, 450 sec. Isp, O2/H2 Deorbit with 25 m ton cargo	1,760 40 % of hover, near empty with crew capsule only, abort to the surface	20:1
37,000 400 km, 450 sec. Isp, O2/H2 Deorbit with 25 m ton cargo	2,957 93 km, 450 sec. Isp 40% of hover before normal landing, 6 m ton capsule	13:1
19,731 93 km, 450 sec. Isp Deorbit with 6 m ton crew capsule	2,957 93 km, 450 sec. Isp 40% of hover before normal landing, 6 m ton capsule	7:1
19,731 93 km, 450 sec. Isp Deorbit with 6 m ton crew capsule	1,760 40 % of hover, near empty with crew capsule only, abort to the surface	11:1
35,665 93 km, 450 sec. Isp, O2/H2 Deorbit with 25 m ton cargo	4,693 93 km, 450 sec. Isp, O2/H2 40% of hover, near empty with 25 m ton cargo	8:1
43,000 400 km, 330 sec. Isp Deorbit with 25 m ton cargo	1,760 40% of hover, near empty with crew capsule only, abort to the surface	24:1

6.6.2 Propellant and Inert Mass Requirements for Different Lander Tasks

The propellant and inert mass (less payload) at deorbit required for a single stage lander to do a variety of tasks is shown in Table 6-13. These numbers come from Table 6-1 thru 6-12. For low orbits, for the three tasks described, a single lander seems capable of handling them all. It will be somewhat oversized for the crew transfer task (6 m tons round trip). The vehicle doing the crew transfer will be required to carry around an additional 2 metric ton minimum of inert mass in order to be able to land the 25 m ton cargo. The 2 extra tons comes from using the scaling equations and the masses in Table 6-10. Section 8.0 addresses this problem in more detail, and also indicates penalties of 2 m tons or so of inert mass will result from a multi-purpose design.

For the higher lunar orbits, certainly 1,000 km circular and above, the vehicle size needed to perform the three tasks diverge considerably, indicating large inert mass penalties for trying to do the three tasks with one vehicle. The vehicles differ in size by as much as a factor of two. The large vehicle sizes for the L2 case indicate staging is needed.

As with other aerospace vehicles designed to perform multiple tasks, performance is degraded as compared to vehicles designed to do single tasks. Does the reduction in the number of vehicles that must be developed offset the cost of performance loss? To attack this problem the design, development, production, and operations costs of the various vehicles must be determined and compared to the additional launch costs and general sizing up costs of the whole system needed to accommodate the multipurpose vehicle. The scenario, or number and type of missions the set of vehicles must perform over their life history must also be defined.

Table 6-13, Propellant and Inert Mass Required for Different Tasks

Parking Orbit, km	Specific Impulse, lbf-sec/lbm	*-----Propellant + Inert Mass Required------------ 25 m tons Down, one way	6 m tons Round Trip	16 m tons Down, Inert Up
93	450	33	26	33
400	450	35	30	38
1,000	450	39	40	48
L2 (between Earth and Moon)	450	55	154	170
93	330	40	37	42
400	330	45	44	50
1,000	330	50	60	65
L2 (between Earth and Moon)	330	75	140	330

*Payload not included

6.6.3 Different Size Vehicles and the LEO to Lunar Orbit Transportation System

Table 6-14 shows the mass of the LEO stack needed to deal with several different possible lander/payload sizes.

The LEO stack masses of the 6 m ton round trip case and the 25 m ton down case differ by 50 to 70 metric tons. Only 26 tons of this difference is in the lander, therefore the additional 25 to 45 tons is added propellant and inert weight in the OTV(s). One option to reduce this would be to break the cargo lander into two parts and deliver them separately. Another would be to carry additional cargo to LLO with the small lander.

Table 6-14, LEO Stack for Different Lander Payloads

93 km parking orbit
455 sec. Isp aerobraked OTVs

--------- LEO Stack Mass --------------------				Lander Mass	Cargo Mass	Lander Isp, sec.
2 stage OTV--------- Load Propellants in:		1 Stage OTV-----------				
LLO	LEO	LLO	LEO			
176	176	192	192	58	25 one way	450
101	127	111	136	32	6 round trip	450
94	94	105	105	50	16 down, inert up	450
199	199	217	217	66	25 one way	330
137	159	148	169	43	6 round trip	330
117	117	128	128	58	16 down, inert up	330

6.7 Propellant Loading Locations

There are several options for lander propellant loading locations. In addition to propellant loading, the lander must be serviced with other consumables, maintained, and periodically tested. Three straight-forward options include:

1) Return the lander to the Space Station after each mission to the lunar surface and service and load it with propellants at the Space Station.

2) Load the lander with propellants in lunar orbit and service and maintain it in lunar orbit.

3) Load the lander with propellants on the lunar surface and service and maintain it on the lunar surface.

Additional options which are combinations of the above:

4) Load the lander with propellants and other consumables in low lunar orbit on most missions, but return it to the Space Station periodically (every 3 or 4 missions) for maintenance, testing, and inspection.

5) Load the lander with propellants in low lunar orbit and perform maintenance, testing, and inspection on the lunar surface. Expend or leave landers on the surface until the time that inspection and maintenance can be supported on the surface.

6) Load hydrogen in low lunar orbit and oxygen on the surface. Oxygen would be produced on the surface. Do maintenance and inspection on the lunar surface.

Each option has unique advantages and disadvantages:

1) Return the lander to the Space Station after each mission to the surface and service and load it with propellants at the Space Station.

Advantages:

a) The concept of maintenance and propellant transfer in space is new. If it can be done successfully, the Space Station will be the first place to try it. The Space Station will already have propellant loading, maintenance, and refurbishment facilities for the Orbital Transfer Vehicles (OTVs). The Space Station will have the largest stock of spares, most personnel, shortest logistics tail, etc. Maintenance man-hours in space will cost the least at the Space Station.

b) Development cost will be reduced in that facilities required for the OTVs can be designed to service the landers as well.

Disadvantages:

a) Bringing the lander back requires a larger stack in LEO. Table 6-15 illustrates this. Given the OTV transportation system described in section 15.0, bringing the lander back costs roughly 25% more LEO mass in one mission than loading propellants in lunar orbit. Loading propellants in lunar orbit will also have costs however. The lander will be left in a given orbit that the next mission must fly too. Some performance loss or loss in mission flexibility will be associated with this. If a facility is required in lunar orbit to handle propellant transfer, then the flights needed to place and support this facility represent a performance loss on the system.

b) It is difficult to integrate the lander with an aerobrake. If the lander cannot be aerobraked into LEO, even larger performance losses will occur. An OTV, specially configured to carry the lander will be required, or the lander will require its own aerobrake and will be an independent vehicle on return to Earth. Given a two stage OTV stack that would result in three separate vehicles aerobraking back to LEO after each mission.

2) Load the lander with propellants in lunar orbit and service and maintain it in lunar orbit.

Advantages:

a) Neglecting the facilities that may be needed to transfer propellants in lunar orbit, and the performance losses to the system incurred by the having to use a fixed lunar orbit, this option results in a low stack mass in LEO. If it is possible to design a lander that can be loaded with propellants, and other consumables and be maintained and checked out in lunar orbit without a fixed facility (a small lunar orbit Space Station), then this is an attractive option. There is debate about the practicality of basing a reusable vehicle at the Space Station however. The further away from Earth a vehicle is based, the more expensive and difficult maintenance, repair, and testing will become, if it is possible at all. Assuming it is possible and practical, the other performance losses would be associated with operation from a fixed orbit. These losses will go up as inclination of the lunar orbit goes up. If the base is equatorial, this will not be a problem.

Disadvantages:

a) Experience to date with vehicle maintenance makes it doubtful that a lander could be maintained in lunar orbit, particularly early in a program when the infrastructure at all locations is small. The lander could be simply expended after one or two missions however. It could also be expended with a one way trip to the surface with a large payload. Flights of this nature should be frequent, early in the program. Maintenance man-hours will be expensive in lunar orbit.

b) A small lunar orbit space station may be required. The lander must be powered and protected in lunar orbit. The penalty of taking this power supply and protection to the surface and back periodically will reduce performance. On the other hand, the power required may be low, on the order of a kilowatt or less. If the lander spends most of its time on the surface, meteoroid protection may not be a significant problem in orbit. The shelter will be needed on the lunar surface.

3) Load the lander with propellants on the lunar surface and service and maintain it on the lunar surface. The propellants are brought from LEO.

Advantages:

a) Loading propellants on the surface in a macro-gravity field eliminates the difficulties of zero-g transfer of propellants. On the other hand, zero-g transfer technology will be developed, regardless of where this lander is based.

Disadvantages:

a) Bringing all propellants from LEO to the lunar surface for the lander essentially doubles the number of flights. Performance is reduced by a factor of almost two (see Table 6-15). A lander loaded with propellant on the surface requires almost the same propellant load as it would if loaded in orbit. For the vehicles in the size range of interest to this study, a propellant load for a crew rotation round trip is on the order of 20 m tons for O_2/H_2 propellants to/from a 93 km orbit. A complete additional cargo flight is needed just to bring down this propellant. The total mass in low lunar orbit required to do a crew rotation is therefore 20 m tons propellant (to the surface) + 25 m tons propellant (to land the other propellant) + perhaps the inert mass and cost of one vehicle if it must be expended on the surface. See Table 6-15 for a comparison of this option to the others. The Table 6-15 numbers assume the propellant delivered to the surface comes down on an expendable lander. Any other scheme will cause the LEO stack to be even more massive.

b) This scheme is only appropriate for a crew rotation. Since the lander is based on the surface it cannot bring the same crew down and up. Two landers are required. A single lander cannot be loaded with propellants on the surface and then ascend with the propellant load needed to bring the next propellant load down without being unreasonably massive.

c) Maintenance man-hours will be more expensive on the lunar surface than at any other location.

Table 6-15, LEO Stack Mass for Several Propellant Loading/Basing Options

Load/Base in:	LEO	LLO*	LS* (prop. del from orbit)	LS* with Surf. O_2 ***	LS* with H_2 & O_2 on Surf.
2 Stage OTV					
Crew Rotation Mission (6 m tons round trip)	127	101	**180	58	44
25 m ton Cargo down, lander expended	176	176	------	------	------
14.0 m ton Cargo to surf. Unloaded lander returns to Orbit.	146	116	-------	71	54
1 Stage OTV					
Crew Rotation Mission (6 m tons round trip)	136	*111	194	65	49
25 m tons Cargo down, lander expended	192	192	-------	-------	-------
12.5 m tons Cargo to surface, Unloaded lander returns to orbit	161	127	-------	80	63

* OTVs carry 6 m ton crew module round trip. 93 km LLO, 450 km Space Station orbit.

** Expended 50 m ton lander delivers 22 m tons of propellant to the surface as cargo. Crew on surface loads a surface based lander with propellant and ascends with 6 m ton crew capsule to rotate surface crew.

*** H_2 loaded in orbit, O_2 on surface.

Appendix C shows the method and values used to calculate these numbers. "Rubber" OTVs are used for all the above cases.

6.8 Propellant and Engine, Type, and Number

The design of the lunar lander propulsion system is more challenging than for any previous manned space vehicle. While all (Mercury, Gemini, Apollo, and Shuttle) required a safe crew return, and the Shuttle required reusability with maintainability on Earth, the lunar lander will additionally need to be maintainable in space and/or on the lunar surface. It is mandatory that this requirement be incorporated in the initial design otherwise the lunar lander will result in a costly, unintentionally expendable vehicle.

The design philosophy for Apollo and the lunar lander are compared on Table 6-17. The primary considerations for both vehicles are simplicity of design and high reliability in order to decrease the probability of failure and loss of crew. To achieve these design goals could require some decrease in performance and increase in weight while still meeting mission objectives. These primary and secondary objectives for Apollo resulted in Earth storable propellants with hypergolic ignition, a single, pressure-fed ablative thrust chamber, and redundant components wherever practical. Cryogenic helium pressurant-storage was used for the descent propulsion system to decrease weight. If the cryogenic pressurant system failed, the ascent stage, which had ambiently stored helium, would be used for abort.

The propellants and engine for the lunar lander require more study before any selections can be made. Cryogenic propellants (requiring an igniter) are considered in addition to the Earth storable, hypergolic combination. A regeneratively cooled chamber becomes necessary instead of the non-reusable ablative chamber. Pump-fed engines are required since the weight penalties for high throttling ratios with pressure fed regenerative engines are prohibitive. This was not a problem for Apollo since the use of the ablative chamber allowed lower tank pressures and also less pressurant. Unlike Apollo, the lunar lander propulsion system will be reusable and will be maintainable in space and on the lunar surface.

The Adaptable Space Propulsion System (ASPS) studies and the Orbital Transfer Vehicle (OTV) studies have narrowed the propellants to N_2O_4/MMH and O_2/H_2 respectively using pump-fed engine cycles. Some of the technology effort for the ASPS and OTV engines is underway and more is planned. The lunar lander propulsion system can benefit from this technology to a great extent. However, a propulsion system designed especially for the lunar lander should also be studied and compared to determine the technical penalties of using the ASPS/OTV technology engines versus the cost and time penalties of developing another engine. Additional technology requirements resulting from the lunar lander studies could be added to the ASPS/OTV engine technology programs. This would decrease cost and development time for the lunar lander engine program.

6.8.1 Propellants

There are many propellant combinations to consider for the lunar lander study. For initial vehicle sizing the Earth storable combination N_2O_4/MMH and the cryogenic combination O_2/H_2 are selected. These propellant combinations are being studied for other space propulsion systems and experience has been gained by their use on operational spacecraft and booster vehicles. All the previous tables and figures can be used to compare the performance of these two propellants. In general, the O_2/H_2 lander and LEO stack is 10 to 30% lighter. The OTVs are all assured to be O_2/H_2. More study of the inert mass is needed to better qualify this difference however. A point design of an O_2/H_2 lander is needed to get good inert weights.

The performance (Isp) of the N_2O_4/MMH is lower than some of the space storable propellants that could be considered, however, it has a high bulk density which is a compensating factor. Thrust chamber cooling throughout the throttling range required for the lunar lander could be a problem depending upon the thrust level and chamber pressure required. The bulk density of O_2/H_2 is very low, however, the performance is the highest possible for a propellant combination acceptable for manned space missions.

The performance of F_2/H_2 and OF_2/H_2 is higher than O_2/H_2 but these combinations have never been used because of problems with material compatibility, toxicity, criticality of material selection and design, rigid cleaning requirements, and extreme reaction with many materials. There are some space storable propellants (OF_2/CH_4, F_2/NH_3, OF_2/MMH, etc) with performance higher than the Earth storable propellants but lower than F_2/H_2 and O_2/H_2. These are not recommended for manned space missions for reasons mentioned previously.

There are other propellant combinations to be investigated such as O_2/C_3H_8, and O_2/C_2H_4 which have higher performance than N_2O_4/MMH, however, the propellant bulk densities are lower. These combinations should be reviewed when the thrust chamber cooling requirements and performance are investigated for high throttling ratios. These propellants could take advantage of surface produced oxygen at some point in the future without the problems of long term hydrogen storage.

6.8.2 Pump vs. Pressure Fed

Pressure-fed propulsion systems with the Earth storable propellant combination N_2O_4/Aer50 were used for the Apollo spacecraft propulsion systems for simplicity and reliability. The Apollo descent stage thrust chamber (non-reusable) was ablatively cooled while the lunar lander thrust chamber (reusable) requires regenerative cooling. The estimated throttling for the lunar lander cannot be achieved with a pressure-fed system using a regeneratively cooled chamber and reasonable tank and system weights. Therefore, the lunar lander will be pump-fed unless some innovative method for thrust chamber cooling is discovered which would then allow a pressure-fed vs. pump-fed comparison.

Achieving the required throttling and cooling with an Earth storable propellant, pump-fed propulsion system will also be difficult and could prove unfeasible. The system would become too complex if two engine designs (different maximum thrust levels) and shutdown of engines became necessary to attain the overall thrust variation.

6.8.3 Number of Engines

The complexity of a pump-fed engine requires at least two engines for a manned space vehicle so that one engine failure will not result in loss of crew. Vehicle control system requirements and effective Isp must be considered in selecting the number of engines, i.e. thrust vector control and loss of Isp due to non-parallel engines if an engine fails.

Four engines have been tentatively selected for the initial study. The engine size is smaller than a two or three engine configuration and the throttling ratio is lower. The maximum thrust required for the O_2/H_2 lunar lander configuration is assumed to be 37,000 pounds. (See Table 6-11). For manned missions, if one engine fails during lunar descent the mission will be aborted to lunar orbit since redundancy would be lost for lunar launch. Thrust would be adequate with two of the four engines operating, but thrust vector control would be a problem. For unmanned missions, if one engine fails during lunar descent the mission will be continued to lunar landing since there is no problem with loss of crew, and at some point in the descent sufficient propellant will not be available to abort to lunar orbit. With these groundrules, the selected maximum thrust level for each of the four engines is 12,334 pounds. This results in a total maximum thrust of 37,000 pounds in the event one engine fails during the unmanned lunar descent, and the lunar lander still has the capability to land, where a normal landing determines minimum thrust on the lunar surface as planned. The throttling ratio required per engine is 13.4:1.

Another approach to obtain pump-fed engine redundancy is the use of a single thrust chamber with two sets of turbopumps and associated controls. This would result in a single thrust chamber of 37,000 lbs. thrust with a slight gain in performance (higher area ratio), simplification of the thrust vector control and a throttling of 10:1. Relying on a single, reusable, regeneratively cooled thrust chamber with the associated deterioration as missions are added would be one reason to negate this approach. An extremely critical inspection of this chamber would be required between missions if this engine system was selected.

Table 6-16, Preliminary Engine Characteristics

The engine characteristics to be used for initial vehicle sizing are:

	O_2/H_2	N_2O_4/MMH
Thrust (lb f.)	12,334	
Chamber Pressure (psia)	1,270	
Mixture Ratio (O/F)	6.0	1.9
Max Isp (sec)	460	340
Ave. 14:1 Isp (sec)	450	330
Nozzle Area Ratio	620	
Nozzle Exit Dia. (inches)	60	
Engine Length (inches)	115	
Weight (lb)	525	

The performance figures for $N_2 0_4$/MMH are satisfactory for preliminary vehicle sizing. Further information on engine cooling is required before additional engine characteristics can be determined. The use of a single, 37,000 pound-thrust, pump-fed engine with 10:1 throttling should be investigated since a large engine results in lower thrust chamber cooling requirements. This investigation should include the use of both propellants for thrust chamber cooling, the integration of redundant turbopump operation, and the possible requirement of a variable-area injector as used on the Apollo descent engine to improve performance throughout the throttling range.

The present technology goal for the OTV engine is an operational life of 500 starts/20 hours burn-time, and a service-free life to 100 starts/4 hours burn-time. Based on the Apollo Lunar Module burn times this would allow approximately 58 operational missions and 11 service-free missions. This is a goal. The Space Shuttle Main Engine (SSME) requires reservicing every mission and is effectively replaced every three missions.

Table 6-17, Lunar Lander Design Philosophy

	Apollo	Lunar Lander
Primary Considerations	Simplicity, Reliability	Simplicity, Reliability
Secondary Considerations	Performance, Weight	Performance, Weight
Propellants	Earth Storable, Hypergolic	1.Cryogenic, Igniter 2.Earth Storable, (Hypergolic)
Engine	Pressure-Fed, Single	Pump-fed, Multiple
Pressurant Storage	Ambient (When Practical) Cryogenic (Descent Stage)	1.Autogenous (Cryogenic) 2.Ambient (Earth Storable)
Thrust Chamber	Ablative	Regenerative (Reusable)
Components	Redundant (Where Practical)	Redundant (Where Practical)
Reusability	Expendable	Reusable
Maintainability	Earth	Earth, Space, and Lunar Surface

6.9 Reusability, Maintenance, and Repairability

Support of the lander for an extended period of time will require a different approach to all of the supportability disciplines than those that have been used for NASA manned spaceflight programs through the Space Shuttle era. This chapter will briefly discuss an approach to reusability, maintenance, and repairability considerations.

Technology available in the late 1980's can, in most cases, produce sufficiently reliable hardware and software to support the lunar lander scenario if proper management emphasis is given to it. The space environment is, in many ways, quite benign and conducive to long-life and high reliability.

The current state of technology may best be demonstrated by examining the current status of unmanned spacecraft operations. The majority of unmanned spacecraft placed in orbit that are geosynchronous or higher are currently designed for a life expectancy of 5 years or greater without maintenance, repairs or replenishing. The reliability of these spacecraft has been relatively high but somewhat less than would be acceptable for manned operations. Several spacecraft launched more than 10 years ago and accordingly reflecting technology approaching 20 years of age are continuing to operate and provide useful information well beyond their initial design life expectancy. Given the current state of technology and the scenario proposed for the lunar lander, it would appear that it is clearly within the present state-of-the-art to produce landers with the required reliability.

Past NASA manned space programs, most notably Apollo and Space Shuttle, have been initiated with the intent of providing in-flight maintenance capability, however these requirements were either deleted from the program or not pursued with sufficient rigor and dedication to provide meaningful results. It will be necessary for the supportability requirements to be given continuous high priority throughout the life cycle of the landers if it is to achieve the current goals of space basing and long useful life.

Many of the required techniques, of necessity, will have been developed by the Space Station program and will be available for use by the lunar lander. However, some lander systems will require materially different approaches and these are identified in the discussion below.

6.9.1 Reusability Approach

If true reusability with acceptable reliability is to be achieved, these considerations must be given high priority from program initiation onward. The current manned spacecraft redundancy requirements will, in general, provide sufficient reliability for the landers. To achieve high reliability it will be desirable to use proven technology in as many of the vehicle systems as possible and still meet the performance requirements. If the lunar lander is adequately maintained and repaired then the reusability goal can be met. The major exception may well occur in the main propulsion system inasmuch as high performance rocket engines with life expectancies of the order needed to satisfy the lander design requirements are not available. Designing to achieve efficient space based maintenance will give rise to new problems and require unique approaches to keep maintenance activity to an acceptable portion of the overall manpower available.

For the purposes of this discussion, it is assumed that the lander will be returned to the Space Station at the conclusion of each mission and that maintenance and repair will primarily be carried out on the Space Station. Limited maintenance and repair on the lunar surface should also be considered, however.

