D180-19201-2

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FUTURE SPACE TRANSPORTATION SYSTEMS ANALYSIS STUDY

Phase I Extension
Transportation Systems Reference Data
December 19, 1975

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Boeing Aerospace Company
Seattle, Washington
Research & Engineering Division

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FUTURE SPACE TRANSPORTATION SYSTEMS ANALYSIS STUDY

PHASE I EXTENSION
TRANSPORTATION SYSTEMS REFERENCE DATA

Volume 2
D180-19201-2
December 19, 1975

Submitted to
The National Aeronautics and Space Administration
Lyndon B. Johnson Space Center
in partial Fulfillment of the Requirements
of Contract NAS9-14323

Approved G R Woodcock

G. R. Woodcock
Study Manager

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(VOLUME II)

TRANSPORTATION AND VEHICLE ANALYSIS

1.0 SYSTEMS ANALYSES

This document presents additional details supporting the results reported in Volume I. It is intended that this volume be of great utility in supporting trade and variational studies of the transportation systems covered in this study. It therefore places considerable emphasis on assumptions, methodology, and working data.

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 in this volume 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 are described.

1.1 GENERAL

1.1.1 Requirements and Guidelines

Requirements were identified to cover the space transportation options covered in Volume I. Since many system design requirements apply to several or more transportation options, the requirements have been collected under the following categories:

A. Applicable to all OTVs and LTVs.

B. Applicable to only OTVs or LTVs.

C. Applicable to type or propellant.

Requirements applicable to staging methods are included under categories A and B. The system design requirements are presented in Table 1-1.

Mission oriented requirements influencing the transportation system design include payload mass and c.g., duration and delta V's. Payload and duration requirements used for initial vehicle sizing are summarized in Table 1-2.

Table 1-1(a). System Design Requirements and Guidelines

		All OTV's and LTV's
1.	Technology base	1980 +
2.	Sizing philosophy	For payloads requiring greatest number of flights.
3.	Payloads	Manned and unmanned.
4.	Design life	
	Expendable	Varies with mission duration.
	Reusable	* 20 Round trips or 5 years.
5.	Earth launch	* Protected by shroud, if launched by vehicle other than Shuttle.
6.	Maximum stage diameter	* 7.93M (26 feet) O.D. with HLLV. 4.42M (14.5 feet) O.D. with Shuttle.
7.	Propellant containment	Separate fuel/oxidizer tanks
8.	Tank shape	0.7 Elliptical heads with cylinder. If less than stage diameter, then use 0.7 heads.
9.	Ullage	
	Cryogenic	5% LH_2, 4% LO_2.
	Storable	3%
10.	Maximum thrust/weight	
	Initial	* 0.3 g
	Burnout	$\leq$3.0g
11.	Engine characteristics	* $>$150
	Expansion ratio	Not unless necessary
	Nozzle retraction	20.6 MN/M^2 (3,000 psia)
	Chamber pressure	F $>$ 220,000 N (50,000 lb_f) 10.3 MN/M^2 (1500 psia) } LO_2/LH_2 F $<$ 220,000 N (50,000 lb_f) 10.3 MN/M^2 (1,500 psia) LO_2/MMH
	NPSP	$<$7 kN/M^2 (1 psi) LO_2, $<$14 kN/M^2 (2 psi) LH_2
12.	Launch loads	
	Shuttle	+ 2.0g, -3.3g (x-axis)
	HLLV	* ~3g
13.	Crash loads	
	Shuttle only	+ 9.0g, -1.5g (x-axis)
14.	Meteoroid protection	* P_O = 0.97 for 5 years.
15.	Reserves	FPR = 2% of total ΔV APS = 10% of total W_p Fuel cell reactants = 10% Propellant biasing = TBD
16.	Docking	Radial misalignment (℄) : 0 - 0.30M (1.0 feet) Angular misalignment: 0 - 5.0 degrees Longitudinal (axial) closing rate: 0.03 - 0.3 M/S (0.1 - 1.0 fps) Lateral (transverse) closing rate: 0-0.09 M/S (0.3 fps) Angular (relative) closing rate: 0-0.5 degree/seconds
17.	Weight growth	25% on dry weight.
18.	Reliability	* 0.95 Per flight (initially include redundancy on mission critical systems/components.)
19.	In-orbit refurb. concept	Fluid transfer "Black box" replacement (no repair).
20.	Refueling	Fuel transfer while under low "g".
21.	Single stage	Docking provisions for payloads. Fuel transfer provisions (unless expendable)
22.	1-½ Stage	
	Drop tank	Will be expendable Only include propellant and required pressurization system and thermal protection. No special meteoroid shielding.
	Main stage	Provide docking provisions for payloads and drop tanks Provide provisions for fuel transfer.

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Asterisk items are working assumptions subject to revision if cost/performance improvements result

Table 1-1 (b). System Design Requirements and Guidelines

OTV's only	
1. Operating mode	• Subsystems will be included to allow operation independent of payloads.
2. Engine characteristics	• Throttling to limit maximum axial acceleration to 3g's.
3. Common stage	• Return first stage to LEO. • Second stage interfaces with payloads. • Ability to dock stages together or with other properly equipped space systems. • Both first and second stage capable of operating independent of payloads.
4. 1-½ Stage	
• Main stage	• Propellant obtained from orbital tanker (not from oversized drop tanks).
• Drop tanks	• Should not remain in destination orbit.
5. Nuclear LH_2 stage	
• Engine	• 330 000N (75,000 lb_f) Nerva type.
• Aft bulkhead	• 0.7 Elliptical (not 10° - 15° conical due to length).
• Tank diameter	• 7.93m (26 feet) O.D. . . . . not launched within shroud.
• Radiation shielding	• Shadow type.
6. Nuclear electric stage	
• Reactor	• Heat pipe cooled.
• Conversion	• 30,000 hours design life
• Radiation shielding	• Brayton cycle
• Thrusters	• Reactor and conversion system fully enclosed.
	• Kaufmann type or Argon UPD.
7. Solar electric stage	
Application	• Power satellite transfer to GEO.

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Table 1-1. (C) System Design Requirements and Guidelines

LTV's only	
1. Operating mode	● Power and avionic: subsystems will be common for the CEM and propulsion system.
2. Payload handling	● The CEM will not be separable from propulsion system. ● The ability to deliver, deploy and :eturn payloads to lunar orbit.
3. Landing condition	● Engine clearance = 0.75m (2.5 feet). Tip-over ratio = 1.2 (landing gear radius/c.g., height).
4. Engine characteristics	● Throttling = 10% - 100%.
5. 1-½ Stage Main stage Drop tank(s)	● Propellant obtained from drop tanks. ● Sized to include propellant for main stage operation. ● Tanks to be separated at end of braking maneuver and allowed to crash on surface up range of landing site.

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Table 1-2. Mission Requirements Summary

Mission	Payload [0] Delivered 10^3 KG	Payload [0] Delivered 10^3 lb	Payload [0] Returned 10^3 KG	Payload [0] Returned 10^3 lb	Active operating duration (days) [1]
GSS – OTV	25	55	15	33	7
GSMS – OTV	6.6	14.5	6.6	14.5	9
ILSS – OTV [2]	39.1	86	7.3	16	40/30 [6]
LTV	11.8	26	7.3	16	15
OLS – OTV [2]	59	130	7	15	22/9 [7]
LTV	15.9	35	11.4	25	28
LSB – OTV [2]	71.4	157	7.3	16	22/9 [7]
LTV	27.7	61	6.4	14	4
M.P. [3] – TMI, MOI, TEI	109.6	241	50.45	111	450/1,000
M.S.S.R. [4] – INJ STG	5.6	12.4	0.14	0.31	1
J.B.P. – INJ STG	2.4	5.3	—	—	1
G.L. – INJ STG	5.9	13	—	—	1
N.W.D. [5] – INJ STG	3.2	7.1	—	—	1
S.E.S. – P.V. – OTV	23,640	52,000	—	—	100
T.E. – OTV	16,360	36,000	—	—	100
P.V. Pilot plant – (OTV)		750	—	—	360

[0] Per flight
[1] Excludes LEO assembly time.
[2] For single stg LO_2/LH_2 LTV.
[3] Opposition/conjunction class.
[4] Direct earth landing - slow return.
[5] Refined waste - each package.
[6] Function of landing site.
[7] Return to E.O. constraint/no constraint.

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A study of tank diameter criteria was made as illustrated in Figure 1-1. Three aeroshell options were considered as shown in Figure 1-2. The hammerhead option was selected. Sizing groundrules for the point designs are depicted in Figure 1-3. The 1½ stage LO_2/LH_2 OTV for the GSS mission was sized with two sets of drop tanks because a single set resulted in drop tanks longer than the main stage, posing operational problems and restrictions. The first set of tanks provides propellant for the boost maneuver. They are separated just prior to circularization at apogee, where a small separation delta V will result in tank disposal by atmosphere entry and burnup. The second set of tanks provides propellant for circularization. They are not separated until after deorbit, when a small separation delta V will again result in tank disposal.

A trade study was run to ascertain the benefits of common bulkheads. Two sizes of OTVs were examined. The results showed, in one case, a very slight mass advantage and, in the other, a very slight mass disadvantage (Table 1-3). Therefore, as they are simpler to design and construct, separate-tank configurations are used except in those instances where the shorter common bulkhead configuration is a significant advantage.

1.1.2 Technology Assumptions

The following technology definitions and selections were developed as working groundrules.

Structures

- Graphite-plastic matrix composites are assumed for unpressurized main structures in reusable vehicles; aluminum skin/stringer is assumed for expendable vehicles or expendable tanks.
- Aluminum is assumed for main propellant tanks with integral stiffening as required.
- Elevated temperature materials are assumed where normal working temperatures for aluminum or composites are exceeded. For example, structural elements of the nuclear electric tug would be titanium due to thermal radiation from hot parts and the heat rejection radiator.
- High temperatures and associated environments are limited to known capabilities of known engineering materials.
- Reusable heat shields will assume shuttle technology where applicable; water cooled or other special heat shields will be used where circumstances merit a departure from shuttle technology. For example, multiple-pass aerobraking maneuvers may use aluminum, titanium,

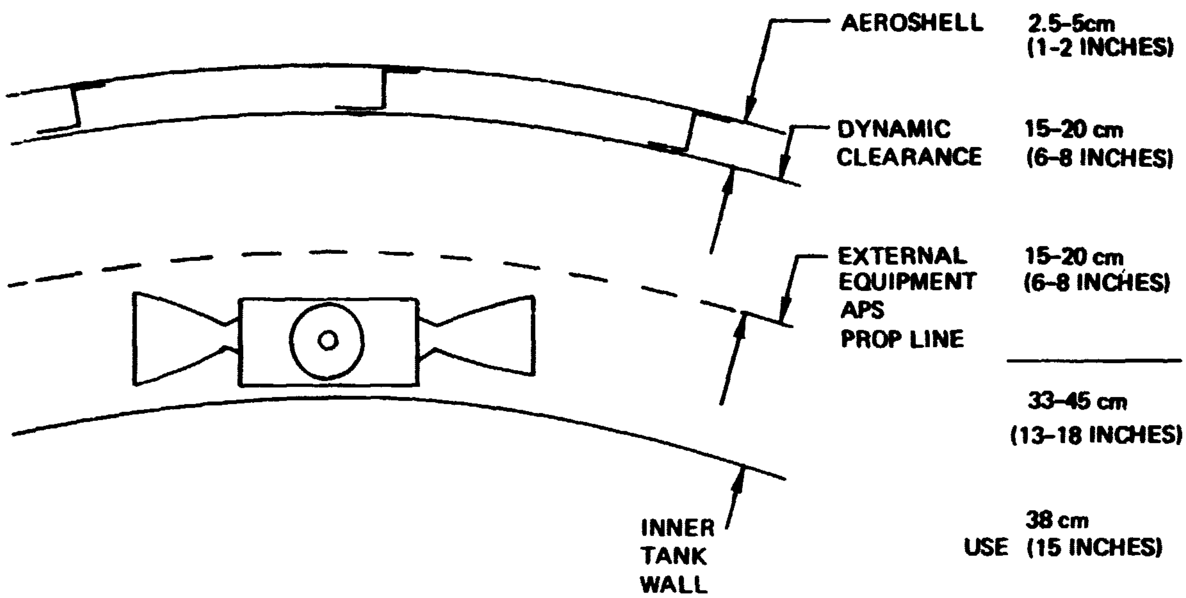
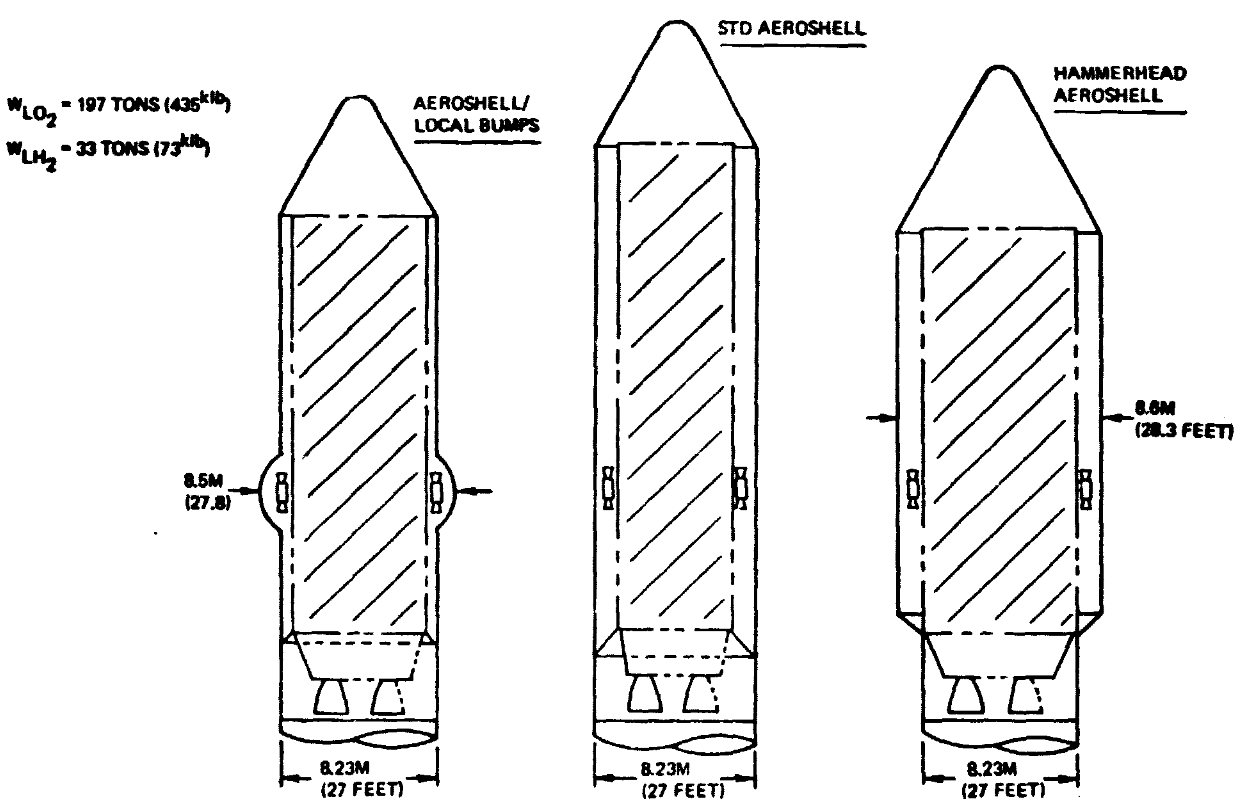


Figure 1-1. Tank Diameter Criteria

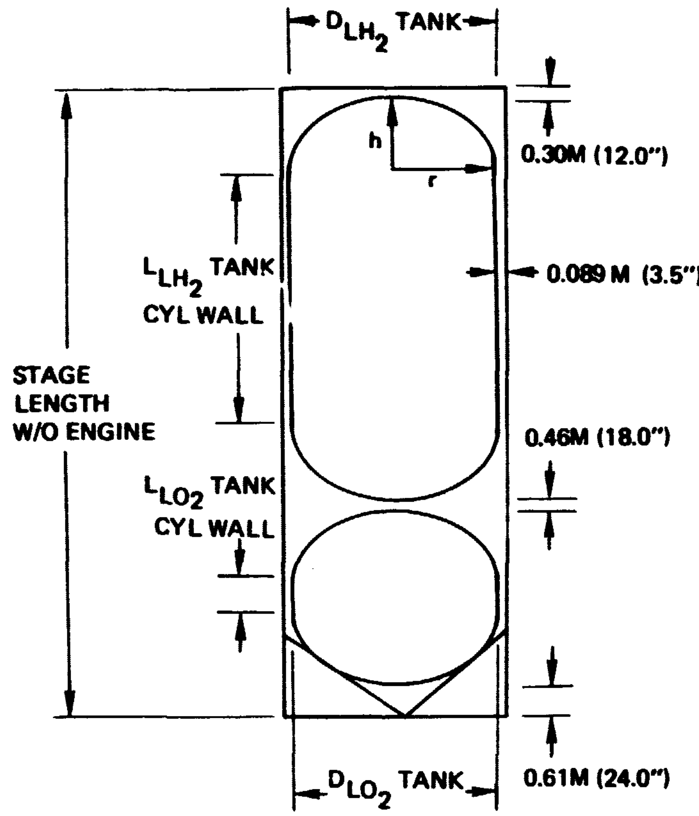


ALLOW.TANK DIAMETER	7.7M (25.4 FEET)	7.5M (24.6 FEET)	8.2M (27 FEET)
Δ TANK LEN (LO_2/LH_2)	0	0.5/1M (+1.7/3.3 FEET)	0/1M (0/-3.1 FEET)
Δ TANK WEIGHT (POUNDS)	0	?	?
Δ A/S WEIGHT (POUNDS)	?	?	?

Figure 1-2. Allowable Tank Diameter LO_2/LH_2 Single Stage GSS Mission

OTV's

- ELLIPSOIDAL TANK HEADS
  (h = 0.707r)
- CLEARANCES AS SHOWN



DROP TANKS

- ELLIPSOIDAL TANK HEADS
  (h + 0.707r)
- NO EXTERNAL WALL/BUMPER
- CLEARANCES AS SHOWN

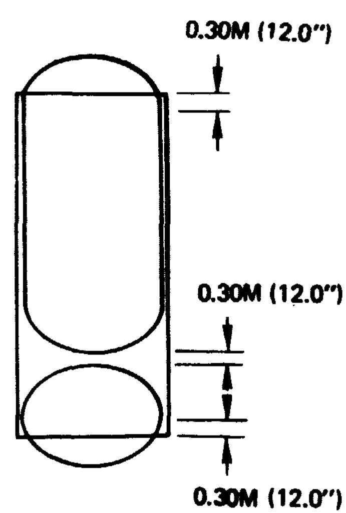


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Figure 1-3. Point Design Sizing Ground Rules

Table 1-3. Common Bulkhead Weight Study

Large OTV 7.62M (25.0 feet) diameter	Two tank configuration		Common bulkhead configuration	
LH_2 Tank bulkheads	1,530 LB_M	694 KG	765 LB_M	347 KG
LH_2 Tank cycle wall	3,210	1,456	4,420	2,005
LO_2 Tank bulkheads	1,530	694	765	347
LO_2 Tank cycle wall	395	179	395	179
Common bulkhead	—	—	1,670	758
Body shell	8,120	3,683	7,220	3,275
Tank insulation	2,265	1,027	2,040	925
Total variable weight	17,050 LB_M	7,733 KG	17,265 LB_M	7,836 KG
Medium OTV 4.42M (14.5 feet) diameter	**Two tank configuration**		**Common bulkhead configuration**	
LH_2 Tank bulkheads	290 LB_M	132 KG	145 LB_M	66 KG
LH_2 Tank cycle wall	1,215	551	1,420	644
LO_2 Tank bulkheads	290	132	145	66
LO_2 Tank cycle wall	305	138	305	138
Common bulkhead	—	—	290	132
Body shell	2,645	1,653	3,345	1,517
Tank insulation	1,040	472	990	449
Total variable weight	6,785 LB_M	3,078 KG	6,640 LB_M	3,012 KG

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or superalloy metal heat shields; the low cost space freighter for power satellite application may merit a water cooled metal re-entry heat shield to minimize refurbishment and turnaround time.

Avionics

- LSI circuit chip technology is assumed available for data processing hardware; data bus techniques are assumed to minimize wire mass.
- Communications and GN&C systems assume shuttle and full-capability tug technology levels. Laser radar is assumed available for rendezvous as required.

Electric Power

- Fuel cells and batteries are assumed for electric power except for electric propulsion primary power.
- Fuel cells tailored to the application, based on shuttle technology, are assumed. Batteries are assumed to be Ni-Cad.
- Nuclear-electric power technology assumptions pertinent to the nuclear electric tugs are summarized in the discussion of the nuclear-electric tug point designs, paragraph 1.2.3 of this volume.

Main Propulsion

- Main engine characteristics will be derived from parametric performance based on Space Shuttle Main Engine (SSME) technology. Expander cycles are assumed for engines below 220 KN (50,000 lb) thrust. Engine performance assumptions were included in the guidelines data of paragraph 1.1.1.
- During the Phase II study effort, cost benefits associated with use of off-the-shelf or modified engines will be analyzed.

Auxiliary Propulsion

- The use of hydrazine monopropellant has been baselined, since gains associated with more advanced technology are minimal in the cases analyzed to date (see Subsystems task discussion below). Storable bipropellant and advanced O_2/H_2 auxiliary propellant technologies are assumed available as needed.

Thermal and Meteoroid Protection

- Multilayer metallized plastic film (MLI) insulation is assumed for thermal protection of all main propellant tanks. A metal skin, non-structural for vehicles with integral tanks, is assumed external to the MLI and is to be thick enough so that, in conjunction with the MLI, it provides sufficient meteoroid protection.

1.1.3 Predevelopment Technology Requirements

Technologies for which basic feasibility is not yet demonstrated, e.g., laser-fusion propulsion, were not assumed in this study. A number of technical capabilities were assumed that have not been flight-demonstrated. In these cases a predevelopment technology program is appropriate. Recommended technology programs are tabulated versus transportation options in Table 1-4.

1.2 Point Designs and Analyses

1.2.1 Heavy Lift

Preliminary analyses of heavy lift options were performed during Phase I. These options were selected as representative of the two classes of heavy lift systems for which potential needs were recognized. A wide range of classes and options are presently under separate study, the Heavy Lift Launch Vehicle (HLLV) study. The FSTSA study will rely principally on data from the HLLV study during Phase II.

1.2.1.1 Shuttle-Derived Systems

Two options were investigated and are depicted in Figures 1-4 and 1-5. 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 1-5.

Representative trajectory data for the SRB/ET vehicles are shown in Figures 1-6 through 1-9.

1.2.1.2 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 1-10 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

Table 1-4. Predevelopment Technology Developments

Technology development ▶ / Transportation options	Fuel transfer & gaging system for refueling	Stage-to-stage docking system	Tank-to-stage docking system	LO/MMH engine technology demonstrator	Nuclear/electric reactor technology	Argon MPD thruster	Deep throttling engine	Large-angle gimbal system	Lunar day/night environment protection	Watercooled entry heat shield	Water soft landing control system
LO_2/LH_2 Single stage	X										
LO_2/LH_2 Common stage	X	X									
LO_2/LH_2 1½ Stage	X		X								
LO_2/MMH Single stage	X			X							
LO_2/MMH Common stage	X	X		X							
Nuclear LH_2	X										
Nuclear electric					X	X					
Solar/chemical hybrid	X		X			X		X			
Lunar transport vehicles (LO_2/LH_2)			X				X		X		
Lunar transport vehicles (LO_2/MMH)			X	X			X		X		
Low cost space freighter										X	X

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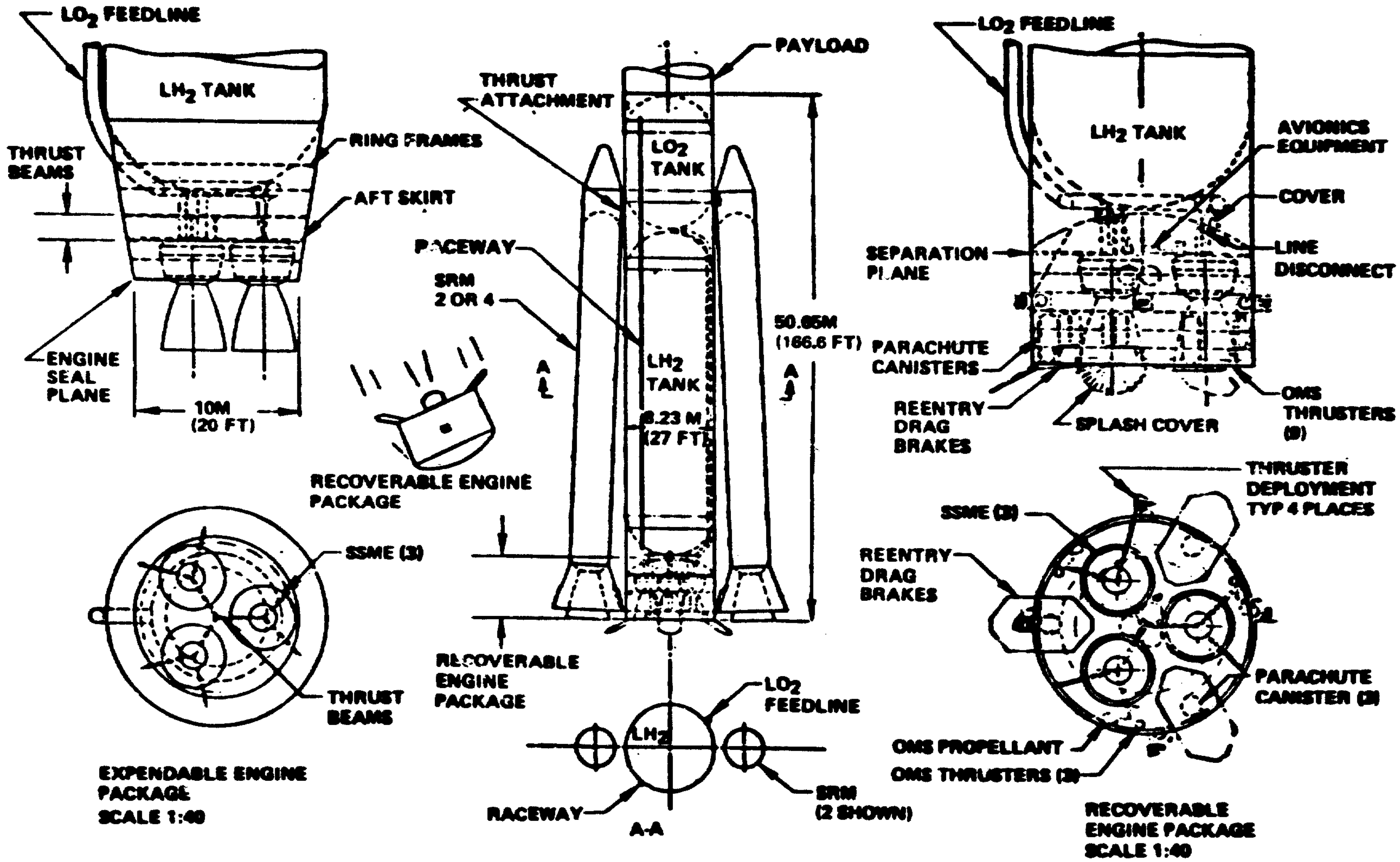

Figure 1-4 SRB/ET Heavy Lift Vehicle

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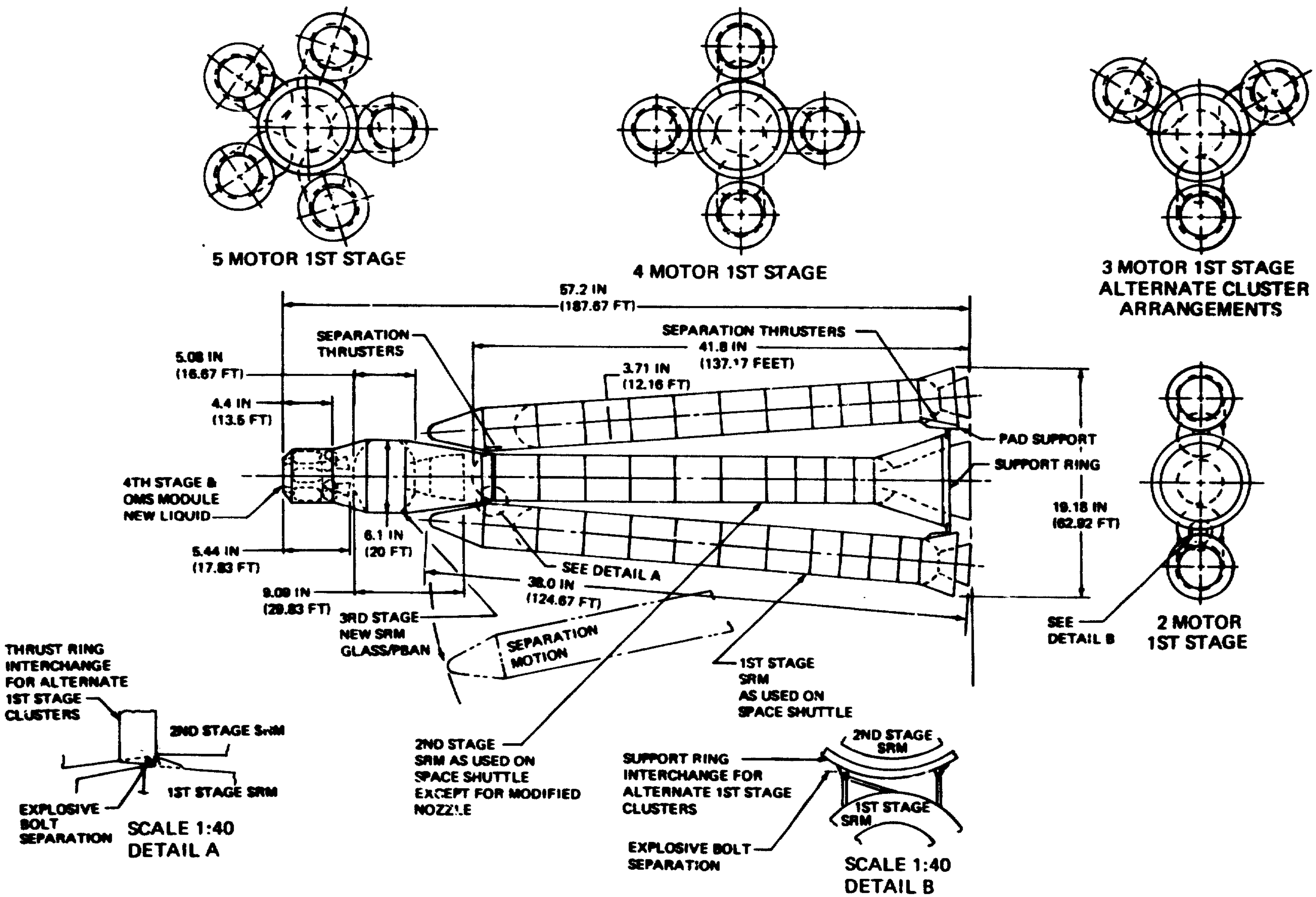

DESIGN BASED ON A CONFIGURATION
FROM JOHNSON SPACE CENTER -
JULY 9, 1976

Figure 1-5. All-SRB Heavy Lift Vehicle

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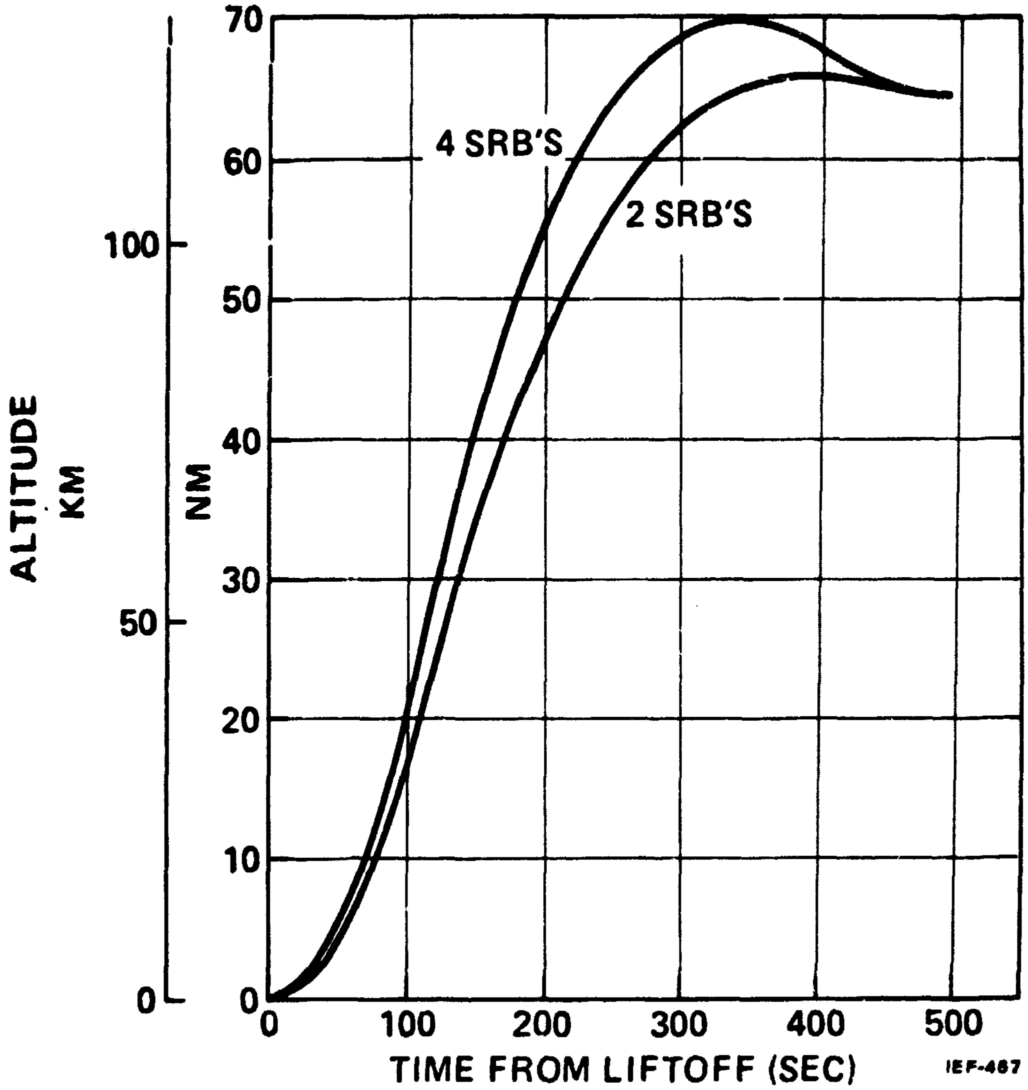

Figure 1-6 SRB/ET Heavy Lift Trajectories: Altitude Profiles

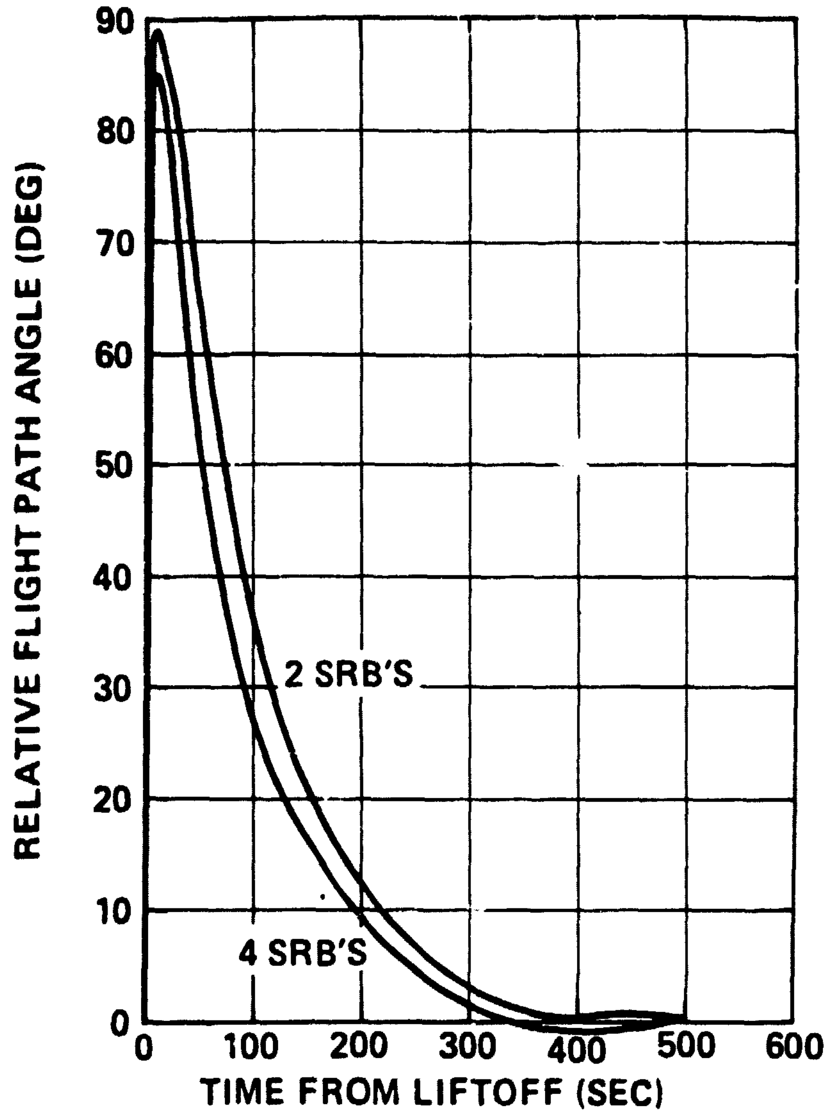

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Figure 1-7 SRE /ET Heavy Lift Trajectories: Flight Path Angle Profiles

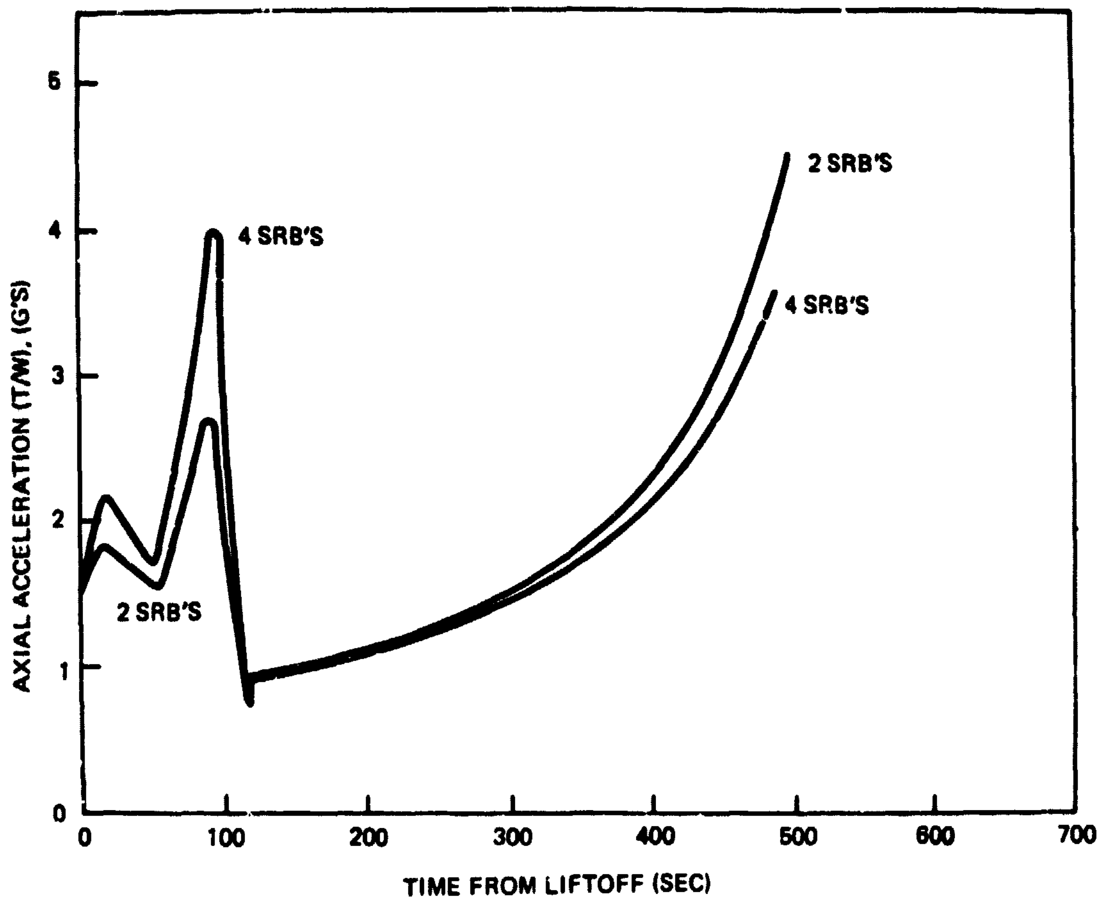

Figure 1-8 SRB/ET Heavy Lift Trajectories: Acceleration Profiles

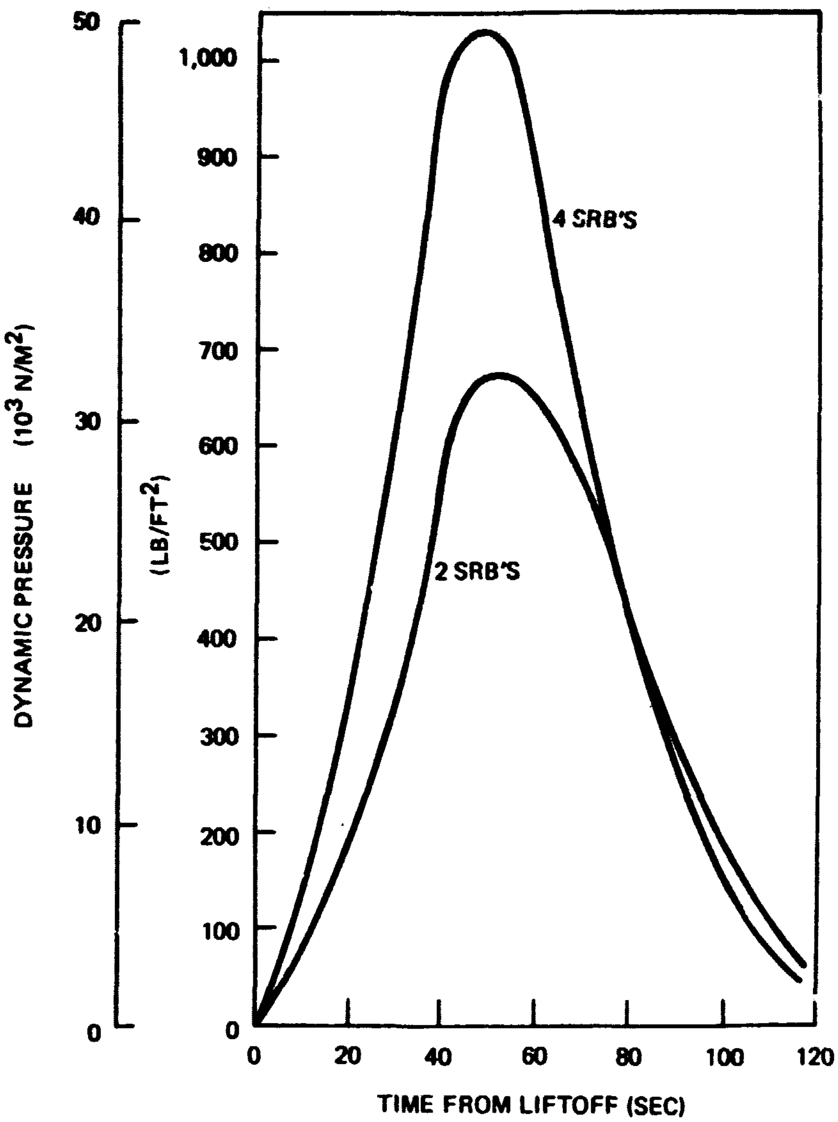

Figure 1-9 SRB/ET Heavy Lift Trajectories: Dynamic Pressure Profiles

Table 1-5. 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 (psf)	Staging velocity m/sec (ft/sec)	Weight in 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 applicable
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)

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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 1-10. 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 1-10, 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

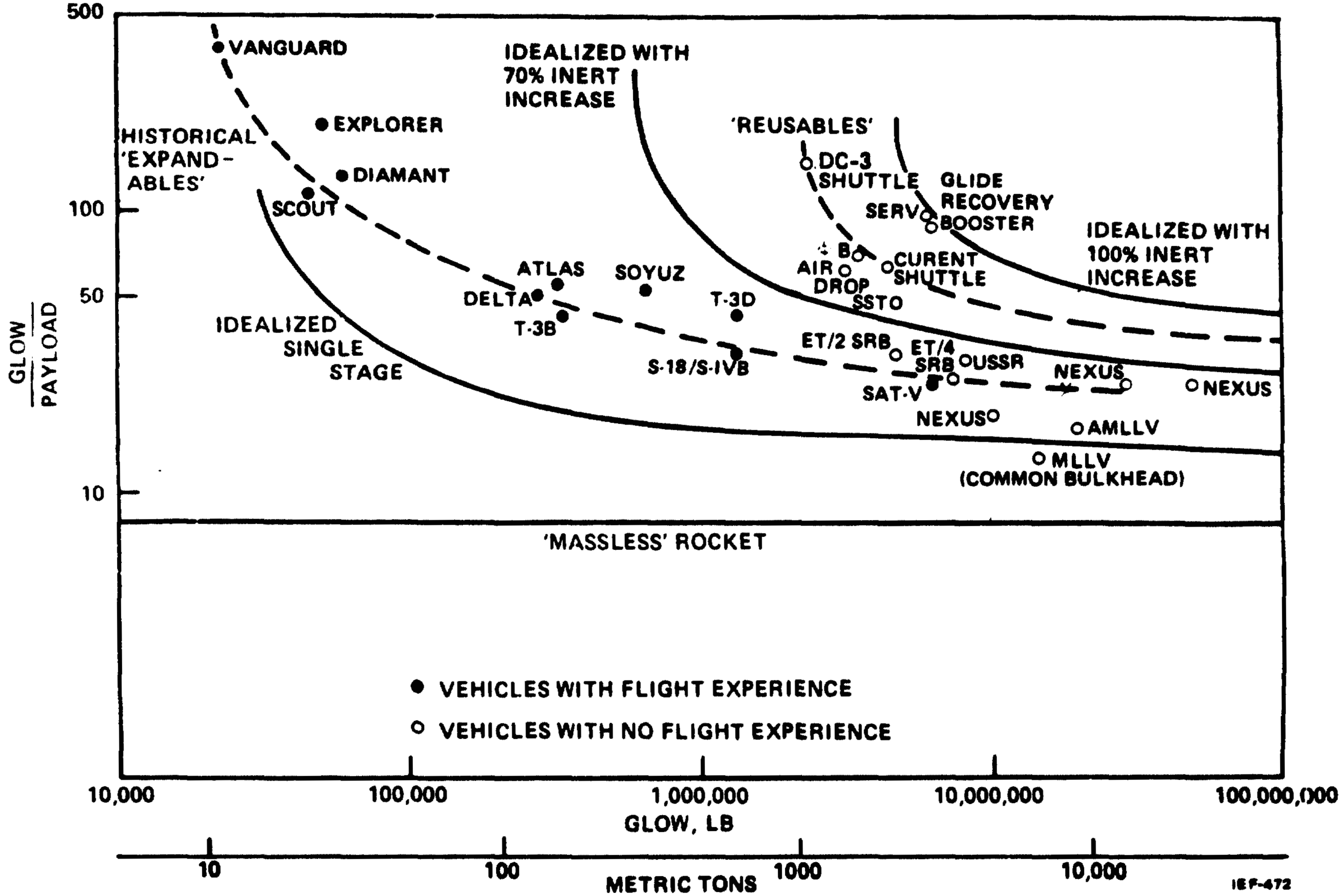

Figure 1-10 Mass/Performance Trending of Earth Launch Systems

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 225 000 kg (500,000 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 110 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 500 km (270 N.M.) assembly altitude. Thus the mass of the LCHLV is not taken to the higher orbit, greatly increasing the payload capability. Figure 1-11 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

GLOW	=	10.32×10^6 KG (22.772×10^6 LBS)
$W_{P_{MAIN}}$	=	7.10×10^6 KG (15.660×10^6 LBS)
$W_{P_{AUX}}$	=	2.04×10^6 KG (4.500×10^6 LBS)

PROPULSION

MAIN 24 Hi P_C LH_2/LOX T_{VAC} = 4.5 MN (1.0×10^6 LBS)

AUX 24 Hi P_C RP-1/LOX T_{VAC} = 2.25 MN (0.5×10^6 LBS)

T/W @ LIFTOFF = 1.30

PAYLOAD = 225,000 KG (500,000 LBS)

PAYLOAD DENSITY = 30 KG/m^3 (1.5 LB/$FEET^3$)

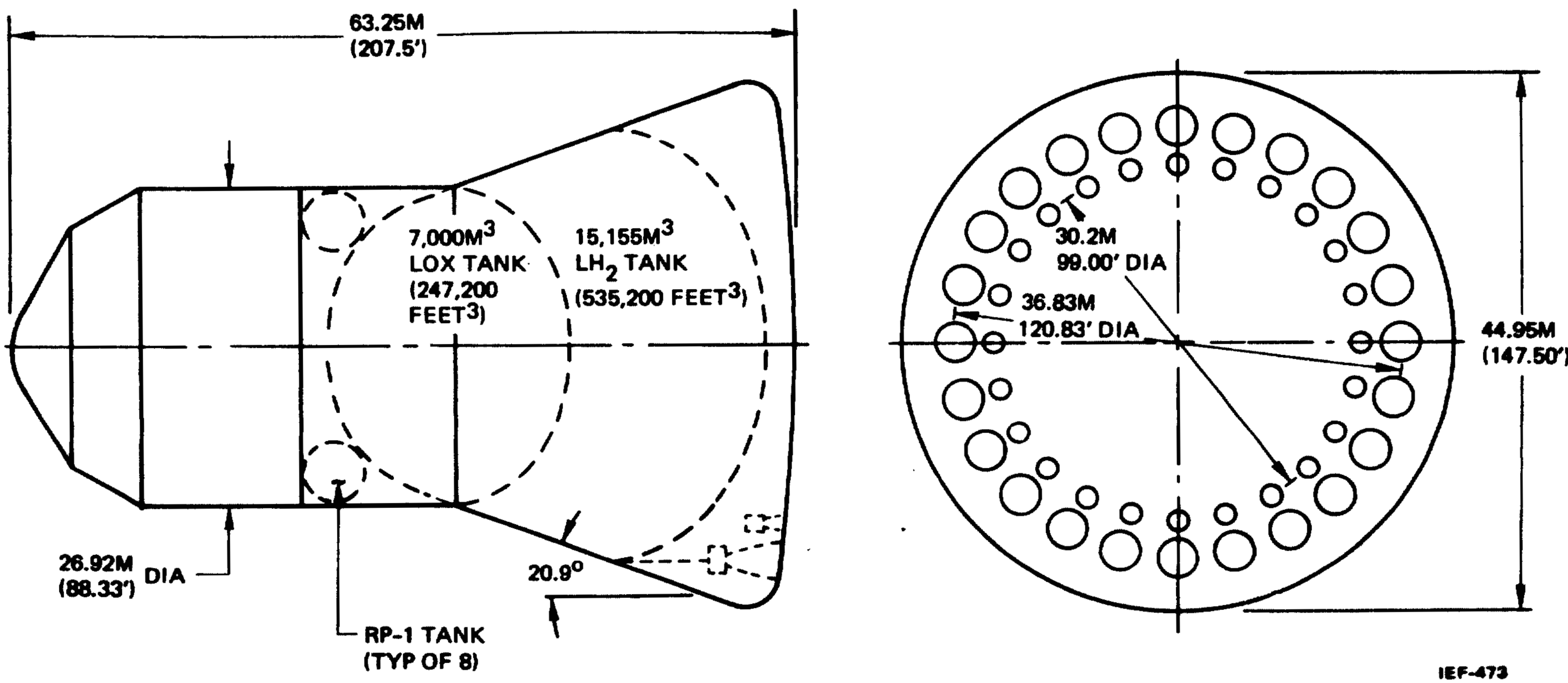

Figure 1-11. Leo Freighter Concept

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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.

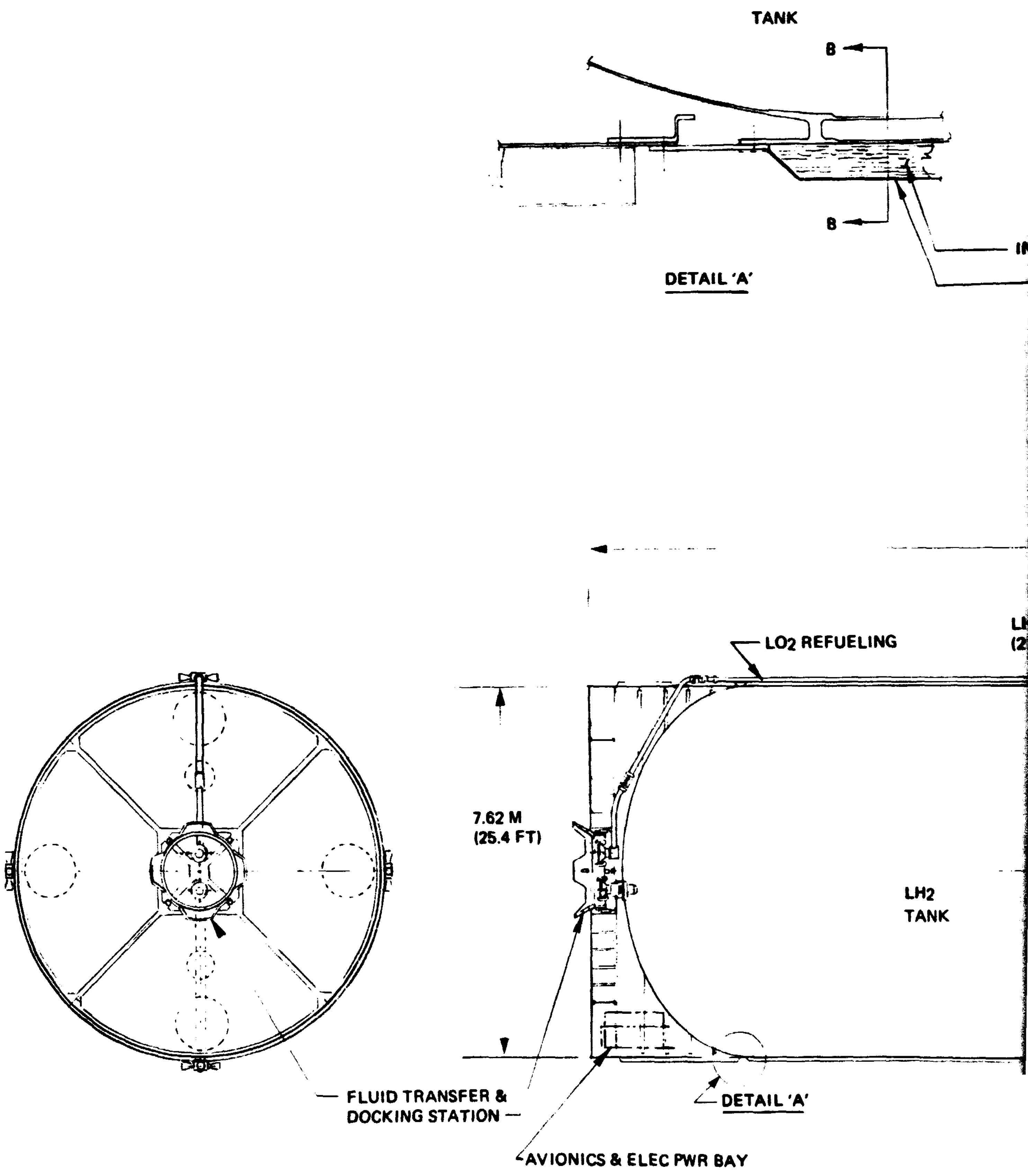
1.2.2 High Thrust Orbit Transfer Vehicles (OTV's)

1.2.2.1 Large Single-Stage LO_2/LH_2 OTV Point Design

This point design is applicable to the following missions:

- Geosynchronous Space Station (GSS)
- Independent Lunar Surface Sortie (ILSS)
- Orbiting Lunar Station (OLS)
- Lunar Surface Base (LSB)
- Manned Planetary Exploration (requires clustering and multistaging)
- Automated Planetary Exploratio: (modified mass properties due to unmanned expendable use)

The configuration inboard profile as drawn (Figure 1-12) was sized to be applicable to the GSS mission; performance analysis using the mass properties developed for the point design resulted in tailored sizes for all applicable missions as reported in Volume I. Table 1-6 presents a detailed mass properties estimate for the point design as drawn; Mass properties parameter variations about the nominal point resulted in scaling equation factors tabulated on Figure 1-13. The boiloff mass rate was estimated on the basis of cryogenic tank surface area and typical insulation performance.



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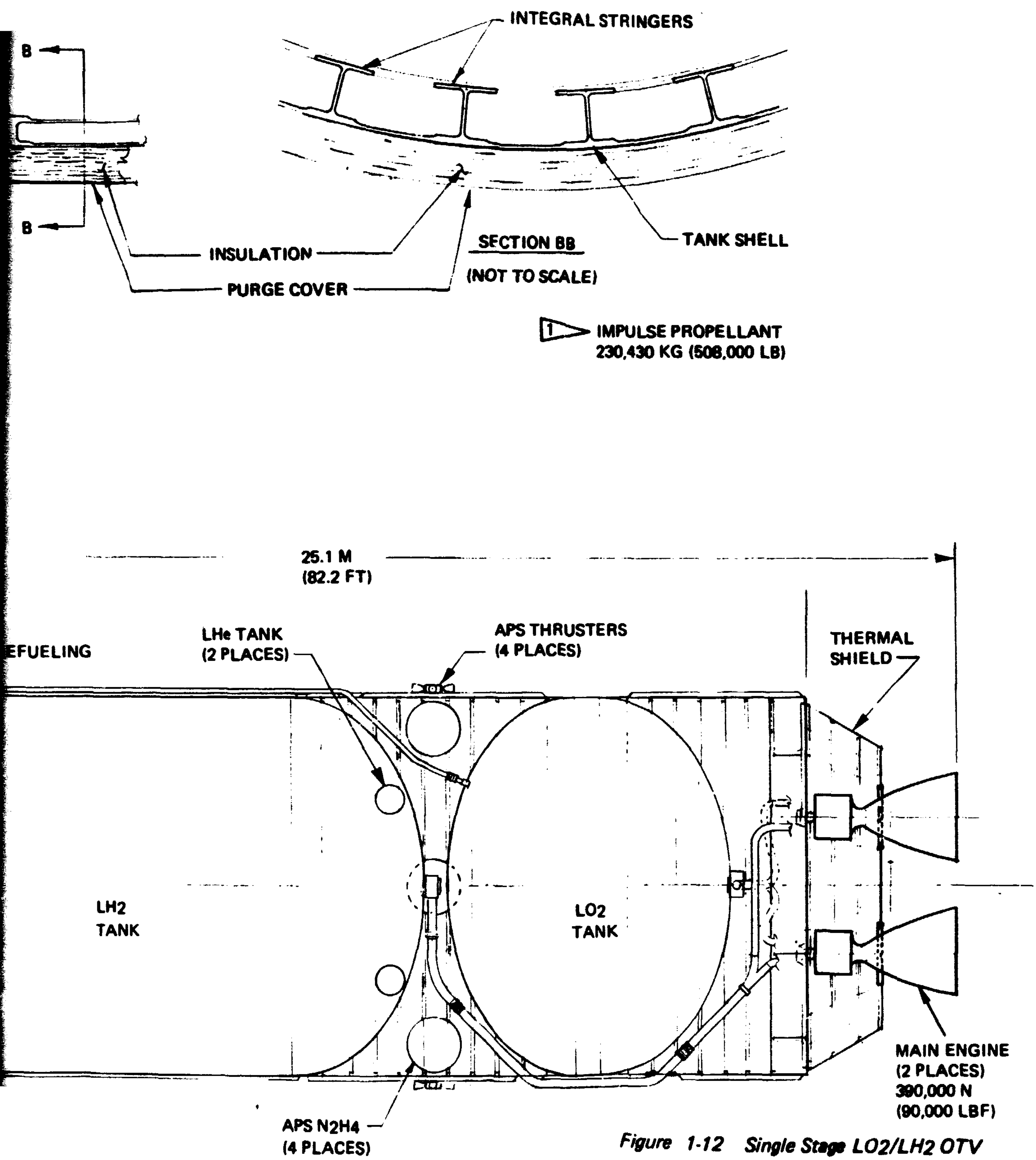

Figure 1-12 Single Stage LO2/LH2 OTV Configuration Point Design 1

Table 1-6. Single Stage LO_2/LH_2 OTV Weight Details Large Size Point Design [1]

	(LBM)	(KG)
Structure and Mechanisms	(17,120)	(7,770)
Body Shell	3,200	
Fuel Tank	7,910	
Oxidizer Tank	3,460	
Thrust Structure	650	
Stage/Payload Interface	900	
Secondary Structure	1,000	
Landing Gear	–	
Main Propulsion	(6,270)	(2,840)
Main Engines	2,860	
Accessories	740	
Pressurization & Vent	1,100	
Propellant System	1,230	
Gimbal System	340	
Auxiliary Propulsion	(1,160)	(530)
Thrusters	360	
Tanks	290	
Pressurization & Vent	160	
Propellant System	350	
Avionics	(570)	(260)
Nav Guid & Control	160	
Data Management	160	
Communications	70	
Instrumentation	140	
Rendez & Docking	40	
Electrical Power	(1,050)	(480)
Fuel Cells	200	
Batteries	160	
Tankage	110	
Processing & Control	140	
Wiring Harnesses	440	
Thermal Control	(2,450)	(1,110)
Main Tank Insulation	1,660)	
Insulation Purge	230	
Equipment Control	150	
Base Protection	250	
Paint & Sealer	160	
Weight Growth (15%)	(4,300)	(1,950)
Total Tank Dry Weight	32,920	14,940

[1] Based upon 230,400 kg (508,000 lbm) impulse propellant

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Figure 1-13 presents the mass properties buildup parametrics. These may be used to develop mass properties summaries for this vehicle type over the applicable range of main impulse propellant loading. Use of the parametrics is illustrated below.

Example Use of Mass Properties Buildup Parametrics

Problem: Develop mass properties for large single-stage OTV sized for 50-man geosynchronous station.

From discussion in Volume I, this OTV was sized at 306 000 kg (675,000 lb) with an impulse propellant loading of 281 000 kg (620,000 lb).

1. For 620,000 lb read the following values from Figure 1-13, sheet 1.

Item	kg	lb	Category
Str and Mech	8 935	19,700	1
Main Propulsion	3 265	7,200	1
Thermal Cont	1 250	2,750	1
APS propellant	3 400	7,500	3
Unusable LO_2	720	1,500	2
Unusable LH_2	885	1,950	2
Using APS propellant, read			
APS dry	725	1,600	1
APS reserves and unusables	385	850	2
From Sheet 2, assuming an average power of 2 kw, and a mission duration of 5 days, read			
Avionics	285	630	1
EPS Fixed	430	950	1
EPS Variable dry	545	1,200	1
Reactant	1 090	2,400	2
Boiloff @ 325 lb/day	740	1,625	3
Assuming six burns, and 900 KN (200,000 lb) thrust,			
Start/stop losses @ 175 lb per burn	475	1,050	3

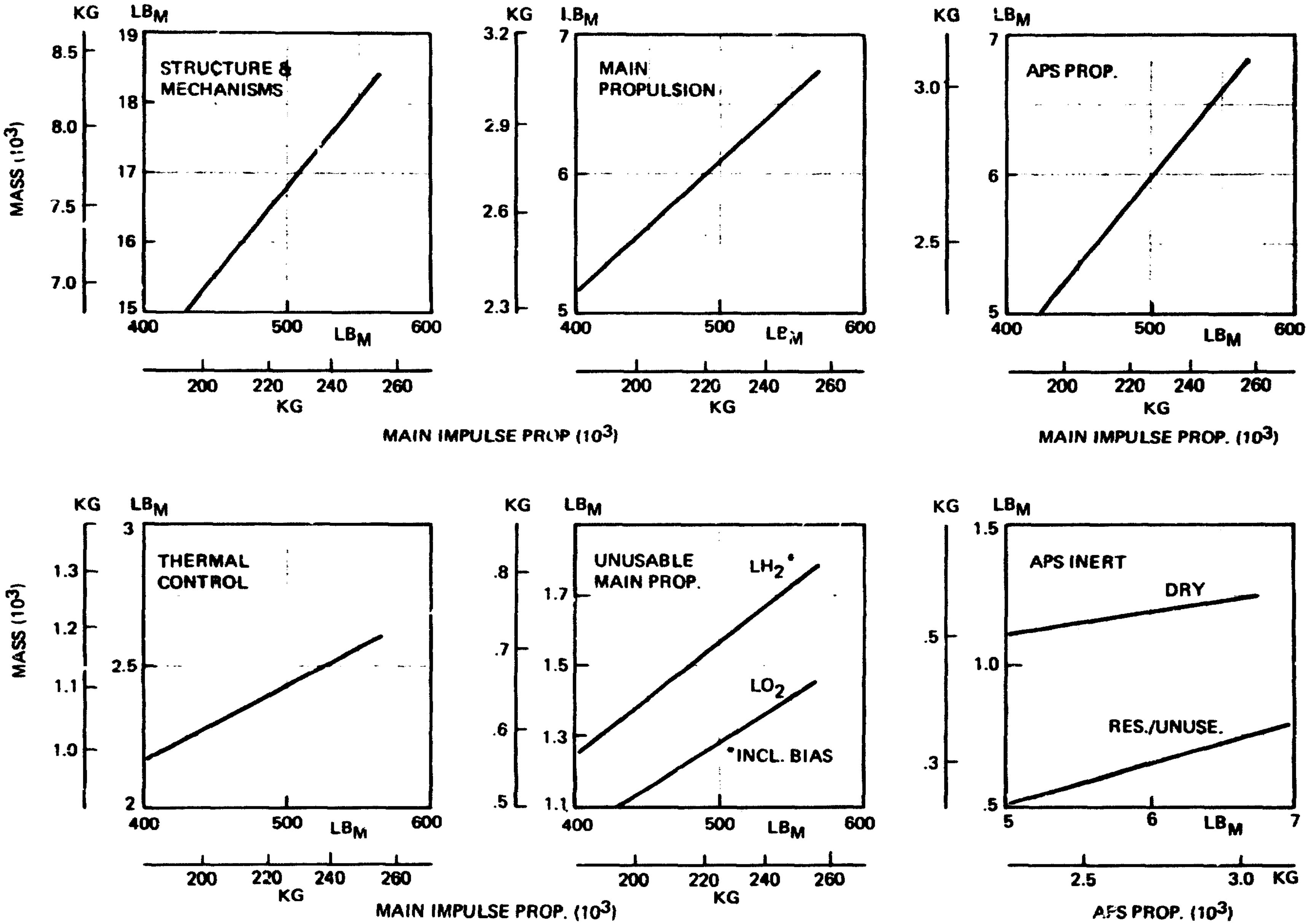
The mass properties statement may now be constructed, observing categories. Note the addition of contingency to dry mass.

Item	kg	lb
Category 1: Dry mass		
Structures and Mech	8 935	19,700
Main Propulsion	3 265	7,200
Thermal Control	1 250	2,750
Auxiliary Propulsion	725	1,600
Avionics	285	630
EPS	975	2,150
Contingency (15%)	2 315	5,105
Total Dry	17 750	39,135
Category 2: Unusable Fluids and EPS reactants		
LO_2	720	1,580
LH_2	885	1,950
APS	385	850
EPS reactant	1 090	2,400
Total Burnout	20 830	45,915
Category 3: Inflight Expendable		
Boiloff	740	1,625
Start/stop losses	475	1,050
APS impulse propellant	3 400	7,500
Main Impulse Propellant	281 000	620,000
Total Start burn	306 445	676,090

The result checks the value read from the stage-level curve within 0.2%. The length of the vehicle may be estimated from Figure 1-13 sheet 3:

	m	ft
Tank length as drawn		
(508,000 lb impulse propellant)	18.14	59.5
Tank length @ 620,000 lb	21.2	69.5

The delta length is 3.05 m (10 ft). The vehicle as drawn was 25.05 m (82.2 ft) in length. The resized vehicle is therefore approximately 28.1 m (92.2 ft) in length, assuming outside diameter is unchanged.



SCALING EQUATION FACTORS: A = 5 650 KG (12,456 LB) B = 0.0460 C = 0 D = 0.1725

Figure 1-13. Subsystem Parametrics — LO_2/LH_2 Single Stage OTV (Sheet 1)

IEF-2

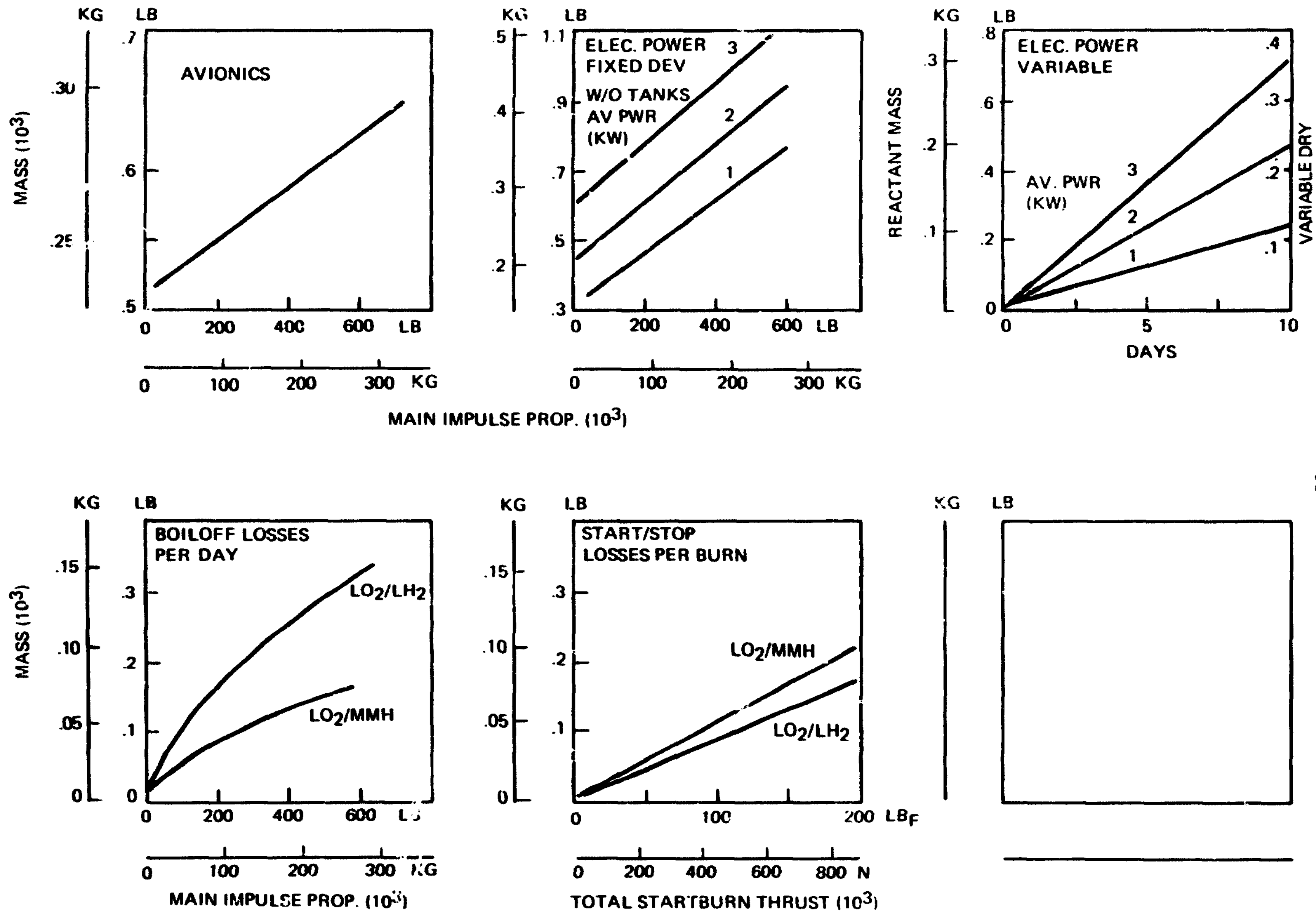


Figure 1-13. Subsystems Parametrics – General For All OTV's (Sheet 2)

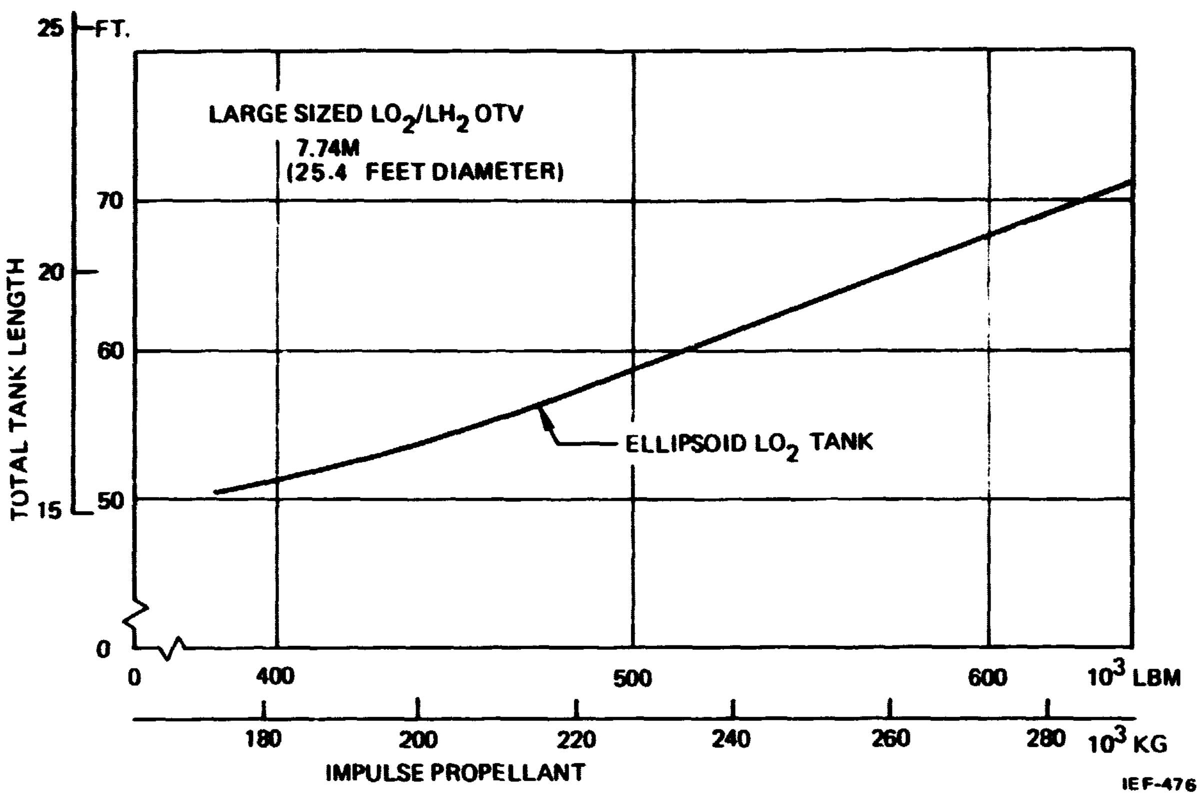


Figure 1-13. Tank Length Versus Impulse Propellant Mass (Sheet 3)

1.2.2.2 Intermediate Single-Stage LO_2/LH_2 OTV Point Design

This point design is applicable to the following missions:

- Geosynchronous Satellite Maintenance Sortie (GSMS)
- Automated Planetary Exploration (modified mass properties due to unmanned expendable use)

The configuration inboard profile as drawn (Figure 1-14) was sized to be applicable to the GSMS mission; performance analysis using the mass properties developed for the point design resulted in tailored sizes for all applicable missions as reported in Volume I. Table 1-7 presents a detailed mass properties estimate for the point design as drawn; Mass properties parameter variations about the nominal point resulted in scaling equation factors shown on Figure 1-15.

Figure 1-15 presents the mass properties buildup parametrics. These may be used to develop mass properties summaries for this vehicle type over the applicable range of main impulse propellant loading.

1.2.2.3 Small Single-Stage LO_2/LH_2 OTV Point Design

This point design is applicable to the following missions:

- Automated Planetary Exploration (modified mass properties due to unmanned expendable use)
- Nuclear Waste Disposal

The configuration inboard profile as drawn (Figure 1-16) was sized to be applicable to the Ganymede Lander mission; performance analysis using the mass properties developed for the point design resulted in tailored sizes for all applicable missions as reported in Volume I. Table 1-8 presents a detailed mass properties estimate for the point design as drawn; Mass properties parameter variations about the nominal point resulted in scaling equation factors shown on Figure 1-17.

Figure 1-17 presents the mass properties buildup parametrics. These may be used to develop mass properties summaries for this vehicle type over the applicable range of main impulse propellant loading.

1.2.2.4 1½ Stage LO_2/LH_2 OTV Point Design

This point design is applicable to the following missions:

- Geosynchronous Space Station (GSS)
- Independent Lunar Surface Sortie (ILSS)

- Orbiting Lunar Station (OLS)
- Lunar Surface Base (LSB)

The configuration inboard profile as drawn (Figure 1-18) was sized to be applicable to the GSS mission; performance analysis using the mass properties developed for the point design resulted in tailored sizes for all applicable missions as reported in Volume I. Table 1-9 presents a detailed mass properties estimate for the point design as drawn; Mass properties parameter variations about the nominal point resulted in scaling equation factors shown on Figure 1-19.

Figure 1-19 presents the mass properties buildup parametrics. These may be used to develop mass properties summaries for this vehicle type over the applicable range of main impulse propellant loading.

1.2.2.5 Large Common-Stage LO_2/LH_2 OTV Point Design

This point design is applicable to the following missions:

- Geosynchronous Space Station (GSS)
- Independent Lunar Surface Sortie (ILSS)
- Orbiting Lunar Station (OLS)
- Lunar Surface Base (LSB)

The configuration inboard profile (Figure 1-20) as drawn was sized to be applicable to the GSS mission; performance analysis using the mass properties developed for the point design resulted in tailored sizes for all applicable missions as reported in Volume I. Table 1-10 presents a detailed mass properties estimate for the point design as drawn; Mass properties parameter variations about the nominal point resulted in scaling equation factors shown on Figure 1-21. The boiloff mass rate was estimated on the basis of cryogenic tank surface area and typical insulation performance.

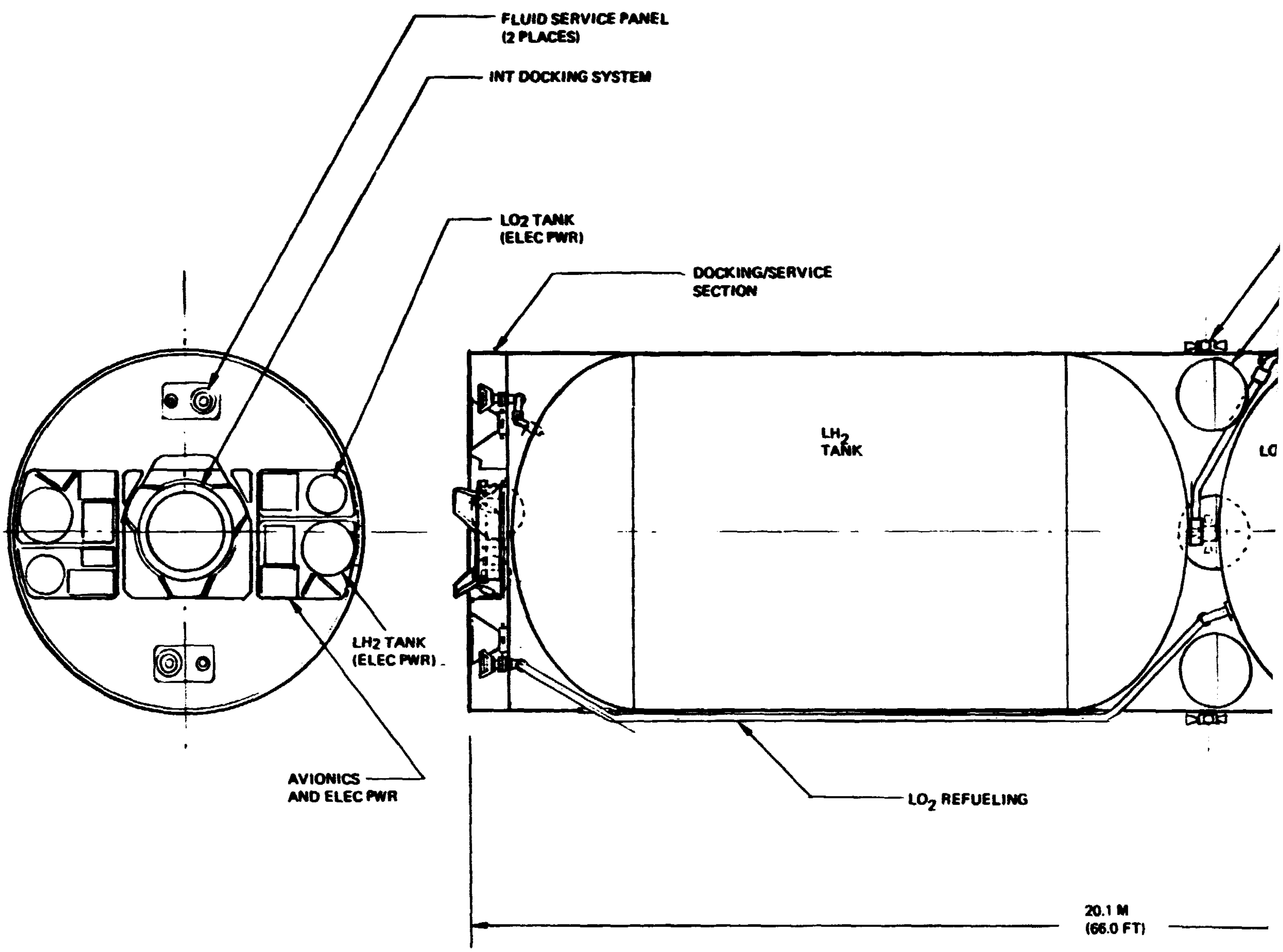
Figure 1-21 presents the mass properties buildup parametrics. These may be used to develop mass properties summaries for this vehicle type over the applicable range of main impulse propellant loading.

