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LUNAR RESOURCES UTILIZATION FOR SPACE CONSTRUCTION

FINAL REPORT

VOLUME III • APPENDICES

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FOREWORD

This final report was prepared by General Dynamics Convair Division for NASA/JSC in accordance with Contract NAS9-15560, DRL No. T-1451, DRD No. MA-677T, Line Item No. 4. It consists of three volumes: (I) A brief Executive Summary; (II) a comprehensive discussion of Study Results; and (III) a compilation of Appendicies to further document and support the Study Results.

The study results were developed from April 1978 through February 1979, followed by preparation of the final documentation. Reviews were presented at JSC on 18 October 1978 and 21 February 1979.

Participants who significantly contributed to this study include General Dynamics Convair personnel, a materials processing and manufacturing consultant, and five technical reviewers who are nationally recognized authorities on lunar materials and/or space manufacturing.

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In addition to these participants, useful supportive information was obtained from two complementary study activities, from personnel at NASA's Johnson Space Center and Lewis Research Center, and from many academic and industrial researchers who are involved with development of manufacturing processes which may be especially suited for in space use.

- Contract NAS09-051-001 "Extraterrestrial Materials Processing and Construction" being performed by Dr. Criswell of LPI under the direction of JSC's Dr. Williams.

- Contract NAS8-32925 "Extraterrestrial Processing and Manufacturing of Large Space Systems" being performed by Mr. Smith of MIT under the direction of MSFC's Mr. von Tiesenhausen.

- Earth Baseline Solar Power Satellite costing information from Mr. Harron, Mr. Whittington, and Mr. Wadle of NASA's Johnson Space Center.

- Ion Electric Thruster information for argon and oxygen propellants provided by Mr. Regetz and Mr. Byers of NASA's Lewis Research Center.

- Electron Beam Vapor Deposition of Metals Information from Dr. Schiller of Forschungsinstitut Manfred Von Ardenne, Dresden, and Dr. Bunshah of UCLA, plus others.

- Solar Cell Manufacturing Information from Mr. Wald of Mobile Tyco Solar Energy Corp., Mr. Minnucci and Mr. Younger of SPIRE Corp., and Mr. Dubik of Schott Optical Glass Co., plus others.

- Glass Manufacture Using Lunar Materials Information from Dr. MacKenzie of UCLA.

The study was conducted in Convair's Advanced Space Programs department, directed by J. B. (Jack) Hurt. The NASA-JSC COR is Earle Crum of the Transportation Systems Office, under Hubert Davis, Manager.

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LIST OF ACRONYMS

ACS	Attitude Control System
COR	Contracting Officers Representative
COTV	Cargo Orbital Transfer Vehicle
CRES	Corrosion Resistant Steel
CTV	Cargo Transfer Vehicle
DOE	Department of Energy
DRD	Data Requirement Description
DRL	Data Requirements List
ECLSS	Environmental Control & Life Support System
EMR	Earth Material Requirements
ET	External Tank (Space Shuttle)
EVA	Extra Vehicular Activity
GDC	General Dynamics Convair
GEO	Geostationary (or Geosynchronous) Earth Orbit
HLLV	Heavy Lift Launch Vehicle
ISP	Specific Impulse
JSC	Johnson Space Center (NASA)
L_2	Lagrangian Libration Point Behind Moon
L_4 or L_5	Lagrangian Libration Point which Forms an Equalateral Triangle with Earth and Moon
LDR	Lunar Derived Rocket
LEO	Low Earth Orbit
LeRC	Lewis Research Center (NASA)
LLO	Low Lunar Orbit
LMR	Lunar Material Requirements
LPI	Lunar and Planetary Institute
LRU	Lunar Resource Utilization
LS	Life Support
LSS	Large Space Structure
LTV	Lunar Transfer Vehicle
MBE	Molecular Beam Epitaxy
MDRE	Mass Driver Reaction Engine
MIT	Massachusetts Institute of Technology
MPTS	Microwave Power Transmission System
MSFC	Marshall Spaceflight Center (NASA)
NASA	National Aeronautics and Space Administration
OTV	Orbital Transfer Vehicle
PLTV	Personnel Lunar Transfer Vehicle
PLV	Personnal Launch Vehicle

LIST OF ACRONYMS (cont'd)

POTV	Personnel Orbital Transfer Vehicle
RDT&E	Research, Development, Test and Evaluation
RMS	Remote Manipulator System (Space Shuttle)
RPL	Rotary Pellet Launcher
SCB	Space Construction Base
SDV	Shuttle Derived Vehicle
SEP	Solar Electric Propulsion
SMF	Space Manufacturing Facility
SPS	Solar Power Satellite or Satellite Power Station
SRB	Solid Rocket Booster (Space Shuttle)
SSME	Space Shuttle Main Engine
SSTS	Space Shuttle Transportation System
TFU	Theoretical First Unit
TT	Terminal Tug
UCLA	University of California at Los Angeles
WBS	Work Breakdown Structure

ENGLISH CONVERSIONS

1 kilogram (kg) = 2.205 lb
1 meter (m) = 39.372 inches = 3.281 ft
1 ton = 1000 kg = 2205 lb
1 square meter = 10.76 square feet
1 micrometer (μm) = 10^{-6} meters = 10^{-3} millimeters (mm) = 3.94×10^{-5} inches
°C = (°F-32) 5/9 = °K-273°
1 kilometer (km) = 0.6214 mile
1 square kilometer = 0.3861 square mile
1 gravitational constant (g) = 9.806 m/sec^2 = 32.2 ft/sec^2
1 Newton = 0.2248 lb_F

$$\text{Specific Impulse } (I_{sp}) = \frac{\text{Newton-second}}{\text{kg}} \left(\frac{\text{N-s}}{\text{kg}}\right) = 9.806 \text{ (ISP in seconds)}$$

Pressure = N/cm^2 = 0.689 lb_F/in^2
1 Pa = 1N/m^2

APPENDIX A

Task 5.2 supplementary data, supporting development of lunar material requirements in Volume II Section 3 of Final Report.

Appendix A contains two sections.

A.1 ESTIMATE OF SPS COMPONENT LEVEL EARTH MATERIAL REQUIREMENTS

Reconciliation of summary material requirements shown in Table A-1, with the component mass breakdown table for the JSC preliminary baseline shown in Table A-2. Ten components have been evaluated.

A.1.1 PHOTOVOLTAIC ARRAY

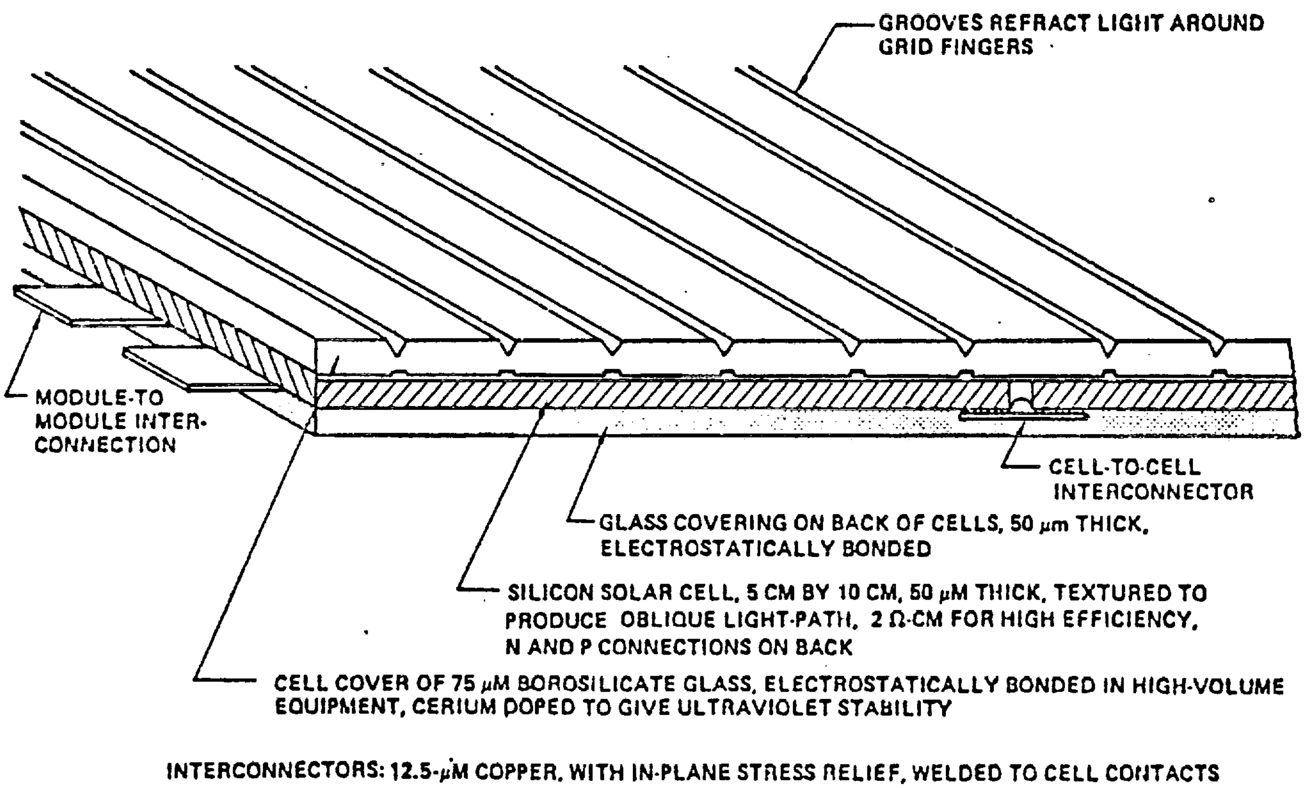


Figure A-1. Low cost annealable blanket structure.

Data Source: Reference 1, Page 124.

From Table A-2.

$$\text{Glass Cell Cover Mass} = 28,313\left(\frac{75}{125}\right) = 16,988\ \text{T} \quad \text{(a)}$$

$$\text{Glass Substrate Mass} = 28,313\left(\frac{50}{125}\right) = 11,325\ \text{T} \quad \text{(c)}$$

Table A-1. 10 GW satellite system materials requirements *

Element	Material	Mass (T)
Energy Collection System		
Structure	Gr-Ep	6,177
	Aluminum	619
Solar Cells	Glass	36,097
	Silicon	14,775
	Copper	1,456
	S. Steel	327
Distribution	Aluminum	2,778
	Copper	116
	S. Steel	67
	Silver	28
Misc. Components	Various	3,209
Power Transmission System		
Structure	Gr-Ep	894
Controls	Aluminum	1,850
	Copper	1,761
	S. Steel	3,449
	Mercury (1)	266
Instrumentation/Buss	Aluminum	1,077
	Copper	1,686
	S. Steel	1,686
Antenna Subarrays	Gr-Ep	5,462
	Copper	5,755
	S. Steel	2,218
	Tungsten	1,132
Misc. Components	Various	4,665
	TOTAL	97,550

(1) Closed System Heat Pipe Application Only

NOTE: Undefined component mass 7,874 T, or 8% of total mass of SPS

* Data Source: A recommended preliminary baseline concept, SPS concept evaluation program, NASA JSC January 25, 1978

Table A-2. Satellite mass summary.

	QUANTITY	MASS, KG
SOLAR ARRAY	1	51,779,200
PRIMARY STRUCTURE		5,385,000
ROTARY JOINT (MECHANICAL)	2	66,800
FLIGHT CONTROL SYSTEM	4	179,000
THRUSTERS	160	(46,800)
MECHANICAL SYSTEMS	4 SETS	(32,200)
CONDUCTORS	4 SETS	(8,000)
POWER PROCESSORS	12	(88,000)
AVIONICS (INSTR. COMM. COMPUTERS)	4 SETS	4,000
ENERGY CONVERSION SYSTEM		43,750,000
SOLAR CELLS	20×10^9	(11,670,850)
SUBSTRATE AND COVERS	78×10^6 PANELS	(28,313,230)
INTERCONNECTS	78×10^6 (1/PANEL)	(1,150,160)
JOINT/SUPPORT TAPES	256 SETS (1/BAY)	(300,360)
CATENARY	256 SETS (1/BAY)	(258,290)
TOLERANCE & OTHER		(2,057,110)
POWER DISTRIBUTION		2,398,400
POWER BUSSES	3	(2,030,000)
CELL STRING FEEDERS	163,000	(38,800)
DISCONNECTS AND SWITCHGEAR	208	(156,000)
ENERGY STORAGE	—	(20,200)
ROTARY JOINT (ELECTRICAL)	2	(39,200)
SUPPORT STRUCTURE	2	(114,200)
MICROWAVE POWER TRANSMISSION SYSTEM	2	25,223,200
ANTENNA STRUCTURE	2	500,000
PRIMARY STRUCTURE	2	(105,000)
SECONDARY STRUCTURE	122 SUBASS'YS	(395,000)
ANTENNA CONTROL SYSTEM	24 UNITS	11,000
MPTS POWER DISTRIBUTION		5,866,200
POWER BUSSES	3	(760,600)
SWITCHGEAR AND DISCONNECTS	912	(273,600)
DC-DC CONVERTERS	456	(2,482,000)
THERMAL CONTROL	456	(1,472,000)
ENERGY STORAGE	—	(598,600)
SUPPORT STRUCTURE	—	(279,400)
SUBARRAYS (6932 x 2)	13,864	18,846,000
WAVEGUIDES	1,663,680	(4,314,000)
KLYSTRONS (97056 x 2)	194,112	(9,316,000)
THERMAL CONTROL	194,112 SETS	(4,174,000)
CONTROL CIRCUITS AND CABLES	194,112 SETS	(1,042,000)

TOTAL SATELLITE MASS (10 GW OUTPUT) - 77,002,400 KG
MARGIN (26.6% BASED ON UNCERTAINTY ANALYSIS) - 20,482,638 KG

PREDICTED ACTUAL MASS - 97,485,038 KG

Table A-3. Photovoltaic array materials.

Material	Total Mass (Table A-1) (T)	Actual Mass Requirement (Table A-2) (T)	Mass Margin (T)	Mass Margin (%)
(a) Glass Cover	21,658	16,988	4,670	27.5
(b) Silicon Cell	14,775	11,671	3,104	26.7
(c) Glass Substrate	14,439	11,325	3,114	27.5
(n) Copper Interconnect	1,456	1,150	306	26.7

A.1.2 THERMAL CONTROL RADIATORS (KLYSTRON)

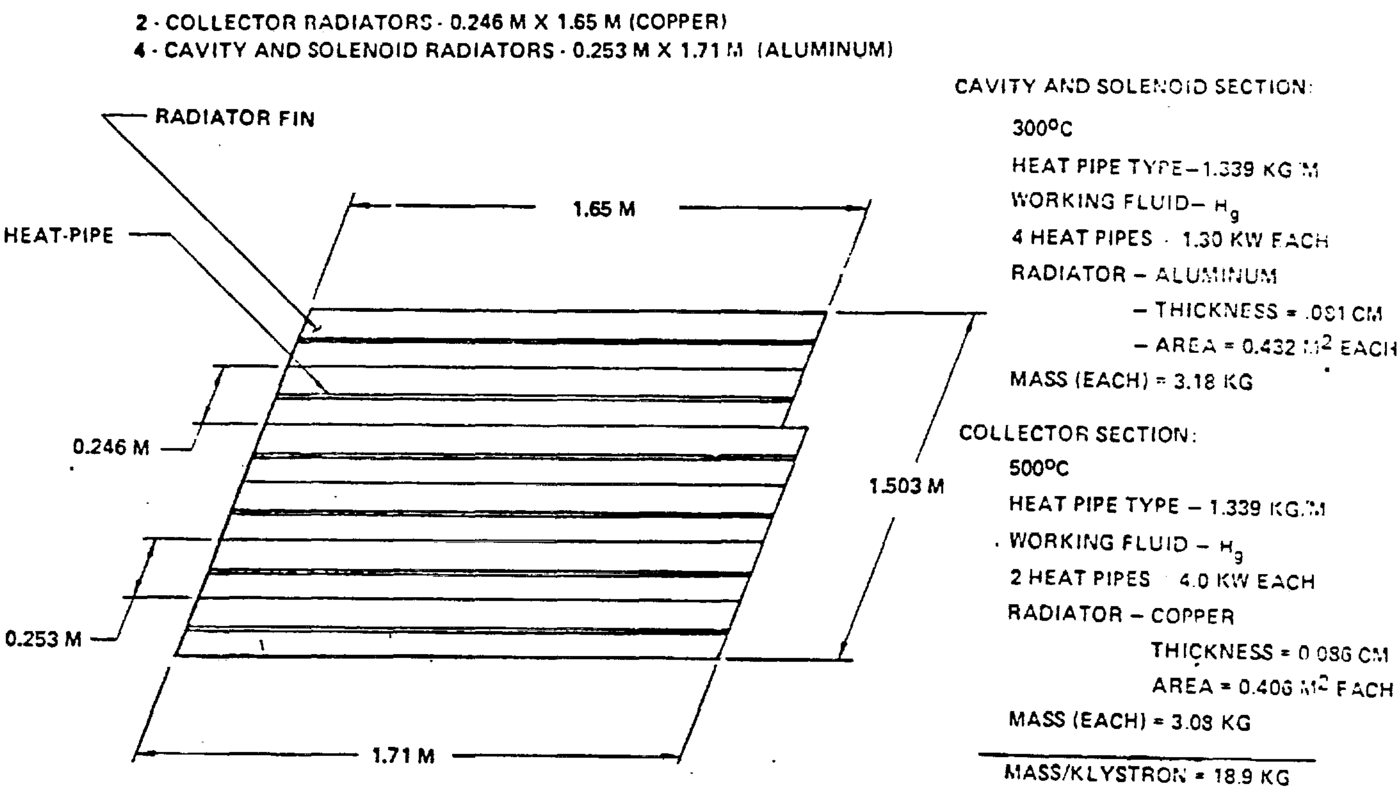


Figure A-2. Typ. Klystron module thermal radiator.

Data Source: Reference 2, Page 177 and 178

$$\rho_{\text{Aluminum}} = 0.002823 \text{ kg/cm}^3$$

$$\rho_{\text{Copper}} = 0.008967 \text{ kg/cm}^3$$

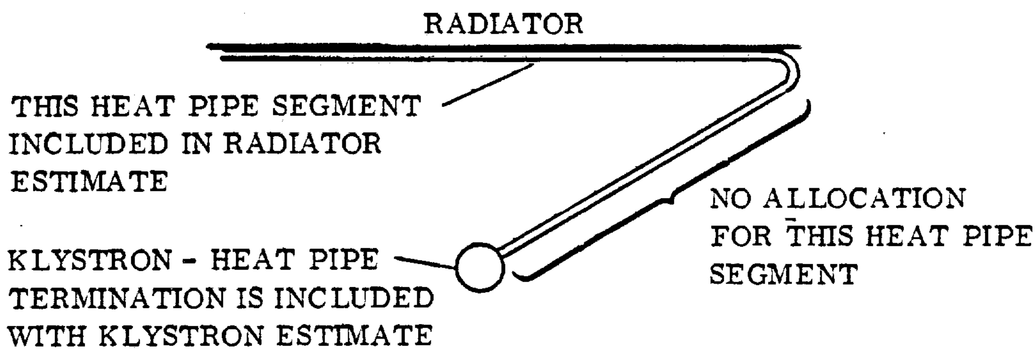
$\text{Mass}_{AL} = 0.002823\ (0.081)\ (0.432)\ (100)^2 4 = 3.95$ kg/Klystron

$\text{Mass}_{AL} = 9.88\ (194.112) = 767$ T

$\text{Mass}_{CU} = 0.008967\ (0.086)\ (0.406)\ (100)^2\ 2 = 6.26$ kg/Klystron

$\text{Mass}_{CU} = 6.26\ (194,112) = 1,215$ T

Heat Pipe Length



$\ell^{H.P.} = \ell_{\text{Radiators}}$

$\ell^{H.P.} = 4(1.71) + 2\ (1.65)$

$\ell^{H.P.} = 10.14$ m/Klystrom

$\text{Mass}_{H.P.} = 1.339\ (10.14)\ (194.112) = 2,636$ T

Total thermal control mass = 767 + 1,215 + 2,636 = 4,618 T
This does not agree with the Table A-2 Klystron thermal control total of 4,174 T, due to an apparent error in the copper radiator weight estimate of Figure A-2.

If all the heat pipe fluid is included in the thermal control estimate the following masses result:

Aluminum Sheet	767 T	(i)
Copper Sheet	1,215 T	(ℓ)
CRES Tubing	1,926 T	(g) adjusted to agree with Table A-2 (should be ~ 2,370 T)
Mercury	266 T	(Other metals)
	4,174 T	

A.1.3 THERMAL CONTROL RADIATORS (DC-DC CONVERTER)

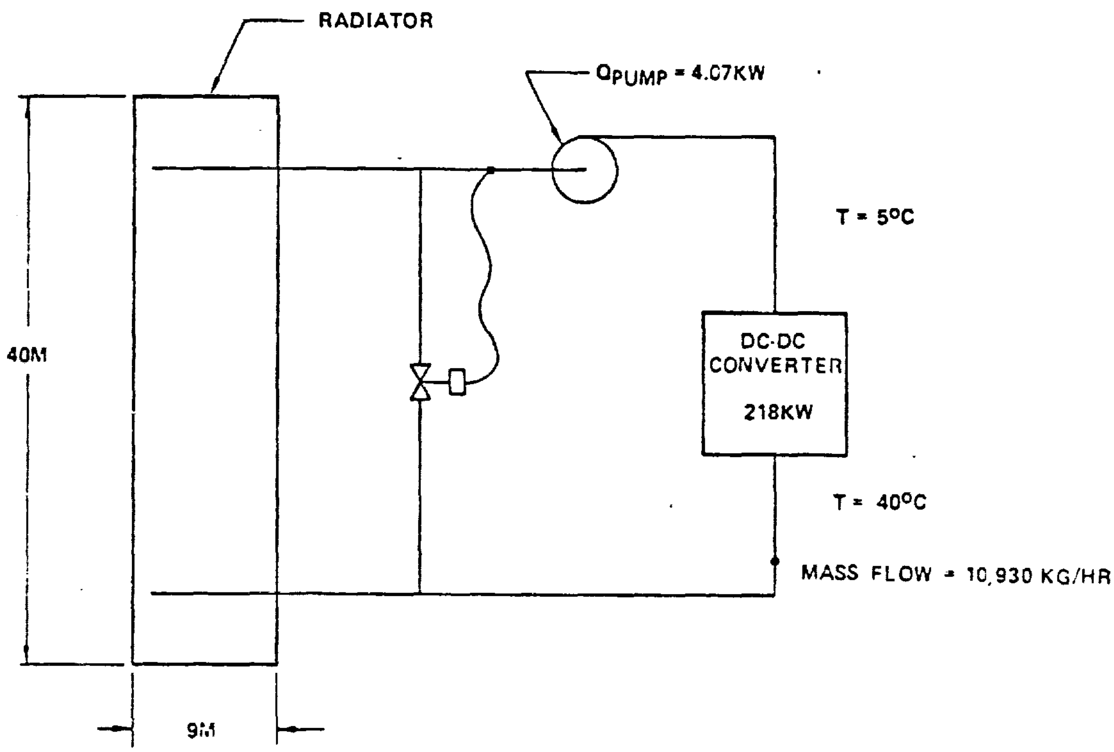


Figure A-3. Active thermal control for DC-DC converter.

Data Source: Reference 2, Page 170.

Assume an aluminum radiator and aluminum tubing

0.122 cm sheet both sides

$$\text{Mass}_{\text{Rad}} = 0.002823\ (2)\ (0.122)\ (360)\ (100)^2 = 2480 \text{ kg/DC-DC converter}$$

$$\text{Mass}_{\text{Rad}} = 2{,}480\ (456) = 1{,}131 \text{ T}$$

Allow 5% for feed tubing = 57 T

Aluminum Sheet Radiator	1,131	1188 T (i)
Aluminum Tubing	57	
Various (Transport Fluid, Pumps, Valves, Etc.)	284	
	1,472 T	

A.1.4 KLYSTRONS

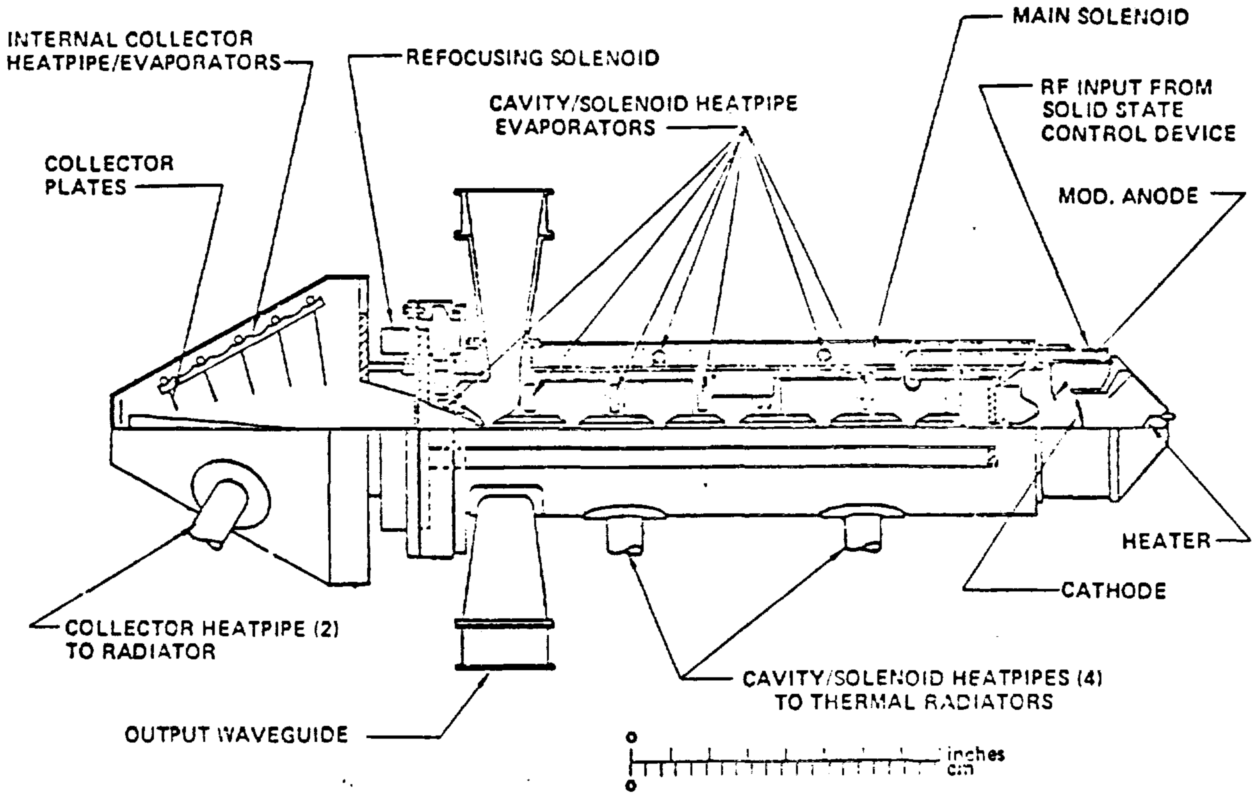


Figure A-4. 70 kW Klystron

Table A-4. Klystron mass estimate.

ITEM	MATERIAL	PRINCIPAL DIMENSIONS (CM)	MASS (kg)	
SOLENOID	COPPER	OD = 11.4, ID = 7.6, L = 41.9		16.4
WIRE		(75% OF SOLENOID VOLUME)	16.0	
INSULATION	ALUMINA	(5% OF SOLENOID VOLUME)	0.4	
CAVITIES ASSEMBLY	COPPER	D = 7.6, L = 41.9, Z = 0.95		7.4
POLE PIECES (2)	IRON	D = 15.2, d = 2.5, Z = 1.02		2.8
SOLENOID HOUSING	STEEL	D = 12.7, L = 41.9, Z = 0.32		4.2
COLLECTOR PLATES				4.6
PLATE 1 (LWR)	TUNGSTEN	D = 15.2, d = 5.1, H = 0.8, t = 0.53	1.7	
PLATE 2	TUNGSTEN	D = 15.2, d = 5.1, H = 1.0, t = 0.30	1.0	
PLATE 3	TUNGSTEN	D = 15.2, d = 5.1, H = 1.3, t = 0.15	0.5	
PLATE 4	TUNGSTEN	D = 15.2, d = 5.1, H = 1.5, t = 0.08	0.2	
PLATE 5	TUNGSTEN	D = 15.2, d = 5.1, H = 1.8, t = 0.08	0.2	
PLATE 6 (UPP)	TUNGSTEN	D = 15.2, d = 5.1, H = 2.0, t = 0.23	1.0	
PROBE	TUNGSTEN	D = 2.5, d = 0, H = 3.8, t = 0.15	~	
COLLECTOR PLATE ISOLATOR	ALUMINA	OD = 18.3, ID = 15.2, H = 15.5, t = 1.22		2.9
COLLECTOR SECTION COVER	STEEL	D = 20.3, H = 19.1, t = 0.13		2.0
OTHER COMPONENTS: REFOCUSING COIL, HEAT PIPES, HI-VOLTAGE CERAMIC SEALS, MODULATING ANODE CONNECTOR, CATHODE CONNECTOR, HEATER, OUTPUT WAVEGUIDES (2), VAC. ION CONNECTOR, CAVITY TUNING PROVISIONS, INTERNAL CABLING, ETC., AND ASSEMBLY AND INSTALLATION HARDWARE.				7.7
				(48 kg)

Data Source: Reference 2, Page 172

Copper Wire	16.0 (194,112) = 3,106	4,542 T (e) (j)
Copper Parts	7.4 (194,112) = 1,436	
Iron	2.8 (194,112) = 544	1,747 T (k) (m)
CRES Parts	6.2 (194,112) = 1,203	
Tungsten (W)	4.6 (194,112) = 893 T	Other Metals
Aluminia	3.3 (194,112) = 640	2,134 T Various
Other	7.7 (194,112) = 1,494	
	9,316 T	(Table A-2)

It appears that the section of heat pipe from the Klystron to the radiator has been inadvertently omitted from the mass estimate. This CRES tubing length is (from Section A.1.2)

$$\ell^{HP} = 0.5\ (1.15)\ \ell^{Radiators}$$

$$\ell^{HP} = 0.58\ (10.14) = 5.85 \quad @\ 1.339\ \text{kg/m}$$

$$\text{Mass}_{HP} = 1.339\ (5.85)\ (194,112) = 1,522\ \text{T}$$

This has been included as CRES margin.

A.1.5 DC-DC CONVERTER

From Table A-2, Mass = 2,482 T
Converter components have been estimated as indicated in Table A-5.

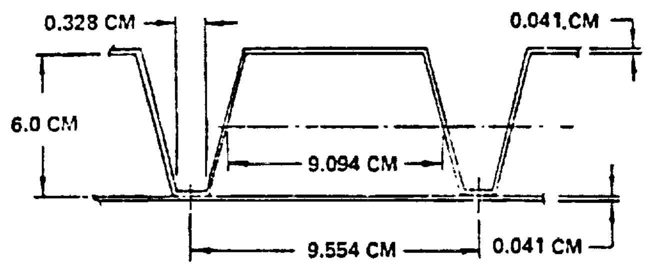
Table A-5. DC-DC converter material requirements.

DC-DC Converter Component	Percent of Total Mass	Material Requirements	Code or Rank	Material Mass (T) Required
Transformer (SENDUST)	40%	5 Percent Alum	(i)	50
		10 Percent Silicon	Various	99
		85 Percent Iron	(k)	844
Transformer Winding	35%	Copper Wire	(e)	868
Electronics Controls & Packaging	25%	Various	Various	621

Silicon has been listed as various, but sufficient silicon mass margin is available (3104 T) to encompass this requirement.

A.1.6 MPTS WAVEGUIDES

TRAPEZOIDAL WAVEGUIDE



STRUCTURAL MAT'L: GR-EP -8PLY
CONDUCTING MAT'L: ALUMINUM (T = 6.67 μM)

PER SUBARRAY	TRAPEZOIDAL
MASS OF GR-EP :	234.5 KG
MASS OF ALUM.:	9.3 KG
MASS OF WAVEGUIDE	243.8 KG
UNASSEMBLED PACKING MASS DENSITY:	1568.4 KG/M^3
VOLUME OF ASSEMBLED WAVEGUIDE:	6.97 M^3
ASSEMBLED PACKING MASS DENSITY:	35.0 KG/M^3
MASS/ANTENNA:	1690.0 MT

Figure A-5. Trapezoidal waveguide.

Data Source: Reference 2, Page 175

From Table A-2

$$\text{Graphite Composite Mass} = 4,314 \ \frac{(234.5)}{(243.8)} = 4,149 \ \text{T (f)}$$

$$\text{Aluminum Coating Mass} = 4,314 \ \frac{(9.3)}{(243.8)} = 165 \ \text{T (i)}$$

A.1.7 ROTARY JOINT (Mechanical)

Data Source: Reference 3, Page 23

$$\text{Mass}_{\text{Graphite}} = 1.1\ (8.79 + 18.38)\ 2 = 60 \ \text{T}$$

$$\text{Mass}_{\text{Alum}} = 1.1\ (0.3 + 0.55 + 0.55 + 0.3 + 0.1)\ 2 = 4 \ \text{T}$$

$$\text{Mass}_{\text{Various}} = \text{Remainder} = 3 \ \text{T}$$

A.1.8 PRIMARY STRUCTURE (Graphite Composite)

Date Source: Reference 3, Page 87

Graphite Thermoplastic	0.91 (5385) = 4900 T (d)
Aluminum Fittings	0.09 (5385) = 485 T
Steel Fittings	0.01 (5385) = 0

A.1.9 ROTARY JOINT (Electrical)

Data Source: Reference 3, Pages 45 and 46

$$\text{Mass}_{\text{Silver}} = 1.05\ (10.74)\ 2 = 23\ \text{T}$$

$$\text{Mass}_{\text{Graphite}} = 1.05\ (0.51)\ 2 = 1\ \text{T}$$

$$\text{Mass}_{\text{Various}} = \text{Remainder} = 15\ \text{T}$$

A.1.10 CONTROL CIRCUITRY AND CABLES

Assume 67% copper wire = 0.67 (1,042) = 698 T (e)

Remaining 33% is insulation,
end fittings, and various = 0.33 (1042) = 344 T

The material requirements matrix shown in Table A-6 was generated using satellite mass summary data and material requirements information developed in the preceding ten subsections. Some discretion was employed in completing this matrix to provide reasonable agreement with the NASA-JSC documented totals and the 26.7 percent material margin. Masses of discrete components are identified in Table A-6 by use of alphabetic superscripts. These components plus smaller amounts of similar components and material margins were collected and ranked into the fifteen discrete material products listed in Table A-7.

A.2 DEVELOPMENT OF EQUIVALENT LUNAR MATERIAL REQUIREMENTS

Each earth material application in Table A-7 was investigated to determine reasonable alternative methods of providing the same function with lunar derived materials. This investigation included development of equivalent material

Table A-6. SPS earth material requirements mass breakdown.

SPS Components	Fused Silica Glass	Silicon Solar Cells	Graphite Comp	Copper	CRES	Alum	Other Metals	Various	(Ref Table A-2) Total
SOLAR ARRAY									
Primary Structure			(d) 4,900			485			5,385
Rotary Joint (Mechanical)			(o) 60			4		3	67
Flight Control System									
Thrusters					47				
Mechanical Systems					32				
Conductors				(e) 8					179
Power Processors								88	
Avionics (Instr. Comm. Computers)								4	
Energy Conversion System									
Solar Cells		(b) 11,671							
Substrate and Covers	(a)(c) 28,313								
Interconnects				(n) 1,150					43,750
Joint/Support Tapes									
Catenary					258				
Tolerance & Other	(a)(c) 181				258			1,919	
Power Distribution									
Power Busses						(h) 2,030			
Cell String Feeders				(e) 39					
Disconnects and Switchgear								156	2,398
Energy Storage								20	
Rotary Joint (Electrical)			1				(Ag) 23	15	
Support Structure			(o) 114						
MICROWAVE POWER TRANSMISSION SYSTEM									
Antenna Structure									
Primary Structure			(o) 105						500
Secondary Structure			(o) 395						
Antenna Control System								11	11
MPTS Power Distribution									
Power Busses						(h) 760			
Switchgear and Disconnects								274	
DC-DC Converters				(e) 868	(k)(Fe) 844	(i) 50		720	5,866
Thermal Control						(i) 1,188		284	
Energy Storage								599	
Support Structure			(o) 279						
Subarrays									
Waveguides			(f) 4,149			(i) 165			
Klystrons				(e)(j) 4,542	(k) (m) 1,747		(W) 893	2,134	
Thermal Control				(l) 1,215	(g) 1,926	(i) 767	(Hg) 266		18,846
Control Circuits and Cables				(e) 698				344	
Margin (~26.7%)	7,603	3,104	2,530	2,254	(1) 2,635	(2) 875	(3) 244	(4) 1,303	20,548
TOTAL (Ref Table 3-9)	36,097	14,775	12,533	10,774	7,747	6,324	1,426	7,874	97,550

(1) 51.5%, (2) 16.1%, (3) 20.6%, (4) 19.8%

Table A-7. SPS earth material mass ranking and application.

RANK	MASS (T)	PERCENT OF TOTAL SPS MASS	MATERIAL	APPLICATION	PERFORMANCE REQUIREMENTS
(a)	21,658	22.2	Borosilicate Glass	Photovoltaic Cell Covers	Structural Support, UV Stability, Emittance, Radiation Protection
(b)	14,775	15.1	Silicon	Solar Cells	Energy Conversion Efficiency, Radiation & Thermal Degradation
(c)	14,439	14.8	Fused Silica Glass	Photovoltaic Cell Substrate	Structural Support, Thermal Control
(d)	6,208	6.4	Graphite Composite	Primary Structure for Solar Array	Structural Stiffness, Buckling Strength, Thermal Stability
(e)	5,980	6.1	Copper Wire	Klystron & DC-DC Converter Coils, Power Cables	Electrical Conductivity, Resistance, Field Strength
(f)	5,257	5.4	Graphite Composite	MPTS Waveguides	Microwave Transmission, Dimensional and Thermal Stability
(g)	3,892	4.0	CRES Tubing	Heat Pipe for Klystron Radiators	Contain Mercury Transport Fluid, High Temperature
(h)	3,535	3.6	Aluminum Sheet	Power Transmission Busses, Array & MPTS	Electrical Conductivity
(i)	2,749	2.8	Aluminum Sheet	Klystron & DC-DC Conv. Radiators	Thermal Conductivity, Surface Emissivity
(j)	1,820	1.9	Copper (Mach Part)	Klystron Solenoid Cavity	Electrical Conductivity, Non-Magnetic, Mercury Compatibility
(k)	1,758	1.8	Iron	Klystron Solenoid & Transformer for DC-DC Converter	Magnetic Properties
(l)	1,539	1.6	Copper Sheet	Klystron Collector Radiators	Thermal Conductivity, Surface Emissivity, High Temperature
(m)	1,524	1.6	CRES (Mach Part)	Klystron Housing	Non-Magnetic, High Temperature
(n)	1,456	1.5	Vacuum Deposited Copper	Solar Cell Inter-Connects	Electrical Conductivity, High Temperature for Array Annealing
(o)	1,210	1.2	Graphite Composite	MPTS Antenna & Other Structure	Structural Stiffness, Thermal Stability, Electrical Insulator

87,800 T 90.0% of Total 97,550 T Earth Baseline SPS requirements. The following procedure was employed to obtain this information:

1) Determine what percentage (by weight) of the earth baseline material requirements can be directly satisfied with lunar resources.

2) Postulate substitute materials which will allow a higher percentage of lunar resource utilization and/or improved in-space production capability. Determine how much more of these substitute materials are required to meet the various performance requirements of the earth baseline materials, such as:

- Structural stiffness (graphite composite)
- Electrical conductivity (power busses, klystrons)
- Radiation protection (glass covers)
- Energy conversion (solar cells)
- Heat dissipation (radiators)
- Dimensional stability (MPTS waveguides)

The substitute lunar derived material mass requirements are defined by the ratio of important performance parameters:

$$\frac{\left[\begin{array}{l}\text{Lunar Material}\\ \text{Performance Parameters}\end{array}\right]}{\left[\begin{array}{l}\text{Earth Material}\\ \text{Performance Parameters}\end{array}\right]} = \left[\begin{array}{l}\text{Lunar material}\\ \text{Performance}\\ \text{Factor}\end{array}\right]$$

a) BOROSILICATE GLASS PHOTOVOLTAIC CELL COVERS

A silicon solar cell must be provided with a cover to increase front-surface emittance from approximately 0.25 to 0.85 and to protect the cell from low-energy proton irradiation. Cerium-doped borosilicate glass was selected as the Earth baseline cover material because its Earth production cost is only a fraction of the best alternate, 7940 fused silica, it matches the coefficient of thermal expansion of silicon, and yet resists darkening by ultraviolet light. Borosilicate or fused silica glass can be electrostatically bonded to silicon to form a strong and permanent adhesiveless joint.

Step 1 Earth Material Composition Data Source: Reference 1, Page 123

Cerium-doped borosilicate glass consists of the following ingredients:

Material	Constituent Percent by Weight	Available in Lunar Resources?
* Boron	8.69%	Yes, but only in few parts per million
* Lithium	0.56%	Yes, but only in few parts per million
Potassium Oxide	0.5%	Yes, but only in hundredths to low tenths of 1%
Alumina	1.1%	Yes
Silica (SiO_2)	70.0%	Yes
Oxygen	Remainder	Yes

* Metallic component of oxides present in the glass

While the majority of borosilicate glass ingredients are available in lunar resources, approximately 10% of this most massive SPS material must still be obtained from earth.

Step 2 Lunar Resource Substitutions

As indicated above, fused silica is the best alternative to borosilicate glass, and has the advantage of being available from lunar materials. Fused silica is very resistant to darkening by ultraviolet radiation.

Step 3 Percent of Substitute Materials Required

Since the density and required thickness of fused silica and borosilicate glass are equal, the lunar derived cover material may be directly substituted for the Earth baseline without any mass increase.

Recommendation: Use fused silica cover material with a mass equivalent to the Earth baseline cerium-doped borosilicate glass. All material (SiO_2) is obtained from lunar resources.

b) SILICON SOLAR CELLS

The Earth baseline SPS assumes the use of 50 μm thick silicon solar cells. Similar cells recently made by Solarex had an air-mass-zero efficiency of 12.5 percent without a back-surface field or anti-reflection treatment. The Earth baseline cells employ sun-facing surface texturing which improves photon collection efficiency, when compared with thicker cells, by lengthening the light path in silicon for infrared photons, and also improves radiation resistance. Each solar cell measures 5 × 10 cm and is produced as a wafer by slicing a single crystal of silicon.

Data Source: Reference 1, Page 123

Step 1 Earth Material Composition

Silicon solar cells are produced from very high purity silicon with minute quantities (~10 ppb) of Group III and Group V (n and p) elements used as dopants.

Step 2 Lunar Resource Substitutions

Since the silicon required to produce solar cells is abundant in lunar materials, there is no need to define substitute lunar materials. It is important, however, to evaluate alternative silicon solar cell manufacturing techniques to evaluate the effects of in-space processing applicability and photovoltaic cell efficiency on overall silicon mass requirements. Three techniques have been proposed for large scale production of silicon solar cells.

1) Sliced silicon crystals (earth baseline) - large diameter single silicon crystals of approximately 15 cm diameter are cut into wafers, polished, sorted and tested. These are labor intensive operations which produce a very high percentage of waste (which is recyclable).

2) Silicon crystal ribbons - A single crystal ribbon (50 μm thick × 5 cm wide) is continuously grown and cut into 10 cm lengths. Polishing, sorting and testing operations are still required, but are somewhat less labor intensive since material waste and recycling is substantially reduced. This process is experimental, but should eventually provide electrical conversion efficiencies equivalent to the baseline.

3) Amorphous silicon sheet - A sheet of silicon is formed by chemical vapor deposition using a fully automated process (non-labor intensive) ideally suited for in-space operations. Unfortunately, the maximum energy conversion efficiency that has currently been achieved with this technique is ~50% of the baseline. While improvements are expected, it is doubtful that single crystal efficiencies can be attained with amorphous sheet. If this production technique were adopted the SPS photovoltaic array area would have to be increased substantially. This increase would impact material requirements for glass covers, substrate, and the array support structure as well as silicon. It also constitutes a redesign of the SPS which is not within the scope of this study.

Step 3 Percent of Substitute Lunar Materials Required

Assuming that either manufacturing methods 1) or 2) above will be used, the quantity of lunar silicon required is identical to the quantity obtained terrestrially for the Earth baseline.

Recommendation: Use identical silicon solar cells with all material obtained from lunar resources.

c) FUSED SILICA GLASS PHOTOVOLTAIC CELL SUBSTRATE

Glass was chosen as the Earth baseline substrate to enable annealing of radiation damage by heating. With all glass-to-silicon bonds made by the electro-static process there are no elements in the blanket which cannot withstand the 500°C (931°F) annealing temperature which at present seems to be required.

Data Source: Reference 1, Page 125

Step 1 Earth Material Composition

Fused silica glass is produced from 99.9+% pure silicon dioxide (SiO_2).

Step 2 Lunar Resource Substitutions

Since the SiO_2 required to produce fused silica glass is abundant in lunar materials, there is no need to define substitute lunar materials.

Step 3 Percent of Substitute Lunar Materials Required

The quantity of lunar fused silica required is identical to the quantity obtained terrestrially for the Earth baseline.

Recommendation: Use identical fused silica glass substrate with all material obtained from lunar resources.

d) PRIMARY SOLAR ARRAY GRAPHITE COMPOSITE STRUCTURE

The SPS structural design proposed by the Boeing Company, from which JSC's baseline was obtained, assumes a space erectable structure of graphite epoxy with aluminum end fittings. Work has been conducted for JSC by General Dynamics Convair on in-space fabricated composite structures made of graphite and E-glass fiber with polysulfone thermoplastic resin. Due to the applicability of this material for automatic in-space fabrication of very large structures, and the degree of attention this concept is receiving, we employed it as the assumed SPS earth baseline material.

The following ground rules were followed for evaluating lunar substitutes for graphite composite material:

1. The baseline SPS array structure selected for construction with earth material was a graphite/glass/thermoplastic composite per JSC Contract No. NAS9-15310. This composite consists of a unidirectional graphite core, woven E-glass facings, and polysulfone resin. The designation for this composite is $120/705_3/120$.
2. It was assumed that the SPS structural configuration should not be optimized or significantly revised for lunar material substitution. To maintain equivalent structural stiffness, which is usually the predominant design

condition for large space structures, beam stiffness (modulus × area) and post stiffness (modulus × moment of inertia) must be held constant. Fortunately, for typical beam post configurations, area and moment of inertia are approximately proportional. Initial material replacement investigations assumed no redesign (diameter revisions) of structural members. Subsequent activities investigated redesign of individual structural members to more efficiently utilize the substitute lunar materials while retaining overall array geometry (node-to-node) and structural stiffness.

3. Candidate lunar construction materials include silica glass, glass fiber composite with thermoplactic resins (earth), glass fiber composite in a metal matrix (all lunar materials), and metal structure.

Step 1 Earth Material Composition

The $120/705_3/120$ composite material consists of the following ingredients:

Material	Ply Data	Constituents % Volume	Total % Volume	Density (g/cm^3)	Material Percent by Weight
120 Glass Fabric	2 plys @ 0.010 cm	E-glass 100%	17%	2.547	24.6% E-Glass
Graphite/ Glass Fabric	3 plys @ 0.019 cm	E-glass 5% Graphite 95%	40%	1.993	45.1% Graphite
Polysulfone Resin (P-1700)	—	Resin 100%	43%	1.246	30.3% Polysulfone

The only material in this graphite composite which is available in lunar resources is E-glass. The remaining 75.4 percent material mass must come from earth. This graphite composite has an elastic modulas of 143.1 GPa and a density of 1.766 g/cm^3.

Step 2 Lunar Resource Substitutions

Using structural stiffness as the primary performance criteria, and by assuming no redesign of structural members or overall arrangement i.e., "direct" material substitution:

$$(EA)_{\substack{\text{Graphite}\\ \text{Composite}}} = (EA)_{\substack{\text{Substitute}\\ \text{Material}}} \qquad \text{where A is proportional to } \frac{W}{\rho}$$

$$W_{\substack{\text{Substitute}\\ \text{Material}}} = \left(\frac{\rho_{\text{Substitute}}}{\rho_{\substack{\text{Graphite}\\ \text{Composite}}}}\right)\left(\frac{E_{\substack{\text{Graphite}\\ \text{Composite}}}}{E_{\text{Substitute}}}\right)\left(W_{\substack{\text{Graphite}\\ \text{Composite}}}\right)$$

W = Total mass of specific material used for manufacturing a SPS component

E = Modules of elasticity (GPa)

A = Structural member cross-sectional area

ρ = Material density (g/cm^3)

Glass Polysulfone Composite

A composite consisting of 60% by volume S-glass and 40% polysulfone thermoplastic resin was assumed. The S-glass is90 percent unidirectional.

Material	E-Elastic Modulas (GPa)	Percent Volume	Density (g/cm^3)	Material Percent by Weight
S-glass	85.5	60%	2.491	75%
Polysulfone Resin (P-1700)	2.5	40%	1.246	25%
Composite	47.3	—	1.993	—

$$W_{\substack{\text{Glass}\\ \text{Polysulfone}}} = \left(\frac{1.993}{1.766}\right)\left(\frac{143.1}{47.3}\right) W_{\substack{\text{Graphite}\\ \text{Composite}}}$$

$$= 3.41\ W_{\substack{\text{Graphite}\\ \text{Composite}}}$$

The equivalent lunar and earth material requirements for this substitute material are contained in Table A-8.

Pure S-Glass

Structural members are entirely manufactured from hi-strength glass, perhaps using the geodetic beam in-space construction technique under development at NASA-JSC, or a foamed glass with gaseous oxygen filled bubbles.

$$W_{\text{Glass}} = \left(\frac{2.491}{1.766}\right)\left(\frac{143.1}{85.5}\right) W_{\substack{\text{Graphite}\\\text{Composite}}}$$

$$= 2.36\ W_{\substack{\text{Graphite}\\\text{Composite}}}$$

All of this material could be obtained from lunar resources as shown in Table A-8

Pure Aluminum

Triangular structural members of aluminum could be manufactured in-space using the metal beam builder concept under development by NASA-MSFC.

$$W_{\text{Aluminum}} = \left(\frac{2.70}{1.766}\right)\left(\frac{143.1}{72.4}\right) W_{\substack{\text{Graphite}\\\text{Composite}}}$$

$$= 3.02\ W_{\substack{\text{Graphite}\\\text{Composite}}}$$

All of this material could be obtained from lunar resources as shown in Table A-8

Pure Titanium

Same as aluminum except:

$$W_{\text{Titanium}} = \left(\frac{4.54}{1.766}\right)\left(\frac{143.1}{106.9}\right) W_{\substack{\text{Graphite}\\\text{Composite}}}$$

$$= 3.44\ W_{\substack{\text{Graphite}\\\text{Composite}}}$$

Unidirectional S-Glass Aluminum Matrix Composite

Stock material would be manufactured by physical vapor deposition of aluminum onto a unidirectional S-glass roving. An aluminum type beam fabricator would be used for in-space construction. A 60 percent fiber content by volume has been assumed for this composite.

Material	E-Elastic Modulas (GPa)	Percent Volume	Density (g/cm^3)	Material Percent by Weight
S-Glass	85.5	60%	2.49	57.0%
Aluminum Matrix	72.4	40%	2.70	43.0%
Composite	80.3	——	2.57	——

$$W_{\substack{\text{Glass/}\\\text{Aluminum}}} = \left(\frac{2.57}{1.766}\right)\left(\frac{143.1}{80.3}\right) W_{\substack{\text{Graphite}\\\text{Composite}}}$$

$$= 2.60 \; W_{\substack{\text{Graphite}\\\text{Composite}}}$$

Both of these material requirements are satisfied by lunar resources as shown in Table A-8.

Unidirectional S-Glass Titanium Matrix Composite

Manufacture of this composite would be accomplished by the same technique previously suggested for the S-glass aluminum composite.

Material	E-Elastic Modulas (GPa)	Percent Volume	Density (g/cm^3)	Material Percent by Weight
S-Glass	85.5	60%	2.49	45.3%
Titanium Matrix	106.9	40%	4.54	54.7%
Composite	94.0	——	3.31	——

$$W_{\substack{\text{Glass/}\\\text{Titanium}}} = \left(\frac{3.31}{1.766}\right)\left(\frac{143.1}{94.0}\right) W_{\substack{\text{Graphite}\\\text{Composite}}}$$

$$= 2.85 \; W_{\substack{\text{Graphite}\\\text{Composite}}}$$

Both of these material requirements are satisfied by lunar resources as summarized in Table A-8..

Step 3 Percent of Substitute Materials Required

Results of this evaluation are contained in Table A-8. As indicated for the glass polysulfone composite, the weight of resin which must be imported from earth is almost equal (85%) the total baseline graphite composite requirement. It is very unlikely that any economic advantage for lunar material utilization can be realized unless earth constituents are reduced to a much smaller percentage of original baseline requirements. The other candidate substitute

Table A-8. MATERIAL COMPARISON

		Earth Baseline	Possible Lunar Material Substitutions					
Property	Material	Graphite Polysulfone $120/705_3/120$	90% Uni S-Glass 25% Resin	Pure S-Glass	Pure Aluminum	Pure Titanium	Unidirectional S-Glass Aluminum Matrix	Unidirectional S-Glass Titanium Matrix
E_{Axial}	(GPa)	143.1	47.3	85.5	72.4	106.9	80.3	94.0
ρ	(g/cm^3)	1.766	1.993	2.49	2.70	4.54	2.57	3.31
α_X	$(\mu m/m/°C)$	-0.380	+4.16	+2.88	+22.32	+9.54	+9.90	+5.90
Equivalent Mass Required								
From Earth		1.00	0.85	0	0	0	0	0
From Moon		0	2.56	2.36	3.02	3.44	2.60	2.85
Total		1.00	3.41	2.36	3.02	3.44	2.60	2.85
$\dfrac{\alpha_X \text{ Alt Mat'l}}{\alpha_X \text{ Graphite-Poly}}$		1	11	7.6	59	25	26	16

materials can all be completely obtained from lunar resources. The total mass requirements for these lunar substitutes, however, substantially exceed the original Earth baseline requirements. Also, except for glass, their coefficients of thermal expansion are considerably higher than the graphite polysulfone Earth baseline.

The most appropriate lunar resource substitute for graphite composite primary structure is glass. It has the lowest coefficient of thermal expansion of any lunar derived structural material, has a modulas of elasticity higher than that for aluminum, and has reasonable good strength characteristics. Its principal drawback is a tendency to shatter when impacted or penetrated. This unacceptable failure mode can be tolerated if the fracture length is sufficiently constrained by the size of elements and their redundancy in the structural member. Two glass construction concepts have been identified which satisfy this requirement:

1) The geodetic strut, shown in Figure A-6, has a large number of short, redundant load carrying elements. Thin glass rods can be used for these elements since multiple fractures can be structurally tolerated as long as they do not propogate through the element nodes.

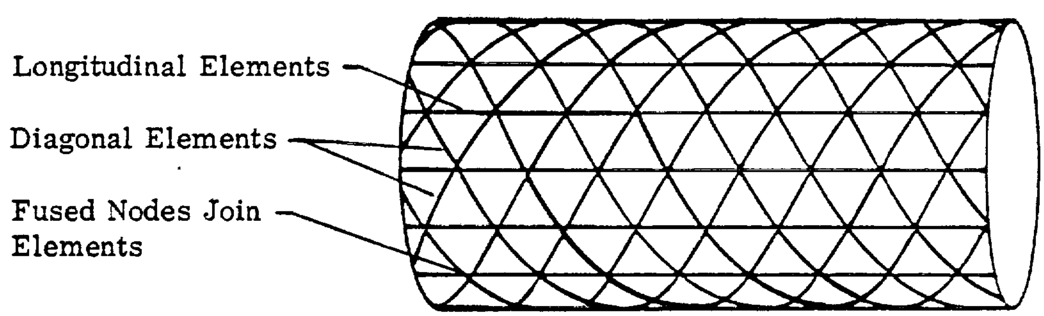


Figure A-6. Geodetic strut configuration.

2) Employ foamed glass, in which a very large number of tiny bubbles create a cellular structure which limits crack propogation to the locally damaged area. Structural members would probably be formed as relatively thin wall foamed glass tubes. A common material similar to foamed

glass is pumice, a low grade volcanic glass which has been frothed by water vapor. Oxygen is a potential foaming gas which can be obtained from lunar resources.

Based on the inherent attractiveness of foamed glass as a graphite thermoplastic replacement, a preliminary structural member resizing investigation was conducted. The assumptions used for this analysis were as follows:

1) S-glass with the structural properties shown in Table A-8.
2) Bubbles of uniform diameter created by low pressure gaseous oxygen were assumed to be distributed in rows and columns.
3) The effective load carrying material lies outside a cylinder with a diameter 0.707 times the bubble diameter.

Applying these assumptions, it was found that the foamed glass could consist of a maximum of 50 percent bubbles by volume, and had an effective $(AE)_{foamed} = 0.88\ (AE)_{solid}$. Applying this relationship to the critical SPS array structure design conditions of reference 5, page 83;

Critical buckling load = 12,824 N

Beam Length = 660 m

It was found that a larger 4.3 m diameter foamed glass tube could withstand both general and local instability criteria with a mass 1.9 times greater than the baseline 0.34 m diameter graphite composite beam elements.

Further mass improvements are expected if a larger foam factor is used. An improved foam factor can be obtained by assuming hexagonal bubble nesting, which is also physically more realistic. Since direct material replacement with glass results in a factor of 2.36 (from Table A-8) and preliminary conservative indications of member resizing for foamed glass result in a factor of 1.90, it may be safely assumed that a realistic factor lies between these two values.

Recommendation: Use foamed glass thin wall tubular structural members with an assumed mass approximately 2.0 times the Earth baseline graphite thermoplastic primary array structure. All material (glass and oxygen) is obtained from lunar resources. If bubbles are created using 14 kPa oxygen at 530°C (approximate glass softening temperature), the oxygen mass is less than 0.1 percent of the glass mass.

e) COPPER WIRE

The Earth baseline SPS employs copper wire in the klystron solenoids, DC-DC converter coils, and as power transmission and control cables in the microwave power transmission system. Electrical conductivity is the primary function inherent in all these applications. The highest temperature environment for these applications occurs in the klystron solenoid which has an operating temperature of 300°C (573°K).

Step 1 Earth Material Composition

Electrical conductors consist of copper alloy 1350, (fomerly EC grade), a high purity (99.99 + %) copper. All of this material must be obtained from Earth since lunar resources do not contain more than 10-30 parts per million of copper.

Step 2 Lunar Resource Substitutions

The best electrical conductor available from lunar material is aluminum. Although its conductivity is slightly lower than that for copper, aluminum's density is considerably less, which results in reduced aluminum mass required to transmit an equivalent amount of electrical energy. Aluminum's only potential disadvantage is its lower melting point; for certain high temperature applications it may be unsuitable. In this instance, however, sufficient margin exists between aluminum's melting temperature (933°K) and its maximum use temperature (573°K) to alleviate any concern.

Step 3 Percent of Substitute Materials Required

$$W_{\text{Aluminum}} = \left(\frac{\rho_{\text{Aluminum}}}{\rho_{\text{Copper}}}\right)\left(\frac{K_{\text{Elect. Cond. Copper}}}{K_{\text{Elect. Cond. Aluminum}}}\right) W_{\text{Copper}}$$

$$W_{\text{Aluminum}} = \left(\frac{2.70}{8.94}\right)\left(\frac{5977.3}{3766.8}\right) W_{\text{Copper}} = 0.479 \; W_{\text{Copper}}$$

Recommendation: Use pure aluminum conductor with a mass 0.479 times that of the Earth baseline copper wire. All aluminum material is obtained from lunar resources.

f) GRAPHITE COMPOSITE MPTS WAVEGUIDES

The Earth baseline MPTS waveguides are manufactured of graphite epoxy with an internal conductive surface of aluminum. A trapezoidal cross-section, shown in Figure A-7 was selected to provide high packing density for these earth manufactured/space assembled waveguides. Earth manufacture was selected due to the close dimensional tolerances required as shown in Figure A-7.

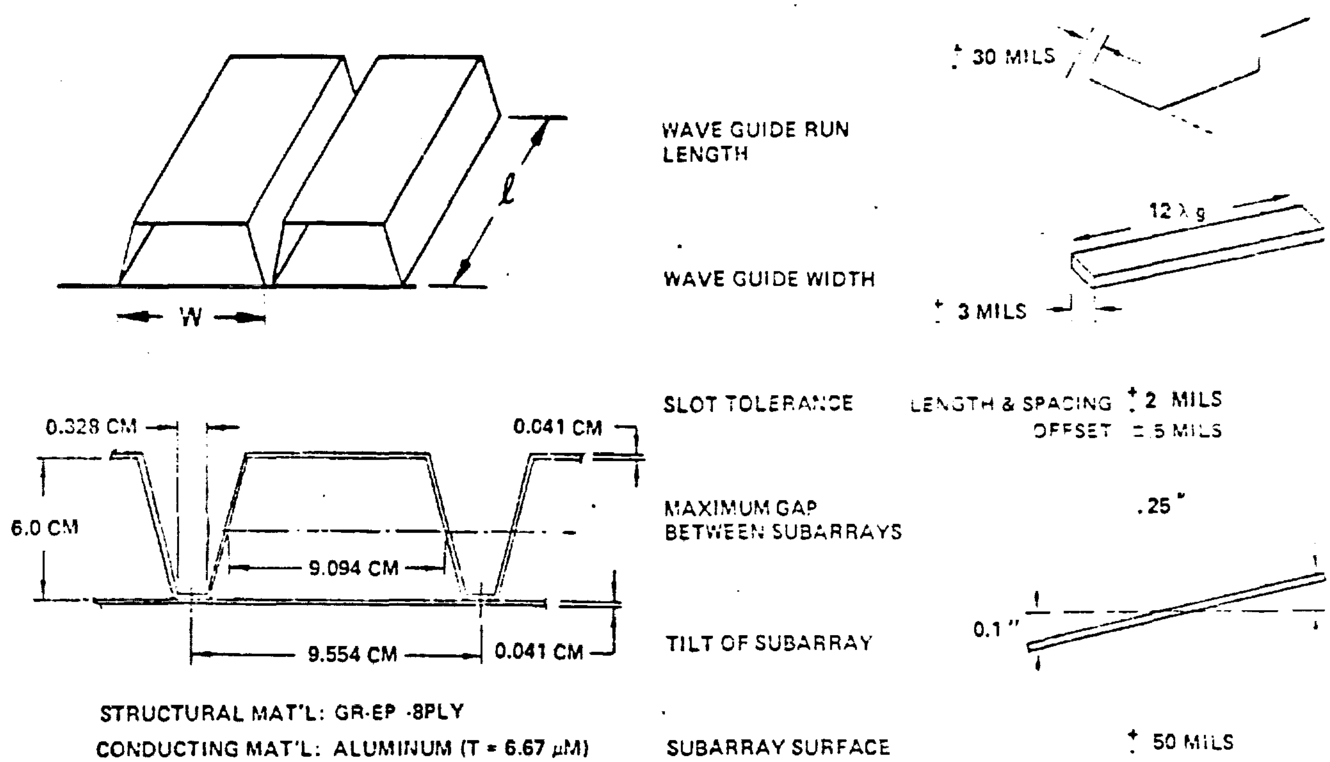


Figure A-7. Waveguide configuration and dimensional tolerances

Data Source: Reference 2, Page 165, 174 & 175

Step 1 Earth Material Composition

Since the waveguide's internal vacuum deposited aluminum coating is separately considered in category (i), the waveguide structure consists entirely of graphite fibers in an epoxy resin.

Material	Ply Data	Total % Volume	Density (g/cm^3)	Material Percent By Weight
VSB-32T Graphite	8 Plys @ 0.005 cm	63%	1.993	72.8% Graphite
Epoxy Resin	——	37%	1.273	27.2% Epoxy
Composite	——	——	1.727	——

These materials are available only from terrestrial resources; none can be obtained from the moon.

Step 2 Lunar Resource Substitutions

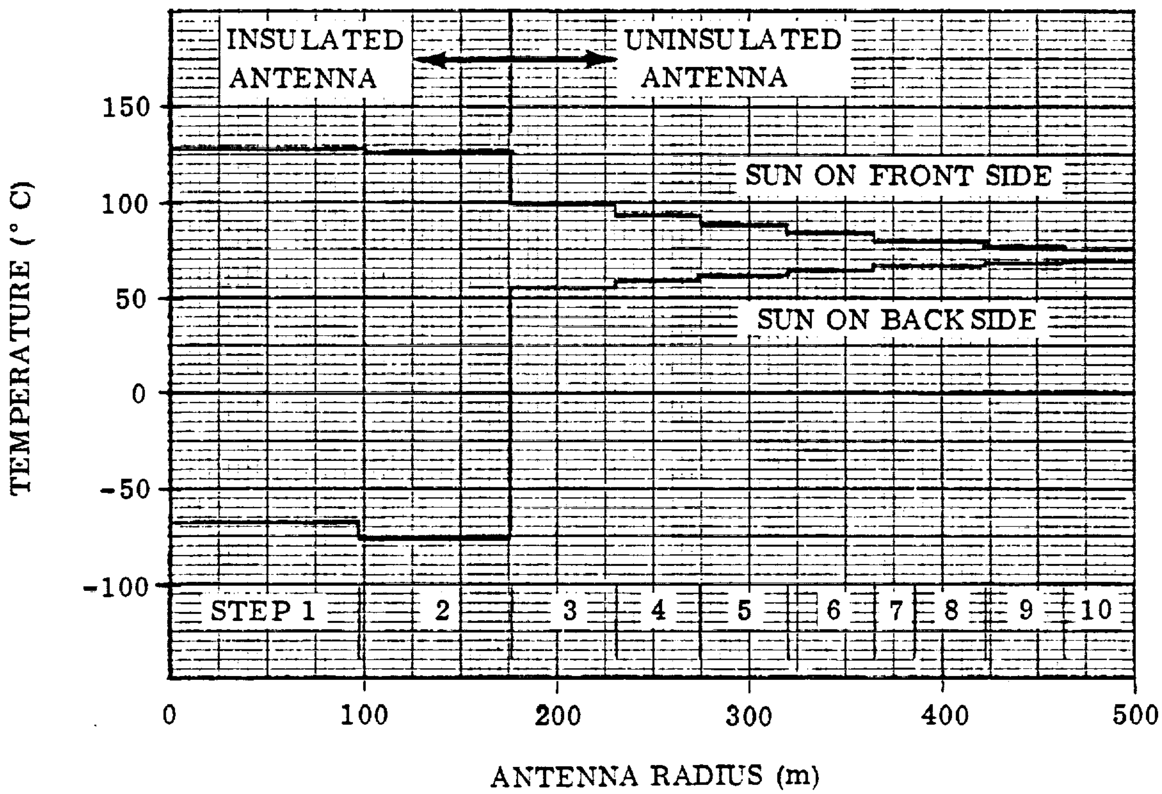
In-space manufactured substitutes employing lunar resources must be capable of meeting these dimensional tolerances over the operating temperature range of the MPTS antenna. This temperature range (ΔT) depends on the antenna's attitude relative to the sun and the local microwave power intensity as shown in Figure A-8. The sun on the front side minus sun on backside ΔT is relatively low (less than 50°C) in the outer uninsulated portion of the antenna, but exceeds 200°C in the antenna's power intensive center portion, which has insulation between the waveguides and klystron radiators. The dimensional effect of this large ΔT is offset by the shorter waveguide lengths used in the high power intensity modules (30 to 36 klystrons per module) located in the center of the antenna.

The maximum permissible coefficient of thermal expansion (CTE^{max}) for MPTS waveguide material has been determined for the 30 klystron module in the Step 2 insulated portion of the antenna.

$$CTE^{max} = \frac{\Delta \ell}{\Delta T \, \ell} = 3.78 \quad \mu m/m/°C$$

Where: $\Delta \ell$ = 0.152 cm (6 mil) from Figure A-7

$\Delta T \, \ell$ = 403.1 °C-m from Figure A-8.



ℓ = LENGTH OF WAVEGUIDE IN EACH ANTENNA STEP

STEP	1	2	3	4	5	6	7	8	9	10
ΔT (C) ℓ (m)	196 1.655	203 1.986	44 2.482	34 1.986	26 2.482	20 3.309	15 4.964	12 4.964	9 4.964	7 4.964
$\Delta T \ell$	324.3	403.1	109.2	67.5	64.5	66.2	74.5	59.6	44.7	34.7

Data Source: Reference 2, Page 188

Figure A-8. MPTS waveguide heating conditions.

CTE values for replacement graphite composite materials were obtained during investigation of primary solar array structure and are listed in Table A-8. This data indicates that the only lunar resource derived substitute material which meets the waveguide CTE requirements is silica glass, with $\alpha_x = 2.88\ \mu m/m/°C$

Step 3 Percent of Substitute Materials Required

Assuming that glass of equal thickness (0.041 cm) is used to replace the graphite epoxy waveguides, the mass of glass required is determined by the density ratio:

$$W_{glass} = \left(\frac{\rho_{glass}}{\rho_{\substack{graphite\\epoxy}}}\right) W_{\substack{graphite\\epoxy}} = \left(\frac{2.491}{1.727}\right) W_{\substack{graphite\\epoxy}}$$

$$W_{glass} = 1.44\ W_{\substack{graphite\\epoxy}}$$

Thin glass waveguides with a vacuum deposited aluminum conductive surface can be entirely derived from lunar resources with a mass only 1.44 times that for the earth baseline. The potential disadvantage of thin glass waveguides is fracture propagation (shattering) due to construction handling or meteroid impact. This problem can be elleviated if thicker foamed glass is used instead of thin sheet glass. Oxygen is a potential foaming gas which can be obtained from lunar resources. If foamed glass is employed, the waveguide wall thickness can be increased while the overall waveguide mass is held equal to or less than that for the earth baseline.

Recommendation: Use foamed glass waveguides with a mass equivalent to the earth baseline graphite epoxy waveguides. All material (glass and oxygen) is obtained from lunar resources.

g) CRES HEAT PIPE TUBING

The Earth baseline SPS employs CRES heat pipes with mercury transport fluid to dissipate klystron losses. The heat pipe evaporators, an integral part of the klystron, pick up the waste heat for transfer to the radiator. The klystron thermal radiator has six sections, two small sections for the collector and the four larger ones for the cavities and solenoid. Six independent heat pipes perform the energy transfer from each klystron to these radiator sections. The collector section radiates at 500°C (773°K) and the cavity/solenoid section at 300°C (573°K).

Step 1 Earth Material Composition

An austenitic stainless steel such as 304L has been assumed for heat pipe tubing. The elemental constituents in this material are:

Material	Constituent Percent By Weight	Available in Lunar Resources ?
Chromium	18.0 - 20.0	Yes, but only in 0.05% to 0.35% concentration.
Nickel	8.0 - 12.0	Yes, but only 100 - 300 parts per million.
Manganese	2.0 Max.	Yes, but only in 0.05% to 0.2% concentration.
Silicon	1.0 Max.	Yes, but not needed.
Carbon	0.03 Max	Yes, but only 100 - 200 parts per million.
Iron	Balance	Yes

CRES 304 L density = 7.95 g/cm^3

While the major constituent of 304L (70% iron by weight) is available from lunar resources, a significant percentage of this SPS material requirement (approximately 20% chromium and 10% nickel) must be obtained from Earth.

The performance requirements which the heat pipe material must satisfy are as follows:

1) Mercury compatibility at 500°C.
2) Non-magnetic in the vacinity of the klystron to preclude field interruptions and beam defocusing.
3) Reasonable thermal conductivity to provide heat transfer from the mercury transport fluid to the space radiator.

Step 2 Lunar Resource Substitutions

Lunar derived materials from which tubing could be manufactured include quartz glass, aluminum, titanium, iron, and alloys of these materials. The primary performance requirements are mercury compatibility in a 500°C operating environment, reasonable thermal conductivity, and non-magnetic properties. Table A-9 provides an assessment of these candidate materials against these performance requirements.

Table A-9 . Heat Pipe Material Evaluation

Candidate Material	Density (g/cm^3)	Melting Temp(°C)	Mercury Compatibility at 500°C	Non-Magnetic	Reasonable Thermal Conductivity	$\frac{\rho_{\text{Substitute Material}}}{\rho_{\text{CRES}}}$
Aluminum	2.70	660	No*	Yes	Yes	0.340
Titanium	4.54	1,660	No*	Yes	Marginal	0.571
Iron	7.87	1,535	No (Acceptable at 300°C)	No	Yes	0.990
Copper Coated Aluminum	2.75	660	No (Acceptable at 300°C)	Yes	Yes	0.346
9 Cr-1 Mo Steel	7.83	~1500	Yes	No	Yes	0.985
Sicromo 5S Croloy 5 Si	7.70	~1500	Yes	No	Yes	0.969
Quartz Glass	2.21	1,720	Yes	Yes	No	0.278

* These metals and their alloys are subject to serious embrittlement and catastrophic fracture when in contact with liquid mercury as well as with its vapor.

The only materials which meet the high temperature mercury compatibility requirement are the two alloy steels and quartz glass.

Modified 9 Cr-1 Mo steel has been evaluated for possible use in the SNAP-8 mercury boiler application (Ref. 4). This alloy has very small additions (generally under 0.10%) of other alloying elements such as niobium, vanadium, boron, zirconium and nitrogen and is stronger than 304L at all temperatures at least up to 600°C. At low velocities (0.6 cm/sec) this alloy exhibits excellent corrosion resistance to mercury at 580°C (853°K) for times up to 5,000 hours. The use of the 9 Cr-1 Mo steel permits up to 90% utilization of lunar material (iron) with only 10% of the ingredients supplied from Earth.

Even greater utilization of lunar materials can be achieved by the use of the low alloy steels which have long been employed in the manufacture of mercury boilers. Steels containing 4-6% chromium, 0.5-0.6% molybdenum and 1-2% silicon (Sicromo 5S and Croloy 5 Si) exhibit corrosion rates in mercury of 0.0075 - 0.010 cm/year at temperatures up to 620°C (Ref. 5). The use of such steels may limit the mass of material which must be transported from Earth to 6 to 7% of the total mass of the heat pipes.

The addition of 0.0001% to 0.001% (1 to 10 ppm) of titanium dissolved in the mercury reduces the corrosion of ferrous alloys by a factor of 10 to 20. The corrosion rate of Sicromo 5S at 538°C and a mercury flow rate of 3 cm/sec was reduced to less than 0.00075 cm/year (Ref. 5).

Quartz glass exhibits excellent mercury compatibility, but the possibility of in-space breakage and the effects of mercury contamination are very undesirable.

Since none of the candidates in Table A-9 meet all three performance requirements, there remains only two choices; 1) retain the earth baseline 304L CRES heatpipes, which allows only 70 percent lunar resource utilization, or 2) use 304L CRES only at the klystron interface, and employ one of the special mercury compatible alloy steels for the majority of the heat pipe, allowing approximately 90 percent lunar resource utilization.

Step 3 Percent of Substitute Materials Required

If tubing diameter and wall thickness remain unchanged for a substitute material, the mass requirement will be the ratio of the replacement materials density to the density of CRES.

$$W_{\substack{\text{substitute}\\\text{material}}} = \left(\frac{\rho_{\substack{\text{substitute}\\\text{material}}}}{\rho_{\text{CRES}}} \right) W_{\text{CRES}}$$

These ratios are included in Table A-9 .

One concern of in-space heat pipe manufacture is the filling of tubes with mercury transport fluid. Mercury is a highly toxic material which must be obtained from Earth since it is unavailable in lunar resources. The in-space handling of mercury will have to be carefully evaluated to guard against spills and contamination of the space manufacturing facility.

Even though the heat pipe transport fluid is a relatively low mass item (0.33 % of total SPS mass), it would be beneficial if a suitable less toxic lunar or earth substitute could be found. Unfortunately the heat pipe operating temperature range eliminates many commonly used earth fluids, and lunar volatiles which would provide a good heat pipe transfer medium are practically non-existent.

While it is recognized that a change in the heat transport fluid necessitates a change in the design of the heat pipe system, consideration should be given to the possible use of the sodium-potassium eutectic composition (NaK) which is widely used as a coolant in nuclear power systems. This material remains liquid over the temperature range of 66°C to 1518°C. High purity iron (Armco Iron) is resistant to attack by NaK at temperatures up to approximately 900°C and thus lunar iron could serve as construction material for the heat pipe system with this coolant.

Fluorochemical liquids which are relatively inert, nontoxic and chemically stable at temperatures up to approximately 400°C are being used as heat transfer fluids. These fluids will not, however, meet the 500°C temperature requirement of the SPS heat dissipation system.

Recommendation: Retain the Earth baseline 304L CRES material for the heatpipe/ klystron interface. At a distance of approximately 0.15 m from the klystron housing, transition to the Cr-Mo-Si alloy steel for the remainder of the heat pipe (approximately 93 percent). These heatpipes will be fully compatible with the titanium treated mercury transport fluid. The chromium and nickel (30% of 304L CRES mass) and the chromium and molybdenum (6-7% of the mass of the remaining alloy steel heat pipes) will be transported from Earth and alloyed with lunar iron and silicon. The small amount of carbon needed (0.15%) can be provided either from Earth or lunar sources. Lunar resources provide approximately 91 percent of the earth baseline material requirements while the remaining 9% must still be obtained from earth.

h) & i) ALUMINUM SHEET CONDUCTORS AND RADIATORS

The earth baseline SPS uses aluminum sheet for a variety of ambient temperature applications including photovoltaic array and MPTS power busses, and radiators for the Klystron solenoid cavity and DC-DC converter transformer. Since commercially pure aluminum can be readily used for these applications, and aluminum is abundant in lunar highlands material, lunar derived aluminum can be directly substituted for these earth aluminum applications.

j) COPPER KLYSTRON SOLENOID CAVITY

The klystron solenoid cavity consists of machined copper parts which form heat pipe evaporator passages and is the core over which the solenoid is wound. The solenoid cavity must be conductive, non-magnetic, and withstand an operating temperature of 300°C. The material must also be compatible with mercury, which is employed as the heat pipe transfer fluid.

Step 1 Earth Material Composition

The machined copper solenoid cavities are assumed to be manufactured from copper no. 101 (oxygen free electronic) which is a high purity copper used for hollow conductors, bus bars and other conductors. If the solenoid cavity requires the use of copper alloys having higher strengths at moderately elevated temperatures, silver bearing copper alloys such as the 114 or 155 grades can be employed for this application. All of these materials must be obtained from Earth since lunar minerals do not contain copper in concentrations of more than 5 to 20 parts per million nor silver in amounts greater than 100 parts per billion.

Step 2 Lunar Resource Substitutions

The best non-magnetic electrical conductor available from lunar material is aluminum. Although it's conductivity is slightly lower than that for copper, aluminum's density is considerably less, which results in reduced aluminum mass required to transmit an equivalent amount of electrical energy. Aluminum is incompatible with mercury, as discussed in paragraph (g) and Table A-9. Aluminum's only other potential disadvantage is its lower melting point, for certain high temperature applications it may be unsuitable. In this instance, however, sufficient margin exists between aluminum's melting temperature (660°C) and its maximum use temperature (300°C) to alleviate any concern.

Step 3 Percent of Substitute Materials Required

$$W_{aluminum} = \left(\frac{\rho_{aluminum}}{\rho_{copper}}\right)\left(\frac{K_{\substack{\text{Elect. Cond.}\\ \text{aluminum}}}}{K_{\substack{\text{Elect. Cond.}\\ \text{Copper}}}}\right) W_{copper}$$

$$W_{aluminum} = \left(\frac{2.70}{8.94}\right)\left(\frac{5977.3}{3766.8}\right) W_{copper} = 0.479 \quad W_{copper}$$

Recommendation: Use aluminum or aluminum alloy for klystron solenoid cavities. If strength requirements dictate the use of an aluminum alloy, an alloy containing 4-5% magnesium should be considered since the latter metal is also available on the moon. Because of aluminum's incompatibility with mercury, it will be necessary to coat all mercury contact surfaces with approximately 0.03 cm thick copper. This can be done by vapor or electrodeposition processes. It is estimated that up to 90% of the mass of the klystron solenoid cavities may be derived from lunar resources. The remaining 10% must be obtained from earth.

k) IRON COMPONENTS

The earth baseline SPS employs machined iron parts as poles in the klystron solenoid, and as the major material component of the DC-DC converter SENDUST transformer. Some commercially pure iron is used for these applications, and iron is abundant in lunar mare material, lunar derived iron can be directly substituted for these earth iron applications.

l) COPPER SHEET KLYSTRON COLLECTOR RADIATORS

Each earth baseline SPS klystron has two 500°C heat pipes to remove waste heat from the collector and dissipate this energy through radiators. The radiators are constructed of flat (or slightly corrugated) copper sheet with the heat pipe routed down the center of the radiator (see paragraph A.1.2).

Step 1 Earth Material Composition

The copper sheet is assumed to be manufactured from commercially pure copper. All of this material must be obtained from earth, since lunar resources contain no copper concentrations worthy of recovery efforts.

Step 2 Lunar Resource Substitutions

The best thermal conductor available from lunar material is aluminum. Although its thermal conductivity is slightly lower than that of copper, aluminum's density is considerably less, which results in reduced aluminum mass required to dissipate an equivalent amount of thermal energy via radiation to space. Aluminum's only disadvantage is its lower melting point, which at 660°C offers a safety margin ΔT of only 160°C with the klystron collector heat pipe operating temperature. This might well be a very undesirable operating temperature for a moderately or highly stressed aluminum part, but the radiator is essentially a zero stress part. Its only function is to act as a cooling fin in a near zero g environment. As long as the operating temperature remains below its melting point, and surface emmitence properties are not degraded, aluminum should be an acceptable substitute for 500°C radiators.

Step 3 Percent of Substitute Materials Required

$$W_{aluminum} = \left(\frac{\rho_{aluminum}}{\rho_{copper}}\right)\left(\frac{K_{\text{thermal cond. copper}}}{K_{\text{thermal cond. aluminum}}}\right)^{*} W_{copper}$$

$$W_{aluminum} = \left(\frac{2.70}{8.96}\right)\left(\frac{3.73}{2.22}\right) W_{copper} = 0.506\ W_{copper}$$

* K values at 500°C

Recommendation: Use pure aluminum sheet with a mass of one half that of the earth baseline copper sheet. All aluminum material is obtained from lunar resources.

m) CRES KLYSTRON HOUSING

The earth baseline SPS klystrons are enclosed within a CRES housing. This is a non-magnetic machined metal part which has an operating temperature requirement of 500°C.

Step 1 Earth Material Composition

An austenitic stainless steel such as 347 has been assumed for the machined klystron housing. The composition of this alloy is:

Element	Constituent Percent by Weight	Available in Lunar Resources ?
Chromium	17.0 - 19.0	No
Nickel	9.0 - 13.0	No
Manganese	2.0 Max.	Yes, up to 0.2% concentration in mare
Niobium + Tantalum	10 × Carbon	Only in PPM concentrations.
Carbon	0.08 Max.	Only in PPM concentrations.
Silicon	1.0 Max	Yes
Iron	Remainder (65-72%)	Yes

CRES 347 density = 8.00 g/cm^3

While the bulk of the 347 CRES ingredients (primarily iron) are available in lunar resources, a significant percentage of this SPS material requirement (28-35%) must still be obtained from earth.

Step 2 Lunar Resource Substitutions

Lunar derived metallics from which the klystron housings could be manufactured include aluminum, and titanium, and alloys of these materials. The primary performance requirement for the housing is operation in a 500°C environment. Table A-9 provides an assessment of these candidate materials.

Step 3 Percent of Substitute Materials Required

If housing diameter and wall thickness remain unchanged for a substitute material, the mass requirement will be the ratio of the replacement materials density to the density of CRES.

$$W_{\substack{\text{substitute}\\ \text{material}}} = \left(\frac{\rho_{\substack{\text{substitute}\\ \text{material}}}}{\rho_{\text{CRES}}} \right) W_{\text{CRES}}$$

These ratios are included in Table A-9 .

Recommendation: Since neither titanium nor aluminum in the unalloyed conditions come close to matching the strength properties of 347 CRES, it may be necessary to either redesign the klystron housings or alloy the above metals to higher strengths. Aluminum can be alloyed with magnesium and silicon, both of which are available on the moon, and titanium can be alloyed with aluminum and manganese; also available in lunar-minerals.

Aluminum or aluminum alloys would be the preferred lunar derived materials for the klystron housings. Aluminum would weigh 0.338 times the weight of the CRES alloy.

n) VACUUM DEPOSITED COPPER SOLAR CELL INTERCONNECTS

The earth baseline SPS uses copper as electrical connections for the photovoltaic array silicon cells. The copper is vacuum deposited onto the silica glass substrate to provide N and P contacts for each solar cell. These connections must provide good electrical conductivity and be capable of withstanding the 500°C annealing temperature employed to counteract array radiation degradation.

Step 1 E arth Material Composition

Vacuum deposited copper is 99.9+% pure. All of this material must be obtained from earth since only minute traces of copper are contained in lunar resources.

Step 2 Lunar Resource Substitutions

The best electrical conductor available from lunar material is aluminum. Although it's conductivity is slightly lower than that for copper, aluminum's density is considerably less, which results in reduced aluminum mass required to transmit an equivalent amount of electrical energy. Aluminum's only potential disadvantage is its lower melting point; it has a melting temperature of 660°C, only 160°C above the array annealing temperature. This might be unacceptable for a highly stressed structural part, but since the photovoltaic sandwich consists of thin silicon and silicon dioxide sheets electrostatically bonded together, the interconnect is a non-structural connection. As long as the annealing temperature remains slightly below its melting point, vacuum deposited aluminum should be an acceptable substitute for solar cell interconnects.

Step 3 Percent of Substitute Materials Required

$$W_{\text{aluminum}} = \left(\frac{\rho_{\text{aluminum}}}{\rho_{\text{copper}}}\right)\left(\frac{K_{\substack{\text{Elect. Cond.}\\ \text{Copper}}}}{K_{\substack{\text{Elect. Cond.}\\ \text{Aluminum}}}}\right) W_{\text{copper}}$$

$$W_{\text{aluminum}} = \left(\frac{2.70}{8.96}\right)\left(\frac{5977.3}{3766.8}\right) W_{\text{copper}} = 0.478\ W_{\text{copper}}$$

Recommendation: Use vacuum deposited pure aluminum with a mass of 0.478 times that of the earth baseline vacuum deposited copper. All aluminum material is obtained from lunar resources.

o) GRAPHITE COMPOSITE MPTS ANTENNA STRUCTURE

The earth baseline SPS employs graphite composite structure similar to that described in paragraph d) for the primary, secondary, and waveguide module support structures in the MPTS antenna, plus in the rotary joint structure and as support for MPTS power distribution busses. The primary performance requirements for graphite composite are a high modulas of elasticity and low coefficient of thermal expansion (CTE). Low CTE is especially important for the primary and secondary MPTS support structure. The 1000 m diameter waveguide surface supported by this structure must remain flat within ~2 arc-min. during varying solar orientations. The material employed for power distribution bus supports must be an electrical insulator.

Step 1 Earth Material Composition

See paragraph d) introduction and Step 1 discussion, pages A-17 and A-18.

Step 2 Lunar Resource Substitutions

As previously determined and described in paragraph d), foamed S-glass appears to be the only lunar material substitute which is capable of meeting the combined performance requirements.

Step 3 Percent of Substitute Material Required

Assuming that structural stiffness dominates most of the applications contained within category o);

$$W_{\substack{\text{foamed}\\\text{S-glass}}} = 2.0\ W_{\substack{\text{graphite}\\\text{composite}}}$$

Recommendation: Use foamed glass thin wall tubular structural members with an assumed mass approximately 2.0 times the earth baseline graphite thermoplastic MPTS structure. All material (glass and oxygen) is obtained from lunar resources.

SUMMARY

The recommended lunar material substitutions have been compiled in Table A-10 for each of the fifteen SPS applications. Substitute material replacement mass factors vary from 0.338 for replacing the CRES klystron housing with aluminum, to 2.0 for replacing graphite composite structure with foamed glass. The total mass derived from lunar material is 88,190 T which requires an additional 440 T of earth supplied alloying materials. This total material quantity (88,630 T) provides the same functions as the 87,800 T of earth baseline SPS materials. The special earth baseline materials (Ag, W, Hg) and electronic components (various)

Table A-10. Compilation of SPS substitute lunar materials.

Earth Baseline Satellite Power System					Lunar Replacement Materials (T) For SPS						Earth Constituent Mat'l. Mass (T)	TOTAL (T)
Rank	Mass (T)	Mass (%)	Material	Application	Silica Glass	Pure Silicon	Aluminum	Iron	Other	TOTAL		
(a)	21,658	22.2	Borosilicate Glass	Photovoltaic Cell Covers	21,658					21,658	0	21,658
(b)	14,775	15.1	Silicon	Solar Cells		14,775				14,775	≪1	14,775
(c)	14,439	14.8	Fused Silica Glass	Photovoltaic Cell Substrate	14,439					14,439	0	14,439
(d)	6,208	6.4	Graphite Composite	Primary Structure for Solar Array	12,404				(O_2)12	12,416	0	12,416
(e)	5,980	6.1	Copper Wire	Klystron & DC-DC Converter Coils, Power Cables			2,865			2,865	0	2,865
(f)	5,257	5.4	Graphite Composite	MPTS Waveguides	5,252				(O_2) 5	5,257	0	5,257
(g)	3,892	4.0	CRES Tubing	Heat Pipe for Klystron Radiators				3,542		3,542	350	3,892
(h)	3,535	3.6	Aluminum Sheet	Power Transmission Busses, Array & MPTS			3,535			3,535	0	3,535
(i)	2,749	2.8	Aluminum Sheet	Klystron & DC-DC Conv. Radiators			2,749			2,749	0	2,749
(j)	1,820	1.9	Copper (Mach Part)	Klystron Solenoid Cavity			785			785	90	875
(k)	1,758	1.8	Iron	Klystron Solenoid & Transformer for DC-DC Converter				1,758		1,758	0	1,758
(l)	1,539	1.6	Copper Sheet	Klystron Collector Radiators			779			779	0	779
(m)	1,524	1.6	CRES (Mach Part)	Klystron Housing			515			515	0	515
(n)	1,456	1.5	Vacuum Deposited Copper	Solar Cell Inter-Connects			697			697	0	697
(o)	1,210	1.2	Graphite Composite	MPTS Antenna & Other Structure	2,418				(O_2)2	2,420	0	2,420
	87,800	90.0		TOTAL MASS (T)	56,171	14,775	11,925	5,300	19	88,190	440	88,630
	PERCENTAGE OF EARTH BASELINE MASS				57.6	15.2	12.2	5.4	-	90.4	0.5	90.9

must still be supplied from earth for the SPS constructed primarily with lunar resources. This earth supplied material has a mass of 97,550 - 87,800 = 9,750 T for each SPS, resulting in a total SPS mass of 98,380 T. Lunar materials employed for SPS construction are produced from only four elements; silicon, oxygen, aluminum and iron.

