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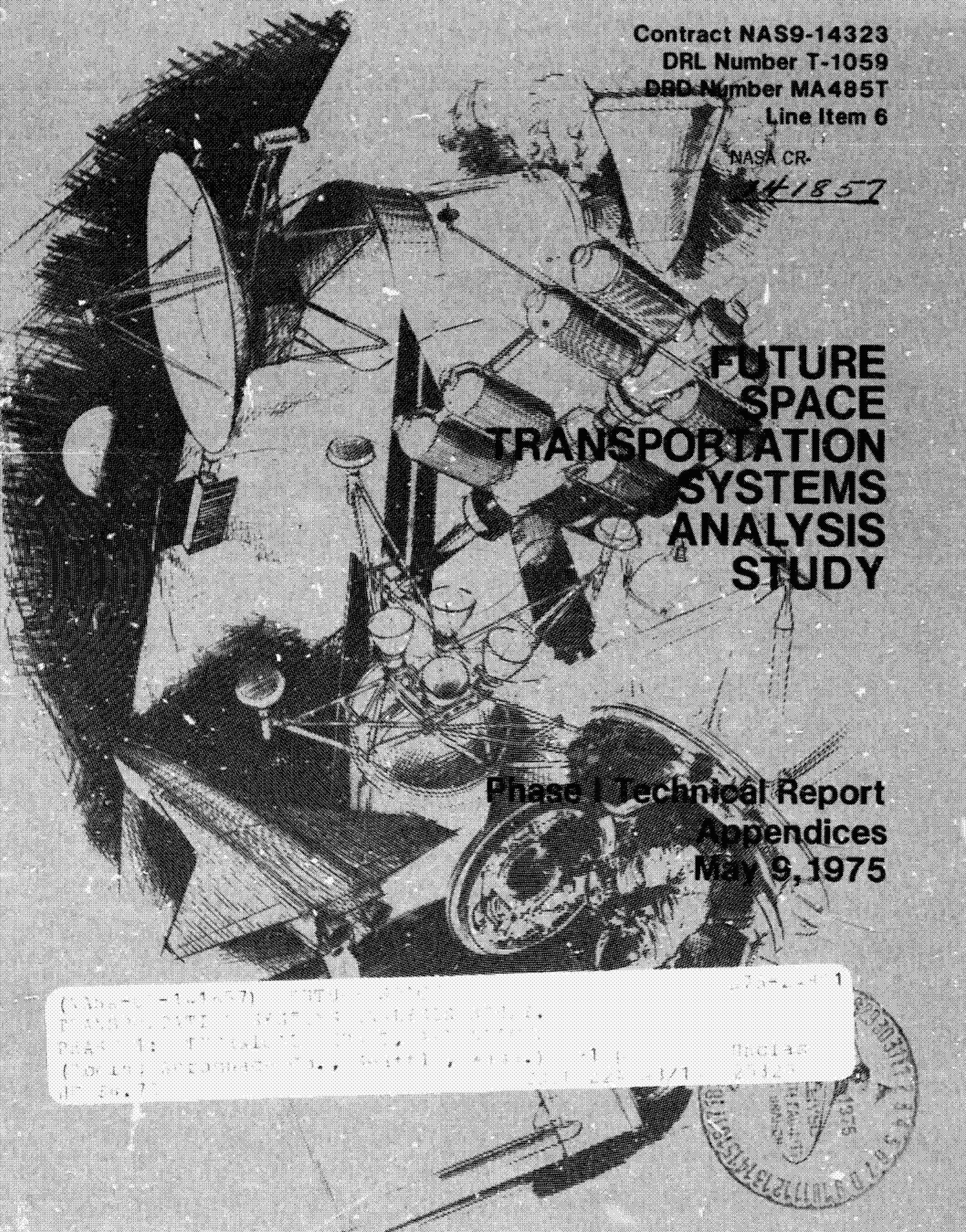

FUTURE SPACE TRANSPORTATION SYSTEMS ANALYSIS STUDY

Phase I Technical Report
Appendices
May 9, 1975

Boeing Aerospace Company
Seattle, Washington
Research & Engineering Division

Contract NAS9-24323
DRL Number T-1059
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FUTURE SPACE TRANSPORTATION SYSTEMS ANALYSIS STUDY

PHASE I TECHNICAL REPORT
APPENDICES
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Submitted to
The National Aeronautics and Space Administration
Lyndon B. Johnson Space Center
in Fulfillment of the Requirements
of Contract NAS9-14323

Approved

G. R. Woodcock
Study Manager

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APPENDIX 1 TRANSPORTATION VEHICLES ANALYSES

This section presents data of a backup nature, supporting performance and mission/transportation system results described in the technical report.

The transportation mass requirements developed for each mission and transportation mode were based on vehicle systems sized to fit the exact needs of each mission (i.e. "rubber" vehicles). The parametric data used to derive the mass requirements for each mission and transportation mode are presented here to enable accommodation of possible changes in mode options or payload definitions. In addition, the vehicle sizing and functional requirements used to derive the parametric data will form the basis for conceptual configurations of the transportation elements in a later phase of study.

1.1 ORBIT TRANSFER VEHICLES

Orbit transfer vehicles are those designed to transfer payloads from one orbit state to another. As such, they are true "space" vehicles. They must, in general, be launched to Earth orbit by Earth launch systems and are not capable of landing on a planetary body. High and low thrust systems are included.

OTV parametric weight curves have been developed to provide equations for stage inert weights of the form $W_I = a + b\ W_P$ in order to enable rapid calculation of required propellant and total stage mass. Inasmuch as propulsion stage inert weight is not a linear function of propellant weight, the curves are limited to specific ranges of propellant mass. The value of "a" in the above equation increases at higher propellant ranges, while the value of the slope "b" decreases. The value of "a" represents those elements that do not vary to a large extent with propellant variation in the applicable propellant range. The value of "b" represents those elements that vary linearly with propellant mass over the applicable range.

1.1.1 LO_2/LH_2 Stages

"Small" and "large" options have been identified, the former compatible with Shuttle launch and the latter with heavy lift launch. In either case, the OTV may be launched partially or fully off loaded if its propellant capacity exceeds the launch mass capability of the launch vehicle. The small option is common with the lunar transport options described below, but it does not include landing legs and associated hardware and does not require throttleable engines. Configuration concepts are described in the discussion of lunar transport vehicles (section 1.2). The small OTV is generically quite similar to the full capability tug as defined by MSFC report 68M00039 (4 volumes), but may incorporate more than one engine, greater propellant loading, and other features depending on mission application.

Figure 1-1 shows the weight of "small" orbit transfer vehicles used in the propellant range of 30 000 kg (66,000 lbm) or less. The dimensions of the small OTV are such that it can be launched in the shuttle cargo bay, with additional payload or crew vehicles. The inert weight of the small OTV's includes residual fluids, reserves and APS propellant proportionate to that included in the large OTV weights. The avionics for the small OTV's, as shown, does not include 320 kg (700 lbm) required for independent operation; that is, certain avionics functions are presumed provided by the payload. The engines for both the large and the small OTV's are considered to be designed specifically for the thrust-to-weight indicated (i.e. "rubber" engines). Weight growth of 15% has been included in the small OTV inert weights.

The 1-1/2 stage transportation elements are essentially basic OTV stages plus drop tanks. Figure 1-2 shows LO_2/LH_2 drop tank weights for "small" sizes that are compatible with shuttle launch. The length of the drop tank system versus propellant weight is shown on Figure 1-3. For simplicity, the cluster structure for attachment and jettison from the OTV main stage has been included in the drop tank inert weights. In more detailed analysis, the cluster structure weight penalty will be split between the main stage and the jettisoned drop tanks. Unusable propellant and minimal flight performance reserves are included.

The large LO_2/LH_2 OTV options include propellant loading appropriate to common stage (slingshot mode) and single stage operation. Figure 1-4 is a typical configuration sized for single-stage operation at 300 000 kg (660,000 lb) propellant loading.

Figures 1-5, 1-6, and 1-7 show weights of large LO_2/LH_2 orbit transfer vehicles, for startburn thrust-to-weight ratios of 0.2, 0.3 and 0.4, respectively. The startburn thrust-to-weight ratio and the startburn weight have a significant effect upon the propulsion subsystem weight. The configuration selected for the parametric analysis includes separate aluminum tanks for LO_2 and LH_2. A multi-layer body shell supports payload, tanks, electrical power, avionics and auxiliary propulsion subsystems. Meteoroid protection is considered to be that inherent in the body shell and insulation design. The nominal large OTV is designed for both independent (unmanned orbital transfer) operation and manned operation with a crew transfer vehicle (OTV). Approximately 320 kg (700 lb) of added avionics and electrical power is required for independent operation of either the large

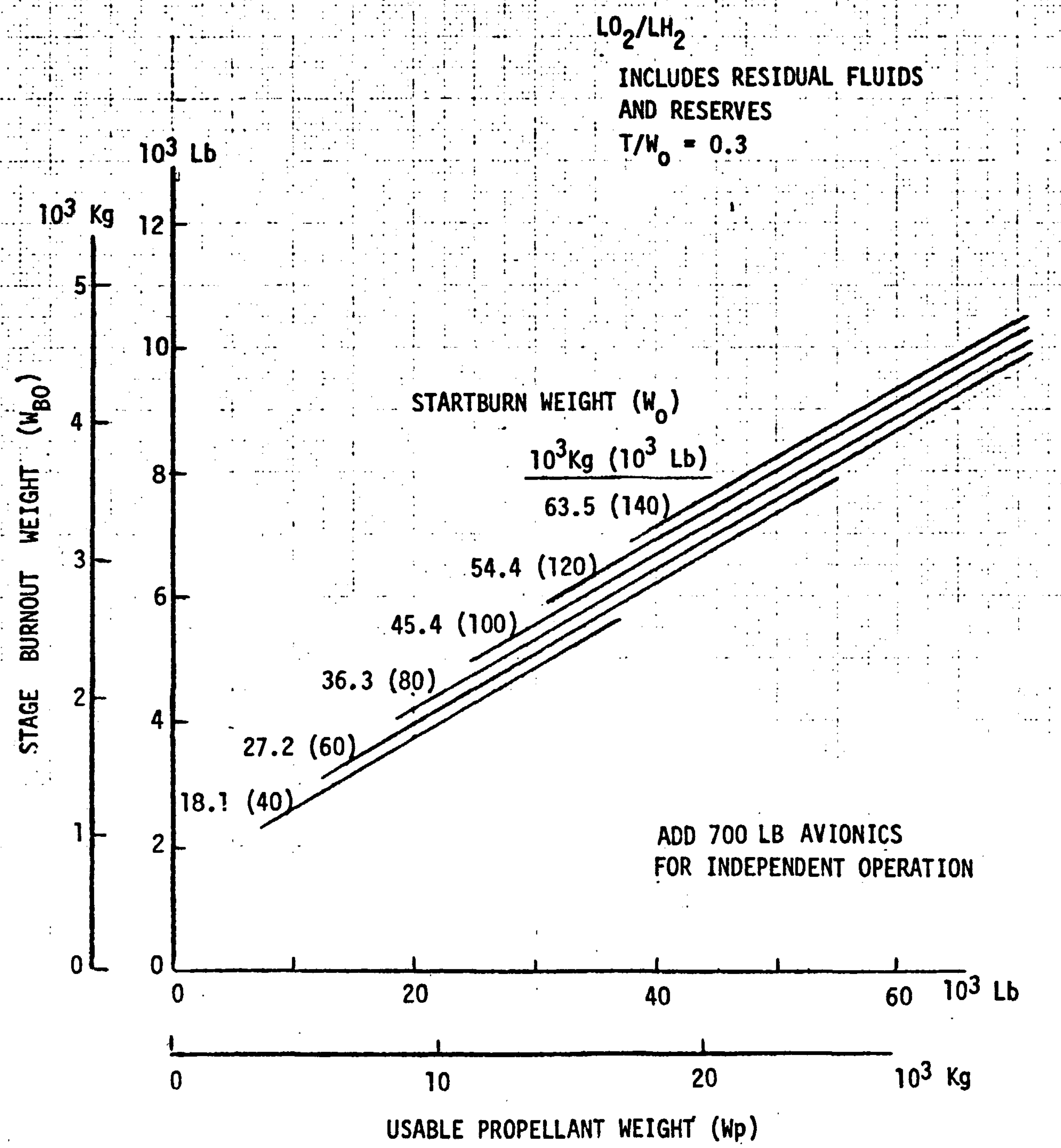

FIGURE 1-1 SMALL ORBIT TRANSFER VEHICLE (OTV)

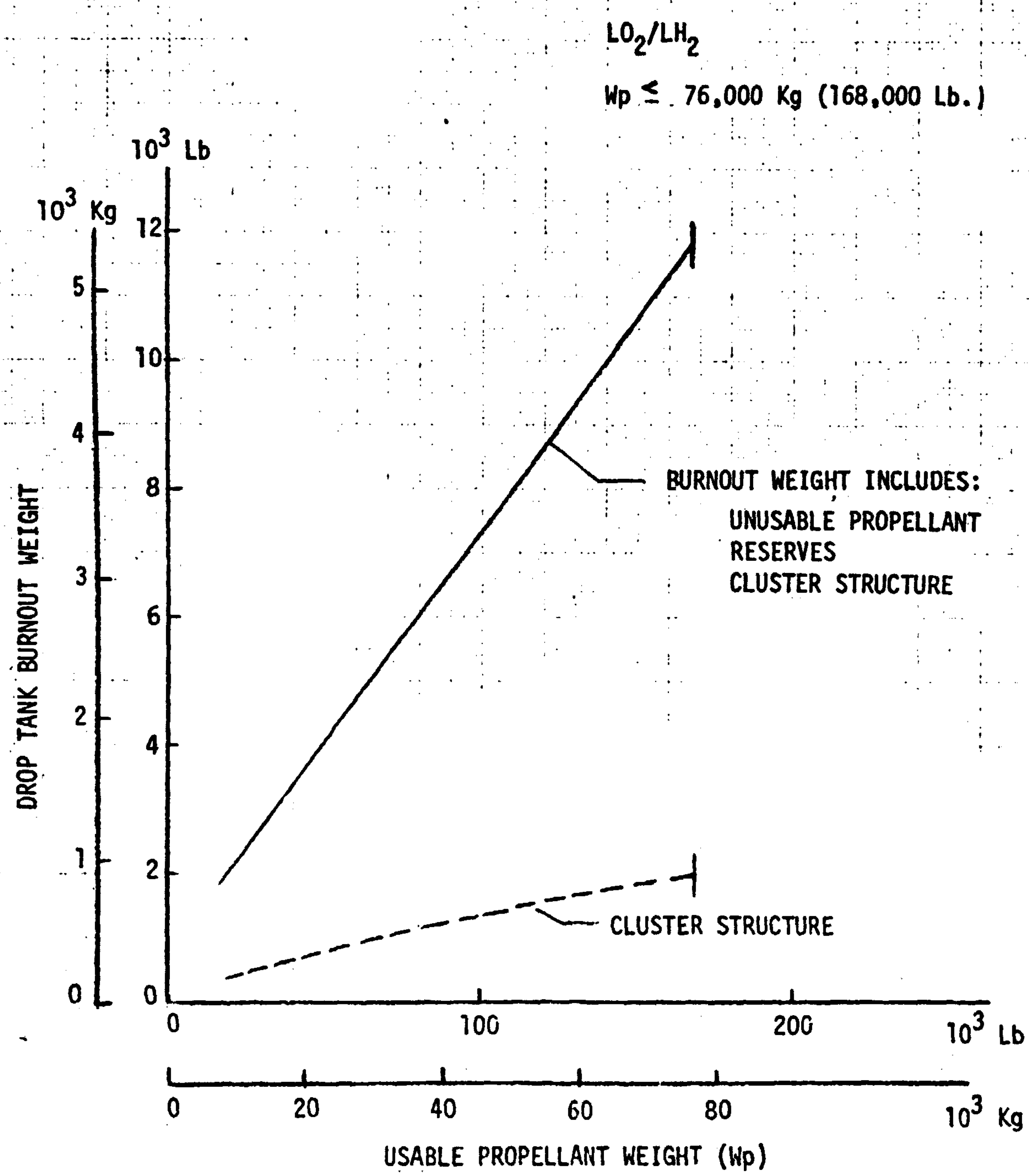

FIGURE 1-2 SMALL DROP TANK SYSTEM WEIGHT.

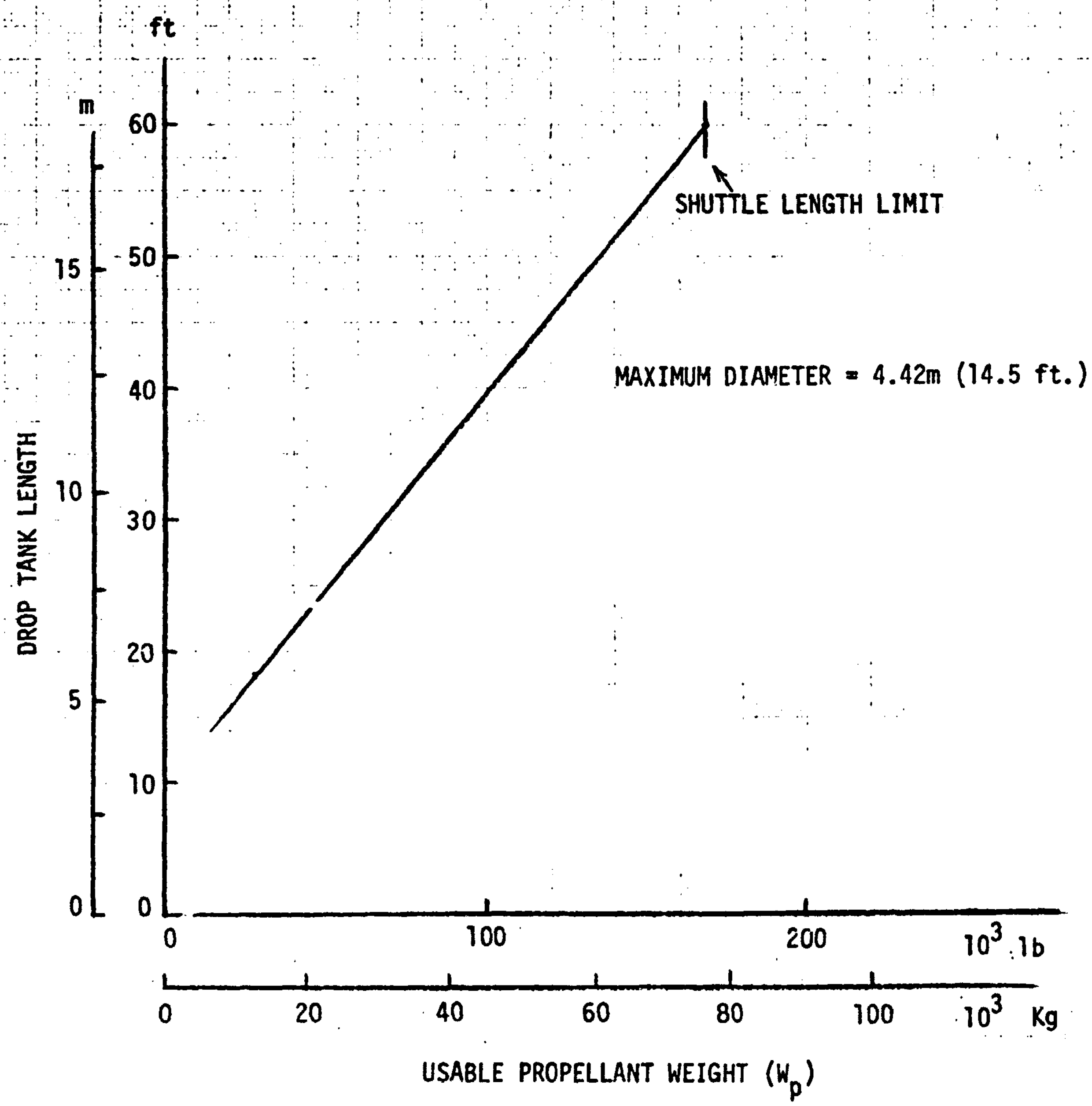

FIGURE 1-3 SMALL DROP TANK SYSTEM LENGTH

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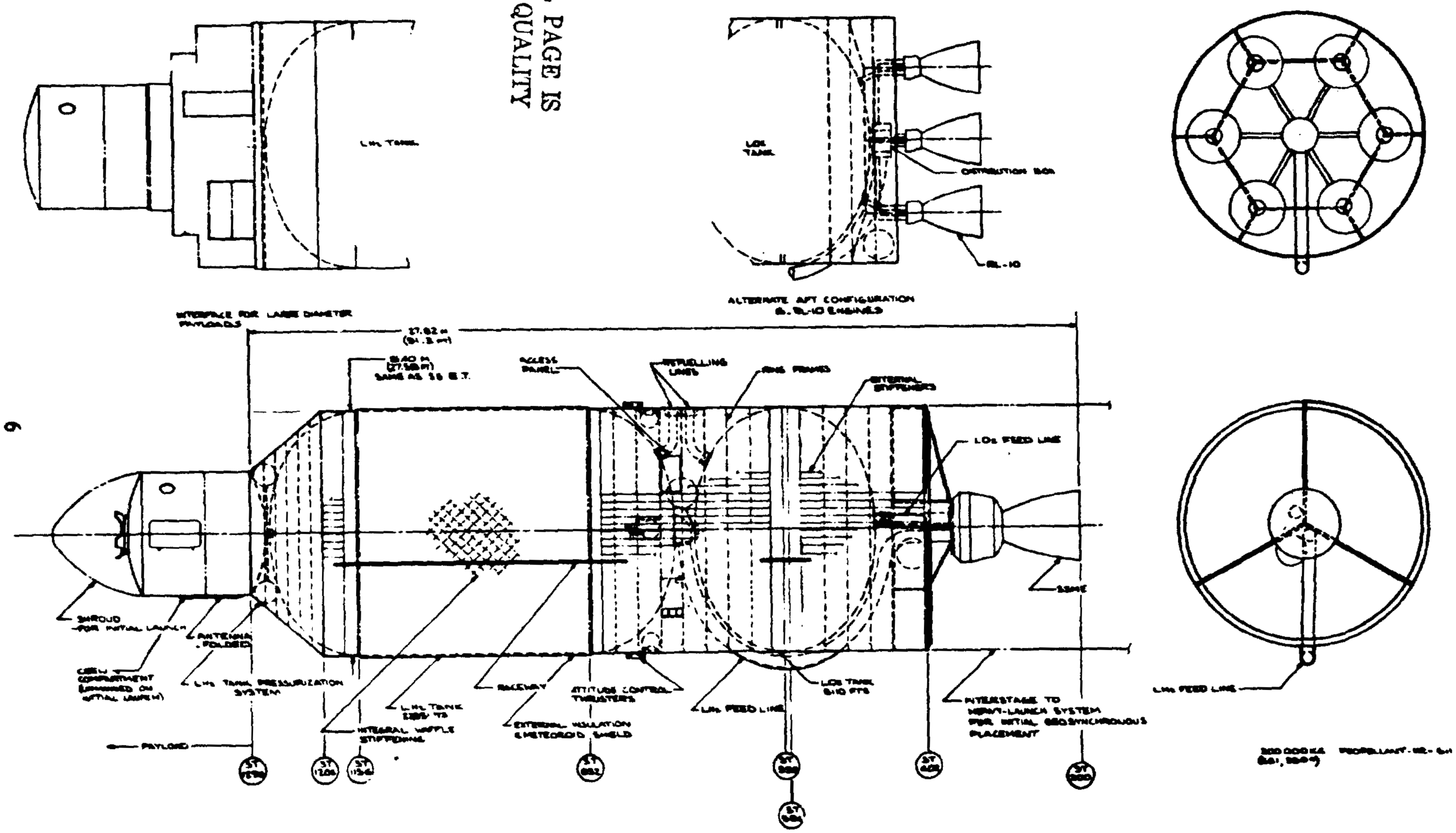

Figure 1-4 Large-Orbit Transfer Vehicle

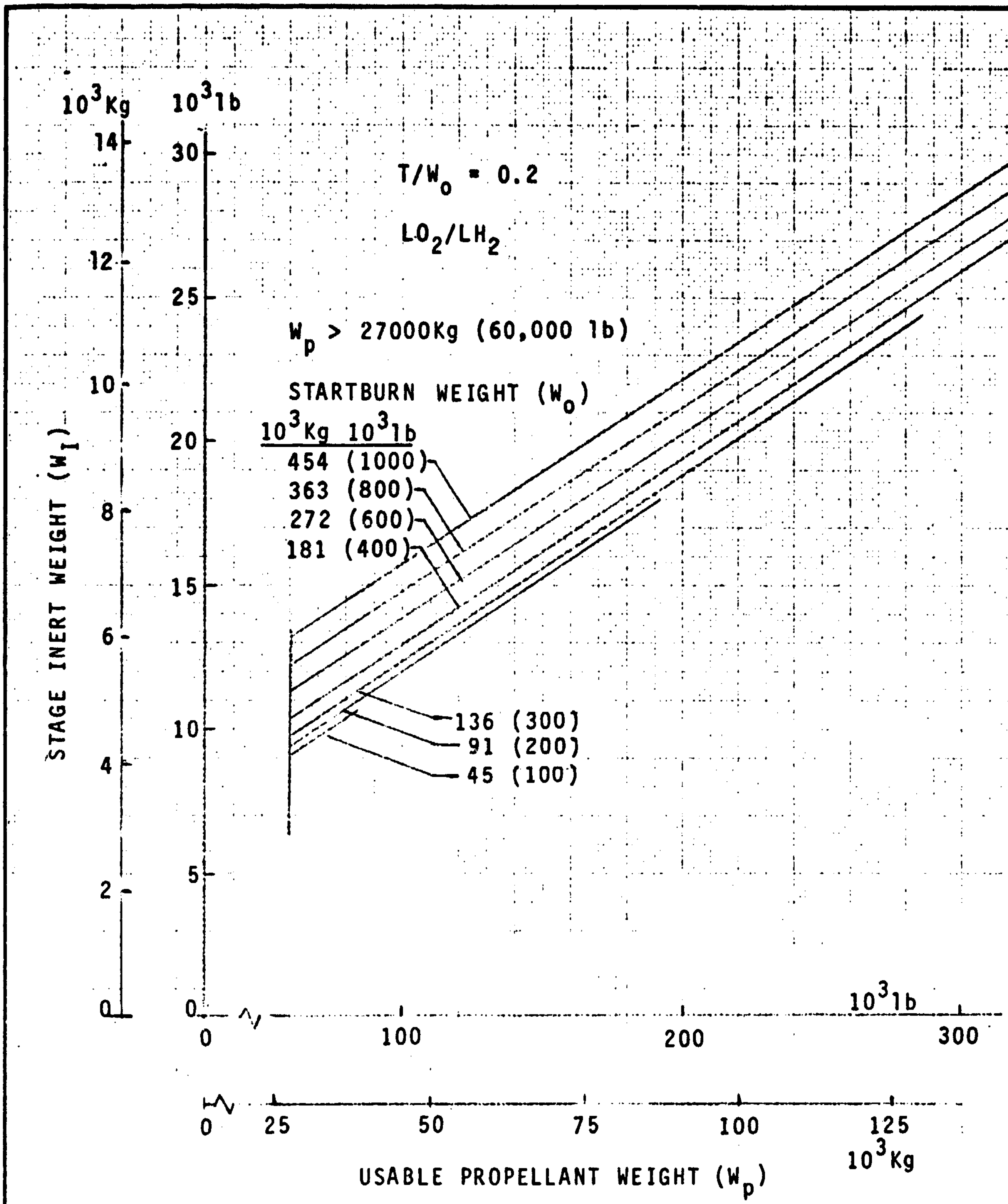

FIGURE 1-5 LARGE ORBIT TRANSFER VEHICLE WEIGHT

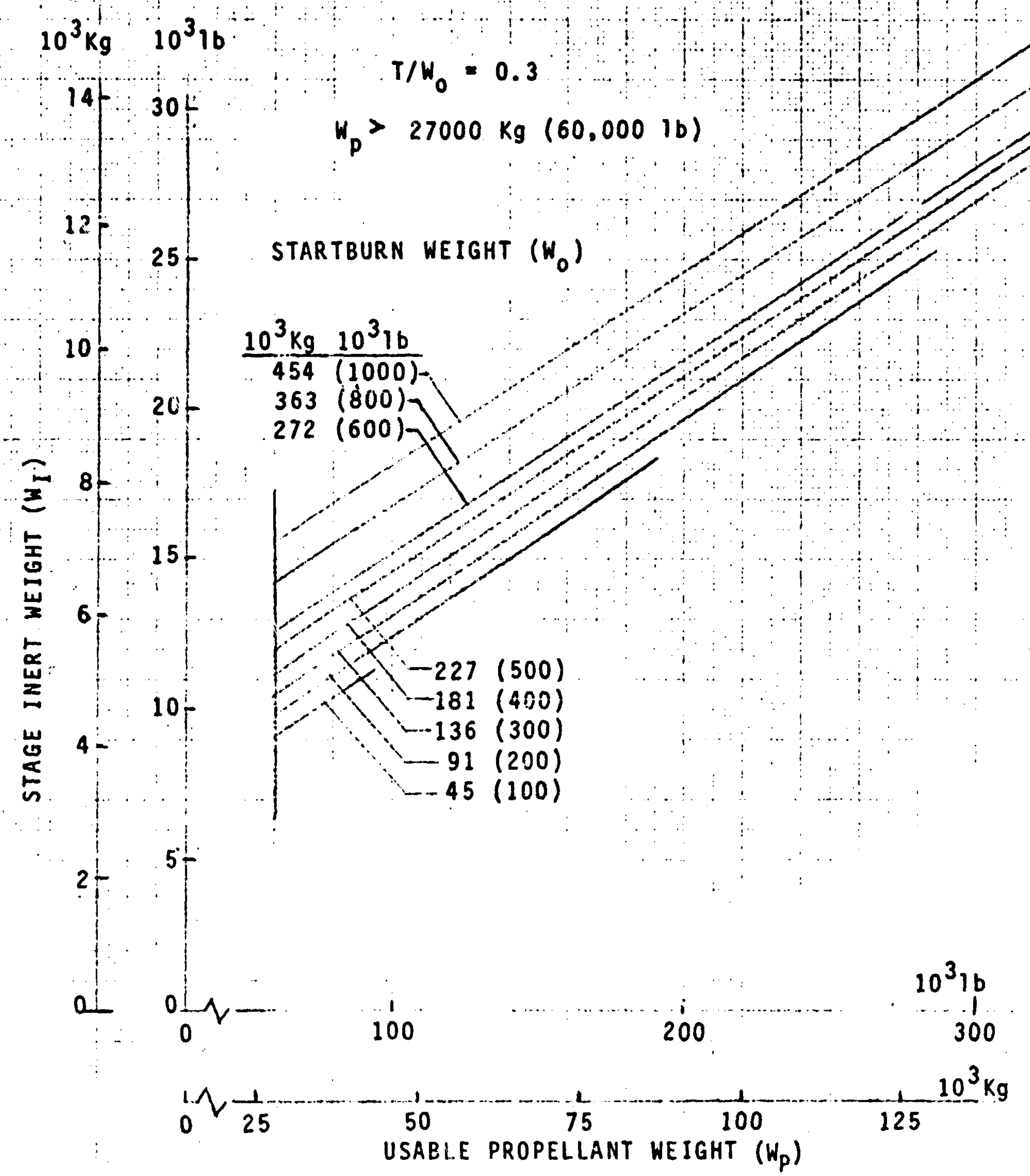

FIGURE 1-6 LARGE ORBIT TRANSFER VEHICLE WEIGHT - LO_2/LH_2

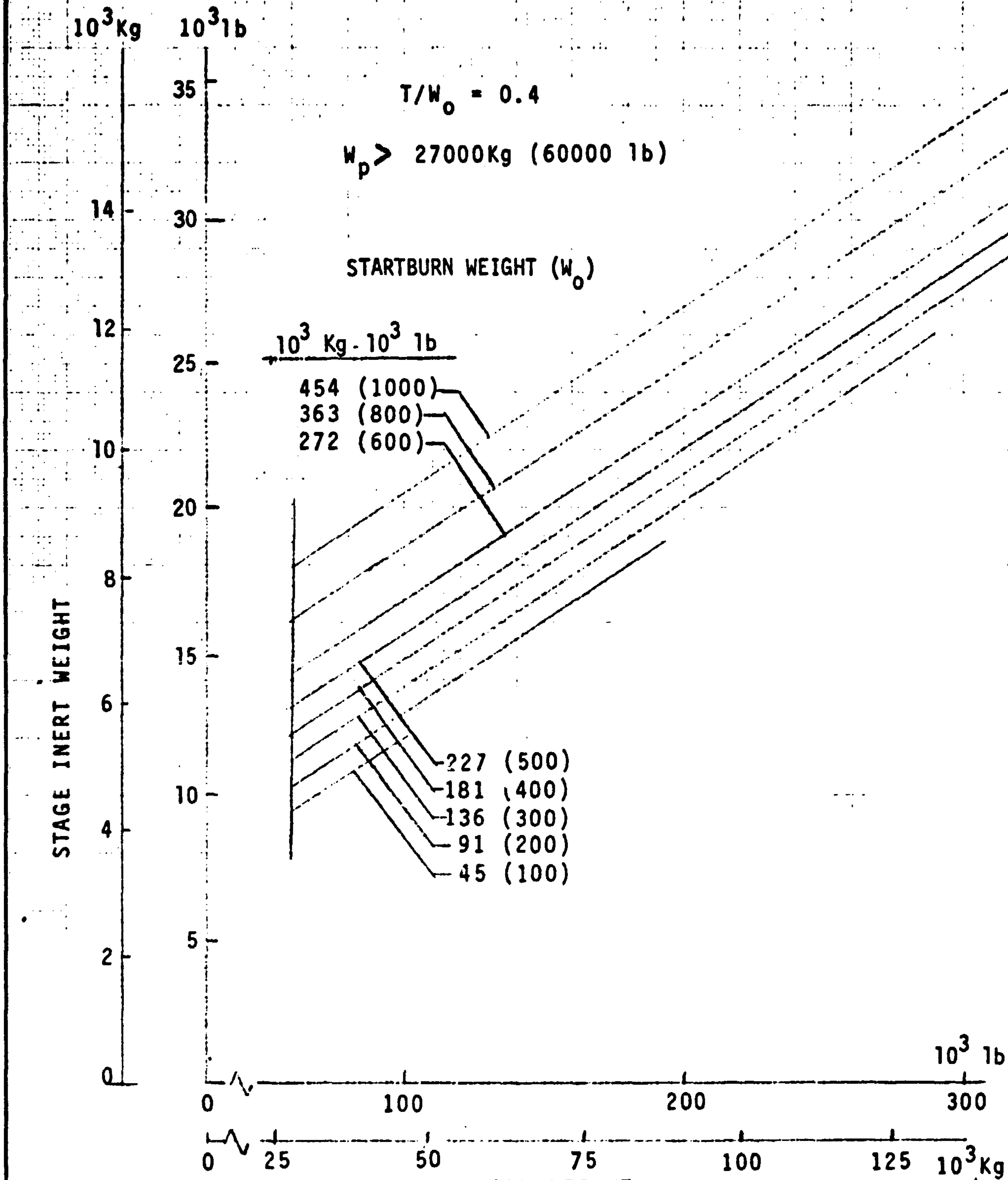

FIGURE 1-7 LARGE ORBIT TRANSFER WEIGHT - LO_2/LH_2

or small OTV's. The stage inert weight (W_I) includes unusable fluids. An allowance (.015 W_0) has been made for auxiliary propulsion system (APS) propellant. Boiloff propellant has not been included as an inert weight. Weight growth of 15% has been included. Tables 1-1 and 1-2 present typical weight statements for large OTV's.

Figure 1-8 shows the effect of increasing the startburn thrust-to-weight ratio (T/W_0) for a translunar injection burn of 3109 m/s (10,200 ft/sec) from orbit. Gravity losses decrease from 108 m/s (354 ft/sec at T/W_0 = 0.2 to 28 m/s (92 ft/sec) at T/W_0 = 0.4. The propellant saved at higher thrust-to-weight ratios is offset by increased engine, feed and thrust structure weight. The OTV typical mission encounters high earth gravity losses for less than half of the total thrusting time, hence a value of T/W_0 = 0.3 has been selected as near-optimum for most of the OTV missions.