1.2.2.6 Intermediate Common-Stage LO_2/LH_2 OTV Point Design

This point design is applicable to the following missions:

- Geosynchronous Satellite Maintenance Sortie (GSMS)
- Nuclear Waste Disposal (NWD)

The configuration inboard profile as drawn (Figure 1-22) was sized to be applicable to the GSMS mission; performance analysis using the mass properties developed for the point design resulted in



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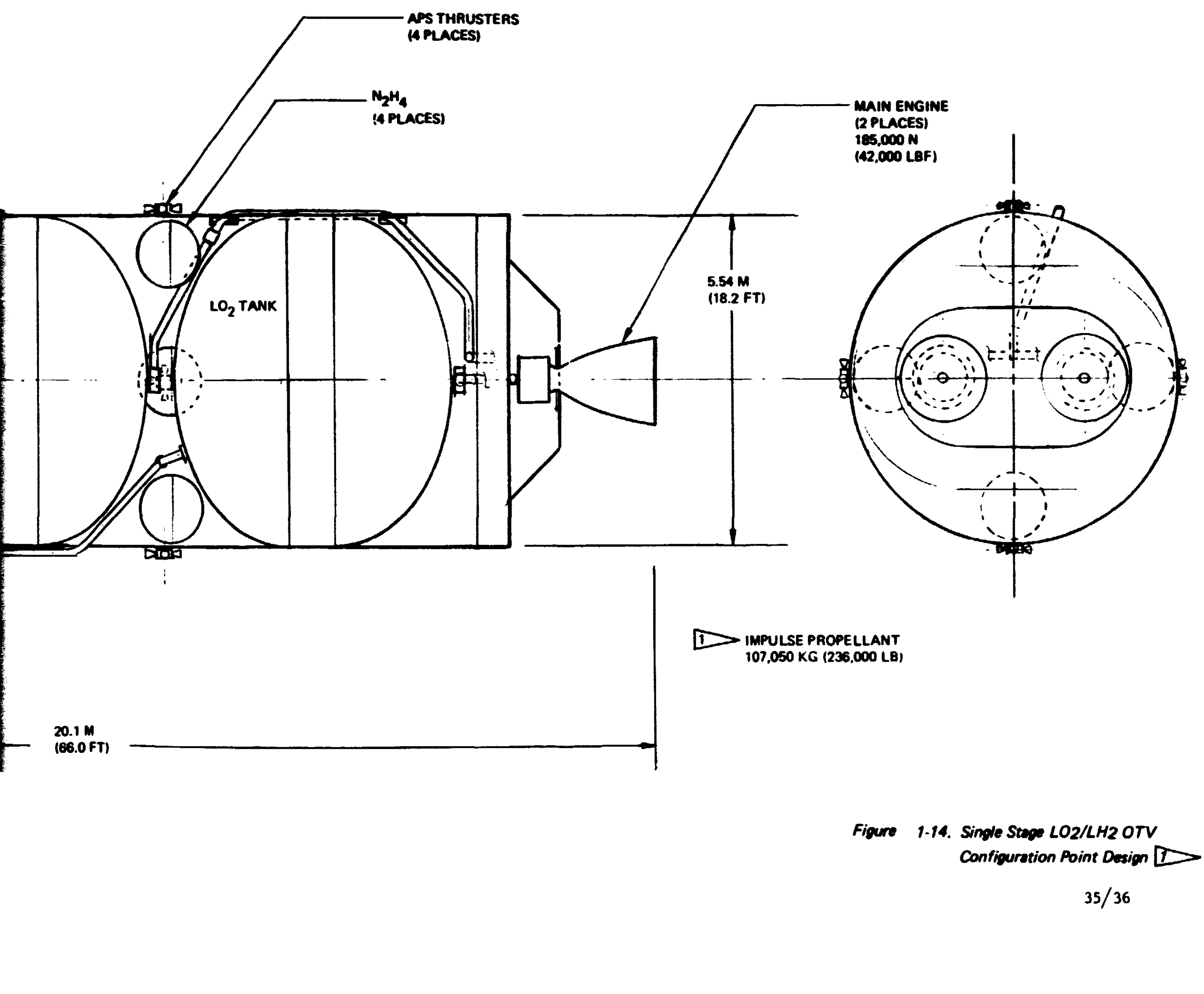

Figure 1-14. Single Stage LO2/LH2 OTV Configuration Point Design 1

Table 1-7. Single Stage LO_2/LH_2 OTV Weight Details Medium Size Point Design [1]

	(LBM)	(KG)
Structure and Mechanisms	(9,390)	(4,260)
Body Shell	1,940	
Fuel Tank	3,670	
Oxidizer Tank	1,780	
Thrust Structure	350	
Stage/Payload Interface	1,000	
Secondary Structure	650	
Landing Gear	–	
Main Propulsion	(3,080)	(1,400)
Main Engines	1,250	
Accessories	320	
Pressurization & Vent	660	
Propellant System	690	
Gimbal System	160	
Auxiliary Propulsion	(800)	(360)
Thrusters	240	
Tanks	360	
Pressurization & Vent	50	
Propellant System	150	
Avionics	(500)	(230)
Nav Guid & Control	160	
Data Management	160	
Communications	70	
Instrumentation	70	
Rendez & Docking	40	
Electrical Power	(800)	(360)
Fuel Cells	160	
Batteries	80	
Tankage	160	
Processing & Control	70	
Wiring Harnesses	330	
Thermal Control	(1,620)	(740)
Main Tank Insulation	1,040	
Insulation Purge	120	
Equipment Control	150	
Base Protection	200	
Paint & Sealer	110	
Weight Growth (15%)	(2,430)	(1,100)
Total Tank Dry Weight	18,620	8,450

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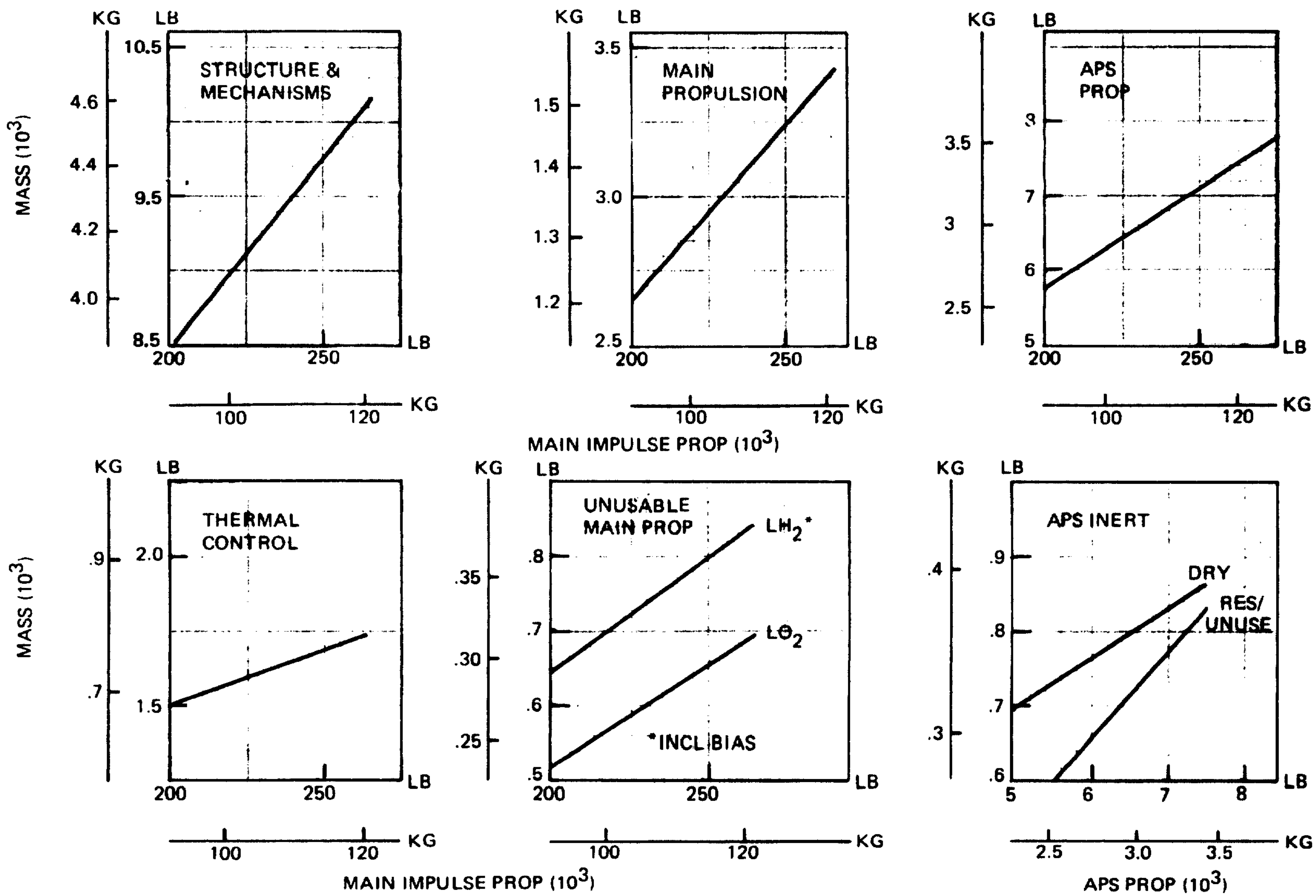
[1] Based upon 107,000 kg (236,000 lbm) impulse propellant



SCALING PARAMETERS A = 3,340 KG (7,370 LB) B = 0.0535 C = 0 D = 1725

IEF-80

Figure 1-15. Single Stage LO_2/LH_2 OTV Subsystem Parametrics (Sheet 1)

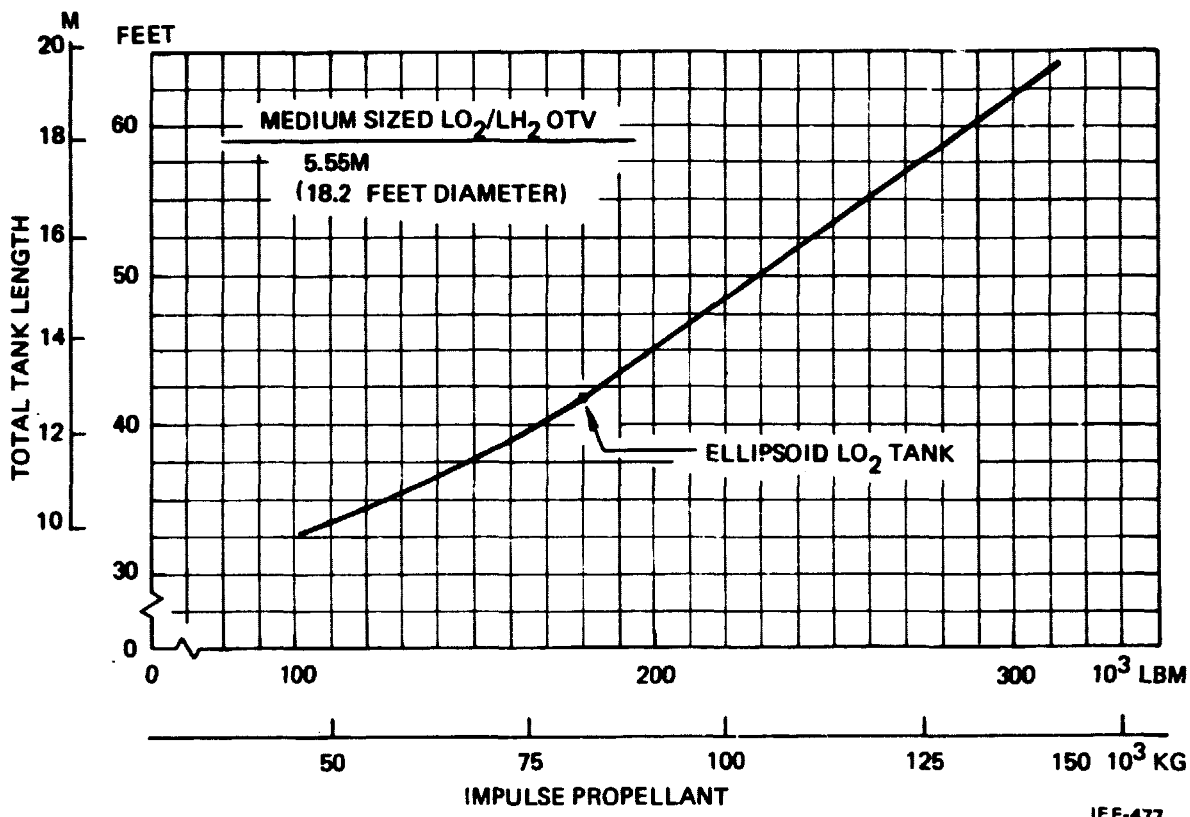


Figure 1-15. Sheet 2

tailored sizes for applicable missions as reported in Volume I. Table 1-11 presents a detailed mass properties estimate for the point design as drawn; Mass properties parameter variations about the nominal point resulted in scaling equation factors shown on Figure 1-23.

Figure 1-23 presents the mass properties buildup parametrics. These may be used to develop mass properties summaries for this vehicle type over the applicable range of main impulse propellant loading.

1.2.2.7 Large Common-Stage LO_2/MMH OTV Point Design

This point design is applicable to the following missions:

- Geosynchronous Space Station (GSS)
- Independent Lunar Surface Sortie (ILSS)
- Orbiting Lunar Station (OLS)
- Lunar Surface Base (LSB)

The configuration inboard profile as drawn (Figure 1-24) was sized to be applicable to the GSS mission; performance analysis using the mass properties developed for the point design resulted in tailored sizes for all applicable missions as reported in Volume I. Table 1-12 presents a detailed mass properties estimate for the point design as drawn; Mass properties parameter variations about the nominal point resulted in scaling equation factors shown on Figure 1-25. The boiloff mass rate was estimated on the basis of cryogenic tank surface area and typical insulation performance.

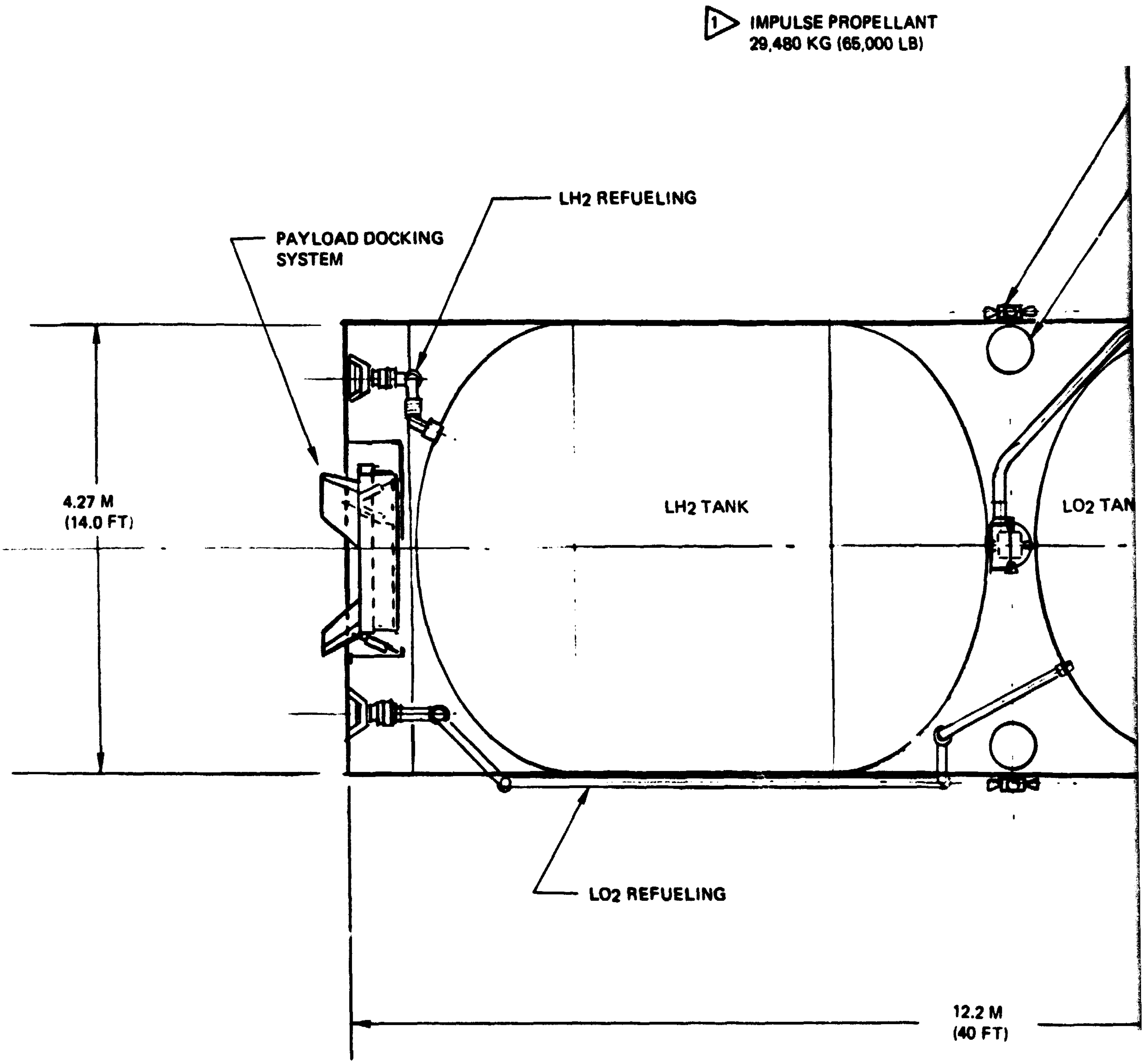
Figure 1-25 presents the mass properties buildup parametrics. These may be used to develop mass properties summaries for this vehicle type over the applicable range of main impulse propellant loading.

1.2.2.8 Intermediate Common-Stage LO_2/MMH

This point design is applicable to the geosynchronous satellite maintenance sortie (GSMS) mission.

The configuration inboard profile as drawn (Figure 1-26) was sized to be applicable to the GSMS mission; performance analysis using the mass properties developed for the point design resulted in tailored sizing for this mission as reported in Volume I. Table 1-13 presents a detailed mass properties estimate for the point design as drawn; Mass properties parameter variations about the nominal point resulted in scaling equation factors shown on Figure 1-27. The boiloff mass rate was estimated on the basis of cryogenic tank surface area and typical insulation performance.

Figure 1-27 presents the mass properties buildup parametrics. These may be used to develop mass properties summaries for this vehicle type over the applicable range of main impulse propellant loading.



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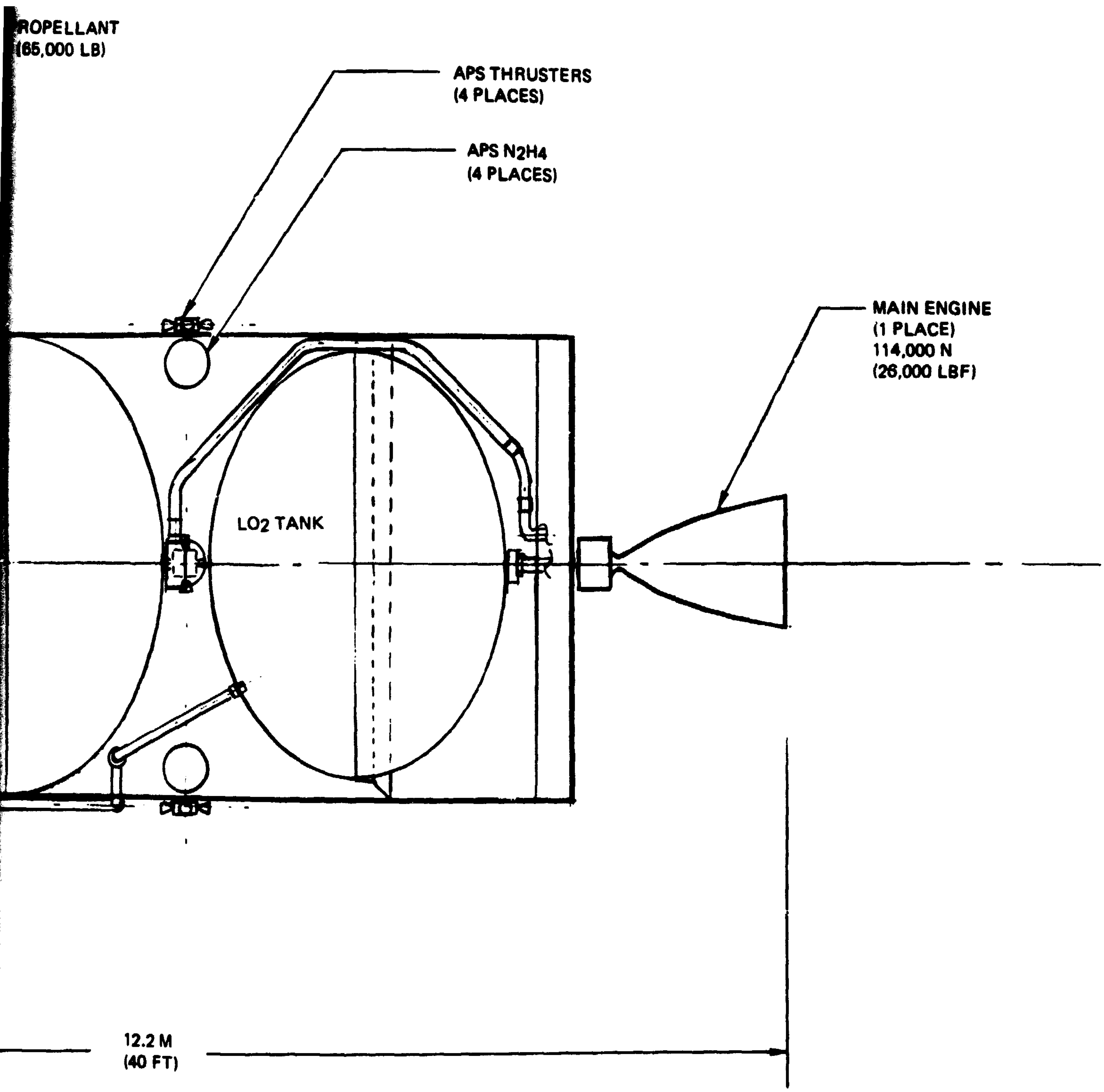

Figure 1-16 Single Stage LO_2/LH_2 OTV Configuration Point Design 1

Table 1-8. Single Stage LO_2/LH_2 OTV Weight Details Small Size Single Burn Point Design [1]

	(LBM)	(KG)
Structure and Mechanisms	(2,740)	(1,240)
Body Shell	920	
Fuel Tank	680	
Oxidizer Tank	420	
Thrust Structure	130	
Stage/Payload Interface	370	
Secondary Structure	220	
Landing Gear	–	
Main Propulsion	(1,200)	(550)
Main Engines	400	
Accessories	140	
Pressurization & Vent	250	
Propellant System	330	
Gimbal System	80	
Auxiliary Propulsion	(220)	(100)
Thrusters	80	
Tanks	10	
Pressurization & Vent	50	
Propellant System	80	
Avionics	(400)	(180)
Nav Guid & Control	160	
Data Management	130	
Communications	60	
Instrumentation	50	
Rendez & Docking	–	
Electrical Power	(340)	(150)
Fuel Cells	80	
Batteries	50	
Tankage	10	
Processing & Control	40	
Wiring Harnesses	150	
Thermal Control	(330)	(150)
Main Tank Insulation	170	
Insulation Purge	–	
Equipment Control	50	
Base Protection	50	
Paint & Sealer	60	
Weight Growth (15%)	(780)	(350)
Total Tank Dry Weight	6,010	2,720

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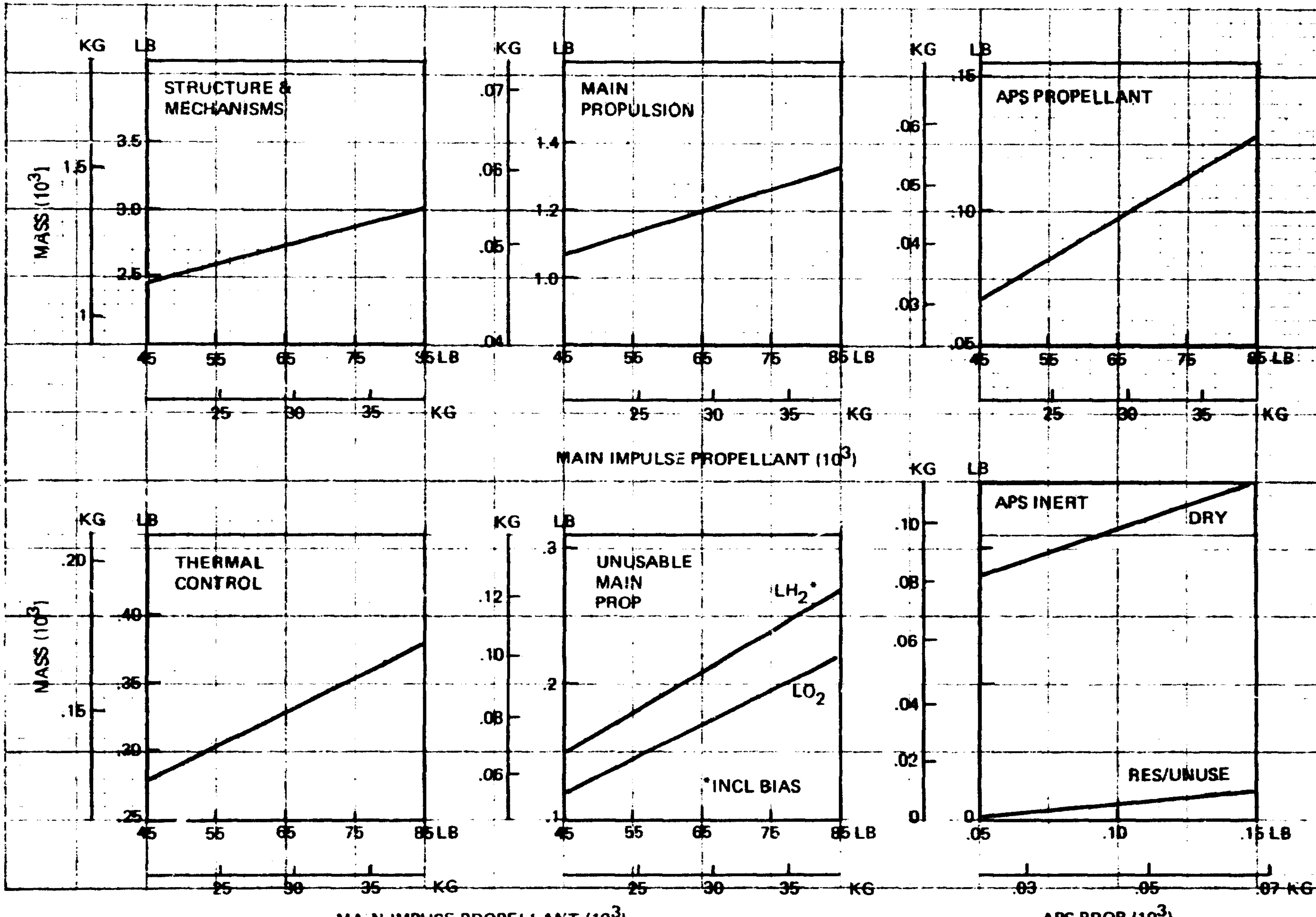
[1] Based upon 29,500 kb (65,000 lbm) impulse propellant



SCALING PARAMETERS A = 1,950 KG (4,300 LB); b = .0320; c = 0; d = .1725

Figure 1-17. Small Single Stage LO2/LH2 OTV Subsystem Parametrics (Sheet 2)

IEF-8

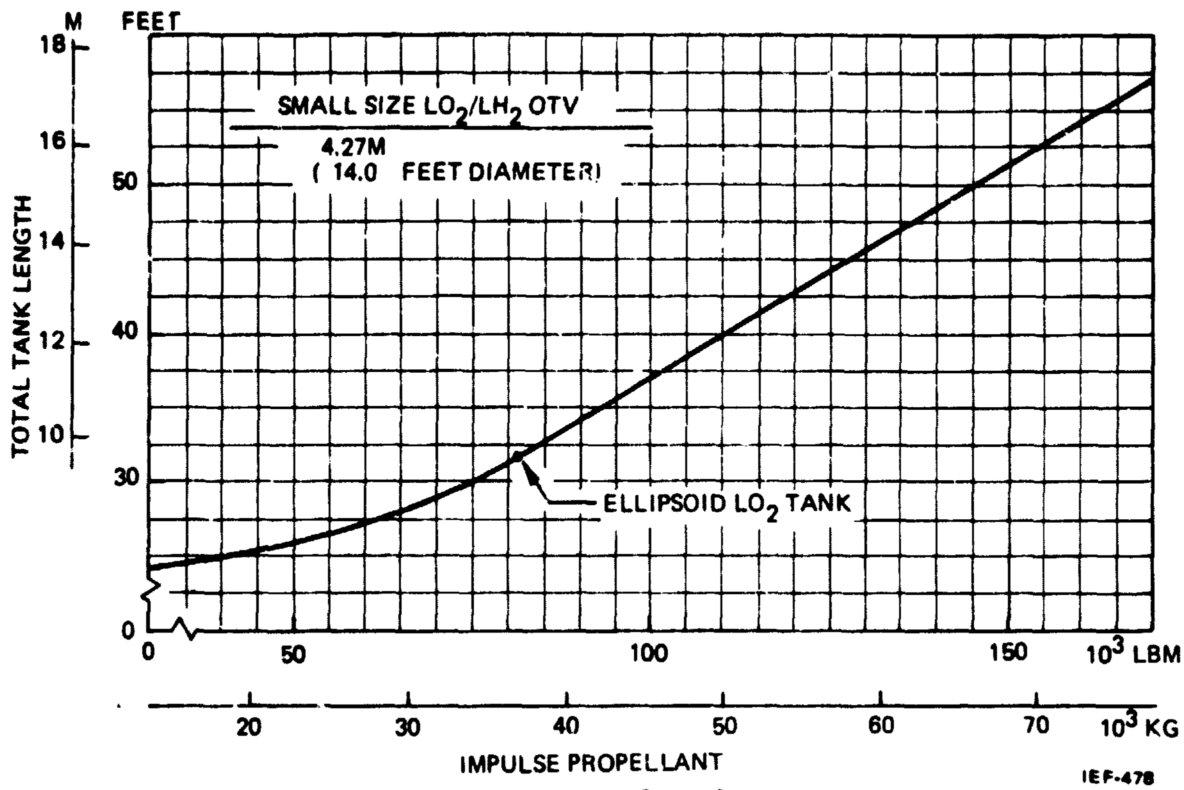
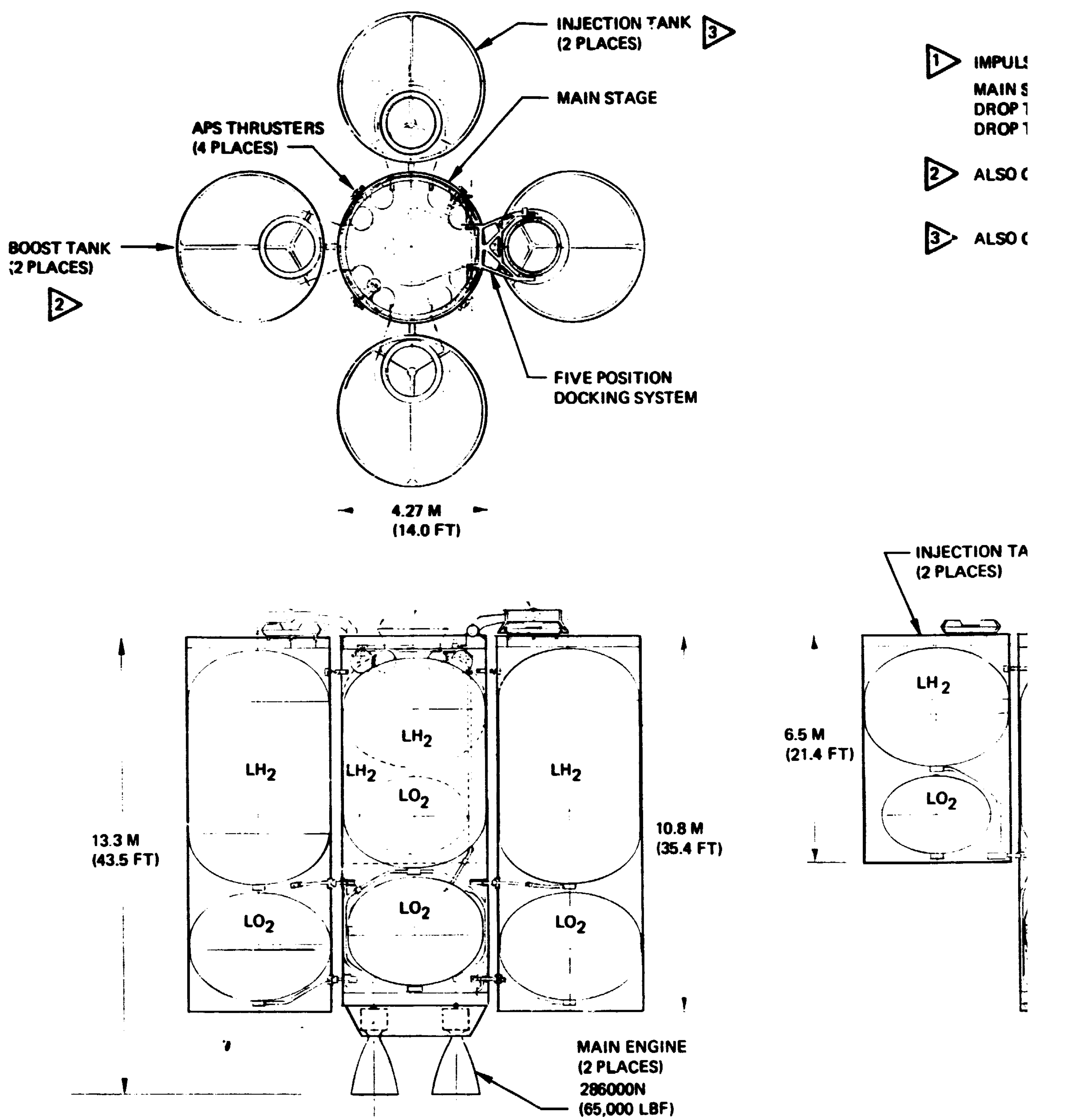


Figure 1-17. Sheet 2



INJECTION TANK (2 PLACES)
3
MAIN STAGE
APS THRUSTERS (4 PLACES)
BOOST TANK (2 PLACES)
2
FIVE POSITION DOCKING SYSTEM
4.27 M (14.0 FT)
1
2
3
LH2
LO2
13.3 M (43.5 FT)
10.8 M (35.4 FT)
MAIN ENGINE (2 PLACES) 286000N (65,000 LBF)
6.5 M (21.4 FT)
(2 PLACES)

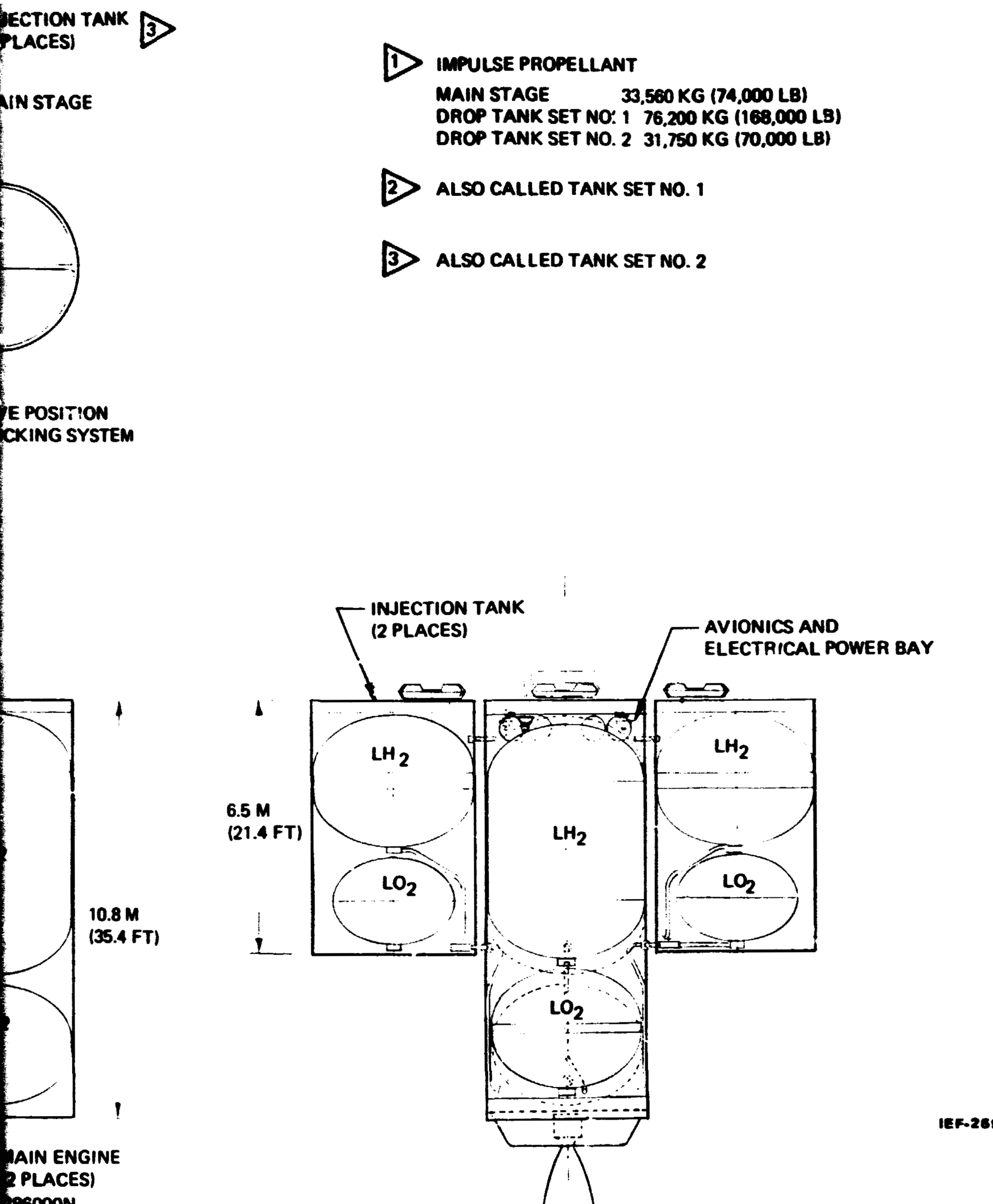


Figure 1-18 1-1/2 Stage LO2/LH2 OTV Configuration Point Design 1

Table 1-9. 1-1/2 Stage LO_2/LH_2 OTV Weight Details Point Design—Main Stage (Sheet 1) [1]

	(LBM)	(KG)
Structure and Mechanisms	(4,090)	(1,860)
Body Shell	970	
Fuel Tank	1,040	
Oxidizer Tank	480	
Thrust Structure	350	
Stage/Payload Interface	660	
Secondary Structure	590	
Landing Gear	–	
Main Propulsion	(4,110)	(1,870)
Main Engines	2,000	
Accessories	480	
Pressurization & Vent	540	
Propellant System	850	
Gimbal System	240	
Auxiliary Propulsion	(850)	(380)
Thrusters	360	
Tanks	210	
Pressurization & Vent	110	
Propellant System	170	
Avionics	(600)	(270)
Nav Guid & Control	160	
Data Management	160	
Communications	70	
Instrumentation	170	
Rendez & Docking	40	
Electrical Power	(770)	(350)
Fuel Cells	200	
Batteries	120	
Tankage	110	
Processing & Control	140	
Wiring Harnesses	200	
Thermal Control	(1,220)	(550)
Main Tank Insulation	750	
Insulation Purge	130	
Equipment Control	150	
Base Protection	110	
Paint & Sealer	80	
Weight Growth (15%)	(1,750)	(790)
Total Tank Dry Weight	13,390	6,070

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[1] Based upon 33,600 kg (74,000 lbm) impulse propellant in main stage

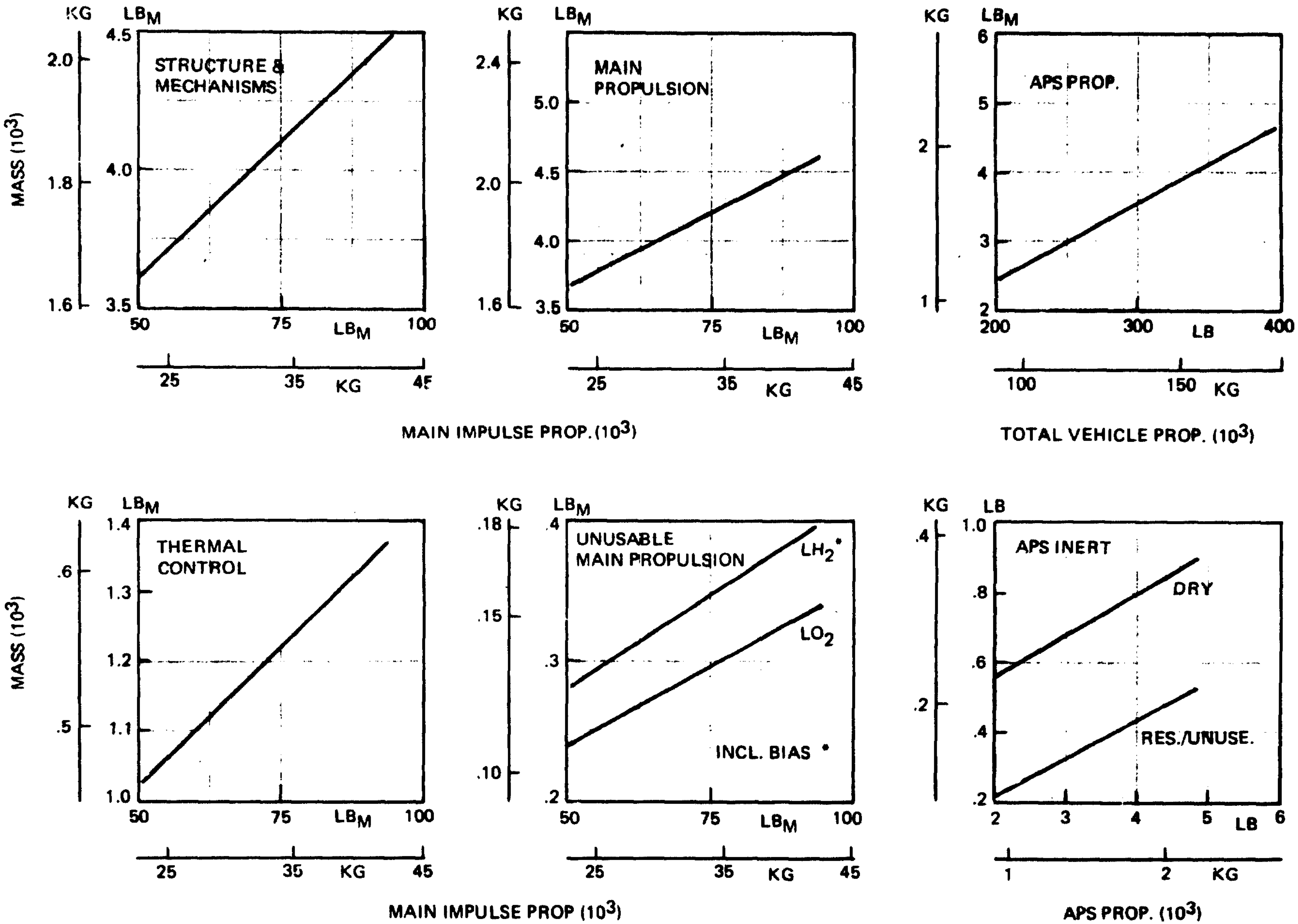
Table 1-9. 1½ Stage LO_2/LH_2 OTV Weight Details Point Design – Drop Tanks (Sheet 2)

	Set No. 1 [1]		Set No. 2 [2]	
	(LBM)	(KG)	(LBM)	(KG)
Structure and Mechanisms	(2,730)	(1,240)	(1,900)	(860)
Body Shell	630		460	
Fuel Tank	860		410	
Oxidizer Tank	360		220	
Thrust Structure	–		–	
Stage/Payload Interface	850		660	
Secondary Structure	30		150	
Landing Gear	–		–	
Main Propulsion	(320)	(140)	(240)	(110)
Main Engines	–		–	
Accessories	–		–	
Pressurization & Vent	100		70	
Propellant System	220		170	
Gimbal System	–		–	
Auxiliary Propulsion	–		–	
Thrusters	–		–	
Tanks	–		–	
Pressurization & Vent	–		–	
Propellant System	–		–	
Avionics	(60)	(30)	(50)	(20)
Nav Guid & Control	–		–	
Data Management	–		–	
Communications	–		–	
Instrumentation	60		50	
Rendez & Docking	–		–	
Electrical Power	(40)	(20)	(30)	(10)
Fuel Cells	–		–	
Batteries	–		–	
Tankage	–		–	
Processing & Control	–		–	
Wiring Harnesses	40		30	
Thermal Control	(360)	(160)	(300)	(140)
Main Tank Insulation	330		280	
Insulation Purge	–		–	
Equipment Control	–		–	
Base Protection	–		–	
Paint & Sealer	30		20	
Weight Growth (15%)	(520)	(240)	(370)	(170)
Total Tank Dry Weight	4,030	1,830	2,890	1,310

IEF-221

[1] Values are for one tank containing 38,100kg (84,000 lbm) impulse prop. Two tanks required

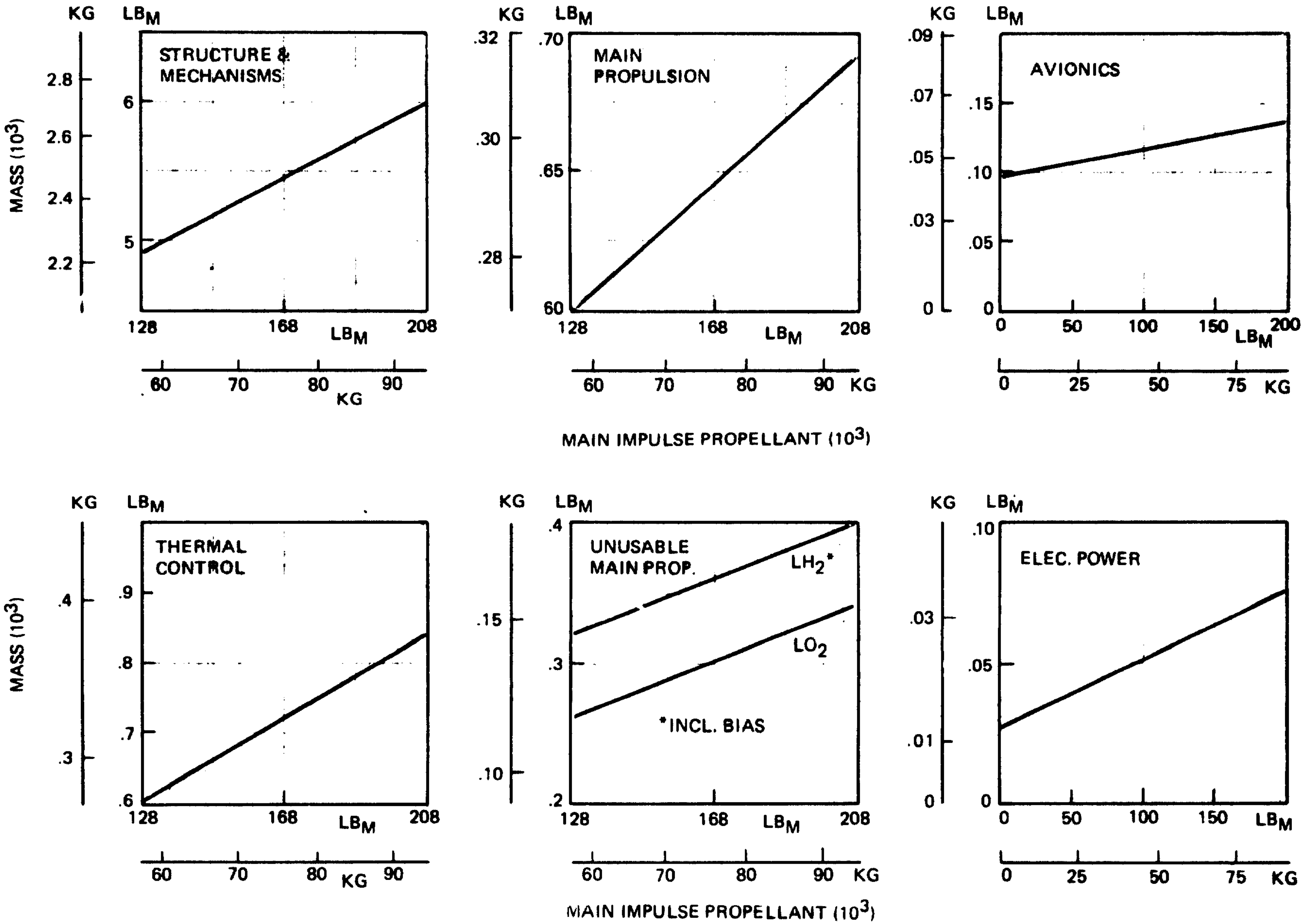
[2] Values are for one tank containing 15,900kg (35,000 lbm) impulse prop. Two tanks required



SCALING PARAMETERS A = 4,090 KG (9,020 LB) B = 0.0733 C = 0 D = 0.1725

IEF-3

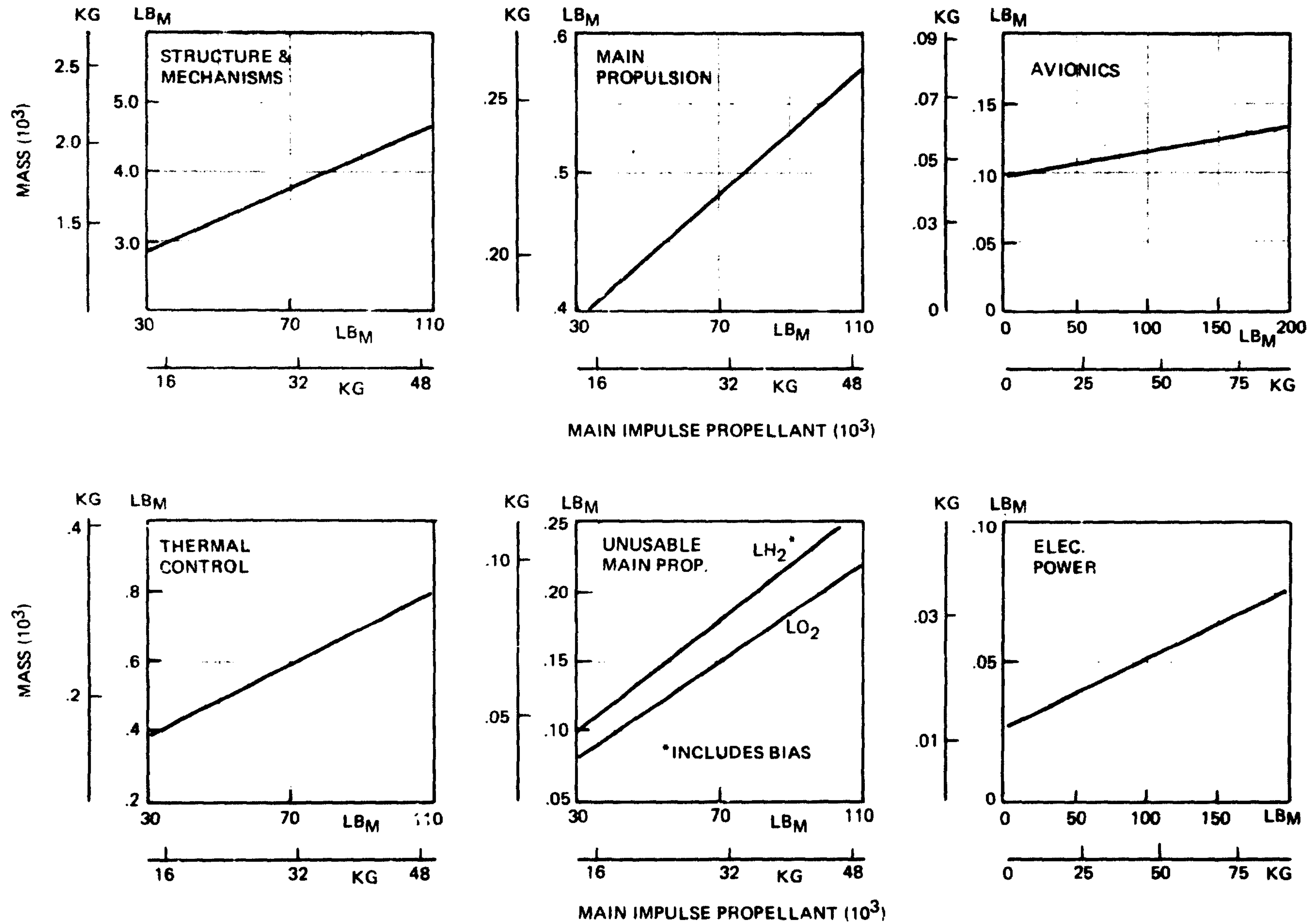
Figure 1-19. Subsystem Parametrice — LO_2/LH_2 1½ Stage OTV - Main Stage (Sheet 1)



SCALING PARAMETERS A = 1,120 KG (2,470 LB) B = 0.022 C = 0 D = 0

IEF-4

Figure 1-19. Subsystem Parametrics — LO_2/LH_2 1½ Stage OTV — Drop Tank Set Number 1 (Sheet 2)



SCALING PARAMETERS A = 920 KG (2,020 LB) B = 0.0320 C = 0 D = 0

IEF-6

Figure 1-19. Subsystem Parametrics — LO_2/LH_2 1½ Stage OTV — Drop Tank Set Number 2 (Sheet 3)

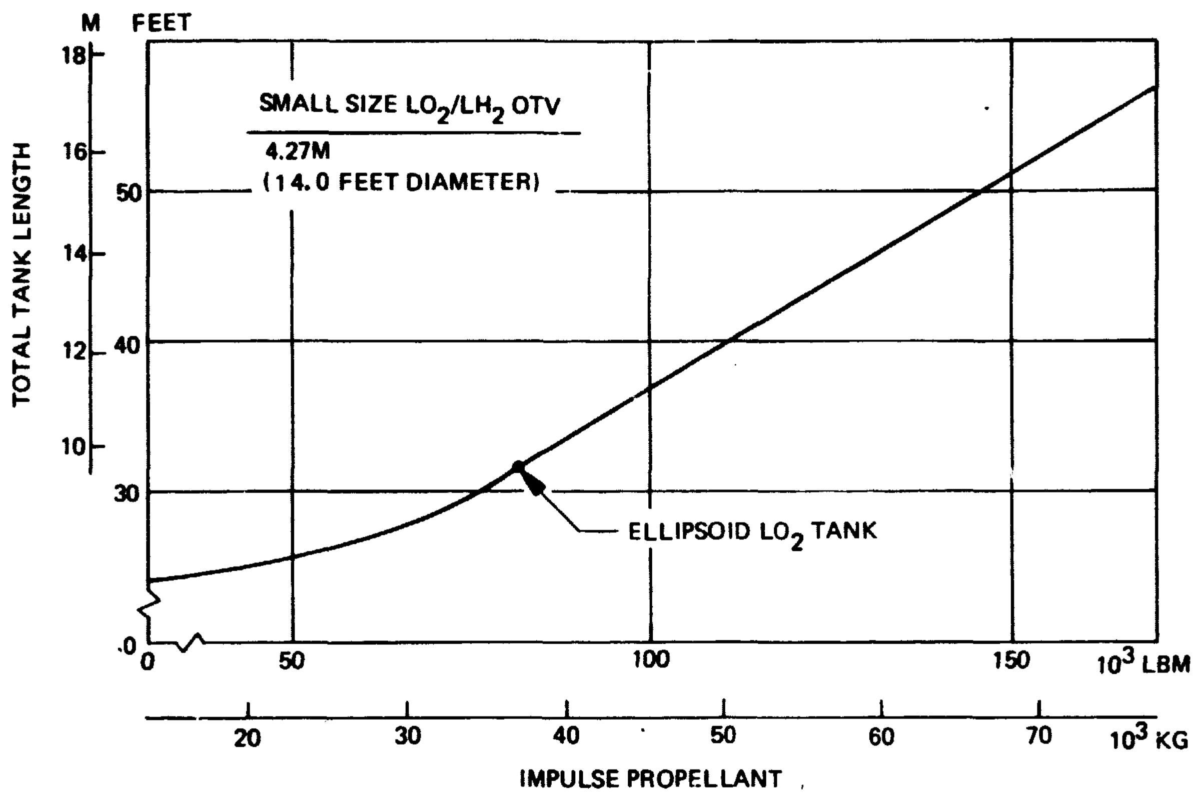
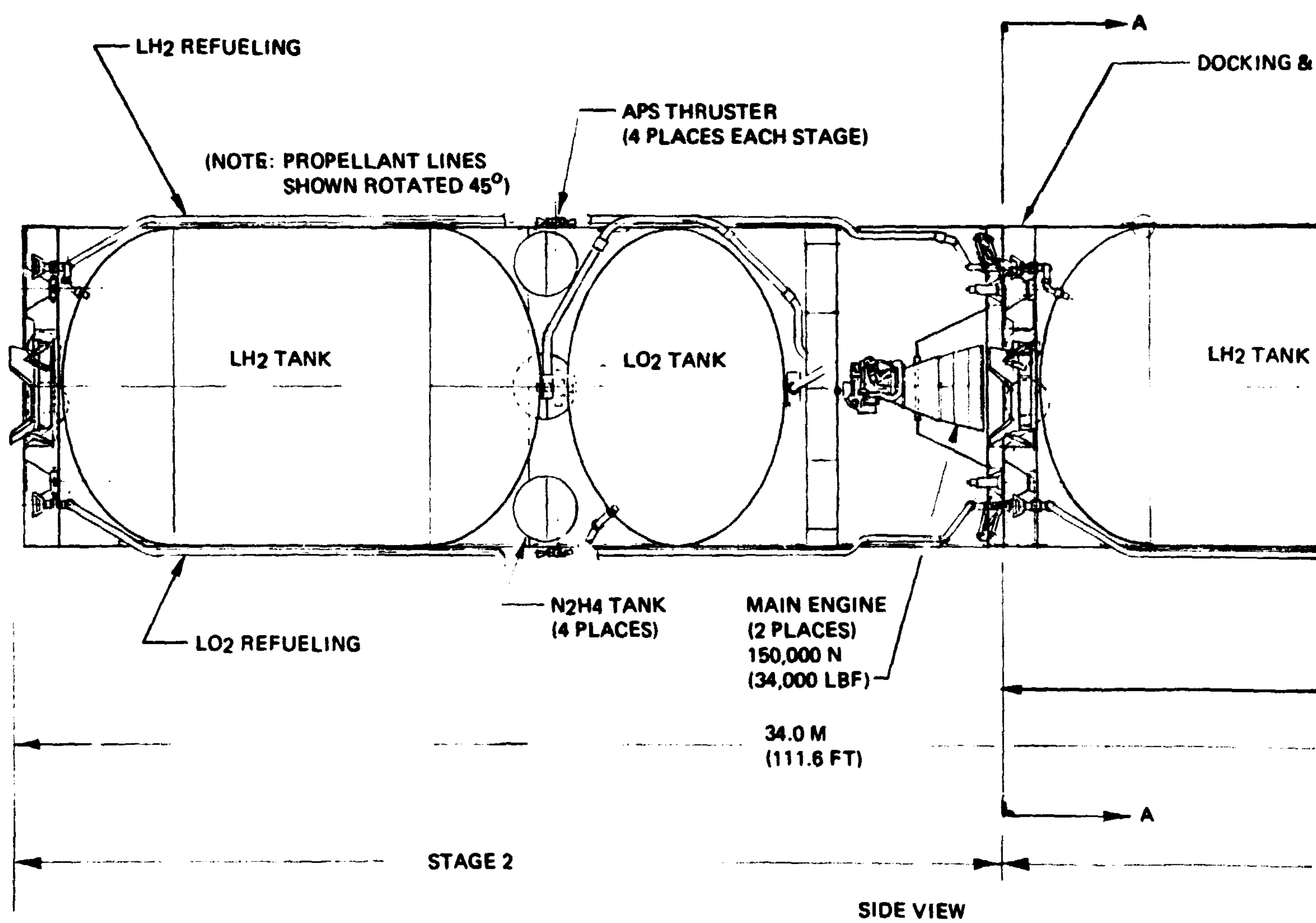


Figure 1-19. Applies to Main Stage and Drop Tanks (Sheet 4)



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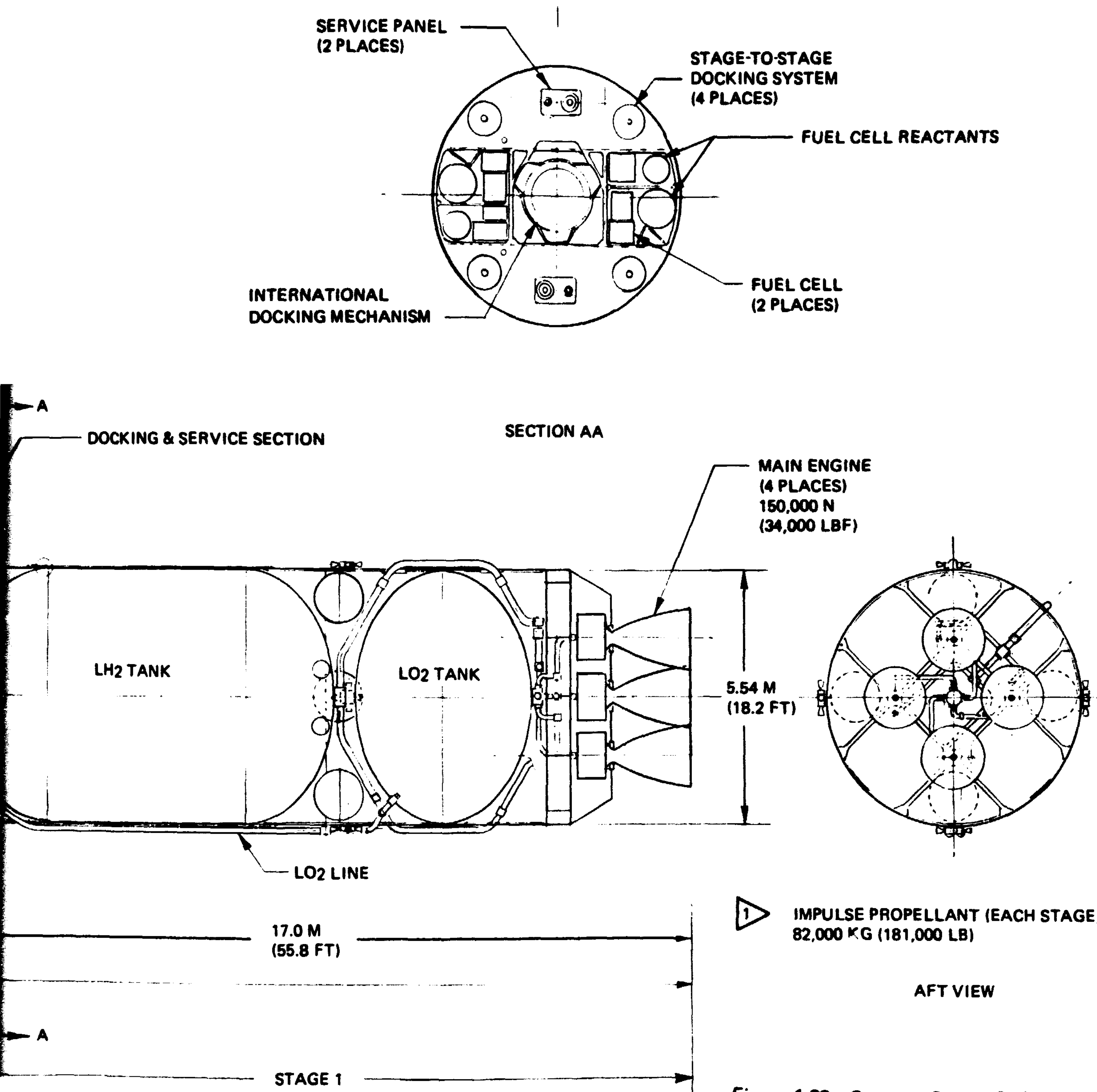

Figure 1-20 Common Stage LO_2/LH_2 OTV Configuration Point Design 1

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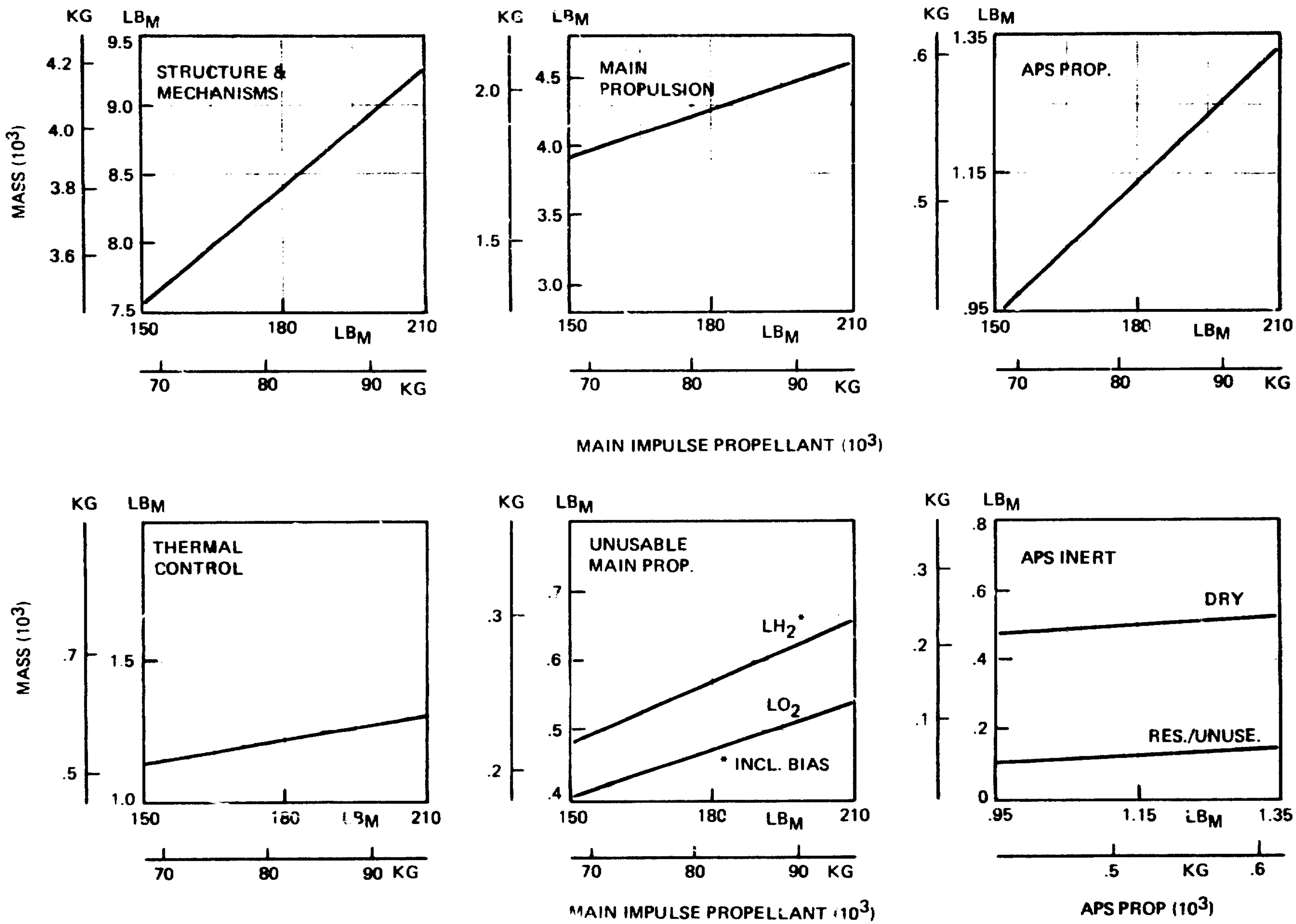
Table 1-10. Common Stage LO_2/LH_2 OTV Weight Details Large Size Point Design [1]

	1st Stage		2nd Stage	
	(LBM)	(KG)	(LBM)	(KG)
Structure and Mechanisms	(8,470)	(3,840)	(8,830)	(4,000)
Body Shell	1,940		2,160	
Fuel Tank	3,270		3,270	
Oxidizer Tank	1,470		1,470	
Thrust Structure	460		280	
Stage/Payload Interface	900		1,000	
Secondary Structure	430		650	
Landing Gear	–		–	
Main Propulsion	(4,270)	(1,940)	(3,100)	(1,410)
Main Engines	2,130		1,070	
Accessories	560		280	
Pressurization & Vent	460		460	
Propellant System	640		1,050	
Gimbal System	480		240	
Auxiliary Propulsion	(510)	(230)	(810)	(370)
Thrusters	240		300	
Tanks	70		260	
Pressurization & Vent	50		50	
Propellant System	150		200	
Avionics	(500)	(230)	(520)	(240)
Nav Guid & Control	160		160	
Data Management	160		160	
Communications	70		70	
Instrumentation	70		90	
Rendez & Docking	40		40	
Electrical Power	(640)	(290)	(800)	(360)
Fuel Cells	160		160	
Batteries	80		80	
Tankage	30		160	
Processing & Control	70		70	
Wiring Harnesses	300		330	
Thermal Control	(1,220)	(550)	(1,460)	(660)
Main Tank Insulation	640		880	
Insulation Purge	120		120	
Equipment Control	150		150	
Base Protection	200		200	
Paint & Sealer	110		110	
Weight Growth (15%)	(2,340)	(1,060)	(2,330)	(1,060)
Total Tank Dry Weight	17,950	8,140	17,850	8,100

[1] Based upon 82,100 kg (181,000 lbm) impulse propellant in each stage

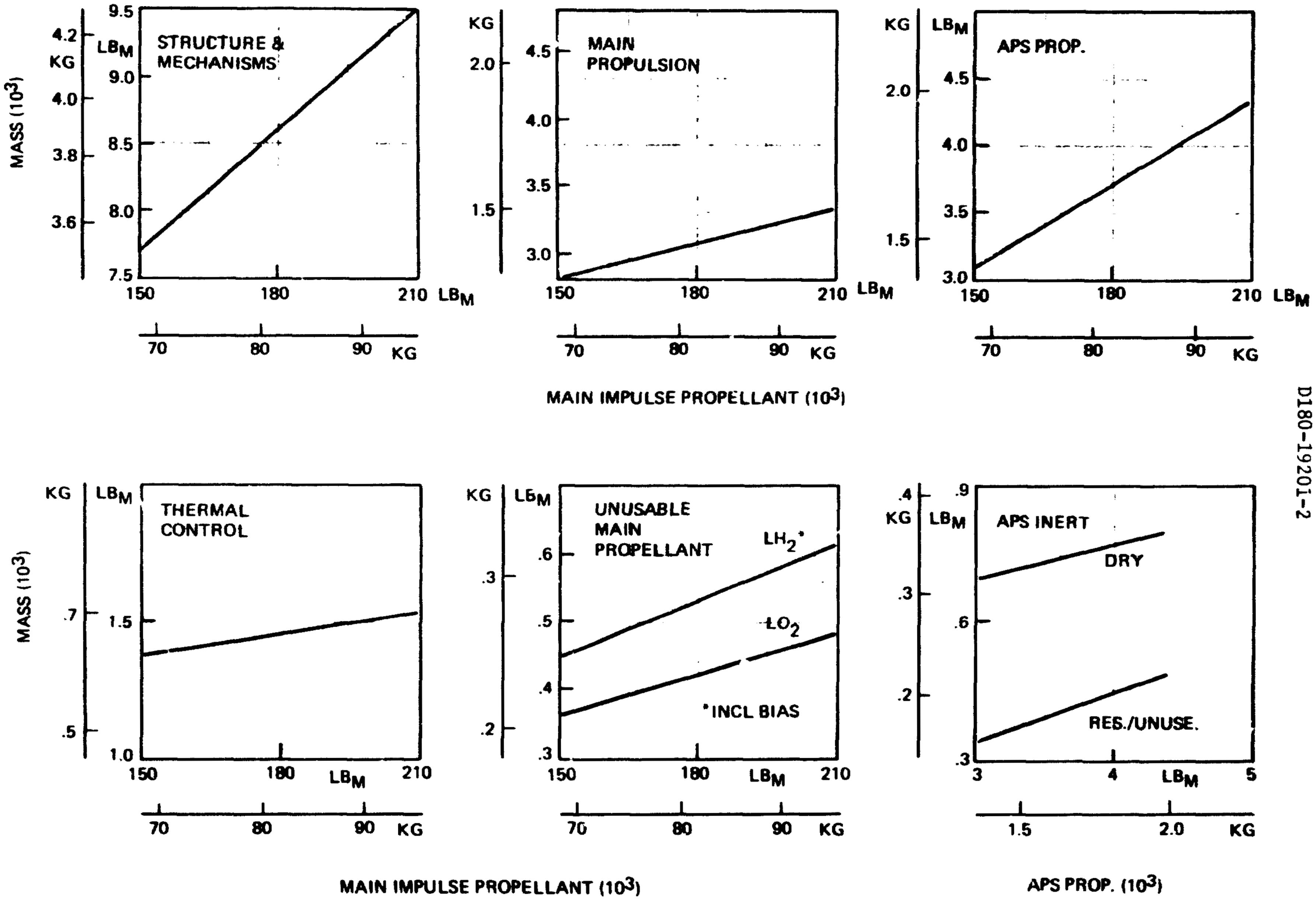
IEF-219



SCALING PARAMETERS A = 4,030 KG (8,885 LB) B = 0.05567 C = 0 D = 0.1725

IEF-6

Figure 1-21. Subsystem Parametrics — LO_2/LH_2 Common Stage - Stage 1 (Sheet 1)



SCALING PARAMETERS A = 4,100 KG (9,040 LB) B = 0.05317 C = 0 D = 0.1725

Figure 1-21. Subsystem Parametrics – LO_2/LH_2 Common Stage – Stage 2 (Sheet 2)

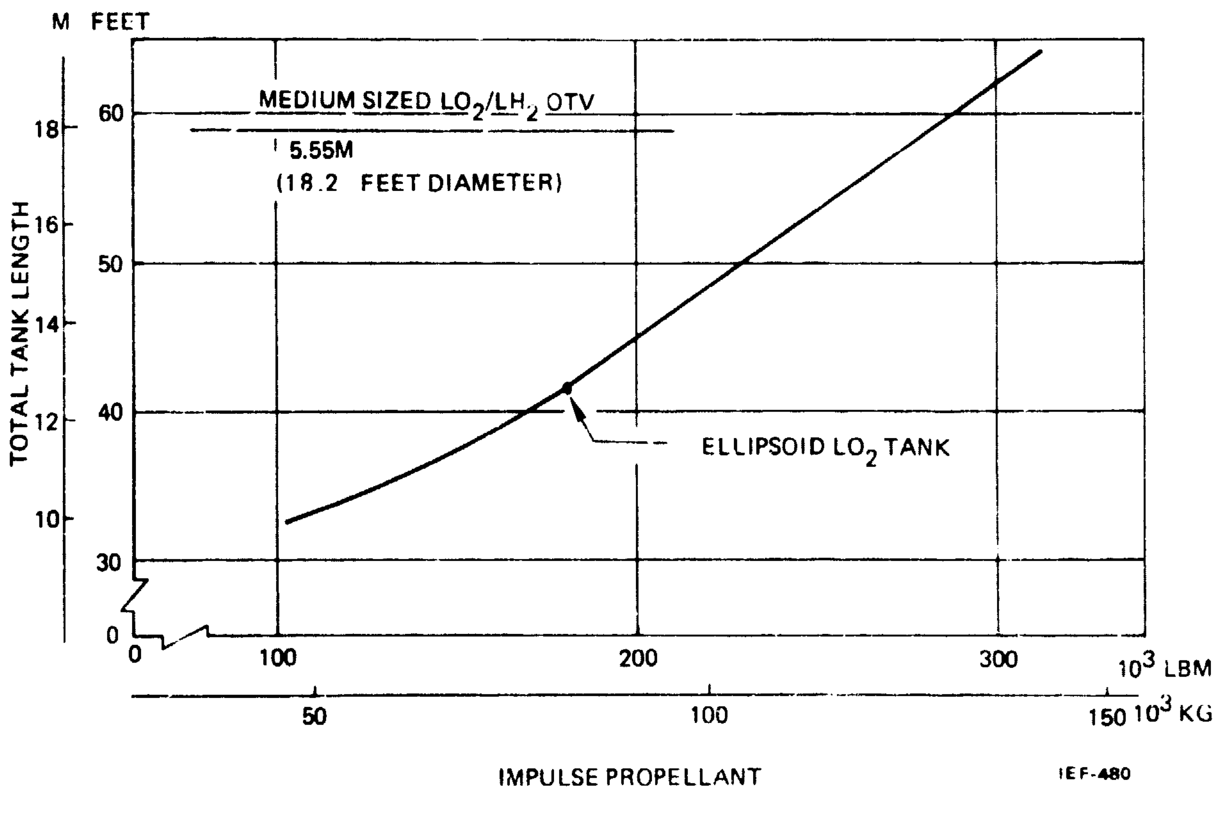
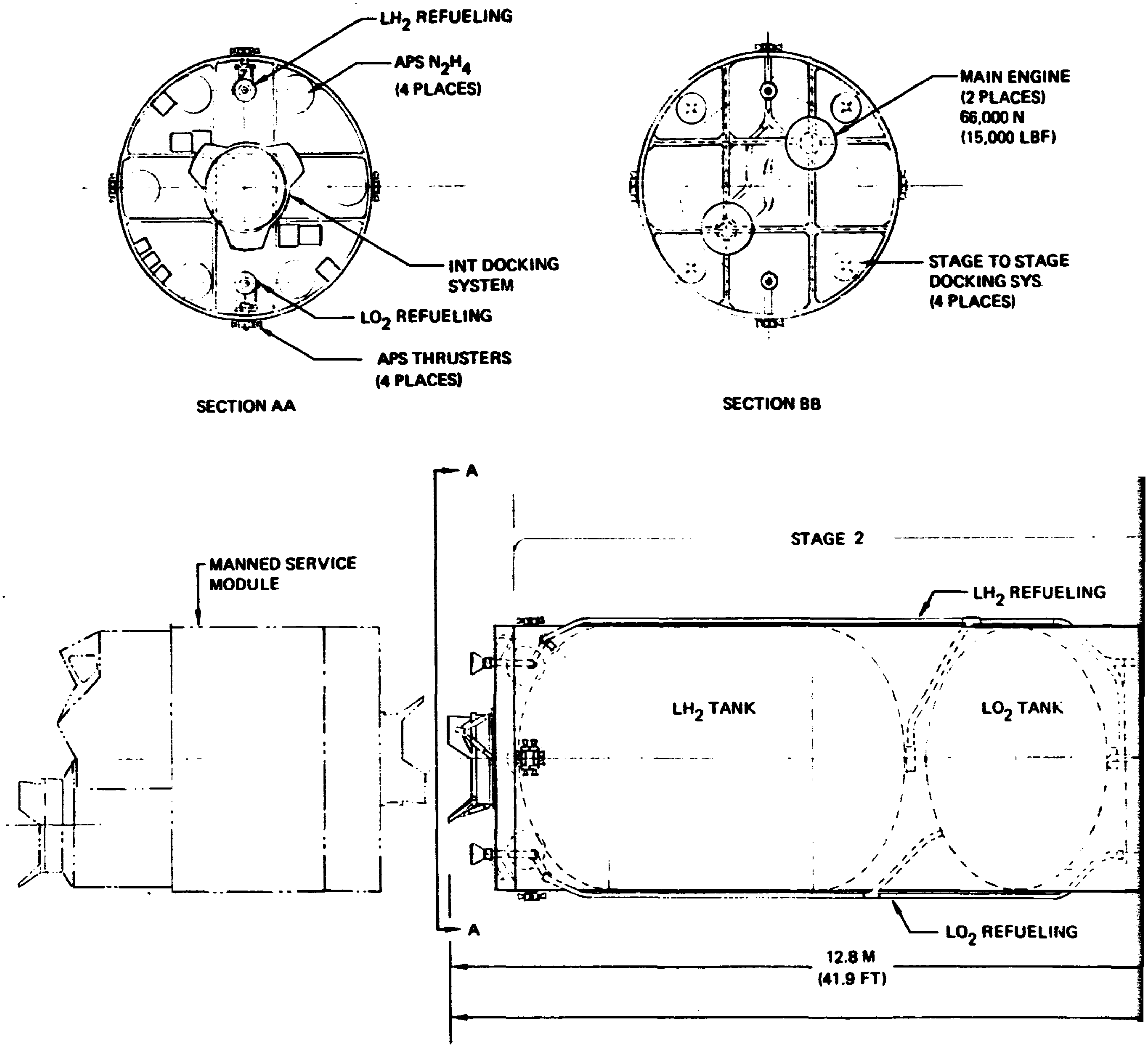


Figure 1-21. Values Are Per Stage (Sheet 3)