REFERENCES

1. Solar Power Satellite System Definition Study Part II, Volume III, SPS Satellite Systems, Report No. D180-22876-3, Boeing Aerospace Company, Seattle, Washington, December 1977.
2. Solar Power Satellite System Definition Study Part II, Volume IV, Microwave Power Transmission Systems, Report No. D180-22876-4, Boeing Aerospace Company, Seattle, Washington, December 1977.
3. Solar Power Satellite System Definition Study Part II, Volume VI, Evaluation Data Book, Report No. D180-22876-6, Boeing Aerospace Company, Seattle, Washington, December 1977.
4. Stone, P. L. "Materials Technology Programs in Support of a Mercury Rankine Space Power System" NASA TN D-7355, Lewis Research Center, September 1973.
5. Liquid Metals Handbook, The Atomic Energy Commission and the Bureau of Ships, Department of the Navy, 2nd Edition, January 1954.

APPENDIX B

Task 5.3 supplementary data, defining earth material requirements sensitivity information developed to support selection of LRU Concepts B, C and D in Volume II, Section 4 of Final Report.

Appendix B contains four sections

B.1 DEFINITION OF GENERALIZED AND SUBSEQUENT DETAILED LUNAR RESOURCES UTILIZATION CONCEPTS

Three generalized options were postulated which represent a broad spectrum of alternatives comprising space based, lunar based, and combination lunar/space based manufacturing scenarios. Iteration of these generalized options via steady state earth material requirements was performed to define an explicit competitive LRU concept representative of each. This was followed by development of detailed steady state material logistics scenarios and sensitivity information for each concept.

Option A - Earth Based - The Earth material utilization scenario, shown in Figure B-1, is based on techniques developed and perfected during NASA's past space accomplishments but implemented on a much larger scale. Two Earth-to-LEO launch vehicles are employed: a fully reusable heavy lift launch vehicle (HLLV) for cargo, and a shuttle derived personnel launch vehicle (PLV). The HLLV is a two-stage fly-back vehicle with chemical propulsion and 424-ton payload capability. Its payload consists of crew support stations, fabrication machinery, assembly jigs, orbital transfer vehicles (OTV), and all construction supplies and OTV propellants. The PLV replaces the Shuttle's tandem burn solid rocket boosters with a series-burn O_2/methane ballistic entry first stage, and has an Orbiter modified to carry 75 passengers with their personal equipment.

Large structural sections are fabricated, inspected and checked out in LEO. These completed satellite sections are transferred to their operational location with unmanned cargo orbital transfer vehicles (COTV). The COTV uses a low-thrust/high-impulse solar powered electric propulsion system and argon propellant. Final assembly of these satellite sections into the complete large space structure is performed at its operational locale, typically GEO. Manned transfer from LEO to GEO is provided by a high-thrust two-stage chemical personnel orbital transfer vehicle (POTV).

Option B - Space Based - The lunar material utilization scenario developed for space manufacturing and space settlements includes unique elements and innovative techniques, and represents the proposals of Dr. Gerard O'Neill of Princeton University. Material brought from earth includes transportation elements and

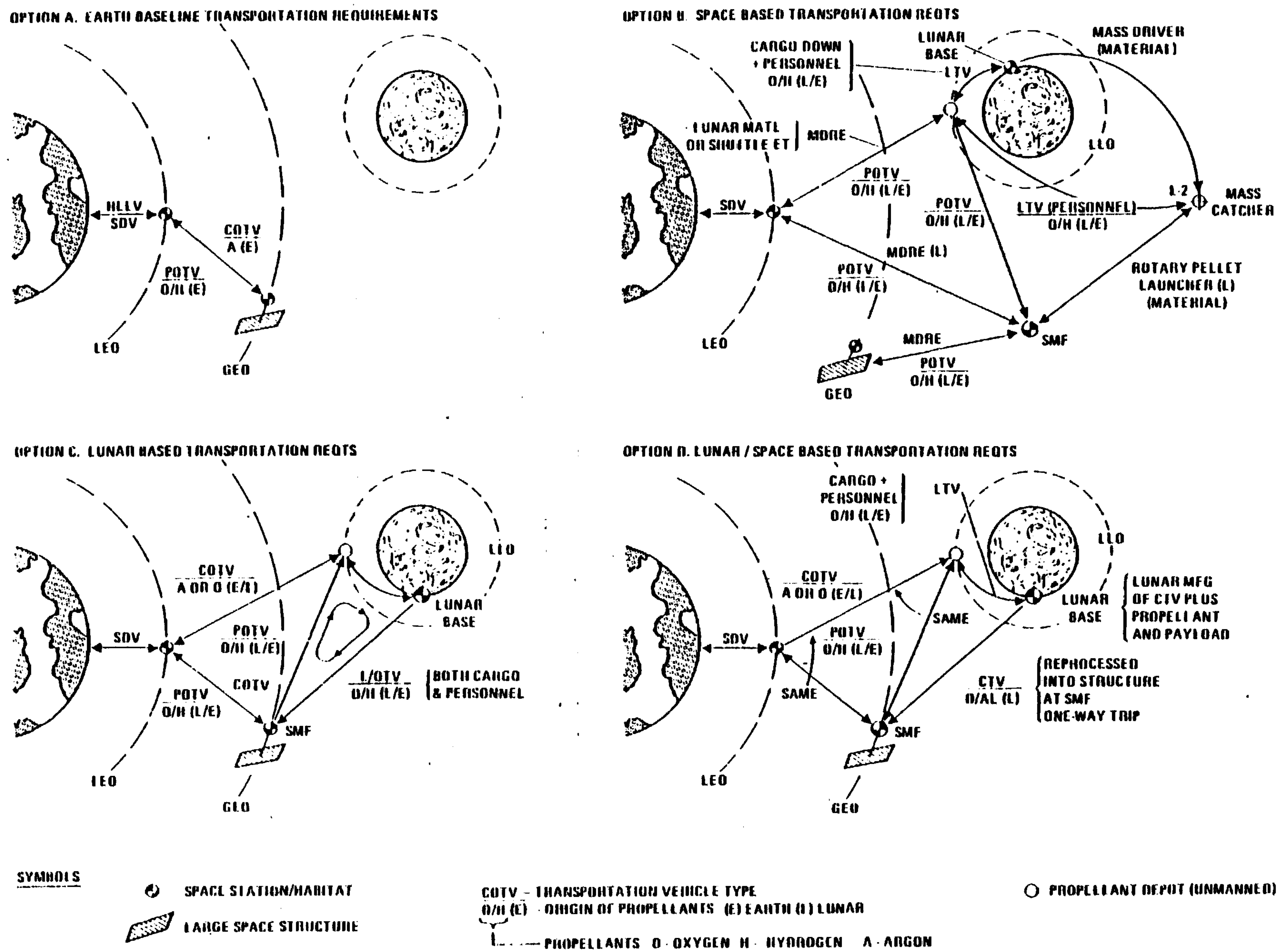


Figure B-1. Four representative implementation options for in-space manufacturing of large structures.

their propellants, lunar mining equipment, material processing and fabrication equipment, personnel plus their habitats and supplies, and a small percentage of large space structure components which cannot initially be manufactured economically in space.

Transfer of these payloads and personnel from earth to LEO is accomplished by Shuttle-derived vehicle (SDV). A relatively small logistics station is constructed of Shuttle external tanks in LEO. This facility is used as a base to assemble transportation, processing, and habitation elements, and to integrate payloads for departure to their operational locales. All personnel transfer to other orbits is accomplished with a high thrust chemical POTV.

Cargo transfer is provided via a low-thrust solar-powered linear electromagnetic accelerator called a mass driver reaction engine (MDRE). This vehicle produces thrust by exhausting any available waste mass (ground-up external tanks or lunar slag) at very high velocity (8,000 m/s). The MDRE delivers lunar base material plus the lunar transfer vehicle (LTV) and its propellants to low lunar orbit (LLO), the mass catcher to L_2, and space manufacturing facility/habitation modules to their selected locale.

The lunar base is established by using the throttlable chemical LTV to land material and personnel. The lunar base consists of mining equipment, a fixed mass driver catapult to launch lunar material to L_2, living accommodations for personnel, a power plant (solar or nuclear), and supplies. Lunar surface operations include material collection, screening, bagging and launch by the mass driver in a steady stream toward L_2. This material is retrieved by the mass catcher at L_2, accumulated in large loads, and subsequently delivered to the space manufacturing facility (SMF), by rotary pellet launcher and terminal tug. At the SMF, this lunar soil is processed into useful structural materials, fabricated into components, and final-assembled into the large space structure.

Although most of these manufacturing operations are highly automated, a significant number of personnel are required for final assembly, machine operation, maintenance and repair, plus support services. Completed earth service satellites are transferred to their operating orbital location (typically GEO) by MDRE. This space manufacturing concept is amenable to bootstrapping, a technique by which a relatively modest initial lunar material throughput can provide products which are then directly applied to increasing the original manufacturing facility's production capability. Thus, sustained bootstrapping can simultaneously provide increased production capability and products. Unfortunately, due to this study's goal of determining a material requirements threshold point, we will be unable to take advantage of bootstrapping. This occurs because the bootstrapping concept results in a steadily increasing production capability and manufacturing rate, so comparison with constant rate manufacturing operations is extremely difficult.

Option C - Lunar Based - This option constitutes a significant departure from the Option B concept in two primary areas: material processing occurs on the lunar surface rather than in-space, and conventional rockets replace the mass driver catapult, mass catcher, and MDRE. Option C has some transportation and support elements that are very similar to those in Option B, such as earth launch and LEO station requirements. OTVs differ from those in B only by the design of cargo transfer stages and their propellant needs (type and quantity).

The COTV is an electric propulsion stage which can use either earth-supplied argon propellant when outbound or lunar-supplied oxygen propellant when inbound. The lunar base is significantly larger since it now provides material processing and component manufacturing in addition to mining and beneficiation.

A chemical lunar/orbital transfer vehicle (L/OTV) is used to transport finished construction supplies to the space manufacturing facility. The L/OTV propellants are lunar derived oxygen and Earth-supplied hydrogen. This vehicle normally makes a round trip from lunar base to SMF to LLO and back to the lunar base. It also supplies oxygen to a propellant depot in LLO for the COTV. Large space structure fabrication and final assembly are accomplished at the SMF which may be coincident to its product's use location.

Option D - Lunar/Space Based - The approach taken by the lunar/space-based option reduces earth propellant requirements. This is accomplished by obtaining both fuel and oxidizer from lunar materials, and is identical to Option C except for the lunar base, SMF, and the transportation between them. To reduce propellant requirements the cargo transfer vehicle (CTV), which transports finished components from lunar base to SMF, is configured as an expendable vehicle. This can only be competitive if the CTV tankage is manufactured at the lunar base from lunar material (aluminum), and reprocessed at the SMF into large space structure components. Therefore, some manufacturing operations are duplicated at these two locations, but the majority of lunar material processing remains at the lunar base. The lunar base must be expanded to include propellant tank fabrication and CTV assembly, checkout, and launch. CTV propulsion (engine) and avionics modules are earth-manufactured subsystems which are recycled from SMF to lunar base for reuse. The return of these subsystems is accomplished by chemical OTVs and LTVs which also perform all personnel transfer.

These three Lunar Resources Utilization options were utilized only as representative techniques encompassing a wide range of space manufacturing scenarios. The earth material requirements analysis technique, described in Volume II, Section 4.2, was employed to determine effects of various options on each of these generalized LRU scenarios. Variable input parameters included lunar material utilization percentage, alternative propellants and propellant sources, different transportation element designs, and efficiencies of material processing, manpower utilization and so forth. The detail LRU systems concepts which resulted from this iterative process are depicted by Figure B-2. Definition of revisions made to the generalized options to obtain these detailed concepts and EMR sensitivity analysis results are contained in Sections B.2, B.3 and B.4 for LRU Concepts B, C and D respectively.

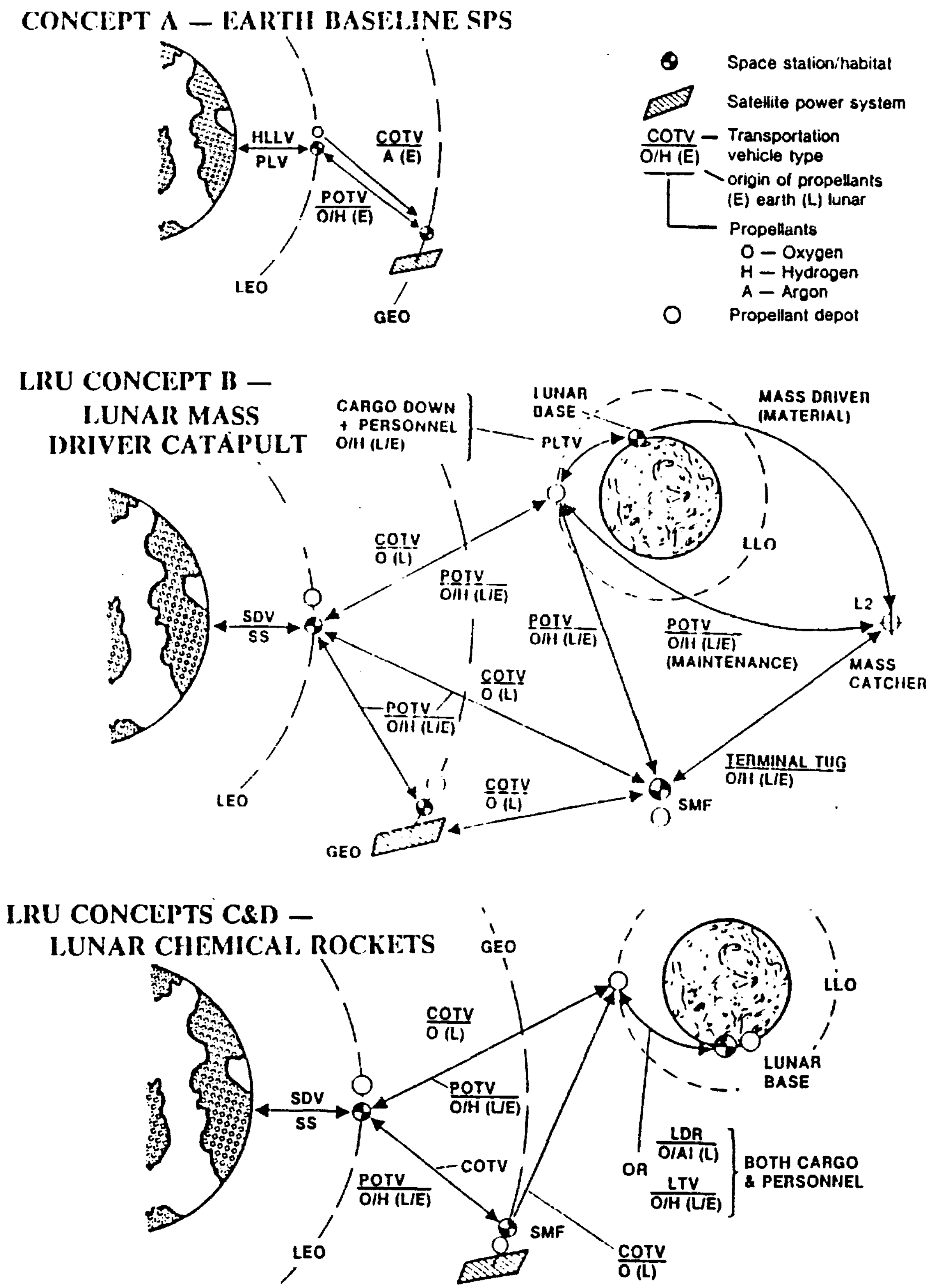


Figure B-2. Updated LRU Space Construction Concepts.

B.2 CONCEPT B - LUNAR MASS DRIVER CATAPULT

This systems concept is characterized by the mass driver catapult/catcher for lunar material transport, and lunar material processing at the space manufacturing facility. Concept B is considered to be the most technologically advanced of the LRU system concepts. Due to its innovative features, it exhibits considerable technical risk but also offers significant potential benefits. Figure B-3 shows the material requirements for the revised version of systems Concept B. This figure illustrates the transportation logistics flow of all materials including payload, propellants, life support (LS) consumables, and lunar material processing chemicals during the steady-state manufacturing phase of operations for LRU at the 89.6 percent level. Crew requirements reflect a SPS production rate of one 10 GW satellite annually.

Analysis of the original option B scenario as described in Section B.1 (shown in Figure B-1) has resulted in one significant revision: the mass driver reaction engine (MDRE) was replaced by an ion-electric COTV employing lunar oxygen as propellant. This change was made necessary by extremely poor MDRE performance when using transfer ΔV's consistent with option A values. Even if theoretical ΔV's are employed for the MDRE, the ion-electric COTV offers significant performance improvements due to its higher specific impulse and reduced propellant requirements. (Ref. 1) Specifically this COTV replacement is recommended because:

1) The COTV specific impulse is approximately six times greater than MDRE.

2) A lunar derived propellant, oxygen, is acceptable for use with an ion-electric COTV. This reduces somewhat the MDRE advantage of using any available waste material as reaction mass.

3) Study personnel feel strongly that if the MDRE were used, it should employ a material such as oxygen for reaction mass. This will eliminate the safety concern of solid high velocity exhaust particles in the vicinity of habitats, manufacturing facilities, and SPS's. Thus similar lunar propellant processing requirements would be imposed for MDRE or ion electric COTV, since both use oxygen propellant.

Figure B-3. LRU Concept B - Mass Driver Catapult.

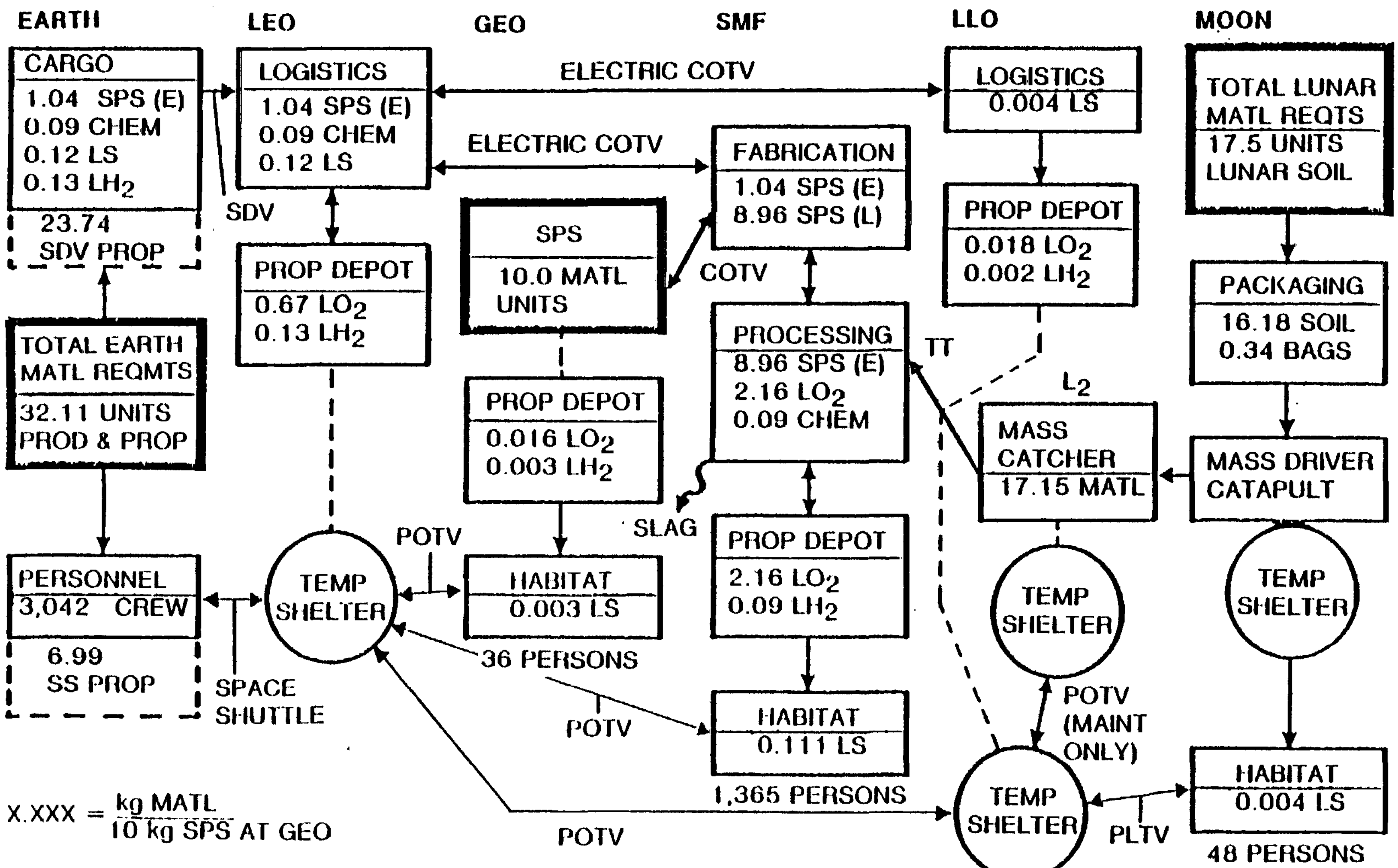


Figure B-3 shows that only 32.11 total earth material units, consisting of 1.38 units of payload plus SDV propellant must be launched from earth to construct 10 units of SPS and deliver it to geosynchronous orbit.

Sensitivity to Lunar Resource Utilization (LRU) Percentage

Material requirements as a function of lunar resource utilization percentage for the new Concept B with ion electric COTV, are displayed in Figure B-4. An interesting trend shown by this data is that both the earth material requirements (EMR) and the lunar material requirements (LMR) decrease with increasing percentages of lunar material in the SPS. The primary reason for this is use of the solar or nuclear powered mass driver catapult (linear electromagnetic accelerator) which provides propellant free (but not power free) launch of material from the moons surface. The remaining primary LMR driver is the oxygen propellant required for cargo transfer from LEO to SMF. As the lunar material percentage increases, the quantity of oxygen propellant needed for transfer of earth materials decreases slightly.

Sensitivity to COTV Type

Similar LRU percentage data is plotted in Figure B-5 for Concept B with a mass driver reaction engine rather than an ion electric COTV. The MDRE is used for all transfer routes previously serviced by the COTV (see Figure B-3). The decreasing trend of both EMR and LMR with LRU percentage increase is considerably more pronounced with the MDRE transfer vehicle, and total LMR is much higher. One reason for this is the assumption that MDRE propellant should be liquid or solid oxygen rather than slag. While lunar slag or ground up external tanks have been proposed for this usage, the continuous expulsion of 8,000 meter/second solid projectiles in the near vicinity of space work areas and habitats is undesirable. The increased LMR for MDRE usage is primarily due to the larger quantities of oxygen propellant required. This increased oxygen requirement is due to the relative performance capability of MDRE and ion electric propulsion. The high specific impulse performance of the ion thrusters, 68,600 N-s/kg versus 7,800 N-s/kg for the MDRE, significantly reduces the lunar material requirement and, to a lessor degree, the earth material requirements. This comparison between MDRE and ion COTV has even been biased in favor of the MDRE by using ΔV requirements nearly half those for the COTV (see Table 4-2 in Volume II, Section 4).

It is due to this material requirements comparison data that the use of an electric ion oxygen propellant COTV has been recommended While the employment of lunar slab as MDRE reaction mass would eliminate most of the EMR/LMR impact of Figure B-5, the technical risk would remain, resolution of the ΔV question would be required, and the environmental hazard for the habitats and SPS would be added.

Sensitivity to Chemical Loss Fraction

Figures B-6, B-7 and B-8 show the effect of increased processing chemical loss (the inability to recover earth chemicals during lunar material processing) on material requirements. The revised baseline with an electric COTV (Figure B-6) exhibits relatively low EMR sensitivity compared to the option B alternative with MDRE (Figure B-7). This extreme MDRE sensitivity is due to the large quantity of oxygen propellants which must be produced for the MDRE to compensate for its relatively poor performance capability. Figure B-8 depicts this MDRE propellant requirement sensitivity on lunar materials requirements. Large changes in soil processing requirements occur in order to supply the oxygen needed to transport larger amounts of processing chemicals from earth. Since the COTV is much more efficient and uses considerably less propellant, the increased LO_2 required to transport chemicals with the electric COTV is very much lower.

Sensitivity to Lunar Soil Oxygen Recovery Ratio

Figures B-9 and B-10 present EMR oxygen recovery data for the revised electric COTV baseline and MDRE alternative, respectively. The information shown reflects EMR sensitivity to the percentage of oxygen which can be efficiently extracted from lunar soil. Nominally, lunar highlands soil consists of approximately 44 percent oxygen. Variations from the assumed 75% recovery (33% oxygen per kg of lunar soil processed) result in extremely low EMR sensitivity for the COTV (Figure B-9), but significant sensitivity for the MDRE (Figure B-10) due to its high demand for oxygen propellant.

Sensitivity to Crew Size

Figure B-11 shows EMR sensitivity to the total number of personnel required for in-space operations. Increased crew size necessitates supply of additional life support and additional POTV flights to tran sport personnel back to arth after their nominal duty tour. The data in Figure B-11 indicates that EMR becomes relatively sensitive to increased crew requirements at higher LRU percentages.

Sensitivity to Cargo Orbital Transfer Vehicle Propellant Requirements

Figures B-11 and B-12 show the effect of cargo transfer vehicle efficiency on EMR. The ion electric COTV (Figure B-11) is almost completely insensitive to increased requirements for lunar derived oxygen propellant due to its high stage efficiency. The MDRE, however, is somewhat EMR sensitive to lunar derived oxygen propellant requirements, particularly at the lower LRU percentages (Figure B-12). This occurs since at lower LRU's, additional LO_2 propellant is required to transport earth materials over the high ΔV LEO to GEO transfer route.

Sensitivity to Terminal Tug Requirements

The terminal tug operates in the vicinity of the space manufacturing facility to capture incoming loads of lunar material, and send maneuvering reaction mass back to the L_2 mass catcher. The tug is assumed to be a conventional LO_2/LH_2 chemical rocket since it must have a relatively high thrust level and must operate near space facilities (precludes use of slag reaction mass). This propellant must be delivered from the Earth (LH_2) and moon (LO_2) by orbital transfer vehicle. Figures B-13 and B-14 compare the EMR sensitivity to ion-electric COTV and MDRE supply of these propellants, respectively. Due to the differential performance capability of these two vehicles, the EMR sensitivity for electric COTV delivery of tug propellants is low (Figure B-13), while MDRE EMR sensitivity is high (Figure B-14).

Figure B-4. Option B - Revised Baseline
(Ion Electric COTV)

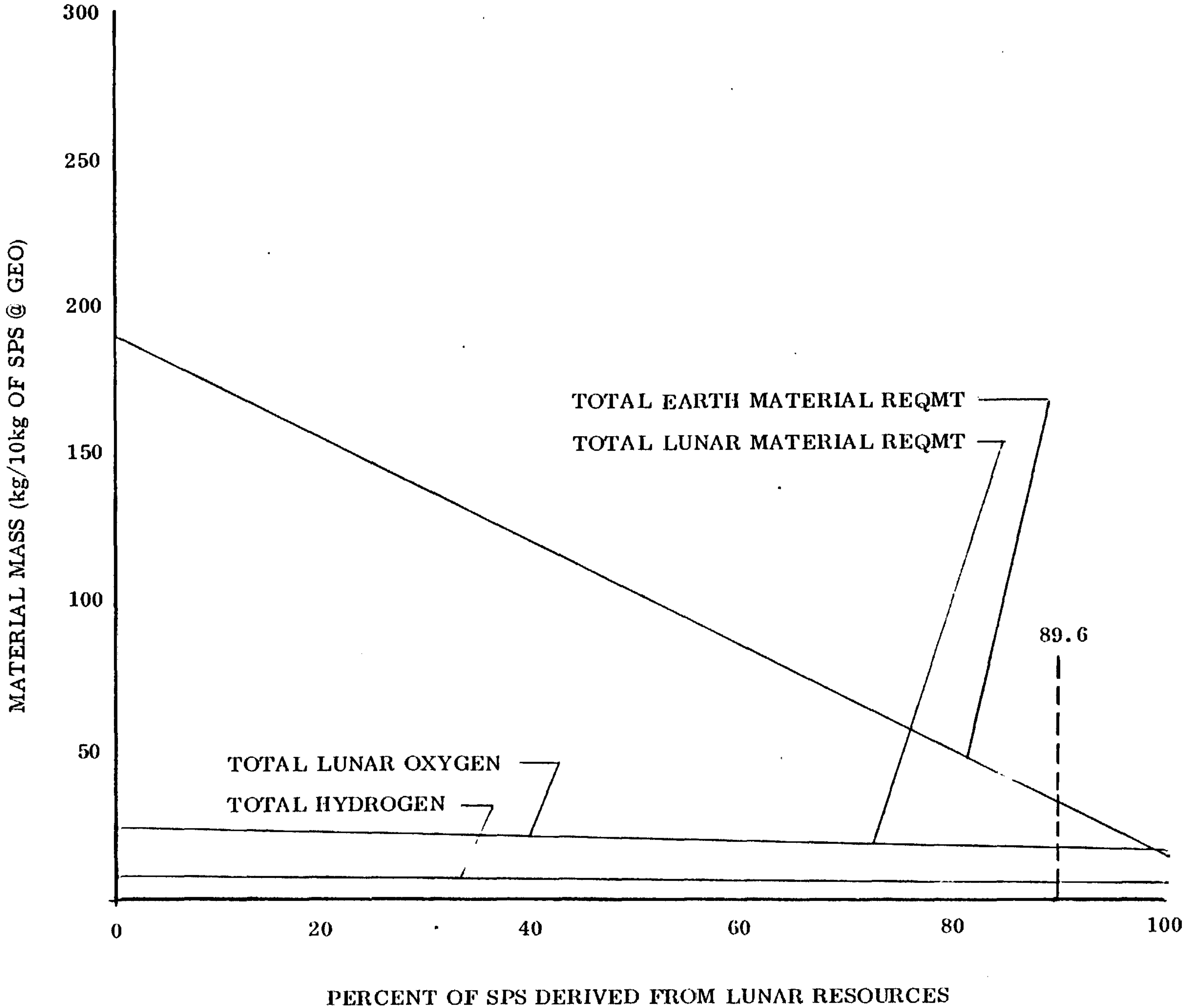


Figure B-5. Option B Alternate
(Mass Driver Reaction Engine)

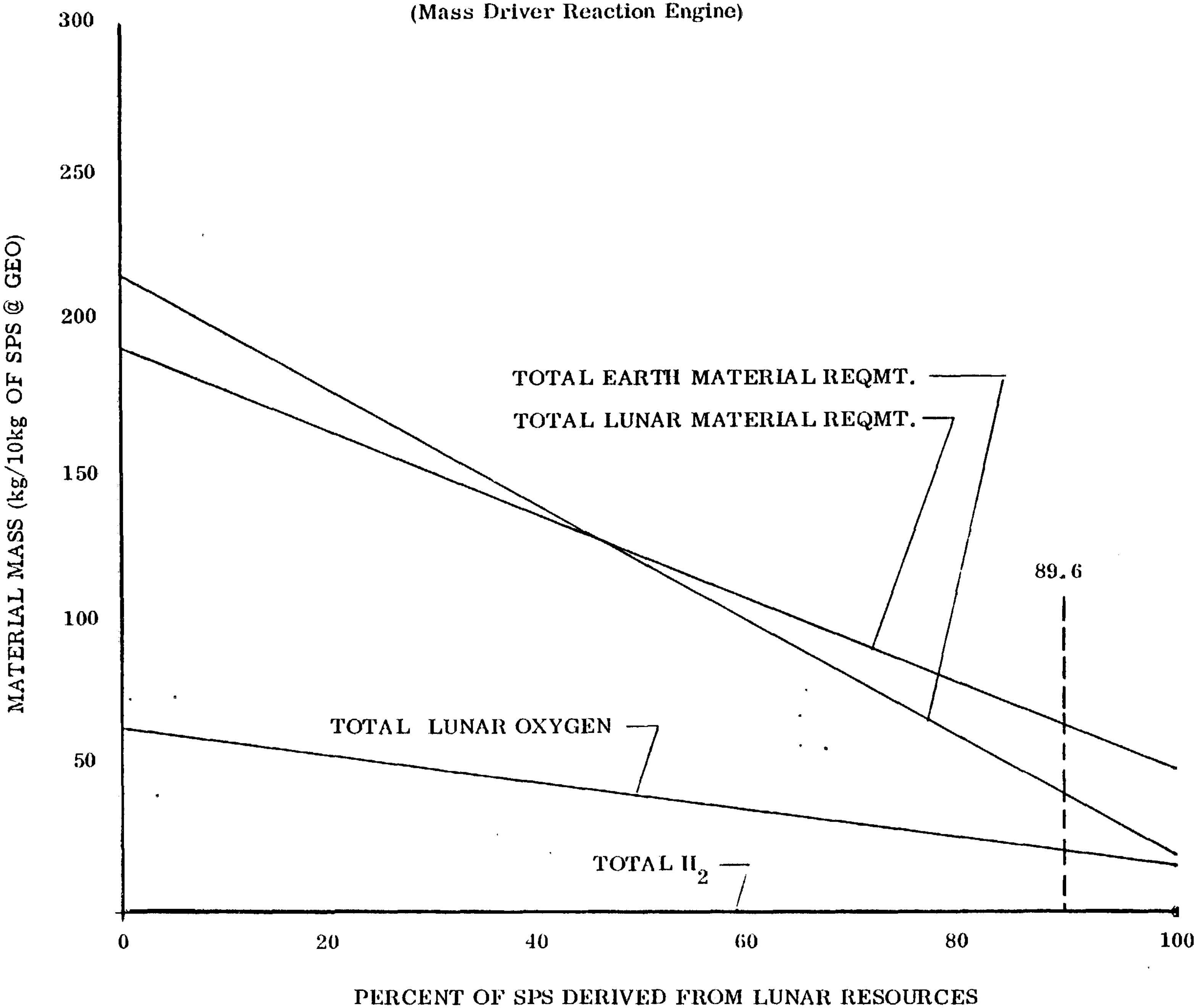


Figure B-6. Option B Revised Baseline with COTV - Sensitivity to Chemical Loss Fraction

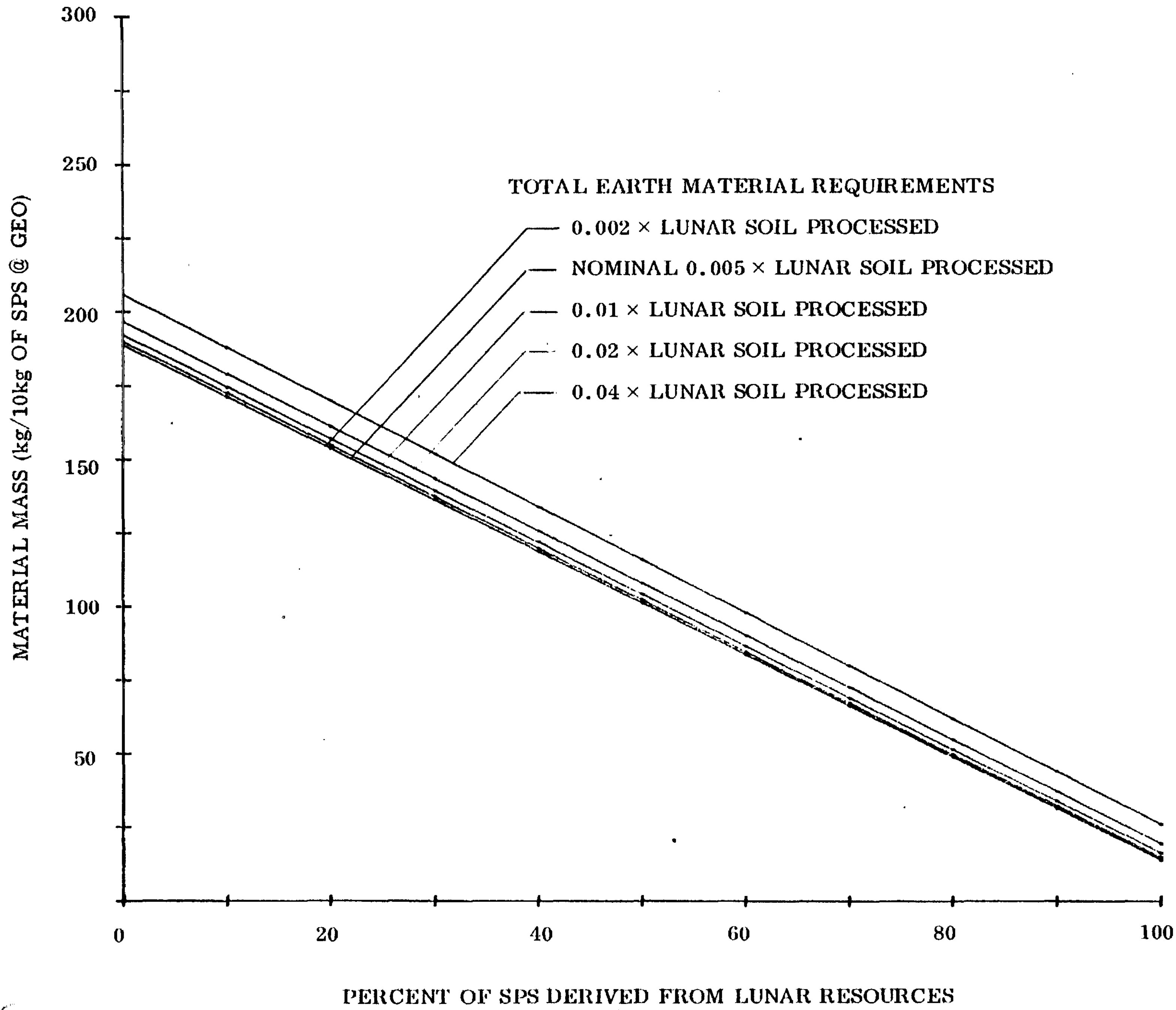


Figure B-7. Option B Alternative with MDRE – Sensitivity to Chemical Loss Fraction

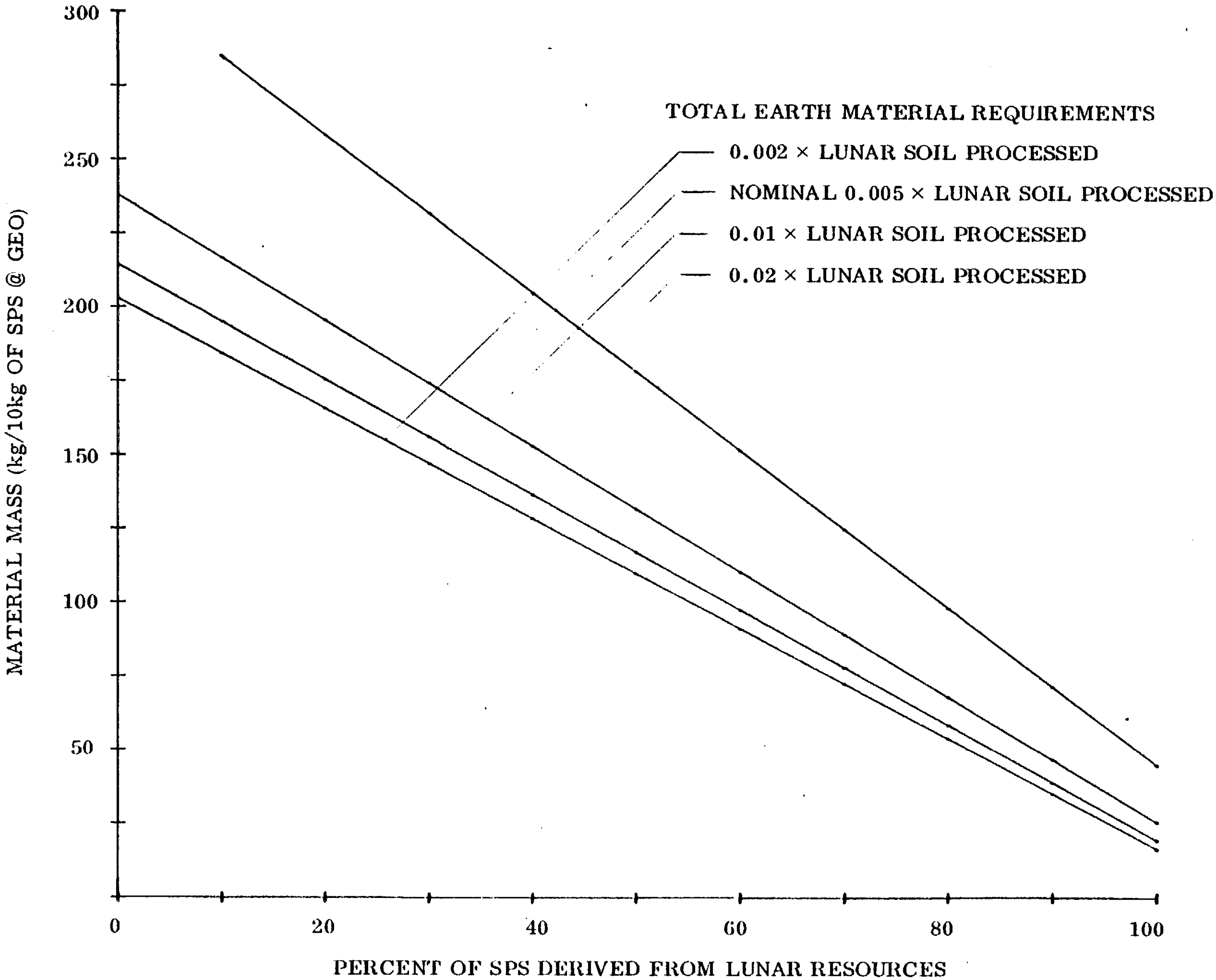


Figure B-8. Option B Alternative with MDRE
LMR Sensitivity to Chemical Loss Fraction

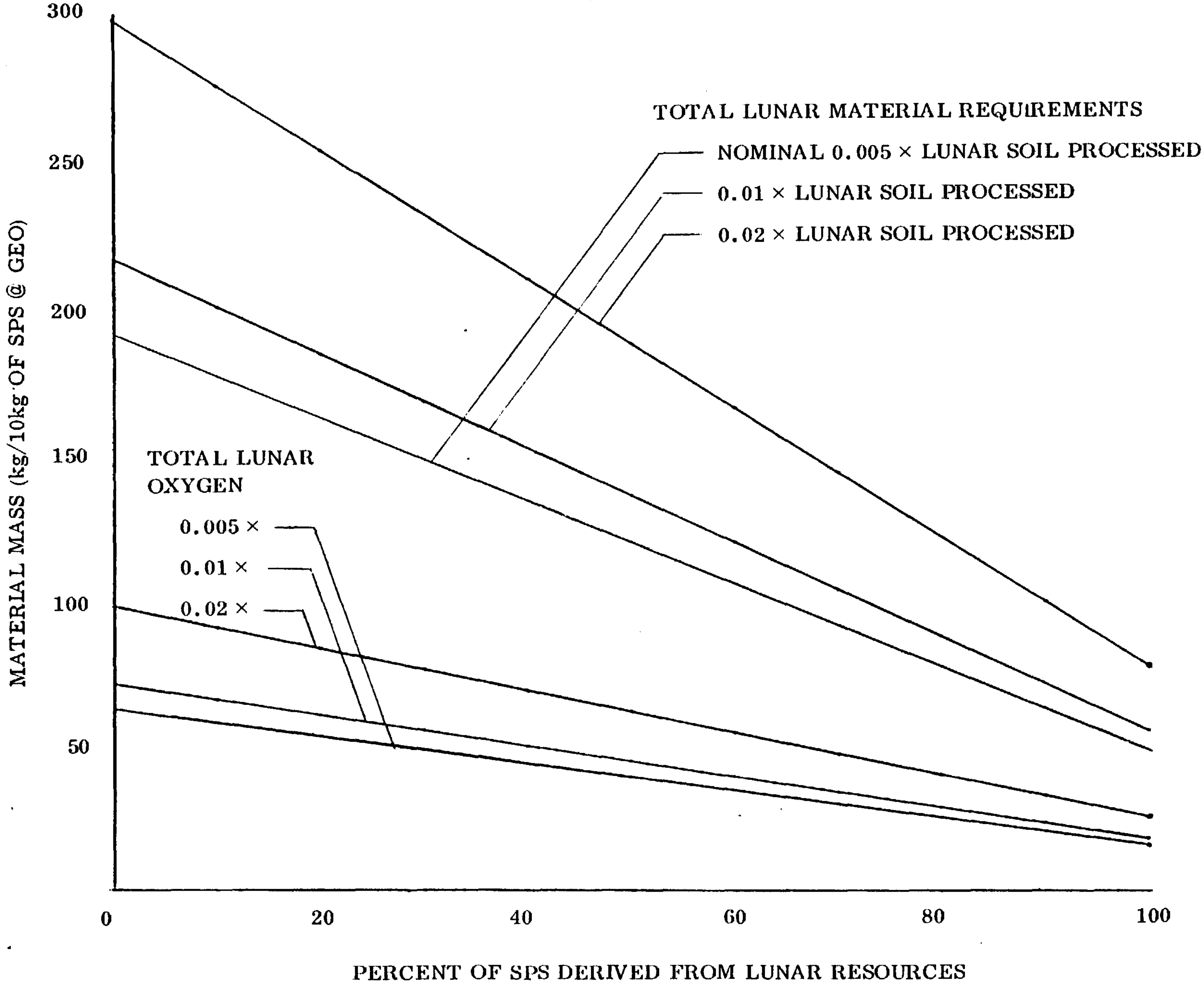


Figure B-9. Option B Revised Baseline with COTV - Sensitivity to Lunar Soil Oxygen Recovery

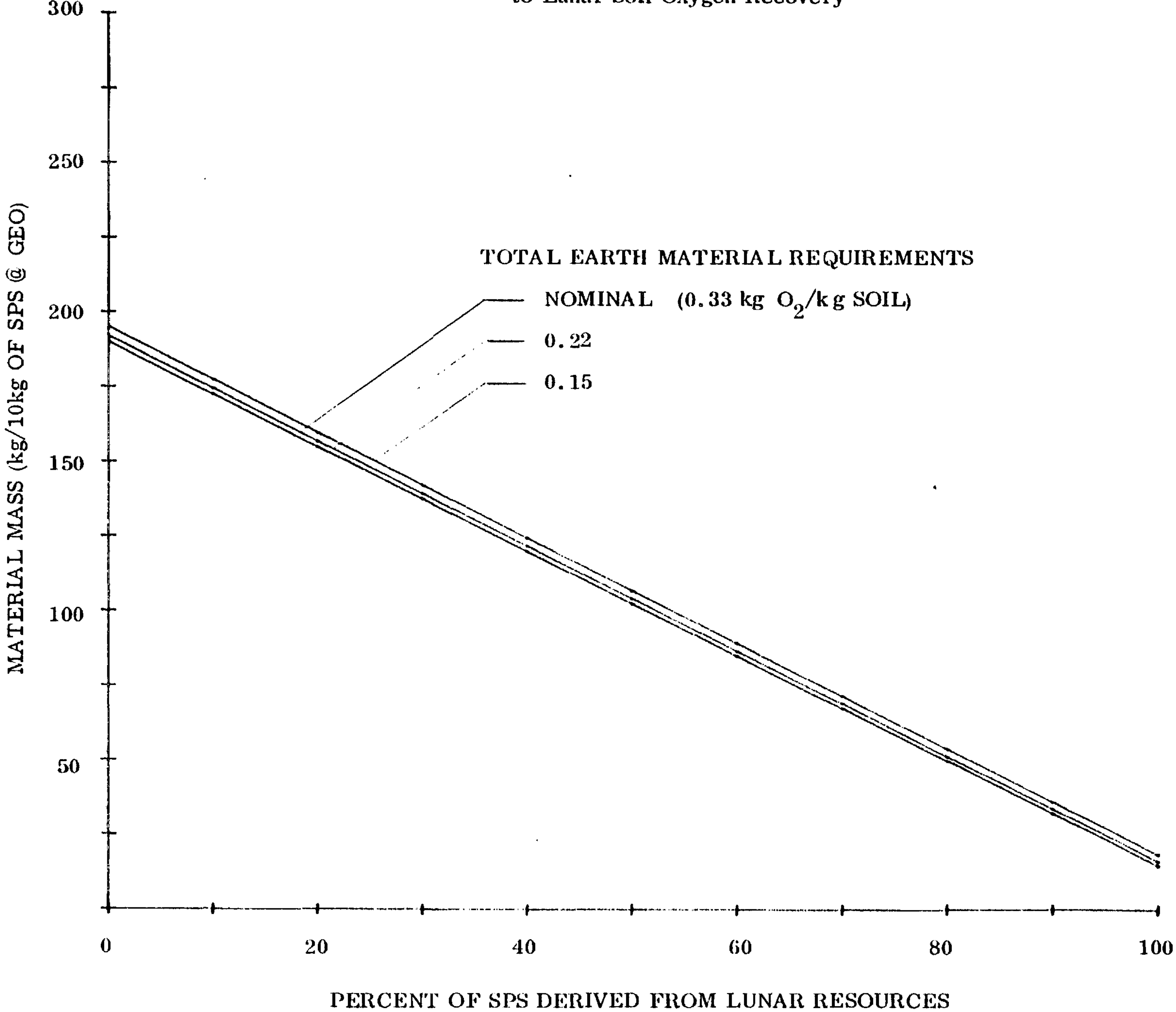


Figure B-10. Option B Alternative with MDRE - Sensitivity to Lunar Soil Oxygen Recovery and Bag Fraction

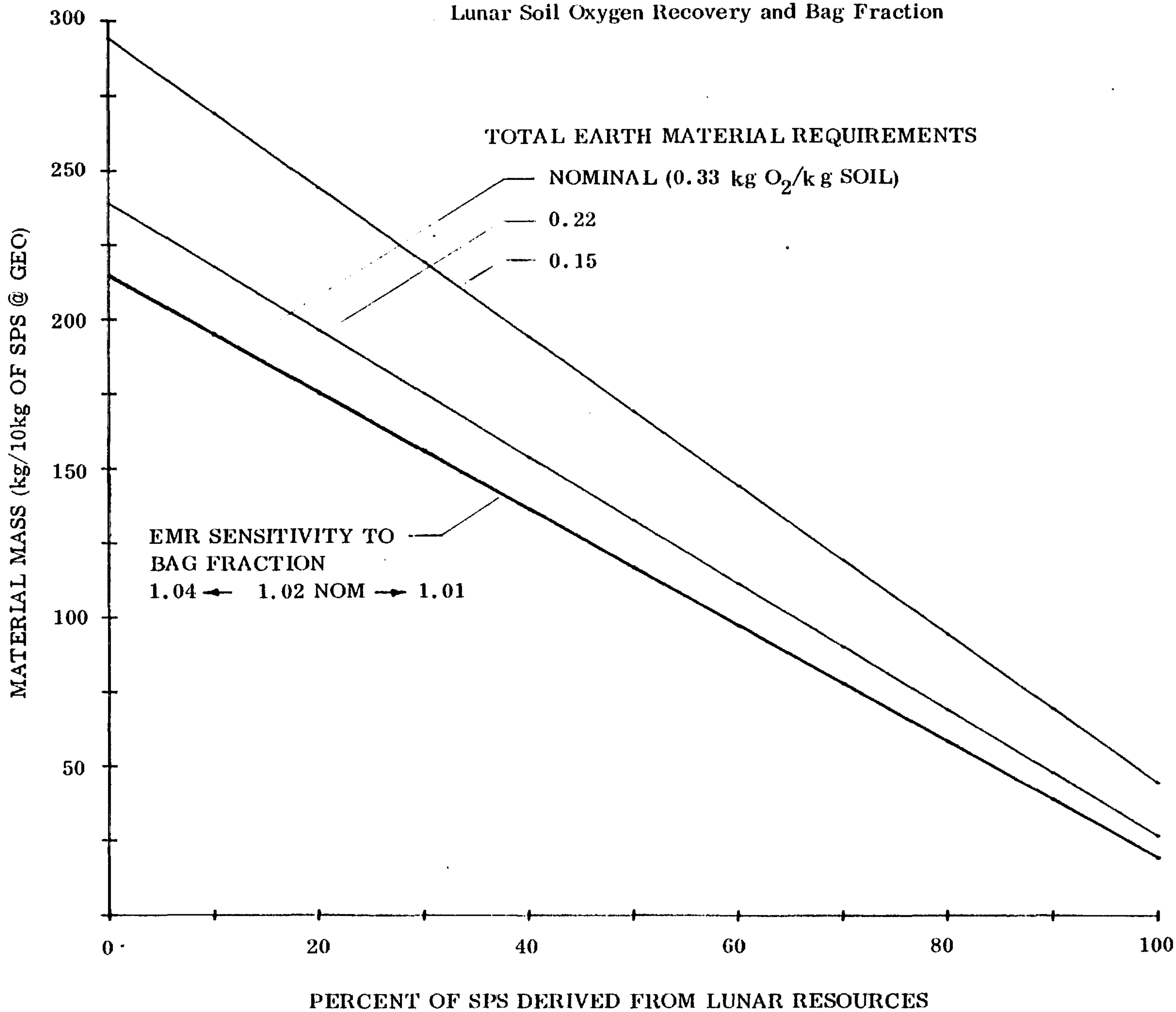


Figure B-11. Option B Revised Baseline with COTV - Sensitivity to COTV Propellant Requirements and Crew Size

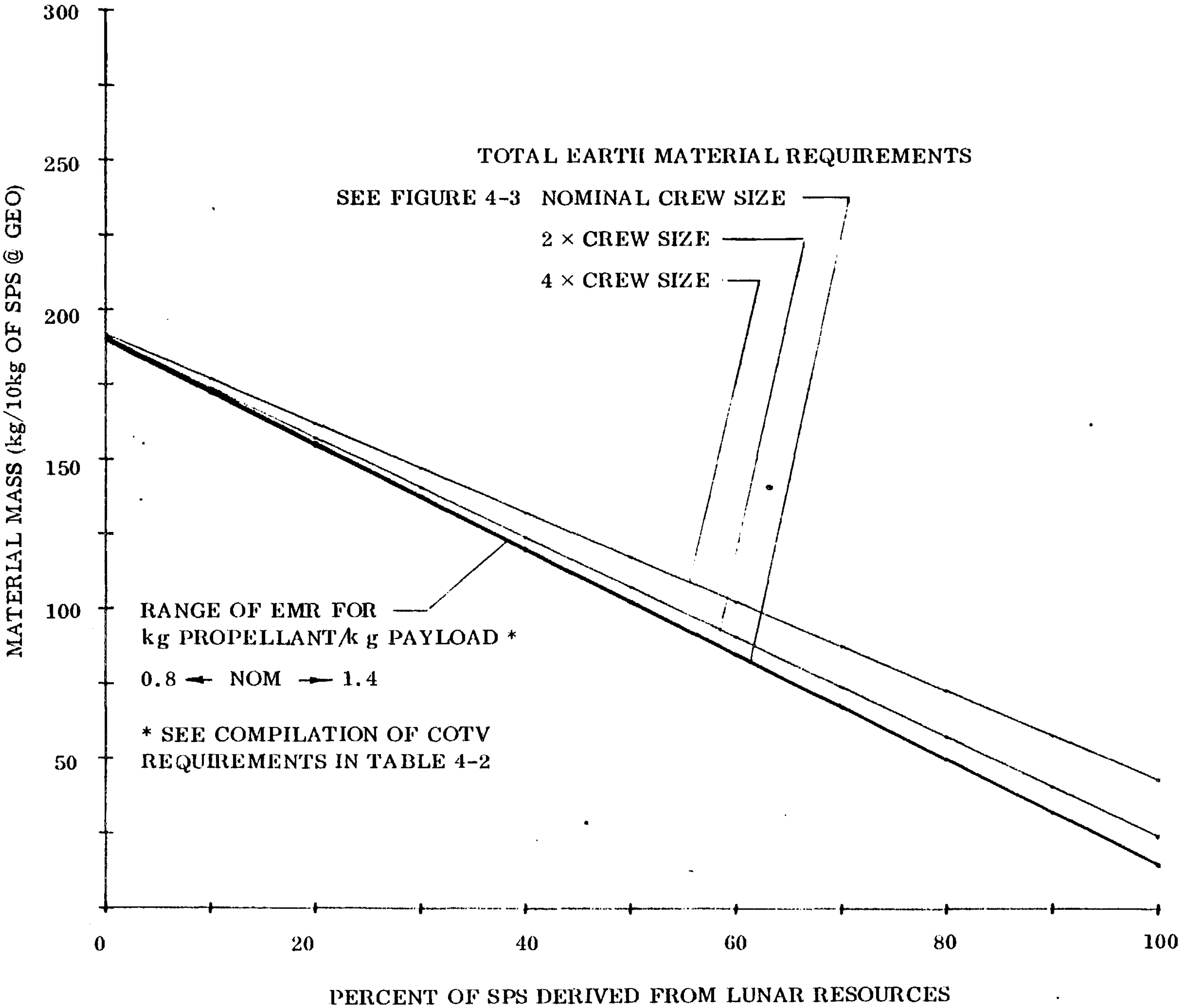


Figure B-12. Option B Alternative with MDRE - Sensitivity to MDRE Propellant Requirements

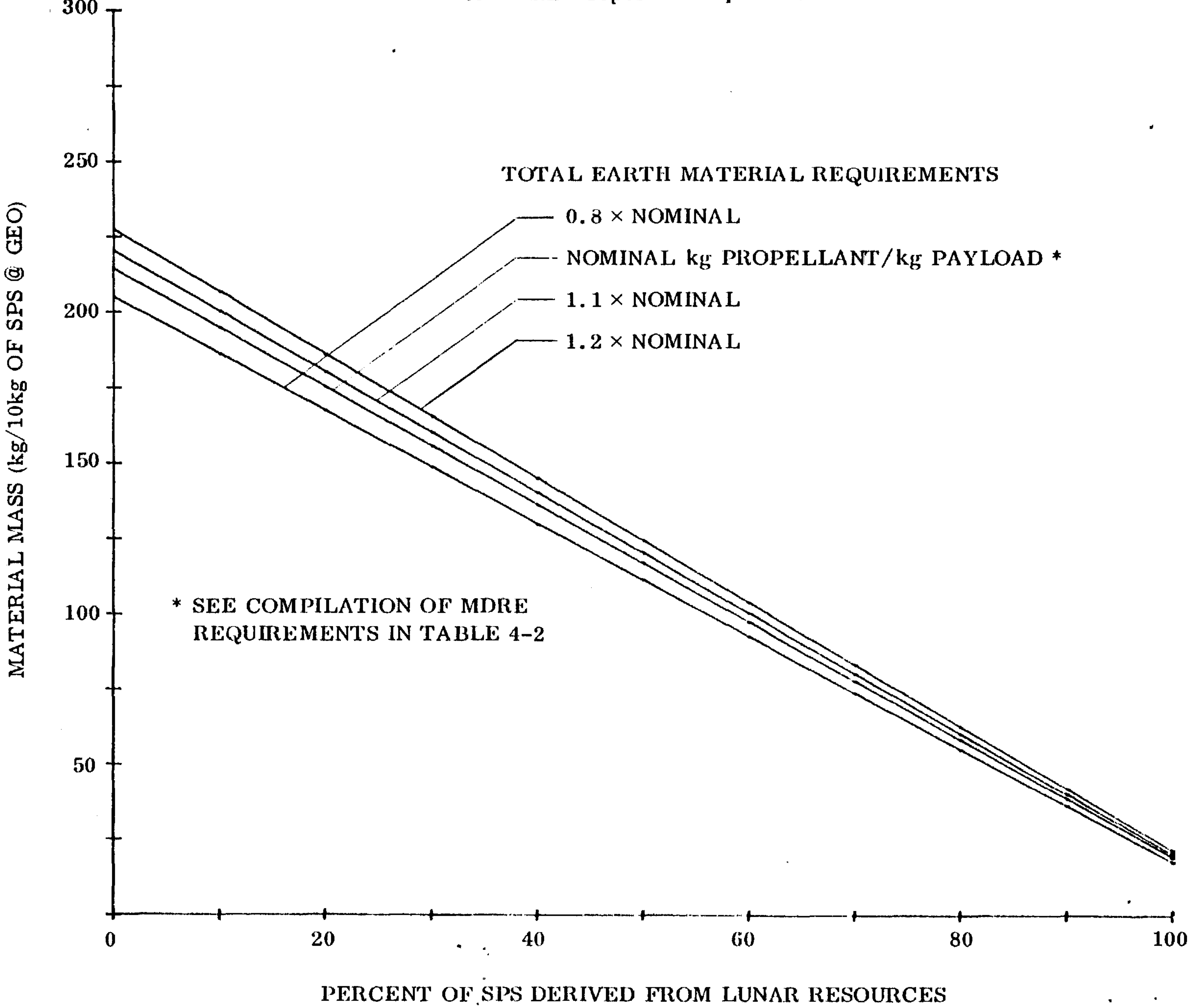


Figure B-13. Option B Revised Baseline with COTV - Sensitivity to Terminal Tug Propellant Requirements

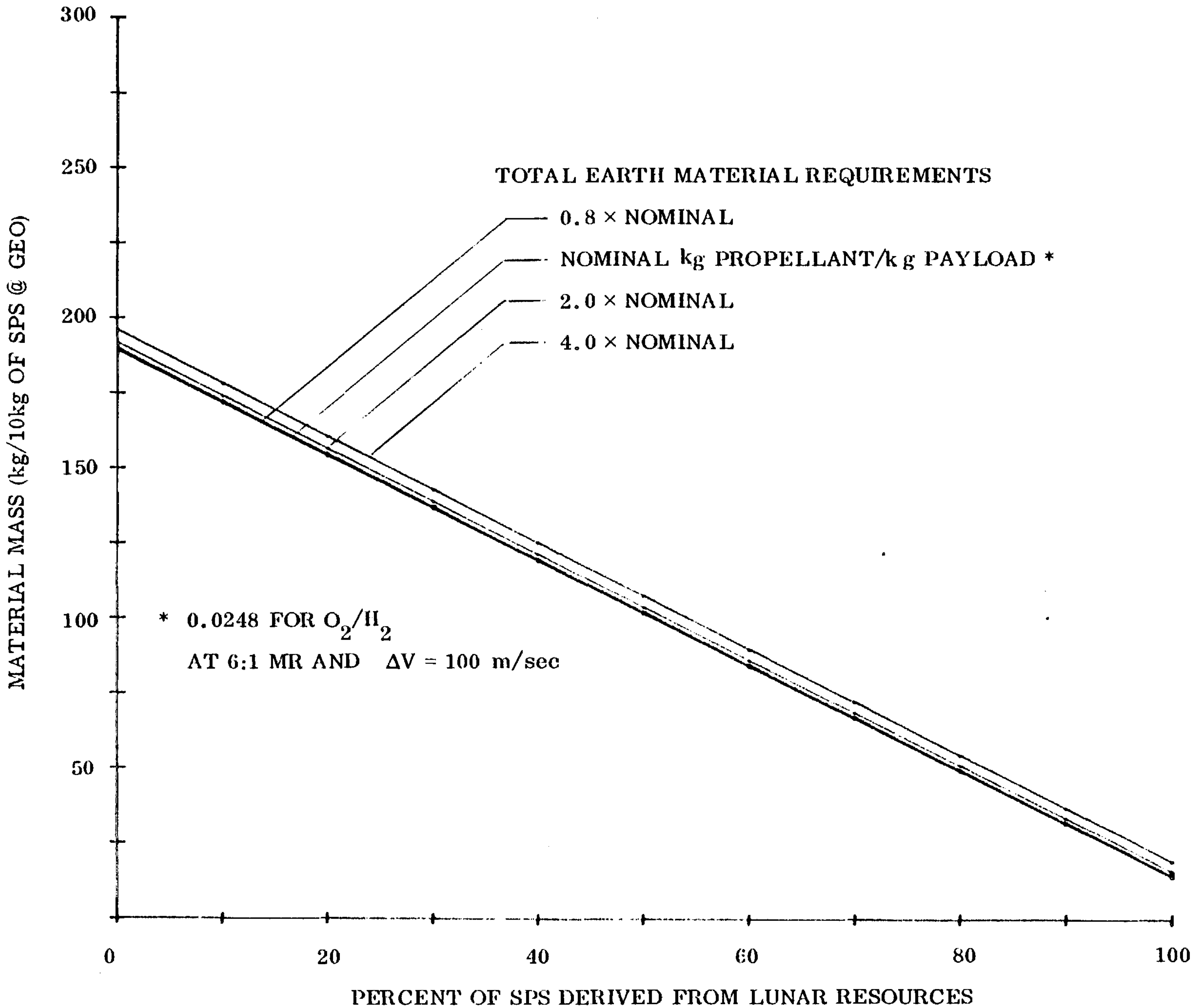
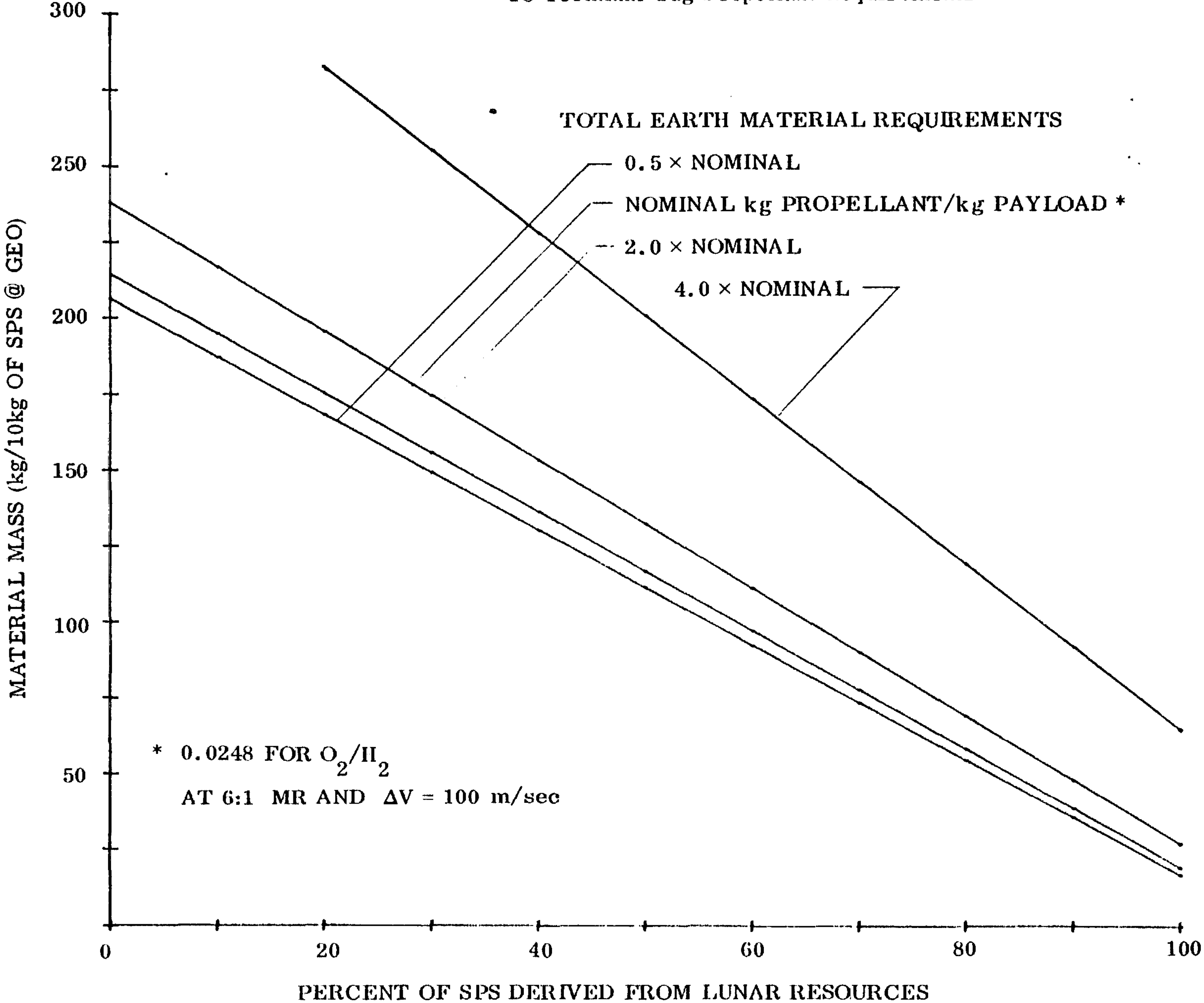


Figure B-14. Option B Alternative with MDRE - Sensitivity To Terminal Tug Propellant Requirements



B.3 CONCEPT C LUNAR LH_2/LO_2 CHEMICAL ROCKET

This systems concept employs conventional LO_2/LH_2 rockets to transport SPS stock materials manufactured at the lunar base into lunar orbit. Since all Concept C transportation routes are serviced by either LO_2/LH_2 chemical rockets or ion electric transfer vehicles, this systems concept exhibits very low technical risk with respect to its transportation elements. The revised version of Concept C is defined in Figure B-15 for the 89.6 percent LRU level. Crew requirements reflect support for the annual production of one 10 GW SPS.

Analysis of the original option C scenario as described in Section B.1 (Figure B-1) has resulted in a revision to the transportation method for delivering lunar manufactured stock material to the GEO fabrication facility. Originally, a large conventional LH_2/LO_2 cargo transfer vehicle (CTV) was assumed for delivery of SPS components directly from the lunar surface to GEO. The revision depicted by Figure B-15 has replaced this single large chemical rocket with two other vehicles:

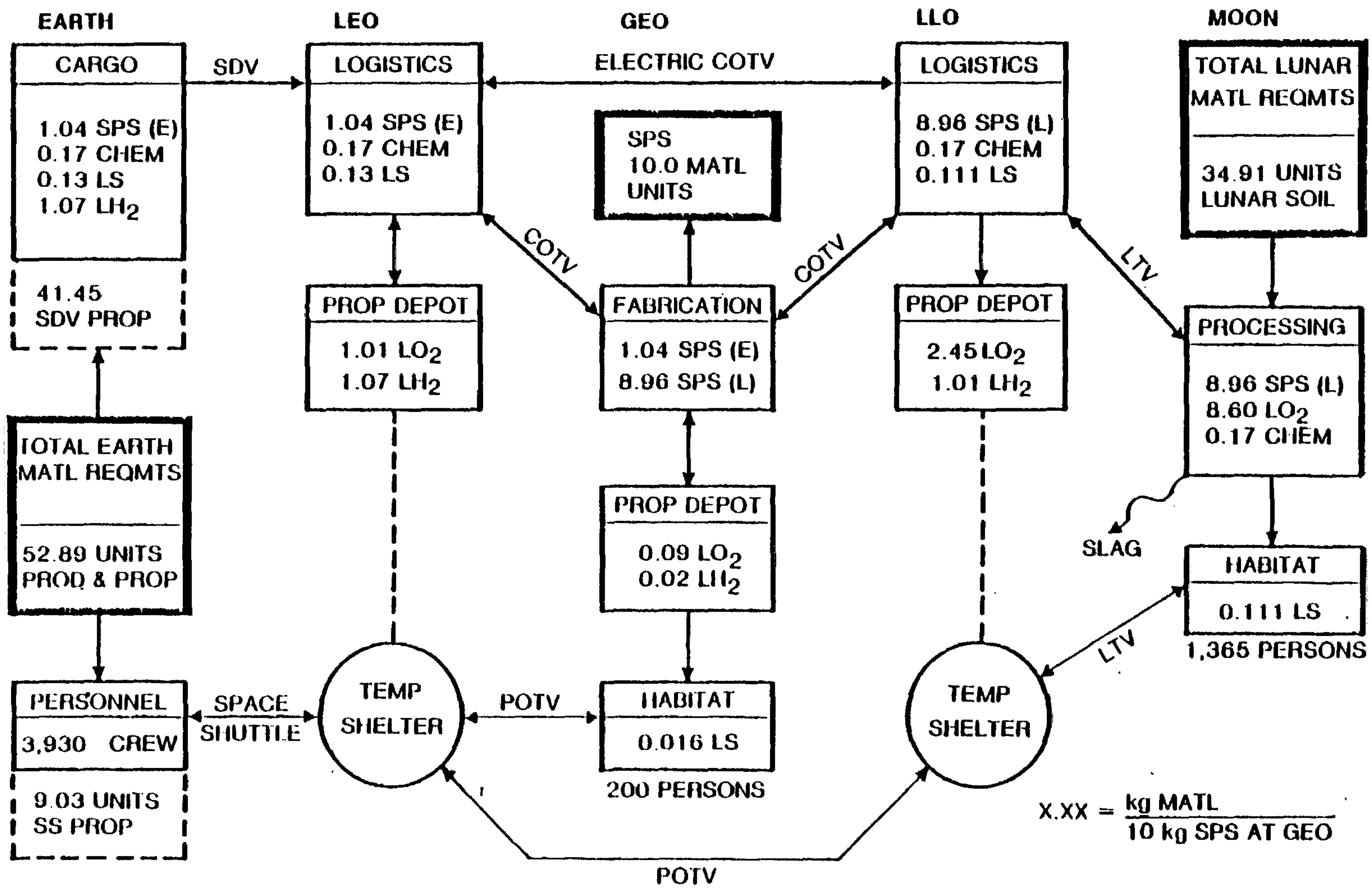
1) A smaller LO_2/LH_2 LTV to deliver SPS components from the lunar surface to LLO.
2) An ion electric COTV using lunar derived oxygen propellant to deliver the components from LLO to GEO.

This revision provides a significant transportation performance improvement, and requires less earth supplied hydrogen and lunar supplied oxygen.

Figure B-15 shows that 52.89 total earth material units, consisting of 2.41 units of payload plus SDV propellant must be launched from earth to construct 10 units of SPS and deliver it to geosynchronous orbit.

Sensitivity to Lunar Resource Utilization (LRU) Percentage - Figure B-16 depicts this sensitivity information for revised Concept C and identifies the relative effects of major mass contributors to total EMR and LMR. The total earth material requirement is primarily SDV propellant. The total lunar material requirement is dependent on the total quantity of oxygen needed, which nominally requires that three

Figure B-15. LRU Concept C - LO_2/LH_2 Lunar Transfer Vehicle.



times this amount of lunar soil be processed. A sufficient quantity of all other lunar derived materials are contained within the soil processed for oxygen recovery. Most of the lunar oxygen is used for delivery of SPS stock materials from the lunar surface to the SMF, which is assumed to be coincidently located to the SPS final assembly and use location in GEO. Some lunar oxygen is recombined with silicon to provide high quality silica glass for SPS solar cell covers and substrate.

Comparison of Alternative LLO to GEO Transfer Techniques - As previously mentioned, the original option C scenario projected use of a single large LO_2/LH_2 LTV to transport lunar products directly to GEO. Since this technique was found to result in rather high EMR and LMR values at higher lunar resource utilization levels, two alternative transport techniques were evaluated. The first of these consisted of two smaller optimized LO_2/LH_2 stages; one from the lunar surface to LLO, the other from LLO to GEO. The second alternative consisted of the smaller chemical LTV for the lunar surface to LLO leg, and an ion electric COTV for LLO to GEO transfer. The effect of these three lunar material/component delivery techniques is graphically displayed in Figure B-17. From the data shown, the rationale for selecting technique 3, which includes the electric COTV, as the Concept C revised baseline is obvious.

Sensitivity to Lunar Oxygen Recovery - Figure B-18 reflects EMR and LMR sensitivity to the percentage of oxygen which can be efficiently extracted from lunar soil. Nominally lunar soil consists of approximately 44 percent oxygen. Variations from the assumed 33 percent recovery (75% extraction efficiency) result in significant LMR sensitivity and minor EMR sensitivity.

Sensitivity to Chemical Loss Fraction - Figure B-19 shows the effect of increased processing chemical loss (the inability to recover earth chemicals during lunar material processing). The EMR are extremely sensitive to process chemical losses, while LMR are relatively insensitive, since the only LMR requirement is for additional propellant.

The EMR sensitivity is due to the fact that processing chemicals make up a non-trivial percentage (nominally 8 percent) of the earth launched cargo. Increases in chemical requirements significantly impact SDV launch requirements and thus total EMR.

Sensitivity to Ion Electric Propulsion Efficiency - Figure B-20 shows the effect of COTV propulsion efficiency on EMR and LMR. Since COTV propellant is assumed to be lunar oxygen, the LMR sensitivity to oxygen propellant requirements is significant, while the EMR sensitivity is somewhat less. The EMR effect is due to the effect on processing chemical requirements.

Sensitivity to Ion Electric COTV Propellant Type and Source - Figure B-21 shows EMR and LMR sensitivity for argon COTV propellant supplied from earth, and oxygen derived from lunar materials at the lunar base. Although argon provides a ~8 % improvement in COTV performance, the EMR is significantly reduced when lunar oxygen is used as propellant. The EMR reduction is due to lower earth launch requirements. The LMR increases for oxygen use, since additional processing of lunar soil is required to produce the oxygen propellant.

Sensitivity to Transfer ΔV Requirements - The introduction to Section 4.2 discusses the difference between low thrust/weight transfer vehicle ΔV requirements for large area payloads and point mass payloads. These differences are identified in Table 4-2 by the performance values given for the ion electric COTV and MDRE, respectively. Figure B-22 compares the effect this difference has on EMR and LMR, assuming that electric COTV's are used for all low g transfer legs as shown in Figure B-15. EMR sensitivity is low and LRU is slightly greater, both due to decreased oxygen processing and processing chemical requirements for the point mass ΔV requirements.

Sensitivity to SDV Propellant Requirements - Figure B-23 indicates the high degree of EMR sensitivity to the quantity of Shuttle derived vehicle propellants needed to lift payload into LEO. Since all SDV propellants are obtained from earth, there is no effect on LMR.

Sensitivity to Lunar Transfer Vehicle Efficiency - Figure B-24 depicts total EMR and LMR sensitivity to the lunar transfer vehicle propellant requirements. A variation in the quantity of LH_2/LO_2 propellants required per kg of SPS stock materials delivered from the lunar surface to GEO, results in small variations of EMR and LMR for high LRU percentages.

Sensitivity to Life Support Requirements (LS) - Figure B-25 shows that should life support requirements quadruple, the EMR at 100% LRU only increases by 20 percent, and LMR is unaffected.

Sensitivity to Personnel Assignment Duration - Figure B-26 depicts the sensitivity to variations in personnel assignments at the GEO assembly facility and lunar mining and processing base. Propellant must be expended to return personnel to earth and transport replacement personnel from earth to their work stations. The nominal assumed stay times are 60 days at GEO and 180 days on the lunar surface. Variations in these durations result in a modest EMR sensitivity, and insignificant LMR sensitivity.