Figure 1-9 shows "large" LO_2/LH_2 drop tank weights versus usable propellant weight. The curve is applicable to the propellant range of 73 000 to 180 000 kg (160,000 to 400,000 lb). The tank dimensions are all above the shuttle capability. Cluster structure, unusable propellant and minimal flight performance reserves are included. None of the systems selected for matching to missions used large drop tanks.

1.1.2 LO_2/MMH Stages

The LO_2/MMH OTV represents a dense propellant OTV. Alternate propellants could be considered, e.g. methane, but MMH provides high performance (Isp estimated as 3,630 m/sec (370 sec)) and high density 874 kg/m^3 (54.5 lb/ft^3). A dense propellant OTV sized for heavy lift launch can be physically small enough for return to Earth empty in the shuttle payload bay. A configuration is shown in figure 1-10.

The dense propellant OTV requires staging to perform the more demanding missions, but less complex assembly in orbit than that required for the small LO_2/LH_2. It would presumably not require on-orbit refueling, if a heavy lift vehicle were available. Because of the lower Isp the mass to be placed on orbit for a given mission is generally significantly more than required for the LO_2/LH_2 options. This disadvantage must be traded with the advantage of simpler operations.

The inert weights of OTV's using LO_2/MMH propellant are shown in figure 1-11. The dimensions of LO_2/MMH vehicles are compatible with launch in the shuttle cargo bay up to about 175 000 kg (385,000 lb) propellant loading. A representative weight statement for a LO_2/MMH OTV is given in table 1-3.

Table 1-1 Single Stage OTV – LO_2/LH_2 (GSS Mission)

	kg	lbm
Structure	6 970	15,370
Propulsion	2 410	5,310
Other Subsystems	2 420	5,330
Weight Growth (15%)	1 770	3,900
Stage Dry Weight	13 570	29,910
Usable Propellant	230 750	508,680
Unusable and Other Fluids	5 180	11,410
Total Stage Weight	249 500	550,000

Resultant $\lambda' = .925$

Table 1-2 Candidate OTV Weights (GSS Mission)

Common Stage OTV – LO_2/LH_2

	kg	lbm
Structure	2 890	6,380
Propulsion	1 380	3,050
Other Subsystems	1 340	2,940
Weight Growth (15%)	850	1,870
Stage Dry Weight	6 460	14,240
Usable Propellant	77 950	171,860
Unusable and Other Fluids	1 770	3,900
Total Stage Weight [1]	86 180	190,000
Total OTV Weight (Two Stages)	192 360	380,000

Resultant $\lambda' = .905$ Each Stage

[1] NOTE: The common stages will vary slightly from each other in APS propellant and probably in propulsion (thrust), APS inerts, and avionics.

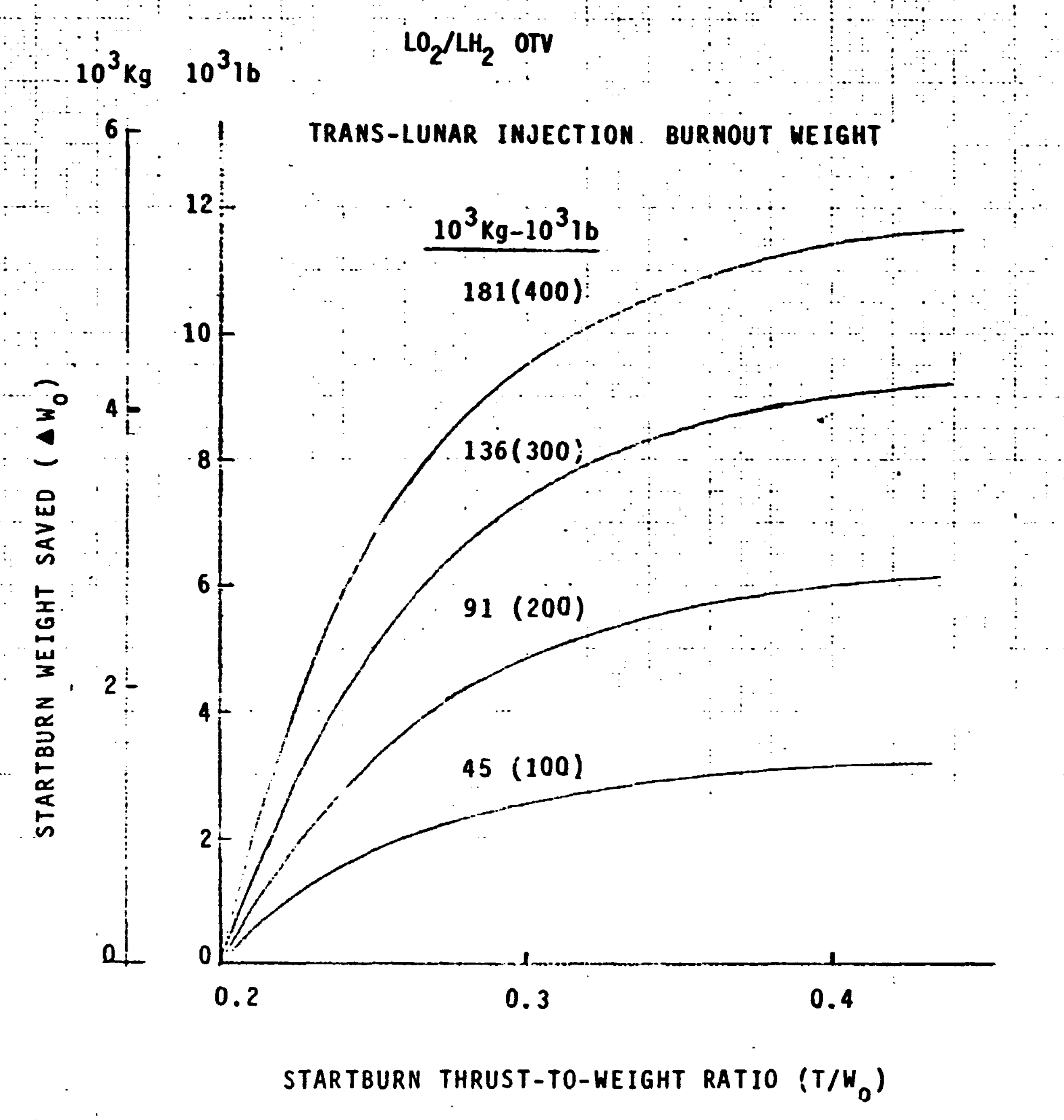

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FIGURE 1-8 EFFECT OF TRANS-LUNAR INJECTION THRUST-TO-WEIGHT RATIO

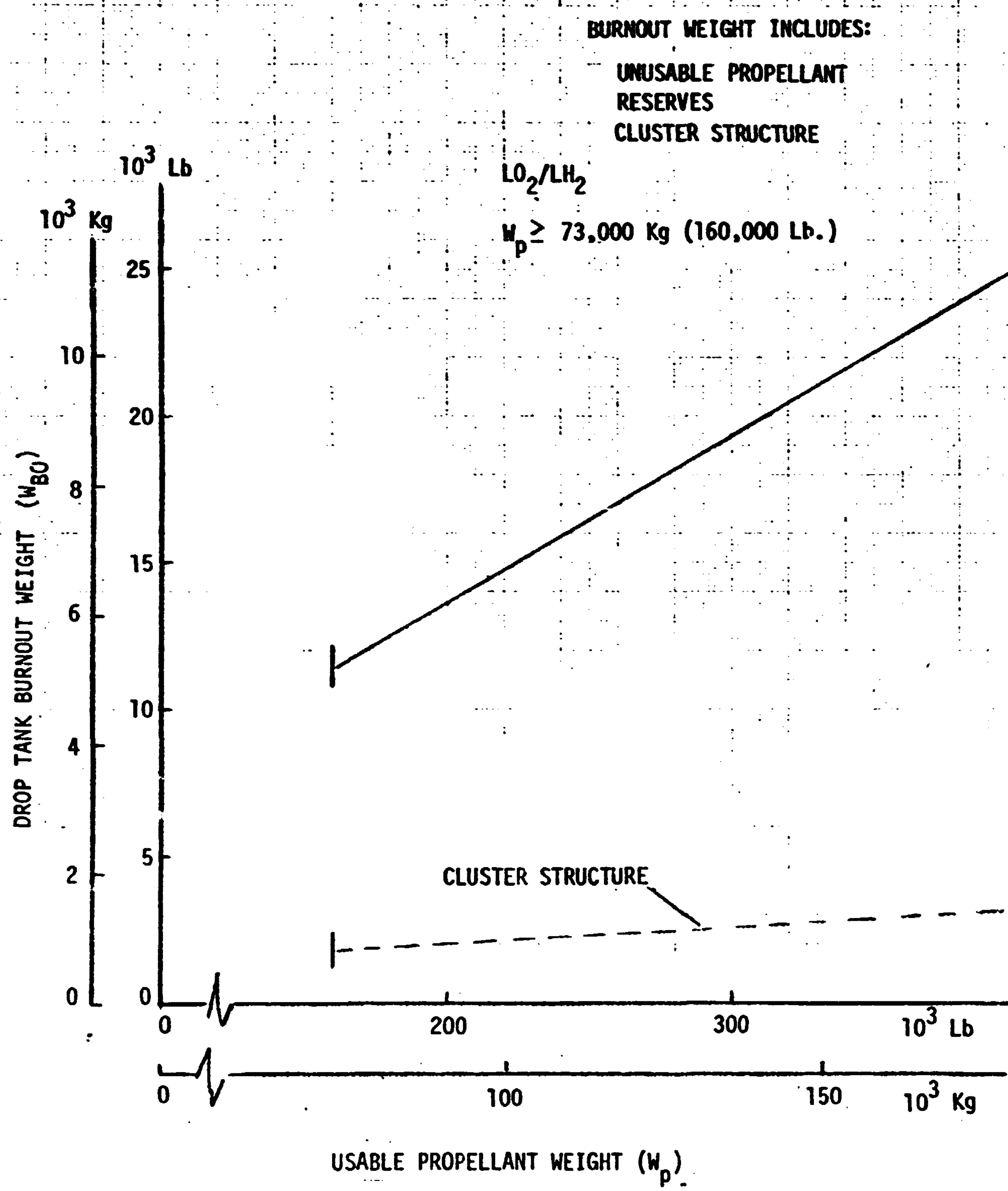


FIGURE 1-9 LARGE DROP TANK SYSTEM WEIGHT

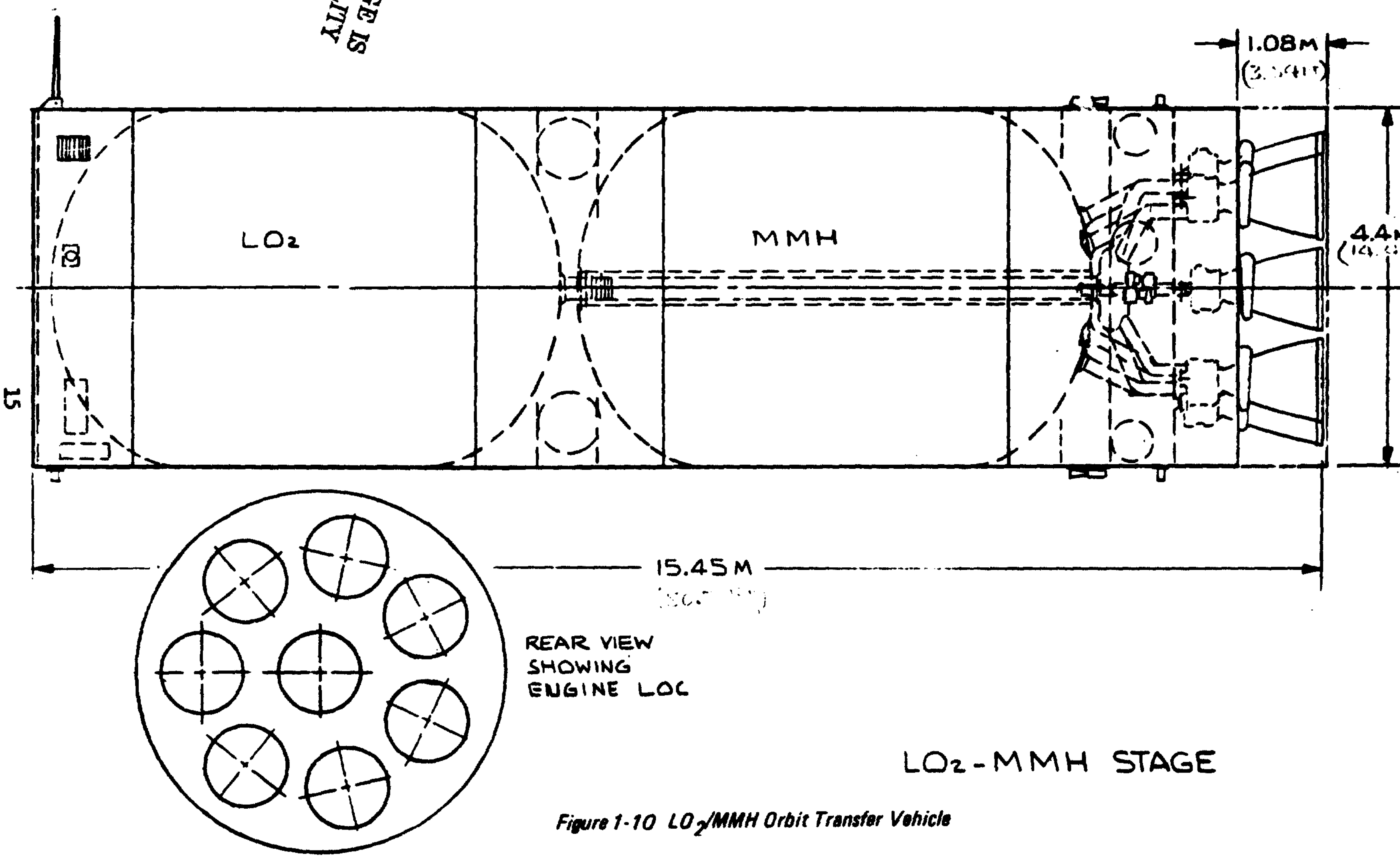


Figure 1-10 LO_2/MMH Orbit Transfer Vehicle

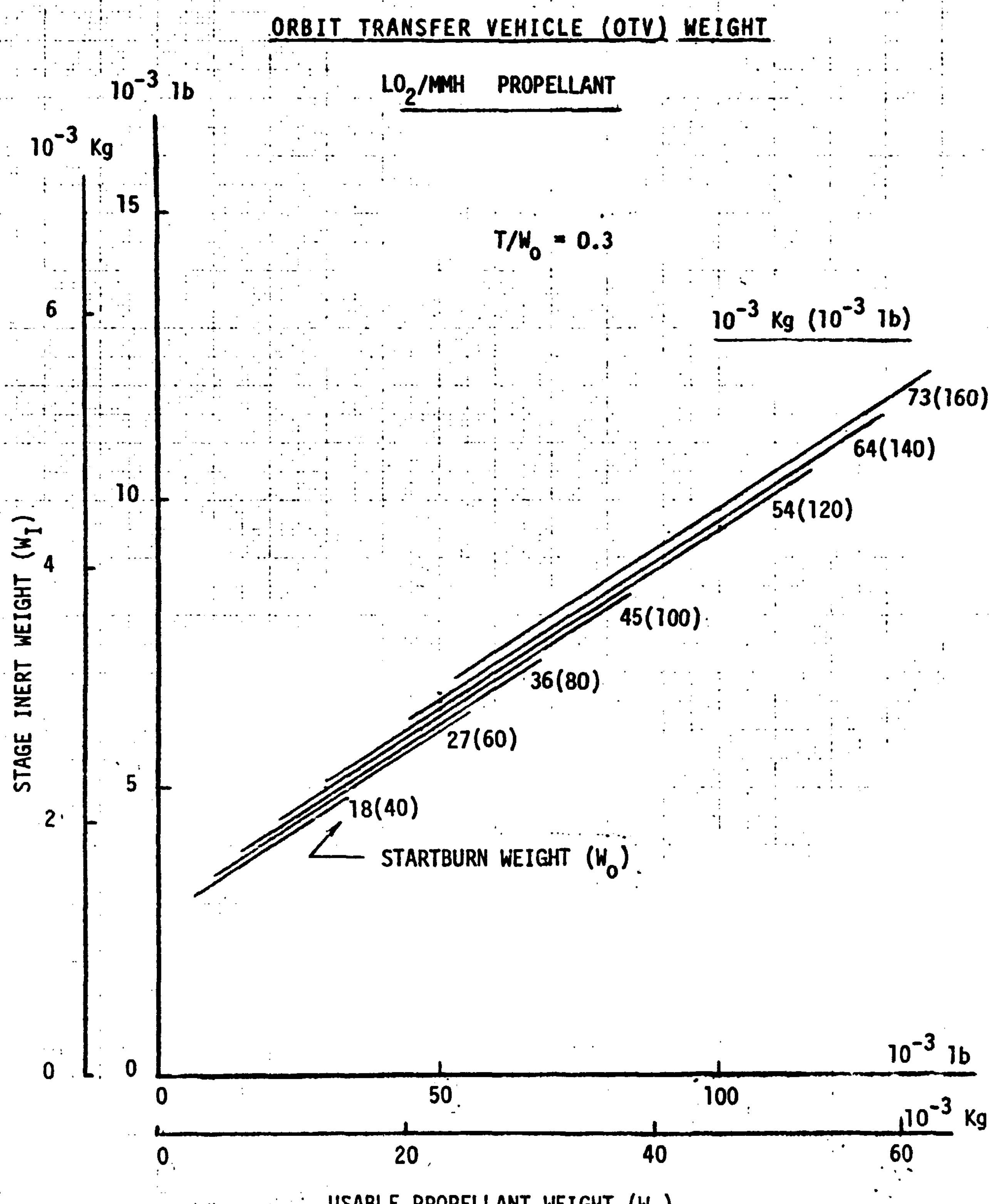


Figure 1-11. LO_2/MMH Orbit Transfer Vehicle Mass

Table 1-3 Common Stage OTV – LO_2/MMH (GSS Mission)

	kg	lbm
Structure	3 730	8,220
Propulsion	1 870	4,130
Other Subsystems	1 560	3,430
Weight Growth (15%)	1 070	2,370
Stage Dry Weight	8 230	18,150
Usable Propellant	136 380	300,700
Unusable and Other Fluids	2 790	6,150
Total Stage Weight [1]	147 400	325,000
Total OTV Weight	294 800	650,000

Resultant $\lambda' = .925$ each stage

1.1.3 Aerobraking Analyses

Introduction and Background—The idea of return to a low Earth orbit from the moon or from a high orbit, employing gradual dissipation of energy through a series of elliptic passes grazing the Earth's atmosphere, was originally suggested by Oberth in the 1920's. In 1971-72 this technique was studied by Boeing for application to the space tug under contract NAS8-27501. The principal conclusions from that study were as follows:

- The aerobraking mode is feasible for the return of the Space Tug from geosynchronous and other high orbit missions.
- The aerobraked Tug's payload capability is maximized by missions having 25 to 35 atmospheric passages during the aerobraking phase. This corresponds to return time 3 to 6 days.

The aerobraking kit to be added included aft heat shields, aerodynamic flares, sidewall insulation, astrionics modifications and payload adapters.

More recently, in 1974, aerobraking was studied by LMSC under contract NAS8-28586. This study synthesized tailored aerobraking vehicles configured expressly for the shuttle-launched round trip mission to geosynchronous orbit with aerobraking.

Performance Potential—The gains that might be achieved through aerobraking are substantial. For example, representative ΔV budgets for all-propulsive and aerobraking geosynchronous round trips from a 28-1/2^{o}, 296 km (160 n.mi.) orbit as follows (table 1-4).

The indicated delta V savings for aerobraking is 2226 m/sec (7302 ft/sec). For a representative space tug of 25 000 kg (55,000 lb) usable propellant loading and jet velocity of 4,500 m/sec (Isp = 459), a round trip payload for propulsive return is estimated as 1 354 kg (2,985 lb), and for aerobraking return 5 036 kg (11,110 lb). The aerobraking return "payload" includes aerobraking hardware. This comparison is based on equal propellant weight. Comparing on equal gross weight requires that the propellant loading of the aerobraking stage be reduced to 22 200 kg (48,900 lb). The payload becomes 4 416 kg (9,735 lb), again including aerobraking provisions. These in the referenced Boeing study totaled 975 kg (2,150 lb) in a typical case leaving a net round trip payload of about 3 440 kg (7,585 lb).

The 30-pass aerobraking mission requires about five days for return to low Earth orbit. The radiation dose to a crewman in a typical crew transport module without added shielding, due to

Table 1-4. Delta V's for Aerobraking

	Propulsive		Aerobraking	
	m/sec	ft/sec	m/sec	ft/sec
Transfer Injection (includes 100 m/sec g loss and 2° plane change)	2547	8356	2547	8356
Ascent midcourse	10	33	10	33
Circularize (includes 26½° plane change)	1787	5862	1787	5862
Deorbit	1787	5862	1844	6050
			(28½° plane change)	
Descent midcourse	10	33	10	33
Trajectory correction during braking passes			100	328
Circularize at 160 n. mi.	2447	8028	64	210
TOTAL	8588	28174	6362	20872

repeated passages through the van Allen belts, would be on the order of 300 rem (10 rem per orbit). Roughly 1 000-1 500 kg (2,200-3,300 lb) of shielding will be required to reduce this to an acceptable level. The remaining useful payload, 1 940 kg (4,275 lb) is not sufficient to provide for a manned round trip to geosynchronous orbit. Enlarging the stage to about 30 000 kg (66,000 lb) propellant loading will provide about 3 500 kg (7,700 lb) net useful payload, about enough for a 2-man round trip to geosynchronous orbit. The gross initial mass is about 40 000 kg (88,000 lb); the system could not be launched fully fueled by the shuttle.

A satellite repair and service mission is likely to require a crew of four, plus 1 000 kg (2,200 lb) or more equipment and spares. The 1 500 kg (3,300 lb) shielding penalty still applies leading to a total payload of 8 550 kg (18,850 lb) including 1 000 kg (2,200 lb) for aerobraking. The required usable propellant is about 41 000 kg (90,000 lb). The system gross mass is 54 000 kg (119,000 lb). The stage and payload will require two shuttle launches with possibly a third for propellant top-off. Stage length is about 13.5 m (44 ft) without payload.

Aerobraking Implementation–The referenced Boeing Study described configurations needing a significant amount of thermal protection. In this investigation we looked for ways to reduce the thermal protection retrofit by deploying a large drag area. The resulting low mass/CdA will reduce heating rates while maintaining an acceptable rate of deceleration. Two potential arrangements are shown in figures 1-12 and 1-13. The first parachute-like device may be unstable in hypersonic flow; the second should be stable and is the recommended low mass/area approach.

Rough estimates of loads and heating were made using a method described by Kostoff in Bellcomm paper B72-01005 dated January 19, 1972. Kostoff gives an equation (corrected here) for deceleration due to an aerobraking pass:

$$\frac{V}{V_o} = \exp\left\{-\frac{C_DA}{M}\rho_p\left[2\pi r_pH\left(\frac{e+1}{e}\right)\right]^{1/2}\right\}\left\{1+\frac{C_DA}{M}\rho_p\left[\frac{\pi Hr_p}{2}\frac{(e+1)}{e^3}\right]^{1/2}\right\}$$

where V/V_o is velocity ratio (exit from the pass)/(entry to the pass).

C_DA/M is the ballistic coefficient for the vehicle in M^2/kg or ft^2/lbm

ρ_p is atmosphere density in kg/m^3 or lbm/ft^3 at perigee

r_p is perigee radius in meters or feet.

H is upper atmosphere scale height in meters or feet, approx. 7,900 m (26,000 ft.)

e is eccentricity of the initial orbit ellipse.

For the case analyzed here the correction term in the second bracket is ≈ 1 and can be ignored.

For a 30-pass mission the initial perigee velocity is 10,340 m/sec (33,923 ft/sec) and the final perigee velocity about 7,910 m/sec (25,950 ft/sec) (90 x 296 km; 48.5 x 160 n.mi. orbit). The velocity reduction of 2,430 m/sec (7,970 ft/sec) requires about 81 m/sec (266 ft/sec) per pass for 30 passes. Thus, the velocity ratio is about (10,340 - 81)/10,340 = 0.992 for the first pass.

Loads–Decelerations are, to first order, independent of C_DA/M. Note that acceleration = D/M = $C_DA\rho V^2/2M$ and that $C_DA\rho/2M$ is a term in the above equation for velocity ratio. The entire exponent must yield V/V_o = 0.992; the exponent must be ln (0.992) = -0.008.

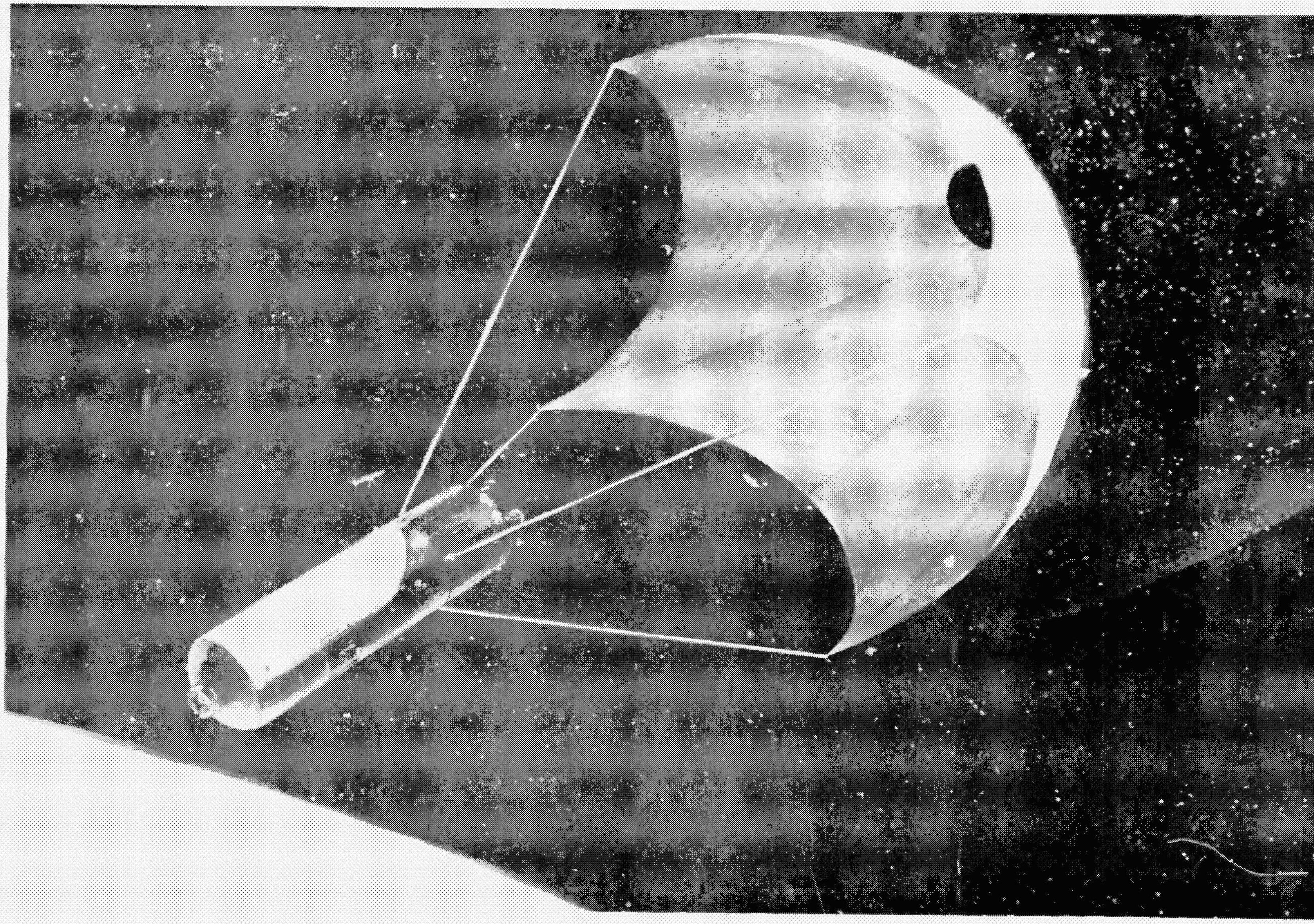


Figure 1-12 Aerobraking Concept

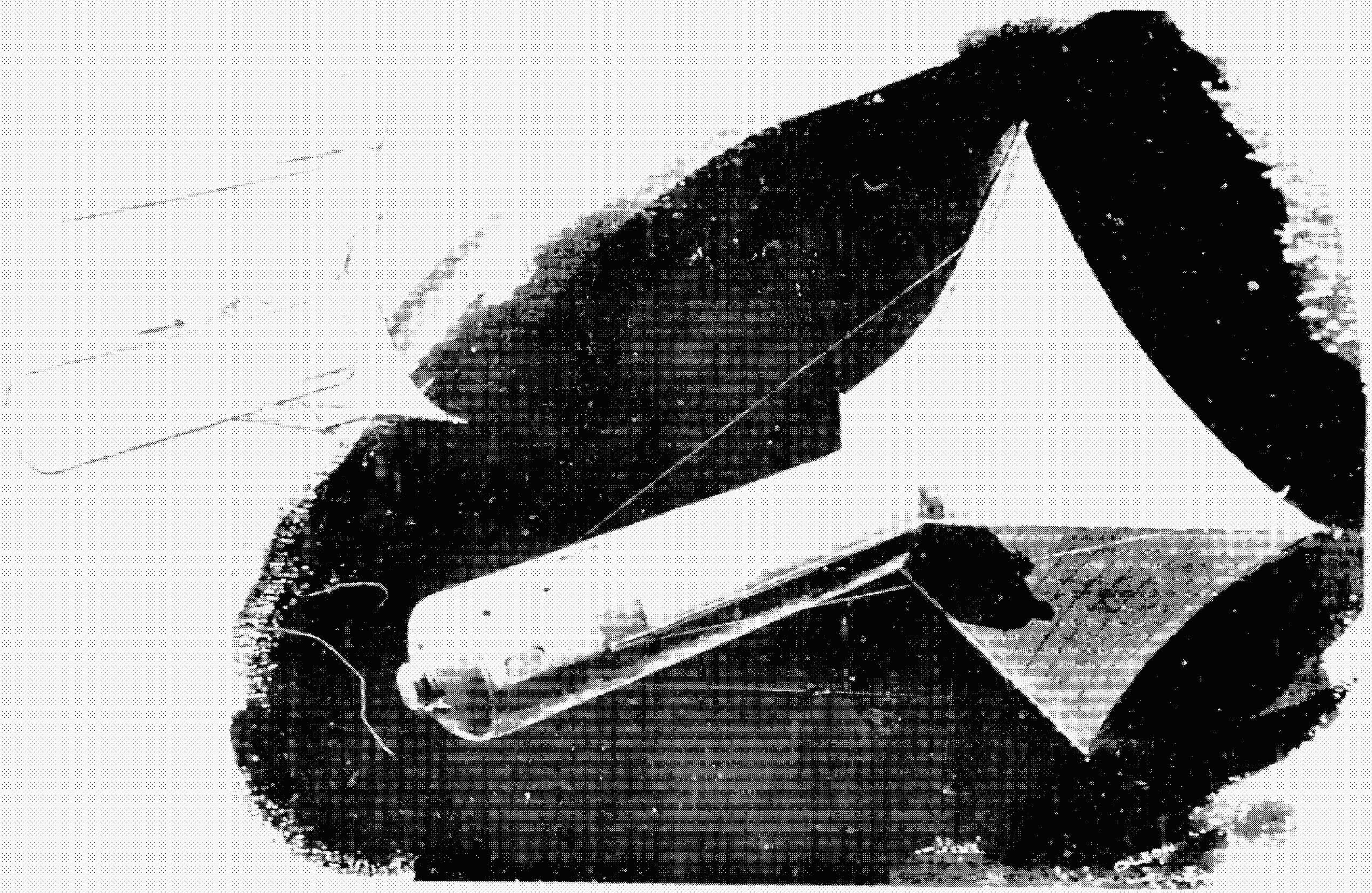


Figure 1-13 Aerobraking Concept

Therefore, $\frac{C_D A\rho}{2M} = \frac{-0.008}{[2\pi r_p H(\frac{e+1}{e})]^{1/2}}$

where $r_p \approx 6.468 \times 10^6$ m (3,492 n.mi.)

$H \approx$ 7,900 m (26,000 ft)

$e \approx 0.734$

Peak acceleration is found to be -0.98 m/sec^2 or about 1/10 g. Also, note that the total effect is about equivalent to the peak deceleration acting for 81/0.98 = 82 sec, a value used to estimate heating. The mass of the example was 13 000 kg (28,660 lb). The peak deceleration load is about 13 000 n (2,900 lb).

Heating–It is estimated that large deployable aerobrakes (if they work) could increase $C_D A/M$ by as much as 10 compared to the metal drag brakes depicted in the referenced study. The heating rates would also be decreased by nearly 10, leading to temperature reductions on the order of $\sqrt[4]{10}$ or 1.7. Equilibrium radiative temperature estimates are shown in table 1-5.

Table 1-5. Aerobraking Temperatures

AREA	REFERENCE STUDY		REDUCED	
	^oK	^oF	^oK	^oF
NOSE	1303	(1886)	767	(920)
SIDEWALLS	706	(812)	416	(288)
SKIRT	633	(680)	373	(211)

Thus, aluminum sidewalls and a Nomex fabric aerobrake may be feasible. The nose temperature appears too high for aluminum. The heating rate is approximately σT^4. Thus, $q \approx 20$ kw/m^2 (317 Btu/hr-ft^2) for 82 sec, a total of 1.64×10^6 joules/m^2 = [illegible].2 Btu/ft^2. For aluminum with specific heat 0.225 and density of 2.7 kg/1, (168 lb/ft^3), the computed temperature rise is about 200^{o}K (360^{o}F) for a 3 mm (1/8 inch) thick heat sink. An aluminum heat sink (non load-bearing) may be sufficient. The 3.175 mm (1/8 inch) shield will have a mass of about 130 kg (287 lb).

1.1.4 Electric Tugs

Recent studies of electric propulsion have emphasized solar photovoltaic panels as a source of electric power. Solar electric propulsion system (SEPS), aided by a chemical rocket boost to Earth escape, were shown to have significant potential for difficult interplanetary missions such as cometary intercepts. More recently, SEPS vehicles have been studied for use as low orbit to geosynchronous orbit tugs. They are expected to experience problems operating in the high flux regions of the van Allen radiation belts due to degradation of solar cells. Accordingly, studies of SEPS systems have emphasized chemical propulsion to a SEPS initiation altitude of about 13 000 km (7015 nm). For transfers from low Earth orbit to synchronous orbit or lunar orbit, however, the transfer to 13 000 km (7015 nm) represents roughly 70 percent of the mission ΔV. This reduces the benefit of the high specific impulse performance of SEPS. In this study, for those missions requiring delivery of large payloads, nuclear-electric tugs and solar-thermal SEPS have been considered.

Nuclear Electric Tugs—A gas cooled nuclear reactor driving a closed-cycle Brayton power conversion system was selected for concept analysis. This system provides good conversion efficiency, avoids high temperature liquid metals, and does not require zero-g phase changes. Like the solar/thermal system, nuclear/Brayton is favored in large sizes. A detailed analysis was beyond the scope of the study, but rough-order of magnitude estimates were made.

Design Considerations—A nuclear system becomes rapidly more efficient at larger sizes because the shield thickness, to first order, is constant as power increases. Hence for a very small reactor, the shield weight overwhelms all else, while for a very large reactor the shield weight is modest by comparison to other items. The selected shielding approach provided enough shielding that the system could be utilized for manned operations without exclusion areas or special operating procedures.

The design of a nuclear system is a highly complex matter. In this analysis we did not consider reactor details such as fuel form, neutron spectrum, or reactivity control, and did not analyze safety systems such as emergency core cooling, scram and restart, afterheat removal, etc. We were concerned primarily with gross performance estimates as a function of size.

The shielding provisions are a primary contributor to system mass. The primary (helium) loop will become radioactive and must be contained within an outer gamma shield. Inside the outer (secondary) gamma shield are the primary loop and a neutron shield, which could be borated water. Inside the neutron shield is a primary gamma shield. It must be a dense material that does not become highly activated by neutrons. Lead may be a suitable material if it can be adequately cooled; zirconium is a possible alternate. Within the inner gamma shield are the neutron reflector and control systems and the reactor itself.