1 IMPULSE PROPELLANT (EA STG)
36,740 KG (81,000 LB)

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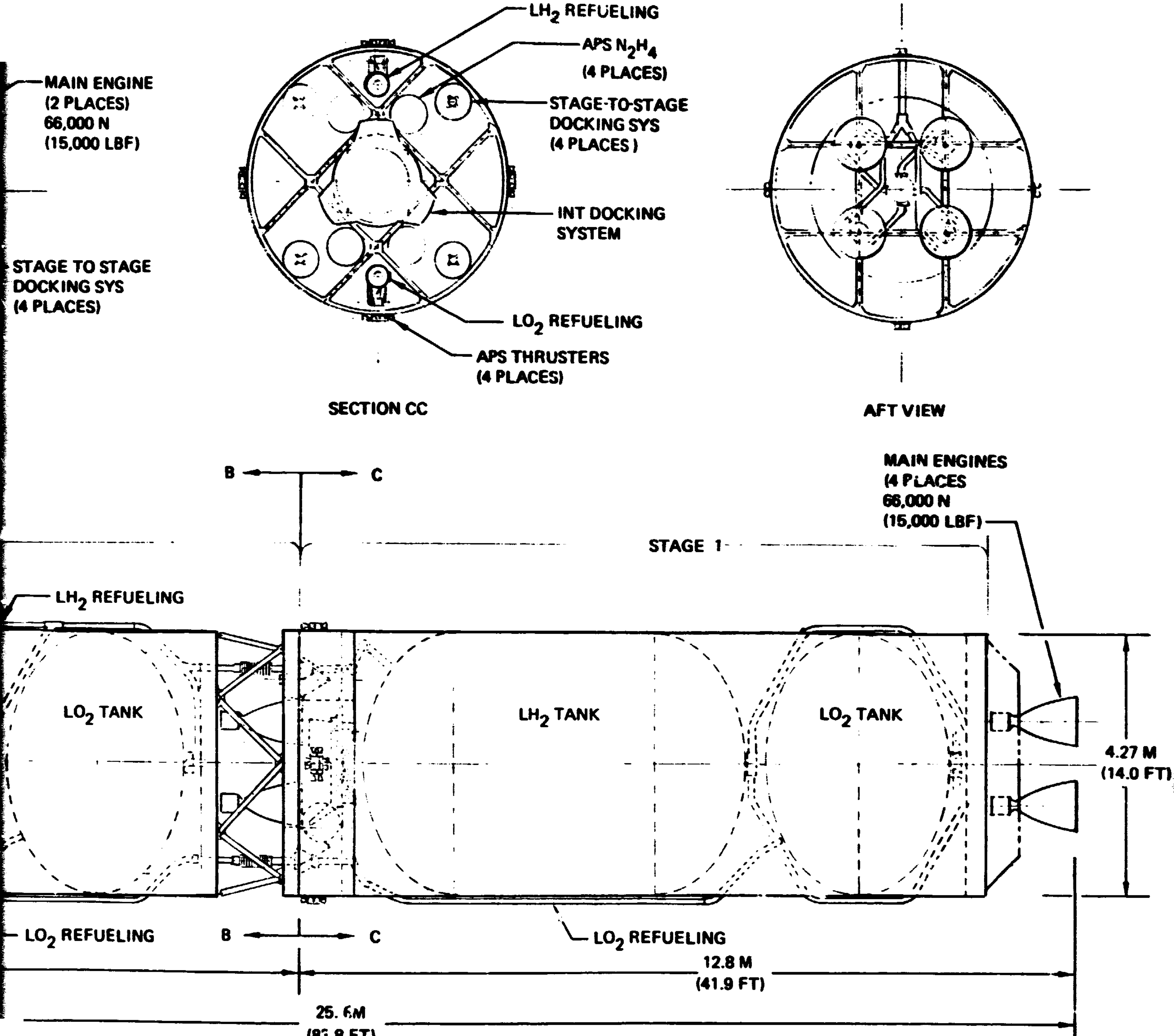

Figure 1-22 Common Stage LO_2/LH_2 OTV Configuration Point Design 1

Table 1-11. Common Stage LO_2/LH_2 OTV Weight Details Medium Size Point Design [1]

	1st Stage		2nd Stage	
	(LBM)	(KG)	(LBM)	(KG)
Structure and Mechanisms	(3,880)	(1,760)	(4,140)	(1,880)
Body Shell	910		1,050	
Fuel Tank	1,240		1,240	
Oxidizer Tank	550		550	
Thrust Structure	260		160	
Stage/Payload Interface	580		680	
Secondary Structure	340		460	
Landing Gear	–		–	
Main Propulsion	(2,550)	(1,150)	(1,940)	(880)
Main Engines	1,000		500	
Accessories	320		160	
Pressurization & Vent	590		590	
Propellant System	440		590	
Gimbal System	200		100	
Auxiliary Propulsion	(410)	(190)	(680)	(310)
Thrusters	160		200	
Tanks	40		230	
Pressurization & Vent	60		60	
Propellant System	150		190	
Avionics	(500)	(230)	(500)	(220)
Nav Guid & Control	160		160	
Data Management	160		160	
Communications	70		70	
Instrumentation	70		70	
Rendez & Docking	40		40	
Electrical Power	(510)	(230)	(700)	(320)
Fuel Cells	160		160	
Batteries	50		50	
Tankage	30		210	
Processing & Control	90		90	
Wiring Harnesses	180		190	
Thermal Control	(850)	(380)	(1,000)	(450)
Main Tank Insulation	390		540	
Insulation Purge	130		130	
Equipment Control	150		150	
Base Protection	100		100	
Paint & Sealer	80		80	
Weight Growth (15%)	(1,310)	(600)	(1,340)	(610)
Total Tank Dry Weight	10,010	4,540	10,300	4,670

[1] Based upon 36,700 kg (81,000 lbm) impulse propellant in each stage

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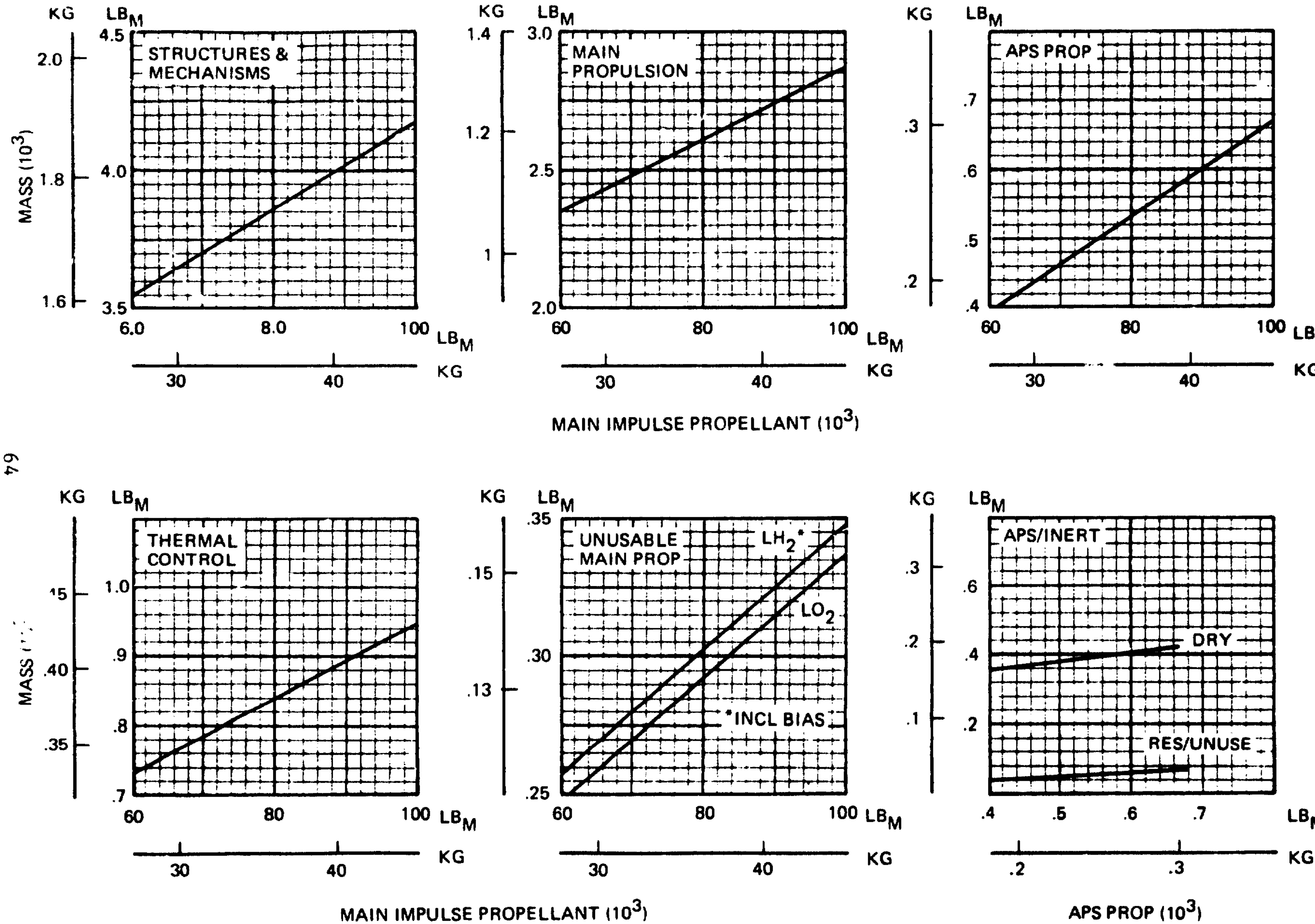
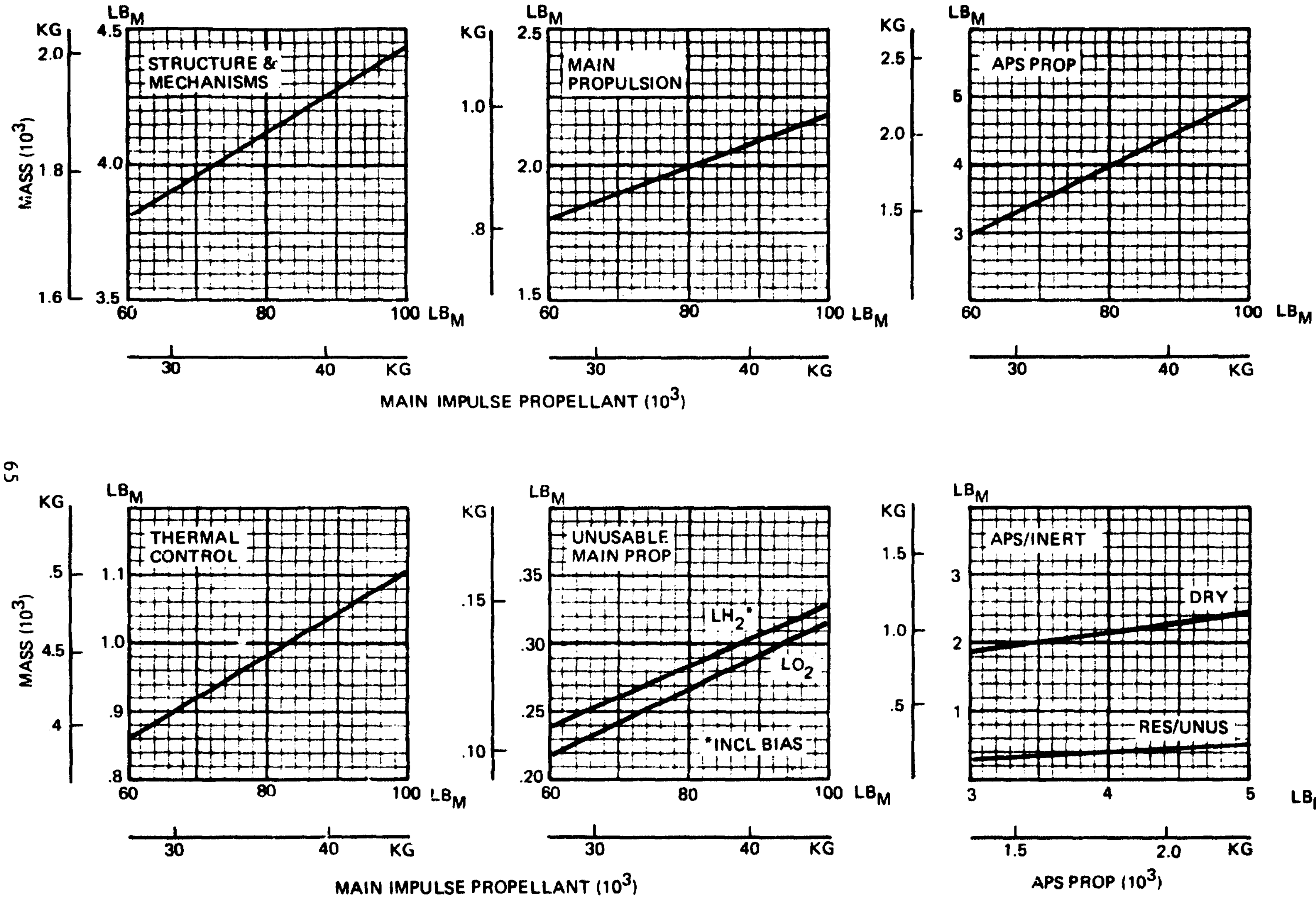


SCALING PARAMETERS A = 3,130 KG (6,900 LB) B = 0.0473 C = 0 D = 0.1725

IEF-11

Figure 1-23. Subsystem Parametrics LO_2/LH_2 Common Stage - Medium Stage 1 (Sheet 1)



SCALING PARAMETERS A = 3,450 KG (7,610 LB) B = 0.0423 C = 0 D = 0.1725

Figure 1-23. Subsystem Parametrics LO_2/LH_2 Common Stage - Medium Stage 2 (Sheet 2)

IEF-12

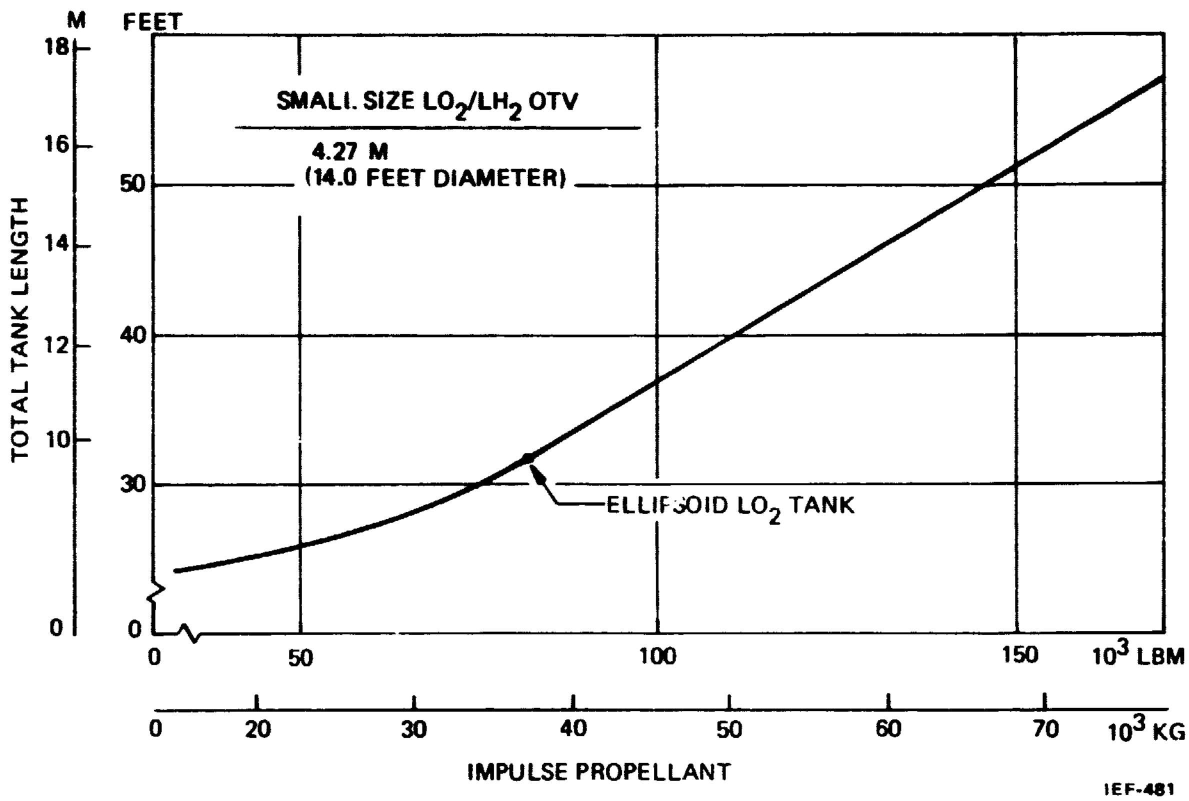
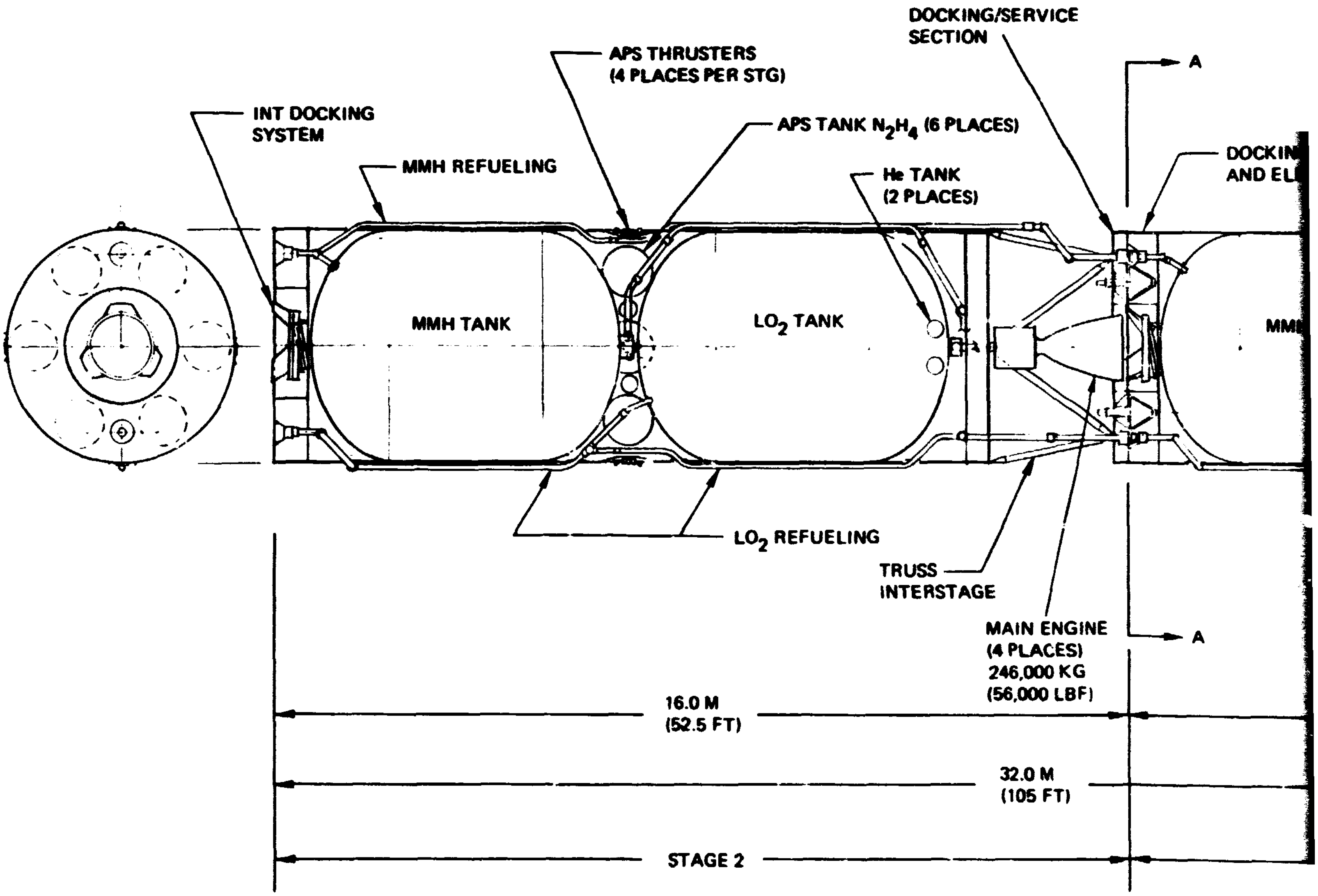


Figure 1-23. (Sheet 3)
Values Are Per Stage

1 IMPULSE PROPELLANT (EA STG)
127,000 KG (280,000 LB)



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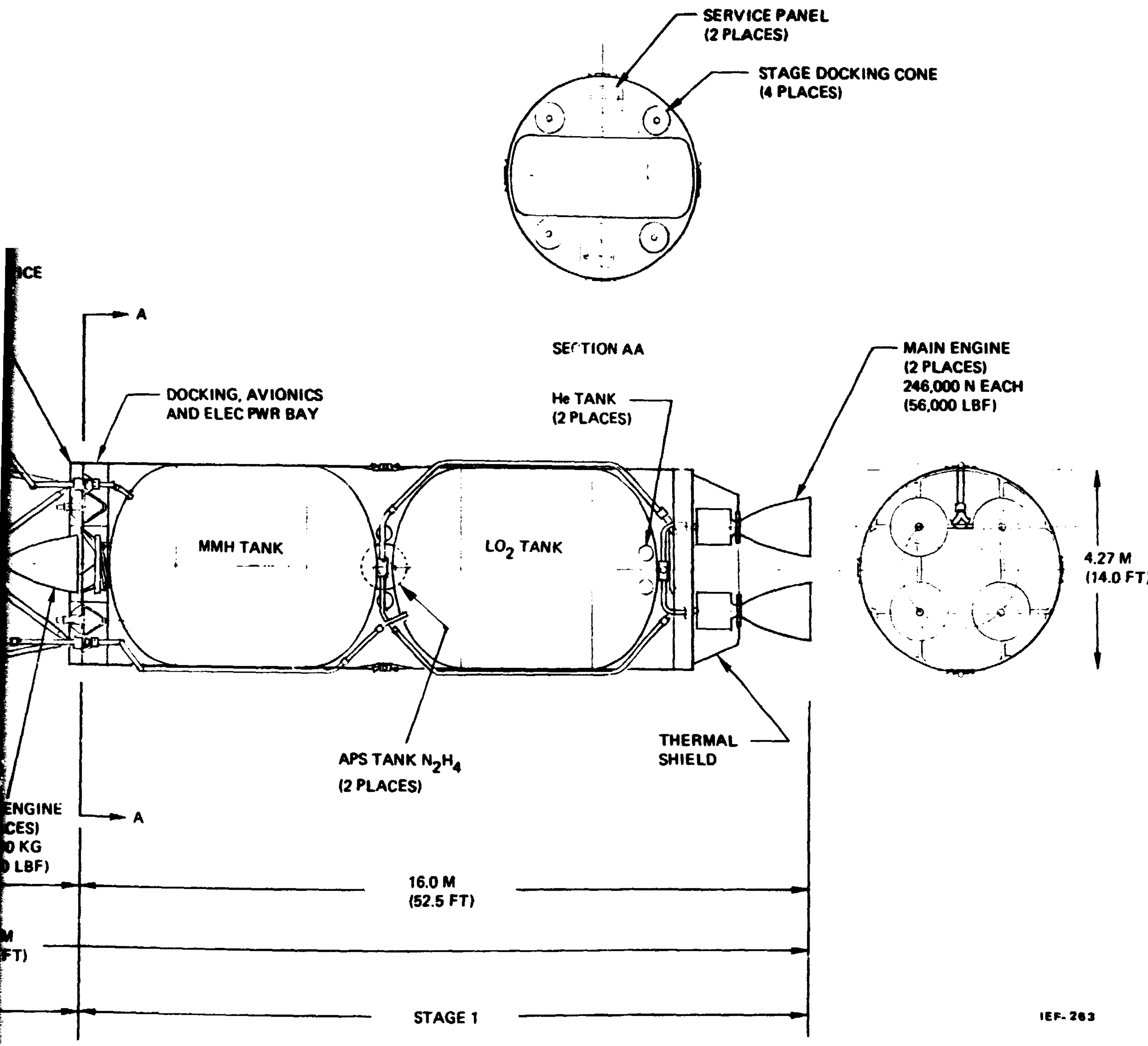

Figure 1-24 Common Stage LO_2/MMH OTV Configuration Point Design 1

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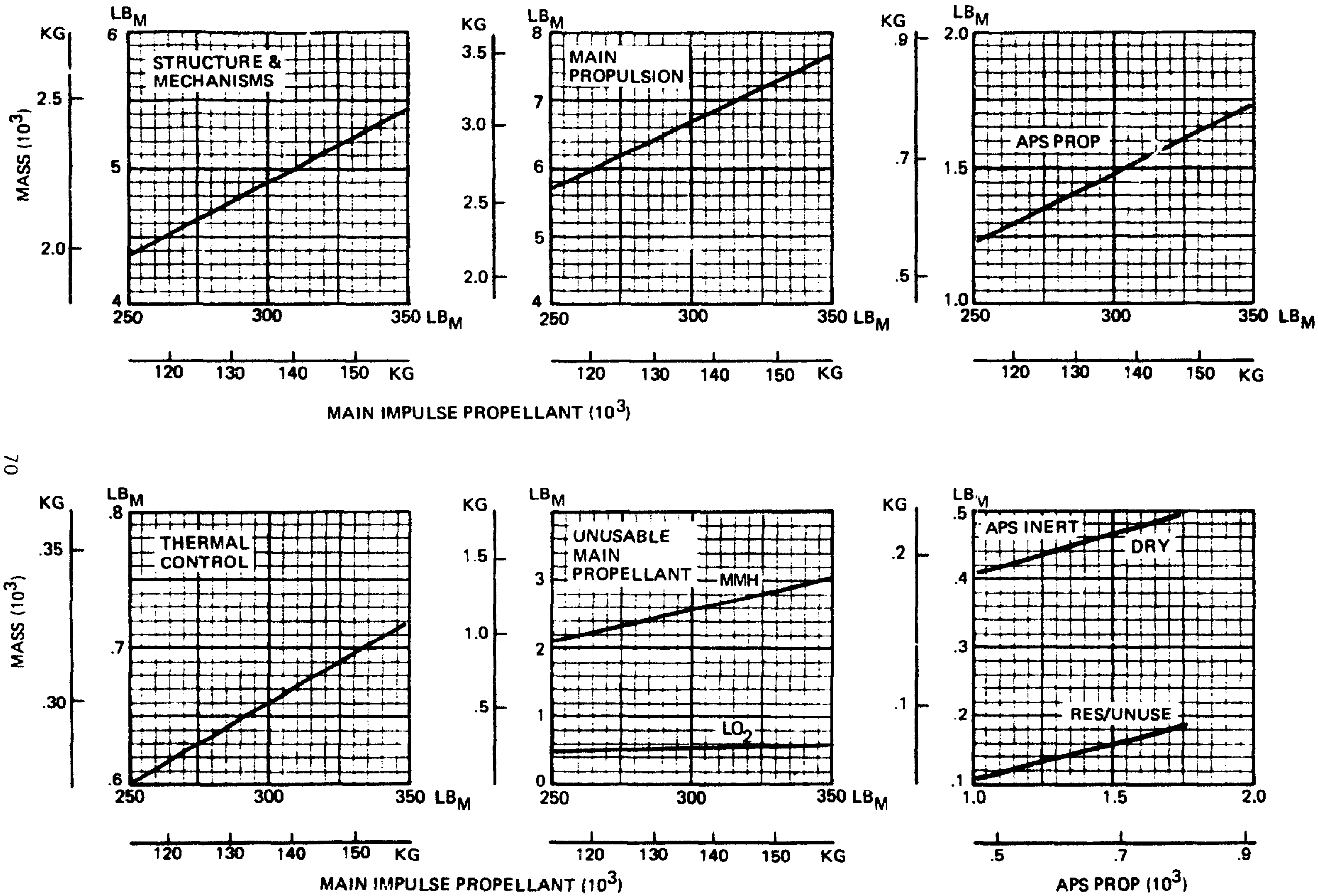
Table 1-12. Common Stage LO2/MMH OTV Weight Details Large Size Point Design [1]

	1st Stage		2nd Stage	
	(LBM)	(KG)	(LBM)	(KG)
Structure and Mechanisms	(4,900)	(2,220)	(5,140)	(2,330)
Body Shell	1,100		1,310	
Fuel Tank	1,110		1,110	
Oxidizer Tank	1,110		1,110	
Thrust Structure	550		310	
Stage/Payload Interface	640		750	
Secondary Structure	390		550	
Landing Gear	–		–	
Main Propulsion	(6,700)	(3,040)	(4,550)	(2,060)
Main Engines	2,950		1,460	
Accessories	650		320	
Pressurization & Vent	1,090		1,090	
Propellant System	1,310		1,380	
Gimbal System	600		300	
Auxiliary Propulsion	(570)	(260)	(890)	(400)
Thrusters	180		240	
Tanks	90		270	
Pressurization & Vent	80		160	
Propellant System	220		220	
Avionics	(580)	(260)	(580)	(260)
Nav Guid & Control	160		160	
Data Management	160		160	
Communications	70		70	
Instrumentation	150		150	
Rendez & Docking	40		40	
Electrical Power	(850)	(390)	(1,050)	(480)
Fuel Cells	200		200	
Batteries	140		140	
Tankage	30		200	
Processing & Control	110		110	
Wiring Harnesses	370		400	
Thermal Control	(660)	(300)	(740)	(340)
Main Tank Insulation	250		330	
Insulation Purge	80		80	
Equipment Control	150		150	
Base Protection	100		100	
Paint & Sealer	80		80	
Weight Growth (15%)	(2,140)	(970)	(1,940)	(880)
Total Tank Dry Weight	16,400	7,440	14,890	6,750

[1] Based upon 136,500 kg (301,000 lbm) impulse propellant in each stage

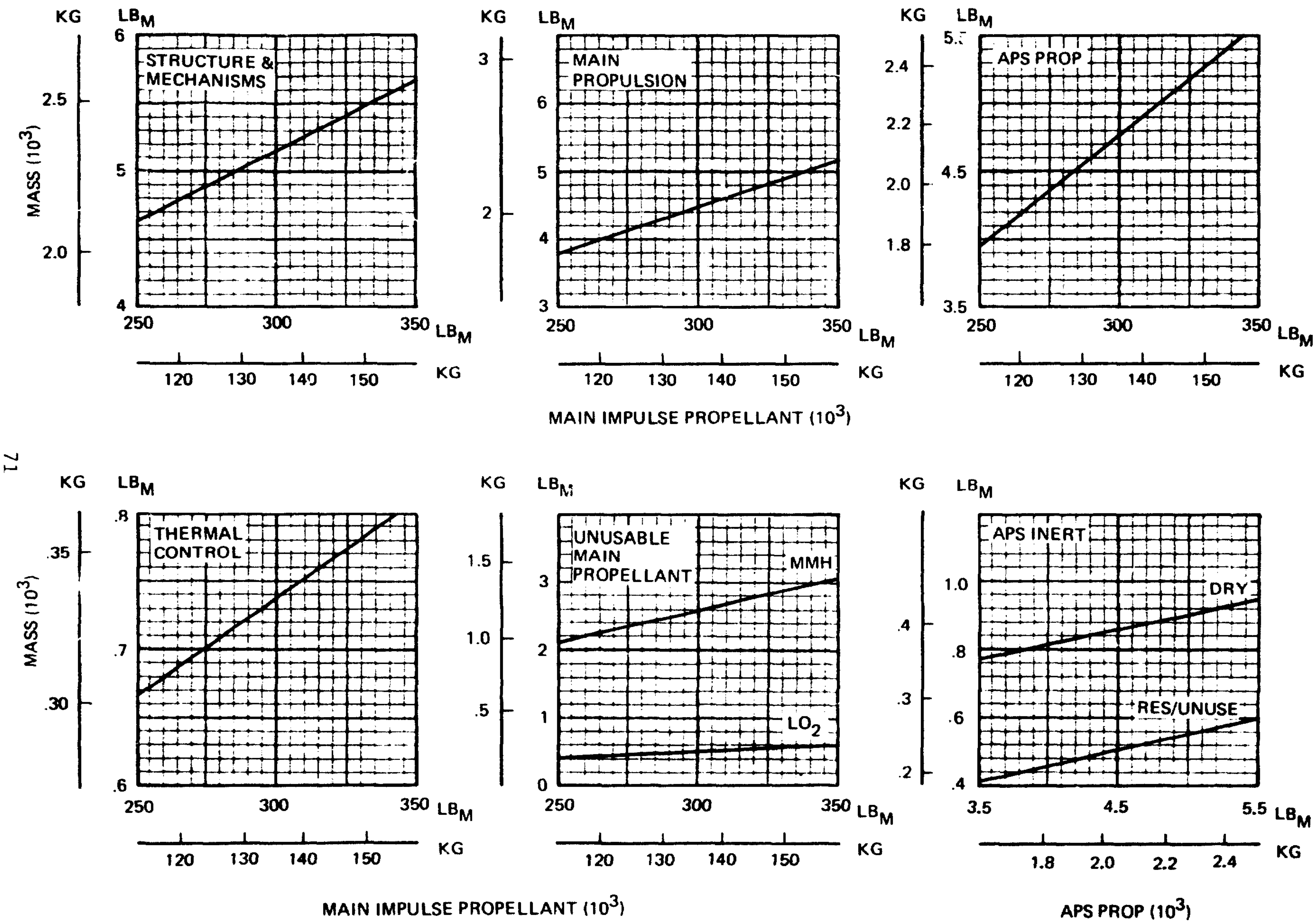
IEF-214



SCALING PARAMETERS A = 2,540 KG (5,590 LB) B = 0.0460 C = 0 D = 0.1725

Figure 1-25. Subsystem Parametrics – LO_2/MMH Common Stage – Stage 1 (Sheet 1)

IEF-14



SCALING PARAMETERS A = 2,890 KG (6,370 LB) B = 0.0394 C = 0 D = 0.1725

IEF-13

Figure 1-25. Subsystem Parametrics LO_2/MMH Common Stage Stage 2 (Sheet 2)

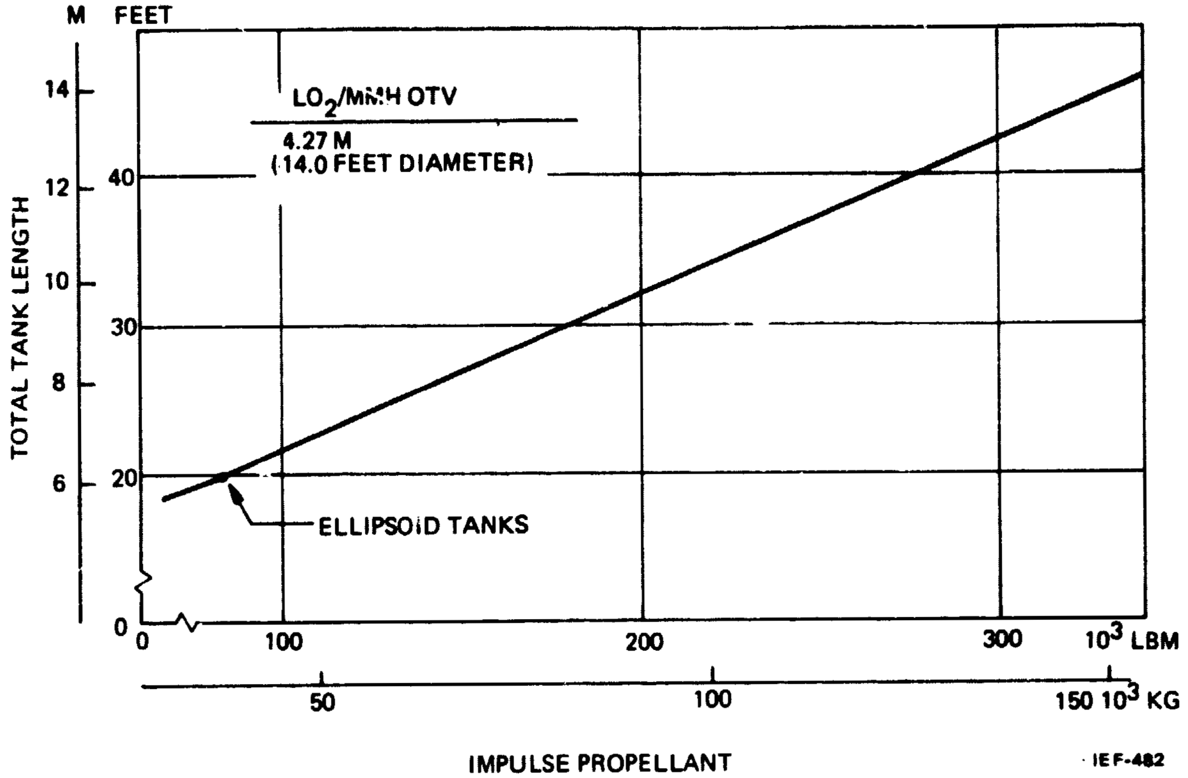
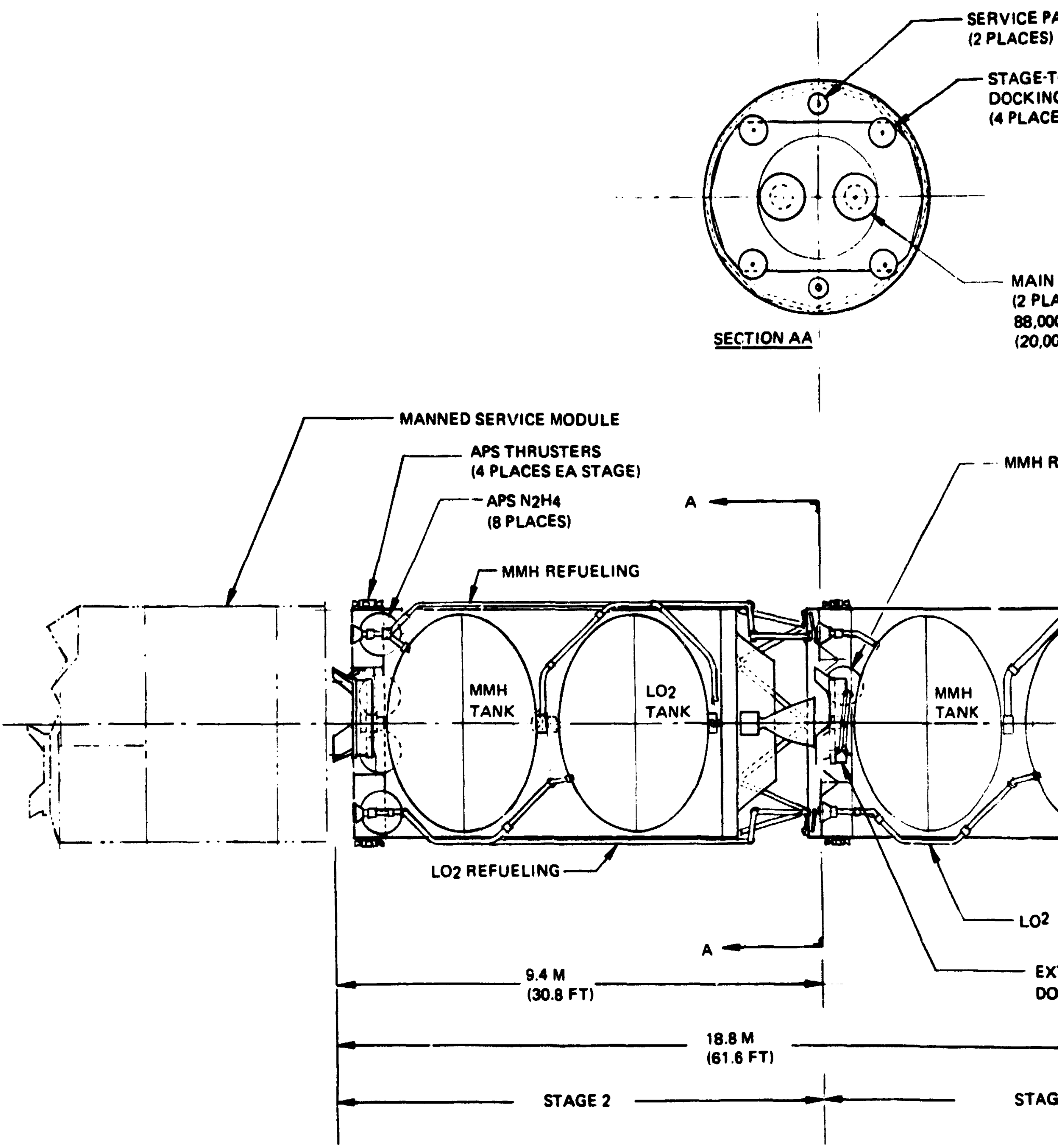


Figure 1-25. (Sheet 3)
Values Are Per Stage



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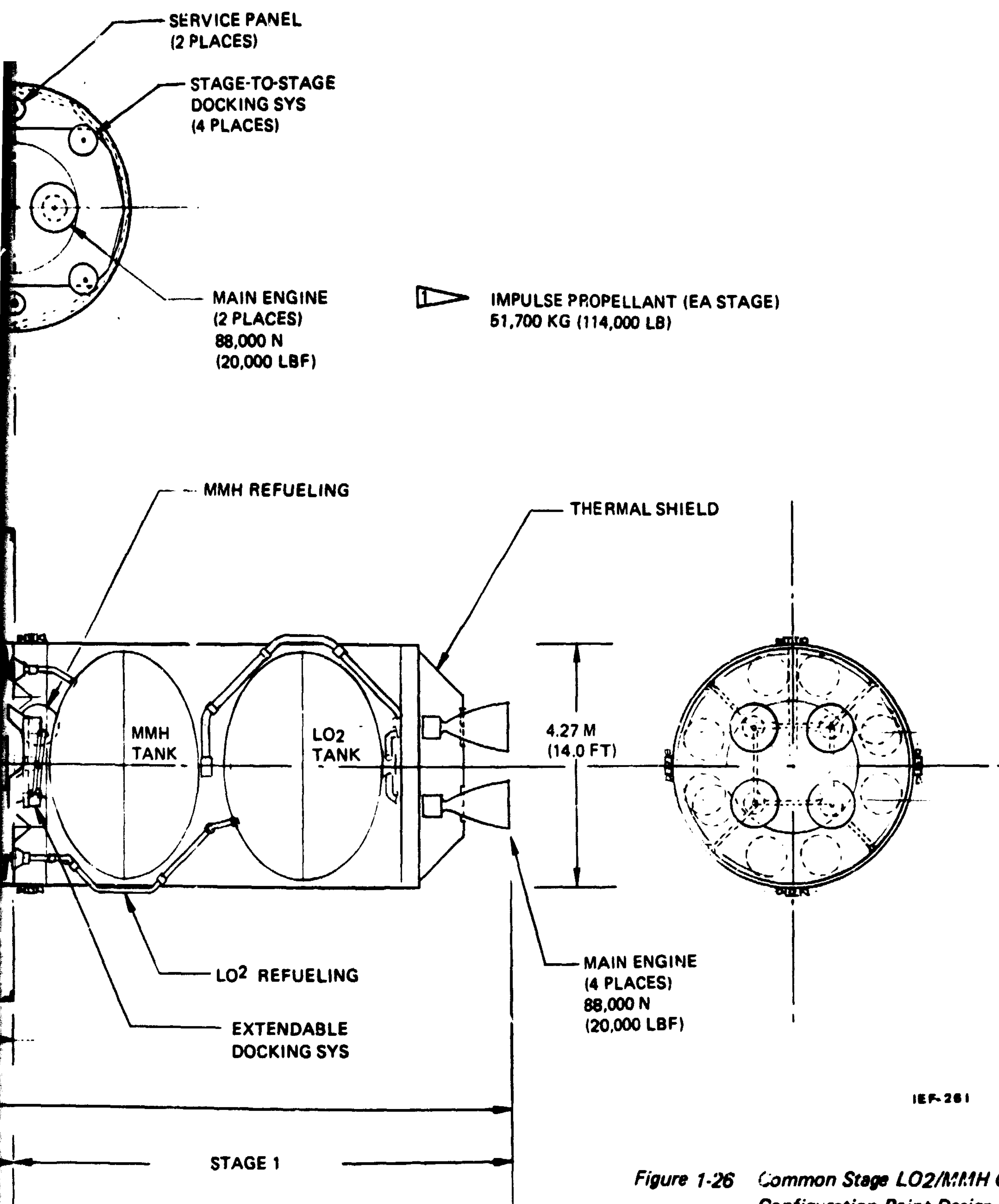

Figure 1-26 Common Stage LO2/MMH OTV Configuration Point Design 1

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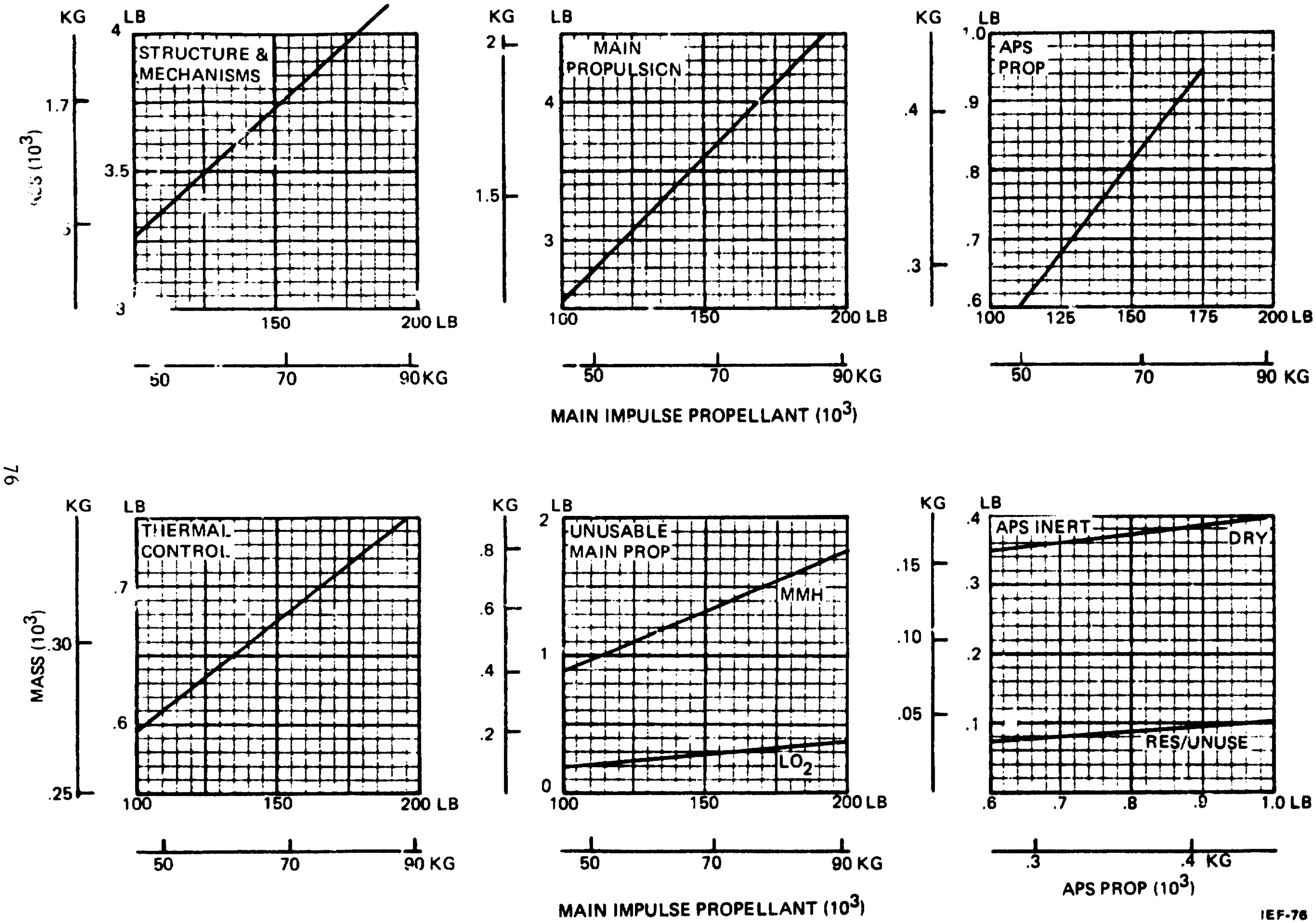
Table 1-13. Common Stage LO_2/MMH OTV Weight Details Medium Size Point Design [1]

	1st Stage		2nd Stage	
	(LBM)	(KG)	(LBM)	(KG)
Structure and Mechanisms	(3,400)	(1,542)	(3,830)	(1,740)
Body Shell	1,100		1,310	
Fuel Tank	510		510	
Oxidizer Tank	590		590	
Thrust Structure	280		280	
Stage/Payload Interface	580		680	
Secondary Structure	340		310	
Landing Gear	–		150	
Main Propulsion	(2,860)	(1,297)	(1,980)	(900)
Main Engines	1,140		570	
Accessories	320		190	
Pressurization & Vent	410		410	
Propellant System	510		570	
Gimbal System	480		240	
Auxiliary Propulsion	(460)	(209)	(790)	(360)
Thrusters	160		210	
Tanks	40		240	
Pressurization & Vent	80		160	
Propellant System	180		180	
Avionics	(500)	(227)	(500)	(220)
Nav Guid & Control	160		160	
Data Management	160		160	
Communications	70		70	
Instrumentation	70		70	
Rendez & Docking	40		40	
Electrical Power	(510)	(231)	(700)	(320)
Fuel Cells	160		160	
Batteries	60		60	
Tankage	30		210	
Processing & Control	90		90	
Wiring Harnesses	170		200	
Thermal Control	(620)	(281)	(720)	(330)
Main Tank Insulation	260		360	
Insulation Purge	70		70	
Equipment Control	150		150	
Base Protection	70		70	
Paint & Sealer	70		70	
Weight Growth (15%)	(1,250)	(567)	(1,280)	(580)
Total Tank Dry Weight	9,600	4,355	9,800	4,450

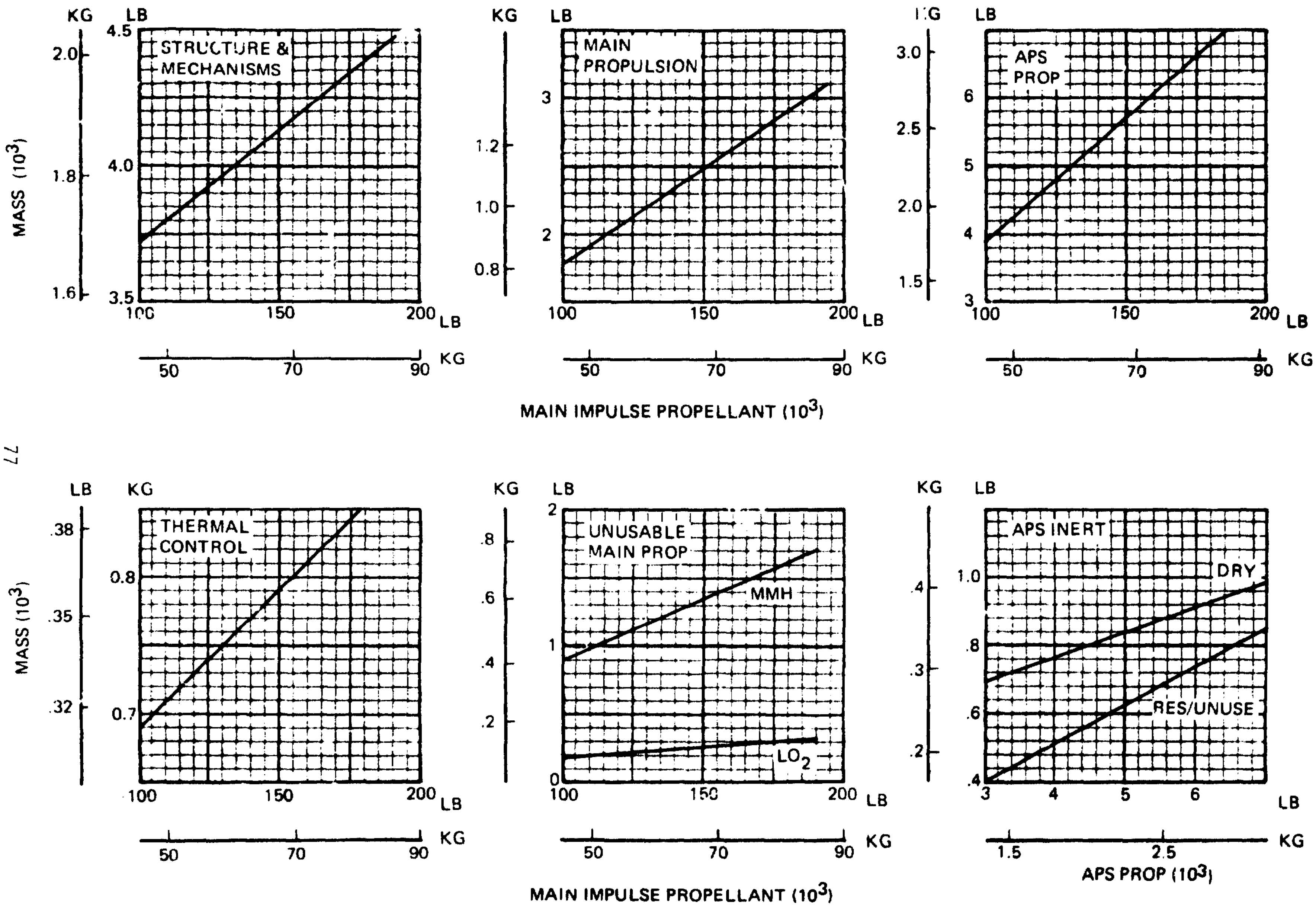
[1] Based upon 51,7[illegible] kg (114,000 lbm) impulse propellant in each stage

IEF-216



SCALING PARAMETERS A = 2,370 KG (5,230 LB) B = 0.0478 C = 0 D = 0.1725

Figure 1-27. Common Stage LO_2/MMH OTV Subsystem Parametrics First Stage (Sheet 1)



SCALING PARAMETERS A = 3,040 KG (6,700 LB) B = 0.0388 C = 0 D = 0.1725

Figure 1-27. Common Stage LO_2/MMH OTV Subsystem Parametrics Second Stage (Sheet 2)

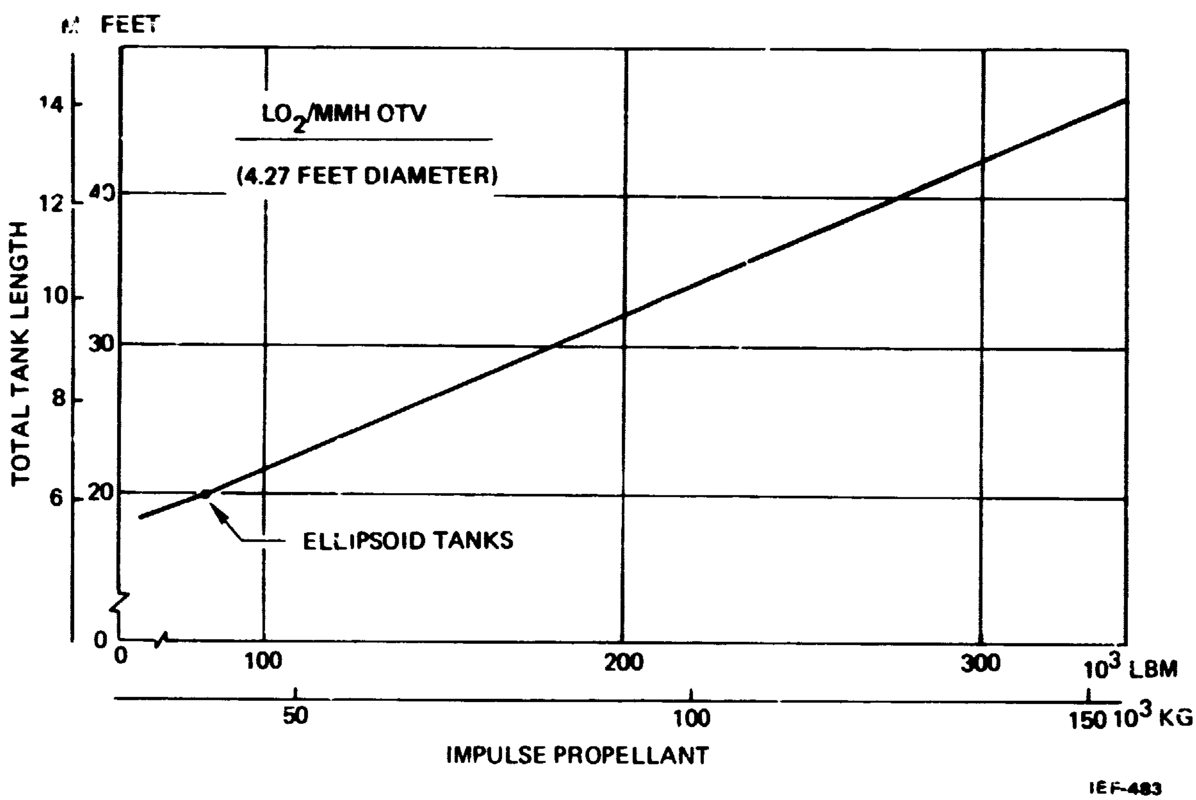


Figure 1-27. (Sheet 3)
Values Are Per Stage

1.2.2.9 Nuclear LH_2 OTV Point Design

This point design is applicable to the following missions:

- Geosynchronous Space Station (GSS)
- Orbiting Lunar Station (OLS)
- Lunar Surface Base (LSB)
- Manned Planetary Exploration (requires clustering and multistaging)

The configuration inboard profile as drawn (Figure 1-28) was sized to be applicable to the GSS mission; performance analysis using the mass properties developed for the point design resulted in tailored sizes for all applicable missions as reported in Volume I. Table 1-14 presents a detailed mass properties estimate for the point design as drawn; Mass properties parameter variations about the nominal point resulted in resulting scaling equation factors shown on Figure 1-29. The boiloff mass rate was estimated on the basis of cryogenic tank surface area and typical insulation performance.

Figure 1-29 presents the mass properties buildup parametrics. These may be used to develop mass properties summaries for this vehicle type over the applicable range of main impulse propellant loading.

1.2.3 Low Thrust OTV's

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.

1.2.3.1 Performance Characteristics of Electric Propulsion for Orbit Transfer

Electric propulsion systems are characterized by two major differences from high-thrust chemical or nuclear systems:

Table 1-14. Nuclear LH_2 OTV Weight Details Point Design [1]

	(LBM)	(KG)	(LBM)	(KG)
Structure and mechanisms	(24,820)	(11,260)		
Body shell	1,410			
Fuel tank	21,080			
Oxidizer tank	—			
Thrust structure	800			
Stage/payload interface	930			
Secondary structure	600			
Landing gear	—			
Main propulsion	(31,950)	(14,490)		
Main engine	25,800			
Shielding	4,000			
Pressurization and vent	880			
Propellant system	500			
Gimbal system	770			
Auxiliary propulsion	(1,360)	(620)		
Thrusters	360			
Tanks	540			
Pressurization and vent	110			
Propellant system	350			
Avionics	(570)	(260)		
Nav guidance and control	160			
Data management	160			
Communications	70			
Instrumentation	140			
Rendez and docking	40			
Electrical power	(800)	(360)		
Fuel cells	160			
Batteries	50			
Tankage	210			
Processing and control	90			
Wiring harnesses	290			
Thermal control	(3,240)	(1,470)		
Main tank insulation	1,860			
Insulation purge	180			
Equipment control	150			
Base protection	—			
Paint and sealer	50			
Weight growth (15%) [2]	(5,540)	(2,510)		
Total stage dry weight	68,280	30,970		

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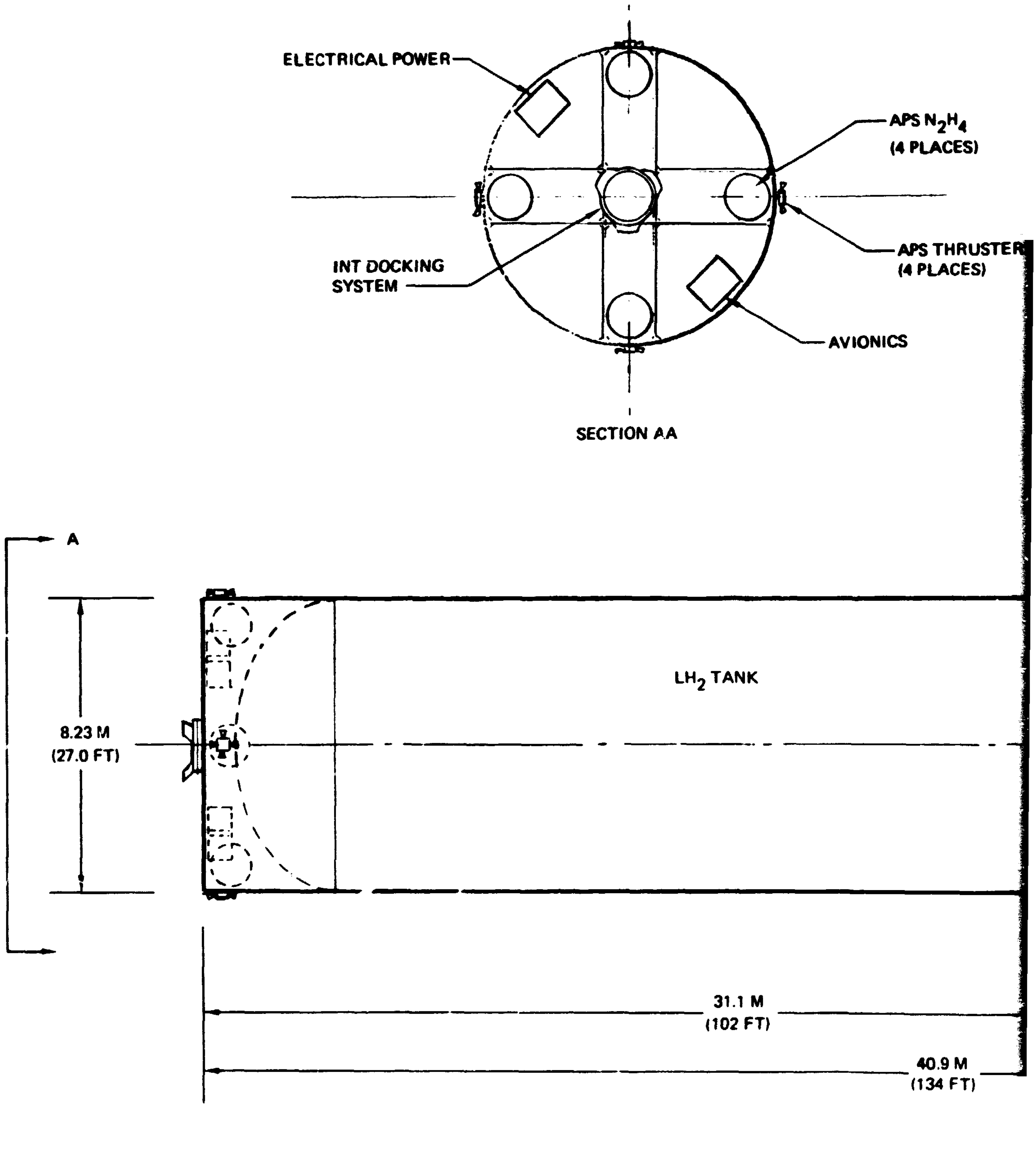
[1] Based upon 100,000 KG (220,000 LBM) impulse propellant.

[2] Weight growth 15% of dry weight less main engine.



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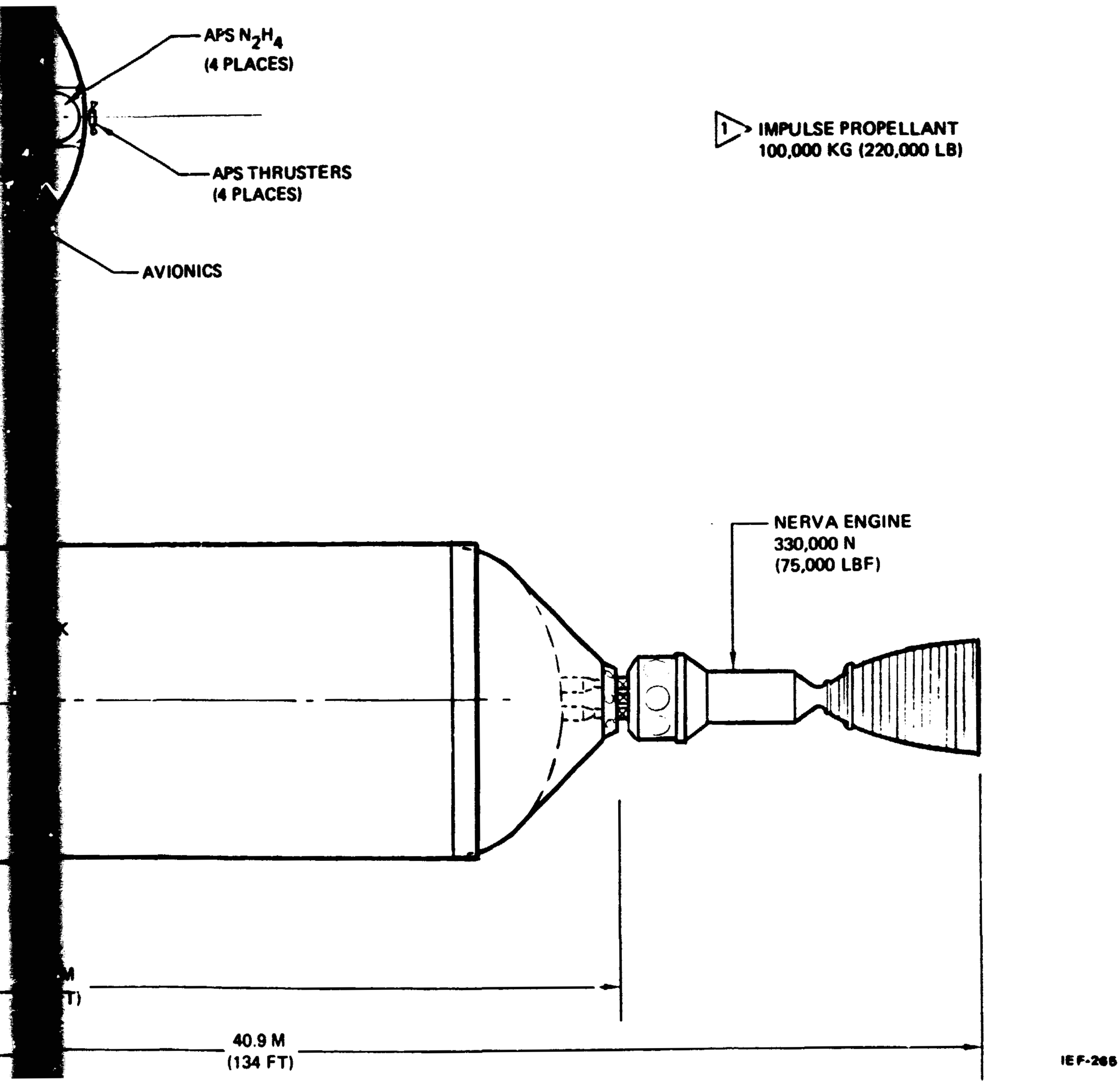
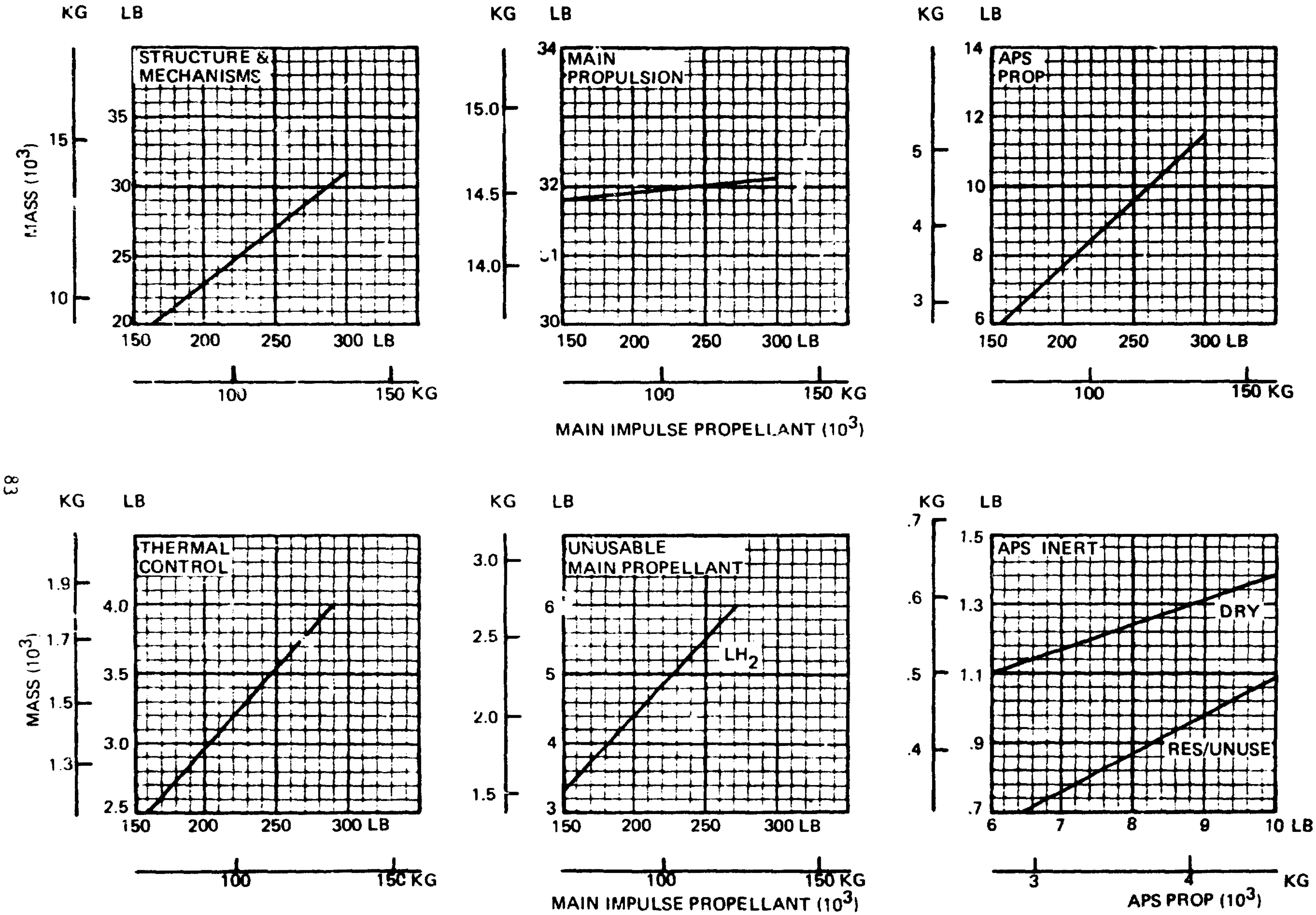


Figure 1-28 Single Stage Nuclear LH_2 OTV Configuration Point Design 1



SCALING PARAMETERS A = 19,775 KG (43,595 lb) B = 0.1323 C = 0 D = 0.1725

Figure 1-29. Nuclear LH_2 OTV Subsystem Parametrics (Sheet 1)

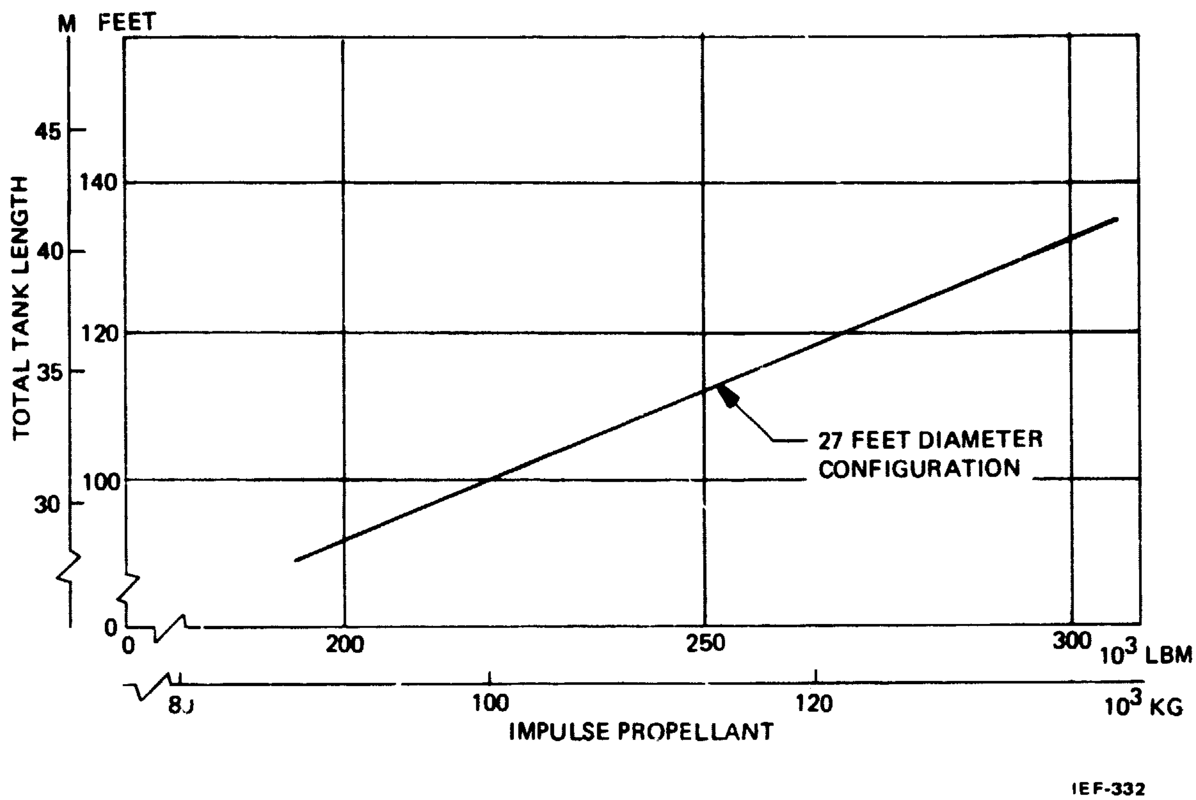


Figure 1-29. (Sheet 2)

1. **Jet velocity (Isp) is not limited by temperature or propellant energy because the propellant is accelerated by electrostatic or electromagnetic body forces on the propellant fluid. Thus any Isp within the practical range of interest (1000-10,000 sec) may be obtained.**

2. Attainable thrust level is limited to comparatively low values by limits on available power. The equivalent power in the jet is $N_T\ p_e$ where N_T is thruster efficiency, typically in the range 0.3 to 0.8, and p_e is electrical power input to the thruster.

Jet power may be expressed as $p_j = mu^2/2$ and thrust as $f = mu$. Thus $p_j = fu/2$ and $f = 2\ p_j/u$. Typical values are:

$$p_j = 10^6 \text{ watts}$$
$$u = 24\ 500 \text{ m/sec (Isp = 2500)}$$

Then thrust is 81.6 newtons = 18.4 lb. Assuming a 50% thruster efficiency and hence 2 megawatts electric power, and power generation system mass of 20 kg/kw_e (44 lb/kw_e), the power generation mass is 40 000 kg (88,000 lb) and the upper limit on acceleration (f/m) is 0.002 m/sec^2, about 2×10^{-4} g's. The Earth's gravitational acceleration is 9.81 m/sec^2 (32.18 ft/sec^2) at sea level and 0.22 m/sec^2 at geosynchronous orbit . . . always at least 100 times the acceleration available from the electric propulsion system.

Under these conditions the effect of electric propulsion is to slowly perturb the space vehicle orbit from its starting condition to some end condition. Thrust is generally applied continuously. For high thrust systems, instantaneous velocity changes (impulsive maneuvers) with unpowered coasting orbits between maneuvers provide a good approximation for flight mechanics analysis. In contrast, for low thrust maneuvers, it is a good approximation to consider the thrusting force as a vanishingly small perturbing force on a path represented by relationships for unpowered orbits.

As an example, consider the approximation of the equivalent delta V for increase in altitude of a circular orbit (without plane changes) by electric propulsion.

The energy of a circular orbit is $E = -\mu m/2r$ where μ is the gravitational potential and r is orbit radius. If the orbit is perturbed,

$$\frac{dE}{dt} = m\frac{\mu}{2r^2}\frac{dr}{dt}$$

By conservation of energy, the rate of change of energy is equal to the rate of doing work by the perturbing force. The latter is

$$\frac{dE}{dt} = F\frac{ds}{dt} = Fv = \frac{dm}{dt}uv$$

where dm/dt is mass flow rate of the thruster, u is effective jet velocity, and v is orbit velocity. Equating energy rates,

$$\frac{dm}{dt}uv = m\frac{\mu}{2r^2}\frac{dr}{dt}$$

Cancelling dt's provides a differential equation. Invoking now the circular orbit condition (in effect saying that the perturbing thrust is vanishingly small),

$$v^2 = \frac{\mu}{r}\ ;\quad 2vdv = -\frac{\mu}{r^2}dr$$

solving for dr, $dr = -2vr^2dv/\mu$. Substituting in the above energy equation,

$$dmuv = m\frac{\mu}{2r^2}\left(\frac{-2vr^2dv}{\mu}\right)$$

Simplifying,

$$\frac{dm}{m} = -\frac{dv}{u}$$

when integrated, the result is the classical rocket equation with ΔV replaced by the difference in orbit velocities. Therefore, the equivalent delta V for a low thrust coplanar ascent is approximately the difference in orbit velocities. As a numerical example, consider orbit transfer between a 500 km (270 n mi) orbit and a geosynchronous orbit, without plane change. The high thrust delta V (impulsive) is a Hohmann transfer at 3 817 m/sec (12,523 ft/sec) whereas the low-thrust delta V is 4 538 m/sec (14,888 ft/sec).

This type of analysis was extended in the FSTSA study to find the appropriate ΔV to synchronous orbit, with a plane change, for low-thrust propulsion. One suggested steering law employs a circular coplanar ascent, for which the ΔV was shown to be approximately $V_1 - V_2$ (the difference in orbit velocities), followed by a continuous thrusting plane change with 90^o yaw or pitch angles alternating every half orbit. The ΔV is easily shown to be $\pi/2\ \gamma\ V_2$ where γ is the plane change in radians. For an initial orbit at 500 km altitude (270 n mi) (r = 6 878 km) and 28-1/2 degrees inclination, V_1 is 7 612 m/sec, V_2 is 3 075 m/sec, (19,323 n mi) and the total delta V is 4 538 +

2 403 = 6 941 m/sec (21,300 ft/sec). Alternative more efficient values on the order of 6 500 m/sec (21,300 ft/sec) have also been suggested with no specific steering law specified.

It should be recognized that if discontinuous thrusting and a very long trip time are acceptable, delta V's as low as about 4 200 m/sec (13,780 ft/sec) could be achieved by approximating the impulsive maneuvers with many short thrusting periods. Trip times would be at least 6 to 8 times longer than with continuous thrusting and therefore generally unacceptable.

There are excellent reasons to believe that the continuous thrusting law stated above is inefficient. The present investigation has indicated a better one of the form

$$\beta = \tan^{-1}(\alpha\, r^{3/2} \sin\theta)$$

where θ is angle along the orbit path, measured from a point 90° from the node, r is instantaneous orbit radius, α is a constant selected to give desired total plane change, and β is yaw thrusting angle. (Note that if θ is measured from the node, the law is $\beta = \tan^{-1}(\alpha\, r^{3/2} \cos\theta)$). Pitch thrusting angle is always zero. Note that this law puts most of the plane change at higher altitude.

This steering law results in quite large yaw angles at higher altitudes. Typical yaw profiles are shown in Figure 1.2-1. Delta V versus plane change is shown in Figure 1.2-2. For the reference case described above, ideal ΔV is approximately 5 775 m/sec (18,950 ft/sec). Presently for FSTSA analyses, an orbit transfer ΔV requirement of 6 000 m/sec (19,685 ft/sec) is being used, allowing 2% for flight performance reserves and 1.9% for thrust vector losses. Self-powered operation for power satellite transfer includes an additional 10% for thrust vector losses associated with gravity gradient torques.

It is cautioned that this steering law may not be practical for some vehicles in view of the large yaw angle requirement.

A comparison with optimal transfers for relatively high starting altitudes indicates this steering law to be near optimal (Figure 1.2-3). With low starting altitudes, there should be some advantage to allowing the orbit to pass through an elliptic phase, a condition excluded by assumption in the above analysis. Low starting altitude optimal data were not available for comparison.