Figure B-16. Option C Revised Baseline (LO_2/LH_2 LTV from Lunar Surface to LLO and Ion COTV from LLO to GEO)

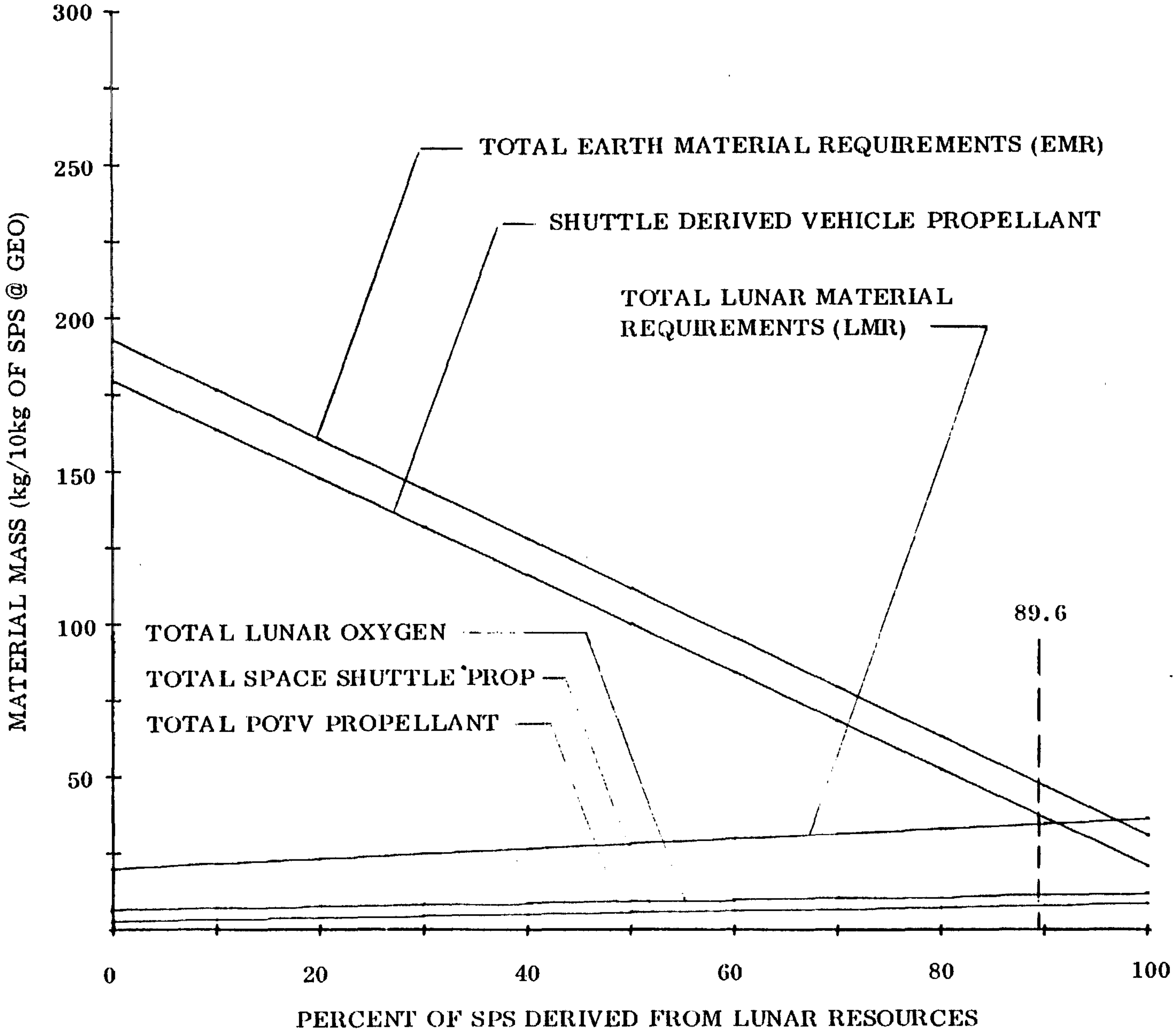


Figure B-17. Option C - Comparison of LLO to GEO Cargo Transfer Techniques

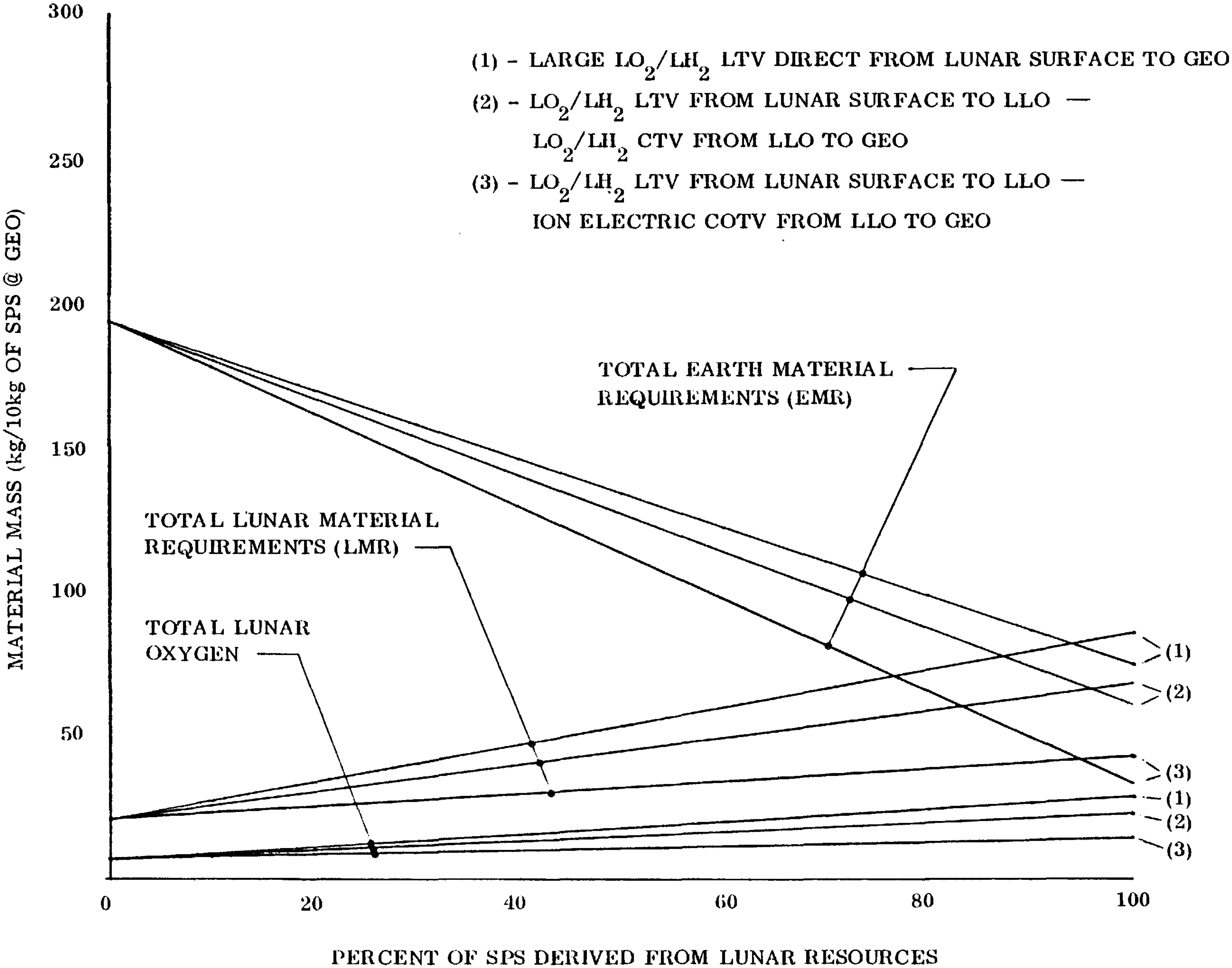


Figure B-18. Option C Revised Baseline - Sensitivity to Lunar Oxygen Recovery

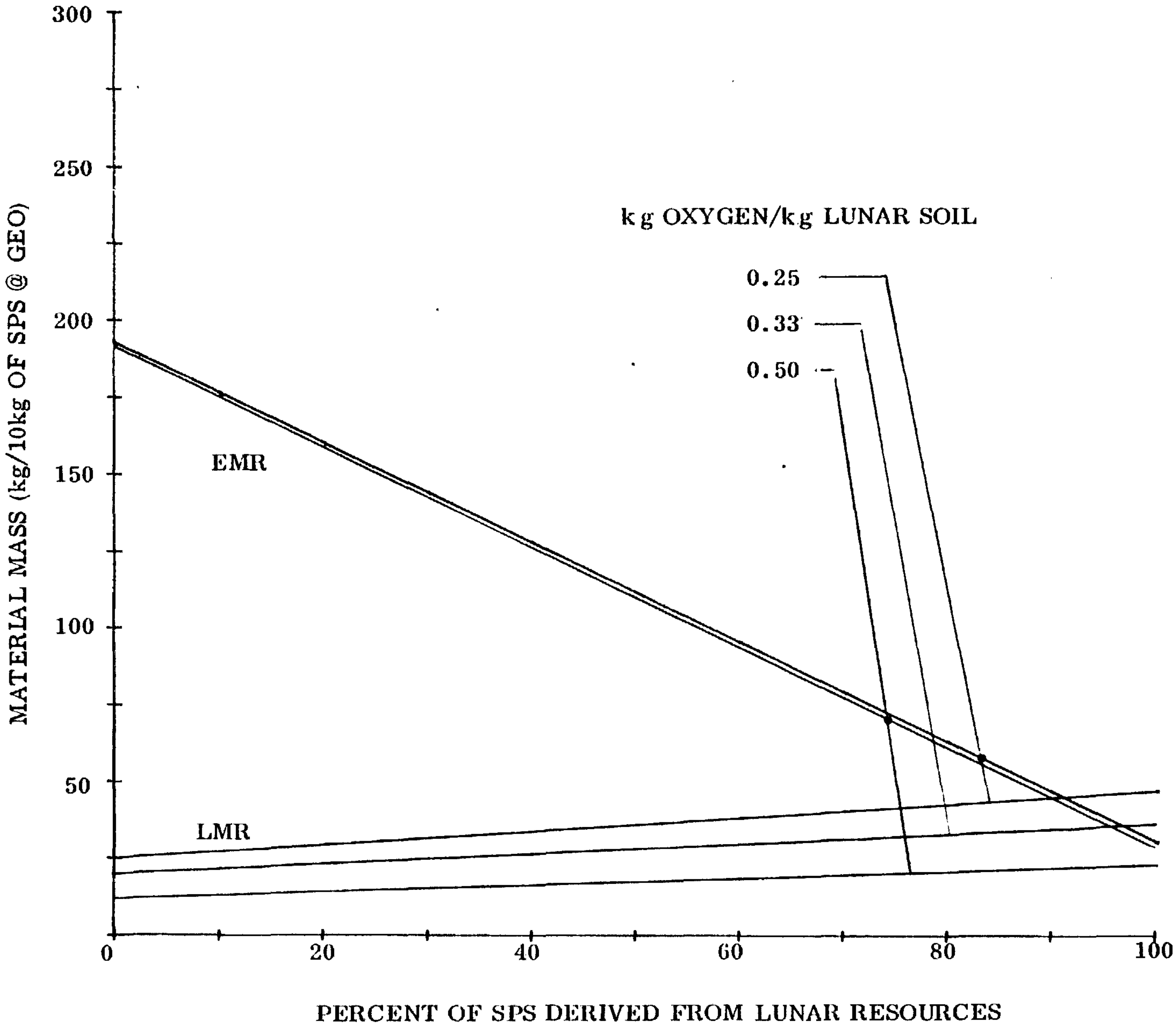


Figure B-19. Option C Revised Baseline - Sensitivity to Chemical Loss Fraction

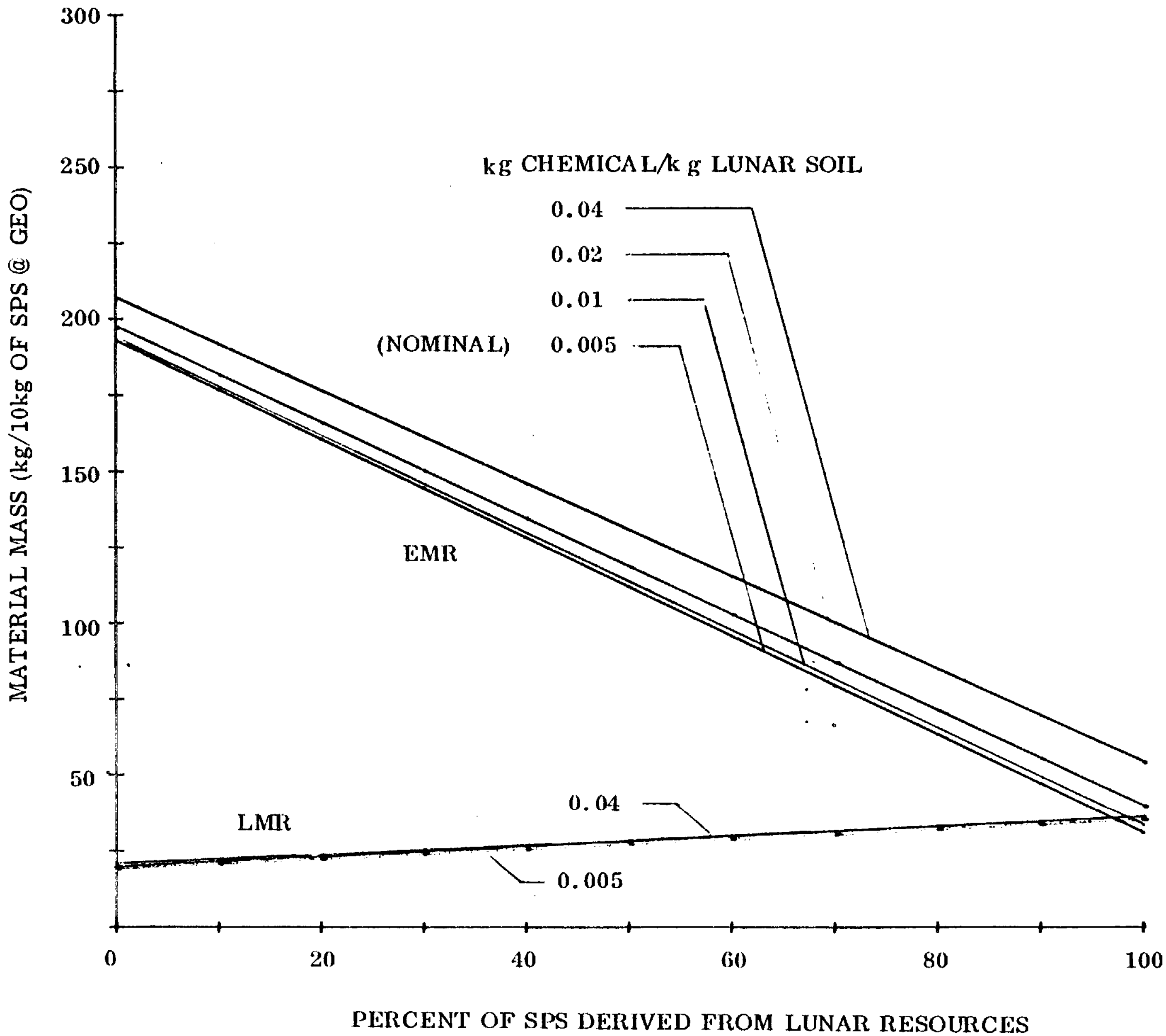
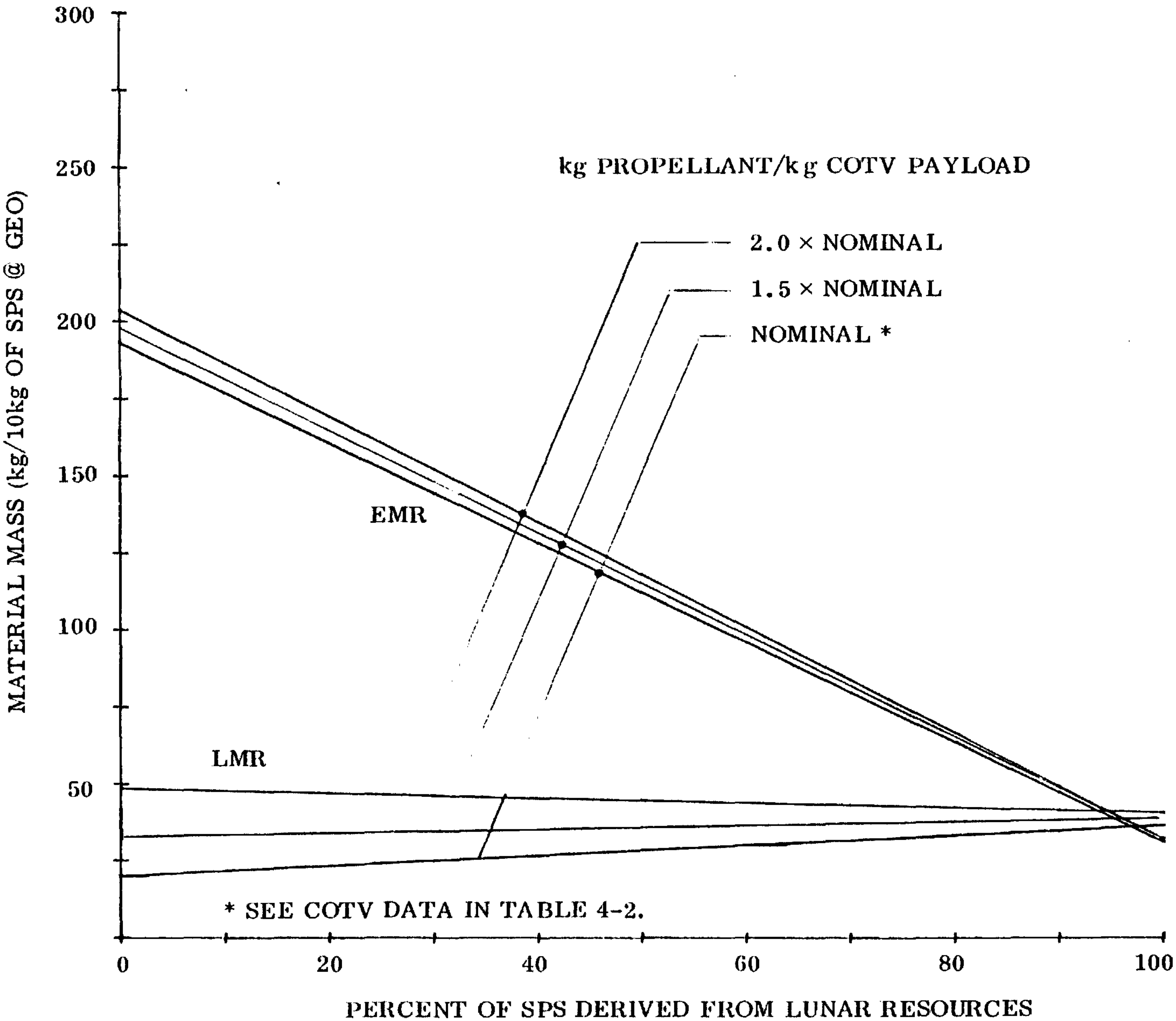


Figure B-20. Option C Revised Baseline - Sensitivity to Ion Electric Propulsion Efficiency



FigureB-21. Option C Revised Baseline - Sensitivity
Propellant Type and Source for Ion Electric COTV

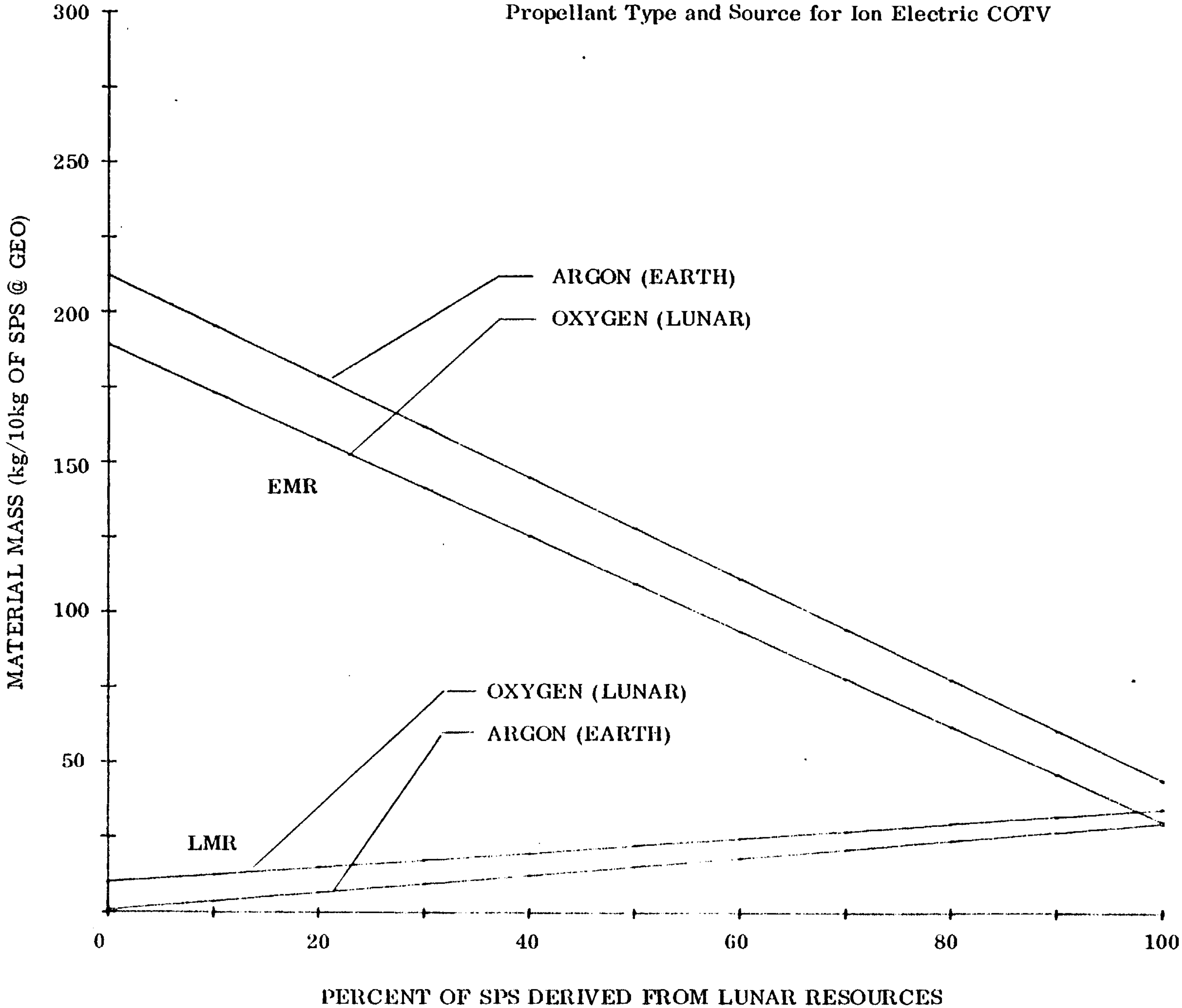


Figure B-22. Option C Revised Baseline - Sensitivity to Transfer ΔV Requirements

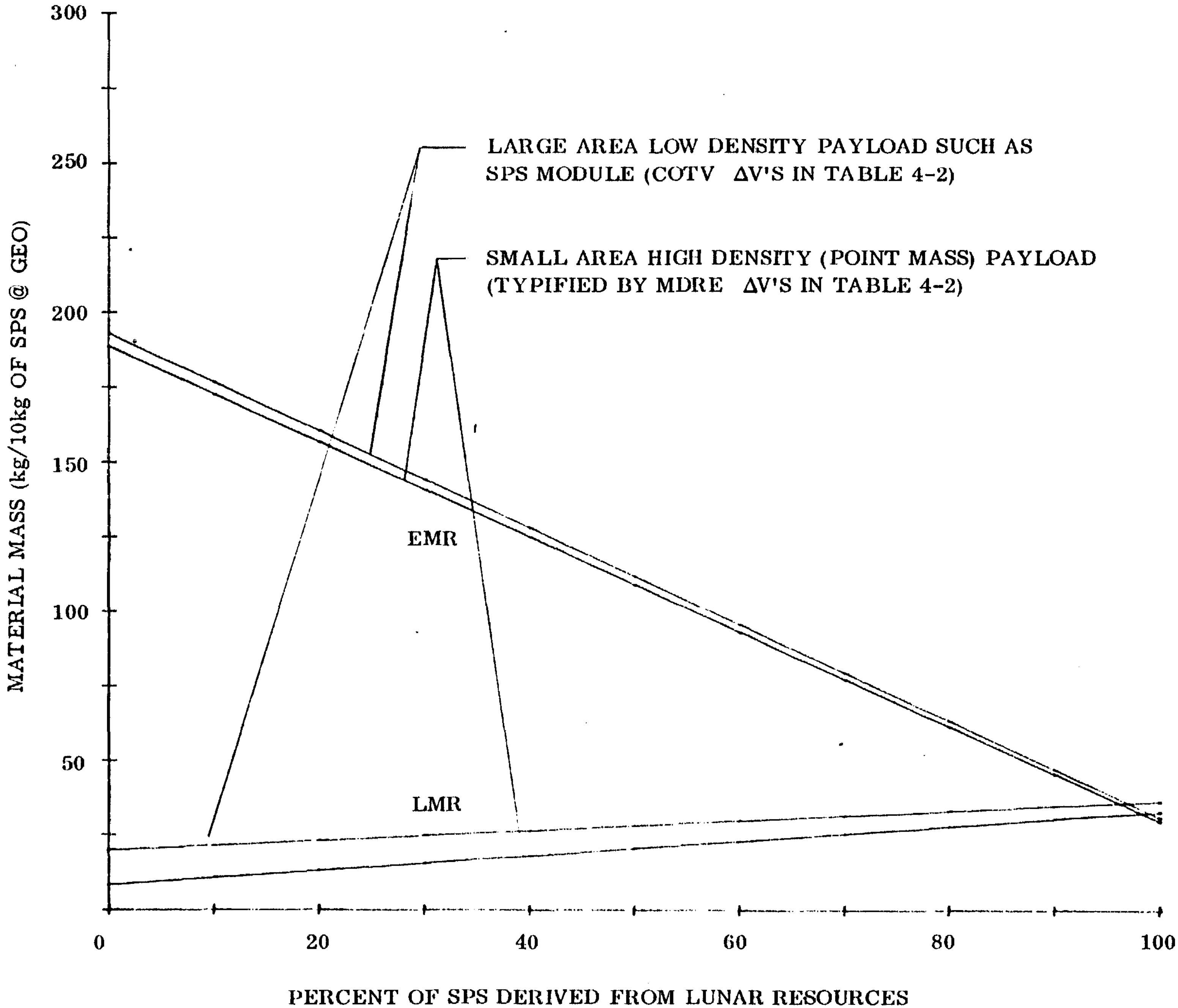


Figure B-23. Option C Revised Baseline - Sensitivity to SDV Propellant Requirements

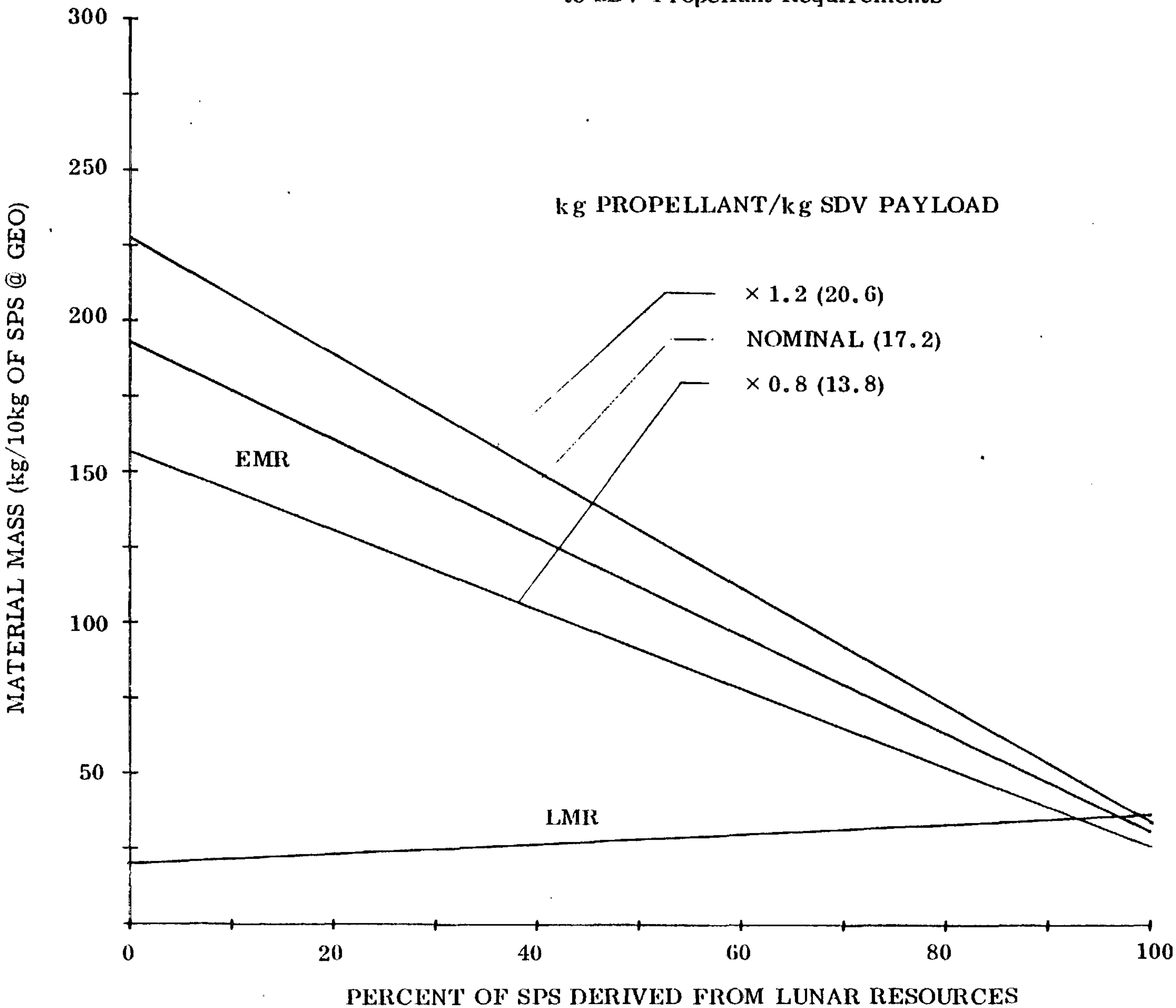


Figure B-24. Option C Revised Baseline - Sensitivity to Lunar Transfer Vehicle Efficiency

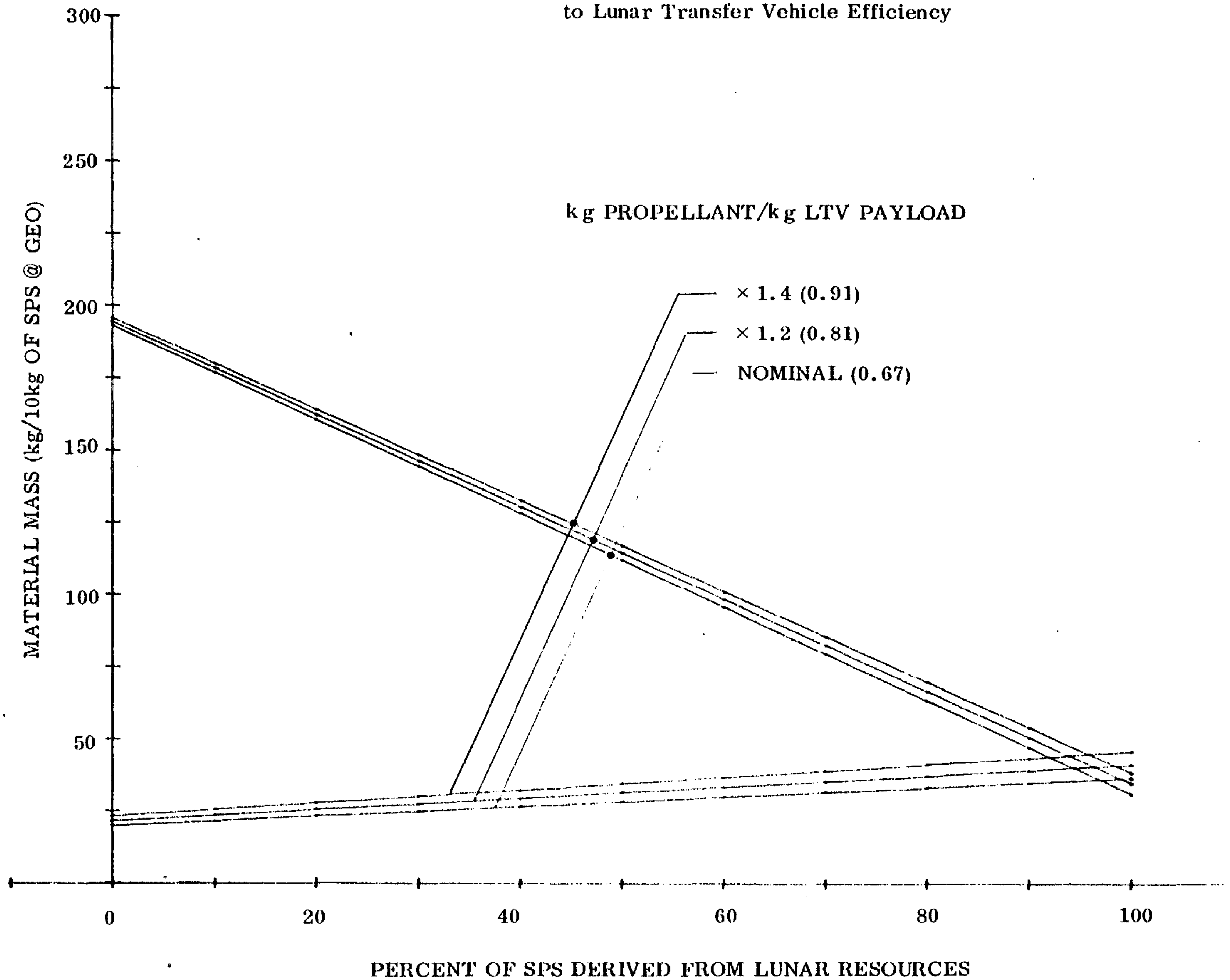


Figure B-25. Option C Revised Baseline - Sensitivity to Life Support Requirements

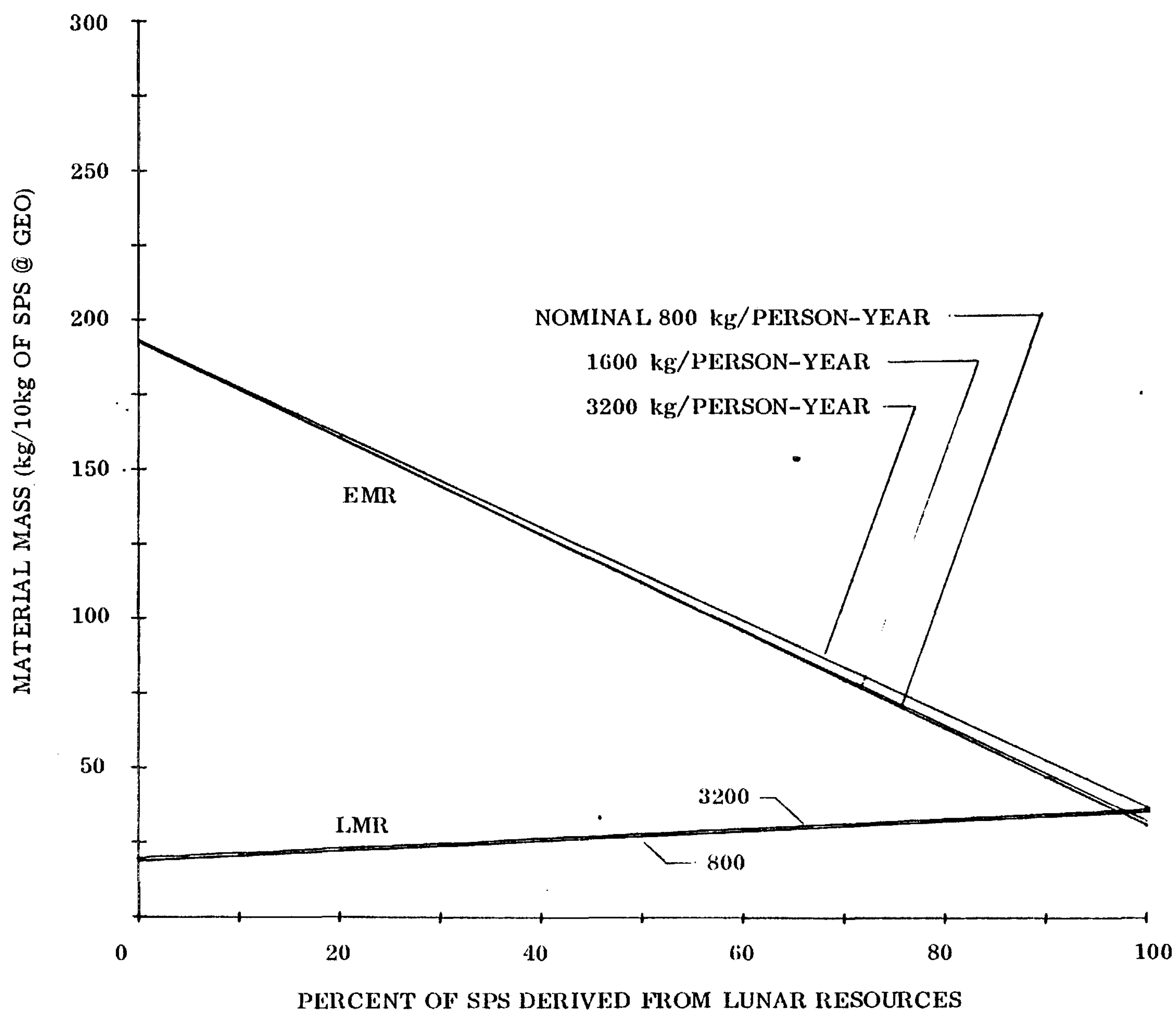
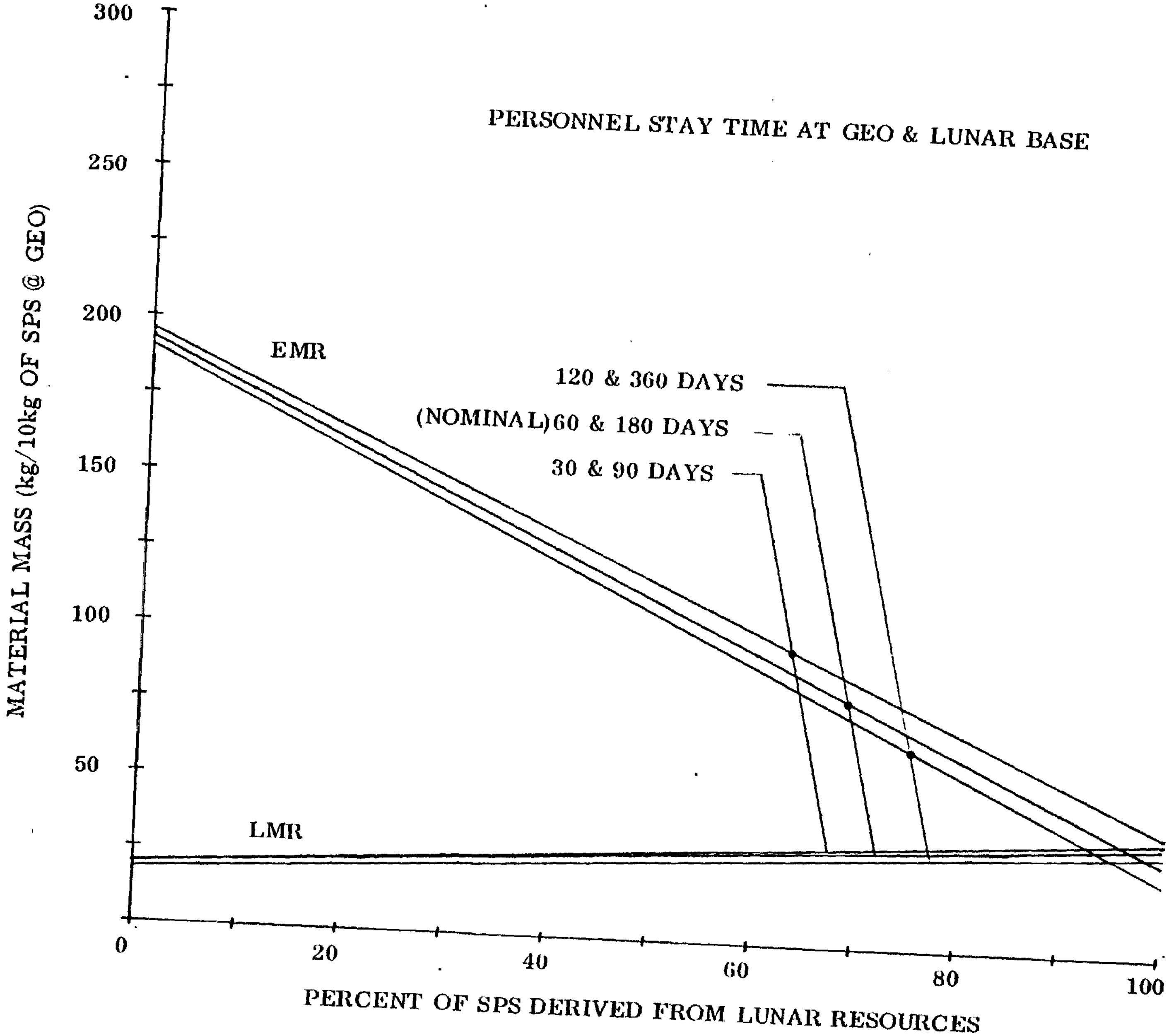


Figure B-26. Option C Revised Baseline - Sensitivity to Personnel Assignment Duration



B.3 CONCEPT D - LUNAR DERIVED ROCKET

Systems Concept D is similar to Concept C except for the vehicle used to transfer construction materials from the lunar surface to low lunar orbit. The LTV has been revised from the LH_2/LO_2 chemical rocket used in Concept C, to a chemical rocket which derives all its propellants (fuel and oxidizer) from lunar materials. This revision reduces considerably the quantity of hydrogen which must be supplied from earth. The baseline all lunar propellant LTV or lunar derived rocket (LDR) uses liquid oxygen as oxidizer and powdered aluminum as fuel. Alternative fuels include mixtures of lunar metals including aluminum, calcium, iron, magnesium, sodium and titanium.

The LDR originally assumed for SPS stock material delivery from the lunar surface to GEO assembly base was a large single stage expendable vehicle (see paragraph B.1). This expendable vehicle is undesirable since extensive fabrication facilities are required at the lunar base to manufacture LDR propellant tanks, and reprocessing facilities are needed in GEO to convert LDR propellant tankage into SPS components. A reusable vehicle for lunar surface to GEO transport of cargo is a more desirable transportation solution. Performance calculations, however, have shown that the lunar derived rocket (LDR) does not have enough specific impulse to make a round trip flight from lunar surface to GEO and back to the lunar base. Therefore, a revised Concept D baseline was developed by replacing the expendable LDR with two other reusable vehicles:

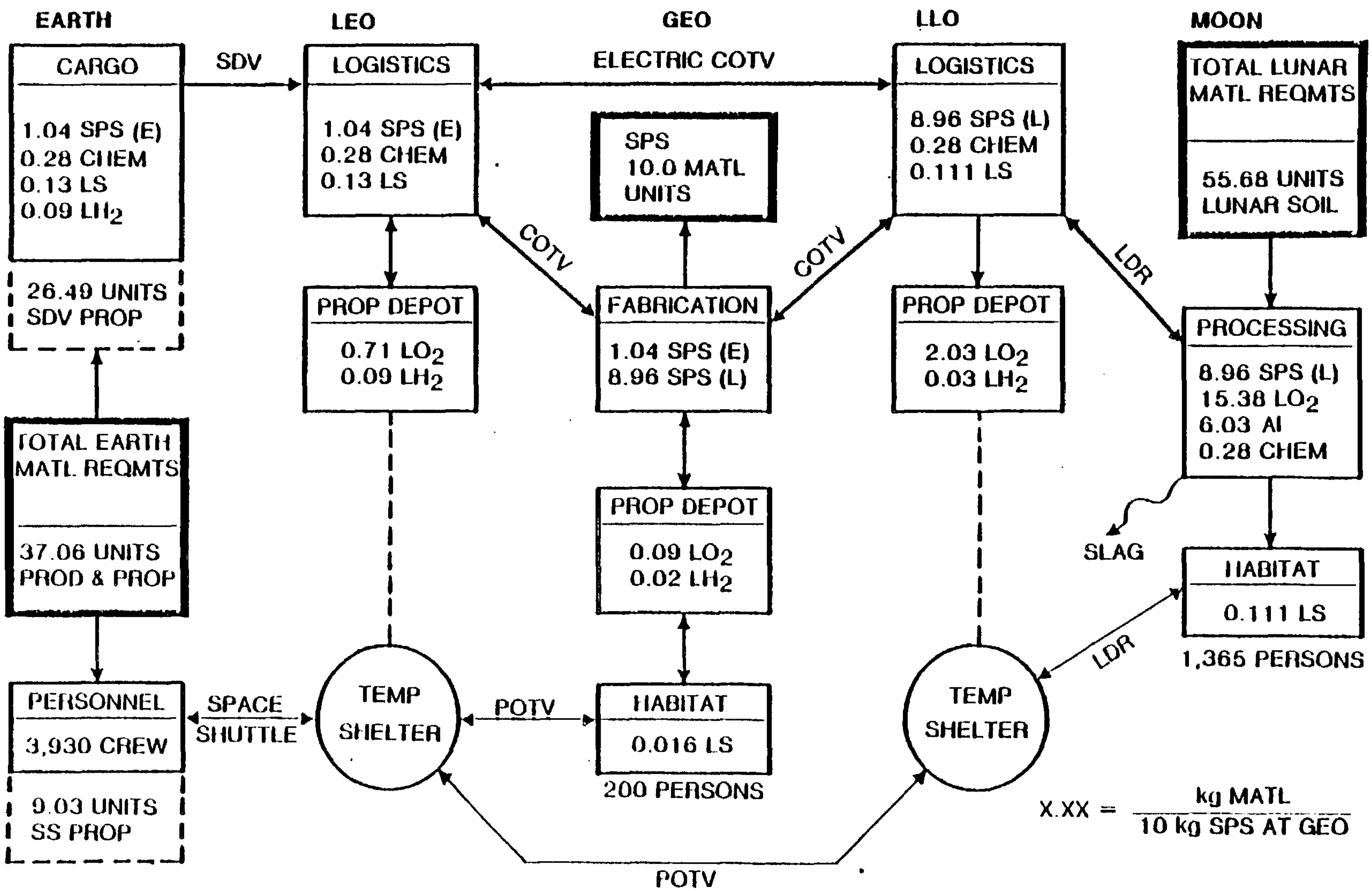
1) A smaller LDR to deliver SPS stock materials from the lunar surface to LLO.
2) An ion electric COTV using lunar derived oxygen propellant to deliver these components from LLO to GEO.

The employment of a reusable LDR reduces manufacturing operations on both the moon (LDR propellant tank construction) and at the GEO assembly facility (tank reprocessing into SPS components), as well as significantly reducing lunar propellant processing requirements. The steady state material flow and personnel requirements for constructing one 10 GW SPS per year is depicted in Figure B-27 for the revised Concept D baseline. This shows that 37.06 total earth material units, consisting of 1.54 units of payload plus SDV propellant must be launched from earth to construct 10 units of SPS and deliver it to geosynchronous orbit.

Baseline Sensitivity to Lunar Resource Utilization (LRU) Percentage - Figure B-28 depicts this sensitivity information and identifies the relative effects of major mass contributors to total EMR and LMR. The total lunar material requirement is dependent on the total quantity of oxygen or aluminum needed, which nominally requires that three times this amount of lunar soil must be processed if oxygen is the controlling requirement, or ten times as much if aluminum controls. These factors assume that 75% of the oxygen or 100% of the aluminum contained in the soil can be successfully extracted. A sufficient quantity of all other lunar derived materials are nominally contained within the soil processed for oxygen or aluminum recovery. Most of the lunar oxygen and aluminum is used as LDR propellant for delivery of SPS materials/ components from the lunar surface to LLO, and the oxygen is also used in the electric OTV for cargo transfer from LLO to the SPS final assembly and use location in GEO.

Comparison of alternative LLO to GEO Cargo Transfer Techniques - As previously mentioned, the original option D scenario projected use of a single expendable lunar derived rocket (LDR) with oxygen/aluminum propellants to transport lunar products directly to GEO. Since this technique was found to be unacceptable, two alternative transport techniques were evaluated. The first of these employed a reusable LDR between the lunar surface and LLO, and used a LH_2/LO_2 orbital transfer vehicle to transport construction materials between LLO and GEO. The second approach was to employ a reusable LDR between the lunar surface and LLO, and use an ion electric COTV (lunar derived oxygen propellant) to transport materials between LLO and GEO. The effect of these two lunar stock material delivery techniques on EMR and LMR is graphically displayed in Figure B-29. From the data shown, the rationale for selecting the second technique, which includes the electric COTV, as the Concept D revised baseline is obvious. Alternative propulsion

Figure B-27. LRU Concept D – Lunar Derived Rocket.



schemes and lunar derived propellants were also considered for the LDR. A discussion of these alternatives and rationale for selection of the O_2/Al pump fed LDR baseline is contained in Appendix E, Section E.4 of Volume III.

Sensitivity to Lunar Oxygen Recovery - Figure B-30 reflects EMR and LMR sensitivity to the percentage of oxygen which can be efficiently extracted from lunar soil. Nominally, lunar soil consists of approximately 44 percent oxygen. Variations from the assumed 33 percent recovery (75% extraction efficiency) result in significant LMR sensitivity and minor EMR sensitivity.

Sensitivity to Lunar Aluminum Recovery - Figure B-31 reflects EMR and LMR sensitivity to the percentage of aluminum which can be efficiently extracted from lunar soil. Nominally, lunar highlands soil consists of approximately 13 percent aluminum. Variations from the assumed 13 percent recovery (100% extraction efficiency) result in very substantial LMR sensitivity and minor EMR sensitivity. By comparing the data in Figures B-30 and B-31 it appears that below 100% aluminum extraction efficiency (0.13 kg aluminum/kg lunar soil), aluminum becomes the controlling extraction requirement for Concept D. This is discussed further in Sections 4.4 and 4.7 of Volume II.

Sensitivity to Chemical Loss Fraction - Figure B-32 shows the effect of increased processing chemical loss (the inability to recover earth chemicals during lunar material processing). The EMR are extremely sensitive to process chemical losses, while LMR are relatively insensitive, since the only LMR requirement is for additional oxygen and/or aluminum propellants.

The EMR sensitivity is due to the fact that processing chemicals make up a non-trivial percentage (nominally 18 percent) of the earth launched cargo. Increases in chemical requirements significantly impact SDV launch requirements and thus total EMR.

Sensitivity to Ion Electric Propulsion Efficiency - Figure B-33 shows the effect of COTV propulsion efficiency on EMR and LMR. Since COTV propellant is assumed to be lunar oxygen, the LMR sensitivity to oxygen propellant requirements is significant, while the EMR sensitivity is negligable.

Sensitivity to Lunar Derived Rocket Efficiency - Figure B-34 depicts total EMR and LMR sensitivity to the lunar derived rocket vehicle propellant requirements from lunar surface to LLO transfer. A variation in the quantity of LO_2/aluminum powder propellants required per kg of SPS stock materials delivered from the lunar surface to LLO, results in slight sensitivity of LMR for high LRU percentages, but has almost no effect on EMR.

Sensitivity to Life Support Requirements (LS) - Figure B-35 shows that should life support requirements quadruple, the EMR at 100% LRU increases by 27 percent, and LMR is unaffected.

REFERENCES

1. Cowgill, Lane, "Low Acceleration Transfers from Low Earth Orbit to Low Lunar Orbit - Analysis for Lunar Resources Utilization Study", General Dynamics Convair Division memo 697-0-78-070, Draft Copy, 9 August 1978. Included as Appendix E.1 of Volume III.
2. Woodcock, G. R., et. al., "Future Space Transportation Systems Analysis Study." Contract NAS9-14323, Boeing Aerospace Company Report D180-20242-3, December 31, 1976.

Figure B-28. Option D Revised Baseline
(LDR from Lunar Surface to LLO and
Ion Electric COTV From LLO to GEO)

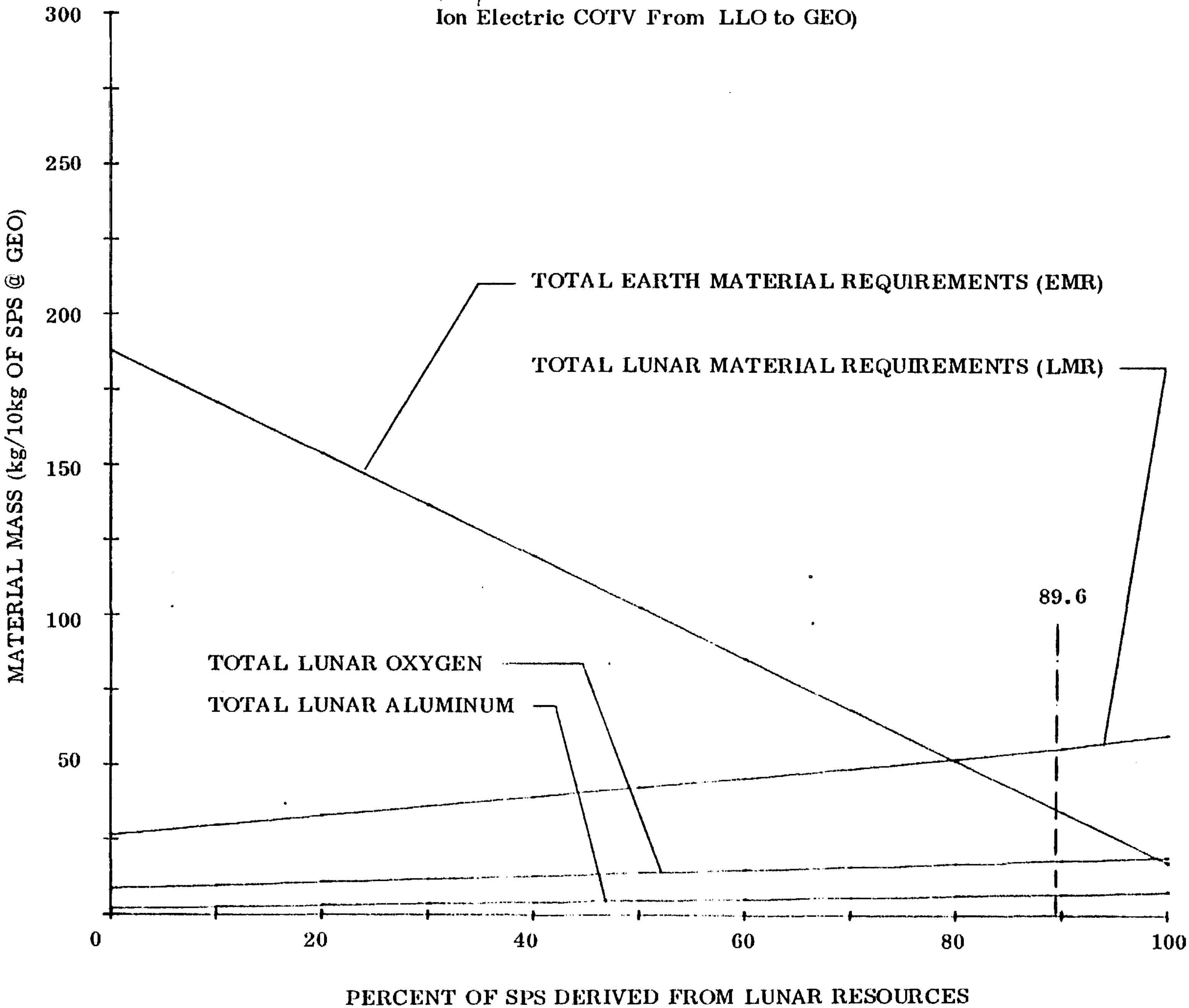


Figure B-29. Option D Comparison of LLO to GEO Cargo Transfer Techniques

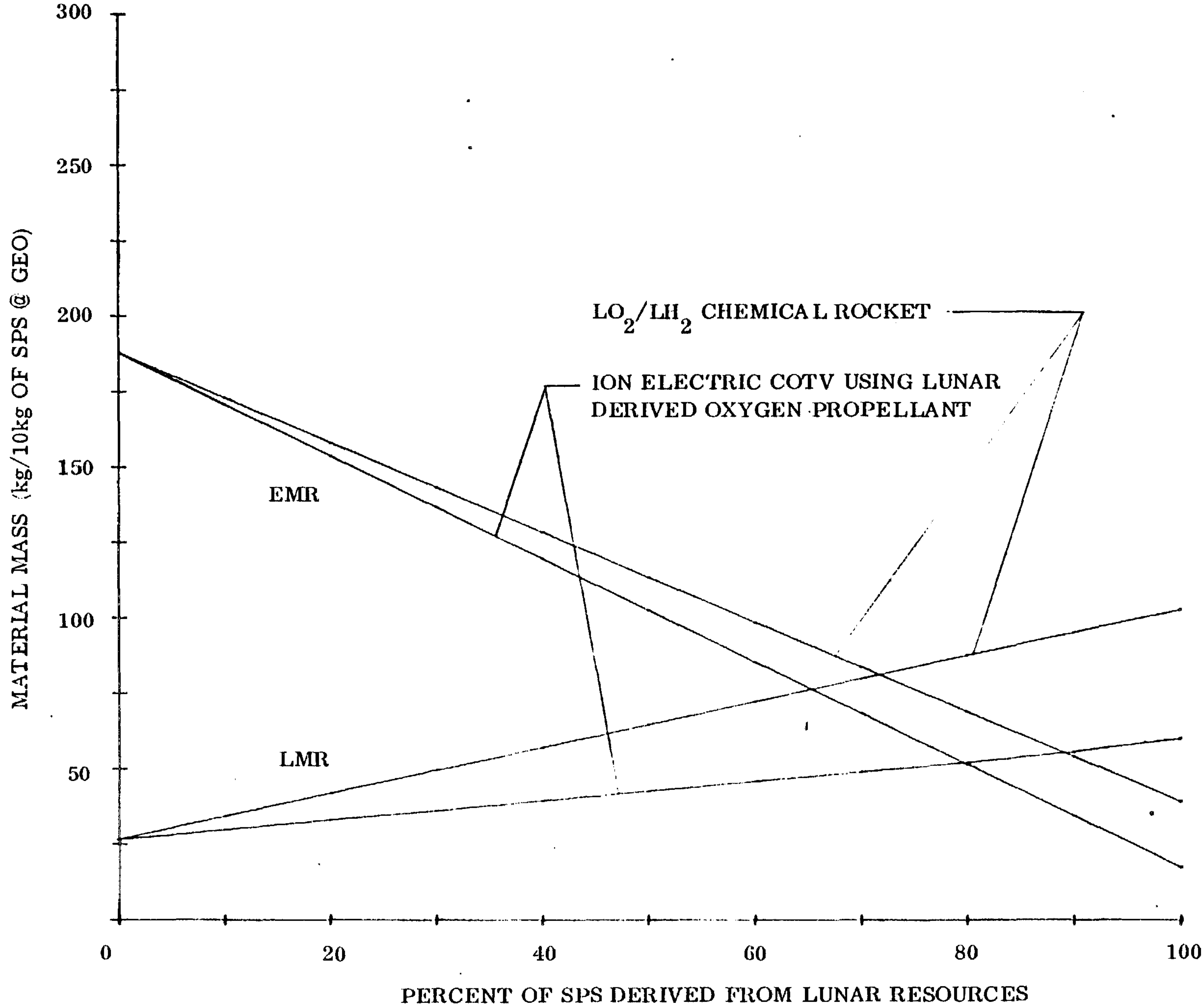


Figure B-30. Option D Revised Baseline - Sensitivity to Lunar Oxygen Recovery

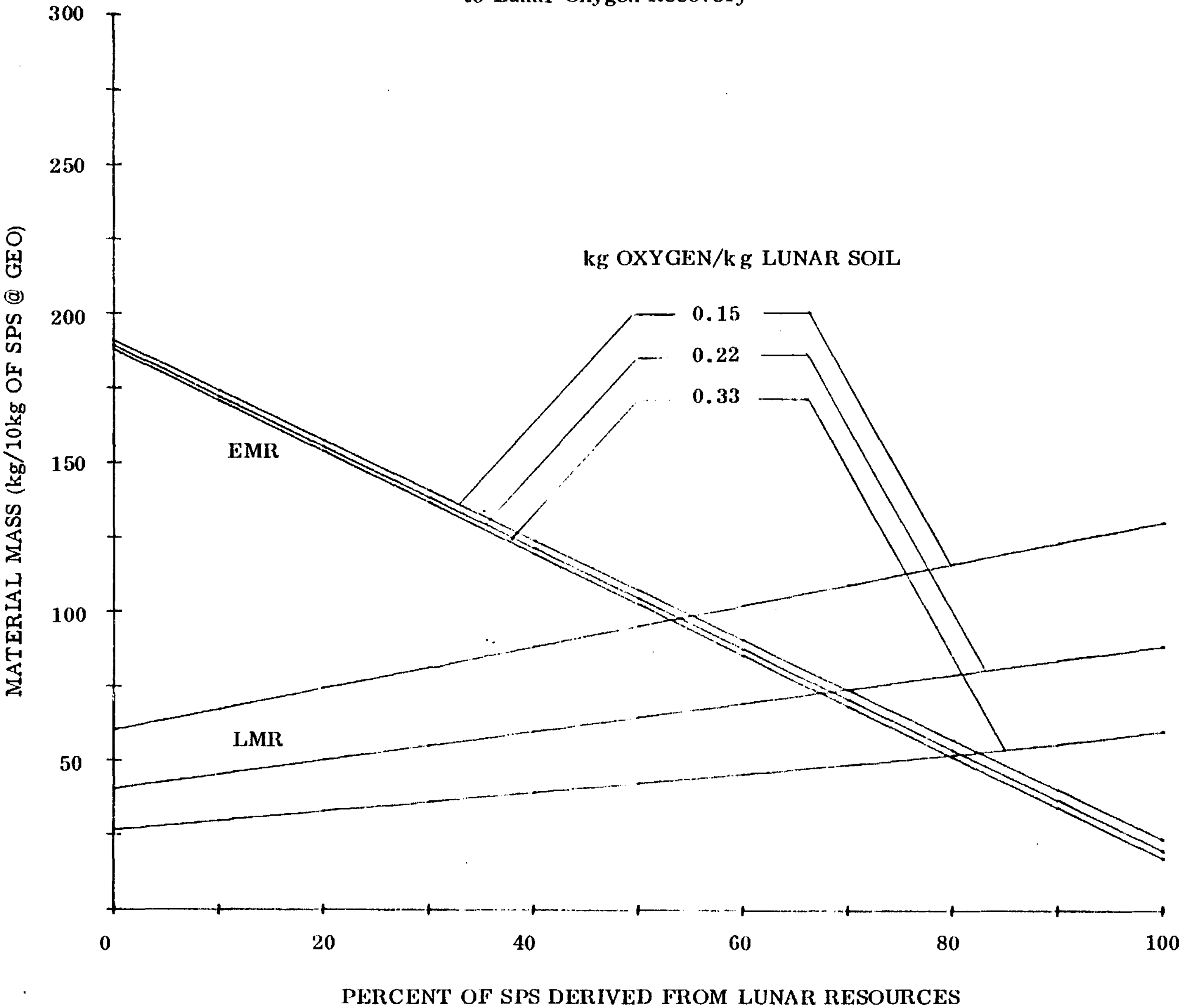


Figure B-31. Option D Revised Baseline - Sensitivity to Lunar Aluminum Recovery

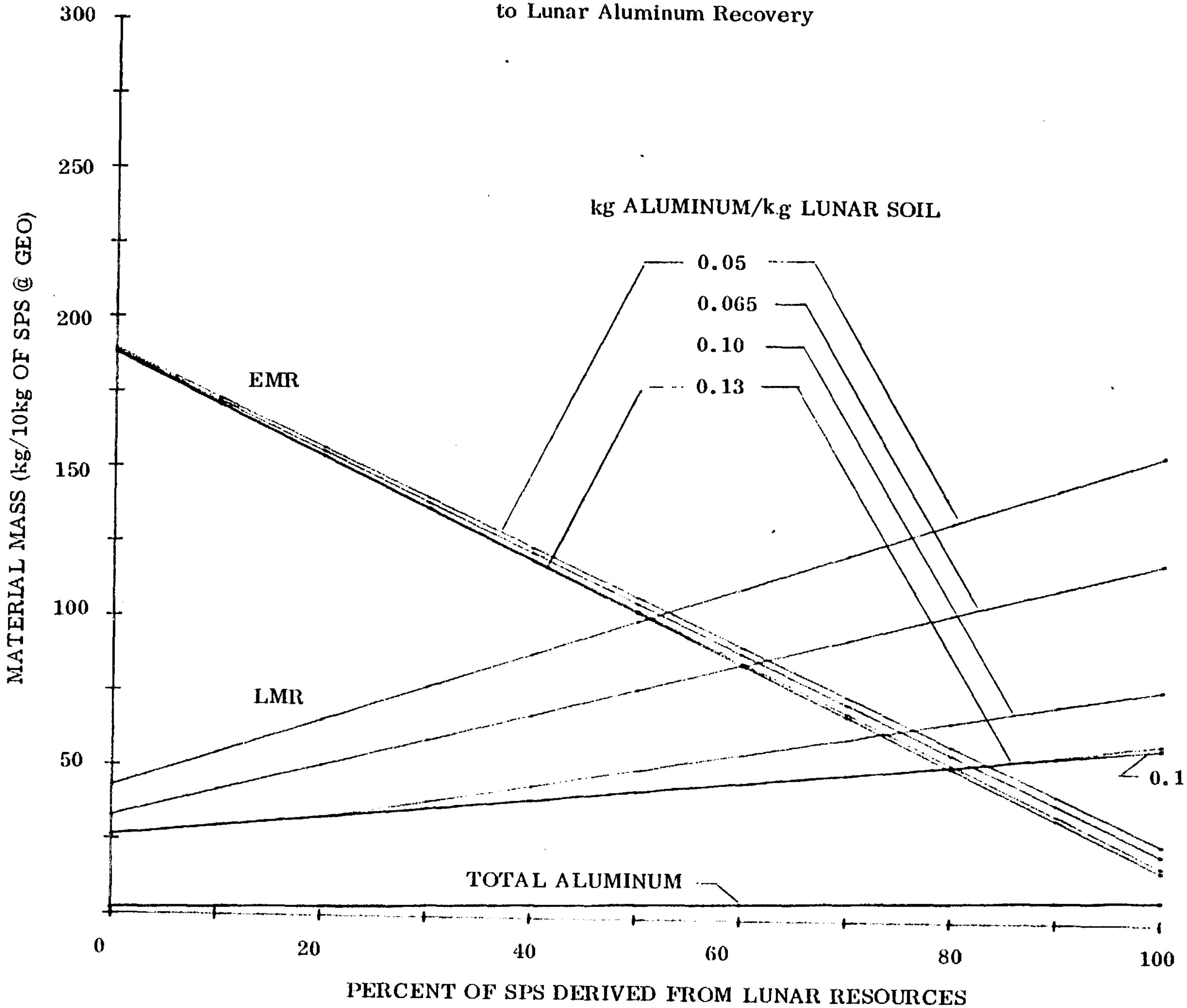


Figure B-32. Option D Revised Baseline - Sensitivity to Chemical Loss Fraction

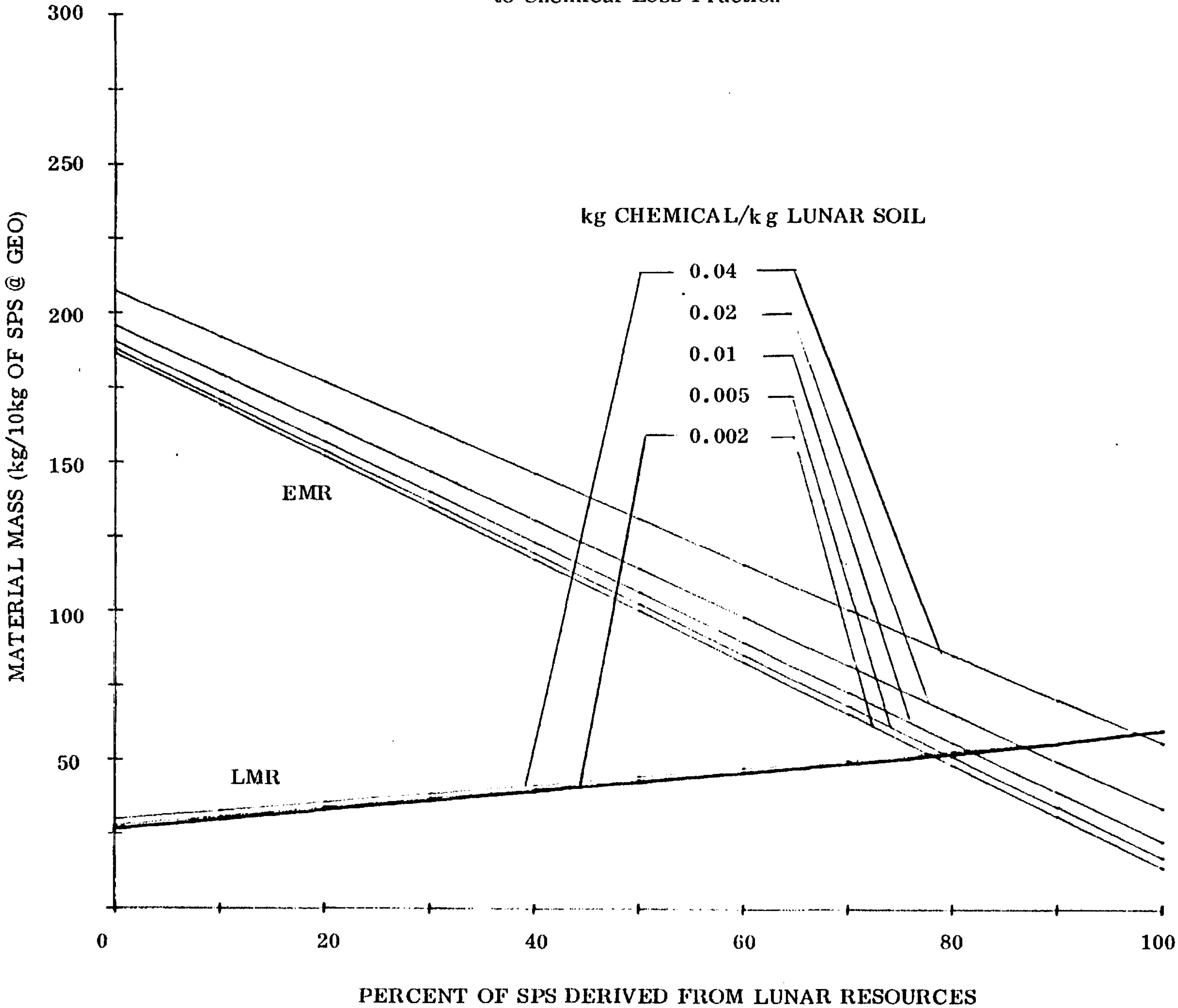


Figure B-33. Option D Revised Baseline - Sensitivity to Ion Electric Propulsion Efficiency

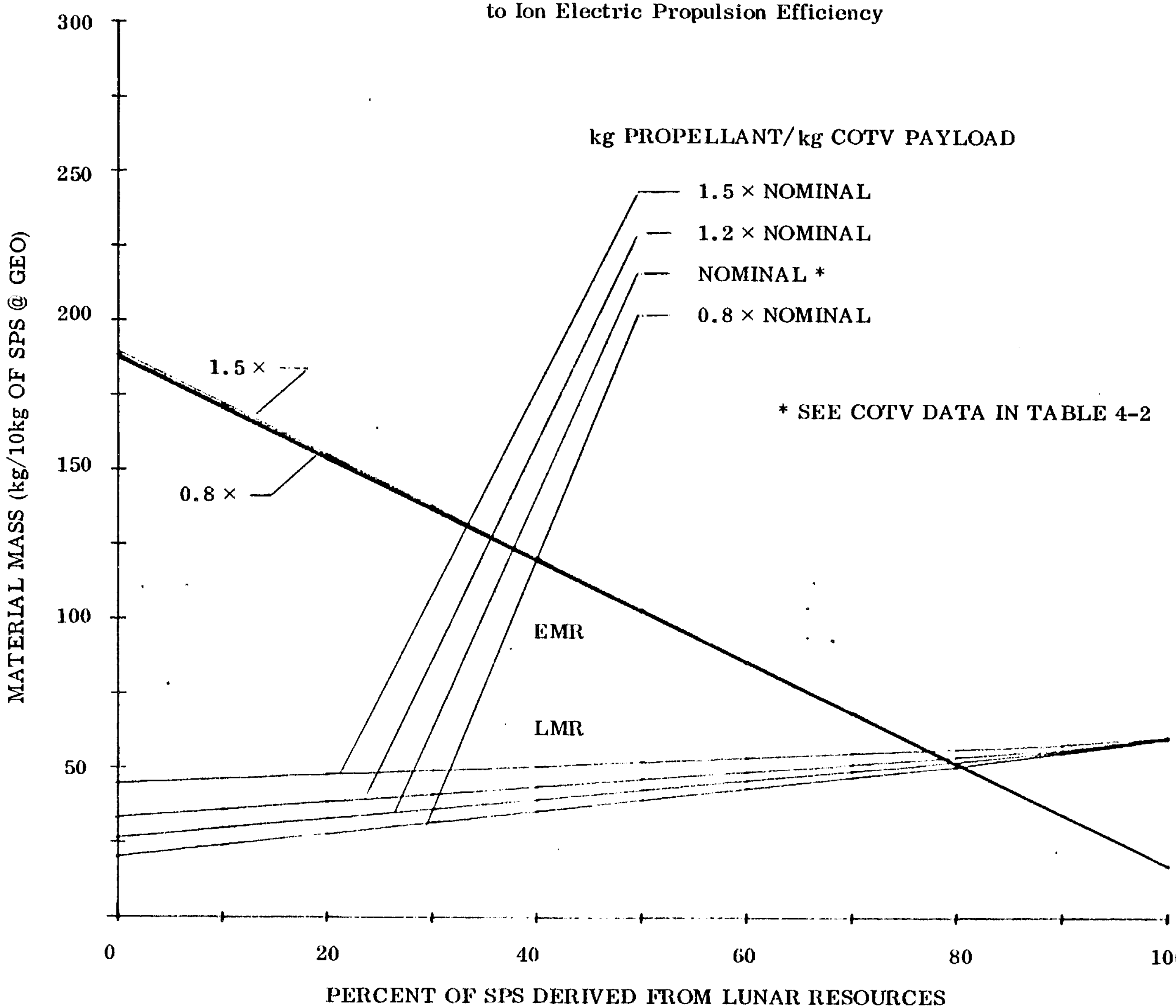


Figure B-34. Option D Revised Baseline - Sensitivity to Lunar Derived Rocket Efficiency

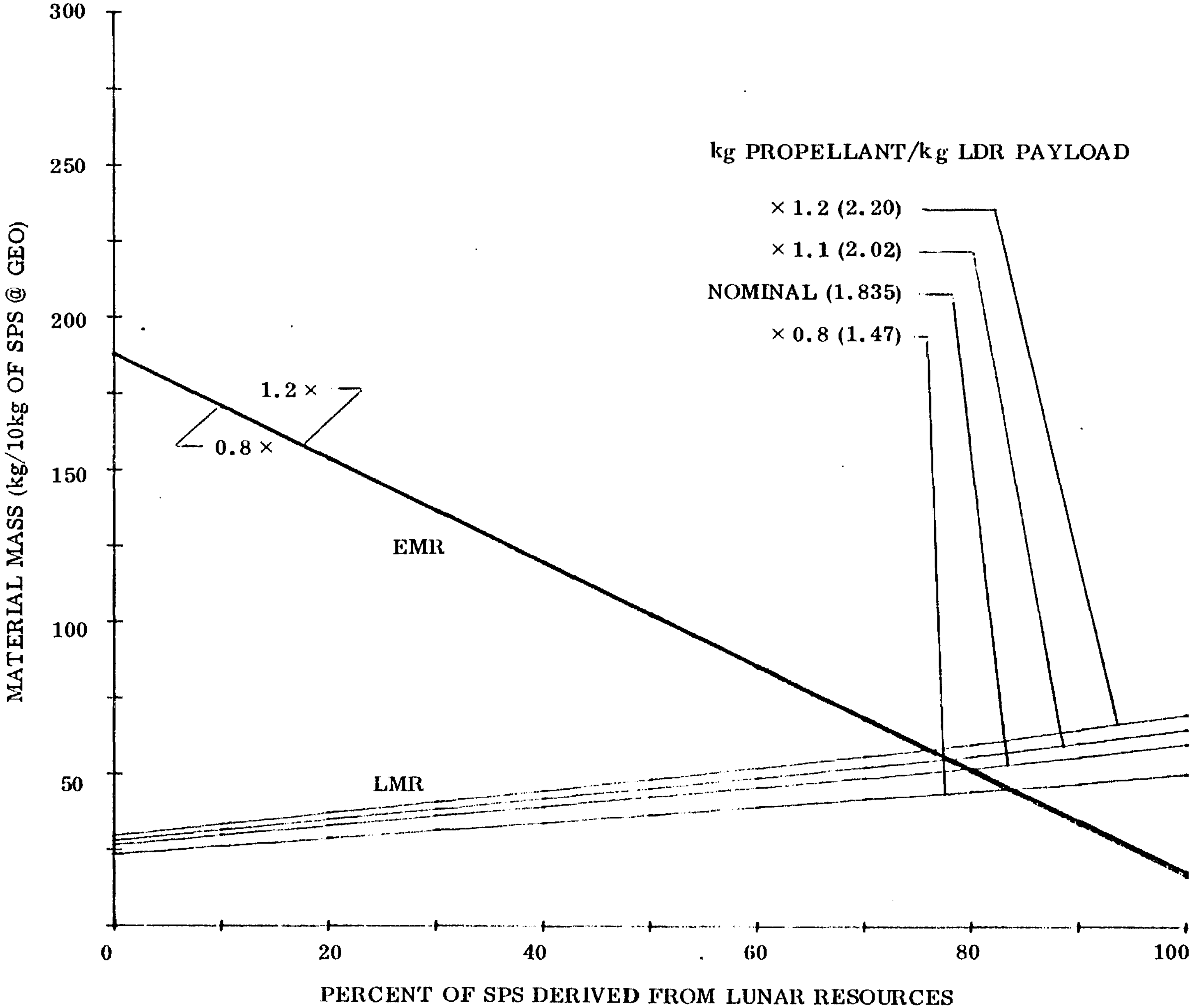
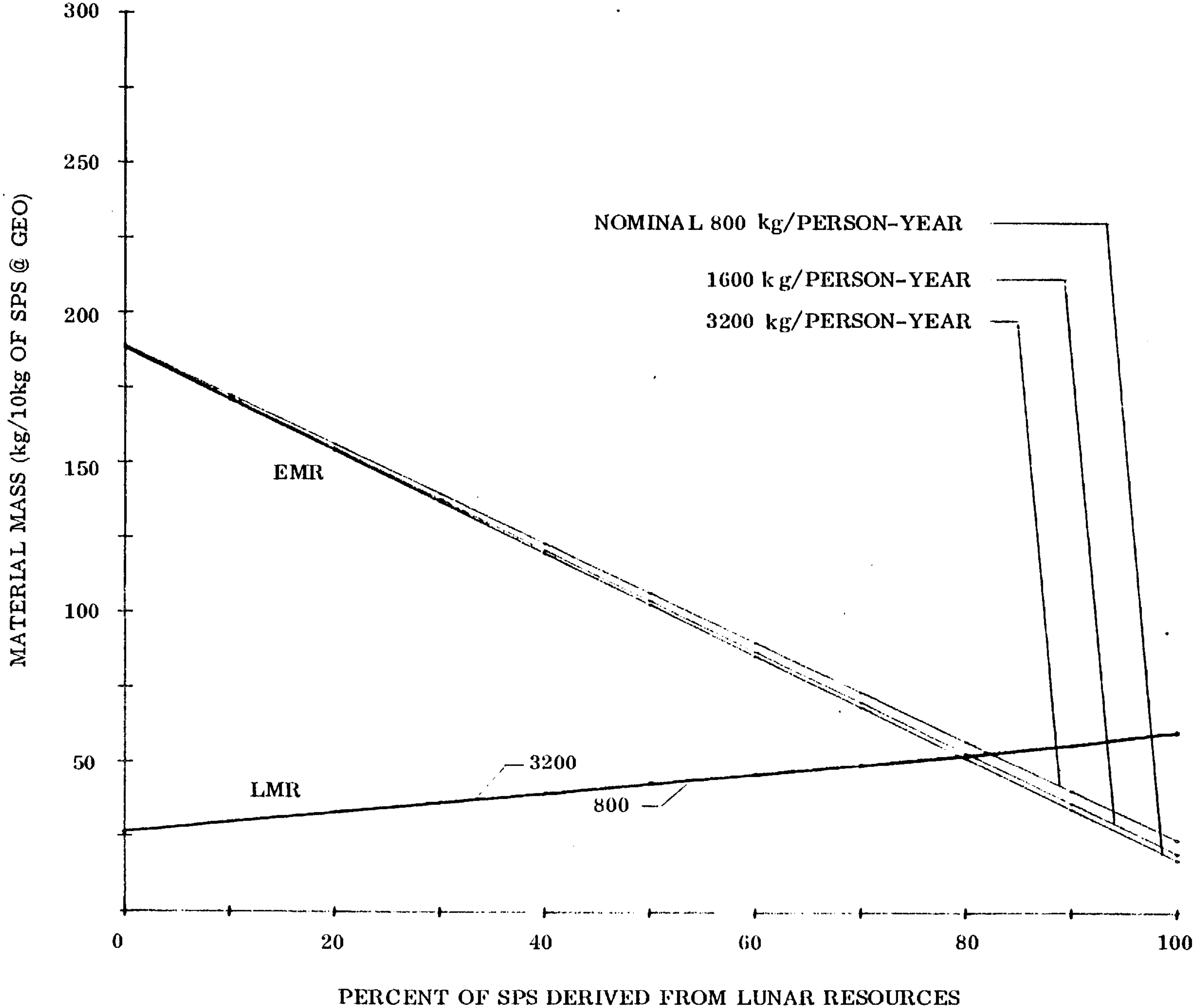


Figure B-35. Option D Revised Baseline - Sensitivity to Life Support Requirements



APPENDIX C

Task 5.3 supplementary data, defining lunar material processing techniques developed to support recommended material recovery methods in Volume II, Section 4.4 of the final report.

Appendix C contains three sections.

C.1 USE OF SOLAR FURNACES FOR MELTING LUNAR MATERIAL

Large solar furnaces of high efficiency have been designed and constructed and are in operation in many countries, including the United States, the USSR, France, Italy, and Japan (Reference 1). The larger units develop power levels up to 1000kW with flux densities in the focal spot as high as $1.7 kW/cm^2$. Earth-based solar furnaces of this capacity are capable of melting two to three metric tons per day of highly refractory oxides (Reference 1). Many of the current solar furnaces consist of two elements; a heliostat composed of several hundred aluminized glass mirrors, each up to 100 × 100cm in size, which direct the sun's rays to a concave concentrator which may be composed of tens to hundreds of glass mirror segments. The heliostat mirrors are capable of following the sun to constantly direct maximum solar energy to the concentrator.

Other types of earth-based solar furnaces have circular arrays of mirrors placed around the base of a tower, and focus solar energy upward to a boiler to generate superheated, high-pressure steam for power generation. The U.S. Department of Energy has constructed a 400kW solar furnace of this type at the Georgia Institute of Technology Engineering Experiment Station (Reference 2). This facility has an array of 550 mirrors which are mechanically linked and driven to focus sunlight at a point 21.3m above the center of the field. The peak flux density in the focal zone is $0.22 kW/cm^2$. Earth-based solar furnaces have been operating at an overall thermal efficiency in the range of 80-85%.

While aluminized glass mirrors have been generally used in solar furnaces, lunar-based furnaces could employ lightweight mirrors made of aluminized Kapton film or may be coated with sodium as suggested by Kraft Erhicke. The kinematic tracking system could also be a lightweight structure made from a graphite/resin composite material.

C.2 ALTERNATIVE OXYGEN PRODUCTION PROCESS

The requirement for lunar-derived oxygen constitutes 31 to 54% of the total weight of lunar material required to satisfy both SPS construction and propellant needs. The amount of oxygen required ranges from 39,250 tons for systems Concept B to 174,500 tons for systems Concept D, and is a primary factor in determining the amount of lunar material which must be mined and processed to meet the concept requirements.

As is the case on earth, oxygen is the most prevalent element in the moon, amounting to 40 to 45% of the mass of the lunar regolith. The oxygen content falls within this range independent of the location and origin of the lunar soils and is a major constituent of all the mineral species found on the moon.

Oxygen may be recovered from lunar soils by a variety of processes among which the following show promise:

1. Direct electrolysis of molten lunar soil.
2. Methane Reduction Process - electrolysis of water produced by reaction of carbon monoxide and hydrogen; the latter two resulting from the reaction between molten lunar soil and methane. (Defined via work accomplished by Aerojet General Corporation, References 3 and 4).
3. Acid Leach Process - electrolysis of water solutions of metal salts resulting from dissolving lunar soil in acids or bases. (Defined via work being performed by the Lunar and Planetary Institute, Referenced 5 and 6.)

Power facilities, chemicals and processing equipment for all of the above processes must be transported from the earth to wherever the extraction of lunar materials is to be performed.

Two interrelated questions involving these (and other) alternative processing techniques must eventually be resolved prior to initiating development of space processing equipment.

1. Where is the best location (lunar surface on in-space) for accomplishing lunar material processing?
2. Which processing technique is preferred?

It is unlikely that either of these questions will be resolved by this study. However, the following discussions are presented to help scope the assessment issue, and identify the important considerations involved.

The three processes identified for the recovery of oxygen and metals from lunar soil have been comparatively evaluated insofar as their current status permits. These three processes are not strictly comparable for a number of reasons, the chief of which is that they have been developed or considered for the extraction of different elements from lunar soil as follows:

Acid Leach Process	Methane Reduction	Electrolysis
Oxygen	Oxygen	Oxygen
Silicon	Silicon	Silicon
Aluminum	slag	Aluminum
Iron		Iron
Calcium		
Magnesium		
Titanium		
Sodium		

Furthermore, the degree of development of the three processes varies widely.

The Chemical Products Division of Aerojet-General Corporation has developed a 3-step process whereby molten rock is reacted with methane to produce carbon monoxide, hydrogen, free silicon and metal oxides (Reference 4). The carbon monoxide and hydrogen are then reacted to form methane and water, following which the water is electrolyzed to produce oxygen and hydrogen. The Aerojet Carbothermal Process was developed under the sponsorship of NASA's Office of Advanced Research and Technology and resulted in the development of laboratory scale reactor units for each of the steps. This process requires four moles of methane per mole of lunar material (anorthite) or approximately 0.23-lbs. of methane per pound of anorthite. Since, however, the methane is constantly regenerated in the second stage of the reaction, the process is efficient and very little makeup methane must be transported from earth after the initial amount is supplied.

Reference 3 listed power and equipment mass requirements for oxygen monthly production rates of 6,000, 12,000 and 24,000-lbs. These data were provided for two systems; one requiring refrigeration cooling and the other radiative cooling, with the latter requiring both less power and lighter processing equipment. The more efficient process was used to scale up to the 100,000 T/year production rate, and the calculated values were reduced by 20% in mass and 10% in power to allow for increased efficiency with size.

The by-products of the Aerojet General process include iron and silicon metal and slag, the latter being a mixture of metal oxides which can be further reduced to recover additional metals and alloying elements. Analyses of the process and equipment requirements have produced plant sizing and cost estimates which indicate considerable economies as compared to the transport of oxygen from the earth (Reference 3).

The acid leach process and various options within the process have been theoretically analyzed. Gaps in current technology and areas for future research and development were identified. Preliminary estimates have been made of equipment and power requirements (Reference 5). While no experimental demonstration has been made of the overall process, many of the individual stages of the acid leach process have been reduced to commercial or pilot plant practice while others have been verified at a laboratory level (Reference 5).

The analysis presented in Reference 5 was based on processing 30,000 T/year. Table IX of the reference listed the following power and equipment mass estimates, while Page 42 of the same reference stated that "the net reagent mass ... is comparable with the process equipment mass."

Item	Mass of Earth Supplied Chemicals & Equipment Metric Tons	Electric Power Requirements
Reagent Inventory	20	30 MW
Process Equipment	20 (see above)	
Compressors	10	
Heat Exchangers	10	
Pipes, Valves	5	
Electrical	6	
Structural & Misc.	25	
Radiators (20 MW)	24	
Elec. Power (30 MW)	120	
Total	240	

Since the annual mass of lunar material to be processed is on the order of 500,000 tons, the above values were multiplied by 16.67.

The electrolysis of lunar soil has been demonstrated on a laboratory scale using earth derived volcanic rocks to simulate lunar material (Reference 7). This study had the objective of extracting oxygen from lunar soil; and demonstrated that oxygen was evolved at the anode and free metals, including silicon, iron, aluminum and others, accumulated on the cathode. To obtain proper fluidity and electrical conductivity at the operating temperature it was necessary to add fluxing compounds (fluorides) to the melt. The experimental work was not carried to the point of recovering and separating the metals deposited on the cathode. This investigation also identified problem areas and recommended further research and development required to make the process practicable.

The free energy of anorthite at 1800°K is -685 Kcal/mole, with each mole containing 128 grams of oxygen. This convers to a requirement of 87.2 MW to produce 100,000 tons of oxygen per year at 100% theoretical efficiency. Assuming 50% electrical efficiency, the power requirement for the electrolytic production of oxygen becomes 175 MW.

The estimate of equipment mass for the solar melting and electrolysis process includes the following:

		Mass, T
Solar mirror and focusing system	-	1,000
Electrical power	-	700
Electrodes	-	25
Piping	-	25
Containment Vessels	-	250
Misc.	-	500
		2,500

The production of oxygen from lunar soil was selected as the basis for comparing these three processes, inasmuch as the requirement for oxygen appears to exceed to the requirement for any other of the lunar-derived materials. The analysis which follows is based upon a number of assumptions; an annual requirement of 100,000 tons of oxygen, constant lunar soil containing 40% oxygen and all processes yielding a 50% recovery of the oxygen. This requires the processing of 500,000 tons of lunar soil, which is within the range of amounts required for the current study. The final assumption is an operational factor of 0.8137 based upon a 330-workday year of 21.6 hours/day, or 7,128 hours/year. This factor is the same used by Dr. R. D. Waldron of LPI in analyzing the acid leach process. Table C-1 lists the power and equipment mass requirements for each of the three processes to produce 100,00 T/year of oxygen.

It must be borne in mind that this comparison does not present a fair picture of the relative merits of the three processes. The acid leach process leads to the extraction of many more elements from lunar soil than do the other two processes and in much greater quantities than are required to fabricate the SPS. The methane reduction process, although requiring less power, requires a much larger mass of equipment and produces only oxygen, with silicon and slag as by-products requiring extensive further processing to extract the elemental materials. While the electrolysis process appears to require both less power and mass of equipment, this process has not been analyzed to the same extent as the acid leach process and probably requires more extensive research and development than acid leaching.

Table C-1. Comparison of Alternative Lunar Soil Processing Methods to Obtain 100, 000T of Oxygen Annually.

	Direct Electrolysis of Molten Lunar Soil	Methane Reduction Process	Acid Leach Process
Elements Other Than Oxygen Extracted	Silicon Aluminum Iron	Silicon	Silicon Aluminum Iron Calcium Magnesium Titanium Sodium
Equipment Mass (T)	2,500	7,200	4,000
Power (MW)	175	240	500
Status	Experimental Work by the Bureau of Mines	Laboratory Prototype by Aerojet General	Various — Portions of Process are Commercial, Others Theoretical
Comments	Secondary Processing Required for Metal & Silicon Recovery	Secondary Processing Required. Large Initial Supply of CH_4 Needed From Earth	Produces More Metals Than Required for SPS Construction

Inasmuch as oxygen is the major lunar material required and determines the amount of lunar material mined and processed, the location selected for oxygen recovery, i.e., on the lunar surface or in a SMF, is very important. The question of optimum oxygen extraction location involves several considerations. If lunar extraction is used, the oxygen must be transported into lunar orbit to enable its use as transfer vehicle propellant. It may be possible to employ a mass driver catapult to launch small canisters of oxygen, but this method is likely to be inefficient and impractical. The alternative is to use chemical rockets which in addition to requiring more oxygen propellant of their own, produce a large quantity of volitiles which may generate a lunar atmosphere. Dr. Richard Vondrak has estimated that continuous release of volatiles at 100 kg/sec in low lunar altitudes would result in a lunar atmosphere with a total mass of 10^5T (compared to the current atmospheric mass of 10T)(Reference 8). This atmosphere would probably impact scientific experimenters, and higher volatile release rates of 1,000 kg/sec would create aerodynamic drag and impact use of mass driver catapults.

For construction of one SPS per year, the Concept C LTV consumes 242.3 T/day of LO_2/LH_2 propellants, which corresponds to an average 2.8 kg/sec and should be acceptable. Although the Concept D lunar derived rocket (LDR) requires 4.2 times more propellant, the LO_2 /Al exhaust products consist of 50% by weight solids which will eventually fall to the lunar surface.

If SMF processing is selected, large quantities of excess material must be transported from the moon, although in-space use of slag obtained from processing operations for shielding or reaction mass may be desirable. In-space manufacturing, with material delivered by mass driver, certainly reduces the lunar atmosphere creation problem, but may create several environmental impacts of its own. Dr. Vondrak has also estimated (Reference 8) that a 600 kg/sec volatile release rate at L_5 could build up an orbital ring capable of diverting the solar wind. This might lead to plasma instability in earth's magnetosphere, which could conceivable dump Van Allen belt radiation into earth's atmosphere. Also, application must be found for waste produced at the SMF, so it doesn't create an navigation problem. This will not be a concern in an expanding space industrialization operation since large quantities of slag will be needed for galactic and solar flare shielding.

To summarize, it is not clear that the alternative processing techniques, processing locations (lunar surface vs SMF), or the environmental considerations provide a strong basis for determining how and where processing should be accomplished, and how materials should be transported. For the purpose of the Lunar Resources Utilization for Space Construction study, we have arbitrarily selected the direct electrolysis processing technique; used in space with Concept B, and on the lunar surface with Concepts C and D. It is unlikely that substitution of an alternative processing technique would significantly affect overall study results.

C.3 ELECTROLYSIS OF LUNAR SOIL

The U. S. Bureau of Mines has done some limited work on the extraction of oxygen by the electrolysis of molten silicate rocks (Reference 7). The rocks were volcanic scoria somewhat similar in composition to vesicular basaltic rocks returned by Apollo 11. Electrolysis was successfully performed at temperatures in the range of 1320-1520°K in boron nitride cells with a silicon carbide cathode and an iridium anode. In order to promote increased fluidity and electrical conduction of the melt, barium and lithium fluorides in amounts up to 75% were added to the melt. Oxygen and other gases were liberated at the anode, while metal dendrites formed on the cathode.

Lunar soil, which is a mixture of plagioclase feldspars (>80% anorthite), pyroxenes, olivine, ilmenite and other minor constituents, melts at temperatures in the range of 1500-1600°K, as compared to pure anorthite which melts at 1820°K. The melting temperature can be further lowered by the addition of fluoride salts of calcium, magnesium or lithium, but these materials do not exist on the moon. In fact, fluorine is virtually absent, being found in lunar soil in amounts of only 30-300ppm. However, judicious mixtures of available lunar soils can lower the melting point somewhat and increase the fluidity as compared to the normal mare or highlands soils. Fluidity can, of course, be increased by increasing the bath temperature.

Experiments performed at the Bureau of Mines showed that the addition of 10% by weight of lithium fluoride caused a considerable increase in the electrical conductivity of molten silicates. More work in this area is needed to determine the minimum amount of the optimum material to be added to lunar soil to develop the proper electrical properties of the molten bath, particularly focusing on materials which are available on the moon. The promise already shown by the limited amount of work conducted by the Bureau of Mines on the electrolysis of molten silicates justifies consideration of this approach for extracting lunar materials.

It has been suggested by Dr. Waldron that solar heating may not be required at all. The electrical power input for electrolysis is sufficient to melt the lunar material if a suitable conductive path can be established through the initial furnace charge.

Experimental work on the electrolysis of silicate rocks was carried out by the Bureau of Mines in laboratory-sized boron nitride crucibles, using silicon carbide cathodes and iridium anodes. A full-scale lunar facility might employ a fused silica brick-lined vessel and corrosion resistant, coated refractory metal anodes. The anode would be enclosed within a perforated thin-walled iridium tube into which the oxygen would diffuse and be removed.

The viscosity of molten anorthite at 1600°C is reported to be 25 poises, while that of a synthetic lunar sample is given as 6-10 poises in the temperature range of 1375-1450°C (Reference 9). These values correspond to the viscosity of a light fuel oil and permit ready diffusion and transfer of gas bubbles.

The combination of high bath temperature and lunar vacuum conditions will complicate actions at the cathode. The lower boling point metals; sodium, potassium and magnesium, will be liberated as vapors. Somewhat higher boiling point metals such as calcium and.manganese will also boil off since they have vapor pressures of 10-500 Torr at temperatures in the range of 1300-1450°C. At 1430°C, aluminum has a vapor pressure of approximately 0.2 Torr, chromium 1.5×10^{-2} Torr, iron 5×10^{-3} Torr, silicon 5×10^{-4} Torr and titanium 7×10^{-5} Torr. Aluminum and silicon are molten at this temperature, and both will tend to rise to the top of the bath since their densities are less than that of molten silicate rock, being 2.3 and 2.5 g/cc, respectively, as compared to 2.9 for molten rock. Vacuum distillation may offer a reasonable approach to achieving separation of the individual metals.

A very recent paper on the electrolysis of lunar material is to be presented at the 4th Princeton/AIAA conference on Space Manufacturing Facilities (Reference 10). Experimental results indicate successful electrolysis of metals and oxygen without large flux quantities.

REFERENCES

1. Trombe, F., "Solar Furnaces for High-Temperature Processing," Solar Energy, Vol. 7, No. 3, 1963, pp 100-107.

2. Altman, R.F., Brown, C.T., and Teague, H.L., "U.S. Department of Energy Advanced Components Test Facility," paper presented at 14th Symposium on Electromagnetic Windows, June 21-23, 1978, Georgia Institute of Technology, Atlanta, Georgia.

3. Rosenberg, S.D., et. al., "The Manufacture of Propellants for the Support of Advanced Lunar Bases." Aerojet-General Corporation, Paper No. 650835. Presented at the National Aeronautic and Space Engineering and Manufacturing Meeting of the Society of Automotive Engineers, Los Angeles, California, October 4-8, 1965.

4. Rosenberg, S.D., Guter, G.A., and Miller, F.E., "The Utilization of Lunar Resources for Propellant Manufacture," Advances in the Astronautical Sciences, Vol. 20, May 1965.

5. Waldron, R.D., Erstfeld, T.E., and Criswell, D.R., "Processing of Lunar and Asteroidal Material," Section III of Extraterrestrial Materials Processing and Construction, Mid-Term Report on Contract NAS 09-051-001, 25 April 1978.

6. Waldron, R.D., and Erstfeld, T., "Processing of Lunar and Asteroidal Materials," Mid-Term Report on Contract NAS 09-051-001, Mod. 24, Lunar and Planetary Institute, April 1978.

7. Kesterke, D.G., "Electrowinning of Oxygen From Silicate Rocks," Bureau of Mines Report of Investigation RI 7587, 1971.

8. Vondrak, R.R., "Environmental Impact of Large Space Facilities," 1975 Princeton Conference on Space Manufacturing, May 1975.

9. "Handbook of Lunar Materials," Lunar and Planetary Sciences Division, NASA-Lyndon B. Johnson Space Center, May 1978.

10. Lindstrom, D.J., and Haskin, L.A., "Electrochemistry of Lunar Rocks," Department of Earth and Planetary Sciences and McDonnell Center for Space Sciences, Washington University, St. Louis, Missouri 63130.

APPENDIX D

Task 5.3 supplementary data, defining space processing and manufacturing requirements including products, production facilities, energy needs, and unrecoverable material losses. This information was developed to support derivation of LRU manufacturing costs and start-up mass estimates in Volume II, Section 4.4 of the final report.

Appendix D contains three sections.

D.1 LRU PROCESSING AND MANUFACTURING REQUIREMENTS

The results of study Task 5.2, contained in Section 3.5 of Volume II, identified total lunar-derived SPS material requirements and the components to be manufactured in space (or on the lunar surface) from these materials. Appropriate lunar materials must be obtained to provide glass, silicon, aluminum, iron and oxygen from which the fifteen SPS product groups in Table A-10 are manufactured. Facilities are required to process raw lunar material into these useful constituents, manufacture the components, and assemble the satellite.