The geometry is schematically shown in figure 1-14. Dimensions in the figure are for a 20-megawatt (jet power) system.

Systems with beam (jet) power of 5, 20, and 50 megawatts were examined. Representative efficiencies and power levels are shown in table 1-6.

Table 1-6 Nuclear Electric Tug Power Estimates

	Efficiency	Power		
Thermal Power		25	102	254
to	.38			
Shaft Power		10	39	96
to	.96			
Electrical Bus		9	37	93
to	.90			
Conditioned Power		8	33	83
to	.60			
Jet (Beam) Power		5 MW	20 MW	50 Mw

$\eta = .197$

Reactor power output on the order of 50 MW/m^3 (1.4 MW/ft^3) is reasonable. Hydrogen-heater rocket reactors have reached over 3000 MW/m^3 (85 MW/ft^3) and contemporary high temperature gas reactor (HTGR) designs provide about 8 MW/m^3 (0.2 MW/ft^3).

Table 1-7 indicates shield masses and dimensions. These shield dimensions are rough estimates of requirements for operation in the vicinity of manned systems. Refined estimates might yield appreciable changes but the data shown are sufficient to indicate general trends. A system schematic is shown in figure 1-15. This schematic is basically the same as the Brayton HTGR presently being studied for utility power applications.

Except for the shield, the thermal radiator is likely to be the most massive item in the power plant system. Figure 1-16 presents representative radiator parameters as a function of radiator effective temperature. Effective temperature may be approximated as:

$$T = T_1 \left[\frac{(T_1/T_2)^2 - 1}{2/3\,[(T_1/T_2)^3 - 1]} \right]$$

Where T_1 and T_2 are radiator inlet and outlet temperatures, respectively. Higher radiation temperatures clearly lead to lower weights, but also to more difficult materials technology.

Table 1-7 Shield Characteristics

Element	Inside Dia. m (ft)	Inside Length m (ft)	Thickness m (ft)	Unit Mass g/cm^2 (lbm/ft^2)	Volume m^3 (ft^3)	Density kg/l (lbm/ft^3)	Mass Mass Metric tons (lbm)
5 MW Jet Power							
Neutron	.75	1.13	.15	30	.739	2	1.48
Reflector	(2.46)	(3.71)	(.49)	(61.4)	(26.10)	(125)	(3,260)
Primary Gamma	1.05	1.43	.10	100	.762	10	7.62
Shield	(3.44)	(4.69)	(.33)	(204.8)	(26.91)	(624)	(16,800)
Neutron	1.25	1.63	1.0	200	28.11	2	56.22
Shield	(4.10)	(5.35)	(3.28)	(409.6)	(992.7)	(125)	(123,940)
Secondary	3.25	5.63	.10	100	7.79	10	77.95
Gamma Shield	(10.66)	(18.47)	(.33)	(204.8)	(275.1)	(624)	(171,850)
TOTAL MASS							143.27 (315,850)
			= 28.65 kg/kw (63.16 lbm/kw) of jetpower				
20 MW Jet Power							
Neutron	1.20	1.80	.15	30	1.80	2	3.596
Reflector	(3.94)	(5.91)	(.49)	(61.4)	(63.6)	(125)	(7.928)
Primary Gamma	1.50	2.10	.10	100	1.506	10	15.056
Shield	(4.92)	(6.89)	(.33)	(204.8)	(53.18)	(624)	(33,192)
Neutron	1.70	2.30	1.0	200	40.96	2	81.920
Shield	(5.58)	(7.55)	(3.28)	(409.6)	(14.46)	(125)	(180,600)
Secondary	3.70	6.30	.10	100	9.90	10	99.0
Gamma Shield	(11.15)	(20.67)	(.33)	(204.8)	(350)	(624)	(218,300)
TOTAL MASS							199.6 (440,000)
			= 9.98 kg/kw (22 lb/kw) of jetpower				
50 MW Jet Power							
Neutron	1.62	2.43	.15	30	2.90	2	5.80
Reflector	(5.31)	(7.97)	(.49)	(61.4)	(102.4)	(125)	(12,790)
Primary Gamma	1.92	2.73	.10	100	2.44	10	24.40
Shield	(6.30)	(8.96)	(.33)	(204.8)	(86.2)	(624)	(53,790)
Neutron	2.12	2.93	1.0	200	55.38	2	110.76
Shield	(6.96)	(9.61)	(3.28)	(109.6)	(1955.7)	(125)	(244,180)
Secondary	4.12	7.93	.10	100	13.44	10	133.44
Gamma Shield	(13.52)	(26.02)	(.33)	(204.8)	(474.6)	(624)	(294,180)
TOTAL MASS							274.40 (605,000)
			= 5.49 kg/kw (12.10 lb/kw) of jetpower				

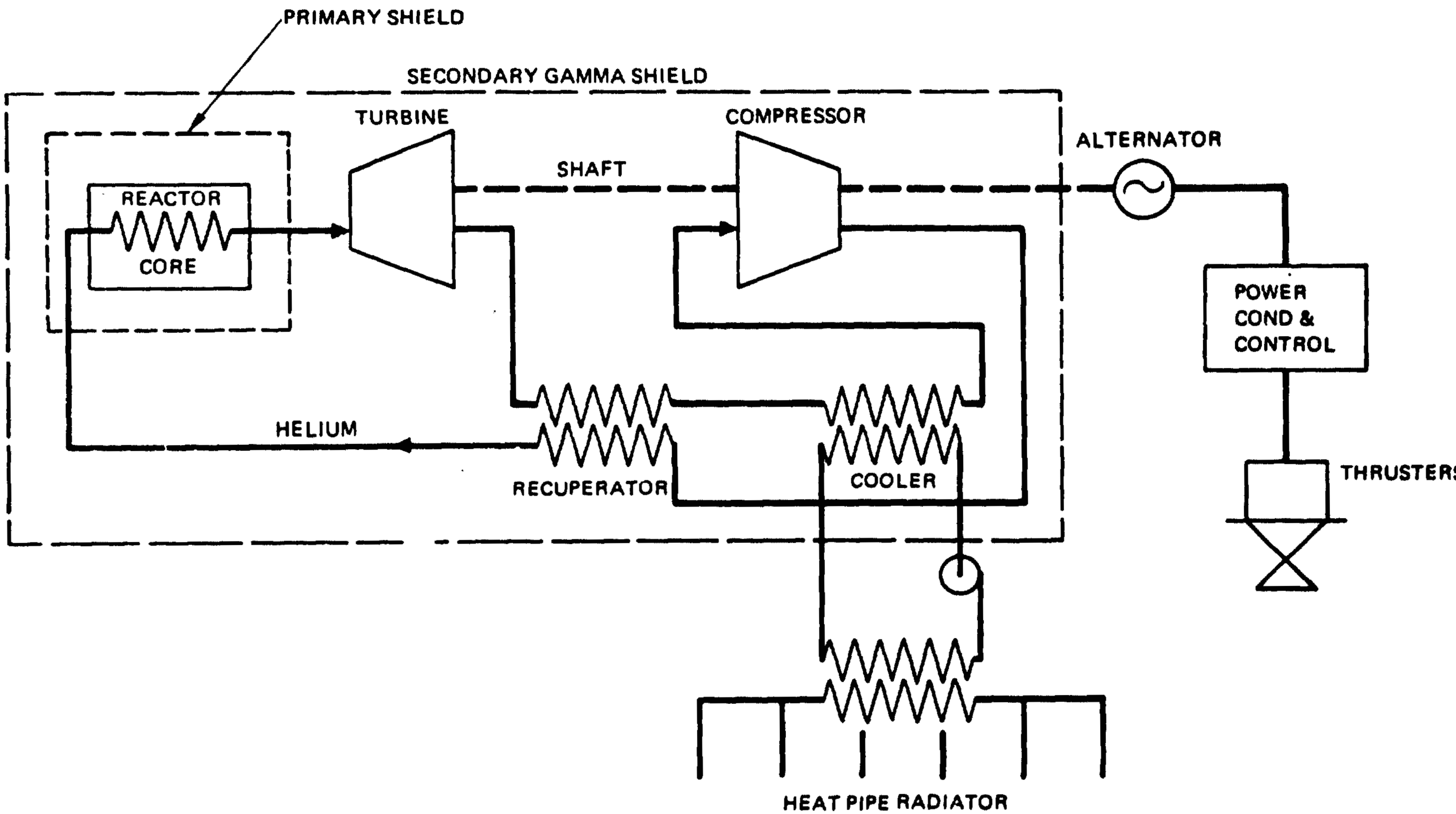


Figure 1-15 Power System Schematic

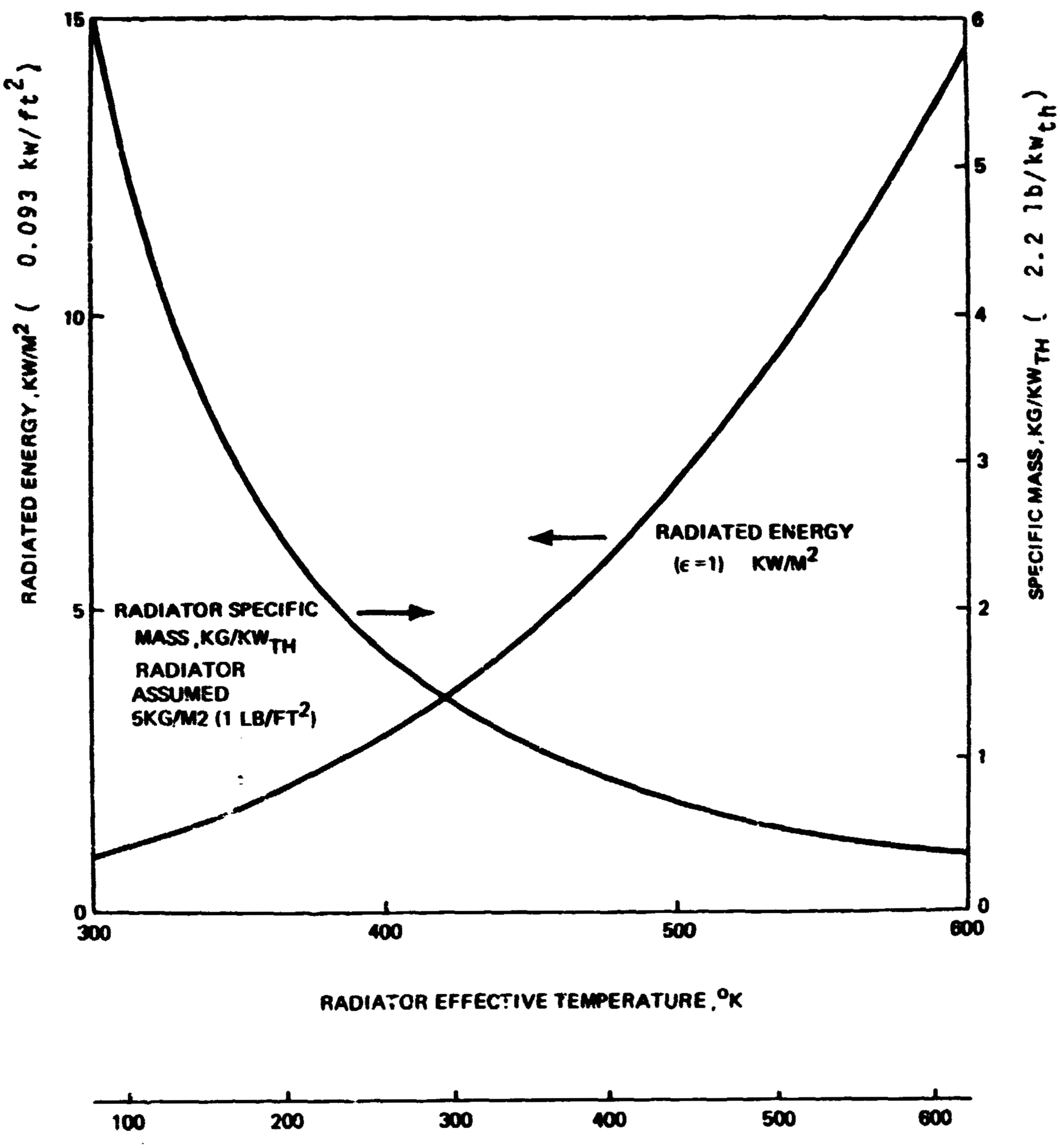


Figure 1-16 Radiator Performance Parameters

Effective temperatures of approximately 390°K (702°R), corresponding to a turbine inlet temperature of 1100K (1520°F), appear reasonable, resulting in a specific mass of 2 kg/kw$_{th}$ (4.4 lb/kw$_{th}$), equivalent to 6.2 kg/kw (13.6 lb/kw) beam power. Table 1-8 summarizes specific weights for the three reactor electric propulsion systems.

Table 1-8

Specific Masses in kg/kw (lb/kw) of Jet Power

	5 MW	20 MW	50 MW
Reactor	0.5 (1.1)	0.5 (1.1)	0.5 (1.1)
Shield	25.5 (56.2)	9.98 (22)	5.1 (11.2)
Turbo machinery and Alternator	2.78 (6.13)	2.78 (6.13)	2.78 (6.13)
Power Conditioning	1.0 (2.2)	1.0 (2.2)	1.0 (2.2)
Thrusters	1.5 (3.3)	1.5 (3.3)	1.5 (3.3)
Radiators	6.2 (13.7)	6.2 (13.7)	6.2 (13.7)
Total	37.48 (82.63)	21.96 (48.43)	17.1 (37.63)
Total Mass kg(lb)	187,400 (413,150)	439,200 (968,600)	855,000 (1,881,500)

Total masses do not include propellant tankage or feed systems.

Operational Considerations–The reactor systems described included enough shielding that special operational procedures with man, such as exclusion areas, would not be required. The 5 and 20 megawatt systems could be launched as subassemblies by the Shuttle and assembled in orbit. Radiators will present the principal assembly challenge; the smallest vehicle requires 6200 m^2 (67,000 ft^2) and the largest vehicle, 62 000 m^2 670,000 ft^2). By comparison a football field is 4050 m^2 (45,000 ft^2). Figure 1-17 shows the 5 MW system, with the shuttle for size comparison.

A reasonable design life target for the reactor is 20,000 hours; this represents a fuel burnup of roughly 40,000 MWD$_{th}$/ton. 20,000 hours will encompass 8 to 10 round trips to geosynchronous or lunar orbit, or one Mars round trip. At the end of life, the reactor must be refueled and refurbished. The masses of the 5 and 20 MW reactors with neutron reflector and inner gamma shield are such that they could be returned to Earth for service. (After a reasonable cooldown period the inner gamma shield should provide sufficient protection from residual radioactivity for handling in the

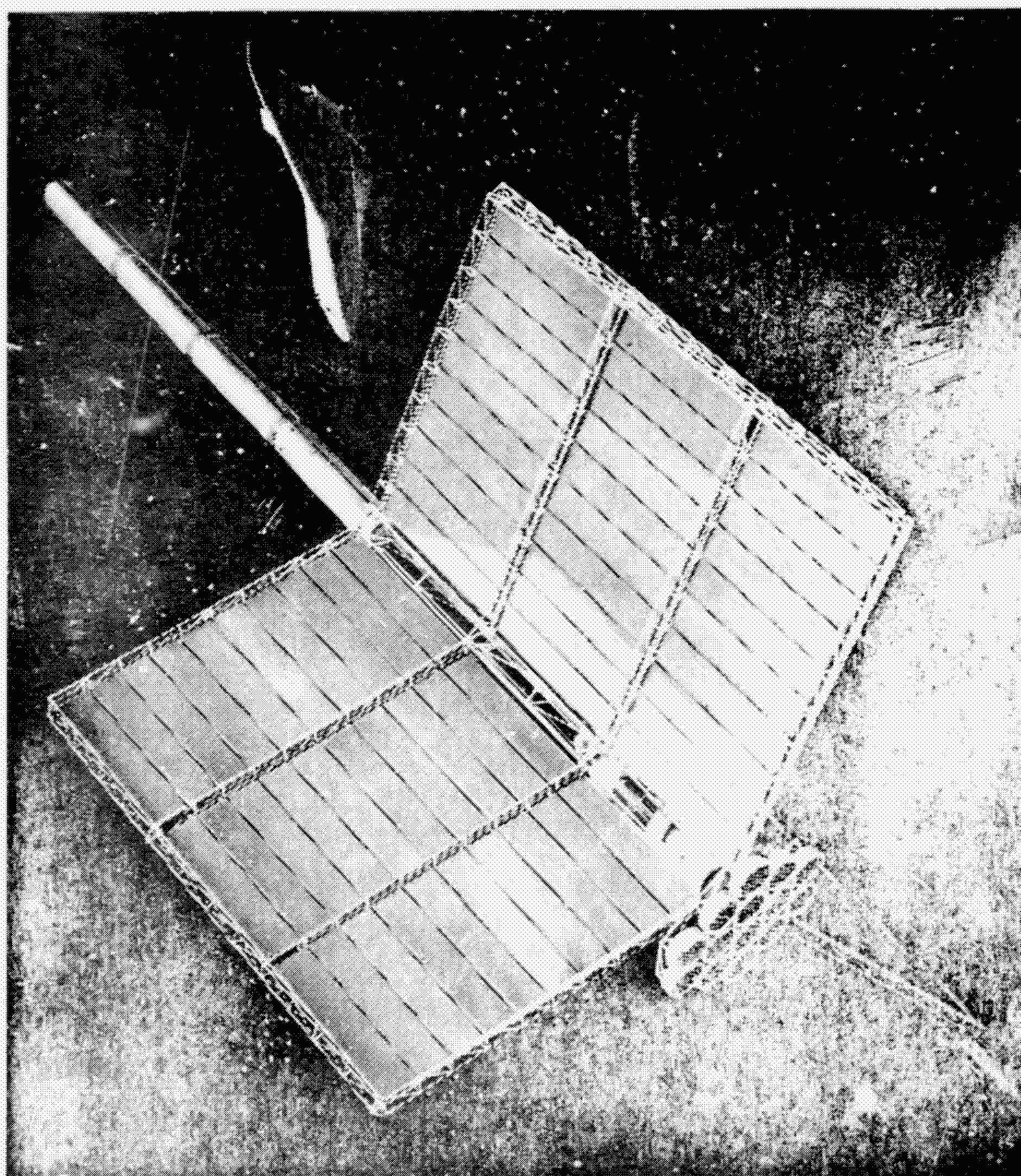


Figure 1-17 5MW Nuclear Electric Tug

cargo bay.) The corresponding mass for the larger reactor is roughly 54 000 kg (120,000 lb). An on-orbit service facility with remote handling and remote manipulators would seem to be required for servicing large reactors. Spent fuel elements would be returned to Earth in shielded containers by several Shuttle flights. Unused (new) fuel elements do not require shielding. If they contain plutonium, special handling is required because of the extreme toxicity of that material.

Solar-Thermal Electric Propulsion System (STEPS) Tug—A solar electric tug employing optical concentrators and heat engines (Brayton turboalternators) for electric power generation provides a potential way of circumventing the major problems associated with solar photovoltaic generation (cannot fly through the van Allen belts) or nuclear generation (radiation hazard and spent reactor disposal). STEPS systems require intermediate-precision pointing (about 10 arc-minutes) to get the concentrated energy into the cavity, and like solar photovoltaic systems, will experience frequent stop/starts in low orbits due to occultation by the Earth.

Two sizes of STEPS were investigated. The smaller size, 50 kw jet power, is too small to employ in a practical manner the multiple-facet stretched plastic film reflector baselined for thermal engine power satellites, so was assumed to use a much heavier unit weight semirigid reflector. The larger size, 20 megawatts jet power, employs the plastic-film facet approach.

A concept of the small STEPS is shown in figure 1-18. Two 40 kw power generation modules are employed. Each module consists of an 12.6m (41 ft) semirigid reflector, support arms, cavity thermal absorber, closed cycle Brayton turboalternator with heat exchangers, and thermal radiators. The semirigid reflector is assembled in orbit from six pie-shaped sections, each formed from graphite face, aluminum core, epoxy-bonded honeycomb, with a 0.13 mm (.005 inch) polished aluminum metal reflector face bonded to the inside graphite face sheet. The estimated unit mass is 5 kg/m^2 (1.02 lb/ft^2). The pie-shaped sections are sized for the shuttle payload bay. A nominal power budget is shown in table 1-9. The thermal efficiency shown is consistent with 1100K (1520°F) turbine inlet temperature.

The estimated specific mass of the basic power unit is therefore 55 kg/kw_j. This estimate does not include propellant, propellant tankage, propellant feed, or payload mass.

Since the small STEPS does not have individually-controlled facets, each module, perhaps the entire vehicle, must be sun-oriented to within about ten arc-minutes.

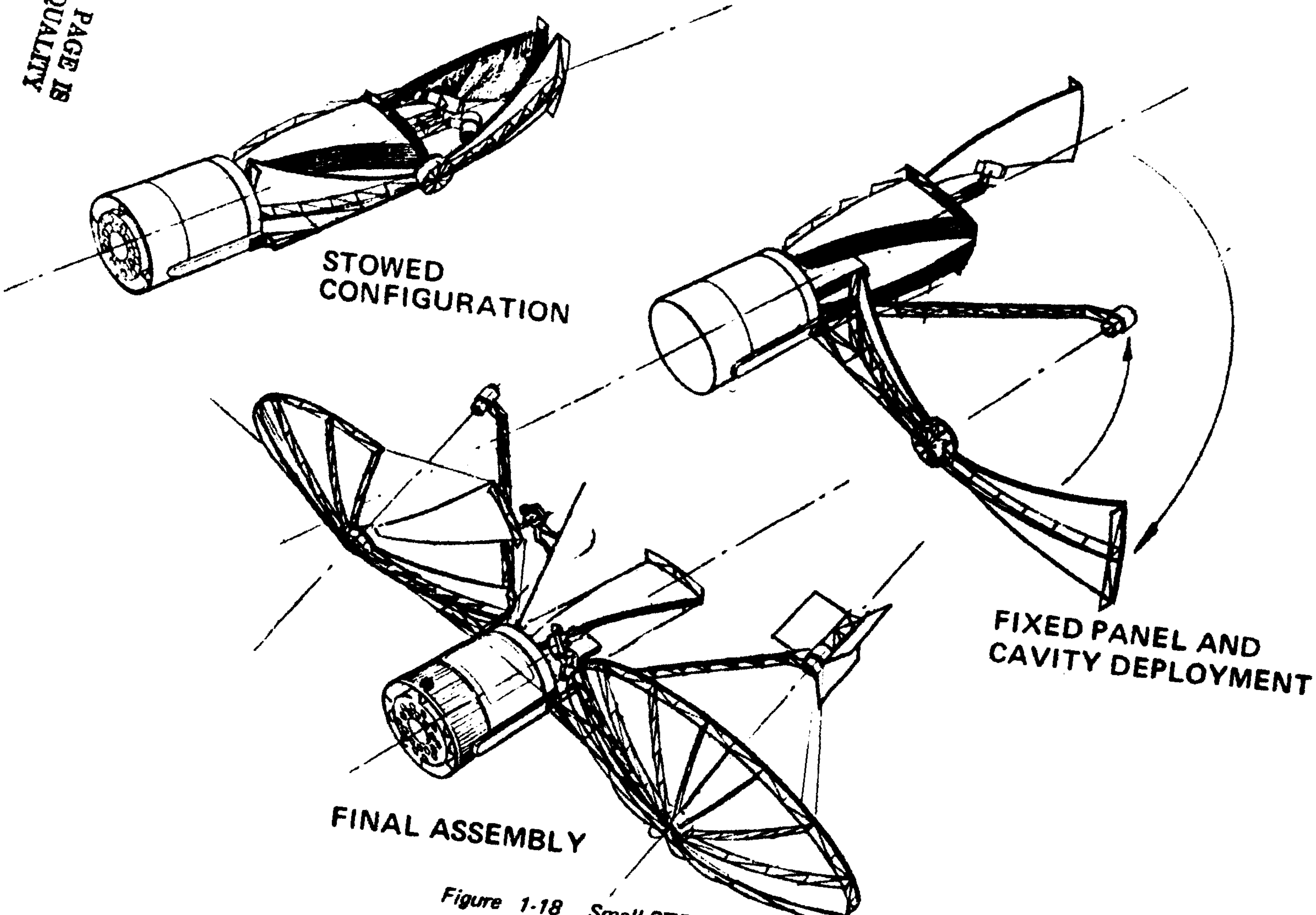


Figure 1-18 Small STEPS Deployment

Table 1-9 Steps Power Budget

	Efficiency	Power Per Module (kw)	Total (kw)
SOLAR FLUX		88	350
TO	.8		
ENERGY INTO CAVITY		70	280
TO	.85		
THERMAL POWER		60	240
TO	.35		
SHAFT POWER		21	84
TO	.95		
ELECTRIC POWER		20	80
TO	.90		
CONDITIONED POWER		18	72
TO	.70		
JET POWER		12.5	50
	η = 14% overall		

A preliminary mass estimate is given in Table 1-10.

Table 1-10 Small STEPS Mass Estimate

ITEM	ESTIMATING BASIS	MASS KG	MASS LB
REFLECTOR	5 kg/m^2 (1 lb/ft^2)	1244	2742
CAVITY	.3 kg/kwt (.66 lb/kwt)	72	159
TURBOALTERNATOR	2.5 kg/kw_e (5.5 lb/kw_e)	200	441
RADIATOR	4 kg/kw_e (8.8 lb/kw_e)	320	705
POWER COND	1.5 kg/kw_e (3.3 lb/kw_e)	120	265
THRUSTERS	2.5 kg/kw_e (5.5 lb/kw_e)	200	441
STRUCTURES SUBSYSTEMS	Typ. of small satellites	600	1322
TOTAL		2756	6075

The large STEPS is depicted in figure 1-19. Its power budget and mass estimates are given in table 1-11, and 1-12. The thermal efficiency is consistent with a 1300K (1880°F) turbine inlet temperature. The thruster efficiency corresponds to a hypothetical thruster, possibly an MPD type, designed to use a common material as propellant.

Table 1-11 Large STEPS Power Budget

	Efficiency	Power (Mv)
SOLAR FLUX		143
TO	.8	
ENERGY INTO CAVITY		115
TO	.85	
THERMAL POWER		97
TO	.40	
SHAFT POWER		39
TO	.95	
ELECTRIC POWER		37
TO	.90	
CONDITIONED POWER		33
TO	.60	
JET POWER		20
	η = 14% overall	

Table 1-12 Large STEPS Mass Estimate

ITEM	ESTIMATING BASIS	MASS KG	MASS LB
REFLECTOR	0.6 kg/m^2	61,500	135,600
CAVITY	0.2 kg/kwt	19,400	42,800
TURBOALTERNATOR	2 kg/kw_e	74,000	163,000
RADIATOR	2 kg/kw_e	74,000	163,000
POWER COND	1 kg/kw_e	37,000	81,500
THRUSTERS	2 kg/kw_e	74,000	163,000
STRUCTURES & SUBSYSTEMS	Equal to reflector	61,500	135,600
		401,400 kg	884,500

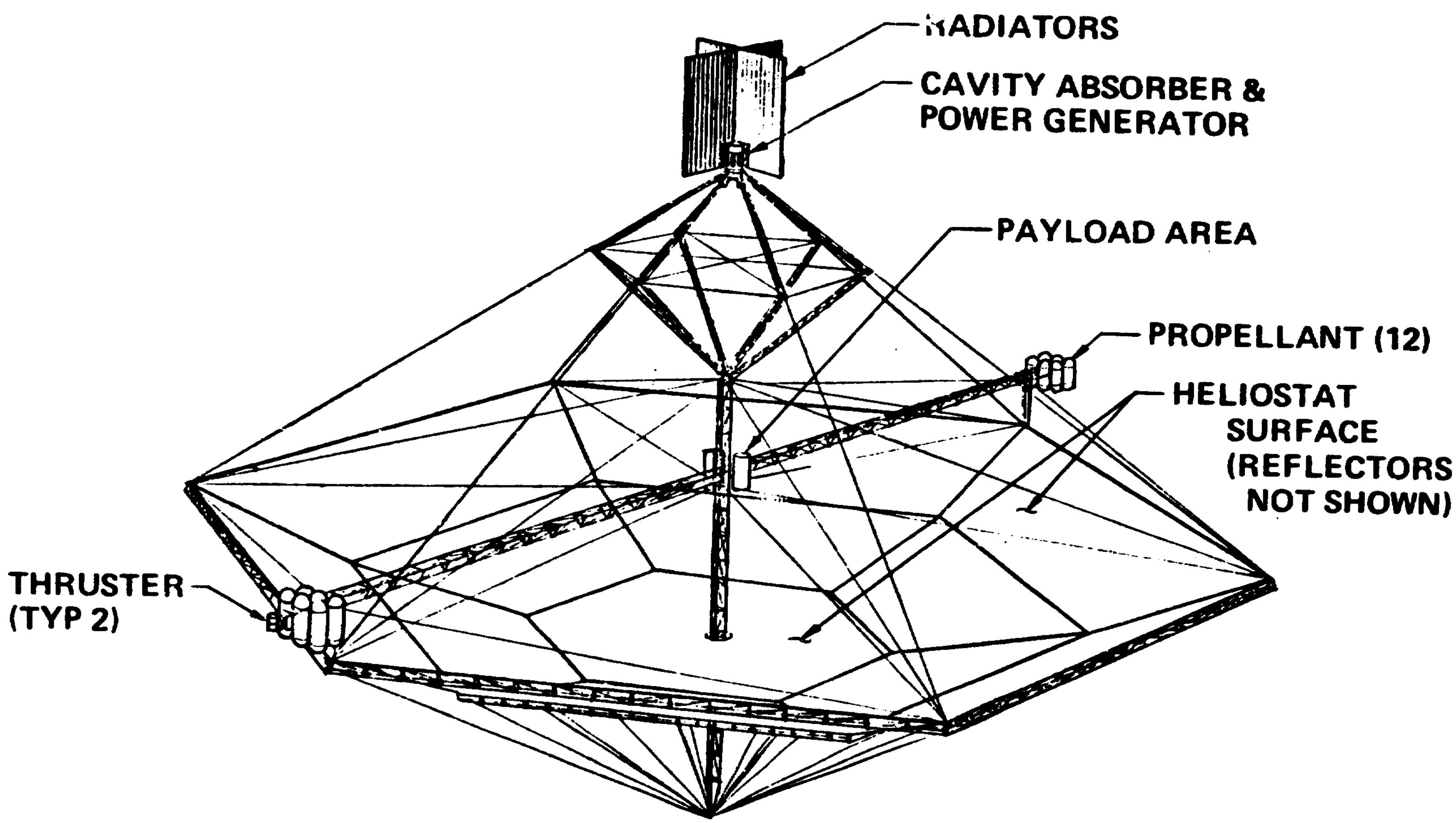


Figure 1-19 Solar Thermal Concentrator Electric Tug

The large STEPS is estimated to have a lower unit weight, 20 kg/kw_j (44 lb/kw_j) than the NET. Since the STEPS, however, experiences occultation for about 10 percent of the LEO-GEO trip its performance would be essentially equal to the NET. Either the large STEPS or the NET should be designed to use readily available material as propellants. The quantities of propellant required for the class of program requiring these vehicles would place a severe strain on available resources of scarce material frequently considered as ion propellants, such as mercury or cesium.

1.2 LUNAR TRANSPORT VEHICLES

The lunar transport vehicles (LTV) differ from the orbit transfer vehicles (OTV) primarily in that they require structural accommodations for lunar landing. These include landing legs and structure for crew egress, payload support and payload handling. Whereas the OTV's are mated to the crew transfer vehicles (CTV) by docking structures, the LTV's are fixed to the crew/equipment modules (CEM). The LO_2/LH_2 LTV baseline chosen for the single stage parametric weight analysis is represented by figure 1-20. A thrust-to-weight ratio of 0.3 was selected for the parametric analysis. Figure 1-21 shows single stage LO_2/LH_2 LTV weights vs. usable propellant. It is assumed that the guidance, navigation and primary communications and power components are in the crew/equipment module rather than the LTV. A single-stage lunar transport mode that is applicable to the Independent Lunar Surface Sortie missions employs jettisoning the separable lunar landing components (legs, descent-only payload provisions, ladder, etc.). A separate plot on figure 1-21 shows the weight of these components.

Figures 1-22 and 1-23 show the weights of ascent and descent stages for the LTV two stage transport mode. The ascent stage contains all of the APS fixed components and propellant.

Figure 1-24 shows the weights of the LO_2/LH_2 LTV 1-1/2 stage vehicles. The main stage is essentially the same as the single stage, with the addition of provisions for additional tank stage supports and propellant transfer. The separated tank stage is similar in configuration to the small OTV drop tanks.

The baseline LO_2/LH_2 vehicles suffer from having the crew and crew module relatively far from the lunar surface during landing and surface operations. Alternate arrangements could be employed if the LTV propulsion system is not intended to perform other roles, e.g. small OTV. For example, Rockwell considered placing the crew module at the lower end of the stage, between propellant tanks and engines.

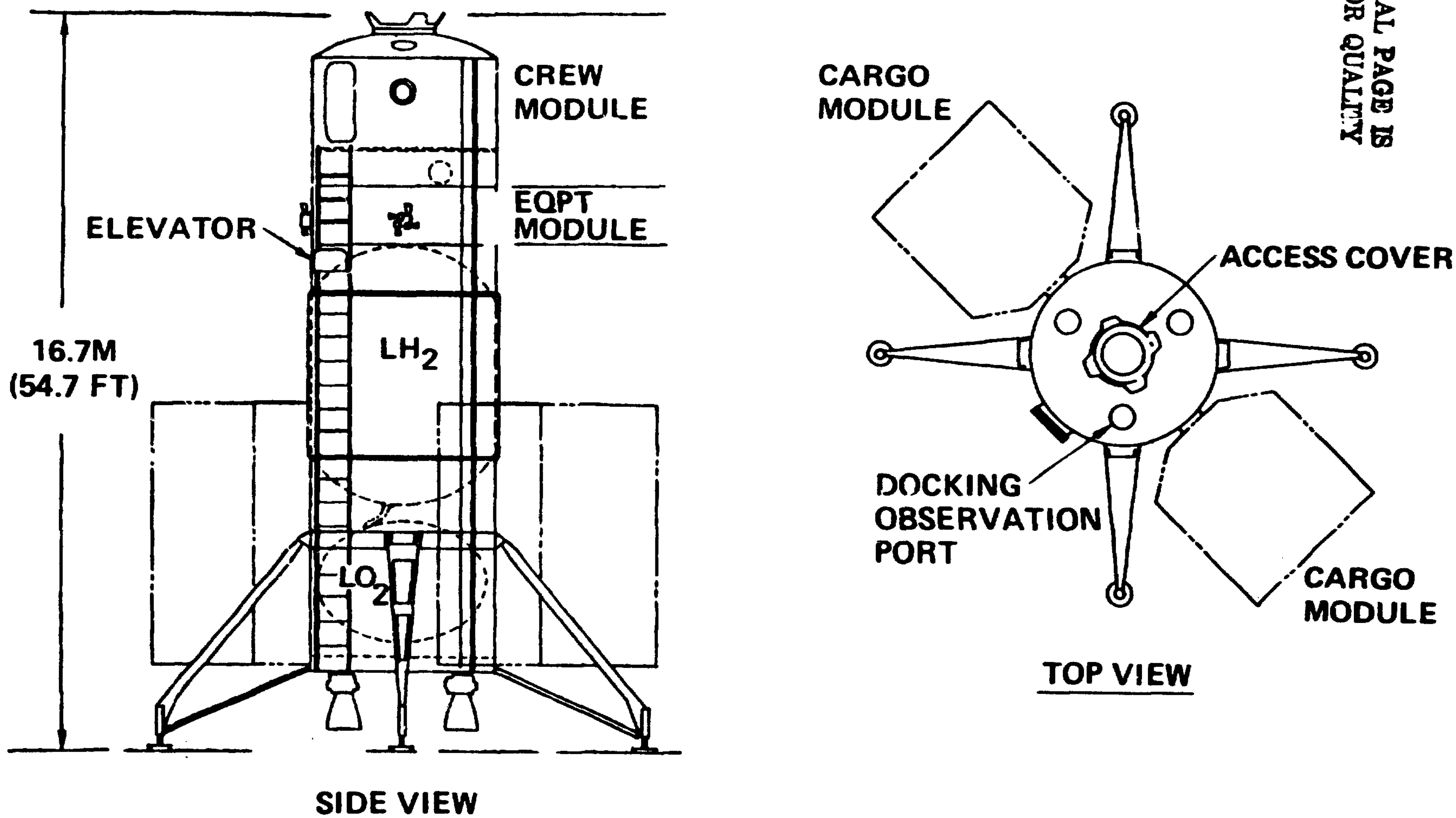


Figure 1-20 Reusable Single Stage L.T.V.