Parameterization of elec propulsion systems is quite complex even for a fixed mission delta V, since nonlinear relationships among several parameters are involved:

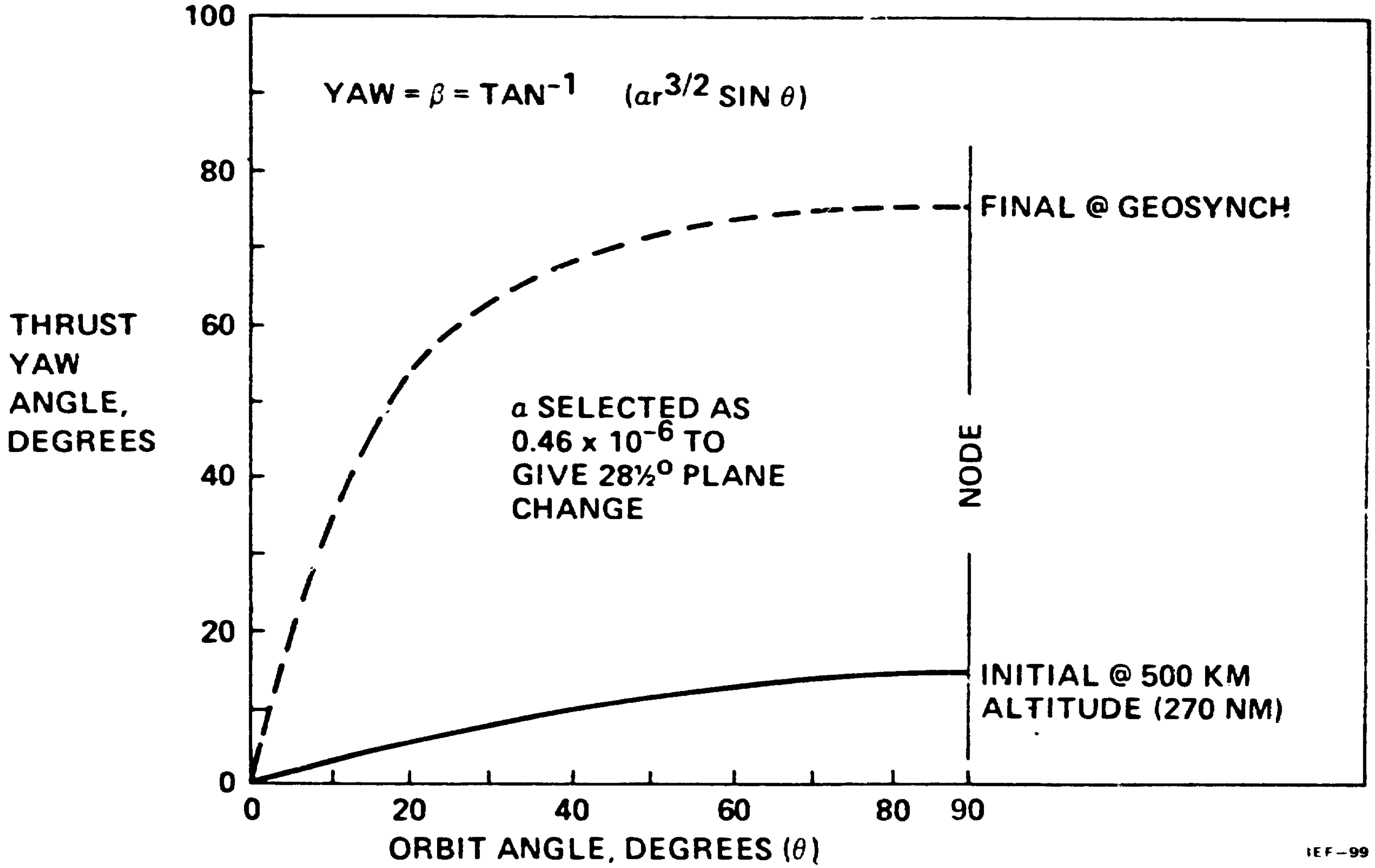


Figure: 1.2-1 Low Thrust Steering Law

c.2

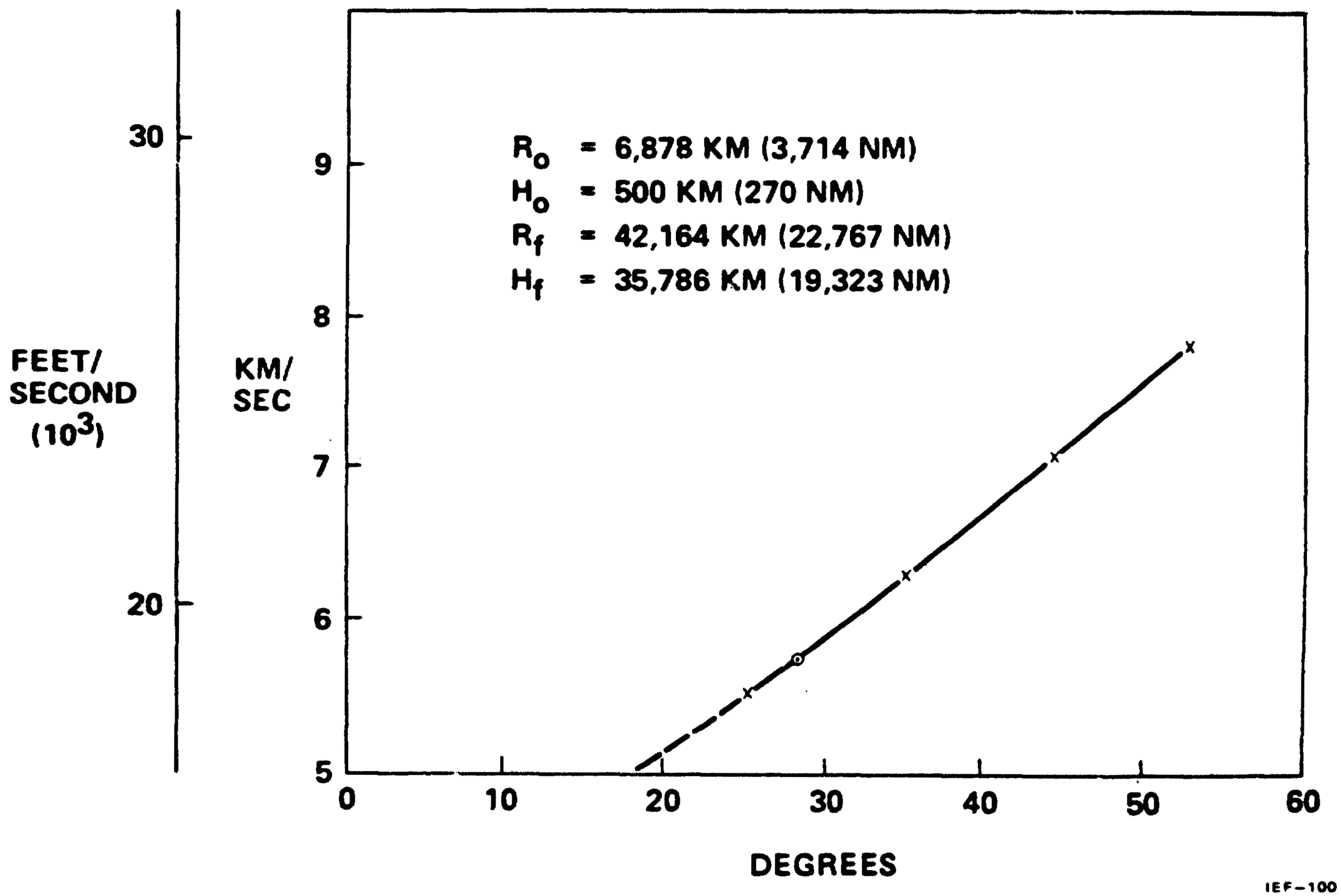
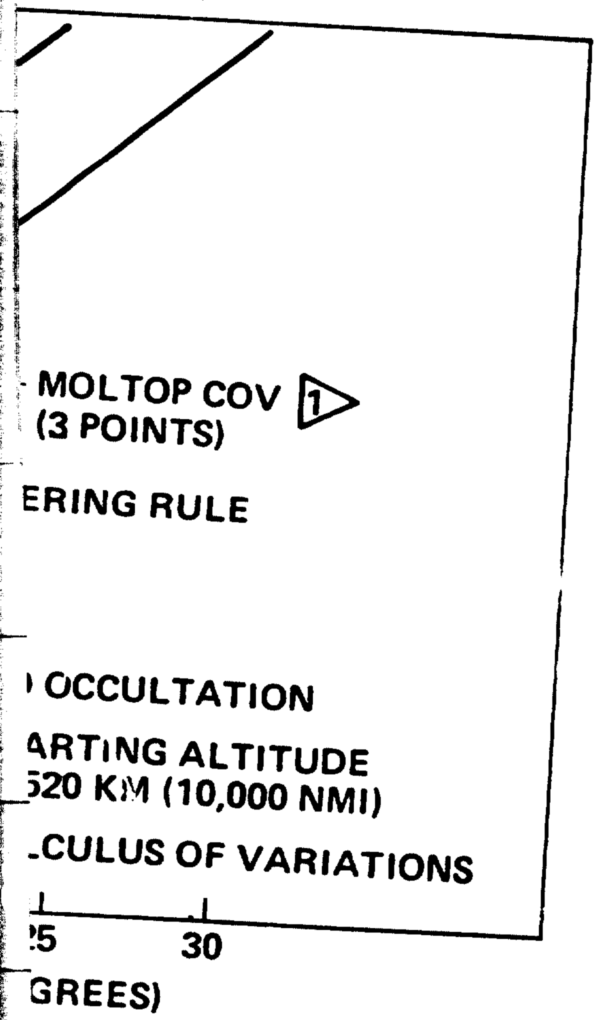


Figure 1.2-2. Delta V Required for Plane Change



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ering Rules

- Specific impulse
- Thruster/power processor efficiency
- Power generation specific power
- Relationship of delivery and return payloads
- Trip time
- Size of the power generation system

A provisional approach follows; improved techniques are in work.

1. Stuhlinger and others have shown that the optimum specific impulse for power limited systems is approximated by

$$u = \sqrt{\alpha t}$$

where α is specific power of the propulsion system in watts of jet power/Kg, t is total (round) trip time in seconds, and u is jet velocity in m/sec. Ghosh and Huson (AIAA 69-275) give an equation:

$$u^2 \left[\frac{(\mu - 1)u}{\Delta V} - \frac{1}{2} \right] = \alpha t$$

for optimum specific impulse; the factor in brackets is roughly unity for cases of practical interest so that this equation is not markedly different than the Stuhlinger equation. Neither of these equations consider variation of thruster efficiency with Isp or differences in delivery/return payloads. The Stuhlinger equation is plotted in Figure 1.2-4.

The optimum Isp plot serves as a guide to Isp selection. Valid reasons often exist not to use the optimum; the following data do not depend on the selected Isp being optimum.

The factor μ, mass ratio of initial mass to final mass, is given by

$$\mu = \exp\left(\frac{\Delta V}{u}\right)$$

where ΔV is ideal ΔV and u is jet velocity, equal to g Isp where g is 9.8066 m/sec^2 (32.174 (ft/sec^2). A typical mission for an electric OTV is a trip from a 500 km (270 n mi) orbit at 28-1/2 degree inclination to a 35 786 km (19,323 n mi) geosynchronous orbit at 0° inclination. The required ideal delta V is approximately 6 000 m/sec (19,680 ft/sec) using a plane change thrusting law described previously. Figure 1.2-5 shows μ versus propulsion Isp for this delta V.

Ghosh and Huson also provide a time equation, it can easily be shown to be:

$$t = \frac{(\mu - 1)u^2}{2} \zeta \frac{M_1}{M_0}$$

where ζ is specific mass in KG/watt of jet power and M_1/M_0 is ratio of payload mass plus propulsive stage mass to propulsive stage along mass. This equation is plotted in Figure 1.2-6 with $\zeta M_1/M_0$ expressed in kg/kw.

The following example is provided: Suppose an ascent payload of 38 950 kg (85,900 lb) and a return payload of 7 600 kg (16,800 lb) are desired. Further suppose that the specific mass of the propulsion system is 60 kg/kw and that a round trip time of 160 days is desired. As a preliminary estimate, ascent and return trip times of 110 days and 50 days may be estimated.

From Figure 2.1-4, Isp = 1550 sec

From Figure 1.2-5, μ = 1.47

From Figure 1.2-6, $\zeta M_1/M_0$ for return trip is about 75.

The propulsion system mass is $$M_0 = \left(\frac{\zeta}{\zeta \frac{M_1}{M_0} - \zeta} \right) M_1$$

$$= \frac{60}{75-60}(7600) = 30\ 400 \text{ kg } (67{,}000 \text{ lb})$$

The propellant for return is $(\mu - 1)\, m_1$

= .47 (30 400 + 7600) = 17 860 kg (39,400 lb)

The total ascent payload includes return propellant and is therefore equal to 56 810 kg (125,243 lb). The value for ascent $\zeta M_1/M_0$ is 172 and ascent trip time is 110 days from Figure 1.2-6. The total initial mass includes ascent propellant and is 130 950 kg (288,500 lb).

1.2.3.2 Nuclear-Electric Tug Concept

During the July NASA/Boeing working session, discussions were held with Mr. John Stearns of JPL on the subject of nuclear-electric tugs (NET's). Sizing and performance estimating data were obtained as reflected in the analyses described below.

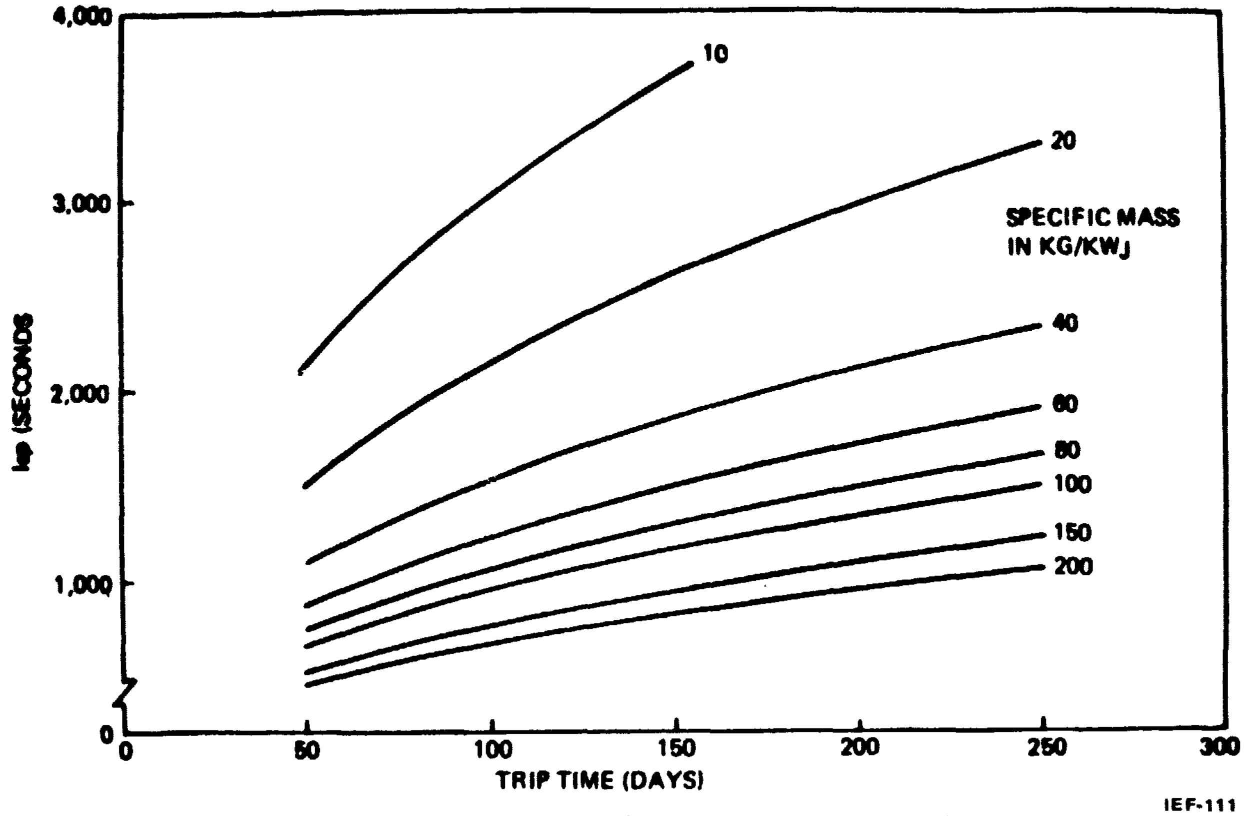

Figure 1.2-4. Optimum I_{sp} (Stuhlinger Approximation)

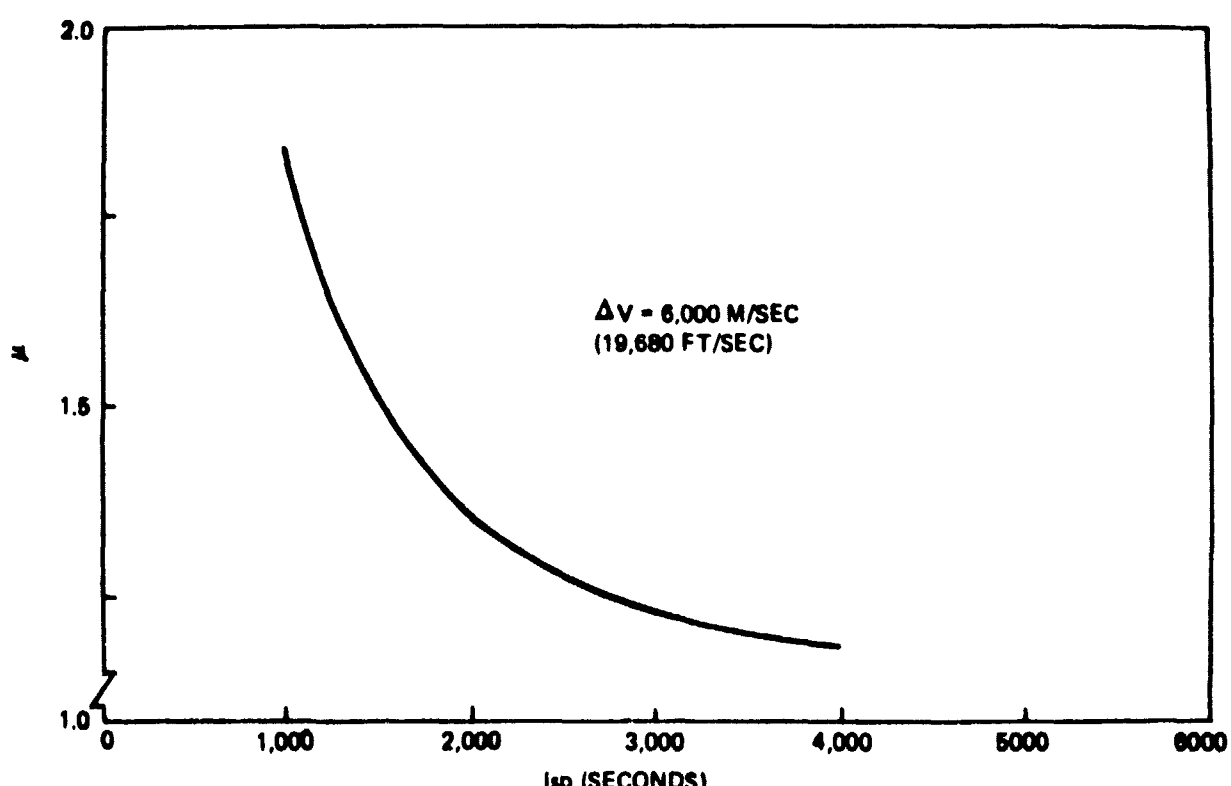

Figure 1.2-5. Mass Fraction Required

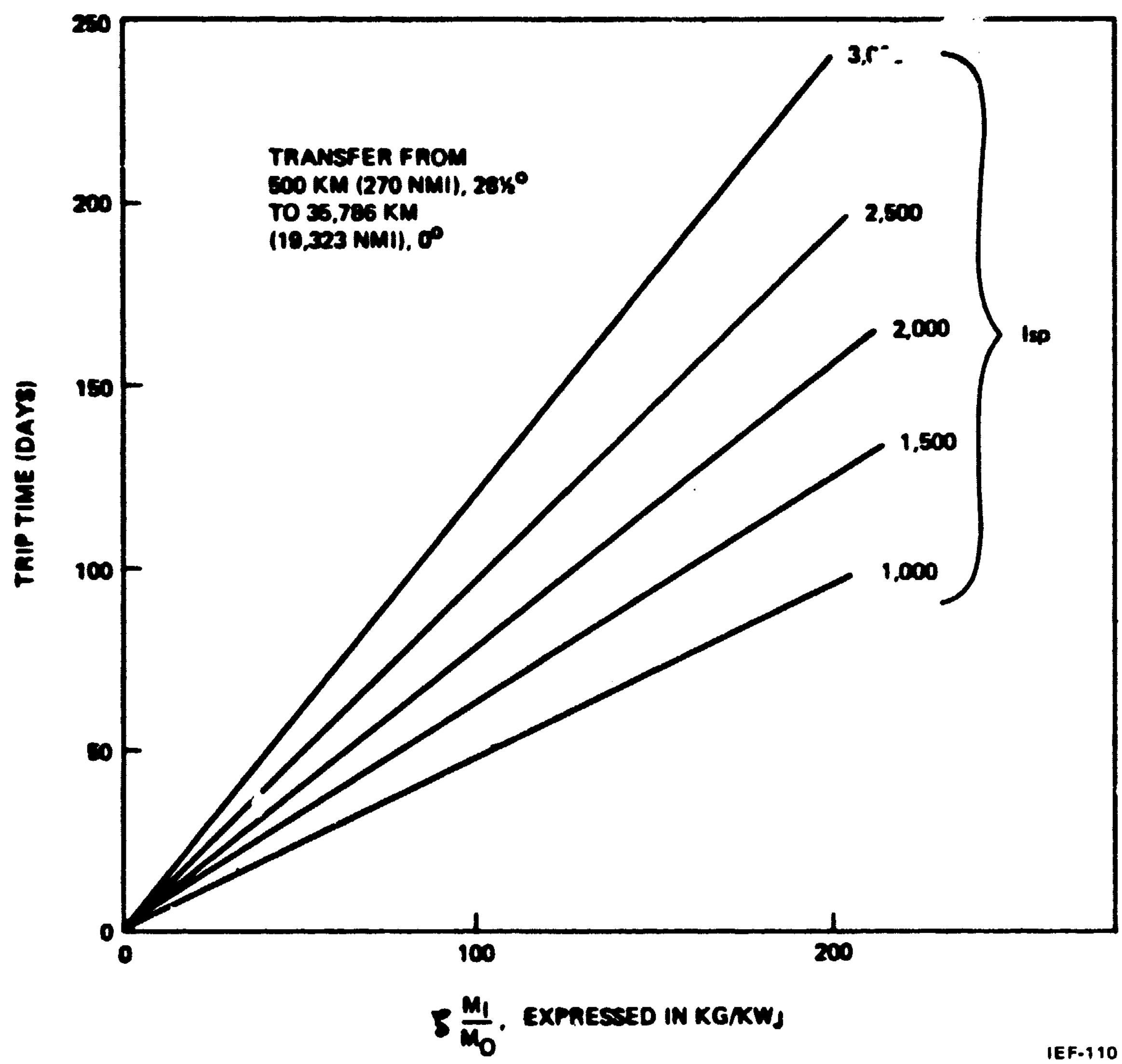

Figure 1.2-6. Low-Thrust Trip Time

Argon MPD Thruster–Experiments at Princeton have used argon magneto plasmadynamic (MPD) devices to create plasmas for plasma physics experiments. Analyses of similar devices used as thrusters indicates that desirable performance characteristics may be obtainable. Table 1.2-1 indicates representative target values:

These performance figures have not been confirmed by test, but are judged to be reasonable when associated with a system like the NET requiring a major development effort.

Reactor Power Generation Systems–The reactor design approach, suggested by Mr. Stearns, employs a high temperature, fast spectrum reactor cooled by heat pipes. Either thermionic or high temperature Brayton conversion systems are potentially practical.

Reactor core assumptions are given in Table 1.2-2.

Power and efficiency budgets are given in Table 1.2-3 and 1.2-4 for the thermionic and Brayton systems.

The thermal power and efficiency assumptions were used to derive the reactor parametrics shown in Figures 1.2-7 and 1.2-8. Figure 1.2-9 illustrates the fuel form concept for the thermionic system including heat pipe. Dimensions shown are representative.

The thermionic system requires only a single active loop, for heat rejection. The cycle concept is shown in Figure 1.2-10. Also shown is the Brayton cycle schematic with a primary liquid metal loop coupled to the typical closed Brayton cycle. Radiator area requirements are shown in Figure 1.2-11. It is anticipated that the Brayton cycle system will require emergency heat removal radiator in the primary loop. The emergency heat removal is required only to handle after-heat and the radiator could be quite small.

Gamma shield dimensions are given in Figure 1.2-12. For the Brayton system the gamma shield encloses the entire primary loop to minimize radiation from neutron activation of the primary loop fluid. Figure 1.2-13 shows outer gamma shield specific mass assuming 100 g/cm^2 (205 lb/ft^2) shielding. The gamma shielding is intended to allow unimpeded manned operation around the NET with the reactor shut down. The shielding allowance is a rough estimate; the manned operation assumption was arbitrary but appears reasonable for the intended mission use. Very little neutron shielding is provided; the unshielded standoff distance for manned operations with the reactor at full power will be on the order of 10 km (5.4 n mi).

Table 1.2-1. Performance of Hypothetical MPD Thruster

Specific impulse – 2,500 seconds (jet velocity is 24,500 m/sec; 80,400 ft/sec)

Efficiency – 45%

Specific mass – 0.1 kg/kw (0.22 lb/kw)

Size – 10^{-5} m^3/kw (3.5×10^{-4} ft^3/kw) for 1 megawatt and larger

Power – 200 volts DC

Propellant feed pressure – 1 atm or less

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Table 1.2-2. Reactor Performance Assumptions

Thermal power density	25 mw/m^3	(0.708 mw/ft^3)
Mass density	7,500 kg/m^3	(470 lb/ft^3)
Reactor life	30,000 hrs	
Maximum heat pipe length	1.2 m	(3.94 ft)
Heat pipe diameter	1 cm	(0.4 in.)
Heat pipe spacing (hex pattern)	2 cm	(0.8 in.)
Heat pipe heat exchanger Heat transfer area	91 m^2/m^3	(27.74 ft^2/ft^3)
Heat pipe temperature	1,600 K	(2,420°F)
Neutron reflector thickness	20 cm	(7.9 in.)

IEF-84

Table 1.2-3. Thermionic Power Budget

	Efficiency	Power in megawatts		
Thermal power		15.6	39	78
to	.15			
DC power		2.34	5.85	11.7
to	.95			
conditioned power		2.22	5.5	11
to	.45			
beam power		1	2.5	5
Overall efficiency = 6.4%				

IEF-85

Table 1.2-4. High Temperature Brayton Power Budget

	Efficiency	Power in megawatts		
Thermal power		7.27	18.2	36.4
to	.35			
shaft power		2.55	6.37	12.73
to	.97			
AC power		2.47	6.18	12.35
to	.90			
conditioned power		2.22	5.5	11.1
to	.45			
beam power		1	2.5	5

Overall efficiency = 13.7%

IEF-86

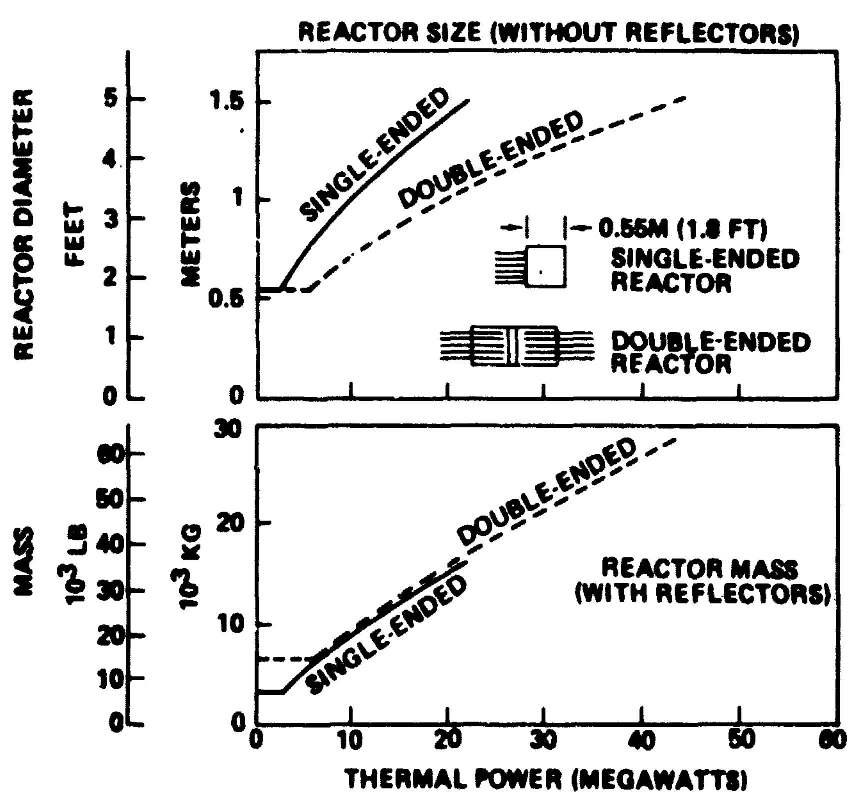

Figure 1.2-7. Reactor Sizing

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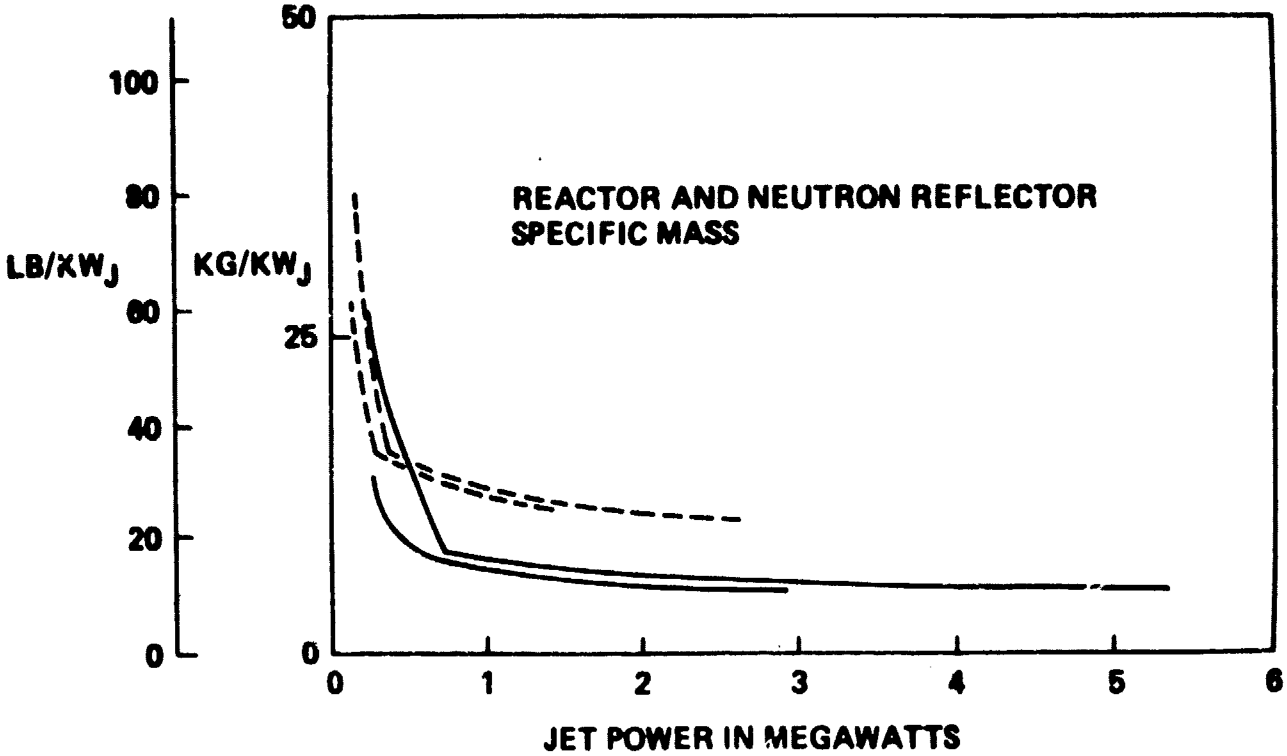


Figure 1.2-8. Reactor Performance IEF-88

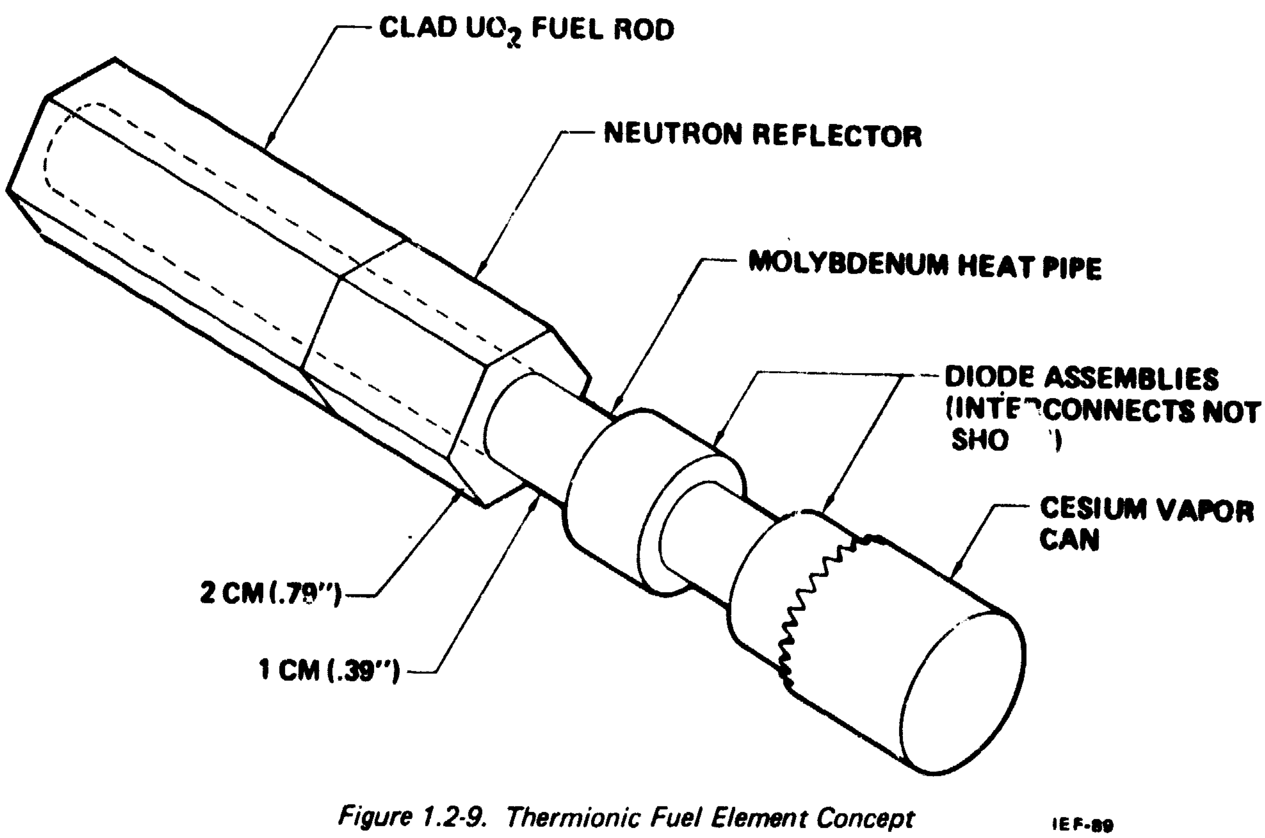


Figure 1.2-9. Thermionic Fuel Element Concept IEF-89

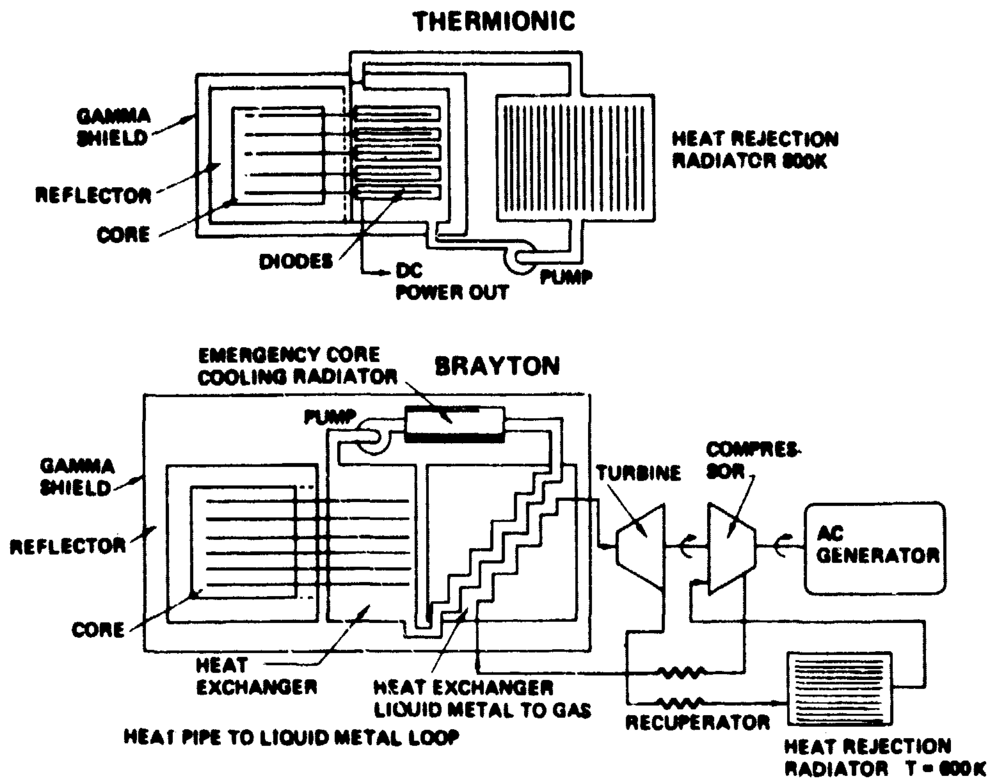


Figure 1.2-10. Power Generation Cycles IEF-90

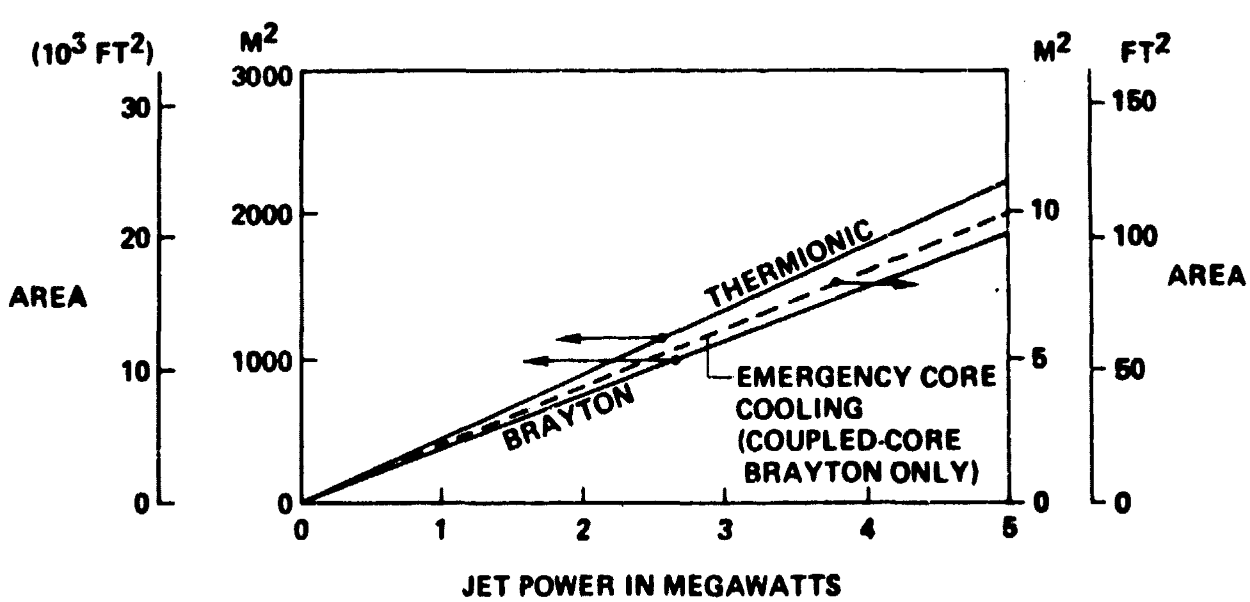
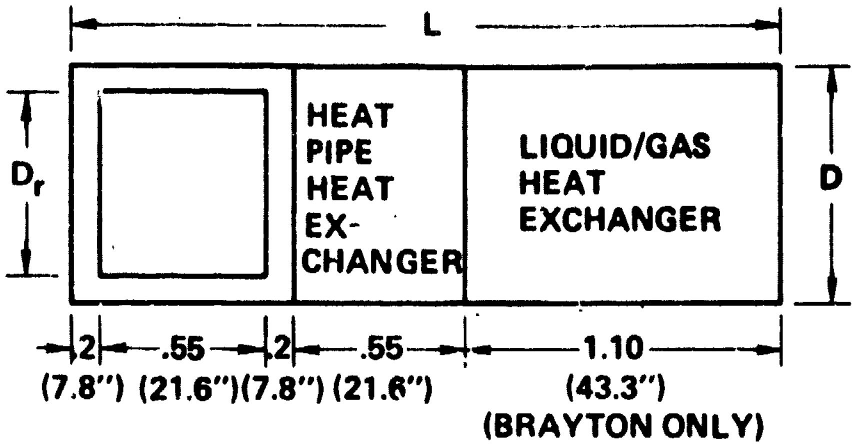


Figure 1.2-11. Radiator Area Requirement IEF-91



GAMMA SHIELD LENGTH

THERMIONIC SINGLE-ENDED	1.5M (4.92 FT)
THERMIONIC DOUBLE-ENDED	2.8M (9.19 FT)
BRAYTON SINGLE-ENDED	2.6M (8.53 FT)
BRAYTON DOUBLE-ENDED	5.0M (16.4 FT)

GAMMA SHIELD DIAMETER = D_r + 0.4M (D_r + 1.3 FT)

Figure 1.2-12. Gamma Shield Dimensions IEF-92

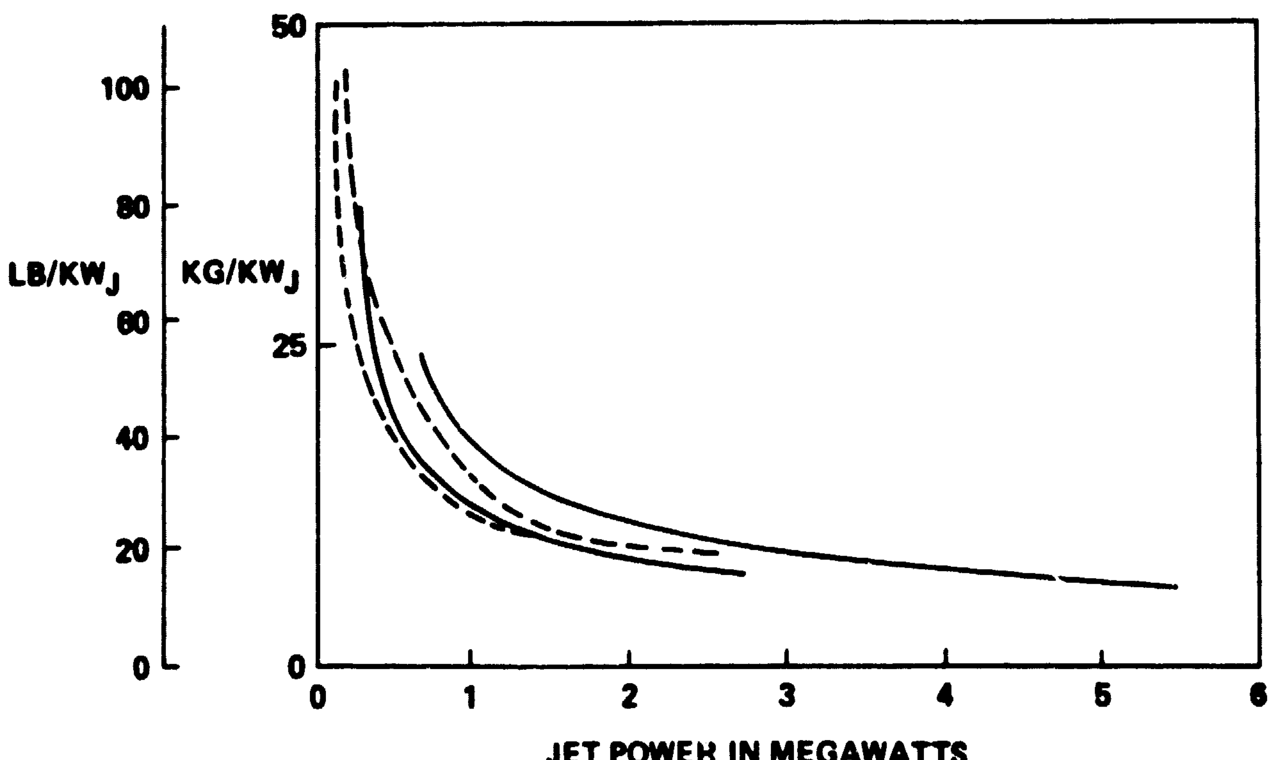


Figure 1.2-13. Outer Gamma Shield Specific Mass

Specific mass estimates for other items are given in Table 1.2-5 and 1.2-6. The Brayton power processing mass is greater than that for thermionics because AC/DC conversion is required. The thermionics radiator is estimated heavier per unit area because it includes a complete liquid loop; the Brayton gas radiator includes gas inventory and containers.

A mass growth allowance of 25% was applied for connecting structure, auxiliary propulsion, avionics, controls, and unidentified items. The completed specific mass estimates, excluding main propellant tankage, are shown in Figure 1.2-14.

Performance and Design Point Selection—These specific mass estimates were used to develop the performance estimates shown in Figures 1.2-15 and 1.2-16. Mass of main propellant tankage was estimated as 5% of the main propellant (argon) required. Since all the power generation systems were similar in specific mass, average values were used for the performance calculations.

The geosynchronous space station delivery mission requires delivery of 61 000 kg (135,000 lb). The 2 megawatt (jet power) tug can perform this delivery in about 75 days, or can deliver the entire station as a single payload in about 110 days.

Reactor disposal can be accomplished by sending the NET to solar system escape (no payload). Propellant required for the 2 MW_j NET is 120 000 kg (265,000 lb); this requirement sizes the propellant tank. 2 MW_j with a single-ended reactor and high temperature Brayton cycle was arbitrarily selected for a point design. The double-ended thermionic system is essentially equivalent in performance; a tradeoff beyond the scope of the FSTSA study would be required to make a selection. Design data are summarized in Table 1.2-7.

The point design is shown in Figures 1.2-17 and 1.2-18. A mass estimate for this point design is provided in Table 1.2-8.

1.2.3.4 Solar Thermal Electric Tug Concept

Solar electric tugs (SEPS) have received considerable attention for a variety of missions. The SEPS vehicles studied have generally used solar photovoltaic conversion, but this restricts system operations to altitudes above the intense portions of the van Allen belts because of radiation degradation of the solar cells.

Table 1.2-5. Thermionics Mass Items

	kg/kwj	lb/kwj
Thermionics heat exchanger and generator	1.5	3.3
Heat rejection loop (radiator @ 30 kw_{th}/m^2 and 10 kg/m^2)	4.42	9.74
Power processor	1.5	3.3
Thrusters	0.1	0.2
Total	7.5	16.5

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Table 1.2-6. Brayton Mass Estimate

	kg/kwj	lb/kwj
Primary loop	1.3	2.9
Turbomachines and generators	4.9	10.8
Heat rejection loop (radiator @ 13 kw_{th}/m^2 and 7 kg/m^2)	2.6	5.7
Power processor	3	6.6
Thrusters	.1	.2
Total	11.9	26.2

IEF-95

Table 1.2-7. Net Point Design Data

Reactor thermal power	14,600 kw
Reactor diameter	1.2 m (3.93 ft)
Electric power	4,940 kw_e
Inert mass	67,600 kg (149,000 lb)
Radiator area	750 m^2 (8,073 ft^2)

IEF-96

Table 1.2-8. Nuclear-Electric OTV Mass Estimates

	KG		LB	
STRUCTURES & MECHANISMS		3,100		6,800
BODY STRUCTURE	1,600		3,500	
PROPELLANT TANKS	1,000		2,200	
SECONDARY STRUCTURE	500		1,100	
MAIN PROPULSION		700		1,540
ELECTRIC THRUSTERS	200		440	
PROPELLANT FILL, DRAIN, VENT & FEED	500		1,100	
AUXILIARY PROPULSION		530		1,160
THRUSTERS	165		360	
TANKS	130		290	
PRESSURIZATION & VENT	75		160	
PROPELLANT SYSTEM	160		350	
AVIONICS		260		570
ELECTRICAL POWER		47,000		103,600
REACTOR, REFLECTOR, & CONTROLS	10,200		22,500	
PRIMARY LOOP & HEAT EXCHANGERS	2,600		5,700	
TURBOGENERATOR & RECUPERATOR	9,800		21,600	
OUTER GAMMA SHIELD	11,500		25,360	
MAIN RADIATOR & SUPPORT STRUCTURE	5,250		11,600	
GENERATOR COOLING LOOP	250		550	
POWER PROCESSOR	6,000		13,200	
POWER PROCESSOR COOLING LOOP	250		550	
POWER DISTRIBUTION	200		440	
AUXILIARY POWER (FUEL CELLS & BATTERIES)	950		2,100	
THERMAL CONTROL		1,110		2,450
CONTINGENCY		12,000		26,450
TOTAL DRY MASS		64,700		142,560

	KG	LB
DRY MASS	64,700	142,560
UNUSABLE MAIN PROPELLANT	2,400	5,300
UNUSABLE AUXILIARY PROPELLANT	325	720
ELECTRICAL POWER REACTANTS	450	990
BURNOUT MASS	67,875	149,570
MAIN PROPELLANT* (MAXIMUM CAPACITY)	120,000	264,600
AUXILIARY PROPELLANT	2,950	6,500
GROSS MASS	190,825	420,670

* MAIN PROPELLANT LOAD VARIES WITH MISSION. CAPACITY SELECTED TO ALLOW DISPOSAL OF SPENT VEHICLE TO SOLAR SYSTEM ESCAPE.

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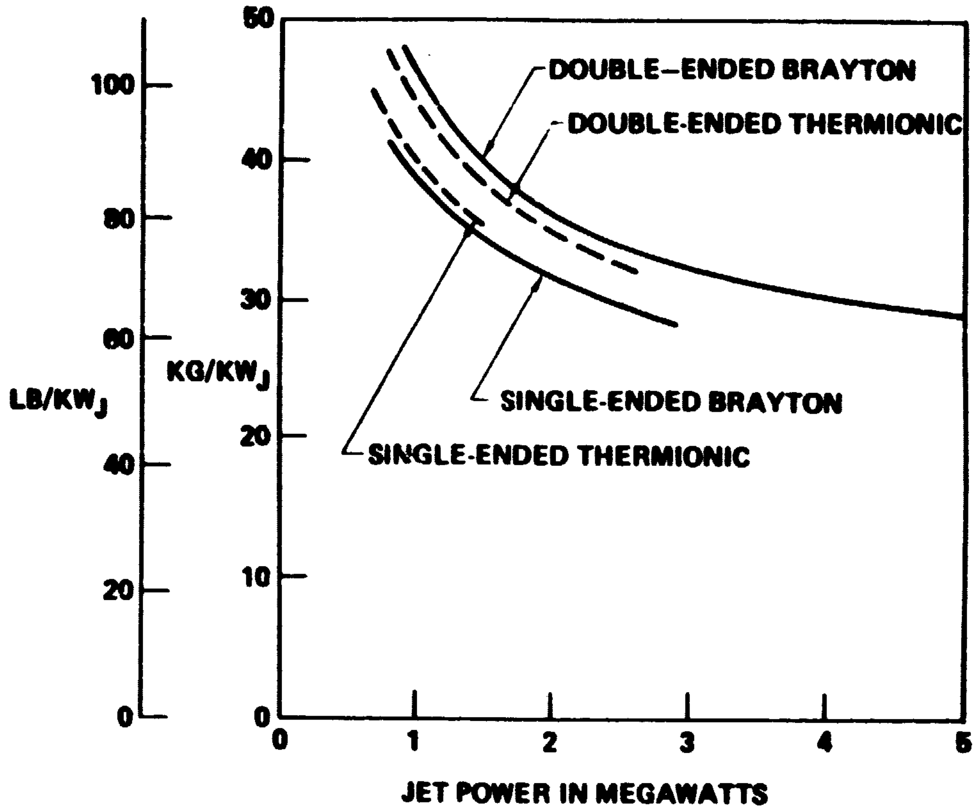

Figure 1.2-14. Overall Specific Mass IEF-97

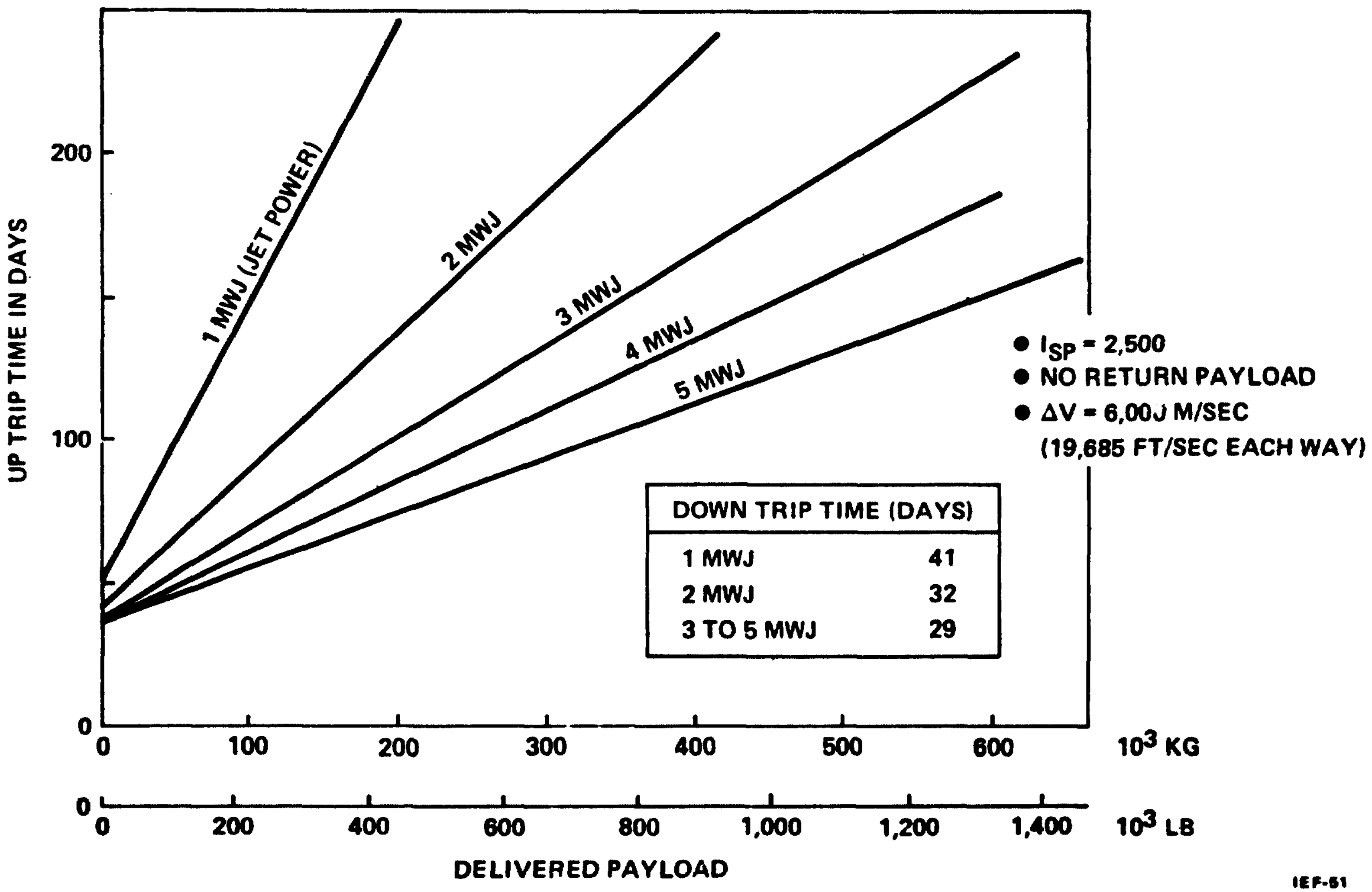

Figure 1.2-15. Nuclear Electric OTV Trip Times

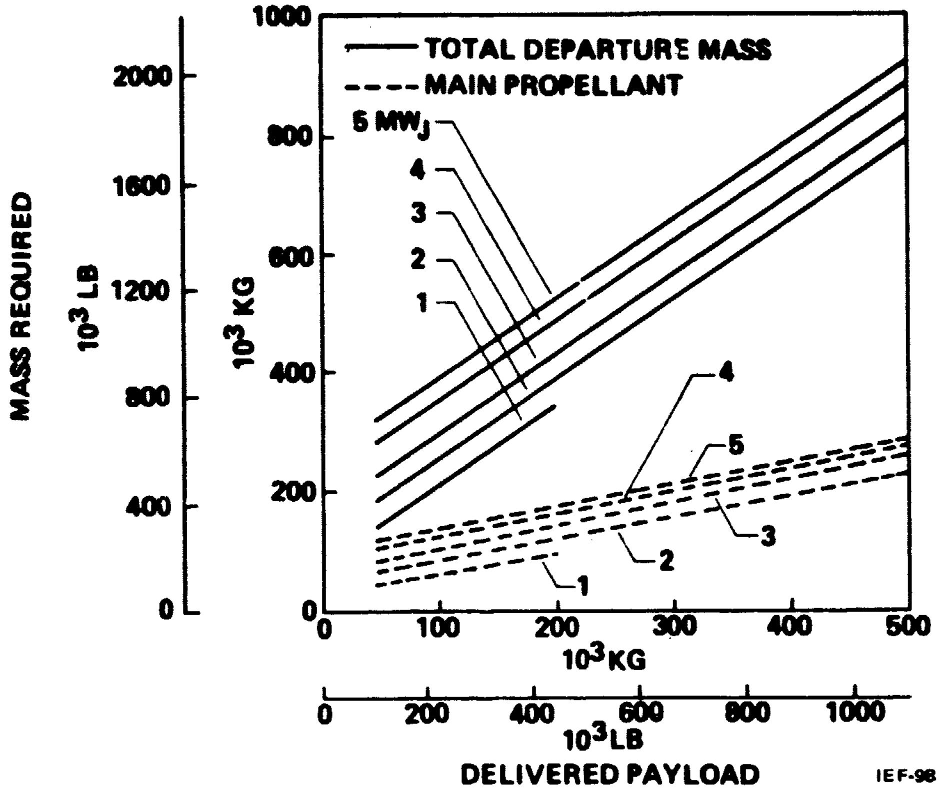

Figure 1.2-16. Nuclear Electric Tug Mass Requirements (I_{sp} = 2,500 Seconds)

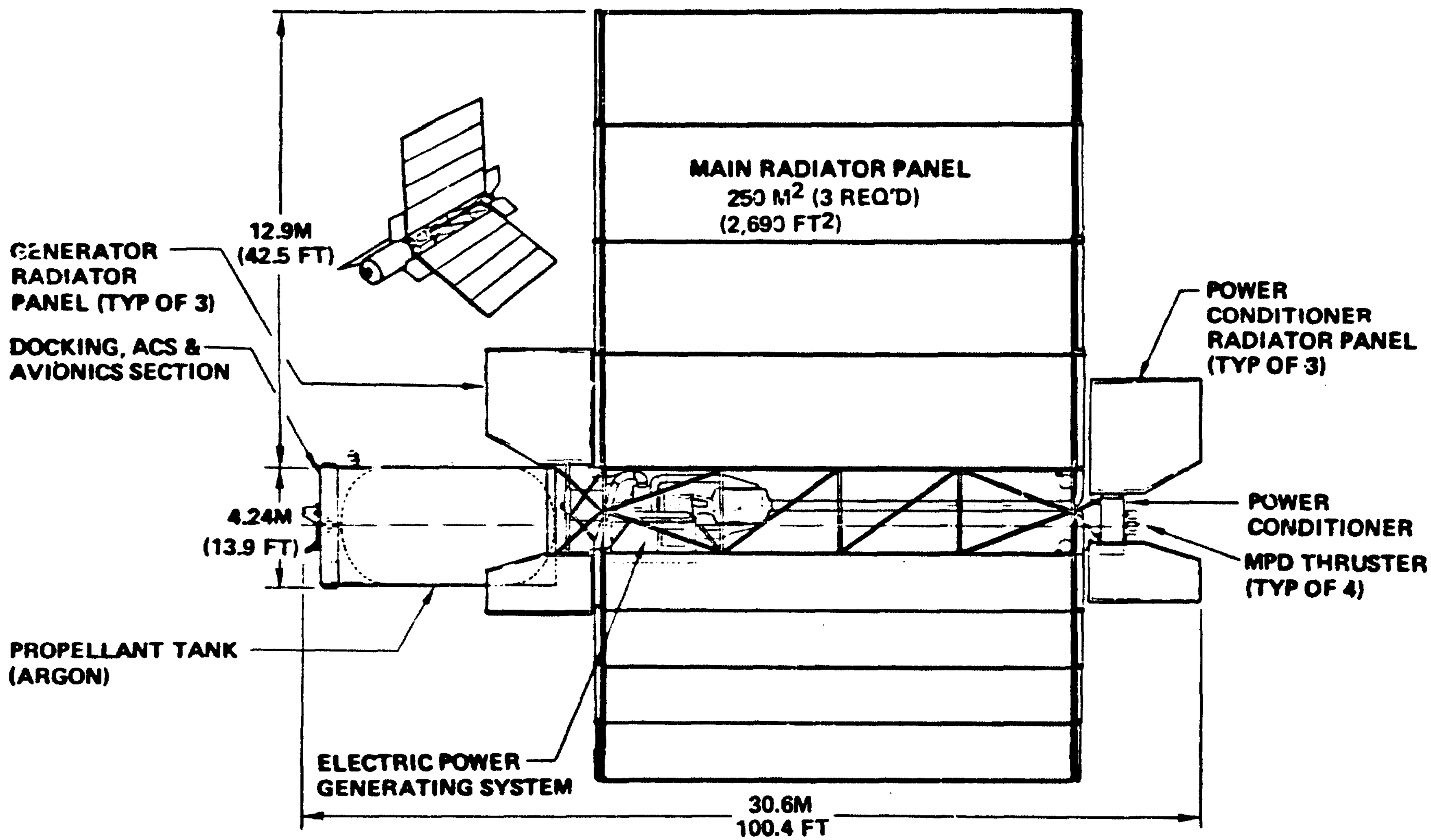

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Figure 1.2-17. Nuclear Electric OTV

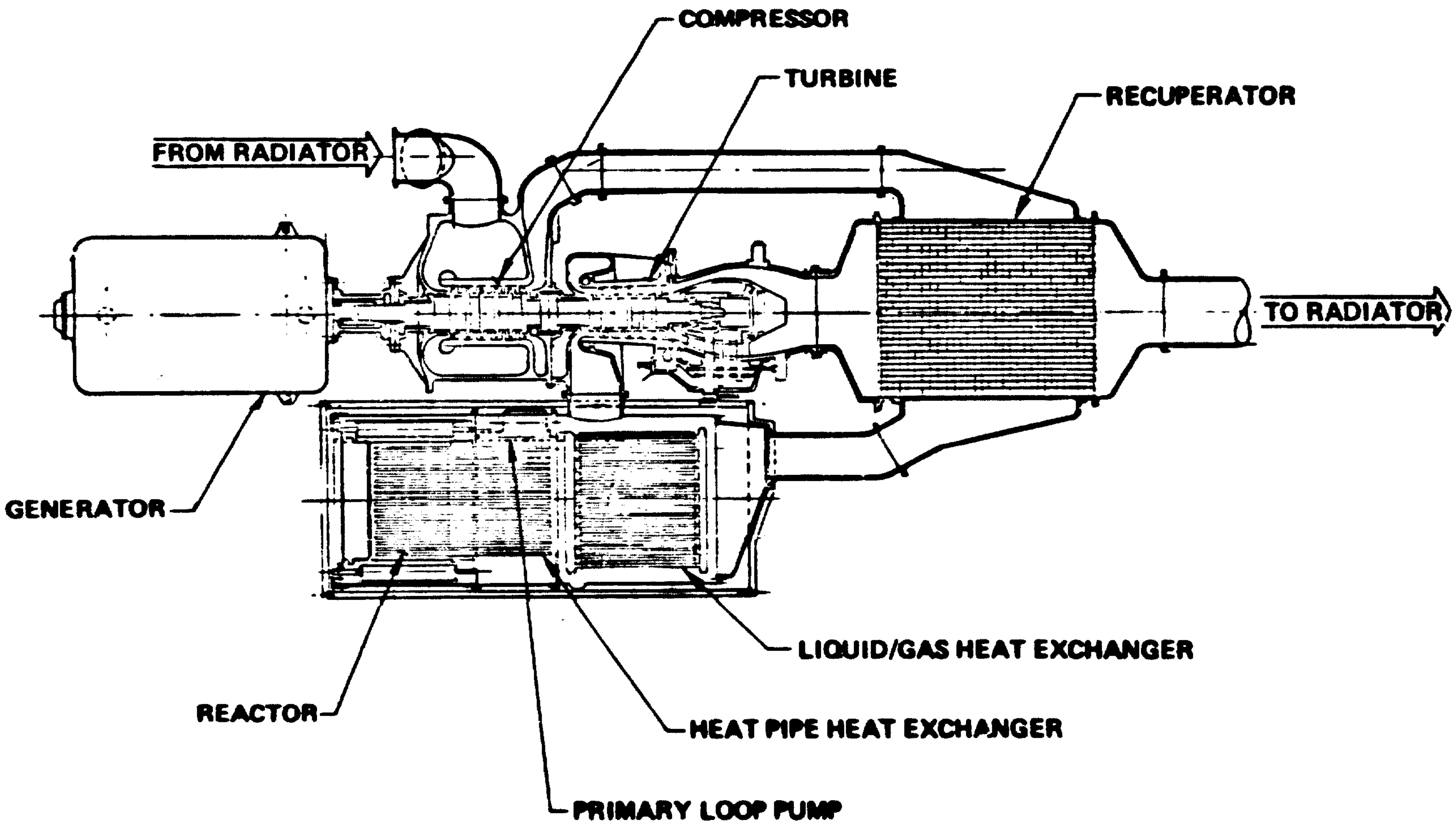

Figure 1.2-18. Nuclear Electric Power Generation System

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This problem could be avoided in principle if a thermal concentrator/heat engine electric tug were used; a "Solar Thermal Electric Propulsion System," STEPS. A STEPS concept was briefly investigated for cargo delivery from low Earth orbits to geosynchronous orbits. Preliminary analyses indicated a jet power of about 1 megawatt might be suitable.

Sizing was based on the efficiency budget estimate of Table 1.2-9 leading to a pair of 60 m (197 ft) dishes for collectors. Table 1.2-10 shows the mass estimate and the estimating basis. Figure 1.2-19 shows the configuration concept.

The use of argon MPD or ion thrusters is assumed, with an Isp of 2500 sec. The argon tank shown is sized for 85 000 kg (187,000 lb) payload up to geosynchronous orbit with zero down. Figure 1.2-20 shows up trip time for the baseline system versus payload, assuming an up trip delta V of 6 000 m/sec (19,680 ft/sec).

The STEPS vehicle shown is believed to be compatible with launch and assembly in orbit by the shuttle. Orbit decay due to air drag will limit operations to altitude of 500 km (270 n mi) and above; even at 500 km, an orbit trim will be required every 10-30 days to avoid excessive decay. Figure 1.2-21 shows the STEPS vehicle adjacent to the Shuttle; Figure 1.2-22 illustrates an assembly operation. At present, it is unclear whether an assembly support vehicle (as illustrated) would be required. The concentrator panels are sized to fit in the Shuttle payload bay. They are molded to paraboloidal sector shape on a precision mold, fabricated from graphite epoxy face sheets and aluminum honeycomb core. The reflective face is aluminum coated. Each panel is adjustable for collimation.

1.2.4 Lunar Transport Vehicles

The lunar transport vehicles (LTV) differ from the orbit transfer vehicles (OTV) in that they require structural and functional accommodations for lunar landing. These include landing legs structure for crew egress, payload support payload handling, and avionics and software. 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). A thrust-to-weight ratio of 0.3 was selected for the parametric analysis. It is assumed that the guidance, navigation and primary communications and power components are in the crew/equipment module rather than the LTV. All the LTV's are considered applicable to the three manned lunar missions (ILSS, OLS, and LSB).

Table 1.2-9. Solar/Thermal Electric Propulsion System (STEPS)

	Power and efficiency budget		
	Efficiency	Power per module (kw)	Total (kw)
Solar flux		3,818	7,636
to	0.85		
energy in cavity		3,245	6,489
to	0.89		
thermal power		2,888	5,776
to	0.40		
shaft power		1,155	2,310
to	0.98		
electric power		1,132	2,264
to	0.95		
conditioned power		1,076	2,151
to	0.45		
jet power		484	968

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Table 1.2-10. Solar/Thermal Electric Propulsion System Mass Estimate, Less Propellant System

Reflectors	5 kg/m^2	28,300 kg (2)
Cavity	0.3 kg/kwt	1,750 kg (2)
Turbogenerator	1.5 kg/kwe	3,400 kg (2 sets)
Radiator	2.5 kw/kwe	5,700 kg
Power conditioner and thrusters	2.6 kg/kwe(c)	5,600 kg
Structure less tanks		2,000 kg
APS		500
Avionics		500
		47,750
Contingency (20%)		9,550 kg
		57,300 kg
	=	60 kg/kwj

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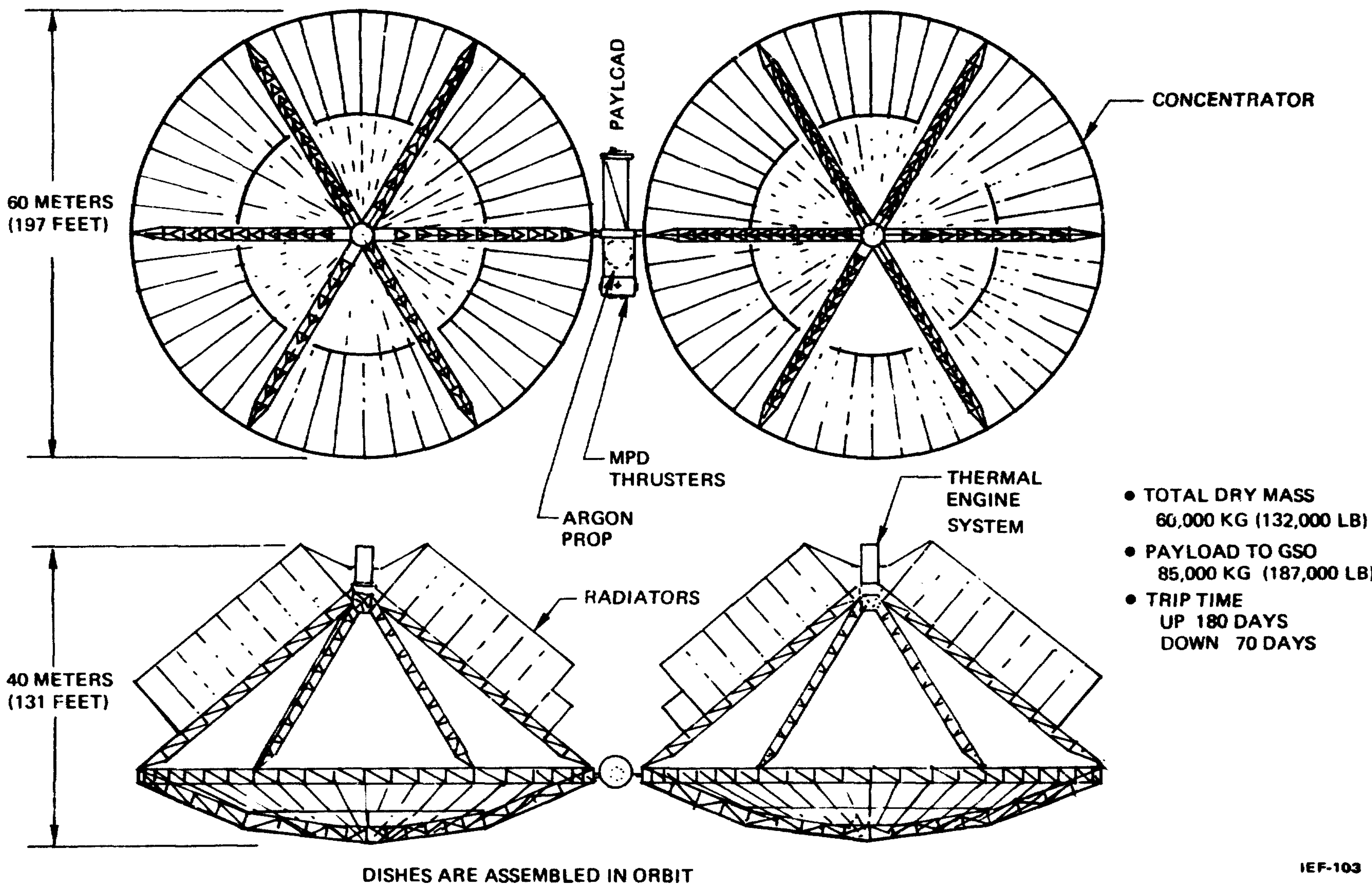


Figure 1.2-19. Two-MWe Solar Thermal Electric Tug

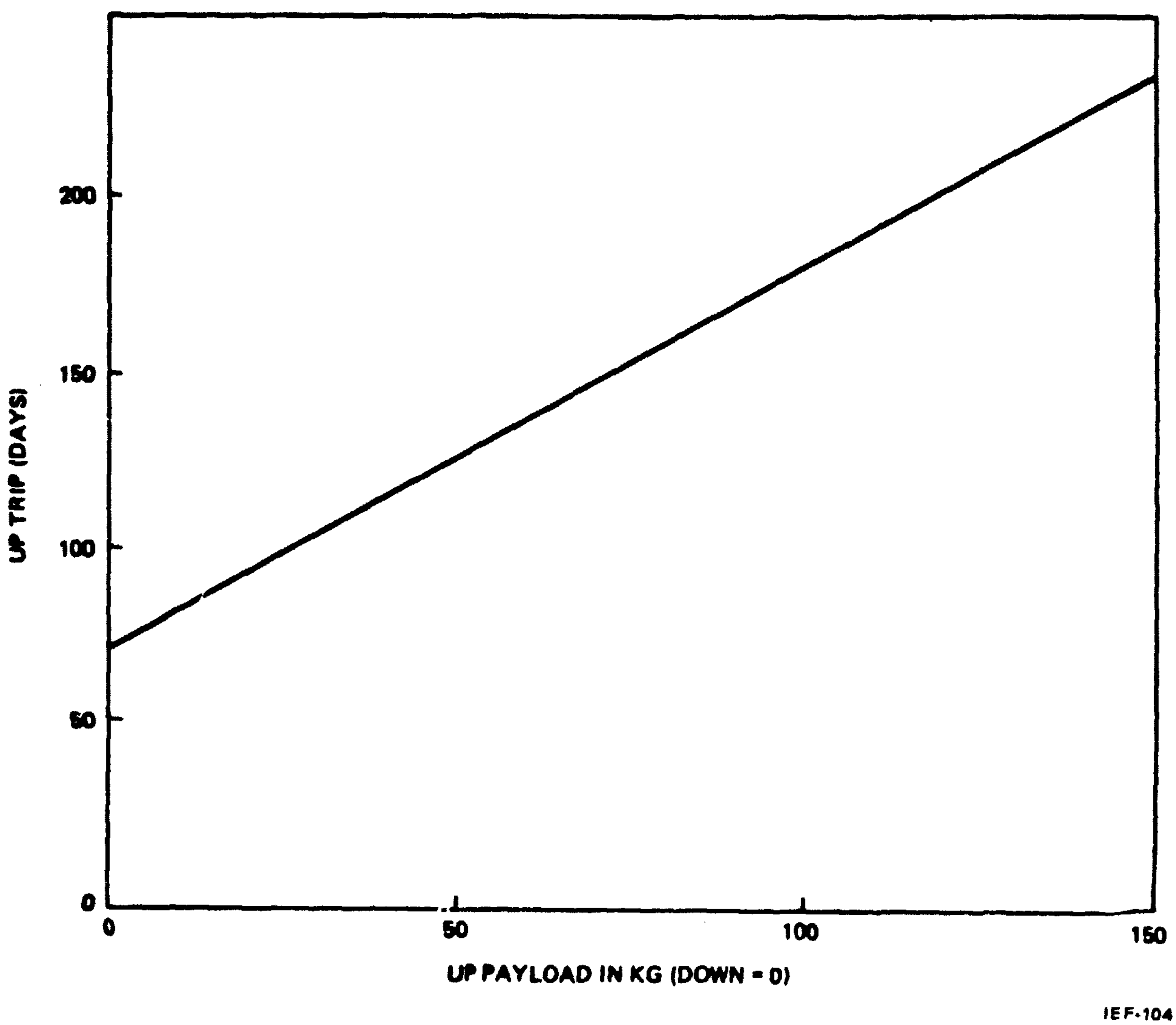
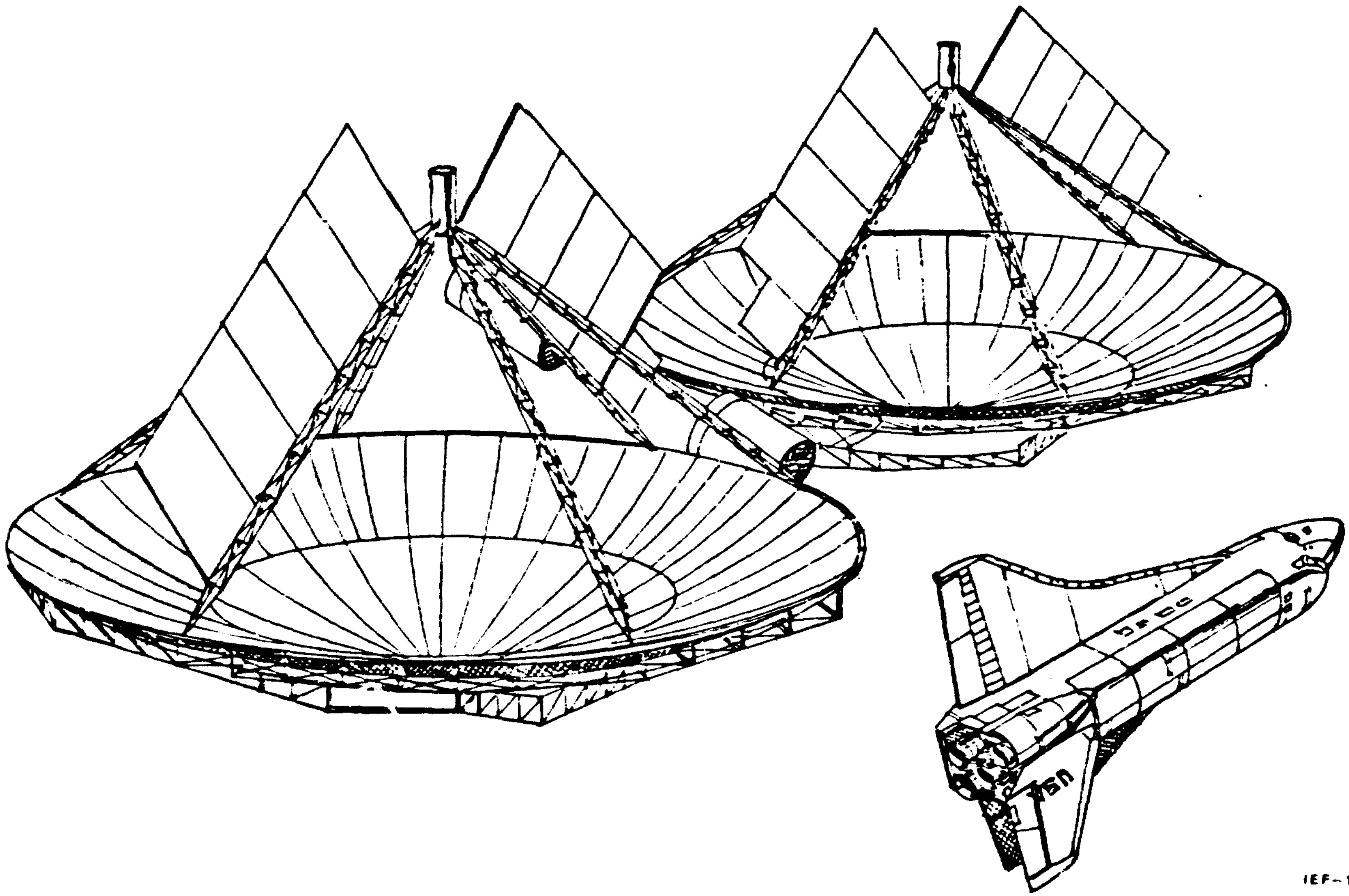


Figure 1.2-20. Baseline Solar Thermal Electric Tug



IEF-105

Figure 1.2-21 STEPS Vehicle Adjacent to Shuttle

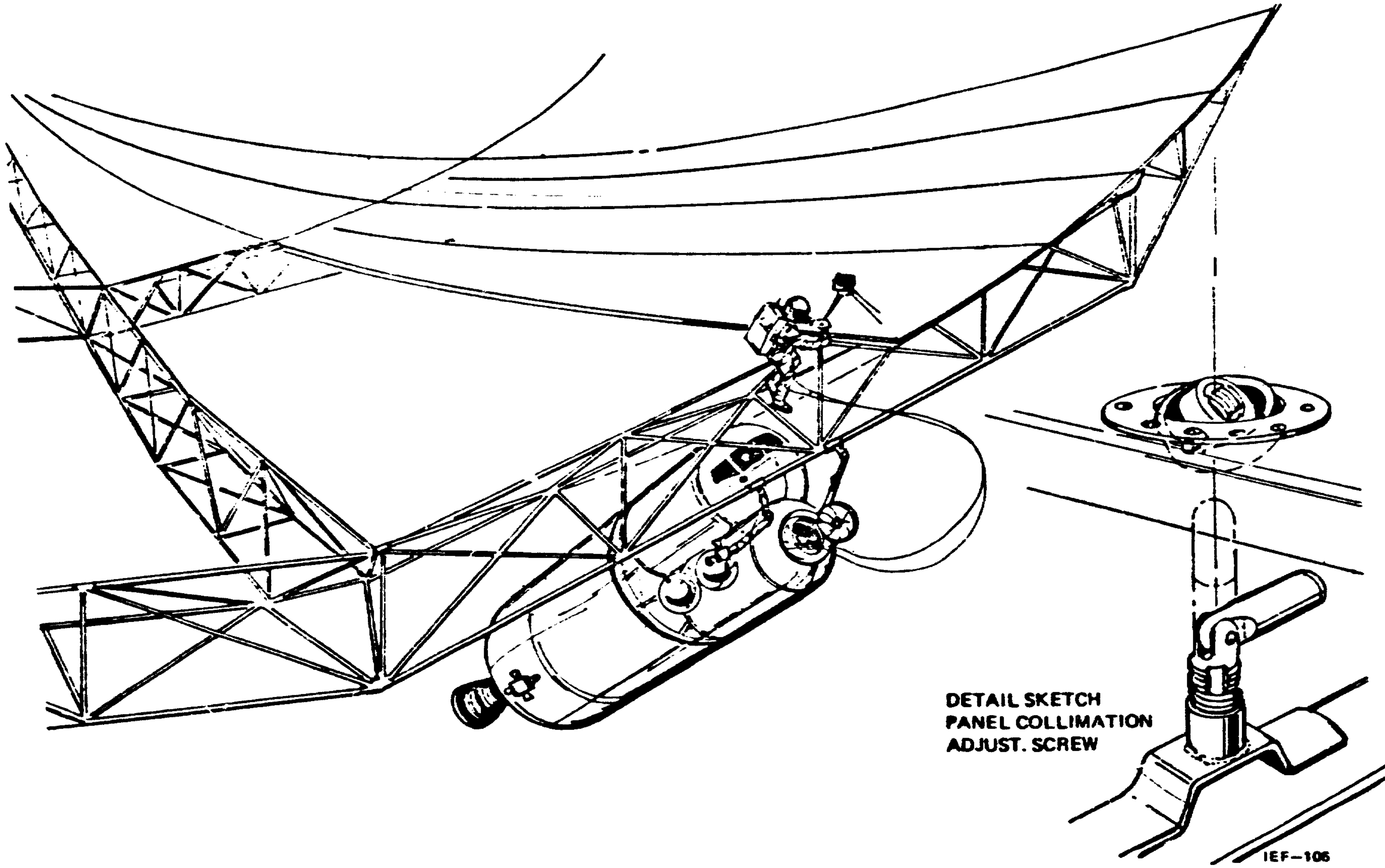


Figure 1.2-22. Assembly Operatic s

1.2.4.1 Shuttle Compatible Single-Stage LO_2/LH_2 LTV Point Design

The configuration inboard profile as drawn (Figure 1.2-23) was sized to be applicable to the OLS mission; performance analysis using the mass properties developed for the point design resulted in tailored sizes for all applicable missions as reported in Volume I. Table 1.2-11 presents a detailed mass properties estimate for the point design as drawn. Mass properties parameter variations about the nominal point resulted in the scaling equation factors shown in Figure 1.2-24. The boiloff mass rate was estimated on the basis of cryogenic tank surface area and typical insulation performance.