To scope the LRU processing and manufacturing task, the flow diagram of Figure D-1 was generated. This figure identifies the lunar material flow, processing steps, and manufacturing steps required to transform raw lunar material into a completed 10 GW solar power satellite. The lower case superscript letters shown in Figure D-1 correspond to those SPS components previously selected in Section 3.5 of Volume II for manufacture with lunar materials. Figure D-1 does not include the earth-supplied materials such as alloying agents, electronics components, and special metals like tungsten and mercury required to manufacture a complete SPS. These earth-supplied ingredients are assumed to be combined with lunar-derived ingredients during the appropriate manufacturing step. The components identified in Figure D-1 correspond to the 89.6% lunar material utilization level for 10 GW SPS construction.

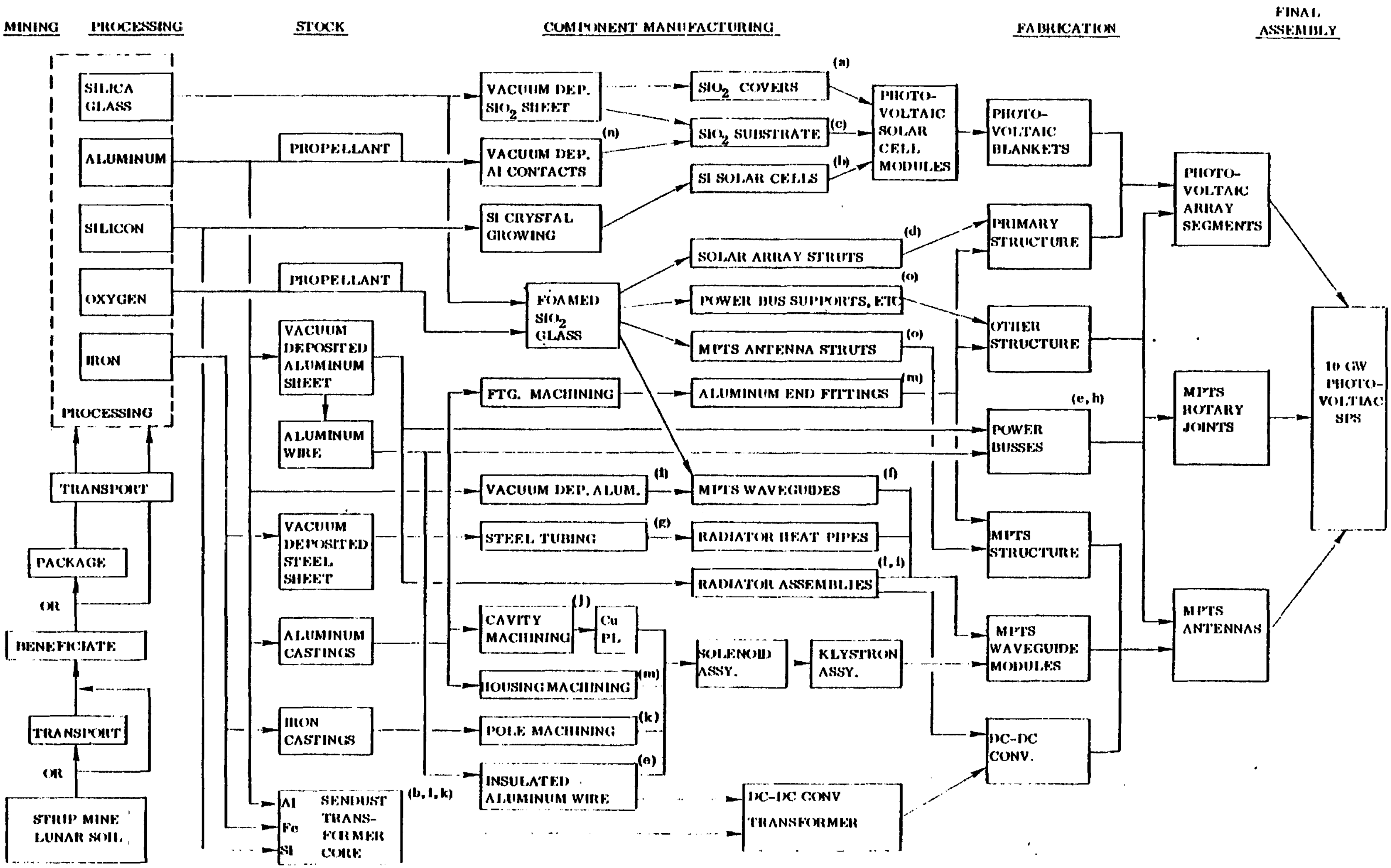


Figure D-1. LRU Processing and Manufacturing Flow Diagram.

D.2 MANUFACTURING DATA SHEETS AND FACILITY REQUIREMENTS

Facility requirements estimates were made for each manufacturing step or collection of similar steps. These facility requirements are based on the product recommendations documented in Section 4.4 of Volume II and industry technology projections for 1990. Facility definition has been separated into four categories; stock manufacturing facilities, parts manufacturing facilities, component assembly facilities and silicon cell panel facilities. A total of 26 mass and power requirements for manufacturing facilities have been documented which are common to all three LRU system concepts. A 27th facility, used to manufacture fiberglass bags for mass driver payload packaging, has also been defined for LRU Concept B. These facilities are described in the following individual data sheets numbered (1) through (27). Source reference information is listed on each data sheet. A summary of LRU common facilities is compiled in Tables D-2 through D-5 which follow the data sheets. Cost estimates for these facilities are contained by item number in Appendix G.

The facility mass and power estimates used in the data sheets for the basic manufacturing equipment (electron beam vapor deposition guns, casting machines, furnaces, etc.) have been based on data for similar earth production equipment. For in-space or lunar surface use the mass and perhaps power consumption associated with these facilities can be reduced considerably. However, a significant quantity of peripheral equipment and tooling is required to support each major manufacturing function. Application of the full earth mass to similar facilities designed for in-space use should adequately account for these undefined peripherals.

Industrial robot quantities are based on assumed material handling and feed requirements for highly automated production equipment. They are very preliminary initial estimates and should be updated following improved understanding of Items (1) through (26).

STOCK MANUFACTURING DATA SHEET

ITEM NUMBER (1)

Aluminum Sheet	Production Requirements	
1.0 mm thick × 10 m wide	9544 T/year	1.20 T/hr
	3381 km/year	7.08 m/min

Aluminum or aluminum allow sheet will be produced by electron beam vapor deposition, using the 1200 kW EH1200/50 electron beam gun.[1] This gun is operated by a high-voltage power supply, 1200 kW, 50kV, with a low-voltage supply for the magnetic lens and beam deflection accessories.

Aluminum or aluminum alloy sheet will be vapor deposted upon an endless belt of thin molybdenum sheet, using seven 1200 kW electron beam guns spaced to provide a sheet of uniform thickness. The aluminum in the form of mixed alloy powder or prealloyed rod, will be continuously fed into an induction-heated crucible, vaporized by the EB guns and deposited on the molybdenum belt. The surface of the molybdenum will be treated to facilitate separation of the aluminum, which will be rolled into coils for further processing.

Iron sheet and plate will be similarly vapor deposited.

The production of aluminum and steel alloy sheet by means of electron beam vapor deposition was suggested by the industrial development and use of large electron beam guns. Guns of 250 kW capacity are commercially used to coat 400 mm wide steel sheet with aluminum, with the steel sheet traveling at a rate of 3 m/sec. EB guns of 150 kW capacity are annually coating millions of square feet of architectural glass. EB guns of 1200 kW capacity have recently been developed which can deposit aluminum at very high rates.

Manufacture of the sheet stock either on the moon or in the SMF eliminates the need for vacuum chambers and pumping equipment. A price in the range of $200-$3000 per kW, including both the electron beam gun and power supply, has been quoted.[2]

Facility Requirements:

Equipment:	Seven 1200 kW electron beam guns, three industrial robots
Power requirements:	8,783 kW
Mass of facility:	34 tons

ITEM NUMBER (1) continued

References:

1. Private communication from Dr. S. Schiller, Forschungsinstitut Manfred von Ardenne, Dresden, and "Deposition by Electron Beam Evaporation with Rates of Up to 50 μm/sec," S. Schiller and G. Jasch, paper presented at third conference on Metallurgical Coatings, April 3-7, 1978, San Francisco, California.
2. Private communication, Mr. Wayne Saindon, Airco Temescal Division, Berkeley, California.

STOCK MANUFACTURING DATA SHEET

ITEM NUMBER (2)

Aluminum Wire,	Production requirements	
1.13 mm diam	2865 T/year	0.36 T/hr
	1×10^6 km/year	2124 m/min

Aluminum wire will be produced from aluminum sheet by slitting the sheet into narrow strips (using steel slitting rolls), electron beam welding strip ends together and then pulling them through wire-drawing dies to form wire of round cross section.

Equipment requirements include:

- 1 set - Slitting rolls and strip coiler
- 1 - electron beam welder
- 8 - wire drawing machines, spool coilers and motor drives
- 2 - industrial robots

Depending upon the amount of cold-work introduced during wire-drawing and its effect upon the electrical conductivity of the wire, a small annealing furnace may also be required.

Equipment mass - 7 tons
Power requirement - 32 kW

Reference to wire drawing process and equipment:

Metals Handbook, American Society for Metals, 1948 Edition.

STOCK MANUFACTURING DATA SHEET

ITEM NUMBER (3)

Steel Sheet	Production requirements	
7.0 cm wide × 0.25 cm thick	4294 T/year	0.54 T/hr
for heat pipe tubing	3106 km/year	390 m/hr

The steel sheet will be produced by electron beam vapor deposition using 1200 kW electron beam guns. Each gun is powered by a 1200 kW, 50 kV power supply with a low voltage power supply for the magnetic lens and beam deflection facilities.

Each gun can evaporate 72.9 kg/hr of steel. 540 kg/hr will require 7.4 (8) electron beam guns. Since the heat pipes will be in contact with liquid mercury at moderately elevated temperatures, an oxidation resistant alloy steel such as CROLOY 5 Si or SICROMO 7 will be used. Both of these steels contain silicon, chromium and molybdenum as alloying elements. The first two can be extracted from lunar soil, but the 0.5% molybdenum must be transported from Earth since its concentration on the moon is less than 1 ppm. The 4294 tons/year of heat pipe steel will require 21.5 tons/year of molybdenum.

Facilities required to produce the steel sheet include (8) 1200 kW electron beam guns and associated power supplies, an endless belt of molybdenum or other refractory metal upon which to vapor deposit the steel sheet, and supporting equipment including 3 industrial robots. Total power requirements will be 9603 kW and equipment weight will amount to 38 tons.

References:

1. "Deposition by Electron Beam Evaporation with Rates of up to 50 μm/sec." S. Schiller and G. Jasch.

2. Personal communication, Dr. S. Schiller, Forschungsinstitut Manfred von Ardenne, Dresden.

3. Personal communication, Dr. Rointan Bunshah, UCLA.

4. Liquid Metals Handbook, Atomic Energy Commission and the Bureau of Ships, Dept. of the Navy, 2nd Edition, Jan 1974.

STOCK MANUFACTURING DATA SHEET

ITEM NUMBER (4)

Iron Sheet	Production Requirements	
1.02 cm thick × 16 cm wide	544 T/year	68.3 kg/hr*
for klystron solenoid	388,224 parts	48.8 parts/hr

*Plus 11.7 kg/hr of excess strip mat'l

The iron pole pieces consist of 1.02 cm thick circular plates 15.2 cm in diameter with 2.5 cm dia center holes. The pole pieces will be fabricated from electron beam vapor deposited iron or iron alloy material. The circular pieces and center holes will be blanked out.

Equipment required will consist of 3-400 kW electron beam guns capable of evaporating 80 kg/hr of iron, associated power supplies (1200 kW), endless belt of molybdenum or other refractory metal, blanking press and dies and 2 industrial robots.

Equipment weight - 12 tons
Power requirement - 1222 kW

References:

Same as for steel sheet (Item Number 3)

STOCK MANUFACTURING DATA SHEET

ITEM NUMBER (5)

Aluminum Castings	Production requirements
Klystron solenoid cavity	688 T/year 86.4 kg/hr
	194,112 parts/year 24.4 parts/hr
Strut assembly nodes	184 T/year 23.1 kg/hr
	230,000 parts/year 28.9 parts/hr

The above two items will be produced as permanent mold castings of aluminum alloys, using an automatic permanent mold casting machine. A 50 kW channel type low frequency induction furnace of approximately 1200 pound capacity can melt 300 lb/hour (135 kg/hr) of aluminum or aluminum alloys. The molten metal can be poured into an 8-10 station automatic casting machine capable of producing up to 100 castings per hour.

The foundry facility will include:

1 - 50 kW induction furnace with power supply and controller
1 - Automatic permanent mold casting machine
4 - Sets of permanent molds and accessories for each casting design
6 - Industrial robots

Total power requirements - 126 kW
Weight of equipment - 28 tons

Reference to permanent mold casting process and equipment —

Metals Handbook, Volume 5, Forging and Casting, Eighth Edition, 1970, American Society for Metals.

STOCK MANUFACTURING DATA SHEET

ITEM NUMBER (6)

Sendust Castings	Production Requirements	
Transformer core	844 T/year - Fe	2.18 tons/part
	99 T/year - Si	
	50 T/year - Al	1.4 parts/day
	993T/year metal	
	456 parts/year	

The transformer cores are manufactured from an 85% Fe-10% Si-5% Al alloy. This alloy can be melted in a high frequency induction furnace and cast into shape in a cored sand mold. The equipment required includes a 600 kW high frequency induction melting furnace, sand mixing and molding equipment, core making and drying equipment, mold flasks and one industrial robot.

The total power required is 750 kW and equipment weight is 50 tons.

Reference:

Metals Handbook, American Society for Metals, Volume 5, Forging and Casting, 8th Edition, 1970.

STOCK MANUFACTURING DATA SHEET

ITEM NUMBER (7)

Glass Filaments	Production requirements	
	750 T/year	94.2 kg/hr

Glass filaments are made by melting glass particles in an electrically heated furnace, pouring the molten glass into a container having a large number of fine orifices through which the glass is continually drawn. The glass filaments may be gathered together into a strand and wound into multifilament threads or may be individually wound on spools.

The manufacture of glass filaments is a standard, highly developed process and no problems are foreseen in transferring this process to the lunar surface or to a SMF.

A surface coating is usually applied to glass filaments prior to winding in order to protect them from abrasion damage. The equipment required for the production of glass filaments includes a melting furnace, container with bushings and orifices, collecting drum winding machine, sizing application equipment, spools and one industrial robot. Requirements for power will total 7 kW and equipment will weigh 4 tons.

Reference:

Kirk-Othmer Encyclopedia of Chemical Technology, Vol 10, 2nd Edition, 1960 pp 565-566.

PARTS MANUFACTURING DATA SHEET

ITEM NUMBER (8)

Aluminum End Fittings for:	Production requirements	
a) Primary support struts	420 T/year	52.7 kg/hr
	1.05×10^6 parts/yr	133 parts/hr
b) MPTS secondary struts	96.5 T/year	12.1 kg/hr
	402,000 parts/yr	51 parts/hr

The end fittings for both sizes of foamed glass struts will be fabricated from 1.0 mm thick aluminum or aluminum alloy sheet. The sheet will be cut to size, roll formed into conical sections and electron beam welded. The cone tips will be flattened and drilled for pin connections.

Equipment requirements include sheet metal cutters, blanking presses, roll formers, electron beam welders and industrial robots. Some of the facility items required to produce the Klystron solenoid and collector housings may be used to fabricate the strut end fittings, see item number (9).

Additional facility and power requirements to fabricate strut end fittings include:

Facility weight	- 8 tons
Power requirements	- 37 kW
Industrial robots	- 2

PARTS MANUFACTURING DATA SHEET

ITEM NUMBER (9)

Aluminum Housings	Production requirements	
Klystron solenoid housing	277 T/year	34.8 kg/hr
	194,112 parts/year	24.4 parts/hr
Klystron collector housing	132 T/year	16.6 kg/hr
	194,112 parts/year	24.4 parts/hr

Both of these parts are fabricated from 1.0 mm thick aluminum or aluminum alloy sheet. The solenoid housing is a cylindrical section with welded attachments and the collector housing is a conical section with various heat pipes, waveguides and other attachments welded to it. Equipment requirements include a sheet metal cutter, roll forming equipment, blanking press and dies, welding jigs and fixtures, metal arc and electron beam welders, welding stations and 2 industrial robots.

Estimated power requirements - 77 kW
Estimated equipment weight - 28 tons

ITEM NUMBER (10)

Copper Plate	Production requirement	
Aluminum klystron cavity	90 T/year Cu	11.3 kg/hr
0.03 cm thick Cu plate	194,112 parts	25 parts/hr

The klystron cavity area to be plated amounts to 1725 cm^2 and each cavity requires the deposition of 0.463 kg of copper. Copper can be electroplated at a rate of 0.006" (0.015 cm)/hr from a pyrophosphate solution at 5 volts and a current density of 1080 amps/m^2.

If 25 parts were to be electroplated simultaneously it would require 23.3 kW and it would require 2 hours to plate the required thickness. Consequently two plating baths can be used to obtain the required production rate. Each bath would be 1.5 × 3 × 0.75 m in size and would require the use of an industrial robot. The total equipment requirements would be:

Item	kW	Tons
2 plating tanks & power supply	46.6	1
electrolyte	—	7
2 industrial robots	22	2.8
Total	69	11

PARTS MANUFACTURING DATA SHEET

ITEM NUMBER (11)

Foamed Glass Components	Production Requirements	
a) MPTS Waveguides	8585 T/year	1.09 T/hr
2.87m wide, 4cm thick	1101 km/year	2.3 m/min
b) Primary Structural Members	27,295 T/year	3.46 T/hr
50cm diameter tube	13,040 km/year	27.3 m/min
1 cm wall		
c) Secondary Structural Members	2143 T/year	0.27 T/hr
25 cm diameter	2090 km/year	4.4 m/min
1 cm wall		
Total requirement	28,023 T/year	4.82 T/hr
	16,231 km/year	2.04 km/hr

Native glass will be recovered from lunar fines by means of electrostatic beneficiation. The glass particles will be ball-milled to particle sizes under 5 μm in conjunction with 0.5% by weight of carbon, 0.3% by weight of SO_3 in the form of sodium sulfate, and 1% by weight of water. The resulting mixture is fed into formed molds and traversed through an electrically heated furnace at 900-1100°C where the foaming action takes place, then through a surface-smoothing mechanism into an annealing furnace where the foamed glass components are stress-relieved at 500-700°C. The annealed glass is then cut into desired lengths, the ends tapered by grinding or machining and the aluminum end-fittings swaged into place. The foamed glass structural members may then be assembled into subassemblies or transferred to the SPS for fabrication into the primary and secondary structure.

Foamed glass is produced in a highly automated plant incorporating the following equipment:

Ball mills	Dryer (120-150°C)
Classifier	Tunnel kilns (40-55m long, 900°-1100°C)
Storage bins	Annealing kilns (500°-800°C)
Conveyer belts	Kiln cars
Extruder	Post-kiln cutter
Pre-kiln cutter	Molds and tooling

Silica (SiO_2) can also be obtained from lunar soil by electrolysis to elemental silicon and oxygen and their chemical reaction to form silica or by the acid leach process recommended by Dr. Waldron. Silica from either of these sources can also be used to manufacture foamed glass components. It is estimated aht 70 industrial robots would be employed in the production of foamed glass parts.

ITEM NUMBER (11) continued

Total power requirements - 2.00 MW*
Estimated equipment weight - 840 Tons

*Scaled up from a 40T/day plant described in Reference 2.

References:

1. "Manufacture and Uses of Foam Glass," B. K. Demidovich, Document #AD/A-005 819, Army Foreign Science & Technology Center, Charlottesville, VA, 25 Oct 1974.

2. "Foam Glass Insulation From Waste Glass," Utah University, Salt Lake City, Utah University, Salt Lake City, Utah, U. S. Dept. of Commerce Report PB-272 761, Aug 1977.

PARTS MANUFACTURING DATA SHEET

ITEM NUMBER (12)

Aluminum Desposition on MPTS Waveguides	Production Requirements	
	191T/year	24.0 kg/hr
	1101 km/year	2.30 m/min

The waveguide segments are 2.87 m wide and are to be coated with 6.67 μm (66,700Å) thick aluminum. The aluminum will be deposited on the foamed glass waveguides by electron beam vapor deposition, using 6-160 kW electron beam guns operated by a 1000 kW, 50 kV power source. The six guns would be placed 3 across the 2.87 m width of the wave guides, with 3 more placed behind the first set but staggered in order to maintain a uniform coating thickness. A low voltage power supply is required for beam deflection.

Facility requirements:	6-160 kW electron beam guns
Power:	1200 kW
Weight:	5 tons

References:

1. "Deposition by Electron Beam Evaporation With Rates of up to 50 μm/sec" S. Schiller and G. Jasch.
2. "Physical Vapor Deposition," Airco Temescal, 1976.

PARTS MANUFACTURING DATA SHEET

ITEM NUMBER (13)

Steel Heat Pipes	Production Requirements	
2.22 cm diam	3808 T/year	3.28 kg/part
2.67 m/pipe	1.16×10^6 parts	146 parts/hr

The heat pipes will be fabricated from 7.0 cm wide, 0.25 cm thick steel sheet will be roll formed into tubing and will be butt welded in an automatic welder. The heat pipes are closed at one end, with the other attached to the klystron; 6 per unit, with 4 attached to the cavity/solenoid section and 2 to the collector. The end closures are made by pressing the tubing ends flat and sealing them shut by welding.

Since the klystron solenoid and collector housings are fabricated from aluminum or aluminum alloy sheet, the attachment of the steel heat pipes to the klystron housings will require the use of Al-steel Detacouple joints.

The heat pipes have a relatively sharp bend just before their attachment to the aluminum radiator sheet.

The Detacouple joints will require manufacture on and transport from earth. Each will be 2.22 cm O.D., 1.72 cm I.D. and 1.5 cm long and will weigh 11.56 grams. 0.75 cm of the length will be aluminum and an equal length will be steel for electron beam welding to the aluminum klystron housing and the steel heat pipe respectively. With 1.16×10^6 Detacouples required, a total of 13.5 tons of such couples will be transported from earth.

The equipment required to manufacture the heat pipes will include roll forming machines, automatic tube welders, a press to form the tube end closures, electron beam welders, a tube bending machine and 5 industrial robots. 107 kW of power will be required and the facilities will weigh 61 tons.

PARTS MANUFACTURING DATA SHEET

ITEM NUMBER (14)

Glass Fiber Insulation Electrical Wiring	Production requirements	
	750 T/year	94.2 kg/hr
	0.1×10^6 km/year	212 m/min

Insulation for 1.13 mm diameter Al wire

Glass fiber insulation may be applied to wire by either braiding or by wrapping with tape. Since untreated glass fiber materials have limited resistance to abrasion and flexing, it is advisable that some surface coating be applied to the glass fibers.

Insulation braiding machines are generally capable of production rates of 1000 ft/ 8 hr day or 0.635 meters/minute, and braid sleeves are currently produced for wire down to 1 mm and less in diameter. At the above production rate, a total of 334 braiding machines are required to apply insulation to the electrical wiring for the SPS. Each braiding machine is approximately $1 \times 1 \times 3$ meters in size, with 16-20 bobbins, each loaded with spools of monofilament glass thread.

Equipment required for applying insulation

Type	No.	Power (kW)	Weight (T)
Braiding machines and support equipment	334	250	334
industrial robots	15	165	21
		415	355

The glass filaments are wash coated with approximately 0.5 to 1.0% by weight of a modified silane coating to provide abrasion resistance. This will require furnishing 5-10 tons/year of the wash coating compound from earth.

Reference

Victor Wire & Cable Corp, Los Angeles, CA, Personal communication.

COMPONENT ASSEMBLY DATA SHEET

ITEM NUMBER (15)

DC-DC Converter	Production requirement	
	2029 T/year	4.45 T/assy
	456 Assemblies	1.4 assy/day

Each DC-DC converter assembly contains a 2.18 ton Fe-Si-Al alloy transformer core, 0.91 tons of aluminum wire and 1.36 tons of electronics, controls and packaging. The 0.91 tons of aluminum represent 334.8 km of wire wrapped around each transformer core at a rate of 3.875 m/sec.

Total weight without housing is 4.45 T, assuming packing density with electronics of 3, volume is 1.84 m^3. If DC-DC Converter is a cube with 1.23 m on a side, surface area is 9.0 m^2. Housing of ribbed Al 0.25 cm thick would weigh 9.0 (.0025) (2.7) or 0.06T.

Winding wire on the transformer core will require 2 machines, each weighing 1 ton. The assembly fixture requires parts storage bins, turntable and controls, locating tools and 2 industrial robots.

Total power requirement	30 kW
Facility weight	12 tons

ITEM NUMBER (16)

Klystron Assembly	Production requirements
	32 kg/assembly
	25 assemblies/hr

Assembly of the klystron involves winding of the solenoid wiring, assembly of the cavity pole pieces and solenoid housing, assembly of the collector plates under the collector section cover and welding the various components and housings together.

The production rate of 25 per hour will require 12 separate production fixtures, each equipped with a turntable, wire winding equipment, EB welders and tooling as well as an automated industrial robot to put the component parts together. The equipment weight is estimated to be 30 tons, with 180 kW of power required for this production operation. One industrial robot has been assumed for each production fixture, resulting in a total requirement for 12.

COMPONENT ASSEMBLY DATA SHEET

ITEM NUMBER (17)

DC-DC Converter Radiator Assembly	Production requirement	
	456/year	1.4 Assy/day

Size - 360 m^2, 9 m × 40 m

The radiator assembly consists of a 9 m × 40 m panel fabricated from two mating 1.0 mm thickness aluminum sheets which are preformed to provide flow passages through the panel. The heat transfer fluid is pumped through these passages and back to the DC-DC converter, via tubing. The tubing pattern is formed in the aluminum sheet by draw forming, with one-half of the tubing segment formed in each sheet. Assembly of the two sheets completes the tubing pattern. The panel will be fabricated from 1 × 2 meter segments each of which is stamped with the appropriate tubing pattern. The mating sections are then roll-seam welded together to make the tubing pressure tight and adjacent segments are butt welded together to build up the 9 × 40 m assembly.

Facilities required for this fabrication include a cutting machine, a forming press and dies, an automated roll seam welder and a fusion or electron beam butt welder as well as 2 industrial robots for material handing.

Power requirement kW	Facility weight tons
24	72

COMPONENT ASSEMBLY DATA SHEET

ITEM NUMBER (18)

Klystron Radiator	Production requirement	
Size - 2.57 m^2, 1.5 m × 1.7 m	194,112/year	24.4 Assy/hr

The radiator assembly consists of 6 Klystron heat pipes attached to aluminum sheet radiator segments varying from 0.246 to 0.253 m in width and 1.65 to 1.71 m in length. The operating temperature of the cavity/solenoid radiator section is 300°C and that of the collector section is 500°C.

The alloy steel heat pipes can be attached to the aluminum radiator segments by means of aluminum brazing either in a furnace or in a salt bath. Aluminum braze alloys can be made from lunar derived materials; a common alloy consists of 89.5% Al-7.5% Si-4% Mg. Brazing can be done at 600-630°C, well below the melting point of aluminum.

The facility-requirements include a cutting machine to produce aluminum strip, a brazing furnace with a conveyor system, fixtures, tooling and industrial robots. In addition, 33.2 tons/year of brazing alloy are required, as well as 60 tons/year of soidum fluoride brazing flux which must be obtained from earth. The tooling fixtures are required to position the heat pipes to the aluminum sheet metal during the furnace braze cycle.

Power Required	Equipment weight
30 kW	14 tons

Industrial robots - 8

COMPONENT ASSEMBLY DATA SHEET

ITEM Number (19)

Structural Member Assembly	Production Requirements	
	726,000 Assemblies/year	92.1 Assy/hr
	2 fittings/assembly	

Foamed glass - primary and secondary structural members
Min length - 6.5 m, Max. length - 144 m

The tubular foamed glass structural members (25 and 50 cm O.D.) are cut to length; the ends heated to the softening temperature, and swaged down to cones with 10 cm O.D. at the ends. Circumferential crimping grooves will be included several cm from the tapered ends for the attachment of the aluminum end fittings (item number 8).

Equipment required

Type	Qty	Power Req'd, kW	Mass, tons
Heating Furnace	3	30	15
Swaging machine	5	15	5
Groove former	3	3	3
Crimping machine	3	1	1
Industrial robots	6	66	8
Total		115 kW	32 tons

ITEM NUMBER (20)

MPTS Waveguide Subarray Assembly	Production Requirements
$114\ m^2$ assembly	13,865/year, 1.74 Assy/hr

Four waveguide half-segments, each 2.87×9.9 m are laser welded together to form a panel 11.48×9.9m in size. Two such panels are then placed face to face and laser welded together to form a completed subarray assembly. The equipment required for the various operations include laser welders, positioning fixtures and industrial robots.

The facility for manufacturing the MPTS waveguide subarray assemblies will weigh 25 tons and require 30 kW of power. Two robots have been assumed to perform handling tasks.

SOLAR CELL PANEL FACILITIES DATA SHEET

ITEM NUMBER (21)

			Production requirements	
a)	Silica glass, solar cell covers	1.17 m wide, 75 μm thick	21,658 T/year 86,463 km/year	2.72 T/hr 181 m/min
b)	Silica glass, solar cell substrates	1.17 m wide, 50 μm thick	14,439 T/year 86,463 km/year	1.81 T/hr 181 m/min

Window glass compositions are currently being produced in 2'×2' (0.61 × 0.61 m) size in thickness down to 0.002" (50 μm). This is being done by the downward drawing process in which molten glass falls through a narrow slit corresponding in width to the desired thickness of glass.

While silica glass has a higher melting point than window glass (1710°C as compared to 1400°C) and the desired width of 1.17m is twice that of current practice, only a moderate advance in current technology is required to produce the silica glass. Silica glass can be melted in alumina (Al_2O_3) or magnesia (MgO) lined electrically heated furnaces. The latter oxides have melting points of 2050° and 2800°C respectively.

Glass sheet can be made in this way at approximately 5 m/minute, and can be made flat and smooth and will not require surface grinding or polishing.

Equipment required:

	Power kW	Weight tons
Melting furnace, 20 tons/hr capacity	18,000	25
10 insulated molten glass tanks:		
5 with 7 Mo dies, each with slits 50 μm × 1.17 m	-	15
5 with 7 Mo dies, each with slits 75 μm × 1.17 m	-	15
15 industrial robots	170	21
Total	18,170	76

References

1. Kirk-Othmer Enclycopedia of Chemical Technology, Vol 10 2nd Edition, 1966 pp 533-604
2. Personal communication, Dr. J. D. MacKenzie, UCLA.

SOLAR CELL PANEL FACILITIES DATA SHEET

ITEM NUMBER (22)

Aluminum deposition on Solar Cell Substrate (20 μm thick interconnects)	Production requirements	
	697 T/year	87.5 kg/hr
	86,463 km/year	181 m/min

Electron beam vacuum deposition facilities currently exist which have the capability to deposit 10 μm of aluminum on 1 m wide steel sheet at a rate of 3 m/sec. This is accomplished with 2 EB axial guns, each of 250 kW power level. Thus 4 electron beam guns, with associated power supplies, would suffice to deposit uniform thickness coatings of aluminum on 1.17 m wide solar cell substrates at 181 m/min (3.017 m/sec).

After vapor deposition of the aluminum on the glass, the coated glass will be masked with the solar cell interconnect pattern and etched to remove the excess aluminum. Facility requirements include etching tanks, maskant and chemicals.

Power requirements - 1200 kW
Weight of equipment - 14 tons

SOLAR CELL PANEL FACILITIES DATA SHEET

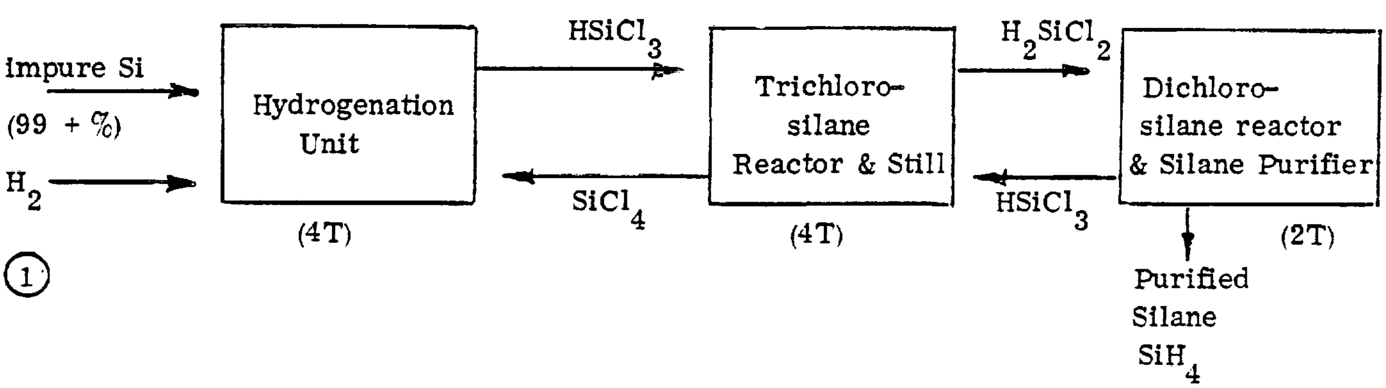
ITEM NUMBER (23)

<u>Silicon Refining to PPB Level</u>

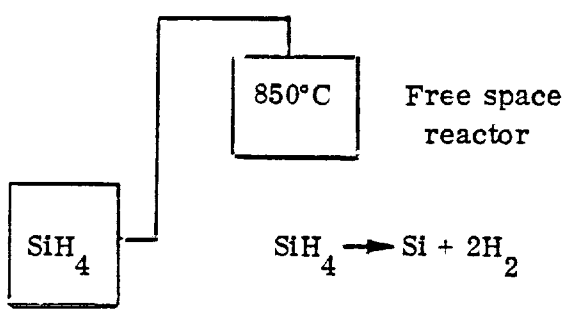
<u>Production Requirements</u>
14,775 T/year

Prior to manufacturing the silicon solar cell ribbon, metallurgical grade silicon (99+% Si) must be purified to the PPB level. This is accomplished by reacting the metallurgical grade silicon with hydrogen and silicon tetrachloride to produce trichlorosilone which is subsequently reduced to dichlorosilane and then to highly purified silane. Silane is then decomposed to silicon. The chemistry of this process is outlined below.

UCC Silane/Silicon Process*



(1)



(2)

(1)

$$Si + 2\,H_2 + 3\,SiCl_4 \rightleftharpoons 4HSiCl_3$$

$$4HSiCl_3 \rightleftharpoons 2H_2SiCl_2 + 2SiCl_4$$

$$2H2SiCl_2 \rightleftharpoons SiH_4 + SiCl_4$$

$$SiH_4 \rightarrow Si + 2H_2$$

Item Number (23) Continued

	Estimated for Plant Capacity of 1000 T/year	Estimates for Plant Capacity of 14,775 T/year
Plant area	3790 m^2	56,000 m^2
Manpower/shift	6	89
Power	1310 kW	19,355 kW
Equipment weight	400 tons	5900 tons

Notes - Si production = 85%
need 1.86 T/hr, @ 85% recovery rate,
plant must process 2.2 T/hr

Facilities include:
- pressure vessels
- pumps
- piping
- fluid bed reactors
- reactors
- storage bins
- conveyors

*page 3-19, Report 5101-67, Proceedings of 9th Project Integration Meeting, April 11-12, 1978.

SOLAR CELL PANEL FACILITIES DATA SHEET

ITEM NUMBER (24)

Silicon Solar Cells	Production Requirements	
	14,775 T/year	1.86 T/hr
7.7 cm wide by 50 μm thick;	1.52×10^6 km	3181 m/min

Automated production of thin (50 μm) silicon solar cells of 15-17% efficiency is currently limited to the "edge-defined film-fed growth" (EFG) process. The "ribbon-to-ribbon" (RTR) process produces thin solar cells, but the efficiency is reduced to approximately 10%. Chemical vapor deposition (CVD) of thick cells followed by laser or electron beam zone refinement produces high efficiencies, but requires 5 to 6 times as much silicon as the EFG process. Sawing or slicing wafers from single crystal silicon ingots also results in high solar cell efficiencies, but is even more wasteful of silicon, and extremely difficult to automate.

There are two major drawbacks to the EFG process, one is that the starting material must be highly purified silicon, and the second is the slow growth rate of the silicon ribbons.

A machine currently being developed will achieve the capability of simultaneously growing 10 ribbons, each 7.7 cm wide at a rate of 7.5 cm/minute. At this rate, it would require a total of 4,283 ribbon growing machines to meet the annual requirement listed above.

Facility requirements

	Power kW	Weight tons
4283 EFG ribbon growing machines, each 2 tons in weight and requiring 31 kW power	132,773	8,566
1070 industrial robots (1 per 4 ribbon growing machines), each requiring 11 kW power and weighing 1.363 tons	11,770	1,460
Total	144,543	10,026

Reference

1. Pages 3-65 through 3-77, Proceedings of 9th Project Integration Meeting, Report 5101-67, April 11-12, 1978, Low Cost Solar Array Project.

SOLAR CELL PANEL FACILITIES DATA SHEET

ITEM NUMBER (25)

	Production requirements	
Cut silicon EFG ribbon, dope, apply contacts & anneal.	7.7 cm wide, $\times 1.52 \times 10^6$ km/year	245 m^2/min, 0.3175 km^2/day

Silicon ribbon is cut into solar cell lengths, ion implanted on both sides to make $n^+ \; pp^+$ cells, electron beam pulse annealed, contact interface lines are ion deposited and the cells are reannealed and prepared for assembly into solar cell panels.

SPIRE Corp. has developed various automated machines capable of performing the above sequence of operations, and has designed a unitized machine capable of performing all of the above operations at a rate of 180 m^2/hr. Each machine will require approximately 550 kW power and will weigh 30 tons, of which 18 tons represent two magnets.

Each SPS will require 1.17×10^8 m^2 of solar cells. Since one SPIRE machine can process 1.42×10^6 m^2/year, a total of 83 such machines will be required.

Equipment requirement:

	Power kW	Weight tons
83 SPIRE machines	45,650	2,490
41 industrial robots	450	60
	46,100	2,550

References

1. "Proceedings, 9th Project Integration Meeting" Report 5101-67, April 11-12, 1978 p 4-60 to 4-116
2. Personal communication, Mr. John A. Minnucci and Mr. Peter Younger, SPIRE Corp.

SOLAR CELL PANEL FACILITIES DATA SHEET

ITEM NUMBER (26)

Solar Cell Module Assembly, $1.29\ m^2$	Production Requirements	
	254 parts/assembly	695 parts/sec
	78,388,736 assy/year	164 assy/min

The silicon cells, cut to size, doped, annealed and with contact interface lines deposited on them are prepared at the rate of $0.3175\ km^2$/day. As they come from the machines, they are automatically positioned on silica glass substrates having vapor deposited aluminum interconnectors on the face in contact with the silicon cells. The silica glass coverplates are positioned over the silicon cells and the assemblies are electrostatically bonded together.

Electrostatic bonding machines under development at SPIRE will be able to bond 50 m^2/hr with a power requirement of 2.07 kW/m^2 and a weight of 7.5 tons per unit.

Equipment required:

	Power kW	Weight tons
254 electrostatic bonding machines	26,226	1905
254 robot fixturing devices	—	1230
254 automated module assembly machines	2,794	345
Total	29,020 kW	3480 tons

References

1. "Proceedings of 9th Project Integration Meeting" Report 5101-67, April 11-12, 1978. p4-60 to 4-116
2. Personal communication, Mr. Peter Younger and Mr. John A Minnucci, SPIRE Corp.

MANUFACTURING FACILITY DATA SHEET

ITEM NUMBER (27)

Glass Bag Manufacture	Production Requirements	
	0.9 T/hr	5 bags/sec

The glass bags are for the transport of lunar soil via the mass driver to the mass catcher at L_2. Each bag will contain 2.5 kg of lunar soil and are required at a rate of 5 per second.

The bags will be produced from seamless woven tubular sections of glass fabric 12 cm in diameter. Each bag will be approximately 12.5 cm long, with the end closures made by heat sealing the tubes.

Glass filaments will be produced in part from lunar glass recovered by the beneficiation of the finer fractions of lunar soil. Silica produced by acid leaching or electrolysis of lunar soil will also be used in the production of glass fabric for use as transport bags. Glass filaments will be produced from molten glass and will be woven into tubes using either circular knitting machines or narrow shuttle looms. The tubular sections will be cut into 12.5 - 13 cm lengths and one end will be heat sealed. The bags will then be loaded with lunar soil using automated loading equipment, the open end heat sealed and the bags conveyed to the mass driver for launching towards L_2.

Equipment required for the production of glass bags include a glass melting furnace, fiberglass production equipment including bushings, collecting drums, winders and spools, tubular weaving and cutting machines, heat sealers and industrial robots. The weight and power requirements are broken down as follows:

	Power kW	Weight tons
glass melting furnace	450	15
fiberglass production facility	25	20
tubular weaving machine and industrial robots	300	120
heat sealing machines	10	2
Total	785 kW	157 tons

The loading of bags with lunar soil will require highly automated facilities and conveyor systems. It is estimated that these systems would require an additional 25 tons of equipment and 175 kW of power. These functions and equipment requirements have been included in the mass driver material handling facility.

Table D-2. Stock Manufacturing Facilities.

Item Number	Stock Products	Production Rate	Equipment Description	Indust Robots	Facility Estimate Mass (t)	Facility Estimate Power (kW)
(1)	Aluminum Sheet 1 mm Thick × 1 m Wide	1.20 T/hr 7.08 m/min	Electron Beam Vapor Deposition, (7) 1200 kW Guns and Fixtures	3	34	8,783
(2)	Aluminum Wire 1.13 mm Dia from Sht	0.36 T/hr 127 km/hr	Sitting Rolls, EB Welder, (8) Wire Drawing Machines	2	7	32
(3)	Steel Sheet 0.25 cm Thick × 7 cm Wide	0.54 T/hr 390 m/hr	Electron Beam Vapor Deposition, (8) 1200 kW Guns and Fixtures	3	38	9,603
(4)	Iron Sheet 1.02 cm Thick × 16 cm Wide	80 kg/hr 7.42 m/hr	Electron Beam Vapor Deposition, (3) 400 kW Guns and Fixtures	2	12	1,222
(5)	Aluminum Castings 0.8 & 3.54 kg/Part	110 kg/hr ~1 Part/min	(1) 50 kW Induction Furnace, (1) Permanent Mold Casting Machine	6	28	126
(6)	Sendust Castings 2.18 T/Part	125 kg/hr ~1.4 Part/day	(1) 600 kW Induction Furnace, Sand Casting Equipment	1	50	750
(7)	Glass Filaments	94 kg/hr	Induction Furnace, Fiber Bushings & Collecting Drum, Spool	1	4	7
	Total	2.51 T/hr		18	173 T	20.5 MW

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Table D-3. Parts Manufacturing Facilities.

Item Number	Parts	Production Rate	Equipment Description	Indust Robots	Facility Estimate Mass (T)	Facility Estimate Power (kW)
(8)	Aluminum End Ftgs For Struts (Sht)	64.8 kg/hr 184 Parts/hr	Blanking Presses, Roll Formers, EB Welders and Fixtures	2	8	37
(9)	Aluminum Housings for Klystron (Sht)	51.4 kg/hr 49 Parts/hr	Blanking Presses, Roll Formers, EB Welders and Fixtures	2	28	77
(10)	Aluminum Klystron Cavity Copper Plate	11.3 kg/hr 25 Parts/hr	Electroplating Tank, Electrolyte, and Handling Fixtures	2	11	69
(11)	Foamed Glass Tubes and Waveguides	4.82 T/hr 2.04 km/hr	Ball Mills, Conveyors, Kilns, Cutters, Molds, & Tooling	70	840	2,000
(12)	Aluminum Deposition on MPTS Waveguides	24 kg/hr 138 m/hr	Electron Beam Vapor Deposition (6) 160 kW Guns & Fixtures	—	5	1200
(13)	Steel Heat Pipes (Sht Material)	3.3 kg/Part 146 Parts/hr	Roll Formers, EB Welders, Press, Tube Benders & Tooling	5	61	107
(14)	Glass Fiber Insulation on Elect Wire	94.2 kg/hr 12.7 km/hr	Glass Filament Coater, (334) Brading Machines	15	355	415
	Total	5.52 T/hr		96	1308 T	3.9 MW

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Table D-4. Component Assembly Facilities.

Item Number	Component Assembly	Production Rate	Equipment Description	Indust Robots	Facility Estimate Mass (T)	Facility Estimate Power (kW)
(15)	dc-dc Converter	1.4 Assy/Day 4.45 T/Assy	Fixture With Storage Bins, Wire Spools, Turntable & Locating Tools	2	12	30
(16)	Klystron Assy	25 Assy/hr 32 kg/Assy	Fixture With Turntable, Wire Winding, EB Welders & Tooling	12	30	180
(17)	dc-dc Converter Radiator Assy	1.4 Assy/Day 360 m^2/Assy	Alum Cutting, Forming Press, Roll Seam Welder & EB Welder	2	72	24
(18)	Klystron Radiator Assy	25 Assy/hr 2.6 m^2/Assy	Alum Cutting, Brazing Furnace, Fixtures & Tooling	8	14	30
(19)	Structural Member Assy	92 Assy/hr ℓ = 6.5-144m	Furnaces, Swaging Machines, Crimping Machines & Fixtures	6	32	115
(20)	MPTS Waveguide Subarray Assy	1.74 Assy/hr 114 m^2/Assy	Lasar Welding Equip, Positioning Fixtures	2	25	30
	TOTAL	144 Assy/hr		32	185 T	0.41 MW

Table D-5. Solar Cell Panel Facilities.

Item Number	Component Assembly	Production Rate	Equipment Description	Indust Robots	Facility Estimate	
					Mass (T)	Power (kW)
(21)	Silica Glass Solar Cell Covers & Substrate 75μm thk × 1.17 in Wide 50μm thk × 1.17 m Wide	2.72 T/hr 1.81 T/hr 181 m/min	Melting Furnace, Molten Glass Tanks & Refractory Dies, Drawing Machines & Annealing Furnace	15	76	18,170
(22)	Aluminum Deposition on Glass Substrate	87.5 kg/hr 181 m/min	(4) 250 kW EB Vapor Dep Guns Plus Masking & Etching Equip	—	14	1,200
(23)	Silicon Refining to PPB Level	2.2 T/hr	Silane/Silicon Process Plant Reactors, Stills, Pumps, Tanks etc	—	5,900	19,360
(24)	Silicon Solar Cells, EFG Process, 50 μm × 7.7 cm	1.86 T/hr 3,181 m/min	(4,283) Ribbon Growing Machines, 10 Ribbons each @ 7.5 cm/min each	1,070	10,030	144,550
(25)	Cut Ribbon, Dope, Apply Contacts & Anneal	1.86 T/hr 695 Parts/sec	(83) 550 kW Integrated Ion Beam Implanters, EB Annealing & Contact Coating	41	2,550	46,100
(26)	Solar Cell Module Assembly 1.29 m^2	164 Assy/min 254 Parts/Assy	Automated Module Assembly Mach, Electrostatic Bonding Equip	254	3,480	29,020
		6.5 T/hr		1,505	22,050	258.4 MW

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D.3 ESTIMATE OF UNRECOVERABLE MATERIAL LOSSES DURING SPACE PROCESSING

Estimates were made of the nonrecoverable losses of both lunar and earth-supplied materials occurring in the various stages of converting metallic and nonmetallic elements into stock materials, parts, components and subassemblies for the SPS.

The nonrecoverable losses of lunar materials at all stages of production are low; in the range of 0.1 to 0.2% since any scrap material can readily be recovered by reprocessing. However, the nonrecoverable losses of many lunar and earth-supplied alloying elements may be much higher, in the order of 5-10%, since it will not generally be worth the effort and expenditure of energy to recover them from scrapped foamed glass, metallic alloys, etc.

Tables D-6 through D-10 list the nominal and total quantities of SPS requirements, starting from the complex assemblies and working back toward the stock materials required to fabricate the parts and components going into assemblies.

Table D-6. Material requirements for SPS Energy Conversion System.

Product or Component	Nominal* Quantity (T/Yr)	Origin: Lunar (L), Earth (E)	Unrecoverable Loss Factor (Percent)	Total Quantity (T/Yr)
Photovoltaic Blankets	54,880	175 μm Thick Sheet of Modules	0.15	54,960
Qty 256 660 m × 660 m	51,570	• Solar Array Modules (L)	0.1	51,620
		78×10^6 1.1 × 1.2 m Modules		
(Produced at SMF Due to	380	• Module Connecting Tape (E)	2.0	390
Handling Considerations)		1.5 cm × 40 μm Plastic Tape		
	2,930	• Blanket Attachment Hdwre (E)	0.5	2,950
		Chord, Springs & Various		
Solar Array Modules	51,620	175 μm Thick Module of Solar Cells	0.42	51,840
Qty 78×10^6 1.1 m × 1.2 m	14,800	• Silicon Solar Cells (L)	1.0	14,950
		20×10^9 7.7 cm Square × 50 μm		
(Produced At SMF Due to	14,800	• Silica Glass Substrate (L)	0.2	14,830
Handling Considerations)		78×10^6 1.1 m × 1.2 m × 50 μm		
	22,020	• Silica Glass Covers (L)	0.2	22,060
		78×10^6 1.1 m × 1.2 m × 75 μm		
Silicon Solar Cells	14,950	50 μm Thick Ribbon (EFG Process)	1.00	15,102
Qty 20×10^9 7.7 × 7.7 cm	14,941	• Purified Silicon (L)	1.0	15,092
(Produced at SMF Due to	8.1	• Aluminum Contacts (L)	10.0	9.0
Facility & Power Reqts)	0.5	• Doping Agents (E)	10.0	0.6
	—	• He Heat Xfer Fluid (E)	25.0	0.5
Silica Glass Substrate				
Qty 78×10^6 1.1 m × 1.2 m	14,830	50 μm Thick Sheet	0.46	14,898
	14,484	• High Purity SiO_2 Glass (L)	0.2	14,513
(Produced at SMF Due to	346	• Aluminum Interconnects (L)	10.0	385
Handling Considerations)				
Silica Glass Covers	22,060	75 μm Thick Sheet	0.42	22,153
Qty 78×10^6 1.1 m × 1.2 m	21,724	• High Purity SiO_2 Glass (L)	0.2	21,768
(SMF-Handling Consid)	346	• Aluminum Interconnects (L)	10.0	385
Refined Silicon for Solar	15,092	Bulk Material (Ingots)	15.0	17,755
Cell Production	15,092	• Metallurgical Grade Si (L)	15.0	17,755
(Production Site Optional)	—	• Processing Chemicals (E)	20.0	154
Silica Glass for Covers	36,281	Bulk Material (Marbles)	0.10	36,317
and Substrate	16,931	• Metallurgical Grade Si (L)	0.1	16,948
(Production Site Optional)	19,350	• Propellant Grade Oxygen (L)	0.1	19,369

*Incl 26.6% Margin

Table D-7. Material requirements for SPS Structural Systems.

Product or Component	Nominal* Quantity (T/Yr)	Origin: Lunar (L), Earth (E)	Unrecoverable Loss Factor (Percent)	Total Quantity (T/Yr)
Primary Structure -	25,763	50 cm Dia Tube With 1 cm Wall	0.21	25,816
Photovoltaic Array &	25,343	• Foamed Glass Tubing (L)	0.2	25,394
MPTS Antenna Base				
(SMF - Handling Consid)	420	• Aluminum End Fittings (L)	0.5	422
Secondary Structure -	2,238	25 cm Dia Tube With 1 cm Wall	0.22	2,243
Conductor Support &	2,141	• Foamed Glass Tubing (L)	0.2	2,145
Waveguide Support	97	• Aluminum End Fittings (L)	0.5	98
(SMF - Handling Consid)				
Strut Assembly Nodes	184	Machined Aluminum Castings	2.23	188.2
(Production Site Optional)	170.6	• Aluminum (L)	2.0	174.1
	13.4	• Alloying Agents (L)	5.0	14.1
Foamed Glass Tubing	27,539	25 & 50 cm Tube with 1 cm Wall	0.34	27,634
(Produced at SMF Due	27,181	• Refined Natural Glass (L)	0.2	27,236
to Low Density and	358	• Foaming Agents (E)	10.0	398
Handling Considerations)		$NaSO_4$, C, H_2O etc.		
Aluminum End Fittings	520	1.0 mm Thick Aluminum Parts	0.5	523
(Produced at SMF Due		• Aluminum Alloy Sheet Stock (L)	0.5	523
to Low Part Density)				

Table D-8 Material requirements for SPS Power Transmission Busses.

Product or Component	Nominal* Quantity (T/Yr)	Origin: Lunar (L), Earth (E)	Unrecoverable Loss Factor (Percent)	Total Quantity (T/Yr)
Sheet Conductors	3,239	1.0 mm Thick Aluminum Sheet	0.1	3,242
(Production Site Optional)		• Aluminum Sheet Stock (L)		
Cable and Wire	802	1.13 mm Diameter Aluminum Wire	0.37	805
Conductors	452	• Aluminum Sheet Stock (L)	0.1	453
(Production Site Optional)		1.0 mm Square Strips		
	350	• Woven Fiberglass Insulation (L)	0.5	352

*Incl 26.6% Margin

Table D-9. Material requirements for SPS MPTS Waveguide Modules.

Product or Component	Nominal* Quantity (T/Yr)	Origin: Lunar (L), Earth (E)	Unrecoverable Loss Factor (Percent)	Total Quantity (T/Yr)
Integrated Klystron Module	20,966	8 cm Thick Panel With Klystrons	0.15	20,997
Qty 13,864 9.9 m × 11.5 m	7,659	• MPTS Waveguide Panel (L) 8 cm Thick Foamed Glass	0.2	7,675
	7,197	• Klystron Assemblies (L/E) 72 kW Klystrons	0.1	7,204
(Produced at SMF Due to Handling Considerations)	1,607	• Radiator Assemblies (L) 1.0 mm Aluminum 1.5 m × 1.7 m	0.1	1,609
	4,159	• Radiator Heat Pipes (L/E) 2.22 cm Dia 0.25 cm Wall CRES	0.1	4,163
	344	• Various Other Components (E) and Attachment Hardware	0.5	346
MPTS Waveguide Panel	7,675	8 cm Thick Foamed Glass Panel	0.20	7,690
Qty 13,902 9.9 m × 11.5 m (Produced at SMF Due to	7,354	• Qty (8) Aluminum Coated (L) Foam Glass Segments/Panel	0.2	7,368
Handling Considerations)	321	• Klystron Mounting Pads (E)	0.5	322
Foamed Glass Panels	7,368	4 cm Thick Foamed Glass Panel	0.62	7,414
Qty 111,400 2.87 m × 9.9 m	7,083	• Refined Natural Glass (L)	0.2	7,097
(Produced at SMF Due to Low Density and	93	• Foaming Agents (E) $NaSO_4$, C, H_2O, Etc.	10.0	104
Handling Considerations)	192	• Vacuum Deposited Aluminum (L)	10.0	213
Klystron Assembly	7,204	72 kW Klystron Assembly	0.33	7,228
Qty 194,112	986	• Solenoid Cavity (L/E) Copper Plated Alum Casting	0.2	988
(Produced at SMF Due to Large Percentage of	351	• Solenoid Housing (L) 1.0 mm Aluminum Alloy Sheet	0.2	352
Earth Components)	167	• Collector Housing (L) 1.0 mm Aluminum Alloy Sheet	0.2	168
	690	• Solenoid Poles (L) 1.0 cm thick Iron Plate	0.2	691
	1,980	• Solenoid Coil Windings (L) Insulated Aluminum Wire	0.2	1,984
	3,030	• Various Other Components (E)	0.5	3,045

*Incl 26.6% Margin

Table D-9. Material requirements for SPS MPTS Waveguide Modules (continued).

Product or Component	Nominal* Quantity (T/Yr)	Origin: Lunar (L), Earth (E)	Unrecoverable Loss Factor (Percent)	Total Quantity (T/Yr)
Solenoid Cavity	988	Copper Plated Aluminum Casting	3.01	1,019
Qty 194,500	874	• Machined Aluminum Casting (L)	2.0	892
(Production Site Optional Although SMF Preferred)	114	• Copper Plating (E)	10.0	127
Solenoid Coil Windings	1,984	1.13 mm Diameter Aluminum Wire	0.65	1,999
	1,890	• Aluminum Sheet Stock (L) 1.0 mm Square Strips	0.2	1,894
(Production Site Optional, Although SMF Preferred)	94	• Plastic Insulation Coating (E)	10.0	105
Klystron Radiator Assy	1,609	1.5 m × 1.7 m Aluminum Sheet	0.20	1,612
Qty 198,470	1,606	• 1.0 mm Alum Sheet Stock (L)	0.2	1,609
	3	• Surface Chemical Treatment For High Emittence (E)	10.0	3
(Produced at SMF Due to Handling Considerations)				
Radiator Heat Pipes	4,163	2.22 cm Dia Pipe 2.67 m Long	0.29	4,175
Qty 1.16×10^6	3,812	• Steel Sheet Strip (L) 7.0 cm Wide × 0.25 cm Thick	0.2	3,820
(Produced at SMF Due to Low Transport Density)	14	• Detacouples (E)	10.0	16
	337	• Mercury Transport Fluid (E)	0.5	339
Alloy Steel for Heatpipes	3,820	Sheet Steel Strip (70.5 Fe - 1.0 Mn - 0.5 Si - 18.0 Cr - 10.0 Ni)	1.65	3,884
	2,693	• Iron (L)	0.2	2,698.4
	19.4	• Alloying Elements (L)	0.2	19.4
	1,108	• Alloying Elements (E)	5.0	1,166

*Incl 26.6% Margin

Table D-10. Material requirements for SPS DC-DC Converter Assembly.

Product or Component	Nominal* Quantity (T/Yr)	Origin: Lunar (L), Earth (E)	Unrecoverable Loss Factor (Percent)	Total Quantity (T/Yr)
DC-DC Converter Assembly	4,335	5.4 MW Transformer With Radiator	0.10	4,340
Qty 456	2,594	• Transformer Assembly (L/E)	0.1	2,597
	1,741	• Radiator Assembly (L/E)	0.1	1,743
Transformer Assembly	2,597	5.4 MW Converter Package	0.20	2,605
Qty 456	1,257	• Sendust Transformer Core (L) Alum-Iron-Silicon Casting	0.2	1,260
(Production at SMF Due to Large Percentage of Earth Components)	553	• Transformer Coil Windings (L) Insulated Aluminum Wire	0.2	554
		• Converter Housing		
	787	• Electronic Controls and (E) Various Other Components	0.5	791
Radiator Assembly	1,743	9 m × 40 m Radiator Panel	0.29	1,748
Qty 456	1,315	• 1.0 mm Alum Sheet Stock (L)	0.2	1,318
	2	• Surface Chem Treatment (E)	10.0	2
(Production at SMF Due to Low Density and Handling Considerations)	66	• Aluminum Tubing (L) 1.0 mm Thick Alum Alloy Sheet	0.5	66
	360	• Various Incl Transport (E) Fluid, Pumps, Valves Etc	0.5	362
Transformer Coil Windings	554	1.13 mm Diameter Aluminum Wire	0.72	558
	528	• Aluminum Sheet Stock (L) 1.0 mm Square Strips	0.2	529
	26	• Plastic Insulation Coating (E)	10.0	29

*Incl 26.6% Margin

APPENDIX E

Task 5.3 supplementary data supporting transportation analysis and vehicle definitions in Volume II, Section 4 (Subsections 4.2 and 4.7)of Final Report.

Appendix E contains four sections.

E.1 — LOW ACCELERATION TRANSFERS FROM LOW EARTH ORBIT TO LOW LUNAR ORBIT - ANALYSIS FOR LUNAR RESOURCE UTILIZATION STUDY

by Lane Cowgill of General Dynamics Convair

SUMMARY:

Data presented in this memo show trajectory characteristics and performance capability for low thrust transfers between low Earth orbit and low lunar orbit using solar electric propulsion (SEP). One way transfer times of six months or less were considered. Trajectory data were generated for initial thrust to weight ratios (T/W) from 6×10^{-5}g to 1×10^{-4}g using SECKSPOT, a computer program for simulating solar electric orbital transfers. The thrust pointing direction time histories for these trajectories were optimized to yield minimum transfer time for the given T/W. The degradation effects of Van Allen belt radiation on solar cell power were considered. Shadowing by both the Earth and the Moon was taken into account. The primary results of this study define transfer time and ideal velocity as functions of initial T/W (Figures 6 and 7).

INTRODUCTION:

This analysis was performed in support of Lunar Resources Utilization (Reference 1), a 10 month study program dealing with the utilization of lunar materials for construction of very large structures in Earth orbit. A significant part of the Reference 1 study is a conceptual definition of the transportation system requirements; such a transportation system would have many elements, including a vehicle designed for transporting massive cargos between low Earth orbit (LEO) and low lunar orbit (LLO). This vehicle is envisioned as having a total mass of up to 15,000 metric tons. The objective of this analysis is to define some of the trajectory and performance characteristics of the low acceleration LEO to LLO transfers for this vehicle. Solar electric propulsion (SEP) was assumed. A possible alternative to SEP is the mass driver reaction engine (MDRE) described in Reference 2. The results obtained in this analysis can be applied to either type of propulsion system.

DISCUSSION AND RESULTS:

Program SECKSPOT (References 3 and 4) was used to generate the low thrust trajectory data for this study. SECKSPOT computes time optimal trajectories for low thrust solar electric orbital transfer. A method of averaging reduces computation time such that analyses of orbital transfers with continuous thrusting lasting months (or years) are feasible. The optimum thrust pointing direction history is calculated using a calculus of variations formulation such that the desired target orbit is achieved with minimum transfer time. The effects of solar cell power degradation due to Van Allen belt radiation were modeled in this study, as were the effects of shadowing by the Earth and the Moon.

GROUND RULES

Trip Time: One round trip between LEO and LLO per year (6 months or less each way)

Total Initial Vehicle Mass: 15,000 metric tons (3.3×10^7 lbs)

SEP Propellant: ARGON (Outbound from LEO to LLO)
OXYGEN (Return trip)

LEO Characteristics: Semi-major Axis, a = 6856 km
Eccentricity, e = 0.
Inclination, i = 31.6 deg

LLO Characteristics: Semi-major Axis, a = 1788 km
Eccentricity, e = 0.
Inclination, i = 0. deg

Constants: Earth Radius = 6378 km
Moon Radius = 1738 km
Earth Gravitation Constant = 398601.2 km^3/sec^2
Moon Gravitation Constant = 4901.8 km^3/sec^2
Moon's Mean Orbital Speed = 1.0183044 km/sec
Moon's Mean Orbital Radius = 384,400 km

Lunar Arrival Date: 1990

Thruster Characteristics (Reference 5)

Propellant	Argon	Oxygen
Specific Impulse, I_{sp} (sec)	7150	7396
Power per Thruster (kw)	160.0	117.2
Thrust per Thruster (N)	3.25	2.03
Thruster Efficiency, η	0.71	0.63
Mass per Thruster (kg)	22	22

Solar Cell Characteristics (Reference 6)

Cell Thickness: 6 mils
Front Shield Thickness: 6 mils
Back Shield Thickness: 20 mils
Base Resistivity: 10 ohm-cm
Power Per Unit Mass: 150 w/kg

TARGETING TECHNIQUE

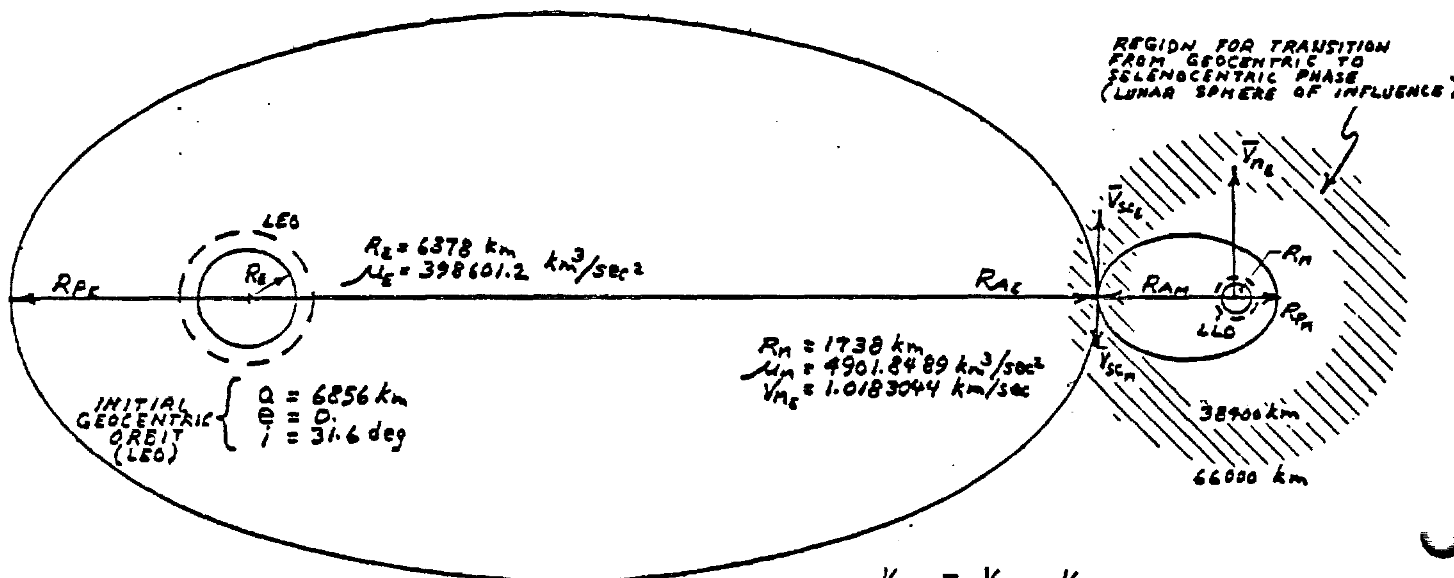
SECKSPOT was developed for simulating solar electric geocentric orbital transfers; however, it was found to be adaptable to the analysis of LEO to LLO transfers by separating the ascent into two parts, corresponding to geocentric and selenocentric phases (the programmed central body constants can be overridden by program input). The method is illustrated in Figure 1. The initial orbit and the final (target) orbit (for both the geocentric and selenocentric phases) were defined via SECKSPOT input by specifying their respective semi-major axes (a), eccentricities (e), and inclinations (i). The interface between the geocentric and selenocentric phases was placed at the boundary of the lunar sphere of influence, specified in Reference 7 as having a selenocentric radius of 66,000 km. For flexibility in targeting, however, the transition from the geocentric to selenocentric phases was allowed to occur within a region, or shell as illustrated, having its outer boundary defined by a radius of 66,000 km and its inner boundary defined by a radius of 38,400 km, at which point the magnitudes of the gravitational accelerations of the Earth and the Moon acting on the spacecraft are equal.

It was orginally intended to vary a_E and e_E (within the ranges that would produce posigrade lunar orbits) to find the optimum combination that would result in minimum ideal velocity for the total mission. However, because of iteration convergence difficulties (typically associated with calculus of variations optimization programs such as SECKSPOT) and the limitations of time and funding, a single set of geocentric target orbit parameters (and corresponding initial selenocentric orbit) was adopted. The elements of this orbit are defined in Figure 4. It should be noted, however, that preliminary efforts at parametrically varying a_E and e_E indicate that performance variations are very small within the allowable range of values.

Inclination, the third orbital element, was targeted in the geocentric phase such that the spacecraft and lunar orbits were coplanar at lunar encounter. In the selenocentric phase, an additional plane change maneuver of about 7 degrees was simulated such that the final orbit (LLO) was coplanar with the lunar equator. Orbital inclination was always specified relative to Earth's equator in order to preserve the validity of the SECKSPOT shadowing calculations for selenocentric as well as geocentric orbits.

Figure 1

METHOD USED FOR TARGETING LEO-TO-LLO TRAJECTORIES USING PROGRAM SECKSPOT



$$V_{sc_M} = V_{M_E} - V_{sc_E}$$

$$R_{A_M} = 384{,}400 \text{ km} - R_{A_E}$$

$$R_{P_M} = \frac{(R_{A_M} V_{sc_M})^2}{2\mu_M - R_{A_M} V_{sc_M}^2}$$

FINAL GEOCENTRIC ORBIT (TARGET)

$$a_E = \frac{R_{A_E} + R_{P_E}}{2}$$

$$e_E = \frac{R_{A_E} - R_{P_E}}{R_{A_E} + R_{P_E}}$$

$$i_E = i_{\text{LUNAR ORBIT PLANE}} = 27.5^\circ \ (yr \approx 1990)$$

INITIAL SELENOCENTRIC ORBIT

$$a_M = \frac{R_{A_M} + R_{P_M}}{2}$$

$$e_M = \frac{R_{A_M} - R_{P_M}}{R_{A_M} + R_{P_M}}$$

$$i_M = 27.5 \text{ deg}$$

$$V_{sc_E} = \sqrt{\frac{\mu_E}{a_E}\left(\frac{1 - e_E}{1 + e_E}\right)}$$

FINAL SELENOCENTRIC ORBIT (TARGET) (LLO)

$$a = 1788 \text{ km}$$
$$e = 0.$$
$$i = 0. \text{ (REL. TO MOON EQUATOR)}$$

Since the inclination of the lunar orbit plane with respect to Earth's equator varies with time, the magnitude of the required orbital plane change is a function of lunar arrival time. Because the moon's orbit precesses about the ecliptic plane, the geocentric inclination of the moon's orbit varies between 18.3 degrees and 28.6 degrees through an 18 year cycle as shown in Figures 2 and 3. The adopted target inclination of about 27.5 degrees corresponds to an assumed lunar arrival in 1990. Figure 2 shows that the moon's equatorial plane is inclined about 6.7 degrees with respect to its orbit plane. Consequently, an orbital plane change maneuver of this magnitude was simulated in the selenocentric phase to achieve the desired equatorial orbit.

To summarize, the transfer from LEO to LLO was targeted in two steps: step one started in the initial low Earth orbit (LLO) followed by a spiral ascent outward to the vicinity of the lunar orbit where capture by the Moon was assumed (the orbit phasing problem was not analyzed). Step 2 started in the corresponding initial selenocentric orbit followed by a spiral inward to achievement of the final low lunar orbit (LLO).

COMMENTS ON ADAPTING SECKSPOT FOR SIMULATING NON-GEOCENTRIC ORBITAL TRANSFERS

Because SECKSPOT was developed for use in simulating geocentric orbital transfers, some adaptation was required to use it for selenocentric transfers. The adaptation was accomplished with three input quantities; the central body gravitation constant (μ), radius (R_o), and specific impulse (I_{sp}). It was necessary to implement an artificial value of I_{sp} because of the way the propulsion equations are implemented in SECKSPOT coding. Thrust, mass flow rate, and acceleration are given by:

$$T = \frac{2\eta D P_e}{I_{sp}\, g_o} \qquad \dot{m} = \frac{2\eta D P_e}{(I_{sp}\, g_o)^2} \qquad a = \frac{2\eta D P_e}{I_{sp}\, g_o\, m}$$

where:

T = thrust (N)	P_e	= initial, undegraded power (watts)
m = mass (kg)	I_{sp}	= specific impulse (sec)
$\dot{m}$ = mass flow rate (kg/sec)	g_o	= mass-to-weight conversion factor (m/sec^2)
a = acceleration (m/sec^2	$I_{sp} g_o$	= exhaust velocity (m/sec)
η = Thruster power efficiency	D	= power degradation factor

Figure 2. Earth-Moon Orbital Relationships

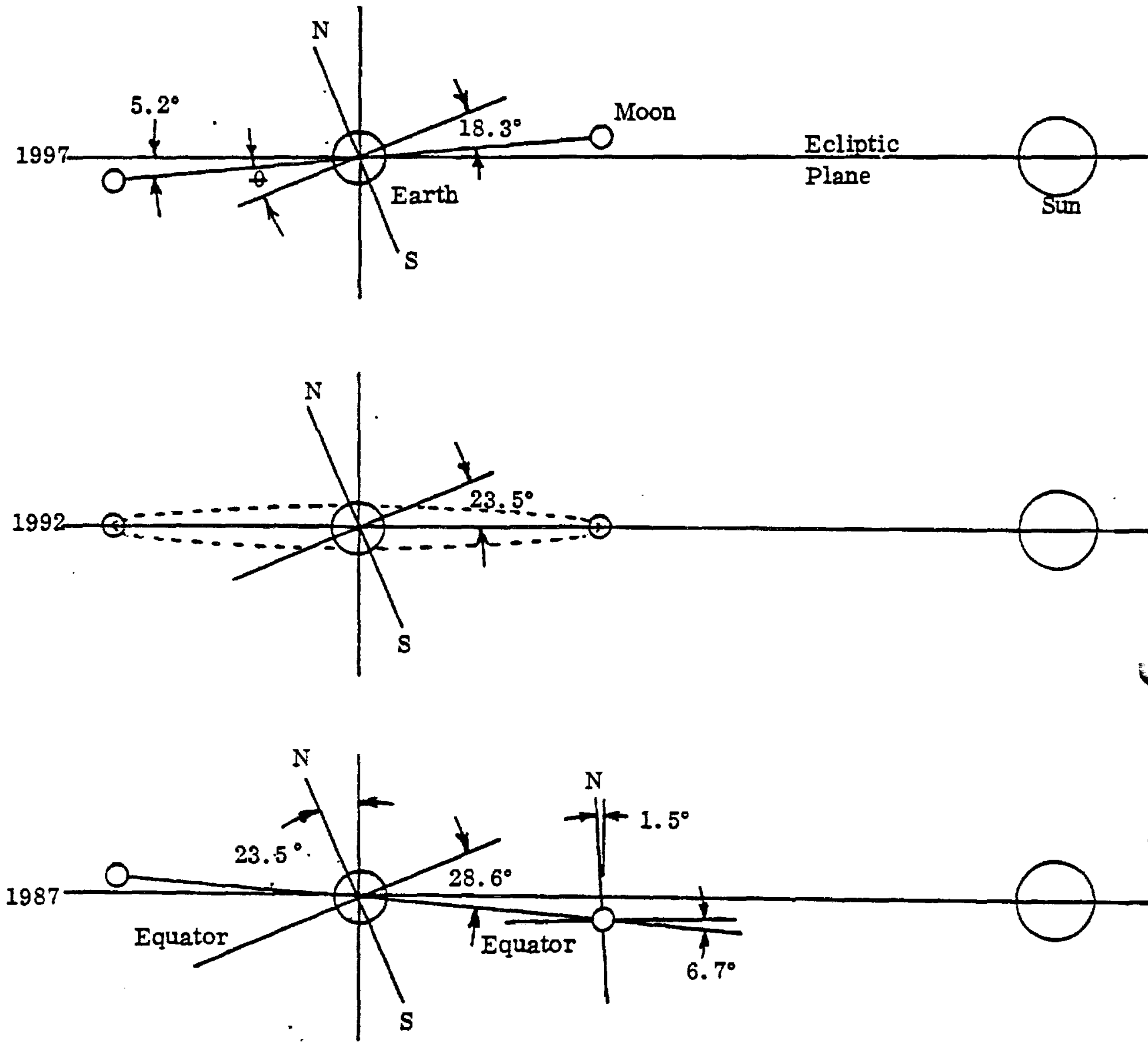
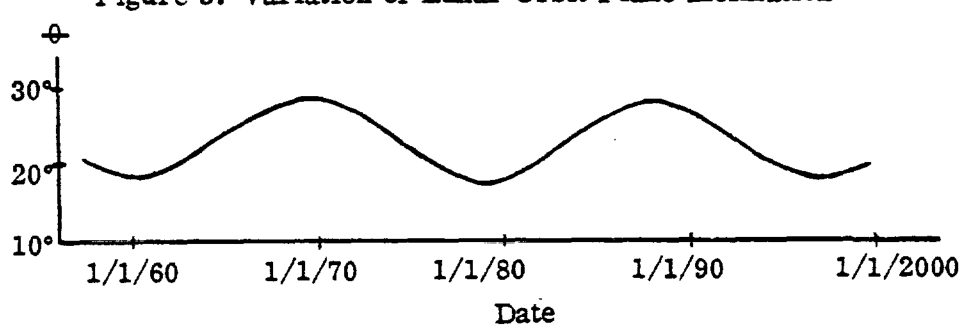


Figure 3. Variation of Lunar Orbit Plane Inclination



SECKSPOT calculates $g_o = \frac{\mu \times 10^3}{R_o^2}$

where μ = gravitation constant (km^3/sec^2)

R_o = central body radius (km)

When the input values of μ and R_o are not Earth values, the calculated values of exhaust velocity and, hence, acceleration are incorrect. Rather than modify SECKSPOT coding, an artificial value of I_{sp} (designated as I'_{sp}) was used in the selenocentric phase trajectory simulations:

$$I_{sp}' = I_{sp} \frac{R_{o_M}^2 \mu_E}{R_{o_E}^2 \mu_M} = I_{sp} \frac{(1738)^2 (398601.2 \text{ Km}^3/\text{sec}^2)}{(6378)^2 (4901.8 \text{ Km}^3/\text{sec}^2)} = 6.038 \, I_{sp}$$

RESULTS

Figure 4 illustrates an ascent trajectory from LEO to LLO for an initial acceleration of 1.0×10^{-4} g. The instantaneous orbit is shown at various times from the start of orbital transfer. The effects of solar cell power degradation and shadowing are included. The spacecraft enters lunar orbit at about 120 days and then spends another 25 days spiraling down to LLO. Figure 5 shows how several parameters of interest vary with time. The discontinuities in the curves at 120 days reflect the transition from geocentric to selenocentric phase. Note that virtually all of the orbital plane change is accomplished at maximum distance from the central body. Note also that shadowing effects decrease with increasing orbit size. At about 50 days, after passage through the Van Allen radiation belt, solar cell power has stabilized at about 66 percent of the undegraded value.

Table 1 shows that the total ideal velocity requirement for an initial acceleration of 1.0×10^{-4} g is 8.176 km/sec and the total transfer time from LEO to LLO is 145 days. Two sets of vehicle masses are shown corresponding to argon and oxygen propellant. The masses shown are based on the thruster and solar array characteristics ground rules presented earlier in this section.

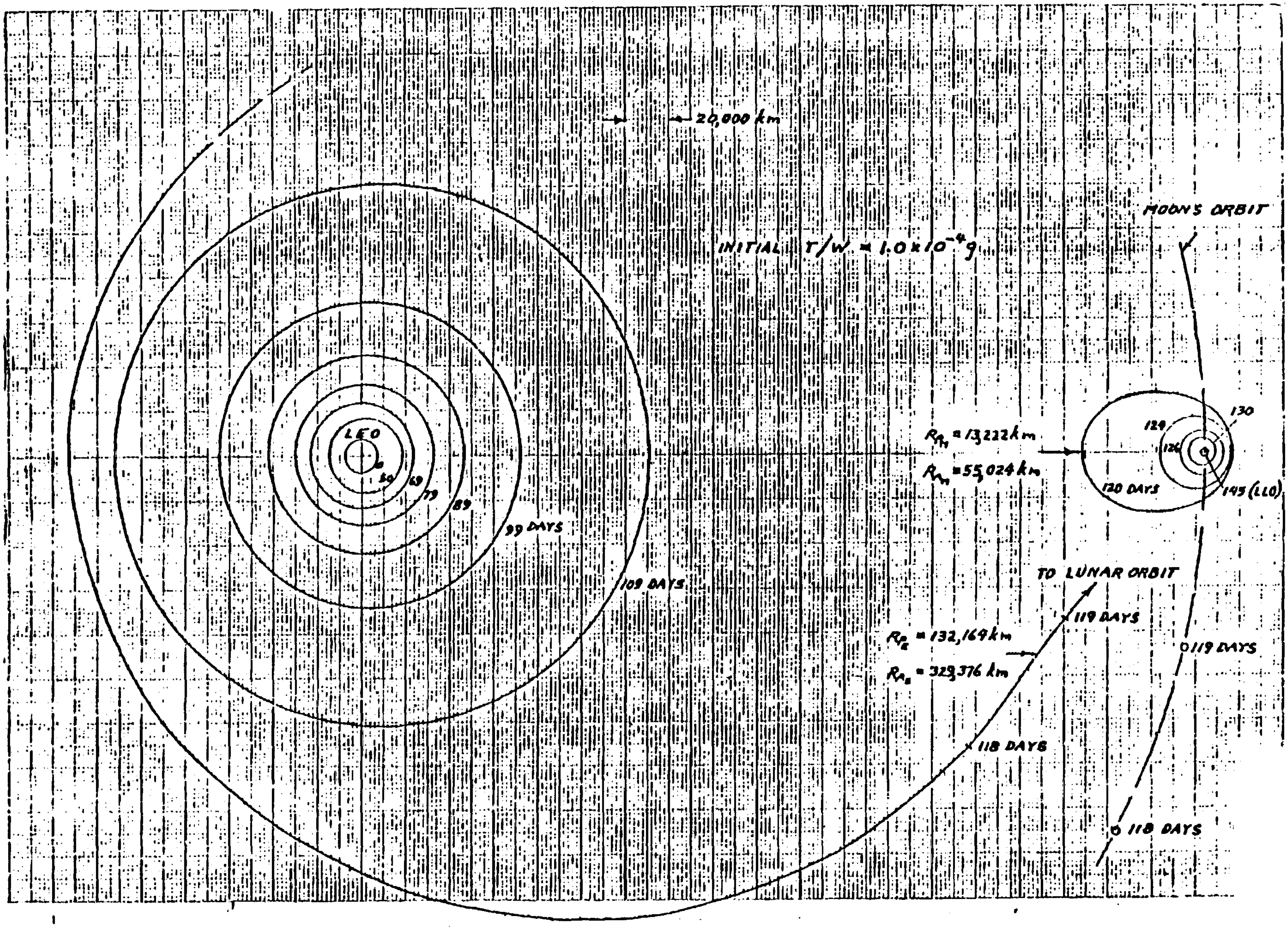


Figure 4. Ascent from LEO to LLO

Figure 5. Variation of Orbit Parameters

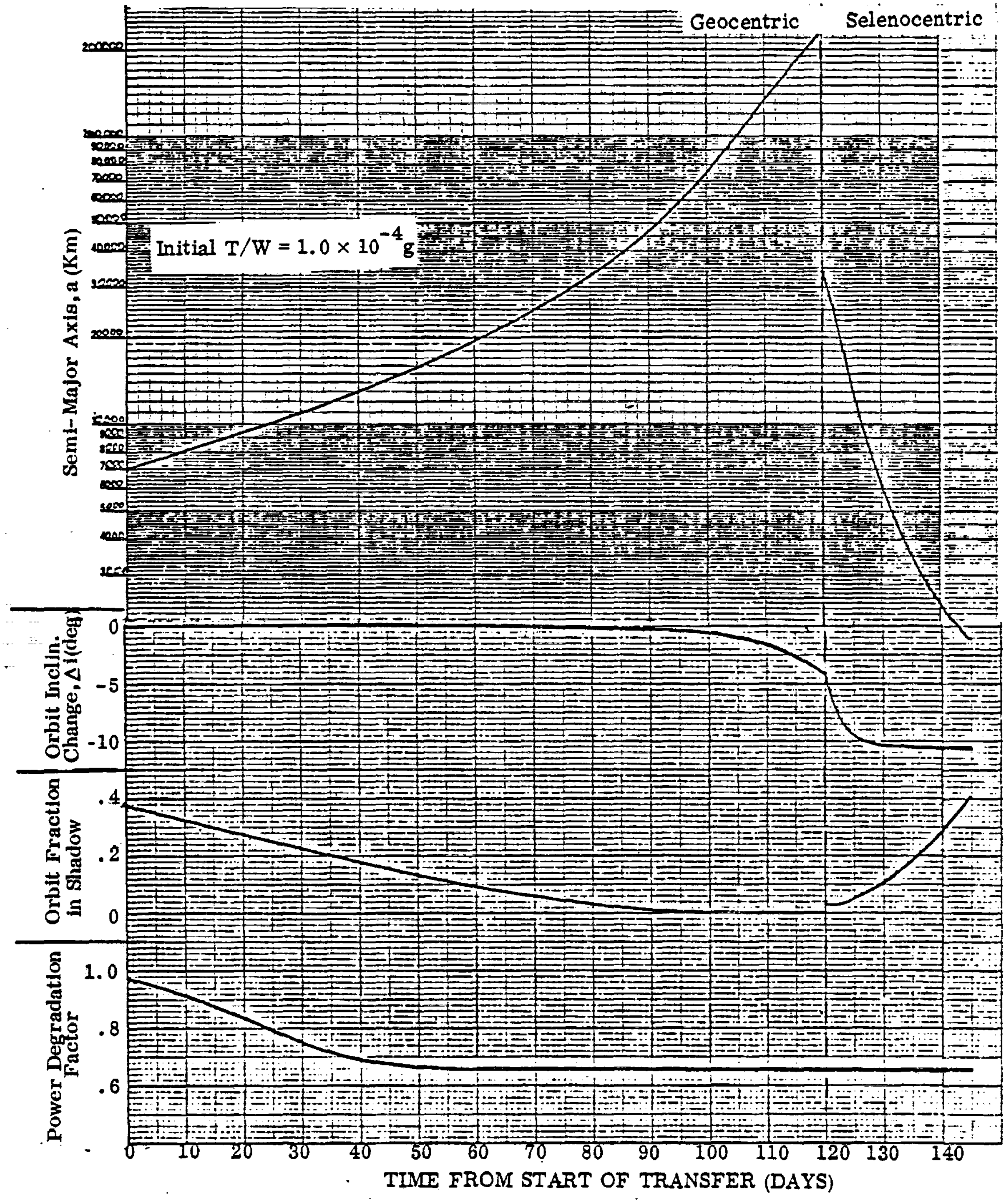


Table 1. Performance Summary

- INITIAL THRUST-TO-WEIGHT RATIO = 1.0×10^{-4} g
- INITIAL MASS = 1.5×10^{7} kg

	IDEAL VELOCITY (km/sec)	TRIP TIME (days)
Geocentric Phase	6.833	120
Selenocentric Phase	1.343	25
Total	8.176	145
Propellant	ARGON	OXYGEN
Mass (kg $\times 10^{-3}$)		
Payload	8233	7396
Solar Array	5013	5844
Thrusters	100	160
Propellant	1654	1600
Total	15000	15000

Figure 6 shows total transfer time from LEO to LLO as a function of initial T/W. Two curves are shown; one includes the effects of power degradation and shadowing (applicable to a solar electric system) and one excludes these effects (corresponding to a nuclear electric system). Figure 7 shows ideal velocity requirement, also as a function of initial T/W, for the same two cases.

REFERENCES:

1. "Lunar Resources Utilization for Space Construction," General Dynamics report No. GDC-PIN77-376, 22 December 1977.

2. O'Neill, Gerard K., "The Low (Profile) Road to Space Manufacturing," Astronautics and Aeronautics, March 1978.

3. Sackett, Lester L., Malchow, Harvey L, and Edelbaum, Theodore N., "Solar Electric Geocentric Transfer with Attitude Constraints: Analysis," The Charles Stark Draper Laboratory, Inc. Report No. R-901, August 1975.

4. Sackett, Lester L., et. al., "Solar Electric Geocentric Transfer with Attitude Constraints: Program Manual," The Charles Stark Draper Laboratory, Inc. Report R-902, August 1975.

5. Byers, Dave (NASA/LeRC), telecon with J.W. Streetman (GDC), 27 June 1978.

6. Wilson, L. G., "Time Optimal Trajectories for Transfers from LEO to GEO Using Solar Electric Propulsion - Analysis for On Orbit Assembly Study," General Dynamics Convair Division Memorandum No. 697-0-77-120, 8 February 1978.

7. Levantovskiy, V.I., "Space Flight Mechanics in Elementary Presentation," edited machine translation from Mekhanika Kosmicheskogo Poleta v Elementarnom Izlozhenii, 1970, Izd vo Nauka, Moscow, pp 1-491.