6.9.2 System Considerations

The system specifications must incorporate reusability, maintenance, and repair requirements and these requirements will play a significant role in determining the overall vehicle configuration as well as dictating many of the subsystem's hardware and software design features. Some of the overall system features will include things such as:

A. Tele-robotics - To reduce the EVA time required for maintenance and repair, the vehicle design should be compatible with tele-robotic servicing. The Space Station Flight Tele-robotic Services (FTS) and the Orbital Maneuvering Vehicle (OMV) will be available for use at the Space Station. It may well be desirable to provide variants of these vehicles at the lunar base. The vehicle maintainability design should accommodate the removal and replacement of externally mounted Orbital Replacement Units (ORUs) by the FTS and/or the return of the ORUs to a pressurized Space Station module to allow more efficient maintenance and repair of the units by the Space Station crew prior to replacement. This will require the development and qualification of highly reliable fluid disconnects, electrical connectors and mechanical attach fittings that can be reused in the space environment while retaining high reliability. For externally mounted ORUs this will be made more difficult by the contamination of equipment with lunar dust and debris.

B. Orbital Replacement Units Configuration - To facilitate Space Station based maintenance and repair, it will be desirable to have the lander systems installed in removable racks and canisters similar to those used on the Space Station. Maintenance and repair considerations for the Space Station will demand that Orbital Support Equipment (OSE) be available and if the Lander ORUs are compatible with the Space Station OSE it will eliminate the need for the design and development of lander unique equipment. During the conceptual design, trade studies should be made to determine the optimum split of system functions to be housed inside the lander's pressurized compartment versus those to be carried externally. Maintenance and repair should be driving factors in these trade studies as it may be desirable to provide the capability to directly dock the pressurized lander compartments to a Space Station module to allow shirt-sleeve accessibility for the maintenance, repair and systems validation functions.

C. Return of the ORUs to the Earth surface for repair and refurbishing - There will inevitably be cases where the OSE is incapable of supporting necessary unplanned repairs, and in these cases it may be desirable to provide the capability to return the equipment to the Earth's surface. To accomplish this, the lander's system should be designed to be compatible with the Space transportation system cargo handling equipment and capabilities to avoid the requirement for new OSE.

6.9.3 Subsystem Considerations

Many of the subsystem components can be either common usage with similar Space Station components or derivatives of them. In these cases, the hardware should have the inherent reusability and maintainability features necessary for the Lander. There are, however, other portions of the subsystems which have either new or more stringent requirements which will necessitate new design. Many of these more stringent requirements will be a direct result of the specified reusability and the demanding mission scenarios that have been adopted. The discussion below identifies some of the major components in the subsystem which will require unique development for the lunar lander.

A. Main Propulsion - For the purposes of this discussion, it is assumed that the main propulsion system is a H_2/O_2 system with pump-fed propellants. The development of engines and pressurization components to meet the lander requirements may well be the most demanding technical task required for the entire lunar program. The propulsion system must be capable of providing variable thrust with a long total operational time with reliability equaling or exceeding that of any rocket engine developed to date. The design of the propulsion system should provide redundancy where possible to reduce the individual component reliability demands while meeting the system values allocated for safety and reliability. The provision for maintainability will require that most active components can be removed and replaced, or repaired and this will demand the development of fluid disconnects with higher reliability than any produced to date. Electric connectors with similar characteristics will be needed, but those developed for electrical power system components will be usable in the propulsion system. If bladdered tanks are used they will undoubtedly need to be removable for replacement during the life of the lander. Other significant considerations for the propulsion system design requirements include the use of fracture mechanics to predict the life of pressurized components and the planned maintenance requirements such as replacement of elastomeric components during the life of the lander. To provide the needed reusability it may be desirable to have major segments of the propulsion system mounted on a common pallet which is removable as a unit for transport into the pressurized module of the Space Station where shirt-sleeve access is possible.

B. Propulsion Reaction Control System - Many of the comments made for the main propulsion system are also applicable to the Reaction Control System (RCS). However, if pressure-fed engines are used, the fracture mechanics aspects of the pressurization system design will be more significant. If the RCS uses Earth storable propellants, it may not be feasible to bring these components into a pressurized environment for maintenance and repair because of the toxic nature of these propellants. An alternative would be the direct transport of the modules to the Earth's surface where facilities with adequate safety provisions are available. If this approach is chosen it will undoubtedly require a higher level of sparing for this system and provisions for storage and validation of these spares must be provided in the Space Station. Another option would be to use O_2/H_2 RCS engines that can be maintained in a pressurized area in space.

C. Guidance, Navigation and Control - All major components of the Guidance, Navigation and Control (GN&C) system are within current technology and can provide the needed reliability. The major reusability consideration will be that of packaging and providing electrical and mechanical disconnects that meet the lander requirements.

D. Environmental Control and Life Support System - The Environmental Control and Life Support System components are expected to be within the envelope of technology developed for the Space Shuttle and Space Station programs. Weight considerations may lead to open loop operations and this may require components similar to those used in earlier programs. Such components will in general be more elementary in nature and will not require a significant development activity.

E. Structural, Thermal and Mechanical Systems - Components which are significantly affected by reusability considerations include the mechanical systems, energy absorbtion systems, lightweight actuation systems, multi-layer insulation and thermal systems.

1. Mechanical Systems - Active mechanical systems will be subjected to contamination from the lunar environment and will be subject to a large number of cycles. While this will influence the design it should not require state-of-the-art advances.

2. Energy Absorbtion Systems - Reusability will rule out some of the energy absorbtion systems employed in past programs such as crushable materials. Electro-mechanical energy absorption systems will have to be capable of operation in the lunar environment and this may require a significant development effort. Fluid energy absorbers may be the best choice, but there is little history of extended operations in space.

3. Actuation Systems - Pyrotechnics have been used for lightweight actuation systems on all of the previous manned space programs. However, their use in a reusable lander will result in large maintenance requirements and other actuation systems with comparable reliability will need to be developed.

4. Multi-layer Insulation - Multi-layer Insulation suitable for use over a long time period under lunar environmental conditions may require special developmental considerations. The ability to repair structural elements damaged during operations will require the development of new repair techniques and procedures. It may be necessary to partially disassemble the structure and return the components to the Earth's surface for repair. It will be necessary to give such requirements careful attention during the design process to assure compatibility with the Space Transportation System (STS).

F. Electrical Power System - If the primary power source for the lander is either fuel cells or batteries, it will be necessary to facilitate their removal and replacement. The development of electro-mechanical components such as circuit breakers, switches, etc. with life expectancies required for the lander may be difficult and trades and analyses will be needed to select components with adequate reliability and acceptable cost and weight. Early attention should be given to the development of the capability to remove and repair EPS components and wiring harnesses. The solution may result in requiring special design considerations in the electrical wiring portion of all subsystems.

G. Communications and Tracking - Communications and tracking are expected to be within the current state-of-the-art and the primary reusability concern will be easy removal and replacement of major elements.

6.9.4 Summary

Reusability, maintainability, and repairability requirements of the lander will present new and different problems. None of those presently identified appear to be insoluble. Proper solutions will demand early and continuing management emphasis and priority. The history of past programs indicates that maintaining the emphasis beyond the early stages of the program is extremely difficult due to other program demands such as budget, schedules, weight, etc. Keeping proper emphasis may well be the biggest challenge in this area.

As the conceptual design matures, continuing evaluation of reusability and maintainability will be necessary and the effort should be expanded to include all aspects of supportability. A cursory examination of the other supportability aspects is recommended at an early date.

7.0 Subsystem Studies

Conceptual designs are proposed for a variety of subsystems: reaction control, data management, guidance, navigation, and control, environmental control, electrical power, crew modules, and thermal control.

7.1 Reaction Control System (RCS)

A baseline proposed RCS system for the initial lunar lander study is generally described in this section. Much of the same design philosophy mentioned for the main propulsion system applies to the RCS. A less complex design and high reliability must be paramount even at the expense of lower performance and higher weight. The use of redundant components (where practical) will increase reliability. Indirect pump-fed systems appear viable contenders for the RCS, especially with O_2/H_2 as main propellants. Reusable, radiation-cooled chambers are most likely candidates for the engines, but technology effort is required to extend the life of these chambers. As with the main propulsion system, long-life reusability and space maintainability will be an enormous new challenge for manned spacecraft RCS design.

The RCS propellants for the O_2/H_2 lunar lander are proposed to be also O_2/H_2 and are loaded into the main propellant tanks. Liquid propellants are extracted from the main tanks, pumped to a higher pressure, gasified by passing through a heat exchanger, and then stored in accumulator tanks as gases to be used in gas/gas RCS thrust chambers. The gas generators to operate the turbopumps use gaseous oxygen/gaseous hydrogen and the exhaust gases are passed through the heat exchanger to gasify the liquid oxygen and liquid hydrogen as mentioned previously. Sixteen thrusters are located in four clusters 90° apart, four engines per cluster, to supply the required control and translation thrust. The thrust of each RCS engine is approximately 100 to 150 pounds depending upon vehicle requirements. The Isp is 370 seconds, steady state.

The RCS propellants for the Earth-storable Lunar Lander are the same as for the main engine, N_2O_4/MMH with separate RCS propellant storage tanks and pressurization system. The engines are pressure fed and the Isp is about 280 seconds, steady state.

Integrating the N_2O_4/MMH main propulsion system and the RCS resulting in smaller RCS tanks and the elimination of the RCS pressurization system is a possibility and warrants investigation.

7.2 Lunar Lander Data Man. System/Guidance, Nav. & Control (DMS/GN&C)

The following key mission requirements were considered in arriving at a conceptual design for the lunar lander DMS/GN&C.

- Checkout and initialization of the lunar lander in lunar orbit or at the Space Station.
- Separation of the lunar lander from the OTV and orientation for the thrusting phase.
- Insertion of the lunar lander into the descent transfer orbit.
- Powered descent to the desired landing site.
- Determination of the launch parameters for insertion into a lunar parking orbit.
- Powered ascent to provide intercept with the OTV at a predetermined aim point.
- Mid-course corrections to reduce dispersions about the aim point.
- Terminal rendezvous and docking with the OTV.

Certain other requirements must be considered if the program objectives are to be met.

1) Manned/Unmanned Operation.

The unmanned operation presents requirements for totally automatic operation with sophisticated fault detection and reconfiguration without the aid of man in the decision making process. Previous NASA programs involving manned operation have taken the approach of implementing manual control and intervention for all mission phases where at all possible. To avoid making the vehicle over complex and to control escalating costs due to software development, simulation facilities, crew training etc; it is proposed that the manual intervention and control be implemented only where the manned decision making capability can significantly augment the automatic system. (eg. abort decisions, final touchdown, and possibly final berthing.)

2) Commonality

It would appear that wherever feasible, the lunar lander design should incorporate components used in the implementation of the NASA Space Station. To the degree that is possible, the program advantages of reduced training, sparing, and support equipment would be multiplied many times over when considered in light of a Space Station based operation. Therefore reasonable predictions have been made of the technology that might be used in both programs.

3) Failure Tolerance

Projections of technology available in the lunar lander time frame indicate further significant reductions in the weight and power of typical avionics components particularly in processors and inertial measurement units (IMUs,) and consequently lower penalties associated with the implementation of multiple identical redundancy. To take full advantage of the simplicity of "effector voting " in the implementation of the (FDIR) fault detection, identification, and reconfiguration a fully triple redundant system is proposed in most areas.

4) Checkout/Maintainability/Testability

Requirements for operation in a space based environment make these functions even more critical than in past programs. The vehicle must be designed from the onset to be entirely self-checking and rely on onboard calibration. Most of the maintainability functions specified for the Space Station are also applicable to the lunar lander.

In addition, the lunar lander design must support a capability for autonomous launch. The Apollo Program demonstrated many aspects of the capabilities needed to launch and operate a vehicle without the benefit of a costly launch check-out facility. With the advances in expert system design and the increases in onboard computer power the autonomous checkout goals should be readily achievable but require that these functions are recognized as primary requirements.

7.2.1 Data Management System Configuration

The DMS is defined as the redundant central processing system, multi purpose displays, data bus network, and general purpose multiplexor-demultiplexors. The software system is also included. Although the DPS processors accomplish the principal function processing, processors are implemented at the subsystem or black box level to perform data compression, FDIR functions and other functions amenable to local processing.

These local processors would be procured to be card compatible with the main processor.
All items required to interface with the standard data bus are procured with a built in Data Bus Interface.

The DMS processor is a 32 bit machine derived from a commercial chip to capitalize on the advantages of off-the-shelf software, support tools, and the many other advantages that accrue from having a readily available ground version of the onboard machine. For the purpose of this conceptual design a version of the Intel 80386 micro-processor was assumed.

Two multi-purpose displays are implemented using flat screen plasma technology. The Operations Management Software supports the monitoring of on-board consumables, system configuration, failure status, and displays this information for the benefit of Space Station checkout crews, or when applicable, to the lunar lander crew members.

The display system also supports the flight displays for mission phases when manual control is available.

The multiplexer-demultiplexor (MDM) is a general purpose, reconfigurable device incorporating a micro-processor for maximum flexibility in dealing with all types of subsystem interfaces. In most cases the MDM discretes, analogs and digital serial data can interface directly with the subsystem effectors and sensors without the necessity for signal conditioning.

7.2.2 GN&C Configuration

The following assumptions are made with respect to interfacing subsystems and arriving at a GN&C configuration.

- Four quadruple RCS jets provide translational and rotatational control torques.
- The same main engines are used for ascent and descent. A minimum of two are provided for redundancy. The engines are provided with slow trim gimbals to deal with offset center of gravity problems and reduce RCS activity.
- Navigation Sensors capable of providing information to update the inertial system are provided by the Communication System. (These sensors are, however, discussed in this section for completeness).
- Intelligence for the automatic sequencing operations would be handled by the GN&C processors.

The IMU proposed is a strapped down system based on ring laser gyro technology. This approach is chosen because of advantages in cost, ruggedness, stability, and ease of integration with optical alignment devices. Projected advances over the next few years also show a clear advantage in weight and power over other types of inertial systems. The ring laser gyro is readily adaptable to a "Hexad" configuration which provides the maximum redundancy for the least weight and power. The "Hexad" configuration contains a built-in, triple redundant inertial sensor assembly (ISA) processor which does the strapdown computations, sensor calibration, redundancy management, checkout, and other local processing assignments. The ISA processor also calculates the vehicle attitude and vehicle body rates required for control system stabilization.

Alignment of the IMU will be required prior to descent and ascent to minimize errors and delta V expenditure. This is accomplished by an automatic star scanner attached to the case of the IMU to minimize boresight errors.

Guidance functions, control equations, jet select logic, and similar processes are mechanized in the DMS processor. To the maximum extent possible these and other critical functions will be implemented in read-only-memory (ROM) to provide the maximum reliability and lowest power and weight penalties. Commands to the main engines and RCS engines are transmitted via the triple redundant data bus to the control electronics sections where electrical voting takes place before transmittal of the command to the actual effectors.

Automatic docking of the lunar lander with the OTV is a requirement, however the OTV is assumed to be equipped with the sensors and intelligence to accomplish this operation and no provision is made on the lunar lander to duplicate this capability. Wherever the capability resides, it must be developed. The sensors and software to do automatic docking do not exist at this time in the free world.

To effect the precision required for automatic touchdown on the lunar surface and to ensure successful rendezvous the onboard inertial system will require updating by some form of non-inertial navigation sensor. This is probably the area in the GN&C requiring the most detailed trade studies.

Lunar navigation requires more and more precision as the vehicle approaches the landing site. First, the orbital parameters and the location of the vehicle and the orbit relative to the landing site must be determined. The vehicle must transfer to an orbit that allows an optimum descent to the site. Once in the vicinity of the site, the vehicle must maneuver to end up roughly over the site with the relative velocity nulled. The vehicle must then descend to land on a location with no more than a few meters of error.

This sequence could be described as orbital, terminal, and landing navigation. A great variety of possible systems exist to support lunar navigation. Table 7-1 lists most of these and notes advantages and disadvantages.

The lander will have an inertial platform and keep track of location with that, but updates will be required. The inertial navigation will be inadequate for the terminal and landing phases. Given adequate orbital and terminal navigation, the crew can handle the landing navigation in the right lighting conditions. The unmanned cargo lander requires a good landing accuracy navigation system for all flights however.

The Apollo landers used a combination of Earth based radar, crew recognition of local features, space sextant work, and inertial navigation to achieve an impressive accuracy. In addition, the vehicles had radar altimeter and radar measured relative velocity. The radar altimeter was used to determine certain checkpoints later in the program. The crew always mounted the landing navigation visually at a minimum. The missions were constrained to having the proper lighting conditions for visual landing.

Table 7-1 shows a variety of possible systems for updating the onboard inertial system and doing landing navigation. The preferred system is the cruise missile type terrain following radar with surface based transponders if required. The basic elements of this system will all be part of the landers anyway, and depending on the surface features and the knowledge of their positions, no surface elements at all may be required. A small surface based radar would be a low cost addition to the onboard terrain following system.

The first requirement for terrain following type navigation is knowledge of the terrain features location to within a certain range of error. As the landing site is approached, this knowledge must become increasingly precise if surface transponders are to be avoided. In addition there must be terrain features with good echos near the landing site. On the other hand, there will be pressure to locate the site on a plain, to improve safety. In any event, the general area of the site must be mapped well enough to allow good terminal navigation. If the first landings on the site are unmanned, a certain element of risk may exist in the initial landings, in the absence of good landing navigation from visual or other sources. If the first landings on the site are manned, they must occur during lighting conditions allowing good visual landing navigation. The first landers can carry a transponder and if required, place another on the surface at a known location. Subsequent landings will then get positions relative to these transponder(s).

Once there is a crew on the surface during cargo landings, a surface based radar can be installed. This is anticipated to be a small unit, with a dish less than a few feet in diameter. It will track the transponder on the incoming lander. The crew on the surface may be provided with the option of terminating the thrust on the incoming unmanned lander if prediction software indicates deviations that may lead to damage to surface equipment. The crew may also give relative position updates to the vehicles that pass overhead in orbit.

For the purpose of generating numbers for power, weight, and volume for the GN&C, LM type Radars were assumed. Figure 7-1 shows the block diagram of the DMS/GN&C and Table 7-2 provides weight, volume, and power estimates.

Table 7-1, Navigation System Advantages and Disadvantages

System	Advantages	Disadvantages
Lunar Orbit Global Positioning Satellite (GPS) type system.	Terminal, perhaps landing accuracy nav. over entire surface.	Many satellites required. Expensive to place.
Earth orbit GPS system or Earth based radar.	Nothing to place or power on lunar surface. Good for orbit determination on the near side.	Accuracy limited. Not adequate for touchdown navigation. GPS accuracy unknown. May require large antenna. Earth side only.
Long and Medium Range Lunar Surface Transmitters: TACAN, LORAN, low feq.	Several low freq. transmitters may provide low accuracy global coverage. Can be placed and powered at base for local nav. and orbit updates. Terminal accuracy.	Heavy ground stations. Large antennae. Accurate over a limited range only. Low feq. does not provide high accuracy for any location. Low freq. global coverage requires several transmitters at different places.
Instrument Landing System or Microwave Landing System at base	Can be placed and powered at base. Landing accuracy.	Terminal and landing nav. only for area close to transmitter.
Lunar Surface Based Radar (located at base)	Enables range safety termination. Can provide updates to vehicles in orbit. Low mass system.	Local area Nav. only.
Cruise missile type onboard terrain matching radar on lander. Transponders on surface if required.	Transponders only on surface in landing area. Very low mass. Landing accuracy nav. probable over entire surface.	Landing accuracy depends on accuracy of surface feature maps.

Figure 7-1, Data Management System

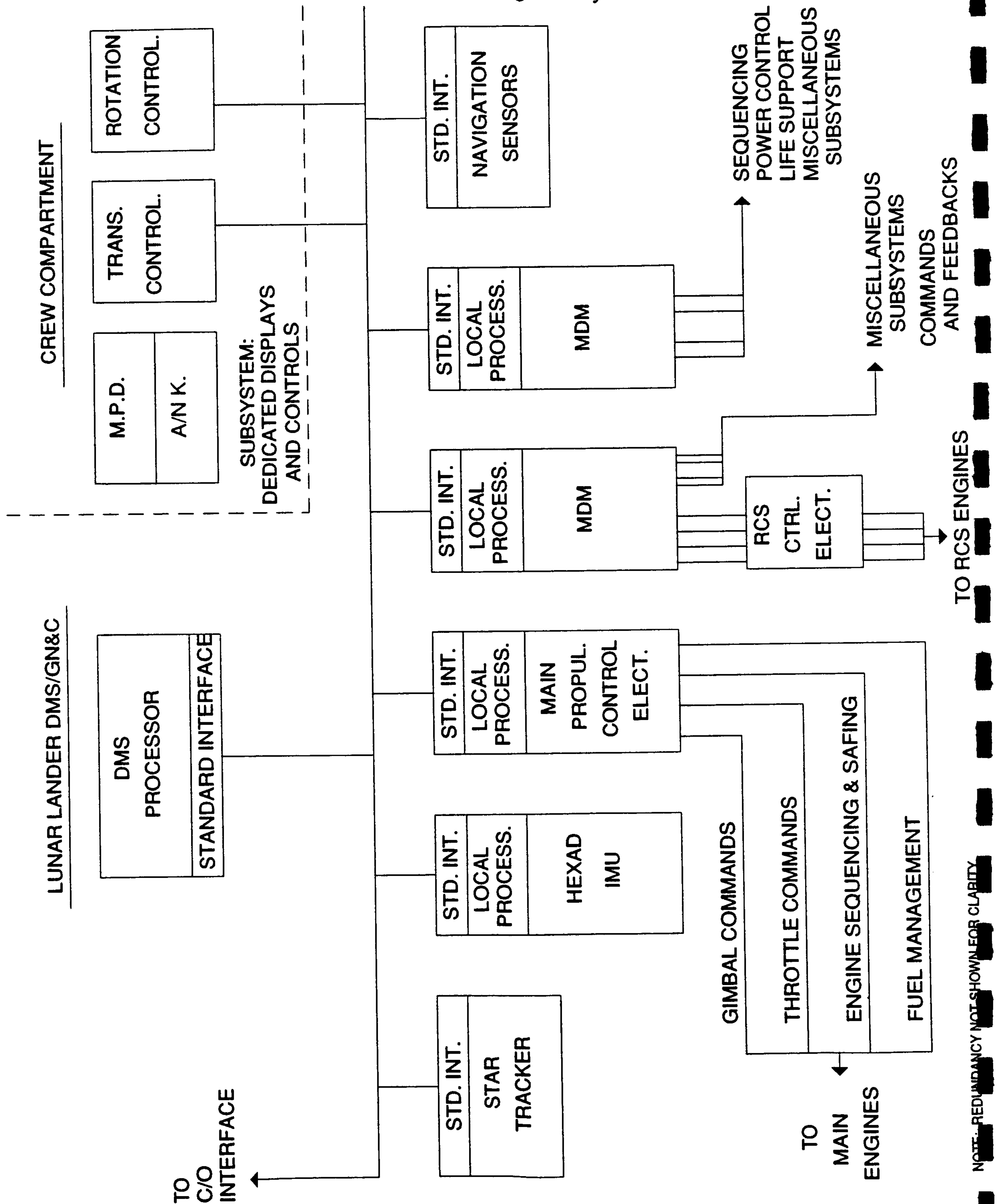


Table 7-2, Lunar Lander DMS/GN&C

Component	Unit (Veh) Weight,kg	Unit (Veh) Power,Watts	Unit cubic ft. Volume (Config)	Number/Veh.
DMS Processor	10 (30)	75 (225)	.27 (.81)	3
MDM	7.7 (46.4)	60 (360)	.25 (1.5)	6
*ANK/Display	8.6 (17.3)	40 (80)	.35 (.7)	2
Hexad IMU	16 (16)	75 (75)	.3 (.3)	1
Star Track	2 (6.1)	10 (30)	.1 (.3)	3
Nav. Sensors				
Landing	13.2 (13.2)	100 (100)	.4 (.4)	1
Rendezvous	20.5 (20.5)	200 (200)	.6 (.6)	1

Total Weight = 149.3 kg (328.5 lbs.)