Figure 1.2-24 presents the mass properties buildup parametrics. These may be used to develop mass properties summaries for this vehicle type over the applicable range of main impulse propellant loading.

1.2.4.2 Large Diameter Single-Stage LO_2/LH_2 LTV Point Design

The configuration inboard profile as drawn (Figure 1.2-25) was sized to be applicable to the OLS mission; performance analysis using the mass properties developed for the point design resulted in tailored sizes for all applicable missions as reported in Volume I. Table 1.2-12 presents a detailed mass properties estimate for the point design as drawn. Mass properties parameter variations about the nominal point resulted in the scaling equation factors shown in Figure 1.2-26. The boiloff mass rate was estimated on the basis of cryogenic tank surface area and typical insulation performance.

Figure 1.2-26 presents the mass properties buildup parametrics. These may be used to develop mass properties summaries for this vehicle type over the applicable range of main impulse propellant ling.

1.2.4.3 1½ Stage LO_2/LH_2 LTV Point Design

The configuration inboard profile as drawn (Figure 1.2-27) was sized to be applicable to the OLS mission; performance analysis using the mass properties developed for the point design resulted in tailored sizes for all applicable missions as reported in Volume I. Table 1.2-13 presents a detailed mass properties estimate for the point design as drawn. Mass properties parameter variations about the nominal point resulted in the scaling equation factors shown in Figure 1.2-28.

Figure 1.2-28 presents the mass properties buildup parametrics. These may be used to develop mass properties summaries for this vehicle type over the applicable range of main impulse propellant loading.

1.2.4.4 Single-Stage LO_2/MMH LTV Point Design

The configuration inboard profile as drawn (Figure 1.2-29) was sized to be applicable to the OLS mission; performance analysis using the mass properties developed for the point design resulted in tailored sizes for all applicable missions as reported in Volume I. Table 1.2-14 presents a detailed mass properties estimate for the point design as drawn. Mass properties parameter variations about the nominal point resulted in scaling equation factors shown in Figure 1.2-30.

Figure 1.2-30 presents the mass properties buildup parametrics. These may be used to develop mass properties summaries for this vehicle type over the applicable range of main impulse propellant loading.

1.2.4.5 1½ Stage LO_2/MMH LTV Point Design

The configuration inboard profile as drawn (Figure 1.2-31) was sized to be applicable to the OLS mission; performance analysis using the mass properties developed for the point design resulted in tailored sizes for all applicable missions as reported in Volume I. Table 1.2-15 presents a detailed mass properties estimate for the point design as drawn. Mass properties parameter variations about the nominal point resulted in scaling equation factors shown in Figure 1.2-32.

Figure 1.2-32 presents the mass properties buildup parametrics. These may be used to develop mass properties summaries for this vehicle type over the applicable range of main impulse propellant loading.

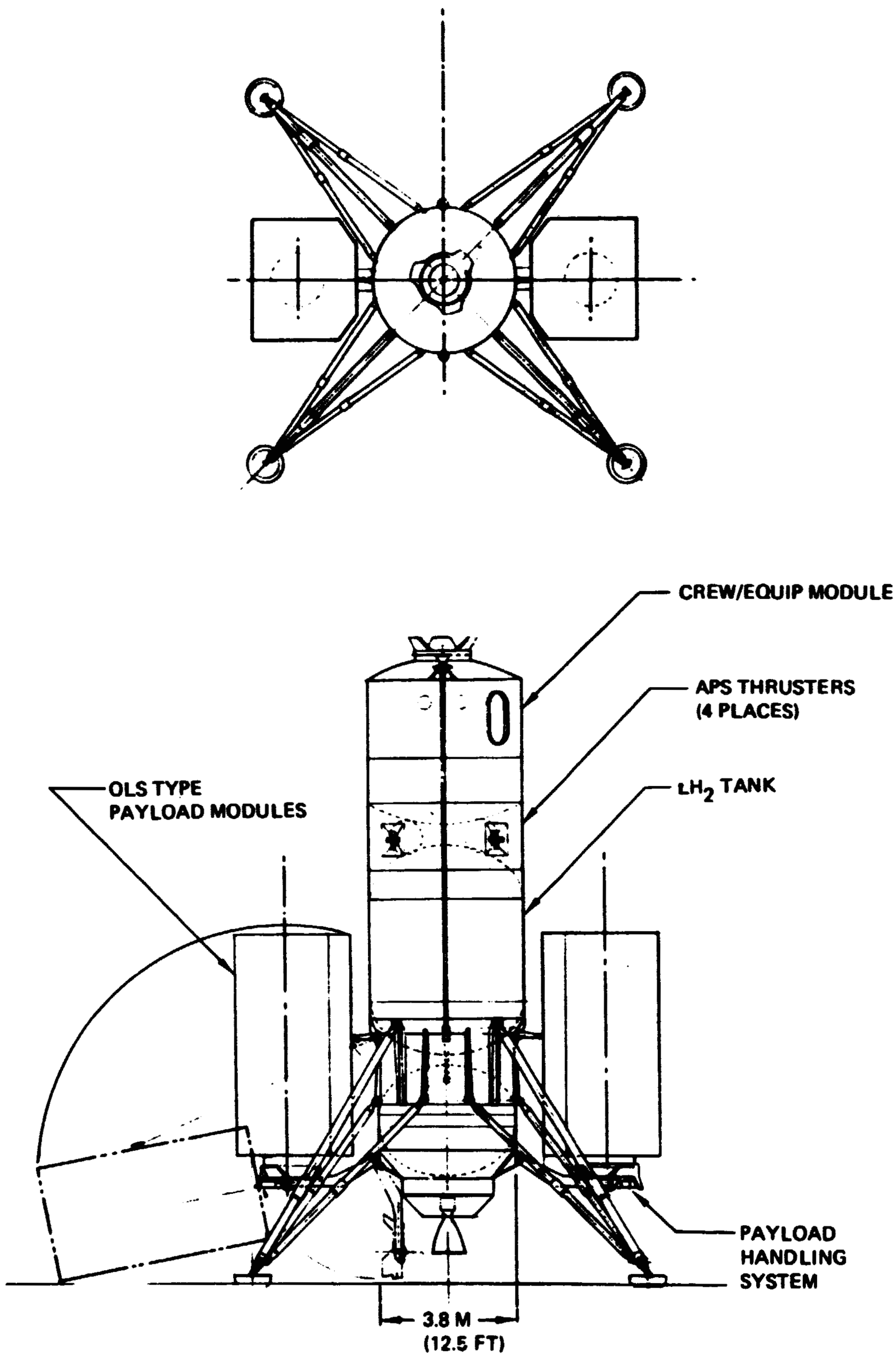
1.2.5 Crew Vehicles

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.

1.2.5.1 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 Figures 1.2-33 and 1.2-34. 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 IMPUSLE PROPELLANT

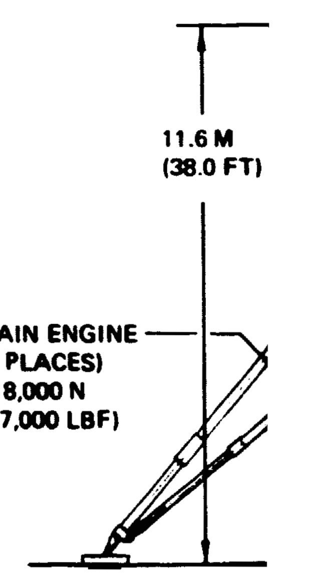


Figure 1..

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1 IMPULSE PROPELLANT 32,200 KG (71,000 LB)

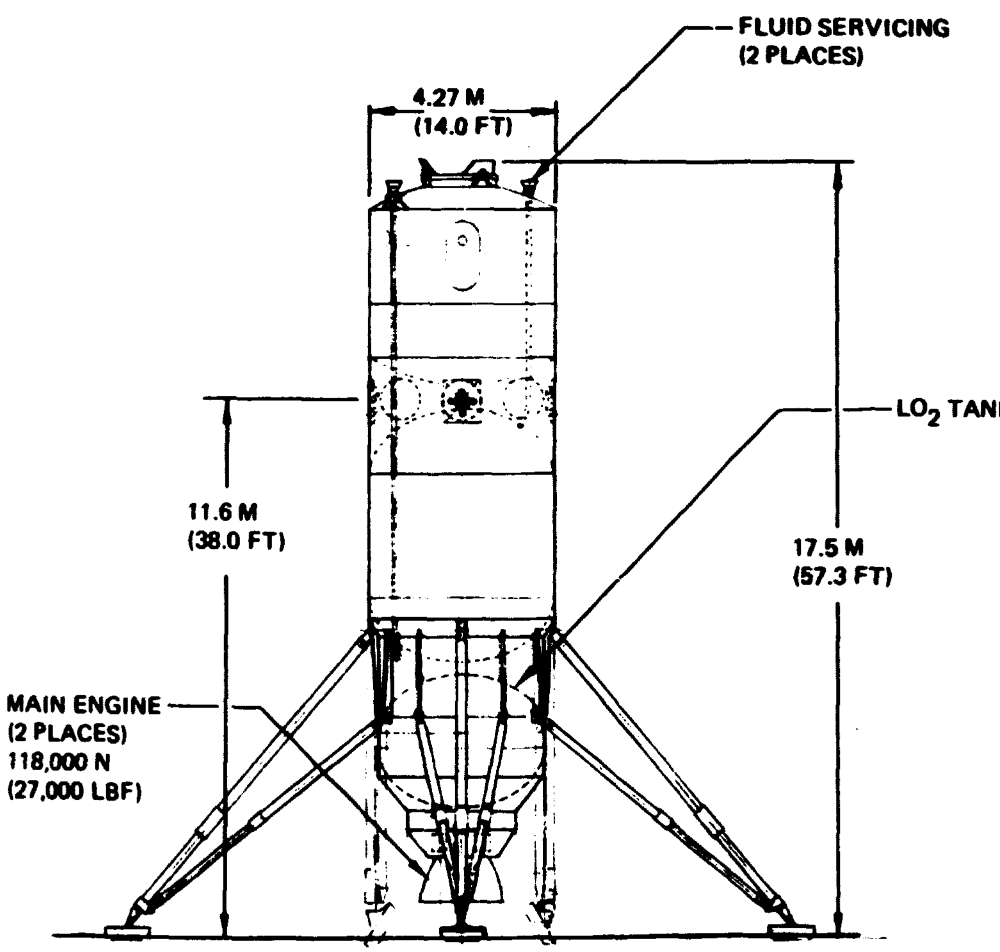


Figure 1.2-23 Single Stage LO_2/LH_2 LTV Configuration (Sm Dia) Point Design 1

Table 1.?-11. Small Diameter Single Stage LO_2/LH_2 LTV Weight Details Point Design [1]

	(Lbm)	(Kg)
Structure and mechanisms	(7,950)	3,606
Body shell	2,100	
Fuel tank	1,030	
Oxidizer tank	440	
Thurst structure	450	
Stage/payload interfaces	1,020	
Secondary structure	210	
Landing gear	2,700	
Main propulsion	(1,850)	839
Main engines	880	
Accessories	180	
Pressurization and vent	300	
Propellant system	360	
Gimbal system	130	
Auxiliary propulsion	(490)	222
Thrusters	180	
Tanks	150	
Pressurization and vent	50	
Propellant system	100	
Avionics [2]	(240	(109)
Nav. guid and control	80	
Data management	40	
Communications	–	
Instrumentation	120	
Rendez and docking	-	
Electrical power [2]	(340)	(154)
Fuel cells	80	
Batteries	50	
Tankage	30	
Processing and control	–	
Wiring harnesses	180	
Thermal control	(830)	(376)
Main tank insulation	710	
Insulation purge	40	
Equipment control	30	
Base protection	30	
Paint and sealer	20	
Weight growth (15%)	(1,760)	(798)
Total stage dry weight	13,460	6,105

IEF-177

[1] Based upon 32,200 kg (71,000 lbm) impulse propellant

[2] Remainder in crew/equipment module

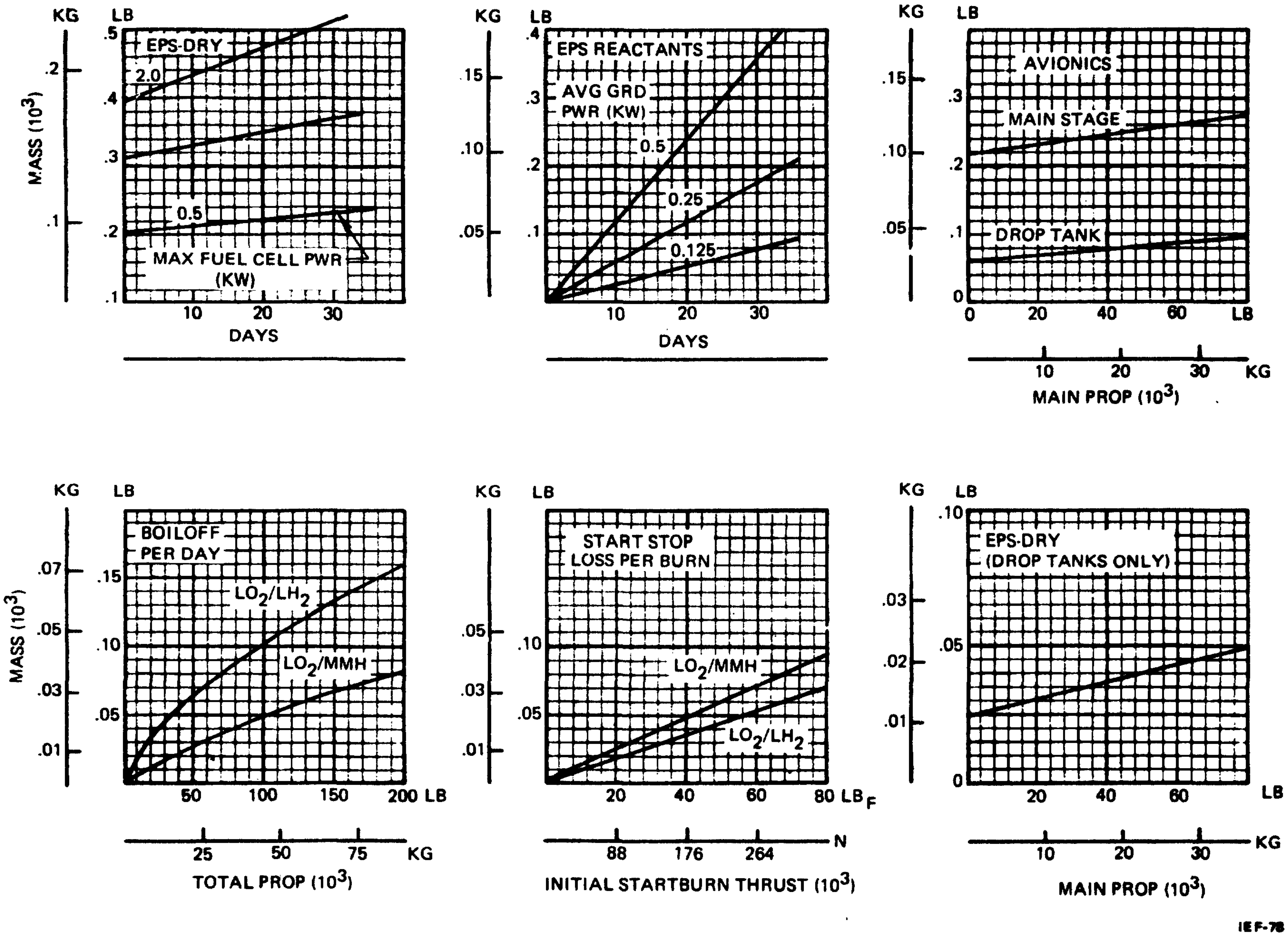
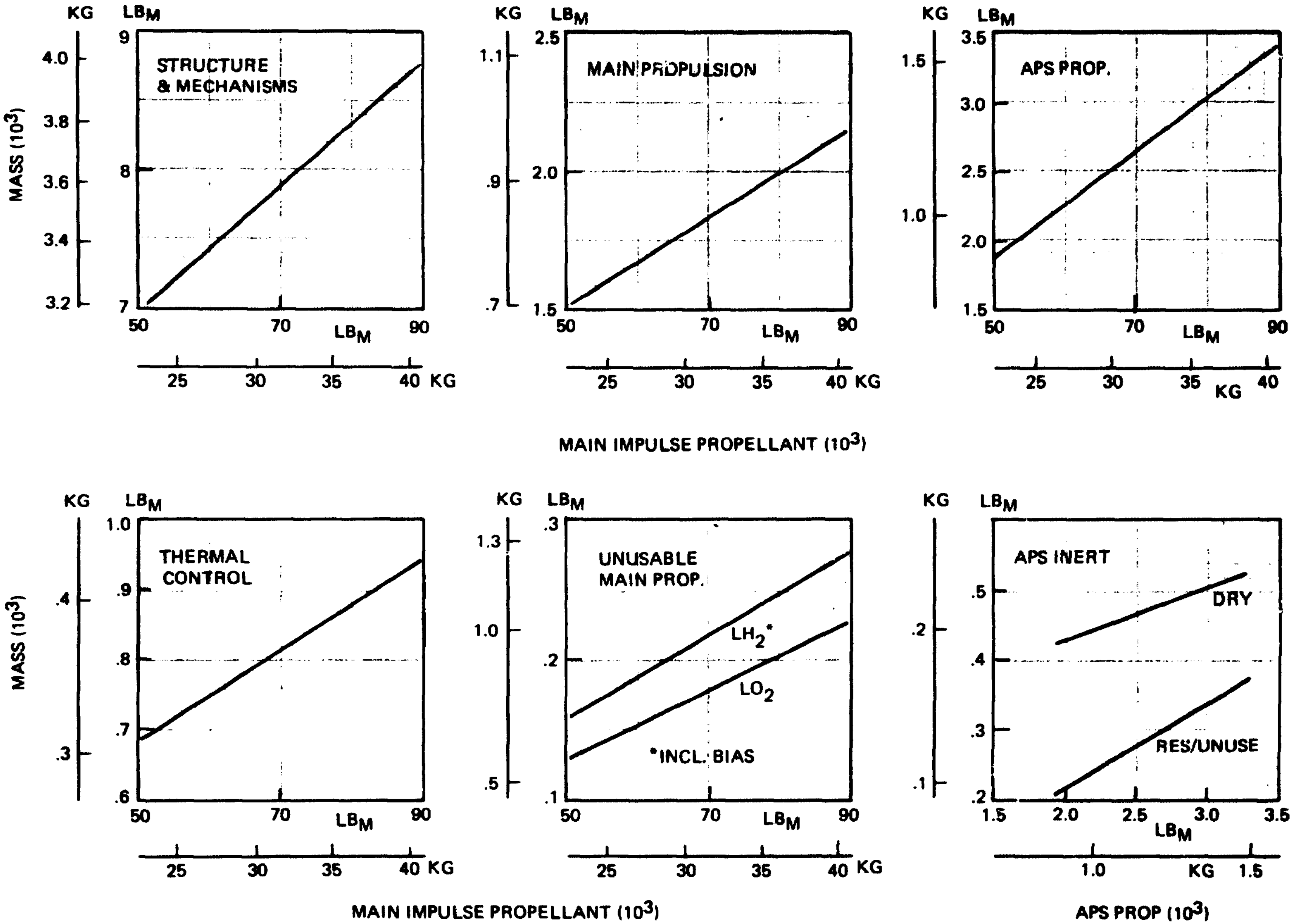


Figure 1.2-24. General LTV Subsystem Parametrics (Sheet 1)



SCALING PARAMETERS A = 3,620 KG (7,980 lb) B = 0.085 C = 0 D = 0.1725

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Figure 1.2-24 Subsystem Parametrics LO_2/LH_2 Single Stage LTV Small Diameter (Sheet 2)

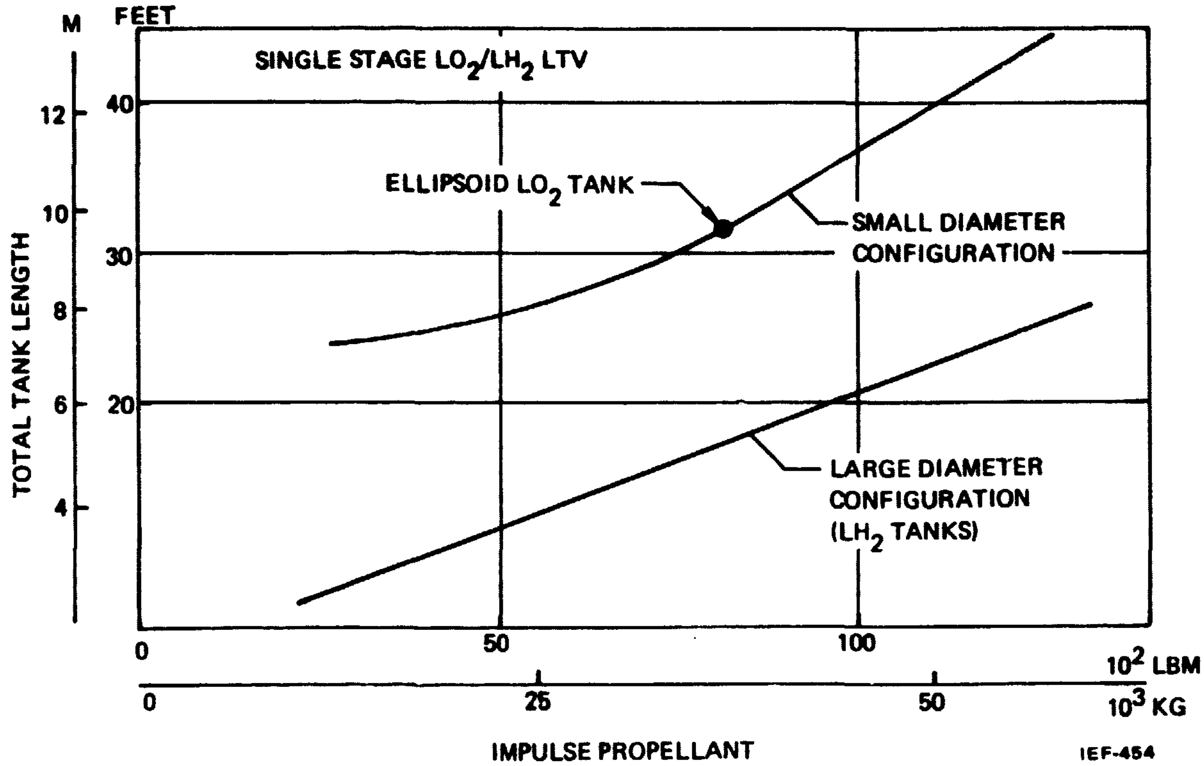
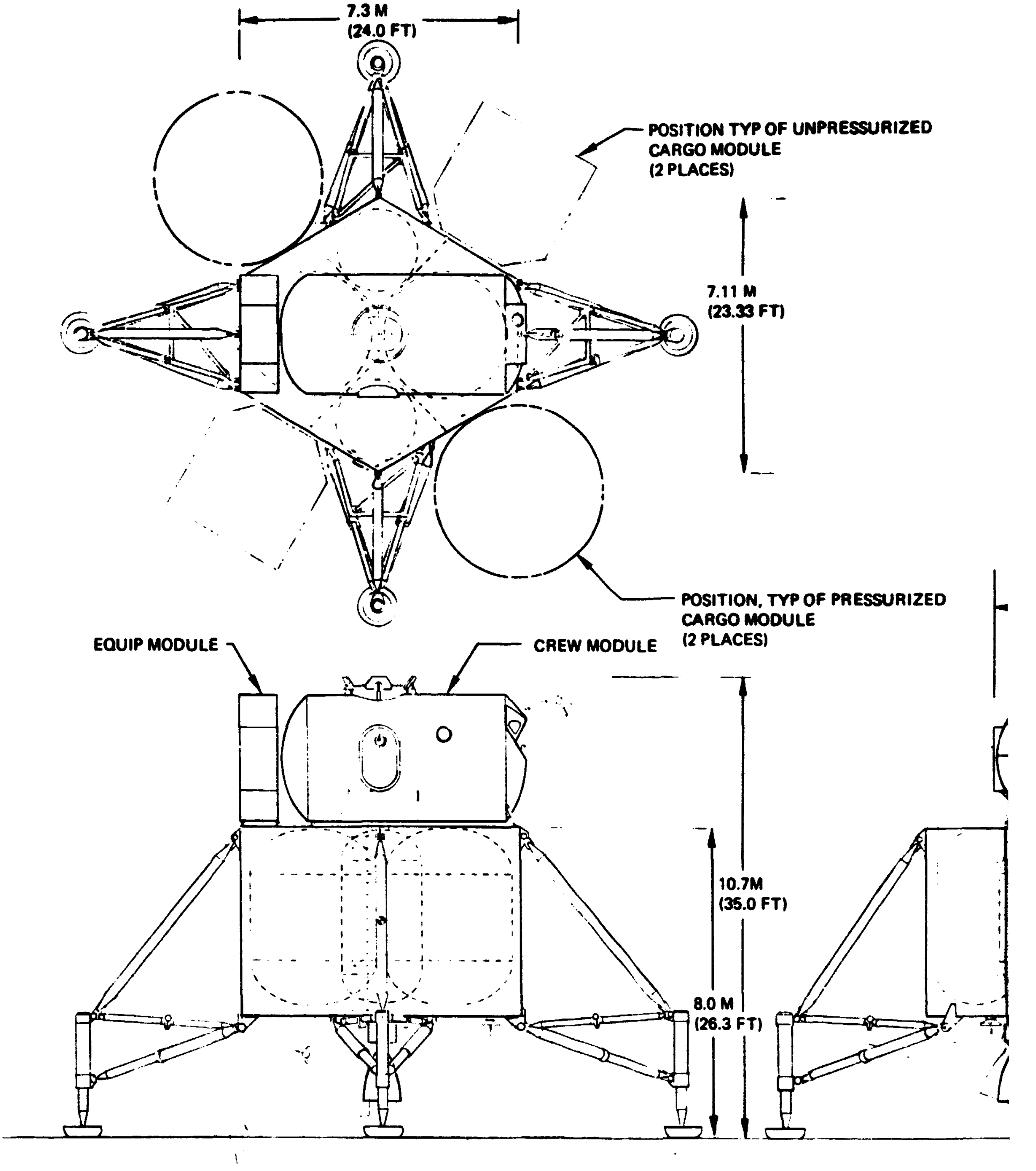


Figure 1.2-24. (Sheet 3)



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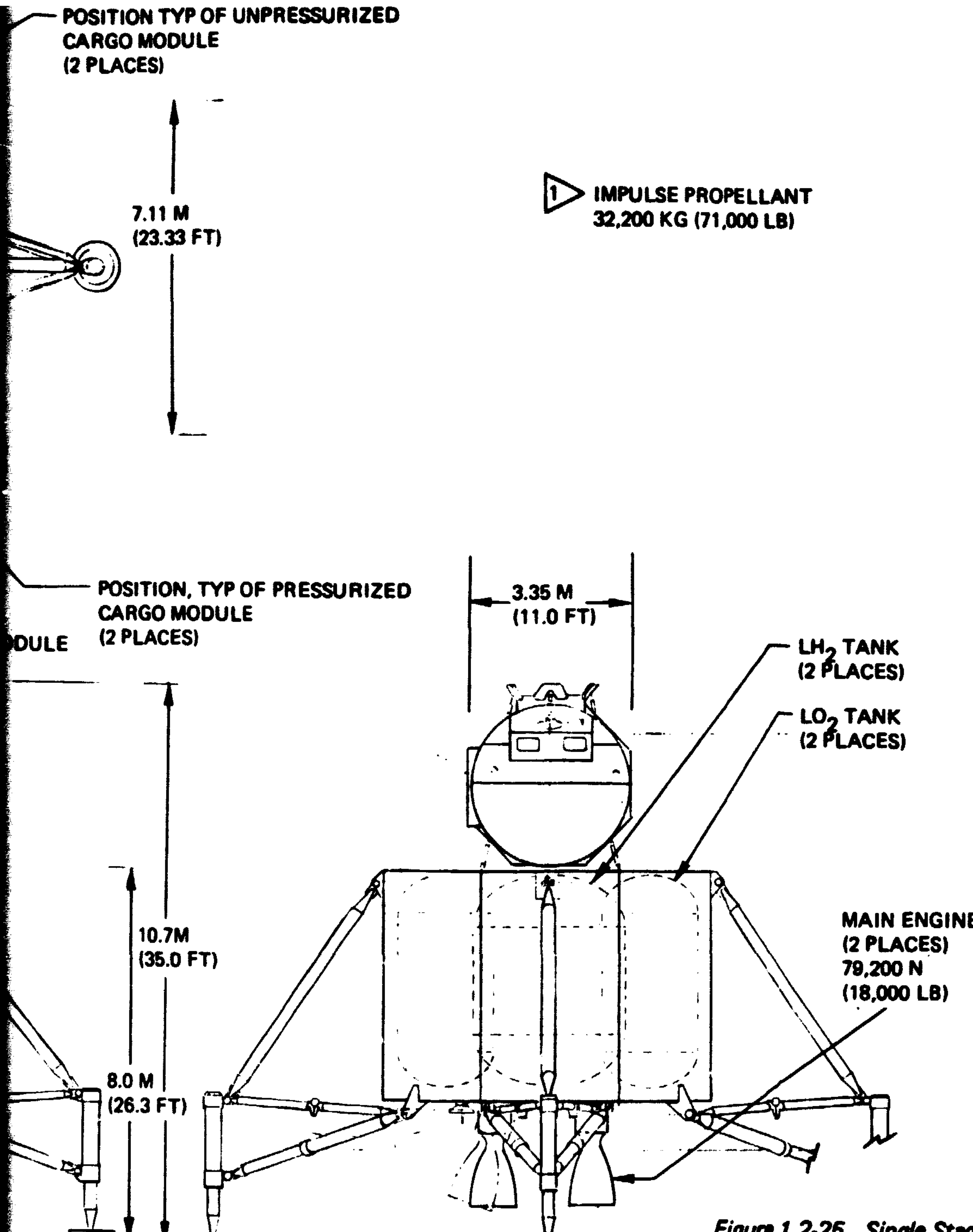

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Figure 1.2-25 Single Stage LO_2/LH_2 LTV Configuration (Lg Dia.) Point Design 1

Table 1.2-12. Large Diameter Single Stage LO_2/LH_2 LTV Weight Details Point Design [1]

	(Lbm)	(Kg)
Structure and mechanisms	(9,410)	4,268
Body shell	3,490	
Fuel tank	780	
Oxidizer tank	550	
Thurst structure	310	
Stage/payload interfaces	1,330	
Secondary structure	730	
Landing gear	2,220	
Main propulsion	(2,060)	(934)
Main engines	880	
Accessories	180	
Pressurization and vent	360	
Propellant system	510	
Gimbal system	130	
Auxiliary propulsion	(490)	(222)
Thrusters	180	
Tanks	160	
Pressurization and vent	50	
Propellant system	100	
Avionics [2]	(240)	(109)
Nav. guid and control	80	
Data management	40	
Communications	–	
Instrumentation	120	
Rendez and docking	–	
Electrical power [2]	(340)	(154)
Fuel cells	80	
Batteries	50	
Tankage	30	
Processing and control	--	
Wiring harnesses	180	
Thermal control	(930)	(422)
Main tank insulation	810	
Insulation purge	40	
Equipment control	30	
Base protection	30	
Paint and sealer	20	
Weight growth (15%)	(2,020)	(916)
Total stage dry weight	15,490	(7,026)

[1] Based upon 32,200 kg (71,000 lbm) impulse propellant

[2] Remainder in crew/equipment module

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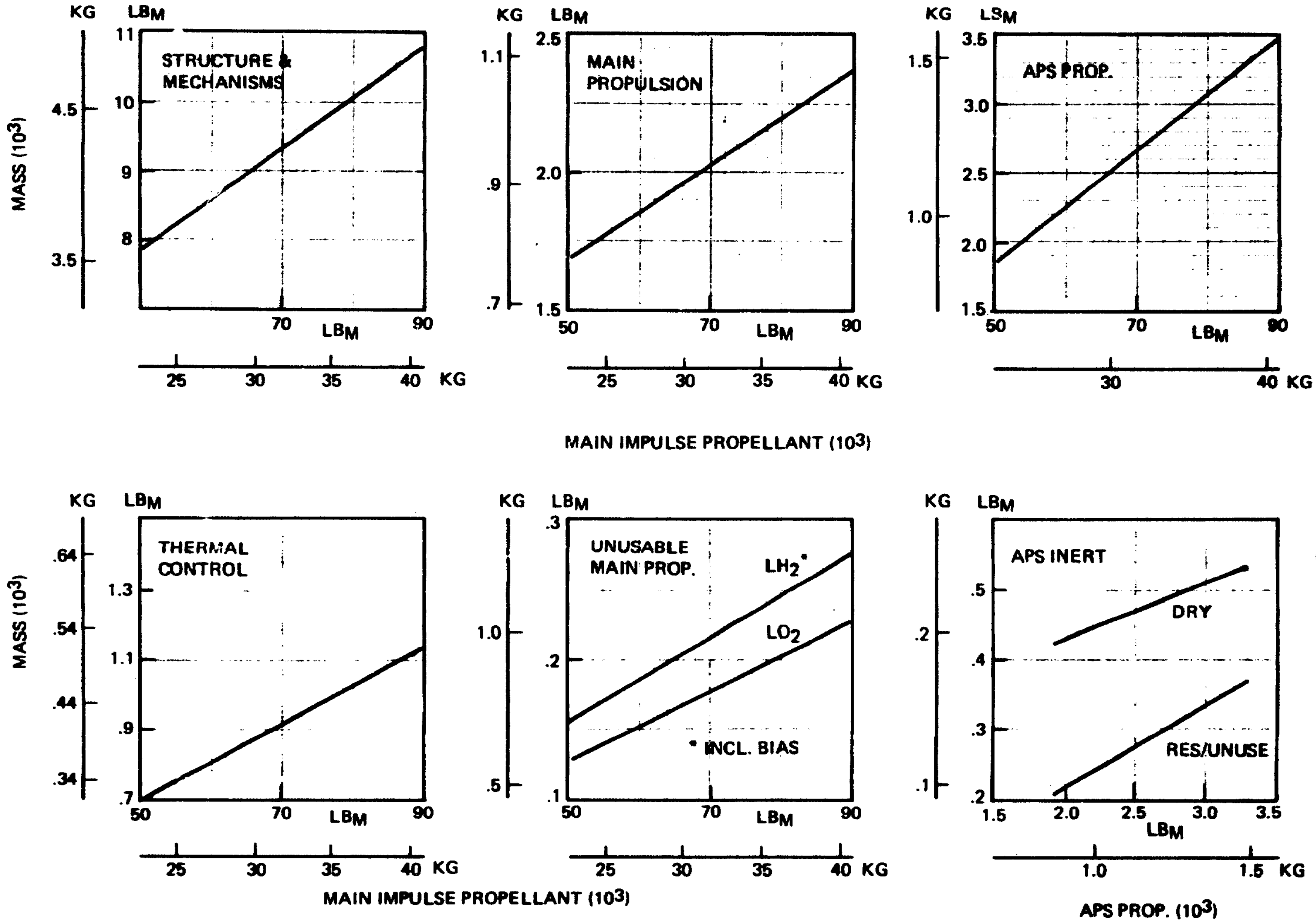


SCALING PARAMETERS A = 3 390 KG (7,470 LB) B = 0.1210 C = 0 D = 0.1725

IEF-9

Figure 1.2-26. Subsystem Parametrics — LO_2/LH_2 Single Stage LTV Large Diameter (Sheet 1)

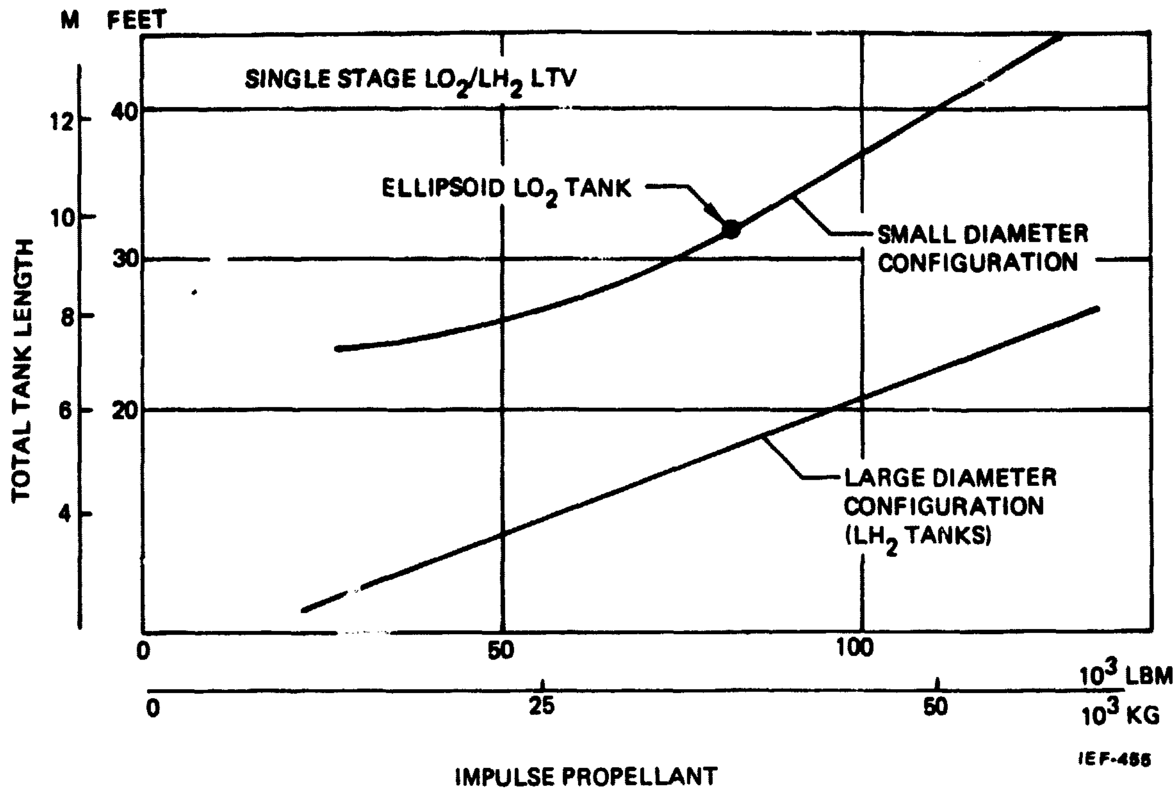
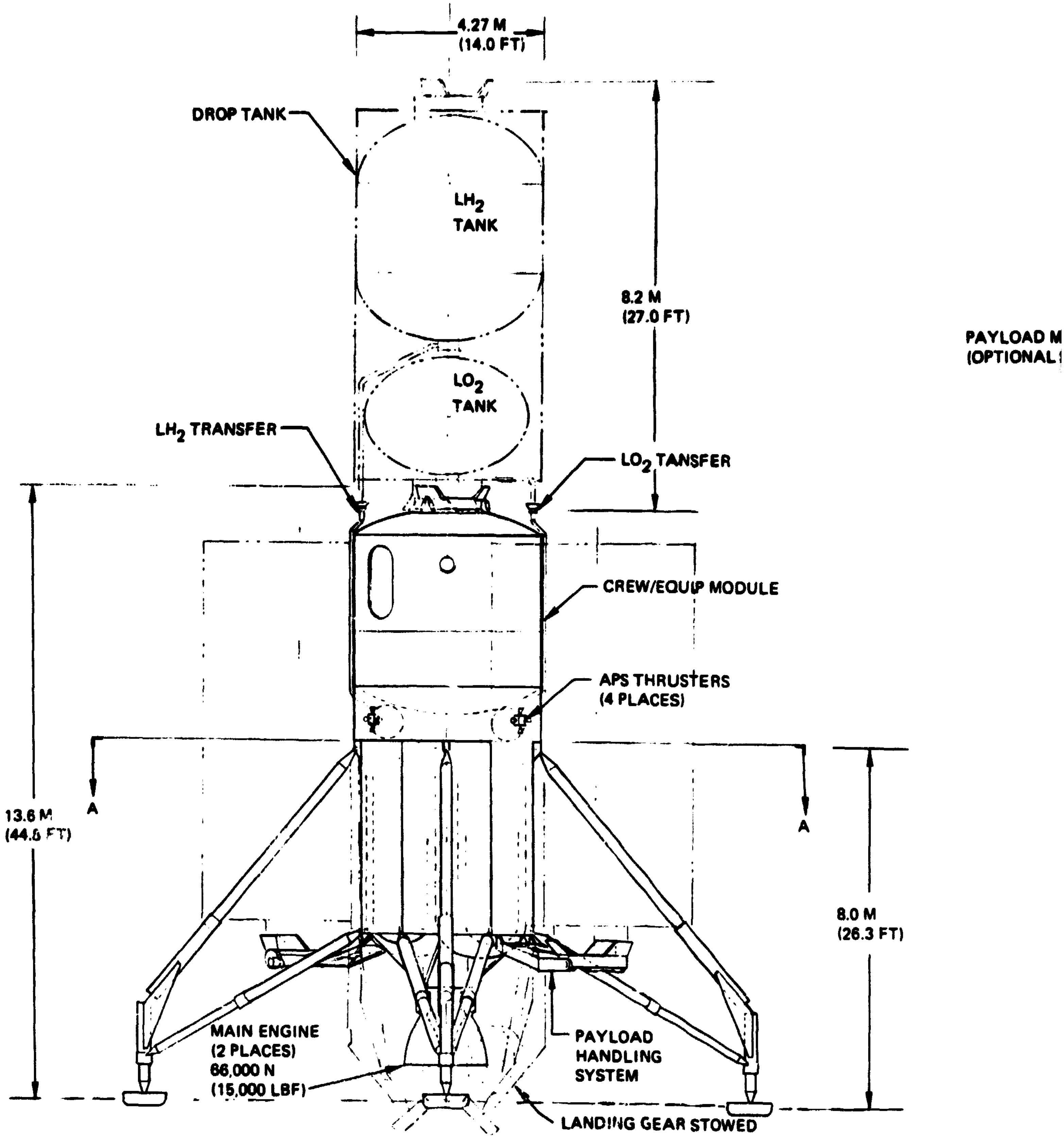


Figure 1.2-26. (Sheet 2)



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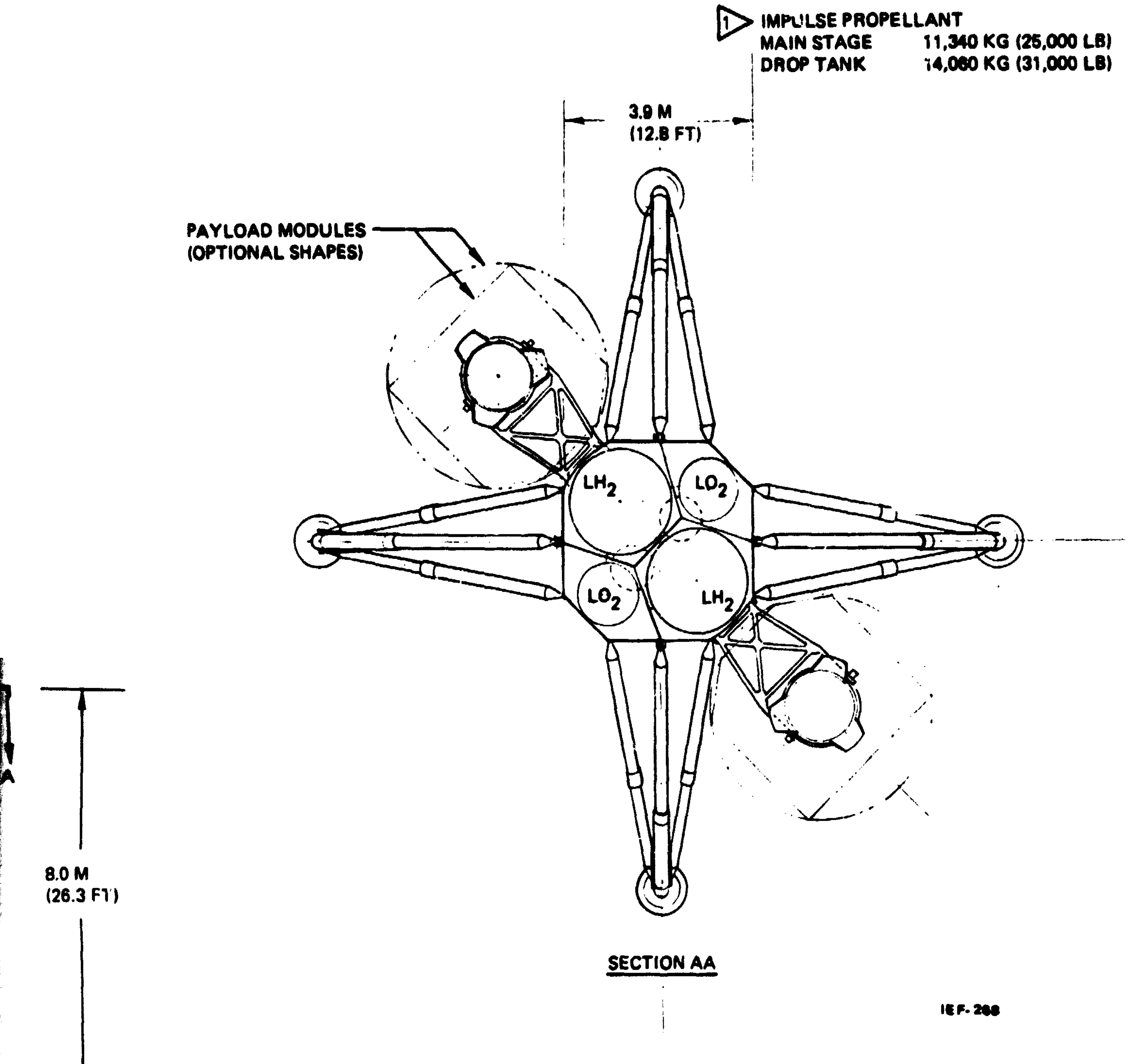
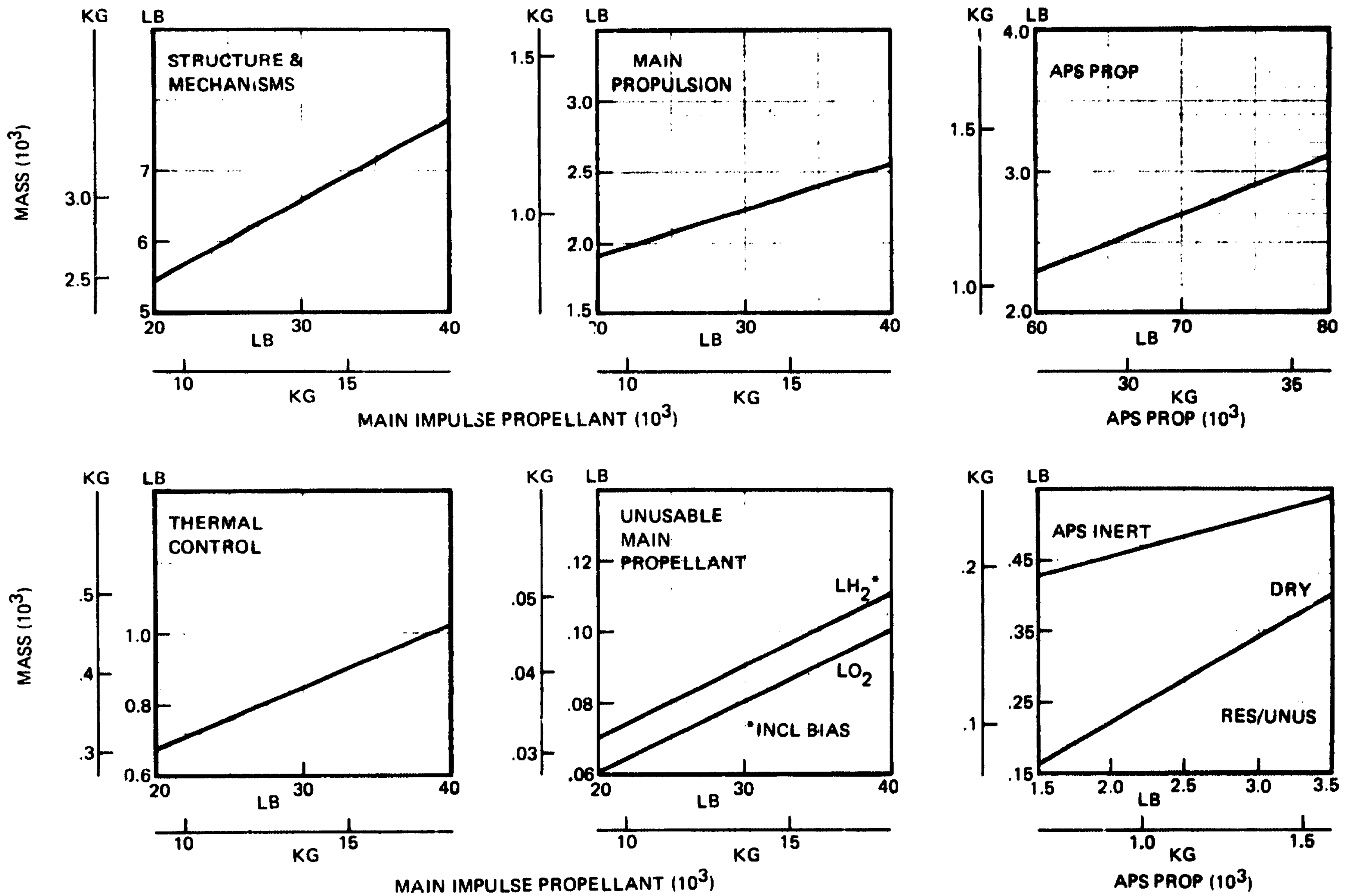
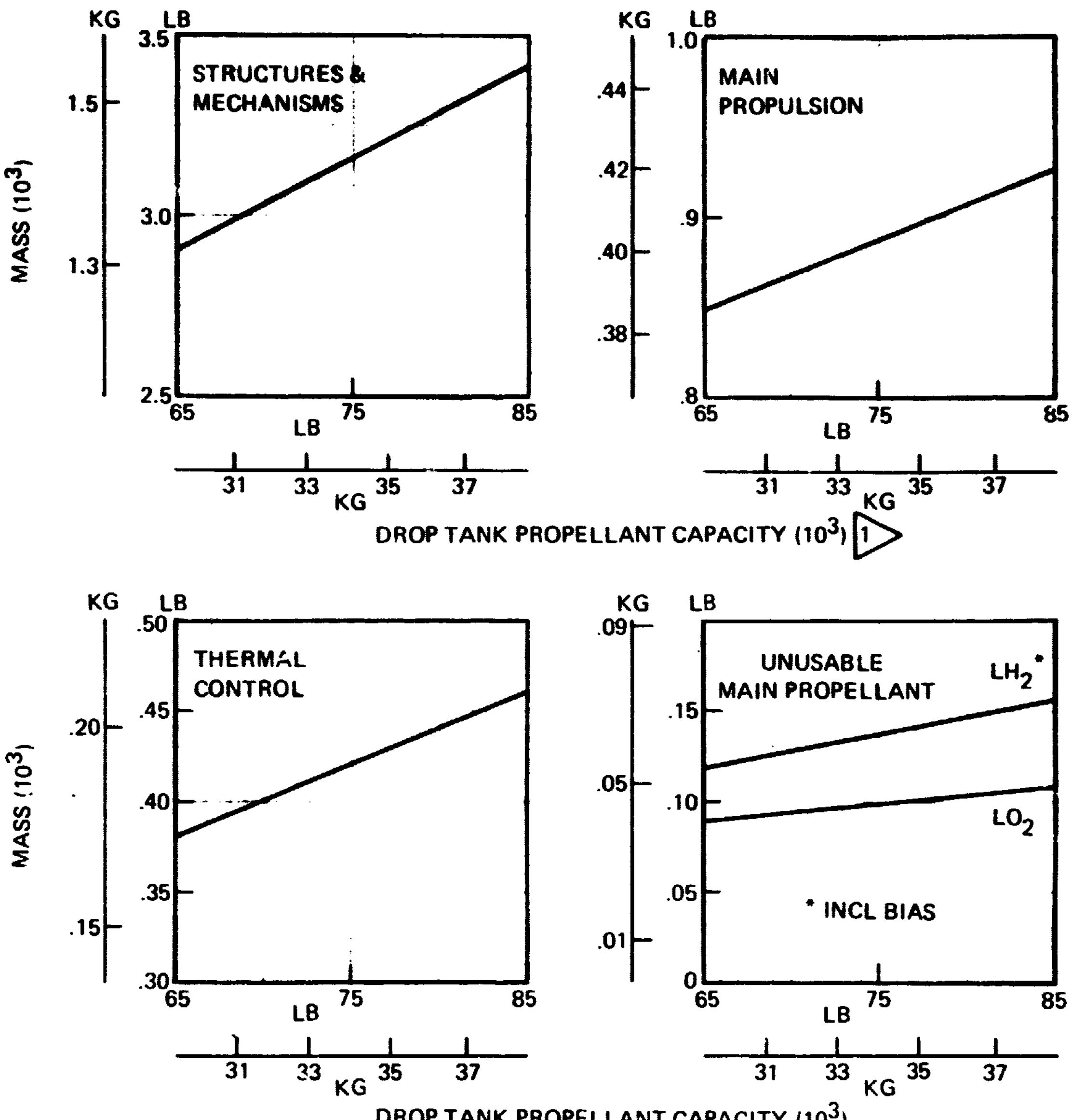


Figure 1.2-27 1-1/2 Stage LO_2/LH_2 LTV Configuration Point Design 1



SCALING PARAMETERS: A = 2,975 KG (6,590 LB); B = 0.1730; C = 0; D = 0.1725

Figure 1.2-28. 1-1/2 Stage LO_2/LH_2 LTV Subsystem Parametrics, Main Stage (Sheet 1)



1 SIZED TO ACCOMMODATE TOTAL LTV PROP.

SCALING PARAMETERS: A = 970 KB (2,140 LB); B = 0.0870; C = 0; D = 0

Figure 1.2-28. 1-1/2 Stage LO2/LH2 LTV Subsystem Parametrics, Drop Tank (Sheet 2)

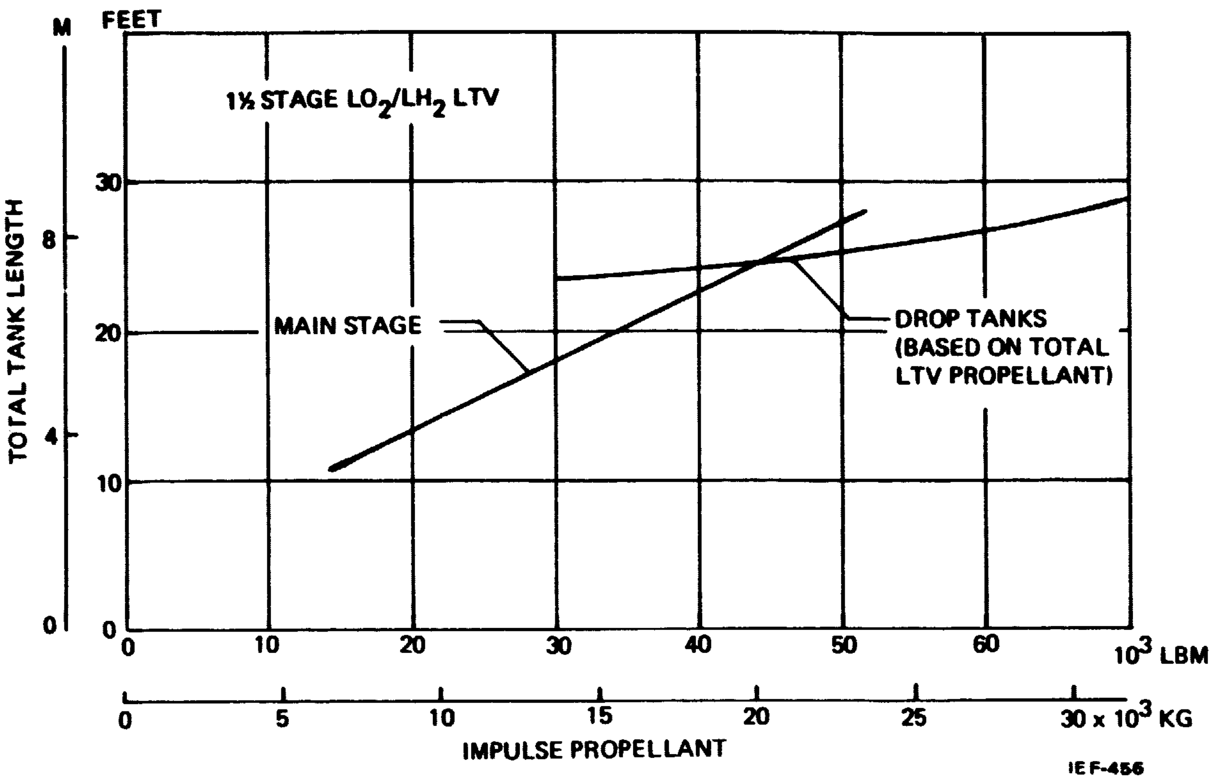


Figure 1.2-28. Sheet 3

Table 1.2-13. 1-1/2 Stage LO_2/LH_2 LTV Weight Details Point Design

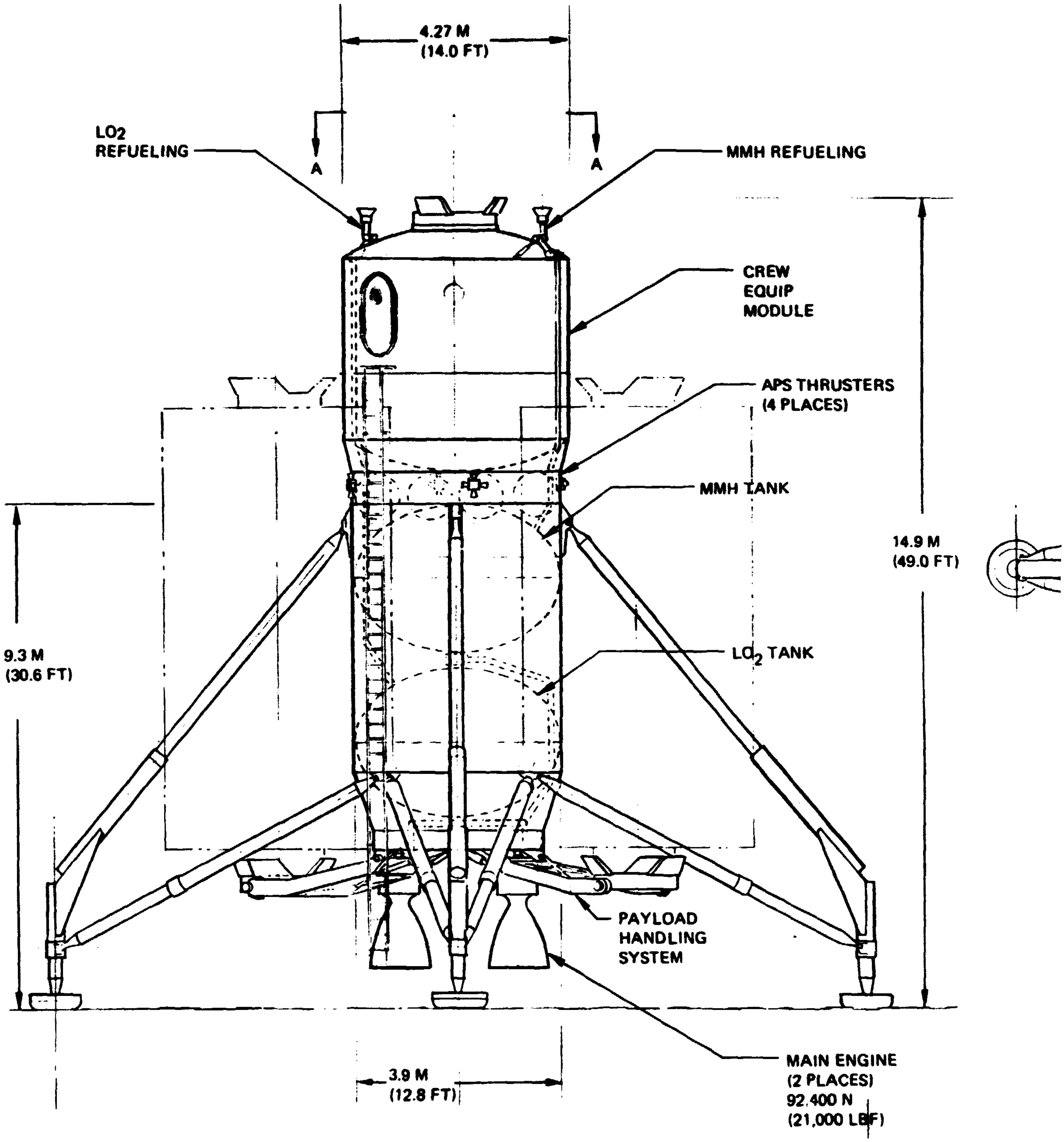
	Main stage [1]		Drop tank [3]	
	(Lbm)	(Kg)	(Lbm)	(Kg)
Structure and mechanisms	(5,970)	(2,708)	(2,650)	(1,202)
Body shell	1,460		670	
Fuel Tank	600		770	
Oxidizer tank	310		360	
Thrust structure	280		–	
Stage/payload interface	950		800	
Secondary structure	280		50	
Landing gear	2,090		–	
Main propulsion	(1,570)	(712)	(810)	(367)
Main engines (2)	720		–	
Accessories	140		–	
Pressurization and vent	260		340	
Propellant system	340		470	
Gimbal system	110		–	
Auxiliary propulsion	(520)	(236)	–	
Thrusters	200		–	
Tanks	170		–	
Pressurization and vent	50		–	
Propellant system	100		–	
Avionics [2]	(240)	(109)	(70)	(32)
Nav., guid and control	80		–	
Data management	–		–	
Communications	–		–	
Instrumentation	120		30	
Rendez and docking	40		40	
Electrical Power [2]	(340)	(154)	(110)	(50)
Fuel cells	80		–	
Batteries	50		50	
Tankage	30		–	
Processing and control	–		–	
Wiring harnesses	180		60	
Thermal control	(760)	(345)	(340)	(154)
Main tank insulation	640		290	
Insulation purge	40		30	
Equipment control	30		–	
Base protection	30		–	
Paint and sealer	20		20	
Weight growth (15%)	(1,410)	(640)	(600)	(272)
Total stage dry weight	10,810	4,903	4,580	2,077

IEF-175

[1] Based upon 11,300 kg(24,900 lbm) impulse propellant

[2] Remainder in crew equipment module

[3] Based upon 25,100 kg (55,400 lbm) total LTV impulse propellant



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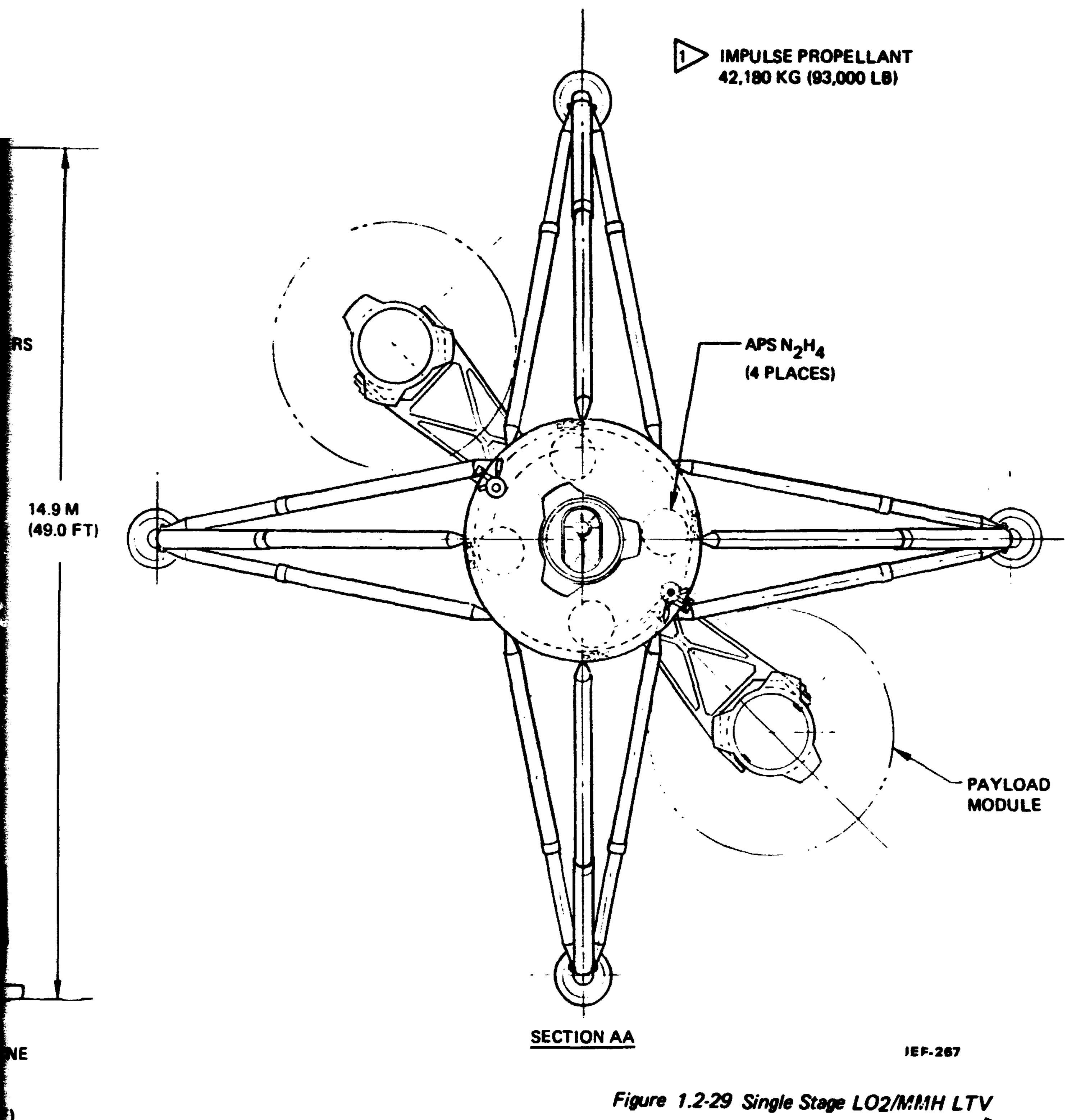

Figure 1.2-29 Single Stage LO2/MMH LTV Configuration Point Design 1

Table 1.2-14. Single Stage LO_2/MMH LTV Weight Details Point Design [1]

	(Lbm)	(Kg)
Structure and mechanisms	(6,650)	(3,016)
Body shell	1,730	
Fuel tank	470	
Oxidizer tank	470	
Thurst structure	460	
Stage/payload interfaces	1,020	
Secondary structure	200	
Landing gear	2,300	
Main propulsion	(2,040)	(925)
Main engines	840	
Accessories	170	
Pressurization and vent	510	
Propellant system	380	
Gimbal system	140	
Auxiliary propulsion	(540)	(245)
Thrusters	200	
Tanks	180	
Pressurization and vent	50	
Propellant system	110	
Avionics [2]	(240)	(109)
Nav. guid and control	80	
Data management	40	
Communications	–	
Instrumentation	120	
Rendez and docking	–	
Electrical power [2]	(330)	(150)
Fuel cells	80	
Batteries	50	
Tankage	30	
Processing and control	–	
Wiring harnesses	170	
Thermal control	(480)	(218)
Main tank insulation	370	
Insulation purge	30	
Equipment control	30	
Base protection	30	
Paint and sealer	20	
Weight growth (15%)	(1,540)	(699)
Total stage dry weight	11,820	5,362

[1] Based upon 32,200 kg (71,000 lbm) impulse propellant

[2] Remainder in crew/equipment module

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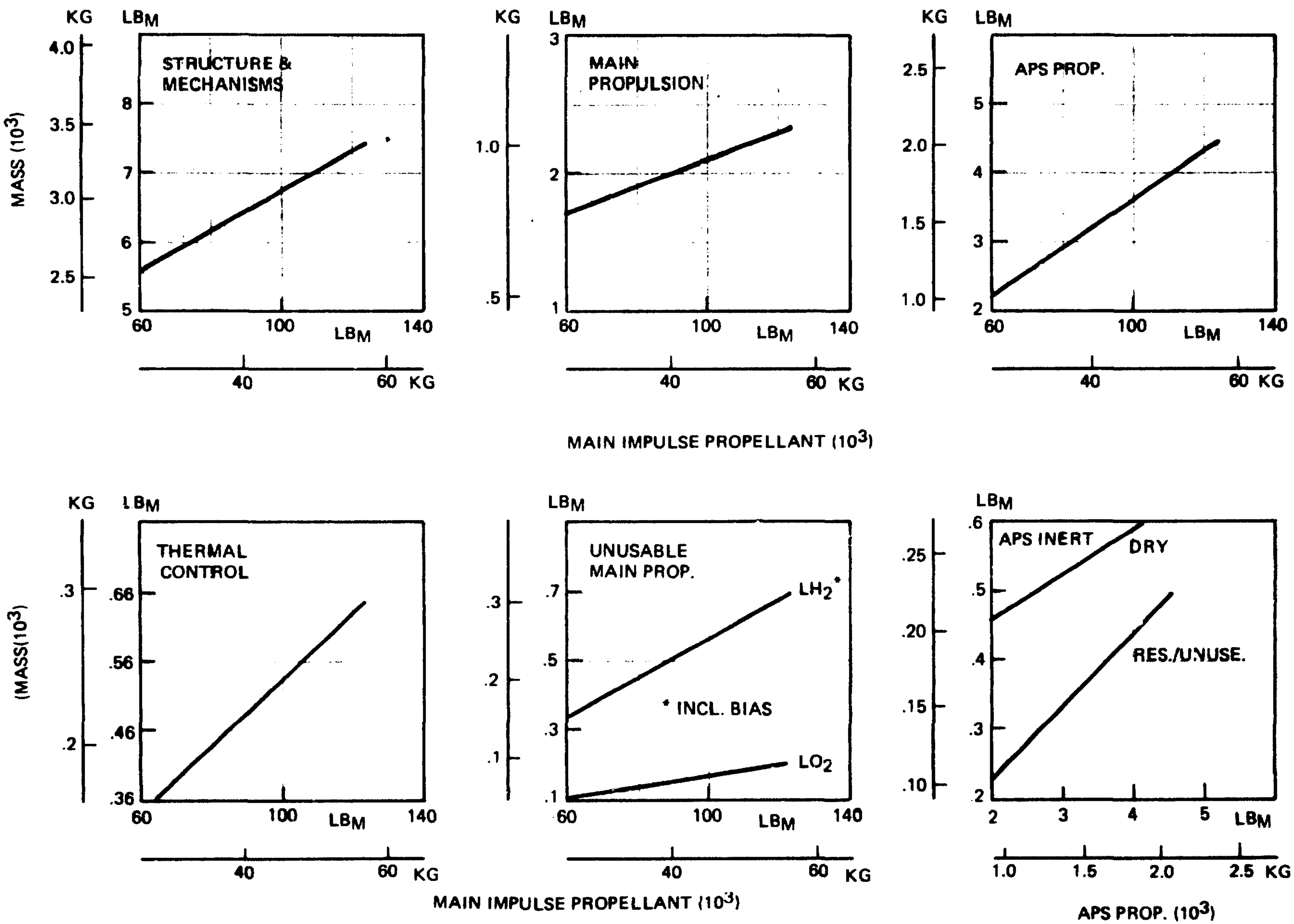

SCALING PARAMETERS A = 3 200 KG (7,050 LB) B = 0.0610 C = 0 D = 0.1725

Figure 1.2-30. Subsystem Parametrics — LO_2/MMH Single Stage LTV (Sheet 1)

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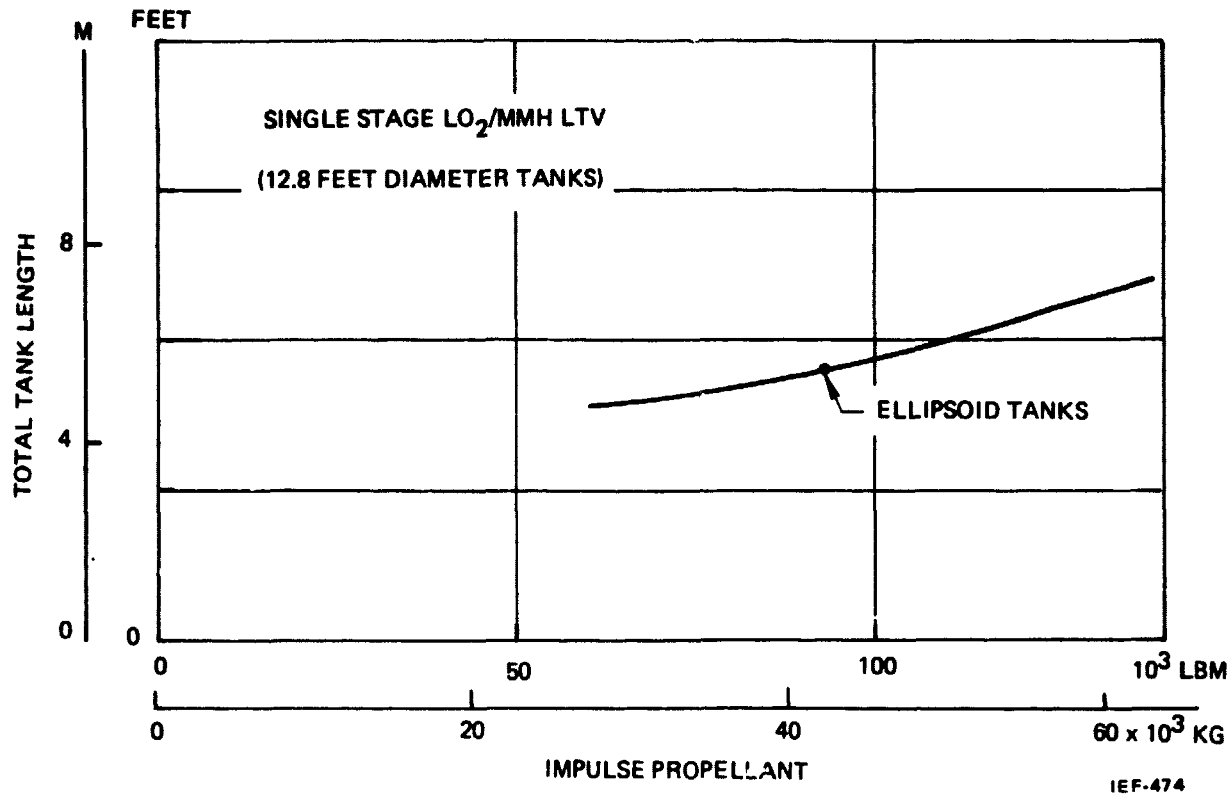


Figure 1.2-30. Sheet 2

Table 1.2-15. 1-1/2 Stage LO_2/MMH LTV Weight Details Point Design

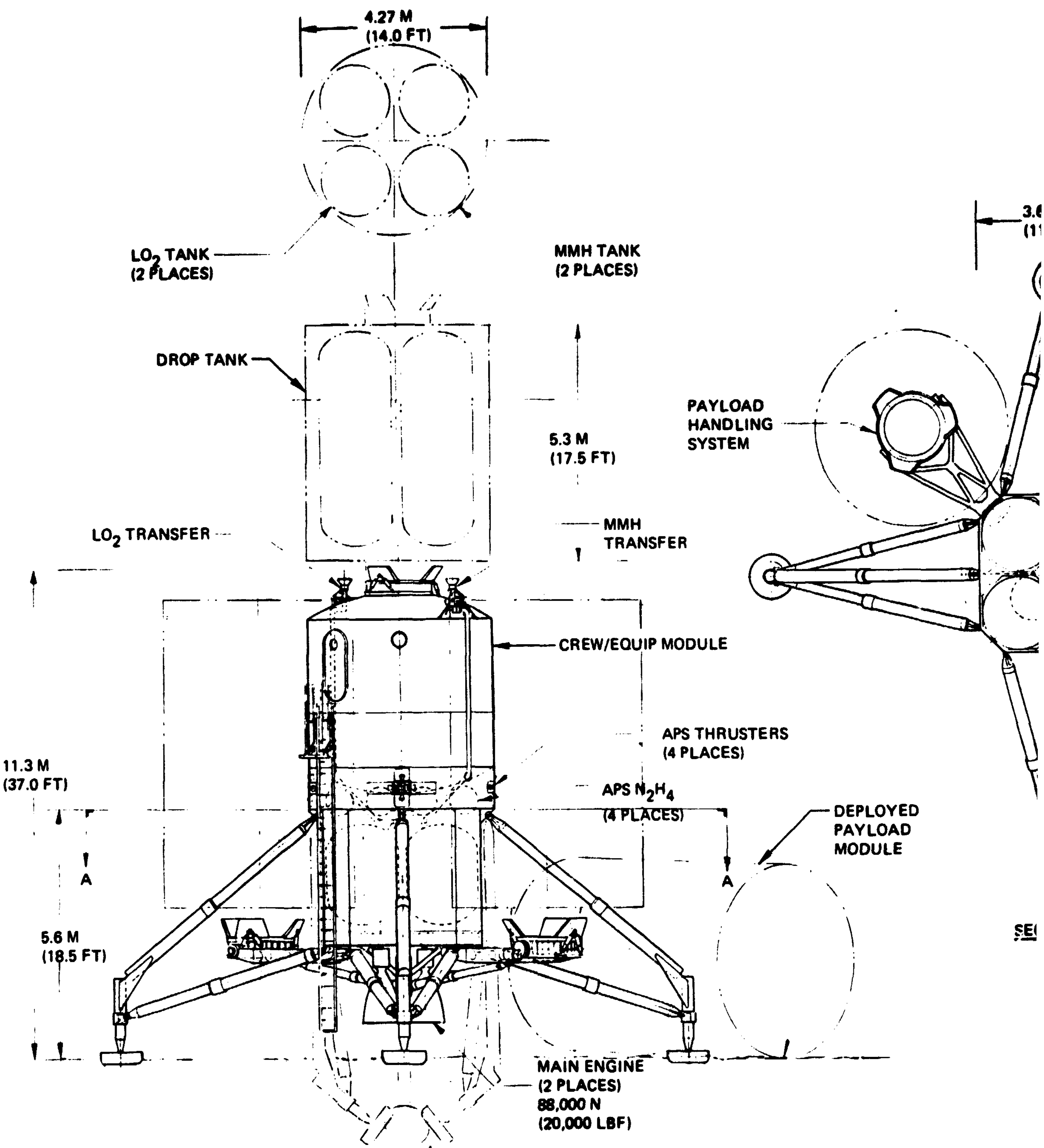
	Main stage [1]		Drop tank [3]	
	(Lbm)	(Kg)	(Lbm)	(Kg)
Structure and mechanisms	(5,480)	(2,486)	(2,490)	(1,129)
Body shell	910		680	
Oxidizer tank	340		480	
Thrust structure	310		480	
Stage/payload interface	950		–	
Secondary structure	390		800	
Landing gear	2,240		50	
Main propulsion	(1,490)	(676)	(570)	(259)
Main engines (2)	790		–	
Accessories	150		–	
Pressurization and vent	200		270	
Propellant system	220		300	
Gimbal system	130			
Auxiliary propulsion	(530)	(240)	–	
Thrusters	200		–	
Tanks	170		–	
Pressurization and vent	50		–	
Propellant system	110		–	
Avionics [2]	(240)	(109)	(70)	(32)
Nav., guid and control	80		–	
Data management	40		–	
Communications	–		–	
Instrumentation	120		30	
Rendez and docking	–		40	
Electrical Power [2]	(330)	(150)	(100)	(45)
Fuel cells	80		–	
Batteries	50		50	
Tankage	30		–	
Processing and control	–		–	
Wiring harnesses	170		50	
Thermal control	(320)	(145)	(420)	(191)
Main tank insulation	210		360	
Insulation purge	30		40	
Equipment control	30		–	
Base protection	30		–	
Paint and sealer	20		20	
Weight growth (15%)	(1,260)	(572)	(550)	(249)
Total stage dry weight	9,650	4,377	4,200	1,905