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Trajectory Analysis

Figure 6. Transfer Time

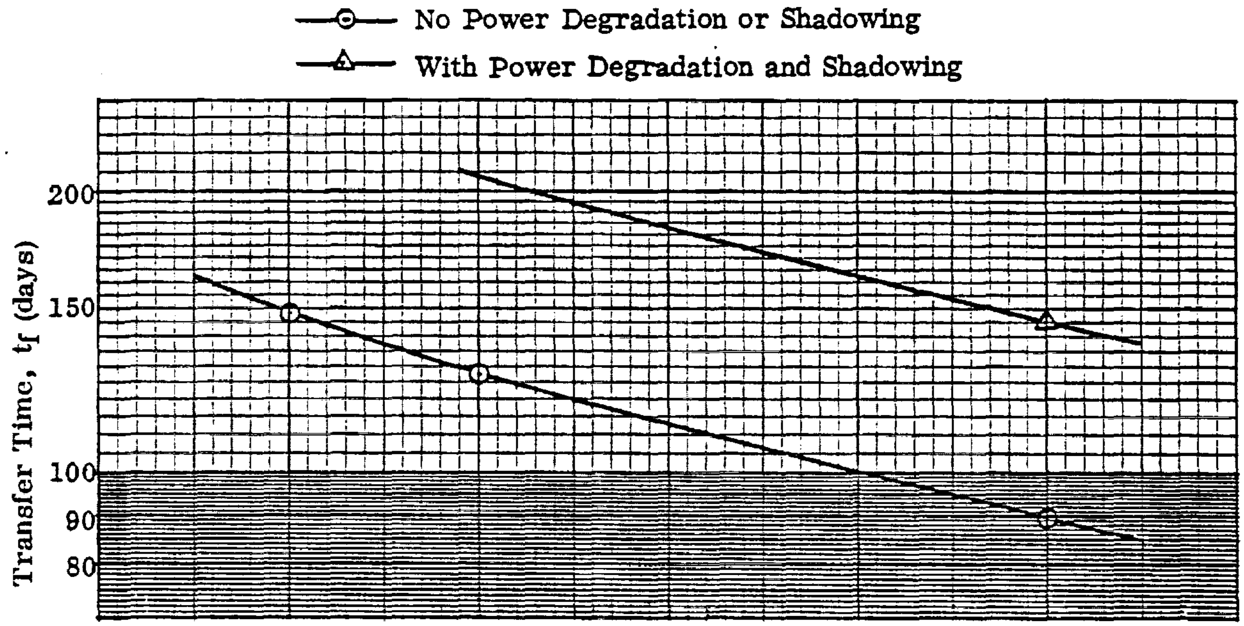
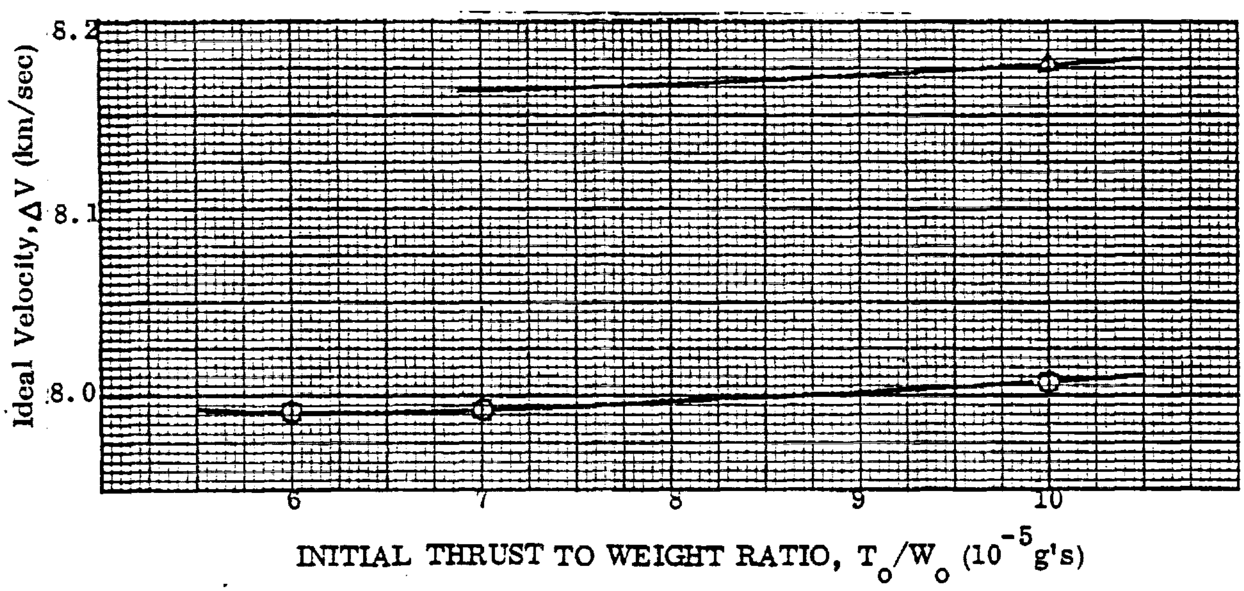


Figure 7. Ideal Velocity



		GEOCENTRIC	SELENOCENTRIC
Start Transfer From LEO	APOAPSIS (km)	6856	—
	PERIAPSIS (km)	6856	—
	INCLINATION (deg)	31.6	—
Lunar Encounter	APOAPSIS (km)	329376	55024
	PERIAPSIS (km)	132164	13222
	INCLINATION (deg)	27.5	6.7
Transfer Complete at LLO	APOAPSIS (km)	—	1788
	PERIAPSIS (km)	—	1788
	INCLINATION (deg)	—	0

E.2 — PRELIMINARY STUDY OF PERFORMANCE AND FEASIBILITY OF A HEAVY PAYLOAD SHUTTLE DERIVED VEHICLE (SDV)

SUMMARY

A study has been made to determine the feasibility and performance of a Heavy Payload Launch Vehicle (HPLV) derived from the Space Shuttle Orbiter and the final Phase B Shuttle Flyback Booster design. This Shuttle Derived Vehicle (SDV) would use a space shuttle orbiter modified structurally to accept the higher payload and a LO_2/Propane version of the final fully reusable heat sink LO_2/H_2 booster defined in 1971 at the end of the Phase B study. It was postulated that this approach would yield a fully reusable vehicle based on existing technology and design with payload in the 350,000 to 500,000 lb. class.

General groundrules used in this study were as follows:

(1) Booster modified to change from LO_2/LH_2 to LO_2/propane propellants. External envelope and structural design unchanged except for required strengthening.

(2) Booster airbreathing flyback engine and associated subsystems removed. Booster lands downrange from launch site.

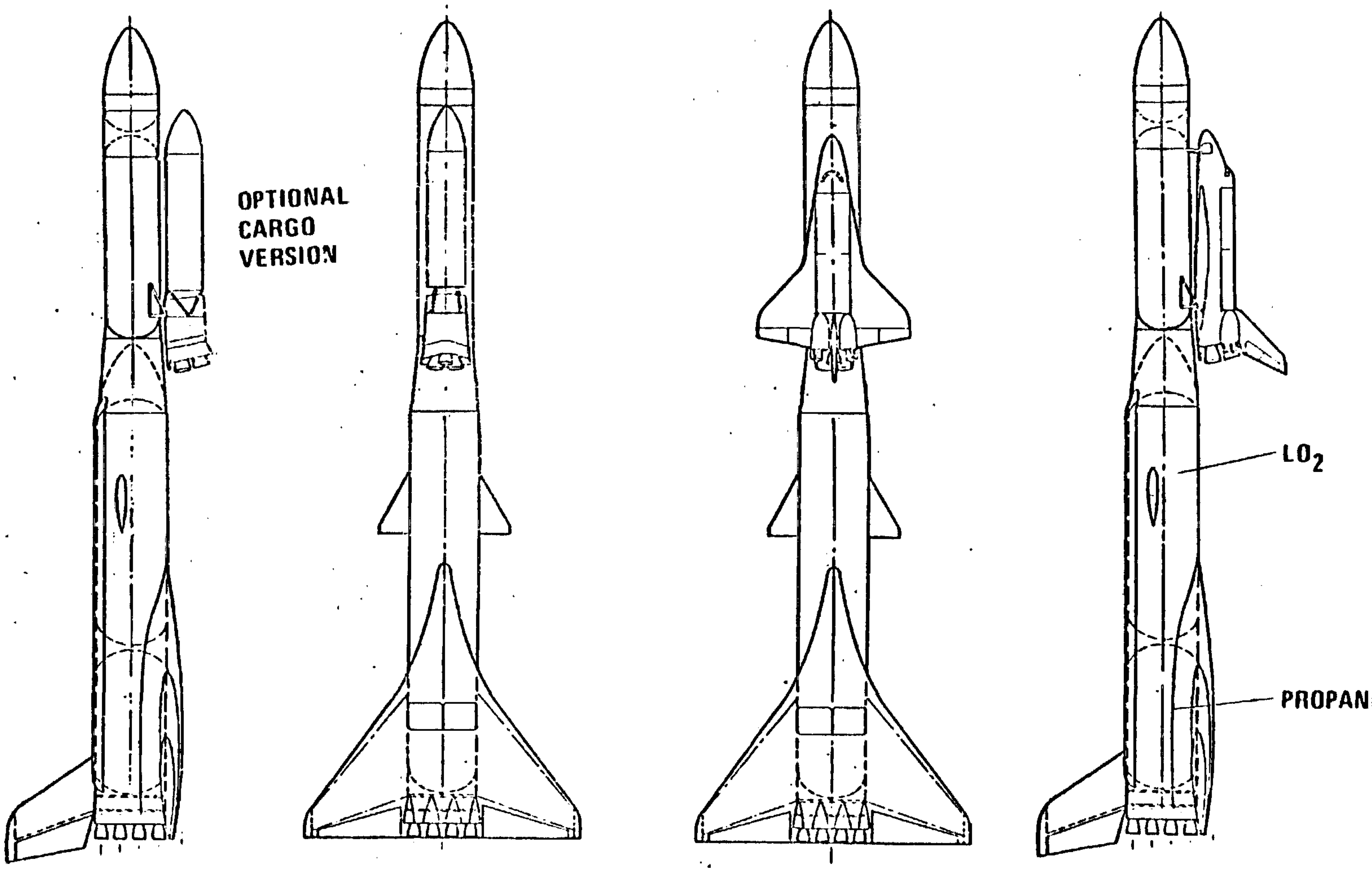
(3) Orbiter modified only as required to support heavier payload.

(4) Orbiter external tanks modified as required to accept boost ascent loads from the tandem mounted booster through the aft bulkhead Y-ring instead of from the side mounted SRBs.

The SDV based on these groundrules is shown in Figure 1. Payload is 373,000 lb. and 295,000 lb. for ETR and WTR launch respectively. Payload for the all cargo version is estimated at 443,000 lb. for ETR launch. Additional weight associated with the higher thrust LO_2/propane engines will probably require reshaping of the booster wing or a 15-20% reduction in projected engine weight to regain the required entry stability margins.

Additional data for the SDV and the LO_2/LH_2 booster on which the SDV booster is based are given in the following sections.

Figure 1. Heavy Payload Shuttle Derived Vehicle (SDV)



Weights (Lb)		Booster	Orbiter	Vehicle
Inert		575,800	1,552,000	
Propellant		6,466,000	285,855	
Total		7,041,800	1,837,855	8,879,655
Payload	ETR			373,000
	WTR			295,000
	Cargo ETR			443,000

Final Baseline Shuttle Phase B Flyback Booster

Figure 2 is an inboard profile of the near-final heat sink booster design designated B-17E developed at the end of the shuttle Phase B Study conducted by the North American Rockwell/General Dynamics Convair Division Team. This vehicle was subsequently modified slightly as shown in Figure 3 for compatibility with a tandem mounted orbiter similar to but smaller than the current orbiter design. Further details of the B-17E booster design are shown in Figures 4 through 17. Depth of design and analysis (documented in References 1 and 2) for this vehicle was sufficient to give high confidence in projected vehicle performance in all areas including entry heating and control.

NOTE

Figures 2 through 17 relating to the previous definition on the flyback booster have been deleted from this LRU Final Report Appendix. This was done to reduce printing costs of material which is detailed back-up information.

Shuttle Derived Vehicle

Modifications made to the B-17E booster design to develop the SDV are summarized in Figure 18. External envelope of the vehicle is not modified. Propellant volume is the same as for the LO_2/LH_2 version but the higher bulk density of the LO_2/Propane increases propellant weight from 2,260,300 to 6,466,000 lb. The different volume ratio for LO_2/Propane necessitates relocating the intertank area as shown. Optimum liftoff thrust level for the booster is 12,000,000 lb. Ten 1,200,000 lb. thrust engines were tentatively selected. This is the minimum number which will clearly give acceptable engine out performance at launch. Twelve 1,000,000 lb. thrust engines may have advantages for packaging within the available base area.

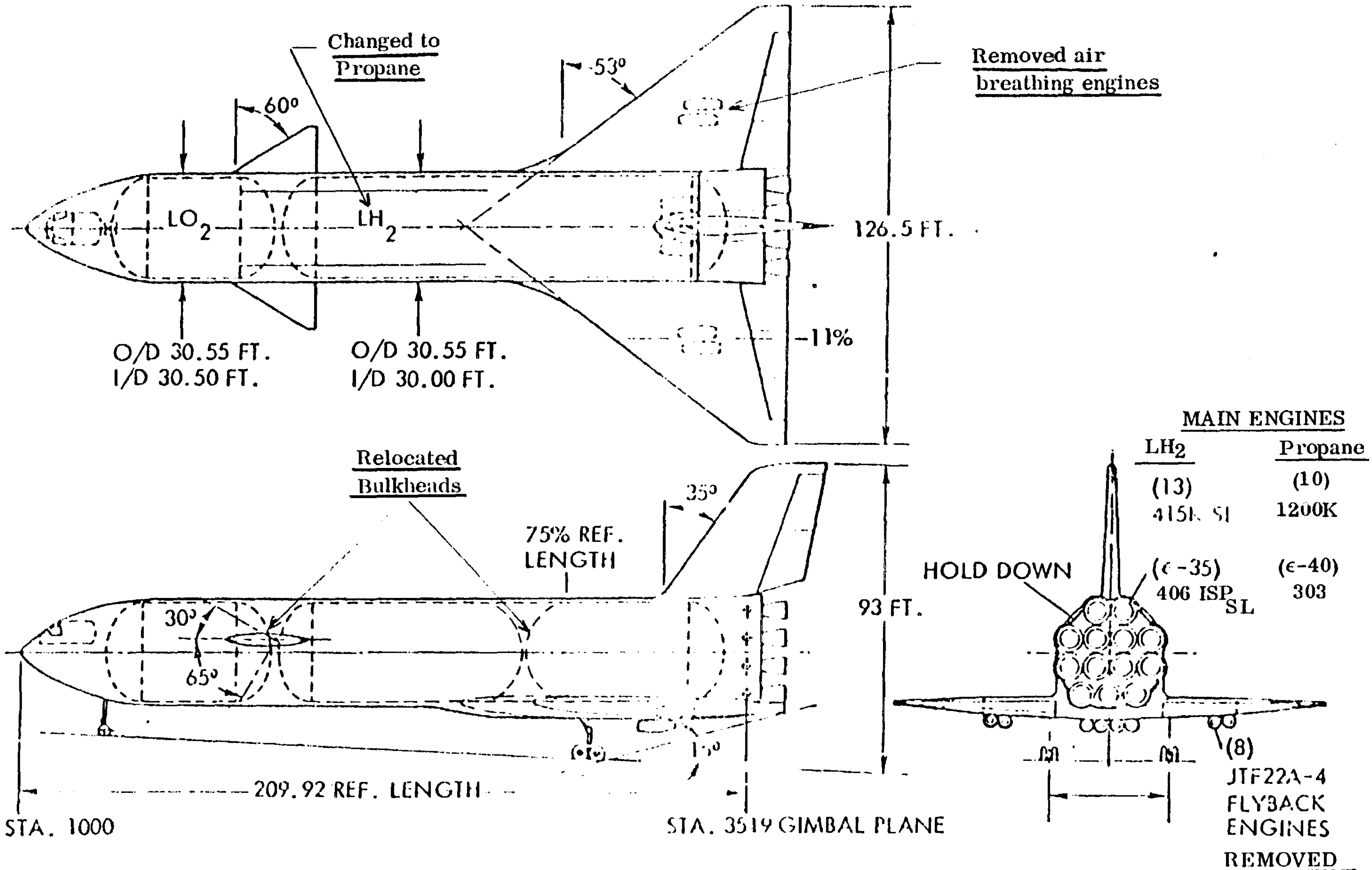
Synthesis summaries for the SDV for a WTR launch to 90 degree inclination and a ETR launch to 28.5 degree inclination are given in Figure 19 and 20. Program iterations were not continued until an exact match was obtained on all weights. These figures summarize weight, volume, geometry, propulsion, and trajectory data. Further trajectory data is given in Figure 21. Detail booster weights are given in Figure 22 & 23.

Trajectory-Trajectory data is summarized in Figure 21. The 3-g maximum acceleration constraint required throttling of the booster engines to 62 percent of liftoff power. A staging flight path angle of 20 degrees was found to give the highest payload. The "roller coaster" altitude profile indicates that a somewhat higher injection altitude may give higher performance. This was not evaluated for this study.

Entry Center of Gravity - Use of the higher thrust and therefore heavier engines required for the SDV moves the empty CG approximately 10 ft. aft of the LO_2/LH_2 booster location. Entry stability appears to be unacceptable if aft movement is more than 6 feet. Aft movement can be limited to less than 6 feet if engine weight can be reduced 15 percent from the value assumed for this analysis. The engine weight model used assumed that the LO_2/Propane engine weighed 80 percent of an equivalent LO_2/LH_2 engine. Further analysis is required to determine if the required further reduction is feasible. If not, the wing may have to be reshaped to regain stability.

Main Propulsion System - The main propulsion system of the booster for the shuttle derived vehicle utilizes ten 1.2 million pound sea level thrust engines burning liquid oxygen and propane. A table of predicted performance of the engines versus that of the Phase B liquid oxygen/liquid hydrogen baseline is shown in the following table:

Figure 18. Modifications to B-17E Heat Sink Booster



SYNTHESIS SUMMARY FOR BOTH ORBITER AND BOOSTER STAGES

GDC B17E REUSABLE/ET/HPLV

	BOOSTER	ORBITER	VEHICLE
WEIGHT (LB)			
PROPELLANT, ASCENT LESS FPR	6465989.	1557601.	
PROPELLANT, ASCENT FPR	0.	8766.	
PROPELLANT, ASCENT TOTAL.	6465989.	1566367.	
PROPELLANT, ORBIT MANEUVER		28386.	
PROPELLANT, TOTAL	6465989.	1594753.	
FLYBACK FUEL	0.	0.	
PAYLOAD		295098.	
STRUCTURE	525696.	228516.	754123.
CONTINGENCY	318[illegible].	6064.	
OTHER	17255.	12923.	
TOTAL	7940651.	2137355.	9178005.
IN ORBIT		579754.	
RETURN CONDITION	574661.	267748.	
ENTRY	557883.	235660.	
LANDING	557883.	235660.	
VOLUME (FT3)			
FUEL TANK	36123.	54672.	
OXIDIZER TANK	69862.	19915.	
PROPELLANT TANKS	105985.	74587.	
PAYLOAD		61359.	
OTHER	209395.	266362.	
TOTAL	315380.	402308.	
GEOMETRY			
LENGTH (FT)	173.1	236.1	
BODY WETTED AREA (FT2)	26598.4	34165.2	
BODY PLANFORM AREA (FT2)	8239.4	10765.5	
THEORETICAL WING AREA (FT2)	5930.0	1818.0	
WING LOADING (PSF)	94.1	129.6	
PROPULSION			
THRUST-TO-WEIGHT		.65969	1.30000
NO. OF ENGINES	10.	3.	
SL THRUST/ENG NOM/UR	1193141./ 1193141.	361869./	11931407./ 11931407.
VAC THRUST/ENG NOM/UR	1330962./ 1330962.	470000./	13309622./ 13309622.
SL ISP NOM/UR	303.0/303.0	353.4/	303.0/303.0
VAC ISP NOM/UR	338.0/338.0	459.0/	338.0/338.0
TRAJECTORY			
MASS RATIO	3.38420	3.68664	
CHARACTERISTIC VELOCITY (FPS)	12955.	19268.	32222.
MAXIMUM DYNAMIC PRESSURE (PSF)			690.8
STAGING DYNAMIC PRESSURE (PSF)			0.
STAGING VELOCITY (RELATIVE) (FPS)			8670.
STAGING ALTITUDE (FT)			285030.
STAGING FLIGHT PATH ANGLE (RELATIVE) (DEG)			20.000
INJECTION VELOCITY (INERTIAL) (FPS)		26141.	
INJECTION ALTITUDE (FT)		304358.	
INJECTION FLIGHT PATH ANGLE (INERTIAL) (DEG)		-.030	
INJECTION INCLINATION (DEG)		93.51	
FLYBACK RANGE (N MI)			

DRAG LOSS AT STAGING	=	149.50
GRAVITY LOSS A STAGING	=	4150.41
DRAG LOSS AT INJECTION	=	149.50
GRAVITY LOSS AT INJECTION	=	4633.14
MISALIGNMENT LOSS AT INJECTION	=	1279.65

Figure 19. SDV Summary - WTR Launch

SYNTHESIS SUMMARY FOR BOTH ORBITER AND BOOSTER STAGES

GDC B17E REUSABLE/ET/HPLV

	BOOSTER	ORBITER	VEHICLE
WEIGHT (LB)			
PROPELLANT, ASCENT LESS FPR	6465955.	1549353.	
PROPELLANT, ASCENT FPR	0.	9963.	
PROPELLANT, ASCENT TOTAL,	6465955.	1559315.	
PROPELLANT, ORBIT MANEUVER		28389.	
PROPELLANT, TOTAL	6465955.	1587704.	
FLYBACK FUEL	0.	0.	
PAYLOAD		373030.	
STRUCTURE	526677.	228251.	754928.
CONTINGENCY	31800.	60 64.	
OTHER	17335.	13188.	
TOTAL	7041767.	2208238.	9250005.
IN ORBIT		656885.	
RETURN CONDITION	575812.	267770.	
ENTRY	558954.	235395.	
LANDING	558954.	235395.	
VOLUME (FT3)			
FUEL TANK	36124.	54427.	
OXIDIZER TANK	69862.	19825.	
PROPELLANT TANKS	105986.	74252.	
PAYLOAD		61359.	
OTHER	209395.	266362.	
TOTAL	315380.	401973.	
GEOMETRY			
LENGTH (FT)	173.1	236.0	
BODY WETTED AREA (FT2)	26598.4	34138.3	
BODY PLANFORM AREA (FT2)	8239.4	10757.0	
THEORETICAL WING AREA (FT2)	5930.0	1818.0	
WING LOADING (PSF)	94.3	129.5	
PROPULSION			
THRUST-TO-WEIGHT		.63852	1.30000
NO. OF ENGINES	10.	3.	
SL THRUST/ENG NOM/UR	1202501./ 1202501.	361869./	12025006./ 12025006.
VAC THRUST/ENG NOM/UR	1341403./ 1341403.	470000./	13414033./ 13414033.
SL ISP NOM/UR	303.0/303.0	353.4/	303.0/303.0
VAC ISP NOM/UR	338.0/338.0	459.0/	338.0/338.0
TRAJECTORY			
MASS RATIO	3.32250	3.35147	
CHARACTERISTIC VELOCITY (FPS)	12754.	17860.	30615.
MAXIMUM DYNAMIC PRESSURE (PSF)			698.2
STAGING DYNAMIC PRESSURE (PSF)			1.
STAGING VELOCITY (RELATIVE) (FPS)			8529.
STAGING ALTITUDE (FT)			276657.
STAGING FLIGHT PATH ANGLE (RELATIVE) (DEG)			20.000
INJECTION VELOCITY (INERTIAL) (FPS)		26140.	
INJECTION ALTITUDE (FT)		304376.	
INJECTION FLIGHT PATH ANGLE (INERTIAL) (DEG)		.000	
INJECTION INCLINATION (DEG)		28.50	
FLYBACK RANGE (N MI)			

DRAG LOSS AT STAGING	=	150.33
GRAVITY LOSS A STAGING	=	4086.53
DRAG LOSS AT INJECTION	=	150.33
GRAVITY LOSS AT INJECTION	=	4544.97
MISALIGNMENT LOSS AT INJECTION	=	1152.08

Figure 20. SDV Summary - ETR Launch

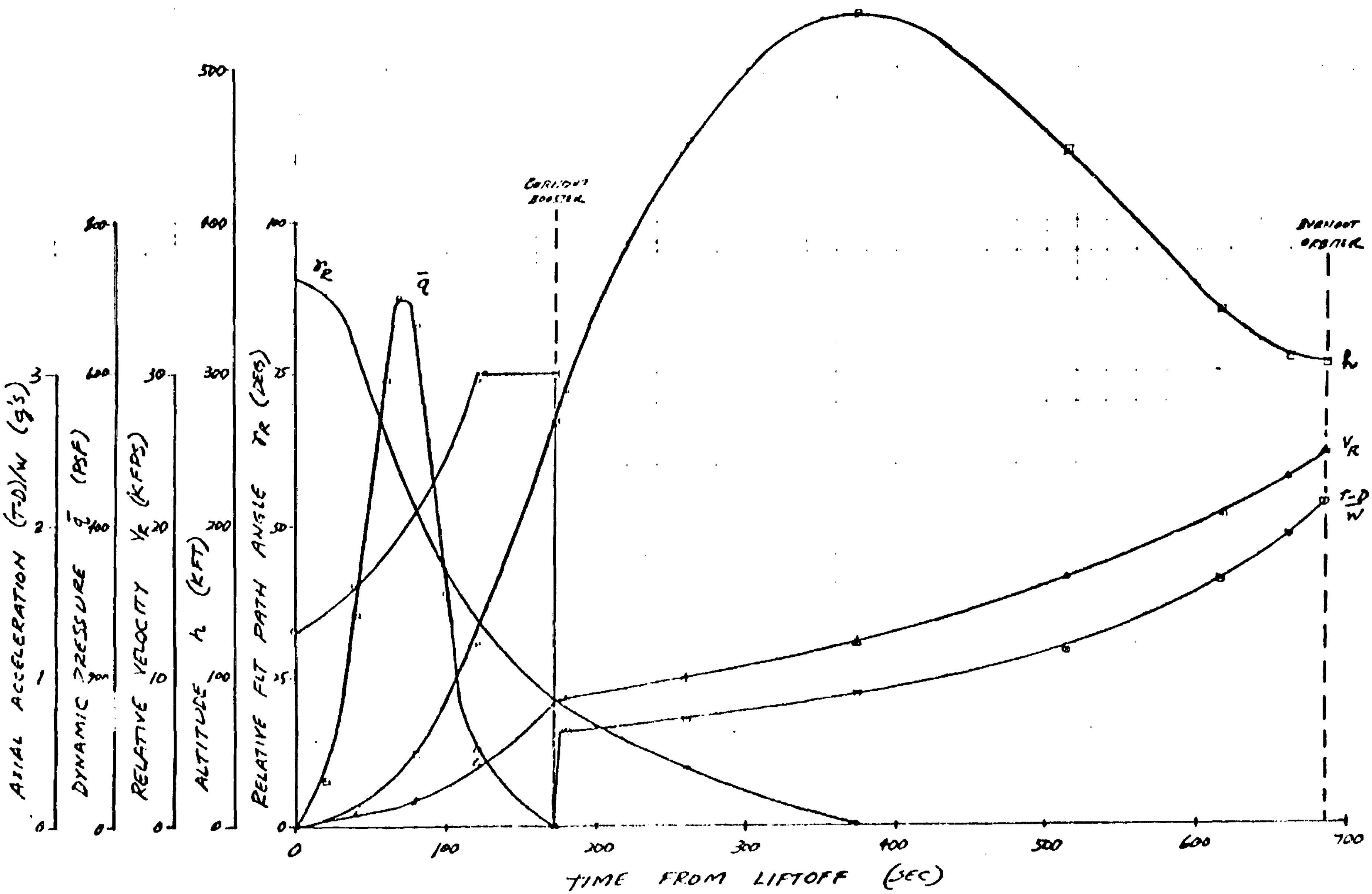


Figure 21. SDV Trajectory Data

GDC 817E REUSABLE/ET/HPLV

WEIGHT BREAKDOWN - BOOST STAGE		POUNDS		KILOGRAMS
AERODYNAMIC SURFACES		93193.		42272.
WING + WING MOUNTED CONTROL SURFACES	57363.		26019.	
HORIZONTAL SURFACES	4747.		2153.	
VERTICAL SURFACES	9383.		4256.	
FAIRINGS,SHROUDS AND ASSOCIATED STRUCTURE	21700.		9843.	
BODY STRUCTURE		119645.		54270.
STRUCTURAL FUEL CONTAINERS	20328.		9221.	
STRUCTURAL OXIDIZER CONTAINERS	26135.		11855.	
BASIC BODY STRUCTURE	26598.		12065.	
SECONDARY STRUCTURE	0.		0.	
THRUST STRUCTURE	46584.		21130.	
INDUCED ENVIRONMENTAL PROTECTION		22086.		10018.
COVER PANELS,NON-STRUCTURAL	18586.		8430.	
VEHICLE INSULATION	3500.		1588.	
LAUNCH,RECOVERY AND DOCKING		76350.		34632.
LAUNCH GEAR	45890.		20815.	
DEPLOYABLE AERODYNAMIC DEVICES	10040.		4554.	
ALIGHTING GEAR	20420.		9262.	
DOCKING STRUCTURE	0.		0.	
PROPULSION		175538.		79623.
ENGINES AND ACCESSORIES	151410.		68678.	
SECONDARY ENGINES AND ACCESSORIES	0.		0.	
FUEL CONTAINERS AND SUPPORTS (NON-STRUCTURAL)	0.		0.	
OXIDIZER CONTAINERS AND SUPPORTS(NON-STRUCTURAL)	0.		0.	
SECONDARY FUEL TANKAGE AND SYSTEMS	0.		0.	
SECONDARY OXIDIZER TANKAGE AND SYSTEMS	0.		0.	
PROPELLANT INSULATION	1921.		871.	
FUEL SYSTEM - MAIN	6655.		3019.	
OXIDIZER SYSTEM - MAIN	13310.		6037.	
PURGE SYSTEMS	2120.		961.	
AIRBREATHING ENGINES AND INSTALLATION	0.		0.	
NACELLES,PODS,PYLONS,SUPPORTS	0.		0.	
AIRBREATHING PROPULSION TANKAGE AND SYSTEMS	123.		56.	
ORIENTATION,SEPARATION AND ULLAGE CONTROL		24389.		11063.
SEPARATION SYSTEMS	0.		0.	
STABILITY AND CONTROL, ENGINE GIMBAL	7933.		3598.	
AERODYNAMIC CONTROLS	9222.		4183.	
SPATIAL ATTITUDE CONTROL SYSTEM	7234.		3281.	
CONTROL PROPELLANT TANKAGE AND SYSTEMS	0.		0.	
PRIME POWER SOURCE AND DISTRIBUTION		3594.		1630.
POWER SOURCE UNITS	3594.		1630.	
POWER SOURCE TANKAGE AND SYSTEMS	0.		0.	
POWER CONVERSION AND DISTRIBUTION		2316.		1051.
HYDRAULIC/PNEUMATIC	2316.		1051.	
GUIDANCE AND NAVIGATION		723.		328.
INSTRUMENTATION		5828.		2644.

Figure 22. SDV Weight Breakdown

COMMUNICATION		553.		251.
PERSONNEL PROVISIONS		1391.		631.
DRY STRUCTURE		(525636.)		(238411.)
DESIGN RESERVE (CONTINGENCY)		31800.		14424.
PERSONNEL		476.		216.
PAYLOAD		2137356.		969488.
CARGO OR UPPER STAGE	2137355.		969488.	
MISSION EQUIPMENT/PAYLOAD	1.		0.	
RESIDUAL PROPELLANT AND SERVICE ITEMS		11141.		5054.
TANK PRESSURIZATION AND PURGE GASES	1726.		783.	
TRAPPED FUEL	4918.		2231.	
TRAPPED OXIDIZER	2662.		1207.	
SERVICE ITEMS RESIDUALS	1836.		833.	
RESERVE PROPELLANT AND SERVICE ITEMS		0.		0.
FUEL-MAIN PROPULSION	0.		0.	
OXIDIZER-MAIN PROPULSION	0.		0.	
POWER SOURCE PROPELLANTS	0.		0.	
LUBRICANTS	0.		0.	
ATTITUDE CONTROL PROPELLANTS	0.		0.	
OPERATING WEIGHT EMPTY		(2706379.)		(1227593.)
IN-FLIGHT LOSSES		1644.		746.
ICE AND FROST	0.		0.	
FUEL VENTED	0.		0.	
OXIDIZER VENTED	0.		0.	
POWER SOURCE PROPELLANTS	150.		68.	
ATTITUDE CONTROL PROPELLANTS	1494.		678.	
LUBRICANTS AND OTHER SERVICE ITEMS	0.		0.	
FLYBACK FUEL	0.		0.	
THRUST DECAY PROPELLANTS		3993.		1811.
PROPELLANTS		6465989.		2932924.
FUEL - MAIN	1747565.		792682.	
OXIDIZER - MAIN	4718425.		2140242.	
FUEL - SECONDARY	0.		0.	
OXIDIZER - SECONDARY	0.		0.	
MASS AT FULL THRUST		9178005.		4163074.
PRE-IGNITION LOSSES		0.		0.
MAXIMUM GROSS WEIGHT		(9178005.)		(4163074.)
GROSS WEIGHT OF ONE BOOSTER		7040651.		3193586.

Figure 22. SDV Weight Breakdown (Cont'd)

GDC 817E REUSABLE/ET/HPLV

		O/F RATIO	ISP	THEORETICAL VELOCITY INCREMENT FEET/SECOND	METERS/SECOND
MASS RATIO DURING FLIGHT PHASE 1=	1.0000	0.0000	0.0	0.0	0.0
MASS RATIO DURING FLIGHT PHASE 2=	1.0000	0.0000	0.0	0.0	0.0
MASS RATIO DURING FLIGHT PHASE 3=	3.3842	2.7000	331.3	12996.4	3961.1
MASS RATIO DURING FLIGHT PHASE 4=	1.0000	6.0000	0.0	0.0	0.0
MASS RATIO DURING FLIGHT PHASE 5=	1.0000	0.0000	0.0	0.0	0.0
MASS RATIO DURING FLIGHT PHASE 6=	1.0000	0.0000	0.0	0.0	0.0

WEIGHT SUMMARY	POUNDS	KILOGRAMS
MAXIMUM GROSS WEIGHT (WGROSS)	9178005.	4163074.
PRE-IGNITION LOSSES	0.	0.
MASS AT IGNITION	9178005.	4163074.
FUEL	0.	0.
OXIDIZER	0.	0.
JETTISON	0.	0.
END OF PRE-FLIGHT PHASE (WTO)	9178005.	4163074.
FUEL	-1747565.	-792682.
OXIDIZER	-4718425.	-2140242.
PAYLOAD ADDED/REMOVED	-2137355.	-969488.
INITIAL SEPARATION	574661.	260662.
THRUST DECAY AND RESIDUALS	-13408.	-6082.
RETURN CONDITION (WRETURN)	561253.	254580.
JETTISON AND EXPENDABLES	-3370.	-1529.
INITIAL FLYBACK	557883.	253052.
FLYBACK FUEL	0.	0.
LANDING WEIGHT (WLAND)	557883.	253052.

Figure 23. SDV Mass Sequence WTR Launch

	Propane	Hydrogen
Number	10	12
Thrust SL	1200 K	414.8
Thrust VAC	1330 K	455.2
ϵ	40	35
Isp SL	303	400
Isp VAC	338	439
MR	2.68:1	6:1

To fit within the available base area, the engines will of necessity operate at 4000 psi chamber pressure. Even at this level, the chamber bells will extend slightly beyond the basic booster mold line. A change to twelve 1,000,000 pound sea level thrust engines, while offering some improvement, still results in chamber bells exceeding the mold line. (See Figures 24 and 25)

There was no analysis made to determine optimum engine nozzle expansion ratio from a performance standpoint. However, a ratio of 40 probably represents a practical maximum for packaging purposes.

Cost estimates for the SDV are given in Figure 26.

REFERENCES

1. Orbiter External Hydrogen Tank Study, Volumes I, II, and III, NR Report No. SD 71-141-1 (MSC-03327), 25 June 1971.

2. Phase B Final Report, Volumes I, II, NR Report No. SD-71-114-1 and SD-71-114-2 (MSC-03307), 25 June 1971.

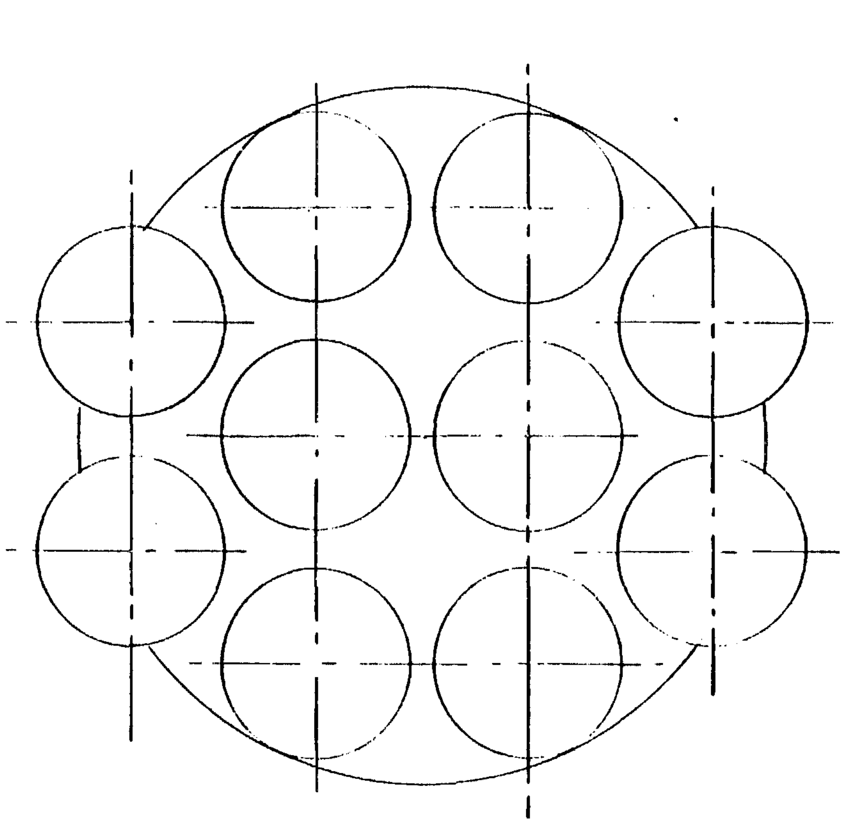


Figure 24. Base Arrangement
Ten 1,200,000 lb LO_2/Propane Engines
Chamber Pressure = 4000 psia
Area Ratio = 40

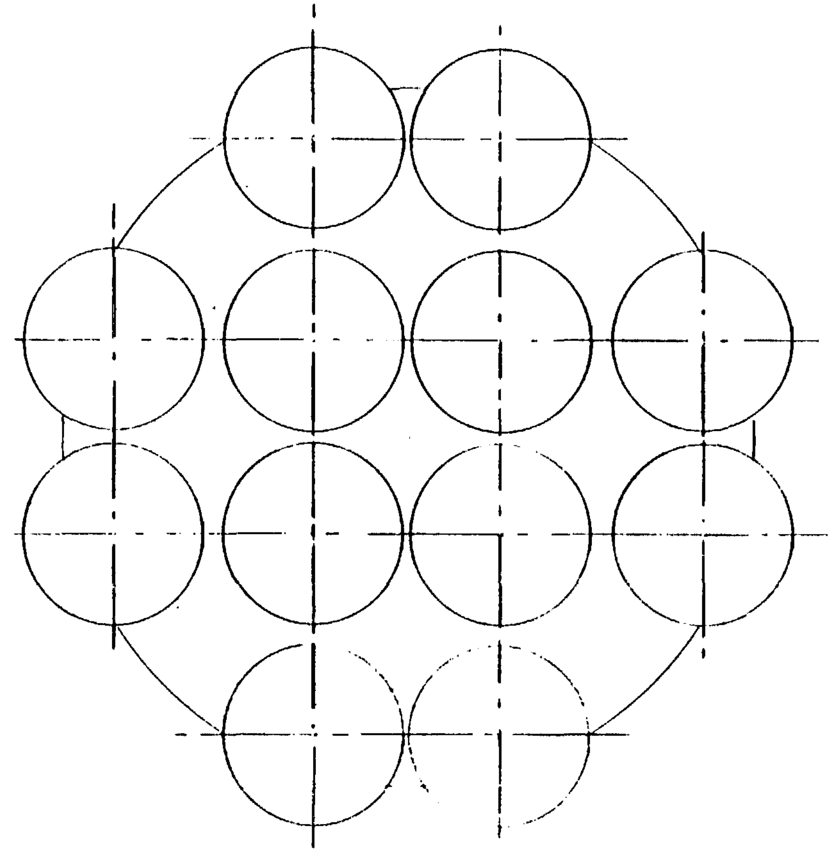


Figure 25. Base Arrangement
Twelve 1,000,000 lb LO_2/Propane Engines
Chamber Pressure = 4000 psia
Area Ratio = 40

Figure 26. COST ESTIMATE FOR SHUTTLE DERIVED VEHICLE BOOSTER

COST ELEMENT	RDT&E (MILLIONS 78$)			FIRST UNIT COST (MILLIONS 78$)		
FLIGHT HARDWARE			1165.57			256.40
STRUCTURES		298.91			132.06	
AERODYNAMIC SURFACES	91.83			66.67		
BODY	101.75			54.46		
TPS	58.68			9.14		
LANDING SYSTEM	46.65			1.79		
PROPULSION		201.68			46.74	
MAIN PROPULSION SYSTEM	68.35			32.11		
ATTITUDE CONTROL	133.33			14.63		
AVIONICS		468.26			30.64	
POWER SOURCE & DISTR.		172.59			26.25	
ECLS		24.13			1.56	
INTEG., ASSY. & C/O					19.15	
TOOLING			465.41			
VEHICLE TEST			1903.55			
GROUND		1219.76				
FLIGHT		683.79				
SYSTEM SUPPORT EQUIP			341.11			
SE&I			171.71			
PROGRAM MGMT			79.91			
TOTAL			4127.26			256.40

NOTES - Figure 26

1. Estimate excludes cost of facilities and facility activation.
2. Main Propulsion System cost excludes main rocket engines.
   ROM cost estimate of the 1.2 million lb. LO_2/propane engines is as follows:

RDT&E	$1.0 - 1.5 Billion
First Unit	$130 Million

3. Basis for estimates is GDC cost data on B^9U Shuttle booster as presented in: Booster Cost Data Book, 270 Day Review, Report 76-118-4-087 dated 2 April 1971. Costs from this report were scaled to the SDV version based on weight and inflated to 1978 dollars using the GNP price deflator index.
4. System Support Equipment includes checkout equipment, handling and transportation equipment, servicing equipment, training materials and aids, training services, initial spares, publication of technical data for maintenance and servicing, propellants and gases for facilities development and booster combined systems tests and transportation.
5. Vehicle test includes both test hardware and test operations.
6. Contractor fee is excluded.
7. Off the shelf orbiter thrusters are assumed and no development cost was included.

E.3 — ELECTRIC PROPULSION SYSTEM FOR LUNAR RESOURCE UTILIZATION FOR SPACE CONSTRUCTION

Enclosure of letter from John Regetz, Jr., of NASA-Lewis Research Center, Dated June 27, 1978.

I. ELECTRIC THRUSTER CONCEPT SELECTION

A. Propellant

Following the guidelines of the letter from E. H. Bock to R. C. Finke (2 May, 1978) only argon and oxygen were considered. An alternate inert gas, xenon, would be a superior choice if increased thrust-to-power were an important issue. In addition, xenon may be stored efficiently (tankage fraction of about [illegible]) at temperatures between about 10 and [illegible] °C.

A review of the use of oxygen was performed. Use of this propellant would require modifications in thruster design from that presently used. In particular, cathodes and insulators would require alteration. The long lead time for technology readiness plus some experience with thruster operation in an oxygen rich atmosphere in our ongoing non-propulsive application efforts lead us to believe that oxygen could be used if the reasons were sufficiently compelling. The characteristics of both oxygen and argon systems were therefore analyzed.

B. Following the study guidelines, a technology readiness date of 1[illegible] was assumed. In that time frame a thruster with an area equal to that of a circular [illegible]-cm diameter thruster appears quite reasonable. The shape would probably be similiar to that shown on Figure 1. This type of thruster, which is composed of sub-modules, is like that presently under development for very high current ion sources for the controlled nuclear fusion program. This approach avoids several potential problems, such as large unsupported spans of close spaced grid accelerator grids.

C. Beam Current and Specific Impulse

The ion beam current that may be obtained from a bombardment thruster is limited by either the accelerator grid physics (the so-called perveance limit) (Ref. 1) or by thermal considerations (Ref. 2). An analysis showed that the thruster under consideration would be beam current limited by the accelerator grid physics.

For both gases the selection of specific impulse is based on mission/system considerations. The beam power from a bombardment thruster is, however, very sensitive to the selected specific impulse. Tables I and II show the major thruster parameters over a range of specific impulse for the selected thruster design for argon and oxygen, respectively. Higher specific impulses may be easily attained, if desired, at small decreases in ion beam current. It should be noted that a single thruster design can operate over the entire range of specific impulse shown on either Table. This fact could be utilized if variable specific impulse is desirable during a single mission.

D. Thruster Mass

The mass of a circular 1[illegible]-cm argon thruster was estimated at 2[illegible] Kg in Reference 2. That mass was increased by ten percent to account for the non-circular shape selected. The baseline thruster mass was then selected at 2[illegible] Kg.

E. Thruster Performance

The input power, output thrust, and thruster efficiency are shown on Tables I and II for argon and oxygen, respectively. A thruster may inherently be operated over a large range of specific impulse. The range of performance shown can be attained with a single thruster design. As stated previously, much higher values of specific impulse can be attained at slightly reduced ion beam currents. The fixed power of a bombardment thruster is nearly invariant with size and was assumed constant at [illegible] (about double that of the present 30-cm thruster design). The performance data for oxygen were calculated under the assumption that the predominate accelerated ion was O^+. This assumption was based on the observed dissociation of diatomic gases (Ref. [illegible]) and heavy modules (Ref. 4) when used in a bombardment thruster.

II. SYSTEM DESIGN

A. Approach

The basic design of the system follows that studied in detail in Reference 5 (Design Concept I[illegible]). In Reference 3 the system power dissipations and masses were derived for a 12 thruster system operated at a total thruster input power of about [illegible] KW. In the selected approach the beam and discharge supplies are powered directly from the solar arrays (the so-called direct drive concept). This approach has been demonstrated with 5, [illegible], 15, and 30-cm diameter thrusters. A system block diagram is presented in figure 2 which shows the various elements in the thrust and interface modules. The thrust module contains thrusters, gimbals, low power supplies, a thermal control system, and structure. The interface module contains reconfiguration units for the beam and discharge loads, a distribution inverter, a DC-DC converter, a thrust subsystem controller, a thermal control system, propellant storage, and structure. In many areas, notably the low power supplies and the reconfiguration units, the use of a gas and operation at 1 AU would very considerably simplify the designs over those used in Reference 5. Such simplifications were not assumed herein but should provide margin in the system definition.

As your requirements are not known, the results of Reference 5 will be put into a form that allows convenient prediction of system parameters after such basic issues as total input power and specific impulse are selected. The various system elements will be described separately below.

B. Thrust Module

1. Thrusters - The mass and performance of the thruster design selected were discussed in section I. It should be pointed out that all power dissipated by the thrusters can be ejected without the need of a separate thermal control system.

2. Low Power Supplies

a. Power Dissipation - In Reference 5 each low power supply dissipated [illegible] watts. This was for a continuous output of [illegible] watts to the thruster. It was assumed that the argon thruster used a fixed power of [illegible] watts and it will therefore be assumed that 52 watts is dissipated in each low power supply.

b. The mass of each low power supply and its harness was estimated to be about 7 Kg. It will be assumed that low power supply mass scales as the square root of the output power which leads to a low power supply mass of about [illegible] Kg per thruster.

3. Gimbal Mass - A gimbal mass of 3 Kg was assumed for the [illegible].75 Kg mercury thruster in Reference 5. For the 22 Kg argon thruster the gimbal mass will be assumed to scale directly for a mass of 7.5 Kg per thruster.

4. Propellant Distribution - For a 12 thruster mercury system a total propellant distribution system mass of 1 Kg was assumed. A gas distribution system is likely to be heavier and an arbitrary 0.5 Kg per thruster will be assumed.

5. Thermal Control - For the heat pipe/radiator system studied in Reference 5 a mass of about 23 Kg per kilowatt of rejected power was derived. This mass will also be assumed herein.

6. Structure - In Reference 5, the thrust module structure consisted of a beam assembly to which the thrusters, gimbals, low power supplies, and some lightweight interface elements were attached. Each kilogram of structure supported about 14 Kg of other thrust module mass. For this study the thrust module struc-

ture will be assumed to be ten percent of the mass of the thrust module less the structure (M_{TM}).

C. Interface Module

1. Ion Beam Reconfiguration Unit

a. Mass - The mass of the reconfiguration was studied in detail in Reference 5 and was given as 45 Kg for 12 thrusters. It is likely that the mass of the reconfiguration unit is more dependent upon the number of thrusters than the total power. A mass of 4 Kg per thruster was then assumed.

b. Dissipated power - The total dissipated power for a thruster input power of 89 KW was estimated at 34 watts in Reference 5. It was assumed herein that 1 watt of power is dissipated per kilowatt of thruster input Power (P_T)

2. Discharge Reconfiguration Unit - In a fashion similiar to that for the beam reconfiguration unit, the mass and dissipated power of the discharge reconfiguration unit was estimated to be 3 Kg per thruster and one watt per kilowatt of thruster input power, respectively.

3. Other Elements - The other elements include the distribution inverter, the controller, the DC-DC converter, and the harness. It is likely that the mass and dissipated power of these elements are more sensitive to the number of thrusters than the total thruster input power. Based on this the mass and dissipated power of the other elements was estimated from Reference 5 to be 2.25 Kg and 10 W per thruster, respectively.

4. Thermal Control - As in the case of the Thrust Module thermal control system, it will be assumed that the mass is 23 Kg per kilowatt of rejected power.

5. Propellant Storage - The propellant storage mass for a cryogenic argon system was presented in Reference 2 and is shown on Figure 3 as a function of propellant mass. The storage for oxygen was assumed to be that for argon times the ratio of the density of liquid argon to liquid oxygen.

6. Structure - In Reference 5, the interface module structure mass of 45 Kg supported a total mass (including the Thrust Module) of 510 Kg. It will be assumed that the interface module mass is ten percent of the sum of the Thrust Module and all the other elements of the interface Module.

D. System Mass and Dissipated Power Summary

Table III shows a summary of the masses and dissipated powers of the proposed gas systems (Argon Tankage was assumed).

E. Sample System Definition Calculation

1. If a specific impulse of 6043 seconds is assumed along with a jet power of 500 KW, Table I shows a total thruster input power of 600.1 KW. Table 1 also shows this would require 5 thrusters of the type selected. An arbitrary 10,000 Kg of propellant is assumed required for the mission.

2. Thrust Module

a. From Table III the total dissipated power is just 0.26 KW.

b. The mass M'_{TM} represents all the thrust module mass except the structure and from Table III is calculated to be 206 Kg.

c. The mass of the thrust module (M_{TM}) is $M'_{TM} + 0.1\, M'_{TM}$ or 227 KG.

3. Interface Module

a. From Table III the dissipated power is just 1.26 KW.

b. The mass M_{IM} is all the interface mass except the structure and from Table III is (assuming 10^4 Kg propellant and the argon tankage mass from Reference 2) 570 Kg.

c. The mass of the interface structure is then given as 0.1 x (570 + 227) = 80 Kg.

4. The system mass and dissipated power are then 877 Kg and 1.52 KW, respectively. Although the scaling factors used are felt to be reasonable it is likely that some contingency factor of order 25 percent should be added to provide margin in system design.

REFERENCES

1. Sovey, J. S.: A 30-cm Diameter Argon Ion Source. AIAA Paper No. 76-1017. Key Biscayne, FLA. 1976.

2. Byers, D. C.; and Rawlin, V. K.: Critical Elements of Electron - Bombardment Propulsion for Large Space Systems. Journal of Spacecraft and Rockets, Vol. 14, No. 11, Nov. 1977, pp. [illegible].

3. Byers, David C.; and Reader Paul D.: Operation of an Electron - Bombardment Ion Source Using Various Gases. NASA TN D-[illegible], Dec. 1971.

4. Byers, David C.; Kerslake, William R.; and Grobman, Jack: Experimental Investigation of Heavy-Molecule Propellants in an Electron-Bombardment Thruster SIC, NASA TN D-24[illegible], Aug. [illegible].

5. Hawthorne, E. I.; et al: Extended Performance Solar Electric Propulsion Thrust System Study. Vol. III Tradeoff Studies of Alternate Thrust System Configurations.

   NASA CR-135281 Final Report
   NASA Contract NAS3-20395, Sept. 1977.

6. J. E. Cake, et al, "Modular Thrust Subsystem Approaches to Solar Electric Propulsion Module," AIAA International Electric Propulsion Conference, Paper No. 76-1062, Key Biscayne, Florida, November 14-17, 1976.

TABLE I

ARGON THRUSTER CHARACTERISTICS

Beam Current	Specific Impulse, sec	Input Power, KW	Thrust, N	Efficiency	P (Jet) / P Total
100	3822	60	1.74	0.54	0.664
↓	4681	80	2.13	0.61	0.748
	6043	120	2.74	0.68	0.832
	7150	160	3.25	0.71	0.874
	8180	200	3.68	0.74	0.899

1 Assumptions

a. Thrust Losses Fixed at 0.05
b. Propellant Utilization Fixed at 0.9
c. Fixed Power Losses 100W
d. Ion Current Derated 22% from Perveance Limit
e. Discharge Energy per beam ampere of 200 W/A

TABLE II

OXYGEN THRUSTER CHARACTERISTICS

Beam Current	Specific Impulse, sec	Input Power, KW	Thrust, N	Efficiency	P (Jet) / P Total
151.1	6039	86.9	1.65	0.56	0.696
	7396	117.2	2.03	0.63	0.774
	9548	177.6	2.62	0.69	0.851
	11297	238.1	3.10	0.72	0.888
	12810	298.5	3.52	0.74	0.910

1 Assumptions

a. Thrust Losses Fixed at 0.05
b. Propellant Utilization Fixed at 0.9
c. Fixed Power Losses, 100 W
d. Ion Current Derated 22% from Perveance Limit
e. Discharge Energy per beam ampere, 200 W/A

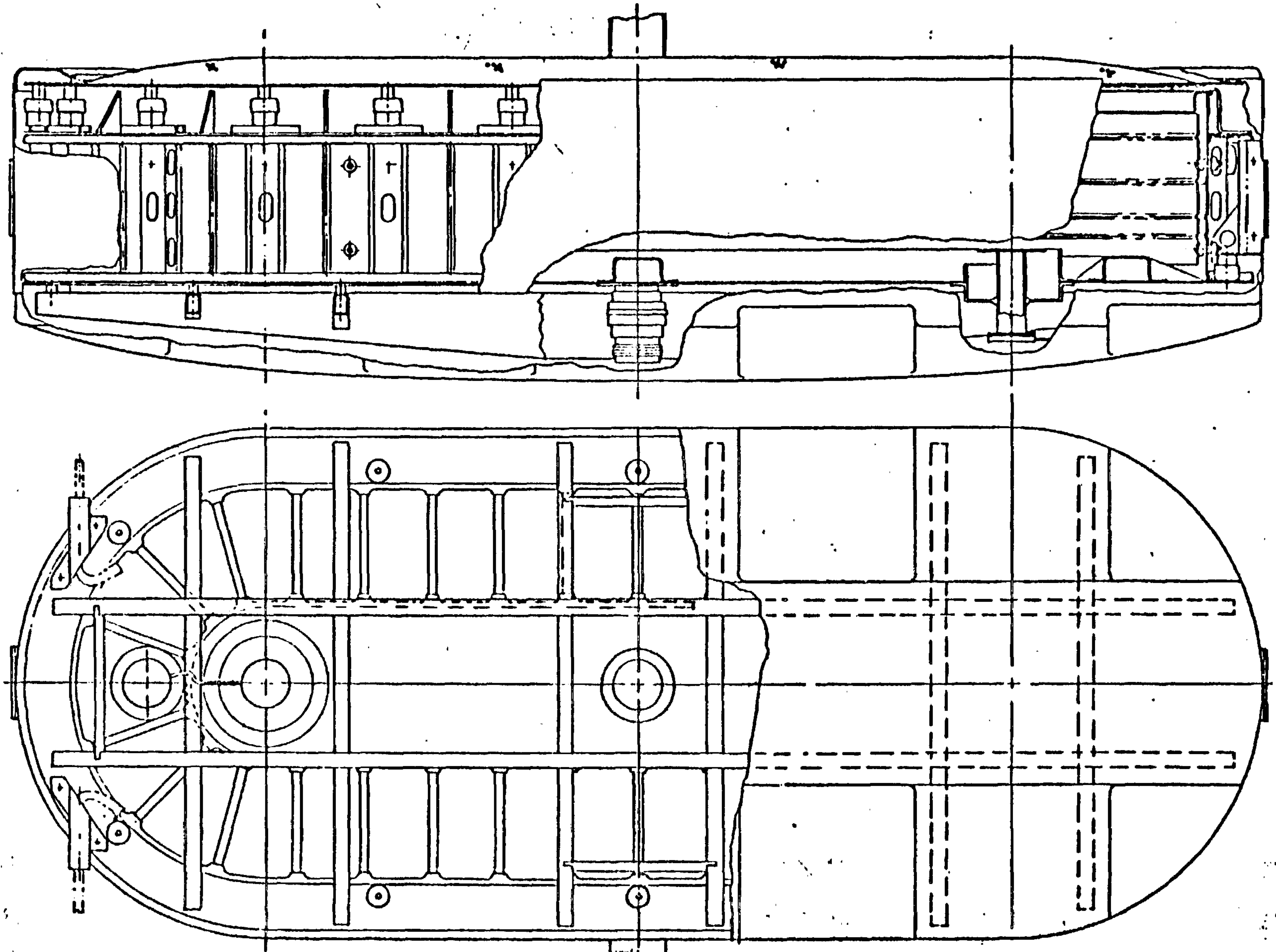
TABLE III

		Mass, KG	Dissipated Power, KW
	THRUST MODULE		
M_S, M_S', M_{TM}, M_{TM}'	Thrusters	$22 \times N$	
	Gimbals	$7.5 \times N$	
	Propellant Dist	$0.5 \times N$	
	Low Power Supplies	$10. \times N$	$0.052 \times N$
	Thermal Control	$23 \times 0.052 \times N$	
	Structure	$0.1 \times M_{TM}$	
	INTERFACE MODULE		
M_{IM}, M_{IM}'	Beam Reconfig.	$4 \times N$	$0.001 \times P_T$
	Discharge Reconfig.	$3 \times N$	$0.001 \times P_T$
	Other Elements	$2.25 \times N$	$0.012 \quad N$
	Thermal Control	$23 \times (0.002\ P_T + 0.012\ N)$	
	Propellant Storage	470 (assumed 10^4 Kg Propellant)	
	Structure	$0.1\ M_g$	

30 x 90-cm THRUSTER CONCEPT
USING 30-cm STRUCTURE APPROACH
- SIDE AND REAR VIEWS -

FIGURE 1a

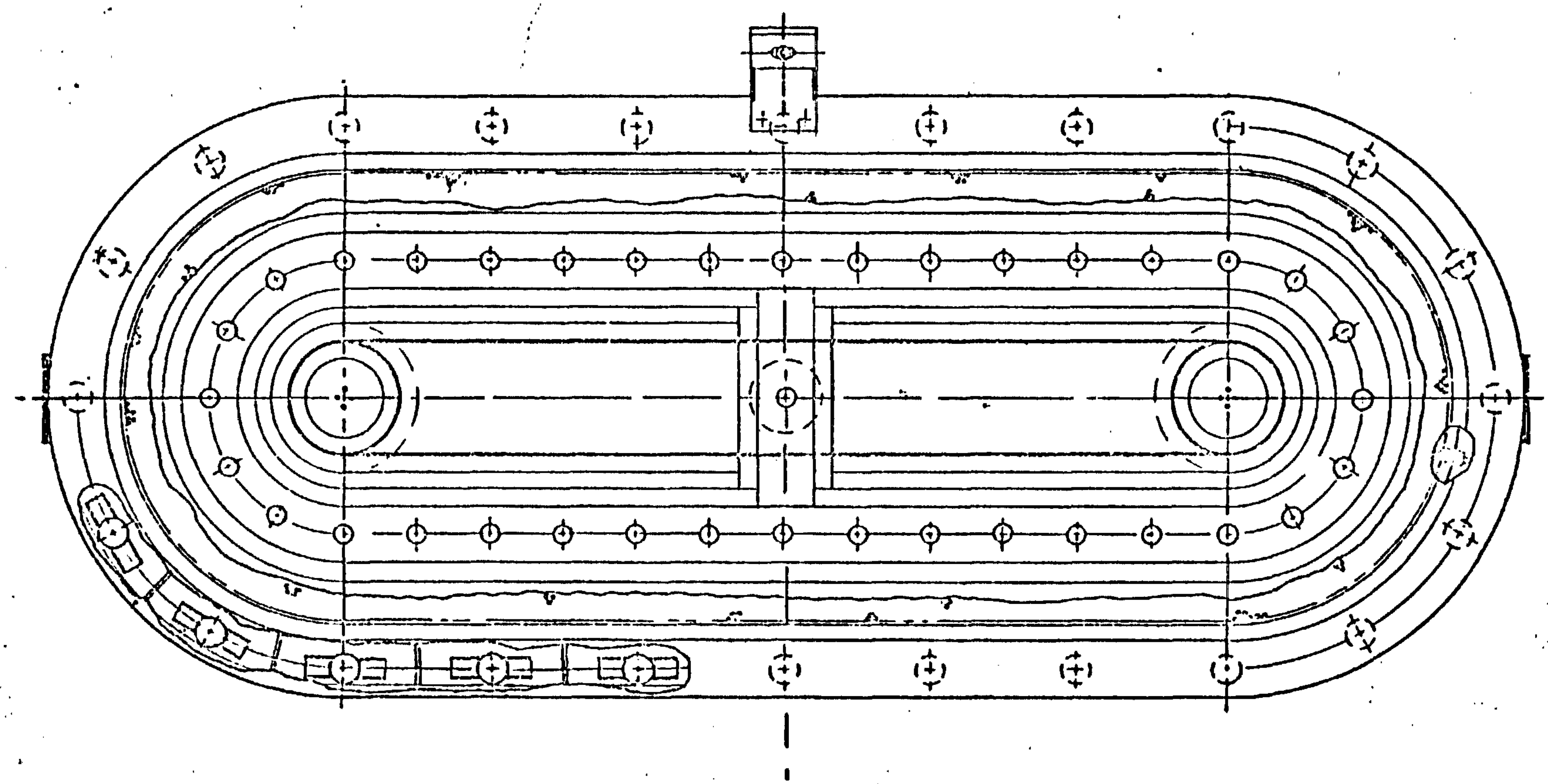
HUGHES
HUGHES AIRCRAFT COMPANY
RESEARCH LABORATORIES



30 x 90-CM THRUSTER CONCEPT
USING 30-CM STRUCTURE APPROACH
- FRONT VIEW, GRIDS REMOVED -

FIGURE 1b

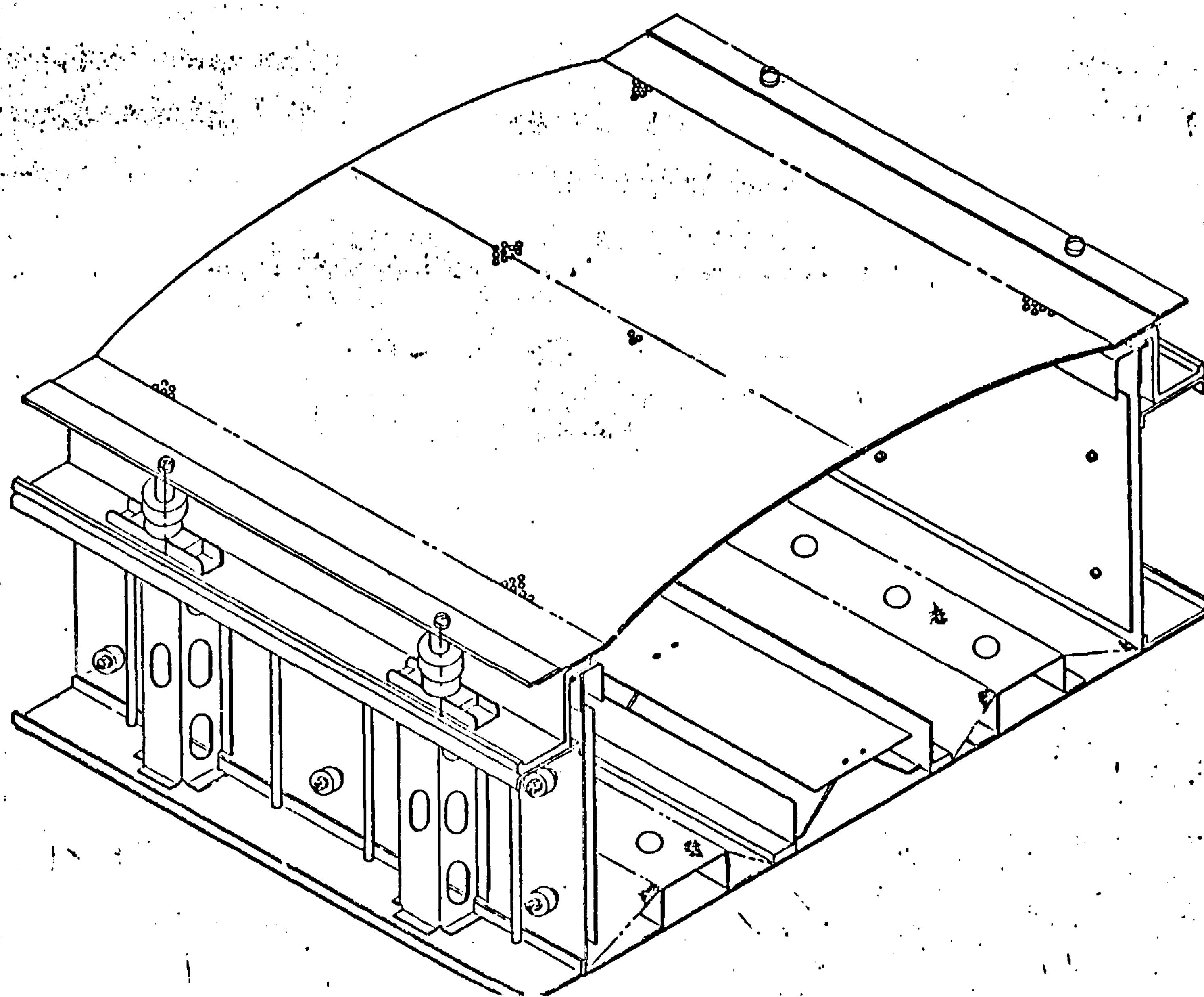
HUGHES
HUGHES AIRCRAFT COMPANY
RESEARCH LABORATORIES



30 x 90-cm THRUSTER CONCEPT
USING 30-CM SCRUCTURE APPROACH
- CROSS SECTION ISOMETRIC -

FIGURE 1c

HUGHES
HUGHES AIRCRAFT COMPANY
RESEARCH LABORATORIES



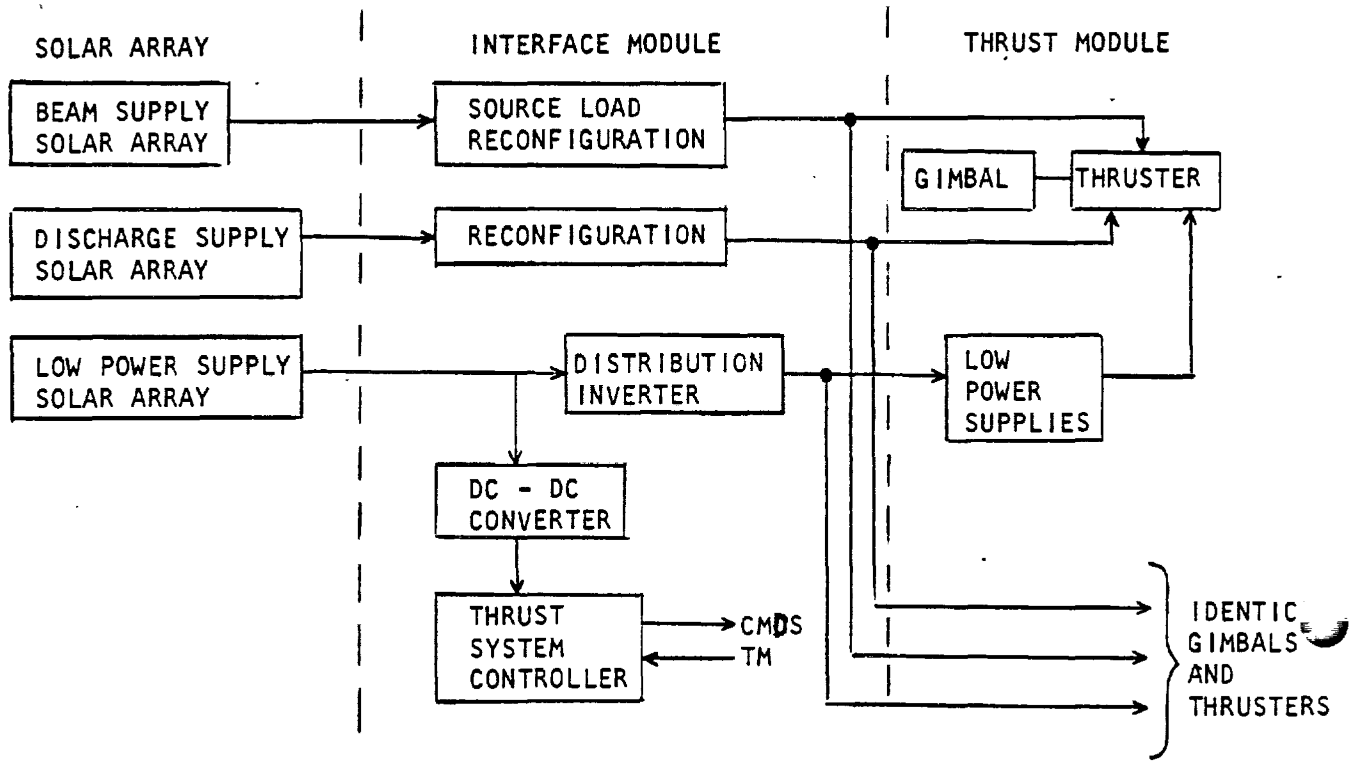
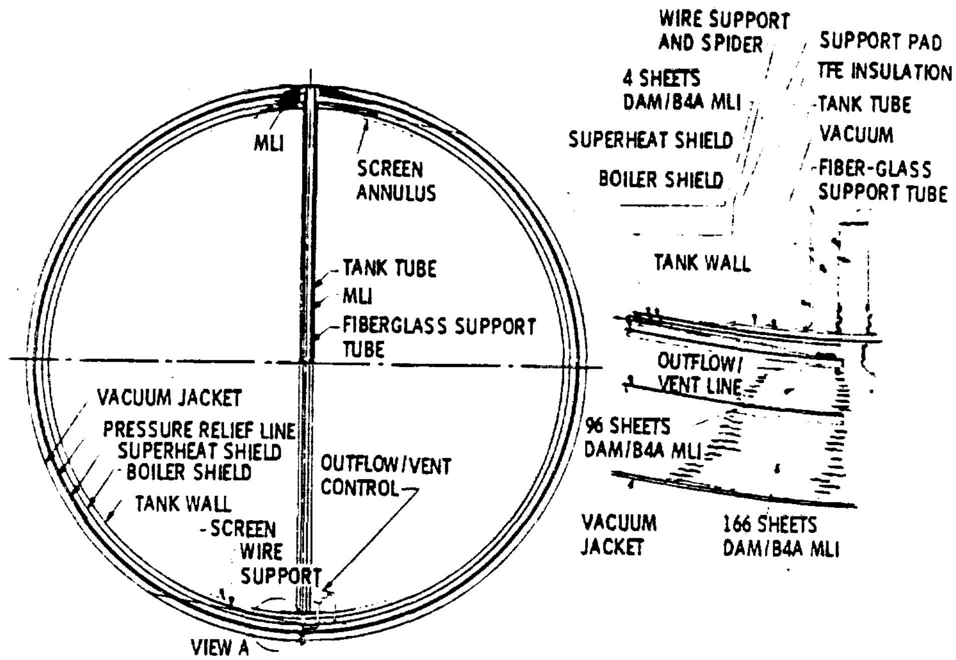
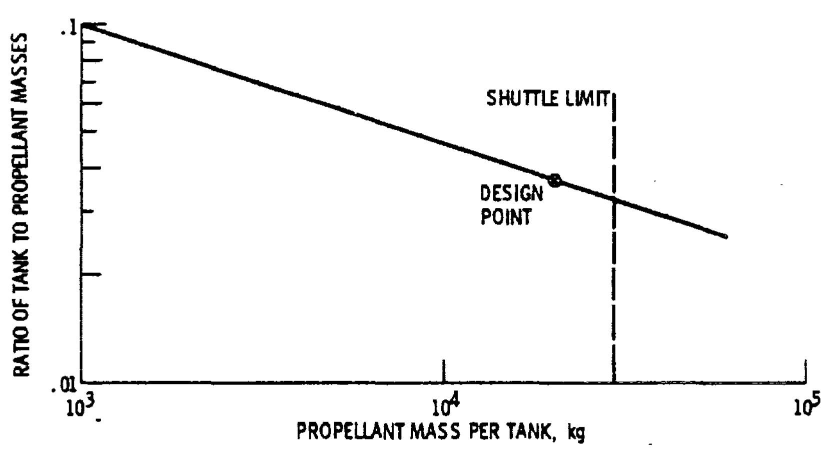

SOLAR ARRAY
INTERFACE MODULE
THRUST MODULE
BEAM SUPPLY SOLAR ARRAY
SOURCE LOAD RECONFIGURATION
GIMBAL
THRUSTER
DISCHARGE SUPPLY SOLAR ARRAY
RECONFIGURATION
LOW POWER SUPPLY SOLAR ARRAY
DISTRIBUTION INVERTER
LOW POWER SUPPLIES
DC - DC CONVERTER
THRUST SYSTEM CONTROLLER
CMDS
TM
IDENTIC
GIMBALS
AND
THRUSTERS

FIGURE 2.



a. - Thermodynamic vent/screen storage system.



b. - Ratio of tank to propellant masses as a function of the propellant mass per tank.

Figure 3

E.4 — PRELIMINARY INVESTIGATION OF THE FEASIBILITY OF CHEMICAL ROCKETS USING LUNAR-DERIVED PROPELLANTS

PAPER NO. 78-1032

AIAA/SAE 14th Joint Propulsion Conf., July 78, Las Vegas

by J. W. Streetman of General Dynamics Convair

Abstract

The cost of the energy required for launch from the earth's surface to earth orbit is a major consideration in the large scale industrialization of space. For example, transportation costs have been estimated to constitute approximately 40 percent of the costs of emplacing an operational fleet of solar power satellites (SPS) in geosynchronous earth orbit (GEO). Transportation requirements from the moon's surface to GEO are much lower than from earth — about 5 to 10 percent in terms of conventional propellant requirements. Recent studies show that the major portions (up to 90 percent) of the solar power satellites can be manufactured from lunar materials. We believe that similar fractions of the structure of most large space industrialization projects can also be lunar-derived.

If these materials can be launched from the moon by a technique that does not depend on the use of substantial quantities of earth-supplied propellants, it may be possible to achieve large cost savings in major space industrialization projects. This paper summarizes the results of a preliminary study of a lunar launch vehicle concept which uses a chemical rocket engine utilizing lunar-derived propellants exclusively.

Potential propellants available are oxygen and a number of metals including aluminum,calcium, magnesium, and iron. Performance of a lunar derived rocket (LDR), using these propellants for launch from the moon's surface to low lunar orbit, was evaluated in the context of an overall transportation scenario for emplacement of a fleet of operational SPS. Use of the LDR reduced earth-supplied material requirements more than 25 percent compared to the use of an oxygen-hydrogen rocket vehicle. The LDR concept has a number of technical risks, but those could be resolved by a feasibility testing program in the normal earth environment.

In this paper, only moon-based concepts for construction of space industrialization program elements are discussed and evaluated. No attempt is made to compare the largely moon based concepts utilizing lunar materials with exclusively earth-based concepts for space industrialization which are currently baselined for such programs as the Solar Power Satellite.

Preliminary Investigation of the Feasibility of Chemical Rockets Using Lunar-Derived Propellants

The General Dynamics Convair Division is performing a study for the NASA Johnson Space Center on Lunar Resources Utilization for Space Construction.* The major thrust of this study is to determine if the use of lunar material in construction of large space industrial facilities at geosynchronous earth orbit (GEO), can substantially reduce their cost. Approximately 40 percent of the overall cost of emplacing solar power satellites (SPS) is in the cost of transportation from the earth's surface. The argument for lower cost through use of lunar derived materials is based primarily on the much lower energy requirements for transportation of SPS construction materials from the moon rather than earth, because of the moon's lower gravitational potential. Only two methods of launching materials from the moon have been studied — conventional chemical H_2/O_2 rockets[1] and the mass driver-catcher.[2,3,4,5] The mass driver-catcher concept reduces propellant (but not energy) requirements to insignificance by catapulting bulk lunar materials from the surface using a nuclear powered electromagnetic launcher. This concept has high technical risk in a number of areas which will not be resolved for some years. Propellant required for moon to GEO transportation is about 5 percent of earth requirements using H_2/O_2 launch vehicles and electric orbiter transfer vehicles. However, to reduce costs, almost all of the propellants used for lunar launch would have to be derived from materials available on the moon.

This paper gives a preliminary technical evaluation of a rocket launch vehicle concept using lunar-derived propellants exclusively, and compares this concept with conventional O_2/H_2 rocket vehicles and the mass driver-catcher for support of an operational SPS program. This data is based on preliminary assessment of transportation energy requirements and transportation vehicle efficiencies. This information will be refined and updated during the current NASA study.

Lunar Materials Available

The lunar resource utilization concept is based on the premise that useful materials can be obtained from the moon, and that deriving these materials from lunar soil is not appreciably more expensive than their extraction on earth. The lunar surface and near subsurface are anhydrous and essentially devoid of carbon and organic material. They consist of rock, complex metal oxides, and silicates. These have been highly pulverized by meteoric impact, and the lunar surface is covered by a fine, silty, and angular sand with a scattering of angular rocks. The depth of the lunar soil, or regolith, varies considerably with location. The regolith depth on mare surfaces ranges from 2 to 10 meters. The highland areas, which are by far the oldest lunar features, have developed regoliths from hundreds of meters to possibly kilometers deep.

Compositions and typical chemical analyses of lunar soils are given in Tables 1 and 2. Notable by their absence (except in trace amounts) are the usual active constituents of rocket fuels hydrogen and carbon. However, oxygen and metals that can be burned with oxygen comprise approximately 80 percent of typical lunar soil.

A variety of techniques for mining the lunar soil and extracting useful materials are being studied; it is generally believed that economically feasible approaches can be developed. Electrolysis of molten metallic silicates holds some promise of permitting direct extraction of oxygen and selected metals at the cathode and anode respectively.

Rocket Vehicle Concepts

The only available lunar materials which are possible candidate rocket fuels are oxygen and various metals. Two concepts for rocket vehicles using these propellants are shown in Figure 1. In the pump-pressure fed concept, the propellants delivered to the thrust chamber are liquid oxygen (LO_2) and a

* *Contract NAS9-15560*

Table 1. Compositions of the lunar regolith.

	Percent by Weight		
	Mare	Highlands	Basic Ejecta
SiO_2	39.9 — 46.2	45.0 — 45.1	45.1 — 48.1
FeO	15.4 — 19.8	5.2 — 7.4	8.6 — 11.6
Al_2O_3	10.3 — 15.5	23.1 — 27.2	17.4 — 20.6
CaO	9.7 — 12.1	14.1 — 15.8	10.8 — 12.9
MgO	8.2 — 11.3	5.8 — 9.3	9.5 — 10.4
TiO_2	2.1 — 9.4	0.5 — 0.6	1.3 — 1.7
Cr_2O_3	0.3 — 0.5	0.1 — 0.2	0.2 — 0.3
Na_2O	0.3 — 0.5	0.4 — 0.5	0.4 — 0.7
MnO	0.2 — 0.3	0.1	0.1 — 0.2

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Table 2. Analysis of lunar materials.

		Typical Range (%)	Apollo 11 Rock (Mare) (%)
Oxygen	O_2	42-45	40.95
Silicon	Si	19-22	18.78
Aluminum	Al	5-15	4.12
Calcium	Ca	7-11	7.34
Iron	Fe	4-15	15.37
Magnesium	Mg	3-6	4.86
Titanium	Ti	0.5-5	7.36
Sodium	Na	0.2-0.4	0.39
Chromium	Cr	0.1-0.3	0.25
Potassium	K		0.25
Manganese	Mn		0.17
Hydrogen	H	50 to 100 parts/million	
Carbon	C	80 to 150 parts/million	
Nitrogen	N	60 to 120 parts/million	

8518-1

fluidized mixture of lunar metal powder and gaseous oxygen (GO_2). LO_2 is supplied from a pressurized tank as in conventional pressure-fed rockets. A portion of the LO_2 is pumped to a higher pressure and, in cooling the thrust chamber, is gassified. Portions of the GO_2 thus produced are used to:

- Pressurize the LO_2 tank
- Run the GO_2 turbine which powers the LO_2 pump
- Fluidize and entrain the lunar metal powder and transport it into the thrust chamber

The lunar metal powder, passivated to prevent reaction with the GO_2 before entering the thrust chamber, is supplied from a tank with its bottom and outlet appropriately contoured for powder flow. A spiral screw feed moves the powder through a duct to a positive displacement feed device which injects pre-measured incremental quantities of powder into the fluidizer and mixer, where it is entrained and mixed with GO_2 flowing to the thrust chamber.

In the hybrid concept, LO_2 is the oxidizer for a solid grain composed of lunar-derived metal powders held together by a suitable binder. The LO_2 feed system and a coolant-pressurant pump are similar to those for the pump-pressure fed concept. It would be desirable for the hybrid concept to use a binder derived from lunar materials. However, no method of synthesizing a suitable material from lunar materials is currently known (6). Use of earth-ferried binder may be acceptable, since it would constitute only a small fraction of the propellant and is evaluated in the following section.

Theoretical Rocket Engine Performance of Candidate Lunar Materials

Theoretical performance of LO_2 and some of the potential lunar derived metallic fuels were derived using the NASA Lewis Research Center computer program[7]. Figure 2 gives performance of aluminum and calcium as functions of nozzle expansion area ratio (ϵ) and equivalence ratio (ER) for two

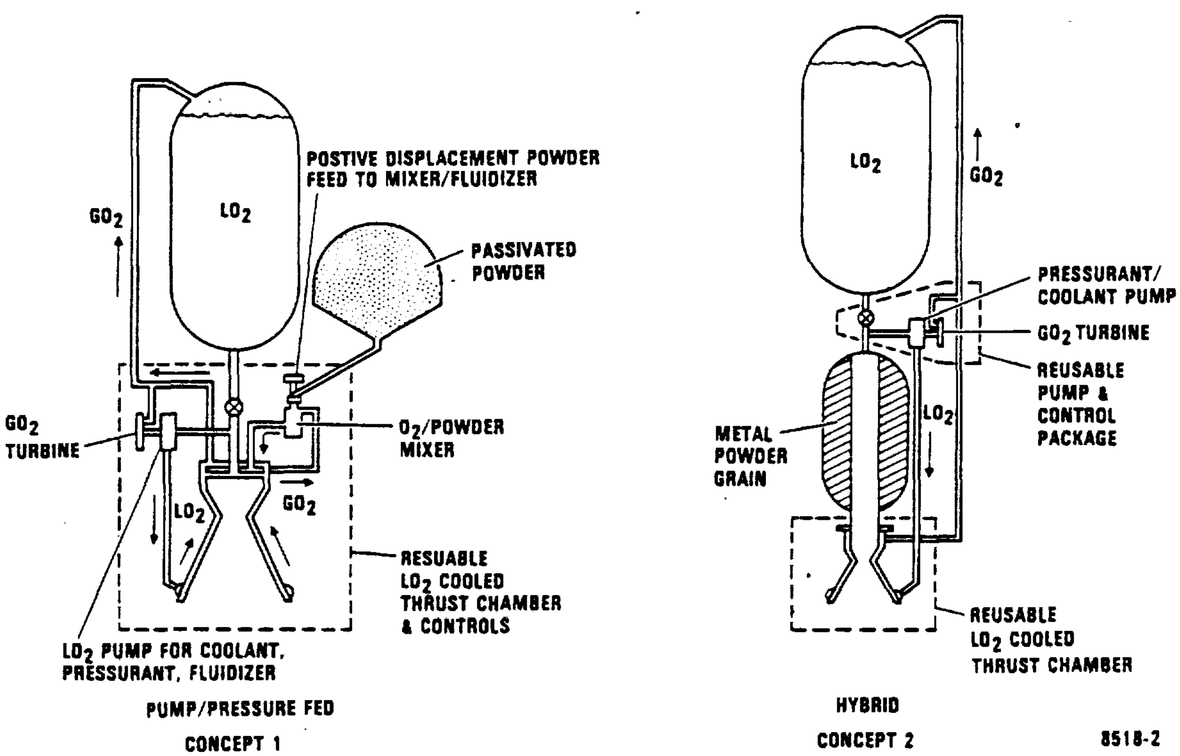


Figure 1. Concepts for rockets using lunar-derived propellants.

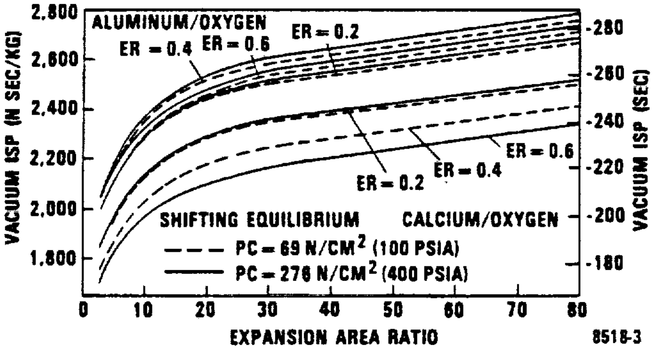


Figure 2. Performance of lunar metal fuels with LO_2.

chamber pressures (100 and 400 psia). ER is defined as the fraction of fuel required for ideal (stoichiometric) combustion and can be converted directly to mixture ratio. Some observations based on this data are:

- Maximum performance (peak specific impulse) is obtained at ERs considerably less than unity, i.e., with a large amount of excess oxygen. This appears to be because the predominant combustion species are the primary metallic oxides in liquid form and oxygen gas. Computer output summarizing performance, thermochemical, and exhaust composition data in the combustion chamber and nozzle at approximately the peak performance points are given in Tables 3 and 4. With aluminum, the Al_2O_3 remains in liquid form throughout the nozzle. With calcium, the liquid oxide begins to solidify just downstream of the nozzle throat. Beyond an area ratio of 5, the exhaust flow is exclusively oxygen and entrained solid CaO. Engines using these propellants can be considered heated oxygen engines, with the heat being supplied by combustion of the metallic fuel.
- Performance variations with chamber pressure Pc and area ratio ϵ are similar to those for conventional fuels.
- Theoretical specific impulse obtained is 80 to 90 percent of that currently available using typical solid propellants or LO_2/RP-1. Utility of this level of performance is discussed in the next section.

Figure 3 gives performance of lunar metals with oxygen and hydroxyl terminated polybutadiene (HTPB) synthetic rubber as functions of nozzle expansion ratio. The HTPB rubber is 10 percent of the total fuel, a fraction appropriate for the binder of the solid grain of the hybrid rocket shown in Figure 1. Data is shown for the ER (equivalent to mixture ratio) which gives peak ISP for the particular fuel used.

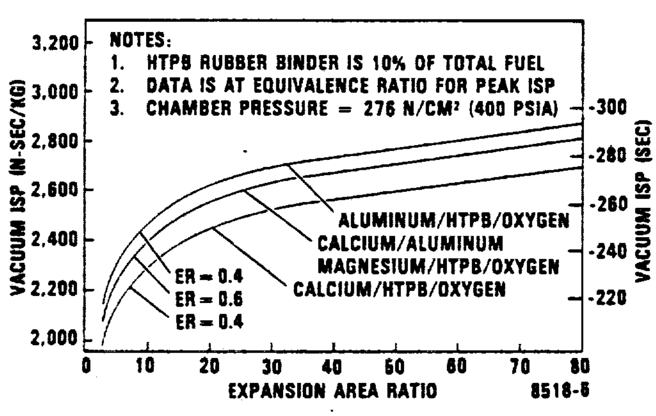


Figure 3. Performance of lunar metals and HTPB binder with LO_2.

Data for both the neat metal powders and the powder-binders are summarized in Figure 4 in the form of ISP vs ER for a nozzle expansion area ratio of 30 and a chamber pressure of 400 psia (276 N/cm²). This data shows that:

- Addition of the HTPB binder improves performance, especially for neat calcium and oxygen. This is as would be expected since HTPB is largely hydrogen and carbon, both more energetic than the metal fuels.
- Highest performance is obtained with aluminum, HTPB, and oxygen at ER of approximately 0.5.
- A combination of calcium, aluminum,. and magnesium (with HTPB binder) and oxygen in their naturally occurring percentages gives nearly the maximum available performance. This point is labeled "lunar soil" in Figure 4.

If extraction of this metals mixture proves to be more economical than extraction of aluminum alone, it could be used with little performance penalty. However, for almost all transportation scenarios for SPS emplacement investigated, oxygen requirements drive the total lunar soil processing requirements, with more then enough aluminum available from soil processed to supply oxygen.

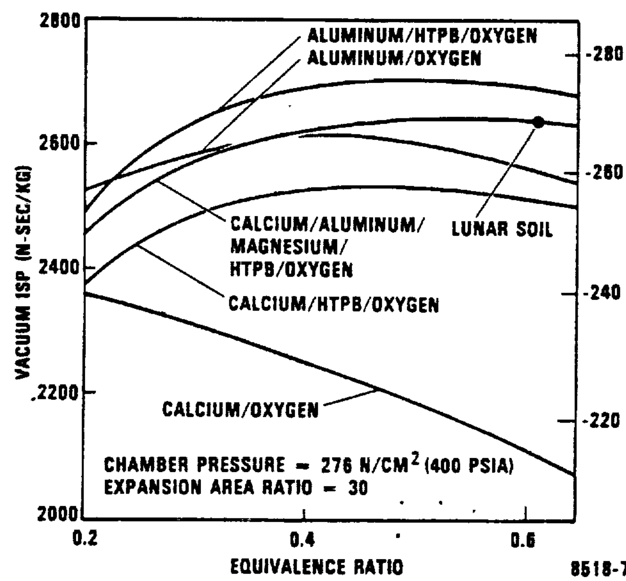


Figure 4. Performance summary, mixture of lunar metals, HTPB, and LO_2.