Total Power = 1,070 W.

Total Vol. = .13 cubic meters (4.61 cubic ft.)

*ANK = alpha-numeric keyboard

7.3 Lunar Lander ECLSS

The Environmental Control and Life Support System (ECLSS) for the lunar lander is designed to provide atmosphere pressurization, revitalization, and temperature control. It also provides food, water, waste management, an airlock, and extra-vehicular mobility unit (EMU) or space suit support. Crew and crew provisions are also considered to be included. The conceptual ECLSS is based on supporting a crew of four on a round trip from low Earth orbit to the Moon and back to low Earth orbit. The system must also be capable of supporting a crew of six on a one way trip from low Earth orbit to the surface of the Moon. Given the debate as to whether the lander crew module must support the crew for the whole trip or just from LLO and back, consumables are parameterized and several options are presented in the tables. The airlock information is based on the STS airlock with an interior volume of 150 cu. ft. Information on the open loop system was derived from STS data while the closed loop system was derived from Space Station data.

Some of the assumptions that were made in the sizing of the system are as follows:

- Personal hygiene accommodations will be similar to the STS design
- Each crewperson will have an EMU
- The EMU will be an open loop system
- An airlock will be included in the system
- Surface stay time will be 3 days
- Cabin pressure will be 1 standard atmosphere
- All critical subsystems will be redundant

Comparison of open and closed systems to determine the cross over point where it pays to go from open loop to a partially closed loop is dependent on several factors. These factors include mass, volume, energy, and operational considerations. From the mass standpoint, the crossover point was approximately 60 days for the atmosphere revitalization system, and 35 days for the water management system. Neither of these two comparisons took into account the impact on other subsystems such as power and thermal control. With the identified power requirements, these impacts should be added to the ECLSS mass impacts to arrive at a reasonable mass break even point. The breakeven point will be at an even longer stay time when the additional power system mass required is considered. For these reasons, the system design selected was the open loop configuration.

The choice of power generation methods can also bias the choice of ECLSS design selection. If fuel cells are used to generate electricity, then the process byproduct, water, makes the choice for water management prefer the open loop concept.

The atmosphere supply and pressurization system source consists of tanks of gaseous high pressure nitrogen and oxygen. If fuel cells are used for electrical power, then the system would get oxygen from a common cryogenic supply tank. These sources are fed through regulators to support the cabin, crew suits, airlock, and EMU service station. Provisions are available for cabin and airlock depressurization and repressurization. Equalization valves are available at each pressure volume interface. Partial pressure sensors will be connected to the regulators to maintain the proper atmosphere composition mix.

Atmosphere revitalization is supported by LiOH canisters for CO_2 removal. Odors and particulates will be removed by activated charcoal and filters. Cabin fans provide the necessary circulation of the atmosphere through the system and habitable volume. Humidity and temperature control will be handled by heat exchangers and water separators. Thermal control for other equipment in the crew compartment will be handled by cold plates and a water loop connected to the thermal control system. Included in this subsystem will be the fire detection and suppression system.

The water management system provides potable water for crew usage, food preparation, and EMU support. Waste water will be collected from the atmosphere revitalization system and stored in a waste water tank or distributed to the thermal control system to assist in providing flash evaporation heat removal.

The waste management system provides for the collection and storage of fecal and urine wastes. Provisions are also available for food waste and loose trash collection and storage.

Food management is supported by provisions for storing, preparing, and warming of crew food.

EMU support is provided for EMU equipment recharging and servicing. Provisions will be available to service two EMU units in a 12 hour period. The airlock is sized to accommodate two suited crewmembers per cycle. With the crew in the airlock, the repressurization air volume is 130 cubic feet. The airlock is also an open loop system. The air is not recovered.

Reduction of cabin pressure should be considered for the lunar lander as well as the lunar base. This has the advantage of reducing pressure vessel masses, airlock atmosphere losses, airlock cycle times, and acclimation times for adaptation to lower pressure systems.

Table 7-3 shows estimated masses for the ECLSS components, power requirements, consumables usages, and crew provisions. The masses were derived from shuttle and Space Station numbers. An additional requirement not shown in the table is for cooling fluids. The current space suit requires 8 lbm/EVA hr of water for evaporating cooling.

Future suits for the lunar surface will be driven toward less consumable intensive thermal control systems. Water boilers may also be needed for lander thermal control for some time period. This water is not accounted for in these tables.

As a point of reference, a partially closed loop system is estimated to require on the order of 4 kilowatts of power and have hardware masses of around 3 metric tons.

Table 7-4 shows the total ECLSS mass for several crew/trip time situations.

Table 7-5 shows the Shuttle ECLSS power requirements, itemized by systems that might be comparable to lunar lander systems. The average power required based on this table is 1.81 kw. The Shuttle is designed for a nominal crew of 7 with a contingency of 10. The lander crew module holds 4 with a contingency of 6. The power requirement is assumed to be roughly linear with crew downsized by 4/7, resulting in a requirement for 1.0 kw ave. power. Increased efficiency in motor design and advanced cooling techniques occurring over the 20-30 year interval between the two vehicles is expected to result in some savings as well.

Table 7-3,Lunar Lander ECLSS

Hardware	Open Loop lbm	kg
Atm Press & Comp Control	400	182
Atmosphere Revitalization	300	136
ECS Heat Transfer	300	136
Food Management	200	91
Water Management	250	114
Waste Management	280	127
Fire Detection	50	23
Airlock	1,000	455
H/W Total	2,780	1,264
Fluids		
Coolant Fluids	200	91
System Water	200	91
Cabin Air	70	32
Fluids Total	470	214
Power	1.0 Kilowatts	
Crew Provisions (4 Crew)		
Crew	680	309
EMU	1,200	545
Seats And Mobility Aids	600	273
Crew Support Provisions	1,280	582
Crew Provisions Total	3,760	1,709 or 427 kg/crew
Consumables		
Nitrogen-Leakage	3.5 lbm/day	1.6 kg/day
Oxygen-Leakage	1.5 lbm/day	.7 kg/day
Oxygen-Metabolic	2.0 lbm/md	.9 kg/man-day
LiOH	5.0 lbm/md	2.3 kg/man-day
Water	8.7 lbm/md	4.0 kg/man-day
Food	5.0 lbm/md	2.3 kg/man-day
Airloss-Airlock	10.0 lbm/cy.	4.5 kg/cycle
Total	20.7 lbm/md + 5.0 lbm/day +10.0 lbm/air lock cy.	9.4 kg/man day 2.3 kg/day 4.5 kg/airlock cycle

Table 7-4, Open Loop ECLSS Mass Required

No. of Crew	Support Time, Days	Consumables (3 airlock cycl.) kg	Hardware kg	Fluids kg	Crew Prov. (+ crew mass) kg	Total kg
6	1	72	1,264	214	2,562	4,112
4	3	133	1,264	214	1,708	3,319
6	15	894	1,264	214	2,562	4,934
4	15	612	1,264	214	1,708	3,798

Table 7-5, Environmental Control and Life Support Power (Reference 9)

Component Description	Component DC Load (Watts)	Average Use Factor (Percent)	Total Comp On Time (Hours)	Energy Required (Kw-hours)
Cabin Fan B	482.90	124.9999	168.7497	101.86
Cab Air Temp Cnt PRI	17.90	12.5000	162.0000	.362
Cab Tmp CN El-PR	3.90	124.9999	168.7497	.822
Cab Air Tmp CN El-SC	3.70	124.9998	168.7497	.780
Cab Air Signal Cond	1.60	124.9997	168.7497	.337
ARS Humidity Sep B	32.10	124.9998	168.7497	6.771
ARS Hum Sep Sig CND	.80	124.9998	164.4944	.164
PPO2 Cntlr-Sys 1	.70	99.9999	168.7497	.118
PPO2 Cntrl-Sys 2	.70	99.9999	168.7497	.118
02 Supply Vlv-SYS 1	9.40	49.9999	168.7497	.793
02 Supply Vlv-SYS 2	9.40	49.9999	168.7497	.793
02 Xover Vlv-SYS 1	11.20	99.9999	168.7497	1.889
02 Xover Vlv-SYS 2	11.20	99.9999	168.7497	1.889
Cabin Press Sensor	.70	99.9999	168.7497	.118
Cab Pres Decay Sensr	2.00	100.0000	168.7497	.337
H20 Byp Loop 1 Sen	.20	99.9999	168.7497	.0337
H20 Byp Loop 2 Sen	.20	99.9999	168.7497	.0337
IMU Air Delta P Sen	.20	99.9999	168.7497	.0337
02 Flow Sensor-Sys 1	1.00	100.0000	168.7497	.168
02 Flow Sensor-Sys 2	1.00	100.0000	168.7497	.168
N2 Flow Sensor-Sys 1	1.00	100.0000	168.7497	.168
N2 Flow Sensor-Sys 2	1.00	100.0000	168.7497	.168
PPO2 Sensor-Sys 1	.80	99.9999	168.7497	.134
PPO2 Sensor-Sys 2	.80	99.9999	168.7497	.134
PPO2 Sensor-Sys 3	.80	99.9999	168.7497	.134
Avion Fan-Bay 1 (B)	173.90	124.9998	168.7497	36.68
Avion Fan-Bay 2 (A)	173.90	124.9998	168.7497	36.68
Avion Fan-Bay 3 (B)	173.90	124.9998	168.7497	36.68
Avion Bay 1 Sig Cond	2.10	124.9999	168.7497	.442
Avion Bay 2 Sig Cond	2.10	124.9999	168.7497	.442
Page Subtotal				229.314

Table 7-5, Continued

Component Description	Component DC Load (Watts)	Average Use Factor (Percent)	Total On Time (Hours)	Energy Required (Kw- Hours)
Avion Bay 3 Sig Cond	2.10	124.9999	168.7497	.442
Smoke DT SNR-L Flt D	8.90	99.9999	168.7497	1.501
Smoke DT SNR-R Flt D	8.90	99.9999	168.7497	1.501
Smoke DT SSR A-Bay 1	8.90	99.9999	168.7497	1.501
Smoke DT SSR B-Bay 1	8.90	99.9999	168.7497	1.501
Smoke DT SSR A-Bay 2	8.90	99.9999	168.7497	1.501
Smoke DT SSR B-Bay 2	8.90	99.9999	168.7497	1.501
Smoke DT SSR A-Bay 3	8.90	99.9999	168.7497	1.501
Smoke DT SSR B-Bay 3	8.90	99.9999	168.7497	1.501
Smoke DT SSR-Cabin	8.90	99.9999	168.7497	1.501
IMU Fan B	47.90	124.9999	168.7497	10.103
IMU Fan Sig Cond	1.80	124.9999	164.4944	.370
H20 Pump - Loop 1 (B	198.00	125.0000	.4000	.0989
H20 Pump - Loop 2	198.00	124.9998	168.7497	41.765
H20 Bypass CN SC-Pri	5.90	124.9998	168.7497	1.244
H20 Bypass CN SC-Sec	5.90	124.9998	168.7497	1.244
Wtr Sep Waste Sys 1	116.70	125.0000	3.5000	.510
Foodwarmer-DBL PHA	254.00	47.6667	17.5000	2.118
Foodwarmer-DBL PHC	254.00	47.6667	17.5000	2.118
Vacuum Vnt Noz Htr	11.40	99.9999	161.5000	1.841
Vacuum Vnt Lne Htr A	28.50	10.1497	150.1650	.434
Pot H20 Noz Htr	21.20	100.0000	13.3000	.281
Pot H20 Dump Ln Htr A	13.10	30.4627	163.5813	.652
Waste Nozzle Htr	22.00	100.0000	.5333	.0117
Waste Dump Line Htr A	15.20	3.0000	143.5000	.0654
Page Subtotal				76.807
Total				306.121

306.121 kw hours/168.75 hrs = 1.81 kw ave. power req.

7.4 Structure

Structural mass is estimated using the scaling equations developed in Section 6.1. The structure is scaled as 45% of the 6.4% of gross or deorbit mass, or .0288 x gross mass. Simply stated it is scaled as 2.88% of the deorbit mass.

7.4.1 Lander Size

The physical size of the lunar lander can be estimated by similarity, using the Lunar Module (LM) of Apollo as a starting point. Storable propellants must be used however. Figure 7-2 from Ref. 5 shows the overall LM dimensions. Selecting a cube in which the LM fits, from footpad to footpad and from the ground to the top of the cabin, the resulting characteristic size, i.e. the side of the cube, D, is approximately 6 meters.

Fig. 7-3 illustrates the geometrical scaling. Assuming that the systems are similar, the characteristic size, D, increase with the 1/3 to 1/2 power of the gross mass, G.. The actual increase is closer to the 1/2 power. Consequently, the estimated characteristic size versus gross mass is:

G, Gross mass, tons	D, Characteristic size, meters
15 (LEM)	6
30	8.2
100	14.5

A more exact method involves laying out appropriately sized propellant tanks, landing gear, etc., as was done in Section 9.0.

7.4.2 Number of Legs

Four legs is the recommended minimum because of stability considerations. It is less likely that a four legged vehicle will flip over landing on an uneven site, such as crater rim, etc. The LM crew could not see the foot pads. One Apollo mission put a foot pad in a small crater and damaged the descent engine. Initial Apollo lander designs had six legs, but this was reduced to four to save weight.

Other Considerations

- O Structural strength - avoid using excessively long legs with pivotal points close to the vehicle center.
- O Foldability and ability for stowage in transport vehicle. The lander should fit in a 30 ft diameter shroud proposed for a heavy lift vehicle.
- O Hydraulics for deployment if required.
- O Shock absorbing capability for vertical and also lateral loads.

7.4.3 Footpad Size

The lander maximum mass on the surface will be in the range of 31 metric tons (25 m tons cargo + 6 m tons inert). Given a footpad design pressure of 1 psi (Ref. 6), and assuming only 3 footpads are touching the surface, circular pads must be a minimum of 1.8 meters in diameter.

Figure 7-2, Apollo Lunar Module

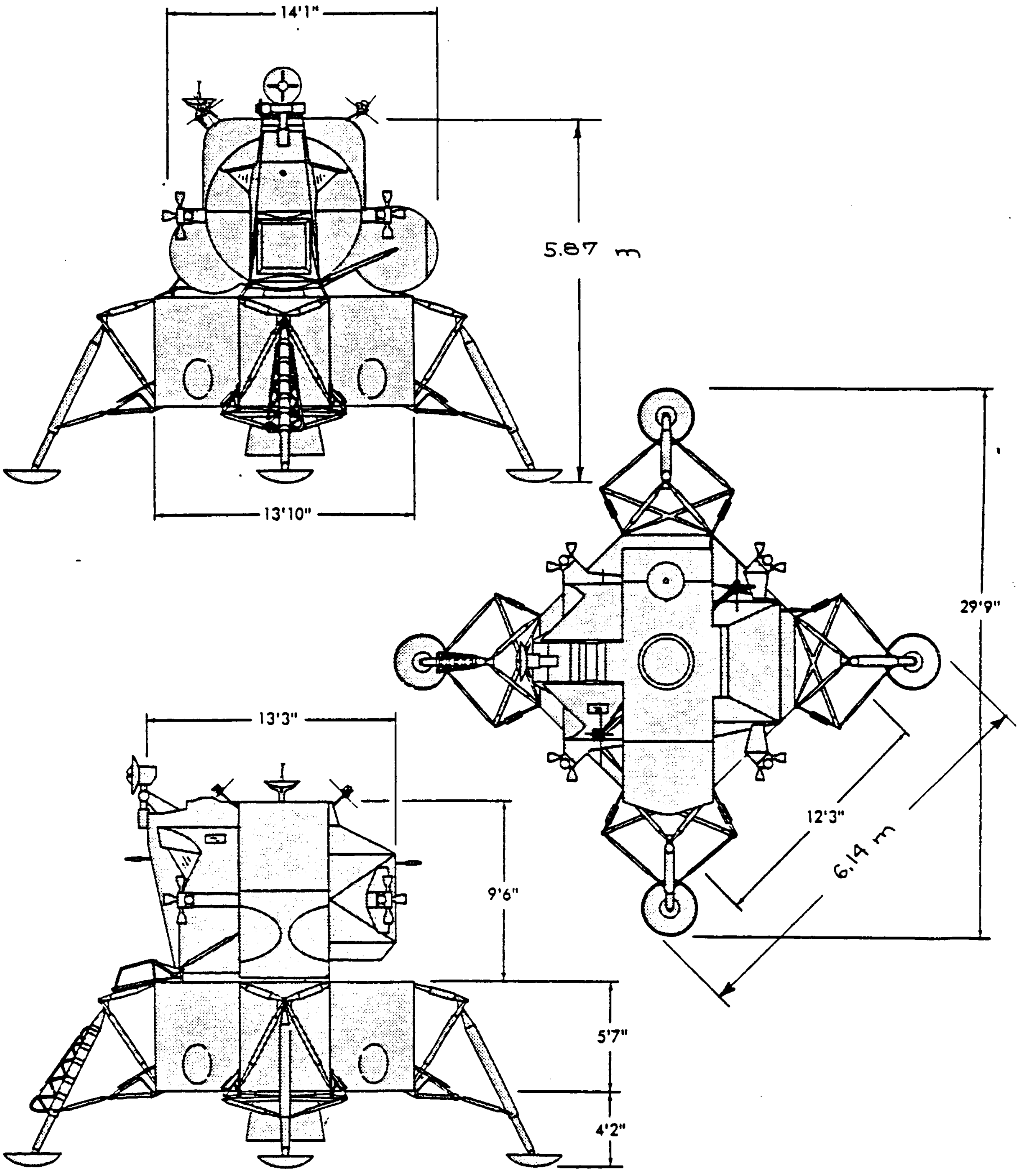
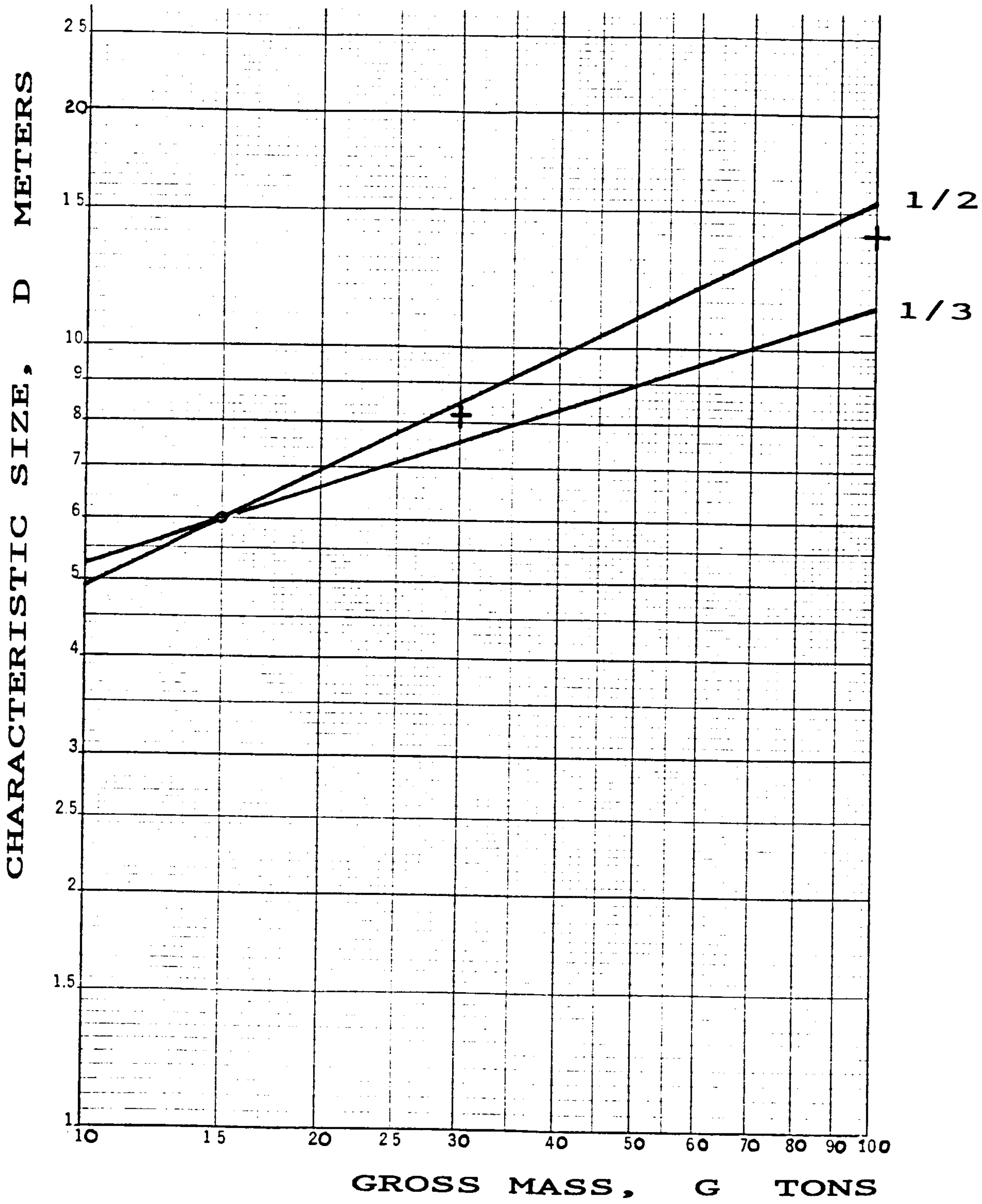


Figure 7-3, Geometric Sizing



7.5 Electrical Power Options

Two scenarios have been discussed with respect to the crew module. In one scenario the crew only enters the module to descend to the surface and lives in another module in-orbit. In the second scenario, the crew lives in the lander module for the complete trip, estimated to be 15 days minimum.