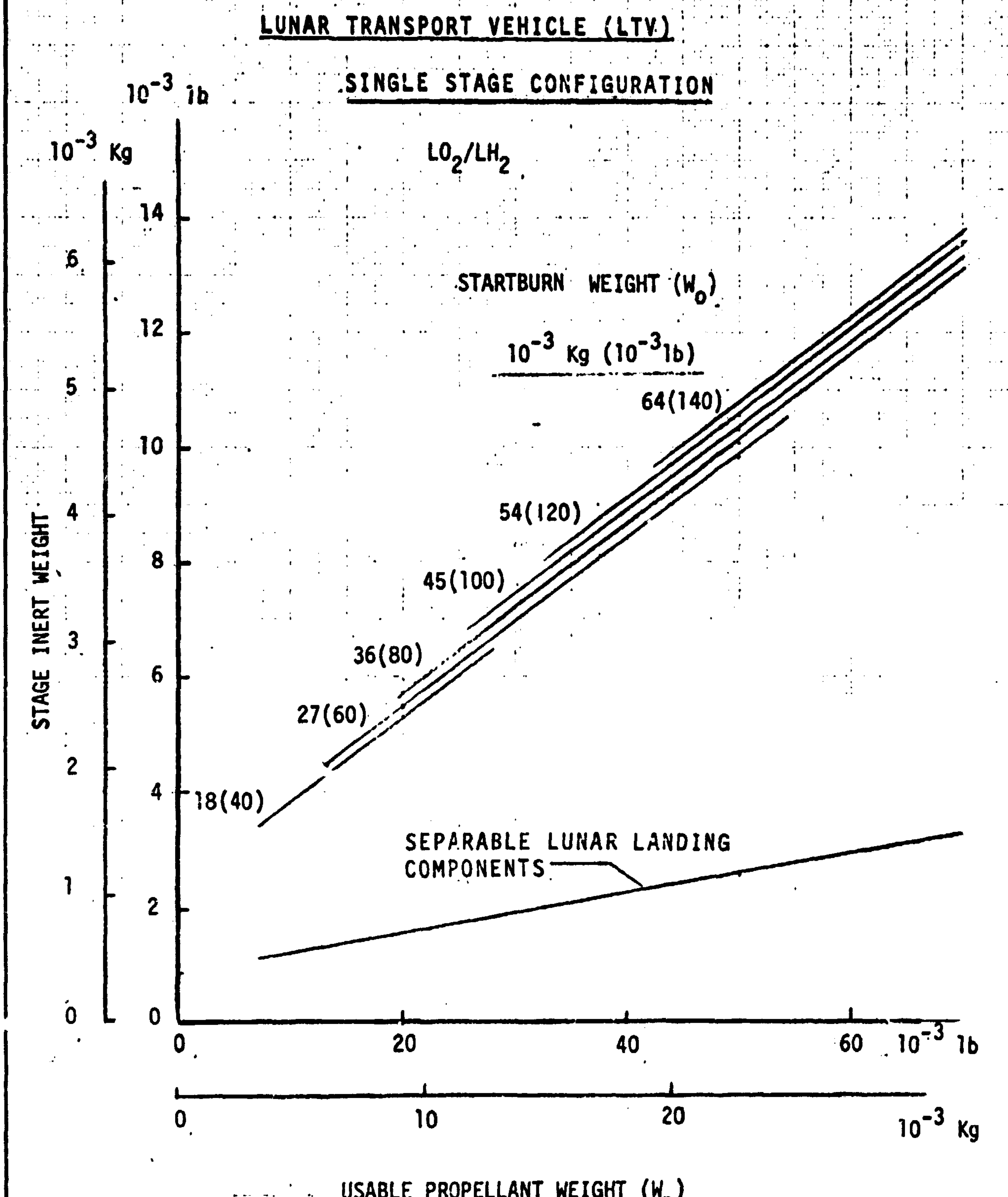
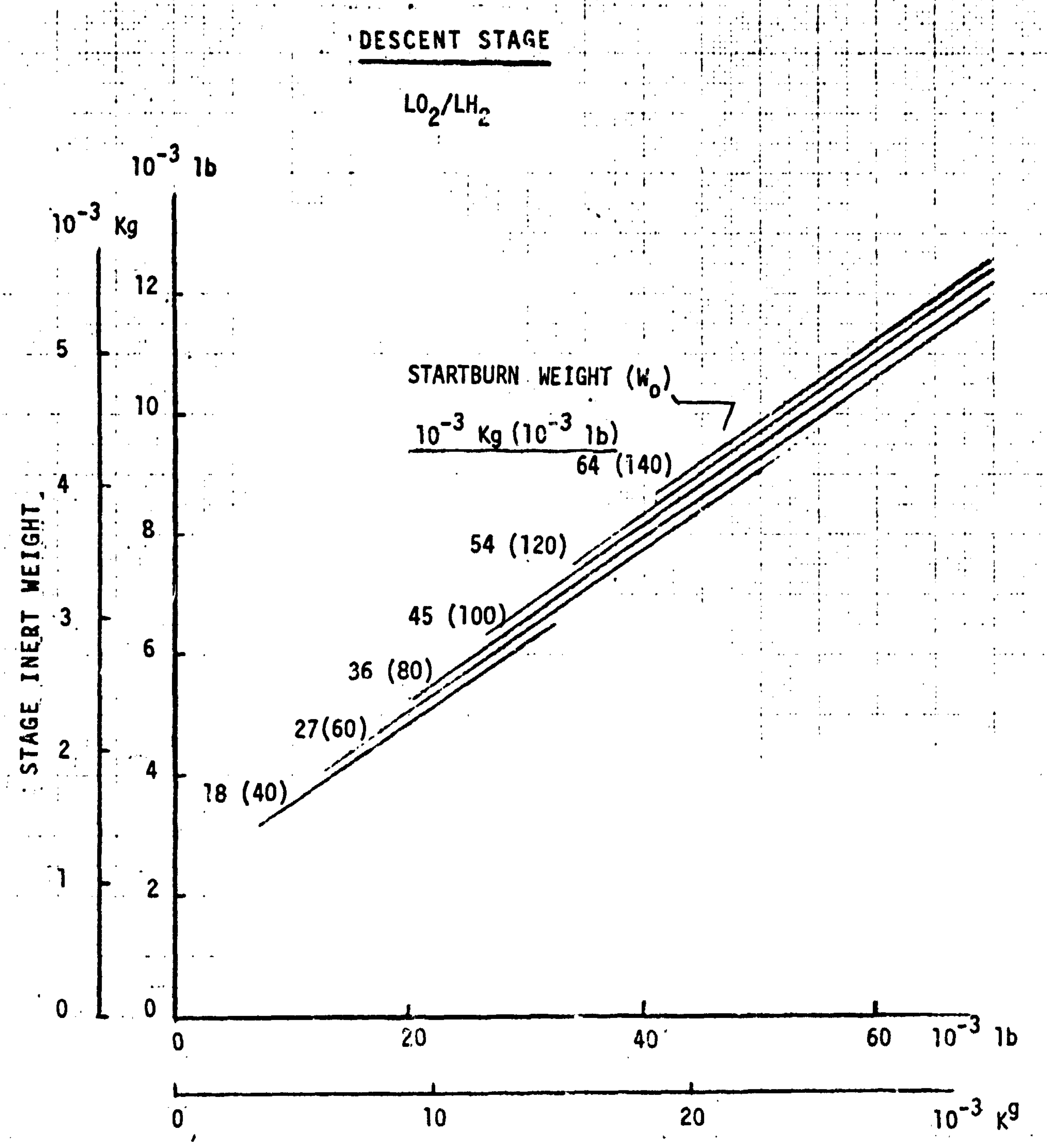


Figure 1-21. LO_2/LH_2 Single Stage Lunar Transport Vehicle Mass



USABLE PROPELLANT WEIGHT (W_p)

FIGURE 1-22 LUNAR TRANSFER VEHICLE (LTV)

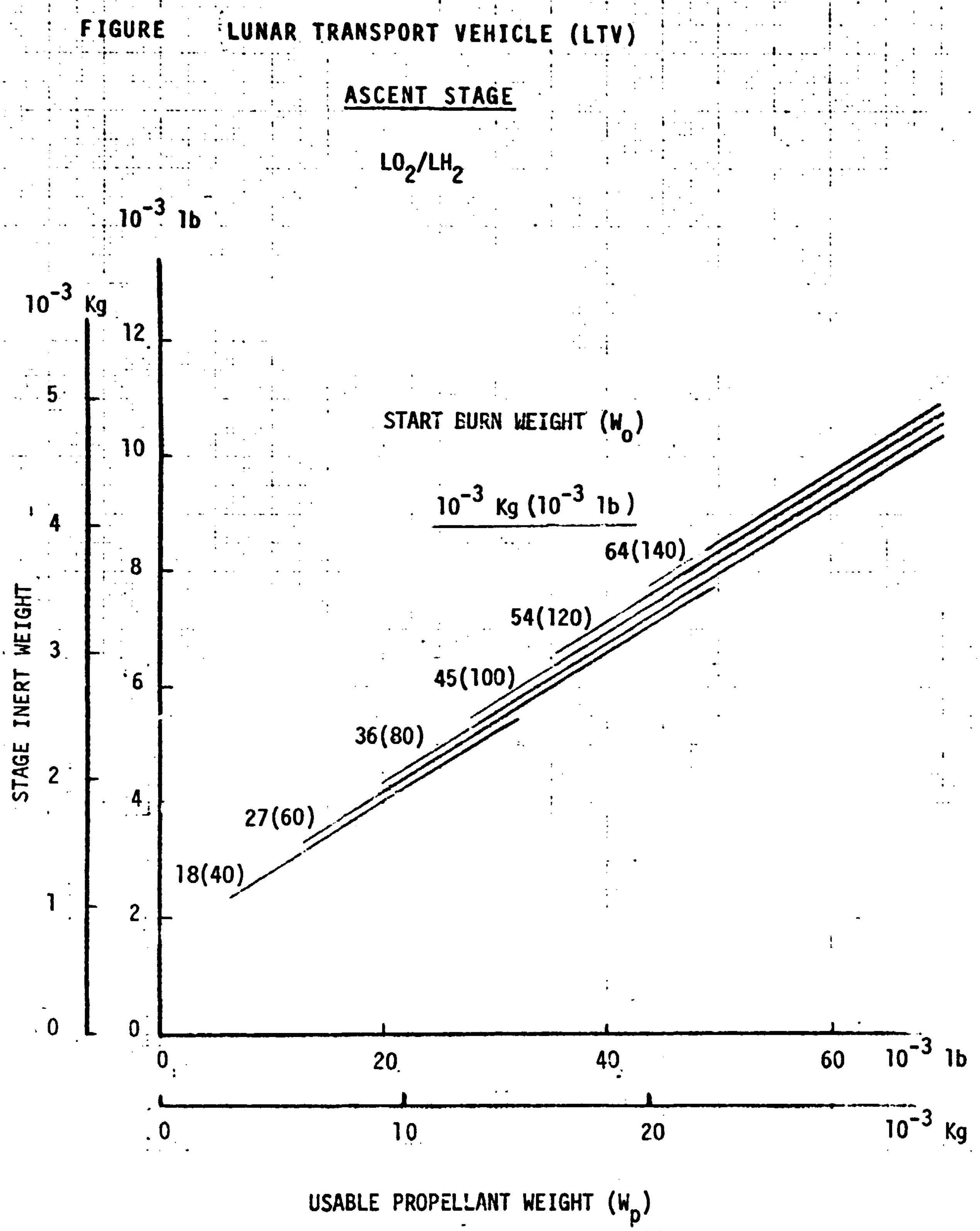


Figure 1-23. Lunar Transport Vehicle Ascent Stage

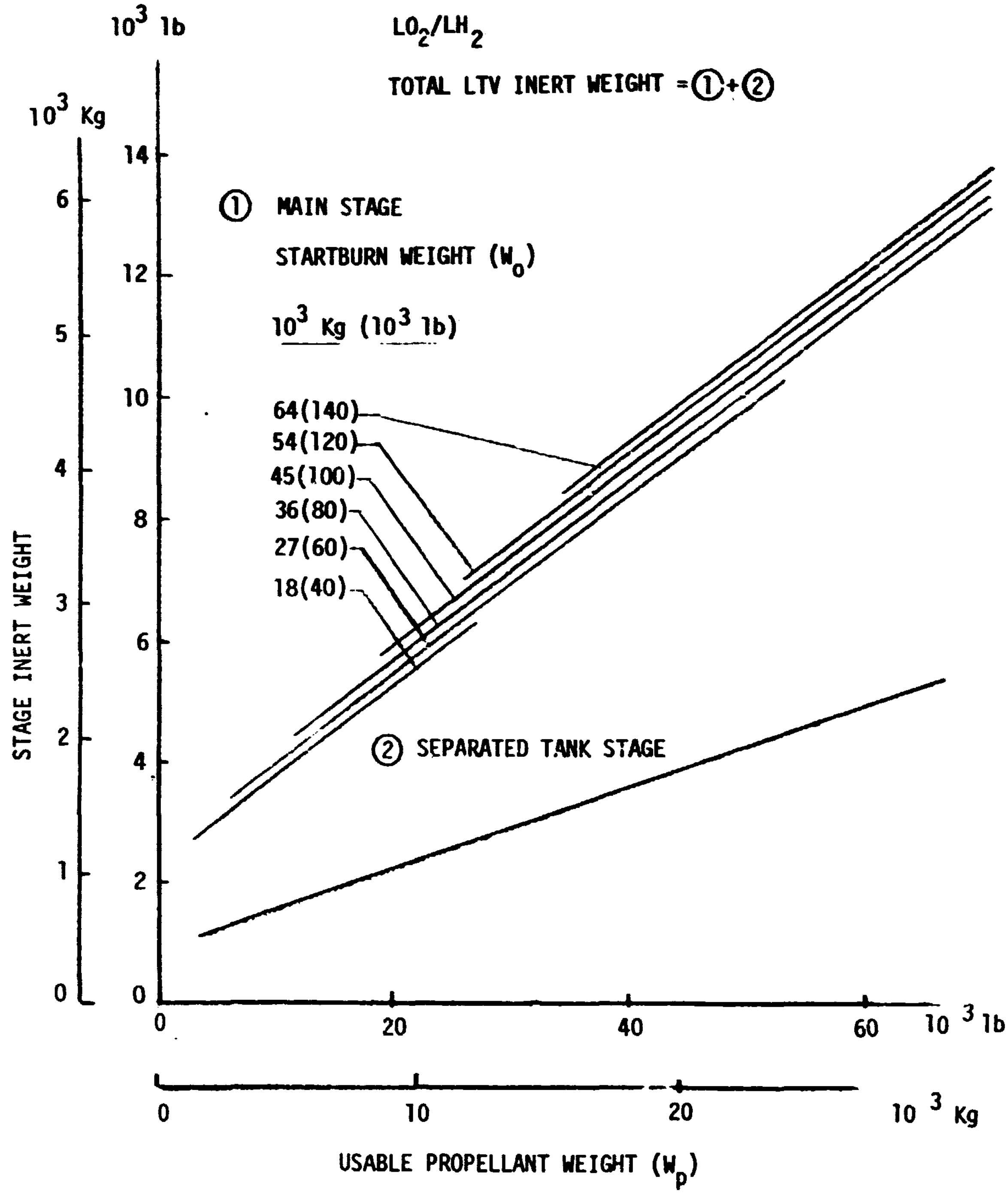


FIGURE 1-24 LUNAR TRANSFER VEHICLE (LTV)
1½ STAGE

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Another alternative is to try to shorten the propulsion system. One arrangement considered briefly is shown in figure 1-25; compared to the baseline. This vehicle, however, would have relatively poor structural efficiency, not a severe penalty on the comparatively low delta V lunar transport mission, but undesirable for alternate uses. The most promising approach at this point appears to be a dense propellant, i.e. LO_2/MMH, LTV. The dense propellant LTV is shown in figure 1-26, and weight estimating parametrics in figure 1-27.

1.3 HEAVY LIFT

1.3.1 Shuttle-Derived Systems

Two options were investigated and are depicted in figures 1-28 and 1-29. The SRB/ET vehicle can use either 2 or 4 SRB's. The all-SRB option is based on a JSC configuration; performance data for this option have been obtained from JSC internal note 74-FM-80 dated November 20, 1974. Performance for the other options was calculated. Data are summarized in table 1-13.

Representative trajectory data for the SRB/ET vehicles are shown in figures 1-30 through 1-33.

Table 1-13 Heavy Lift Options

Performance to 100 n. mi. Orbit, East Launch

Vehicle	GLOW 10^6 KG (10^6 lb)	T/W Liftoff	Max. Q KN/M^2	Staging Velocity (psf) (ft/sec)	Weight in m/sec kg (lb)	Payload kg (lb) Orbit	Propellant left with no Payload
All-SRB, (5 SRB 1st Stage)	3.97 (8.75)	1.77	38 (800)	1 615 (5,300)	78 000 (172,000)	71 000 (156,000)	Not Applic.
2 SRB/ET	2.02 (4.45)	1.49	31 (650)	1 340 (4,400)	152 000 (336,000)	79 000 (175,000)	68,000 (150,000)
4 SRB/ET	3.21 (7.07)	1.711	48 (1,000)	2 060 (6,760)	193 000 (425,000)	120 000 (265,000)	104,000 (230,000)

1.3.2 Low Cost Heavy Lift

The task of transportation to low orbit of many millions of kilograms (pounds) per year for power satellites at low cost is a significant challenge, needing a low cost heavy lift vehicle (LCHLV).

Figure 1-32 shows how a significant performance parameter, the ratio of gross lift off weight (GLOW) to payload delivered relates to GLOW itself, for many of the launch vehicles which have

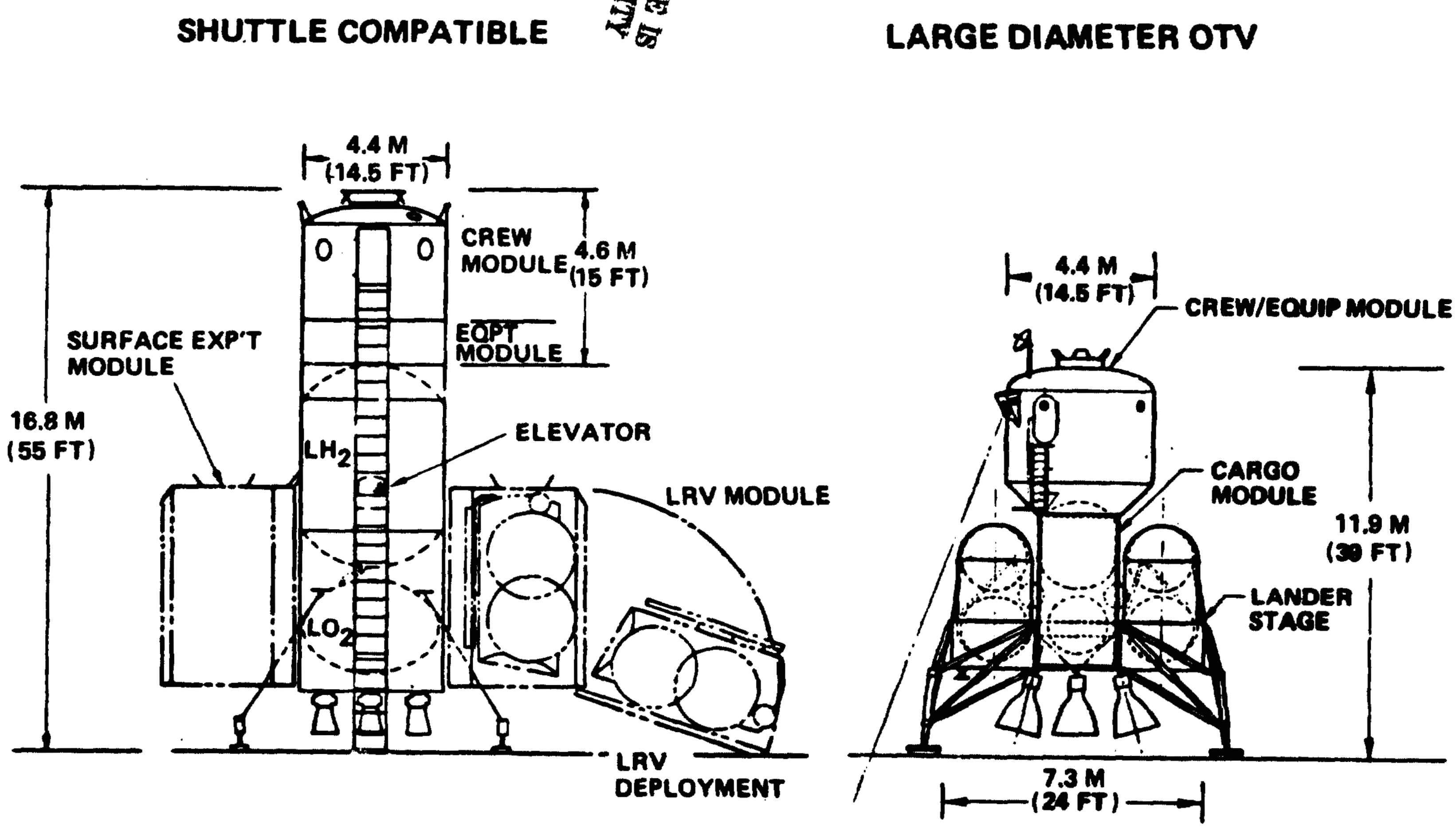
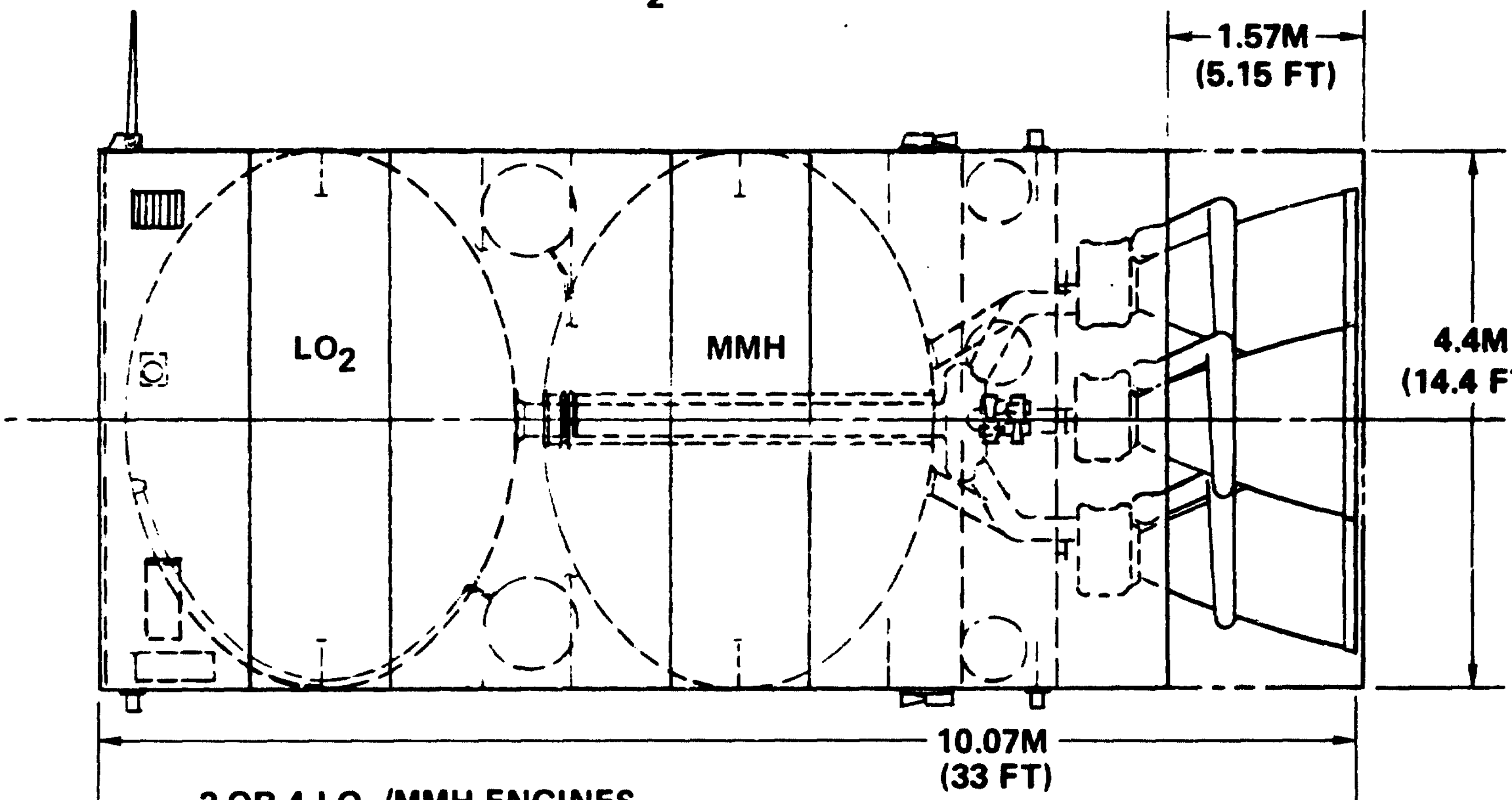


Figure 1-25 Small OTV Configurations Lunar Landing/Ascent Application

PROPELLANT MASS: 60 000 KG (132,000 LB)
MIX RATIO: 1.3 LO_2: 1 MMH



2 OR 4 LO_2/MMH ENGINES
89,000 N (20,000 LB) THRUST EACH
I_{SP} = 3630 M/SEC (370 SEC)

Figure 1-26 LO_2—MMH Stage Concept

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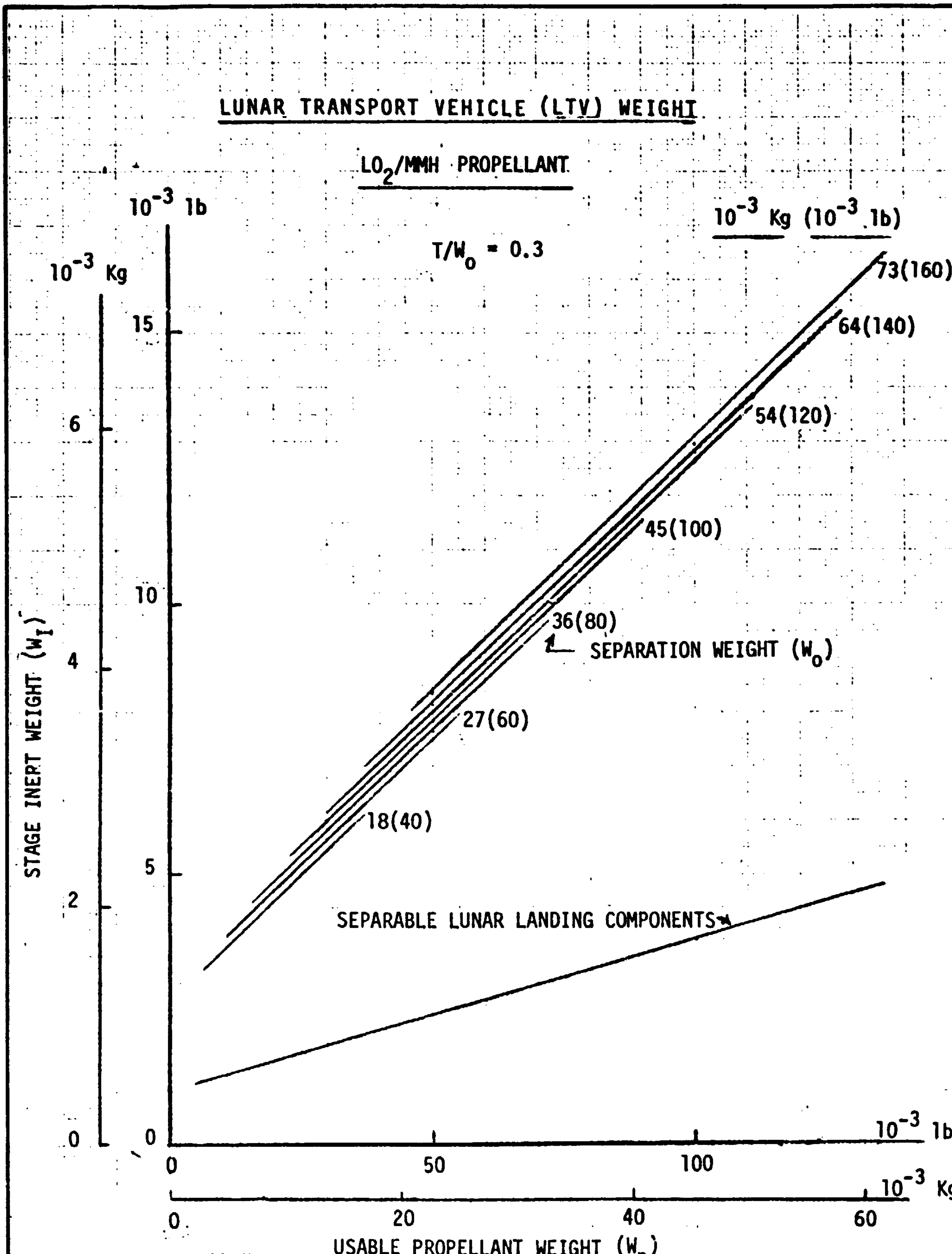