Mission Evaluation

The utility of the lunar derived rocket (LDR) has been evaluated in the context of suitability for support of an operational SPS program in which the SPS is fabricated (at least in part) from lunar materials. The SPS program is considered to typify other large space industrialization programs. For this assessment, use of oxygen and aluminum, both neat and with 10 percent HTPB, at an overall ISP efficiency of 90 percent shifting equilibrium was assumed for the LDR. Data for these systems is summarized in Table 5. The LDR based on this performance has a maximum single stage ideal incremental velocity (ΔV) capability of about 17,000 ft/sec, and has reasonable efficiency for missions up to approximately 12,000 ft/sec. Transportation requirements are outlined in Figures 5 and 6. Components of the SPS brought up from earth and all earth-moon cargo are assumed to be transported using a reusable solar electric cargo orbital transfer vehicle (COTV). The COTV uses oxygen propellant derived from lunar materials (or optionally, Argon

Table 3. Oxygen and aluminum performance.

```
PC =   400.0 PSIA
CASE NO.       2
```

CHEMICAL FORMULA		WT FRACTION (SEE NOTE)	ENERGY CAL/MOL	STATE	TEMP DEG K	DENSITY G/CC
FUEL	AL 1.00000	1.000000	0.000	S	298.15	0.0000
OXIDANT	O 2.00000	1.000000	-3102.000	L	90.18	1.1490

C/F= 2.2237 PERCENT FUEL= 31.0206 EQUIVALENCE RATIO= .4000 PHI= .4000 REACTANT DENSITY= 0.0000

	CHAMBER	THROAT	EXIT	EXIT	EXIT	EXIT	EXIT	EXIT (BASELINE)
PC/P	1.0000	1.7859	12.606	24.554	58.129	133.73	215.78	673.60
P, ATM	27.218	15.956	2.1592	1.1085	.4682	.2035	.1261	.0404
T, DEG K	4488	4313	3776	3622	3438	3274	3186	2990
RHO, G/CC	4.0680-3	2.4968-3	3.9801-4	2.1525-4	9.7103-5	4.4911-5	2.8824-5	1.0017-5
H, CAL/G	-66.9	-151.4	-436.7	-522.0	-626.0	-720.3	-771.9	-887.8
S, CAL/(G)(K)	1.6534	1.6534	1.6534	1.6534	1.6534	1.6534	1.6534	1.6534
M, MOL WT	54.946	55.378	57.106	57.708	58.500	59.279	59.732	60.828
(DLV/DLP)T	-1.20308	-1.19227	-1.15474	-1.14400	-1.13170	-1.12149	-1.11632	-1.10572
(DLV/DLT)P	3.9996	3.9520	3.6920	3.5968	3.4806	3.3814	3.3314	3.2325
CP, CAL/(G)(K)	1.9950	2.0209	2.0294	2.0131	1.9878	1.9646	1.9536	1.9368
GAMMA (S)	1.0952	1.0930	1.0858	1.0838	1.0815	1.0794	1.0784	1.0759
SON VEL,M/SEC	861.7	841.3	772.5	752.0	726.9	704.1	691.5	663.1
MACH NUMBER	0.0000	1.0000	2.2770	2.5952	2.9757	3.3214	3.5127	3.9524
AE/AT		1.0000	3.0000	5.0000	10.000	20.000	30.000	80.000
CSTAR, FT/SEC		4308	4308	4308	4308	4308	4308	4308
CF		.641	1.340	1.486	1.647	1.781	1.850	1.996
IVAC LB-SEC/LB		164.3	211.2	226.3	243.6	258.5	266.3	283.2
ISP, LB-SEC/LB		85.8	179.4	199.0	220.6	238.5	247.7	267.3

MOLE FRACTIONS

AL	.00226	.00204	.00122	.00097	.00070	.00048	.00038	.00021
AL+	.00011	.00009	.00004	.00003	.00002	.00001	.00001	.00001
ALO	.02280	.01970	.01169	.00942	.00690	.00492	.00399	.00230
ALO-	.00003	.00003	.00001	.00001	.00000	.00000	.00000	.00000
ALO2	.01924	.01736	.01079	.00890	.00677	.00504	.00420	.00263
ALO2-	.00006	.00005	.00002	.00001	.00001	.00001	.00000	.00000
AL2O	.00070	.00058	.00024	.00016	.00009	.00005	.00003	.00001
AL2O2	.00054	.00047	.00024	.00018	.00011	.00007	.00005	.00002
AL2O3(L)	.22248	.22577	.23784	.24158	.24612	.25016	.25234	.25714
E	.00002	.00001	.00001	.00001	.00001	.00001	.00001	.00000
O	.35311	.35268	.34617	.34300	.33627	.32782	.32212	.30625
O-	.00001	.00000	.00000	.00000	.00000	.00000	.00000	.00000
O2	.37942	.38120	.39103	.39572	.40301	.41143	.41686	.43142
O3	.00001	.00001	.00000	.00000	.00000	.00000	.00000	.00000

ADDITIONAL PRODUCTS WHICH WERE CONSIDERED BUT WHOSE MOLE FRACTIONS WERE LESS THAN .50000E-05 FOR ALL ASSIGNED CONDITIONS

AL(S) AL(L) ALO+ AL2O+ AL2O2+ AL2O3(S) O+ O2-

8518-5

Table 4. Oxygen and calcium performance.

PC = 400.0 PSIA
CASE NO. 1

	CHEMICAL FORMULA	WT FRACTION (SEE NOTE)	ENERGY CAL/MOL	STATE	TEMP DEG K	DENSITY G/CC
FUEL	CA 1.00000	1.000000	0.000	L	298.15	0.0000
OXIDANT	O 2.00000	1.000000	-3102.000	L	90.18	1.1490

O/F= 1.3959 PERCENT FUEL= 33.3786 EQUIVALENCE RATIO= .2000 PHI= .2000 REACTANT DENSITY= 0.0000

	CHAMBER	THROAT	EXIT	EXIT	EXIT	EXIT	EXIT	EXIT	
PC/P	1.0000	1.7195	11.485	22.726	55.418	130.62	213.66	691.11	
P, ATM	27.218	15.829	2.3698	1.1975	.4911	.2084	.1274	.0394	
T, DEG K	3682	3512	3200	3054	2846	2657	2551	2300	
RHO, G/CC	5.0338-3	3.0972-3	5.0594-4	2.7006-4	1.2065-4	5.5569-5	3.5624-5	1.2373-5	
H, CAL/G	-64.6	-133.6	-351.9	-427.6	-519.4	-600.5	-644.2	-740.2	
S, CAL/(G)(K)	1.4977	1.4977	1.4977	1.4977	1.4977	1.4977	1.4977	1.4977	BASELINE
M, MOL WT	55.878	56.369	56.058	56.507	57.375	58.135	58.528	59.300	
(DLV/DLP)T	-1.03571	-1.03116	-1.03414	-1.03005	-1.02243	-1.01593	-1.01262	-1.00615	
(DLV/DLT)P	1.6015	1.5496	0.0000	1.6108	1.4875	1.3703	1.3051	1.1647	
CP, CAL/(G)(K)	.6494	.6297	0.0000	.7222	.6469	.5727	.5273	.4201	
GAMMA (S)	1.1170	1.1151	.9670	1.1066	1.1058	1.1064	1.1075	1.1136	
SON VEL,M/SEC	762.3	759.9	677.5	705.1	675.4	648.4	633.5	599.3	
MACH NUMBER	0.0000	1.0000	2.2869	2.4719	2.8884	3.2662	3.4765	3.9674	
AE/AT		1.0000	3.0000	5.0000	10.000	20.000	30.000	80.000	
CSTAR, FT/SEC		3844	3844	3844	3844	3844	3844	3844	
CF		.649	1.323	1.487	1.665	1.807	1.879	2.029	
IVAC LB-SEC/LB		147.0	189.3	204.0	220.5	234.2	241.3	256.3	
ISP, LB-SEC/LB		77.5	159.1	177.7	198.9	215.9	224.6	242.5	

MOLE FRACTIONS

CA	.00018	.00014	.00024	.00017	.00007	.00002	.00001	.00000
CA+	.00001	.00000	.00000	.00000	.00000	.00000	.00000	.00000
CAO(S)	0.00000	0.00000	.28493	.31962	.32318	.32616	.32767	.33059
CAO(L)	.31681	.31901	.03274	0.00000	0.00000	0.00000	0.00000	0.00000
CAO	.00092	.00064	.00064	.00039	.00015	.00005	.00002	.00000
O	.09228	.06109	.08845	.07876	.05956	.04257	.03375	.01646
O2	.56978	.59910	.59300	.60106	.61705	.63120	.63854	.65295
O3	.00001	.00000	.00000	.00000	.00000	.00000	.00000	.00000

ADDITIONAL PRODUCTS WHICH WERE CONSIDERED BUT WHOSE MOLE FRACTIONS WERE LESS THAN .50000E-05 FOR ALL ASSIGNED CONDITIONS

CA(S) CA(S) CA(L) E O+ O- O2-

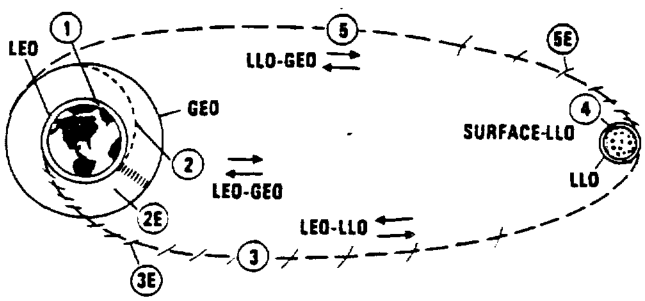
NOTE. WEIGHT FRACTION OF FUEL IN TOTAL FUELS AND OF OXIDANT IN TOTAL OXIDANTS

8518-4

Table 5. Performance of lunar-derived rocket.

	Pump/Pressure Fed	Hybrid
Propellants		
Oxidizer	LO_2	LO_2
Fuel	Al powder	Al powder 90% HTPB binder 10%
Mixture ratio	2.22	1.86
Equivalence ratio	0.4	0.6
Area ratio	80	80
Specific impulse (sec)		
Theoretical	283	297
Delivered	255	267
Stage mass fraction	0.9	0.9

8518-8



APPROXIMATE ENERGY RQMT - ΔV FT/SEC

		CHEMICAL	ELECTRIC	VEHICLES
1	EARTH - LEO	31,000	–	SHUTTLE, SDV
2	LEO-GEO(R)	14,200	19,100	EOTV, POTV
3	LEO-LLO(R)	13,000	26,800	EOTV, POTV
4	LLO-LUNAR SURFACE (R)	6,000	–	H_2O_2 OR LDR OTV
5	LLO-GEO(R)	7,100	7,700	EOTV, H_2/O_2 OTV, LDR OTV

E - ELECTRIC PROPULSION
(R) EACH LEG, ROUND TRIP

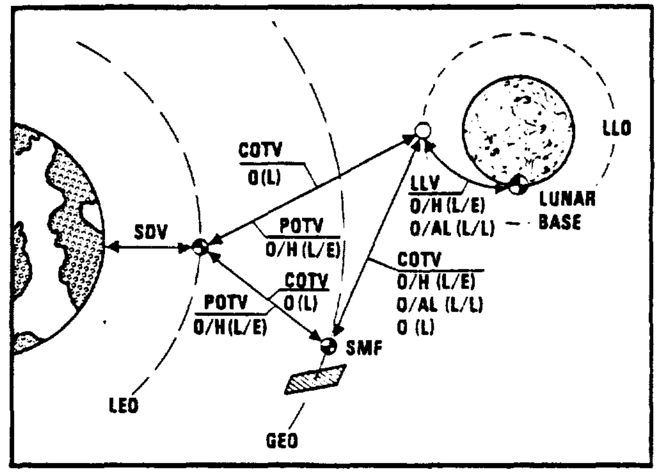
8518-9

Figure 5. Transportation system energy requirements.

from earth) at an ISP of 7,000 seconds. SPS construction materials obtained from the moon are brought from low lunar orbit (LLO) to GEO using either an electric COTV or a chemical OTV. Propellants for the chemical OTVs considered use lunar derived oxygen and either earth delivered hydrogen, or lunar derived aluminum. Transportation from the lunar surface to LLO is by a chemical lunar launch vehicle (LLV), using the same propellant options as the chemical OTV. All OTVs and launch vehicles are reusable so all missions are round trip.

The primary evaluation criteria used in the assessment of lunar derived propellants versus other possible options are the earth material requirements (EMR) and lunar material requirements (LMR) for fabrication of 5 solar power satellites per year, each weighing 100,000,000 kilograms (kg). Assumptions used in computation of these requirements are summarized in Table 6.

Moon-bound cargo is the primary driver of the EMR. For the LDR option, it consists primarily of life support materials (for the lunar base) and chemicals and other materials used in lunar soil processing. For the O_2/H_2 and hybrid LDR rocket option, H_2 and HTPB for the OTV or LLV must also be ferried.



SPACE STATION/HABITAT
LARGE SPACE STRUCTURE
PROPELLANT DEPOT (UNMANNED)

COTV — TRANSPORTATION VEHICLE TYPE
O/H(E) — ORIGIN OF PROPELLANTS (E) EARTH (L)D LUNAR
PROPELLANTS
O — OXYGEN
H — HYDROGEN
AL — ALUMINUM

8518-10

Figure 6. Lunar-based construction transportation scenarios.

Table 6. Assumptions for lunar transportation analysis.

Steady-state operations — buildup phase complete & all earth, lunar & space facilities in place
All hydrogen propellants are delivered from earth
All other propellants used above LEO are obtained from the moon (oxygen, aluminum)
Processing of lunar soil results in 33% oxygen recovery
Chemicals expended (lost) in lunar processing equal 0.5% of soil processed
Ecosystems are partially closed. Crew requirements including food & water from earth are 0.8 ton/year
Manpower requirements — operational payload/manyear
GEO - 500 ton/manyear (qty of five 10GW SPS/yr)
Lunar — 81.8 tons/manyear
Operational payload is manufactured 90% from lunar material

8518-11

Performance for the options evaluated are summarized in Table 7. Options 1, 2 and 3 all use the LDR with neat aluminum for lunar surface to LLO, and trade the use of the LDR, chemical O_2/H_2 and electric O_2 orbit transfer vehicles from LLO to GEO. The LDR OTV is seen to be in a very distant third place in terms of EMR and LMR. The chemical O_2/H_2 and electric O_2 options are close together, but the electric is clearly the more efficient in terms of performance.*

Option 4 utilizes the hybrid concept LDR from the lunar surface to LLO, but is otherwise the same as Option 3. Compared with Option 3, EMR is increased 16 percent because of the requirment to haul up HTPB binder from the earth, while LMR is decreased eight percent because of the higher ISP for lunar launch.

Options 3 and 5 compare the use of the LDR (aluminum powder) and a conventional chemical O_2/H_2 launch vehicle from the surface to LLO. The LDR is the more efficient in terms of EMR (27 percent lower) but 58 percent higher in

* *Transfer time for the electric O_2 vehicle is ≈ 10-40 days compared to 2 days for the chemical O_2/H_2 vehicle.*

Table 7. Summary comparison of lunar transportation options evaluated.

Option	Propulsion Used: Surface — LLO	Propulsion Used: LLO-GEO	Material Requirements (lb/lb SPS): EMR	LMR	Lunar O_2	Lunar Al
1.	LDR O_2/Al	LDR O_2/Al	22.7	27.5	7.3	3.57
2.	LDR O_2/Al	Chemical O_2/H_2*	5.46	9.46	3.15	1.05
3.	LDR O_2/Al	Electric O_2	3.51	5.53	1.83	0.72
4.	LDR O_2/Al-HTPB*	Electric O_2	4.08	5.08	1.61	0.66
5.	Chemical O_2/H_2*	Electric O_2	4.79	3.49	1.16	0.13
6.	Mass driver†	Electric O_2 ††	3.24	1.75	0.57	0.13

* H_2 and HTPB delivered from earth. All other propellants lunar-derived
† Surface-2:1 resonance orbit
†† 2:1 resonance orbit - GEO

8518-12

LMR. EMR is believed to be the better measure of cost, ultimately the real comparison criterion, since the difference in EMR is primarly in launch vehicle propellants (H_2, O_2 and methane) while the difference in LMR is in bulk lunar soil processed. Cost of processing the lunar soil has not been assessed.

Option 6 uses the mass driver-catcher concept, with the catcher located at the lunar libration point L-2, and the space manufacturing facility in 2:1** resonance orbit around the earth. The EMR for this option is only eight percent lower than for Option 3.

The overall life cycle cost, including development, space facility buildup, fleet manufacture and emplacement; and SPS emplacement and operation, will be estimated for these concepts in the Johnson Space Center study in progress.

Conclusions

The use of a chemical rocket vehicle which burns lunar-drived metallic fuels for launch from the lunar surface to low lunar orbit results in a substantial reduction in earth material and energy requirements compared with the use of LH_2, which is earth manufactured and delivered to the moon. Performance in terms of EMR is within eight percent of the most advanced systems (mass driver-catcher) which have been postulated. Major areas of technical uncertainties in the rocket vehicle system for use of the metallic fuels include oxygen cooling of the thrust chamber, powder feed (for the pump-pressure concept), and the basic powder-liquid combustion process. All of these technical uncertainties are amenable to resolution by straight-forward testing in the normal earth environment.

In this paper, only moon-based concepts for construction of space industrialization program elements are discussed and evaluated. No attempt is made to compare the largely moon-based concepts, utilizing lunar materials, with exclusively earth-based concepts for space industrialization which are currently baselined for such programs as the SPS.

** *The space manufacturing facility makes two orbits around the earth while the moon makes one.*

References

1. Streetman, J.W., *Vehicle Sizing Support to the Space Settlements Summer Study at Ames Research Center,* General Dynamics Convair Division memo dated 29 July 1977.
2. O'Neill, G.K., *Mass-Driver Reaction Engine as a Shuttle Upper Stage,* Paper 77-536, Third Princeton/AIAA conference on Space Manufacturing Facilities, Princeton, NJ, May 9-12, 1977.
3. Heppenheimer, T.A., *The Lunar Mass Transport Problem in Space Colonization,* Paper Reprint, AAS/AIAA
4. Anon; *Space Settlements — A Design Study,* 1975 NASA Ames/OAST Summer- Faculty Fellowship Program in Engineering System Design, NASA SP-413 NASA Science & Techncial Information Office, 1977.
5. Driggers, G., & Newmar, J., *Establishment of a Space Manufacturing Facility,* 1976 NASA Ames/OAST Study on Space Manufacturing from Non-Terrestrial Materials, November 25, 1976.
6. Lee, S.M., *Lunar Building Materials — Some Considerations on the Use of Inorganic Polymers,* NASA Ames/OAST Summer Study on Space Settlements and Industrialization, 1977, Paper 3, Group 5.
7. Gordon, S. and McBride, B., *Computer Program for Calculation of Complex Chemical Equilibrium Compositions, Rocket Performance, Incident and Reflected Shocks, and Chapman-Jouguet Detonation,* Interim Revision March 1976, NASA SP-273, NASA Lewis Research Center, 1971.

APPENDIX F

Supplementary data for Section 5.1.2 contains explanatory notes to Table 5-4, Earth Baseline Life Cycle Cost. Also contains figures referenced by the notes. (Reference Figures F-1 through F-8.)

The following notes accompany Table 5-4 of Section 5 in Volume II:

NOTE 1

Transportation

HLLV

From Davis pitch, Fig. F-1 , cost of 15 units is \$15.1 billion. TFU - \$1.38 billion. He used a 14 year program @ 1 SPS/yr.

Learning curve coefficient can be derived from this data:

$$y = a\,x^{1+b}$$

$$15.1 \times 10^9 = (1.38 \times 10^9)\,(15)^{1+b}$$

$1 + b = \underline{.884}$ This is a 92.3% learning curve

Referring to the Benson pitch, Fig. F-2 , the data does not quite agree. Benson shows an initial fleet cost of \$6.04 billion for 6 vehicles. Using Davis TFU and learning curve: Total = $1.38\,(6)^{.884}$ = \$6.73 billion, which is slightly higher.

For our purposes, use Davis' numbers:

Total cost of HLLV production = $\underline{(1.38 \times 10^9)\,N^{.884}}$

Figure F-3 (Davis) shows 391 HLLV flights are required per SPS. For a 30 SPS fleet, 30 × 391 = 11,730 flights. Ground rules in Fig. F-4 (Davis) give a 500 flight mission life. Total number of HLLV's required for the program then, is:
11,730/500 = 23.5 → 24

Use Davis assumption that initial fleet size is 6. This amount will be included in RDT&E. The remaining 18 vehicles will be included in the Production Phase under vehicle replacement.

Cost to Develop and Produce Initial HLLV Fleet of 6

Development (WBS 1411) (Ref. Davis pitch Fig. F-1 and Benson pitch Fig. F-5)

	Cost (Billions \$)
1000 Ton CH_4/O_2 Engine	.8
Second Stage Engine	.1
Airframe and Integration	10.2
	\$11.1 Billion

NOTE 1 (continued)

Production of Initial Fleet (6 HLLV's) (WBS 1412)

$$\text{Cost} = 1.38 \times 10^9 \ (6)^{.884} = \$6.726 \times 10^9$$

Production of Remaining 18 HLLV's (WBS 2131)

$$\text{Cost} = 1.38 \times 10^9 \ (24)^{.884} - 6.726 \times 10^9$$

$$= 22.908 \times 10^9 - 6.726 \times 10^9 = \underline{\$16.182 \times 10^9}$$

NOTE 2

PLV Costs

Development

Fig. F-1 shows development costs as follows:

Shuttle/Ballistic Booster Airframe	$1.9 billion
Shuttle/Ballistic Booster Integration	.5 billion
Total	$2.4 billion (WBS 1421)

The development to the CH_4/O_2 engine was allocated to the HLLV.

Production

Fig. F-4 shows life to be 500 missions. Fig. F-3 shows 36 PLV flights are required per SPS for a total of 36 × 30 = 1080 flights. Thus

$\frac{1080}{500} = 2.16$ PLV's are required. Assume 2 vehicles will be built for the initial fleet and one replacement will be required during the production phase (WBS 2130) TFU cost is not provided in the baseline document. This information is given in the JSC Redbook on page X - D - 31 for a propane/Lox PLV.

PLV TFU Cost $.354 billion

External tank TFU cost $.011 billion

Assume the learning curve is the same as a HLLV, then total production costs for a PLV can be expressed as:

$$TC = .354 \ N^{.884} \text{ (billions of 77 \$)}$$

$$= .354 \ (2)^{.884}$$

$$= \$.653 \text{ billion for the initial fleet (WBS 1422)}$$

NOTE 2 (continued)

$TC = .354\ (3)^{.884} - .653 = \underline{\$.282}$ billion for replacement vehicles (WBS 2131)

Since external tanks are expendable they will be considered separately. Learning is assumed the same as the HLLV.

$TC = .011\ N^{.884}$

A total of 1080 ETs will be required, one for each flight. Assume 36 will be fabricated as part of the initial fleet. The remaining 1044 will be fabricated in the production phase under WBS element 2130.

WBS 1422 Initial Fleet Cost

$.011\ (36)^{.884} = \$.261$ billion (ET)

WBS 2130 Replacement Vehicles

$.011\ (1080)^{.884} - .261 = \$5.022$ billion (ET)

Total initial fleet production WBS 1422

PLV	.653 billion
ET	.261 billion
Total	$.914 billion

NOTE 3

POTV Costs

Development

From Fig. F-1 we can obtain the following costs:

Personnel OTV, 2 stage + crew module	$1.5 billion
Shuttle/OTV Passenger Module	.5 billion
	$2.0 billion

Production

Fig. F-4 gives the life of a POTV as 50 and Fig. F-3 shows that 5 flights are required per SPS. Then there are a total of 30 × 5 = 150 flights which requires 150/50 = 3 POTV's. Assume an initial fleet size of 2 with 1 vehicle replacement during the production phase.

NOTE 3 (continued)

The SPS baseline does not provide TFU costs for the POTV directly but they can be determined from Figs. F-1 or F-2.

Fig. F-1

Total cost is .7 billion for 4 units

Assume learning is the same as HLLV

$.7 - \text{TFU}\ (4)^{.884}$

TFU cost = \$.206 billion

Fig. F-2

Total cost is .350 billion for 2 units

$.35 = \text{TFU}\ (2)^{.884}$

TFU cost - \$.190 billion

Use TFU cost of \$.20 billion

Initial Fleet (WBS 1432)

$\text{T.C} = .20\ (2)^{.884}$

= \$.369 billion

Production of remaining POTV (WBS 2130)

$\text{TC} = .20\ (3)^{.884} - .369$

= \$.159 billion

NOTE 4

COTV Costs

Development

Fig. F-1 provides development costs as follows:

SPS Electric/Cryo Thruster Modules (COTV) \$1.7 billion

Production

Fig. F-3 shows 1 COTV is required per SPS and this is expended, thus for 30 SPS's, 30 COTV's would be required. Assume an initial fleet of 1 COTV with the remaining 29 being fabricated during production.

NOTE 4 (continued)

Fig. F-1 gives the avg. cost per SPS as \$1.7 billion. Since this is only a soft estimate no learning curve will be applied.

Initial Fleet Cost (WBS 1442)	\$1.7 billion
Production of Remaining 29 (WBS 2130)	
29×1.7	\$49.3 billion

NOTE 5

Figure F-6 provides a first unit cost of \$12.829 billion. H. Benson of NASA, in a telecon with J. Fox of GDC, reported that NASA has added a cost for the large contingency in satellite mass. He recommended a 25% increase to the costs in Figure F-6 to allow for this contingency.

Figure F-6 may be used to determine a learning curve exponent for the satellite.

$$AC = TFU\,(N)^{b}$$

Where AC = Avg. cost/SPS (billions 77\$)

TFU = First unit cost (billions 77\$)

N = 30 Satellites

b = cumulative avg. cost learning curve exponent

$$7,140.656 = 12,829\,(30)^{b}$$

$$b = -.172 \qquad (88.8\%\ \text{learning})$$

For total cumulative cost, TC, the exponent is 1 + b, or .828. With the 25% increase total satellite production cost can be computed by:

$$\underline{TC = 16.036\,N^{.828}} \qquad (\text{billions of } 77\$)$$

For 30 satellites, TC = <u>\$268.011 billion</u>

NOTE 6

Figure F-7 shows a cost of \$4.446 billion per SPS. Since each satellite requires 2 recetnnas this cost is assumed to represent 60 ground system sites. Rectennas

NOTE 6 (continued)

will have to be fabricated at a rate of 2 per year in order to support satellite production.

Total cost over a 30 year production period is:

$4.446 × 30 = $133.38 billion

NOTE 7

1. Facility Maintenance (WBS 2121)

The SPS base does not provide facility maintenance costs. It is assumed that maintenance and operation of the propellant production and SPS hardware facilities is reflected in the propellant and hardware costs. The maintenance and operation of the launch recovery facilities will be considered here.

Assume maintenance costs are 5% of the launch/recovery facilities cost per year:

.05 (C (1311)) Y

Where: C (1311) = Development and fabrication cost of launch/recovery facilities

Y = Number of years of production phase

= 30 years

Maintenance Cost = .05 (2.8) 30 = $4.200 billion

2. Launch & Recovery Operations (WBS 2122)

It is assumed that the costs for launching and recovering vehicles are contained in the "personnel" and "other" categories in Figure F-3. Since the split between launch/recovery operations and vehicle operations is not known, all costs in Figure F-3 will be inserted under the transportation category (WBS 2130) and WBS 2122 will be zero.

NOTE 8

1. Vehicle Replacement (WBS 2131)

Vehicle replacement cost is the sum of the production costs previously identified in Notes 1, 2, 3 and 4.

NOTE 8 (continued)

Vehicle	Cost (Billions of 77$)
HLLV	16.182
PLV	
Vehicle	.282
External Tank	5.022
POTV	.159
COTV	49.300
TOTAL	$70.945

2. Vehicle Maintenance (WBS 2132)

Vehicle Maintenance is assumed to be included in Figure F-3, under "personnel" or "other". As mentioned in Note 7, these categories also are assumed to include launch/recovery operations but no split is shown. The costs under these categories in Figure F-3 will be shown under vehicle maintenance:

	Cost/Flight ($M)	Number of Flights	Total (Billions 77$)
HLLV	8.9	11,730	104.397
COTV	-	30	-
PLV	9.0	1,080	9.720
POTV	27.6	150	4.140
			$118.257

Spares (WBS 2133)

This category is zero for the baseline. Spares costs are included under WBS 2131, Vehicle replacement.

Propellants/Gases (WBS 2134)

Cost/Flight is contained in Figure F-3.

	Cost/Flight ($M)	Number of Flights	Total (Billions 77$)
HLLV	2.1	11,730	24.633
COTV	-	30	-
PLV	.5	1,080	.540
POTV	.2	150	.030
		Total	$25.203

Further evaluation is required to assess costs for the COTV.

NOTE 9

Construction System Operations

Figure F-8 provides an initial base cost for the construction systems as well as operations costs per SPS of $1.216 billion. Although not explicitly stated, it is assumed that the $1.216 billion is entirely a recurring charge for construction system maintenance, refurbishment and logistics support.

Total cost for WBS 2320 = 30 × $1.216 = $36.480 billion.

NOTE 10

The SPS baseline document does not provide satellite or rectenna operations costs. The JSC Report, Solar Power Satellite Concept Evaluation, on page X-D-7 provides an estimate of 3% of satellite hardware in orbit per year for satellite operations and maintenance. It will be assumed that rectenna operations are also 3% of hardware costs per year.

Since program operational life has not been established, costs will be based on the 30 year period when satellites are being constructed. Operations costs may be determined by the general relation:

$$C = (.03A)\,\frac{n\,(n+1)}{2}$$

Where A = average hardware cost of per satellite or rectenna

n = number of years in operations phase

C = operations phase cost for satellite or rectenna at a 1/year construction rate.

Satellite

Average cost can be calculated from WBS 2111

$$A = \frac{268{,}011}{30} = \$8.934 \text{ billion}$$

$$\text{WBS 3100, cost of satellite operations} = .03\,(8.934)\,\frac{30\,(30+1)}{2}$$

$$= \underline{\$124.629 \text{ billion}}$$

NOTE 10 (continued)

Rectenna

Average cost can be calculated from WBS 2112

$$A = \frac{133.38}{30} = \$4.446 \text{ billion}$$

$$\text{WBS 3200, Cost of Earth Rectenna Operations} = .03\,(4.446)\,\frac{30\,(30+1)}{2}$$

$$= \underline{\$62.002 \text{ billion}}$$

FIGURE F-1

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SPS SYSTEMS DEFINITION STATUS REPORT	TRANSPORTATION SYSTEMS	
	H. P. DAVIS	1/25/78

SPS TRANSPORTATION SYSTEM NONRECURRING COSTS

	ESTIMATED COSTS, $B	NOTES
BOOSTER/PLV 1000 TON CH_4/O_2 ENGINE RDT&E	0.8	COMMON TO BOTH VEHICLES
SHUTTLE BALLISTIC BOOSTER AIRFRAME RDT&E	1.9	INCLUDES ET TANK MODS.
SHUTTLE/COTV PASSENGER MODULE RDT&E	0.5	BASED UPON SPACELAB
SHUTTLE/BALLISTIC BOOSTER INTEGRATION RDT&E	0.5	INCLUDES FACILITY MODS
PERSONNEL OTV, 2 STAGE + CREW MODULE RDT&E	1.5	RL10-DERIVATIVE ENGINE
HLLV 2ND STAGE ENGINE RDT&E	0.1	SSME MODIFICATION
HLLV AIRFRAME & INTEGRATION RDT&E	10.2	WINGED, FLYBACK, 2 STAGE
SPS ELECTRIC/CRYO THRUSTER MODULES (COTV) DDT&E	1.7	
KSC LAUNCH & RECOVERY FACILITIES	2.8	500 FLTS/YR CAPABILITY
SUBTOTAL	$20.0	
THE FOLLOWING INVESTMENTS OVER 14 YEARS ARE AMORTIZED IN THE COST PER FLIGHT ESTIMATES		
PROPELLANT PRODUCTION FACILITIES	3.5	NEW RATES & KSC LAUNCH
PLV FLEET, 525 FLIGHTS, 4 UNITS	1.3	MODIFIED EXISTING ORBITERS + 525 ET'S OF 550 TON CAPACITY
POTV FLEET, 75 FLIGHTS, 4 UNITS	0.7	
HLLV FLEET, 5550 FLIGHTS, 15 UNITS	15.1	SPARES + ATTRITION = 4 UNITS TFU = 1.38
COTV FLEET, 8 LARGE + 24 SMALL PANELS PER SPS	1.7 PER SPS	EXPENDED - VERY SOFT ESTIMATE

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SPS SYSTEMS DEFINITION STATUS REPORT — 1/25/78

COST $ X 10^6

	INITIAL FLEET COST	COST/SPS
HLLV (6 REQ'D)		
VEHICLE AND SPARES	6,040	2,268
FUEL		821
PERSONNEL		1,760
OTHER		1,720
COTV (SET OF 8)	2,820	2,820
PLV (2 REQ'D)	650	470
POTV (2 REQ'D)	350	230
TOTAL	8,740	10,089

FIGURE F-3

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SPS SYSTEMS DEFINITION STATUS REPORT

TRANSPORTATION SYSTEMS

H. P. DAVIS

1/25/78

SPS TRANSPORTATION COST PER FLIGHT

VEHICLE	NO/FLTS SPS	COST/FLIGHT, $M					COST/ SPS $B	NOTES
		VEHICLE & SPARES	PROPELLANT	PERSONNEL COSTS	OTHER	TOTAL		
HLLV	391	4.53	2.1	3.58	3.4	13.6	5.32	FLYBACK, KSC OPERATION AT 1 SPS/YR REQUIRES STUDY
						16.8*	6.57*	ADJUSTED BOEING NO.
COTV	1	--	--	--	--	--	2.82	BOEING PART II, VOL. 6 "THRUPUT"; REQUIRES STUDY
PLV	36	3.7	0.5	6.0	3	13.2	.47	REQUIRES STUDY
POTV	5	18.7	0.2	16	11.6	46.5	.23	REQUIRES STUDY
*RECOMMENDED FOR INTERIM USE						TOTAL	8.8 TO 10.0*	

FIGURE F-4

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SPS SYSTEMS DEFINITION STATUS REPORT	TRANSPORTATION SYSTEMS	
	H. P. DAVIS	1/25/78

SPS TRANSPORTATION SYSTEM COST GROUND RULES

- o 1978 DOLLARS
- o "PREVIEW" ESTIMATE OF MORE FORMALLY-GENERATED COSTS SCHEDULED FOR 7/78
- o ONE 10 GWe SPS PER YEAR
- o 14 YEAR PROGRAM DURATION
- o KSC LAUNCH SITE W/FLYBACK BOOSTER
- o ALL PROPELLANTS DERIVED FROM COAL, AIR, AND WATER
- o 500 MISSION LIFE OF LAUNCH VEHICLES
- o 50 MISSION LIFE OF SPACE-BASED VEHICLES
- o SPS ORBIT TRANSFER SYSTEM (ION PROPULSION) NOT RECOVERED, NOT REDUCED BY SPS UTILIZATION FOR RCS & STATIONKEEPING

FIGURE F-5

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SPS SYSTEMS DEFINITION STATUS REPORT

1/25/78

DDTE COST $ X 10^6

POWER CONVERSION, TRANSMISSION, RECEPTION	3,344
TECHNOLOGY VERIFICATION	2,926
TRANSPORTATION	
HLLV	11,100
COTV	1,700
OTV	1,500
PLV	1,900
CONSTRUCTION BASE	6,939
SPS HARDWARE FACILITIES	10,366
LAUNCH FACILITIES	2,800
	42,575

FIGURE F-6

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SPS SYSTEMS DEFINITION STATUS REPORT

1/25/78

COST/SPS $ X 10^6

SATELLITE

o POWER COLLECTION		
	- STRUCTURE	360
	- ROTARY JOINT	17.856
	- ATTITUDE CONTROL	152.9
	- INSTRUMENTATION/COMM.	124.4
	- SOLAR CELL BLANKETS	3,749
	- POWER DISTRIBUTION	115
o POWER TRANSMISSION (TOTAL OF 2 REQUIRED)		
	- STRUCTURE	64.5
	- ATTITUDE CONTROL	201.7
	- INSTRUMENTATION/COMM.	666.3
	- KLYSTRONS	524.1
	- THERMAL CONTROL	274
	- WAVEGUIDES	258.4
	- POWER DISTRIBUTION	632.5
	TOTAL	7,140.656

NOTE: COST OF FIRST SPS $12,829

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SPS SYSTEMS DEFINITION STATUS REPORT

1/25/78

COST/SPS $\$ \times 10^6$

GROUND SYSTEM

o RECTENNA

- DIPOLES	1,574
- POWER COLLECTION AND CONDITIONING	580
- STRUCTURE, FENCING AND ASSEMBLY	1,788
- LAND AND SITE PREPARATION	504
TOTAL	4,446

FIGURE F-8

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SPS SYSTEMS DEFINITION STATUS REPORT

1/25/78

SPACE FABRICATION AND ASSEMBLY, $ X 10^6

	INITIAL BASE COST	COST/SPS 1/YR FOR 30 YRS
CONSTRUCTION FACILITY/LEO		
FACILITY	4,115	165
CONSTRUCTION EQUIPMENT	1,445	169
SUPPLY AND REFURBISHMENT	156	156
CONSTRUCTION FACILITY/GEO		
FACILITY	3,990	159
CONSTRUCTION EQUIPMENT	1,980	247
SUPPLY AND REFURBISHMENT	191	191
SUBTOTAL	11,877	1,087
TRANSPORTATION	1,925	129
TOTAL	13,802	1,216

APPENDIX G

Tasks 5.3 and 5.4 supplementary data, identifying details of LRU element cost development required to support economic analysis activity reported in Section 5 of Volume II.

Appendix G consists of 7 sections:

G.1 PROPELLANT DEPOTS

G.1.1 Cost estimates for the depots will be based on the following study:

Orbital Propellant Handling and Storage Systems for Large Space Programs, GDC Report CASD-ASP-78-001, Vol II, 14 April 1978. (JSC-13967), pp 9-1 through 9-43.

Estimates were provided in the Orbital Propellant Handling (OPH) study for two different sizes of propellant depot facilities: 5 million lbs and 40 million lbs capacity (Reference Section G.7, Table G-42) and for 2 different sizes of tanks: 1 million lb and 2 million lb (Reference Section G.7, Table G-43). The tanks in the OPH study were bipropellant, with provisions for LH_2 and LO_2. In the Lunar Resources Study the storage tanks will be either all LO_2 , LH_2 or aluminum. Costs for this difference will be adjusted accordingly. Data is shown in the following table. Depot configuration descriptions are contained in Section 4.5.1 of Volume II.

Table G-1. Orbital Propellant Handling Study Data.

Element/Size	Cost (Millions of 77$) R&D	Cost (Millions of 77$) First Unit	Reference
Propellant Depot Platform			
5 mil. lb. capacity (2268 metric tons)	85.77	21.12	Table G-42
40 mil. lb. capacity (18141 metric tons)	135.41	40.77	Table G-42
Tank Modules			
1 mil. lb. capacity (453.5 metric tons)	125.38	6.22	Table G-43
2 mil. lb. capacity (907 metric tons)	225.81	10.23	Table G-43

Tank capacities in Table G-1 are given in terms of the combined LH_2/LO_2 propellant weights at a 6:1 mixture ratio. The weights of the individual propellant capacities were calculated as follows:

	LO_2(6/7)	LH_2(1/7)
1 million lb tank	.857	.143
2 million lb tank	1.714	2.86

The above capacities can be transformed into single propellant capacities using density ratios according to $W_2 = W_1 \frac{P_2}{P_1}$. The following densities were used to determine the single propellant capacities shown in Table G-2: (1) LH_2 - 4.4 lb/ft^3, (2) LO_2 -71.2 lb/ft^3 (3) powdered aluminum - 86.4 lb/ft^3.

Table G-2. OPH Tank Module-Single Propellant Capacities
Equivalent Single Propellant Size (Metric Tons).

OPH Size	LH_2	LO_2	A
1 Million lb	1438.095	88.889	1745.125
2 Million lb	2876.190	177.778	3490.250

The data in Tables G-1 and G-2 was used to derive the scaling relationships in the following sections.

G.1.2 Propellant Depot Platform Scaling Relationships

Assume costs vary with propellant depot capacity on a nonlinear basis according to $y = ap^b$

A. First Unit Cost:

$$b = \frac{\log 40.77 - \log 21.12}{\log 18141 - \log 2268} = .316$$

$$a = \frac{40.77}{18141^{.316}} = 1.839$$

$$\text{TFU } \underline{\underline{\text{Cost} = 1.839p^{.316}}} \text{ (Mil of 77\$)}$$

B. Development Cost:

$$b = \frac{\log 135.41 - \log 85.77}{\log 18141 - \log 2268} = .220$$

$$a = \frac{135.41}{18141^{.22}} = 15.658$$

$$\underline{\underline{\text{Cost} = 15.658p^{.220}}} \text{ (Mil of 77\$)}$$

where p = propellant capacity of structure in metric tons

CERs are plotted in Figure G-1.

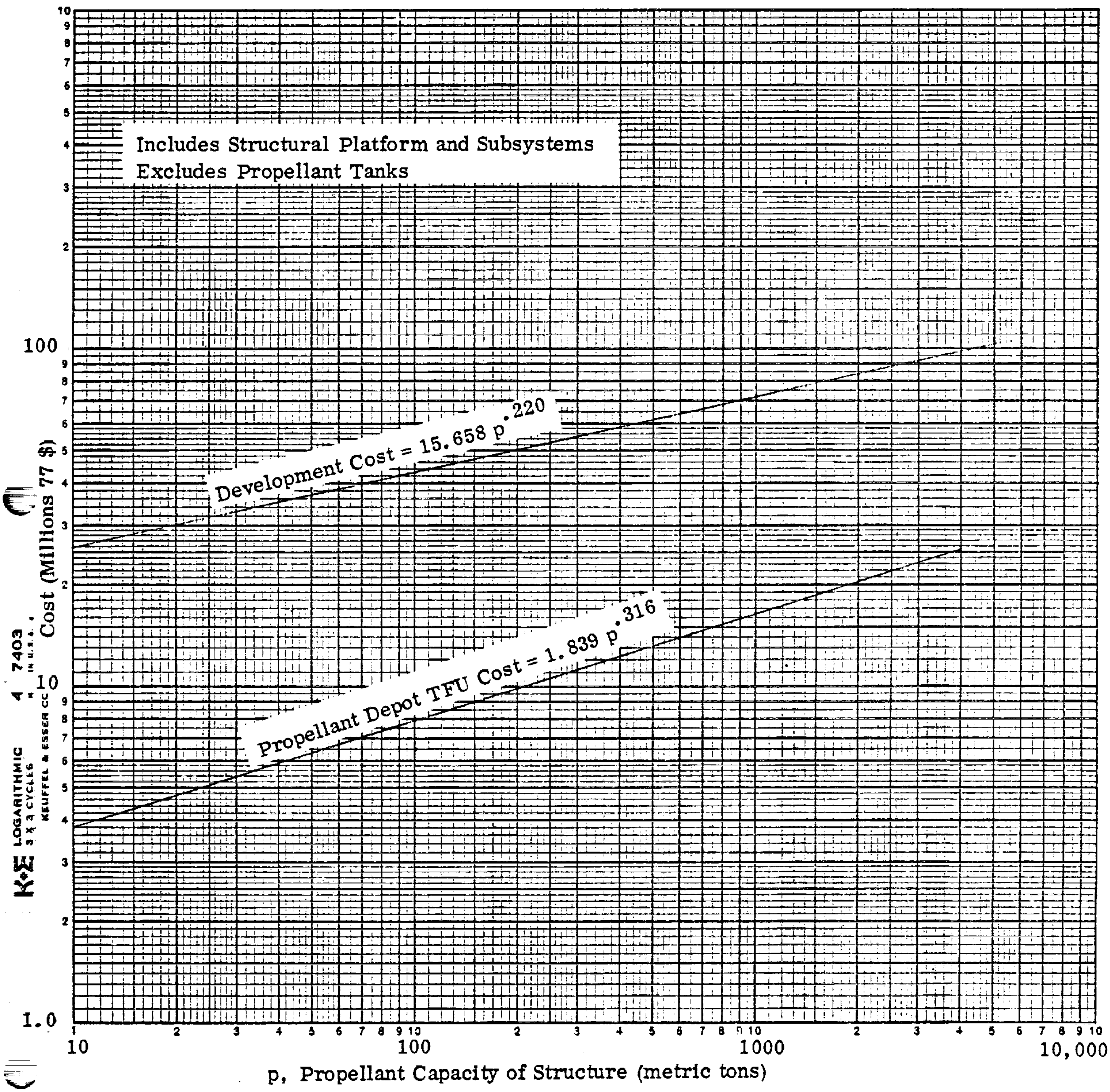

Figure G-1. Propellant Depot Scaling Relationships

G.1.3 Tank Module Scaling Relations

Assume costs vary exponentially with tank size, measured in tons of propellant, according to: $y = aT^b$

where T = propellant capacity of each tank (tons)

a = constant

b = slope coefficient

Using the costs in Table G-1 and capacities in Table G-2 the following tank scaling relations were found:

	Development	First Unit
LH_2 tank	$2.777T^{.849}$	$.248T^{.718}$
LO_2 tank	$.261T^{.849}$	$.034T^{.718}$
Aluminum tank	$.222T^{.849}$	$.029T^{.718}$

The above relationships are plotted in Figure G-2.

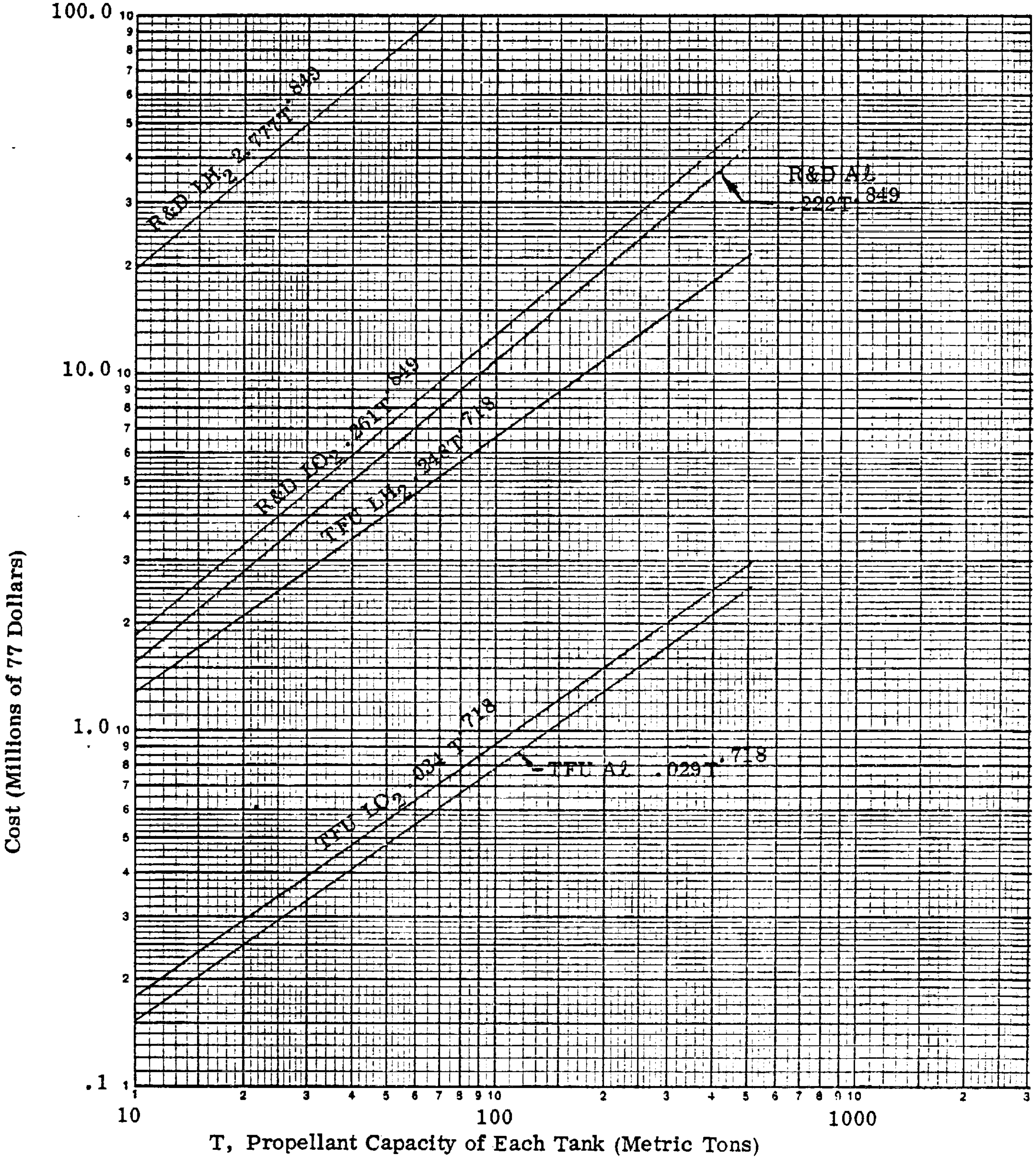

Figure G-2. Propellant Tank Scaling Relationships.

G.1.4 Learning Curves

The OPH Study used a 90% learning for propellant tanks and no learning for the propellant depot (because of fabricating only 1-2 units).

For the Lunar Resources Study use 90% for both tanks and depot. The equation can be expressed as follows:

$$\text{Total Production Cost} = \underline{a x^{.848}}$$

a = 1st unit cost (from Sections G.1.2 and G.1.3)

x = quantity to be produced

G.1.5 Propellant Depot Sizing

Propellant capacity requirements for each propellant depot were determined by analyzing the usage of the depot in each scenario. An absolute minimum storage capacity, with no contingency, was first identified and then a minimum capacity, with contingency was recommended. Absolute minimum capacities are shown in Section G.7, Tables G-44, G-45 and G-46. Recommended capacities are shown in Table 4-31 on page 4-105 of Volume II.

Minimum propellant capacity was found by identifying the vehicles which will use each depot and determining how much propellant would be required to tank these vehicles for one trip. In order to allow for any contingencies the depot should be sized slightly larger. Arbitrary 1-6 month propellant supplies were selected as the basis for recommended minimum capacities, depending on depot location.

Standardized depot platform and tank sizes were selected to avoid the excessive costs of multiple development and to improve unit costs through learning. Two platform sizes were chosen: 1000 metric ton capacity and 5000 metric ton capacity. It was assumed that these platforms could be ganged together as required to meet propellant capacity requirements. Two standard tanks sizes were chosen: 200 tons for LO_2 and aluminum and 100 tons for LH_2. These tank modules are installed on the platforms to attain the required capacity. The required quantities of platforms and tanks are shown in Table G-3.

G.1.6 Cost Estimates

A. Assumptions/Ground Rules

1. All costs are in constant 1977 dollars.
2. Development of the two depot platforms is considered similar. Development cost of the second platform is assumed to be 40% of the cost had it been a single development.
3. Costs of standard size units are as follows: (References Figures G-1 and G-2):

	R&D	TFU
Tanks:		
200 ton LO_2	23.454	1.526
100 ton LH_2	138.540	6.768
200 ton A	19.949	1.302
Propellant Depot Platform:		
1st development - 5000 ton capacity	101.979	27.130
2nd development - 1000 ton capacity	28.628	16.315

B. LRU Option B

1. Propellant Depot Platform
   a. R&D Costs

5000 ton	101.979
1000 ton	28.628
Total R&D	$130.607 million

Table G-3. Propellant Depot Requirements.

Depot Location	Recommended Min. Capacity (metric tons)			No. of tanks required			No. of Platforms Req'd 1000 Ton	5000 Ton
Option B	LO_2	LH_2	A	LO_2	LH_2	A		
LEO	3158	688	-	16	7	-	-	1
GEO	82	12	-	1	1	-	1	-
LLO	56	8	-	1	1	-	1	-
SMF	9939	480	-	50	5	-	1	2
Option C								
LEO	4588	5101	-	23	51	-	-	2
GEO	454	65	-	3	1	-	1	-
LLO	6728	2396	-	34	24	-	-	2
Moon	7178	-	-	36	-	-	See Note (2)	
Option D								
LEO	3254	486	-	17	5	-	-	1
GEO	454	65	-	3	1	-	1	-
LLO	6117	133	-	31	2	-	2	1
Moon	12927	-	5135	65	-	26	See Note (2)	

Notes: (1) Standard tank sizes: LO_2 200 Tons; LH_2 100 Tons; A 200 tons

(2) No platform is required. These tanks are used with the LO_2 liquefaction facility on the lunar surface and to store the aluminum propellant manufactured on the moon.

B. <u>LRU Option B</u> (cont)

1. Propellant Depot Platform (cont)

b. Production

(1) 5000 ton (3 required)

$$\text{Cost} = 27.130\ (3)^{.848}$$

$$= \$68.873 \text{ million}$$

(2) 1000 ton (3 required)

$$\text{Cost} = 16.315\ (3)^{.848}$$

$$= \$41.418 \text{ million}$$

Total Platform production $110.291 million

2. Propellant Tanks

a. R&D Costs

LO_2 tanks	23.454
LH_2 tanks	138.540

$161.994 million

b. Production

(1) LO_2 tanks (68 required)

$$\text{Cost} = 1.526\ (68)^{.848}$$

$$= \$54.641 \text{ million}$$

(2) LH_2 tanks (14 required)

$$\text{Cost} = 6.768\ (14)^{.848}$$

$$= \$63.442$$

Total Tank Production $118.083 million

3. Total Propellant Depot Cost - Option B

R&D	130.607 + 161.994	=	292.601
Production	110.291 + 118.083	=	228.374
			$520.975 million

C. LRU Option C

1. Propellant Depot Platform

a. R&D Costs

5000 ton	101.979
1000 ton	28.628
	$130.607 million

b. Production

(1) 5000 ton (4 required)

Cost $= 27.130\ (4)^{.848}$

$= \$87.902$ million

(2) 1000 ton (1 required)

Cost $= 16.315\ (1)^{.848}$

$= \$16.315$ million

Total Platform Production 104.217

2. Propellant Tanks

a. R&D Costs

LO_2 Tanks	23.454	
LH_2 Tanks	138.540	
		$161.994 million

b. Production

(1) LO_2 Tanks (96 required)

Cost $= 1.526\ (96)^{.848}$

$= \$73.202$ million (Avg Cost $.763 million)

(2) LH_2 Tanks (76 required)

Cost $= 6.768\ (76)^{.848}$

$= \$266.312$ million

Total Tank Production $339.514

3. Total Propellant Depot Cost - Option C

R&D	130.607 + 161.994	=	292.601
Production	104.217 + 339.514	=	443.731
			$736.332 million

Allocation between lunar and space based depots:

Lunar Based - 36 tanks (production only) @ .763 = $ 27.468 million

(no charge for R&D)

Space Based - Balance $708.864 million

D. <u>LRU Option D</u>

1. Propellant Depot Platform

a. R&D Costs

5000 ton	101.979
1000 ton	28.628
	$130.607 million

b. Production

(1) 5000 metric ton (2 required)

$$\text{Cost} = 27.130\ (2)^{.848}$$

$$= \$48.834 \text{ million}$$

(2) 1000 metric ton (3 required)

$$\text{Cost} = 16.315\ (3)^{.848}$$

$$= \$41.418 \text{ million}$$

Total Platform Production $ 90.252 million

2. Propellant Tanks

a. R&D Costs

LO_2 tanks	$ 23.454
LH_2 tanks	138.540
A	19.949

$181.943 million

D. LRU Option D (cont)

2. Propellant Tanks (cont)

b. Production

(1) LO_2 Tanks (116 required)

Cost = $1.526\ (116)^{.848}$

= \$85.944 million (avg Cost \$.741 million)

(2) LH_2 Tanks (8 required)

Cost = $6.768\ (8)^{.848}$

= \$39.471 million (avg cost \$4.934 million)

(3) A Tanks (26 required)

Cost = $1.302\ (26)^{.848}$

= \$20.630 million (avg. Cost \$.793 million)

Total Tank Production \$146.045 million

3. Total Propellant Depot Cost - Option D

R&D	130.607 + 181.943	=	\$312.550
Production	90.252 + 146.045	=	236.297
			\$548.847 million

Allocation between lunar and space based depots:

Lunar Based (tanks only; no R&D charge) \$ 68.783 million

LO_2 - 65 req'd @ .741 = 48.165

Aluminum - 26 req'd @ .793 = 20.618

Space Based (balance of \$548.847) \$480.064 million

E. Propellant Depot Operations Costs

Annual Operating Costs of the depot consist of spares, maintenance and operating labor. On-site maintenance and operating labor are covered in "Construction Maintenance Crew" costs. An annual allowance for maintenance of 3% of production will be made to cover earth based support of maintenance operations. For spares a 1% per year allowance will be made.

<u>Option B</u>

Production: $520.975 (Ref Table G-4)

Operations: 4% (520.975) = $20.839 million/year

<u>Option C</u>

Lunar based production: $27.468 (Ref. Table G-4)

Lunar based operations: 4% (27.468) = $1.099 million/year

Space based production: $708.864

Space based operations: 4% (708.864) = $28.355 million/year

<u>Option D</u>

Lunar based production: $68.783 (Ref Table G-4)

Lunar based operations: 4% (68.783) = $2.751 million/year

Space based production: $480.064 (Ref. Table G-4)

Space based operations: 4% (480.064) = $19.203 million/year

Table G-4. Summary Cost Table - Propellant Depot. (Millions of 77 $)

LRU CONCEPT	R&D	PRODUCTION	TOTAL	LUNAR BASED	SPACE BASED
B	292.601	228.374	520.975	-	520.975
C	292.601	443.731	736.332	27.468	708.864
D	312.550	236.297	548.847	68.783	480.064

G.2 HABITATS

This category encompasses a much wider range of elements than the propellant depot of the previous section. Habitat includes any living quarters, shelters, or space stations on the lunar surface or in any of the orbits being considered. The four groups of habitats were previously defined and discussed in Section 4.5.2. Costs or cost scaling relationships will be derived in this section for each of the habitat elements.

G.2.1 LEO Modular Space Station

The basis for space station cost estimates will be: Modular Space Station, Phase B Extension, Program Cost and Schedules, Report SD71-226-1 and -2, North American Rockwell, January 1972. The space station in the referenced report consists of 6 replaceable station modules, 2 core modules, 1 power module, 1 cargo module and 3 RAM modules. We will eliminate the RAM from consideration. Figure 4-21 shows the basic configuration.

Weights and costs for the 12 man station are shown in Tables G-47 and G-48 of Section G.7. Costs were adjusted to account for inflation and to account for the addition of aluminum shielding on one of the modules to provide solar flare protection.

Scaling relationships were derived under the following assumptions:

(1) Cost varies logarithmically with total space station dry weight according to:

$$y = aW^b.$$

(2) The exponent b has a value of .5 for development and .67 for first unit cost.

The following equations were derived:

(1) First Unit Cost

$$C = 594.8 \left[\frac{W}{W_S}\right]^{.67} \quad \text{(Millions of 77 dollars)}$$

(2) Development Cost

$$C = 2301.6 \left[\frac{W}{W_S}\right]^{.5} \quad \text{(Millions of 77 dollars)}$$

where: W_S = Dry Weight of Rockwell Space Station (Ref Table G-47 of Section G.7)

W = Dry Weight of LEO Modular Space Station in LRU Study

In this study it is desirable to have cost expressed as a function of crew size. This enables estimates to be directly made for the space station once crew size is known. Table G-47 in Section G.7 provides a weight to man ratio of 8.3 m tons/man (with shielding). W then, in the previous equations, can be expressed as: W = 8.3 M, where M = space station crew size. Substituting this expression and W_S = 99.4 m tons into the weight scaling relations we obtain:

(1) First Unit Cost

$$= 594.8 \left[\frac{8.3M}{99.4}\right]^{.67}$$

$$= \underline{\underline{112.7\ M^{.67}}} \quad \text{(Millions 77 dollars)}$$

(2) Development Cost

$$= 2301.6 \left[\frac{8.3M}{99.4}\right]^{.5}$$

$$= 665.1\ M^{.5} \quad \text{(Millions 77 dollars)}$$

The relationships are plotted in Figure G-3.

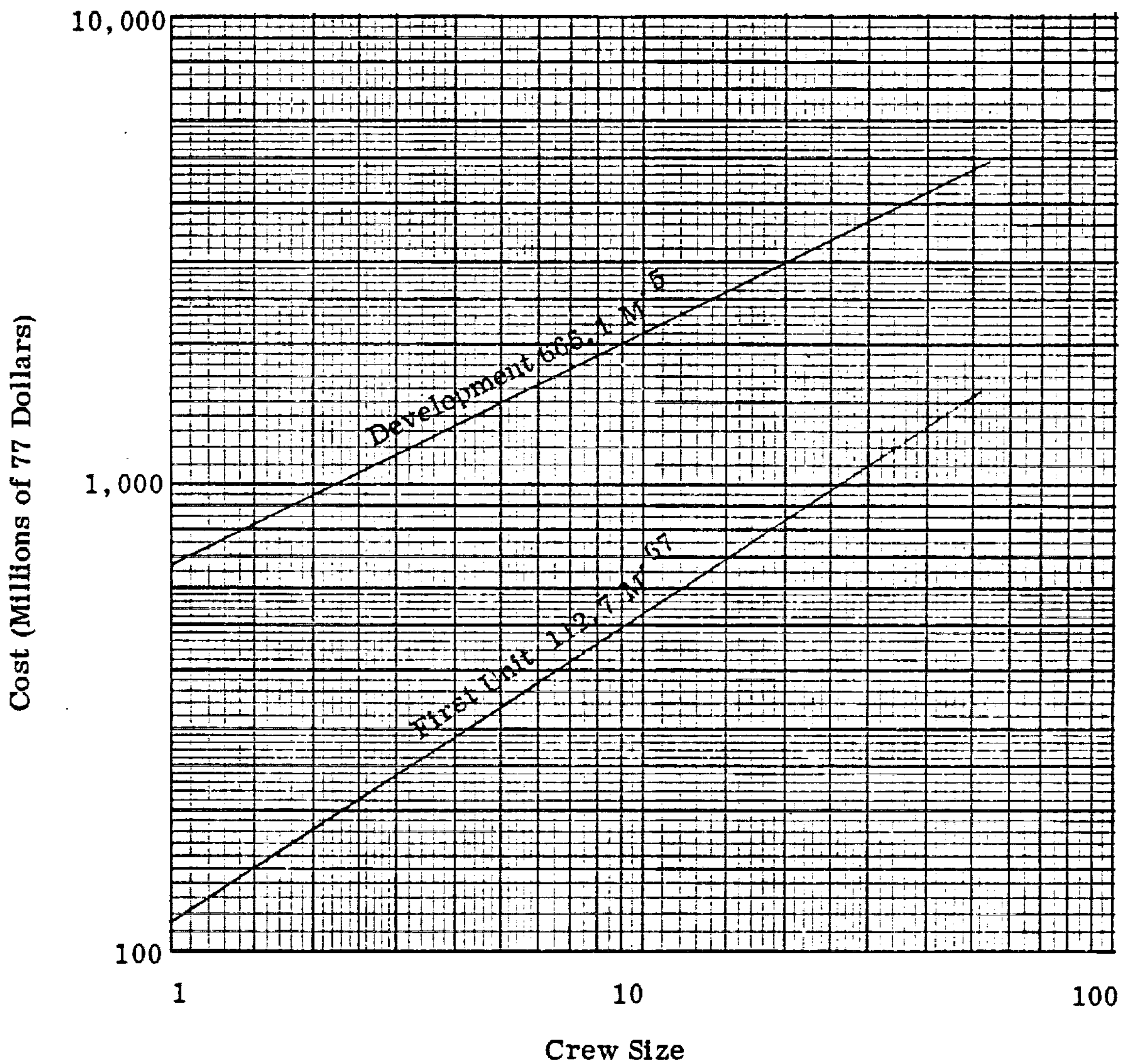

Figure G-3. LEO Modular Space Station Cost Scaling Relationships.

G.2.2 GEO Modular Space Station

The GEO Modular Space Station is the same as the LEO Modular Space Station except for the addition of a Solar Flare Shelter. Shelter characteristics were previously defined in Table 4-36. The approach to cost determination will be to use the relations for the LEO station and add an allowance for the Solar Flare Shelter.

The Solar Flare Shelter consists of a structure and subsystems similar to the other modules except it is spherical. The structure is shielded with some form of bulk lunar material to provide crew protection during periods of high radiation. The basic shelter cost can be scaled from one of the station modules. The cost of designing a means to install the lunar shielding and the cost of installing it is more difficult because the concept has not yet been defined. It will be inexpensive compared to shielding installed on Earth due to the lack of processing required for the material. The primary costs will be development and transportation from the moon to GEO. The following costs will be assumed:

Development: $10/lb of lunar shielding

Production: $1/lb of lunar shielding

For the basic shelter cost assume the structure and subsystems are similar to Station Module 1 shown in Table G-47 and G-48 of Section G.7. The scaling relationships derived are:

(1) First Unit Cost $= 54.0 \left[\frac{W_{SF}}{W}\right]^{.67}$ (Millions of 77$)

(2) Development Cost $= 353.5 \left[\frac{W_{SF}}{W}\right]^{.50}$ (Millions of 77$)

where W = Wt. of Station Module 1 in Table G-47= 8.55 m tons.

W_{SF} = Wt. of Solar Flare Shelter without lunar shielding

Using the data from Table 4-36 and the above relationships, shelter costs were estimated as a function of crew size. This data is shown in Table G-5.

Table G-5. Solar Flare Shelter Costs.
(Millions of 1977 Dollars)

Crew Size	D, Development	F, First Unit
12	166.4	19.2
24	222.4	28.1
50	293.2	40.5
100	358.2	52.5
200	441.0	68.5
400	625.8	108.6
800	889.1	172.2
1600	1264.7	273.4
3200	1801.8	434.2

Total cost for the GEO modular space station can be determined by the scaling relationships in Section G.2.1 combined with the Solar Flare Shelter cost in Table G-5. The basic space station relationships, from Section G.2.1, adjusted to remove the aluminum shielding for LEO are:

(1) First Unit Cost $= 111.1M^{.67}$

(2) Development Cost $= 647.0M^{.5}$

Total cost can be expressed as:

(1) First Unit Cost $= \underline{\underline{111.1M^{.67} + F}}$ (Millions of 77 dollars)

(2) Development Cost $= \underline{\underline{647.0M^{.5} + D}}$ (Millions of 77 dollars)

where M = GEO Space Station Crew Size

F = First Unit Cost of Shelter (Ref. Table G-5)

D = Development Cost of Shelter (Ref Table G-5)

The equations are shown in Figure G-4.

G.2.3 Temporary Shelter

The temporary shelter was defined in Figure 4-24. It consists of two space station crew accommodation modules and a short core module from the modular space station with no radiation shielding. The shelter will accommodate 6 persons. For costing purposes assume the two major modules are identical to Rockwell Space Station Modules #1 and #2 and the short core is identical to the Growth Core Module.

From Tables G-47 and G-48 of Section G.3 the following data was derived:

Development Cost = \$751.3 Million

First Unit Cost = \$124.8 Million

Total Weight = 45843 lbs (20.79 m tons)

Costs were adjusted to 1977 dollars and programmatic costs were allocated by weight. Using a weight to man ratio of 3.47 m tons/man, the following scaling relations were obtained:

(1) Development Cost $= 751.3 \left[\frac{3.47M}{20.79}\right]^{.5}$

$$= \underline{\underline{306.9\ M^{.5}}}$$

(2) First Unit Cost $= 124.8 \left[\frac{3.47M}{20.79}\right]^{.67}$

$$= 37.6\ M^{.67}$$

These cost scaling relationships are depicted in Figure G-5.

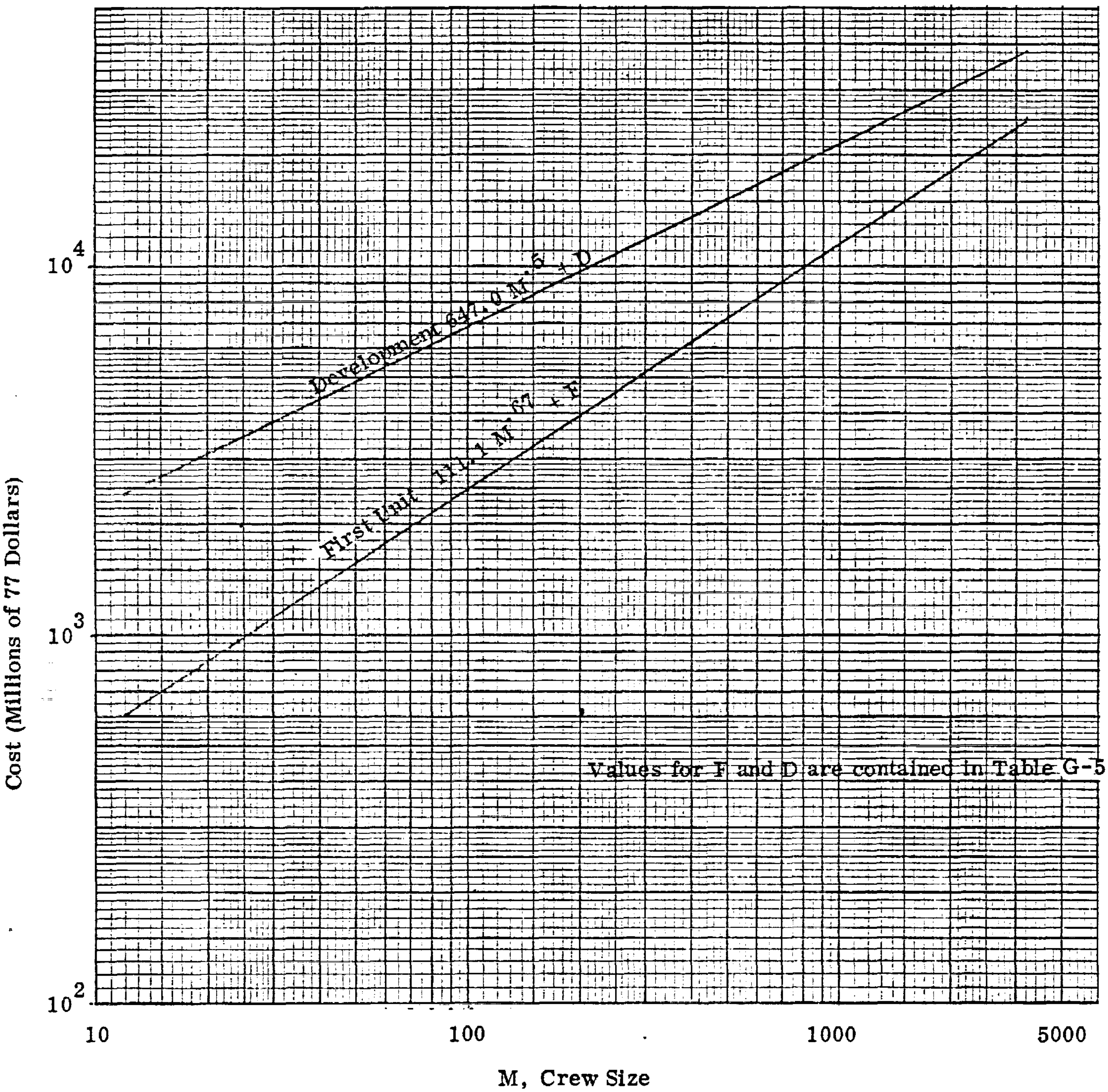

Figure G-4 GEO Modular Space Station Cost Scaling Relationships

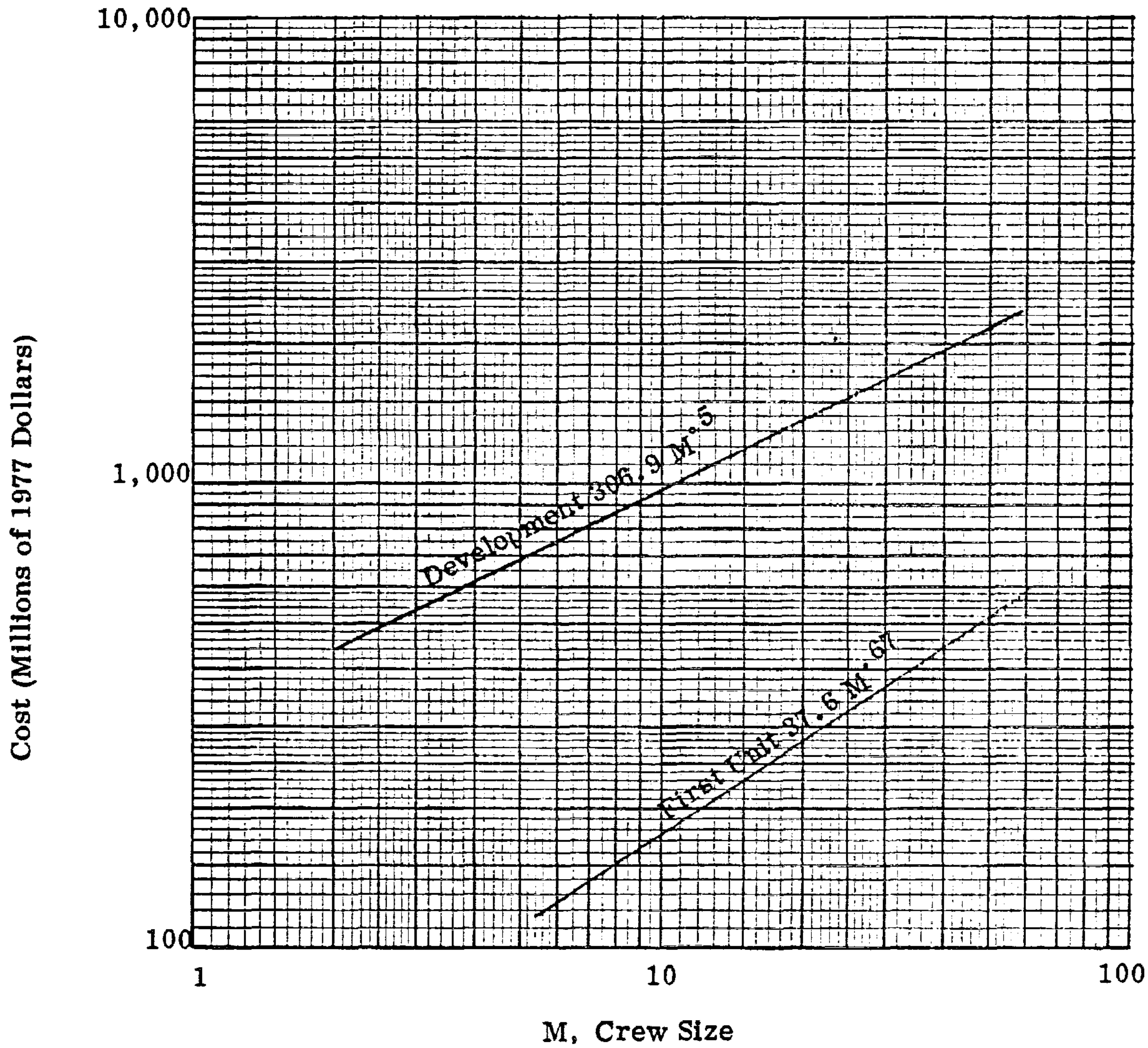

Figure G-5. Temporary Shelter Cost Scaling Relationships.

G. 2. 4 Lunar Base Habitat (Small Crew)

The basis for the cost estimates of the lunar base habitats will be the Lunar Base Synthesis Study, Final Report, Vol IV, North American Rockwell, 15 May 1971. In the synthesis study, 13 modules were used for the base. We will use only 8 of these modules as shown in Figure 4-25. The Rockwell cost data is shown in Table G-49 of Section G.7 and size data is presented in Table G-7.

The cost data was adjusted to account for the deletion of the five modules and is shown in Table G-5. Items other than module hardware were scaled down using a weight ratio. Adjustments were made for inflation using the GNP Price Deflator. The adjusted data is shown in Table G-6.

Table G-6. 12 Man Lunar Base Costs.

Cost Element	R & D Millions of 1970 Dollars	R & D Millions of 1977 Dollars	Production Millions of 1970 Dollars	Production Millions of 1977 Dollars
Modules - Hardware				
1) Crew & Medical	63.1	97.616	14.8	22.896
2) Crew & Operations	48.5	75.030	25.6	39.603
3) Sortie & Transient	25.0	38.675	17.1	26.454
4) Lab & B/U	40.3	62.344	23.7	36.664
5) Assy & Recreation	23.0	35.581	11.0	17.017
6) Base Maintenance	13.1	20.266	7.1	10.984
7) Drive-in Garage	9.6	14.851	4.3	6.652
8) Drive-in Warehouse	8.0	12.376	4.6	7.116
GSE	30.793	47.637	2.145	3.318
Systems Test Hardware	107.546	166.374	—	—
Launch Operations Support	—	—	10.647	16.471
Facilities	32.325	50.007	—	—
Logistics & Training Equip.	9.192	14.220	4.060	6.281
System Engr'g Support	24.665	38.157	3.447	5.333
Project Mgmt	24.665	38.157	3.83	5.925
Total		711.291		204.714

Notes:

(1) All cost elements, except for module hardware, were scaled down by weight to account for modules excluded from this study:

$$\text{Cost} = (\text{Rockwell Moon Base Cost}) \times \left[\frac{\text{Lunar Resources Hardware Weight}}{\text{Rockwell Hardware Weight}}\right]$$

$$= (\text{Rockwell Moon Base Cost})\ .766$$

(2) Costs adjusted to 1977 dollars using GNP Price Index (1970 = 91.36; 1977 = 141.3)

Table G-7. Size Data for Lunar Base Habitat.

Mod #	Module	Crew Size	Gross Dry Wt (lbs)
1	Crew & Medical	4	8291
2	Crew & Operations	4	9292
3	Sortie & Transient	4	8818
4	Lab & B/U Command	-	8640
5	Assy & Recreation	-	7574
6	Base Maintenance	-	6297
7	Drive in Garage	-	4807
8	Drive in Warehouse	-	5024
	Total	12	58,743 (26.64 tons)

Notes:

(1) Data based on Lunar Based Synthesis Study, Rockwell

(2) Weight of five habitats deleted from the Rockwell Scenario is 17,914 pounds

(3) Habitat weight to man ratio $= \frac{26.64}{12} = 2.22$ metric tons/man

Scaling relationships can be derived from the cost data in Table G-6 using the assumptions that: (1) cost is a logarithmic function of weight according to: $y = aW^b$, and that (2) the exponent b is .5 for development and .67 for first unit cost. These exponent values are representative of a system which is basically structure and are typical of those used throughout the industry.

(1) For development the scaling relation is:

$$\text{Cost} = 711.291 \left[\frac{W}{W_{lb}}\right]^{.5} \quad \text{(Millions of 77 dollars)}$$

where: W = Weight of the lunar base under consideration (lbs)

W_{lb} = Weight of lunar base whose costs are shown in Table G-6.

= 26.64 metric tons

(2) First unit cost can be expressed as:

$$\text{Cost} = 204.714 \left[\frac{W}{W_{lb}}\right]^{.67}$$

where: W and W_{lb} are the same as above

Using the above scaling relationships, costs for any size lunar base can be estimated. For the purposes of this study it is desirable to express these relations in terms of crew size. From Table G-7 we find that the lunar base weight to crew size ratio is 2.22 metric tons per man. Lunar base habitat weight then may be expressed as 2.22M, where M = crew size of the habitat. The scaling relationships can now be expressed as follows:

(1) Development Cost

$$C = 711.291 \left[\frac{2.22M}{26.64}\right]^{.5}$$

$$= \underline{\underline{205.332\ M^{.5}}}$$

(2) First Unit Cost

$$C = 204.714 \left[\frac{2.22M}{26.64}\right]^{.67}$$

$$= \underline{\underline{38.734\ M^{.67}}}$$

The above relations are plotted in Figure G-6.

G.2.5 Large Lunar Base (Shuttle Tanks)

This 1200 person base is described in Figures 4-26 and 4-27. The base consists of the LH_2 tank portions of expended Shuttle external tanks. The cost of the tanks themselves is negligible since they are normally expended. The primary costs are in the furnishings and equipment and their installation, in the tank modifications required and transportation. Cost/lb for the Shuttle tank derived base are not unlike those for the small lunar base. For the small lunar base most of the assembly and installation tasks were performed on Earth. Modules were then transported to the moon. In the case of the large lunar base the assembly will probably take place in LEO and the completed tank module will then be transported to the lunar surface.

It will be assumed that the cost scaling relationships derived for the small lunar base hold also for the large lunar base. For the large lunar base the dry weight may be expressed as: $W = 2.55\ M$, where M = number of people habitat will support. This excludes the weight of the external tank hardware since this is essentially a no charge item. Combining this with the scaling

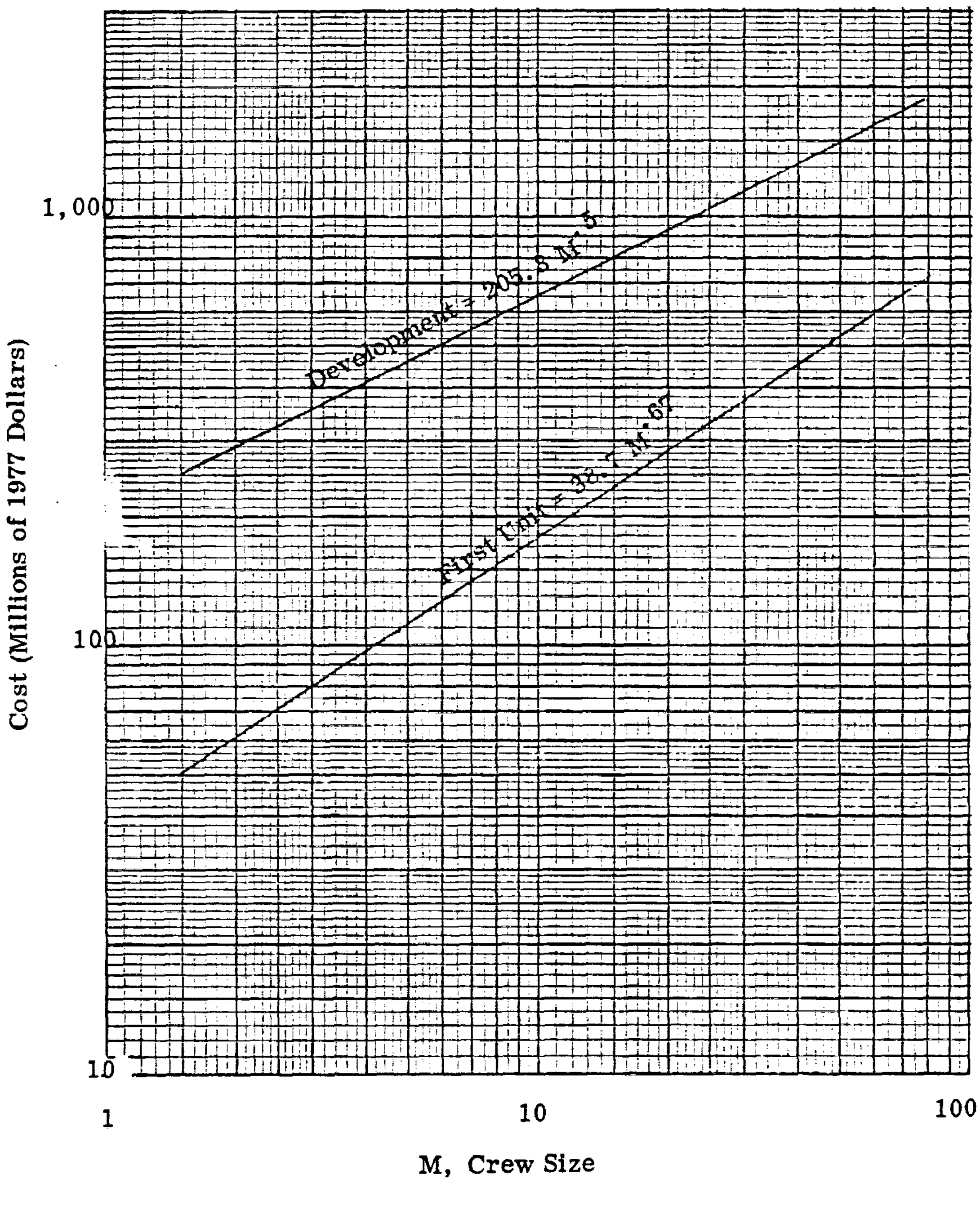

Figure G-6. Lunar Base Habitat (Small Crew) — Cost Scaling Relationships.

relations in Section G.2.4 the following relationships were obtained for the Large Lunar Base: These relationships are plotted in Figure G-7.

(1) Development Cost $= 711.291 \left[\frac{2.55M}{26.64}\right]^{.5}$

$= \underline{\underline{220.1\ M^{.5}}}$ (Millions of 77 dollars)

(2) First Unit Cost $= 204.714 \left[\frac{2.55\ M}{26.64}\right]^{.67}$

$= 42.5\ M^{.67}$ (Millions of 77 dollars)

G.2.6 Space Manufacturing Facility Habitat

The Space Manufacturing Facility (SMF) is shown in Figures 4-29 and 4-30. Like the large lunar base, ET hydrogen tanks are utilized as the basic habitats. Since the tanks are normally an expended item they are essentially a free item. The major costs are in equipping the tanks with their subsystems, flooring, partitions, etc., in LEO, transferring them to GEO and assembling them into a single installation. There is also a requirement for shielding the SMF using lunar material. Initially this would be some type of "sandbag" configuration. As the facility began manufacturing, the raw lunar material in the sandbags could be converted to a more permanent material, such as bricks, which are more securely attached. Cost of the shielding is difficult to define without a better definition of the configuration. Major costs of the shield include: Mining and installation labor, operation of the transportation elements which transfer the lunar soil to orbit, special equipment to process the soil into bricks or other permanent configuration.

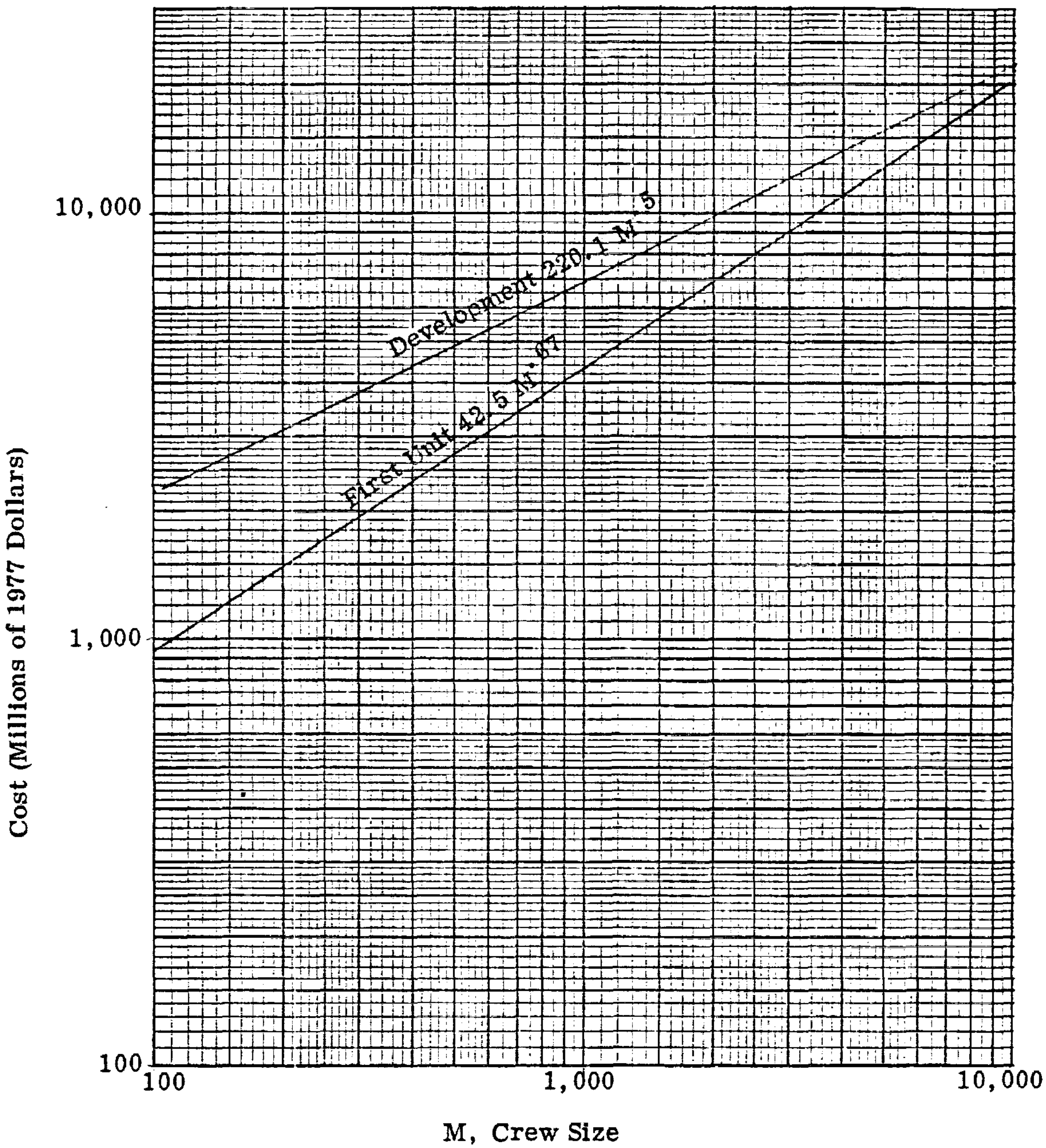

Figure G-7. Large Lunar Base Cost Scaling Relationships.

Except for size, the SMF is similar to the LEO Modular Space Station described in Section G. 2. 1 . The cost scaling relationships developed in that section are assumed to hold also for the SMF (excluding shielding). Weight of the external tanks will be excluded from the scaling relations since they are no charge items. Since the scaling relation is for a low earth orbit station, an allowance could be made for transporting all SMF material from LEO to GEO. This transportation cost would include the cost of operating the COTV: propellants, maintenance and spares. These costs are probably a couple of orders of magnitude lower than the cost of the SMF and will have no significant effect on total cost. The transportation cost then is assumed negligible.

The Modular Space Station relationships in Section G. 2. 1 included shielding. For the SMF the shielding will be considered separately and the relations can be expressed as follows:

(1) First Unit Cost $= 589.4 \left[\frac{W}{W_S}\right]^{.67}$

(2) Development Cost $= 2247.4 \left[\frac{W}{W_S}\right]^{.5}$

where: W_S = Modular Space Station Weight without radiation shielding

= 74.8 metric tons (ref Table G-47)

W = 2.94 M (ref. Table 4-32)

M = SMF crew size

Substituting values, the above relationships become:

(1) First Unit Cost $= \underline{\underline{67.4\ M^{.67}}}$

(2) Development Cost $= \underline{\underline{445.6\ M^{.5}}}$

The basis for the shielding cost of the SMF is the work done in the 1977 Ames Space Settlement Summer Study in "Habitat Design- An Update," by Bock, Lambrou and Simon, 1977. In that study total manufacturing cost was \$.21 per kg. Assume development costs are ten times that, or \$2.10/kg. Shielding weight for a 1500 man facility is 85,500 tons or 57 metric tons per person. Shielding costs can be expressed as follows:

(1) Development Cost = 57 tons/person × \$2100/ton

= \$.120 million/person

= .120M (millions of 77 dollars)

(2) First Unit Cost = 57 tons/person × \$210/ton

= \$.012 million/person

= .012M (millions of 77 dollars)

where M = Number of people in SMF crew

Total SMF costs are the combined total of the SMF and the required shielding:

(1) Development Cost $= 445.6\ M^{.5} + .120\ M$

(2) First Unit Cost $= 67.4\ M^{.67} + .012\ M$

These relationships are plotted in Figure G-8.