For this reason the lunar lander mission is broken down into two scenarios as regards the electric energy storage provisions:

1. Power up in Lunar Orbit; descent, 3 days on surface; ascent to Lunar Orbit - 144 kWh at 2 kW average.

2. Power up in Low Earth Orbit, 1 day; 3 days to Lunar Orbit; 1 day in Lunar Orbit; descent, 3 days on surface; ascent, 1 day in Lunar Orbit; 3 days to LEO; 3 days in LEO - 720 kWh at 2 kW average. (15 days)

The lander may stay much longer than three days on the surface, but it is assumed that external power will be provided. In either case it is assumed that the power system would be serviced at the Space Station in LEO.

It is assumed that the Lander would be self-powered for only 3 days on the surface. When surface power and provisions for reactants are available for the entire surface stay, off-loading at the Space Station may be realized, adding to payload capability.

The 2 kw average power req. is an estimate based on the Apollo LM (peak power 2.3 kw) and calculations indicating DMS/GN&C and ECLSS will each require about a kilowatt. This may be reduced, but there will be other power requirements. A more conservative estimate might be 3 kw average power required.

Several candidate battery systems were considered and the results are presented in Table 7-6. Only ambient temperature (0 - 100 degrees C) batteries were studied. Following is a brief discussion of those selected for analysis:

Ag/Zn Long-Life Secondaries - This is the best of the current battery technology. Batteries of this type (as primaries) have flown on all manned spacecraft to date. In order to achieve "long life" (2 years, 10 cycles), appreciable derating must be applied, yielding a system energy density of approximately 82 Wh/kg. No manufacturer is specifying greater than 5 cycles per year at 75% depth-of-discharge.

Ni/Cd - This is the rechargeable battery of longest experience. Its low depth-of-discharge limit (25%) limits usable capacity severely and yields a system energy density of 40-45 Wh/kg.

Ni/H_2 - This couple represents a hybrid battery/fuel cell system whereby the reactant fuel, H_2, is stored under pressure in individual cell pressure vessels. It has a usable depth-of-discharge of 80%, and yields a system energy density of 40-45 Wh/kg.

Li/TiS_2 - This couple is still at the laboratory development stage, with 5 Ah cells under test. It operates at room temperature, is projected to be capable of 100 cycles at 75% depth-of-discharge, and yields a system energy density of 110 Wh/kg.

The Shuttle-derived fuel cell yields the system of lowest weight and greatest flexibility. For large energy (>50kWh) requirements the fuel cell becomes the candidate of choice primarily due to the large energy content of the reactants, H_2 and O_2, supplying approximately 2200 Wh/kg (tankage not included). The Shuttle fuel cell is state-of-the-art, requiring no development for the lander application except to remove 2 of the 3 substacks and re-package for mounting in the lander. This modification provides a fuel cell with 1 stack of 32-cells operating between 0.7 to 4 kW between the voltage limits of 28 to 32.5 Volts and weighing 68 kg. Two units are required providing 100% redundancy, permitting failure of one unit without compromising the mission.

The volume displacement of each modified fuel cell is approximately 0.058 m^3, measuring 28 x 28 x 71 cm.

The reactant storage system can be configured one of three ways:

1. Dedicated high-pressure gaseous storage

2. Dedicated liquified form

3. Integrated with propellant storage

Because the lander propellants will be supplied by the Space Station facility and the quality is electrolytic grade (the quality utilized by the fuel cells), option 3 is the most attractive option for a lander using O_2/H_2 propellant. The results of the fuel cell reactant storage analysis are presented in Table 7-7.

Basically, the impact of adding the fuel cell reactants to the propellant tanks is nil; 31 kg H_2 adds 26 mm to the diameter of each H_2 tank - an increase of 0.7% for each parameter, and 244 kg O_2 adds 6 mm to the diameter of each O_2 tank - an increase of 0.9% and 0.3% respectively for each parameter. This provides energy storage of 200% of that required for the 15-day mission. Getting the reactants out of the large tanks when only small quantities are left may be a problem however.

The fuel cell operating temperature range is between 80 and 95°C. It is provided with a fluid loop heat exchanger that is integrated with the ECLSS thermal control loop, just as in the Shuttle Orbiter. Heat rejection will be approximately 4,400 btu's per hour at the 2 kW power level.

Fuel cell product water is potable and useful for crew consumption and evaporative cooling. It is produced at the rate of about 3/4 liter per hour at the 2 kW power level for a total of 260 kg for the 15-day mission. It is delivered to the fuel cell interface in liquid for transfer to the ECLSS system. Therefore, storage and plumbing are not included in the power system design. However, for single tank storage, a tank of 0.8 m in diameter is required.

The baseline system used in most of the section 8.0 lander calculations is a dual redundant fuel cell system using dedicated tanks for cryogen storage. Table 7-7 estimates the total mass of the system that provides 2kw for 3 days as 478 kg. An equivalent system which uses the main propellant tanks for reactants might weigh 274 kg (dual redundant, not counting tank mass increase). The low mass lander shown in Table 8-3 and in the various other plots and tables uses this system.

Table 7-6, Power System Options (15 Day Mission)

System	Energy Density (Wh/Kg)	System Wt., (Kg)
Batteries (50% Redundancy)		
Ag/Zn	55	13,090
Ni/Cd	44	16,364
Ni/H_2	44	16,364
Li-Al/FeS_2	77	9,350
Li/TiS_2	110	6,545
H_2/O_2 Fuel Cells (100% Redundancy)		
Ded. Cryo Tanks	391	1,842
Integ. w/Prop. Tanks*	1,051	685

* Added Wt. of Propellant tanks for slight increase in diameter not included. Reactants are included.

Table 7-7, Fuel Cell System Analysis (No Redundancy)*

	Reactants, Kg	Tank Dia., M	Tank Wt., Kg	F.C. Wt. Kg	Sys. Wt. Fc, Rx, Tank	EnergyDensity Whrs/kg
Gaseous						
720 kwh:(15 days)						
H_2	30.9	1.57	442	68	1,000	720
O_2	243.7	1.46	215			
144 kwh:(3 days)						
H_2	6.2	0.92	88	68	254	567
O_2:	48.8	0.73	43			
Cryo (15 days)						
720 kwh:						
H_2	30.9	0.94	224	68	921	782
O_2	243.7	0.74	354			
144 kwh: (3 days)						
H_2	6.2	0.55	45	68	239	603
O_2	48.8	0.43	71			
or						

* 1 Fuel Cell, 1 Set of Tanks.
Included in Weights: 10% F.C. wt. for mounting
10% Tank wt. for plumbing/mounting
5% Reactant wt. for ullage.

7.6 Crew Module

Reference 8 provides a scaling equation for crew module mass, 1,250 + 525N, where N is the number of occupants for the mission and the equation is given in kilograms. It has been suggested that a new crew capsule be designed to hold six persons. Using this scaling factor this would give a crew capsule mass of 4,400 kilograms. Lunar lander vehicle crew modules used in the Apollo program, designed for 2 people, were in the range of 2,000 kilograms, being in line with the approximation. The mass of the crew module may contain other subsystems as well. Table 7-8 shows an approximate weight statement for the crew module shown in Figure 7-4.

7.7 Thermal Control

The lunar lander may be required to sit on the lunar surface near the equator through a 14 Earth day long lunar day. Initial estimates indicate the peak heat input may be on the order of 1 to 2 kw. The following estimates of heat input to the vehicle come from Ref. 11. The vehicle was assumed to be covered with a "tent" with thermal properties as shown below. As the average temperature of the vehicle goes down, as will be the case for cryogenic propellants, the heat input will rise.

Case 1
Ave. temperature of vehicle assumed to be 306° K = 33°C

	Outside Surface of Tent	Inside Surface of Tent
Emissivity	.91	.01
Absorbtivity	.21	----

(These numbers are characteristic of multilayer insulation on the inside and white on the outside surface of the tent).

Heat output at noon at Lacus Veris (13°S latitude) = 1.1 w/m^2 = -251 watts in @ *229 m^2 surface area.

Case 2
Ave. temperature of vehicle assumed to be 288°K = 15°C

	Outside Surface of Tent	Inside Surface of Tent
Emissivity	.01	.01
Absorbtivity	.01	

(Those numbers are for multi-layer insulation on the inside and outside surfaces of the tent)

Heat input at noon at Lacus Veris (13°S latitude) = 4.3 w/m^2 = 985 watts in @ *229 m^2 surface area.

* Estimate for lander shown in Section 9.1.

The Apollo LM used water boilers to handle this problem. This will probably not be adequate for longer stay times and a radiator system will be required. This radiator might be a trailer mounted plug-in device. Hydrogen storage in the lander on the surface may also require refrigeration. Again, a trailer mounted unit may be needed. Ref. 6 presents a conceptual design for a trailer mounted radiator to handle this type of heat rejection.

Multi-layer insulation (MLI) is needed to reduce the cooling and extra power required to pump out heat leak due to solar radiation. Both aluminized Kapton foil (usual choice) or Beta are acceptable as long as surface reflectance requirements are satisfied. Surface reflectance of Beta, in the solar and IR spectrum, in order to be suitable, must be similar to that of Kapton, on both sides of the foil. Protection from engine exhaust can be provided by covering the MLI with a coarse net, with a 1/2 inch to 1 inch pitch.

Hydrogen boiloff will be significantly influenced by all these factors. One reference (7) estimates roughly 1% per day boiloff for the in-space operations. This would be on the order of 50 kg per day for the landers discussed in this report. For 15 days, 750 kg would be lost.

A detailed thermal analysis of the lander cycle, transit, landing, surface stay,(particularly the 180 day stay) and return is required. The chief concern is the boiloff of hydrogen during a long surface stay.

Table 7-8, Crew Module Weight Statement (All masses are in kilograms)

Aluminum Shell (Cylinder 430 cm diam., 267 cm high, 6 mm wall)	1,094
ECLSS (Crew of 4 for 3 days)	
Consumables	72
Hardware	1,264
Fluids	214
Crew Provisions (Includes crew, suits, and seats)	1,708
Controls and Displays	50
Two hatches	53
Total Crew Module	4,455
Contingency	500
Additional Payload	1,000
Lander Ascent/Descent Total Payload	5,955

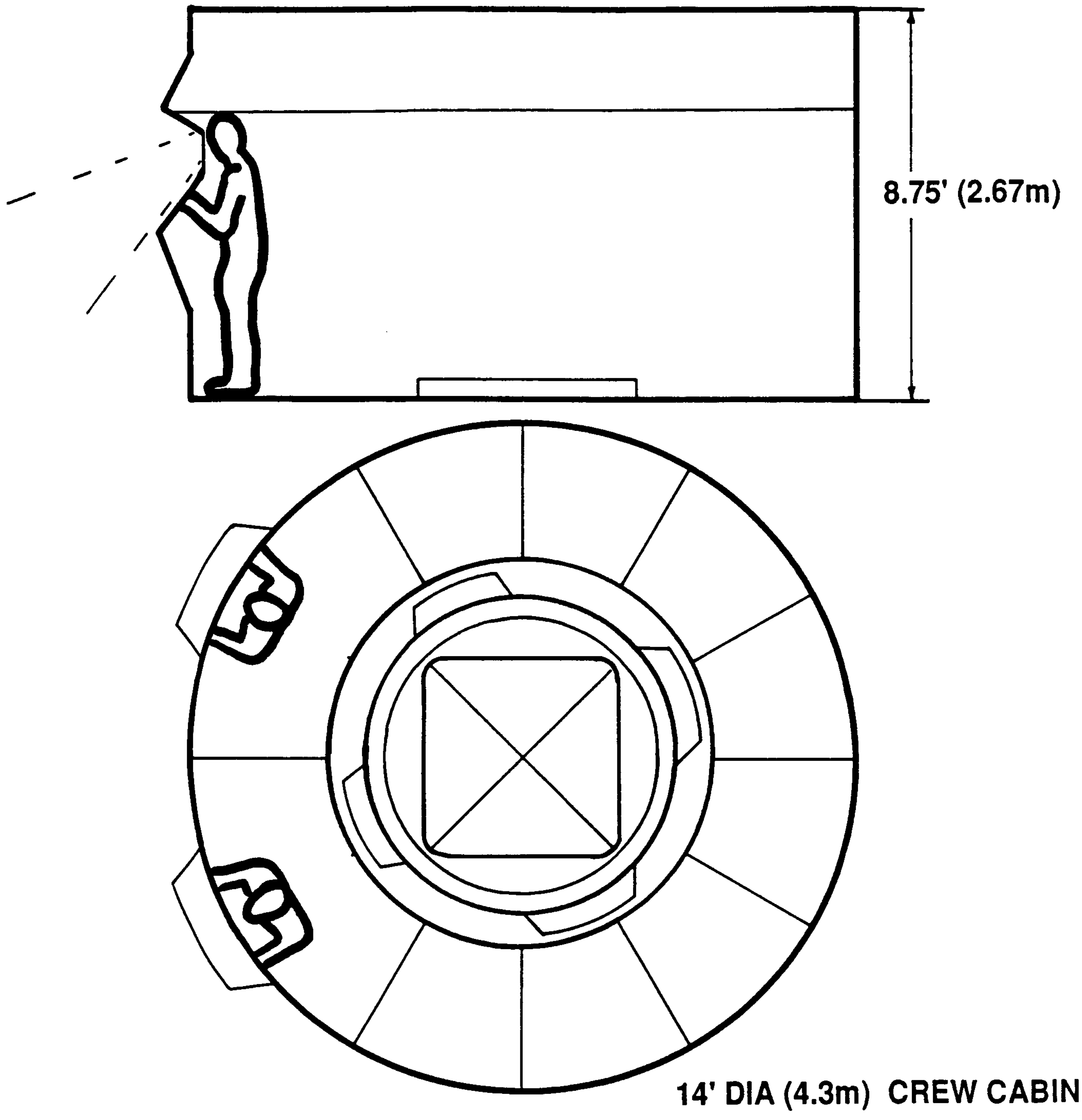


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FIGURE 7-4, CREW MODULE

8.0 Weight Statements

The weight statements shown in Tables 8-1, 8-2, 8-4, and 8-5 represent second interation vehicles, one step past those discussed previously in this report and shown in the previous tables and plots. As expected, these second interation vehicles are heavier. All the knowledge gained in the subsystem studies was used to refine the earlier weight statements. The Table 8-3 weight statements are for the first interation landers shown in the previous plots and figures.

8.1 LO_2/LH_2 Multi-purpose Lander

Table 8-1 shows three dedicated landers scaled specifically for three different tasks. The system masses shown are combined to produce one lander to do all three tasks (Table 8-2). The cargo landing task results in the largest deorbit mass which scales the structures, engines, RCS dry mass, and landing systems. The round trip with a crew module results in the largest propellant mass which scales the tanks and thermal protection. The electrical power system uses four dedicated tanks for redundant reactant storage. The delta V includes an additional .43 km/sec for a 15° plane change.

Table 8-2 shows the multi-purpose vehicle (MPV). The MPV pays a penalty of 2,300 kg for the crew module case, for being able to do all three tasks.

Table 8-3 shows a weight statement for the landers scaled in the plots. This lander does not carry a weight allotment for the airlock/tunnel and has a minimum electrical power system.

8.2 N_2O_4/MMH Multi-purpose Lander

Tables 8-4 and 8-5 show dedicated and multi-purpose landers for performing the tasks discussed. Both cryogenic and storable vehicles assume the same power systems.

Table 8-1, LO_2/LH_2 Dedicated Landers

All masses kg and Delta V's - km/sec, Isp=450 lbf-sec/lbm

Delta V, Ascent	0	*2.28	*2.28
Payload, Ascent	0	6,000	0, (inert mass only returned to LLO)
Delta V, Descent	2.10	2.10	2.10
Payload, Descent	25,000	6,000	14,000
Total Inert Mass	8,828	9,062	9,301
Structure	1,681	1,322	1,523
Engines	822	646	744
RCS dry	411	323	372
Landing systems	784	617	711
Thermal Prot.	1,604	2,017	1,934
Tanks	2,406	3,025	2,901
DMS/GN&C	150	150	150
** Electrical Power	478	478	478
Airlock/Tunnel	455	455	455
Total Propellant Mass	24,530	30,838	29,570
Ascent Prop.	0	10,789	6,866
Descent Prop.	21,951	17,265	19,888
Unusable Prop. (3%)	659	842	803
FPR Propellant (4%)	878	1,122	1,070
Usable RCS	833	656	755
Unusable RCS (5%)	42	33	38
FPR RCS (20%)	167	131	151
Deorbit Mass (less payload)	33,358	39,900	38,871
Deorbit Mass (with payload)	58,358	45,900	52,871

* Delta V = 1.85 + .43 km/sec for a 15° plane change in a 93 km circular orbit.

** Electrical power provided for 3 days only, (2 kw). 100% redundant fuel cells, with dedicated tanks are included.

Table 8-2, LO_2/LH_2 Multi-purpose Lander Weight Statement

All masses are kg, all Delta Vs, km/sec, Isp=450 lbf - sec/lbm

Delta V, Ascent	0	*2.28	*2.28
Payload, Ascent	0	6,000	0, Inert mass returned to LLO
Delta V, Descent	2.10	2.10	2.10
Payload, Descent	25,000	6,000	14,000
Total Inert Mass	9,823	9,823	9,823
Structure	1,681	1,681	1,681
Engines	822	822	822
RCS Dry	411	411	411
Landing Syst.	784	784	784
Thermal Prot.	2,017	2,017	2,017
Tanks	3,025	3,025	3,025
DMS (GN&C)	150	150	150
** Elect. Power	478	478	478
Airlock/Tunnel	455	455	455
Total Prop. Mass	25,251	32,395	30,638
Ascent Prop.	0	11,334	7,240
Descent Prop.	22,597	18,137	20,486
Unusable Prop.(3%)	678	884	832
FPR Prop. (4%)	904	1,179	1,109
Usable RCS	858	689	778
Unusable RCS (5%)	43	34	39
FPR (20%)	172	138	156
Deorbit or Gross Mass (less payload)	35,074	42,218	40,461
Deorbit or Gross	60,074	48,218	54,461

* Delta V = 1.85 + .43 km/sec for a 15° plane change in a 93 km circular orbit.

** Electrical power provided for 3 days only, (2kw). 100% redundant fuel cells have dedicated redundant tankage.

Table 8-3, LO_2/LH_2 Dedicated Landers, First Generation Estimates

The numbers in this table correspond to those shown in the section 6 plots.

All masses are kg, all Delta Vs, km/sec, Isp=450 (lbf - sec/lbm).

Delta V, Ascent	0	*1.85	*1.85
Payload, Ascent	0	6,000	0, Inert mass returned to LLO
Delta V, Descent	2.10	2.10	2.10
Payload, Descent	25,000	6,000	14,000
Total Inert Mass	7,930	5,802	6,762
Structure	1,636	930	1,236
Engines	800	454	604
RCS Dry	400	227	302
Landing Syst.	764	434	577
Thermal Prot.	1,562	1,339	1,450
Tanks	2,342	2,008	2,175
DMS/GN&C	145	145	145
** Elect. Power	245	245	245
Airlock/Tunnel	0	0	0
Total Prop. Mass	23,878	20,473	22,166
Ascent Prop.	0	6,036	3,853
Descent Prop.	22,036	11,725	16,147
Unusable Prop.	641	558	600
FPR Prop. (4%)	855	744	800
Usable RCS	811	461	613
Unusable RCS	40	23	31
FPR RCS (20%)	162	92	123
Deorbit or Gross Mass (less payload)	31,808	26,275	28,928
Deorbit or Gross Mass (with payload)	56,808	32,275	42,928

* Delta V includes no allowance for plane change.

** Electrical power uses main propellant tanks as reactant source.

Table 8-4, N_2O_4/MMH Dedicated Landers

All masses are kg, all Delta Vs, km/sec, Isp= 330 (Gf-sec/lbm)

Delta V, Ascent	0	*2.28	*2.28
Payload, Ascent	0	6,000	0, Inert mass returned to LLO
Delta V, Descent	2.10	2.10	2.10
Payload, Descent	25,000	6,000	14,000
Total Inert Mass	7,222	7,667	7,476
Structure	1,955	1,831	1,882
Engines	956	895	920
RCS Dry	478	448	460
Landing Syst.	912	855	879
Thermal Prot.	718	1,006	884
Tanks	1,077	1,509	1,326
DMS (GN&C)	150	150	150
** Elect. Power	478	478	478
Airlock/Tunnel	455	455	455
Total Prop. Mass	35,650	49,918	43,884
Ascent Prop.	0	15,440	8,929
Descent Prop.	32,185	30,152	30,995
Unusable Prop.(3%)	966	1,368	1,198
FPR Prop. (4%)	1,287	1,824	1,597
Usable RCS	969	908	933
Unusable RCS	49	45	47
FPR RCS (20%)	194	182	187
Deorbit or Gross Mass (less payload)	42,872	57,585	51,361
Deorbit or Gross Mass (with payload)	67,872	63,585	65,361

* Delta V = 1.85 + .43 km/sec for a 15° plane change in a 93 km circular orbit.

** Electrical power provided for 3 days only, (2kw). 100% redundant fuel cells/tank sets.

Table 8-5, N_2O_4/MMH Multi-purpose Landers

All masses are kg, all Delta Vs, km/sec, Isp=330 (Gf-sec/lbm).

Delta V, Ascent	0	*2.28	*2.28
Payload, Ascent	0	6,000	0, Inert mass returned to LLO
Delta V, Descent	2.10	2.10	2.10
Payload, Descent	25,000	6,000	14,000
Total Inert Mass	7,899	7,899	7,899
Structure	1,955	1,955	1,955
Engines	956	956	956
RCS Dry	478	478	478
Landing Sys.	912	912	912
Thermal Prot.	1,006	1,006	1,006
Tanks	1,509	1,509	1,509
DMS/GN&C	150	150	150
** Elect. Power	478	478	478
Airlock/Tunnel	455	455	455
Total Prop. Mass	36,398	50,767	45,429
Ascent Prop.	0	15,702	9,406
Descent Prop.	32,861	30,665	31,927
Unusable Prop.	986	1,391	1,240
FPR Prop. (4%)	1,314	1,855	1,653
Usable RCS	990	923	961
Unusable RCS	50	46	48
FPR RCS (20%)	198	185	192
Deorbit or Gross Mass (less payload)	44,297	58,666	53,328
Deorbit or Gross Mass (with payload)	69,297	64,666	67,328

* Delta V = 1.85 + .43 km/sec for a 15° plane change in a 93 km circular orbit.

** Electrical power provided for 3 days only, (2 kw). 100% redundant fuel cells/tank sets.

9.0 Conceptual Designs

9.1 LH_2/LO_2 Multi-purpose Lander

Figure 9-1 and 9-2 show a conceptual design of an LH_2/LO_2 Multi-purpose Lander. The tanks are sized to hold roughly 30 metric tons total of propellant. The H_2 tanks are 3.9 meters in diameter. The O_2 tanks are 2.76 meters in diameter.

Important features of this lander include:

1) Airlock/servicing tunnel down center of lander to allow easy access on surface and pressurized volume for LRUs. Many engine connections can be made and broken inside the pressurized volume.