Figure 1-27. LO_2/MMH Lunar Transport Vehicle Mass

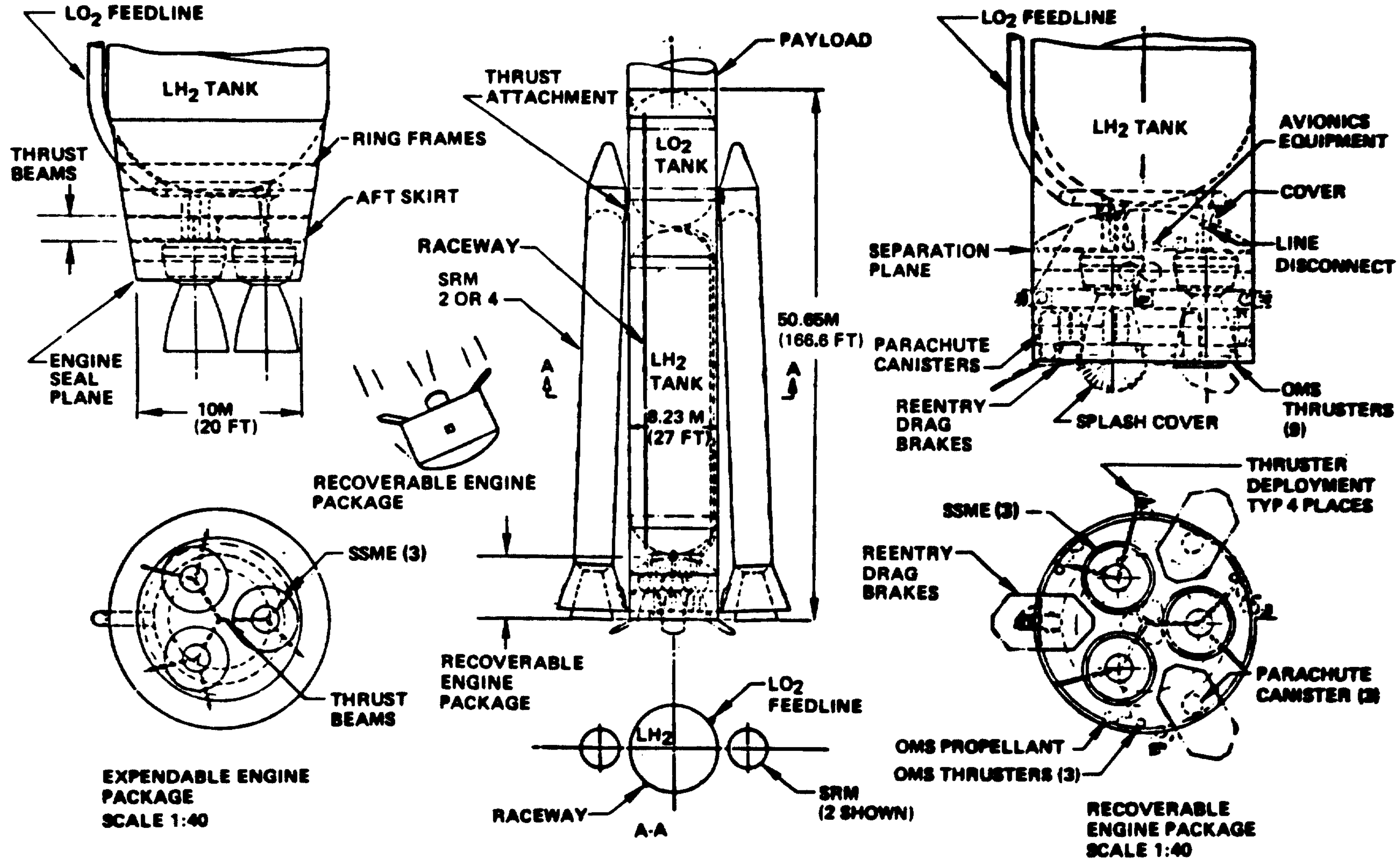

Figure 1-28 SRB/ET Heavy Lift Vehicle

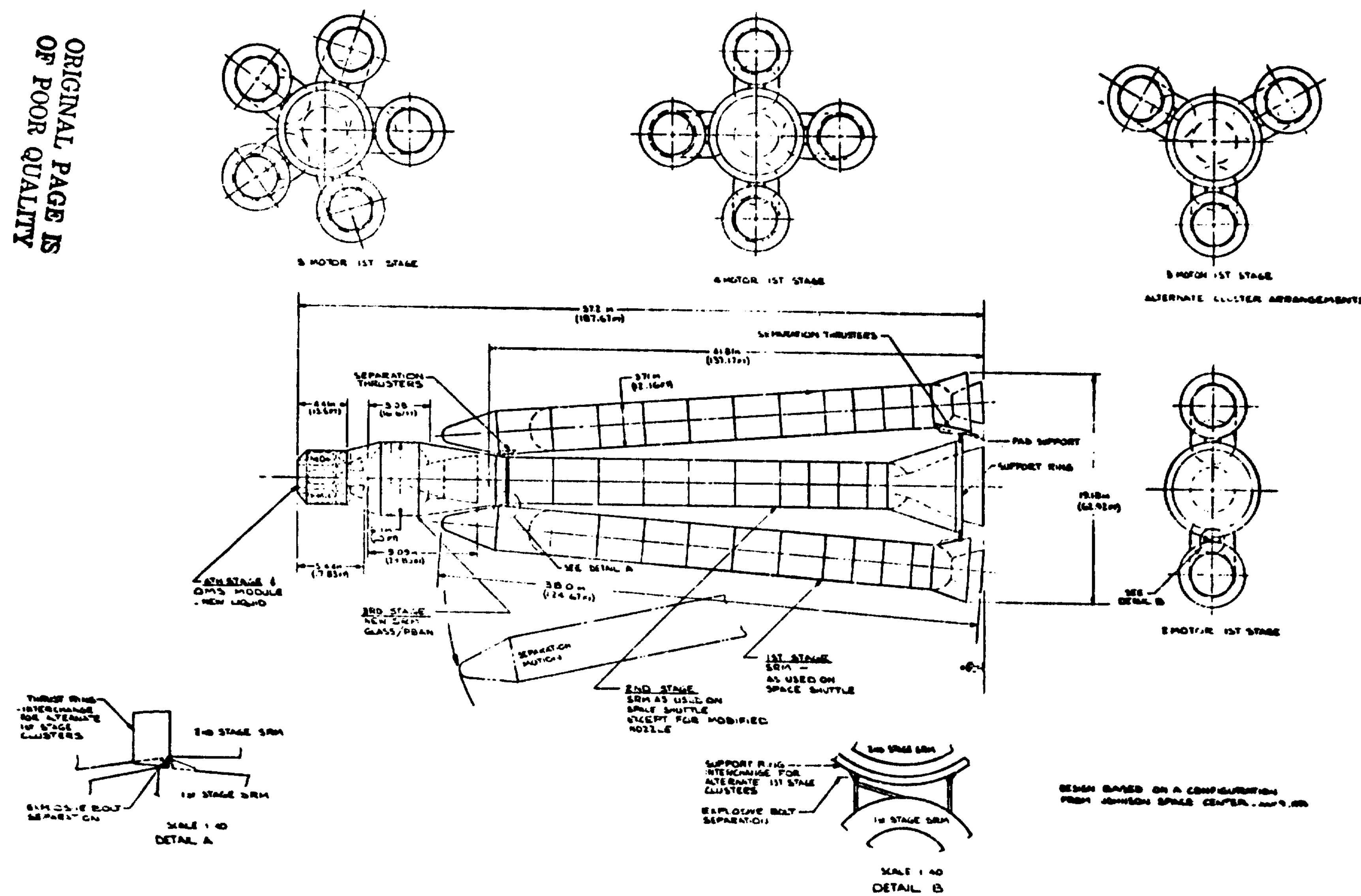

Figure 1-29 All-SRB Heavy Lift Vehicle

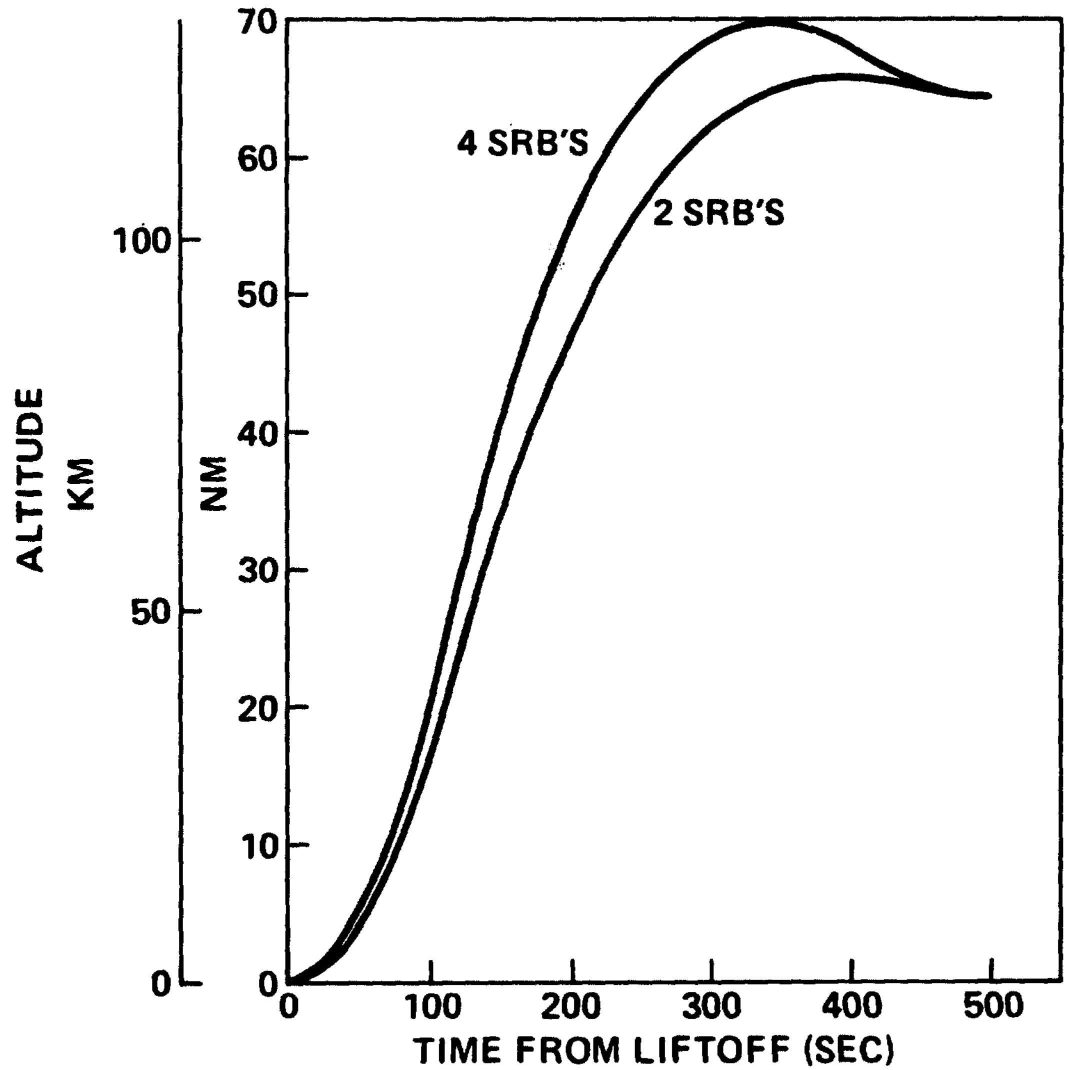

Figure 1-30 SRB/ET Heavy Lift Trajectories: Altitude Profiles

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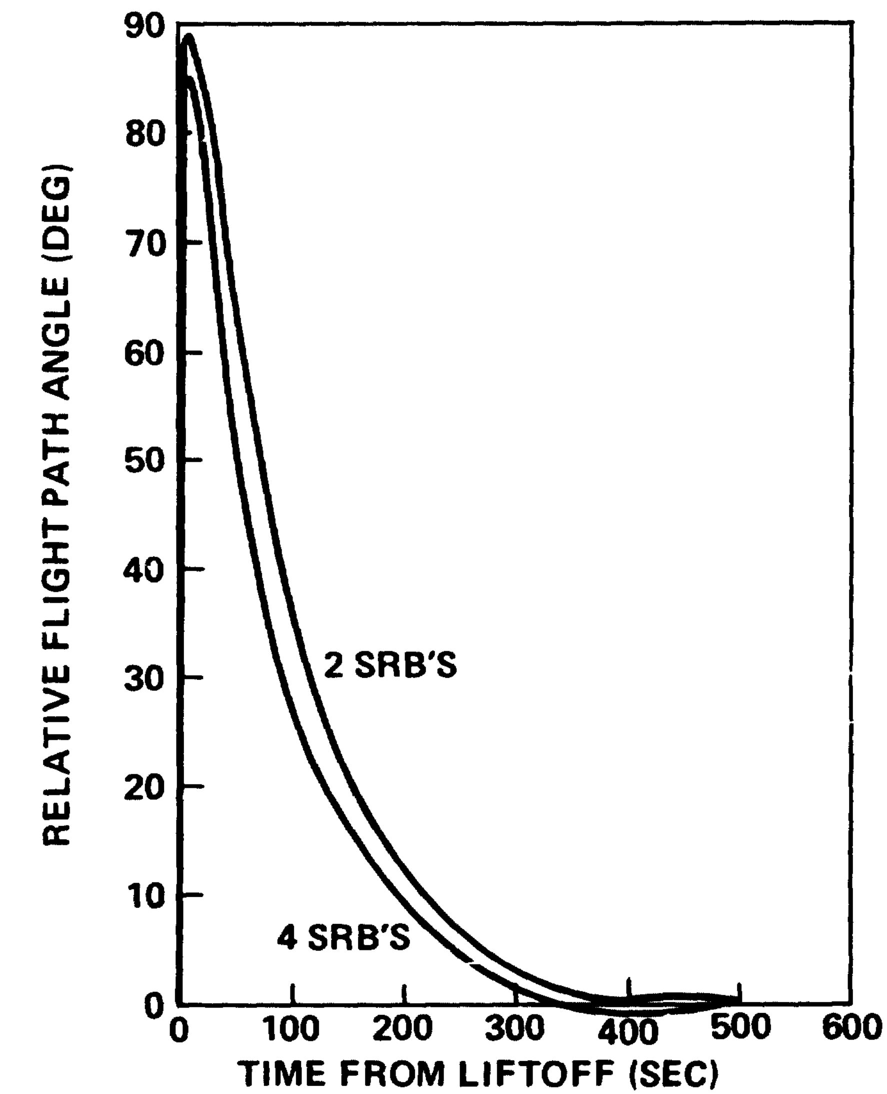

Figure 1-31 SRB/ET Heavy Lift Trajectories: Flight Path Angle Profiles

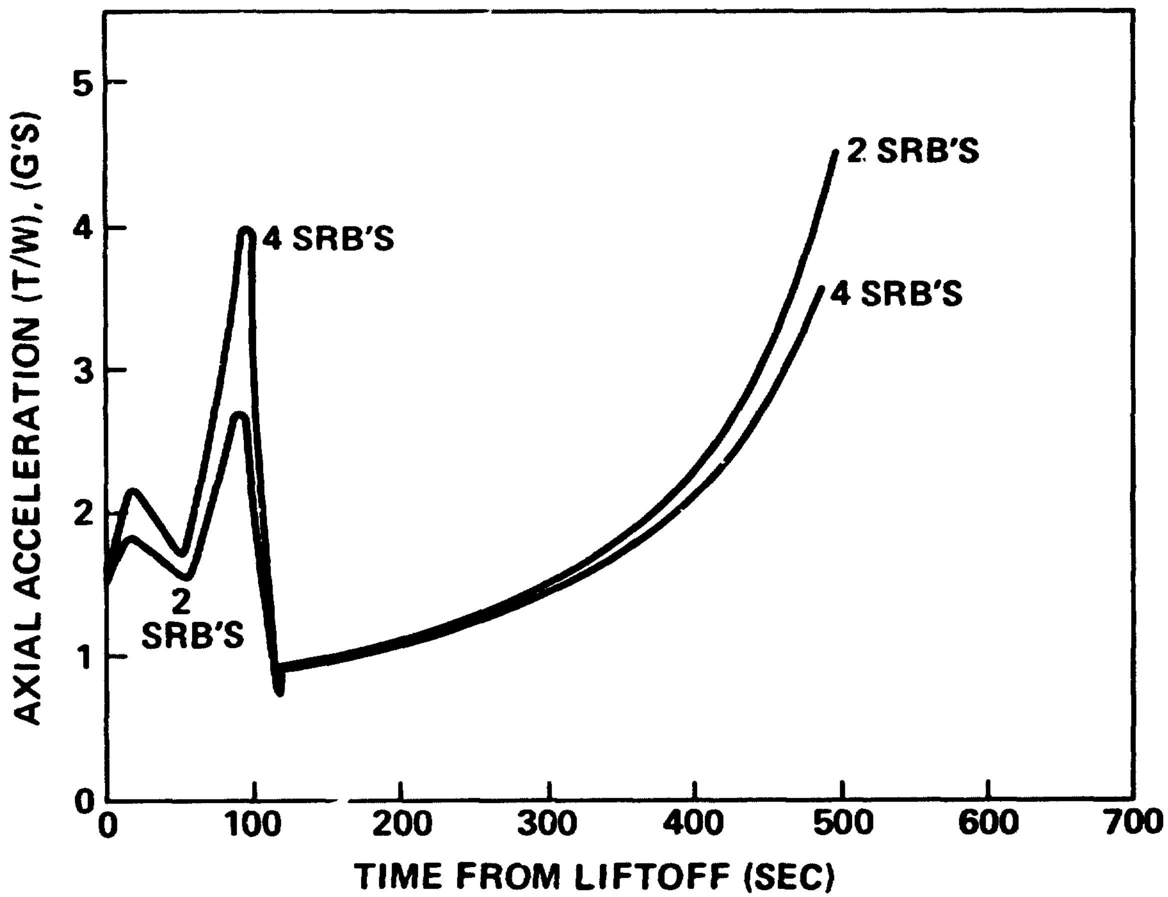

Figure 1-32 SRB/ET Heavy Lift Trajectories: Acceleration Profiles

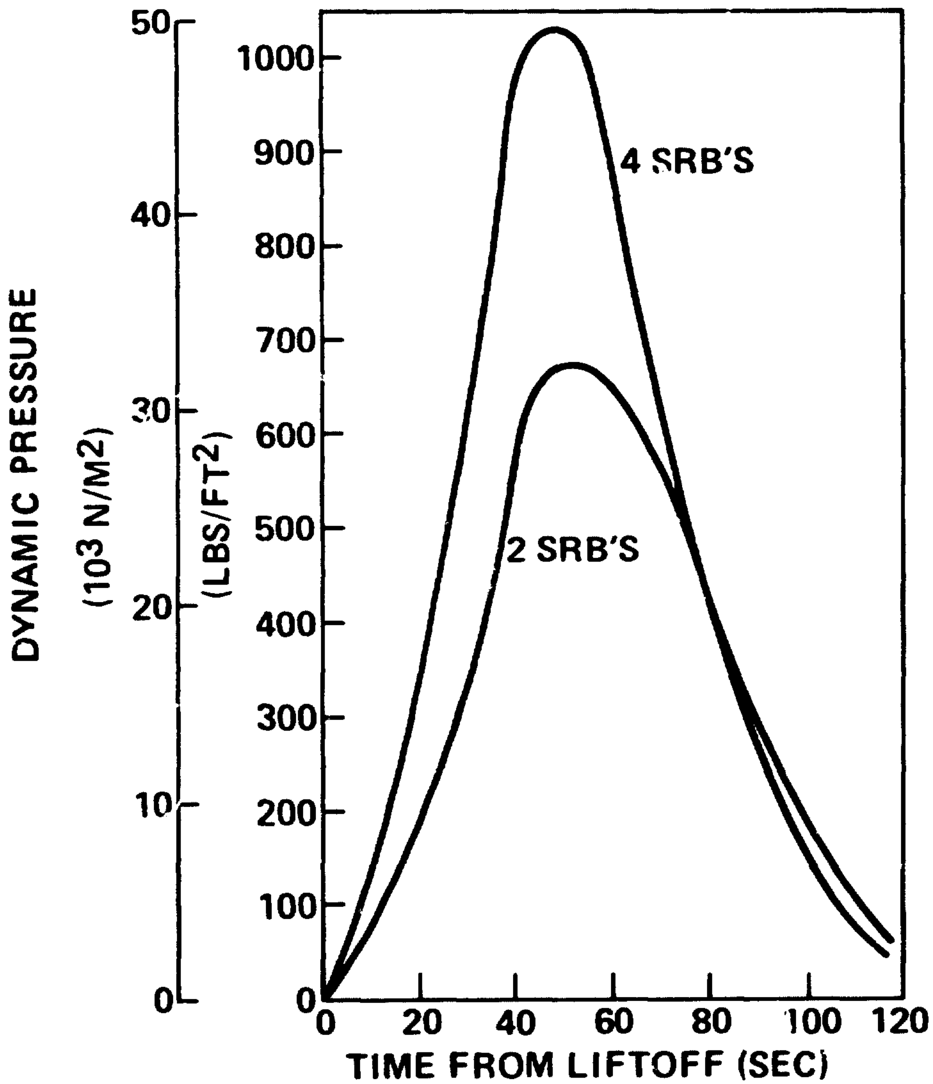

Figure 1-33 SRB/E Heavy Lift Trajectories: Dynamic Pressure Profiles

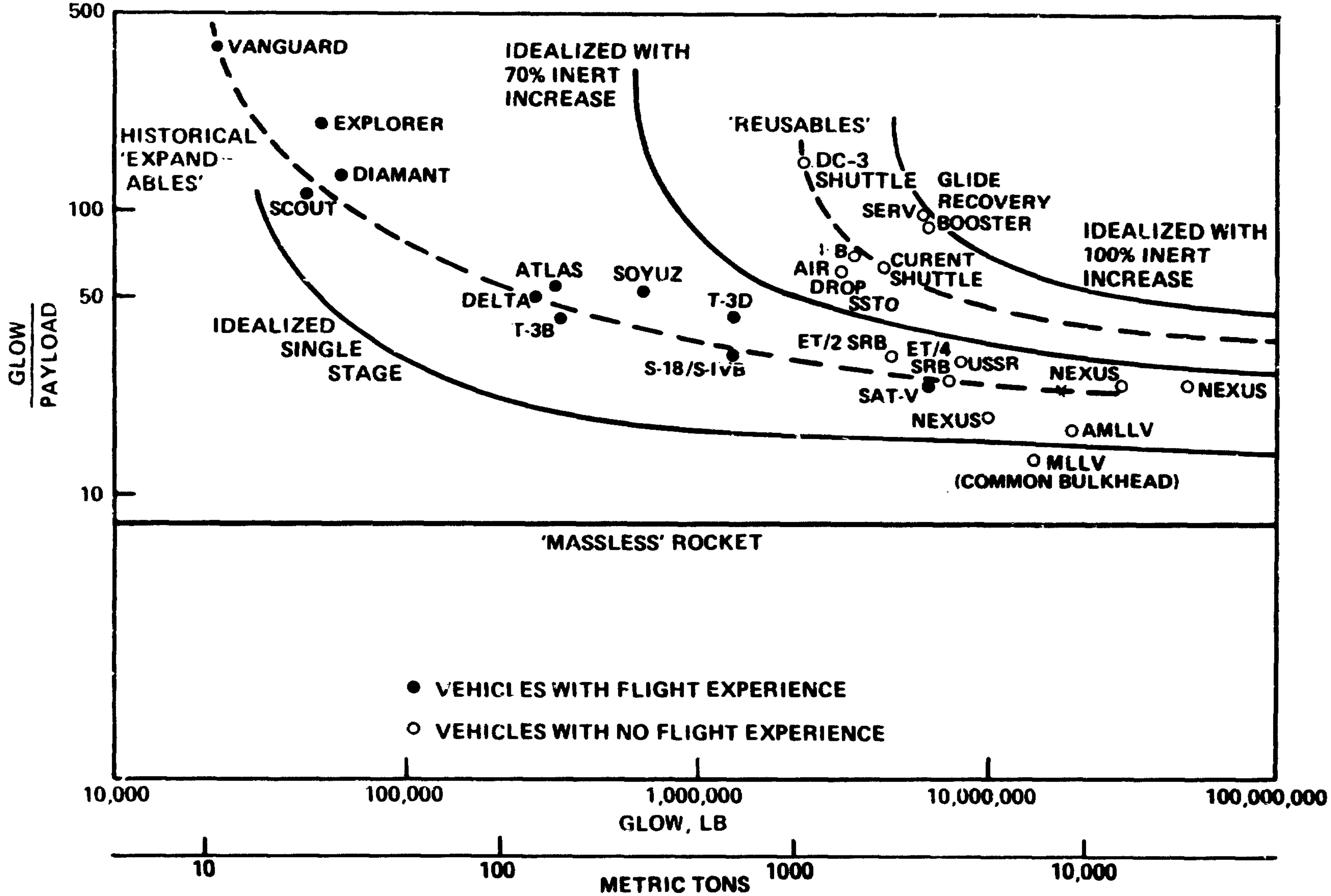

Figure 1-34 Mass/Performance Trending of Earth Launch Systems

been built or studied. Despite the many type variations (liquid/solid, low specific impulse/high impulse, two stage/three stage, etc.), all expendable rockets fall within a fairly narrow band. It is clear that increasing size leads to an increase in the percentage of payload carried. Also shown is a line corresponding to a "massless rocket," i.e., one in which there is no inert weight and which consists initially of only payload and propellant (LOX/LH_2, I_{sp} = 455 sec). This represents a lower limit for expendable rockets with this propellant. A sizing curve for an idealized vehicle is also shown in figure 1-34. Note that the curve generally parallels but lies below the historical expendable band.

Reusable rockets are heavier than expendables since the return/recovery system must be carried in addition to the payload. Many of the reusable vehicles studied, plus the current Space Shuttle, are also shown in the figure. Again a band is indicated; when the inert weight of the idealized rocket is increased by 70 percent and 100 percent of that the two boundary curves shown are produced.

Considering cost indicators, a very large expendable, typified by "X" on figure 1-34, would have the following characteristics:

ITEM	MASS 10^6 KG	MASS 10^6 LB	
Payload	.45	1.00	
Inert Wt. Engines	.22	0.48	$\left(\frac{\text{GLOW}}{\text{PAYLOAD}} = 23\right)$
Inert Wt. Other	.63	1.38	
Propellant	9.14	20.14	
Glow	10.44	23.00	

Employing D. Koelle's cost model wherein all costs are in terms of direct hours, based on his analysis of 68 space vehicle projects, a prediction of the recurring cost of the inert weight and engines was derived. The hardware cost of the engines and other inert weight contributed \$425/kg (\$193 per pound) of payload, without consideration of propellant costs, amortization of development, etc.

The predicted reusable vehicle of GLOW = 10 400 metric tons (23 MLB) has a payload of only approximately 227 000 kg (500.000 lb). In effect the payload is decreased by the addition of the recovery system.

In estimating the recurring cost of this vehicle, it was assumed that the basic airframe lasts for 1,000 flights, and that the engines last 100 flights per set. Eliminating the airframe learning factor and increasing its complexity factor by fifty per cent, the stage cost distributed over 1,000 flights is \$5.64 per kg (\$2.56 per pound) of payload for hardware amortization, indicating that a large reusable may attain the nominal target of \$45/kg (\$20/lb).

Several possibilities were suggested for vehicle configuration. Drop tanks and expandable payload housings appear too expensive. No significant down payload is required, and a cross range capability of 320 km (200 miles) would probably suffice. The payload bay should be of the greatest feasible volume. A large single stage, ballistic recovery (VTOVL) vehicle was selected as a representative concept for power satellite use with nominal payload capability of 250 000 kg (550,000 lb).

This is a vertical take-off/land system, with a general shape similar to the Gemini or Apollo Spacecraft. The take-off is accomplished with the thrust of the LO_2/LH_2 main engines (ME) and the LO_2/hydrocarbon auxiliary engines (AE). The AE burn approximately 70 seconds. Total burn time for the ME until injection into the initial orbit is 410 seconds. Acceleration is limited to four g's. Immediately after orbit insertion the payload door opens and the payload and a small "tug type" propulsion system is released. This "kicker" propulsion system raises the payload to the 463 km (250 N.M.) assembly altitude. Thus the mass of the LCHLV is not taken to the higher orbit, greatly increasing the payload capability. Figure 1-35 shows an inboard profile of the LCHLV and "kicker." After payload separation the payload bay is closed and the AE are used to raise the LCHLV orbit to 185x185 km (100 x 100 N.M.) The deorbit maneuver is performed by the AE. Reentry heat transferred to the vehicle is absorbed by a watercooled thermal protection system (TPS). The resultant steam is used to cool the engine bells.

The rationale for the water TPS is as follows: it is much heavier, possibly as much as 40 000 kg (90,000 lb) more than either an ablative or metallic reradiative TPS. However reradiators require refurbishment and an ablative TPS would of course require replacement. An associate has noted that if transport aircraft required even such a simple operation as the application of a single coat of paint between flights that it would double the cost of airline tickets. Along these lines, we estimate that approximately 0.1 kg (0.2 lb) of ablator would have to be replaced per flight per pound of payload, and that the production and installation cost of ablator panels would be at least \$88/kg (\$40 per pound), adding up to \$17/kg (\$8 per pound) of payload to the operational cost of the vehicle, an increase of perhaps 25 to 50 percent, at a payload increment of only approximately 20 percent.

The LCHLV could be targeted initially for a landing approximately 160 km (100 miles) off shore of the Cape. After a safe trajectory is assured, the flight path could be depressed for a landing in the recovery basin. In prior flight programs, spacecraft were consistently recovered within 3 km (2

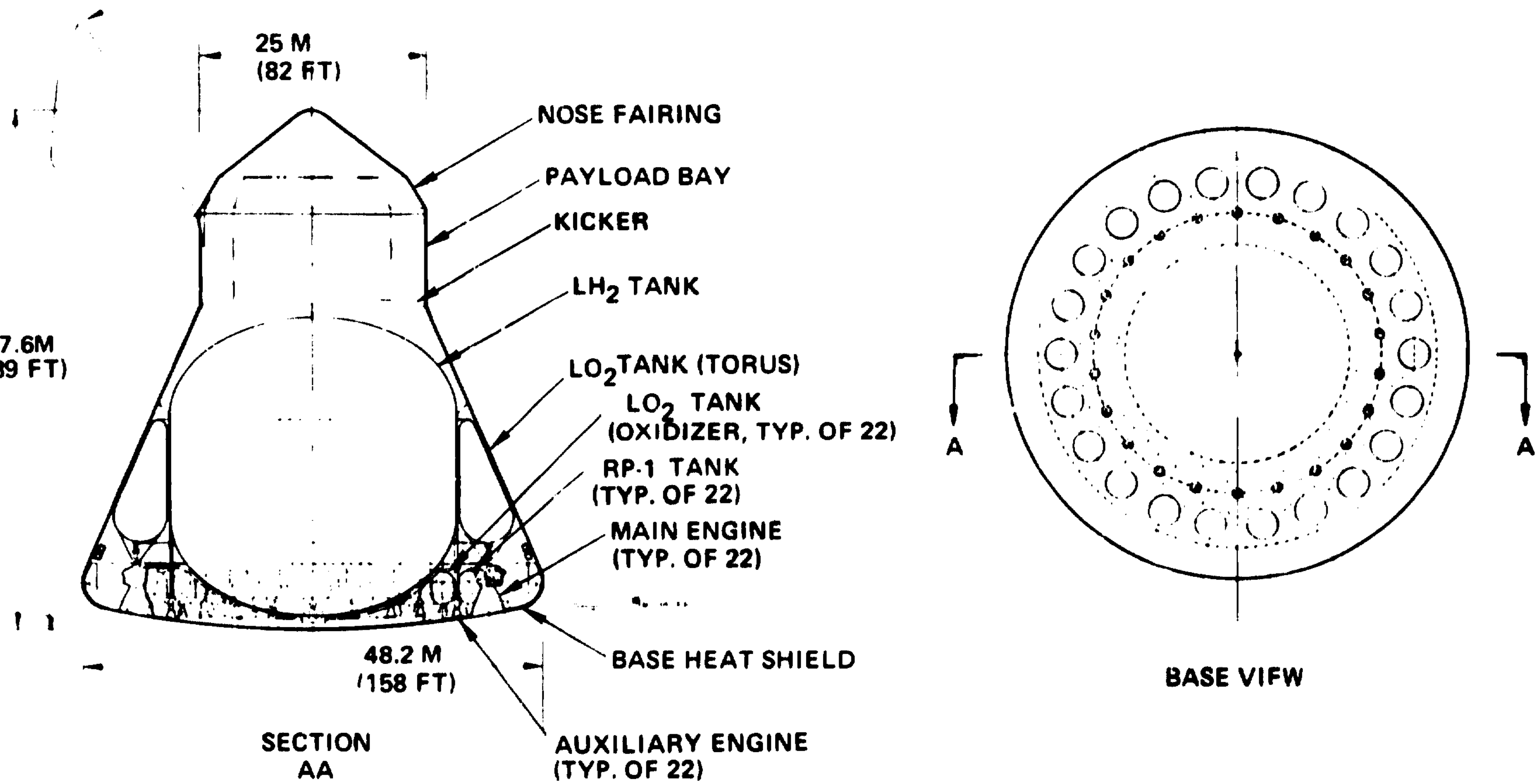

Figure 1-35. Low Cost Heavy Lift Vehicle

miles) of the target point without control from the ground, and despite parachute drift with the wind. With ground control, we might expect better accuracies. A basin diameter of 4600 m (15,000 feet) is believed adequate. Aeromaneuvers would be accomplished using an off-set center of gravity and roll control to position the resultant lift vector. Terminal descent velocity is approximately 100 m/sec (300 ft/sec). A weight optimization of the landing rocket system indicates a minimum total weight for the engines propellant and associated tanks with a deceleration of four to five g's. Consequently, the braking activity does not begin until an altitude of approximately 460 m (1,500 feet) is reached. The LO_2/hydrocarbon engines used will have a thrust to weight ratio of perhaps 110 to 120, compared to the 60 to 70 of LO_2/LH_2 engines. These landing engines are used at liftoff to provide a major portion (approximately one third) of the total thrust with a corresponding savings in ME weight. The AE must be throttable to perform the landing maneuver. During ascent, this capability serves for attitude control, to AE cutoff. After that the ME provide control. Gimballed engines are not used; the gimbal points would be too near the c.g. to be effective, and the fixed engines are easier to thermally protect.

1.4 CREW VEHICLES

Parametric mass data were developed for three types of crew vehicles: a crew transport vehicle, a direct-entry Apollo-shape crew vehicle, and a crew and equipment module for lunar missions.

1.4.1 Crew Transport Vehicle (CTV)

The CTV is applicable to short-duration crew transfer missions such as geosynchronous orbit or lunar orbit crew rotation. It includes an optimal emergency 400 m/sec (1,300 ft/sec) propulsion system, needed for lunar crew rotation missions.

Parametric mass data are shown in figure 1-36 and 1-37. Values shown are not cumulative; i.e., total mass is derived by summing indicated masses for inert, crew and reserves, propellant, and consumables mass. The CTV's were assumed 4.4m (14-1/2 ft) diameter for compatibility with shuttle launch to orbit.

1.4.2 Direct-Entry Vehicle

Mass parametrics for an Apollo-type direct entry vehicle, capable of geosynchronous altitude or lunar return direct entry, are shown in figure 1-38.

1.4.3 Crew and Equipment Module (CEM)

The CEM is similar to the CTV except that much longer missions are considered and it does not include a propulsion system. The CEM is not capable of controlled flight on its own; it must be attached to a propulsion vehicle (usually lunar lander). Mass parametrics are shown in figures 1-39, 1-40, and 1-41.

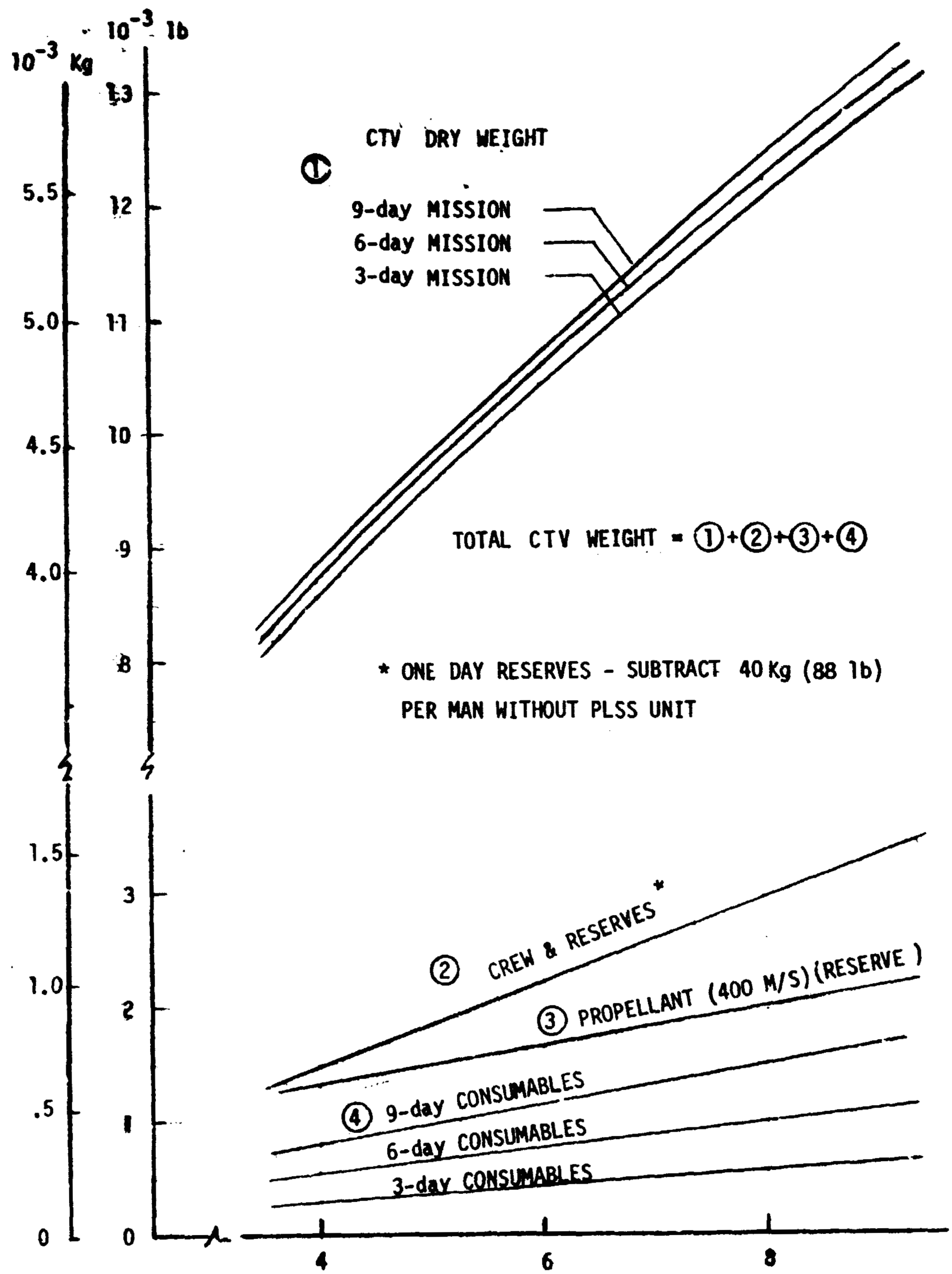

Figure 1-36.