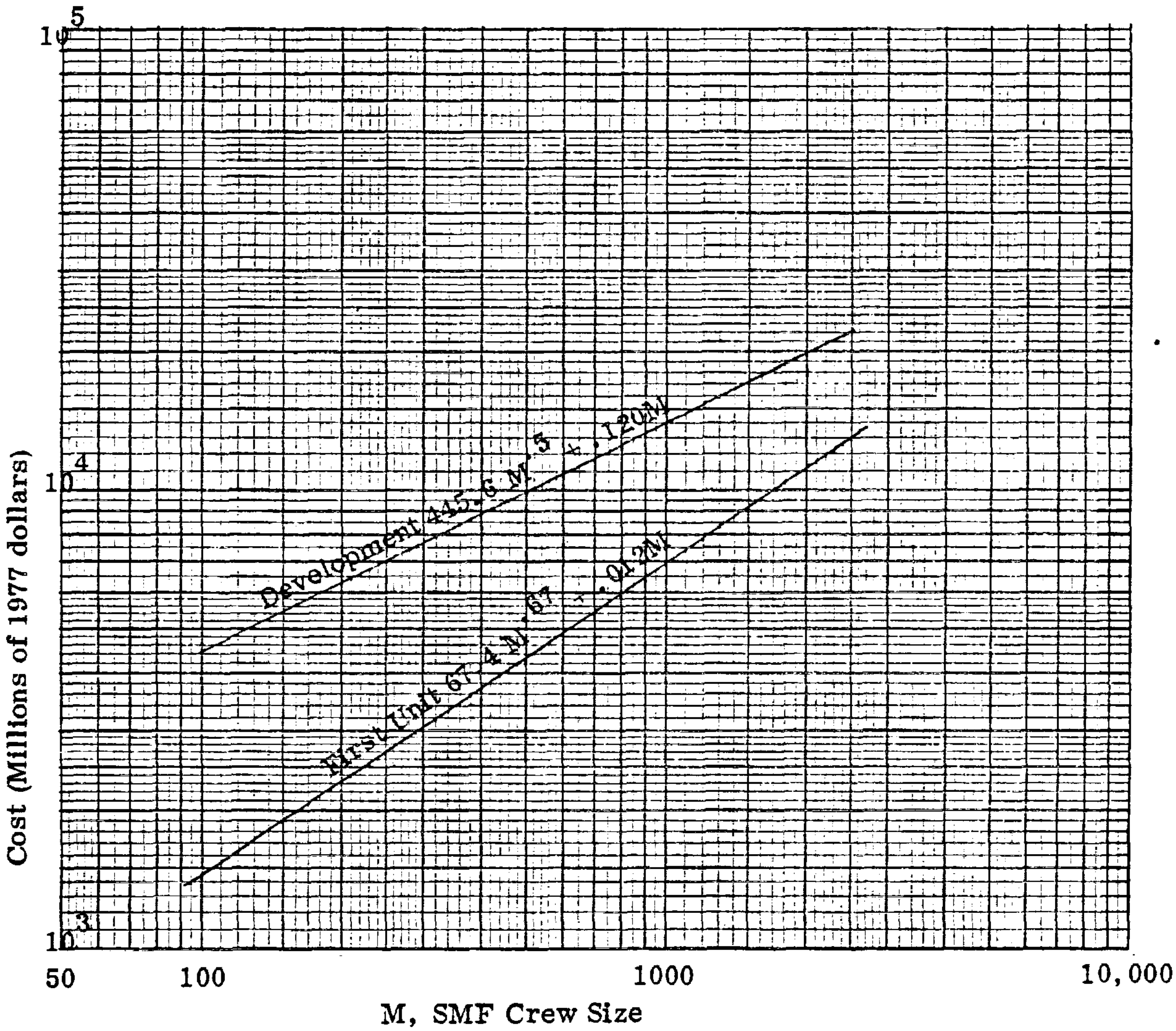

Figure G-8 . SMF Manufacturing Facility Cost Scaling Relationships.

G.2.7 Habitat Operations Cost

Additional recurring costs exist for each of the habitats described previously which are operational in nature. These are the costs of maintaining the habitats. The costs can be broken down into two categories: spares and maintenance. Table 5B-7 in the Appendix provides a means to estimate these Operations Costs as a function of first unit cost. Using the data from Table 5B-7 the following annual cost percentages were obtained:

Spares - 0.4% of first unit cost/year

Maintenance - 5.4% of first unit cost/year

The above percentages apply to each habitat discussed in the previous sections. First unit costs of the GEO Modular Space Station and Space Manufacturing Facility should be adjusted to delete the cost of lunar material shielding since lunar shielding maintenance requirements are probably insignificant.

G.3 TRANSPORTATION

Transportation elements include all personnel and cargo carrying vehicles in the LRU options. There are 15 different types of vehicles. These were previously defined and discussed in Section 4.6.2. This section contains the costs of each of these elements.

G.3.1 Heavy Lift Launch Vehicle

This is the SPS baseline configuration and is defined in Figure 4-36. Cost methodology is contained in Note 1 of Table 5-4, which is included in Appendix F.

G.3.2 Personnel Launch Vehicle

This is the SPS baseline configuration and is defined in Figure 4-37. Cost methodology is contained in Note 2 of Table 5-4, which is included in Appendix F.

G.3.3 Personnel Orbital Transfer Vehicle

This is the SPS baseline configuration and is defined in Figure 4-38. Cost methodology is contained in Note 3 of Table 5-4 (Appendix F). Costs include provisions for the passenger and crew modules.

G.3.4 Cargo Orbital Transfer Vehicle

This is the SPS baseline configuration and is defined in Figure 4-39. This is a non-reusable vehicle. Costs are shown in Note 4 of Table 5-4 (Appendix F).

G.3.5 Passenger and Crew Modules

These modules are defined in Figure 4-40. They are identical to the ones included in the POTV costs for the earth baseline. Module costs are split out here for use with LRU transportation elements. References used were:

(1) Solar Power Satellite Concept Evaluation, Activities Report July 1976 to June 1977, Vol. II, JSC-12973, July 1977.

(2) Initial Technical, Environmental and Economic Evaluation of Space Solar Power Concepts, Vol. II, JSC-11568, Aug 1976.

Table G-8 shows the costs provided by the referenced documents. Even though costs in Ref. (1) were based on weight statements in Ref. (2) an increase is noted. It is assumed that in Ref. (1) a more detailed cost analysis was performed and that it provides more credible cost numbers. These estimates will be used for the LRU concepts.

Table G-8. Passenger and Crew Module Costs.

Source	Passenger Module		Crew Module		Remarks
	Development	TFU	Development	TFU	
Ref (1)	287	13	524	24	mils 77 $
Ref (2)	120	6	365	34	mils 76 $

Operating costs for the modules includes the cost of spares and maintenance. Assume these two items are 1 percent/year and 3 percent/year of first unit cost. Costs were computes as follows:

Passenger Module

Spares: .01 (13) = $.13 million/year/module

Maintenance: .03 (13) = $.39 million/year/module

Crew Module

Spares: .01 (24) = $.24 million/year/module

Maintenance: .03 (24) = $.72 million/year/module

G.3.6 Shuttle Derived Vehicle (SDV)

The SDV is described in Figure 4-42. Cost and definition of the SDV booster is shown in Appendix E. Adjusting costs in the Appendix for inflation and to include the main engines the following is obtained:

Booster Development Cost = \$5311.50 (millions of 1977 dollars)

Booster First Unit Cost = \$364.72 (millions of 1977 dollars)

Costs for the cargo pod were obtained from Future Space Transportation Systems Analysis Study, BAC Report D180-20242-3, Vol. 3, Dec 1976, Tables 2.2-1 and 2.3-4. Development cost, including modifications to the external tanks, and first unit cost, including the modified tank are:

Cargo Pod Development Cost = \$1520.64 (millions of 1977 dollars)

Cargo Pod First Unit Cost = \$121.44 (millions of 1977 dollars)

The First Unit Cost includes both expendable and reusable hardware. The expendable hardware portion (external tank and shroud) is \$18 million.

Production cost of the flight hardware can be determined using a 90 percent learning curve for the number of units built plus a 30 percent allowance for production program level costs. Program level costs include such items as program management and sustaining engineering. The following relation can be used:

$$\text{Production Cost} = 1.3\ (\text{First Unit Cost})\ N^{.848}$$

where N = Number of units produced

SDV operations costs consist of propellants, refurbishment of reusable hardware and maintenance. Propellant rates are based on: Solar Power Satellite Concept Evaluation, Activities Report July 1976 to June 1977, Vol. II, NASA/JSC, Figures VI-E-5, 6 & 9. It was assumed that the government borrows money at a 9 percent interest rate to finance the propellant production facilities and the coal price is \$17/ton. The same reference shows LO_2 losses at 56% and LH_2 losses were assumed to be 10%. Table G-9 provides the cost per flight of propellants.

The remaining operations costs were estimated using the operations cost per flight of the SDV described in Shuttle Derivative Vehicles Study, Vol. I, BAC Report.

Table G-9. SDV Propellant Costs.

Element	Mixture Ratio	Total Propellant (millions lbs)	Propellant Breakdown (millions lbs) LO_2	LH_2	C_3H_8
Booster	2.68:1	6.466	4.709		1.757
Cargo Pod	6:1	.286	.245	.041	
Losses			2.774	.003	.176
Total per flight			7.728	.044	1.933
Cost per pound ($)			.021	.54	.37
Total Cost per flight (mils 77 $)			.162	.024	.715

D180-228-75-1, Dec. 1977. Operations costs for the SDV are shown in Figure 4-6 of that report and total $13.605 million per flight. The following adjustments were made for the LRU SDV: (1) launch facility operations costs were removed ($1.905 million). These will be included under facility operations, (2) propellant costs were removed and will be replaced with the costs calculated above ($1.088 million) (3) an arbitrary 15 percent of the costs was removed for SRB refurbishment and spares since the LRU version does not contain SRB's ($2.041 million). Total LRU SDV cost is: $13.605 - 1.905 - 1.088 - 2.041 + .901 = $9.472 million/flt. This includes spares and refurbishment of reusable hardware.

G.3.7 Space Shuttle

The current space shuttle configuration is shown in Figure 4-43. Minor modifications would be necessary to fit the 75 passenger module into the cargo ban. These costs are assumed negligible and no development cost will be used. For the purposes of the LRU study a charge of $20 million per flight will be made.

G.3.8 LRU Personnel Orbital Transfer Vehicle (POTV)

A description of the LRU POTV is provided in Figure 4-44. It is similar to one stage of the POTV used in the earth baseline. Table X-D-13 of NASA/JSC's Solar Power Satellite Concept Evaluation gives second stage POTV costs as follows:

Development	\$328 million (1977 \$)
First Unit	20 million (1977 \$)

Dry weight of the stage is 11,000 lbs; slightly smaller than the 14,774 lb LRU POTV.

The LRU POTV costs were determined from scaling relationships similar to those used previously:

$$\text{Development Cost} = 328 \left[\frac{14774}{11000}\right]^{.5}$$

$$= \underline{\underline{\$380}} \text{ million (1977 \$)}$$

$$\text{First Unit Cost} = 20 \left[\frac{14774}{11000}\right]^{.67}$$

$$= \$24.37 \text{ million (1977 \$)}$$

For production assume a 90 percent learning curve for hardware and allow 20 percent of hardware cost to cover program level costs. Total production cost can be expressed as follows:

$$\text{Total Production Cost} = 1.2\ (24.37)\ N^{.848}$$

$$= 29.24\ N^{.848}$$

where: N = Number of vehicles produced

There are three primary categories for vehicle operations cost: propellants, spares and maintenance. Annual costs for spares and maintenance are assumed to be 1 percent/year and 3 percent/year, respectively, of first unit cost for each vehicle in the fleet.

Spares: .01 (24.37) = \$.244 million/year/POTV

Maintenance: .03 (24.37) = \$.731 million/year/POTV

Total propellant weight is 59.4 metric tons (130,977 lbs). At a mixture ratio of 6:1 the amounts of fuel and oxidizer required per flight is shown in Table G-10. The LO_2 is manufactured from lunar soil and LH_2 is supplied from earth. The cost of LO_2 will be reflected in the LRU facilities development, production and facilities costs. The cost of LH_2 is based on future earth rates.

Table G-10. POTV Propellant Costs (millions 1977 dollars).

	Propellant Weights (lbs)	Losses (lb)	Total Flight (lbs)	Cost/ Flight
LO_2	112,266	13472	125,738	—
LH_2	18,711	3742	22,453	.012

Notes: (1) Propellant losses were assumed to be 20% for LH_2 and 12% for LO_2.

(2) Propellant cost: LH_2: \$.54/lb

Ref: SPS Concept Evaluation, JSC-12973 July 1977, pps. X-D-41, VI-E-20 & 21.

G.3.9 Cargo Orbital Transfer Vehicle

The LRU COTV configuration was defined by Figure 4-45 with variations shown in Table 4-53. Since no cost studies have been performed on similar type vehicles, rough order magnitude costs were determined using cost estimating relationships (CER's). For estimating purposes the COTV was broken down into structural, ion propulsion and solar array elements. CER's are shown in Table G-11 and a weight statement is provided in Table G-12. These tables, together with the data furnished in Tables 4-52 and 4-53 provide the basis for the cost estimates which follow.

The development and production costs for each COTV and each LRU option are shown in Tables G-13, G-14 and G-15. The following notes apply to the tables.

1. Diameter of the ion thruster is the diameter of a circle of equivalent area to the oval shaped thrusters used.
2. One power processing unit per 70 thrusters was assumed.
3. Vehicle First Unit Cost for Ion Thrusters and PPU computed according to:

$$C = (TFU')N^{1tb}$$

where

TFU'	=	Element First Unit Cost
N	=	Number of Thrusters or PPU's per vehicle
1 + b	=	Slope exponent of Total Cost Learning Curve
	=	.848 for 90% curve

4. Learning curve for ion propulsion and vehicle production assumed at 90%.
5. Design of all COTV's is common, except for vehicle size and quantities of elements. Because of the modular design and commonality, development costs of the second and third COTV's are assumed to be only 30% of sole development values.

Table G-11 Cost Estimating Relationships for COTV

Cost Element	Cost Estimating Relationship (Millions of 1977 Dollars) Development	Element First Unit Cost	Remarks
Structure			
Truss	$.55\ W^{.187}$	$.004\ W^{.667}$	Note 2
Tankage/Misc	$10.14\ W^{.187}$	$.007\ W^{.667}$	Note 2
Solar Array	—	—	Note 1
Ion Thrusters	$2.15\ D^{.32}$	$.016\ D^{.74}$	Note 3
Power Processing Units	$12.46\ P^{.18}$	$.27\ P^{.46}$	Note 3

Notes: (1) Basic Development cost for solar array is absorbed by SPS solar array development. Assume a nominal development charge for the COTV of $50 million. For First Unit Cost assume $500/kW (Ref. 1, Table X-C-2 and Ref. 3, Table 3.8).

(2) W = Structural Weight in lbs. CER's are from Reference 2.

(3) D = Thruster Diameter in cm. P = Power Processor output in kW. CER's are from Reference 2.

References: (1) Solar Power Satellite Concept Evaluation, Activities Report, July 1976 to June 1977. Vol. II, NASA/JSC.

(2) Parametric LCC Analysis Technique for Space Systems, 1978 IRAD Study by General Dynamics, Convair Division, Report Pending.

(3) Space-Based Solar Power Conversion and Delivery Systems Study, Vol. IV, Report C-78127, prepared for ECON, Inc. by Arthur D. Little, Inc., March 1977.

Table G-12. COTV Weight Relationships.

Truss Structure	22.8 kg/thruster
Tankage	.08 kg/kg of propellant
Misc. Structure/ACS	14.9 kg/thruster
Ion Thruster	22.0 kg
Power Processing Unit	19.3 kg/thruster
Solar Array	781.3 kg/thruster

6. Program level costs include such items as system test, tooling, program management, sustaining engineering and assembly and checkout. The following allowances were made for these costs: (1) Development - 40% of hardware development costs; (2) First Unit - 10% of hardware first unit cost and (3) Production - 20% of hardware production cost.

There are three primary cost categories for vehicle operation: propellants, spares and maintenance/refurbishment. Vehicle life is assumed to be 50 flights. Spares and maintenance costs are assumed to be 1 percent/yr and 3 percent/yr of first unit cost, respectively, for each vehicle. These costs are shown in Table G-16. They are based on the first unit costs shown in Tables G-13, G-14 and G-15. Propellant costs are also shown in Table G-16. They were computed using the future technology propellant production methods described in Solar Power Satellite Concept Evaluation, JSC-12973, July 1977, (9% interest rate assumed). The cost of LO_2 is reflected in the propellant manufacturing facilities cost since it is manufactured from lunar soil and no charges are shown for it in Table G-16.

Table G-13. COTV Costs for Option B.
(Millions of 1977 Dollars)

Cost Element	Development	First Unit		Production		CER
		Element	Vehicle	Total	Average	Variables
$COTV_2$ (2 req'd) - Total	349.17		137.29	296.54	148.27	
Structure						
Truss	3.19		2.11			12066 #
Tankage	69.95		6.86			30517 #
Misc Structure	54.29		2.78	247.12		7885 #
Solar Array	50.00		14.00			$.28 \times 10^5$ kW
Ion Thrusters	8.88	.42	43.82			N = 240, D = 84 cm
Power Processing Units	63.10	17.05	55.24			N = 4, P = 8204 kW
Program Level Costs	99.76		12.48	49.42		
$COTV_3$ (2 req'd) - Total	143.58		1070.03	2311.27	1155.64	
Structure						
Truss	1.51		10.71			137,750 #
Tankage	38.78		6140			815,850 #
Misc. Structure	25.68		14.11	1926.06		90,021 #
Solar Array	15.00		160.00			3.2×10^5 kW
Ion Thrusters	2.66	.42	345.51			N = 2740, D = 84 cm
Power Processing Units	18.93	17.05	381.02			N = 39, P = 8204 kW
Program Level Costs	41.02		97.28	385.21		
$COTV_4$ (3 req'd) - Total	144.19		2001.55	6097.43	2032.48	
Structure						
Truss	1.74		17.83			295,611 #
Tankage	35.04		42.74			473987 #
Misc. Structure	29.62		23.49			193,184 #
Solar Array	15.00		345.00			6.9×10^5 kW
Ion Thrusters	2.66	.42	660.21			N = 5880, D = 84 cm
Power Processing Units	18.93	17.05	730.32			N = 84, P = 8204 kW
Program Level Costs	41.20		181.96	1016.24		
Total - All COTV's	636.94			8705.24		

Table G-14 COTV Costs for Option C.
(Millions of 1977 dollars)

Cost Element	Development	First Unit Element	First Unit Vehicle	Production Total	Production Average	CER Variables
$COTV_1$ (2 req'd) - Total	409.39		497.66	1074.95	537.48	
Structure						
Truss	3.67		3.48			W = 25536 lb
Tankage	104.31		28.56			W = 258,955 lb
Misc Structure	62.46		4.59	895.79		W = 16688 lb
Solar Array	50.00		75.00			$P = 1.3 \times 10^5$ kW
Ion Thrusters	8.88	.42	161.80			N = 1120, D = 84 cm
Power Processing Units	63.10	17.05	178.99			N = 16, P = 8204 kW
Program Level Costs	116.97		45.24	179.16		
$COTV_2$ (5 req'd) - Total	143.16		1901.92	8935.12	1787.02	
Structure						
Truss	1.48		10.08			W = 125,674 lb
Tankage	38.95		62.37	7445.93		W = 835,254 lb
Misc. Structure	25.24		13.28			W = 82129 lb
Solar Array	15.00		325.00			$P = 6.5 \times 10^5$ kW
Ion Thrusters	2.66	.42	625.00			N = 5512, D = 84 cm
Power Processing Units	18.93	17.05	693.29			N = 79. P = 8204 kW
Program Level Costs	40.90		172.90	1489.19		
$COTV_3$ (3 req'd) - Total	138.47		1843.89	5617.14	1872.38	
Structure						
Truss	1.48		9.94			W = 123,120 lb
Tankage	35.69		45.64			W = 523,026 lb
Misc. Structure	25.15		13.10	4680.95		W = 80,460 lb
Solar Array	15.00		315.00			$P = 6.3 \times 10^5$ kW
Ion Thrusters	2.66	.42	614.21			N = 5400, D = 84 cm
Power Processing Units	18.93	17.05	678.37			N = 77. P = 8204 kW
Program Level Costs	39.56		167.63	936.19		
Total - All COTVs	691.02			15,627.21		

Table G-15. COTV Costs for Option D.
(Millions of 1977 Dollars)

Cost Element	Development	First Unit: Element	First Unit: Vehicle	Production: Total	Production: Average	CER Variables
$COTV_1$ (2 req'd)- Total	409.39		497.66	1074.95	537.48	
Structure						
Truss	3.67		3.48			W = 25,536 lbs
Tankage	104.31		28.56			W = 258,955 lbs
Misc Structure	62.46		4.59	895.79		W = 16688 lb
Solar Array	50.00		75.00			$P = 1.3 \times 10^5$ kW
Ion Thrusters	8.88	.42	161.80			N = 1120, D = 84 cm
Power Processing Units	63.10	17.05	178.99			N = 16, P = 8204 kW
Program Level Costs	116.97		45.24	179.16		
$COTV_2$ (3 req'd)	142.37		1891.62	5762.54	1920.85	
Structure						
Truss	1.48		10.06			W = 125,400 lbs.
Tankage	38.39		59.19			W = 772,279 lbs.
Misc. Structure	25.23		13.26	4802.12		W = 81950 lbs.
Solar Array	15.00		320.00			$P = 6.4 \times 10^5$ kW
Ion Thrusters	2.66	.42	623.85			N = 5500, D = 84 cm
Power Processing Units	18.93	17.05	693.29			N = 79, P = 8204 kW
Program Level Costs	40.68		171.97	960.42		
$COTV_3$ (3 req'd)	138.47		1843.89	5617.14	1872.38	
Structure						
Truss	1.48		9.94			W = 123,120 lbs.
Tankage	35.69		45.64			W = 523,026 lbs.
Misc. Structure	25.15		13.10	4680.95		W = 80 460 lbs.
Solar Array	15.00		315.00			$P = 6.3 \times 10^5$ kW
Ion Thrusters	2.66	.42	614.21			N = 5400, D = 84 cm
Power Processing Units	18.93	17.05	678.37			N = 77, P = 8204 kW
Program Level Costs	39.56		167.63	936.19		
Total - All COTV's	690.23			12,454.63		

Table G-16. COTV Operations Cost (millions of 1977 dollars).

LRU Option	Vehicle	Propellant Weight (lbs) LO_2	Propellant Weight (lbs) LH_2	Propellant Cost per Flight LO_2	Propellant Cost per Flight LH_2	Propellant Cost per Flight Total	Cost per year for each vehicle Spares	Cost per year for each vehicle Maintenance
	$COTV_2$	418,696	9,155	See Note (2)	.005	.005	1.373	4.373
B	$COTV_3$	11,193,463	244,756		.132	.132	10.700	32.100
	$COTV_4$	6,503,099	142,196		.077	.077	20.016	60.048
	$COTV_1$	3,552,865	77,687		.042	.042	4.977	14.931
C	$COTV_2$	11,459,685	250,577		.135	.135	19.019	57.057
	$COTV_3$	7,175,917	156,908		.085	.085	18.439	55.317
	$COTV_1$	3,552,865	77,687		.042	.042	4.977	14.931
D	$COTV_2$	10,595,670	231,684		.125	.125	18.916	56.748
	$COTV_3$	7,175,917	156,908		.085	.085	18.439	55.317

Notes:

1. Propellant weights based on Table 4.6-6 total propellant. LO_2 is 98% and LH_2 2% of the total. Losses were assumed to be 20% for LH_2 and 12% for LO_2 and are included in the total weight. Propellant costs are: LH_2 - \$.54/lb (Ref. JSC SPS Concept Evaluation, JSC-12973, July 1977).
2. LO_2 costs are reflected in lunar based propellant production facilities costs.

G.3.10 Lunar Transfer Vehicle

The LTV configuration is described in Figure 4-46. It consists basically of a landing structure supporting a LH_2/LO_2 tank and two side mounted cargo pods. A rough order magnitude estimate of this vehicle was made using cost estimating relationships. A weight statement is shown in Table G-17. Cost estimating relationships used are shown in Table G-18. Development and First Unit Costs are shown in Table G-19.

Table G-17. LTV Weight Breakdown.

Element	Weight (kg)
LO_2/LH_2 Tankage	7255
Cargo Pods	3630
Landing Structure	10000
Engines 4 @ 1070 kg	4280
Subsystems	4835

For production a 90 percent learning curve is assumed for hardware. Program level costs, which include initial spares, sustaining tooling and engineering and program management are assumed to be 20 percent of total hardware costs. Total production can be expressed as follows:

$$\text{Production Cost} = 1.2\ (28.22)\ N^{.848}$$

$$= 33.864\ N^{.848}\ \text{(millions of 77 \$)}$$

where N = Number of vehicles produced

The operations costs of the LTV consists of propellants, spares and maintenance. Spares and maintenance costs are assumed to be 1 percent/year and 3 percent/year, respectively, of first unit cost for each LTV.

Table G-18. LTV Cost Estimating Relationships.
(millions of 1977 dollars)

Cost Element	Development	First Unit Cost	Remarks
Structure/Subsystems	$10.14\ W^{.187}$	$.007\ W^{.667}$	Note 1, 2
Engines (LO_2/LH_2)	$3.39\ T^{.38}$	$NF\left[.308 + 10.857 \times 10^{-6}\ T^{.904}\right]$	Note 3

Notes:

(1) Subsystems includes hydraulics, pneumatics, propellant feed and electrical whose characteristics are not defined. Assume CER's for Structure will apply to these subsystems taken as a whole.

(2) W = Weight in pounds of element being considered

(3) T = Vacuum Thrust per engine (lbs)
F = Propulsion complexity factor = 3.15
N = Number of engines per vehicle

(4) Allow the following for Program Level Costs:
Development - 40% of Hardware Development Cost
First Unit - 10% of Hardware First Unit Cost
Production - 20% of Hardware Production Cost

Ref: (1) Parametric LCC Analysis Technique for Space Systems, 1978 IRAD Study by General Dynamics, Convair Division, Report Pending.

Table G-19. LTV Cost Summary.
(millions of 1977 dollars)

Cost Element	Development	Vehicle First Unit	CER Variables
Structure:			
Tankage	61.97	4.46	W = 16,000 lbs
Cargo Pods/Landing	69.73	6.79	W = 30,050 lbs
Engines (LO_2/LH_2)	325.72	11.00	T = 165,000 lbs, N = 4, F = 3.15
Subsystems	57.44	3.40	W = 10,660 lbs
Program Level Costs	205.94	2.57	
Total	720.80	28.22	

Spares: .01 (28.22) = \$282 million/year/vehicle

Maintenance: .03 (28.22) = \$.847 million/year/vehicle

Total propellant weight, per Figure 4-46 , is 242.3 metric tons per round trip flight from the lunar surface to LLO. The LO_2 is manufactured from lunar soil and LH_2 is supplied from the earth. The cost of LO_2 will be reflected in the LRU facilities development, production and operations costs. The cost of LH_2 is based on future earth rates. Table G-20 shows the propellant breakdown and cost per flight for propellants.

Table G-20. LTV Propellant Costs.

	Propellant Wt (lbs)	Losses (lb)	Total/flight (lbs)	Cost/flight, (millions, 1977 \$)
LO_2	467488	56099	523587	—
LH_2	66784	13357	80141	.043

Notes:

(1) Mixture Ratio 7:1

(2) Propellant losses were assumed to be 20% for LH_2 and 12% for LO_2.

(3) Propellant Costs: LH_2: \$.54/lb. Based on Allgeier and McBryar Propellant Study, SPS Concept Evaluation, JSC-12973, July 1977.

G. 3. 11 PLTV

The Personnel Lunar Transfer Vehicle (PLTV) configuration is shown in Figure 4-51. With the exception of the cargo pods, the design is similar to the LTV discussed in Section G. 3. 10 . The cost estimating relationships for the LTV, shown in Table G-18, are also applicable to the PLTV and will be used for the cost estimate. A weight breakdown is shown in Table G-21. Four 13, 825 lb thrust engines are assumed. Development and First Unit Costs are presented in Figure G-22.

Table G-21. PLTV Weight Breakdown.

	Weight	
	Kg	lbs
Engines (4)	828	1826
LH_2 Tank	877	1934
LO_2 Tank	526	1160
Other Structures	1934	4264
Subsystems	935	2062

Table G-22. PLTV Costs (Millions of 1977 Dollars).

Cost Element	Development	Vehicle First Unit	CER Variables
Structure			
Tankage	45.58	1.49	W = 3094 lbs
Other	48.40	1.85	W = 4264 lbs
Engines (LO_2/LH_2)	126.96	4.64	T = 13, 825 lbs, N = 4, F = 315
Subsystems	42.25	1.14	W = 2062 lbs
Subtotal	263.19	9.12	
Program Level Costs	105.28	.91	
Total	368.47	10.03	

Notes: (1) Program Level Costs are 40% of hardware development and 10% of hardware first unit.

For production of multiple units a 90 percent learning curve will be assumed. An allowance of 20 percent of hardware costs will be made to cover program level costs. Production cost can be expressed as follows:

$$
\begin{aligned}
C &= 1.2\,(10.03)\,N^{.848} \\
&= 12.04\,N^{.848} \text{ (millions of 1977 dollars)}
\end{aligned}
$$

Option B, the only LRU option for which the PLTV is used, requires only 1 vehicle. Assume that one backup is required and that a total of 2 will be produced for initial production. No replacements will be required over the 30 year program life due to the low usage rate of the vehicle. Production cost for the two units is:

$$
\begin{aligned}
C &= 12.04(2)^{.848} \\
&= \$21.67 \text{ (millions of 1977 dollars)}
\end{aligned}
$$

Each PLTV requires a passenger module and crew module. Passenger modules are costed with the POTV. They are merely transferred from one vehicle to the next with the personnel onboard. It is assumed that the crew modules will be dedicated to the PLTV and two will be required. Cost of the crew module will be for production only. Development will be included with the POTV costs. From Table G-8 Crew Module First Unit Cost is \$24 million. Assuming 90 percent learning and 20 percent for program level cost, the cost of the two units is:

$$
\begin{aligned}
C &= 1.2\,(24)\,(2)^{.848} \\
&= \$51.84 \text{ (millions of 1977 dollars)}
\end{aligned}
$$

Total production cost is: \$21.67 + 51.84 = \$73.51 million.

Operations costs for the PLTV consist of propellants, spares and maintenance. Using the same relations as for the LTV the following costs are obtained:

Spares: .01(34.03) = \$ .34 million/year/vehicle

Maintenance: .03(34.03) = \$1.02 million/year/vehicle

Round trip flight propellant requirements for the PLTV are 41.1 metric tons. The LO_2 is manufactured from lunar soil at the SMF and the associated costs are reflected in facilities development, production and operations costs. The LH_2 is brought up from earth and future earth rates will apply to its costs. Using the Allgeier and McBryan Propellant Study, LH_2 cost is \$.54 per lb. Total LH_2 required per round trip is: 1/8 (41.1) (2205) = 11329 lbs. Cost per flight for propellants is: .54 × 11329 = \$6118/flight.

G.3.12 Lunar Derived Rocket (LDR)

The LDR configuration is shown in Figure 4-47. It is similar in design to the Lunar Transfer Vehicle. Instead of a hydrogen tank it contains two aluminum powder tanks. The LH_2/LO_2 engines are replaced with A/LO_2 engines. Dry weight of the LDR is 180 metric tons. This compares with 30 metric tons for the LTV.

LDR costs will be determined by scaling up the LTV vehicle, excluding the engines. Engine costs will be estimated separately because of their uniqueness. LDR mass, excluding engines, is 67.5 metric tons; LTV mass is 25.7 metric tons. Using the LTV costs in Table G-19 the following LDR costs (excluding engines) are obtained:

$$\text{Development:} \quad C = 720.80 \left[\frac{67.5}{25.7}\right]^{.5}$$

$$= \$1168.154 \text{ (millions of 1977 dollars)}$$

First Unit: $C = 28.22 \left[\frac{67.5}{25.7}\right]^{.67}$

$= \$53.893$ (millions of 1977 dollars)

Costs for the Aluminum/Oxygen engines will be estimated using the CER's in Table G-18. Complexity factors of 10 and 4 will be used for development and first unit costs respectively. Thrust level for each engine is 1290 KN, or 290,000 lbs and 4 are required.

Development: $C = 10(3.39)\,(290,000)^{.38}$

$= \$4035.626$ (millions of 1977 dollars)

First Unit: $C = 4(4)\left[.308 + 10.857 \times 10^{-6}(290,000)^{.904}\right]$

$= \$19.988$ (millions of 1977 dollars)

Total costs for the LDR are the sum of the engines and the values scaled from the LTV:

Vehicle Development:	\$1168.154 + 4035.626 = \$5203.78
Vehicle First Unit:	\$ 53.893 + 19.988 = \$ 73.881

Using a 90 percent learning curve for production and allowing 20 percent of hardware costs for program level costs, vehicle production cost can be expressed as follows:

$$C = 1.2\,(73.881)\,N^{.848}$$

$$= 88.657\,N^{.848} \text{ (millions of 1977 dollars)}$$

Like the PLTV in the previous section, each LDR requires a dedicated crew module. From Section G.3.11, first unit cost is \$24 million and production cost can be expressed as follows:

$$C = 1.2(24)\, N^{.848}$$
$$= 28.8\, N^{.848} \text{ (millions of 1977 dollars)}$$

No development cost will be charged to the PLTV. This will be allocated entirely to the POTV.

Operations costs consist of propellants, spares and maintenance. Since the LO_2 and aluminum are manufactured from lunar soil, there costs are included in the development, production and operation of the LRU facilities and will not be included as part of the LDR operations. Spares and maintenance are estimated to be:

Spares:	.01 [73.881 + 24]	= \$ .979 million/yr/vehicle
Maintenance:	.03 [73.881 + 24]	= \$2.936 million/yr/vehicle

G.3.13 Mass Catcher

The Mass Catcher is unique to LRU Option B. It is a combination of the catcher described in Figure 4-49 and the Terminal Tug in Figure 4-50. The combined concept was also discussed on page 4-141 of Volume II.

Table G-23 contains an estimated weight breakdown of the catcher assembly and includes a 5 percent contingency. Cost estimates will be made using the weights and the cost estimating relationships in Table G-24. Table G-25provides the results of the calculations for development and first unit costs.

Table G-23. Mass Catcher Weight Breakdown.

Element	Weight: Metric Tons	Weight: Pounds
Engines (8 @ 4.8 KN Thrust each)	4	8,820
Major Structural Ring (Despun)	412	908,460
Bag Rupture Screen	240	529,200
Bag Spin Bearing	206	454,230
Catcher Bag		
Steel Cable	1,200	2,646,000
Kapton	390	859,950
Propellant Tankage	360	793,800
Propellant Tank Shielding	50	110,250
Avionics	2	4,410
Contingency	136	299,880
	3,000	6,615,000

The following assumptions were made for the cost estimates in Table G-25:

1. There are 8 propellant tanks, each weighing 793,800/8 = 99,225 lbs.
2. Tank Shields are 110,250/8 = 13,781 lbs each.
3. Structural Ring is divided into 8 identical segments, each weighing 908,460/8 = 113,558 lbs.
4. Catcher Bag Spin Bearing is divided into 16 identical segments, each weighing 454,230/16 = 28,390 lbs.
5. Assume Fluid Systems Weight is 40 tons, or 88,200 lbs.
6. Program Level Costs are 40% for Development and 10% for First Unit Cost.

For production a 90% learning curve will be assumed and 20% of the hardware cost will be allowed for program level costs. Using the vehicle first unit cost in Table G-25, production cost can be expressed as follows:

$$C = 1.2\,(578.711)\,N^{.848}$$

$$= 694.453\,N^{.848} \text{ (millions of 1977 dollars)}$$

where N = Number of vehicles produced

Table G-24. Mass Catcher Cost Estimating Relationships.
(Millions of 1977 Dollars)

Element	Cost Estimating Relationships: Development	Cost Estimating Relationships: First Unit	Remarks
Engines (LO_2/LH_2)	$3.39\ T^{.38}$	$3.15\left[.308 + 10.857 \times 10^{-6} T^{.904}\right]$	T = Vacuum Thrust (lbs)
Structure			
Tankage	$10.14\ W^{.187}$	$.008\ W^{.667}$	W = Subsystem Weight (lbs)
Tank Shielding	$10.14\ W^{.187}$	$.005\ W^{.667}$	
Ring	$10.14\ W^{.187}$	$.005\ W^{.667}$	
Rupture Screen	$.55\ W^{.187}$	$.004\ W^{.667}$	
Catcher Bag Spin Bearing	$10.14\ W^{.187}$	$.018\ W^{.667}$	
Fluid Systems	$3.04\ W^{.30}$	$.096\ W^{.43}$	
Catcher Bag	$.55\ W^{.187}$	$.004\ W^{.667}$	
Avionics	$.231\ W^{.5}$	$.021\ W^{.667}$	

References: (1) Parametric LCC Analysis Techniques for Space Systems, 1978 IRAD Study by GDC, Report Pending
(2) Shuttle System Payload Data Activity, GDC Report PSD-CO-015, Sept. 1974.

Table G-25. Mass Catcher Development and First Unit Costs.
(Millions of 1977 Dollars)

Element	Development	Element First Unit	Vehicle First Unit	CER Variable
Engines	48.184	.989	5.768	T = 1080 N = 8
Structure:				
Tankage	87.178	17.212	100.381	W = 99,225 lbs N = 8
Tank Shielding	60.268	2.883	16.814	W = 13,781 lbs N = 8
Ring	89.405	11.771	68.649	W = 113,558 lbs N = 8
Rupture Screen	6.467	26.285	26.285	W = 529,200 N = 1
Spin Bearing	68.989	16.809	176.456	W = 28,390 N = 16
Fluid Systems	92.579	12.848	12.848	W = 88,200 N = 1
Catcher Bag				
Steel Cable	8.737	76.900	76.900	W = 2,646,000, N = 1
Kapton	7.081	36.337	36.337	W = 859,950, N = 1
Avionics	15.340	5.663	5.663	W = 4410, N = 1
Subtotal	484.228	207.697	526.101	
Program Level Costs	193.691		52.610	
Total	677.919		578.711	

Notes: (1) Vehicle First Unit Cost is the total cost of the elements in each subsystem assuming a 90% learning curve: C = (Element TFU) $N^{.848}$, where N = Number of elements in the subsystem.

Operations costs consist maintenance, spares and propellants. Crew labor for operating the vehicle is costed under the "construction/maintenance crew" categories. Annual costs are as follows:

Spares:	1% (578.711)	= \$ 5.787 million/year/vehicle
Maintenance:	3% (578.711)	= \$17.361 million/year/vehicle

Each catcher uses 5585 metric tons of LO_2 and 800T LH_2 per round trip. The LO_2 is manufactured from lunar soil and costs for it are reflected in the LO_2 manufacturing facilities. The LH_2 is earth supplied and costs \$.54 per pound. Cost per flight is: $\$.54 \times 800 \times 2005 = \$.953$ million/flight.

G.3.14 Mass Driver Catapult

The mass driver catapult configuration is shown in Figure 4-48. Costs of the unit will be determined from cost estimating relationships. Power to the unit will be supplied by the lunar based nuclear power station and its cost will not be included.

A weight breakdown is shown in Table G-26. These weights will be the basis for the cost estimates which follow. Due to the complexity of Mass Driver Catapult and the lack of detail definition of the configuration (e.g., lack of subelement quantities and types, lengths) confidence in the cost estimate will be low. The Cost Estimating Relationships (CER's) are similar to those used previously and are shown in Table G-27. The development and production costs are presented in Table G-28. Since only one unit is required, first unit and production costs are the same.

Operations costs consist of maintenance and spares. These are estimated to be 4 percent of the hardware cost per year or:

$$C = .04\ (269.105)$$
$$= \$10.764 \text{ million/year}$$

Table G-26. Mass Driver Catapult - Weight Breakdown.

	Weight	
	Metric Tons	Pounds
Electronics		
Windings	60.4	133,182
Feeders	10.0	22,050
Capacitors	10.0	22,050
SCR	3.2	7,056
Structures		
Radiators	32.0	70,560
Launcher Tube	58.2	128,331
Tunnel	20.0	44,100
Misc.	10.0	22,050
Support Facilities		
Trim Stations	60.0	132,300
Loading Facilities	20.0	44,100
Stockpile Bins	20.0	44,100
Packaging Units	35.0	77,175
Soil Binders	50.0	110,250

Table G-27. Cost Estimating Relationships - Mass Driver Catapult.

Element	CER (millions of 1977 dollars)		Remarks
	Development	First Unit	
Electronics	$.231\ W^{.5}$	$.021\ W^{.667}$	W = Subsystem Weight (lbs)
Structures	$10.14\ W^{.187}$	$.013\ W^{.667}$	
Support Facilities	$10.14\ W^{.187}$	$.013\ W^{.667}$	
Soil Binder	10	10 W	Note (1)

Notes: (1) Soil Binder costs are assumed.

Table G-28. Mass Driver Catapult Costs.
(millions of 1977 dollars)

Element	Development	Production	CER Variable
Electronics			
Windings	84.301	54.982	W = 133,182 lbs
Feeders	34.302	16.568	W = 22,050 lbs
Capacitors	34.302	16.568	W = 22,050 lbs
SCR	19.404	7.748	W = 7,056 lbs
Structures			
Radiators	81.793	22.281	W = 70,560 lbs
Launcher Tube	91.474	33.205	W = 128,331 lbs
Tunnel	74.911	16.285	W = 44,100 lbs
Misc.	65.804	10.256	W = 22,050 lbs
Support Facilities			
Trim Stations	91.996	33,886	W = 132,300 lbs
Loading Facilities	74.911	16.285	W = 44,100 lbs
Stockpile Bins	74.911	16.285	W = 44,100 lbs
Packaging Units	83.176	23.653	W = 77,175 lbs
Soil Binder	10.000	1.103	W = 110,250 lbs
Subtotal	821.285	269.105	
Program Level Costs	328.514	80.732	
Total	1149.799	349.747	

Notes: (1) Program Level Costs are assumed to be 40% of hardware cost for Development and 30% for first unit.

G.4 EARTH BASED FACILITIES

This category includes the design and construction of earth facilities required for the SPS program. Two such facilities were identified: (1) propellant production facilities and (2) SDV launch/recovery facilities.

G.4.1 Propellant Production Facilities

Propellant production requirements for the LRU options are not nearly as large as the Earth Baseline requirements. This is due primarily to the use of lunar resources in manufacturing oxygen and to the decreased usage of earth based launch vehicles.

The Earth Baseline propellant requirements totaled 3.865×10^6 metric tons per SPS (Ref. Table G-50 in Section G.7) or 10589 metric tons per day. Facility costs were \$3.5 billion (Ref. Figure F-1 in Appendix F). This size plant and cost is supported by the propellant plant CER on page X-D-154 of Solar Power Satellite Concept Evaluation, Activities Report July 1976 to June 1977, JSC-12973. The relationship is: $C = 11.694T^{.6}$, where T is plant capacity in tons per day. A factor of 20% was applied for Program Management and Integration. This yields: $1.2\ (11.694)\ T^{.6} = 14.033\ T^{.6}$.

Table G-29 shows the propellant facilities requirements for the LRU options and the resulting facilities costs using the above equation. Propellant requirements are the total propellants required to launch all ground based vehicles plus the propellants carried from earth for space use. The recurring costs of producing propellant from these facilities is included in the operations cost of each launch or space vehicle.

G.4.2 Launch/Recovery Facilities

Launch/Recovery facilities for the LRU options are required for the Shuttle Derived Vehicle. Costs for these facilities were scaled from the \$2.8 billion Launch/Recovery facility cost of the Earth Baseline (Ref. Figure F-1). It was assumed that gross vehicle liftoff weight (GLOW) in tons/year varies exponentially with launch/recovery facilities cost.

Earth Baseline - 391 HLLV flts/yr @ 11,041 tons = 4.317×10^6 tons/yr.

Scaling Relationship: $C = \$2800 \text{ million} \left[\frac{\text{GLOW/year}}{4.317 \times 10^6}\right]^{.67}$

Costs for the facilities in each LRU option, based on the above scaling relation, are shown in Table G-30.

Facility operations costs consist of launch/recovery operations and maintenance costs. Launch/recovery operations costs are included in the operations costs of the SDV. Facility maintenance costs are assumed to be 5 percent of facility costs per year and are shown in Table G-30.

Table G-29. Propellant Production Facilities

Propellant Use	Option B	Option C	Option D
LH_2 carried to space	1,279	10,527	886
SDV Propellant	233,555	407,785	260,609
Space Shuttle Propellant	68,768	88,837	88,837
Total Propellant (tons/SPS)	302,602	507,149	350,332
Capacity Requirement (tons/day)	1,000	1,400	1,000
Facilities Cost (millions of 1977 dollars)	885.422	1083.496	885.422

Notes:

(1) Earth supplied propellant requirements for sizing the propellant production facilities were obtained from Figures 4-4, 4-6 and 4-7. As an example, the SDV propellant in Option C, from Fig 4-6, is: 41.45 × 9838 tons/SPS = 407785 tons/SPS.

(2) Facility capacity determined by dividing total propellant required per SPS by 365 days and rounding up to the nearest thousand.

(3) Facilities CER: $14.033T^{.6}$, where T = tons/day capacity.

Table G-30. Launch/Recovery Facility Costs.
(Millions of 1977 dollars)

LRU Option	GLOW (tons/year)	Facilities Costs	Annual Maintenance
B	68 SDV flts/yr × 4196 tons = $.285 \times 10^6$	453.244	22.662
C	120 SDV flts/yr × 4196 tons = $.504 \times 10^6$	664.071	33.204
D	76 SDV flts/yr × 4196 tons = $.319 \times 10^6$	488.794	24.440

G.5 LRU MANUFACTURING FACILITIES AND EQUIPMENT

The elements in this section are the facilities and equipment required to remove the lunar material and convert it into usable products. They include mining and beneficiation equipment, processing facilities, manufacturing equipment and LO_2 liquefaction equipment.

G.5.1 Lunar Mining Equipment

Mining equipment is described in Figure 4-10. Lunar loaders and haulers will be similar to present day earth equipment with modified power plants. Costs of equipment today are:

12.5 Ton Loader	$.105 million (Caterpillar Model 966C)
50 Ton Hauler	$.412 million (Caterpillar Model 777)

The type of power plant has not yet been defined. It could be powered by fuel cells or batteries combined with an electric motor. The cost to develop and install these power systems would far outweigh the above prices for mass produced equipment. Assume the cost to develop and produce each piece of equipment is as follows:

12.5 Ton Loader	$15 million each
50 Ton Hauler	$10 million each

Total cost for two loaders and two haulers is $50 million.

Operations cost of the equipment consists of spares, maintenance and labor for operating the equipment. Spares and maintenance are assumed to be 1% and 3% of total hardware cost per year. Maintenance costs represent an allowance for earth based support of maintenance operations. The actual maintenance labor, as well as operating labor, is covered as a single item, "Construction/Maintenance Crew."

Total Operating Cost = 4% (50) x 30 years = $60 million

G. 5. 2 Lunar Material Beneficiation Equipment

The beneficiation equipment concept is shown in Figure 4-10. The configuration is not well enough defined to use cost estimating relationships on a subsystem basis. It will be assumed that a structural type cost estimating relationship for a truss type structure will apply to the entire system. CER's are from: Parametric LCC Analysis Technique for Space Systems, 1978 IRAD Study by GDC, Report pending.

Development Cost = $1.104\ W^{.187}$ (millions of 1977 dollars)

First Unit Cost = $.005\ W^{.667}$ (millions of 1977 dollars)

An allowance of 40% for development and 30% for production will be made for program level costs. This includes system test, tooling, program management, sustaining engineering and assembly/checkout. Applying these factors to the above equations the following CER's are obtained:

Development Cost = $1.546\ W^{.187}$ (millions of 1977 dollars)

Production Cost = $.007\ W^{.667}$ (millions of 1977 dollars)

Operations costs consist of spares, maintenance and labor for operating the equipment. Labor costs are included under a single category: "Construction/Maintenance Crew" and are not included here. Annual costs are as follows:

Spares: 1% (Production Cost)

Maintenance: 3% (Production Cost)

Maintenance costs represent the cost of earth based support for repair and maintenance operations. The actual maintenance operations are carried out by the resident crews.

G. 5. 3 Processing Facility

The processing facility has not been defined in sufficient detail to determine costs with a high level of confidence. A rough order of magnitude estimate will be made however, and updated as the configuration is further defined. Table C-1, on page C-7 of Appendix C, provides processing equipment weight estimates for three different approaches.

For the present, assume the acid leach process will be used.

Facility equipment masses vary among the different LRU options. Other than the radiator, no breakdown of subsystem weights has been defined. An assumed breakdown for costing purposes is shown in Table G-31. Cost estimating relationships are provided in Table G-32. Costs for LRU Option C are shown in Table G-33. Processing facility costs for the other options can be scaled by weight as shown below.

Development: $C = 1371.957 \left(\frac{W}{10405}\right)^{.5}$

$= 13.450\ W^{.5}$

Production: $C = 2829.410 \left(\frac{W}{10405}\right)^{.67}$

$= 5.756\ W^{.67}$

Operations: 4%/year (Production Cost)

where: W = Total Processing Facility Weight of Options B or D (metric tons)

Table G-31. Processing Facility Weights (metric tons)

Element	Option B	Option C	Option D
Radiators	6500	7500	11,500
Structural Enclosure (25%)	444	726	1,370
Processing Machinery (60%) Silica Glass, Silicon Aluminum, Oxygen Iron	1065	1743	3,288
Fluid Systems (14%)	248	407	767
Electronics (1%)	18	29	55
Total	8275	10405	16,980

Notes: (1) Percentages represent assumed breakdown of elements out of the total facility equipment mass.

(2) Various processing machinery elements are assumed to be of equal weight.

Table G–32. Processing Facility Cost Estimating Relationships

Element	CER's (millions of 77 $) Development	First Unit	Reference
Structures/Radiators	$4.614\ W^{.187}$	$.013\ W^{.667}$	(1)
Processing Machinery	$10.14\ W^{.187}$	$.007\ W^{.667}$	(1)
Fluid Systems	$3.04\ W^{.30}$	$.096\ W^{.43}$	(1)
Electronics	$.231\ W^{.5}$	$.021\ W^{.667}$	(2)

Notes: (1) W = Weight in lbs.

References: (1) <u>Parametric LCC Analysis Technique for Space Systems</u>, 1978 IRAD Study by GDC, Report Pending.

(2) <u>Shuttle System Payload Data Activity</u>, GDC Report PSD-CO-015, Sept. 1974.

Table G-33 Lunar Processing Facility Costs - Option C
(millions of 1977 dollars)

Element	Total Cost Development	Total Cost Production	Annual Cost Operations
Radiators	29.943	1634.157	
Structural Enclosure	66.725	178.748	
Processing Machinery			
Silica Glass	127.841	59.004	
Aluminum	↕	↕	
Silicon			
Oxygen			
Iron	127.841	59.004	
Fluid Systems	185.684	34.839	
Electronics	58.414	33.705	
Subtotal	979.971	2176.469	
Program Level Costs	391.988	652.941	
Spares			21.765
Maintenance			65.294
Total	1371.959	2829.410	87.059

Notes:
(1) Program Level Costs: 40% of Hardware Development Cost and 30% of Production Cost.
(2) 750, 10 ton radiator units required. 85% learning assumed.
(3) Annual Operations Costs: Spares 1% and Maintenance 3% of Hardware Production cost.

G.5.4 Liquefaction Equipment

A. Lunar Surface Facility

The LO_2 lunar surface liquefaction facility is defined in Figure 4-18. Costs will be determined for this facility using cost estimating relationships. Costs for other sizes of facilities then can be scaled from this base cost. Costs for the storage tanks are not included in this section. They are covered with propellant depots in Section G.1.

The following is a weight breakdown of the facility shown in Figure 4-18:

Element	Weight tons	Weight lbs.
Structural Enclosure	66.6	146,853
Radiator	815.3	1,797,737
Heat Exchangers/Pumps	5.9	13,010
Liquefaction Equipment	185.5	409,028
Avionics, Controls	6.7	14,774
Total	1080.0	2,381,402

Cost estimating relationships for the above elements are shown in Table G-34. Development and production costs are shown in Table G-35. From these costs, and the above weight, the following scaling relationships can be derived for other sizes of facilities:

$$\text{Development Cost} = 382.151 \left(\frac{W}{1080}\right)^{.5}$$

$$= 11.628\ W^{.5} \quad \text{(millions 1977 dollars)}$$

$$\text{Production Cost} = 515.520 \left(\frac{W}{1080}\right)^{.67}$$

$$= 4.785\ W^{.67} \quad \text{(millions 1977 dollars)}$$

Operations Cost = 4%/year (Production Cost)

Where: W = Liquefaction equipment weight (tons) for Options B, C or D.

B. SMF Oxygen Liquefaction Facility

The SMF oxygen liquefaction facility is shown in Figure 4-19. It is similar to the lunar surface liquefaction facility except it has its own solar array power supply. Cost for the facility, excluding power supply, can be estimated using the scaling relationships from Step A. Costs of the solar array power supply and associated systems can be estimated using the scaling relationships for Photovoltaic Power Stations in Section G.6.1. The combined costs are shown below:

SMF Oxygen Liquefaction Facility Costs (Concept B) -

Development: $C = 11.628\ W^{.5} + 24.04\ P^{.5}$

Production: $C = 4.785\ W^{.67} + 22.54\ P^{.67}$

where: W = Liquefaction Facility weight, in metric tons, excluding power source

P = Power output (megawatts)

Operations costs are estimated at 4 percent of production cost per year.

Table G-34. Liquefaction Equipment Cost Estimating Relationships

Subsystem Type	CER in millions of 1977 dollars Development	First Unit
Structures	$4.614\ W^{.187}$	$.013\ W^{.667}$
Fluid Systems	$3.04\ W^{.30}$	$.096\ W^{.43}$
Avionics	$.231\ W^{.50}$	$.021\ W^{.667}$

Sources: (1) Same as shown in Table G-32.

Table G-35. Base Cost - Liquefaction Equipment

Element	Development	First Unit	Quantity	Production
Structural Enclosure	17.238	1.431	100	48.712
Radiator	28.821	8.950	100	304.665
Heat Exchanger & Pumps	52.138	5.642	1	5.642
Liquefaction Equipment	146.690	24.850	1	24.850
Avionics	28.078	12.685	1	12.685
Hardware Total	272.965			396.554
Program Level	109.186			118.966
Total	382.151			515.520

Notes:

(1) Structural Enclosure is equipment tunnel. Assume there are 100, 30 meter long sections, each weighing 1151 lbs.

(2) For costing purposes assume there are 100 radiator elements, each 30 meters in length. Element weight - 17977 lbs.

(3) Program Level Costs: 40% for Development; 30% for Production

(4) An 85 percent learning curve was assumed for Structures production.

G.5.5 Manufacturing Facilities

A manufacturing flow diagram is shown in Figure D-1. The individual components of the manufacturing process are identified in Tables D-2, D-3, D-4 and D-5 of Appendix D. Facilities were divided into four major categories: (1) Stock Manufacturing, (2) Parts Manufacturing, (3) Component Assembly, and (4) Solar Cell Panel Facilities. Depending on the LRU option, some of the facilities may be placed on the moon and some in space. Individual facilities may also be split between the moon and space.

The approach taken here was to use cost comparables to establish equipment costs. In this method costs are estimated using the same or similar products. The comparables method was pursued primarily because of data availability, that is, data is readily available on current or proposed products from commercial sources. It should be noted that certain items might not be usable in an off the shelf condition, but any attempt to derive a modification factor would be specious.

Costs were categorized to correspond to the four major categories mentioned above. The derivation of those costs is shown in sections A through D below. For each element within each category the product to be manufactured was identified and given the same item number as the manufacturing process tables on pages D-29 through D-32 of Appendix D. The equipment necessary for the particular operation and the parameters necessary for identifying cost comparables were then identified. Next, the cost comparables themselves, including source description, cost source and any applicable analyses were presented. In some cases equipment will not be costed on an item by item basis but rather, as part of a process. Finally costs were summarized in tabular form for each category. Several of the categories share the use of manufacturing equipment. Rather than allocate value by the percentage of use of a particular item, total costs will be assigned to the first item using that equipment, as presented by the study.

In Section E the allocation of equipment between space and the moon is made. Costs are then allocated accordingly and adjustments are made for system level costs and design changes to give total manufacturing facility costs for each option.

A. STOCK MANUFACTURING FACILITIES

Stock manufacturing facilities consist of Items (1) through (7). Costs are derived below and are summarized in Table G-36.

Item (1) - Aluminum Sheet

Equipment Required:

1. 7/1200 KW, 50 KV electron beam guns and power supplies, including magnetic lens and beam deflection accessories.
2. 3 Industrial Robots

Cost Comparables:

1. Airco Temescal Model EH 1200/50 electron beam gun @ $2000 - 3000 per KW, including power supply. The high end of the price range will be used in order to include the magnetic lens and beam deflection accessories.
2. Unimation Model 2005C Industrial Robot - $60,000 each.

Equipment Cost:

1200 KW @ $3000 = $3.6 million for one unit.
Assume 95% learning for 7 units.
Cost = $3.6\ (7)^{.926}$ = $21.82 million

Robots 3 @ .06 = $.18 million

Total Cost = $22.00 million

Item (2) - Aluminum Wire - Conductors and Coils

Equipment Required:

1. 1 roll slitter and strip coiler
2. 1 Electron Beam Welder
3. 8 Wire drawing machines utilizing 1/4" aluminum strip to produce 1.13 mm wire at 2124 M/minute
4. 2 Industrial Robots

Cost Comparables:

1. Niagra 1R4 Shear \$.025 million
2. Sciaky model VX .3 Electron beam welder - \$643,500
3. Roth R2R3 Wire Drawing Machine
   First Unit = .275 million
4. Unimation Model 2005C Industrial Robot - \$60,000 each

Equipment Cost:

Roll Slitter - \$.025
Electron Beam Welder - \$.644
Wire Drawing Machines - 8 units @ 95% learning - .275 $(8)^{.926}$ = \$1.886
Industrial Robots - 2 @ .06 = \$.120

Total Cost = \$2.675 million

Item (3) - Steel Sheet for Heat Pipe Tubing

Equipment Required:

1. 8 - 1200KW, 50 KV electron beam guns and power supplies including magnetic lens and beam deflection accessories
2. 3 Industrial Robots

Cost Comparables:

1. Airco Tenescal Model EH 1200/50 electron beam gun @ \$3000 per KW including power supply. Price includes magnetic lens and beam deflection access. Each item: \$3000 x 1200 KW = \$3.6 million
2. Unimation Model 2005C Industrial Robot - \$60,000 each.
   2 @ .06 = \$.120 million

Equipment Cost:

Electron Beam Guns - 8 units @ 95% learning 3.6 $(8)^{.926}$ = \$24.692 million
Industrial Robots - 3 @ .06 = \$.180 million

Total Cost = \$24.872 million

Item (4) - Iron Sheet - Poles for Klystron Solenoid

Equipment Required:

1. 3 - 400 KW electron beam guns w/associated power supplies
2. 1 Blanking Press & Dies
3. 2 Industrial Robots

Cost Comparables:

1. Airco Tenescal quote of \$2000 - 3000 per KW including power supply. \$3000 chosen. \$1.2 million per item.
2. Niagra PN-6048 (60" x 48") - \$.06 million.
3. Unimation Model 2005C Industrial Robot - \$60,000 each

Equipment Cost:

Electron Beam Guns - 3 @ 1.2 = \$3.60
Blanking Press - \$.06
Industrial Robots - 2 @ .06 = \$.12

Total Cost = \$3.78 million

Item (5) - Aluminum Castings - Klystron Solenoid Cavity & Strut Assembly Nodes

Equipment Required:

1. 1 50 KW induction furnace with power supply & controller - .136 ton/hr. capacity
2. 1 Automatic Permanent Mold Casting Machine 8-10 stations with 100 castings/hr. capacity
3. 4 sets of permanent mold & accessories
4. 6 Industrial Robots

Cost Comparables:

1. .136 ton/hr. induction furnace per GDC Facilities personnel (Bill Ladd) - \$250,000
2. 8-10 Station Automatic Mold Casting Machine, American Die Casting Institute (John Nelson) - \$.255 million
3. Permanent molds & accessories, GDC Facilities personnel estimate \$.200 million
4. Unimation Model 2005C Industrial Robot - \$60,000 each

Equipment Cost: Furnace - \$.250
Casting Machine - \$.255
Permanent Molds - \$.200
Robots - 6 @ .06 = \$.360
Total Cost = \$1.065 million

Item (6) - Sendust Casting - Transformer Core

Equipment Required:

1. 600 KW high frequency induction melting furnace - .127 tons/hr. capacity
2. Sand mixing and molding equipment
3. 1 Industrial Robot

Cost Comparables:

1. .136 ton/hr. induction furnace in item (5) has essentially same capacity - $.25 million
2. Sand mixing and molding equipment $.03 million - analyst judgement
3. Unimation Model 2005C Industrial Robot - $60,000 each

Equipment Cost:

Furnace - $.25
Sand Mixing/Molding Equipment - $.03
Industrial Robot - $.06

Total Cost = $.34 million

Item (7) - Foamed Glass Components - MPTS Waveguides, Primary Structural Members, Secondary Structural Members

Equipment Required:

1. Foam Glass Manufacturing Facility - 104 ton/day capacity
2. 70 Industrial Robots

Cost Comparables:

1. Based on a study by the University of Utah for the EPA, "Foam Glass Insulation From Waste Glass," Rpt. PB-272761, a foam glass manufacturing facility with the required capacity would cost approximately $1.8 million. Manpower requirements are in the order of 85 people.
2. Unimation Model 2005C Industrial Robot - $60,000 each

Equipment Cost:

Robots were substituted for 70 of the 85 persons required for the facility on a man for man basis.

Manufacturing Facility - $1.80
Industrial Robots - 70 @ .06 = $4.20

Total Cost = $6.00 million

Table G-36 Cost Summary - Stock Manufacturing Facilities

Item Number	Description	Total Cost (millions of 1977 dollars)
(1)	Aluminum Sheet	22.00
(2)	Aluminum Wire	2.68
(3)	Steel Sheet	24.87
(4)	Iron Sheet	3.78
(5)	Aluminum Castings	1.07
(6)	Sendust Casting	.34
(7)	Foamed Glass	6.00

B. PARTS MANUFACTURING FACILITIES

Parts manufacturing facilities are assigned Item numbers (8) through (14) and are described below. Costs are summarized in Table G-37.

Item (8) - Aluminum End Fittings - Primary Support Struts, MPTS Secondary Struts

Equipment Required:

1. 1 Sheet Metal Cutter
2. 1 Roll Forming Machine
3. 1 Blanking Press & Dies
4. 1 Electron Beam Welder
5. 2 Industrial Robots

Cost Comparables:

5. Unimation Model 2005C Industrial Robot - $60,000 each

Equipment Cost:

Equipment Items 1 through 4 above are used in parts manufacturing, Item (9), Aluminum Housings for Klystron, and no charge will be made here. Only the robots are costed in this category.

Industrial Robots - 2 @ .06

Total Cost = $.12 million

Item (9) - Aluminum Components; Klystron Solenoid Housing, Klystron Collector Housing

Equipment Required:

1. 1 Sheet Metal Cutter
2. 1 Roll Forming Equipment
3. 1 Blanking Press & Dies
4. 1 Welding Jig & Fixtures
5. 1 Metal Arc Welder
6. 1 Electron Beam Welder
7. 2 Industrial Robots

Item (9) - Continued

Cost Comparables:

1. Niagra IR4 Shear - $.025 million

2. Farnham 10 ft. roll former - $.183 million

3. Niagra PN-6040 blanking press - $.06 million

4. & 5. Linde SVI 400 welder - $.005 million

6. Previously purchased - cost not included under this item

7. Unimation Model 2005C Industrial Robot - $60,000 each

Equipment Cost:

Sheet Metal Cutter - $.025
Roll Former - $.183
Blanking Press - $.06
Welding Jig/Welder, 2 @ .005 - $.01
Industrial Robots - 2 @ .06 = $.12

Total Cost = $.398 million

<u>Item (10)- Copper Plating - Klystron Cavity Aluminum Parts</u>

Equipment Required:

1. .5T rubber coated electroplating tank & accessory power unit

2. 1 Industrial Robot

Cost Comparables:

1. .5T rubber coated electroplating tank & access power unit. GDC Facilities Engineering estimate $.40 million

2. Unimation Model 2005C Industrial Robot - $60,000 each

Equipment Cost:

Electroplating Tank - $.40
Industrial Robot - $.06

Total Cost = $.46 million

Item (11) - Foamed Glass Tubes and Waveguides

Facility required for these components was costed under Item (7) of Stock Manufacturing.

Item (12 - Aluminum Deposition on MPTS Waveguides

Equipment Required:

1. 6 - 160 KW Electron Beam Guns

Cost Comparables:

1. Airco Tenescal quote of \$2,000 - 3,000 per KW, including power supply. Cost per unit - \$160 x \$3,000 = \$480,000

Equipment Cost:

Electron Beam Guns - 6 @ 95% learning
$C = .48(6)^{.926}$

Total Cost = \$2.52 million

Item (13) - Steel Heat Pipes (Sheet)

Equipment Required:

1. 5 roll forming machines - 3 meter
2. Automatic tube welder
3. 3 presses for end closure
4. 3 Electron Beam Welders
5. 5 Tube Bending Machines
6. 5 Industrial Robots

Cost Comparables:

1. Farnham 10 ft. Roll Forming Equipment - .183 million each

2. & 4. Sciaky VX .3 Electron Beam Welder - .644 million each

3. End Closure Press - \$.024 million each - analyst's judgement

5. Tube Bending Machine - \$.012 million - analyst's judgement

6. Unimation Model 2005C Industrial Robot - \$60,000 each

Item (13) - Continued

Equipment Cost:

Roll Forming Machines - 5 @ .183 = \$.915
Tube Welder - \$.644
Presses - 3 @ .024 = \$.072
Electron Beam Welders - 3 @ .644 = \$1.932
Tube Bending Machines - 5 @ .012 = \$.060
Industrial Robots - 5 @ .06 = \$.30

Total Cost = \$3.923 million

Item (14) - Glass Fiber Insulation on Electrical Wiring

Equipment Required:

1. 1 Glass Filament Coater
2. 334 Braiding Machines (2 ft/minute rate)
3. 15 Industrial Robots
4. 1 Melting Furnace
5. 1 Bushing Winding Machine

Cost Comparables:

1. Glass Filament Coater - \$.02 million - analyst's judgement
2. New England Buff Co. Braiding Machines - \$.005 million each
3. Unimation Model 2005C Industrial Robots - \$60,000 each
4. Melting Furnace - \$.10 million, analyst's judgement
5. Bushing Winding Machine - \$.02 million, analyst's judgement

Equipment Cost:

Glass Filament Coater - \$.02
Braiding Machines - 334 @ 95% learning, $.005(334)^{.926}$ = \$1.086
Industrial Robots - 15 @ .06 = \$.90
Melting Furnace - \$.10
Bushing Winding Machine - \$.02

Total Cost = \$2.126 million

Table G-37 Cost Summary - Parts Manufacturing Facilities

Item Number	Description	Total Cost (millions of 1977 dollars)
(8)	Aluminum End Fittings	.12
(9)	Aluminum Housings	.40
(10)	Copper Plating	.46
(11)	Foamed Glass Tubes/Waveguides	(See Note 1)
(12)	Aluminum Deposition on Waveguides	2.52
(13)	Steel Heat Pipes	3.92
(14)	Glass Fiber Insulation	2.13

Note (1): Costs for this facility is included under Item (7) of Table G-36.

C. COMPONENT ASSEMBLY FACILITIES

Component assembly facilities are assigned Item numbers (15) through (20). Costs are derived below and are summarized in Table G-38.

Item (15) - DC-DC Converter Assembly

Equipment Required:

1. Assembly Fixture, including storage bins, turntable and controls, wire spools and locating tools (9 tons)
2. 2 Industrial Robots

Cost Comparables:

1. No assembly fixture comparable available. Cost estimate using structural CER: $.004W^{.667}$
   $.004(19845)^{.667} = \$2.94$ million
2. Unimation Model 2005C Industrial Robot - $60,000 each

Equipment Cost:

Assembly Fixture - $2.94
Industrial Robots, 2 @ .06 = $.12

Total Cost = $3.06 million

Item (16) - Klystron Assembly

Equipment Required:

1. 6 Electron Beam Welders
2. 12 Industrial Robots

Cost Comparables:

1. Sciaky VX.3 Electron Beam Welder - $.644 million
2. Unimation Model 2005C Industrial Robot - $60,000 each

Equipment Cost:

Electron Beam Welder - 6 @ 95% learning - $.644(6)^{.926} = \$3.384$
Industrial Robots - 12 @ .06 = $.72

Total Cost = $4.104 million

Items (17) & (18) - Radiator Assembly, Klystron and DC-DC Converter

1. 2 Cutting Machines to produce aluminum strip
2. 2 Brazing Furnaces w/conveyer system
3. 10 sets, Fixtures and Tooling (2 tons total weight)
4. 1 Cutting Machine to prepare 1 x 4M segments
5. 2 Forming Press & Die
6. 2 Automated Roll Seam Welder
7. 2 Fusion or Electron Beam Butt Welders
8. 10 Industrial Robots

Cost Comparables:

1 & 4. Niagra IR4 48 inch shear - $.025 million

2. Brazing Furnace 350-1100 deg. C. GDC Facilities estimate (B. Ladd) - $.045 million

3. Fixtures & Tooling - no cost comparable. Analyst's judgement - $1.25 million

5. Farnham Roll Forming Equipment, 10 ft width - $.183 million

6 & 7. Sciaky VX. 3 Electron Beam Welder - $.644 million

8. Unimation Model 2005C Industrial Robot - $.06 million each

Equipment Cost:

Cutting Machines - 3 @ .025 = $.075
Brazing Furnace - 2 @ .045 = $.090
Fixtures/Tooling - $1.250
Forming Press & Die - 2 @ .183 = $.366
Welders - 4 @ .644 = $2.576
Industrial Robots - 10 @ .06 = $.600

Total Cost = $4.957 million

Item (19) - Structural Member Assembly (Foamed Glass)

Equipment Required:

1. 3 - Heating Furnaces
2. 5 - Swaging Machines
3. 3 - Groove Cutters
4. 3 - Crimping Machines
5. 6 - Industrial Robots

Cost Comparables:

1. Induction Furnace under Item 5 of Stock Manufacturing Facilities - \$.25 million

2 & 4. Farnham 10 ft. Roll Former - \$.183 million

3. Niagra 1R4 Shear Machine - \$.025 million

5. Unimation Model 2005C Industrial Robot - \$.06 million each

Equipment Cost:

Heating Furnaces - 3 @ .25 = \$.75
Swaging Machines - 5 @ .18 = \$.90
Groove Cutters - 3 @ .025 = \$.075
Crimping Machines - 3 @ .18 = \$.54
Industrial Robots - 6 @ .06 = \$.360

Total Cost = \$2.625 million

Item (20) - MPTS Waveguide Subarray Assembly

Equipment Required:

1. 1 - Electron Beam Welder
2. 1 - Industrial Robot

Cost Comparables:

1. Sciaky Model VX.3 Electron Beam Welder - \$.644 million
2. Unimation Model 2005C Industrial Robot - \$.06 million each

Equipment Cost:

Electron Beam Welder - \$.644
Industrial Robot - \$.06

Total Cost = \$.704 million

Table G-38. Cost Summary - Component Assembly Facilities

Item Number	Description	Total Cost (millions of 1977 dollars)
(15)	DC-DC Converter	3.06
(16)	Klystron	4.10
(17)	DC-DC Converter Radiator	4.96
(18)	Klystron Radiator	
(19)	Structural Member	2.63
(20)	MPTS Waveguide Subarray	.70

D. SOLAR CELL PANEL PRODUCTION FACILITIES

Facilities for solar panel production are assigned Item numbers (21) through (26). Costs are derived below and are summarized in Table G-39.

Item (21) - Silica Glass Solar Cell Covers & Substrate

Equipment Required:

1. 1 - Melting Furnace (20 tons/hr capacity)
2. 10 - Insulated Molten Glass Tanks with 14 Molybdenum dies with slits (Weight 3 tons/tank)
3. 15 Industrial Robots

Cost Comparables:

1. Melting furnace scaled from .136 ton/hour induction furnace in Item (5), Stock Manufacturing by factor of .67. $C = .25\ (20/.136)^{.67}$ = \$7.08 million
2. Insulated Glass tank with dies - No cost comparable available. Estimate based on simple structural CER: First Unit $.004(6615)^{.667}$ = \$1.414 million
3. Unimation Model 2005C Industrial Robots - \$.06 million each

Equipment Cost:

Melting Furnace - \$7.08
Glass Tanks - 10 @ 90% learning - $1.414(10)^{.848}$ = \$9.96
Industrial Robots - 15 @ .06 = \$.54

Total Cost = \$17.58 million

Item (22) - Aluminum Deposition on Glass Substrate

Equipment Required:

1. 4 - 250 KW Electron Beam Guns w/power supplies for coating solar cell substrates with aluminum
2. 1 - Etching Tank & Maskant Film Interconnect Pattern for etching

Cost Comparables:

1. Airco Tenescal model EH 1200/50 electron beam gun w/power supply 3000/KW - \$.75 million each
2. Etching Tank & Maskant Film - \$.10 million - Analyst's judgement

Item (22) - Continued

Equipment Cost:

Electron Beam Guns - 4 @ \$.75 = \$3.00
Etching Tank/Maskant Film - \$.10

Total Cost = \$3.10 million

<u>Item (23) - Silicon Refining to PPB Level</u>

Equipment Required:

1. Silane/Silicone Process Equipment with 19272 ton/year capacity

Cost Comparable:

1. <u>Low Cost Solar Array Project Proceedings: 9th Project Integration Meeting</u>, Report 5101-67, April 1978 provides the following data on page 3-19: UCC Silane/Silicon Process Plant Cost - \$6.0 million
   Plant Size - 1000 metric tons/year

Equipment Cost:

Scaling above plant up based on capacity, cost is:

$$6.0\left(\frac{19272}{1000}\right)^{.67} = \$43.556 \text{ million}$$

<u>Item (24) - Silicon Solar Cells</u>

Equipment Required:

1. 4,283 Ribbon Growing Machines (edge-defined film-fed growth (EFG) process). Annual production - 117.04 x 10^6 m^2.

2. 1070 - Industrial Robots

Cost Comparables:

1. <u>Low Cost Solar Array Project Proceedings: 9th Project Integration Meeting,</u> Report 5101-67, April 1978 provides the following data on page 3-76: Cost of EFT equipment is \$16 per square meter of annual cell production.

2. Unimation Model 2005C Industrial Robots - \$.06 million each

Equipment Cost:

EFG Equipment - 117.04 x $10^6 m^2$ x 16 = \$1872.64
Industrial Robots - 1070 @ \$.06 = \$64.20

Total Cost = \$1936.84 million

Item (25) - Cut Ribbon, Dope, Apply Contacts & Anneal

Equipment Required:

1. 83 - 550KW Ion Beam Implanters, Electron Beam Annealer and contact coating equipment. Mass 30 tons.
2. 166 - Industrial Robots

Cost Comparables:

1. 200 KW Ion Beam Implanter, 2 ton mass (per A. Hurlich, GDC Mat'ls. Research) - $1 million
2. Unimation Model 2005C Industrial Robots - $.06 million each

Equipment Cost:

Beam Implanter cost, scaled up by weight to obtain First Unit Cost, using a .67 scaling exponent: $1.0\,(30/2)^{.67} = \$6.137$

Ion Beam Implanter @ 95% learning
$6.137(83)^{.926} = \$367.30$
Industrial Robots - 166 @ .06 = $9.96

Total Cost = $377.26 million

Item (26) - Silicon Solar Cell Module Assembly

Equipment Required:

1. 164 - Electrostatic Bonding Machines, 7.5 tons each
2. 164 - Automated Module Assembly Machines, 11.6 tons each
3. 254 - Industrial Robots

Cost Comparables:

1. Cincinnati Cost Breaker 90, Hydraulic Press Brake Model 135CB $75,000
2. Pratt & Whitney, Aztec 15, 4 axis Horizontal Machining Center, mass 10.25 metric tons - $.259 million
3. Unimation Model 2005C Industrial Robot - $.06 million each

Item (26) - Continued

Equipment Cost:

Bonding Machines - assume 95% learning
$.075(164)^{.926} = \$8.43$

Assembly Machines - assume 95% learning
$.259(164)^{.926} = \$29.12$

Industrial Robots - .06 x 254 = $15.24

Total Cost = $52.79 million

Item (27) - Glass Bag Manufacturing

Equipment Required:

1. 1 - Melting Furnace (.9 tons/hr capacity), 450 KW
2. 1 - Fiberglass Production Equipment (Bushings, drums, Insulated Molten Glass Tanks), 20 metric tons, 25 KW
3. 30 - Tubular Weaving Machines for 12 cm dia tubes at rate of 150 cm/min, 4 tons each, 10 KW
4. 10 - Heat Sealing Machines, 2 tons, 10 KW

Cost Comparables:

1. Melting Furnace scaled from .136 ton/hour induction furnace in Item (5), Stock Manufacturing by a factor of .67.
   $C = .25\ (.9/.136)^{.67} = \$.887$ million
2. No cost comparable. Estimate based on simple structural CER:
   First Unit $= .004\ (44100)^{.667}$
   $= \$5.01$ million
3. Pratt & Whitney, Aztec 15, 4 axis Horizontal Machining Center, mass - 10.25 metric tons - $.259 million
4. No cost comparable - Estimate @ $60,000 each

Equipment Cost:

Melting Furnace - $.887
Fiberglass Production Equipment - $5.01
Weaving Machines - 95% learning
$.259(30)^{.926} = \$6.04$

Heat Sealing Machines - 10 @ .06 = $.60

Total Cost = $12.537 million

Table G-39. Cost Summary - Solar Cell Panel Production Facilities

Item Number	Description	Total Cost (millions of 1977 dollars)
(21)	Solar Cell Covers	17.58
(22)	Aluminum Deposition	3.10
(23)	Silicon Refining	43.56
(24)	Solar Cells	1936.84
(25)	Cut Ribbon/Dope/Anneal	377.26
(26)	Cell Module Assembly	52.79
(27)	Glass Bag Manufacturing	12.54

E. LRU OPTION MANUFACTURING FACILITY COSTS

All development and production costs of facilities will be allocated to one of two RDT&E cost elements in the WBS: C(1326), Lunar Based Manufacturing Equipment or C(1333), Space Based Manufacturing. Cost to operate and maintain these facilities over their operational life is included under the SPS Production Phase, C(2226) and C(2323). Operations cost of all facilities are assumed to be 4 percent of production hardware cost per year (1% for spares, 3% for earth support of maintenance operations). Labor costs for operating the facilities are included under WBS elements C(2210) or C(2310), which are lunar and space based construction/maintenance crew costs.

The allocation of the manufacturing equipment between space and the lunar surface is the same for Options C and D. For Option B, all manufacturing equipment is in space except for Item (27), Glass Bag Manufacturing. The unadjusted facility hardware costs, and their allocation to space or the lunar surface, are shown in Table G-40. Cost adjustments and the resulting LRU manufacturing facility costs are shown in Table G-41.

An adjustment of 100 percent of the hardware cost was made to allow for any design changes in the equipment to make it compatible with a space environment and to allow for uncertainties. This is the cost for hardware development. System level costs, in the amount of 40% of the design change allowance, were added to the development costs to allow for initial tooling, system testing, training and program management.

Production costs, or those costs shown in Tables G-36, 37, 38 & 39 , were adjusted by 20 percent to allow for hardware accessories which may be required to integrate the equipment into a single facility or for installation. An allowance of 30 percent was made for Production Program Level Costs. It includes such items as sustaining engineering and tooling, system test and checkout and initial spares.

Table G-40. Unadjusted Manufacturing Element Costs and their Allocation
(millions of 1977 dollars)

Category	Item Number	Option C or Option D		Option B	
		Lunar Based	Space Based	Lunar Based	Space Based
Stock Manufacturing	(1)	22.00			22.00
	(2)	2.68			2.68
	(3)	24.87			24.87
	(4)	3.78			3.78
	(5)	1.07			1.07
	(6)	.34			.34
	(7)		6.00		6.00
Parts Manufacturing	(8)		.12		.12
	(9)		.40		.40
	(10)	.46			.46
	(11)	–	–		–
	(12)		2.52		2.52
	(13)		3.92		3.92
	(14)		2.13		2.13
Component Assembly	(15)		3.06		3.06
	(16)		4.10		4.10
	(17)		4.96		4.96
	(18)		–		–
	(19)		2.63		2.63
	(20)		.70		.70
Solar Cell Panel Manufacturing	(21)		17.58		17.58
	(22)		3.10		3.10
	(23)	43.56			43.56
	(24)		1936.84		1936.84
	(25)		377.26		377.26
	(26)		52.79		52.79
	(27)	–	–	12.54	
	Total Unadjusted Equipment Cost	98.76	2418.11	12.54	2516.87

Table G-41. LRU Manufacturing Facility Costs

	Option C or Option D		Option B	
	Lunar Based	Space Based	Lunar Based	Space Based
Unadjusted Equipment Cost (Ref. Table 5-44)	98.76	2418.11	12.54	2516.87
Plus:				
Hardware Accessories (20%)	19.75	483.62	2.51	503.37
Subtotal	118.51	2901.73	15.05	3020.24
Program Level Costs (30%)	35.55	870.52	4.52	906.07
Total Production Cost	154.06	3772.25	19.57	3926.31
Plus Development:				
Allowance for Design Change (100%)	98.76	2418.11	12.54	2516.87
System Level Costs (40%)	39.50	967.24	5.02	1006.75
Total Development & Production Cost	292.32	7157.60	37.13	7449.93
Annual Operations Cost (millions $/year)	4.74	116.07	.60	120.81

G.6 POWER STATIONS

Two types of power stations are used for the LRU options: nuclear and photovoltaic. The nuclear system is shown in Figure 4-32 and the GEO-based photovoltaic system is shown in Figure 4-33. An alternate photovoltaic system is lunar-based rather than GEO-based. Due to the similarity of the photovoltaic configurations, costs can be determined by the same methods.

G.6.1 Photovoltaic Power

The photovoltaic power stations are similar in configuration to the solar power satellite. The similarity of the systems allows power station costs to be estimated based on the Earth Baseline SPS costs. In the JSC briefing "A Recommended Preliminary Baseline Concept," dated January 25, 1978, the following data are obtained:

Satellite RDT&E Cost	\$6.27 billion
Satellite First Unit Cost	12.829 billion
Power Output	17 GW (approximate transmitted power)
Satellite Weight	97.49×10^6 Kg

Due to the similarity of the power station subsystems to the SPS, a 50 percent commonality factor will be assumed for development. Assuming an exponential relationship between cost and power output, development cost of the solar power station can be expressed as follows:

$$C = .5(6270)\left(\frac{P}{17000}\right)^{.5}$$

$$= 24.04\ P^{.5} \quad \text{(millions of 1977 dollars)}$$

where: P = power station output (megawatts)

The scaling relationship for first unit cost is:

$$C = 12829\left(\frac{P}{17000}\right)^{.67}$$

$$= 18.78\ P^{.67} \quad \text{(millions of 1977 dollars)}$$

For production of several power stations a 90% learning curve will be applied. Program level costs for production are assumed to be 20 percent of the hardware cost. Production cost can be expressed as follows:

$$C = 1.2 \left[18.78\ P^{.67}\right]\ N^{.848}$$

$$= 22.54\ P^{.67}\ \ N^{.848} \quad \text{(millions of 1977 dollars)}$$

where: P = Solar power station power output (megawatts)
N = Number of Solar power stations

Operations costs are estimated at 4% of production cost per year.

G.6.2 Nuclear Power

The nuclear power system concept is shown in Figure 4-32. Basis for the estimate is the 120 KW_e Nuclear Brayton Power Module described in Space Station Systems Analysis Study, SCB Alternate EPS Evaluation, MDAC Report No. G6959, Aug. 1977. System costs for the Brayton cycle power module are shown in Figure G-51 of Section G.7. Instead of 120 KW_e power sources, it will be assumed that 1000 KW_e nuclear power sources will be developed and a number of these will be used to satisfy power requirements. Scaling relationships will be used to estimate development and first unit costs from the MDAC data.

Development cost for a 120 KW_e system is $189 million (1977 dollars). Assuming a 1000 KW_e system is twice as complex, cost can be computed as follows:

$$\text{Development Cost} = 189 \left(\frac{1000}{120}\right)^{.5} \times 2$$

$$= \$1091 \text{ million (1977 dollars)}$$

In addition to the basic system, a conversion and distribution system must be developed which carries the power to the required locations. This is assumed to be an additional 20 percent for a total development cost of $1309 million.

From Figure G-51, production cost for four units is 19.8 million (1977 dollars). Assuming a 90 percent learning curve was used the first unit cost is:

$$\text{TFU Cost (120 KW}_e\text{ System)} = \frac{19.8}{4^{.848}} = 6.1 \text{ million (1977 dollars)}$$

Scaling the above first unit cost up to a 1000 KW_e system we obtain:

$$\text{TFU Cost (1000 KW}_e\text{ System)} = 6.1\left(\frac{1000}{120}\right)^{.67}$$

$$= 25.3 \text{ million (1977 dollars)}$$

An additional 10% will be allowed for the conversion and distribution system, giving a total first unit cost of $27.8 million.

The above first unit cost is for a 1000 KW_e system. For larger systems the 1000 KW_e elements can be ganged together to reach the required power level. It is assumed that the cost of additional units follow a 90 percent learning curve and production cost can be expressed as follows:

$$C = 27.8\ N^{.848} \quad \text{(millions of 1977 dollars)}$$

where: N = Number of 1000 KW_e elements

Operations costs include the cost of spares, maintenance, fuel (U 238) and labor to operate the facility. Operating labor and maintenance will be included under the Construction/Maintenance Crew elements. An allowance will be made for maintenance which includes earth activities in support of the maintenance function. Costs are as follows:

	Annual Cost
Spares	1% (Production Cost)
Maintenance	3% (Production Cost)
Fuel	.5% (Production Cost)
	4.5% (Production Cost)

Table G-42. Propellant Depot Facility Cost Estimate.
(Millions of 1977 $)

	5 M lb Capacity			40 M lb Capacity		
Cost Element	Size	Dev	Prod	Size	Dev	Prod
Structure	15 K lb	11.31	.05	40 K lb	18.46	.10
Avionics/Software	500 lb	20.92	2.23	625 lb	26.15	2.79
Solar Array	33.3 m^2	-	.01	266 m^2	-	.02
Electrical Power System	1000 lb	5.35	.81	8000 lb	15.13	3.24
Fluid System/Plumbing	1500 lb	4.55	2.82	4000 lb	7.42	5.43
Reliquifiers	2200 lb	13.95	5.72	6000 lb	22.86	11.09
Radiators	300 lb	.46	.17	800 lb	.75	.32
RCS System	400 lb	5.61	2.24	640 lb	7.35	4.09
Subtotal		62.15	14.05		98.12	27.08
Floating Items		23.62	4.78		37.29	9.12
Initial Spares			2.11			4.06
Initial Transportation			.18			.51
Total		85.77	21.12		135.41	40.77

Ref.: Orbital Propellant Handling and Storage Systems for Large Space Systems, Vol. 2, GDC Rpt. CASD-ASP-78-001 (JSC-13967), April 1978.

Table G-43. Tanker Module Cost Estimates.

(Millions of 1977 $)

	1 MLB O/H Tanker			2 MLB O/H Tanker			1 MLB Water Tanker			2 MLB Water Tanker		
Cost Element	Weight	Dev	Unit	Weight	Dev	Unit	Weight	Dev	Unit	Weight	Dev	Unit
Tanker Module	-	54.21	-	-	86.78	-	-	39.65	-	-	44.19	-
Structure	12653 lb	-	.97	36750 lb	-	1.87	-	-	-	-	-	-
LH_2 Tank	8557 lb	-	3.70	15365 lb	-	5.19	-	-	-	-	-	-
LO_2 Tank	5557 lb	-	.74	11166 lb	-	1.10	-	-	-	-	-	-
Water Tank	-	-	-	-	-	-	8406	-	1.44	15000	-	1.93
Tooling (Structure)	-	12.16	-	-	30.41	-	-	-	-	-	-	-
Tooling (Tank)	-	7.48	-	-	12.89	-	-	8.55	-	-	14.07	-
Floating Items	-	28.06	-	-	49.43	-	-	18.32	-	-	22.14	-
Ground Test Units	-	23.47	-	-	46.30	-	-	4.94	-	-	7.26	-
Initial Spares	-	-	.81	-	-	2.07	-	-	.23	-	-	.34
Total		125.38	6.22		225.81	10.23		71.46	1.67		87.66	2.27

Ref.: Orbital Propellant Handling and Storage Systems for Large Space Systems, Vol. 2, GDC Rpt. CASD-ASP-78-001 (JSC-13967), April 1978.

Table G-44. Propellant Depot Sizing - Option B.