IEF-174

[1] Based upon 16,500 kg (36,400 lbm) impulse propellant

[2] Remainder in crew equipment module

[3] Based upon 38,700 kg(85,300 lbm) total LTV impulse propellant



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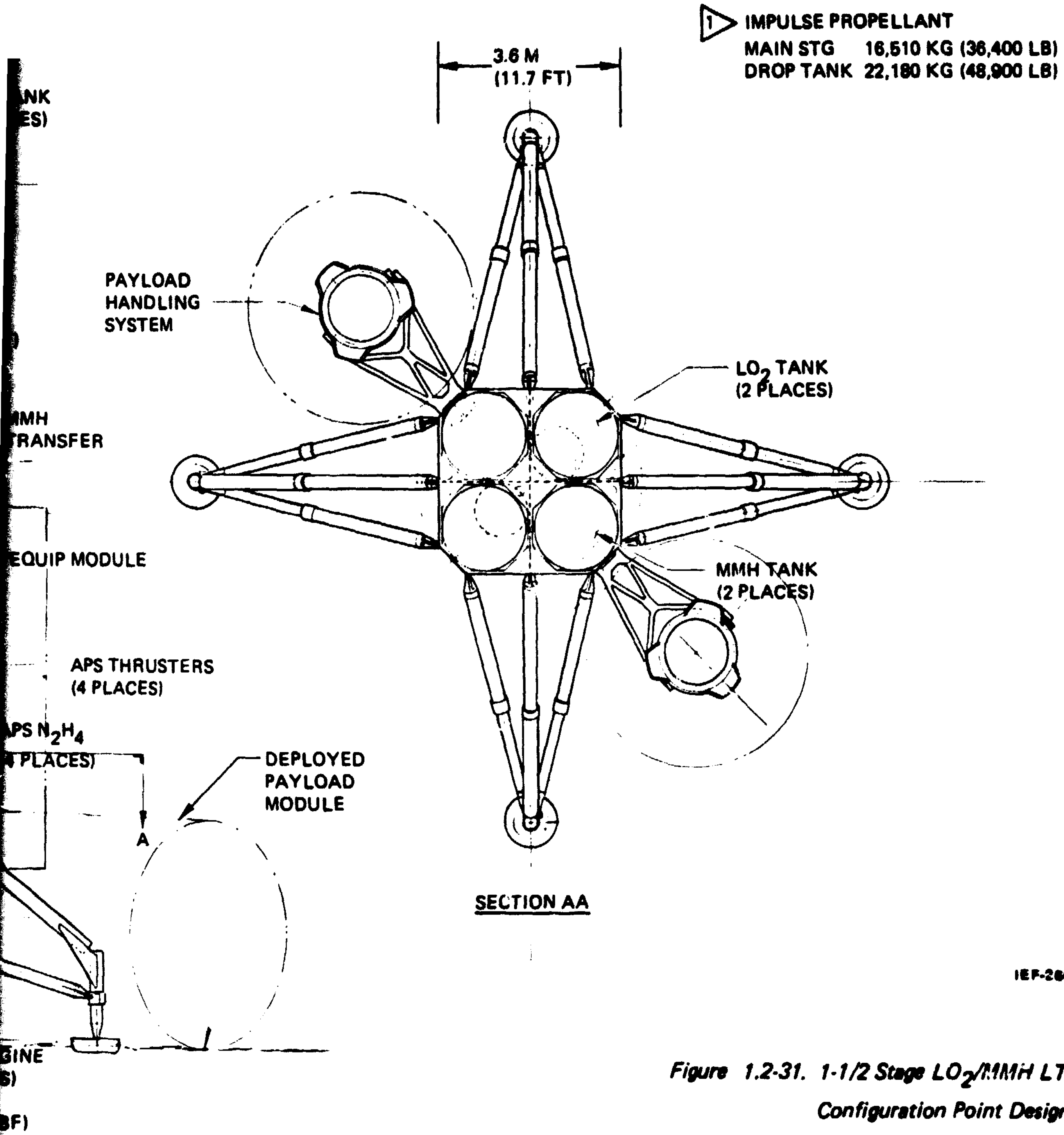

Figure 1.2-31. 1-1/2 Stage LO_2/MMH LTV Configuration Point Design 1

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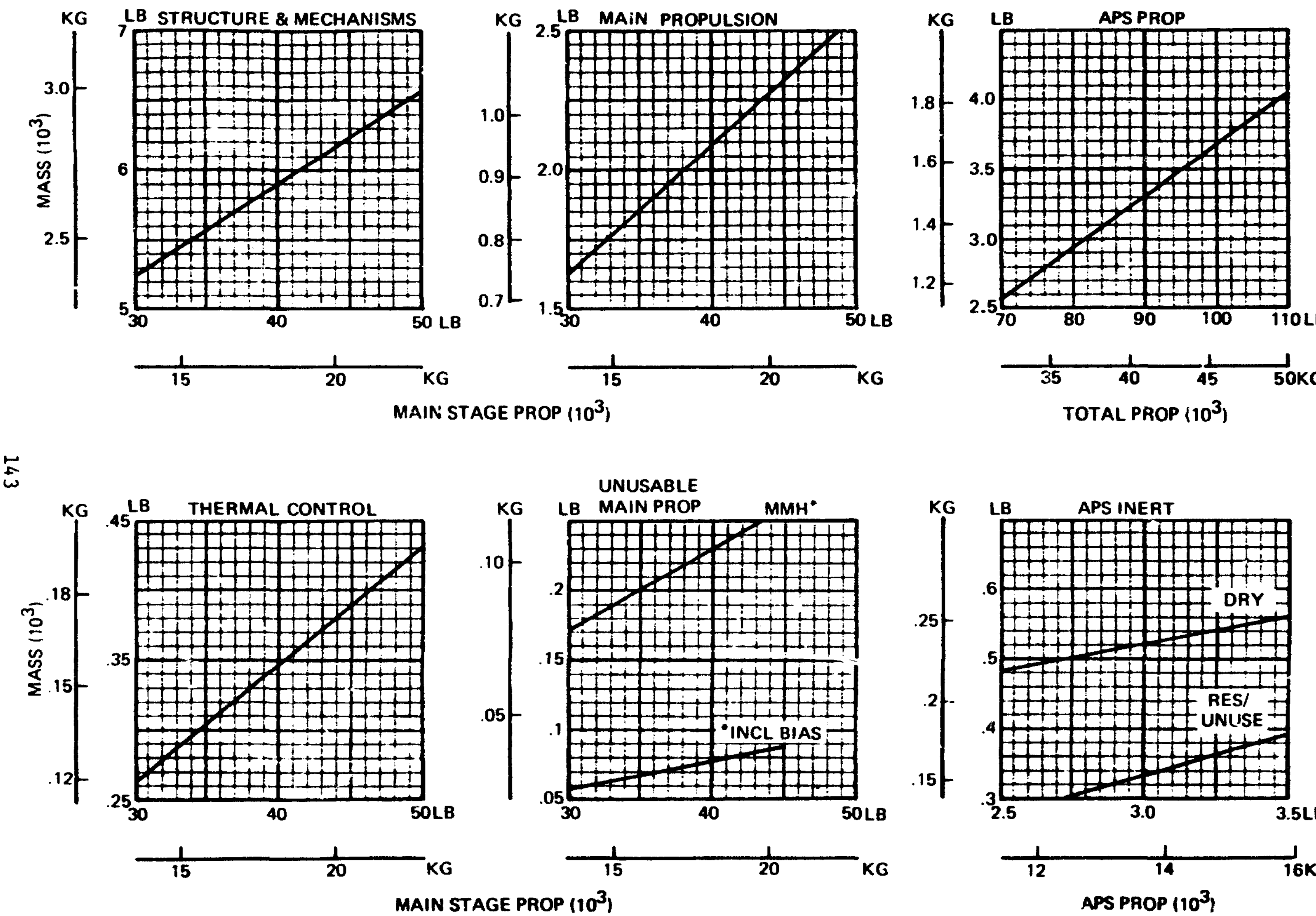
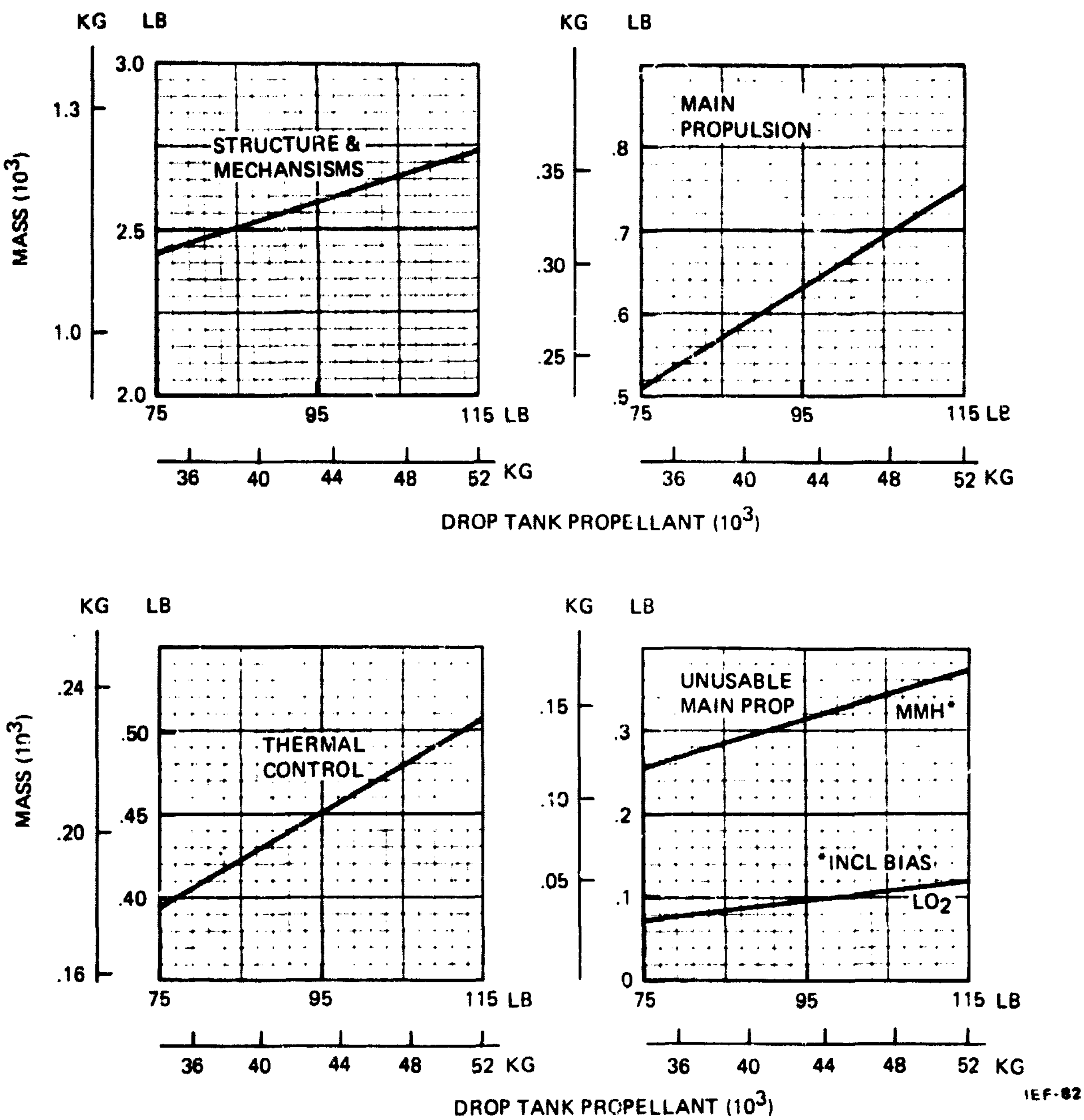


SCALING PARAMETERS A = 2 690 KG (5,930 LB) B = 0.1220 C = 0 D = 0.1725

1EF-81

Figure 1.2-32. 1-½ Stage LO_2/MMH LTV Subsystem Parametrics Main Stage OLS Mission (Sheet 1)



SCALING PARAMETERS A = 1 020 KG (2,250 LB) B = 0.04 C = 0 D = 0

Figure 1.2-32. 1½ Stage LO_2/MMH LTV Subsystem Parametrics Drop Tank Lunar Mission (Sheet 2)

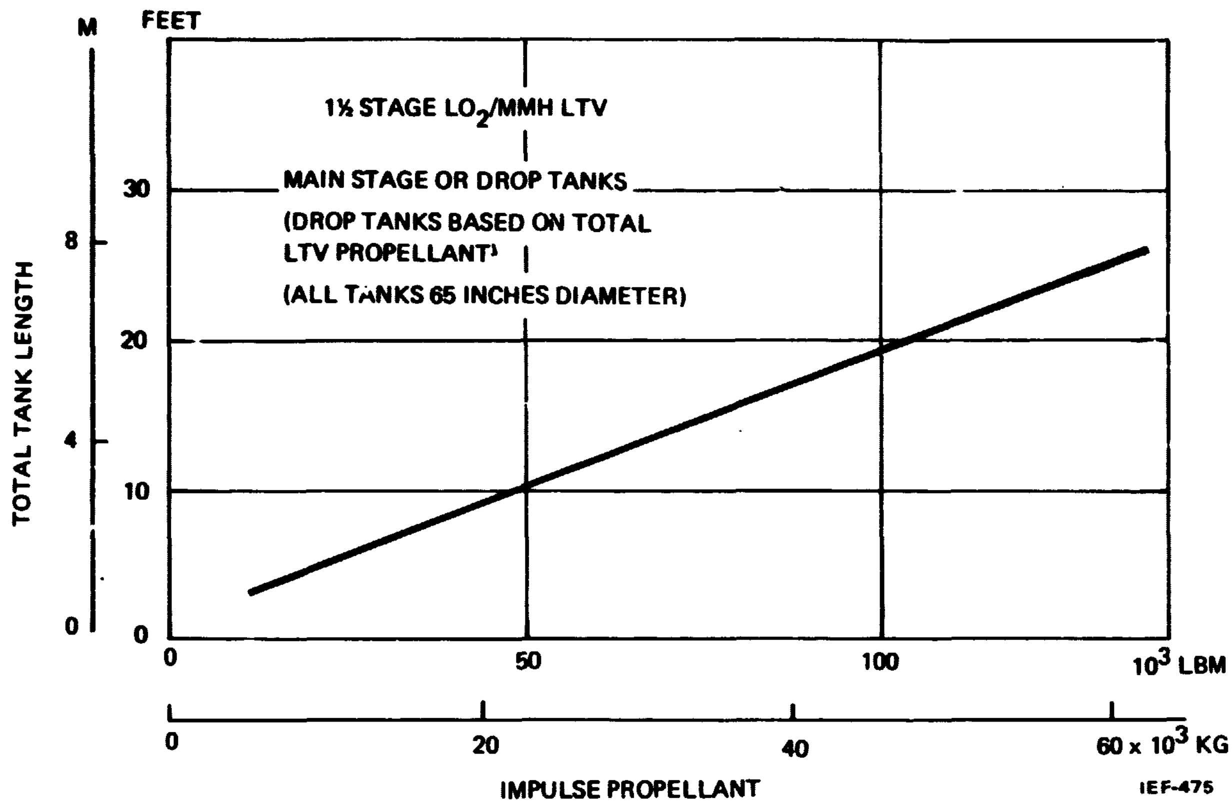


Figure 1.2-32. Sheet 3

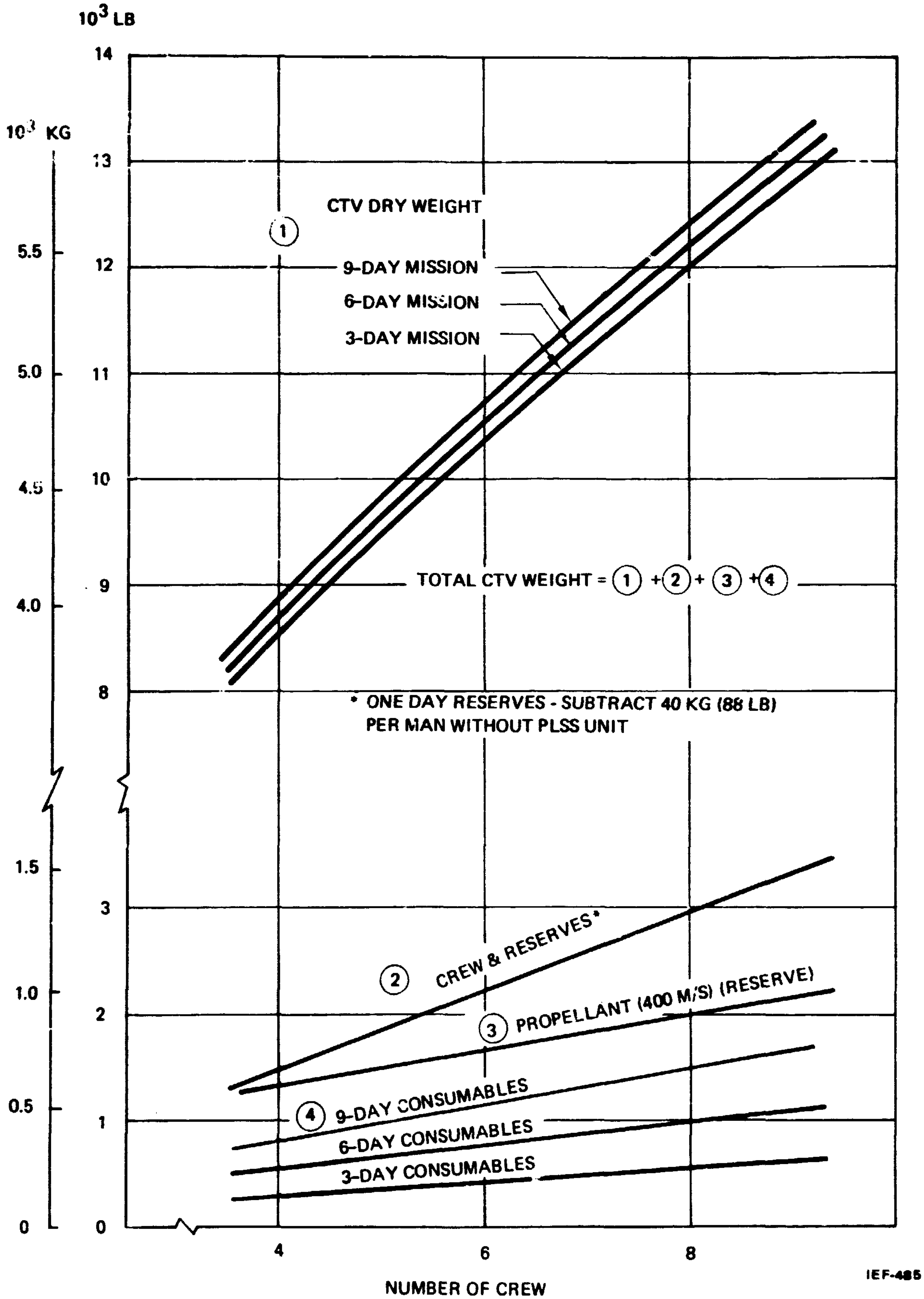


Figure 1.2-33. Crew Transfer Vehicle (CTV)

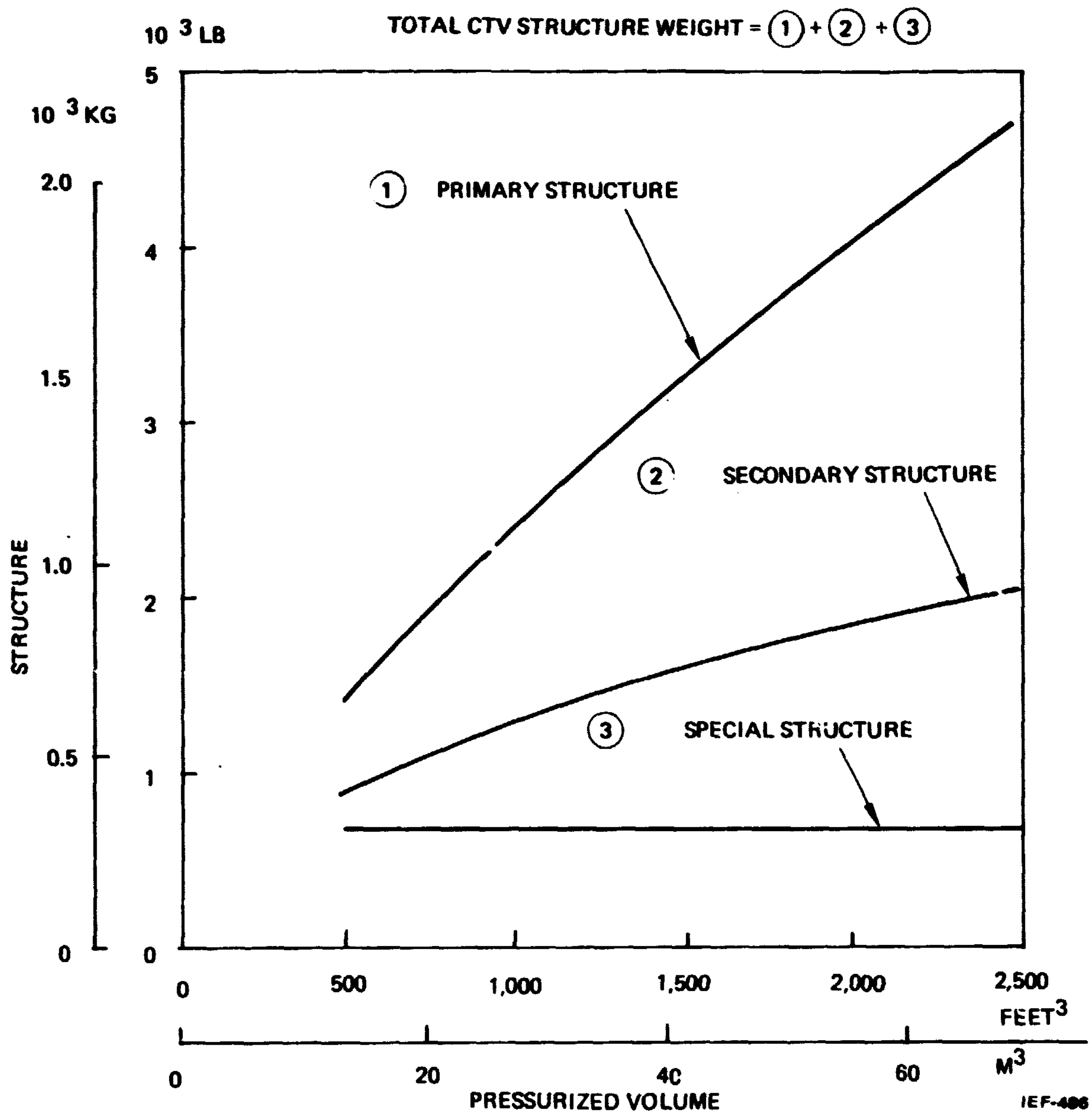


Figure 1.2-34. Crew Transfer Vehicle Structure Mass

1.2.5.2 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 1.2-35.

1.2.5.3 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 1.2-36, 1.2-37, and 1.2-38.

1.2.6 Satellite Energy Systems

All of the orbit transfer vehicles described in paragraphs 1.2.2 and 1.2.3 are potentially applicable to power satellite orbit transfer, if separate-power transfer is used (see discussion of Satellite Energy Systems, section 3.10, in Volume I), and to crew transfers to and from geosynchronous orbit. Crew transfer requirements are not well understood at present. Satellite module self-powered orbit transfer presently seems to be the most attractive method. If this is adopted, it leads to a unique propulsion system, not generally applicable to alternate uses. A concept of such a system is described in section 3.10, Volume I.

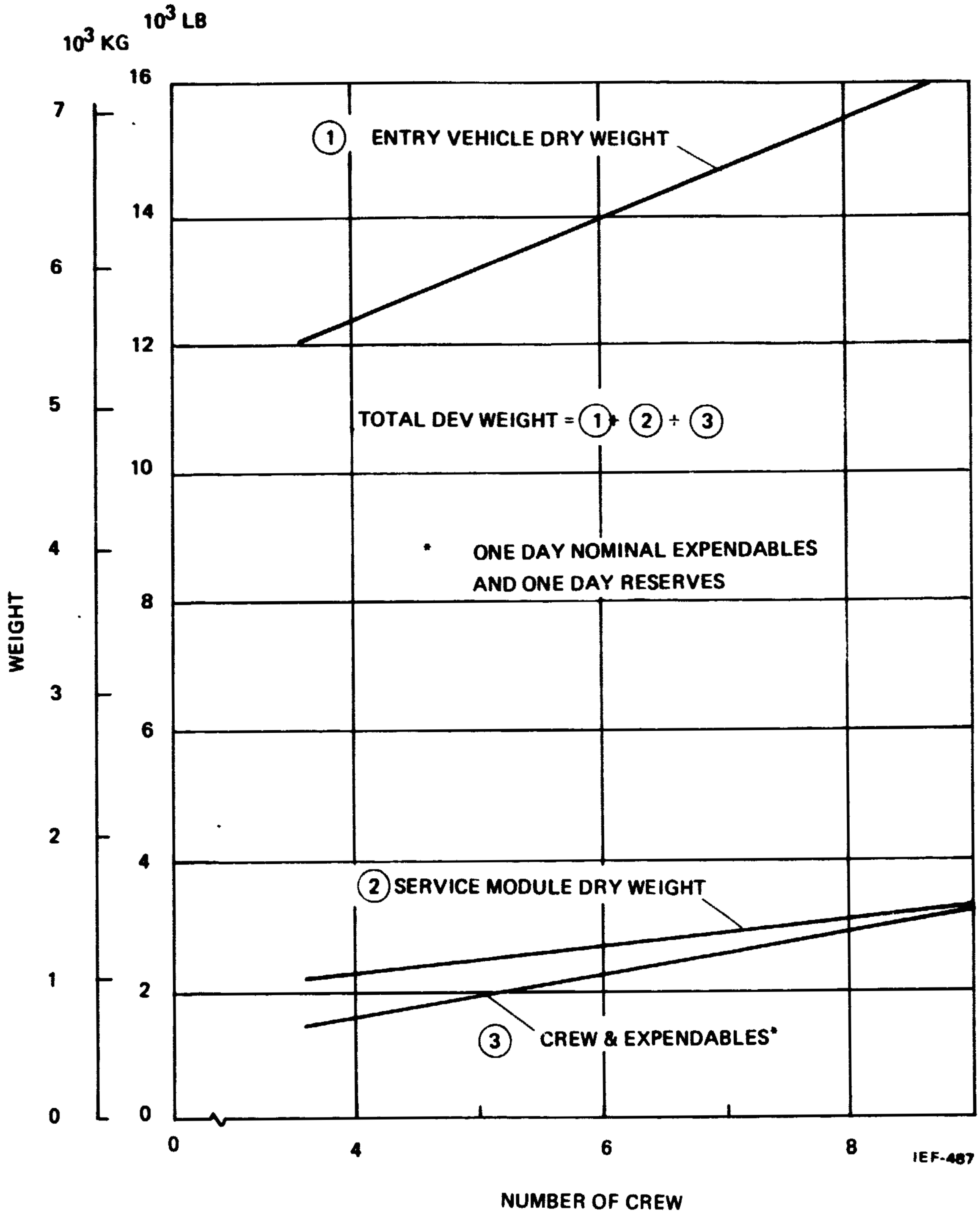


Figure 1.2-35. Direct Entry Vehicle (DEV) Apollo Shape

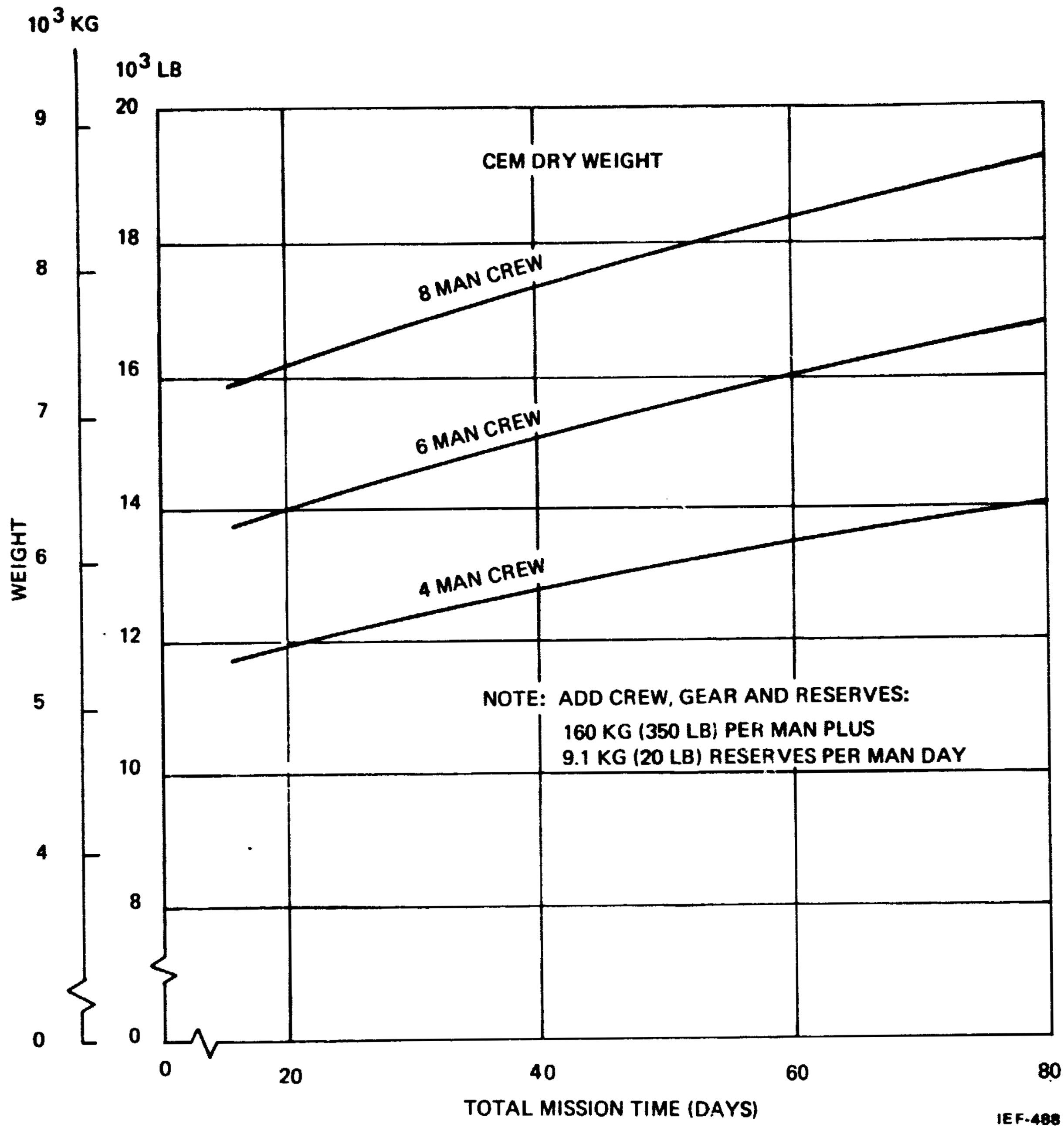


Figure 1.2-36. Lunar Crew/Equipment Module Inert Mass

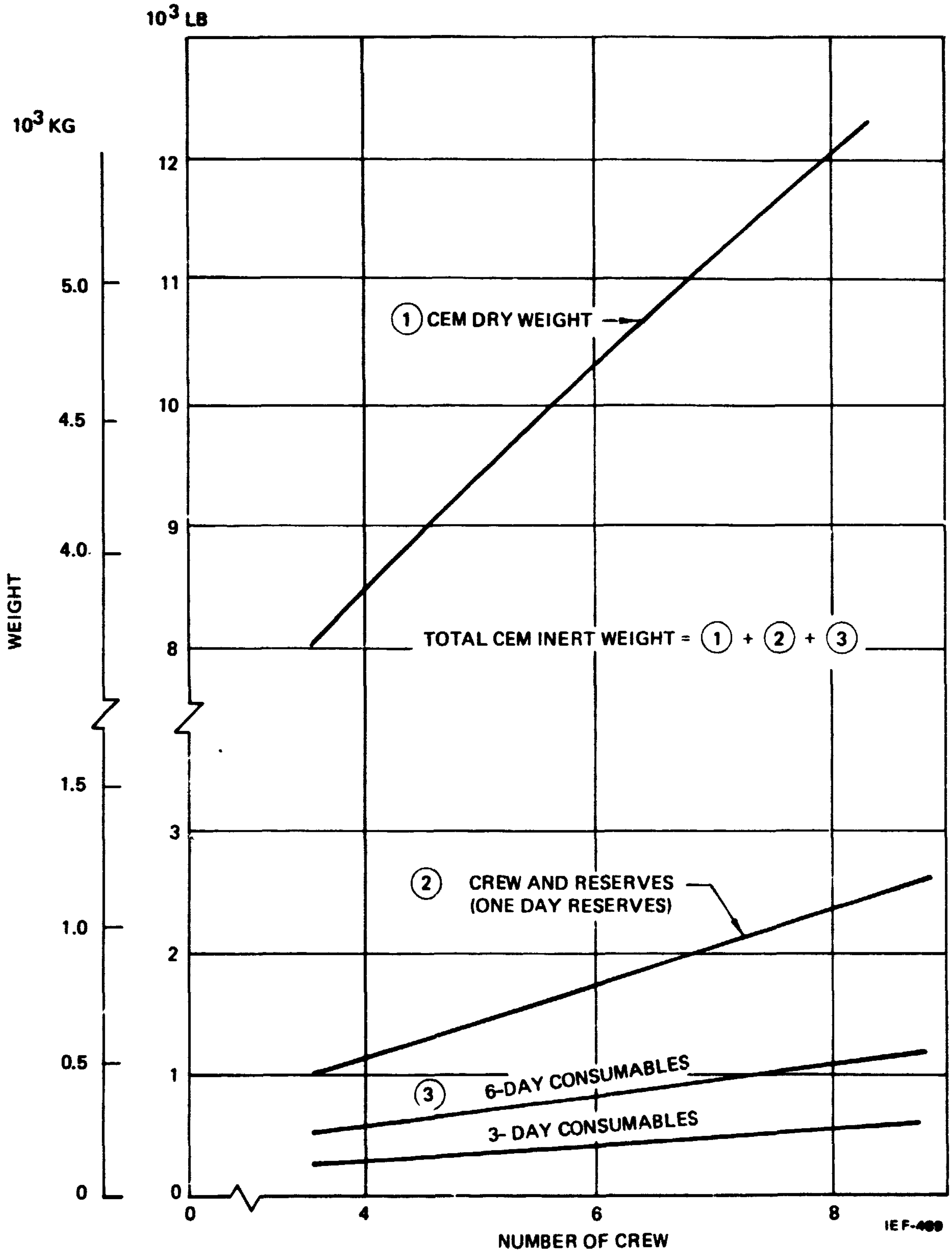


Figure 1.2-37. LSB Crew/Equipment Module (CEM)

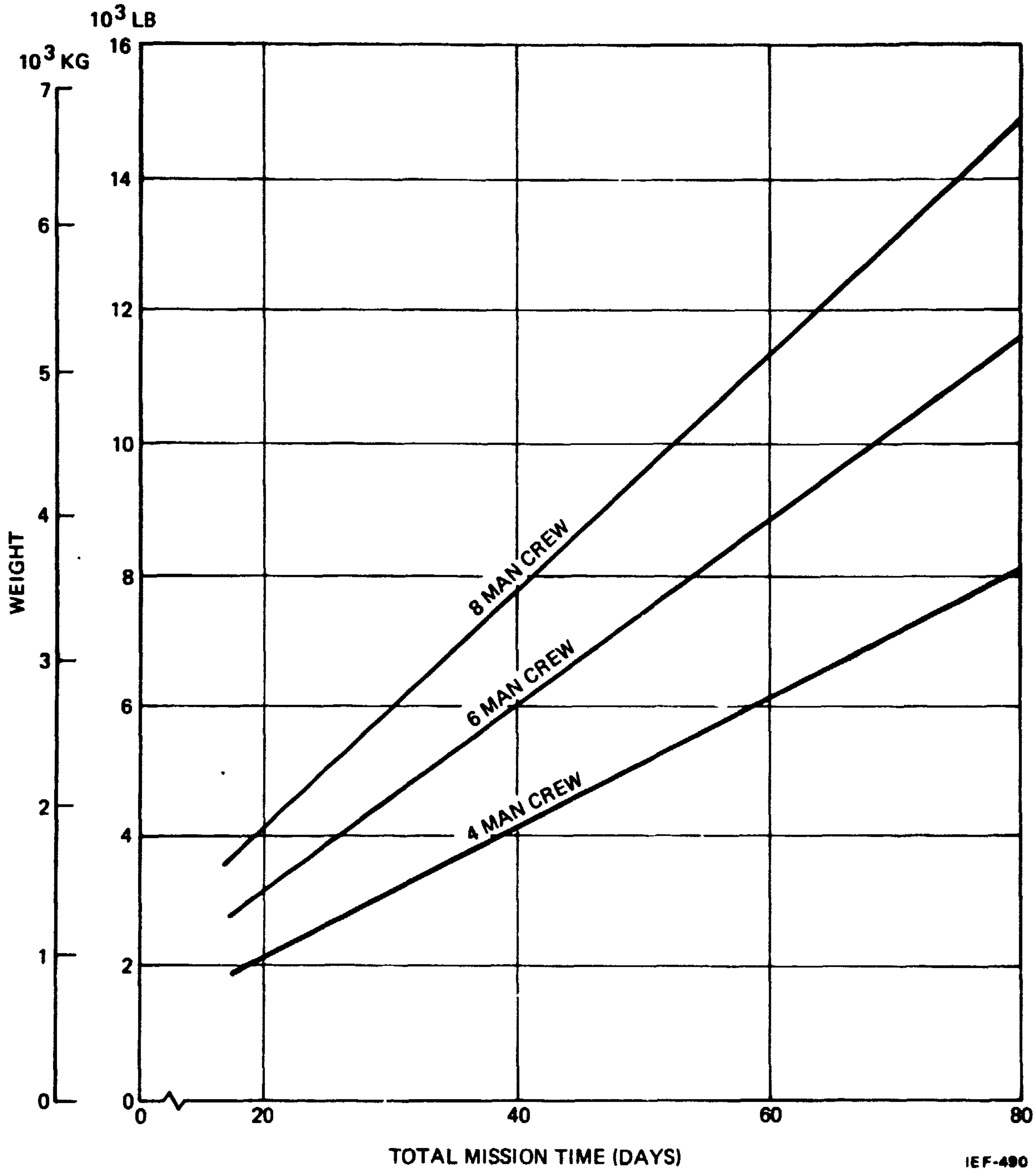


Figure 1.2-38. Lunar Crew/Equipment Module Consumables Mass

1.3 SUBSYSTEMS ANALYSES

Subsystems analyses resulted in the technology assumptions and choices stated in paragraph 1.1.2 of this volume, and in the subsystems descriptions in Volume I. Most of the subsystems choices were established on the basis of study precedent or prior use on existing systems or in earlier studies. These choices are summarized in Table 1.3-1.

A tradeoff was conducted to evaluate existing versus advanced technology for auxiliary propulsion. An advanced O_2/H_2 APS system was evaluated for the LO_2/LH_2 OTV. The schematic is shown in Figure 1.3-1. Tanks are initially filled with liquid propellants and then thermally pressurized to maintain a nominal working pressure of 1.4 Mn/M^2 (200 psia). The temperature equalizing cooling jacket on the thruster assures gas delivery to the combustion chamber, with the two gases at approximately equal temperatures, maintaining mixture control. Pumps and accumulators are assumed not required. The following assumptions were made:

	LO_2/LH_2	N_2H_4
Isp	400	220
Mixture Ratio	4.5	Monopropellant
Pressurization	Thermal	GHe
Residuals	20%	10%
Thrustors	Equal mass	

Comparison results are as follows:

	LO_2/LH_2		N_2H_4	
	kg	lb	kg	lb
Impulse Propellant	1 622	3,575	2 948	6,500
Tanks	285	628	132	290
Thrusters	163	360	163	360
Press and Vent	27	60	73	160
Propellant Feed	159	350	159	350
Residuals and Reserve	340	750	327	720
Endburn mass	974	2,148	854	1,830
Effective inert mass (includes 1/3 of impulse propellant)	1 515	3,340	1 837	4,047

Table 1.3-1 Subsystems Selections

Subsystems: Item	Alternatives	OTV LO$_2$-LH$_2$				OTV LO$_2$MMH	OTV NUC		LTV LO$_2$/LH$_2$				LTV LO$_2$/MMH		
		1 Stage	1½ Stage		Common stage	Common stage	LH$_2$	Elec.	1 Stage 4.42m (14½ feet)	1 Stage 8.23m (27 feet)	1½ Stage		1 Stage	1½ Stage	
			Core	Tanks							Core	Tank		Core	Tank
Structures & mechanism															
Main tanks	Integral	X	X	X	X	X	X	X	X		X	X	X		
	suspended									X				X	X
Body shell	Aluminum skin-stringer			X				X				X			X
	Composite honeycomb	X	X		X	X	X		X	X	X		X	X	
Docking	International standard	X	X	[1]	X	X	X	X	X	X	X	X	X	X	X
	specialized				[2]	[2]									
Main propulsion															
Pressurization	High pressure ambient H$_e$			X								X			X
	Cold H$_e$ & heat exchanger					MMH							MMH	MMH	
	Engine tap off	X	X		X	0$_2$	X		X	X	X		0$_2$	0$_2$	
	Flash boiling							X							
Auxiliary propulsion															
Propellant	Hydrazine	X	X		X	X	X	X	X	X	X		X	X	
	Bipropellant storable														
	0$_2$/H$_2$														
Standard GN&E package		X	X		X	X	X	[3]	X	X	X		X	X	
Optional GN&C equipment															
	Star tracker	X	X		X	X	X	X							
	Sun sensor	X	X		X	X	X	X	X	X	X		X	X	
	Landmark tracker								X	X	X		X	X	
	Horizon senso·	X	X		X	X	X	X							
	Radar altimeter								X	X	X		X	X	

[1] Tanks rotate into position after docking [2] Stage-to-stage [3] Low-thrust software required

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The LO_2/LH_2 system is lighter by 322 kg (710 lb.), as compared to an effective stage inert mass of about 20 000 kg (44,000 lb.). This small savings was considered not to be sufficient value to justify the risk and cost of the advanced system.

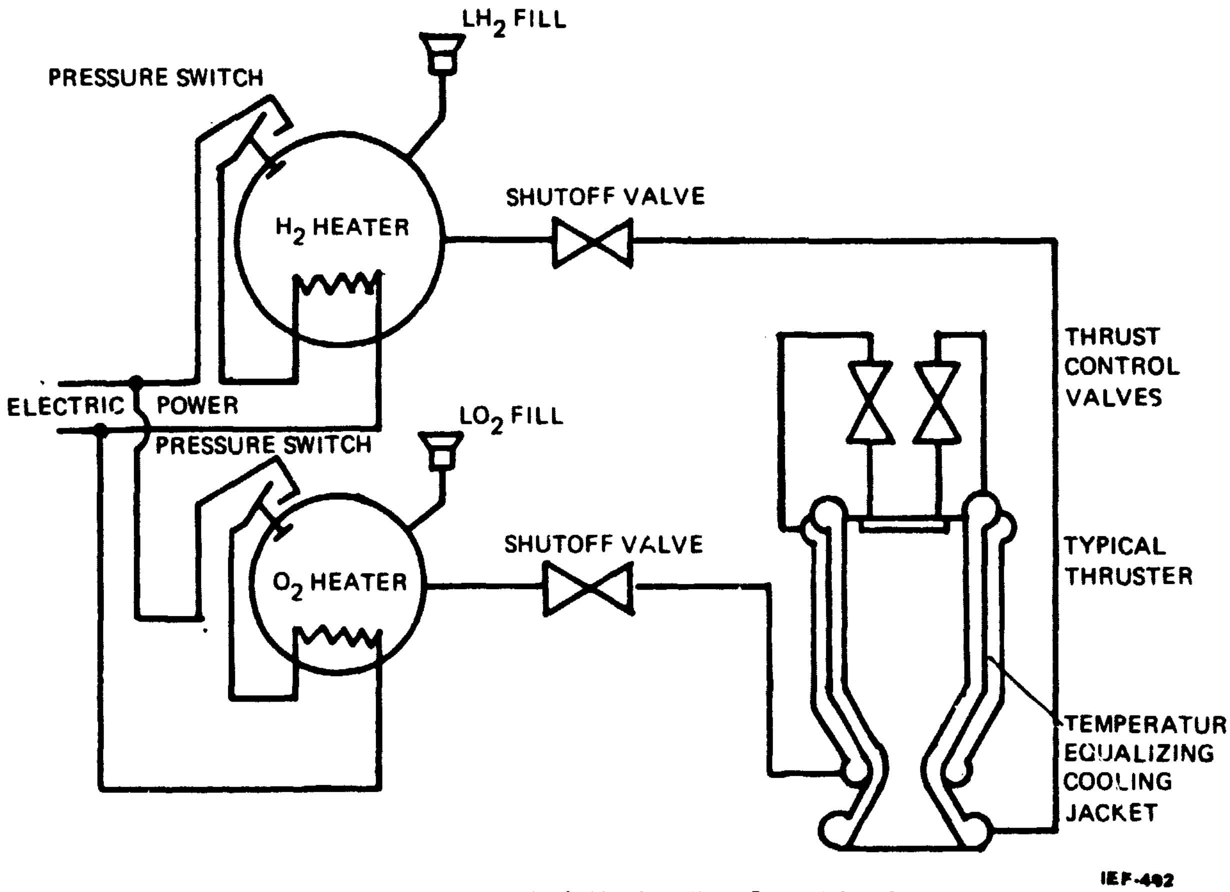


Figure 1.3-1. Advanced LO_2/LH_2 Auxiliary Propulsion Schematic

1.4 COST ANALYSES

Cost analyses were based on the Boeing PCM methodology described in Section 2.3 of Volume I. Results of the cost analyses were used to develop the higher level model reported here. Figure 1.4-1 diagrams the procedure for buildup of costs using high level CER's. The CER's are shown in Figure 1.4-2. The CER's include off-the-shelf (OTS) and modified existing hardware (MOD) factors for DDT&E flight hardware development cost estimates as defined for the point estimates used to develop the CER's. Plot points shown on the CER plots are point estimates developed by the PCM model for the various vehicles studied - they are *not* historical experience points.

Mass properties statements used as inputs to costing by this model will generally include an unallocated mass contingency. Representative historical cost growth is included in the CER's; they correlate experienced cost with experienced system element mass. The mass contingency allowance in the mass properties statements is based on historical experience and is applied to identified mass properties to project actual experience mass properties. Its cost equivalent must therefore be reflected in the cost statement. Sheet 14 of Figure 1.4-2 can be used to determine the percent of DDT&E and unit cost totals that should be added in as cost equivalent of the mass contingency.

1.4.1 Cost Element Definitions

Program Management—This element includes that effort relating to the technical and business management of the Program. It includes the contractor's effort of directing and assuring that approved plans are implemented by the responsible organizations; and controlling the program in a cost-effective and technically excellent manner.

Specific areas of effort are:

- Planning and Controls
- Finance Management
- Configuration Management
- Data Management
- Facility Coordination
- Personnel Training and Certification

System Engineering and Integration—This element includes the activities directed at assuring a totally integrated engineering effort. It includes the effort to establish system, subsystem, GSE and Test requirements and criteria, to define and integrate technical interfaces to optimize total system definition and design, to allocate performance parameters to the subsystem level, to identify, define and control interface requirements between system elements, to monitor design and equipment to determine CEI compliance, to provide and maintain system mass properties analyses, support and documentation, to develop and maintain system specification to provide parts, standards and materials and processes surveillance and to integrate product assurance activities. Fundamental to this WBS element is the documentation of system-level design requirements as derived from NASA-established requirements and guidelines and through functional analyses.

Specific areas of effort are:

- System Design and Integration
- Configuration
- Flight Hardware Requirements
- Operations Requirements
- GSE Requirements
- System Test Requirements
- Mass Properties
- Interfaces
- Materials, Processes, and Standards
- Product Assurance
- Service and Maintenance Requirements

Software–This element includes the costs of the design, development, production, checkout, maintenance and delivery of computer software. Included are test, on-board and mission or flight software.

GSE–This element includes the costs to design, develop, fabricate, assemble, test, and deliver all ground support equipment. Also included under GSE are mockups and simulators where required. Cost of development of test procedures and reports associated with the acceptance and qualification of GSE are included.

Flight Hardware–This element includes the costs to design, develop, fabricate, assemble, and test all flight article subsystems, the assembly of these subsystems and the test and checkout of the flight article. Included are the costs associated with all test procedures and reports preparation and the Quality Control inspection effort. Also included are costs of operation/test-unique support equipment (including factory support and special test equipment), and the cost of handling and transportation of items between operation/test locations.

Ground Test Hardware—This element includes the cost of engineering liaison, fabrication, assembly and test of ground test hardware. Ground test hardware includes the static, dynamic, thermal and firing (if required) test articles. *Excluded* is engineering subsystem design effort.

Flight Test Hardware—This element includes the fabrication, assembly and checkout of the flight test vehicle(s) including spares to support the test.

Test Labor—This element is the manpower to conduct the ground and flight tests.

Tooling—This element includes (a) initial and (b) production (if required) tooling jigs and fixtures. Initial tooling is that needed to fabricate and assemble the test hardware and first unit. This is "soft" tooling. Production tooling is "hard" tooling designed for repetitive use in fabricating and assembling recurring production units. Production tooling includes sustaining and replenishment tooling.

Spares—This element includes the costs of developing and documenting requirements for, and the fabrication, assembly, test, storage, delivery, and accountability of spare components, assemblies, or subsystems to be used as test, production or mission support spares. Excluded are production spares, such as fasteners, electronic parts, etc.

DDT&E (Non-Recurring Cost)—This element consists of the "one-time" cost of designing, developing, testing, and evaluating an item. Specifically it includes: development engineering and development support, major test hardware, captive and ground test, flight test, ground support equipment, tooling and special test equipment; manufacturing, test, mission control or launch site activation (if required), initial spares and other program peculiar costs not associated with repetitive production.

First Unit Cost (Recurring Cost)—This is the first production-configured flight or mission article in a hardware production program. If there is only one designated flight or mission article in the program, this would be called the first unit as differentiated from any developmental hardware such as a prototype. First unit cost is that cost associated with producing the first flight or mission article through acceptance of the hardware by the government and includes all costs associated with: (1) the fabrication, assembly and checkout of flight or mission hardware, (2) ground test and factory checkout of flight or mission hardware.

NOTE: Initial spares are priced in DDT&E and cover the support of the first unit; additional spares would be a function of a production program for the vehicle and would be included in recurring production costs for spares. Maintenance of tooling and special test equipment would also be part of production recurring costs.

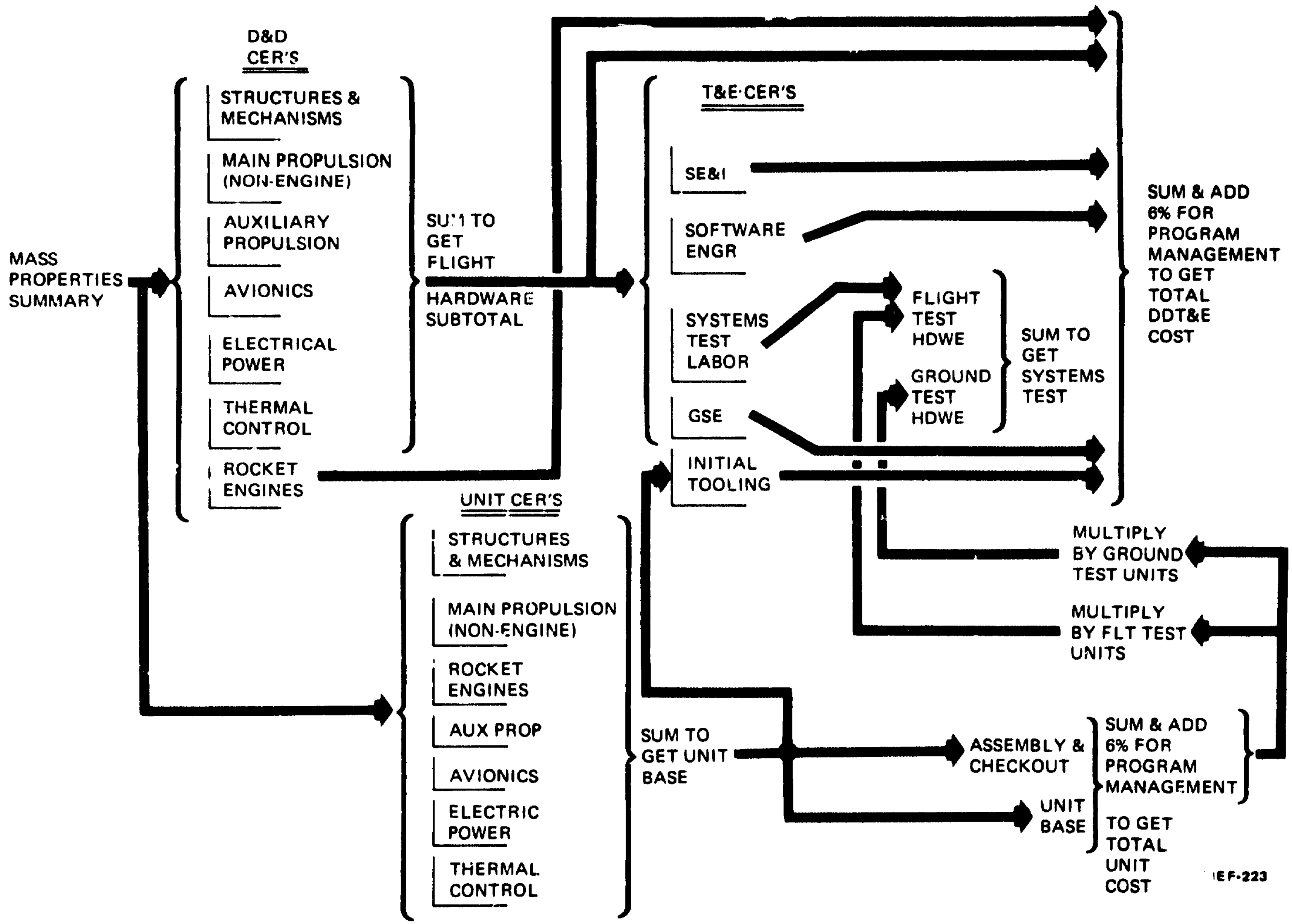


Figure 1.4-1. FSTS High Level Cost Model

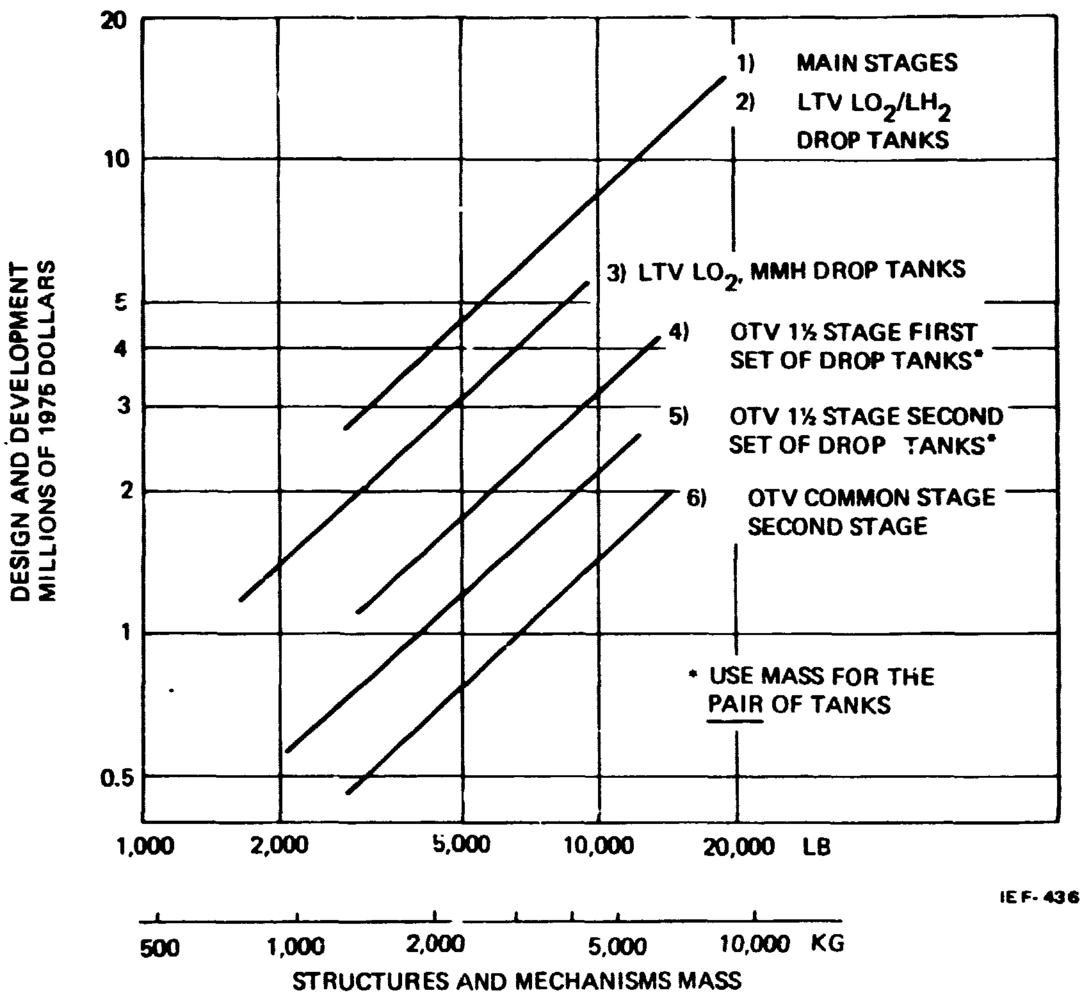


Figure 1.4-2. Structures and Mechanisms Design and Development (Sheet 1)

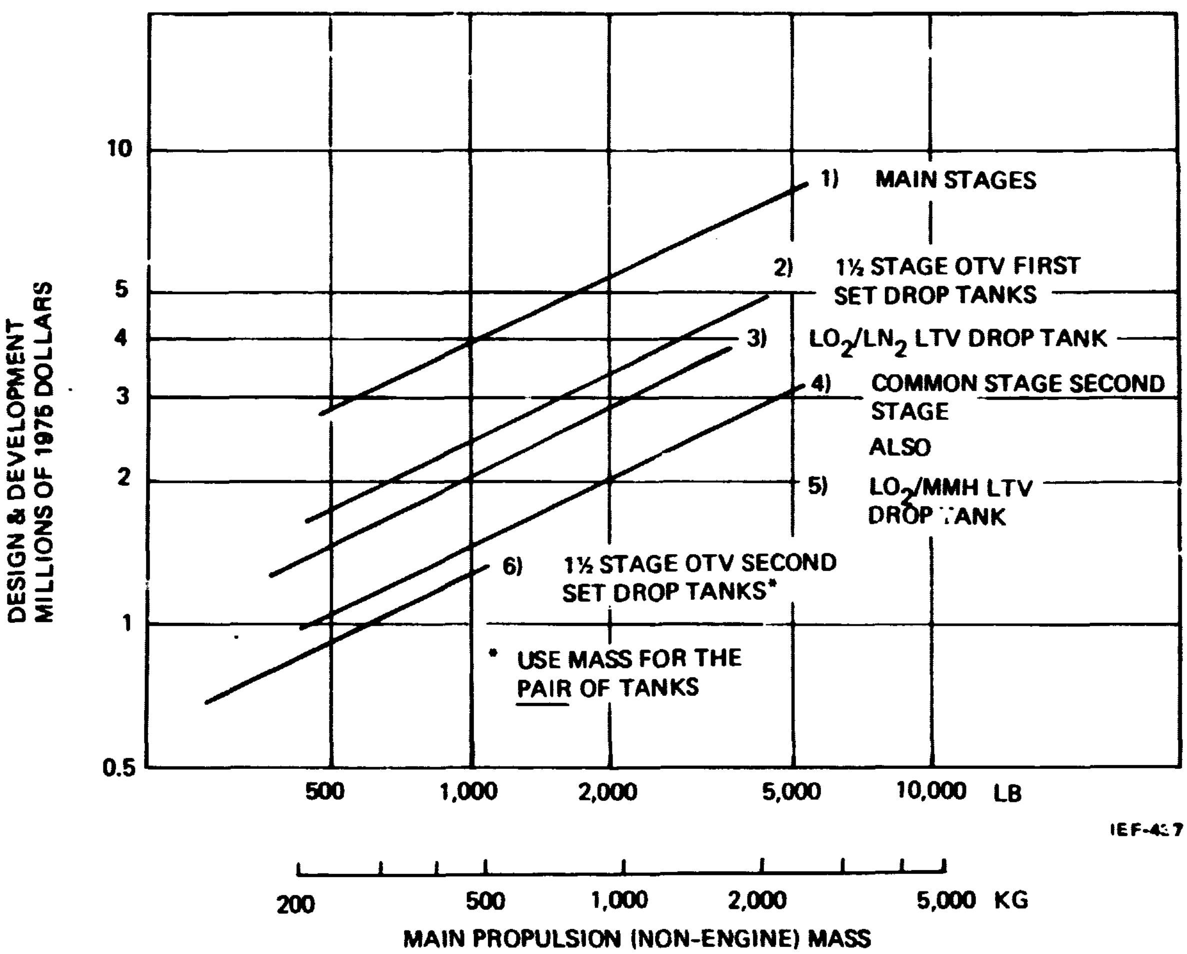
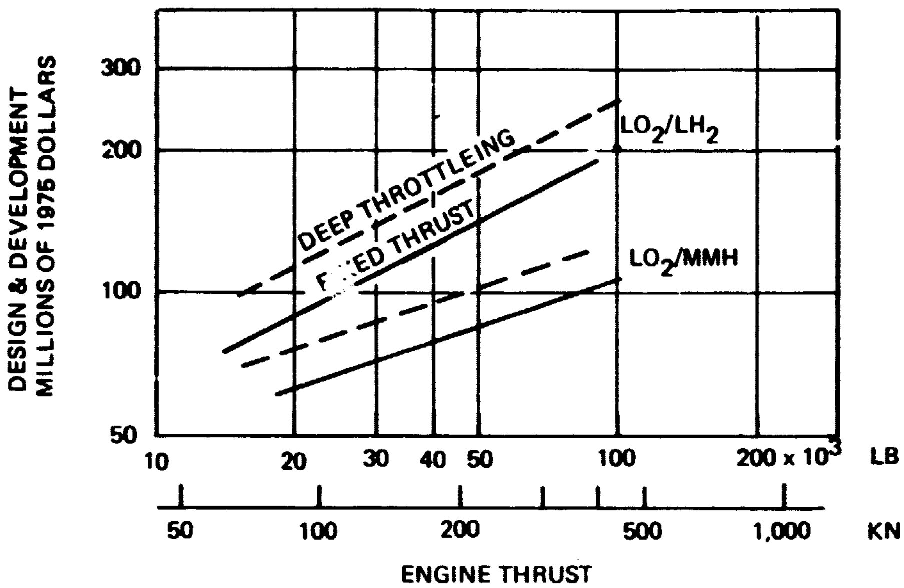


Figure 1.4-2. Main Propulsion (Non-Engine) Design and Development (Sheet 2)



HYDRAZINE AUXILIARY PROPULSION (APS)
DESIGN & DEVELOPMENT

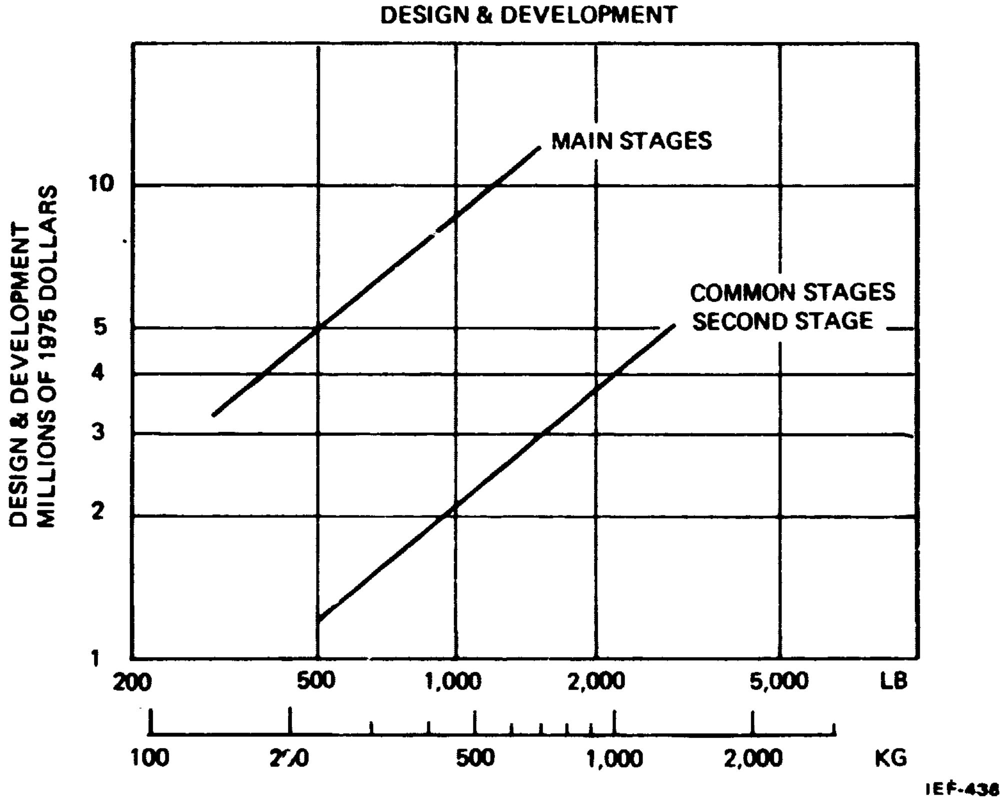


Figure 1.4-2. Rocket Engines Design and Development (Sheet 3)

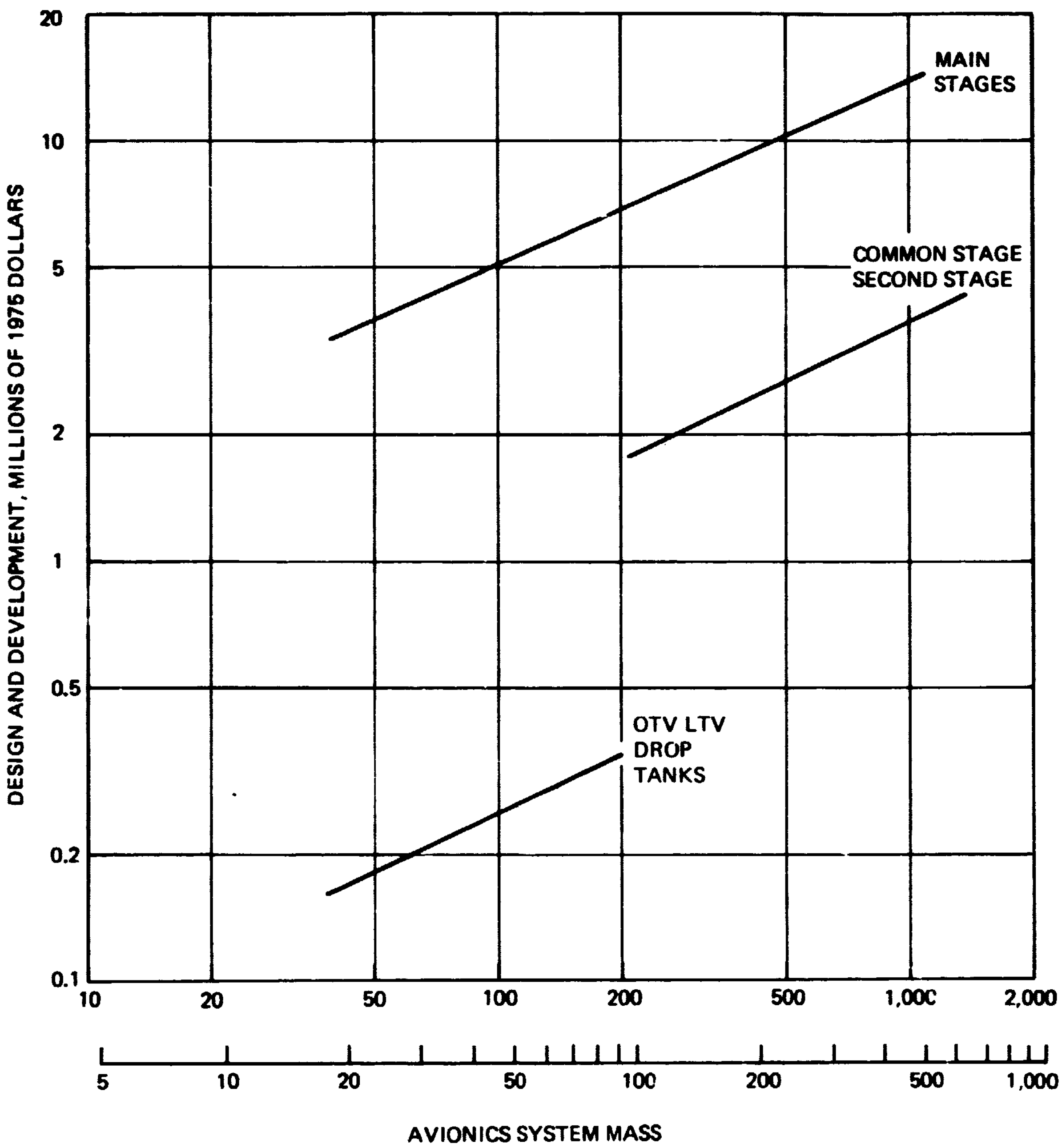


Figure 1.4-2. Avionics Design and Development
(Sheet 4)

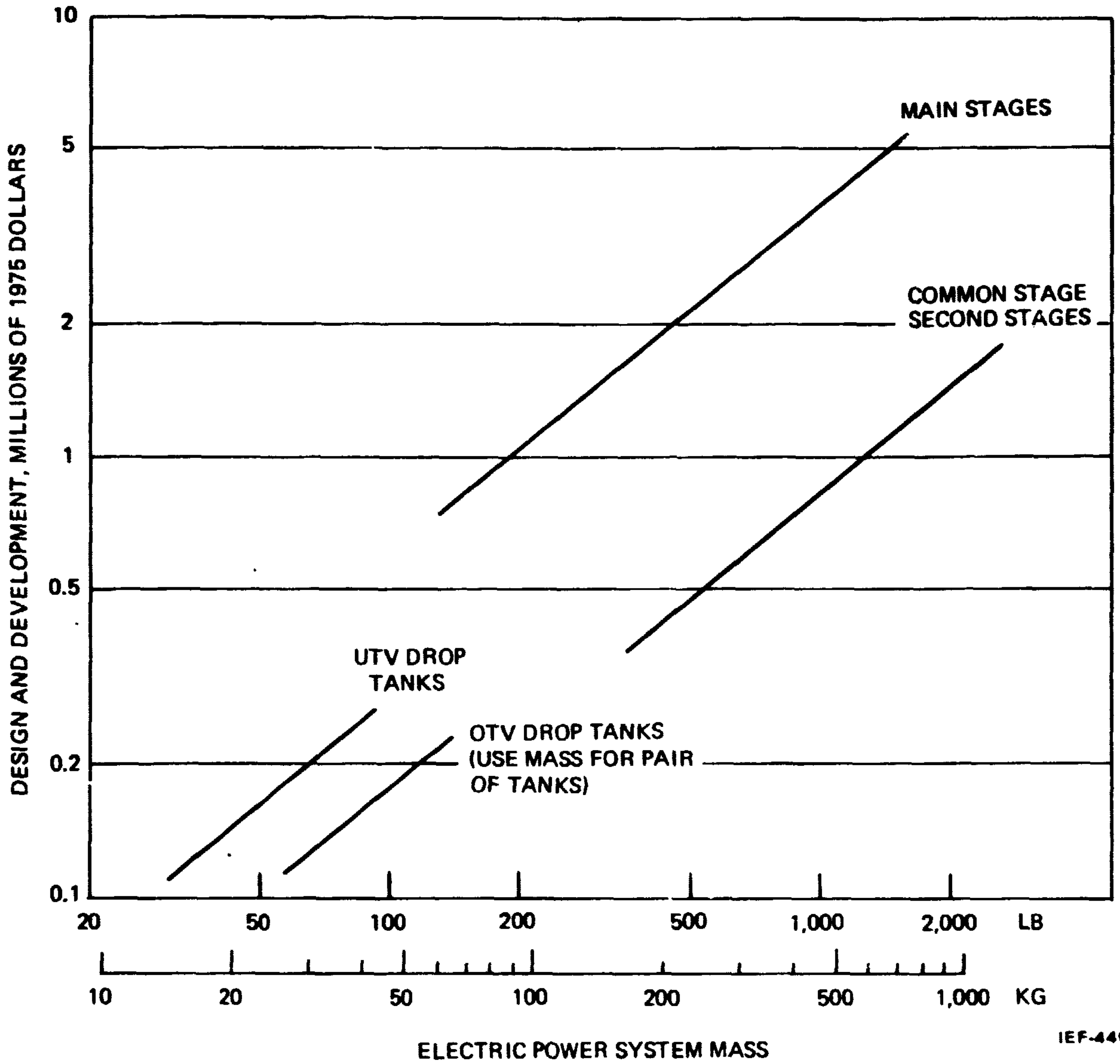


Figure 1.4-2. Electrical Power Systems Design and Development (Sheet 5)

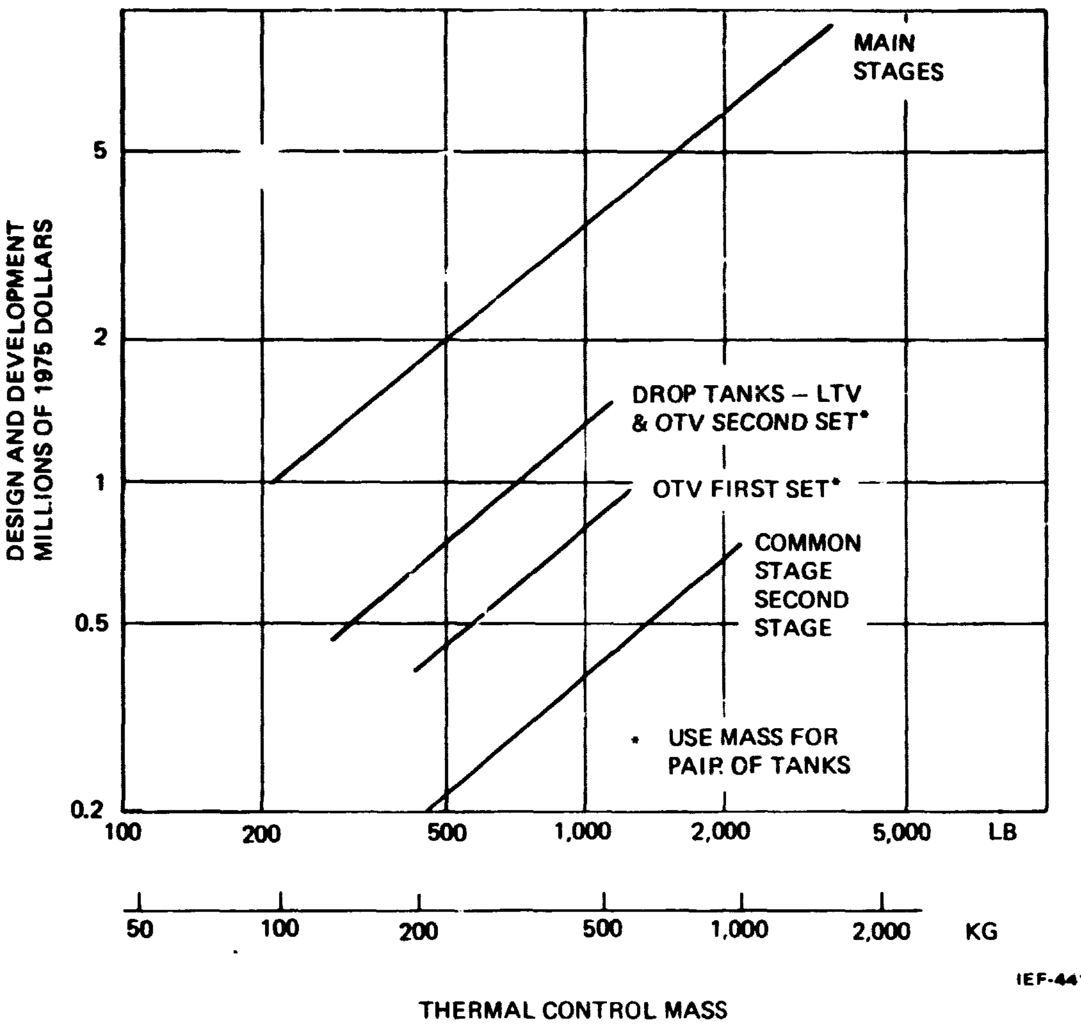


Figure 1.4-2. Thermal Control System Design and Development (Sheet 6)

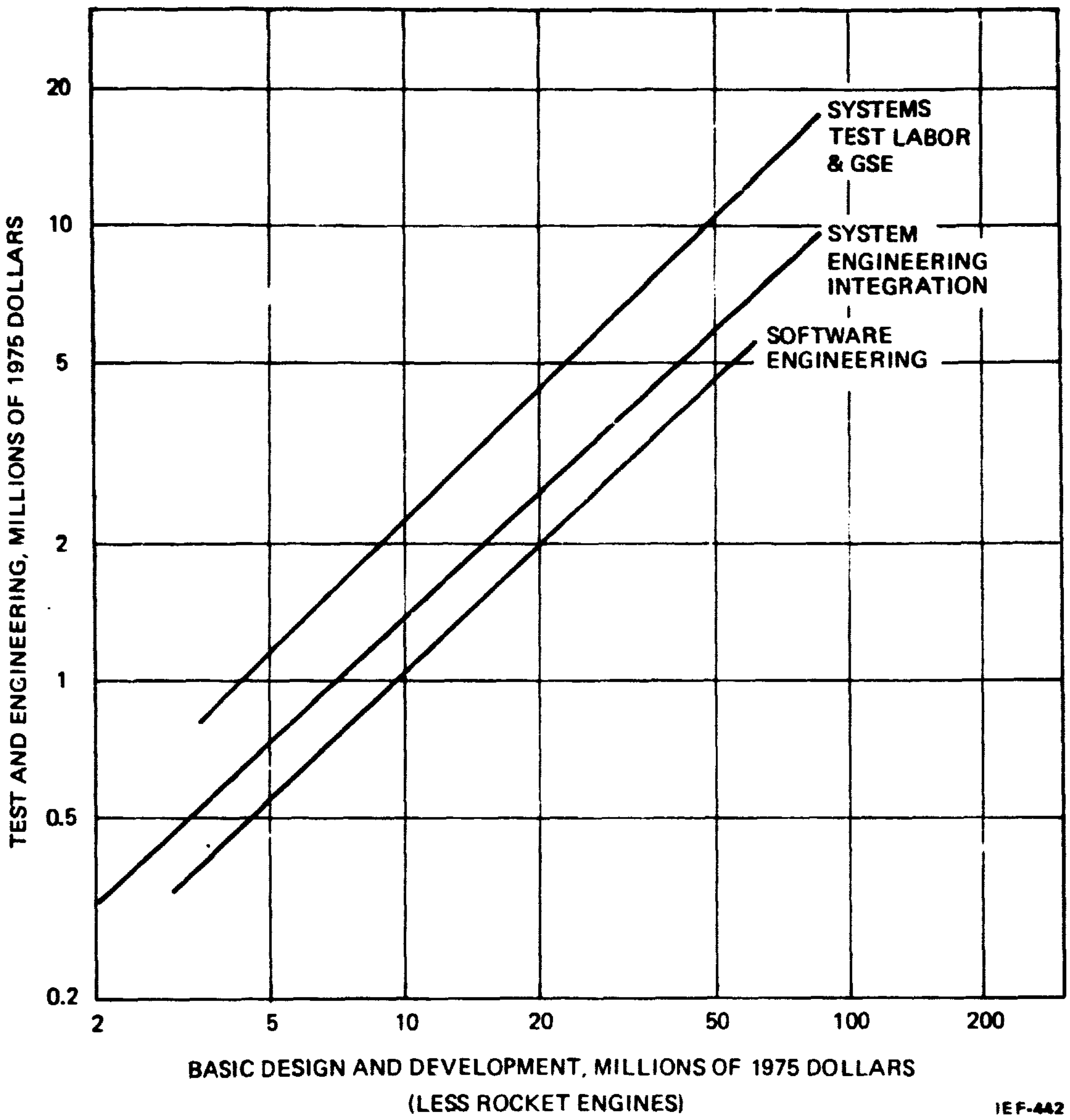
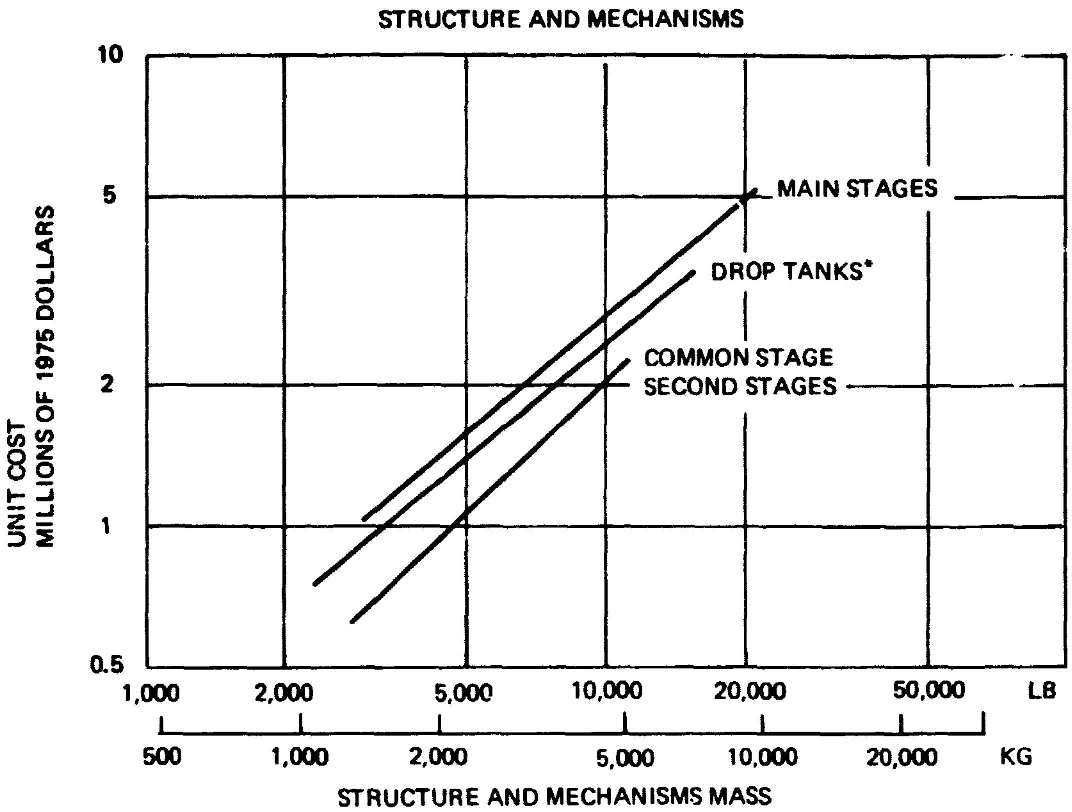