2) Removable crew module. The lander is flyable without the crew module.

3) Lander fits in 30' heavy lift vehicle shroud with landing gear stowed.

4) Electro-mechanical shock absorbers on landing gear.

5) Emergency ascent with one or two crew possible without crew module. Crew would ride in suits in airlock/servicing tunnel.

Figure 9-3 shows this lander being serviced on the lunar surface and illustrates how the airlock/servicing tunnel allows pressurized access to a surface vehicle. An engine is being removed in the figure.

Figure 9-4 shows this lander in lunar orbit, about to dock with a large (single stage) OTV. The OTV is designed to return the lander to the Space Station for servicing. The OTV delivers the lander to low lunar orbit, single stage, and waits in orbit for it to return. The OTV tanks are sized to hold 118 m tons of LO_2/LH_2 propellants.

Figure 9-5 shows the lander on the surface at the poles. The lander may also serve as a suborbital "hopper" if propellant loading on the lunar surface is provided.

C-2

Figure 9-1, LO_2/LH_2 Reusable Lunar Lander, Side View
Scale: 1/2 inch = 1 meter

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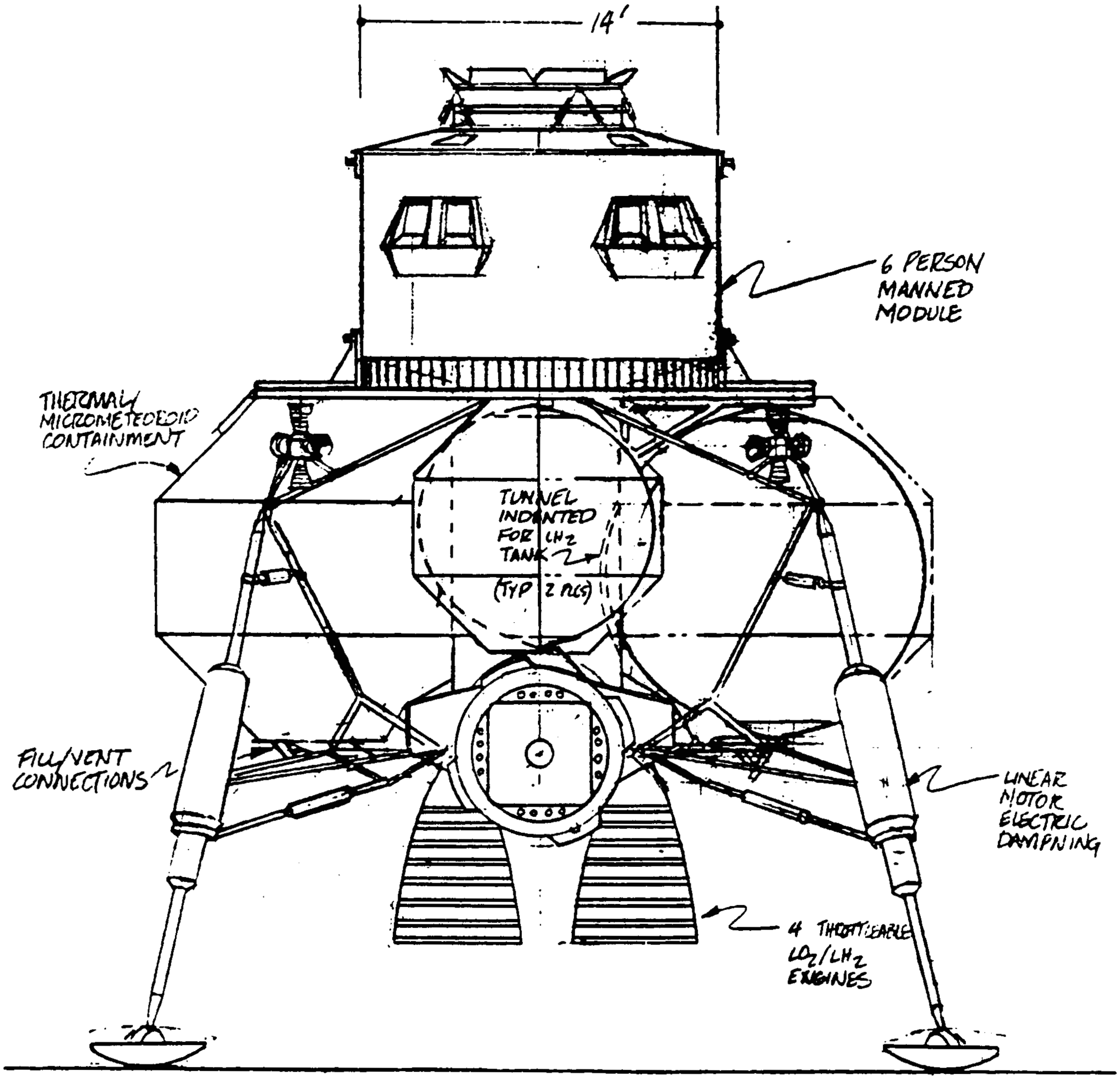
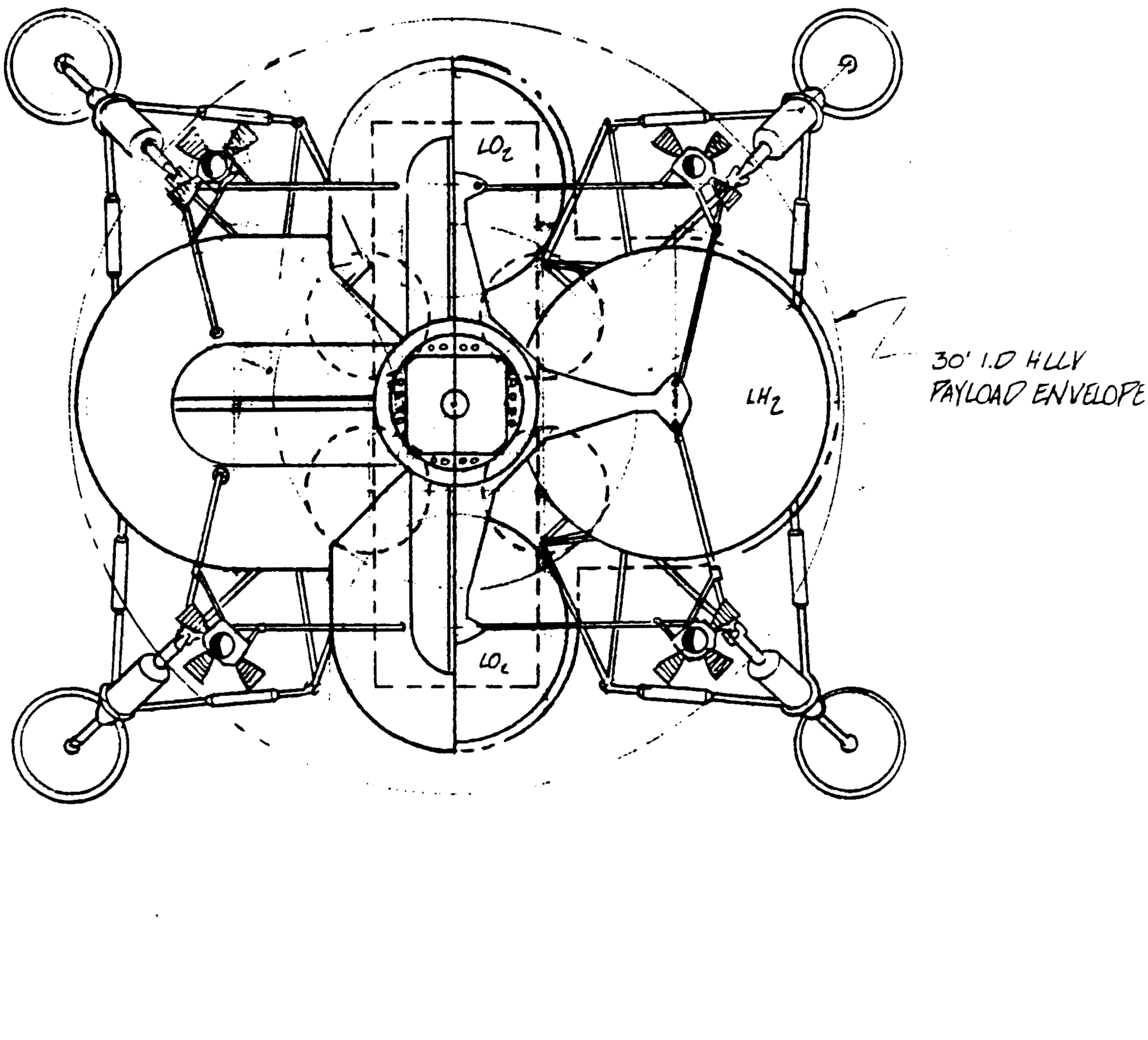


Figure 9-2, LO_2/LH_2 Reusable Lunar Lander, Top View
Scale: 1/2 inch = 1 meter



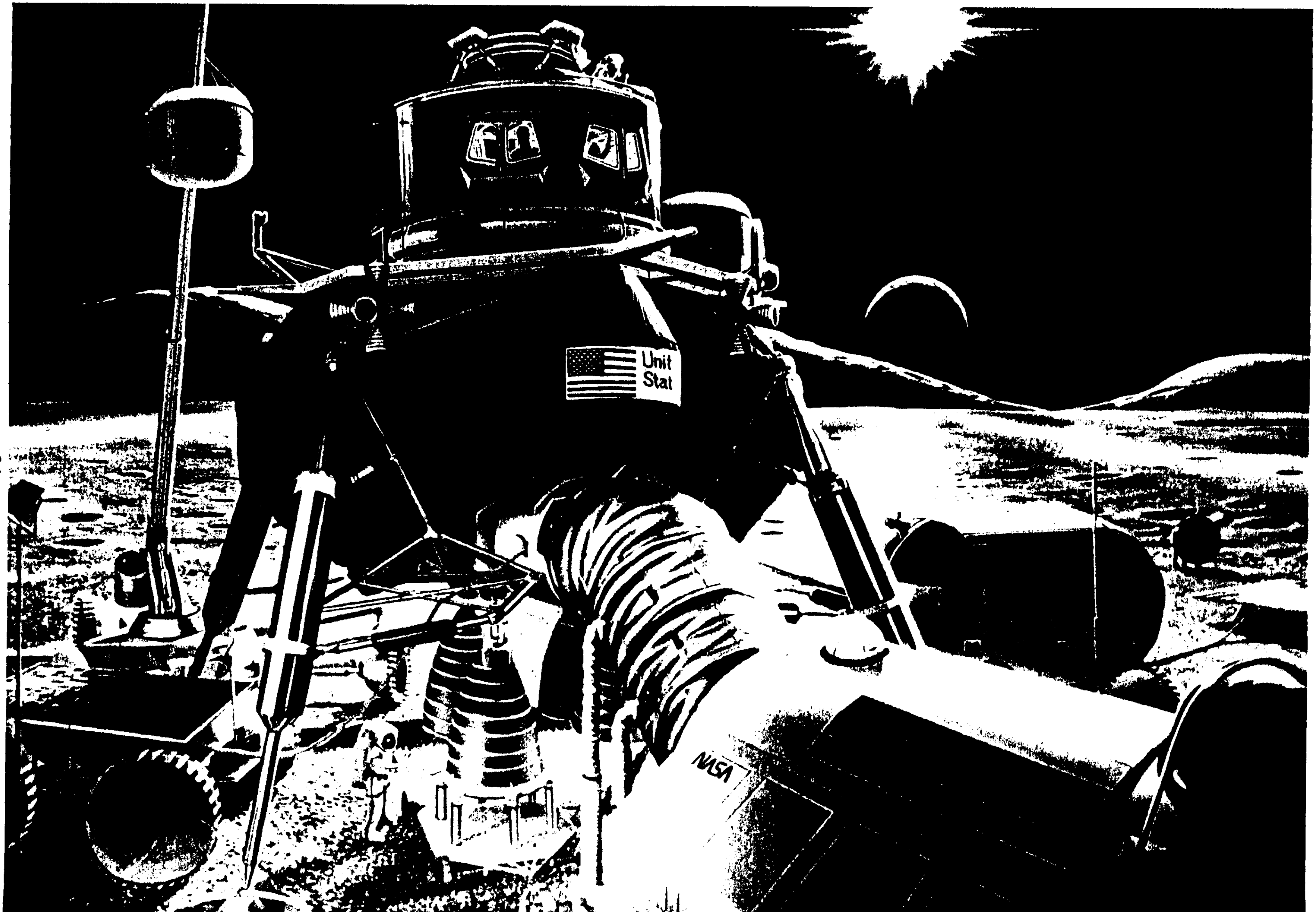


Figure 9-3, Lander on Surface

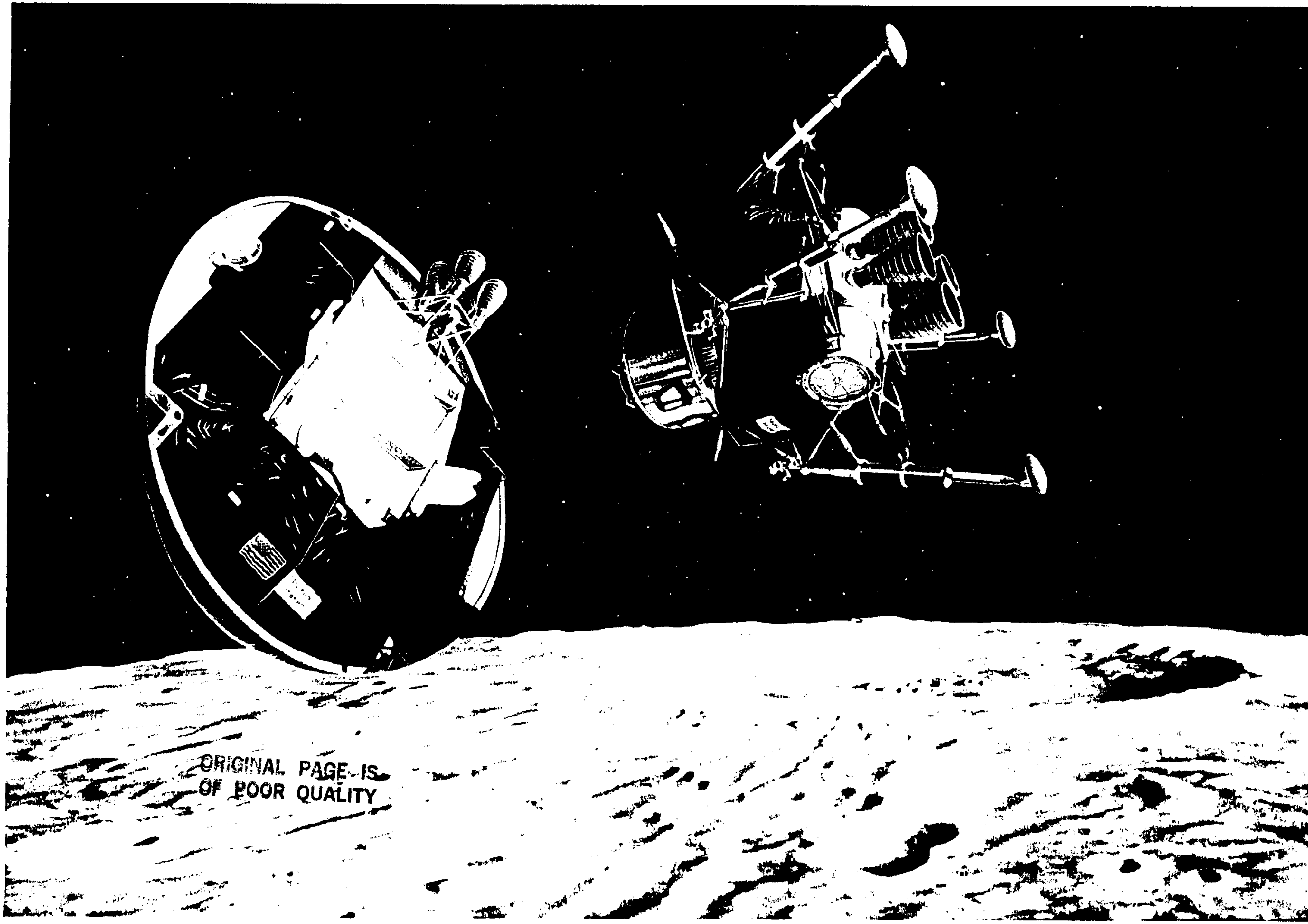


Figure 9-4, Lander and OTV in LLO

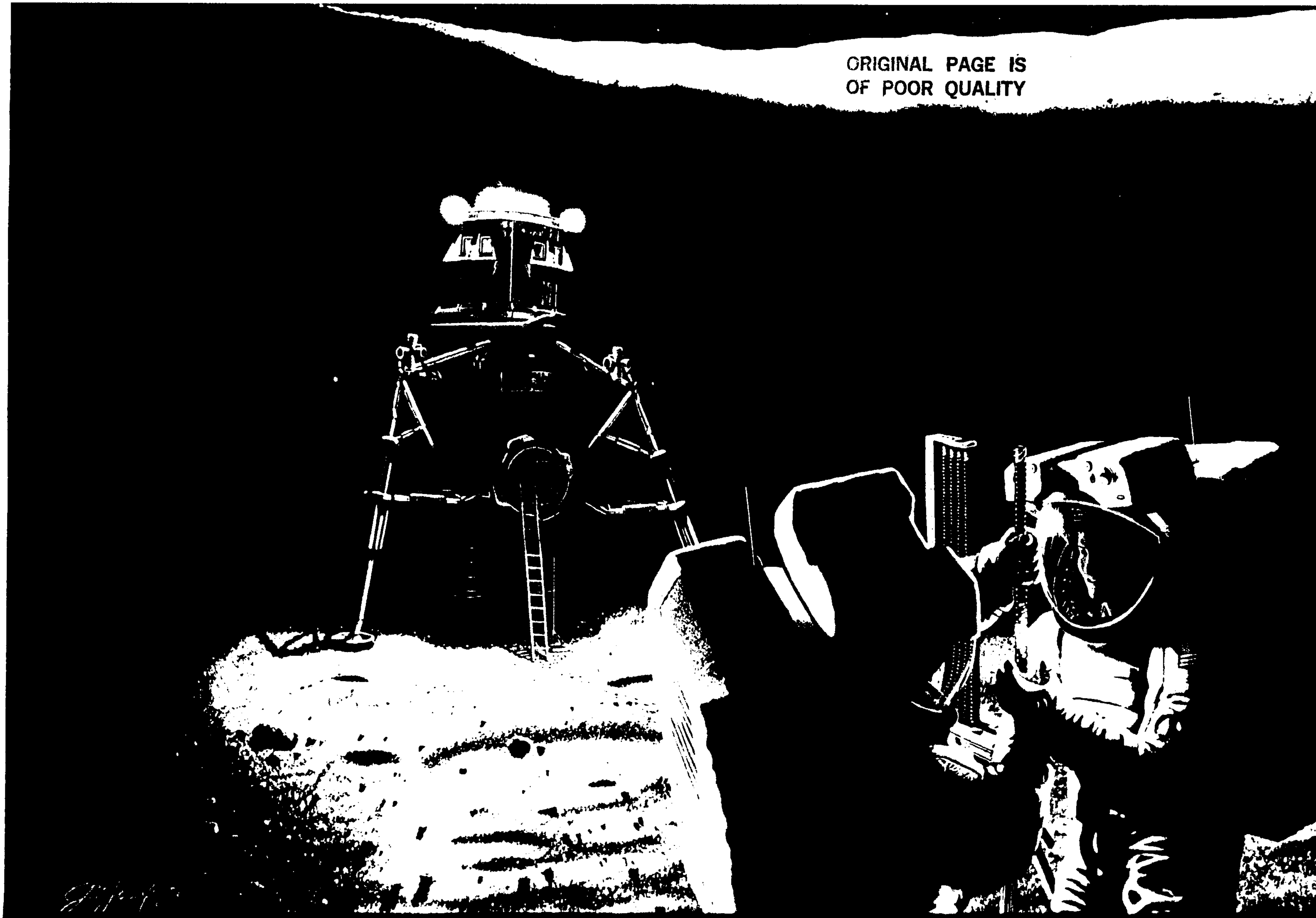


Figure 9-5, Lander on Surface at Pole

9.2 N_2O_4/MMH Multi-purpose Lander

Figure 9-3 shows a lander with equivalent capability to the Section 9.1 lander, except using N_2O_4/MMH propellants. This lander, though considerably heavier than the LH_2/LO_2 lander, is much smaller, due to higher propellant density. Its features are essentially the same as the 9.1 lander.

The propellant capacity of this lander is 35 m tons divided into four tanks of 16 m^3 each. Tank diameter is 2.5 meters for all tanks.