CALC			REVISED	DATE	CREW TRANSFER VEHICLE (CTV) WEIGHTS	
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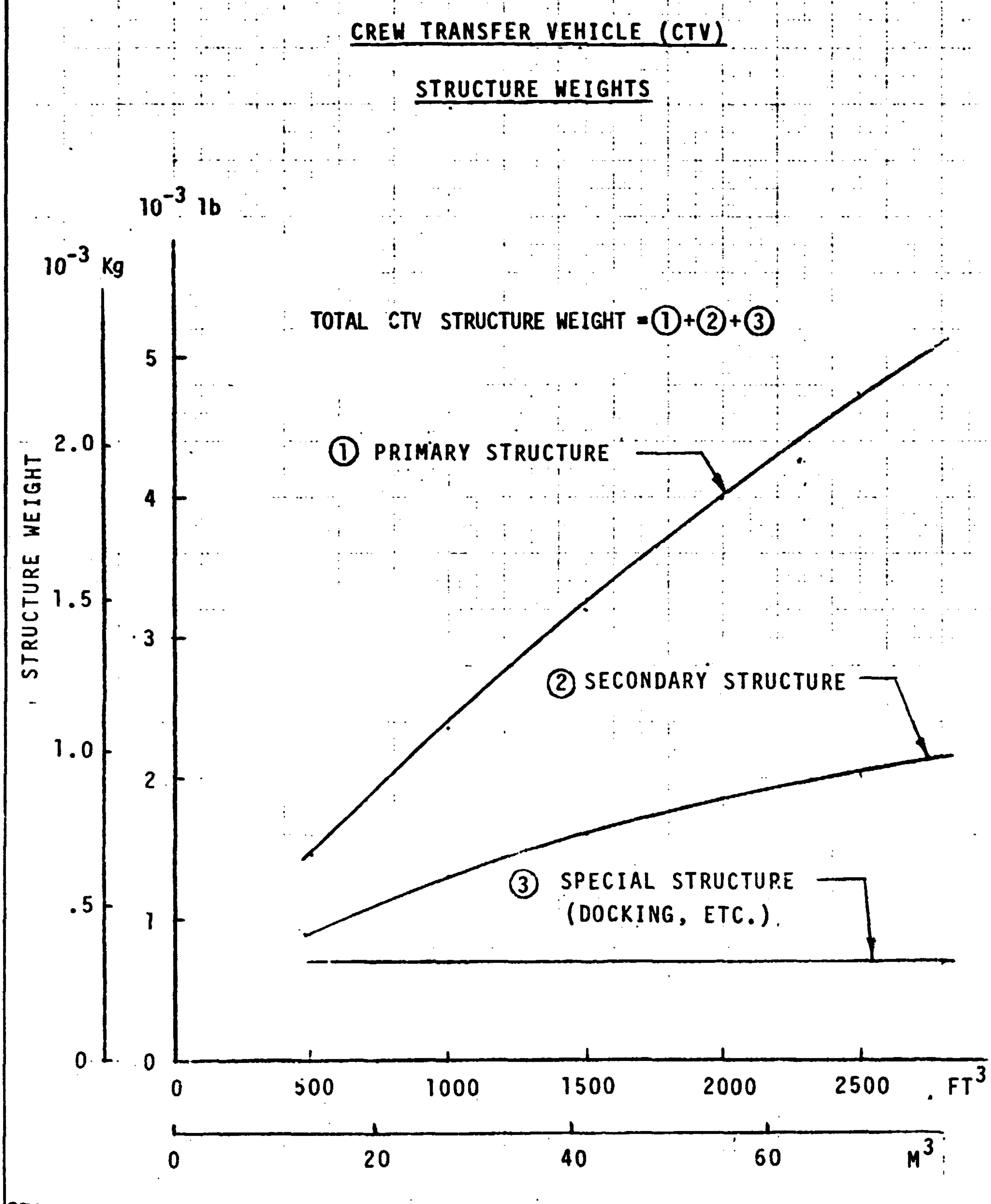

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Figure 1-37. Crew Transfer Vehicle Structure Mass

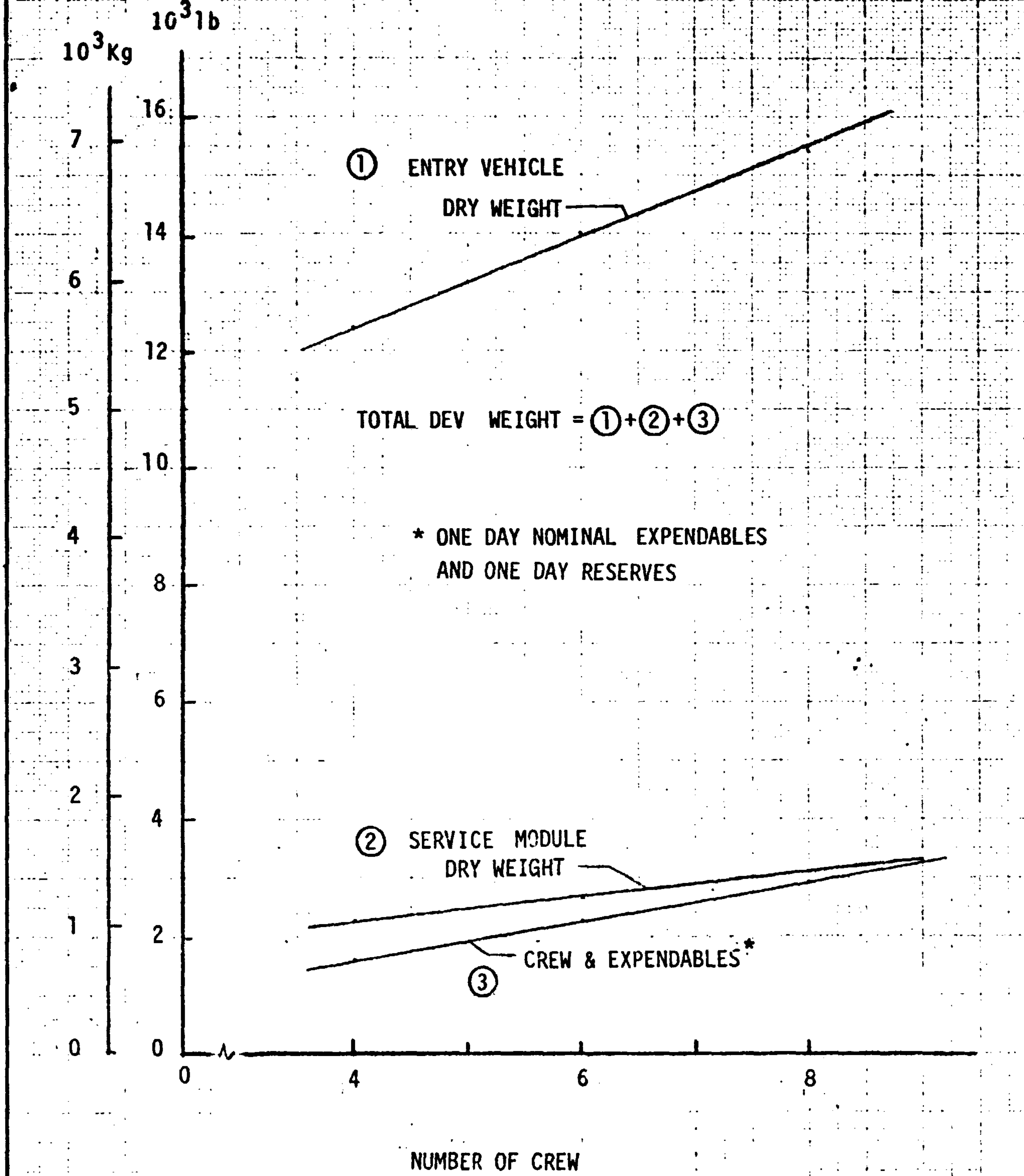

FIGURE 1-38 DIRECT ENTRY VEHICLE (DEV) APOLLO SHAPE

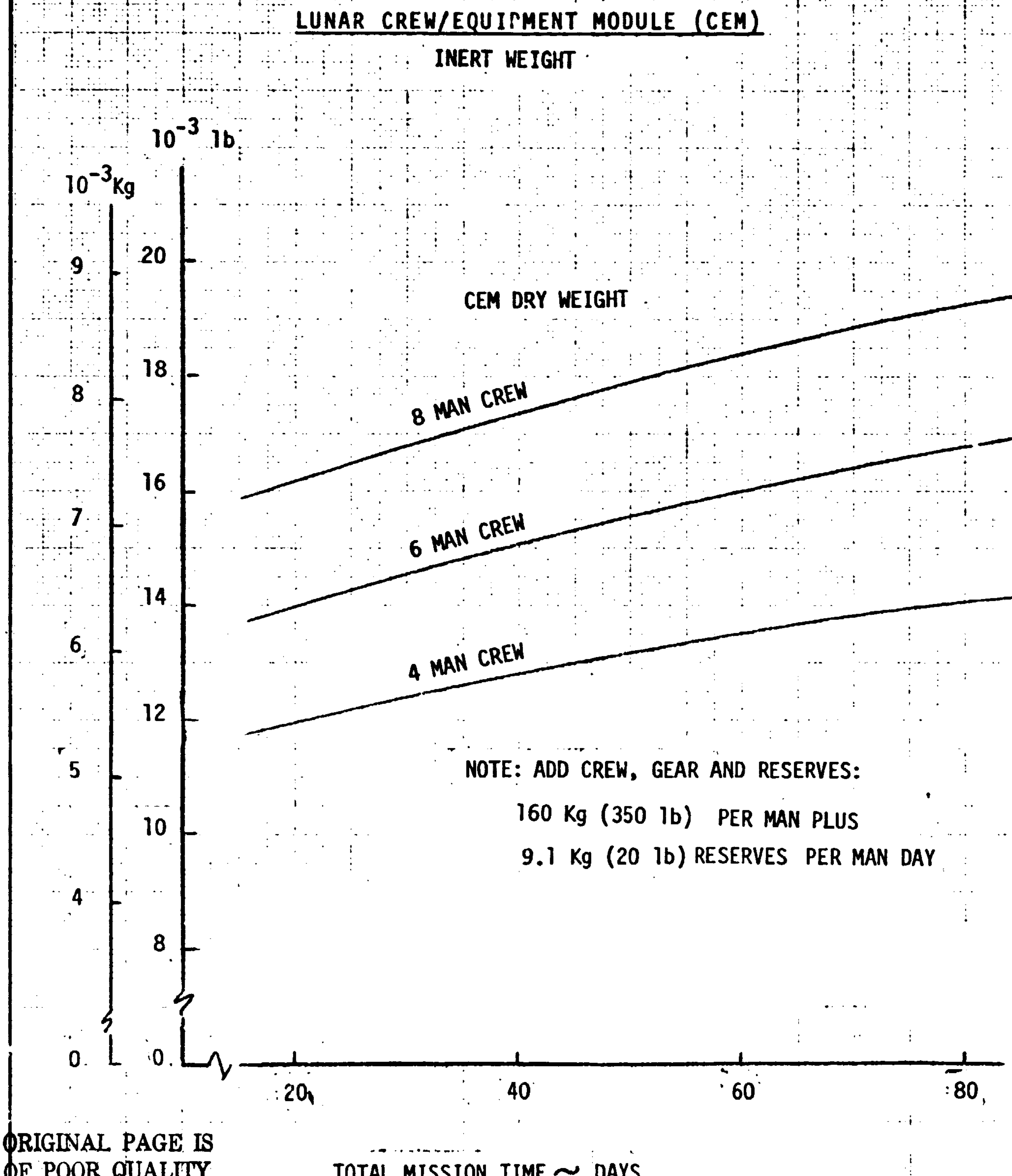

Figure 1-39. Lunar Crew/Equipment Module Inert Mass

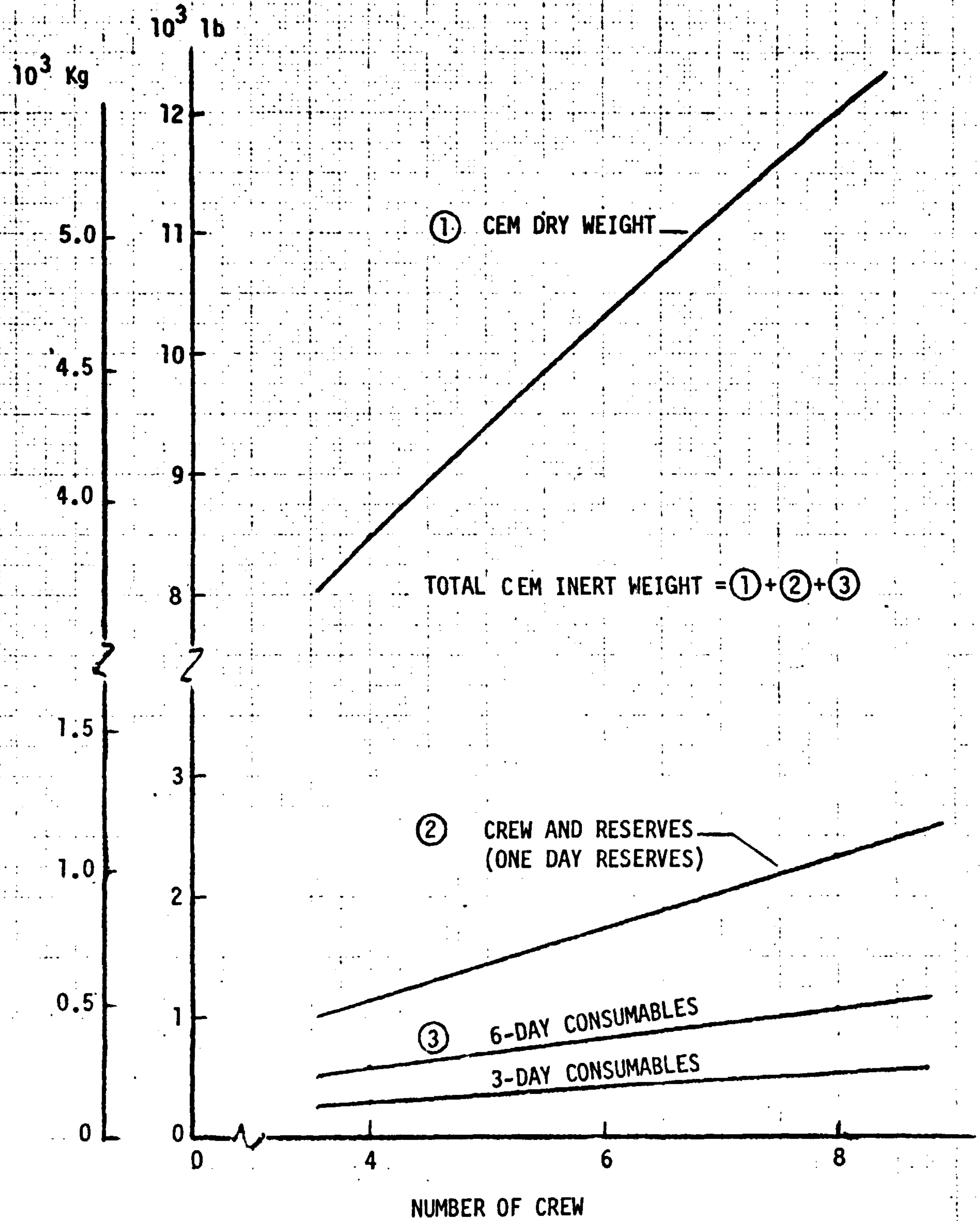

FIGURE 1-40 LSB CREW/EQUIPMENT MODULE (CEM)

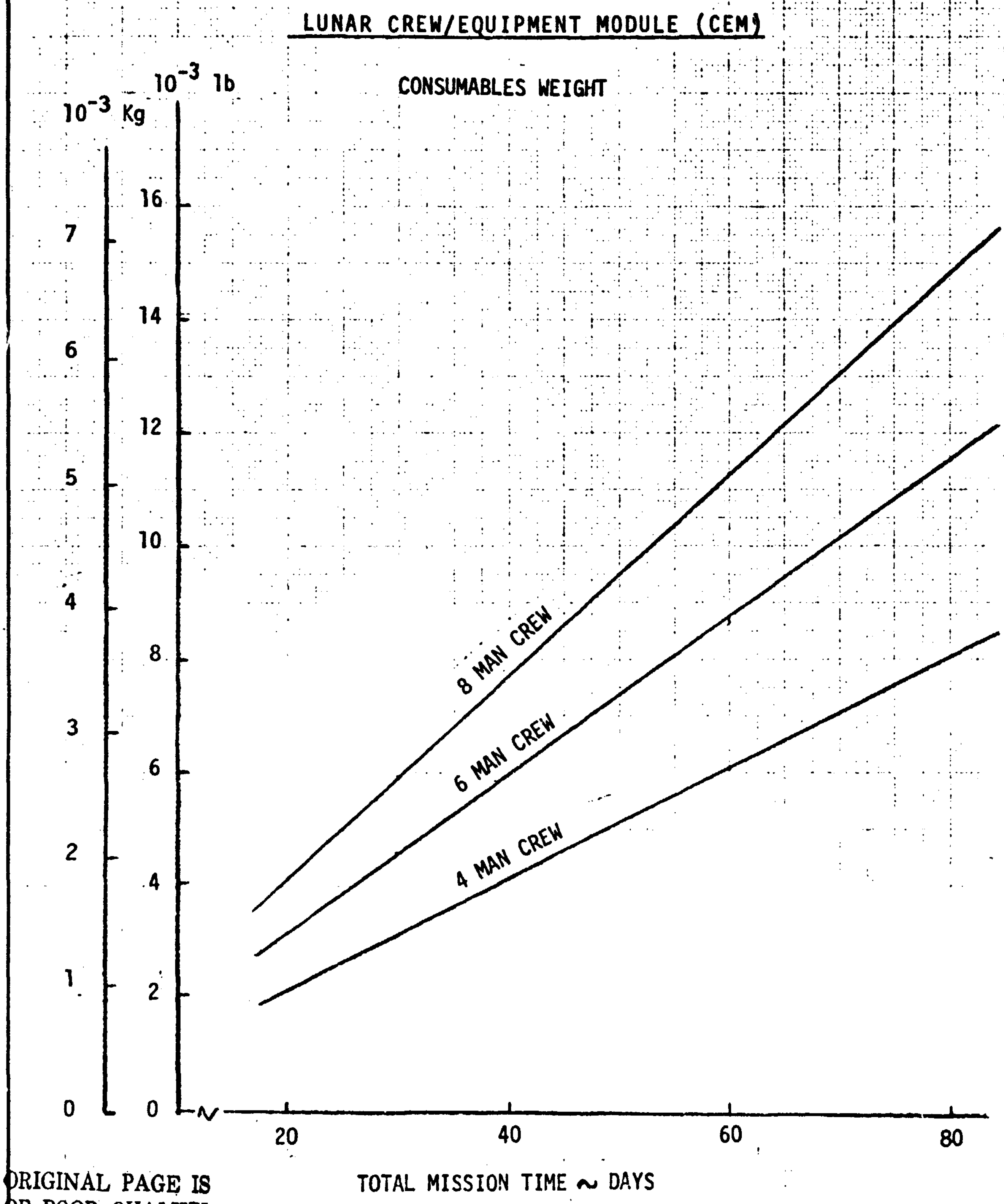

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Figure 1-41. Lunar Crew/Equipment Module Consumables Mass

APPENDIX 2 WEIGHT GROWTH PREDICTIONS FOR FUTURE SPACE TRANSPORTATION REQUIREMENTS

The following presents an examination of the weight growth approach applied to Future Space Transportation Systems Analysis (FSTSA) requirements. The information presented responds to an action item levied at the October 24 working-session review of FSTSA by the NASA study management team. Parameters that affect weight growth, past weight histories, and the current state of future space-mission design are discussed. Weight growth factors of from 10 percent to 41 percent have been derived for various missions or vehicles as currently defined in the FSTSA study. The growth percentage to be used depends upon the probability desired for not exceeding the selected weight growth.

2.1 APPROACHES TO WEIGHTS GROWTH PREDICTION

With few exceptions (Mariner Mars '71 and smaller Earth satellites), positive weight growth has always been present in aerospace programs. Values from the start of the program definition phase (phase B) range from 8.7 percent (Saturn SIC) to 57.0 percent (Apollo lunar module) for recent programs. In the case of aircraft, boosters, and missiles, weight growth has been accommodated by increased propellant and thrust to maintain constant performance. However, the high energy requirements of future space missions, high costs of major design changes, accuracy of weight prediction required for shuttle payloads, and traffic model analyses motivate accurate prediction of expected weight growth for each possible space mission.

Past weight growth studies have taken one of two approaches. One is to chart growth versus time for known vehicles and average the data. If the historical vehicles are closely correlated with the vehicle (and design phase) in question, this method gives an average and indicates some weight extremities that may be encountered.

A more recent approach has been to consider as many applicable growth factors as possible and plot them as cumulative distributions or frequency distributions. A "probability-of-not-exceeding" value is then chosen and applied to the expected weight growth. This method was used to arrive at expected space shuttle weight growth.

The method used in this study uses both approaches. Vehicles used for growth data are correlated as to technology (airplanes, manned spacecraft, boosters, etc.), generation (first-of-the-line or follow-on), and phase relationship (where in the program the weight estimate is made). In addition, new or expected technology advances not considered at the time of the mission studies used as sources were analyzed for weight effects. Also, the amount of weight detail in the mission studies was examined for possible omissions or oversimplification. Adjustments to the expected weight growth arrived at by analytical/empirical means are identified.

Since the purpose of the FSTSA Study is to forecast future requirements, weight growth "probabilities of not exceeding" of 50% were used as indicative of most probable weight growth. Higher confidence levels, up to 90%, are often used to match specific transportation systems to specific requirements. Most probable growth is appropriate to the general requirements predictions of this study since the mission implementations are representative and are not firm system or design selections.

2.1.1 Definition of Weight Growth

Two factors have been generally applied to basic identified weights early in aerospace vehicle design. These have been "contingency" and/or "growth allowance." Contingency is the weight allowance included for deficiencies in identified weight resulting from lack of detail in design definition. Growth allowance is the weight allotted for effects of design changes. "In-scope" growth is due to changes required to meet original specifications and "out-of-scope" growth is due to specification changes. (The term "margin" often used in studies only applies to the difference between identified weight plus contingency/growth and a delivery system capability.) Figure 2-1 shows a typical weight history.

It is impractical to establish a precise separation between contingency and growth allowance weights when analyzing past program weight histories. The weight growth allowance considered in this study will include contingency, in-scope growth, and out-of-scope growth, but it will not include number of crew, major change in time of mission, or other sizeable mission requirement changes.

2.1.2 Parameters That Affect FSTSA Weight Growth

The following parameters affect the value of weight growth allowance that should be placed upon FSTSA study missions or vehicles:

- Type of spacecraft (manned, unmanned, rovers, etc.)
- Generation of the spacecraft
- Program phase
- Completeness of weight estimates used
- Remaining configuration options
- Design definition completeness

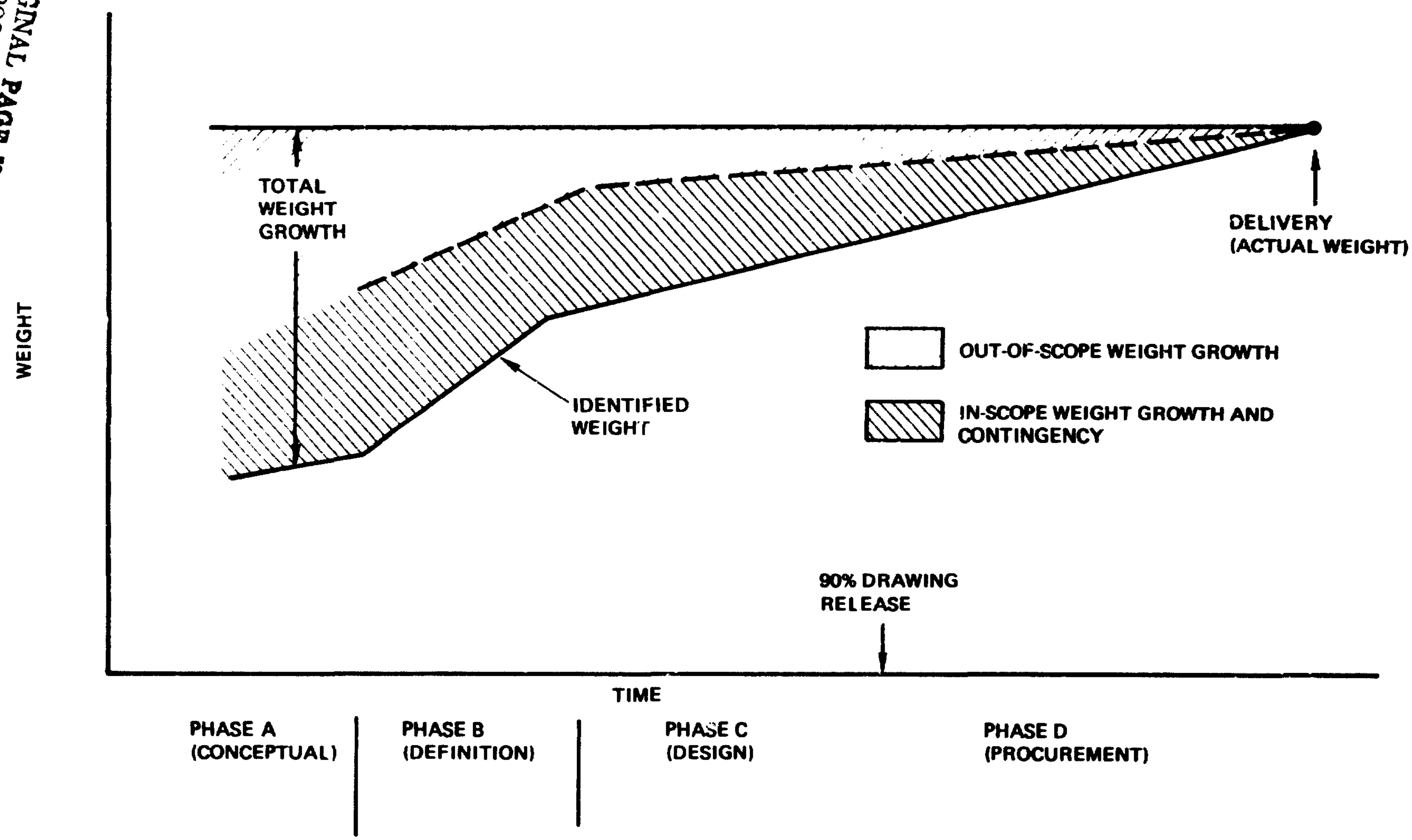

Figure 2-1. Typical Aerospace Vehicle Weight History

A first generation spacecraft is the first of its kind and, as such, would be expected to have a higher weight growth than a second or third generation spacecraft such as ballistic entry vehicle or propulsion stage. Most of the manned missions considered in the FSTSA study are first generation, although some second-generation vehicles are used.

With the possible exception of the space station and the space tug (IUS or tug), the missions being considered by FSTSA studies can be considered as at the start of program phase B.

In general, the completeness and detail of weight estimates for past studies fall short of what is desired. An exception is the Lunar Surface Base Study that included many weight details. Most of the configuration options have been exercised in the studies to arrive at optimum subsystems, considering state-of-the-art (SOA) technology development. Lower cost might dictate heavier subsystems in some areas (metabolic supply); however, advancements in SOA not foreseen at the time of the mission studies may offset such considerations. An example is the large-scale integrated (LSI) circuits now in development that will reduce weight, volume, and power requirements.

2.1.3 Past Weight Histories

Figures 2-2 through 2-6 show a summary of weight histories of aerospace vehicles that represent the engineering technologies that will be involved in FSTSA mission vehicle designs. These are:

- Jet aircraft
- Transportation vehicles
- Manned spacecraft
- Unmanned spacecraft
- New concepts

In general, weight histories show a rapid increase in estimated weight during phase B or early phase C (design definition). Reported weight histories need to be evaluated with use of detailed weight estimates from as early in the program as possible. Since it is contingency-plus-growth allowance that is being examined, any such factors in the early weight histories used for empirical data must be known. Most of the vehicles used for data in this study are Boeing products or Boeing evaluated (Apollo Technical Evaluation and Integration Contract). The Boeing products have been used for three reasons: (1) Detailed historical weight data are readily available, (2) the vehicle designs span the technologies applicable to FSTSA missions, and (3) the weight estimating procedures use fairly consistent and rigorous methodology.