Figure 1.4-2. Test and Engineering (Sheet 7)



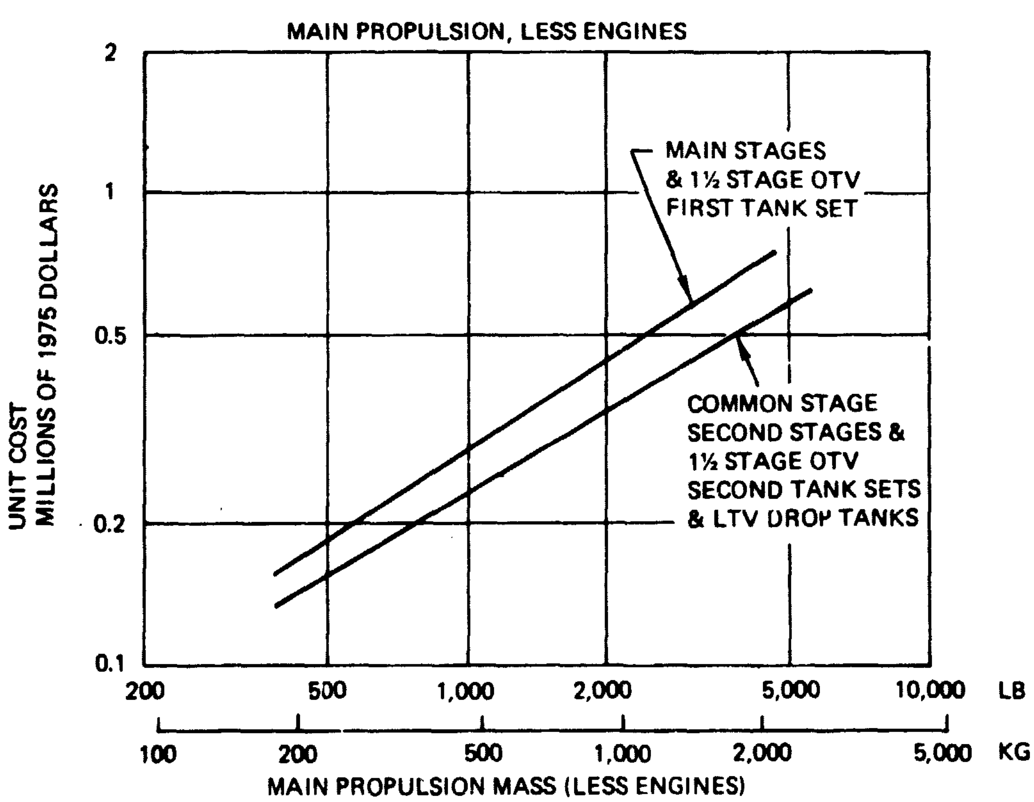
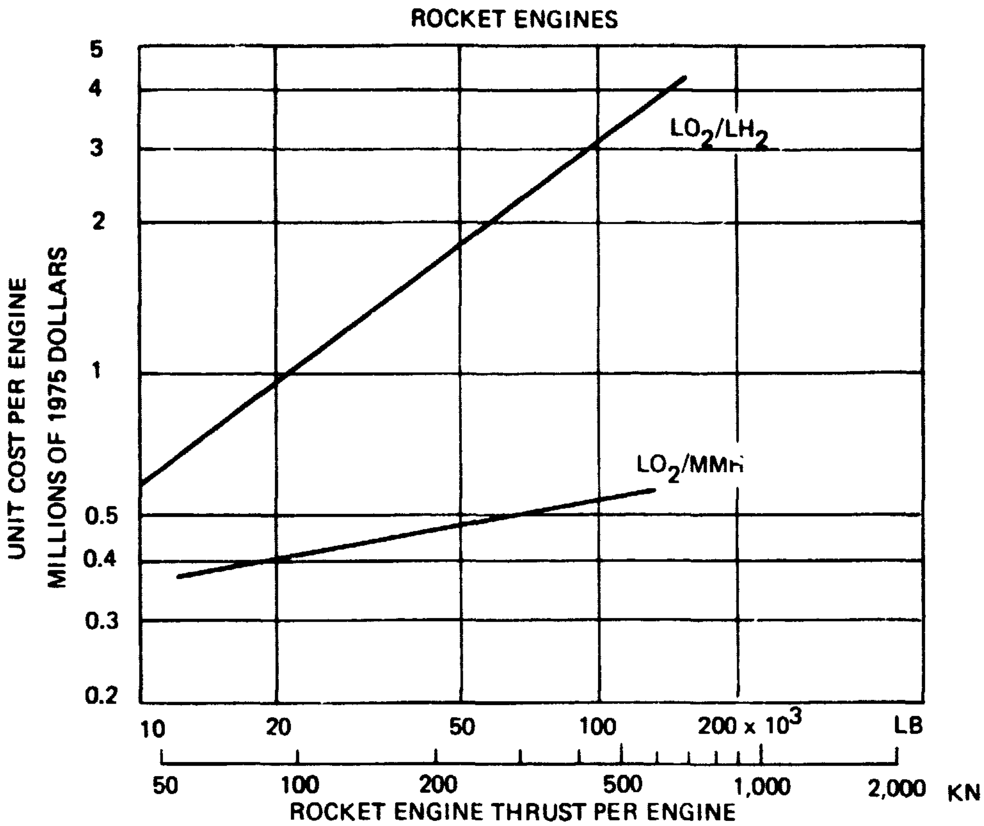


Figure 1.4-2. Subsystem Parametrics
(Sheet 8)



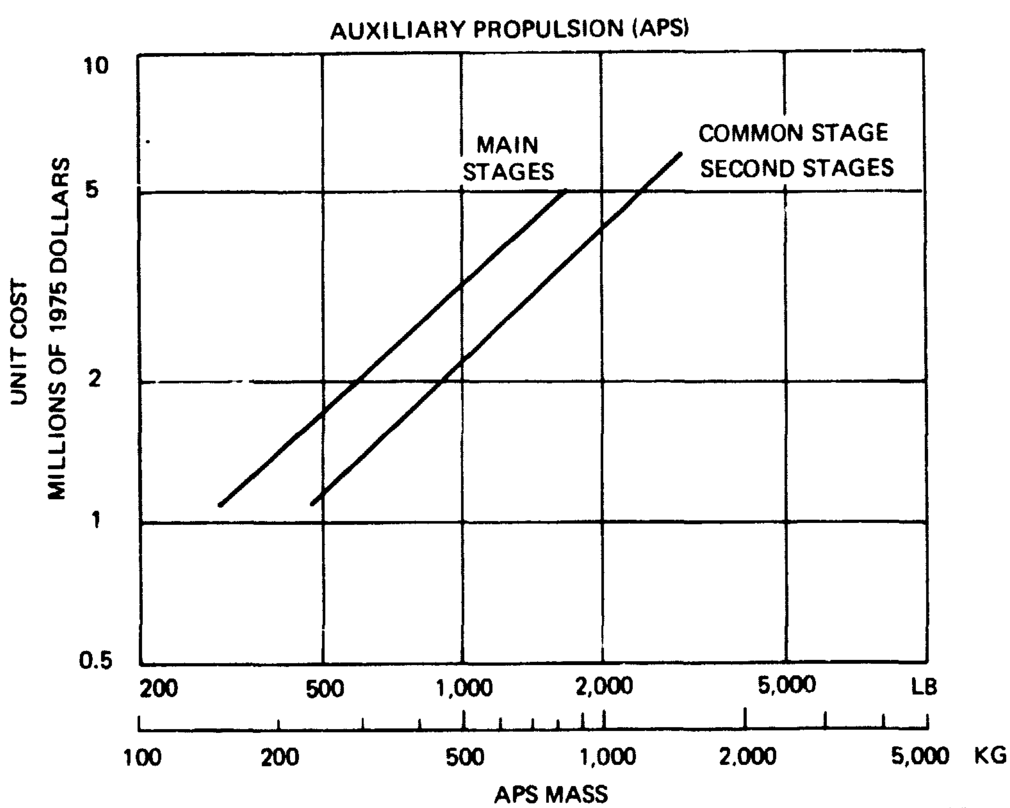


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Figure 1.4-2. Subsystem Parametrics (Sheet 9)

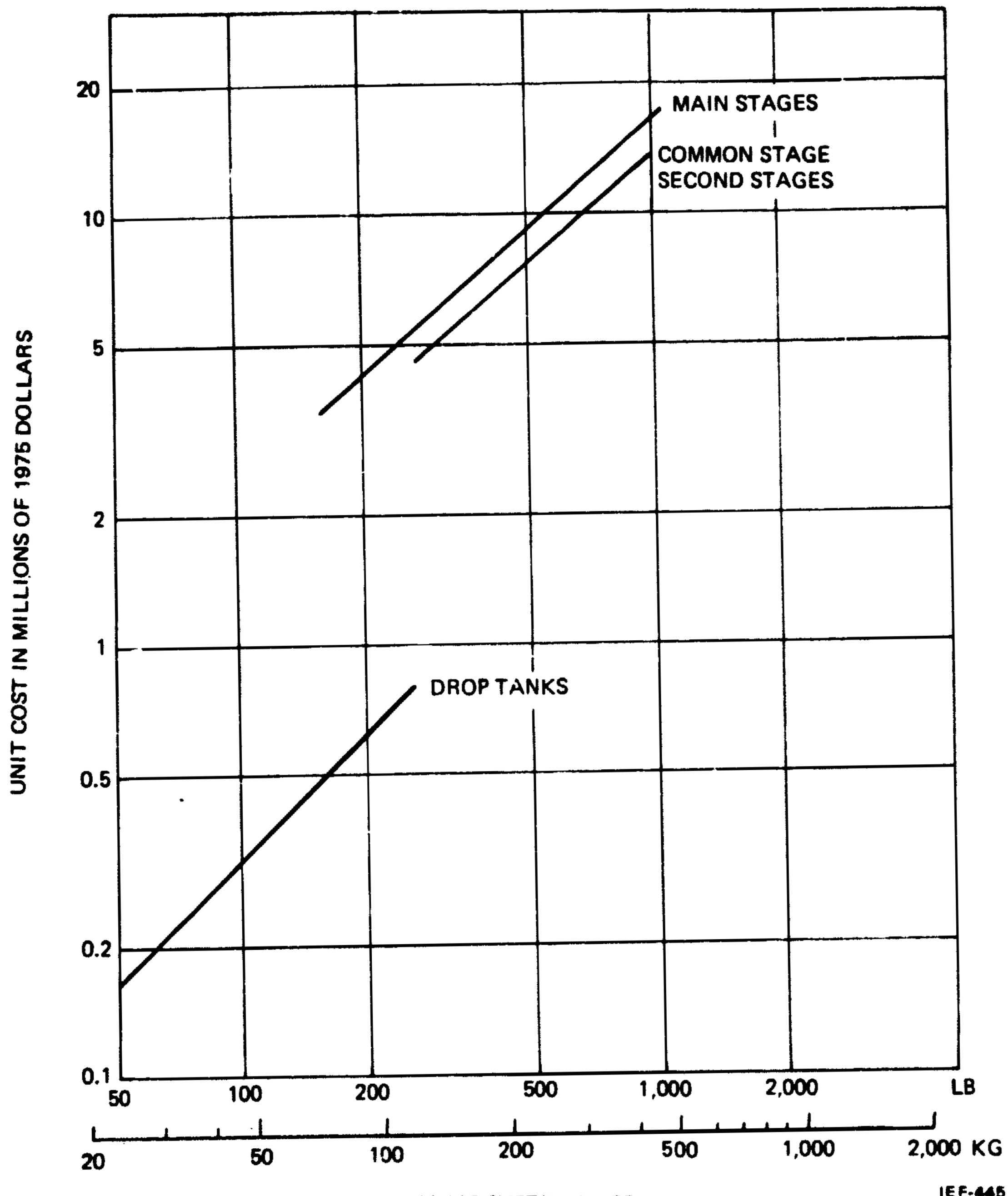


Figure 1.4-2. Avionics System Unit Cost (Sheet 10)

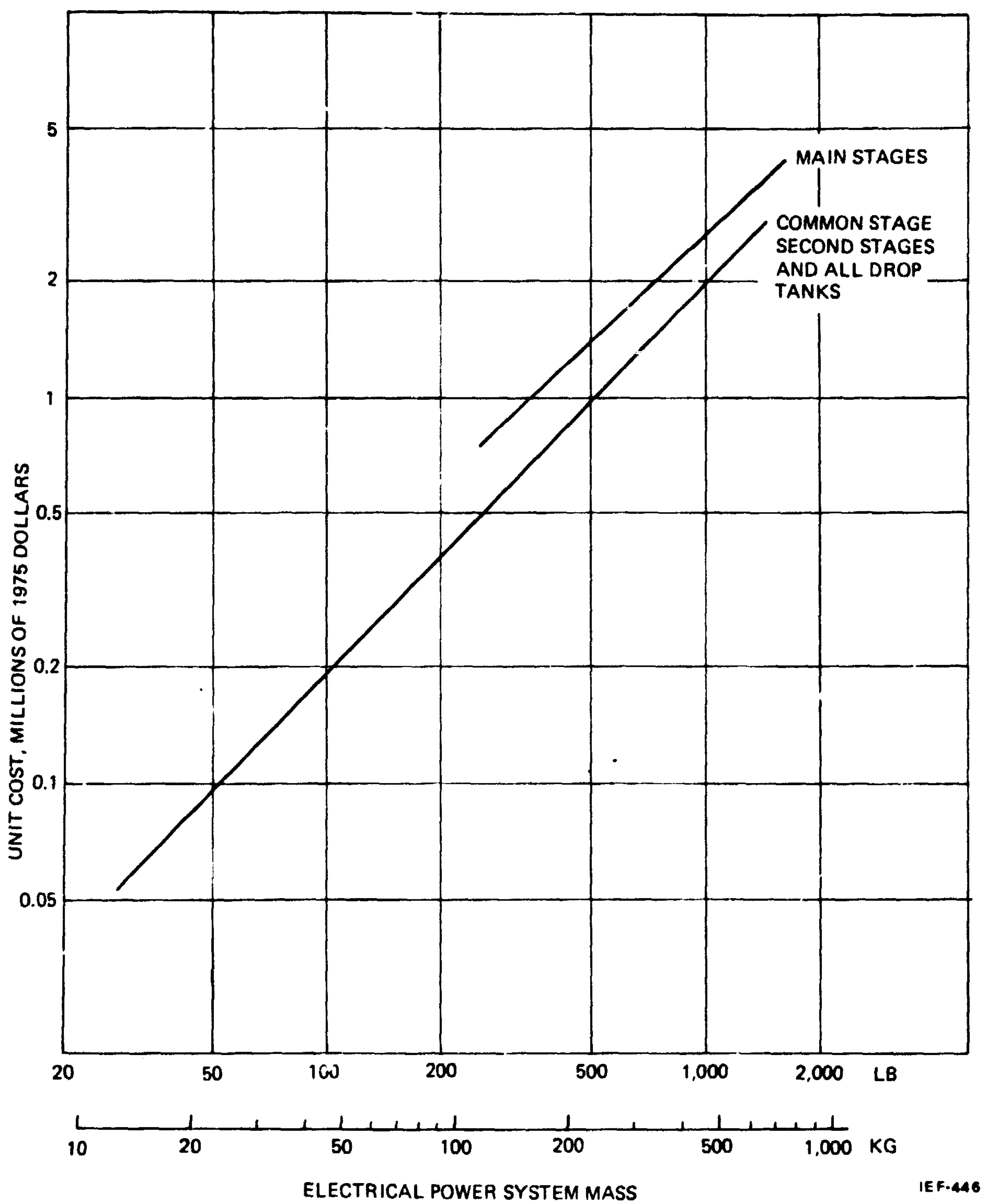


Figure 1.4-2. Electrical Power System Unit Cost (Sheet 11)

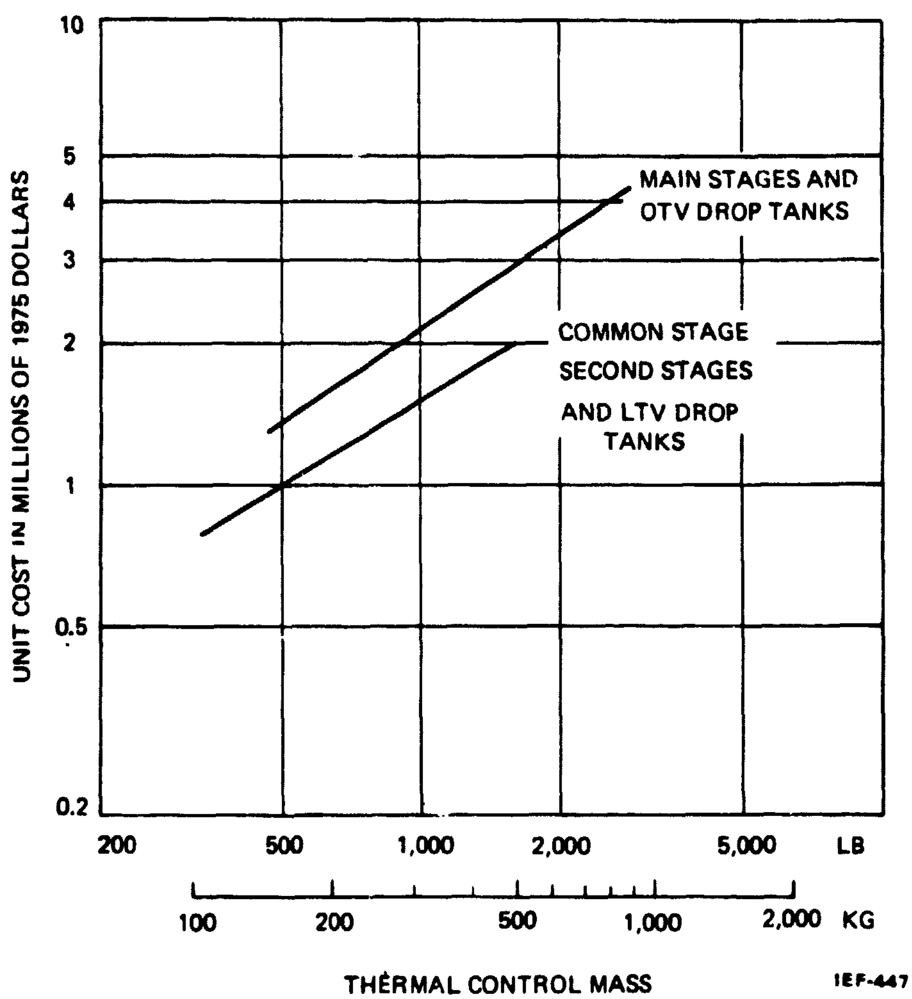


Figure 1.4-2. Thermal Control Unit Costs (Sheet 12)

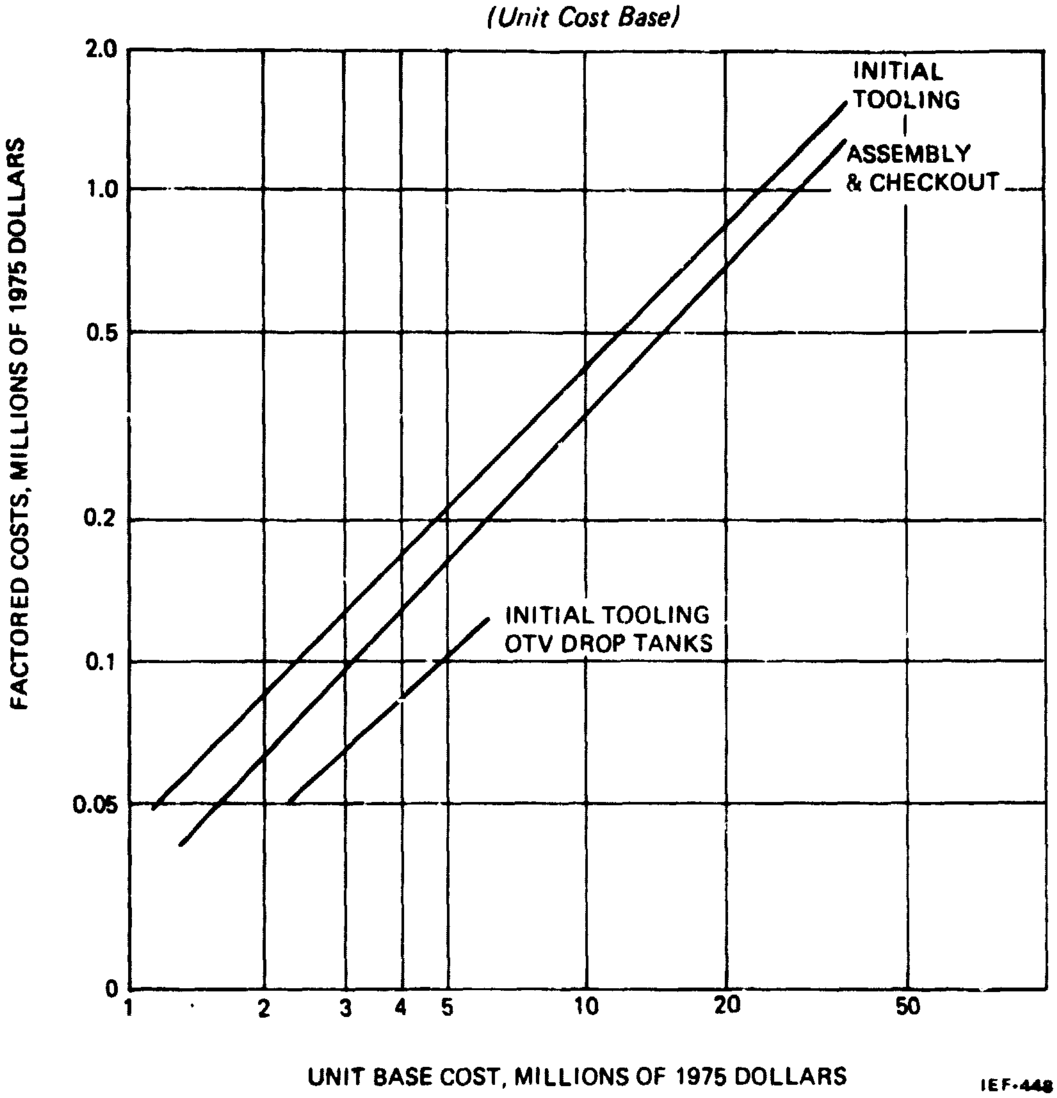


Figure 1.4-2. Factored Costs
(Sheet 13)

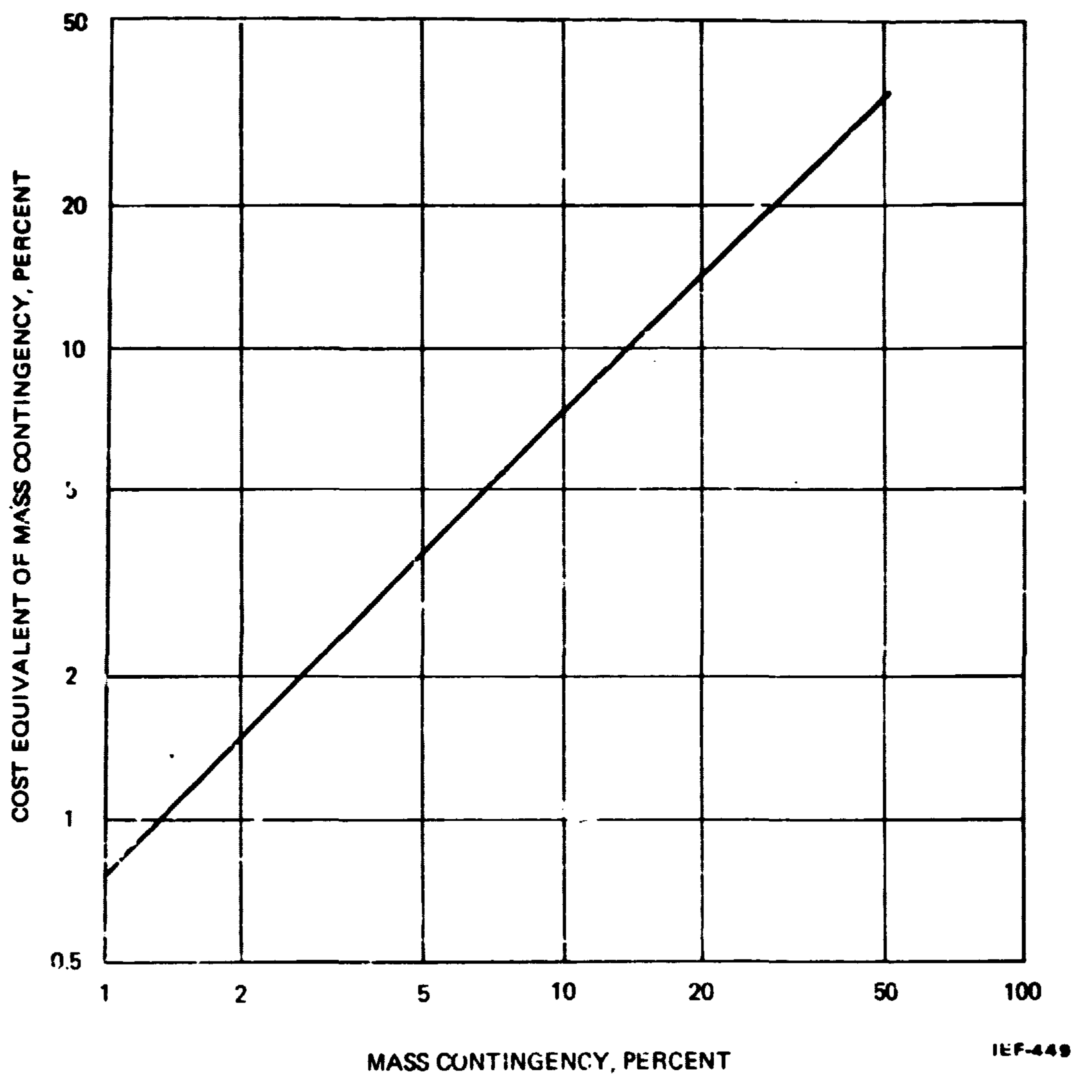


Figure 1.1-2. Cost Equivalent Factor for Mass Contingency (Sheet 14)

2.0 METHODOLOGY NOTES

2.1 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.

2.1.1 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.

2.1.1.1 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.

2.1.1.2 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

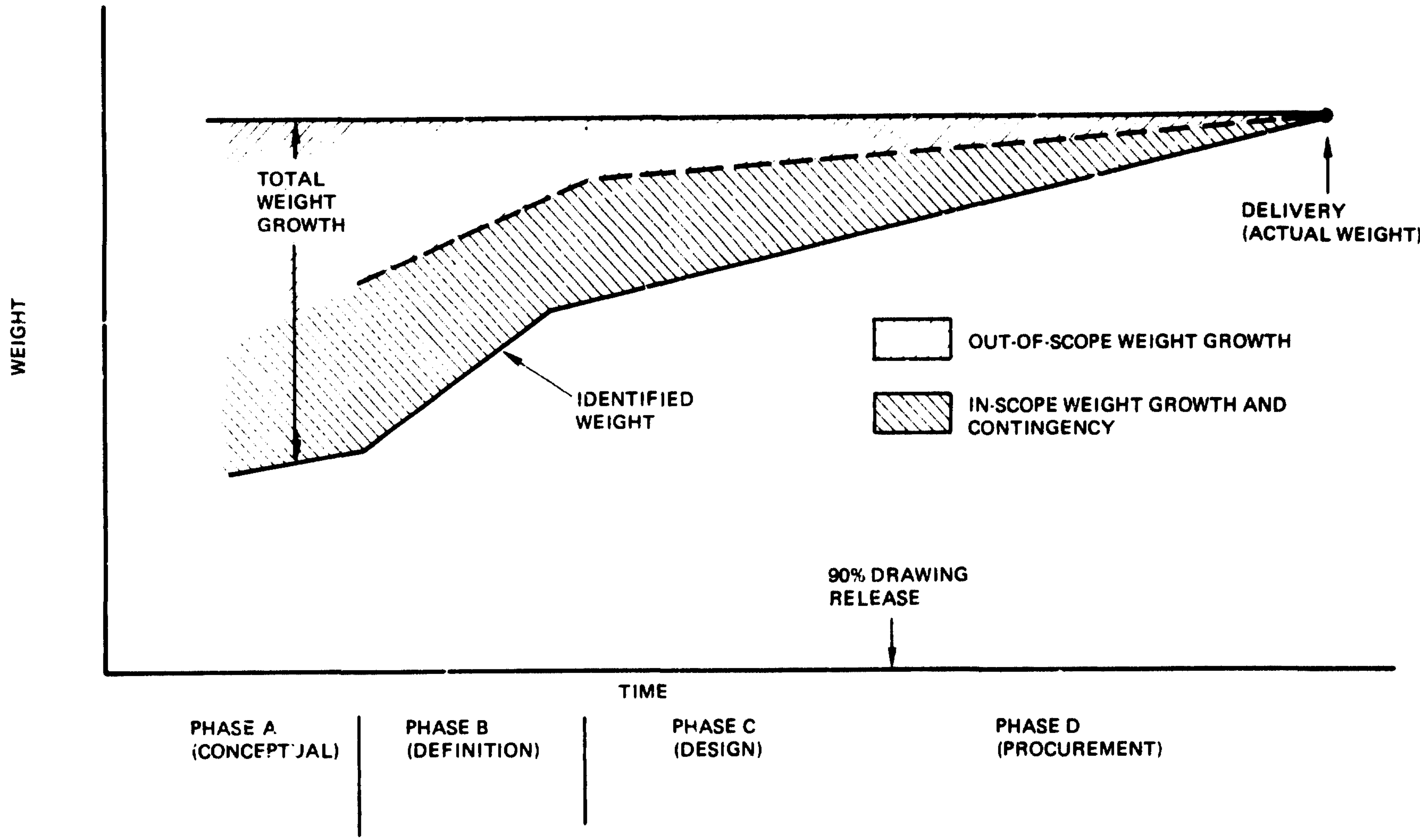


Figure 2-1. Typical Aerospace Vehicle Weight History

A first generation spacecraft is the first of its kind and, as such, would be expected to have a higher weight growth han 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.

2.1.1.3 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.

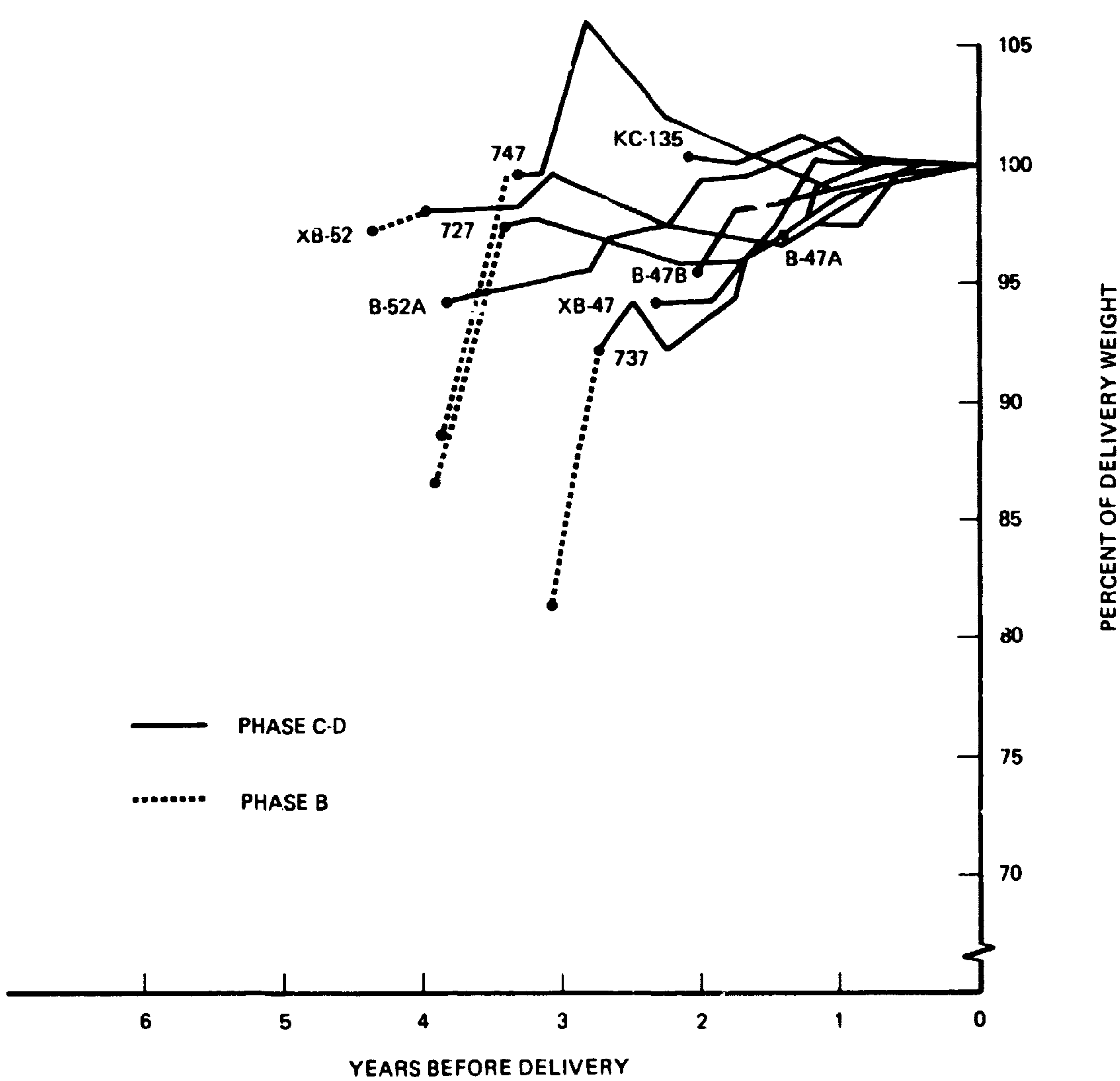


Figure 2-2. Weight Growth – Jet Aircraft

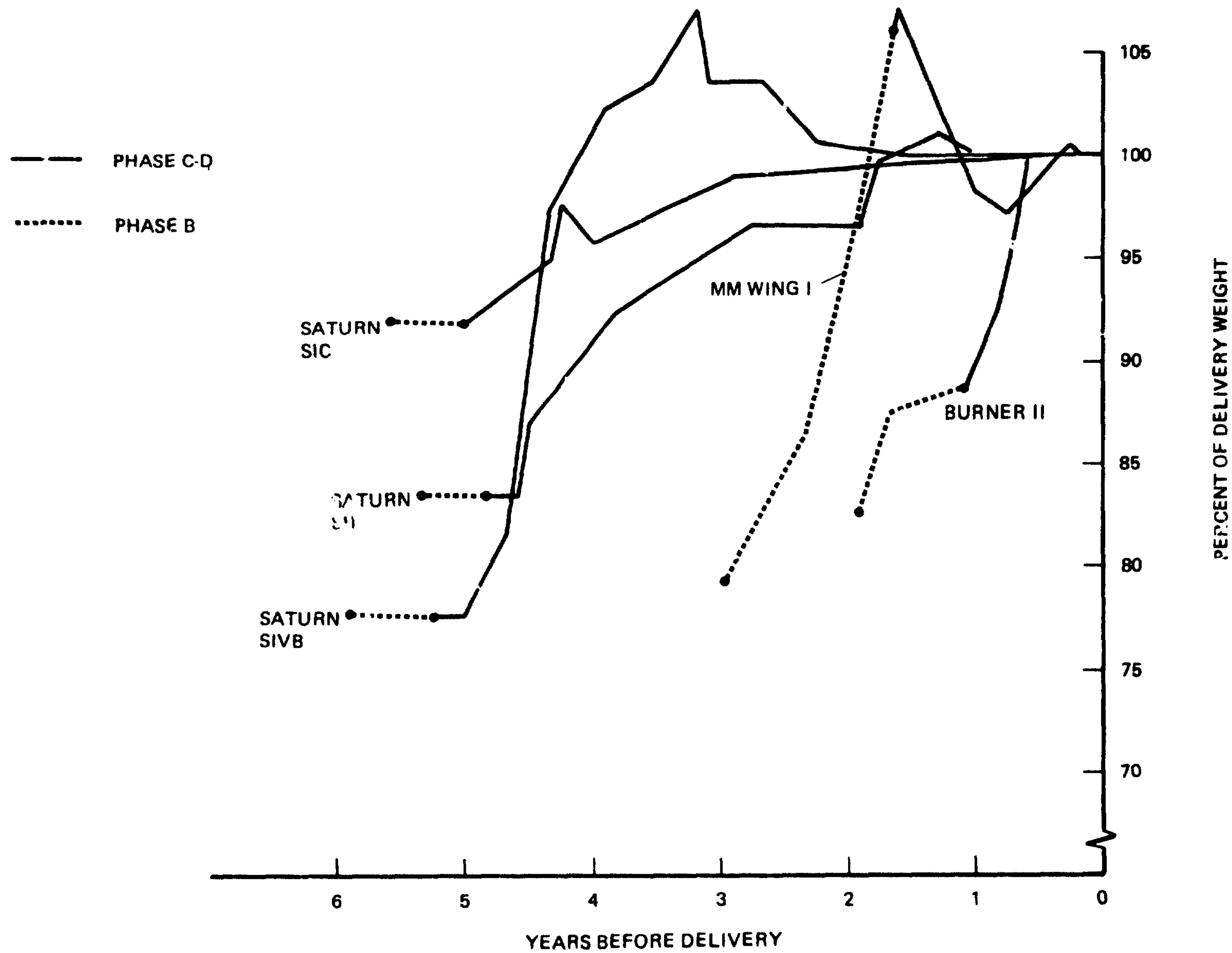


Figure 2-3. Weight Growth – Boosters

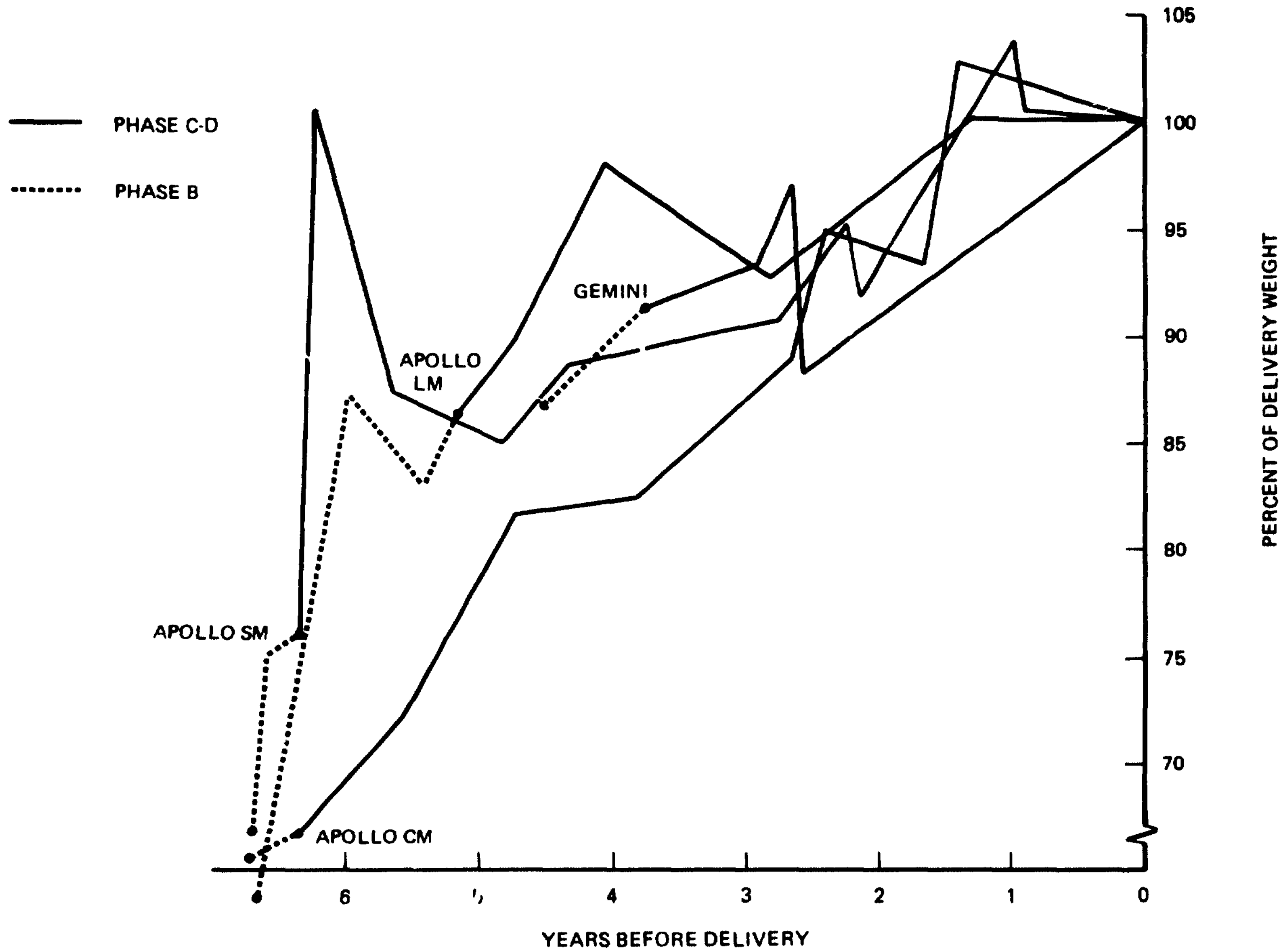


Figure 2-4. Weight Growth – Manned Spacecraft

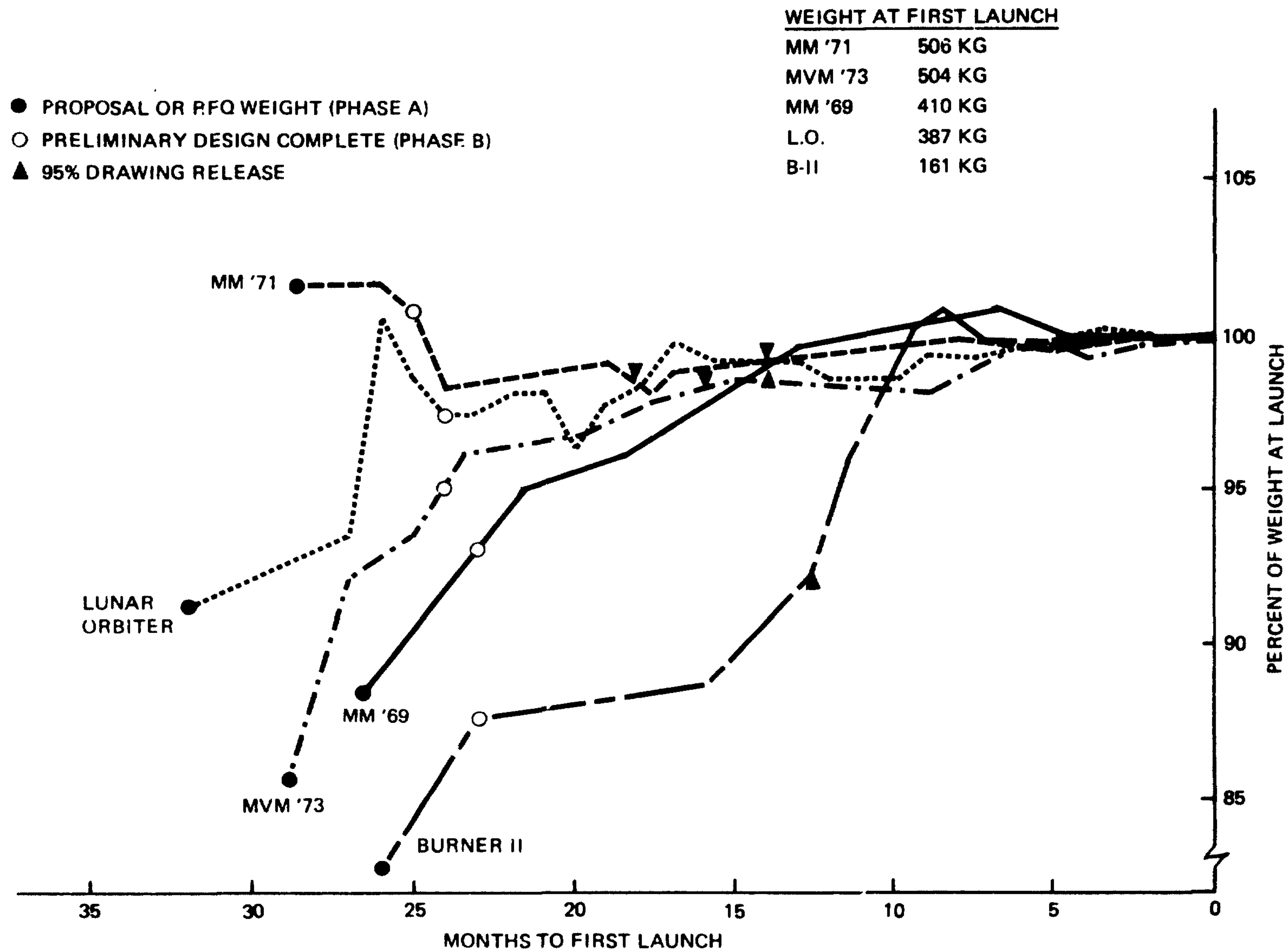


Figure 2-5. Weight Histories of Unmanned Spacecraft

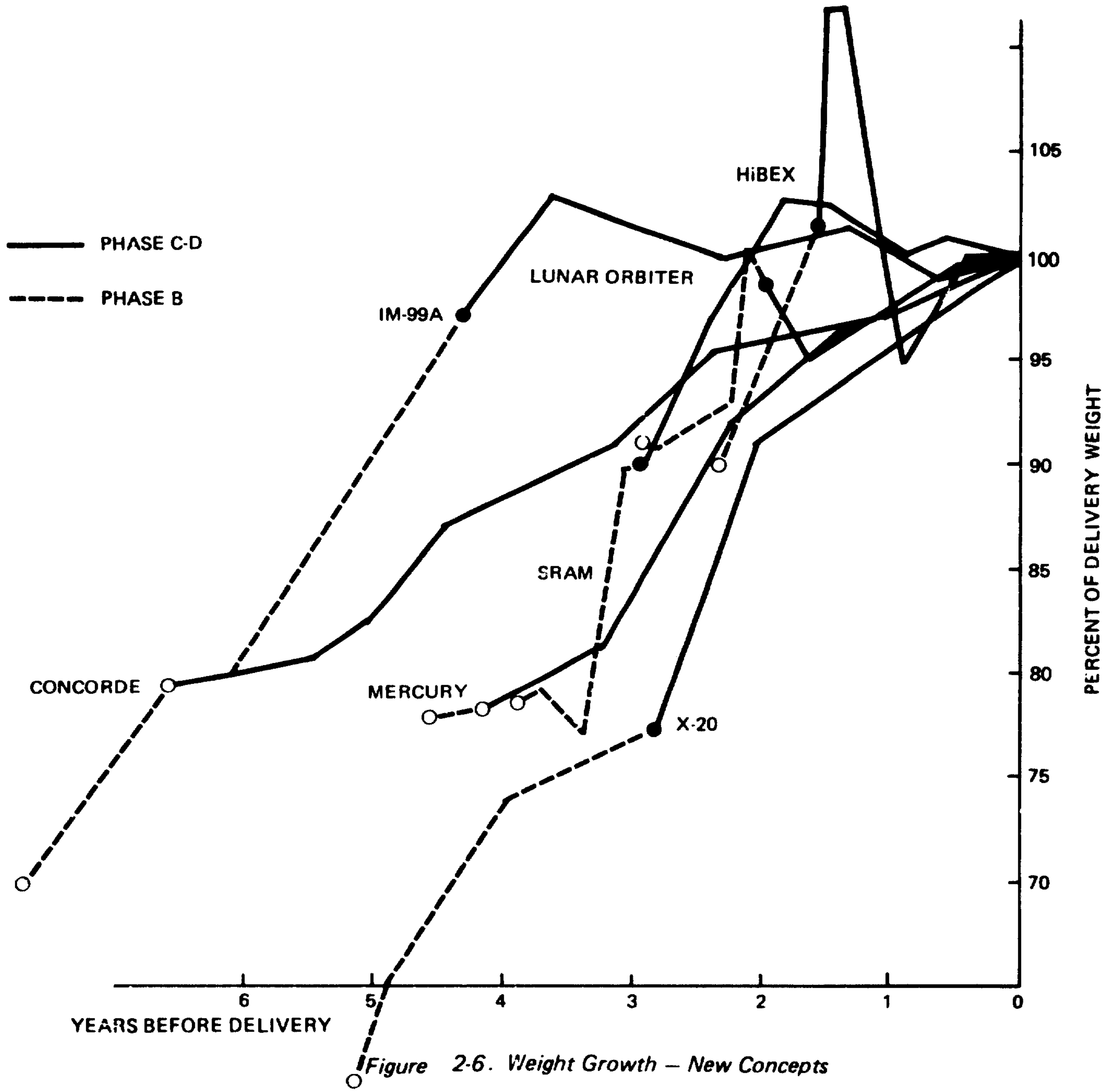


Figure 2-6. Weight Growth – New Concepts

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.

2.1.2 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	% GROWTH 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			

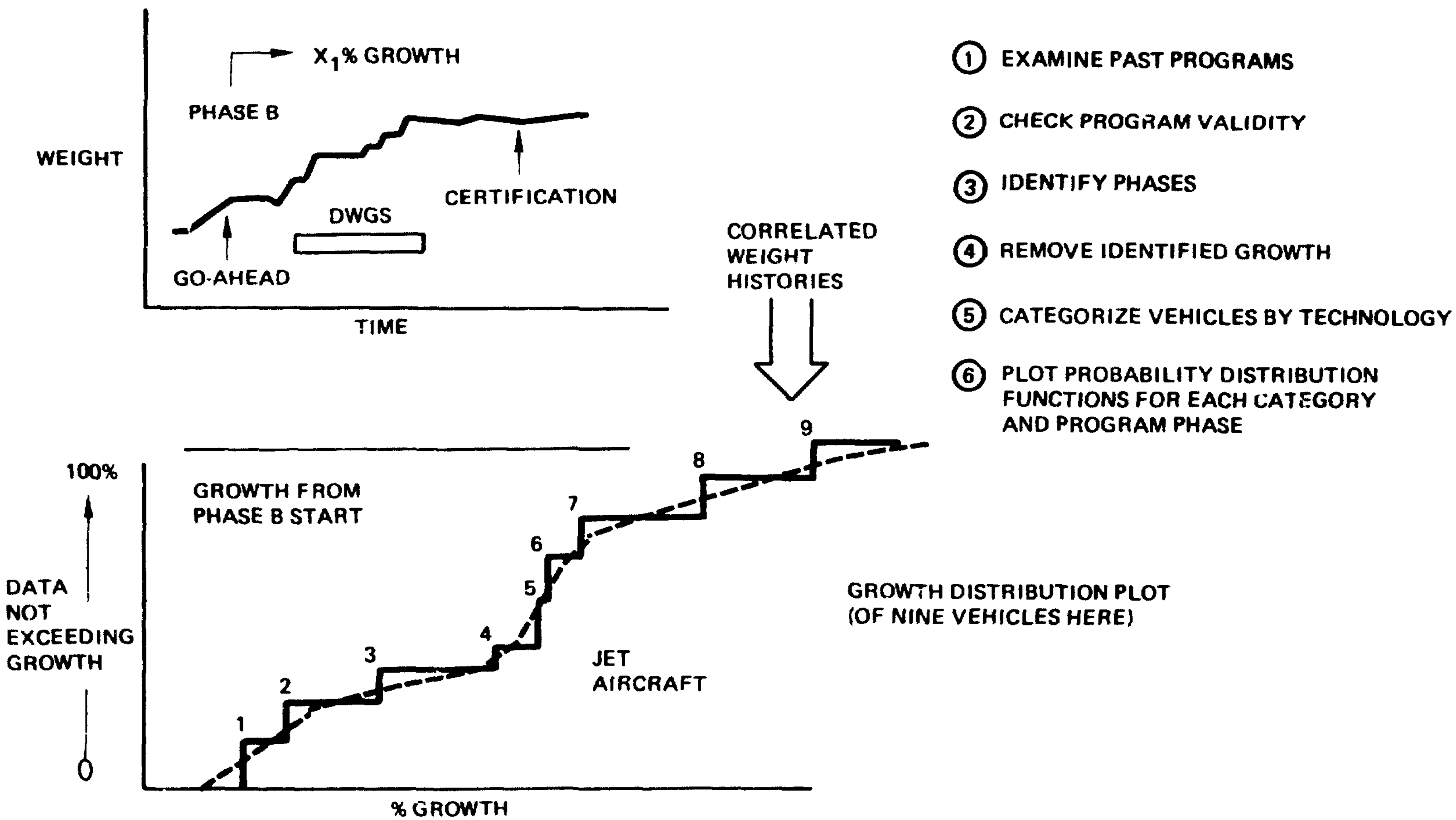


Figure 2-7. Vehicle Development and Weight History Evaluation and Derivation of Growth Distribution Plots

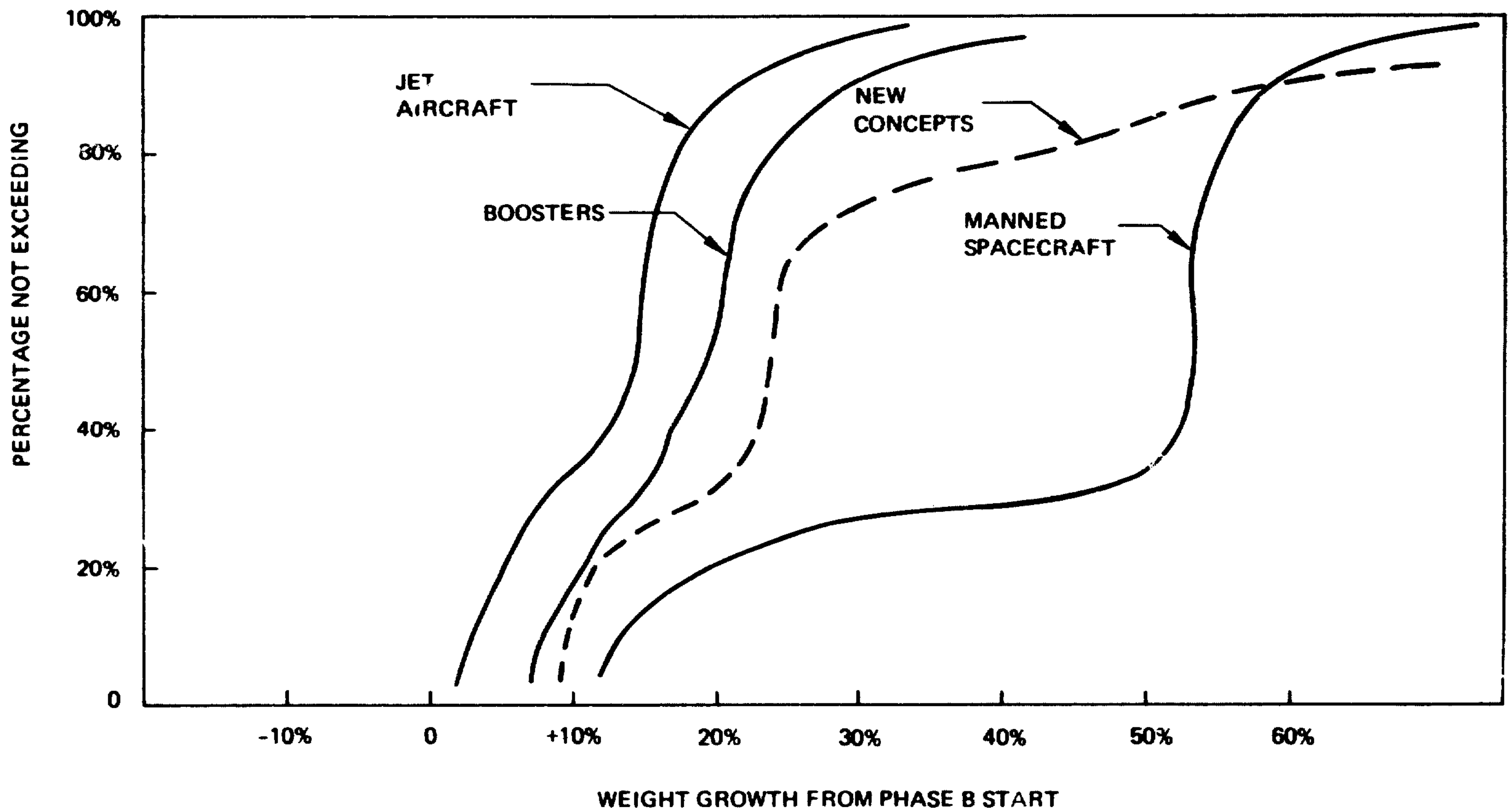


Figure 2-8. Past Program Growth Distribution

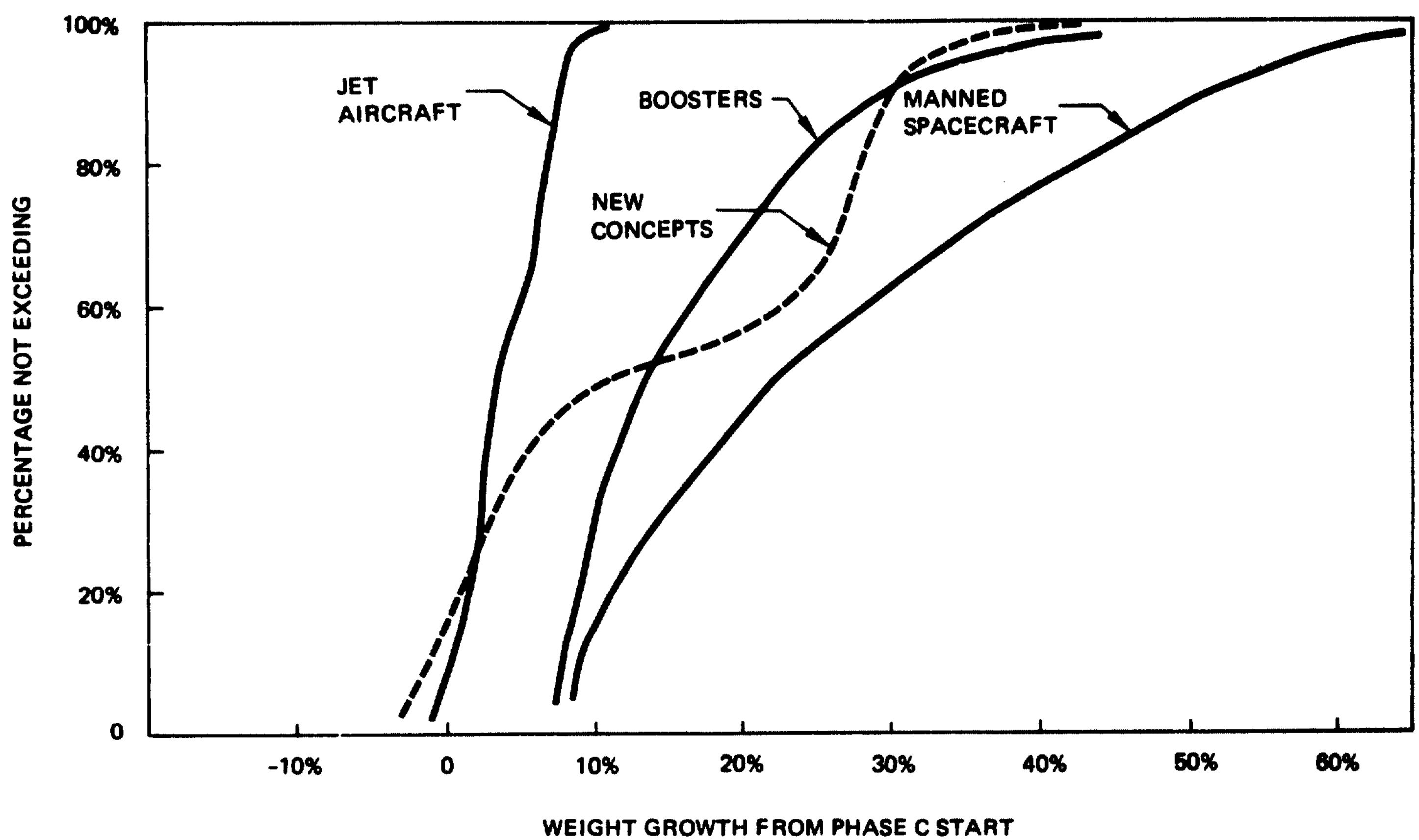


Figure 2-9. Past Program Growth Distribution

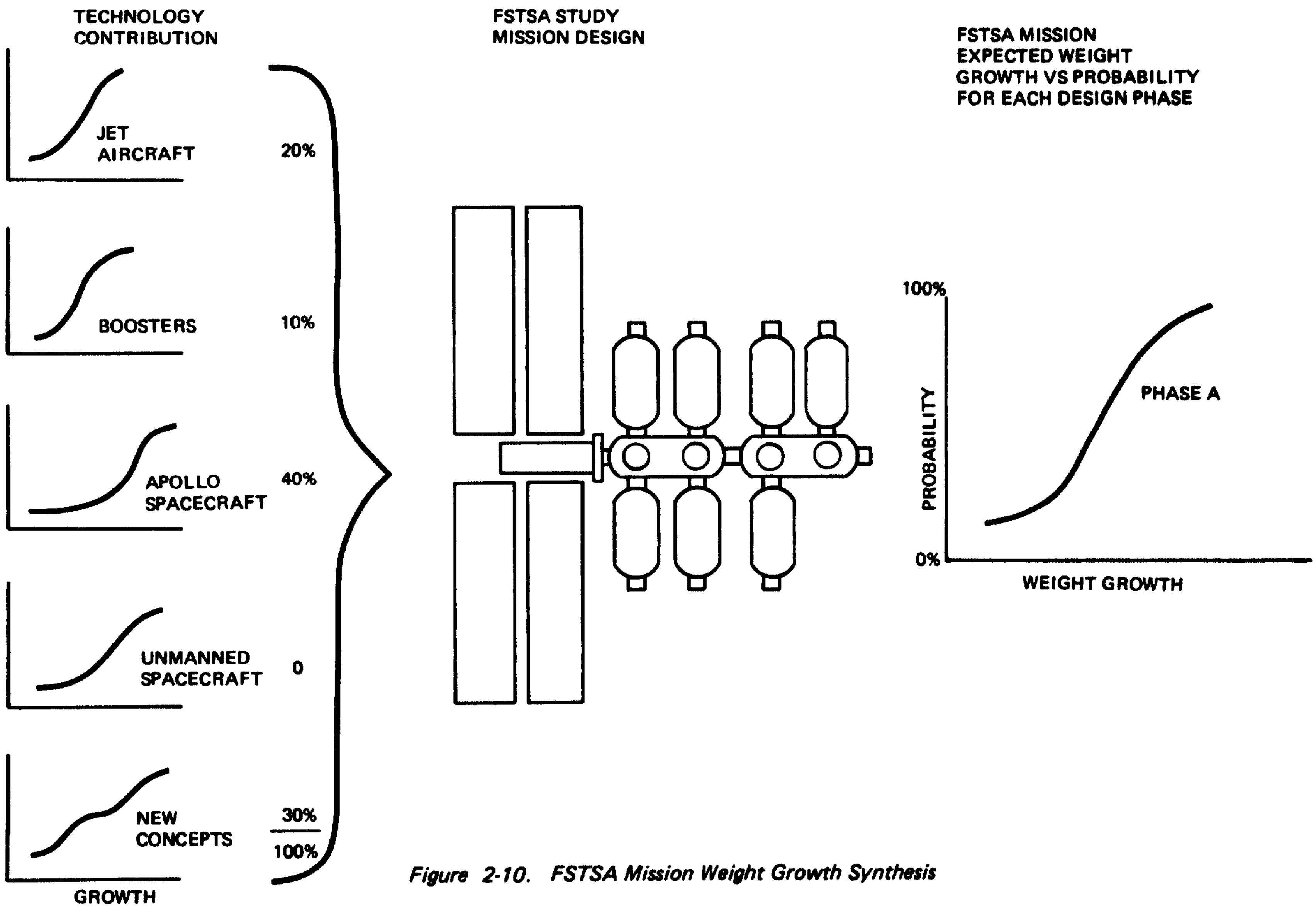


Figure 2-10. FSTSA Mission Weight Growth Synthesis

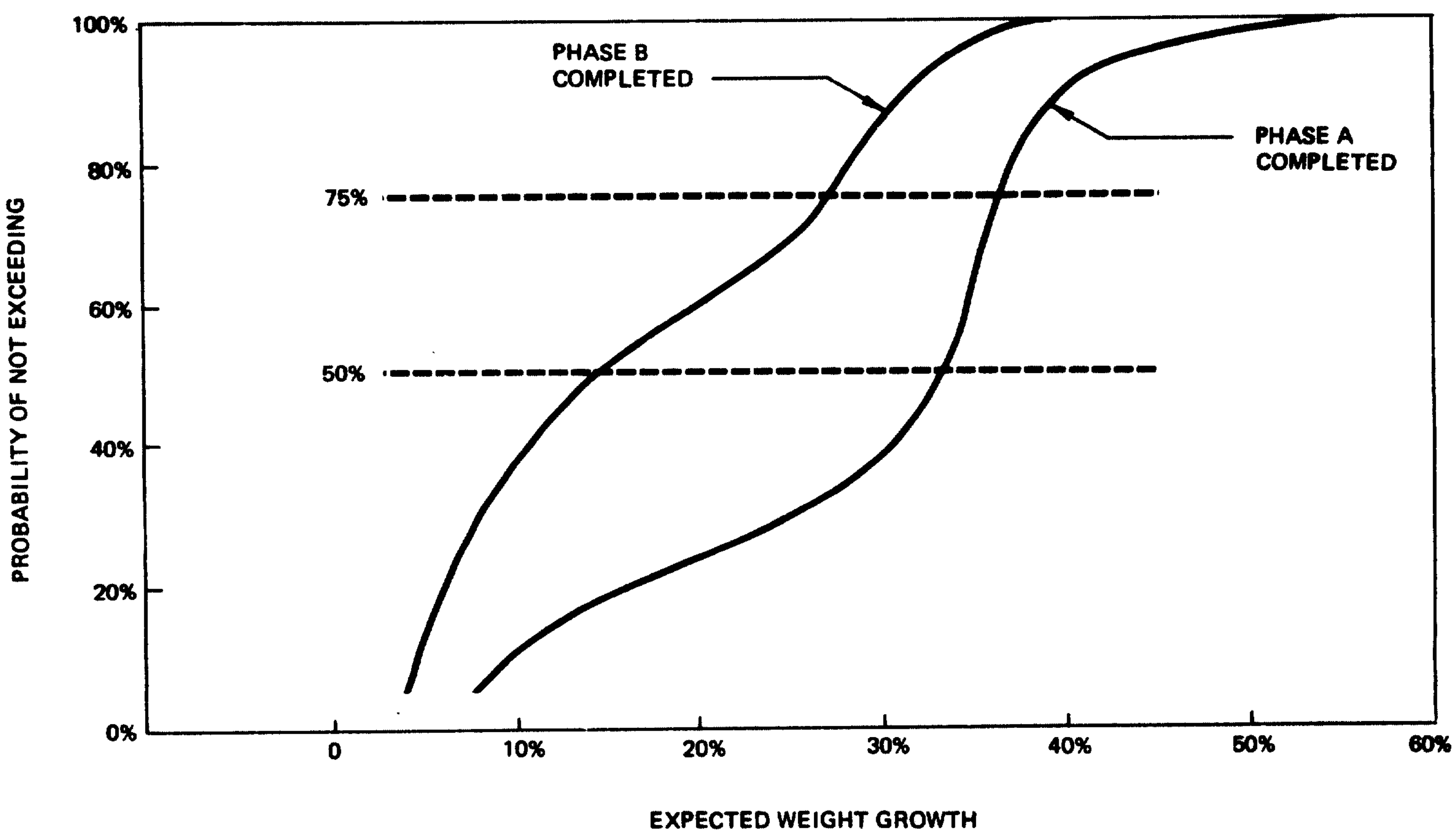


Figure 2-11. Expected Weight Growth for Earth Orbital Space Station (EOSS)

2.1.3 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	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%

2.2 PERFORMANCE

Geosynchronous Missions, High Thrust OTV's

Ideal impulsive delta V's were based on transfers from a 28.75 degree 500 km (270 n mi) orbit to a 0^{o}, 35 786 km (19 323 n mi) orbit. Ideal delta V's were computed using a simple point-mass Earth with gravitational potential strength of 398 601.2 km^3/sec^2 ($1.407596 X 10^{16} ft^3/sec^2$) and equatorial radius of 6 378 km (20,925,000 ft). The perigee burn includes 2.25^{o} plane change for an ideal ΔV of 2,394 m/sec (7,854 ft/sec); the apogee burn includes 26.5 degrees plane change at 1,773 m/sec (5,816 ft/sec). Figure 2.2-1 shows sensitivity of the ideal delta V to starting altitude and inclination. Actual delta V budgets included a nominal 100 m/sec (328 ft/sec) gravity loss on the first burn, a total of 50 m/sec (164 ft/sec) for each rendezvous and docking, 10 m/sec (33 ft/sec) ascent and return midcourse corrections, and small delta V's for orbit wait and standoff maneuvers. (Standoff is a separation and coast maneuver used to achieve a distance of several km between an OTV and a service vehicle or facility prior to initiating main engine firing.) A flight performance reserve of 2% of ideal delta V was applied to each mission.

Low Thrust OTV's

Performance methods were discussed in paragraph 1.2.2.

Lunar Missions, High Thrust OTV's.

Ideal impulsive delta V's were based on transfers from a 31.6 degree 528 km (285 n mi) Earth orbit to a 111 km (60 n mi) polar lunar orbit. The selected Earth orbit has a repeating ground track and its nodal regression is synchronized with the moon's motion such that orbit/moon configurations repeat every 2 lunar sidereal months (55 days). Non-symmetric transfers (90 hours translunar and 110 hours transEarth) provide a favorable mission profile in that a reasonable stay time at the moon (15 days) is obtained with small plane changes at lunar orbit insertion and departure. The round trip requires a total of 23½ days. Principal ideal delta V's are:

	m/sec	ft/sec
Translunar injection (TLI)	3,115	10,219
Lunar Orbit Insertion (LOI)	915	3,001
TransEarth Injection (TEI)	860	2,821
Earth Orbit Insertion (EOI)	3,115	10,219
(total)	8,005	26,260

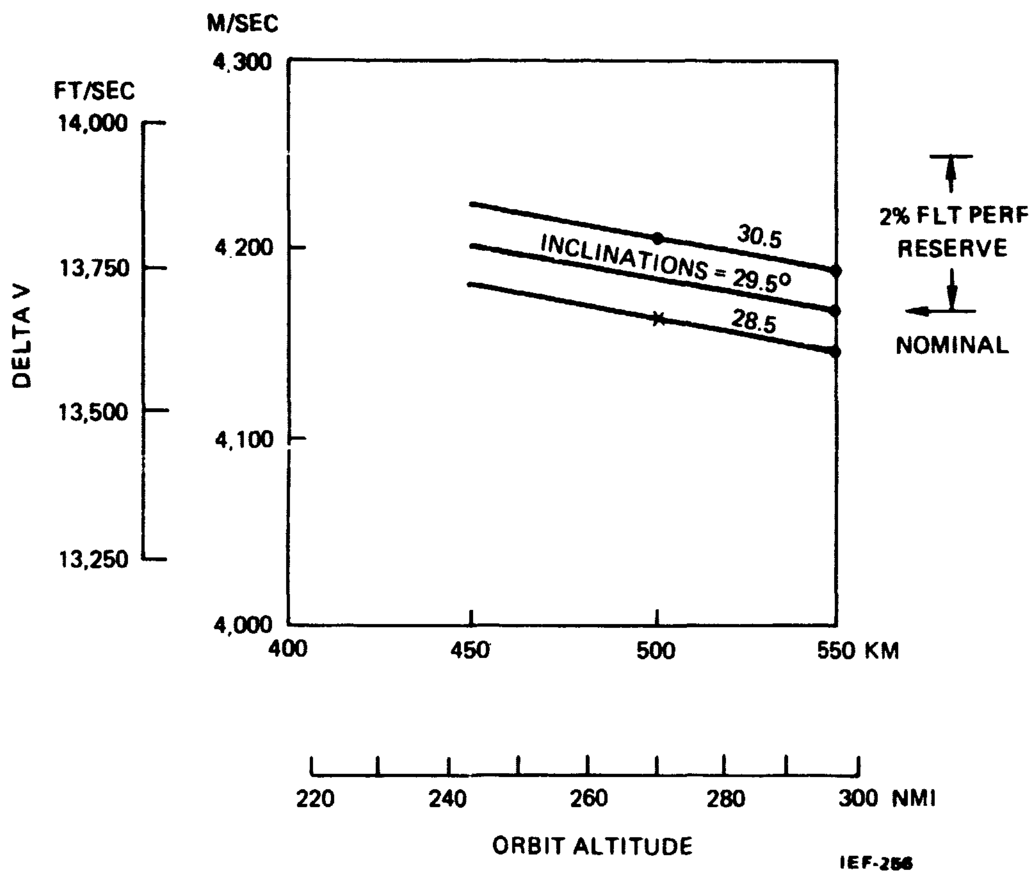


Figure 2.2-1. Ideal Delta V One-Way Geosynchronous Transfers

Assigned additions were 100 m/sec (328 ft/sec) gravity loss for the TLI maneuver, 30 m/sec (9 ft/sec) for translunar and transEarth midcourse and coast, 50 m/sec (162 ft/sec) total for each rendezvous and docking, 3 m/sec (10 ft/sec) for standoffs, and 2% of ideal delta V flight performance reserve.

Lunar Landing, LTV's

Lunar landing used a typical Apollo delta V budget for descent from and ascent to a 111 km (60 n mi) circular lunar orbit. Principal delta V's were as follows:

	m/sec	ft/sec
Powered Descent Initiation (PDI)	22	72
Braking	1,620	5,314
Landing	492	1,614
Ascent	1,846	6,056
Rendezvous & Docking	175	573
TOTAL	4,155	13,629

Other Missions

Delta V's for manned and unmanned planetary missions were taken from various references. Delta V's for nuclear waste disposal missions were taken from NASA TMX 2911 and calculated from mission requirements.

2.3 PROPULSION

PROPELLANT PERFORMANCE SUMMARY

PROPULSION SYSTEM PARAMETER	PERFORMANCE ACHIEVABLE BY DATE 1980	1990	2000
Liquid Boosters			
O_2+H_2, Isp vac.	455	458	466
Isp S.L.	363	373	416
O_2+RP-1, Isp vac.	340	354	362
Isp S.L.	270	285	320
O_3+H_2, Isp vac.	N.A.	490	490
Isp S.L.	N.A.	403	403
Chemical Space Engines			
N_2O_4+A-50, Isp vac.	325	338	338
O_2+RP-1, Isp vac.	305	362	362
O_2+MMH, Isp vac.	359	366	366
FLOX + CH_4, Isp vac.	413	421	421
OF_2+B_2H_6, Isp vac.	439	451	451
O_2+H_2, Isp vac.	453	462	462
F_2+H_2, Isp vac.	471	479	479
F_2+N_2H_4, Isp vac.	419	425	425
F_2+Li+H_2, Isp vac.	513	523	523
O_2+B_e+H_2, Isp vac.	N.A.	552	552

Theoretical Kinetics Values (Ivac) at p_c = 1500 psia and A_e = 200 supplied by Philip A. Masters of Lewis Research Center

Propellant	O/F	Isavc (kinetic)
H_2/F_2	11.0	497.0
CH_4/Flox (82.6% F_2)	5.5	438.3
N_2O_4/A-50	2.0	354.0
MMH/O_2	1.3	383.6
OF_2/B_2H_6	3.5	473.8

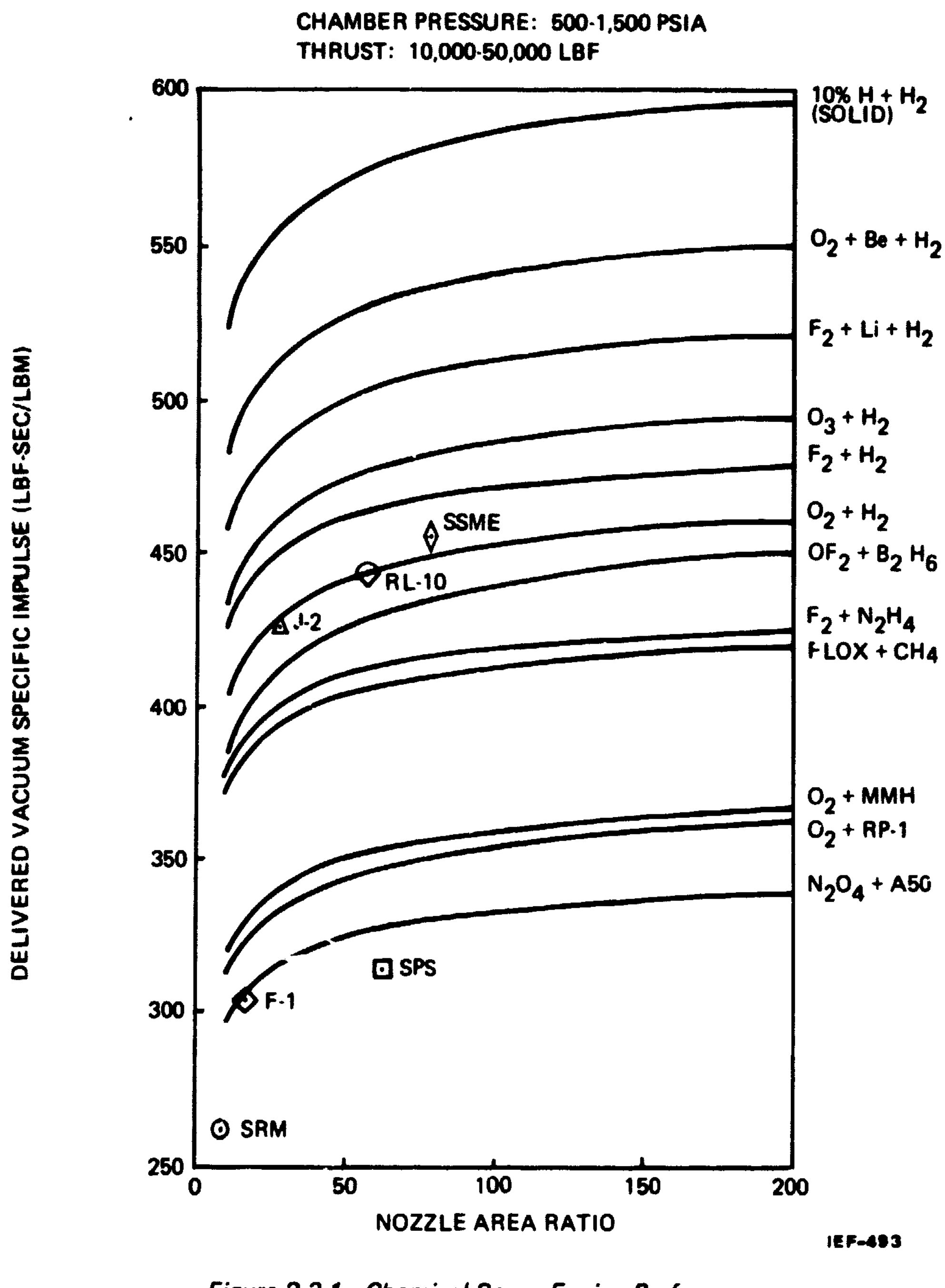


Figure 2.3-1. Chemical Space Engine Performance

OXYGEN-HYDROGEN BOOST PROPELLANT CHARACTERISTICS

Oxidizer:	Oxygen – O_2
Fuel:	Hydrogen – H_2
Mixture Ratio:	6
Bulk Density:	22.54 lbs/ft^3

Property	O_2	H_2
Molecular Weight	32	2.016
Stored Density – lbs/ft^3	71.3	4.42
Freezing Point – ^oR	97.8	24.8
Normal Boiling Point – ^oR	162.3	36.5
Stability	Stable	Stable

Performance Factors

Well developed technology for hydrogen oxygen engines has demonstrated specific impulse efficiency of 96.3% for the 15,000 pound thrust RL-10. Booster thrust class hydrogen oxygen engines such as the Shuttle main engine are expected to achieve approximately 97.6% specific impulse efficiency.