Figure 9-4, Advanced Storable Reusable Lunar Lander, Side View
Scale: 1/2 inch = 1 meter

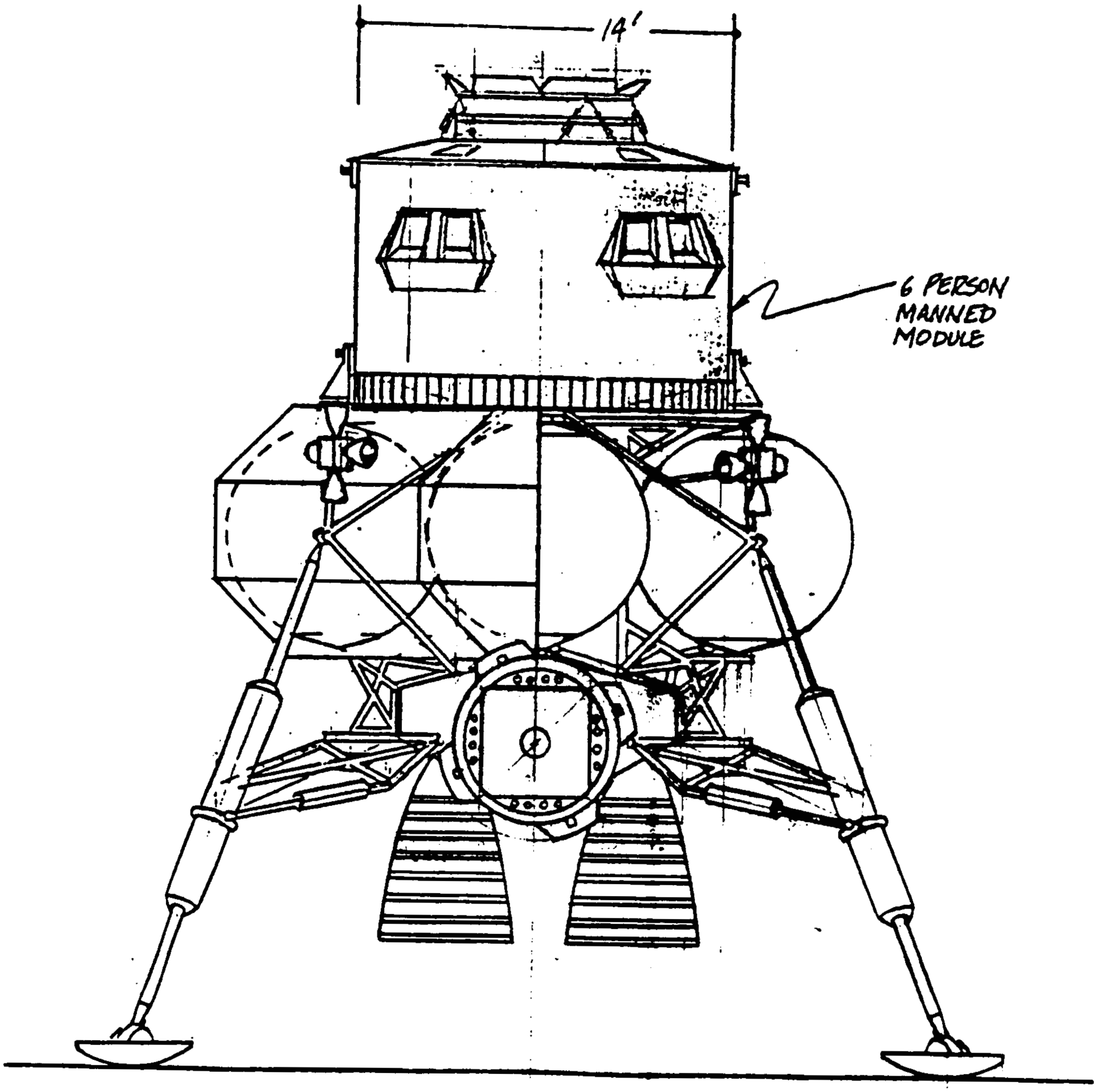


Figure 9-5, Advanced Storable Reusable Lunar Lander, Top View
Scale: 1/2 inch = 1 meter

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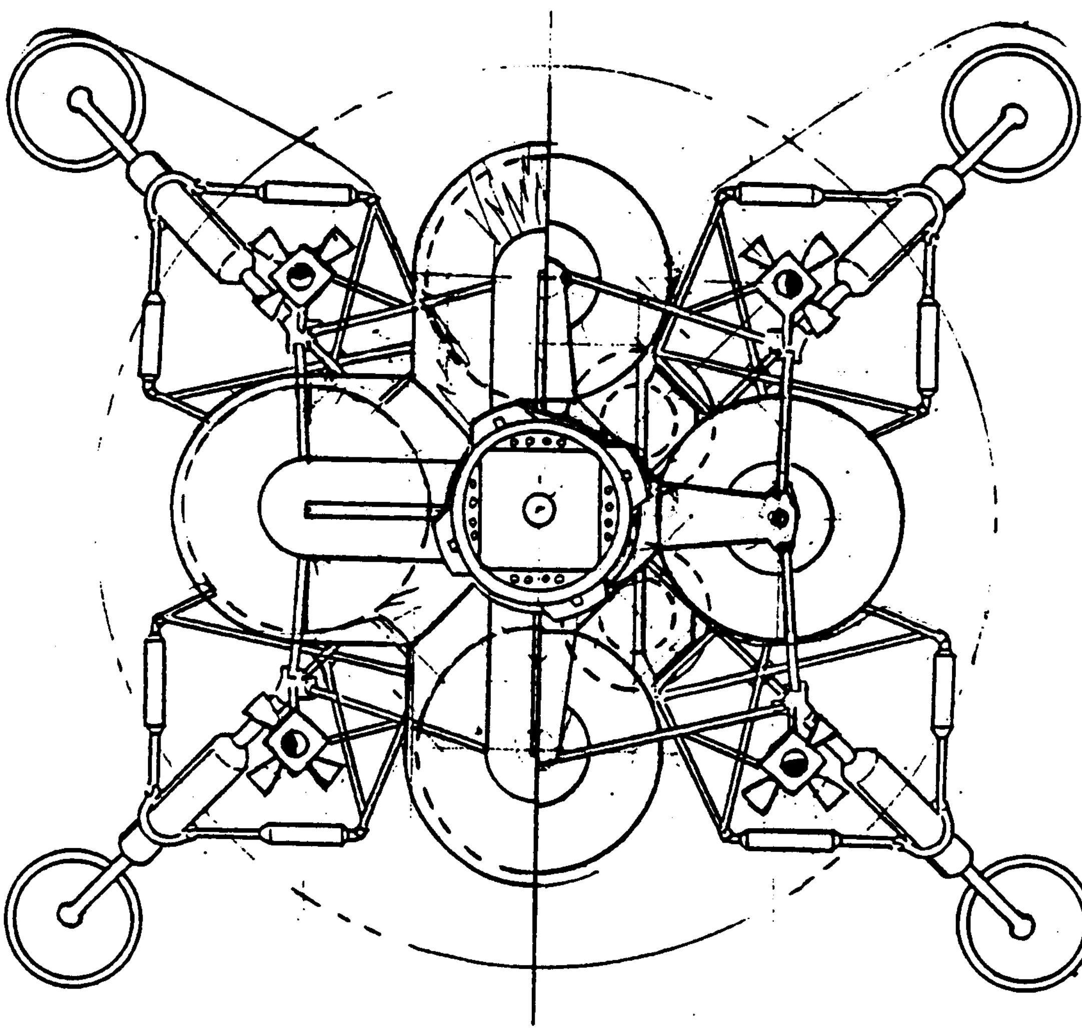

10.0 Vehicle Cost

Lunar lander (LLV) production costs were determined using a cost estimating relationship (CER) model. With this method, design and fabrication cost curves are developed for each vehicle component, relating the component's historical costs to its weight. Components from the Gemini, Apollo, Skylab, and Shuttle programs were considered when developing the CER's. Where several significantly distinct classes of a given component existed, a separate CER was created for each class. The cost curves generated using this method usually had a correlation coefficient of 0.9 or better. All costs have been adjusted for inflation, and are expressed in 1988 dollars. Program management wrap factors are included in the CERs.

Table 10-1 summarizes the lander production costs. Total design and development cost is estimated to be $1,539 million, and total fabrication cost is estimated to be $759 million per vehicle. Total program cost for ten vehicles is $9,129 million.

To verify the reasonableness of these estimates, they were compared to actual Apollo LM engineering and fabrication costs. Table 10-2 shows this comparison. Estimated design and development costs were within 7% of actual LM costs (when adjusted for inflation), and estimated fabrication costs were within 2% of actual LM costs.

The weights on which the costs estimates are based were first-cut estimates, and are likely to differ from the actual weights of the completed vehicle. The program costs described here can only be considered gross estimates and are likely to contain considerable error. Indeed, consultation with Owen Morris, former manager of the Apollo Spacecraft Program, indicates that the component cost estimates for the separation system, docking adapter, engines, waste management system, and crew accommodations may be low. Estimates for thermal control may be high.

Table 10-1, Summary of Lunar Lander Vehicle Production Costs

Component	Weight (kg)	Design/Dev Cost ($M)	Per Unit Fabrication Cost ($M)
Structures	2,145	118	8
Tanks	2,700	88	7
Separation System	1	1	*
Thermal Control	2,000	19	2
Payload/Docking Adapter	455	19	2
Landing Gear	788	7	1
Guidance, Navigation, and Control	150	188	59
Communications and Data	23	60	11
Power Distribution	68	35	3
Power Generation	455	31	13
Reaction Control System	413	65	8
Liquid Rocket Engines	825	462	9
Environment Control & Life Supt	545	83	6
Atmosphere Management	364	14	4
Water Management	182	3	1
Waste Management	127	7	4
Crew Accommodations	1,000	36	2
	-------	------	-----
Subtotal	12,241	$1,236	$140
System Test Hardware			$172
System Test Operations			66
Ground Support Equipment			306
Syst Eng and Integration		201	14
Program Management		102	61
		-----	-----
Subtotal		$ 303	$619
Total Cost, One Vehicle		$1,539	$759
Number of Vehicles in Program			X 10
		------	------
Total Program Cost		$1,539	$7,590

* - less than $1 million

Table 10-2, Comparison of Lunar Lander Vehicle Costs to Apollo LM Costs

<u>Design/Development Costs</u>

*Apollo LM (1967 $M)	378
Apollo LM (adj. to 1988 $M)	1,672
New lunar lander (1988 $M)	1,539

<u>Fabrication Costs</u>

Apollo LM (8 units, 1967 $M)	1,354
Apollo LM (1 unit, 1967 $M)	169
Apollo LM (1 unit, adj. to 1988 $M)	745
Lunar Lander Vehicle (1 unit, 1988 $M)	759

* These numbers come from a 1967 document. Other significant development costs were incurred after 1967 which are not shown here.

11.0 Conclusions and Recommendations

The following major conclusions resulted from this study:

1) Single stage landers do not pay a large penalty over two stage landers (15-30%) when operated from low lunar orbit. Higher orbits, such as L2, need two stages.

2) Maintainability in space must be designed in from the start and will result in additional inert weight. It must be made a program priority.

3) Low lunar parking orbits minimize LEO stack mass. The very low orbits are unstable over relatively short periods of time (months). Very low orbits may also cause problems for abort situations.

4) Loading propellants and reservicing a reusable lander at the LEO Space Station is possible with some penalty (10 to 25%) in LEO stack mass.

5) A multi-purpose lander, staging from LLO, landing cargo one way, or crew modules round trip is possible, with some penalty in inert mass (5 to 10%) over dedicated designs.

6) Some plane change capability (10° - 15°) is desirable to allow wide launch windows from the surface up. This is not needed for an equatorial base but becomes more important as base latitude goes up. This plane change capability could also reside in the OTV. 15° plane change capability might increase lander mass 10%.

7) Total thrust on the order of 35 to 40,000 lbf at a throttling ratio of 13:1 to 20:1 (depending on the assumption made) is needed for a multi-purpose lander.

8) A regeneratively cooled, pump fed engine will probably be required due to difficulties with regenerative cooling over wide throttling ranges with pressure fed systems.

The following major recommendations resulted from this study:

1) More detailed sizing and weight statements must be generated for a point design LH_2/LO_2 lander in order to get a good point for scaling equations and insure the predicted performance gain for LH_2/LO_2 is accurate.

2) Thermal analysis for long stay times on the lunar surface are needed to determine hydrogen loss and inert weight penalties from added insulation or refrigeration.

3) More engine sizing work is needed. The lander study effort should be continued until definite conclusions can be reached concerning engine type, size, etc. The engine is the long lead, chief development item.

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13.0 Appendix A, Annotated Bibliography

The following list of documents is a top level list of references of interest. Many, but not all were acquired by Eagle in the process of the study. A number of the LM books are the property of Owen Morris (O.M. library), project manager for the Apollo LM spacecraft. The general format of the references is:

Title
  Source; Author; Location; Date; microfiche
  Topics

1) JPL requirements for spacecraft landing and recovery
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   Interplanetary spacecraft, Lunar spacecraft, Spacecraft landing, Spacecraft recovery, Onboard equipment, Space probes, Spacecraft design

2) Advanced Lunar Transportation System Studies Progress Report
   Lockheed; 79N76150*; 08/01/62
   Lunar flight, Lunar landing, Mission planning, Spacecraft design, Manned space flight, Manned spacecraft

3) Direct flight Apollo study
   McDonnell Aircraft Co.; 79N77451; 10/03/62
   Apollo project, Design analysis, Flight mechanics, Command modules, Lunar landing modules, Lunar spacecraft

4) Study of hybrid propulsion systems for Apollo lunar landings and takeoffs
   Lockheed Propulsion Co.; 73N74477; 10/19/62
   Apollo project, Hybrid propulsion, Lunar landing, Lunar spacecraft

5) Direct-Ascent vs. Parking-Orbit Trajectory for Lunar-Soft-Landing Missions
   JPL/CIT; Clarke, V. C.; 68N86178*; 12/03/62
   Ascent trajectories, Parking orbits, Surveyor project, Launch windows, Lunar landing, Lunar launch, Lunar luminescence

6) Lunar logistics system study
   Martin Marietta; 75N74049; 12/07/62
   Lunar logistics, Lunar spacecraft, Apollo project, Lunar landing, Mission planning, Spacecraft design

7) Lunar flight handbook, volume 2
   Martin Marietta; 73N73941*; 01/01/63
   Handbooks, Lunar orbits, Lunar trajectories, Space flight, Bibliographies, Earth-Moon trajectories, Lunar landing

8) An Analysis of the Value of a Lunar Logistic System. Part 3 - Lunar Logistic System Operational Plans
   Bellcom, Inc.; Fudali, R. F.; 79N71699*; 03/18/63
   Lunar exploration system, Lunar logistics, Manned spacecraft, Systems engineering, Lunar module, Payloads

9) Study of powered descent trajectories for manned lunar landings, Project Apollo
NASA/JSC; Bennett, F. V.; 70N76261*; 08/09/63
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Lunar exploration, Lunar landing, Space logistics, Cost estimates, Ground support equipment, Management planning, Project management

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14) Propellant utilization/propellant management
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15) Launch Operations Critical Events Analysis: Final Report, 1 Nov. 1963 - 30 Sep. 1964
General Electric; 78N75916*; 09/15/64
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16) LEM propulsion zero gravity test plan
Grumman Aircraft Engr.; Salek, J.; 71X84480; 01/27/65
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17) Lunar landing site accessibility for July 1969
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51) Radar Section: Study Guide LM-3
Grumman; O.M. Lib; 02/01/68
Radar, Subsystems interface, Rendezvous radar, Transponder electronics assem, Landing radar, Radar logic, Control panels

52) Lunar module ascent engine (bac model 8258)
72T17994; 03/28/68; mf

53) LM landing point flexibility provided by the lunar flying unit on single launch lunar missions
Bellcom, Inc.; Valley, D. R.; 79N71821*#; 04/16/68; mf
Analysis (mathematics), Lunar flight, Lunar landing, Lunar landing sites, Lunar surface vehicles, Mission planning, Apollo applications

54) Apollo spacecraft engine specific impulse
Boeing; Cuffe, J.P.B.; 70N35783; 05/06/68; mf
Apollo spacecraft, Propulsion system performance, Specific impulse, Ascent propulsions systems, Descent propulsion systems, Lunar module, Performance tests

55) Guidance, navigation, and control lunar module functional description and operation using flight program luminary vol. 2
73T18719; 05/28/68; mf

56) Lunar module structural adequacy review
73T16022; 05/31/68; mf

57) Improved Lunar Cargo and Personnel Delivery System. Volume 1 - Management Summary Final Report
Lockheed; 68X17198*#; 06/28/68; mf
Lunar logistics, Lunar module, Management planning, Spacecraft design, Systems engineering, Lunar exploration, Transportation

58) Improved Lunar Cargo and Personnel Delivery System. Volume 2 - Study Summary: Final report
Lockheed; 68X17195*#; 06/28/68; mf
Lunar logistics, Mission planning, Spacecraft design, Systems analysis, Lunar module, Research projects, Support systems

59) Improved Lunar Cargo and Personnel Delivery System. Volume 3A - Conceptual Designs and Subsystem Analysis: Final Report
Lockheed; 68X17197*#; 06/28/68; mf
Lunar logistics, Spacecraft design, Systems analysis, Lunar module, Mission planning, Systems engineering

60) Improved Lunar Cargo and Personnel Delivery System. Volume 5 - Evolutionary Development Plan: Final Report
Lockheed; 68X17196*#; 06/28/68; mf
Lunar logistics, Mission planning, Spacecraft design, Cost estimates, Lunar module, Scheduling, Systems engineering

61) Improved Lunar Cargo and Personnel Delivery System. Volume 6 - Supporting Research and Technology Requirements: Final Report
Lockheed; 68X17194*#; 06/28/68; mf
Lunar Logistics, Mission planning, Research projects, Spacecraft design, Systems engineering, Lunar module, Support systems

62) Thermal stress analysis of the Apollo block 2 spacecraft lunar module adapter and service module vol. 1
73T15994; 06/28/68; mf

63) Thermal stress analysis of the Apollo block 2 spacecraft lunar module adapter and service module
75T14055; 06/28/68; mf

64) Lunar module 2 elementary functional diagrams
75T14204; 07/15/68; mf

65) Lunar module performance and interface specification blocks
North American Rockwell Corp.; 73X75427; 07/15/68
Ground support equipment, Lunar module, Specifications, Command service modules, Lunar spacecraft

66) Propulsion and RCS subsystem study guide-Lunar Module LM-3
Grumman Aircraft Engr.; Strasburger, W.; 71X10087; 08/01/68; mf
Ascent propulsion systems, Descent propulsion systems, Lunar module, Thrust control, Control equipment, Display devices, Ground support

67) Design requirements specification rocket engine-ascent, lunar module performance, design and construction requirements part 1
73T13946; 08/30/68; mf

68) Surveyor I design and performance
Parks, R.J.; 68A42133; 09/01/68
Launch vehicle configurations, Space missions, Spacecraft design, Surveyor 1 lunar probe, Aerospace systems, Conferences, Data Acquisitions

69) Contract specification for lunar module system
74T13336; 10/01/68; mf

70) Quantative analyses of lunar module slopes in candidate apollo landing sites
71T15801; 10/22/68; mf

71) Lunar module pressure vessel operating criteria specification
72T17863; 10/25/68; mf

72) Apollo spacecraft engine specific impulse, part 2
Boeing; Cuffe, J.P.B.; 70N36361; 10/28/68; mf
Ascent propulsion systems, Descent propulsion systems, Propulsion system performance, Specific impulse, Apollo flights, Liquid propellant rocket engines, Lunar module

73) Extended lunar module status review for NASA LM derivative working group
72T18802; 11/28/68; mf

74) Summary report lunar module (LM) soil mechanics study
74T15749; 11/28/68; mf

75) Thoughts on LM landing requirements at science sites
Bellcom, Inc.; Silberstein, I.; 79N73398*#; 12/18/68; mf
Lunar exploration, Lunar landing sites, Lunar module, Requirements, Lunar landing modules, Lunar topography, Lunar trajectories

76) Preflight Analysis of the LM Steerable Antenna Tracking System Operation During Lunar Ascent on the Apollo 12 Mission
TRW; Chan, R. J.; 70X12121*#; 01/01/69; mf
Apollo 12 flight, Lunar landing, Lunar launch, Lunar module, Steerable antennas, Graphs, Spacecraft communication

77) Universal lunar module systems handbook LM-4 and subsequent vehicles
75T14457; 01/17/69; mf

79) Lunar module 3 elementary functional diagrams
75T15019; 02/01/69; mf

80) Apollo lunar module propellant studies
71T15685; 02/17/69; mf

81) Lunar module 4 revision-a elementary functional diagrams
75T14808; 03/01/69; mf

82) Lunar module 5 revision-a elementary functional diagrams
72T15057; 04/15/69; mf

83) Landing dynamics of the lunar module
73T10411; 04/16/69

84) Exploration of the Moon
NASA/MSC; Meyer, A. J.; 69A27908*; 05/01/69
Ferry spacecraft, Lunar exploration, Lunar surface vehicles, Lunar topography, Soft landing spacecraft, Apollo spacecraft, Flying platforms

85) Apollo mission G/LM-5/APS preflight performance report
TRW; Thompson, P.F.; 73X80075; 06/01/69
Apollo project, Ascent propulsion systems, Flight simulators, Lunar module, Mission Planning, Performance prediction, Systems analysis

86) Lunar module primary guidance, navigation and control subsystem equipment performance and interface specification
74T10808; 06/20/69; mf

87) Manual attitude control of the lunar module
73T12463; 06/28/69; mf

88) Ideas for improvement of LM descent trajectory
Bellcom, Inc.; Bush, G. L.; 79N71886*; 06/30/69
Descent trajectories, Lunar module, Manual control, Landing sites, Lunar trajectories

89) One-man flying vehicle study. Final Review
Bell Aerospace Systems; Carey, L.; 69X18779*#; 07/25/69; mf
Lunar flying vehicles, Propulsion system, Spacecraft configurations, Flight control, Landing gear, Temperature control

90) Optimum landing of a spacecraft on the moon
Volt Information Sciences; Isayev, V.K.; 69N35489; 08/20/69; mf
Lunar landing, Lunar spacecraft, Trajectory optimization, Equations of motion, Lunar orbits, Thrust control, Thrust loads

91) Optimum spacecraft lift-off from the moon's surface
Volt Information Sciences, Inc; Isayev, V.K.; 69N35490; 08/20/69; mf
Equations of motion, Lifting bodies, Lunar launch, Lunar spacecraft, Trajectory optimization, Dynamic models, Lunar landing modules

92) Lunar module ascent engine program
72T17631; 08/25/69; mf

93) Lunar module ascent engine program Final report
Rockwell; 70X10901; 08/25/69; mf
Ascent propulsion systems, Injectors, Lunar module, Systems engineering, Ablation, Combustion stability, Engine tests

94) Lunar module ascent engine program Final report
Rockwell; 70X10901; 08/25/69; mf
Ascent propulsion systems, Injectors, Lunar module, Systems engineering, Ablation, Combustion stability, Engine tests

95) Apollo 10 LM-4. Ascent propulsion system, final flight evaluation
TRW; Griffin, W.G.; 70X10856; 09/16/69; mf
Apollo 10 flight, Ascent propulsion systems, Lunar module, Postflight analysis, Propulsion system performance, Fuel consumption, Graphs (charts)

96) Lunar module ascent stage thermal math model
73T11576; 09/16/69; mf

97) Lunar module descent stage thermal math model
72T17975; 09/26/69; mf

98) Lunar module 6-elementary functional diagrams
75T15016; 10/01/69; mf

99) Apollo 11 LM-5 ascent propulsion system, final flight evaluation
TRW; Thompson, P.F.; 70X12078; 11/01/69; mf
Apollo 11 flight, Ascent propulsion systems, Lunar module, Postflight analysis, Erosion, Ignition, Performance prediction

100) Lunar Module LM-10 through LM-14 vehicle familiarization manual
70T11351; 11/01/69; mf

101) Lunar module LM 10 thru LM 14
72T14713; 11/01/69; mf

102) Apollo 12(mission H-1) dispersion analysis.Volume 4: Descent and ascent dispersion analyses. Part 2: Lunar ascent
NASA-JSC; West, J.V.; 75N70475; 11/04/69
Apollo 12 flight, AScent propulsion systems, Lunar module, Postflight analysis, Monte Carlo method, Statistical analysis

103) Flight Performance Evaluation Report LM-5
Grumman; O.M. Lib; 11/18/69
Mission operations, Mission objectives, Vehicle performance charac., Significant mission events, Flight problems, Flight discrepancies, Detailed objectives