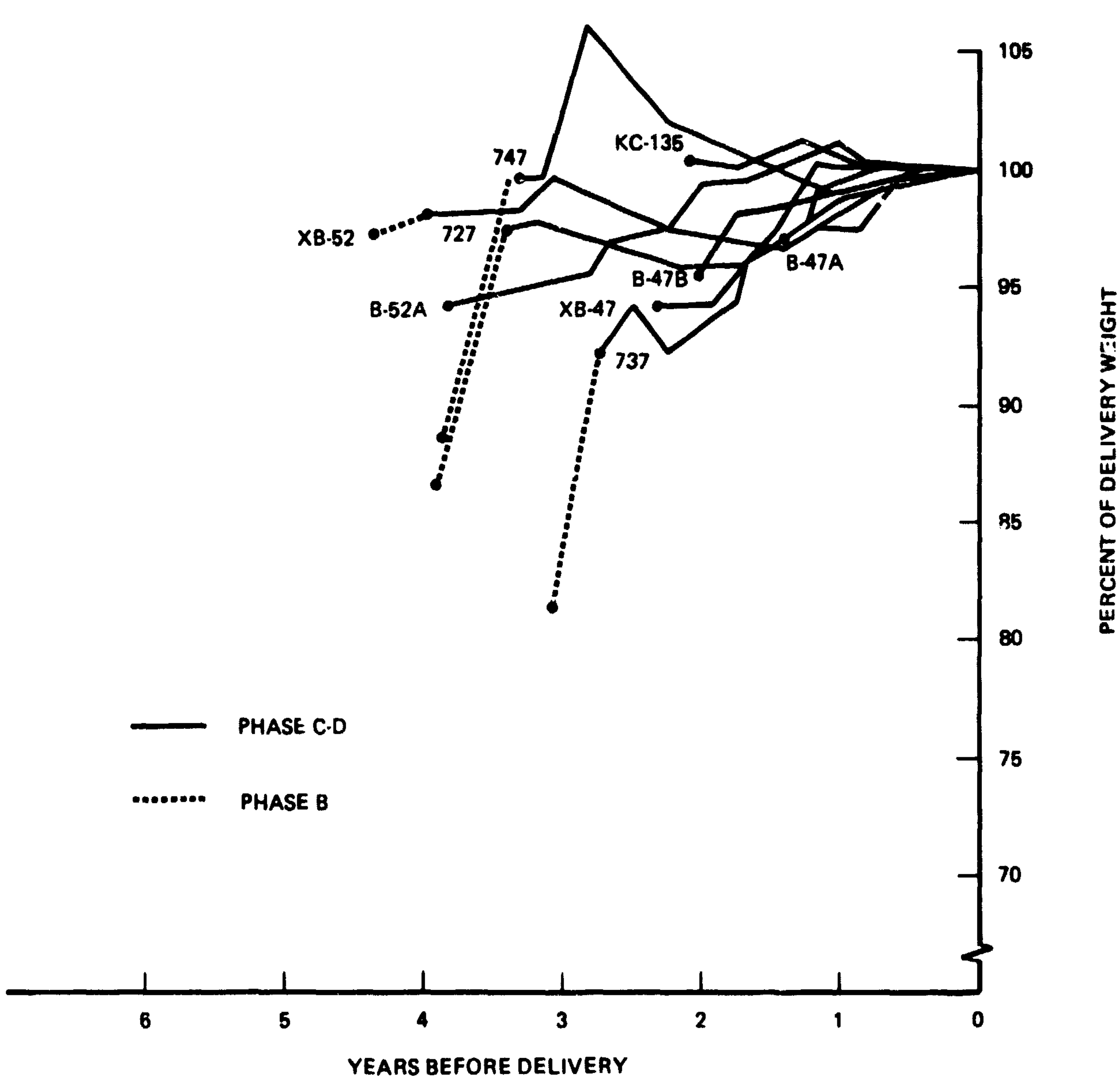

Figure 2-2. Weight Growth – Jet Aircraft

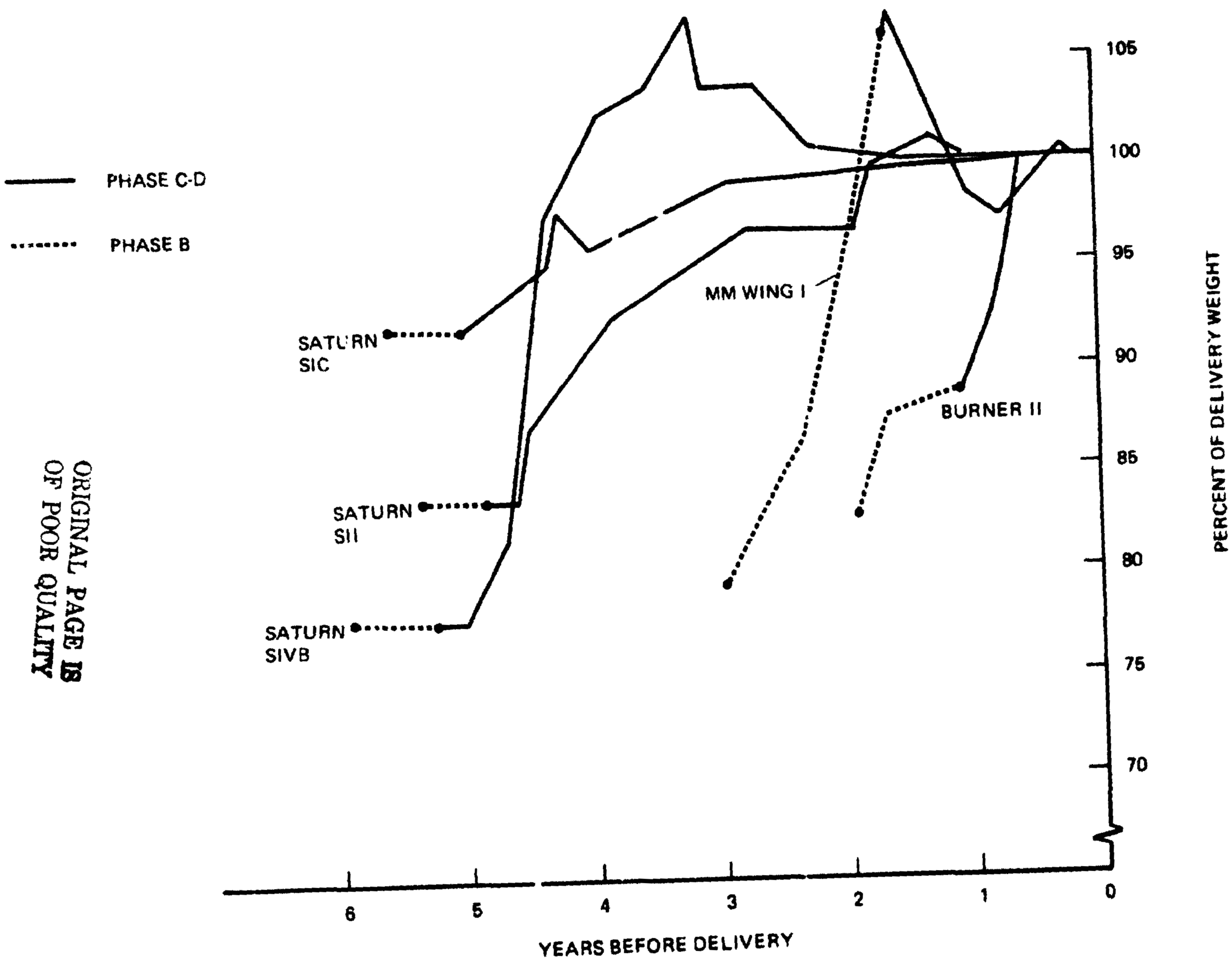

Figure 2-3. Weight Growth — Boosters

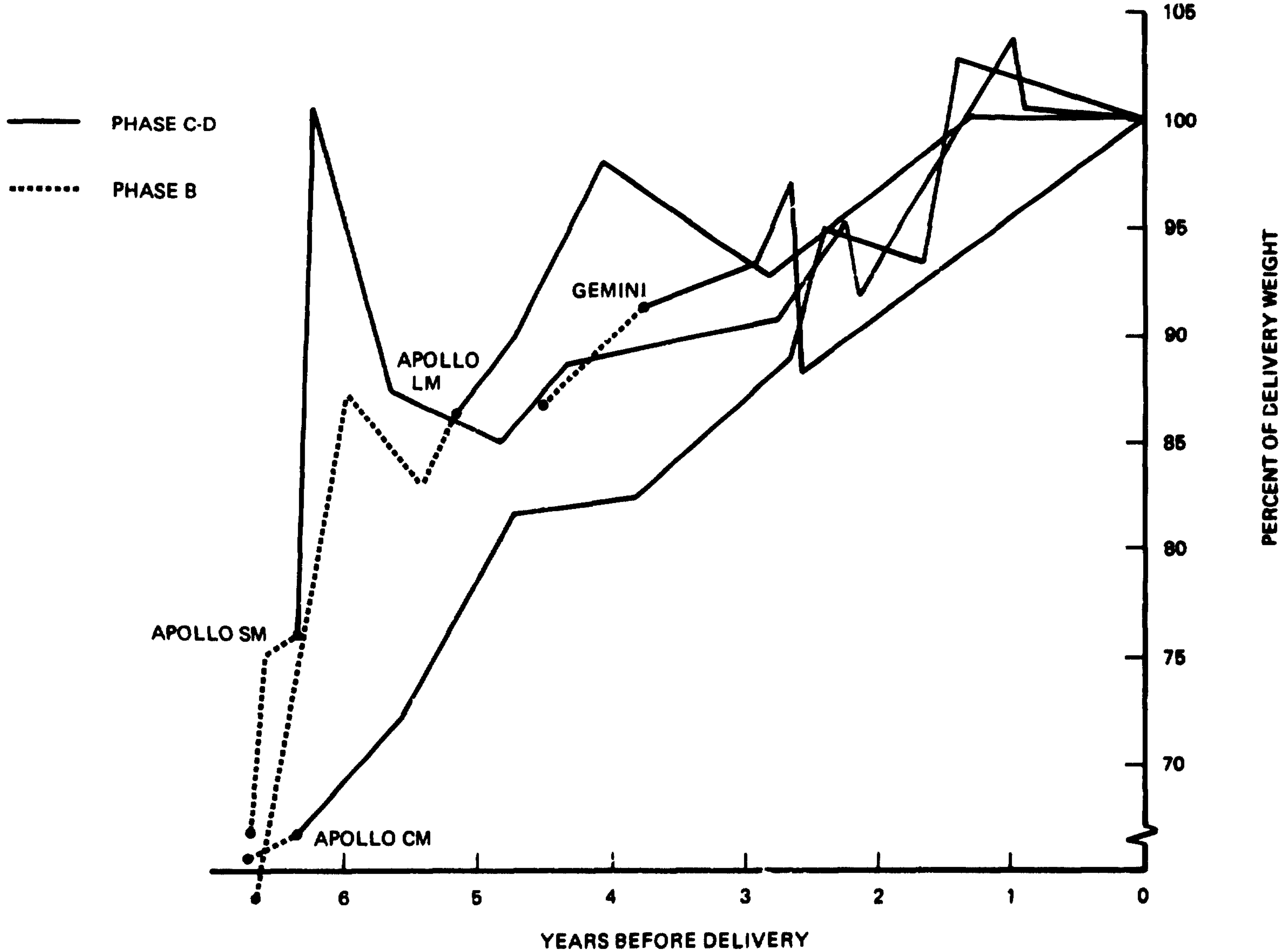

Figure 2-4. Weight Growth – Manned Spacecraft

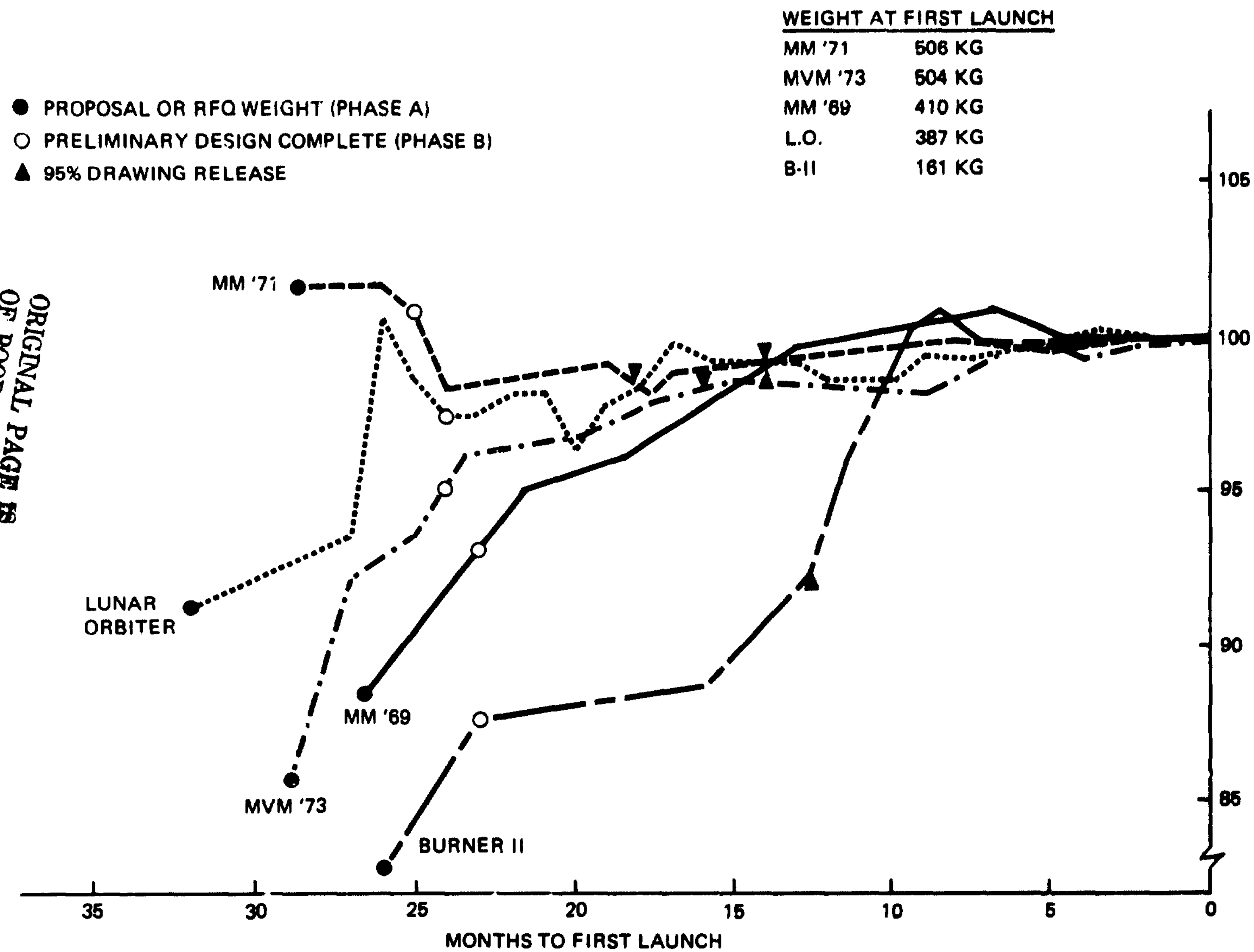

Figure 2-5. Weight Histories of Unmanned Spacecraft

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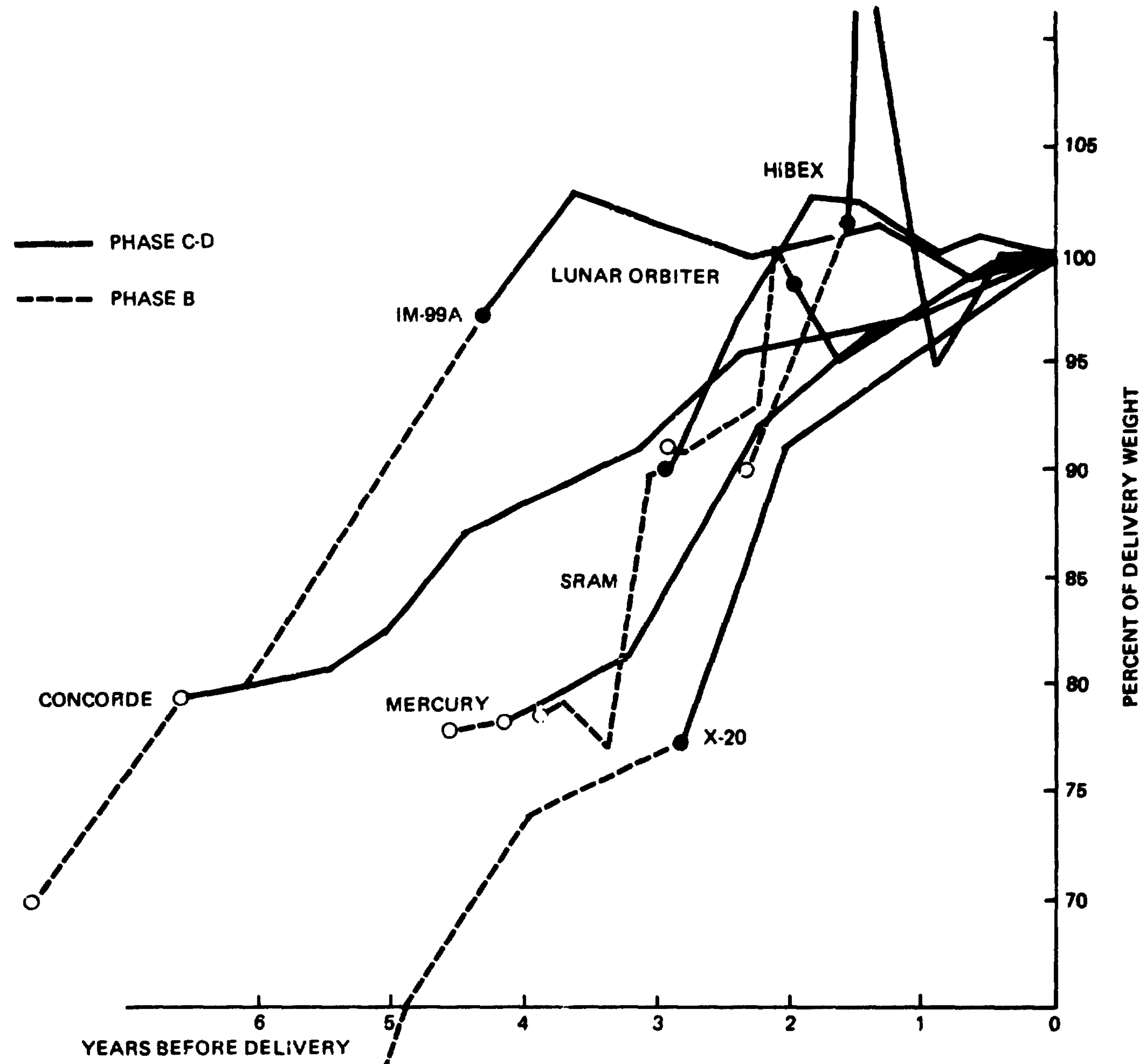


Figure 2-6. Weight Growth — New Concepts

Weight histories can be reviewed by several methods. Most commonly used are plots from a "start" weight (start plots). This method was referred to by Rockwell in SD 70-155-1, "Summary Report for the Space Station Program." A difficulty with these plots is that the start weight is generally ambiguous–it may be a phase B weight, phase A, back-of-the-envelope, or a specification weight. Unless the weights can be correlated to a common point in the design phase, no correlation can be established.

Another method is to plot weight change backwards from an end date that can reflect a common, final actual weight (end plots). These were used in figures 2-2 to 2-6. When plotted as a percent (X) of final weight, the growth indicated at any point back in the program is (100-X)/X. For purposes of this study, these plots were examined for possible regrouping of vehicles into common populations for use as samples in program phase versus growth distribution plots. As an example, jet airplanes indicated fair commonality regarding phases, with the exception of the Concorde. For this reason, Concorde was put into the new concept category, although it could well be placed in either category. The lunar orbiter was placed in both the new concept and unmanned spacecraft categories and the Burner II in booster and in unmanned spacecraft since it is designed for both roles.

Table 2-1 summarizes the result from review and correlation of the various past aerospace vehicles.

2.2 METHOD FOR DETERMINING FSTSA EXPECTED WEIGHT GROWTH

Common "start" dates have been chosen for each historical program as end of phase A and end of phase B, periods spanning those of the FSTSA missions design status. The growth of each of the vehicles in a given technology population is then plotted for growth from phase A and from phase B with each vehicle given equal rank in a distribution plot. Figure 2-7 illustrates the method. Plots for each technology are shown on figures 2-8 and 2-9.

The FSTSA mission vehicle designs had to draw from the various aerospace disciplines represented by these past technologies. The next step in FSTSA growth analysis was to assign a fraction of each technology to the FSTSA design being evaluated and combine these into an FSTSA growth distribution. This is illustrated in figure 2-10. Figure 2-11 shows the resultant expected growth distribution for the low Earth-orbit space station.

Similar plots were used for each FSTSA mission. A value of probability-of-not-exceeding is chosen to arrive at weight growth from the program phase of the mission in question. This may be phase A, phase B, or in between. For requirements-forecasting purposes, a 50 percent probability of not exceeding was used.

Table 2-1. Summary of Weight Growth

TECHNOLOGY				% GROWTH	
				END OF PHASE A	END OF PHASE B
JET AIRCRAFT			UNMANNED SPACECRAFT		
727-22	14.7	2.3	MM '71	-1.7	-0.8
737-100	23.2	7.2	LUNAR ORBITER	9.6	2.7
747-21	14.1	0.3	MM '69	13.1	7.4
XB-47	6.1	6.1	MVM '73	16.7	5.3
B-47A	(1)	2.9	BURNER II	20.8	14.0
B-47B	(1)	3.9	LUNAR ROVER	27.6	20.0
XB-52	2.8	2.1			
B-52A	(1)	6.0			
KC-135	(1)	-0.5	NEW CONCEPTS		
			CONCORDE	46.9	25.9
BOOSTERS			SRAM	27.5	12.0
SATURN S-1C	8.7	8.7	IM-99A	25.4	3.1
SATURN S-II	19.5	19.5	HiBEX	11.2	-2.0
SATURN S-IVB	28.8	28.8	MERCURY	28.5	27.4
MM WING I	23.6	-6.9	LUNAR ORBITER	9.5	1.3
BURNER II	21.0	12.8	X-20	68.0	33.0
MANNED SPACECRAFT					
GEMINI	15.0	9.7			
APOLLO CM	53.4	49.4			
APOLLO SM	52.0	30.0			
APOLLO LM	57.0	16.8			

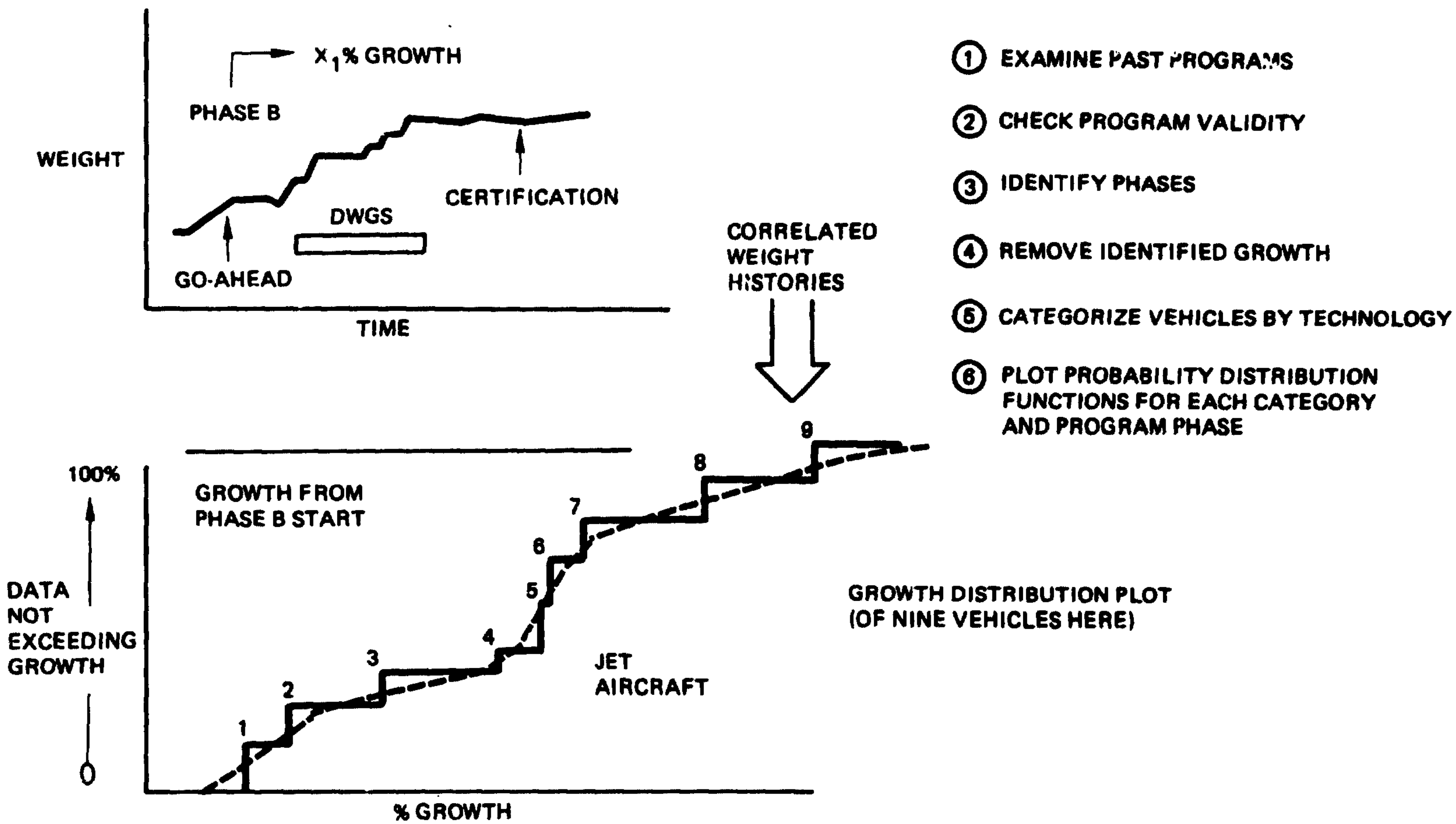


Figure 2-7. Vehicle Development and Weight History Evaluation and Derivation of Growth Distribution Plots

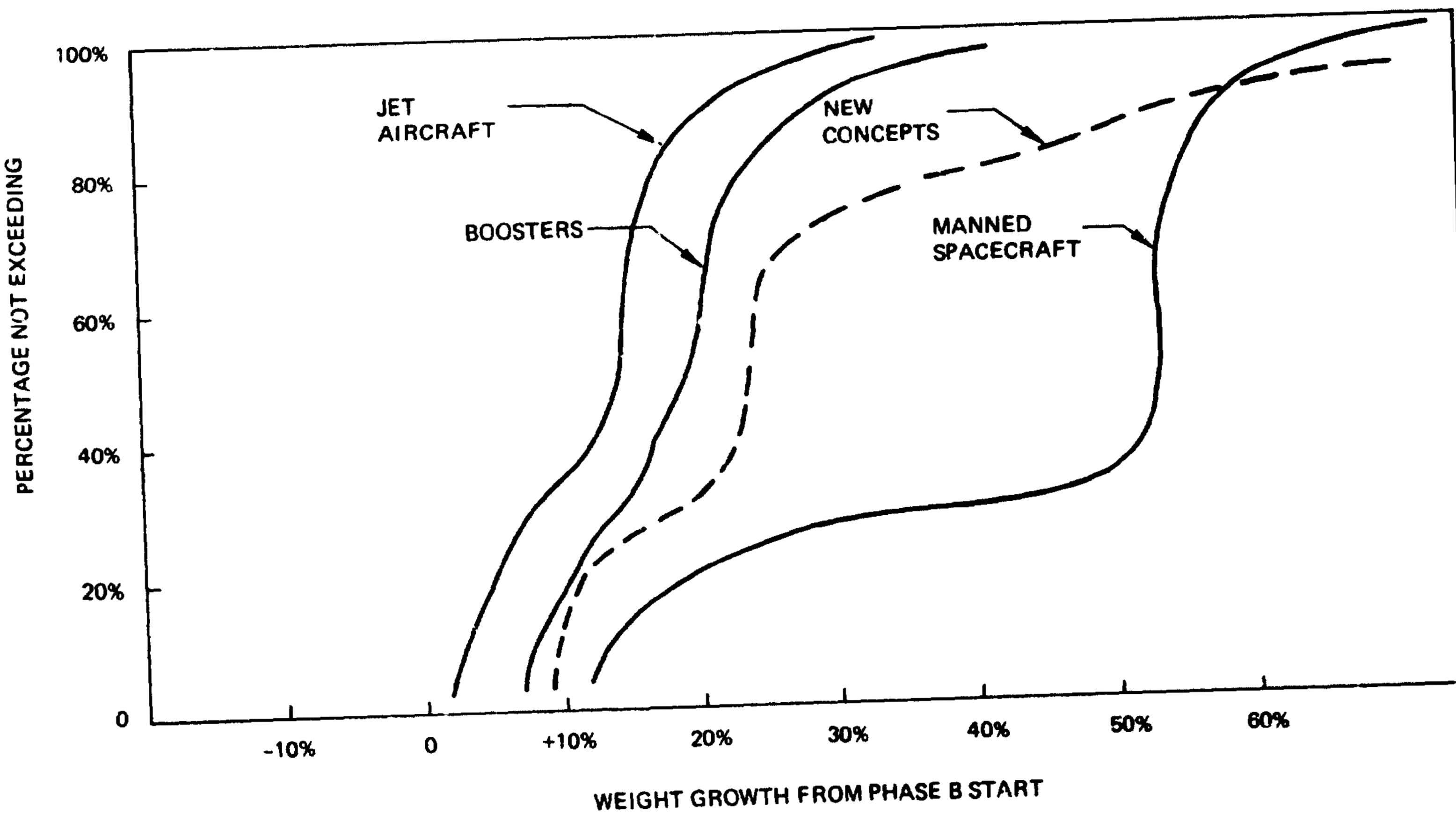


Figure 2-8. Past Program Growth Distribution

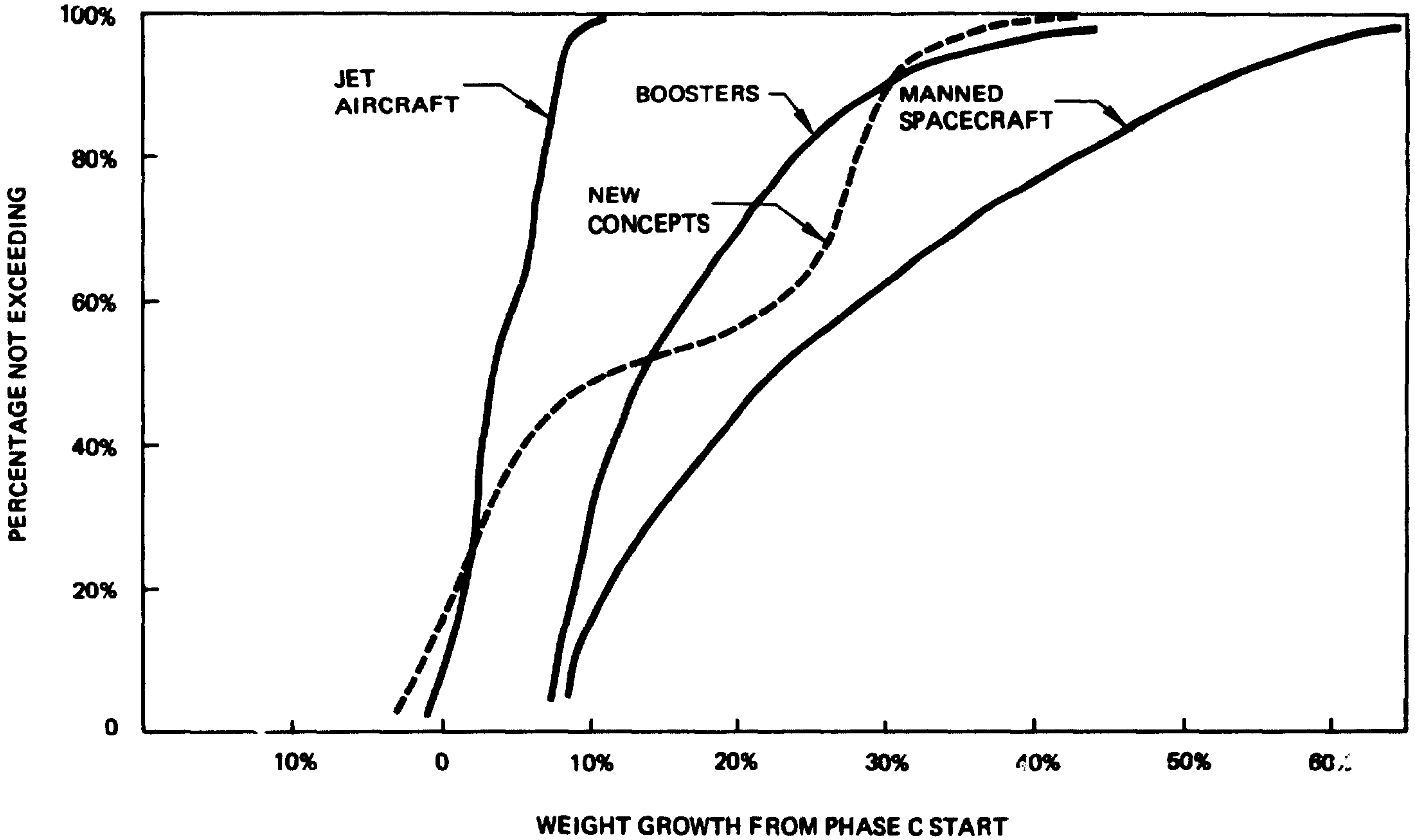


Figure 2-9. Past Program Growth Distribution

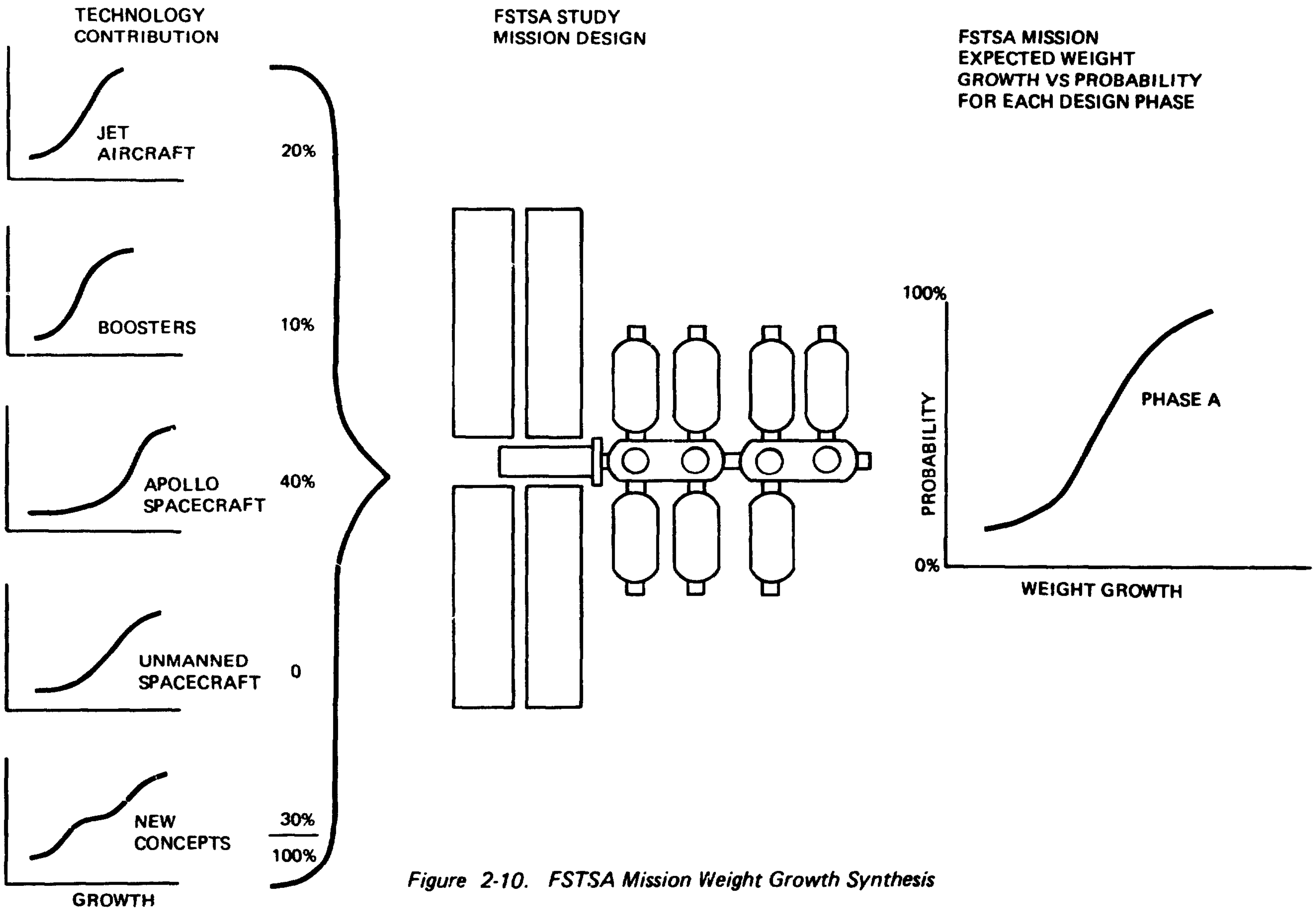


Figure 2-10. FSTSA Mission Weight Growth Synthesis

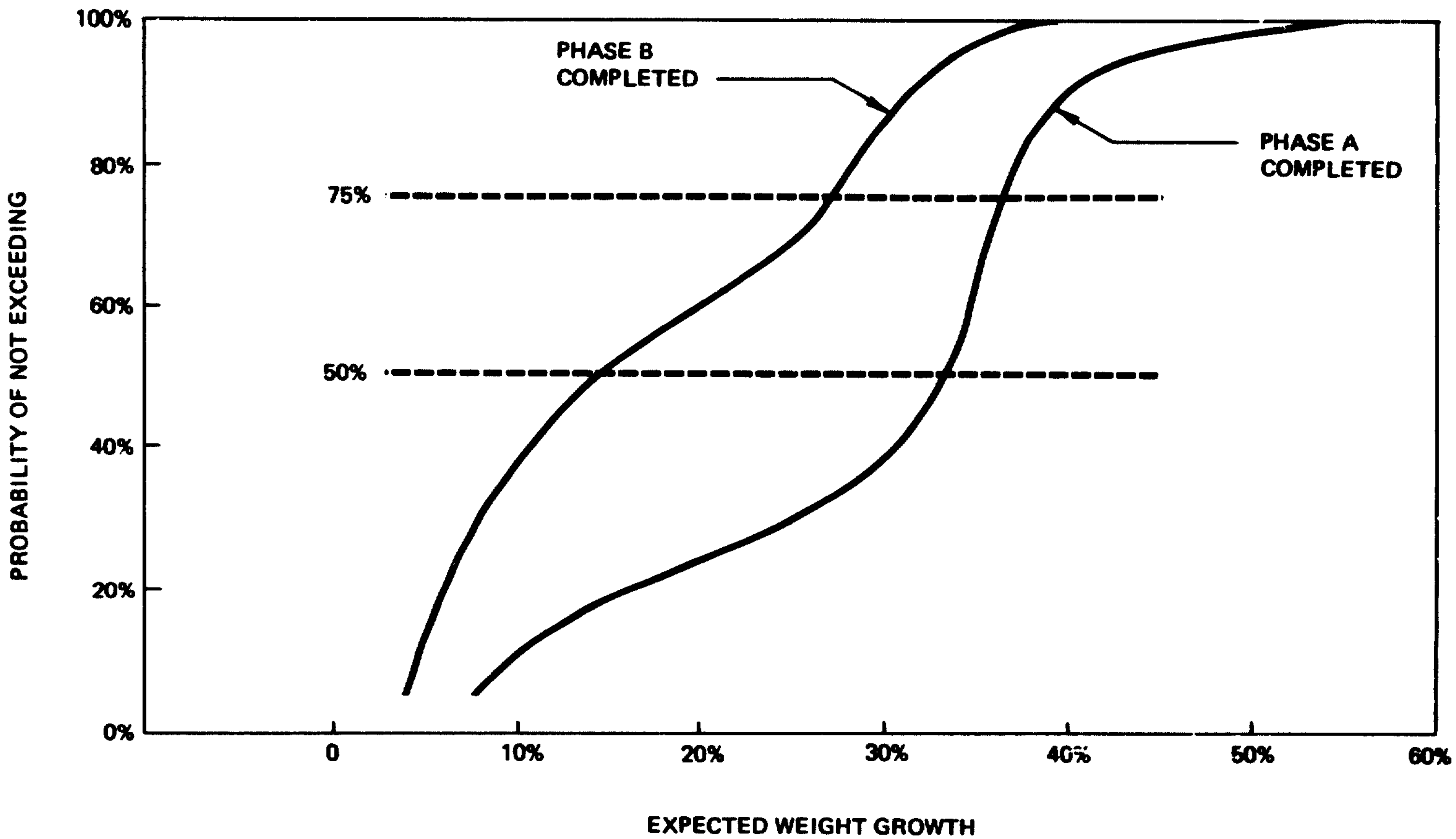


Figure 2-11. Expected Weight Growth for Earth Orbital Space Station (EOSS)

2.3 RESULTS

Table 2-2 shows the results for the major FSTSA missions or vehicles. Values of probability-of-not-exceeding of 50 percent and 75 percent are shown. Note that a reasonable determination of the program phase is necessary. The EOSS has essentially completed phase B, so 50 percent probability weight growth of only 14.9 percent would be expected based solely on historical data. However, the lack of design detail and detailed weight estimates in the EOSS reports indicates that a step further back in the phase relationship may be necessary for weight-estimating purposes. If EOSS were at the end of phase A (phase B start), a value of 33.2 percent would be found. An adjustment that splits the difference between phase A and B appears reasonable. Since the OLS is a direct derivation of the EOSS, this approach was used for the OLS expected weight growth also.

Table 2-2. FSTSA Mission Expected Weight Growth

MISSION	ASSUMED PHASE COMPLETION	EXPECTED WEIGHT GROWTH WITH 50% PROBABILITY OF NOT EXCEEDING	EXPECTED WEIGHT GROWTH WITH 75% PROBABILITY OF NOT EXCEEDING
EARTH ORBITAL SPACE STATION	1/2 B	24%	32%
ORBITING LUNAR STATION	1/2 B	24%	32%
LUNAR SURFACE BASE	A	33%	37%
GEOSYNCH SPACE STATION	A	33%	37%
SPACE BASE	A	37%	41%
INDEPENDENT LUNAR SORTIE	B	20%	27%
AUTOMATED LUNAR	A	20%	26%
AUTOMATED PLANETARY	A	20%	26%
SOLAR POWER STATION	A	20%	24%
MANNED PLANETARY	A	34%	38%
MANNED SPACE PROPULSION	A		
CHEMICAL	A	15%	33%
NUCLEAR	A	31%	36%
UNMANNED SPACE PROPULSION			
CHEMICAL	B	10%	19%
NUCLEAR	A	19%	26%
MANNED LAUNCH VEHICLE	A	27%	29%
UNMANNED LAUNCH VEHICLE	B	12%	20%

APPENDIX 3

ANALYSIS OF SPACE DISPOSAL OF TOTAL SOLIDIFIED NUCLEAR WASTE

Disposal of refined waste was described in section 5.9 of the technical report. It was shown, concurring with earlier NASA studies, that refined waste disposal is practical using the space shuttle and a modified full-capability tug for transportation.

Nuclear waste is presently processed to a solidified form consisting of about 25 percent fission product oxides, less than 1 percent actinides, the remainder being inert (nonradio-active) material. The waste is typically canned in "pots" 0.3m in diameter by 2.4m in length (1 x 8 ft). It would be desirable, if economically practical, to dispose of total waste in this form, eliminating completely the need for long-term Earth storage. Accordingly, a brief study of total waste disposal was performed.

3.1 TOTAL WASTE DISPOSAL PAYLOAD CONCEPT

This concept assumes disposal of total solidified waste, based on current waste solidification technology. The total waste is roughly 1/10th as radioactive per unit mass as the partially refined waste discussed above. The total waste package is illustrated in figure 3-1. It appears practical to provide a portable shield for safe handling and for flight crew protection. It is unlikely, however, that such a massive shield could be designed to survive abort entry and impact. The launch system and operational procedures must provide protection from public exposure. The shield is assumed returned to Earth for reuse.

Requirements are stated in table 3-1. Data shown are typical. Waste can be repackaged to some degree in order to tailor the mass per package to capabilities of the transportation system.

3.2 TRANSPORTATION ANALYSES

3.2.1 Transportation Mode Candidates

The total waste requirement is very demanding, both in terms of total mass and in terms of economics, i.e., transportation cost. Consequently, only very low cost Earth launch options were considered. Orbit transfer options included 1-1/2 stage and common stage (slingshot mode) LO_2/LH_2 OTV's and an electric propulsion option powered by decay heat of the waste itself.

The low cost Earth launch options included a low cost heavy lift vehicle (LCHLV) and a second generation single-stage-to-orbit (SSTO) shuttle. Where the LCHLV is used as the only Earth launch

PRODUCT: SPRAY MELT (TYPICAL OF CURRENT WASTE SOLIDIFICATION PROCESSES)
COMPOSITION - UP TO 25% FISSION PRODUCT OXIDES
DENSITY - TYPICALLY 3000 KG/M^3 (190 lb/ft^2)
DECAY HEAT - 25KW/M^3(0.7kW/ft^3) (TEN YEARS AFTER FUEL REMOVAL FROM REACTOR).
VOLUME - 2.5 LITERS/1000 MWd_{th} = 8500 KG/GW_{ye} = 0.088 ft^3 /1000 MWd_{th}

PACKAGING: 3000 KG (6,600 lb) WASTE IN ONE SHUTTLE FLIGHT

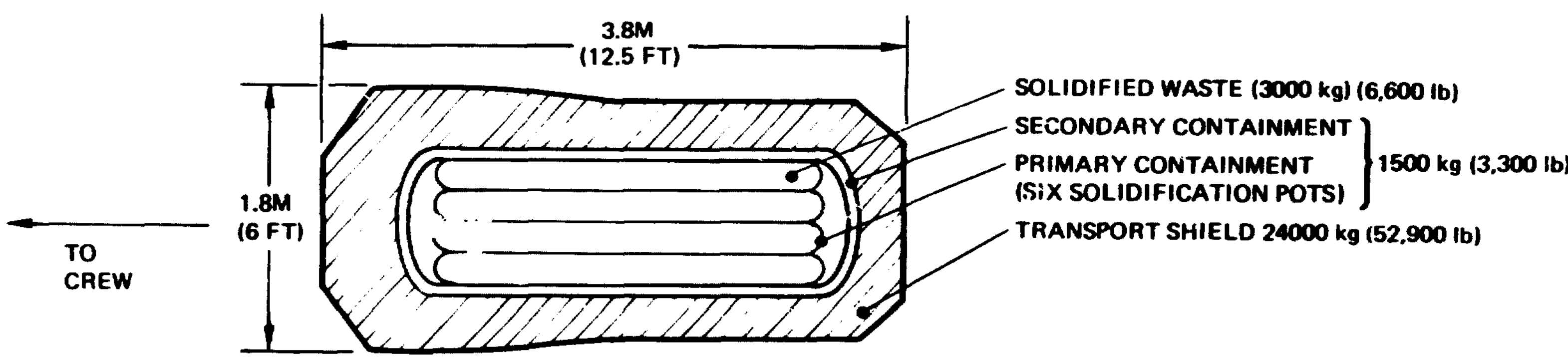


Figure 3-1. Nuclear Waste Disposal Total Waste Packaging Option

Table 3-1. Total Nuclear Waste Dispos· . equirements

REPRESENTATIVE PACKAGE MASS	4500 kg (9900 LB)
REPRESENTATIVE PACKAGE SIZE (DxL)	1 m X 3 m (3 FT X 10 FT)
SHIELD MASS	24,000 kg (52,900 LB)
SHIELD SIZE	1.8 m X 3.8 m (6 FT X 12.5 FT)
PACKAGES/YR TO BE TRANSPORTED (TYPICAL)	1100
MASS/YR TO BE TRANSPORTED	
PACKAGES + SHIELD TO EARTH ORBIT	31.4×10^6 kg (70×10^6 LB)
PACKAGES ONLY TO SOLAR SYSTEM ESCAPE	4.95×10^6 kg (10.9×10^6 LB)

option, gliders similar to the shuttle orbiter, but without main propulsion systems, delivered to orbit by the LCHLV, are used as waste carriers to provide the needed intact-abort capability. The LCHLV is described in Appendix 2. SSTO concepts have been published in the literature, notably by Salkeld, and have been studied by Boeing on IR&D. The Boeing concept is illustrated in figure 3-2. No effort was spent on SSTO concepts by this study.