Depot Location	Use	Mission	Tons P/L Per SPS	kg Prop / kg P/L	Total Prop (Tons/Yr)	Oxid (Note 3)	Fuel (Note 3)	Trips/Yr	Metric Tons/Trip Oxidizer	Metric Tons/Trip Fuel
LEO	$COTV_2$	LEO ▶ LLO	236	.3168	74.765	73.576	1.189	2	36.788	0.594
	$COTV_3$	LEO ▶ SMF	13153	.2912	3830.154	3769.254	60.899	2	1884.627	30.450
	$POTV_1$	LEO ▶ GEO	85	2.2	187	163.625	23.375	6	27.271	3.896
	$POTV_2$	LEO ▶ SMF	1073	2.1	2253.3	1971.638	281.663	38	51.885	7.412
	$POTV_3$	LEO ▶ LLO	38	1.87	71.06	62.178	8.883	2	31.089	4.441
					6416.979	6040.271	376.009		2031.660	46.743
								min capacity ——▶	2078.453 metric tons	
GEO	$POTV_1$	GEO ▶ LEO	85 ①	2.2	187	163.625	23.375	6	27.271	3.896
								min capacity ——▶	31.167 metric tons	
SMF	$COTV_3$	SMF ▶ LEO	6591	.2912	1919.299	1888.782	30.517	2	944.391	15.258
	$COTV_4$	SMF ▶ GEO	98596	.06	5915.76	5821.699	94.061	3	1940.566	31.35
	$POTV_2$	SMF ▶ LEO	1073	2.1	2253.3	1971.638	281.663	38	51.885	7.412
	TT	SMF ▶ L_2	10,000+168,722	.0248	4432.306	3878.267	554.038	2	1939.134	277.019
					14520.665	13560.386	960.279		4875.976	331.039
								min capacity ——▶	5207.015 metric tons	
LLO	$POTV_3$	LLO ▶ LEO	38 ①	1.87	71.06	62.178	8.883	2	31.089	4.441
	PLTV (Round Trip)	LLO ▶ Moon	38 ②	1.52	57.76	50.540	7.220	2	25.270	3.610
					128.82	112.718	16.103		56.359	8.051
								min capacity ——▶	64.410 metric tons	

NOTES:

1 POTV P/L wt. calculated by assuming 65000# wt for 75 people or 866.67#/Person (.393 metric tons/person).

2 Assume same wt/person for PLTV as POTV.

3 COTV propellants are 1.59% LH_2, 98.41% LO_2. POTV and TT propellants are 12.5% LH_2, 87.5% LO_2

Table G-47. Modular Space Station Weights.

Module	Dry Weight (lbs)
Initial Core	20944
Power	22262
Station Module 1	18855
2	16705
3	16245
4	18302
5	15676
Station Module 6	14820
Growth Core	10283
Cargo Module	10940
Solar Flare Shielding	54243
Total	219,275 (99.4 metric tons)
Total W/O Shielding	165,032 (74.8 metric tons)

Notes: (1) Power Module weight includes 9702 lbs for solar array

(2) Solar Flare Shielding estimate taken from page 4-111 in Section 4.5.2.

(3) Station is for 12 man crew. Weight to man ratio is 8.3 with shielding and 6.2 metric tons/man without shielding

Ref: Modular Space Station, Phase B Extension, Rockwell Report Nos. SD71-226-1 and -2, Jan. 1972.

Table G-48. Modular Space Station Costs.

Cost Element	Development Mils 72 $	Development Mils 77 $	Production Mils 72 $	Production Mils 77 $	Operations Mils 72 $	Operations Mils 77 $
Initial Core	219.7		38.0			
Power Module	172.3		113.7			
Station Module 1	250.7		38.3			
2	108.7		15.4			
3	48.7		15.1			
4	60.4		36.1			
5	34.6		25.3			
Station Module 6	10.7		12.7			
Growth Core	51.1		19.3			
Cargo Module	51.5		29.9		52.9 (refurb)	
Mission Ops					201.3	
Spares					24.5	
Programmatic	585.2		74.2		82.5	
Subtotal	1593.9	2247.4	418.0	589.4	361.2	509.3
Solar Flare Shielding		54.2		5.4		
Total	-	2301.6		594.8		509.3

Notes: (1) Adjustments to 1977 dollars made using GNP price deflator; multiplier is 1.41

(2) Costs for the aluminum solar flare shielding based on the assumptions that Development Cost is $1000/lb and Production is $100/lb.

(3) Operations costs are for a 15 year period.

Ref: Modular Space Station, Phase B Extension, Rockwell Report Nos. SD71-226-1 & -2, Jan. 1972.

Table G-49. Rockwell Study - Lunar Base Cost Data.
(Millions of 1970 Dollars)

	Nonrecurring	Recurring	Total
Crew and medical module	62.1	14.8	77.9
Crew and operations module	48.5	25.6	74.1
Sortie and transient module	25.0	17.1	42.1
Lab and backup command module	40.3	23.7	64.0
Assembly and recreation module	23.0	11.0	34.0
Base maintenance module	13.1	7.1	20.2
Drive-in garage module	9.6	4.3	13.9
Drive-in warehouse module	8.0	4.6	12.6
Mobile cargo modules	5.8	10.1	15.9
Deep drill cover module	7.1	4.3	11.4
Support operations equipment module	98.4	56.3	154.7
Observatory shell modules	2.7	5.4	8.1
Mobility equipment transport modules	0.3	16.1	16.4
Ground support equipment	40.2	2.8	43.0
Systems test hardware	140.4	-	140.4
Launch support operations	-	13.9	13.9
Facilities	42.2	-	42.2
Logistics and training equipment	12.0	5.3	17.3
System engineering support	32.2	4.5	36.7
Project management	32.2	5.0	37.2
Total	$644.1	$231.9	$876.0

Ref.: Lunar Base Synthesis Study, Vol. IV, Cost and Resource Estimates, North American Rockwell, Rpt. SD71-477-4, May 1971, page 7-5.

Table G-50

SPS SYSTEMS DEFINITION STATUS REPORT	TRANSPORTATION SYSTEMS	
	H. P. DAVIS	1/25/78

SPS TRANSPORTATION REQUIREMENTS

VEHICLE	FLIGHT/YEAR	FLEET SIZE	PROPELLANT REQUIREMENTS METRIC TONS	
PLV 1ST STAGE	36 (2)	2 (5)	O_2 CH_4	44,000 12,500
PLV 2ND STAGE	36 (2)	2 (5)	O_2 H_2	17,000 2,850
POTV - BOTH STAGES	5 (2)	2	O_2 H_2	2,160 360
HLLV 1ST STAGE	391 (1)	6 (4)	O_2 CH_4	2.0×10^6 670,000
HLLV 2ND STAGE	391 (1)	6 (4)	O_2 H_2	812,000 133,000
COTV ~ LARGE PANELS	8 (3)	8	AR O_2 H_2	11,800 5,040 840
COTV ~ SMALL PANELS	24 (3)	24	AR O_2 H_2	13,000 5,400 900

TOTALS:	
O_2	$\approx 3 \times 10^6$
H_2	$\approx$140,000
CH_4	$\approx$700,000
AR	$\approx$ 25,000

NOTES:

(1) PLUS 61 FLTS 1ST YEAR TO DELIVER LEO & GEO CONST. BASES
(2) 480 PEOPLE IN LEO, 60 PEOPLE IN GEO, 90 DAY STAY TIME, 80% LOAD FACTOR
(3) ASSUMED SINGLE FLIGHT/UNIT
(4) 4 DAY TURNAROUND, 25% SPARES
(5) 14 DAY TURNAROUND, 40% SPARES

Figure G-51

SOLAR AND REACTOR BRAYTON SYSTEM COSTS

10 YR PROGRAM COSTS IN MILLIONS OF DOLLARS ('78)

	SOLAR BRAYTON		REACTOR BRAYTON	
DDT&E				
POWER MODULE	103		179	
INTEGRATION PACKAGE	14		21	
SUBTOTAL		117		200
PRODUCTION				
POWER MODULES (4)	8		18	
INTEGRATION PACKAGE	2		3	
SUBTOTAL		10		21
OPERATIONS (10 YEARS)				
INITIAL LAUNCH (2)	40		40	
SUPPORT LAUNCHES (2.1)	42		42	
SPARES AND REPLACEMENT HARDWARE	5		5	
RCS PROPELLANT COST	18		9	
SUBTOTAL		105		96
TOTAL		232		317
PROBABLE UPPER BOUND*		348		634

*BASED ON HISTORICAL AND RISK ASSESSMENT

Source: Space Station Systems Analysis Study, SCB Alternate EPS Evaluation, Task 10 MDAC Report No. MDC G6959, August 1977

APPENDIX H

Supplementary notes to LRU oncept cost tables in Section 5.3.1.

This appendix is divided into 3 sections:

H.1 NOTES TO TABLE 5.5, page 5-18 of Volume II

NOTE 1.1

Nuclear Brayton Power Station with a 50 MWe capacity.

From Section G. 6.2, costs are as follows:

Development	\$1309
Production - $27.8(50)^{.848}$	\$ 766.958
	\$2075.958 million

NOTE 1.2

There are two lunar based habitats. One is a 48 person habitat for small crews as discussed in Section G. 2.4 (Ref. Fig. G-6 for Scaling Relations). The second is a 12 person Temporary Shelter as discussed in Section G. 2.3 (Ref. Fig. G-5 for Scaling Relations). Costs for Development and Production are:

Small Habitat:	$205.3(48)^{.5} + 38.7(48)^{.67}$	=	\$1940.137
Temporary Shelter:	$306.9(12)^{.5} + 37.6(12)^{.67}$	=	\$1261.852
			\$3201.989 million

NOTE 1.3

Beneficiation Equipment weight is 9 tons, or 19,845 lbs.

Costs can be determined from Section G. 5.2.

Development:	$1.546(19845)^{.187}$	=	\$ 9.837
Production:	$.007(19845)^{.667}$	=	\$ 5.148
			\$14.985 million

NOTE 1.4

There will be a temporary requirement for propellant storage on the lunar surface during startup. Tanks with capacities of 7.5 tons for LH_2 and 52.5 tons for LO_2

are required for the POTV. Assume standard tanks from Section G.1.6 are used:

LO_2 tank	1.526
LH_2 tank	6.768
	\$8.294 million

NOTE 2.1

Photovoltaic Power Station with a capacity of 650MW and mass of 5030 metric tons. Using the relations in Section G.6.1 the following costs are obtained

Development:	$24.04(650)^{.5}$	=	612.902
Production:	$22.54(650)^{.67}$	=	1728.247
			\$2341.149 million

NOTE 2.2

LEO Modular Space Station - 75 person crew. From Figure G-3 costs are:

Development:	$665.1(75)^{.5}$	=	5759.935
Production:	$112.7(75)^{.67}$	=	2033.363
			\$7793.298 million

NOTE 2.3

GEO Modular Space Station with 36 person capacity and solar flare shelter. Costs can be determined from Fig. G-4:

Development:	$647.0(36)^{.5} + 255.1$	=	4137.1
Production:	$111.1(36)^{.67} + 33.8$	=	1259.6
			\$5396.7 million

NOTE 2.4

LLO Temporary Shelter with 12 person capacity. Development costs are included under lunar based habitats (Ref. Note 1.2). From Fig. G-5:

Production Cost $= 37.6\ (12)^{.67}$

$= \$198.719$ million

NOTE 2.5

2:1 Resonance Orbit SMF Habitat with 1365 person capacity.

Costs can be determined from Fig. G-8 .

Development:	$445.6(1365)^{.5} + .120(1365)$	=	16,626.896
Production:	$67.4(1365)^{.67} + .012(1365)$	=	8,512.072
			\$25,138.968 million

NOTE 2.6

Space Based Beneficiation Equipment weight is 18 tons, or 39,690 lbs.

Costs can be determined from Section G. 5.2.

Development:	$1.546(39690)^{.187}$	=	11.199
Production:	$.007(39690)^{.667}$	=	8.174
			\$19.373 million

NOTE 2.7

Processing Facility weight is 8275 tons per Table G-31. Scaling relationships are contained in Section G. 5.3.

Development:	$13.450(8275)^{.5}$	=	1223.507
Production:	$5.756(8275)^{.67}$	=	2426.752
			\$3650.259 million

NOTE 2.8

Liquefaction Facility weight is 64 metric tons, excluding power supply. Power required is 2.32 MW. Costs can be determined from the scaling relationships in G. 5.4.

Development:	$11.628(64)^{.5} + 24.04(2.32)^{.5}$	=	129.641
Production:	$4.785(64)^{.67} + 22.54(2.32)^{.67}$	=	117.241
			\$246.882 million

NOTE 3.1

This is the cost to operate the vehicles during facility construction and includes spares, maintenance and propellants. Startup period is 3 years. During this time a gradual buildup of the vehicle fleet occurs. Assume an average maintenance period of 1 1/2 years, instead of 3 years to account for the buildup. Cost of spares and maintenance is calculated below.

SDV	(Ref. Note 6.2):	$8.571 million/flt × 615 flts	=	$5271.165
COTV	(Ref. Note 12.2):	(84.194 + 253.090) 1.5 years	=	505.926
POTV	(Ref. Note 12.2):	(6.754 + 20.251) 1.5 years	=	40.508
PLTV	(Ref. Notes 9.2 & 9.3):	(2.04 + .68) 1.5 years	=	4.080
Mass Driver	(Ref. Notes 9.2 & 9.3):	(8.073 + 2.691) 1.5		16.146
Mass Catcher	(Ref Note 12.2):	(11.574 + 34.722) 1.5		69.444
				$5907.269 million

User charges for the space shuttle are calculated at $20 million per flight and 42 flights are required for startup. Total cost is 42 × 20 = $840 million.

Total propellant requirements for startup were presented in Section 4.8 for space vehicles. SDV propellant requirements per flight are provided in Table G-9. The following Table summarizes total propellant requirements for startup operations.

All propellants for startup are assumed to be earth supplied and the cost per pound as shown in Table G-9 applies.

Total Propellant for Startup (Millions of Pounds)

User	LO_2	LH_2	C_3H_8
SDV	4752.7	27.1	1188.8
COTV	56.0	.9	—
POTV/PLTV	12.4	1.8	—
Catcher	3.5	.3	—
Total	4824.6	30.1	1188.8
$/lb	.021	.54	.37
Total Cost (millions $)	101.317	16.254	439.856

Total transportation cost is the sum of the above elements: 5907.269 + 840 + 101.317 + 16.254 + 439.930 = $7304.770 million.

NOTE 3.2

Initial Depot Propellant Supply is provided in Section 4.8. Costs are:

LO_2: 3349 tons × 2205 × $.021/lb = $ .155 million

LH_2: 1195 tons × 2205 × $.54/lb = $1.423 million

$1.578 million

NOTE 3.3

Construction/Maintenance Crews during facility activation average approximately 800 persons. At a cost of $.120 million/man year total cost for the 3 year period is: 3 × .120 × 800 = $288 million.

NOTE 3.4

Operations cost for the SDV launch/recovery facilities is \$22.662 million/year (Ref. Note 5.3). Total for 3 years is \$67.986 million.

NOTE 3.5

Annual cost of lunar based operations is \$78.866 million (1/30 of cost element C(2220)). For the 3 year activation period assume the average annual cost is half the steady state value, or \$39.433 million. Operations cost for the facility activation period is: $3 \times 39.433 =$ \$118.299 million.

NOTE 3.6

Annual cost of space based operations is \$1008.067 million (1/30 of cost element C(2320)). For the 3 year activation period assume an average annual cost of half the steady state value or \$504.034 million. Operations cost for the facility activation period is: $3 \times 504.034 =$ \$1512.102 million.

NOTE 4.1

Each POTV has a 50 flight life. Total flights are as follows:

Startup Operations		50
Steady State:	$POTV_1 - 6 \times 30$	180
	$POTV_2 - 38 \times 30$	1140
	$POTV_3 - 2 \times 30$	60
Total		1430 flights

Number of vehicles required: $1430/50 = 28.6 \approx 29$. Need 11 vehicles for startup operations for safety reasons. The remaining 18 will be manufactured during the SPS production phase. Costs can be determined from the relationships in Section G.3.8.

Vehicle Development:			\$380 million
Total Production: $29.24(29)^{.848}$	=		508.266 million
Initial Production: 11/29 (508.266)	=		\$192.791 million
Replacement Vehicles: 18/29 (508.266)	=		\$315.475 million

Each operating POTV requires passenger and crew modules. A total of 11 are required and will be fabricated during initial production. No replacements are assumed. Costs can be obtained from Section G.3.5.

Development: Passenger Module	287
Crew Module	524
	\$811 million
Production:	
Passenger module: $13(11)^{.848}$	99.322
Crew Module: $24(11)^{.848}$	183.363
	\$282.685 million

Total costs are as follows:

Development:	
Vehicle	380
Module	811
	\$1191 million
Initial Production:	
Vehicle	192.791
Module	282.685
	\$475.476 million
Replacement Vehicles:	
Production	\$315.475 million

NOTE 4.2

Each COTV has a 50 flight life, but the number required over the program life is based on the annual launch requirements during steady state. The trip/year constraint applies since each COTV can only make 1 trip/year. From Table 4-39 the flights/year and thus the total number of vehicles required is:

$COTV_2$	2
$COTV_3$	2
$COTV_4$	3

All vehicles will be manufactured during the initial production phase. Table G-13 provides the following costs:

Development:	\$636.94 million
Initial Production:	\$8705.24 million

NOTE 4.3

Each SDV has a 500 flight life. The following launch and vehicle requirements exist:

Startup period:	615 flights/500	=	1.23
Steady State:	$(68 \times 30)/500$	=	<u>4.08</u>
			$5.31 \approx 6$

Four SDV's will be required for startup in order to accomplish startup within a 3 year period. The remaining two will be replacements, manufactured during the SPS production phase. Costs were discussed in Section G.3.6 and are shown below:

Development:		
Booster		\$5311.50
Cargo Pod		\$1520.64
		\$6832.14 million
Reusable Hardware Production (90% learning):		
Booster $1.3\ (364.72)\ (6)^{.848}$	=	2166.58
Cargo Pod $1.3\ (103.44)\ (6)^{.848}$	=	614.47
Total		\$2781.05 million
Initial Production (4/6):		\$1854.03
Replacement (2/6):		927.02

The cargo pod shroud and external tanks are expendable and a total of 2655 shipsets of these elements will be required.

Expendable Hardware Production (85% learning):

$1.3\ (18)\ (2655)^{.766}$ = \$9818.59 million

Assume 650 shipsets will be fabricated during initial production and the remaining 2005 are made during SPS production:

Initial Production: (650/2655) 9818.59 = \$2403.80 million

Replacement: (2005/2655) 9818.59 = \$7414.79 million

NOTE 4.4

Two mass catchers are required. From Section G.3.13, Table G-25, Development cost is \$677.919 million; Production cost is: $694.453\ (2)^{.848}$ = \$1250.018 million.

NOTE 5.1

A total of 10.4% of the SPS mass must be earth supplied. This amounts to 10232 metric tons per satellite. Approximately half of this material is complex avionics equipment and half is miscellaneous material which is not obtainable on the moon or not desirable to manufacture on the lunar surface. These materials are not well enough defined for detailed estimates so a general electronics type CER will be used to provide a single estimate for the entire amount: TFU cost = $.021\ W^{.667}$. This CER is from Shuttle System Payload Data Activity, GDC Report PDS-CO-015, Sept. 1974. First Unit Cost, including 10% for Program Level Costs, is:

$$\begin{aligned} \text{TFU} &= .021\,(22{,}561{,}560)^{.667} \times 1.1 \\ &= \$1686.229 \text{ million} \times 1.1 \\ &= \$1854.852 \text{ million} \end{aligned}$$

Production costs, using a 90% learning curve and 30% for Program Level Costs are:

$$\begin{aligned} \text{Production Cost} &= 1854.852(30)^{.848} \times 1.3 \\ &= \$43{,}137.149 \text{ million} \end{aligned}$$

NOTE 5.2

Production costs of earth rectenna are identical to the SPS earth baseline and are \$133.38 billion.

NOTE 5.3

From Table G-30 in Section G.4 , annual maintenance cost of the Launch/Recovery Facilities is \$22.662 million. Total for 30 years is 30 × 22.662 = \$679.86 million.

Launch/Recovery Operations costs are included in the SDV operations cost on a per flight basis and will not be included here.

NOTE 6.1

All transportation charges for Earth Based Fab/Assy refer to the operations cost of the SDV and Space Shuttle. Vehicle Replacement cost for the SDV was calculated in Note 4.3 and the total is:

Reusable Hardware	$ 927.02
Expendable Hardware	$7414.79
	$8341.81 million

NOTE 6.2

From Section G.3.6, cost per flight for spaces and maintenance is $8.571 million/flight for the SDV.

$$68 \text{ flights/year} \times 20 \text{ years} \times 8.571 = \$17484.84 \text{ million}$$

NOTE 6.3

From Section G.3.6, propellant costs are $.901 million per SDV flight.

$$68 \text{ flights/year} \times 30 \text{ years} \times .901 = \$1838.040 \text{ million}$$

NOTE 6.4

From Section G.3.7, cost per Shuttle flight is $20 million.

$$41 \text{ flights/year} \times 30 \text{ years} \times 20 = \$24,600 \text{ million}$$

NOTE 7.1

Number of people assigned to lunar base during steady state operations is 48. Assuming a rate of $.120 million per man year, the cost is:

$$48 \times .120 \times 30 = \$172.800 \text{ million}$$

NOTE 8.1

Spares, fuel and maintenance costs of the nuclear power station are 4.5% of production cost per year (Ref. Section G.6.2). From Note 1.1, production cost is $766.958 million. Total cost is:

$$.045 \times 766.958 \times 30 \text{ years} = \$1035.393 \text{ million}$$

NOTE 8.2

From Section G.2.7 annual habitat operations cost is 5.8% of production cost/ year. From Note 1.2 production costs are $38.7\ (48)^{.67} + 37.6\ (12)^{.67}$ = \$716.496 million. Total operations cost is:

.058 x 716.496 x 30 years $1246.703 million

NOTE 8.3

From Section G.5.2, beneficiation equipment annual cost is 4% (Production Cost). Using the production cost from Note 1.3 total cost is: .04(5.148) x 30 years = \$6.178 million.

NOTE 8.4

Table G-41 Section G.5.5 provides the lunar based manufacturing operations cost as \$.60 million per year. Total cost is: 30 years x .60 = \$18 million.

NOTE 9.1

Transportation charges for Lunar Based Fab/Assy refer to the PLTV and mass driver. The PLTV fleet and mass driver are fabricated during initial production and no replacement vehicles are required.

NOTE 9.2

From Section G.3.11, annual maintenance cost for the PLTV is \$1.02 million/ year/vehicle. For a fleet size of two annual cost is \$2.04 million. From Section G.3.14 annual maintenance cost of the mass driver catapult is \$8.073 million. Total operating cost for the 30 year period is: 30(204 + 8.073) = \$303.39 million.

NOTE 9.3

Annual cost for PLTV spares, from Section G. 3.11 is \$.34 million/year/vehicle. For a fleet size of two, annual cost is \$.68 million. From Section G. 3.14 mass driver spares cost is \$2.691 million/year. Total cost for spares for the 30 year period is: 30 (.68 + 2.691) = \$101.130 million.

NOTE 9.4

Per Section G. 3.11, propellant cost per PLTV flight is \$6118. Total cost for 30 years is:

30 years × 2 flts/yr × \$.006	=	\$.36 million

NOTE 10.1

There are 1365 people stationed in the 2:1 Resonance Orbit and 36 in GEO. Using a rate of \$.120 million/man year, total cost is:

1401 × .120 × 30 years	=	\$5043.6 million

NOTE 11.1

From Section G.6.1, power station operations cost is 4% of production cost per year. Production cost is \$1728.247 million (Ref. Note 2.1). Total operating cost is

.04 (1728.247) × 30 years	=	\$2073.896 million

NOTE 11.2

Habitat operations costs, from Section G.2.7, is 5.8% of hardware cost per year (excluding the cost of lunar shielding). Costs were calculated as follows:

Habitat	Production Cost	30 Yr. Operations Cost
LEO	\$2033.363 (Ref. Note 2.2)	\$3538.052 million
GEO	\$1259.6 (Ref Note 2.3) less \$1.4 for shielding = \$1258.2	\$2189.268 million

Habitat	Production Cost	30 Yr. Operations Cost
LLO	\$198.719 (Ref. Note 2.4)	\$345.771 million
SMF	$67.4\ (1365)^{.67}$ = \$8495.69 (Ref. Sec. G.2.6)	\$14,782.501 million
Total Space Habitat Operations		\$20,855.592 million

NOTE 11.3

From Table G-41 in Section G. 5. 5 annual operations cost of the space based manufacturing facility is \$120. 81 million/year. For 30 years the cost is:

30×120.81 = \$3624.30 million

NOTE 11.4

Annual operations cost for the propellant depots is \$20. 839 million/year (Ref. Section G. 1). For 30 years the cost is:

30×20.839 = \$625.170 million

NOTE 11.5

From Section G. 5. 2 annual cost is 4% of production. Note 2. 6 shows space based Beneficiation equipment production cost as \$8. 174 million. Total operations cost is:

$.04\ (8.174) \times 30$ years = \$9.809 million

NOTE 11.6

From Section G. 5. 3, annual operating cost is 4% of production cost per year. Note 2. 7 provides processing facility production cost. Total operating cost for 30 years is:

$.04(2426.752) \times 30$ = \$2912.102 million

Note 2.7 provides processing facility production cost. Total operating cost for 30 years is:

$$.04(2426.752) \times 30 \qquad = \$2912.102 \text{ million}$$

NOTE 11.7

From Section G. 5.4 and Note 2.8 total operations cost for the liquifaction equipment is:

$$.04(117.241) \times 30 \text{ years} \qquad = \$140.689 \text{ million}$$

NOTE 11.8

The LRU options require only one satellite construction facility in GEO, compared with two in the earth baseline. Figure F-8, gives annual costs to maintain the GEO construction facility as follows:

Facility	.159
Construction Equip	.247
Supply/Refurbishment	.191
	$.597 billion/yr

Total Cost = $.597 \times 30$ = $17.910 billion

NOTE 12.1

Transportation costs under Space Based Fab/Assy refer to the cost of operating the COTV's, POTV's and Mass Catcher. Vehicle replacement cost was previously calculated in the Notes 4.1 and 4.2. No Mass Catcher replacements are required. Total is $315.475 million.

NOTE 12.2

COTV operating costs are shown in Section G.3.9, Table G-16. Operations costs are as follows:

	Steady State flts/year	Propellant cost per flight	Fleet Cost/year Propellants	Spares	Maintenance
$COTV_2$	2	.005	.010	2.746	8.746
$COTV_3$	2	.132	.264	21.400	64.200
$COTV_4$	3	.077	.231	60.048	180.144
Total Annual Cost			.505	84.194	253.090

POTV operations costs are the costs of maintaining not only the POTV in operating condition, but the passenger and crew modules as well. Fleet size is 11 vehicles and a total of 46 flights per year are required. Sections G. 3.5 and G. 3.8 provide the following operations costs:

Spares:	.13 + .24 + .244 = \$.614 million/year/vehicle
Maintenance:	.39 + .72 + .731 = \$1.841 million/year/vehicle
Propellant:	\$.012/flt

Total annual costs for the 11 vehicle POTV fleet are:

Spares:	\$.614 × 11	=	\$6.754 million
Maintenance:	\$1.841 × 11	=	\$20.251 million
Propellant:	\$.012 × 46 flts/yr	=	\$.552 million

Section G. 3.13 provides operations costs for the Mass Catcher. Annual costs for the two vehicle fleet are:

Spares:	2(\$5.785)	=	\$11.574 million
Maintenance:	2 (\$17.361)	=	\$34.722 million
Propellant:	2 flts/yr × \$.953	=	\$1.906 million

Total operating costs for the COTV, POTV and Mass Catcher over a 30 year period are summarized below:

	Spares	Maintenance	Propellant
COTV	84.194	253.090	.505
POTV	6.754	20.251	.552
Mass Catcher	11.574	34.722	1.906
Total Annual Cost (millions \$)	102.522	308.063	2.963
Total 30 yr. Cost (millions \$)	3075.660	9241.890	88.890

H. 2 NOTES TO TABLE 5-6, Page 5-22 of Volume II

NOTE 1.1

Nuclear Brayton Power Station with a 600 MW_e capacity from Section G. 6. 2 costs are as follows:

Development	$1309
Production $27.8\ (600)^{.848}$	$6308
	$7617 million

NOTE 1.2

Large Lunar Base Habitat with 400 person capacity. Using the relations from Section G. 2. 5 the cost of development and production is:

$$C = 220.1\ (400)^{.5} + 42.5\ (400)^{.67}$$

$$= \$6755.798 \text{ million}$$

NOTE 1.3

Beneficiation Equipment weight is 27 tons, or 59535 lbs. CER's are contained in Section G. 5. 2.

Development	$1.546(59535)^{.187}$	= $12.08 million
Production	$.007(59535)^{.667}$	= $10.71
		$22.79 million

NOTE 1.4

Liquefaction plant weight, excluding propellant storage tanks, is 486 tons. Using the scaling relations from Section G. 5. 4 the following costs are obtained:

Development	$11.628\ (486)^{.5}$	256.344
Production	$4.785\ (486)^{.67}$ =	301.945
		$558.289 million

NOTE 2.1

Photovoltaic Power Station in GEO with a capacity of 260 MW and mass of 2,015 metric tons. Using the relations in Section G.6.1 the following costs are obtained:

Development	$24.04\ (260)^{.5}$	= 387.633
Production:	$22.54\ (260)^{.67}(1)$	= 935.375
		$1323.009 million

NOTE 2.2

LEO Modular Space Station with a 75 person crew size. Costs can be determined from Figure G-3.

Development	$665.1\ (75)^{.5}$ =	5759.935
Production	$112.7\ (75)^{.67}$ =	2033.363
		$7793.298 million

NOTE 2.3

GEO SMF Habatat with a 1165 person capacity. Costs can be determined from Figure G-8.

Development	$445.6(1165)^{.5} + .120(1165)$ =	15,349.062
Production	$67.4(1165)^{.67} + .012(1165)$ =	7,654.066
		$23,003.128 million

NOTE 2.4

LLO Temporary Shelter with a 12 person capacity. Costs can be determined from Figure G-5.

Development	$306.9\ (12)^{.5}$	= 1063.133
Production	$37.6\ (12)^{.67}$	= 198.719
		1261.852 million

NOTE 3.1

Transportation costs during the activation of LRU facilities consist of spares, maintenance and propellant costs. This is the cost to operate the vehicles during facility construction.

The following table summarizes total propellant requirements and for startup operations. Propellant costs are based on the rates shown in Section G. 3.6.

Total Propellant for Startup (Millions of Pounds)

User	LO_2	LH_2	C_3H_8
SDV	6893.4	39.2	1724.2
COTV	72.8	1.2	—
POTV/LTV	58.1	8.3	—
	7024.3	48.7	1724.2
Total Cost (Millions $)	$147.5	$26.3	$638.0

Maintenance and spares costs for the vehicles are shown below. Average maintenance period assumed to be 1 1/2 years.

SDV:	8.571 million/flight × 892 flts (startup)	=	$7645.332 million
COTV:	(Ref Note 12.2) (160.366 + 481.098) 1.5 years	=	$ 962.196 million
POTV:	(Ref Note 12.2) (6.754 + 20.251) 1.5 years	=	$ 40.508 million
LTV:	(Ref Note 9.2) (1.974 + 5.929) 1.5 years	≐	$ 11,855 million

User charges for the space shuttle are calculated at $20 million per flight. For the 80 flights during startup total cost is $1600 million.

Total transportation cost is the sum of the above elements - $11,071.691 million

NOTE 3.2

Initial propellant supply for the propellant depots is as follows:

LO_2 11.7 million lbs × $.021/lb = $ .246 million

LH_2 16.8 million lbs × $ .54/lb = $9.072 million

$9.318 million

NOTE 3.3

Construction/maintenance crews during facility activation average 200 persons on the lunar surface and 600 persons in GEO. At a cost of $.120 million per manyear, total costs for the 3 year period are: 3 × .120 × 800 = $288 million.

NOTE 3.4

Operations of the launch/recovery facilities for the SDV cost 33.2 million/year (Ref Note 5.3). For the 3 year startup period total cost is $99.6 million.

NOTE 3.5

Annual cost of lunar based operations is \$525.80 million (1/30 of cost element 2220). For the 3 year activation period assume average annual cost is half steady state value or \$262.90 million. Operations cost for the facility activation period then is: $3 \times 262.90 = \$788.70$ million.

NOTE 3.6

Annual cost of space based operations is \$754.43 million (1/30 of cost element 2320). For the 3 year activation period assume average annual cost is half the steady state value or \$377.22 million. Operations cost for the facility activation period then is: $3 \times 377.22 = \$1131.66$ million.

NOTE 4.1

Each POTV has a 50 flight life. Total flights are as follows:

Startup operations		32
Steady State $POTV_1$	$18/yr \times 30$	540
$POTV_2$	$38/yr \times 30$	1140
		1712 flights

Number of vehicles required: $1712/50 = 34.24 \approx 35$

Need 11 vehicles for startup operations for safety reasons.

Remaining 24 will be manufactured during the SPS production phase.

Costs can be determined from the relationships in Section G. 3. 8.

Vehicle Development: \$380 million

Total Production: $29.24\,(35)^{.848} = \$596.138$ million

Allocating between Initial Production and Replacement:

Initial Production: 11/35 (596.138) = $187.358 million

Replacement Vehicles: 24/35 (596.138) = $408.780 million

Each operating POTV requires passenger and crew modules. Assume they have a life of 500 flights each. Based on the life only 4 sets would be required. Since 11 COTV's are required for startup 11 sets will be required in initial production. Section G.3.5 provides the estimating relationships.

Development:	Passenger Module	287
	Crew Module	524
		$811 million

Production:

Passenger Module	$13\,(11)^{.848}$	= 99.322
Crew Module	$24\,(11)^{.848}$	= 183.363
		$282.685 million

Total costs are as follows:

Vehicle Development		$380
Module Development		$811
POTV Development		$1191 million
Initial Production:	Vehicle	187.358
	Modules	282.685
		$470.043 million
Replacement Vehicle Production		$408.780 million

NOTE 4.2

Each COTV has a 50 flight life, but the number required over the program life is based on the annual launch requirements during steady state. The trip/year constraint applies since each COTV can only make 1 trip/year. From Table 4-51 the flights/year and thus the number of vehicles required is:

$COTV_1$	2
$COTV_2$	5
$COTV_3$	3

All vehicles will be manufactured during the initial production phase. Table G-13 provides the following costs:

Development	$ 691.02 million
Production	15,627.21 million

NOTE 4.3

SDV has a 500 flight life. The following launch and vehicle requirements exist:

Startup period	892 flights ÷ 500	= 1.78
Steady State	(118 flts/yr × 30)÷500	= 7.08
Total required		8.86 ~ 9

In order to limit the length of time for the startup period, all SDV's will be manufactured during initial production and used for startup.

The majority of SDV hardware is reusable. The cargo pod shroud and external tank are expendable and 4432 shipsets of these elements will be required.

Costs were discussed in Section G.3.6. Totals are shown below.

Development:	Booster	$5311.50
	Cargo Pod	$1520.64
		$6832.14 million

Reusable Hardware Production (90% learning):

Booster: $(1.3)\ 364.72\ (9)^{.848} = 3055.63$

Cargo Pod: $(1.3)\ 103.44\ (9)^{.848} = \underline{\ 866.62\ }$

Total Initial Production $3922.25 million

Expendable Hardware Production (85% learning):

$$1.3\ (18)\ (4432)^{.766} = \$14,538.22 \text{ million}$$

Assume 900 sets are fabricated in initial production and the remaining 3532 are made during SPS production.

Initial Production: (900/4432) (14,538.22) = $2952.26 million

Replacement: (3532/4432) (14,538.22) = $11585.96 million

NOTE 4.4

The following LTV flights are required:

Startup		86
Steady State	365/yr × 30	= 10950
		11036

Using a 500 flight life, 11036/500 ≈ 22 vehicles are required, of which 7 are required for startup.

Costs were discussed in Section G. 3.10.

Development = $720.80 million

Production: $33.864\ (22)^{.848} = \$465.71$

Splitting production costs between initial production and replacement vehicles:

Initial Production: 7/22 (465.71) = $148.18 million

Replacement 15/22 (465.71) = $317.53 million

NOTE 5.1

A total of 10.4% of the SPS mass must be earth supplied. This amounts to 10232 metric tons per satellite. Approximately half of this material is complex avionics equipment and half is miscellaneous material which is not obtainable on the moon or not desirable to manufacture on the lunar surface. These materials are not well enough defined for detailed estimates so a general electronics type CER will be used to provide a single estimate for the entire amount: TFU cost $= .021\ W^{.667}$. This CER is from Shuttle System Payload Data Activity, GDC Report PDS-CO-015, Sept. 1974. First Unit Cost, including 10% for Program Level Costs, is:

$$\begin{aligned} \text{TFU} &= .021\ (22{,}561{,}560)^{.667} \times 1.1 \\ &= \$1686.229 \text{ million} \times 1.1 \\ &= \$1854.852 \text{ million} \end{aligned}$$

Production costs, using a 90% learning curve and 30% for Program Level Costs are:

$$\begin{aligned} \text{Production Cost} &= 1854.852\ (30)^{.848} \times 1.3 \\ &= \$43{,}137.149 \text{ million} \end{aligned}$$

NOTE 5.2

Production costs of earth rectenna are identical to the SPS earth baseline and are \$133.38 billion.

NOTE 5.3

Facility Maintenance is assumed to be 5% of facility construction costs/year.

$$5\%\ (.664 \text{ billion}) = \$.0332 \text{ billion/year}$$

$$\text{Total} = .0332 \times 30 \text{ years} = \$.996 \text{ billion}$$

Launch/Recovery Operations costs are included in the SDV operations cost on a per flight basis and will not be included here.

NOTE 6.1

All transportation charges for Earth Based Fab/Assy refer to the operations cost of the SDV and Space Shuttle. Vehicle Replacement costs for SDV's is calculated in Note 4.3. It consists of the expendable external tanks and shrouds for the cargo pad.

Total \$11.586 billion

NOTE 6.2

From Section G.3.6, cost per flight for spares and maintenance is \$8.571 million/flight

118 flights/year × 30 yrs × 8.571 million = \$30.341 billion

NOTE 6.3

From Section G.3.6, propellant costs are \$.901 million per flight for the SDV.

118 flights/year × 30 years × \$.901 million = \$3189.54 million

NOTE 6.4

From Section G.3.7, cost per Shuttle flight is \$20 million

53 flights/year × \$20 × 30 years = \$31,800 million

NOTE 7.1

Number of people assigned to lunar base during steady state operations is 400. Using a rate of $.120 million per manyear, cost is:

$400 \times .120 \times 30 = \$1440$ million

NOTE 8.1

From Section G.6.2, spares, fuel and maintenance costs total 4.5% of production. Referring to Note 1.1, .045 (6308) = $283.68 million/year

Total cost = $283.86 \times 30 = \$8515.8$ million

NOTE 8.2

From Section G.2.7, spares and maintenance costs for the large lunar base are 5.8% of production costs per year. From Note 1.2 production costs total $2353.798 million. Operations costs are 5.8% (2353.798) = $136.520 million/year. Total for 30 years = $30 \times 136.520 = \$4095.609$ million

NOTE 8.3

From Section G.5.2 annual cost is 4% (Production Cost). Using the production cost from Note 1.3, total cost is: $.04\ (10.71) \times 30$ years = $12.852 million.

NOTE 8.4

From Section G.5.3, annual operating cost is $ 87.059 million per year. Total for 30 years is $30 \times 87.059 = \$2611.77$ million

NOTE 8.5

Section G.5.5 provides annual operations costs for lunar based manufacturing equipment of $4.74 /year. For 30 years total cost is: 30 × 4.74 = $142.20 million.

NOTE 8.6

From Section G. 5.4 and Note 1.4, total operations costs are: .04 (301.945) (30 years) = $362.334 million.

NOTE 8.7

From Section G-1, annual operating cost of the lunar based tank depot is $1.099 million/year. Total cost is: 30 years × 1.099 = $32.97 million.

NOTE 9.1

Transportation charges for Lunar Based Fab/Assy refer to the LTV only. Note 4.4 provides the calculation for LTV replacement as $317.53 million.

NOTE 9.2

Discussion of LTV operations costs is contained in Section G. 3.10. Fleet size is 7 vehicles. Annual launch rate is 365 per year or 52 launches per vehicle per year.

Spares	.282 × 7 vehicles	= 1.974
Maintenance	.847 × 7 vehicles	= 5.929
Propellant	$.054/flt × 365	= 19.710
Annual Operations		$27.613 million/year

Total cost over 30 years:

Spares	1.974×30	= \$59.220 million
Maintenance	5.929×30	= \$177.870 million
Propellant	19.710×30	= \$591.300 million

NOTE 10.1

Number of people stationed in Space Base during steady state operation is 1165. Using a rate of \$.120 million/manyear, cost is:

$1165 \times .12 \times 30 = \$4194$ million

NOTE 11.1

From Section G.6.1, power station operations cost is 4% of production cost per year. Note 2.1 gives production cost of \$ 935.376 million. Total operations cost, then is:

$.04\ (935.376)\ (30\ \text{years}) = \$1122.451$ million

NOTE 11.2

Habitat Operations Cost as described in Section G.2.7 is 5.8% of hardware first unit cost per year. Lunar shielding costs are excluded since maintenance is nil. Costs were calculated as follows:

Habitat	Production Cost	Operations Cost @ 5.8%
LEO (75 person cap.)	$112.7\ (75)^{.67} = \$2033.363$ (Ref. Fig G-3)	\$117.935 million/year or \$3538.05 million total
GEO (SMF-1165 person cap.)	$67.4\ (1165)^{.67} = \$7640.086$ (Ref. Fig. G-8)	\$443.125 million/year or \$13,293.750 million total

Habitat	Production Cost	Operations Cost @ 5.8%
LLO (Temp.	$37.6\ (12)^{.67} = \$198.719$	\$11.526 million/year
Shelter 12 person)	(Ref Fig. G-5)	or \$345.780 million total
Total Space Habitat Operations		\$17,177.58 million

NOTE 11.3

Section G.5.5 defines manufacturing operations costs in space as \$116.07 per year. Total cost for 30 years is: $30 \times 116.07 = \$3482.10$ million.

NOTE 11.4

Section G.1 provides an annual operating cost of \$28.355 million for the Concept C propellant depots. For the 30 year production period total operating costs are: $30 \times 28.355 = \$850.650$.

NOTE 11.5

The LRU options require only one satellite construction facility in GEO, compared with two in the earth baseline. Figure F-8 gives annual costs to maintain the GEO construction facility as follows:

Facility	.159
Construction Equip	.247
Supply/Refurbishment	.191
	\$.597 billion/yr

Total Cost = $.597 \times 30 = \$17.910$ billion

NOTE 12.1

Transportation costs under Space Based Fab/Assy refer to the cost of operating the COTV's and POTV's. Vehicle replacement costs were previously calculated in the notes accompanying development costs:

POTV	$408.780 million	(Ref Note 4.1)
COTV	$ None	(Ref Note 4.2)

NOTE 12.2

COTV costs are discussed in Section G.3.9. Operations costs are as follows:

	Steady State flts/year	Propellant cost per flight	Cost/Year for Fleet Propellants	Spares	Maintenance
$COTV_1$	2	.117	.234	9.954	29.862
$COTV_2$	5	.376	1.880	95.095	285.285
$COTV_3$	3	.236	.708	55.317	165.951
Total Annual Cost			2.822	160.366	481.098

POTV

POTV operations costs consist of the costs of maintaining the POTV, the passenger module and crew module in operating condition. Fleet size is 11 vehicles and a total of 56 flights per year are required. Cost of the crew is contained in "Construction/Maintenance Crew." Sections G.3.5 and G.3.8 provide the following operations costs.

Spares: .13 + .24 + .244 = \$.614 million/year/vehicle

Maintenance: .39 + .72 + .731 = \$1.841 million/year/vehicle

Propellant: \$.015/flt

Total costs for the vehicle fleet are:

Spares: .614 × 11 = \$6.754 million/year

Maintenance: 1.841 × 11 = \$20.251 million/year

Propellant: .015 × 56 flts/yr = \$.84 million/year

Total operating costs for both vehicles over a 30 year period is shown below.

Spares: (160.366 + 6.754) 30 = \$ 5013.60

Maintenance: (481.098 + 20.251) 30 = \$15040.47

Propellant: (2.822 + .84) 30 = \$ 109.86

H.3 NOTES TO TABLE 5-7, page 5- 26 of Volume II

NOTE 1.1

Nuclear Brayton Power Station with a 960 MW capacity.

From Section G. 6. 2, costs are as follows:

Development:		1309
Production:	$27.8(960)^{.848}$	9397.467
		\$10,706.467 million

NOTE 1.2

Large Lunar Base Habitat with a 400 person capacity.

Using the relations from Section G.2.5, Fig. G-7, the cost of development and production is:

$$220.1(400)^{.5} + 42.5(400)^{.67} = \$6,755.798 \text{ million}$$

NOTE 1.3

Beneficiation Equipment weight is 60 tons, or 132,300 lbs.

From Section G.5.2 costs are as follows:

Development:	$1.546\ (132,300)^{.187}$	=	14.026
Production:	$.007\ (132,300)^{.667}$	=	18.246
			\$32.272 million

NOTE 1.4

Processing Facility weight is 16,980 metric tons (including 11500 tons for radiators).

Using the scaling relationships in Section G.5.3, the following costs are obtained:

Development:	$13.450(16,980)^{.5}$	=	1752.634
Production:	$5.756(16,980)^{.67}$	=	3928.068
			\$5680.702 million

NOTE 1.5

Liquefaction plant weight, excluding propellant storage tanks, is 836 metric tons. Using the scaling relationships in G. 5.4 the following costs are obtained:

Development:	$11.628(836)^{.5}$	=	336.208
Production:	$4.785(836)^{.67}$	=	434.270
			$770.478 million

NOTE 2.1

Photovoltaic Power Station in GEO with a 260 MW capacity.

From Section G. 6.1 cost is:

Development:	$24.04(260)^{.5}$	=	$387.633
Production:	$22.54(260)^{.67}$	=	935.376
			$1323.009 million

NOTE 2.2

LEO Modular Space Station with a 75 person crew size. Costs can be determined from Figure G-3.

Development	$665.1(75)^{.5}$	=	5759.935
Production	$112.7(75)^{.67}$	=	2033.363
			$7793.298 million

NOTE 2.3

GEO SMF Habitat with a 1165 person capacity. Costs can be determined from Figure G-8.

Development	$445.6(1165)^{.5} + .120(1165)$	=	15,349.062
Production	$67.4(1165)^{.67} + .012(1165)$	=	7,654.066
			$23.003.128 million

NOTE 2.4

LLO Temporary Shelter with a 12 person capacity. Costs can be determined from Figure G-5.

Development	$306.9(12)^{.5}$	1063.133
Production	$37.6(12)^{.67}$	198.719
		\$1261.852 million

NOTE 3.1

This is the cost to operate the vehicles during facility construction and includes spares, maintenance and propellants. Startup period is 3 years. During this time a gradual buildup of the vehicle fleet occurs. Assume an average maintenance period of 1 1/2 years, instead of 3 years to account for the buildup. Cost of spares and maintenance is calculated below.

SDV	(Ref. Note 6.2):	\$8.571 million/flt × 1269 flts	= \$10,876.599
COTV	(Ref. Note 12.2):	(42.332 + 126.996) 1.5 years	= 253.992
POTV	(Ref. Note 12.2):	(6.754 + 20.251) 1.5 years	= 40.508
LDR	(Ref. Note 9.2):	(6.853 + 20.552) 1.5 years	= 41.108
			\$11,212.207 million

User charges for the space shuttle are calculated at \$20 million per flight and 80 flights are required for startup. Total cost is 80 × 20 = 1600 million.

Total propellant requirements for startup were presented in Section 4.8 for space vehicles. SDV propellant requirements per flight are provided in Table G-9 . The following Table summarizes total propellant requirements for startup operations. All propellants for startup are assumed to be earth supplied and the cost per pound as shown in Table G-9 applies.

Total Propellant for Startup (Millions of Pounds)

User	LO_2	LH_2	C_3H_8	Al
SDV	9806.8	55.8	2453.0	
COTV	133.1	2.2	—	
POTV	19.4	2.8	—	
LDR	115.2	—	—	51.9
Total	10074.5	60.8	2453.0	51.9
$/lb	.021	.54	.37	.40
Total Cost (millions $)	211.565	32.832	907.610	20.760

Total transportation cost is the sum of the above elements: 11,212.207 + 1600 + 211.565 + 32.832 + 907.610 + 20.760 = $13,984.974 million.

NOTE 3.2

Initial Depot Propellant Supply is provided in Section 4.8. Costs are:

LO_2:	4308 tons × 2205 × $.021/lb	=	.199 million
LH_2:	684 tons × 2205 × $.54/lb	=	.814 million
Al:	270 tons × 2205 × $.40/lb	=	.238 million
			$1.251 million

NOTE 3.3

Construction/Maintenance Crews during facility activation average approximately 800 persons. At a cost of $.120 million/man year total cost for the 3 year period is: 3 × .120 × 800 = $288 million.

NOTE 3.4

Operations cost for the SDV launch/recovery facilities is $24.440 million/year (Ref. Note 5.3). Total for 3 years is $73.320 million.

NOTE 3.5

Annual cost of lunar based operations is \$744.167 million (1/30 of cost element C(2220)). For the 3 year activation period assume the average annual cost is half the steady state value, or \$372.083 million. Operations cost for the facility activation period is: $3 \times 372.083 = \$1116.249$ million.

NOTE 3.6

Annual cost of space based operations is \$745.267 million (1/30 of cost element C(2320)). For the 3 year activation period assume an average annual cost of half the steady state value or \$372.634 million. Operations cost for the facility activation period is: $3 \times 372.634 = \$1117.902$ million.

NOTE 4.1

Each POTV has a 50 flight life. Total flights are as follows:

Startup Operations		82
Steady State:	$POTV_1 - 18 \times 30$	540
	$POTV_2 - 38 \times 30$	1140
		1762 flights

Number of vehicles required: $1762/50 = 35.24 \approx 36$. Need 11 vehicles for startup operations for safety reasons. Remaining 25 will be manufactured as replacements during the SPS production phase. Costs can be determined from the relations in Section G.3.8.

Vehicle Development		\$380 million
Total Production: $29.24(36)^{.848}$	=	\$610.551 million
Initial Production: 11/36 (610.551)	=	\$186.557 million
Replacement Vehicles: 25/36 (610.551)	=	\$423.994 million

Each operating POTV requires passenger and crew modules. A total of 11 are required and they will be fabricated during initial production. No replacements are assumed. From Section G.3.5 costs are as follows:

Development: Passenger Module			287
Crew Module			524
			$811 million
Initial Production			
Passenger Module:	$13(11)^{.848}$	=	99.322
Crew Module:	$24(11)^{.848}$	=	183.363
			$282.685 million

In summary, total costs are as follows:

Vehicle Development	380
Module Development	811
	$1191 million
Initial Production: Vehicle	186.557
Modules	282.685
	$469.242 million
Replacement Vehicles:	$423.994 million

NOTE 4.2

Each SDV has a 500 flight life. The following launch and vehicle requirements exist:

Startup Period:		1269 flights
Steady State: 76×30	=	2280 flights
		3549 flights

A total of $3549/500 = 7.1 \approx 8$ vehicles will be required. Assume all will be manufactured during initial production and used for startup. Costs can be determined from Section G. 3.6.

Development:	Booster		\$5311.50
	Cargo Pod		\$1520.64
			\$6832.14 million

Reusable Hardware Production (90% learning):

Booster	$1.3\ (364.72)\ (8)^{.848}$	=	2765.179
Cargo Pod	$1.3\ (103.44)\ (8)^{.848}$	=	784.246
			\$3549.425 million

The cargo pod shroud and external tanks are expendable and a total of 3549 shipsets will be required.

Expendable Hardware Production (85% learning):

$1.3(18)\ (3549)^{.766}$ = \$12263.009 million

Assume 1300 shipsets will be fabricated during initial production and the remaining 2249 are made during the SPS production phase. Cost can be split as follows:

Initial Production:	(1300/3549) 12263.009	=	\$4491.945 million
Replacement:	(2249/3549) 12263.009	=	\$7771.064 million

NOTE 4.3

Each LDR has a 500 flight life. Total flights are as follows:

Startup Period	138
Steady State - 365 × 20	10950
	11088 flights

A total of 11088/500 = 22.176 ≈ 23 vehicles will be required over the program life. A fleet size of 7 is required for startup operations and for steady state. Initial production will be 7 vehicles and 16 replacements will be manufactured during the SPS production phase. In addition to the basic vehicle, 7 dedicated crew modules are required. These will be fabricated during initial production also. From Section G. 3.12 costs are as follows:

Development:			\$5203.78 million
Vehicle Production:	$88.657(23)^{.848}$	=	\$1266.072 million
Initial Production:	7/23 (1266.072)	=	\$ 385.326 million
Replacement:	16/23(1266.072)	=	\$ 880.746 million
Crew Module Initial Production:	$28.8(7)^{.848}$	=	\$149.981 million

NOTE 5.1

A total of 10.4% of the SPS mass must be earth supplied. This amounts to 10232 metric tons per satellite. Approximately half of this material is complex avionics equipment and half is miscellaneous material which is not obtainable on the moon or not desirable to manufacture on the lunar surface. These materials are not well enough defined for detailed estimates so a general electronics type CER will be used to provide a single estimate for the entire amount: TFU cost = $.021\ W^{.667}$. This CER is from Shuttle System Payload Data Activity, GDC Report PDS-CO-015, Sept. 1974. First Unit Cost, including 10% for Program Level Costs, is:

$$\text{TFU} = .021\ (22,561,560)^{.667} \times 1.1$$

$$= \$1686.229 \text{ million} \times 1.1$$

$$= \$1854.852 \text{ million}$$

Production costs, using a 90% learning curve and 30% for Program Level Costs are:

$$\text{Production Cost} = 1854.852\ (30)^{.848} \times 1.3$$

$$= \$43,137.149 \text{ million}$$

NOTE 5.2

Production costs of earth rectenna are identical to the SPS earth baseline and are \$133.38 billion.

NOTE 5.3

From Table G-30in Section G.4 annual maintenance cost of the Launch/Recovery Facilities is \$24.440 million. Total for 30 years is: 30 × 24.440 = \$733.200 million.

Launch/Recovery operations costs are included in the SDV operations cost on a per flight basis and will not be included here.

NOTE 6.1

All transportation charges for Earth Based Fab/Assy refer to the operations cost of the SDV and the cost of using the Space Shuttle. Vehicle replacement costs for SDV's is calculated in Note 4.2. The replacement hardware consists entirely of expendable hardware and the total cost is $7771.064 million.

NOTE 6.2

From Section G.3.6, cost per flight for spares and maintenance is $8.571 million per flight.

76 flts/yr × 30 yrs × 8.571 = $19541.88 million

NOTE 6.3

From Section G.3.6, propellant costs are $.901 million per SDV flight.

76 flts/yr × 30 yrs × .901 = $ 2054.280 million

NOTE 6.4

From Section G.3.7, cost per Shuttle flight is $20 million.

53 flights/yr × 30 yrs x 20 = $ 31,800 million

NOTE 7.1

During steady state operations there are 400 people stationed at the lunar base. At a rate of $.120 million per man year, total cost is:

400 × .120 × 30 yrs = $1440 million

NOTE 8.1

Spares, fuel and maintenance costs of the nuclear power station are 4.5% of production cost per year (Ref. Section G.6.2). From Note 1.1, production cost is:

$.045 \times 9397.467 \times 30$ yrs = \$12,686.580 million

NOTE 8.2

Annual habitat operations cost, from Section G. 2.7, is 5.8% of production cost. From Note 1.2 production cost is: $42.5(400)^{.67}$ = \$2353.798 million. Total operations cost is:

$.058 \times 2353.798 \times 30$ years = \$ 4,095.609 million

NOTE 8.3

From Section G. 5.2, beneficiation equipment annual cost is 4% of production. Production cost is \$18.246 million (Ref. Note 1.3). Total operating cost is:

$.04(18.246) \times 30$ years = \$21.895 million

NOTE 8.4

From Section G.5.3 and Note 1.4, annual operating cost for the processing facility is:

$.04(3928.068) \times 30$ yrs = \$4713.682 million

NOTE 8.5

Annual operations cost for lunar manufacturing are \$4.74 million/yr (Ref. Table G-41). For 30 years the cost is \$142.20 million.

NOTE 8.6

From Section G.5.4 and Note 1.5, lunar liquefaction operations cost is:

$.04(434.270) \times 30$ yrs = \$521.124 million

NOTE 8.7

Annual operating cost of the lunar based propellant depot is \$2.751 million/year (Ref. Sec. G.1). Total cost for 30 years is \$82.53 million.

NOTE 9.1

Transportation charges for Lunar Based Fab/Assy are for the Lunar Derived Rocket (LDR). Replacement vehicle costs were derived in Note 4.3 and total $880.746 million.

NOTE 9.2

In Section G.3.12 LDR annual maintenance cost was found to be $2.936 million per vehicle and spares $.979 million per vehicle. Propellant costs are included in the lunar propellant production facilities costs. LDR operations cost for a 30 year period and for a fleet size of 7 are as follows:

Maintenance:	2.936(7)(3)	=	$616.560 million
Spares:	.979(7)(3)	=	$205.590 million

NOTE 10.1

Number of people stationed in space during steady state operation is 1165. At a rate of $.120 million per man year, cost is:

$1165 \times .12 \times 30$ = $4194 million

NOTE 11.1

Using the data in Note 2.1 and Section G. 6.1, GEO power station operations cost over 30 years is:

$.04(935.376) \times 30$ = $1122.451 million

NOTE 11.2

Space Habitat operations costs, from Section G. 2.7, is 5.8% of hardware cost per year (excluding the cost of lunar shielding). Production costs, adjusted to remove shielding costs, are shown below along with the 30 year operations costs.

Habitat	Production Cost	30 yr Operations Cost
LEO	\$2033.363 (Ref. Note 2.2)	3538.050
GEO	$67.4(1165)^{.67} = \$7640.086$ (Ref. Fig. G-9)	13,293.750
LLO	$37.6(12)^{.67} = \$198.719$ (Ref. Note 2.4)	345.771
Total Space Habitat Operations		\$17,177.571 million

NOTE 11.3

From TableG-41 in Section 5.2.5.5, annual operations cost of the space based manufacturing facility is \$116.07 million/year. Total cost for 30 years is:

30×116.07 = \$3482.10 million

NOTE 11.4

Section G.1 provides an annual operating cost of \$19.203 million for the space based depots. Total for 30 years is:

\$19.203(30) = \$ 576.090 million

NOTE 11.5

The LRU options require only one satellite construction facility in GEO, compared with two in the earth baseline. Figure F-8 gives annual costs to maintain the GEO construction facility as follows:

Facility	.159
Construction Equip	.247
Supply/Refurbishment	.191
	\$.597 billion/yr

Total Cost = $.597 \times 30$ = \$17.910 billion

<u>NOTE 12.1</u>

Transportation costs under "Space Based Fab/Assy" refer to the cost of operating the COTV and POTV fleets. Vehicle replacement cost was calculated in Notes 4.1 for the POTV as $423.994 million. No replacements are required for the COTV.

<u>NOTE 12.2</u>

COTV operating costs are shown in Section G.3.9, Table G-16.

Spares:	$42.332 million/yr		
Maintenance:	$126.996 million/yr		
Propellant:	$.252 million/flt × 8 flts/yr	=	$2.016 million/yr

POTV operations costs are the costs of maintaining the POTV, passenger and crew modules in operating condition. Fleet size is 11 vehicles which fly 56 missions per year: Sections G.3.5 and G.3.8 provide the following operations costs:

Spares:	.13 + .24 + .244 = $.614 million/year/vehicle
Maintenance:	.39 + .72 + .731 = $1841 million/year/vehicle
Propellant:	$.012/flt

Total annual costs for the 11 vehicle POTV fleet are:

Spares:	.614 × 11 = $6.754 million/yr
Maintenance:	1.841 × 11 = $20.251 million/yr
Propellant	.012 × 56 flts/yr = $.672 million/yr

Total operating costs for the COTV and POTV over a 30 year period are as follows:

Maintenance:	(126.996 + 20.251) 30	=	$4417.41 million
Spares	(42.332 + 6.754) 30	=	$1472.58 million
Propellant	(2.016 + .672) 30	=	$ 2.688 million

APPENDIX I

Supplementary data for Section 5.3.3, Cost Reconciliation (page 5-31) and Section 5.4.3, Threshold Sensitivity to Manufacturing Costs (page 5-58).

I.1 Cost Reconciliation Tables I-1 through I-5

I.2 Sensitivity Analysis Tables I-6 through I-8

Table I-1. Categorization of RDT&E & Facility Costs for Reconciliation (billions $).

	Concept B	Concept C	Concept D	Concept Earth Baseline
MANUFACTURING				
Earth Based				
SPS Hardware	6.270	6.270	6.270	6.270
SPS H. W. Facilities	—	—	—	10.366
	6.27	6.27	6.27	16.636
Lunar Based	5.388	19.525	24.358	—
Space Based				
Construction System	20.741	20.741	20.741	20.741
Facility Activation	9.293	13.390	16.581	
Equip/Facilities	52.756	41.248	41.019	
	82.790	75.379	78.341	20.741
TRANSPORTATION				
Earth Based				
Launch/Recovery Facilities	.453	.664	.489	2.8
Propellant Production Facilities	.885	1.084	.885	3.5
HLLV				17.826
PLV				3.314
POTV				2.369
COTV				3.400
SDV	11.090	13.706	14.873	
	12.428	15.454	16.247	33.209
Lunar Based				
PLTV	.443	—	—	—
Mass Driver	1.500	—	—	—
LTV	—	.869	—	—
LDR	—	—	5.739	—
	1.943	.869	5.739	—
Space Based				
POTV	1.667	1.661	1.660	—
COTV	9.342	16.318	13.145	—
Mass Catcher	1.928	—	—	
	12.937	17.979	14.805	—

Table I-2. Categorization of Production Costs for Reconciliation (billions $).

	CONCEPT B		C		D		Earth Baseline	
MANUFACTURING								
Earth Total		176.517		176.517		176.517		401.391
Satellite	43.137		43.137		43.137		268.011	
Earth Rectenna	133.380		133.380		133.380		133.380	
Lunar Total		2.539		17.274		23.765		
Crew	.173		1.440		1.440			
Fac/Equip Ops	2.366		15.834		22.325			
Space Total		53.196		44.737		44.462		36.480
Crew	5.044		4.194		4.194		—	
Fac/Equip Ops	—		—		—			
Mfg System	30.242		22.633		22.358			
Constr System	17.910		17.910		17.910		36.480	
TRANSPORTATION		66.073		99.632		69.921		218.605
L/R Facility Ops	.680		.996		.733		4.2	
Earth Based	52.265		76.917		61.167		214.405	
Lunar Based	.405		1.146		1.704			
Space Based	12.723		20.573		6.317			

Table I-3. Comparison of Concept B With Earth Baseline.

Category	Earth Baseline			LRU Concept B			Difference		
	NR	R	T	NR	R	T			
Transportation			251.8			93.3			158.5
Earth Based	33.2	218.6	251.8	12.4	53.0	65.4		186.4	
Lunar Based	-	-	-	1.9	.4	2.3		- 2.3	
Space Based	-	-	-	12.9	12.7	25.6		-25.6	
Manufacturing									148.4
Earth Based			418.0			182.8		235.2	
Satellite	16.6	268.0	284.6	6.3	43.1	49.4	235.2		
Rectenna	-	133.4	133.4	-	133.4	133.4	0		
Lunar Based	-	-	-	5.4	2.6	8.0		- 8.0	
Space Based			57.2			136.0		-78.8	
Construction System	20.7	36.5	57.2	20.7	17.9	38.6	18.6		
Manufacturing System	-	-	-	62.1	35.3	97.4	-97.4		

Notes: 1. Costs are in billions of 1977 dollars

2. NR = Non-Recurring Development and Facility Cost Amortization

R = Recurring Production Costs

T = Total costs, excluding the Operations Phase which is the same for all Concept.

Table I-4. Comparison of Concept C with Earth Baseline.

Category	Earth Baseline NR	Earth Baseline R	Earth Baseline T	LRU Concept C NR	LRU Concept C R	LRU Concept C T	Differences		
Transportation			251.8			134.0			117.8
Earth Based	33.2	218.6	251.8	15.5	77.9	93.4		158.4	
Lunar Based	–	–	–	.9	1.1	2.0		- 2.0	
Space Based	–	–	–	18.0	20.6	38.6		-38.6	
Manufacturing									135.6
Earth Based			418.0			182.8		235.2	
Satellite	16.6	268.0	284.6	6.3	43.1	49.4	235.2		
Rectenna	–	133.4	133.4	–	133.4	133.4	0		
Lunar Based	–	–	–	19.5	17.3	36.8		-36.8	
Space Based			57.2			12.0		-62.8	
Construction System	20.7	36.5	57.2	20.7	17.9	38.6	18.6		
Manufacturing System	–	–	–	54.6	26.8	81.4	-81.4		

Note: See Notes for Table I-3.

Table I-5. Comparison of Concept D with Earth Baseline.

Category	Earth Baseline			LRU Concept D			Difference		
	NR	R	T	NR	R	T			
Transportation			251.8			106.6			145.2
Earth Based	33.2	218.6	251.8	16.2	61.9	78.1		173.7	
Lunar Based	-	-	-	5.7	1.7	7.4		- 7.4	
Space Based	-	-	-	14.8	6.3	21.1		-21.1	
Manufacturing									121.4
Earth Based			418.0			182.8		235.2	
Satellite	16.6	268.0	284.6	6.3	43.1	49.4	235.2		
Rectenna	-	133.4	133.4	-	133.4	133.4	0		
Lunar Based	-	-	-	24.4	23.8	48.2		-48.2	
Space Based			57.2			122.8		-65.6	
Construction System	20.7	36.5	57.2	20.7	17.9	38.6	18.6		
Manufacturing System	-	-	-	57.6	26.6	84.2	-84.2		

Note: See Notes for Table I-3.

Table I-6. Allocation of Manufacturing Cost Differences to LRU Concepts for Sensitivity Analysis.

Concept	Manufacturing Location	Total [1]	RDT&E (WBS 1320 & 1330)	Production: Total	Production: Constr. Crew (WBS 2210 & 2310)	Production: Fac/Equip Ops (WBS 2220 & 2320)
B	Lunar	$\frac{8}{105.4}(129.8) = 9.85$	$\frac{5.4}{8}(9.85) = 6.65$	$\frac{2.6}{8}(9.85) = 3.20$	$\frac{.173}{2.539}(3.20) = .22$	$\frac{2.366}{2.539}(3.20) = 2.98$
	Space	$\frac{97.4}{105.4}(129.8) = 119.95$	$\frac{62.1}{97.4}(119.95) = 76.48$	$\frac{35.3}{97.4}(119.95) = 43.47$	$\frac{5.044}{53.196}(43.47) = 4.12$	$\frac{48.152}{53.196}(43.47) = 39.35$
C	Lunar	$\frac{36.8}{118.2}(117) = 36.43$	$\frac{19.5}{36.8}(36.43) = 19.30$	$\frac{17.3}{36.8}(36.43) = 17.13$	$\frac{1.44}{17.274}(17.13) = 1.43$	$\frac{15.834}{17.274}(17.13) = 15.70$
	Space	$\frac{81.4}{118.2}(117) = 80.57$	$\frac{54.6}{81.4}(80.57) = 54.04$	$\frac{26.8}{81.4}(80.57) = 26.53$	$\frac{4.194}{44.737}(26.53) = 2.49$	$\frac{40.543}{44.737}(26.53) = 24.04$
D	Lunar	$\frac{48.2}{132.4}(102.8) = 37.42$	$\frac{24.4}{48.2}(37.42) = 18.94$	$\frac{23.8}{48.2}(37.42) = 18.48$	$\frac{1.44}{23.765}(18.48) = 1.12$	$\frac{22.325}{23.765}(18.48) = 17.36$
	Space	$\frac{84.2}{132.4}(102.8) = 65.38$	$\frac{57.6}{84.2}(65.38) = 44.73$	$\frac{26.6}{84.2}(65.38) = 20.65$	$\frac{4.194}{44.462}(20.65) = 1.95$	$\frac{40.268}{44.462}(20.65) = 18.70$

NOTES:

1. Total amount of manufacturing to be allocated obtained from Tables I-3 (Concept B, \$129.8 billion), I-4 (Concept C, \$117 billion), I-5 (Concept D, \$102.8 billion). Amounts exclude costs related to construction system.
2. Allocations for Total RDT&E and Total Production based on ratios from Tables I-3, I-4 and I-5. Allocation of Total Production between Construction Crew and Facility/Equipment Operations based on cost ratios from Life Cycle Cost Tables 5-5, 5-6 and 5-7.

Table I-7. Program Phase Cost Uncertainty Ranges for Sensitivity Analysis (billions of 1977 dollars).

Cost Element	Uncertainty Range (± %)	Concept B Nominal Cost	Concept B ± 3σ	Concept B σ^2	Concept C Nominal Cost	Concept C ± 3σ	Concept C σ^2	Concept D Nominal Cost	Concept D ±3σ	Concept D σ^2	Earth Baseline Nominal Cost	Earth Baseline ±3σ	Earth Baseline σ^2
RDT&E													
SPS Hardware	65.0	6.270	4.076	1.846	6.270	4.076	1.846	6.270	4.076	1.846	6.270	4.076	1.846
Construction System	80.0	20.741	16.593	30.591	20.741	16.593	30.591	20.741	16.593	30.591	20.741	16.593	30.591
Facilities & Equipment													
Earth Based	65.0	1.338	.870	.084	1.748	1.136	.143	1.374	.893	.089	16.666	10.833	13.039
Lunar Based	132.2	11.988	15.848	27.907	38.825	51.327	292.714	43.298	57.240	364.046	—		
Space Based	132.2	129.236	170.850	3243.302	95.288	125.971	1763.181	85.749	113.360	1427.832	—		
Facility Activation	117.2	9.293	10.891	13.180	13.390	15.693	27.364	16.581	19.433	41.960	—		
Transportation													
HLLV	65.0	—	—	—	—	—	—	—			17.826	11.587	14.917
PLV	65.0	—	—	—	—	—	—	—			3.314	2.154	.516
POTV	95.0	1.667	1.584	.279	1.661	1.578	.277	1.660	1.577	.276	2.369	2.251	.563
COTV	140.0	9.342	13.079	19.006	16.318	22.845	57.989	13.145	18.403	37.630	3.400	4.760	2.518
SDV	95.0	11.090	10.536	12.333	13.706	13.021	18.838	14.873	14.129	22.182	—	—	—
LTV/PLTV	170.0	.443	.753	.063	.869	1.477	.242	—	—	—	—	—	—
LDR	170.0	—	—	—	—	—	—	5.739	9.756	10.576	—	—	—
Mass Driver	185.0	1.500	2.775	.856	—	—	—	—	—	—	—	—	—
Mass Catcher	185.0	1.928	3.567	1.414	—	—	—	—	—	—	—	—	—
Nominal Cost		204.886			208.816			209.430			70.586		
Sum of the Variances				3350.861			2193.185			1937.028			63.990
RDT&E Uncertainty Range (± 3σ)			173.660			140.494			132.035			23.998	
Production													
Earth Based Fab/Assy													
SPS System Hardware	32.5	176.517	57.368	365.677	176.517	57.368	365.677	176.517	57.368	365.677	401.391	130.452	1890.860
Launch/Recovery Facilities Ops	32.5	.680	.221	.005	.996	.324	.012	.733	.238	.006	4.200	1.365	.207
Transportation	51.1	52.265	26.707	79.254	76.917	39.305	171.650	61.167	31.256	108.551	214.405	109.561	1333.734
Lunar Based Fab/Assy													
Construction/Maintenance Crew	47.5	.393	.187	.004	2.87	1.363	.206	2.56	1.216	.164	—	—	—
Facility & Equipment Ops	62.5	5.346	3.341	1.240	31.534	19.709	43.161	39.685	24.803	68.355	—	—	—
Transportation	55.0	.405	.223	.006	1.146	.630	.044	1.704	.937	.098	—	—	—
Space Based Fab/Assy													
Construction/Maintenance Crew	47.5	9.164	4.353	2.105	6.684	3.175	1.120	6.144	2.918	.946	—		—
Facility & Equipment Ops	62.5	87.502	54.689	332.318	64.583	40.364	181.028	58.968	36.855	150.921	36.480	22.800	57.760
Transportation	55.0	12.723	6.998	5.441	20.573	14.226	22.486	6.317	3.474	1.341	—	—	—
Nominal Cost		344.995			381.820			353.795			656.476		
Sum of the Variances				786.050			785.384			696.059			3282.561
Production Uncertainty Range (± 3σ)			84.110			84.074			79.149			171.881	
Operations													
Satellite	62.5	124.629	77.893	674.147	124.629	77.893	674.147	124.629	77.893	674.147	124.629	77.893	674.147
Earth Rectenna	62.5	62.022	38.764	166.959	62.022	38.764	166.959	62.022	38.764	166.959	62.022	38.764	166.959
Nominal Cost		186.651			186.651			186.651			186.651		
Sum of the Variances				841.106			841.106			841.106			841.106
Operations Uncertainty Range (± 3σ)			87.005			87.005			87.005			87.005	

Table I-8. Theoretical First Unit Costs - Sensitivity Analysis
(billions of 1977 $)

Concept	Nominal Production Cost	TFU
B	344.995	19.285
C	381.820	21.343
D	353.795	19.777

Notes:

1. TFU Cost = $\dfrac{\text{Total Production Cost}}{30^{.848}}$

   Assumes 90% learning

2. Production Costs are from Table I-7.

APPENDIX J

Task 5.5 supplementary data, identifying technology development tests required for major Earth Baseline and LRU Concept B system elements.

Appendix J consists of 4 Tables

Table J-1. Transportation System Elements.

TECHNOLOGY DEVELOPMENT PHASE - TERRESTRIAL AND SPACE TESTS	
EARTH BASELINE	LRU PECULIAR
SDV - Liquid flyback booster with modified orbiter plus external tank. Required for delivery of demo satellite, and subsequently used as a personnel launch vehicle. • New booster engine development • Booster structure/aerodynamics • Cargo pod and recoverable SSME propulsion module (Shuttle derived) • Flight test program • Expanded ground support operations (Utilizes Apollo and Shuttle program technology)	SDV - Similar to earth baseline except its operational use for cargo delivery during commercial program may influence (increase) the required payload capability. Other possible design impacts include: • Reduced turnaround/refurb schedule • Glide return and horizontal landing of cargo propulsion module rather than ballistic (to reduce turnaround).
POTV - Two stage LH_2/LO_2 vehicle. New configuration based on mostly existing technology. • New engine development (ASE or RL-10 derivative). • Orbital propellant transfer • Flight test program • On-orbit maintenance	POTV - Similar to earth baseline except only a single stage vehicle is required since propellant loading is feasible at each destination. Early coordination should permit use of one POTV stage for the LRU scenario without extensive modification.
COTV - Ion electric propulsion system powered by photovoltaic array. Uses argon propellant. • Large ion-engine performance • Ion engine life/maintenance • Engine cluster performance • Photovoltaic array & power system* • Structure & space construction* *Utilizes technology developed for satellite power and construction. See Table J-2.	COTV - Similar to earth baseline except oxygen propellant is substituted for argon, and several COTV configurations are required, necessitating a modular design approach. Supplementary development activities are mostly propellant related. • Ion-engine performance with oxygen • Engine life/maintenance with oxygen • Engine cluster performance (modular) • Array/structure configuration (modular) • Flight test program

Table J-1. Transportation System Elements (contd).

TECHNOLOGY DEVELOPMENT PHASE - TERRESTRIAL AND SPACE TESTS	
EARTH BASELINE	LRU PECULIAR
HLLV - Two stage fully reusable flyback vehicle with 450T payload capability to LEO. • New engine developments (may use or adapt SSME and/or SDV booster engines). • Vehicle structure/aerodynamics (two dissimilar stages). • Flight test program. • Expanded ground support operations.	HLLV not required; SDV should be suitable for delivery of required earth equipment, supplies, and personnel for start-up and steady state operations.
No corresponding vehicle requirement.	LTV - Similar to LRU single stage POTV with following changes: • Lunar landing structural kit (legs). • Landing avionics. • Throttable engine (for landing). • Flight test program/maintenance.
No corresponding equipment requirement.	MASS DRIVER CATAPULT - Electromagnetic accelerator constructed on lunar surface to catapult material into space. • "Lunar concrete" foundations. • Accelerator/return track structure • Drive coils and sequence control • Bucket conditioning and loading • Terminal guidance stations • Automatic monitoring & control system • Site preparation (similar to mining) • Equipment life/maintenance
No corresponding equipment requirement.	MASS CATCHER - Device for receiving, accumulating, and transporting lunar material from L_2 to SMF. • Catcher structure for arresting and retaining incoming material stream.

Table J-1. Transportation System Elements(contd).

TECHNOLOGY DEVELOPMENT PHASE - TERRESTRIAL AND SPACE TESTS	
EARTH BASELINE	LRU PECULIAR
	• High thrust chemical propulsion system (similar to POTV or LTV). • Low thrust ion propulsion system (similar to LRU COTV system with modular thrusters and oxygen propellant). • Power supply, probably nuclear, suitably protected from potential damage which could be caused by incoming material stream. • Guidance and control system for automated operation, maneuvering, stationkeeping, orbital transfer, and SMF rendezvous. • Vehicle life/maintenance.

Table J-2. Satellite System Elements.

TECHNOLOGY DEVELOPMENT PHASE - TERRESTRIAL AND SPACE TESTS	
EARTH BASELINE*	LRU PECULIAR
RECTENNA - 5 GW ground receiving antenna considerations: Collection efficiency RF-DC conversion efficiency Factors influencing rectenna size Low-cost rectenna elements Sensitivity to beam power density and grid loads Pilot beam interfaces Maintenance	RECTENNA - Identical to earth baseline. All materials used for antenna construction are obtained from earth. LRU is not expected to affect overall power transmission parameters such as frequency, power density, power distribution, antenna aperatures, etc.
PHOTOVOLTAIC ENERGY CONVERSION - Large solar cell array, silicon or GaAlAs cells with glass substrate/covers. • Solar cell blankets: Thermal cycling Electron/proton and ultraviolet radiation effects Fabrication techniques Annealing techniques and performance • Solar concentrators (reflectors): (if required) Radiation effects Micrometeoroid effects Application of vapor deposited coatings in orbit • Electrical and mechanical performance of very large arrays • High voltage/plasma interactions	PHOTOVOLTAIC ENERGY CONVERSION - Similar to earth baseline except max substitution of lunar materials precludes consideration of GaAlAs cells. If a compromise LRU/earth baseline design is not possible, additional development and testing of cells constructed primarily with lunar materials will be required, i.e., silicon cells with SiO_2 covers and substrate. Recent analyses have shown concentrators to be ineffective with silicon solar cells; therefore, a LRU compatible photovoltaic array configuration will probably not include reflectors. If reflectors are needed, sodium coated aluminum foil is a possible LRU compatible candidate for reflector construction.

*Obtained from "Solar Power Satellite Concept Evaluation," Vol II Detailed Report, July 1977, NASA-Johnson Space Center.

Table J-2. Satellite Systems Elements (contd).

TECHNOLOGY DEVELOPMENT PHASE - TERRESTRIAL AND SPACE TESTS	
EARTH BASELINE *	LRU PECULIAR
STRUCTURES - Photovoltaic array support structure and primary/secondary structures for the microwave power transmission system antennae. • Structural Systems Solar collector structure/attitude control interactions Antenna stiffness/pointing accuracy/attitude control interactions Antenna subarray chassis/thermal control Structural elements for space construction Numerical characterization of SPS structural performance Similitude modeling for subscale testing Transient response of structure during eclipse • Material Considerations Availability of graphite for SPS construction Graphite composite lifetime Graphite composite cables Tension cable lifetime Joining techniques and properties Electrostatic charging phenomena	STRUCTURES - Significant design changes will be necessitated by substitution of lunar materials. These substitutions will require substantial supplementary development testing in both the systems and materials category. Potential lunar material substitutions will be chiefly influenced by dimensional stability requirements during thermal cycling. A lunar ceramic material (foamed glass), possibly containing high strength fibers (from either the moon or earth) may be suitable for structural applications requiring a low coefficient of thermal expansion. Other substitute structural materials include alloys of lunar derived aluminum, titanium, or steel, for applications where dimensional stability requirements are less critical.
POWER DISTRIBUTION Thin sheet conductors Power bus insulation Power switching System verification	POWER DISTRIBUTION - Design changes and supplementary development tests should be minimal. Primary power busses are manufactured of lunar rather than earth aluminum. Cable conductors will substitute aluminum insulated with woven glass for plastic coated copper. Insulators will be lunar ceramic rather than plastic composite.

Table J-2. Satellite System Elements (contd).

TECHNOLOGY DEVELOPMENT PHASE - TERRESTRIAL AND SPACE TESTS

EARTH BASELINE *

POWER TRANSMISSION - Phased Array Microwave Antenna

- Microwave System
  - Transmission frequency
  - Ionosphere power density limits
  - Heat dissipation from microwave generators
  - Transmitting antenna construction and operation
  - Interfaces with transmitting antenna
  - Microwave system-level problems
  - Microwave effects on other areas
- Microwave Generation (Klystrons)
  - Efficiency
  - Reliability
  - Low noise
  - Low weight
  - Stability
- Antenna Subarrays
  - Slotted waveguide antenna designs
  - Efficiency
  - Power level effects
  - Waveguide materials and fabrication techniques
- Thermal Control
  - Microwave generator thermal design
  - MPTS thermal control
  - Thermal design of rotary joint
  - Thermal control of power distribution system
- Phase Control

LRU PECULIAR

POWER TRANSMISSION - Design changes will be necessitated by substitution of lunar materials. The following are a preliminary indication of possible LRU development tasks associated with these substitutions.

- Microwave system - minor, if any, system level supplementary testing should be needed.
- Microwave generation - substitution of aluminum for copper (and possibly CRES) parts in klystron will require substantial additional development and testing to demonstrate equivalent performance.
- Antenna subarrays - substitution of foamed glass (or a lunar ceramic material) for graphite composite waveguides will require substantial additional development and testing.
- Thermal control - the following substitutions should be considered
  - Aluminum rather than copper radiators
  - Alloy steel or aluminum rather than CRES heat pipes. An alternative transfer fluid to replace mercury should also be considered to alleviate material compatibility problems.
- Phase Control - identical to earth baseline. Consists of high technology electronic assemblies manufactured on earth.

Table J-2. Satellite System Elements (contd).

TECHNOLOGY DEVELOPMENT PHASE - TERRESTRIAL AND SPACE TESTS	
EARTH BASELINE*	LRU PECULIAR
POWER CONDITIONING • DC-DC Converters • Converter thermal control	POWER CONDITIONING - Design changes will be necessitated by substitution of lunar materials. The following are a preliminary indication of possible LRU development tasks associated with these substitutions. • DC-DC Converters - substitution of aluminum wire for copper windings in transformer coil, and manufacture of transformer core from lunar materials will require supplementary development testing to demonstrate equivalent performance. • Thermal Control - use of lunar rather than earth supplied aluminum radiators should not require supplementary development.
OTHER SPS SYSTEMS • Communications and Instrumentation • Stabilization and Control • Antenna Pointing Control • Propulsion and Reaction Control MPD arc-jet thruster 100-cm ion thruster • Rotary Joint Slip rings and brushes	OTHER SPS SYSTEMS - These systems primarily consist of high technology components of relatively low mass. With minor exceptions, lunar material substitutions are probably not worth considering. All these components will be obtained from earth, therefore no supplementary development is required.

Table J-3. Manufacturing System Elements.

TECHNOLOGY DEVELOPMENT PHASE - TERRESTRIAL AND SPACE TESTS	
EARTH BASELINE	LRU PECULIAR
No corresponding space facility.	LUNAR MINING - Development of equipment to excavate, mechanically separate and transport lunar soil from the strip mine to processing/logistics base. • Skip loader (performance) • Screening, magnetic, and electrostatic beneficiation equipment (performance) • Hauler (performance) • Equipment Life/Maintenance
No corresponding space facility.	MATERIAL PROCESSING - Electro-chemical reduction of beneficiated lunar soil into its constituent elements, i.e., aluminum, iron, oxygen, silicon & others. • Chemical processing equipment • Electrolysis equipment • Process chemical recovery equipment • Peripheral equipment
No corresponding space facility.	MATERIAL REFINING - Development of processing equipment to refine silicon from metallurgical grade to PPB level required for solar cell production.
No corresponding space facility.	PROPELLANT PRODUCTION FACILITY - Development of equipment needed to liquefy lunar derived oxygen and store LO_2 for use as transportation vehicle propellant. • Liquefaction equipment • Large radiator construction • Storage & transfer (pumping) equipment

Table J-3. Manufacturing System Elements (contd).

TECHNOLOGY DEVELOPMENT PHASE - TERRESTRIAL AND SPACE TESTS

EARTH BASELINE	LRU PECULIAR
No corresponding space facility.	SPACE MANUFACTURING FACILITY - Development of equipment required to manufacture SPS components and subassemblies from processed lunar materials. • Stock Manufacturing – Aluminum sheet and wire – Iron and steel sheet – Aluminum and sendust castings – Glass filaments • Parts Manufacturing – Aluminum fittings and housings – Foamed ceramic struts and waveguides – Steel heat pipes and glass insulation • Component Assembly – DC-DC converters and klystrons – DC-DC conv. and klystron radiators – Structural members and waveguide modules • Solar cell panel manufacturing – Glass covers and substrate – Silicon solar cells – Solar cell module assembly
Construction facility configurations and services are scenario dependent. • If SPS modules are fabricated in LEO, transferred via self-powered mode, and assembled in GEO, the SPS orbital construction facility will be located in LEO. • If SPS is fabricated in GEO, the LEO construction facility consists of assembly fixtures to manufacture COTV's for bulk material transfer to GEO. The SPS orbital construction facility is located in GEO.	

Table J-3. Manufacturing System Elements (contd).

TECHNOLOGY DEVELOPMENT PHASE - TERRESTRIAL AND SPACE TESTS

EARTH BASELINE

Using the second case, which is more compatible with the LRU option, as an example:

LEO COTV CONSTRUCTION FACILITY - Large structural framework and equipment suitable for manufacture of an ion electric COTV structure, deployment of solar array blankets and attachment/integration of subsystems.

Technology Development Requirements:

- Automatic fabrication of elemental truss
- Assembly of elemental trusses into long truss
- Deployment and attachment of solar cell blankets
- Space installation of power distribution cables
- Integrity verification of space-fabricated structures
- Assembly of jigs and fixtures for orbital construction

GEO SPS CONSTRUCTION FACILITY - Large structural framework and equipment suitable for manufacture of a SPS. The solar array and microwave antennae are fabricated and joined using this fixture. Many development items duplicate those needed for the COTV construction facility.

- Automatic fabrication of elemental truss
- Assembly of elemental trusses into long truss
- Large space radiator construction
- Deployment and attachment of solar cell balnkets
- Space installation of power distribution cables
- Integrity verification of space-fabricated structures
- Assembly of jigs and fixtures for orbital construction.

LRU PECULIAR

LEO CONSTRUCTION FACILITY - Similar to earth baseline COTV fabrication fixture. Several COTV configurations are required to service alternative transfer routes which promotes a modular COTV design and fixturing approach with elements similar to those needed for earth baseline construction fixture and assembly elements. Identical technology requirements.

GEO CONSTRUCTION FACILITY - Very similar to earth baseline SPS construction facility. Differences will be limited to those caused by SPS design and construction details associated with lunar material substitutions. The potential use of ceramic structure and waveguides to replace graphite composite will result in fixturing revisions and changes in joining/attachment techniques.

Table J-4. Infrastructure System Elements.

TECHNOLOGY DEVELOPMENT PHASE - TERRESTRIAL AND SPACE TESTS	
EARTH BASELINE	LRU PECULIAR
LEO HABITAT - Modular space station to support COTV construction, transportation vehicle servicing, and construction material logistics functions. • Life support systems (solar flare shelter) • Power supply and heat rejection • Attitude control and positioning	LEO HABITAT - Similar to earth baseline habitat. Crew size and duty assignments may differ somewhat, but station functional requirements and development needs should be identical.
LEO PROPELLANT DEPOT - Storage of LH_2, LO_2 and LA_R propellants for COTV and POTV, plus vehicle docking and propellant transfer provisions • Large storage tanks (insulated) • Boiloff reliquefaction equipment • Power, attitude control, heat rejection and other satellite support systems.	LEO PROPELLANT DEPOT - Similar to earth baseline depot except for propellant capacity and type. Argon will not be required since the COTV will utilize oxygen propellant obtained from lunar resources.
GEO HABITAT - Large modular space station to support SPS construction, SPS maintenance, and transportation vehicle servicing. Same basic technology requirements as LEO habitat applied to a larger habitat.	GEO HABITAT - Similar to but smaller than the earth baseline habitat to accommodate personnel for SPS maintenance and vehicle servicing. Same basic technology requirements.
No corresponding space facility (Although a GEO depot to supply POTV propellants might be cost effective).	GEO PROPELLANT DEPOT - Similar to LEO propellant depot except for propellant quantity. Propellant storage of LH_2 and LO_2 required.
No corresponding space facility.	SMF HABITAT - Similar to earth baseline GEO habitat except for larger size and different location. Habitat is sized to accommodate processing and manufacturing personnel in addition to those required for SPS construction and transportation vehicle servicing. Same basic technology requirements applied to an even larger habitat.

Table J-4. Infrastructure System Elements (contd).

TECHNOLOGY DEVELOPMENT PHASE - TERRESTRIAL AND SPACE TESTS	
EARTH BASELINE	LRU PECULIAR
No corresponding space facility.	SMF PROPELLANT DEPOT - Same as GEO propellant depot
No corresponding space facility.	LLO PROPELLANT DEPOT - Same as GEO propellant depot
No corresponding facility.	LUNAR SURFACE PROPELLANT DEPOT (if required) - Limited to contingency supplies of LO_2/LH_2 and storage of LO_2 manufactured on the lunar surface.
No corresponding facility.	LUNAR SURFACE HABITAT - Modular living quarters to support lunar mining, processing, and transportation operations. • Life support & environmental control • Power supply and heat rejection • Personnel access to and from the lunar surface
CONSTRUCTION POWER STATION - Photovoltaic array and power conditioning equipment based on SPS technology to provide electrical energy needed for SPS construction. • See Table J-2 for technology development requirements.	SMF POWER STATION - Similar larger photovoltaic array to provide electrical energy for lunar material processing, stock manufacturing, SPS component manufacturing, module sub-assembly, and SPS construction.
No corresponding facility.	LUNAR POWER STATION - Electrical power generation to supply mining, processing, manufacturing, transportation, and personnel support requirements on the lunar surface. Implementation options include nuclear Brayton, photovoltaic with energy storage capability, and photovoltaic with orbital reflectors. • Nuclear reaction for lunar use (maintenance) • Lunar derived shielding ("lunar concrete") • Waste heat radiators (lunar surface) • High capacity energy storage devices • Orbital solar reflector satellites (optional for photovoltaic supply)