104) A Summary of the Acoustic and Vibration Testing in Support of the Apollo Spacecraft Program: Vol. 2
General Electric - Houston; Peverley, R. W.; O.M. Lib; 12/01/69
Apollo spacecraft design, Vibration test criteria, SLA noise generation, Flight data analysis, Acoustic tests, Component list, Spacecraft vibration

105) Apollo 11 Mission Report
NASA/MSC; O.M. Lib; 12/01/69
Mission description, Pilot's report, Lunar descent and ascent, Communications, Trajectory, CSM performance, Lunar Mod. performance

106) Lunar Module Subsystem Assembly and Installations
Grumman; O.M. Lib; 12/01/69
Ascent stage structure, Descent stage structure, Landing gear, Electrical Power, Environmental control, Propulsion, Reaction control

107) Ascent propulsion system and descent propulsion system propellant budget program user's manual
TRW; TRW; 70X12093; 12/16/69; mf
Apollo project, Ascent propulsion systems, Descent propulsion systems, Fuel consumption, Computer programming, Computer programs, Flow charts

108) Master end-item specification for lunar module
75T13952; 12/17/69; mf

109) Lunar module cooling system chemical analysis program
72T10480; 12/22/69; mf

110) A Summary of the Acoustic and Vibration Testing in Support of the Apollo Spacecraft Program: Vol. 1
General Electric - Houston; Perverley, R. W.; O.M. Lib; 01/01/70
Qualification test, Acceptance vibration test, Test philosophy, Flight environments, Vibration and acoustic test, Revised vibration environments, Flight vibration da

111) Apollo mission H-2/LM-7/APS preflight performance report
TRW; Johnston, C.G.; 70X14002; 01/01/70; mf
Ascent propulsion systems, Lunar module, Preflight Analysis, Engine tests, Flight characteristics, Performance prediction, Simulation

112) Direct lunar landing survey and optimization program. Part 2 - Users guide final report.
Dynamics Research Corp; 70x13710*#; 01/03/70; mf
Lunar landing sites, Optimization, Surveys, Computer Programs, Lunar Orbits, Lunar trajectories, Mathematical models

113) Direct lunar landing survey and optimization program. Part 3 - Program description final report
Dynamics Research Corp; 70x13711*#; 01/03/70; mf
Lunar Landing Sites, Optimization, Surveys, Computer Programs, Lunar Orbits, Lunar Trajectories, Mathematical Models

114) Time Specific Apollo Lunar Surface Accessibility for Relaxed Free Return Missions - Computer Program Description
Bellcom, Inc.; Caldwell, S. F.; 79N71658*; 01/09/70
Apollo project, Lunar launch, Space missions, Spacecraft trajectories, Computer Programs, Launch dates, Launch windows

115) Direct lunar landing survey and optimization program. Part 1 - Mathematical formulation Final Report
Dynamics Research Corp.; 70X13709*#; 03/01/70; mf
Lunar landing sites, Optimization, Surveys, Computer programs, Lunar orbits, Lunar trajectories, Mathematical models

116) Lunar module 7, 8, and 9 elementary functional diagrams
75T14892; 03/10/70; mf

117) Lunar descent and ascent trajectories
NASA/JSC; Bennet, F. V.; 75N70840*; 04/21/70
Apollo 11 flight, Apollo 12 flight, Lunar trajectories, Postflight analysis, Ascent trajectories, Descent trajectories

118) Flight performance of the LM ascent and descent propulsion systems
Botwin, R.; 70A33577; 06/01/70; mf
Apollo flights, Ascent propulsion systems, Descent propulsion systems, Flight characteristics, Lunar module, Correlation, Data acquisition

119) Flight performance of the LM ascent and descent propulsion systems
Botwin, R.; 70A33577; 06/01/70; mf
Apollo flights, Ascent propulsion systems, Descent propulsion systems, Flight characteristics, Lunar module, Correlation, Data acquisition

120) Lunar module subsystem performance analysis program (LM SPAP) documentation
70T01003; 06/28/70

121) Lunar module subsystems model performance analysis Apollo 12
75T15486; 06/28/70

122) Apollo 13 Mission Report
NASA/MSC; O.M. Lib; 08/01/70
CSM performance, Lunar module performance, Mission consumables, Pilot's report, Anomaly summary, Mission support performance, Vehicle descriptions

123) Ascent Propulsion System Final Flight Evaluation: Apollo 11 Mission Report
NASA/JSC; Thompson, P. F.; 72X10107*#; 08/01/70; mf
Apollo 11 flight, Ascent propulsion systems, Postflight analysis, Lunar launch, Lunar module, Lunar module ascent stage

124) Ascent propulsion system final flight evaluation: Apollo 11 mission report
NASA-JSC; Thompson, P.F.; 72X10107; 08/01/70; mf
Apollo 11 flight, Ascent propulsion systems, Postflight analysis, Propulsion system performance, Lunar launch, Lunar module, Lunar module ascent stage

125) Apollo mission 12, trajectory reconstruction and postflight analysis, volume 1
TRW, NASA/JSC; 74N75212*; 08/10/70
Apollo 12 flight, Postflight analysis, Lunar landing, Lunar trajectories

126) Apollo 9 Mission Report. Performance of the Lunar Module Reaction Control System
70T00807; 08/28/70; mf

127) Lunar Module Systems Handbook. LM-8 Vehicle
70T00482; 09/15/70; mf

128) An investigation of LM primary propulsion systems restart capabilities
Grumman Aerospace; Backshall, R.G.; 71X70861; 10/01/70
Ascent propulsion systems, Descent propulsion systems, Lunar module, Cold flow tests, Injectors, Liquid propellant rocket engines, Restartable rocket engines

129) LM Fire-In-the-Hole Tests - A Simulation of the Lunar Launch Sequence
Grumman; Deane, I. J.; 71X70859; 10/01/70
Ascent propulsion, Lunar launch, Lunar module, Propulsion performance, Test firing, Heat shielding, Scale models

130) Lunar module propulsion flight anomalies
Grumman Aerospace; Wishney, I.P.; 71X70858; 10/01/70
Abnormalities, Ascent propulsion systems, Descent propulsion systems, Failure analysis, Lunar module, Propulsion system performance, Apollo flights

131) Plane Change Penalty for Unscheduled Abort from the Lunar Surface
Bellcom, Inc.; Schreiber, A. L.; 79N71902*; 10/01/70
Abort trajectories, Lunar launch, Orbit calculation, Trajectory analysis, Orbital Rendezvous, Rendezvous trajectories

132) Lunar Module Support Manual Transportation and Handling for LM-10
70T00916; 10/15/70; mf

133) Luna 17 space probe construction, landing stage, and roving vehicle chassis
Academy of Sciences; Anisov, K.S.; 72N18244; 01/01/71; mf
Lunar roving vehicles, Lunar spacecraft, Chassis, Lunar landing modules, Lunar probes

134) Moon samples by automation: Unmanned Soviet spacecraft Luna 16, describing landing techniques
Spaceflight; Gatland, K. W.; 71A16147; 01/01/71
Automatic control, Lunar landing, Lunik Lunar probes, Experiment design, Flight tests, Lunar rocks, Lunar topography

135) Performance analysis of the ascent propulsion subsystem of the Apollo spacecraft
ND526277; Hooper, J.C., III; 71X10468; 01/01/71; mf
Ascent propulsion systems, Lunar module, Performance prediction, Postflight analysis, Propulsion system performance, Computer programs, Computerized simulation

136) Performance analysis of the ascent propulsion subsystem of the Apollo spacecraft
ND526277; Hooper, J.C., III; 71X10468; 01/01/71; mf
Ascent propulsion systems, Lunar module, Performance prediction, Postflight analysis, Propulsion system performance, Computer programs, Computerized simulation

137) Technique for correcting large primary guidance and navigation control subsystem errors during lunar module powered descent
71T11834; 01/01/71

138) The Apollo docking system
North American Rockwell; Bloom, K.A.; 72N13392; 01/01/71; mf
Apollo spacecraft, Dynamic structural analysis, Lunar landing modules, Spacecraft docking, Conferences, Lunar spacecraft, Mechanical engineering

139) Delta V Performance of the Apollo LM Ascent trajectory
Bellcomm, Inc.; Yang, T. L.; 79N71976*#; 02/16/71; mf
Apollo 14 flight, Ascent trajectories, Lunar launch, Lunar module, Trajectory analysis, Trajectory optimization

140) Apollo 14 Mission Report
NASA/MSC; O.M. Lib; 04/01/71
Lunar surface experiments, Anomaly summary, Inflight demonstrations, CSM performance, Lunar module performance, Pilots report, Vehicle description

141) Operations handbook-lunar module 10 and subsequent-(volume 1-subsystems data)
71T12898; 04/01/71; mf

142) Unmanned lunar logistics vehicle may support the astronauts
Hendel, F.J.; 71A25529; 04/01/71; mf
Lunar exploration, Lunar logistics, Lunar spacecraft, Spacecraft design, Conferences, Instrument packages, Lunar flying vehicles

143) Landing dynamics of the lunar module (performance characteristics for the LM-10 vehicle)
71T11766; 04/14/71; mf

144) Landing dynamics of the lunar module-(method of analysis for the LM-10 vehicle)
71T12926; 04/14/71; mf

145) Out-of-plane performance requirements for lunar module ascent
71T12806; 05/28/71; mf

146) LM-11 actual weight report
Grumman Aerospace; Cunnius, D.; 72N12872; 06/01/71; mf
Center of gravity, Lunar module, Weight(mass), Ascent propulsion systems, Descent propulsion systems, Tables (data)

14.0 Appendix B, Apollo Lunar Module Weight Statement

14.1 Top Level Lunar Module (LM 11) Weight Statement at Earth Launch, Broken into Stages

			lbm	kg
A.		Ascent Stage Inert * Weight at E.L.	4,748.2	2,158.3
	1.0	Structure	1,364.8	620.4
	2.0	Stabilization and Control	79.2	36.0
	3.0	Navigation and Guidance	78.1	35.5
	4.0	Crew Provisions	138.7	63.0
	5.0	Environmental Control	295.6	134.4
	7.0	Instrumentation	128.2	58.3
	8.0	Electrical Power Supply	731.1	332.3
	9.0	Propulsion System	471.9	214.5
	10.0	Reaction Control	242.3	110.1
	11.0	Communications	114.6	52.1
	12.0	Controls and Displays	234.4	106.5
	13.0	Explosive Devices	28.4	12.9
	22.0	Manufacturing Variation		
		Hardware - Sub-Total	(3,907.3)	(1,776.0)
	14.0	Government Furnished Equipment	664.6	302.1
	15.0	Liquids and Gases - Excludes Propellant	135.7	61.7
	17.0	Propellant - Non-Tanked	(40.6)	(18.5)
		Main	14.1	6.4
		Reaction Control	26.5	12.0
B.		Descent Stage Inert* Weight at E.L.	5,795.2	2,634.2
	1.0	Structure	1,372.4	623.8
	2.0	Stabilization and Control	13.3	6.0
	3.0	Navigation and Guidance	44.0	20.0
	4.0	Crew Provisions	148.3	67.4
	5.0	Environmental Control	207.5	94.3
	6.0	Landing Gear	479.8	218.1
	7.0	Instrumentation	6.7	3.0
	8.0	Electrical Power Supply	785.0	356.8
	9.0	Propulsion System	1,085.5	493.4
	11.0	Communications	13.8	6.3
	12.0	Displays and Controls	3.3	1.5
	13.0	Explosive Device	24.6	11.2
	22.0	Manufacturing Variation		
		Hardware - Sub-Total	(4,184.2)	(1,901.9)
	14.0	Government Furnished Equipment	1016.0	461.8
	15.0	Liquids and Gases - Excludes Propellant	518.2	235.5
	17.0	Propellant - Non-tanked	76.8	34.9
Total Inert Weight at Earth Launch			10,543.4	4,792.5
Ascent Stage at Earth Launch			4,748.2	2,158.3
Descent Stage at Earth Launch			5,795.2	2,634.2
RCS Propellant tanked			604.5	274.8
Ascent Main Propellant tanked			5,229.1	2,376.8
Descent Main Propellant tanked			19,524.9	8,875.0
		Total Vehicle Earth Launch	35,901.9	16,319.0

* Inert weight without tanked propellant.

14.2 Apollo Lunar Module Level 2 Weight Statement

The following LM weight statement was collected from data that compared LMs of the Apollo Program. The comparisons were furnished by Buddy Heineman in the Advanced Programs Office at JSC. Two weight statements used, dated 1/17/70 and 9/18/70, were not completely consistent with each other, having a deviation of 0.37%. The mass summary included in this report uses data from both because complete data for one was not available. The mass summary is meant to be used for comparison or estimation for lander conceptual designs. The mass statement includes generations one, two, and four of the LM used on Apollo 17. Several lower levels of detail are available in reference 1.

Summary Mass Statement LEM 11

All masses in kg's

1.	Ascent structure		627.9
1.1	Front face		82.7
	1.1.1	Front face skins	21.0
	1.1.2	Window shielding	0.7
	1.1.3	Beams vertical	10.3
	1.1.4	Beam caps	1.0
	1.1.5	Stiffeners skin	4.9
	1.1.6	Stiffeners skin	3.9
	1.1.7	Window frames	6.4
	1.1.8	Interstage mts ext	6.0
	1.1.9	EVA handrail instl	2.9
	1.1.10	CBN 340 supts	1.0
	1.1.11	CBN supts	0.1
	1.1.12	CBN eps sups	1.2
	1.1.13	CBN comm sups	0.3
	1.1.14	FF windows	10.9
	1.1.15	FF hatch	5.9
	1.1.16	FF jsf	6.2
1.2	Cabin		91.6
	1.2.1	Cabin skins	13.6
	1.2.2	Window shielding	0.3
	1.2.3	Cabin imu beams	8.5
	1.2.4	Cabin longerons	9.8
	1.2.5	Frames cabin skins	7.6
	1.2.6	Frames upr dkg wnd	2.7
	1.2.7	CBN 340 supts	16.7
	1.2.8	ECS supts	1.6
	1.2.9	CBN eps supts	1.5
	1.2.10	CBN rcs supts	3.4
	1.2.11	CBN cons sups	0.9
	1.2.12	Cabin deck	10.7
	1.2.13	Cabin window	1.8
	1.2.14	Cabin jsf	12.5
1.3	Midsection		259.2
	1.3.1	Tunnel skins	7.0
	1.3.2	MS skins	10.4
	1.3.3	MS bulkheads	44.2
	1.3.4	Beams Y22	7.5
	1.3.5	Beams Y17	5.4
	1.3.6	Beams Y37	1.9
	1.3.7	Beams engine	6.8
	1.3.8	Beams bulkheads	9.8
	1.3.9	MS longerons	4.3
	1.3.10	MS stiffeners	14.5
	1.3.11	MS frames	10.5
	1.3.12	MS interstage mts	9.9
	1.3.13	MS s&C sups	0.4

		1.3.14	MS n&g sups	0.4
		1.3.15	MS 340 sups	24.5
		1.3.16	MS ecs sups	7.8
		1.3.17	MS inst sups	0.5
		1.3.18	MS eps sups	5.5
		1.3.19	MS prop sups	13.7
		1.3.20	MS rcs sups	22.6
		1.3.21	MS comm sups	6.3
		1.3.22	MS gfe sups	1.5
		1.3.23	MS deck X2335	6.4
		1.3.24	MS deck X2535	8.3
		1.3.25	MS decks X277	3.5
		1.3.26	MS decks X277	10.0
		1.3.27	MS deck X310	0.9
		1.3.28	MS hatch	4.9
		1.3.29	MS jsf	9.8
	1.4	AEB total		31.8
		1.4.1	AEB racks-wo-cp	7.4
		1.4.2	AEB horizontal bms	4.4
		1.4.3	AEB colp-p-ate-asy	8.5
		1.4.4	AEB etg & trusses	2.8
		1.4.5	AEB ecs supts	0.5
		1.4.6	AEB inst supts	0.1
		1.4.7	AEB eps supts	2.4
		1.4.8	AEB prop supts	0.5
		1.4.9	AEB pcs supts	4.4
		1.4.10	AEB anta supts	0.2
		1.4.11	AEB jsf	0.6
	1.5	A/S thermo protection		162.5
		1.5.1	FRT face cb shield	17.7
		1.5.2	FF/F cabin insul	12.5
		1.5.3	Front face cab jsf	5.7
		1.5.4	Front face cab supt	1.9
		1.5.5	Midsection shield	26.5
		1.5.6	Midsection insul	25.0
		1.5.7	Midsection jsf	2.4
		1.5.8	M/S shield supts	47.5
		1.5.9	AEB shielding	8.8
		1.5.10	AEB insulation	7.3
		1.5.11	AEB jsf	1.3
		1.5.12	AEB supports	5.9
	1.6	Ascent miscl/lcd		0.1
2.	Descent structure			650.5
	2.1	Forward section		99.4
		2.1.1	Web fwd end clos	2.7
		2.1.2	Upr cap fwd clos	0.1
		2.1.3	Lwr cap fwd clos	0.8
		2.1.4	Post-left fwd clos	2.9
		2.1.5	Post-right fwd clos	2.9
		2.1.6	Stiffeners fwd clos	2.0
		2.1.7	LG fittings fwd clos	0.9
		2.1.8	JSF fwd closure	1.4

	2.1.9	Forward left-panel	9.3
	2.1.10	Forward right-panel	10.5
	2.1.11	Forward upper deck	7.3
	2.1.12	Forward lower deck	5.4
	2.1.13	Fwd equipment bay	15.7
	2.1.14	Fwd equip bay right	4.8
	2.1.15	Fwd interstg mount	2.0
	2.1.16	Fwd int-stg mt col	1.3
	2.1.17	Fwd int-st sup trus	6.9
	2.1.18	Fwd oxid tank supt	15.0
	2.1.19	Fwd truss supports	9.8
	2.1.20	S-iv-b attach ftg	3.5
	2.1.21	Fwd egress platform	3.9
	2.1.22	Equipt spt fwd sct	0.3
2.2	Center section lft		69.0
	2.2.1	Web center lft clo	2.5
	2.2.2	Upr cab cen-lf clo	0.5
	2.2.3	Lwr cap cen-lf clo	0.8
	2.2.4	Post aft lft clos	2.9
	2.2.5	Post fwd lft clos	3.0
	2.2.6	Stif cent-lft-clos	1.9
	2.2.7	Lg ftg cent-lft clos	1.0
	2.2.8	Jsf center-lft clos	1.1
	2.2.9	Fwd pan center-lft	7.9
	2.2.10	Aft pan center-lft	10.1
	2.2.11	Upr deck center-lf	7.5
	2.2.12	Lwr deck center-lf	5.1
	2.2.13	Truss & sut cnt-lft	21.2
	2.2.14	S-iv-b attach fig	3.5
2.3	Mid-center section		51.1
	2.3.1	LF pan mid-center	6.6
	2.3.2	Rt pan mid-center	6.4
	2.3.3	Fwd pan mid-center	6.7
	2.3.4	Aft pan mid-center	6.3
	2.3.5	Eng act supt m-c-s	2.8
	2.3.6	Eng trus supt col	3.6
	2.3.7	Eng supt trusses	4.8
	2.3.8	Blast deflector	3.9
	2.3.9	Jsf cruciforms etc.	2.2
	2.3.10	Equipt spt cnt mid	7.8
2.4	Center section rt		74.2
	2.4.1	Web center-rt clos	2.6
	2.4.2	Upr cap cen-rt clo	0.5
	2.4.3	Lwr cap cent-rt clo	0.8
	2.4.4	Post fwd rt clos	3.0
	2.4.5	Post aft rt clos	2.9
	2.4.6	Stif cent-rt clos	1.9
	2.4.7	Lg ftg cent-rt clos	0.9
	2.4.8	End clos center rt	1.0
	2.4.9	Fwd pan center-rt	9.8
	2.4.10	Aft pan center-rt	8.8
	2.4.11	Upr deck center-rt	7.3

	2.4.12 Lwr deck center-rt	5.5
	2.4.13 Truss & supt. center	23.4
	2.4.14 S-iv-b attach ftg	3.4
	2.4.15 Cold plate & rails	0.2
	2.4.16 Equipt spt cnt rt	2.2
2.5	Aft section	124.3
	2.5.1 Web aft end clos	3.0
	2.5.2 Upr cap aft clos	0.5
	2.5.3 Lwr cap aft clos	0.8
	2.5.4 Post left aft clos	3.2
	2.5.5 Post right aft clos	3.2
	2.5.6 Stiffeners aft clos	1.2
	2.5.7 Lg ftg aft closure	0.9
	2.5.8 Jsf aft end clos	1.4
	2.5.9 Aft left panel	9.6
	2.5.10 Aft right panel	9.3
	2.5.11 Aft upper deck	6.8
	2.5.12 Aft lower deck	5.4
	2.5.13 Sci eq bay lwr dk	8.9
	2.5.14 Sci eq bay-upr dk	3.6
	2.5.15 S e b diag cap ii	5.6
	2.5.16 Sci eq bay rt clos	0.8
	2.5.17 Sci eq bay cntr pan	1.0
	2.5.18 Sci eq bay lft clo	0.8
	2.5.19 Sci eq bay inb-pan	1.5
	2.5.20 Equip bay right	7.8
	2.5.21 Access panels	0.6
	2.5.22 Aft lg supt truss	6.9
	2.5.23 Aft oxid tank supt	15.0
	2.5.24 Misc tank supports	7.3
	2.5.25 Aft truss & support	0.5
	2.5.26 S-iv-b attach ftg	3.4
	2.5.27 Equipt spt aft sct	15.3
2.6	Base heat shield	0.3
2.7	Miscellaneous	3.9
	2.7.1 Land gear shocks	1.5
	2.7.2 Miscellaneous	2.4
2.8	Mess stow + rel struct	10.6
2.9	Alsep remot deploy	14.5
2.10	Thermal protection	179.2
	2.10.1 Upper shielding	25.2
	2.10.2 Upper insulation	12.5
	2.10.3 Upper jsf	0.2
	2.10.4 Upper supports	2.0
	2.10.5 Side shielding	6.8
	2.10.6 Side insulation	24.2
	2.10.7 Side jsf	0.9
	2.10.8 Side supports	2.2
	2.10.9 Lower shielding	7.8
	2.10.10 Lower insulation	12.7
	2.10.11 Lower jsf	0.5
	2.10.12 Lower supports	1.9