3.2.2 Transportation Sequences

Figures 3-3 and 3-4 show the t· nsportation sequences investigation for the SSE destination. The first mode employs a LCHLV and a common-stage LO_2/LH_2 OTV. Intact abort capability during Earth launch is provided by the gliders shown. One shielded waste package is carried in each glider. In orbit, the waste packages are extracted from their shields and installed on the OTV system. The shields are returned to Earth by the g. ·:ers. The OTV's oper· te in slingshot mode with the boost stage recovered and the second stage expended along with the payloads to solar system escape.

The second mode employs a SSTO to launch the waste packages and small OTV/drop tank systems to orbit. The waste package goes up last; the shield is recovered by the SSTO. The OTV operates in a perigee kick mode; the drop tanks contain enough LO_2/LH_2 to establish a one day elliptic orbit. At first perigee the injection stage fires to SSE with the payload. All OTV elements are expended.

Table 3-2 provides a summary mission history for the 1-1/2 stage OTV system.

The LCHLV was assumed to have a low orbit payload capability of 200 000 kg (440,000 lb) as for the power satellite program. The SSTO was assumed to have 30 000 kg (66,000 lb) low orbit capability, with return payload capability of 24 000 kg (53,000 lb). The gliders used with the LCHLV were also assumed to have 24 000 kg (53,000 lb) return payload capability.

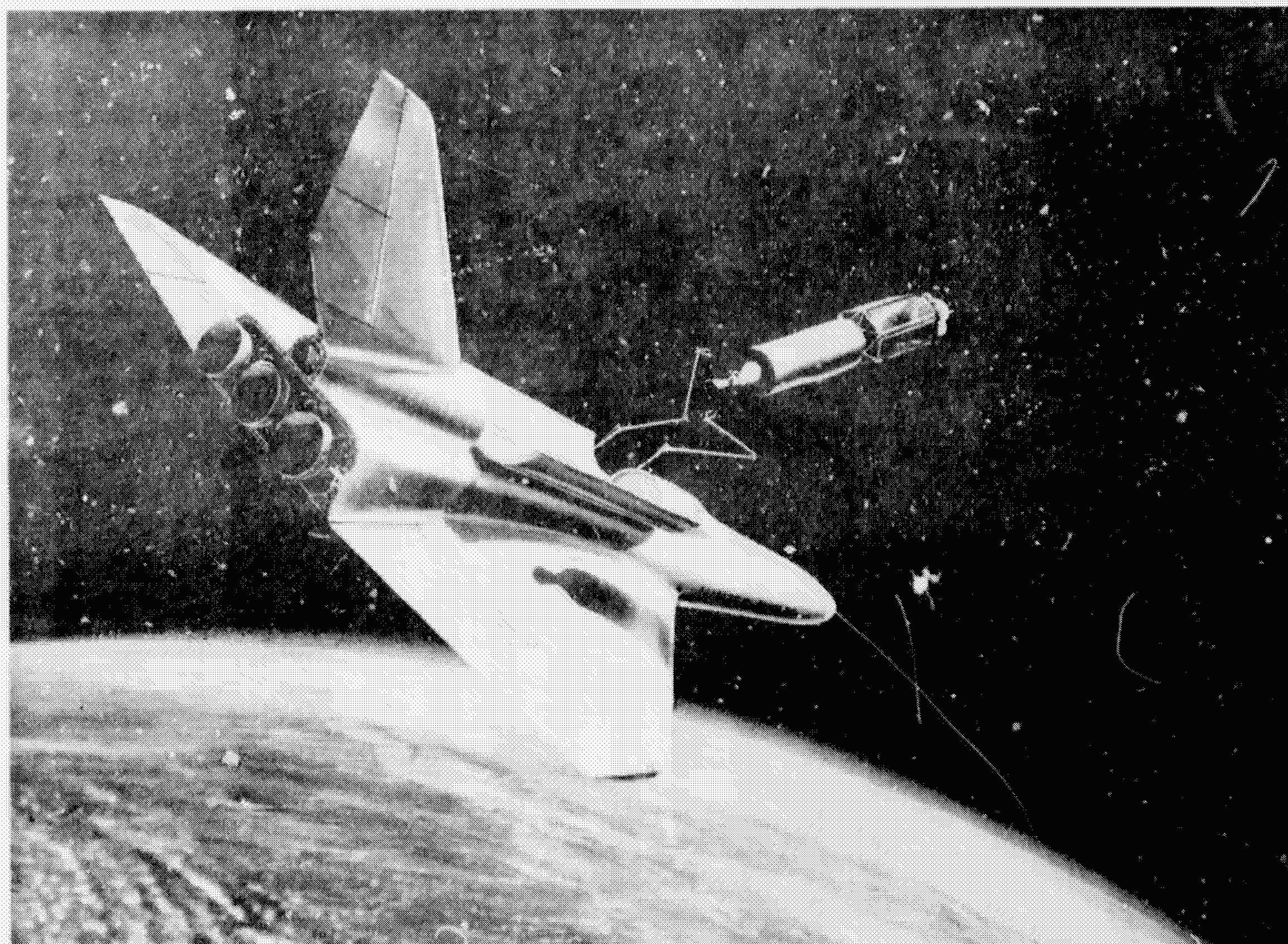


Figure 3-2. Boeing Concept Single Stage-to-Orbit (SSTO)

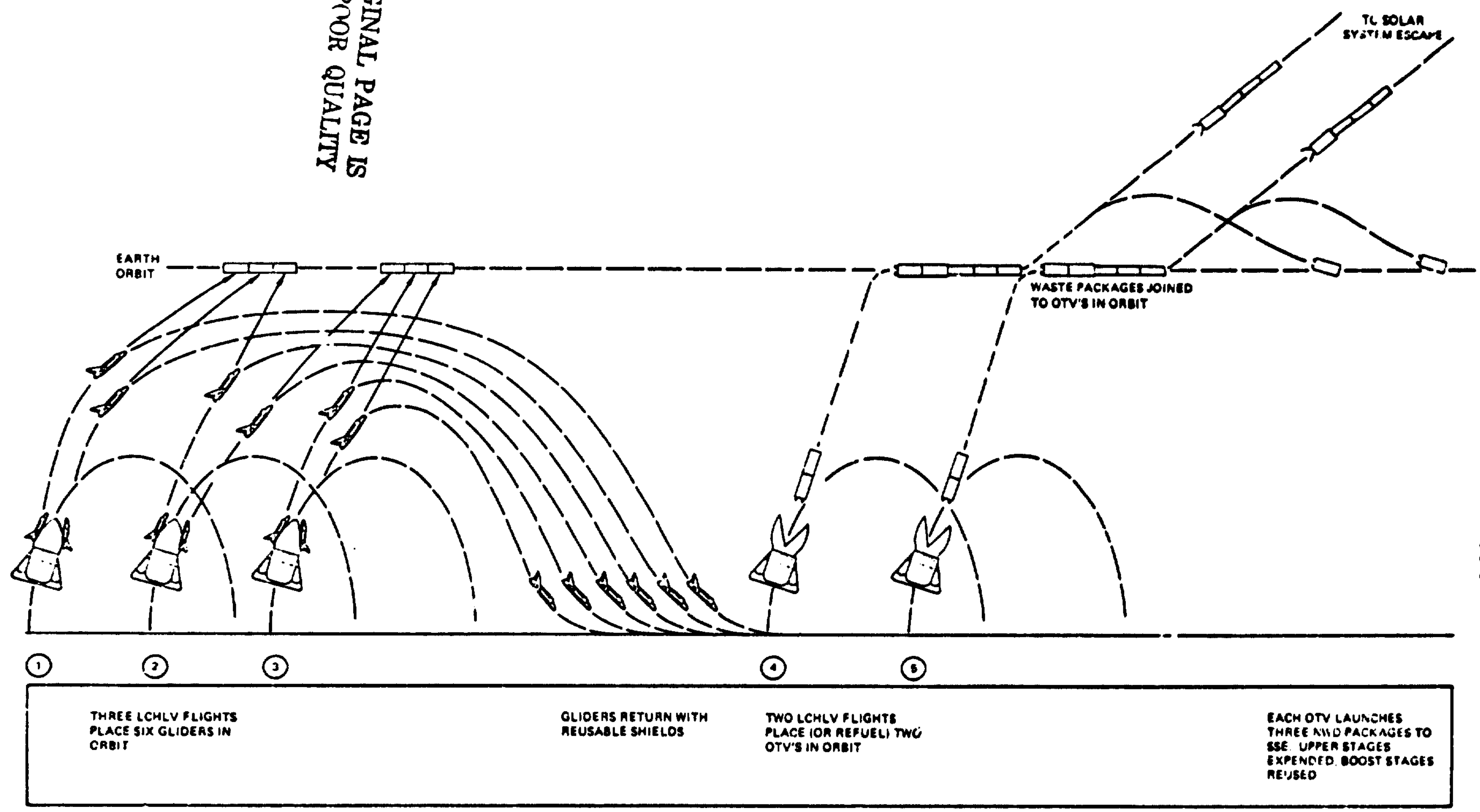
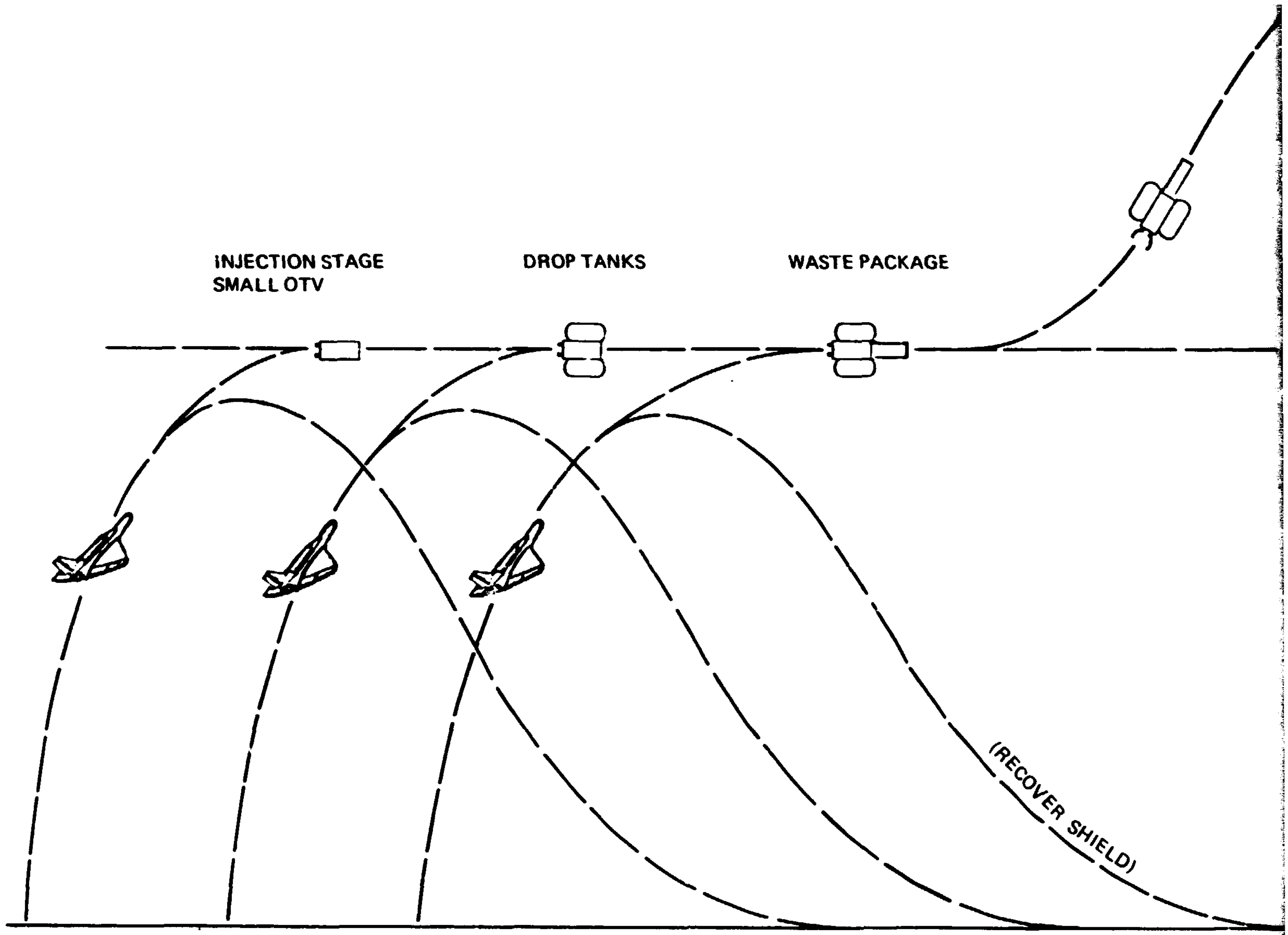


Figure 3-3. Transportation Mode for Nuclear Total Waste Disposal



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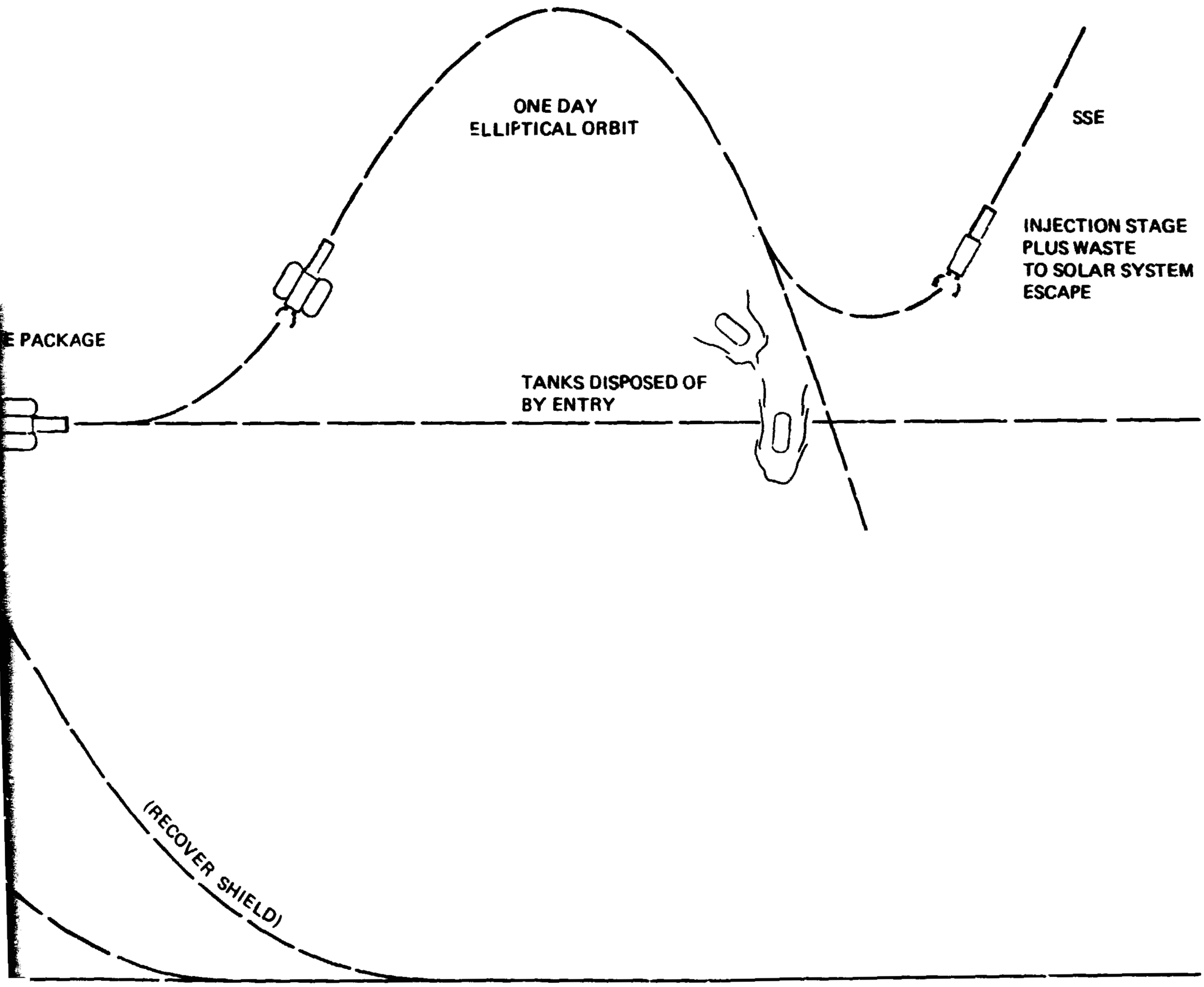

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Figure 3-4 Nuclear Waste Transportation Sequence Employing SSTO and 1-½ Stage OTV

3.2.3 Earth Launch Summary

A summary of Earth launch and OTV requirements for the various options and modes is shown in table 3-3. The ROM busbar surcharge values shown are in cents/kwh, 1975 dollars, and are transportation cost only. They do not include waste processing or packaging costs. Numbers of flights per waste package are indicated with flights per year in parentheses based on 50 and 1,100 waste packages per year, respectively.

Table 3-3. Earth Launch Requirements

MODE	LCHLV FLIGHTS PER YEAR	SSTO FLIGHTS PER YEAR	OTV FLIGHTS (EXPENDED) PER YEAR	OTV FLIGHTS (REUSED) PER YEAR	ROM ($/KWH) COST
LCHLV	913	-	363	363	.0024
SSTO	-	3300	1100	0	.0020

3.3 Special Study: Nuclear Waste Disposal in Space Utilization of Waste Decay Heat

It was suggested that the decay heat of nuclear fission waste products might be used to drive a propulsion system to accomplish disposal of the waste to SSE. A typical conceptual system includes a closed-cycle heat engine operating from the decay heat, generating electricity to drive an electric propulsion system (figure 3-5). Refined and total waste options are examined by the FSTSA study. Only the total waste option appears to be a candidate for this transportation mode because (a) the refined waste as defined by Lewis Research Center has very little thermal power, and (b) it can be handled economically by Shuttle/FCT.

This is an energy-limited problem. The energy available in the waste is finite and must be sufficient to provide the necessary energy change to accomplish the mission. An estimate of the energy available in solidified total waste is presented in figure 3-6. This decay is nearly a straight line on the log/log-plot and therefore may be approximated by $q = at^b$ where q is thermal panel at time t after core shutdown and a and b are curve-fit constants. Decay heat data were obtained from a MIT study and adjusted for representative mass properties of solidified waste. The above expression can be readily integrated to determine total thermal energy available over any period t_1 to t_2. Results are shown in figure 3-7.

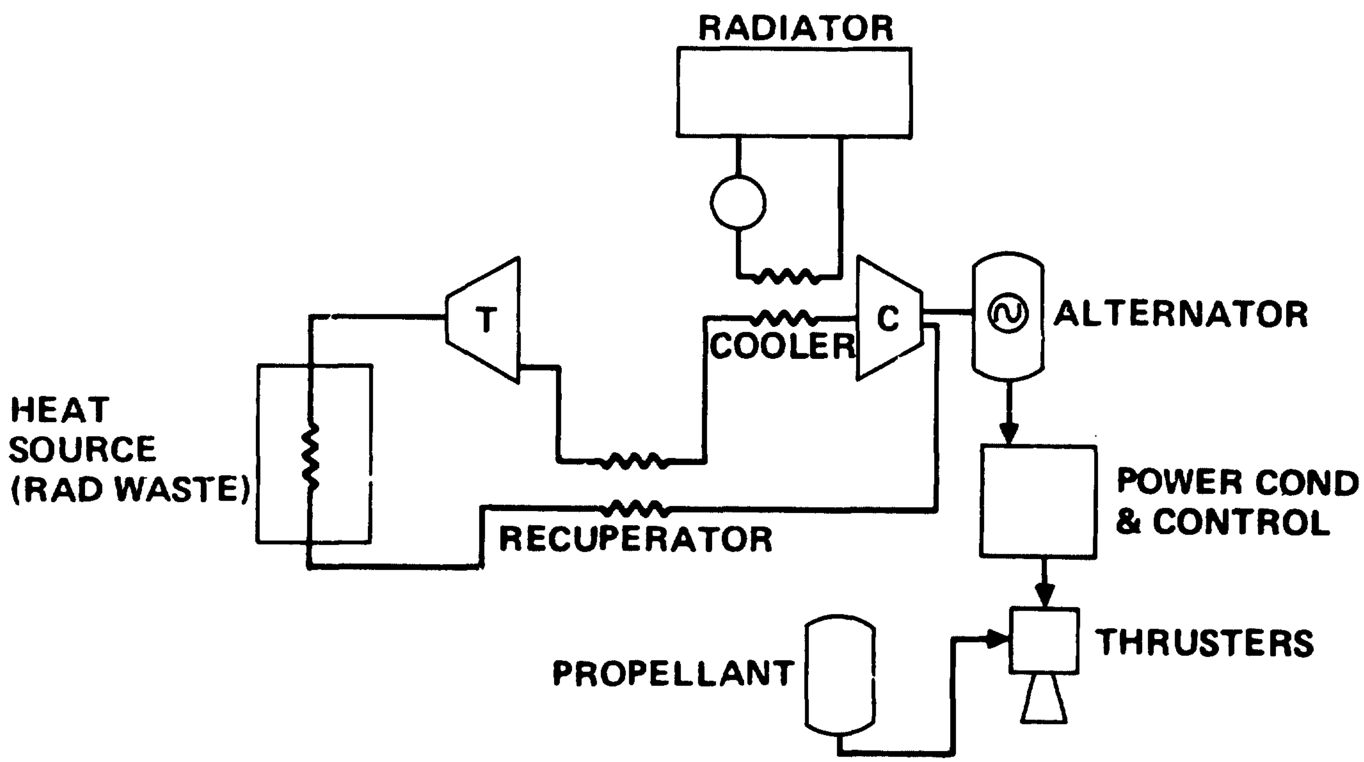

Figure 3-5. System Schematic

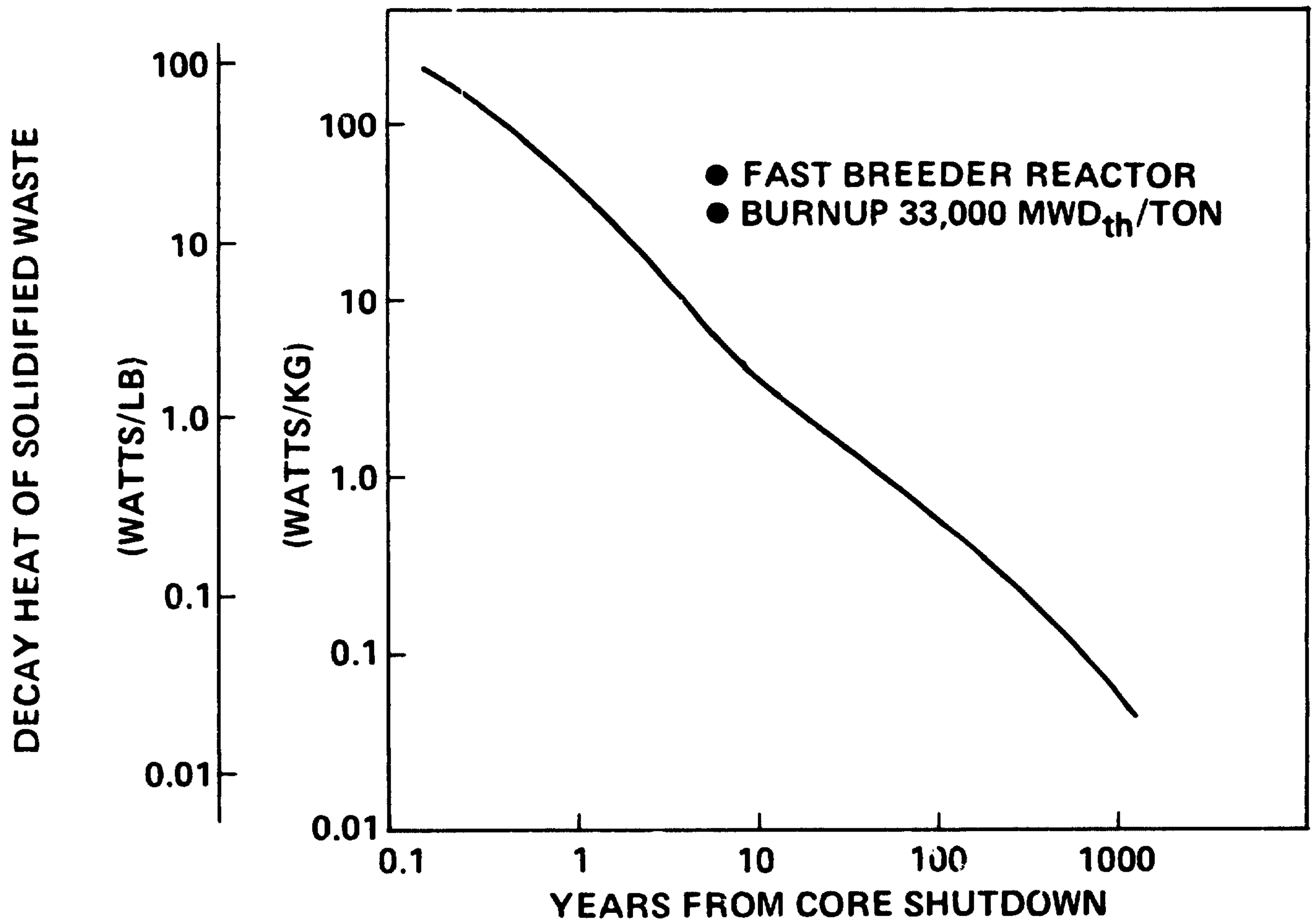

Figure 3-6. Thermal Decay for Solidified Waste

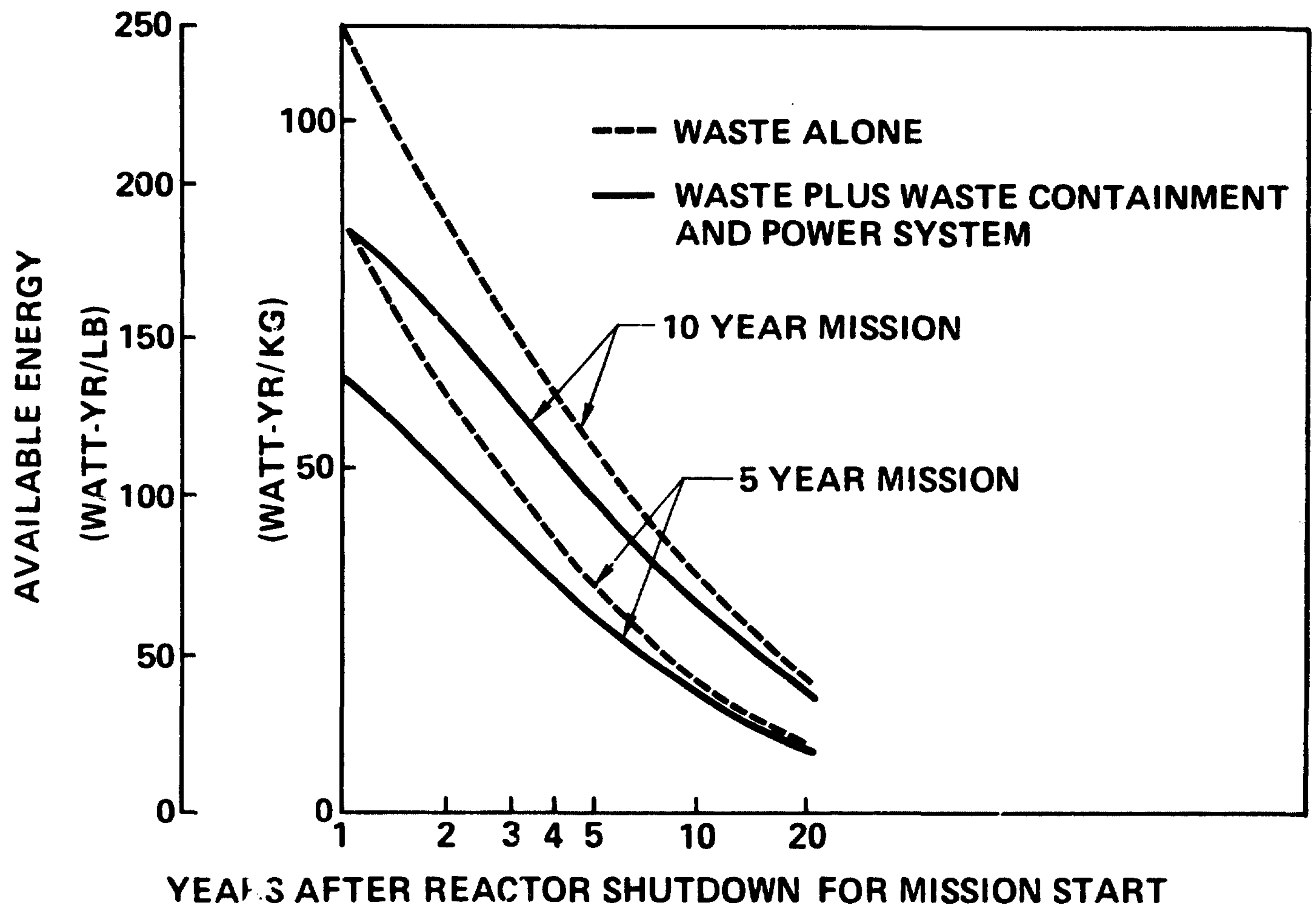

Figure 3-7. Energy Available From Total Solicified Waste

The energy required for solar system escape from low Earth orbit at low thrust is roughly equivalent to a delta V of 25 km/sec (82,000 ft/sec). This large delta V arises because the low thrust system must first escape Earth at nearly the full 7.73 km/sec (25,360 ft/sec) required at infinitely low thrust plus a large proportion of the additional 30 km/sec (98,420 ft/sec) required to escape the solar system at infinitely low thrust. (An impulsive maneuver from low Earth orbit with no gravity losses, can reach solar system escape with a delta V of about 8.8 km/sec (29,000 ft/sec)).

The energy required to achieve a ΔV of 25 km/sec (82,000 ft/sec) is a function of jet velocity (Isp) and of the efficiency of converting thermal energy to jet energy. The required energy versus Isp has a minimum.

This function is plotted in figure 3-8 for cycle and thruster system efficiencies of 40% and 70%.

Comparing this result with figure 3-7 and recognizing the uncertainties in such a brief analysis, the following observations are made:

- There is a question as to whether enough energy for self-propulsion is available in nuclear waste as presently processed. Careful examination of this question and its ramifications should precede any system definition activities.
- A system designed to utilize waste energy for disposal will be sensitive to the "quality," i.e., thermal power, of the waste. It could not dispose of "old" waste and low grade wastes (contaminated shoes, clothing, tools, etc.) except as a payload on high quality wastes.
- The system will have to combine long life with low cost. Propulsive periods on the order of 5-10 years are required.
- A large number of vehicles will be under powered flights in various stages of the escape mission at any one time. All would presumably require some degree of monitoring. We have not made an estimate of the number of vehicles (the number clearly depends on the size of each) but a number in the range between 100 and 1,000 is likely.

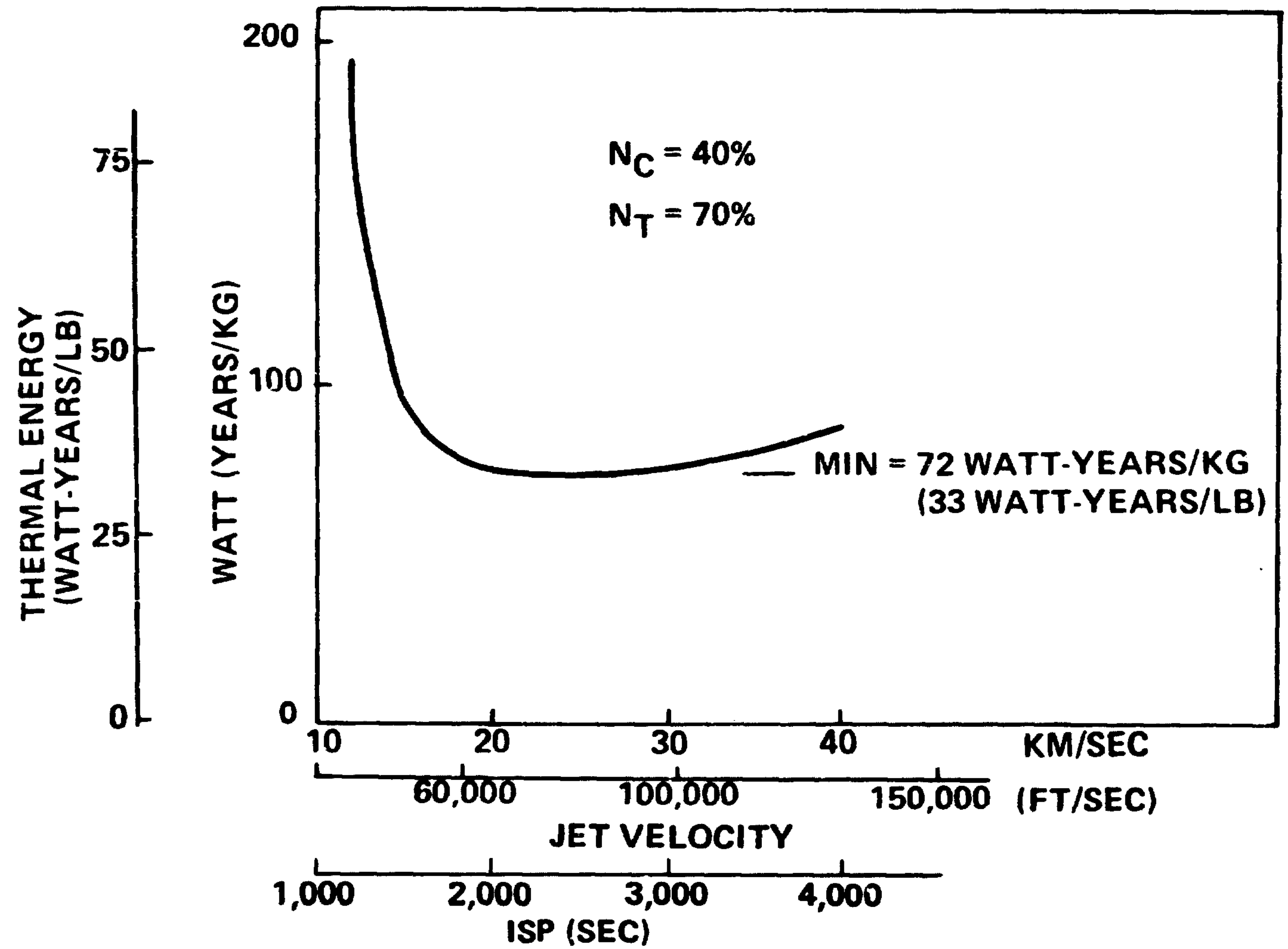

Figure 3-8. Energy Required for Solar System Escape

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