2.10.13	Rhbs shielding	14.0
2.10.14	Rhbs insulation	30.1
2.10.15	Rhbs jsf	3.0
2.10.16	Remove bhs supports	27.3
2.10.17	Outrig shielding	3.5
2.10.18	Outrig insulation	4.4
2.11	Descent structure	24.0
3.	Stab control	41.9
3.1	Stab and control ascent	35.9
3.1.1	Atca	10.8
3.1.2	Rga	0.9
3.1.3	Abort guidance syst	24.2
3.2	Stab and control descent	6.0
3.2.1	DECA	6.0
4.	Navigation and guidance	55.1
4.1	Navigation and guid ascent	35.6
4.1.1	Pen Radar Sect.	35.6
4.2	Navigation and guid descent	19.5
4.2.1	Land Radar Sect.	19.5
5.	Crew provisions	166.7
5.1	Crew prov ascent	67.6
5.1.1	Outer lighting	12.0
5.1.2	Inner lighting	2.0
5.1.3	Miscellaneous	4.0
5.1.4	Waste management	3.1
5.1.5	Furnishings	26.1
5.1.6	Paint	2.7
5.1.7	Crew prov ascent	17.7
5.2	Crew prov descent	99.1
5.2.1	Descent	24.4
5.2.2	Mesa module	74.7
6.	Environmental cont	228.0
6.1	Lunar stay ascent	134.8
6.1.1	Ham std 190 pkg	46.5
6.1.2	190 pkg-hdw	4.5
6.1.3	O2+H2O coolant as	16.9
6.1.4	Atmos revit sect	1.3
6.1.5	-290 subtotal	3.6
6.1.6	Hts pri loop	11.2
6.1.7	Hts secnd loop	3.6
6.1.8	Total gox tanks	4.6
6.1.9	390 ox module	4.0
6.1.10	390 pkg-hdw	3.4
6.1.11	Plss O2 recharge	1.6
6.1.12	Total H2O tanks	4.9
6.1.13	490 H2O module	2.6
6.1.14	490 pkg-hdw	2.2
6.1.15	Plumbing gaec	13.0
6.1.16	Cold plates r&d	1.3
6.1.17	Total primary cp	1.2
6.1.18	Translunar phase	8.3
6.2	Lunar stay-descent	93.2

6.2.1	Inert d/s lm	42.0
6.2.2	O2 system	27.4
6.2.3	H2O system	10.3
6.2.4	Glycol system	1.4
6.2.5	Plss O2 recharge	3.7
6.2.6	LiOH cartridges	8.4
7.	Landing gear installation	220.7
7.1	Primary-strut-assembly	97.7
7.1.1	Inner-cyl-assys	44.2
7.1.2	Outer-cyl-assys	40.2
7.1.3	Cartridge-assys	11.1
7.1.4	Jsf-primary-strut	2.2
7.2	Secondary strut assembly	30.6
7.2.1	Inner-cyl-assys	6.9
7.2.2	Outer-cyl-assys	6.6
7.2.3	Cartridge assys	17.1
7.3	Landing pad assemblies	21.2
7.3.1	Honeycomb-panel-as	18.1
7.3.2	Bumper assembly	0.5
7.3.3	Hub assembly	2.3
7.3.4	Jsf-landing-pad-as	0.3
7.4	Deploy truss assembly	37.4
7.4.1	Cross member tube a	5.0
7.4.2	Cross member tube a	7.3
7.4.3	Side brace tube as	15.4
7.4.4	Side brace tube as	3.2
7.4.5	Misc Deploy comps	3.8
7.4.6	Jsf deploy truss as	2.7
7.5	Mechanisms	7.8
7.5.1	Deployment spring	1.5
7.5.2	Lock spring assys	0.6
7.5.3	Down lock hatch as	1.3
7.5.4	Crank cam idler	1.0
7.5.5	Surface probe mech	3.3
7.5.6	Jsf mechanisms	0.1
7.6	Thermal insul lg	21.0
7.7	Egress ladder assy	3.2
7.7.1	Egress ladder	3.0
7.7.2	Jsf egress ladder	0.2
7.8	Jsf inst and gear	1.8
7.8.1	Jsf instl pri struc	0.7
7.8.2	Jsf instl sec stru	0.1
7.8.3	Jsf instl and pad	0.8
7.8.4	Jsf deploy truss	0.1
7.8.5	Jsf lg instl	0.1
8.	Instrumentation	61.5
8.1	Instrument ascent	58.5
8.1.1	Signal conditioner	32.2
8.1.2	Pcmtea	10.3
8.1.3	Data storage unit	1.1
8.1.4	Caution and warning	8.3
8.1.5	Aeb jsf ascent	0.1

		8.1.6 Fcs sensors ascent	2.7
		8.1.7 Prop sensors ascent	1.6
		8.1.8 Rcs sensors ascent	2.2
	8.2	Instrumentation descent	3.0
		8.2.1 Ecs sensors descent	0.8
		8.2.2 Prop sensors descent	1.2
		8.2.3 Mech des sensor ds	0.9
		8.2.4 Instr sensors ds	0.1
9.	Electrical Power		688.5
	9.1	Eps ascent stage	332.4
		9.1.1 Battery ascent	112.8
		9.1.2 Elect cont assy as	9.5
		9.1.3 Invertor ascent	13.9
		9.1.4 Electronic unit as	9.3
		9.1.5 Panel 11 ascent	18.4
		9.1.6 Panel 16 ascent	15.8
		9.1.7 GN&C hrns assy asc	2.3
		9.1.8 S&C hrns assy asc	0.1
		9.1.9 Ecs hrns assy asc	5.5
		9.1.10 Instr hrns assy as	0.8
		9.1.11 Eps hrns assy asc	5.9
		9.1.12 Prop hrns assy asc	1.4
		9.1.13 Rcs hrns assy asc	0.5
		9.1.14 Comm hrns assy asc	2.0
		9.1.15 Eds hrns assy asc	2.3
		9.1.16 Multi subsys hrns	105.4
		9.1.17 Misc hrns assy asc	0.6
		9.1.18 Total inst hdwr as	22.3
		9.1.19 Delta wt chgs asc	0.3
		9.1.20 Wire adjustment	3.3
	9.2	Eps descent stage	356.1
		9.2.1 Battery descent	304.0
		9.2.2 Elect cont assy ds	18.4
		9.2.3 Electronic unit ds	1.5
		9.2.4 GN&C hrns assy dsc	1.0
		9.2.5 Ecs hrns assy dsc	0.1
		9.2.6 Instr hrns assy ds	0.3
		9.2.7 Eps hrns assy dsc	0.7
		9.2.8 Prop hrns assy dsc	2.1
		9.2.9 Comm hrns assy dsc	1.2
		9.2.10 Eds hrns assy dsc	3.1
		9.2.11 Multi-subsys hrns	14.2
		9.2.12 Eps instl hdwr dsc	5.6
		9.2.13 Eps descent stage	3.9
10.	Total Propulsion		706.6
	10.1	Total inert ascent	214.0
		10.1.1 Propellnt tnk inst	59.3
		10.1.2 Prop. quan. sensors	0.8
		10.1.3 Propellant plumbing	10.4
		10.1.4 Helium tanks asc	48.9
		10.1.5 Pressuriz plumb as	11.9
		10.1.6 Pressuriz sys asc	2.0

	10.1.7	Ascent engine istl	79.6
	10.1.8	Engine & misc asc	1.1
10.2		Total inert desc	492.6
	10.2.1	Prop tank inst des	216.5
	10.2.2	Prop fd disnct ins	2.6
	10.2.3	Propellant plumb d	20.8
	10.2.4	Helium tank desc	51.7
	10.2.5	Pressuriz plumb de	30.9
	10.2.6	Engine & misc desc	170.1
11.		Reaction control	109.7
11.1		RCS propellant sys	45.4
	11.1.1	Fuel tanks	8.0
	11.1.2	Oxidizer tanks	9.5
	11.1.3	Rc asc prop tie in	6.1
	11.1.4	Fuel system	10.0
	11.1.5	Oxidizer system	10.0
	11.1.6	Pr flt & iso vv ass	1.8
11.2		Pressurization sys	18.2
	11.2.1	Helium tanks	7.5
	11.2.2	Plumbng pres sys a	5.4
	11.2.3	Plumbng pres sys b	5.3
11.3		Thruster instls	46.1
	11.3.1	Thrust chbr assys	37.4
	11.3.2	Hardware cluster 1	2.2
	11.3.3	Hardware cluster 2	2.2
	11.3.4	Hardware cluster 3	2.2
	11.3.5	Hardware cluster 4	2.1
12.		Communicants	58.4
12.1		Communicants asct	52.1
	12.1.1	VHF xceiver & dipl	6.0
	12.1.2	Sig processor assy	4.8
	12.1.3	VHF in-flt ants	2.2
	12.1.4	UHF ranging assy	1.4
	12.1.5	EVA antenna assy	0.9
	12.1.6	S-band transceiver	9.1
	12.1.7	Pwr ampl & diplex	8.6
	12.1.8	In-flt antennas	0.4
	12.1.9	Steerable antenna	12.7
	12.1.10	Communicants asct	6.0
12.2		Communicants desc	6.3
	12.2.1	Erectable antenna	6.3
13.		Controls & display	107.9
13.1		Asc control & display	106.4
	13.1.1	Support structure	14.4
	13.1.2	Panel 1	21.5
	13.1.3	Panel 2	19.2
	13.1.4	Panel 3	9.1
	13.1.5	Panel 4a	0.4
	13.1.6	Panel 4b	0.4
	13.1.7	Panel 5	2.5
	13.1.8	Panel 6	4.3
	13.1.9	Panel 8	5.1

13.1.10	Panel 12	9.3
13.1.11	Panel 14	4.2
13.1.12	Non-panel items	15.9
13.1.13	Asc control & displ	0.1
13.2	Des control & display	1.5
13.2.1	Des control & displ	1.5
14.	Elect expl device	24.0
14.1	Elect expl device	24.0
14.1.1	Asct explos device	8.6
14.1.2	Ascent structure	4.3
14.1.3	Dsct explos device	6.4
14.1.4	Descent structure	4.7
15.	Total gfe at E.L.	782.1
15.1	Earth launch ascent	300.8
15.1.1	Drogue	9.1
15.1.2	Bpa installed hdwe	0.6
15.1.3	Mit nav and guid	120.2
15.1.4	Crew prov. ascent	167.4
15.1.5	Instr scien eqp as	2.9
15.1.6	Electrical ascent	0.6
15.2	E.L. equip descent	481.3
15.2.1	Plss batteries dsc	16.1
15.2.2	Bpa installed hdw	0.1
15.2.3	Crew provis desct	22.3
15.2.4	Scient. equip desc	442.8
16.	Liquids & gases	296.8
16.1	Liquids & gas asct	61.7
16.1.1	Total coolant asc	11.2
16.1.2	Tanked gox ascent	2.2
16.1.3	Water residual asc	0.5
16.1.4	Water tanked asc	40.8
16.1.5	Nitrogen asc H2O tk	0.1
16.1.6	Helium ascent aps	5.9
16.1.7	Helium ascent rcs	1.0
16.2	Liquids & gas desc	235.1
16.2.1	Coolant descent	1.2
16.2.2	Gox descent	43.5
16.2.3	Water residual dsc	0.2
16.2.4	Water tanked dsc	166.0
16.2.5	Nitrogen dsc H2O tk	0.5
16.2.6	Helium descent dps	23.7
17.	Total delta-v	10,812.6
17.1	Main propel delta-v	10,582.1
17.1.1	Delta-v propel asc	2,257.4
17.1.2	Delta-v propel dsc	8,324.7
17.2	Rcs Propel delta-v	230.5
17.2.1	Rcs propel delta-v	230.5
18.	Total non delta-v	743.0
18.1	Total unuse. main	687.3
18.1.1	Trapped aps	6.4
18.1.2	Unused aps prop	21.8
18.1.3	Disp & Malfunction	37.5

18.1.4	Unuse. oxid ascent	55.2
	18.1.5 Trapped dps	34.8
	18.1.6 Unused dps prop	74.1
	18.1.7 Unused dps prop	175.4
	18.1.8 Disp & malfunction	282.1
18.2	Total rcs propellant	55.7
	18.2.1 Unuse.rcs propelnt	55.7
Total mass of LEM		16,381.9

15.0 Appendix C, Lunar OTV Calculations

The following printout shows the steps in an OTV sizing calculation representative of the calculations performed in this study. The calculation starts with the OTV and payload at the end of mission in LEO and works backward to the start of the mission. Significant assumptions include the following:

1) 15% of the Earth entry mass of an OTV/payload is aerobrake mass.

2) Other OTV inert mass is sized using the formula:

   Inert mass = A + B * Wp where

   A = engines, etc. = 2.5 m tons for 2 stage, 4.5 for single stage.

   B = .05

   Wp = Propellant mass

3) Unusable and flight performance reserve propellants are held at 2.3 % of the total propellant.

4) If the lander is returned to LEO for service, no additional OTV crew module is carried. If the lander is loaded with propellants in LLO, the OTV must carry an additional crew module.

5) When two OTV stages are used, they are sized to hold equal quantities of propellant. Table 15-1 shows the mass breakdowns for several cases of interest.

6) TLI, LOI, and EOI delta Vs come from an Eagle produced program and assume a 93 km LLO, a 450 km Space Station Orbit, and all operations in the same plane.

Other assumptions can be seen in the printout.

Table 15-1, OTV Weight Statements

All masses are metric tons unless otherwise indicated.

Item	2 Stage OTV Load propellants for lander in: LEO	2 Stage OTV Load propellants for lander in: LLO	1 Stage OTV Load propellants for lander in: LEO	1 Stage OTV Load propellants for lander in: LLO
- 6 m ton crew module, 32 m ton lander (including payload)				
Propellant related inerts (5% of prop.)	2.1	1.6	4.5	3.5
Other inert engines, etc	2.5	2.5	4.5	4.5
Aerobrake mass (15% of entry mass)	3.2	2.0	4.3	3.0
Total OTV Inert	7.8	6.1	13.3	11.0
Unusable and FPR prop. (2.3% of Total prop.)	0.9	0.7	2.0	1.6
Total OTV prop. capacity	39	31	89	71
- 25 m ton one way down payload, 57 m ton (including payload) expended lander				
Propellant related inerts - tanks (5% of prop.)	2.5	-	4.5	-
Other inert - engines, etc.	2.5	-	4.5	-
Aerobrake mass (15% of entry mass)	1.2	-	2.4	-
Total OTV inert	6.2	-	11.4	-
Unusable and FPR prop. (2.3% of total prop.)	1.2	-	2.7	-
Total OTV prop. capacity	51	-	116	-

3-22-88 Propellant Loading Locations Comparison, Crew Rotation Only Mission

Loading -> Options	1. Return lander to Space Sta. 2 Stage OTV	2. Load Prop. in lunar orbit 2 Stage OTV	3. Return lander to Space Sta. 1 Stage OTV	4. Load Prop. in lunar orbit 1 Stage OTV

Earth Aerocapture and Perigee Raise and Rendezvous
(The 2nd Stage OTV returns itself, a crew capsule and whatever other mass must be returned to the Space Station to LEO)

Perigee raise & rendezvous in LEO, km/sec	0.24	0.24	0.24	0.24
% of entry mass that is aerobrake, %	15	15	15	15
% of prop. that is inert mass, % (B)	5	5	5	5
Perigee raise Isp, sec	455	455	455	455
OTV Prop. related inerts, m ton B*Wp	2.1	1.6	4.5	3.5
Other OTV (A) inert mass, m tons	2.5	2.5	4.5	4.5
Unusable & FPR propellants, m tons	0.9	0.7	2.0	1.6
OTV Crew compartment, m tons	0	6	0	6
Returned lander inert, m tons	6	0	6	0
Returned crew compartment, m tons	6	0	6	0
Aerobrake mass metric tons	3.2	2.0	4.3	3.0
Mass of veh. & payload after per. raise & rend.	21	13	27	19
Mass ratio, per. raise & rend. burn	1.06	1.06	1.06	1.06
Perigee raise	1.14	0.71	1.51	1.03

3-22-88 Propellant Loading Locations Comparison, Crew Rotation Only Mission

Loading -> Options	1. Return lander to Space Sta. 2 Stage OTV	2. Load Prop. in lunar orbit 2 Stage OTV	3. Return lander to Space Sta. 1 Stage OTV	4. Load Prop. in lunar orbit 1 Stage OTV
& rend. propellant, m tons				
Mass of veh. & payload before per. raise & rend.	22	14	29	20

Departure from Lunar Orbit or Trans Earth Insertion (TEI)
(The 2nd stage OTV departs lunar orbit with a crew module
and whatever else is being returned to LEO)

Delta V, TEI km/sec	0.846	0.846	0.846	0.846
Delta V, midcourse, km/sec	0.06	0.06	0.06	0.06
Total Delta V km/sec	0.906	0.906	0.906	0.906
TEI Isp, sec	455	455	455	455
TEI mass ratio	1.23	1.23	1.23	1.23
Mass befor TEI & MC, m tons	27	17	35	24
TEI & MC Prop. m tons	5	3	6	4
Mass befor TEI & MC less returned equipment m tons	15	17	23	24

Lunar Orbit Insertion (LOI)
(2nd Stage OTV arrives in lunar orbit with a payload
from LEO)

Delta V, LOI km/sec	0.846	0.846	0.846	0.846
Mid course km sec	0.06	0.06	0.06	0.06
MC + LOI km/sec	0.906	0.906	0.906	0.906
LOI Isp, sec	455	455	455	455

3-22-88 Propellant Loading Locations Comparison, Crew Rotation Only Mission

Loading -> Options	1. Return lander to Space Sta. 2 Stage OTV	2. Load Prop. in lunar orbit 2 Stage OTV	3. Return lander to Space Sta. 1 Stage OTV	4. Load Prop. in lunar orbit 1 Stage OTV
LOI mass ratio	1.23	1.23	1.23	1.23
Payload Mass to lunar orbit m tons	32	21	32	21
Total Mass after LOI,MC, m tons	47	38	55	45
Total Mass before LOI,MC, m tons	57	46	68	55
LOI Prop m tons	11	8	12	10
Total OTV Prop used for Per. & rend.,MC,TEI,LOI,MC	17	12	20	16

2nd Stage TLI Burn
(The 2nd stage OTV makes its final escape burn from LEO at the perigee of a high ellipse after staging from the first stage OTV)

2nd stage burn (TLI total - 1st stage burn),km/sec (TLI total = 3.101 km/sec)	1.471	1.521	3.101	3.101
2nd Stage Isp,	455	455	455	455
2nd Stg mass ratio	1.39	1.41	2.00	2.00
Mass befor 2nd stage burn, m tons	80	65	136	111
2nd Stage Prop m tons	22	19	68	55
Total OTV Prop used for Per. & rend.,MC,TEI,LOI,MC,2nd TLI	39	31	89	71

Perigee Raise and Rendezvous for returned 1st Stage OTV
(Before the 1st TLI burn can be calculated, the propellant needed to put the 1st stage OTV back into LEO after aerocapture must be determined)

Circ. & Rend. after aerocapture, km/sec	0.24	0.24	0.00	0.00

3-22-88 Propellant Loading Locations Comparison, Crew Rotation Only Mission

Loading -> Options	1. Return lander to Space Sta. 2 Stage OTV	2. Load Prop. in lunar orbit 2 Stage OTV	3. Return lander to Space Sta. 1 Stage OTV	4. Load Prop. in lunar orbit 1 Stage OTV
1st Stg Isp	455	455	455	455
Cric.& Rend. mass ratio	1.06	1.06	1.00	1.00
1st stg OTV Prop. related inerts, m ton B*Wp	2.1	1.6	0	0
Other OTV (A) inert mass, m tons	2.5	2.5	0.0	0.0
1st stg Aero-brake mass, m tons	3.2	2.0	0.0	0.0
Unused Prop. & Flight Perf. Reserve (2.25% of total propellant)	0.9	0.7	0	0
Mass after circ.& rend. burn, m tons	8.70	6.80	0.00	0.00
Mass before circ.& rend. burn, m tons	9.18	7.18	0.00	0.00
Propellant req for circ. & rend.	0.48	0.38	0.00	0.00

1st Stage TLI Burn
(The stack of 2 OTVs departs LEO with a cargo for low lunar orbit)

1st stage burn km/sec	1.63	1.58	0.00	0.00
1st Stage Isp,	455	455	455	455
1st Stg mass ratio	1.44	1.43	1.00	1.00
1st stg OTV Prop. related inerts, m ton B*Wp	2.1	1.6	0.0	0.0
Other OTV (A) inert mass, m tons	2.5	2.5	0	0
1st stg Aero-brake mass, m tons	3.2	2.0	0.0	0.0

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3-22-88 Propellant Loading Locations Comparison, Crew Rotation Only Mission

Loading --> Options	1. Return lander to Space Sta. 2 Stage OTV	2. Load Prop. in lunar orbit 2 Stage OTV	3. Return lander to Space Sta. 1 Stage OTV	4. Load Prop. in lunar orbit 1 Stage OTV
Mass after 1st stage burn, m tons	88	71	136	111
Mass befor 1st stage burn, m tons	127	101	136	111
1st Stage LOI. Prop., m tons	39	30	0	0
Total 1st Stg. Prop., m tons LOI, Per. raise & rend.	39	31	0	0
Total OTV Prop used for EOI,MC,TEI,LOI,MC,2nd TLI,1st TLI perigee raise and rend., for both	78	62	89	71

Condensed Input (Variables)

Payload to lunar orbit, m tons	32	21	32	21
Returned landr inert, m tons	6	0	6	0
Returned landr crew module, m tons	6	0	6	0
OTV Isp, sec	455	455	455	455
Aerobrake fraction, %	15	15	15	15
TLI Total Delta V, km/sec	3.101			
LOI/TEI Delta V, km/sec	0.846			

Condensed Output

Stack Mass in LEO, m tons	127	101	136	111
1st Stg Prop. capacity, m tons	39	31	0	0

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3-22-88 Propellant Loading Locations Comparison, Crew Rotation Only Mission

Loading -> Options	1. Return lander to Space Sta. 2 Stage OTV	2. Load Prop. in lunar orbit 2 Stage OTV	3. Return lander to Space Sta. 1 Stage OTV	4. Load Prop. in lunar orbit 1 Stage OTV
OTV inert less crew module & Aerobrake mass	5	4	9	8
Aerobrake Mass metric tons	3	2	4	3

Iteration Steps

1. Guess delta V split between 1st and 2nd Stage OTV.
   Change 1st to get propellant masses the same.

1st stge delta V	1.63	1.58	0.00	0.00
1st stage Propellant	39	31	0	0
2nd stage Propellant	39	31	89	71

2. Check propellant related inert. Set at proper % of OTV propellant

% Desired	5	5	5	5
% Actual	5	5	5	5
OTV prop. related inerts	2.10	1.60	4.50	3.50

3. Check unused propellant and FPR as a fraction of total propellant

% Desired	2.3	2.3	2.3	2.3
% Actual	2.3	2.3	2.3	2.3
Set unused and FPR mass, m tons	0.90	0.70	2.00	1.60

4. Check aerobrake as a fraction of entry mass

% Desired	15	15	15	15
% actual	15	15	15	15
Set new aerobrake mass, m tons	3.20	2.00	4.30	3.00

5. Return to 1. and repeat as required.