Summary

Date	Engine	Vacuum Specific Impulse Lbf-Sec/Lbm
1975	J-2	426
1975	SSME	455
1980	SSME	455
1990	New	458
2000	New	466

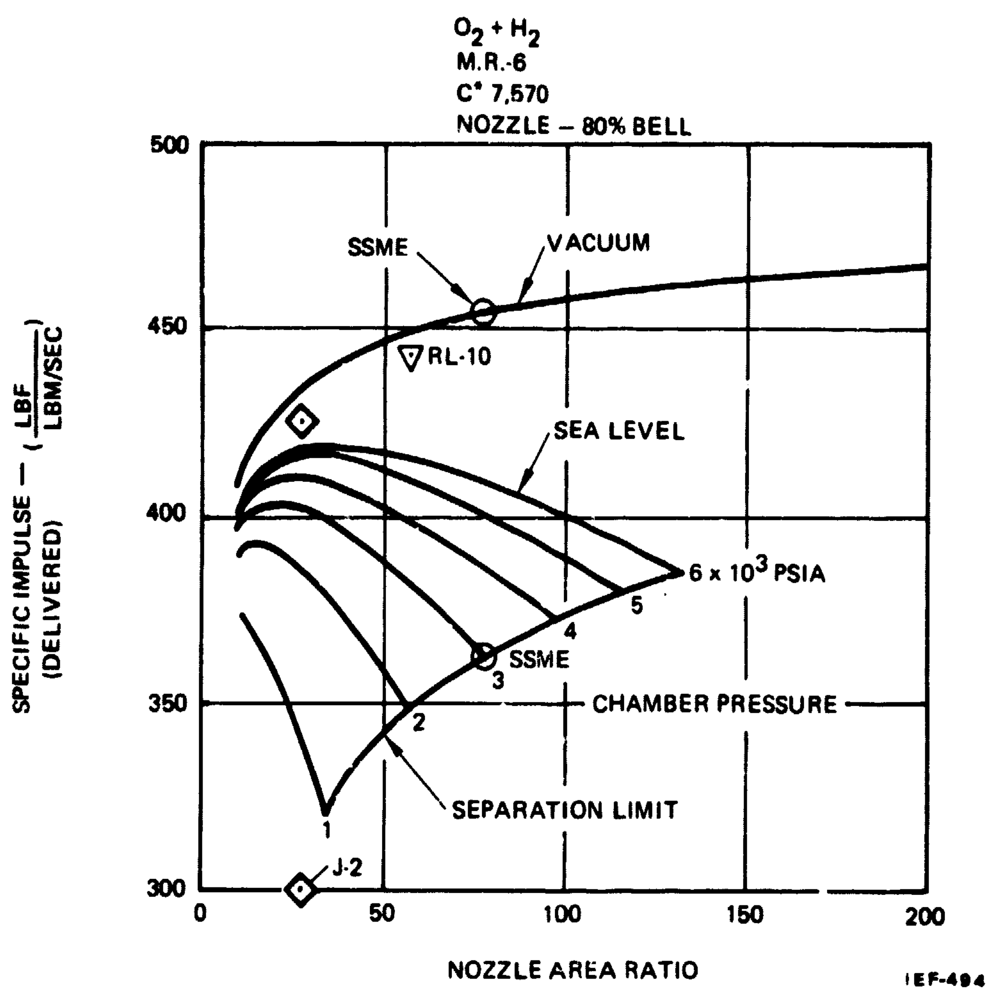


Figure 2.3-2. Oxygen Hydrogen Boost Engine Performance

OXYGEN–RP-1 PROPELLANT CHARACTERISTICS

Oxidizer:	Oxygen – O_2
Fuel:	RP-1 – H/C = 2.0
Mixture Ratio:	2.6
Bulk Density:	63.75

Property	O_2	RP-1
Molecular Weight	32	163
Stored Density lbs/ft^3	71.3	49.94
Freezing Point – ^oR	97.8	Below 420
Normal Boiling Point – ^oR	162.3	851.8
Stability	Stable	Stable

Performance Factors

The F-1 engine using oxygen and RP-1 propellants developed 90.2% vacuum specific impulse efficiency. The gas generator cycle and low combustion efficiency contributed to the low specific impulse efficiency. Use of a pre-burner cycle with reasonable combusion efficiency improvement could provide 94.5% specific impulse efficiency.

Summary

Date	Engine	Vacuum Specific Impulse Lbf-Sec/Lbm
1975	F-1	305
1980	F-1	305
1990	New	362
2000	New	362

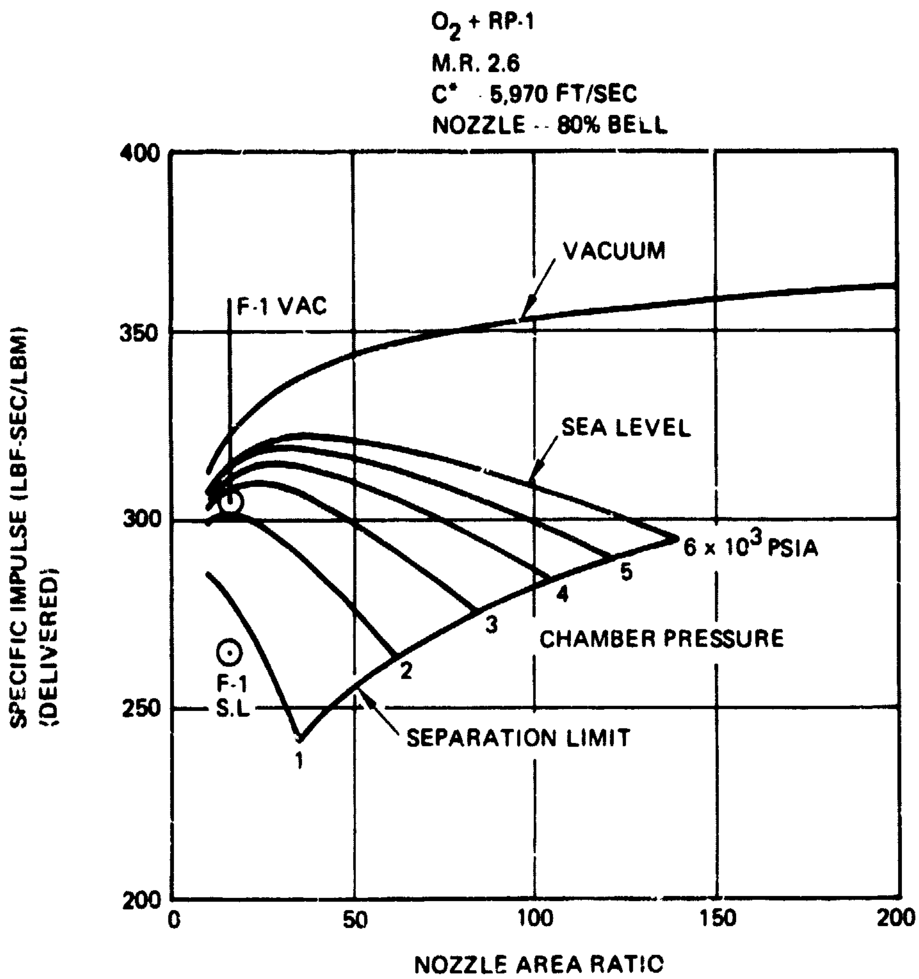


Figure 2.3-3. Oxygen RP-1 Boost Engine Performance

NITROGEN TETROXIDE–AEROZINE 50 PROPELLANT CHARACTERISTICS

Oxidizer:	Nitrogen Tetroxide – N_2O_4
Fuel:	Aerozine 50 – 50/50 Mixture of hydrazine – N_2H_4 and unsymmetrical dimethylhydrazine – $(CH_3)_2 N_2H_2$
Mixture Ratio:	2
Bulk density:	74.67 lbs/ft^2

Property	N_2O_4	A–50
Molecular Weight	94.016	41.8
Stored Density – lbs/ft^3	89.52	56.06
Freezing Point – ^oR	471.5	478.5
Normal Boiling Point – ^oR	529.8	617.9
Stability	Stable	Stable at room temperature

Performance Factors

Nitrogen tetroxide–Aerozine 50 propellants have been used for several primary propulsion and reaction control engines. Specific impulse increases above current engines depend primarily on operation of higher pressures and thrusts. Ablative chamber materials improvements to permit higher pressures are needed to provide specific impulse gains.

Summary

Date	Engine	Vacuum Specific Impulse Lbf-Sec/Lbm
1975	SPS	314
1980	New	325
1990	New	338
2000	New	338

OXYGEN–MONOMETHYL HYDRAZINE PROPELLANT CHARACTERISTICS

Oxidizer:	Oxygen – O_2
Fuel:	Monomethyl hydrazine – $(CH_3) N_2H_3$
Mixture Ratio:	1.3
Bulk Density:	62.91 lbs/ft^3

Property	O_2	$(CH_3)N_2H_3$
Molecular Weight	32	46.074
Stored Density – lbs/ft^3	71.3	54.56
Freezing Point – ^oR	97.8	397.7
Normal Boiling Point – ^oR	162.3	649.7
Stability	Stable	Stable Below 500^{o}F

Performance Factors

The use of oxygen instead of nitrogen tetroxide as oxidizer for the hydrazine fuels provides approximately 8% specific impulse increase. The performance expected for oxygen and methane is the same as the oxygen monomethyl hydrazine.

Summary

Date	Engine	Vacuum Specific Impulse Lbf-Sec/Lbm
1980	New	359
1990	New	366
2000	New	366

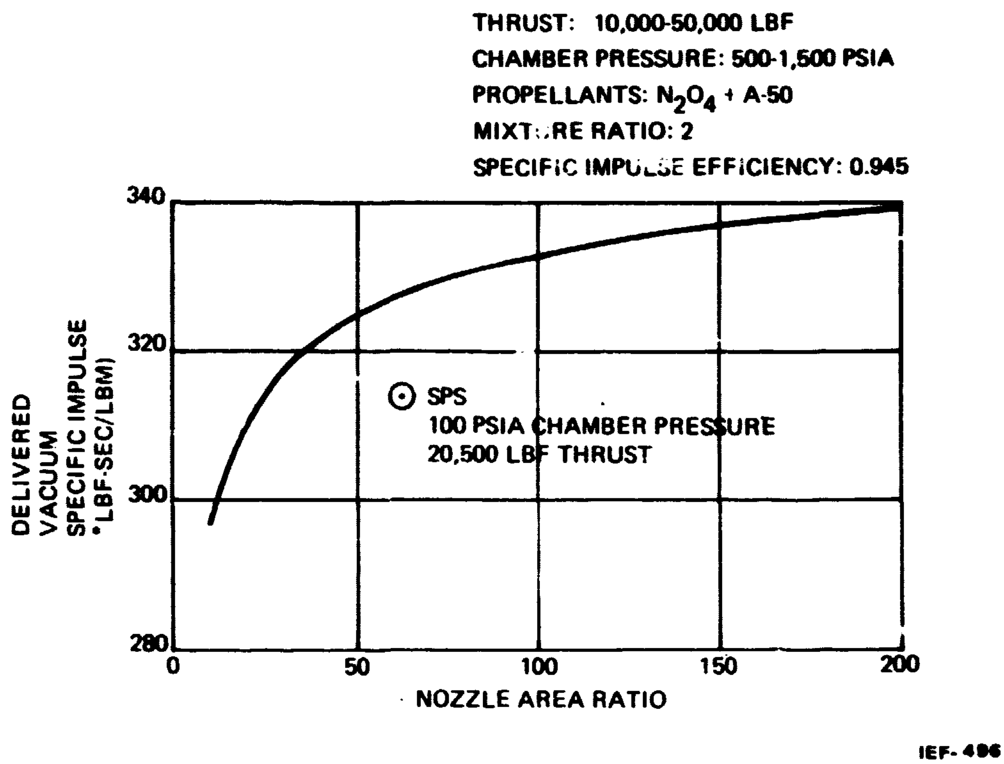


Figure 2.3-4. Nitrogen Tetroxide—Aerozine 50 Space Engine Performance

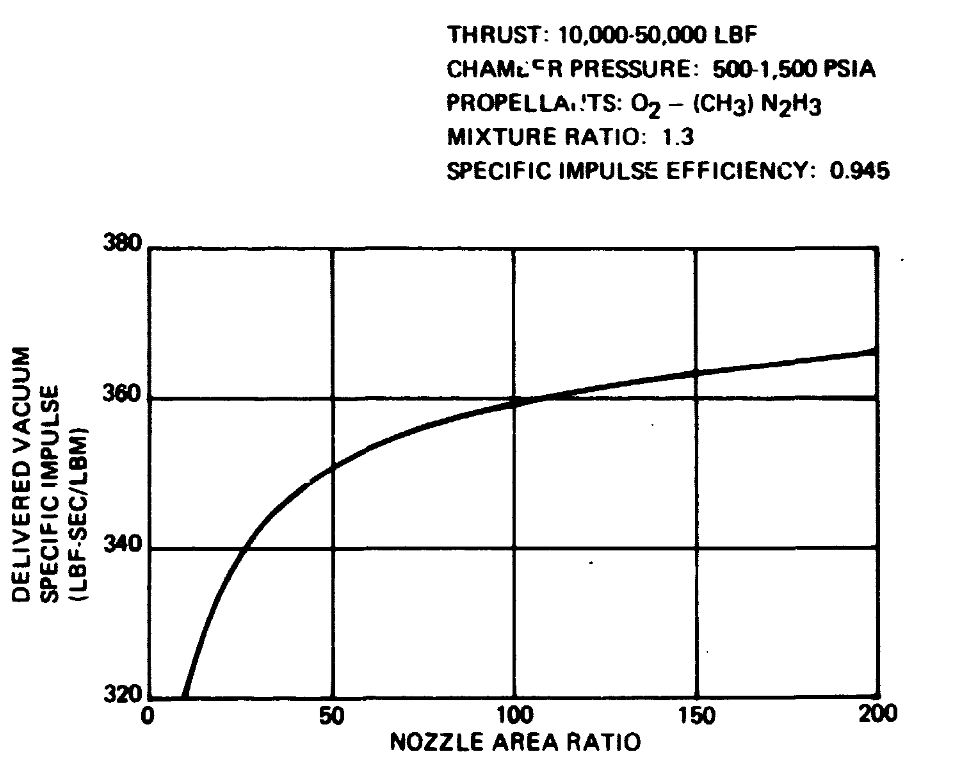


Figure 2.3-5. Oxygen—Monomethyl Hydrazine Space Engine Performance

FLOX–METHANE PROPELLANT CHARACTERISTICS

Oxidizer:	FLOX 82.6% F_2 + 17.4% O_2
Fuel:	Methane – CH_4
Mixture Ratio:	5.5
Bulk Density:	63.54

Property	FLOX	CH_4
Molecular Weight	36.88	16.042
Stored Density – lbs/ft^3	86.67	25.75
Freezing Point – ^oR	96.4	163.2
Normal Boiling Point – ^oR	154.9	201.2
Stability	Stable	Stable

Performance Factors

Flox methane testing conducted by Pratt and Whitney demonstrated high combustion efficiencies and lower than expected kinetic losses.

Summary

Data	Engine	Vacuum Specific Impulse Lbf-Sec/Lbm
1980	New	413
1990	New	421
2000	New	421

OXYGEN DIFLUORIDE–DIBORANE PROPELLANT CHARACTERISTICS

Oxidizer:	Oxygen Difluoride – OF_2
Fuel:	Diborane – B_2H_6
Mixture Ratio:	3.5
Bulk Density:	61.07 lbs/ft^3

Property	OF_2	B_2H_6
Molecular Weight	54	27.69
Stored Density – lbs/ft^3	94.8	27.2
Freezing Point – ^oR	88.9	193.8
Normal Boiling Point – ^oR	231.1	325.3
Stability	Slow Decomposition	Slow Decomposition

Performance Factors

Chamber cooling is a major development problem because fuel decomposition limits regenerative cooling capability. Ablative or transpiration cooled chambers may be required. Low pressure engines with low thrust have reduced efficiency due to kinetic losses.

Summary

Date	Engine	Vacuum Specific Impulse Lbf-Sec/Lbm
1980	New	439
1990	New	451
2000	New	451

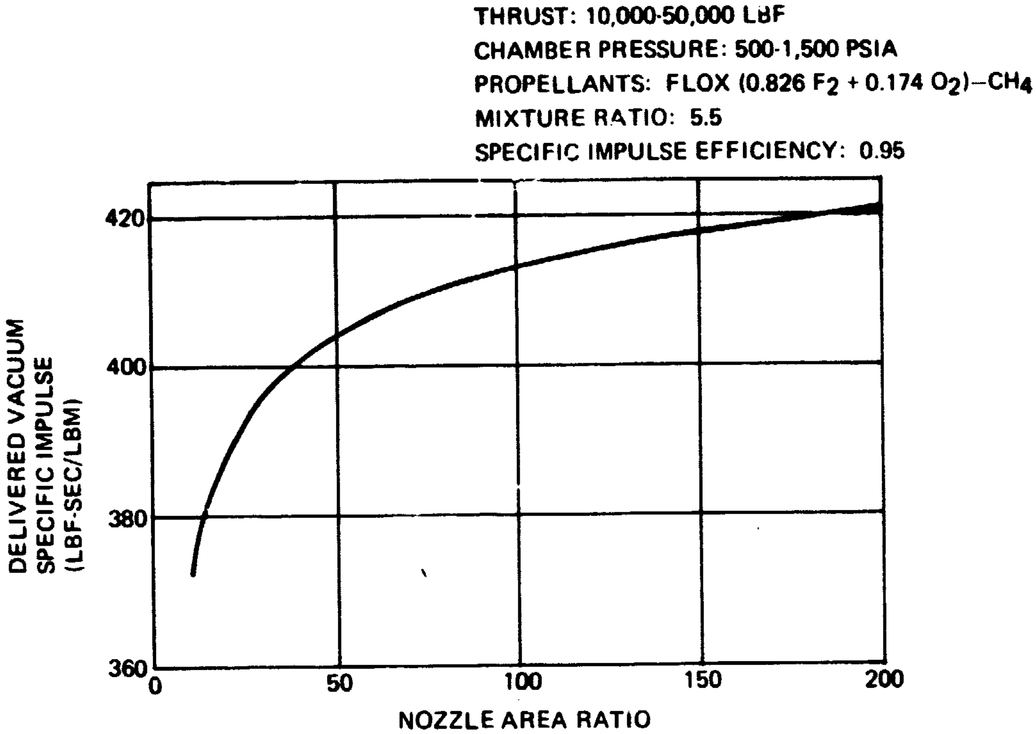


Figure 2.3-6. Flox–Methane Space Engine Performance

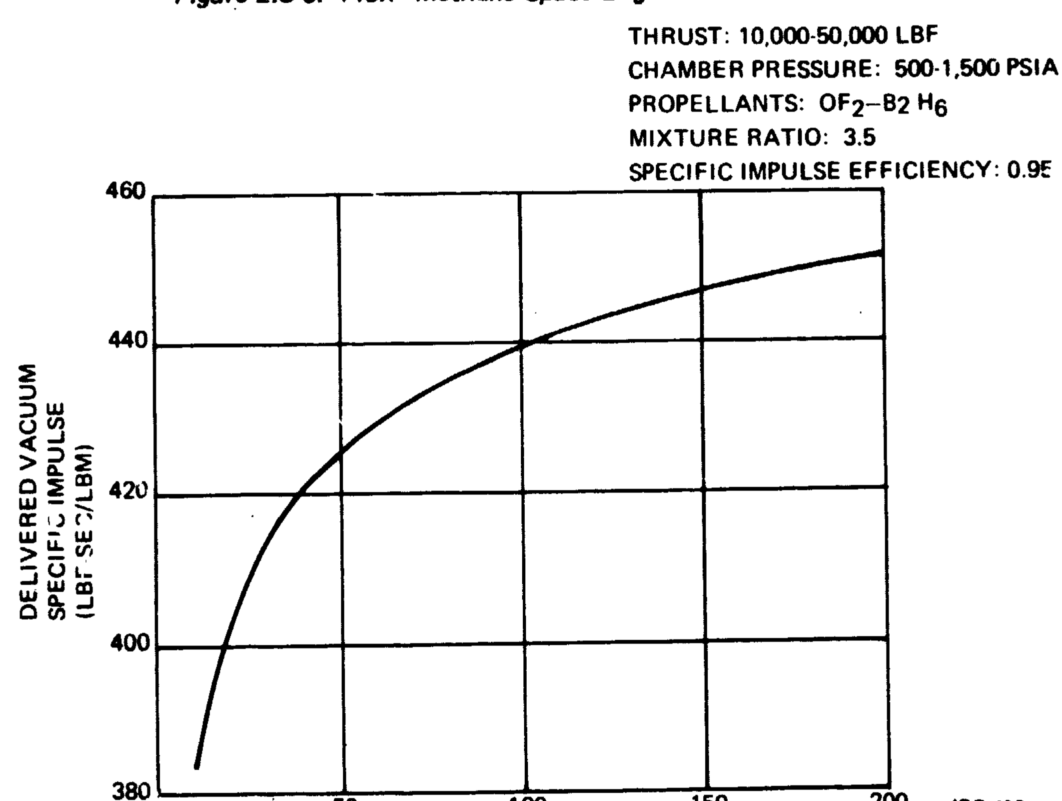


Figure 2.3-7. Oxygen Difluoride–Diborane Space Engine Performance

OXYGEN–HYDROGEN PROPELLANT CHARACTERISTICS

Oxidizer: Oxygen – O_2

Fuel: Hydrogen – H_2

Mixture Ratio: 6

Bulk Density: 22.54 lbs/ft^3

Property	O_2	H_2
Molecular Weight	32	2.016
Stored Density – lbs/ft^3	71.3	4.42
Freezing Point – ^oR	97.8	24.8
Normal Boiling Point – ^oR	162.3	36.5
Stability	Stable	Stable

Performance Factors

Well developed technology for hydrogen oxygen engines has demonstrated specific impulse efficiency of 96.3% for the 15,000 pound thrust RL-10. Booster thrust class hydrogen oxygen engines such as the Shuttle main engine are expected to achieve approximately 97.6% specific impulse efficiency.

Summary

Date	Engine	Vacuum Specific Impulse Lbf-Sec/Lbm
1975	RL-10	444
1980	New	453
1990	New	462
2000	New	462

FLUORINE-HYDROGEN PROPELLANT CHARACTERISTICS

Oxidizer:	Fluorine – F_2
Fuel:	Hydrogen – H_2
Mixture Ratio:	11
Bulk Density:	34.95 lbs/ft^3

Property	F_2	H_2
Molecular Weight	38	2.016
Stored Density – lbs/ft^3	93.96	4.42
Freezing Point – ^oR	96.4	24.9
Normal Boiling Point – ^oR	153.1	36.7
Stability	Stable	Stable

Performance Factors

Fluorine hydrogen engines provide the highest specific impulse available from stable bi-propellant combinations. Low pressure engines with low thrust have reduced efficiency due to kinetic losses.

Summary

Year	Engine	Vacuum Specific Impulse Lbf-Sec/Lbm
1980	New	471
1990	New	479
2000	New	479

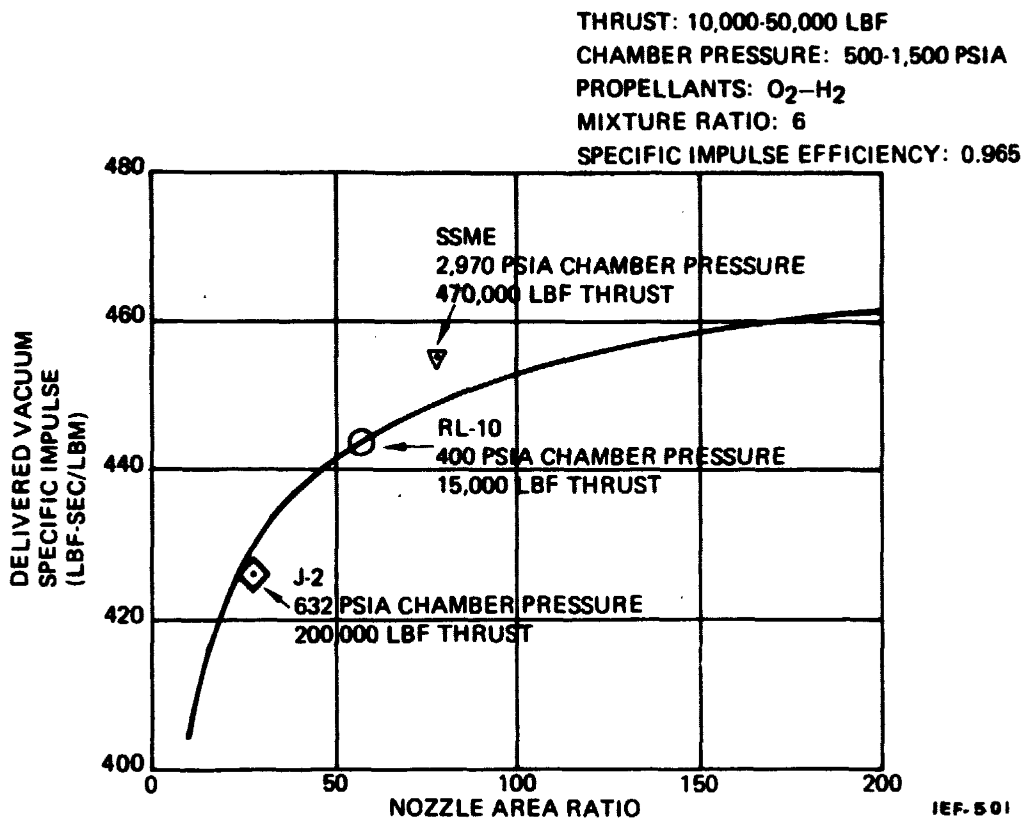


Figure 2.3-8. Oxygen--Hydrogen Space Engine Performance

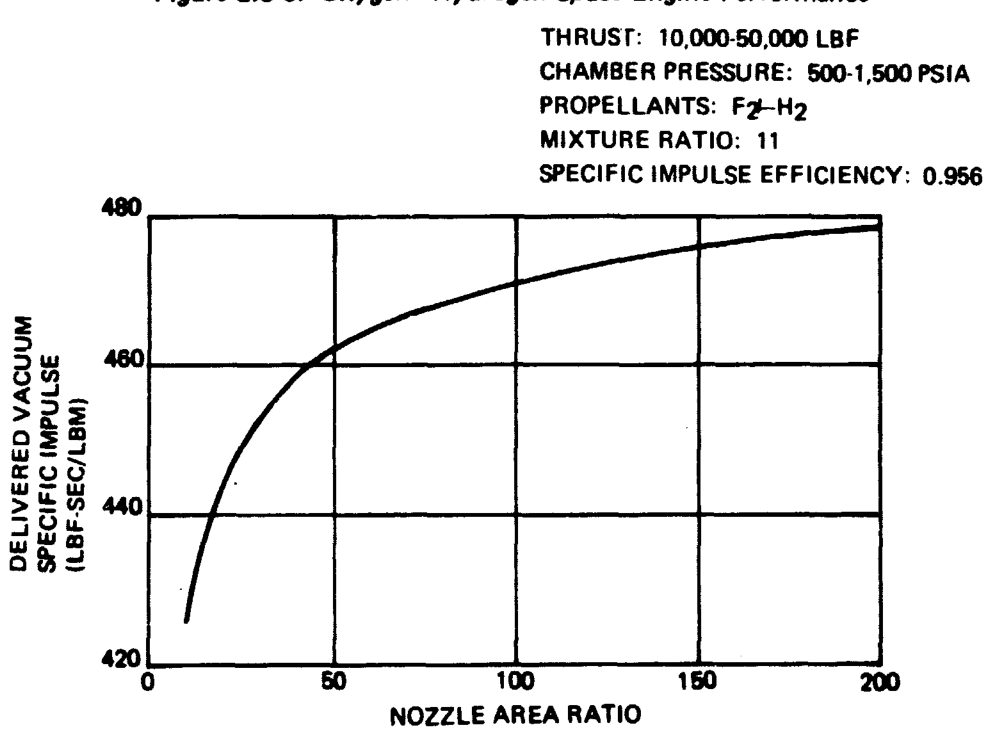


Figure 2.3-9. Fluorine—Hydrogen Space Engine Performance

FLUORINE–HYDRAZINE PROPELLANT CHARACTERISTICS

Oxidizer:	Fluorine – F_2
Fuel:	Hydrazine – N_2H_4
Mixture Ratio:	2.2
Bulk Density:	81.23

Property	F_2	N_2H_4
Molecular Weight	38	32.048
Stored Density – lbs/ft^3	93.86	62.68
Freezing Point – ^oR	96.4	494.2
Normal Boiling Point – ^oR	153.1	695.1
Stability	Stable	Stable

Performance Factors

Fluorine hydrazine specific impulse is maximum at approximately the stoichiometric mixture ratio of 2.37. High flame temperatures indicate need for ablative chamber materials developments.

Summary

Date	Engine	Vacuum Specific Impulse Lbf-Sec/Lbm
1980	New	419
1990	New	425
2000	New	425

FLUORINE-LITHIUM-HYDROGEN PROPELLANT CHARACTERISTICS

Oxidizer:	Fluorine – F_2
Fuel:	Lithium – L_i
	Hydrogen – H_2
Mixture Ratio:	53.1°/F_2, 19.4% L_i, 27.58% H_2
Bulk Density:	13.56

Property	F_2	L_i	H_2
Molecular Weight	38	6.941	2.016
Stored Density – lbs/ft^3	93.86	33.1	4.42
Freezing Point – °R	96.4	813.9	24.9
Normal Boiling Point – °R	153.1	2862	36.7
Stability	Stable	Stable	Stable

Performance Factors

Satisfactory combustion of this tripropellant was demonstrated by Rocketdyne under NASA contract. Handling and maintaining lithium in the liquid state is feasible with current technology.

Summary

Date	Engine	Vacuum Specific Impulse Lbf-Sec/Lbm
1980	New	513
1990	New	523
2000	New	523

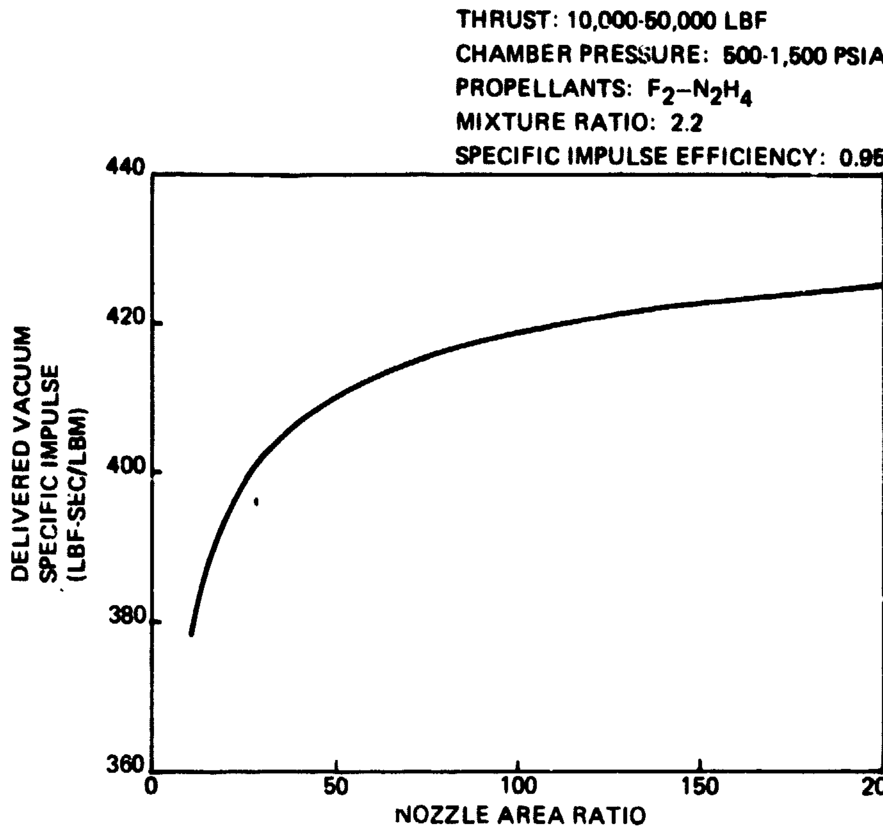


Figure 2.3-10. Fluorine–Hydrazine Space Engine Performance

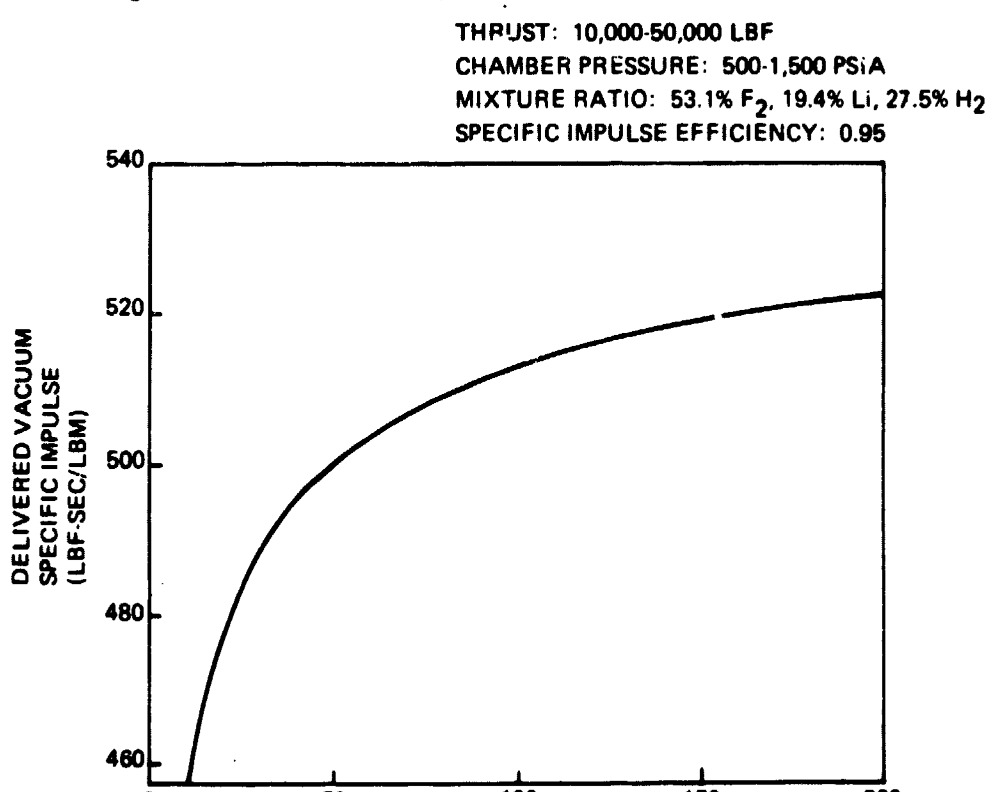


Figure 2.3-11. Fluorine–Lithium–Hydrogen Space Engine Performance

OXYGEN–BERYLLIUM–HYDROGEN PROPELLANT CHARACTERISTICS

Oxidizer:	Oxygen – O_2
Fuel:	Beryllium – Be
	Hydrogen – H_2
Mixture Ratio:	46.9% O_2, 26.6% Be, 26.5% H_2
Bulk Density:	14.6 lbs/ft^3

Property	O_2	Be	H_2
Molecular Weight	32	9.0122	2.016
Stored Density – lbs/ft^3	71.3	57.69*	4.42
Freezing Point – ^oR	97.8	2792	24.8
Normal Boiling Point – ^oR	162.3	5837	36.5
Stability	Stable	Stable	Stable

*Taken as one-half solid density

Performance Factors

Satisfactory methods of handling beryllium have not been developed. Efficient combustion and recovery of the available energy has not been demonstrated.

Summary

Date	Engine	Vacuum Specific Impulse Lbf-Sec/Lbm
1990	New	552
2000	New	552

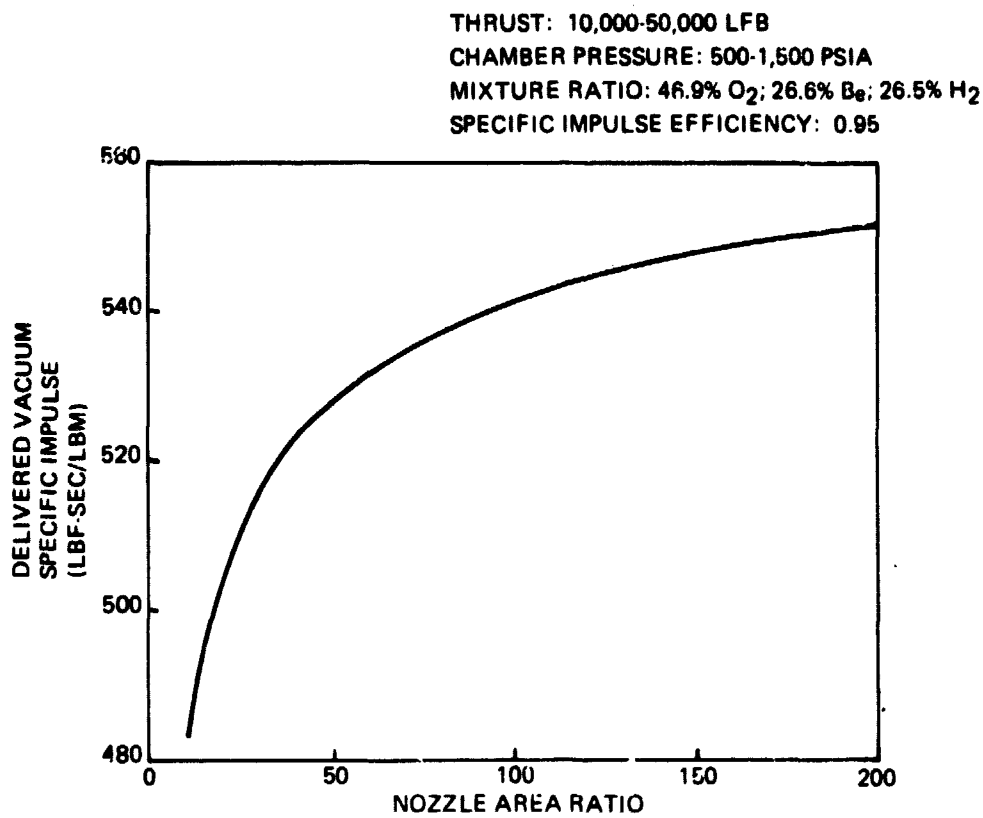


Figure 2.3-12. Oxygen–Beryllium–Hydrogen Space Engine Performance

Table 2.3-1. Rocket Engine Comparison (From JSC Propulsion and Power Division)

Engine name	Oxidizer/ fuel	Thrust vac. (lb_f)	Chamber pressure (psia)	Nozzle area ratio	Weight (lb_m)	Thrust/ weight (lb_f/lb_m)	Cycle time
H-1	LOX/RP-1	230,000	705	8:1	1,997	115.2	Gas generator
LR91-AJ3	LOX/RP-1	80,000	682	25:1	1,115	71.7	Gas generator
LR87-AJ3	LOX/RP-1	344,400	587	8:1	2,704	127.4	Gas generator
F-1	LOX/RP-1	1,748,000	980	16:1	18,740	94:6	Gas generator
F-1	LOX/RP-1	1,663,000	980	10:1	16,587	100.3	Gas generator
LR91-AJ5	N_2O_4/A-50	100,000	827	492:1	1,041	96.9	Gas generator
LR91-AJ11	N_2O_4/A-50	100,850	827	492:1	1,258	80.2	Gas generator
LR87-AJ5	N_2O_4/A-50	474,500	790	8:1	3,792	125.1	Gas generator
LR87-AJ11	N_2O_4/A-50	520,000	809	15:1	4,133	125.8	Gas generator
Agena	IRFNA/UDMH	16,000	500	45:1	290	55.2	Gas generator
J2	LOX/LH_2	230,000	718	22:5	3,454	66.6	Gas generator
J2S	LOX/LH_2	265,000	1,200	10:1	3,800	69.7	Gas tap
SSME	LOX/LH_2	470,000	3,000	775:1	6,339	74.1	Staged comb

JSC Propulsion and Power Division

Comments on Boeing Propulsion Data from April 7, 1975 memorandum
(2-5730-0000-139)

- The maximum chamber pressure reasonable with RP-1 regenerative cooling is limited to approximately 2000 psia. This is caused by the very high coolant velocity and corresponding pressure drop required to prevent coking the coolant tubes (need to keep the coolant wall temperature below 800 to 1000°F).

High pressure oxygen cooling has yet to be verified in this country. The bulk temperature rise must also be maintained below a certain level to prevent oxidation with certain chamber materials.

In summary a significant amount of technology would be required to go beyond the 2000 psia chamber pressure.

- If propellant cost is a significant factor O_2/RP-1 should be considered for a space engine application. While the O_2/RP-1 combination has 2.6% lower performance than the O_2MMH, when one considers combustion kinetics and thrust chamber cooling, the actual delivered performance difference is expected to only be approximately 1%.

- A new man rated FLOX/CH_4 engine could not be available by 1980, 1985 would be a more reasonable earliest availability.

- Because of problems such as: propellant cost, reusable chamber cooling, and reusable turbo machinery, it is recommended that OF_2/B_2H_6 only be considered for small pressure fed propulsion modules where long life and reusability is not required.

- A new man rated F_2/H_2 engine could not be available by 1980, 1985 would be a more reasonable earliest availability.

- Because of the major unsolved problems associated with the handling, storage and cost associated with ozone, it is recommended that the earliest O_3/H_2 engine availability would be 2000.

- A man rated F_2/Li/H_2 engine could not be available by 1980, 1990 would be a more reasonable earliest availability.

- It is very doubtful that a man rated O_2/Be/H_2 engine could be available by 1990, 2000 would be a more reasonable earliest availability.

2.4 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 2.4-1).

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 2.4-1. 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)	2,547	8,536	2,547	8,536
Ascent midcourse	10	33	10	33
Circularize (includes 26½° plane change)	1,787	5,862	1,787	5,862
Deorbit	1,787	5,862	1,844 (28½° plane change)	6,050
Descent midcourse	10	33	10	33
Trajectory correction during braking passes			100	328
Circularize at 160 nmi	2,447	8,028	64	210
Total	8,588	28,174	6,362	20,872

IEF-507

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 2.4-1 and 2.4-2. 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:

$$\frac{V}{V_o} = \exp\left\{ -\frac{C_D A}{M} \rho_p \left[2\pi r_p H \left(\frac{e+1}{e}\right)\right]^{1/2} \right\} \left\{1 + \frac{C_D A}{M} \rho_p \left[\frac{\pi H r_p}{2} \frac{(e+1)}{e^3}\right]^{1/2}\right\}$$

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

$C_D A/M$ 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 $C_D A/M$. Note that acceleration = D/M = $C_D A \rho V^2/2M$ and that $C_D A \rho/2M$ 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.

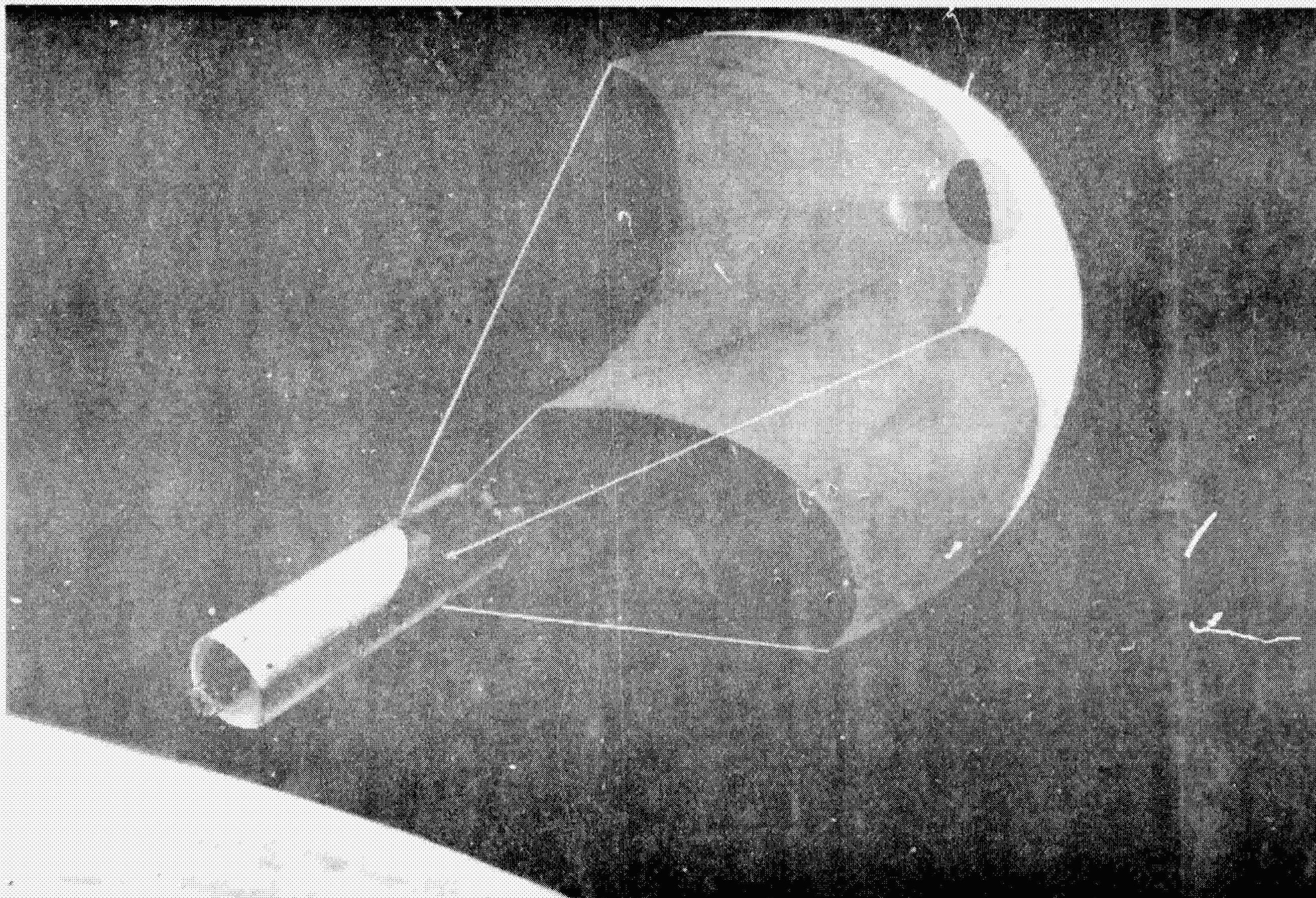


Figure 2.4-1 Aerobraking Concept

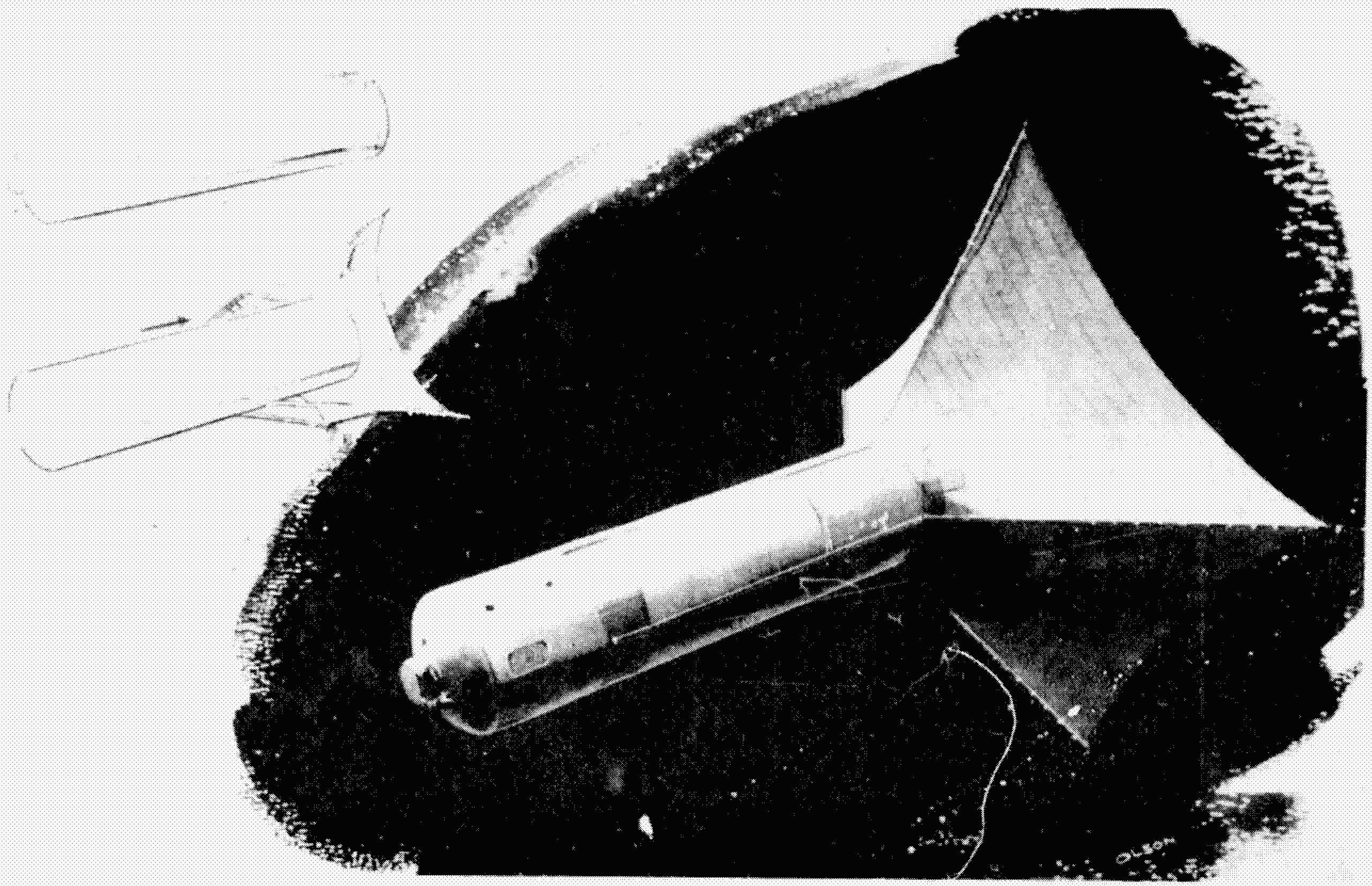


Figure 2.4-2 Aerobraking Concept

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 2.4-2.

Table 2.4-2 Aerobraking Temperatures

AREA	REFERENCE STUDY		REDUCED	
	^{o}K	^{o}F	^{o}K	^{o}F
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 = 7.2 Btu/ft^2. For aluminum with specific heat 0.225 and density of 2.7 kg/1. (168 lb/ft^3), : 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).

3.0 ANALYSIS OF SPACE DISPOSAL OF TOTAL SOLIDIFIED NUCLEAR WASTE

Disposal of refined waste was described in section 3.9 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

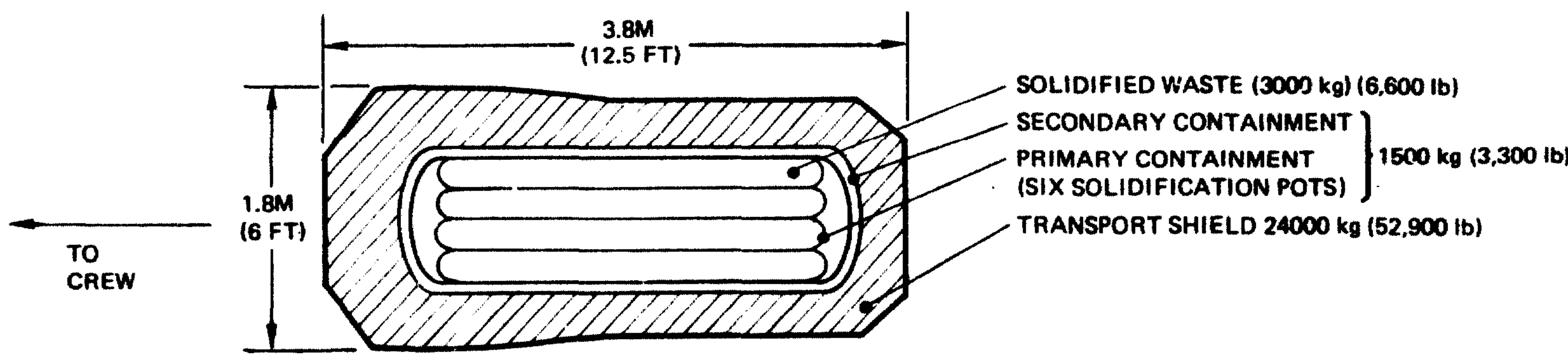


Figure 3-1. Nuclear Waste Disposal Total Waste Packaging Option

Table 3-1. Total Nuclear Waste Disposal Requirements

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 transportation 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 gliders. The OTV's operate 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.

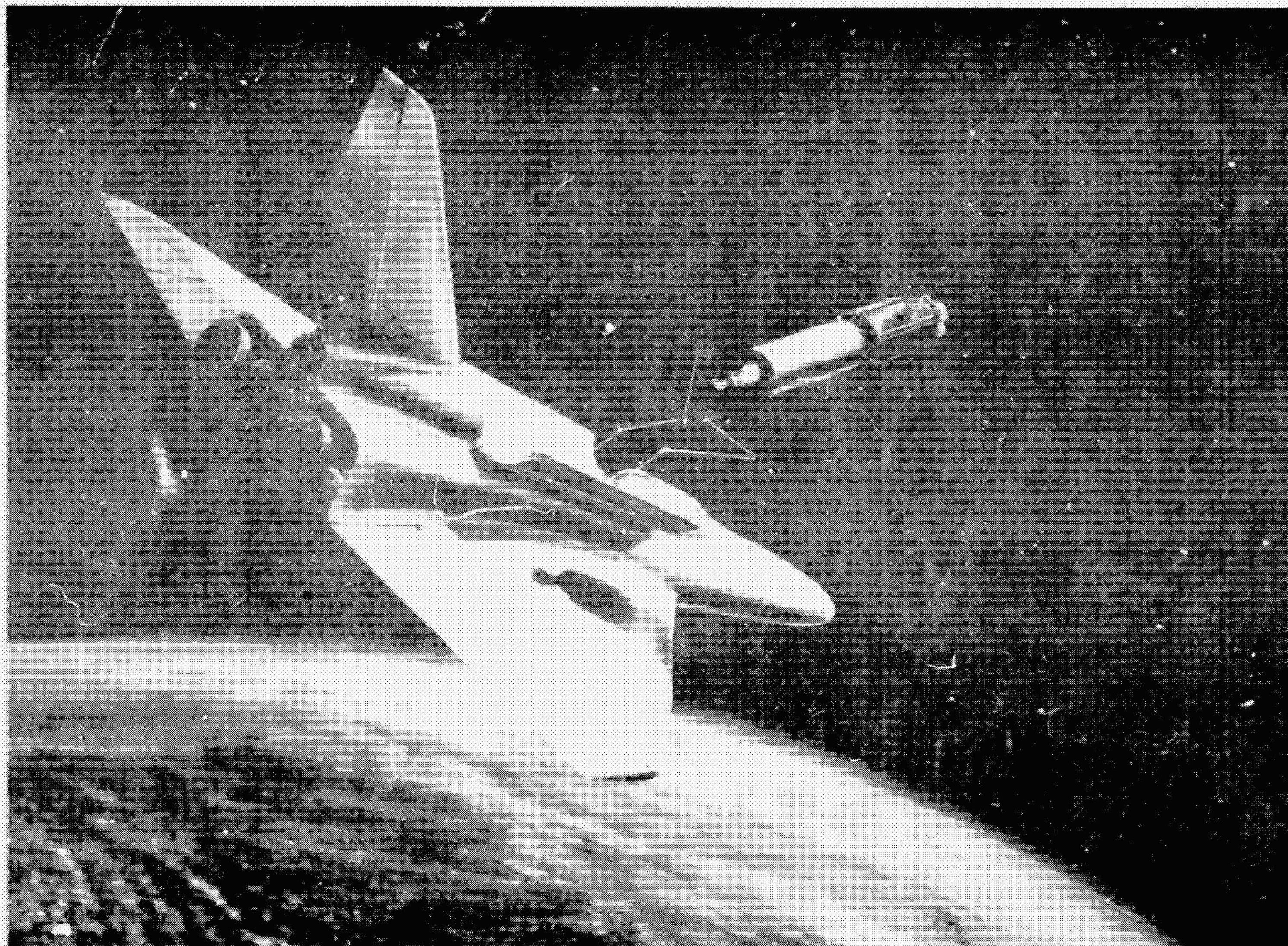
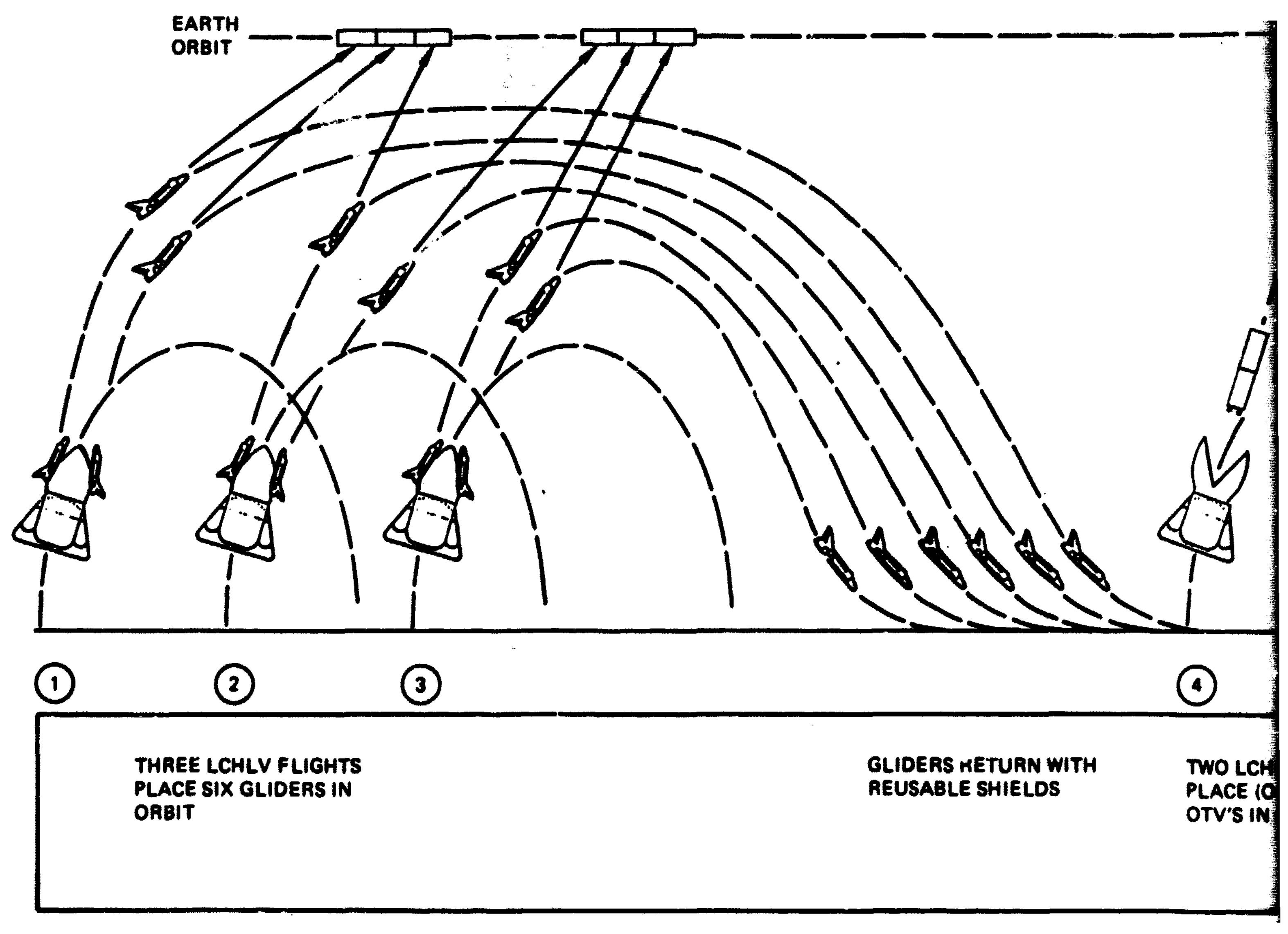


Figure 3-2. Boeing Concept Single Stage-to-Orbit (SSTO)



FOLDOUT FRAME 1

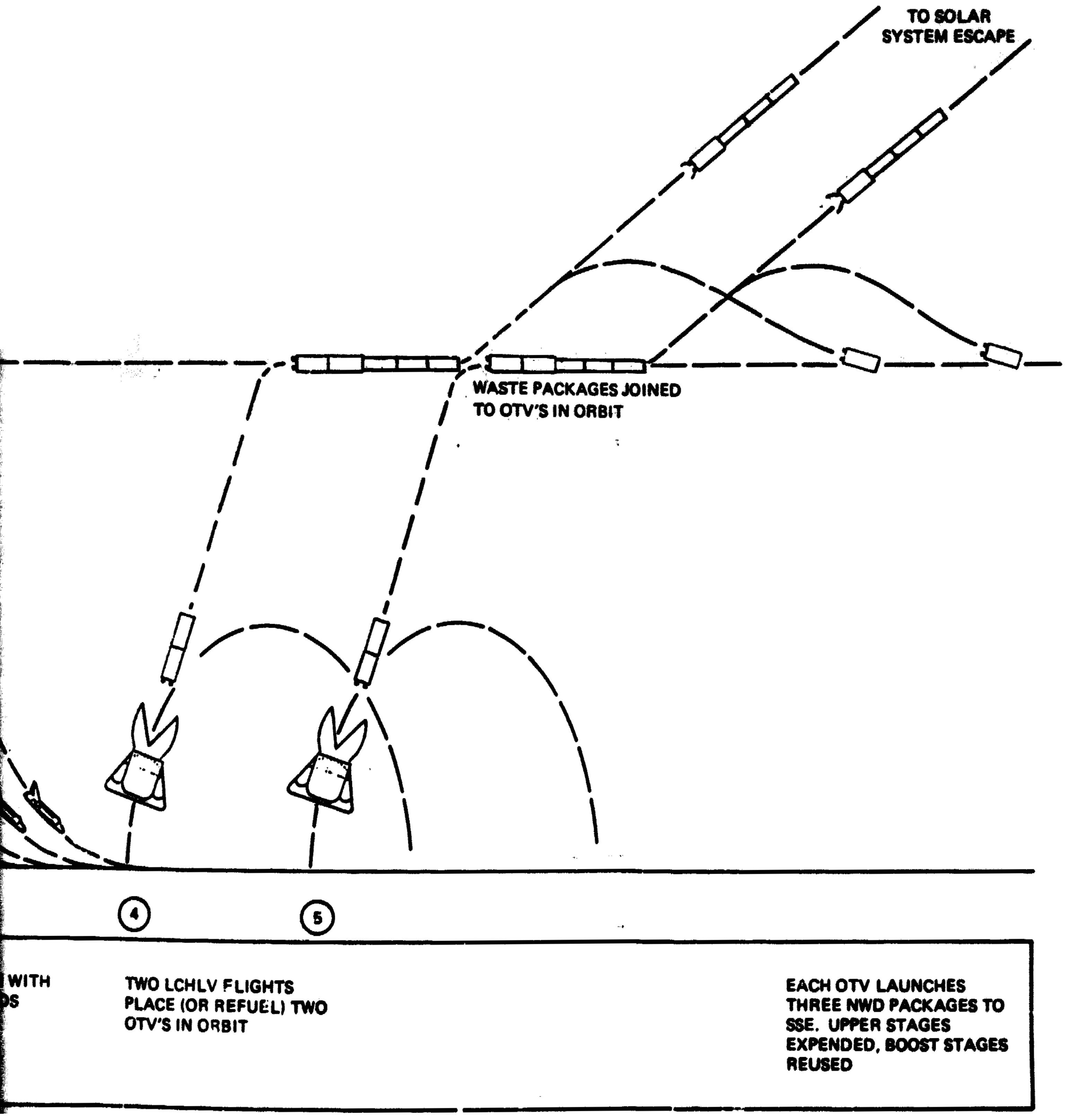
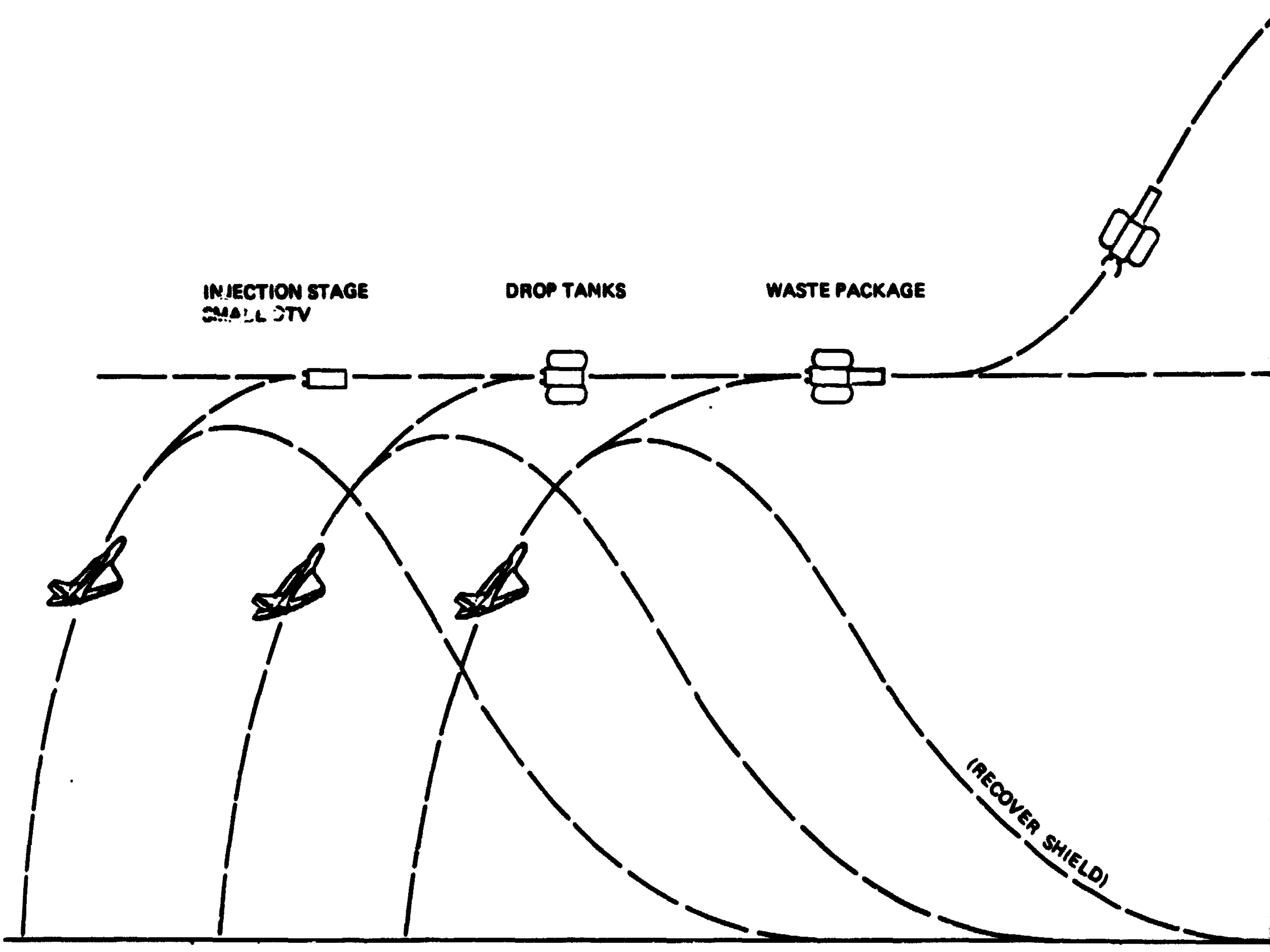


Figure 3.3 Transportation Mode for Nuclear Total Waste Disposal



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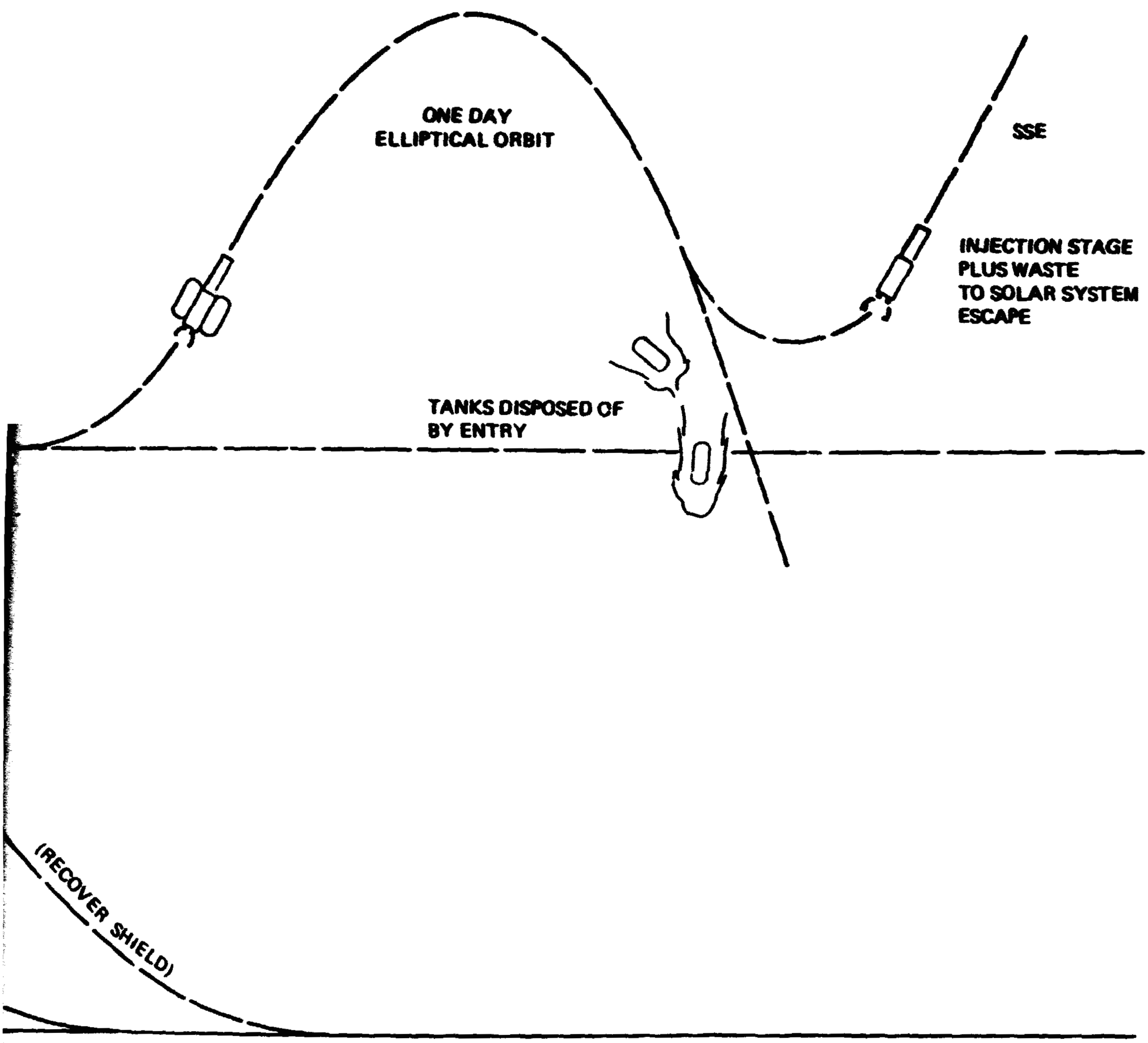

Figure 3-4 Nuclear Waste Transportation Sequence Employing SSTO and 1-½ Stage OTV

FOLDOUT FRAME 2

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.

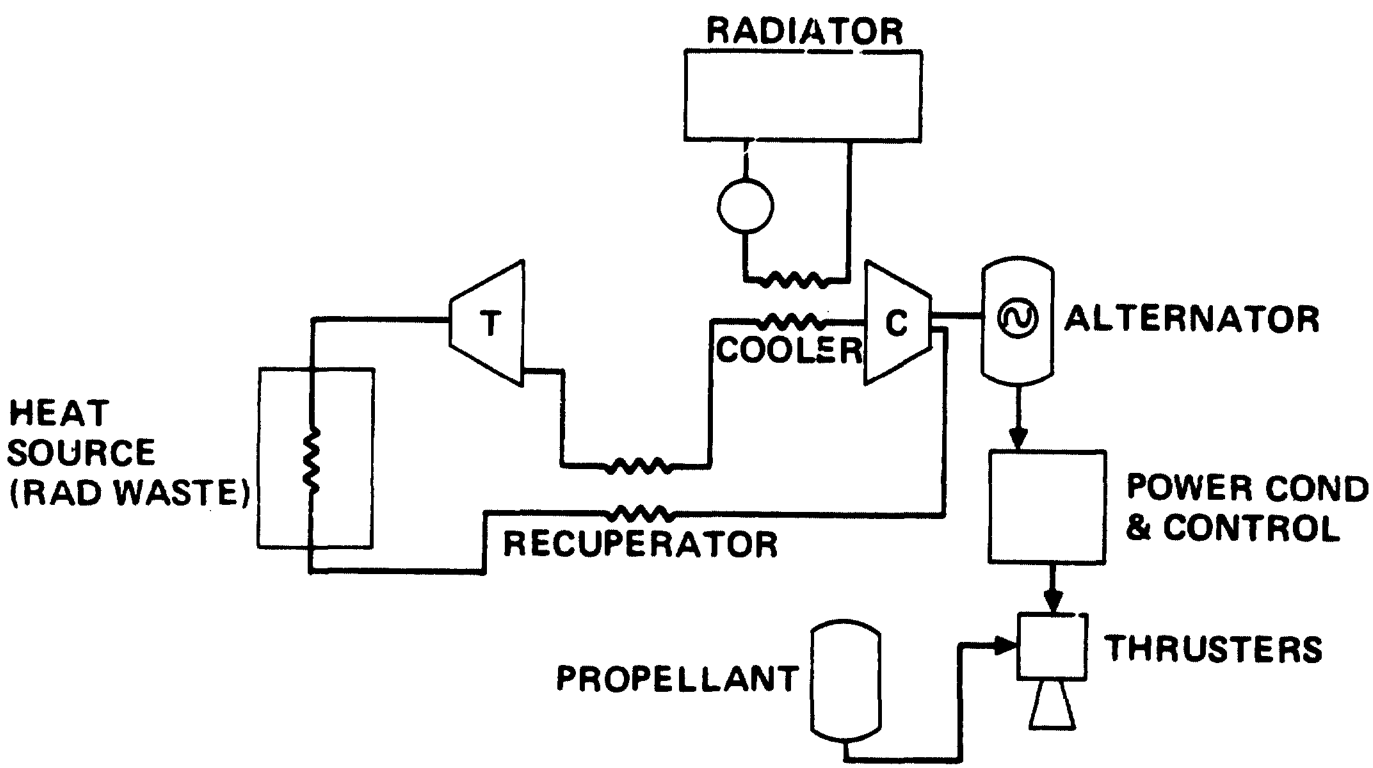

Figure 3-5. System Schematic

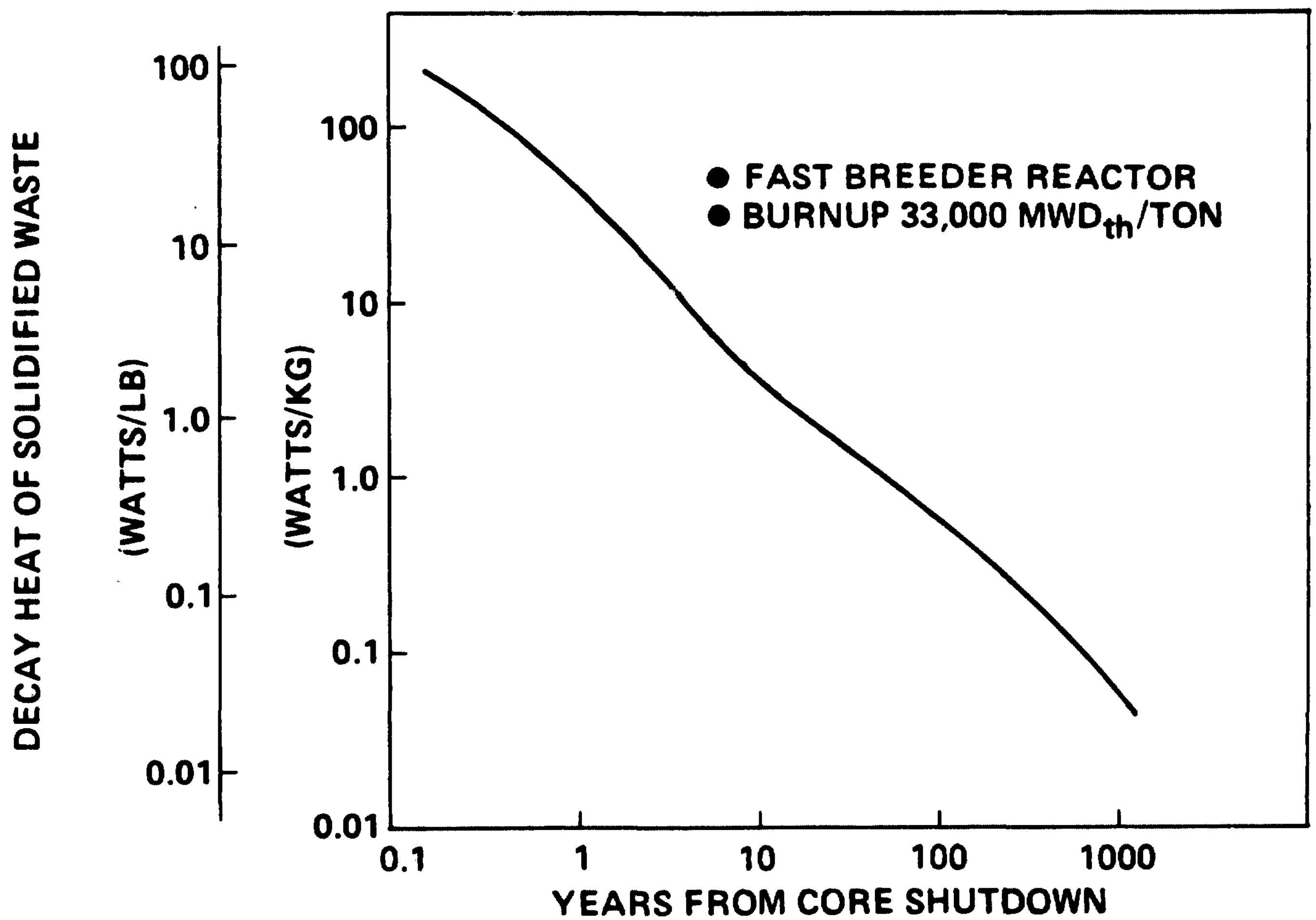

Figure 3-6. Thermal Decay for Solidified Waste

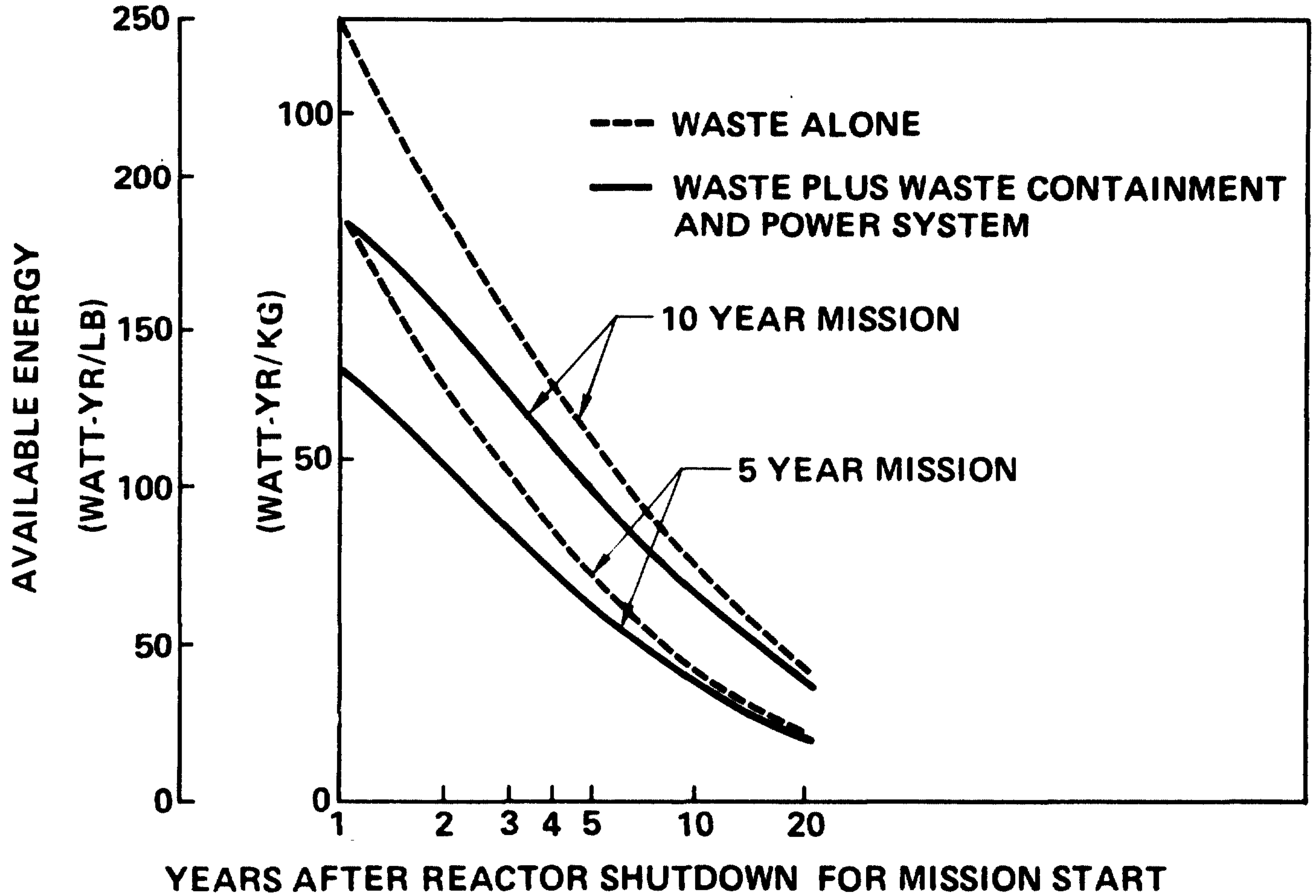

Figure 3-7. Energy Available From Total Solidified Waste

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.

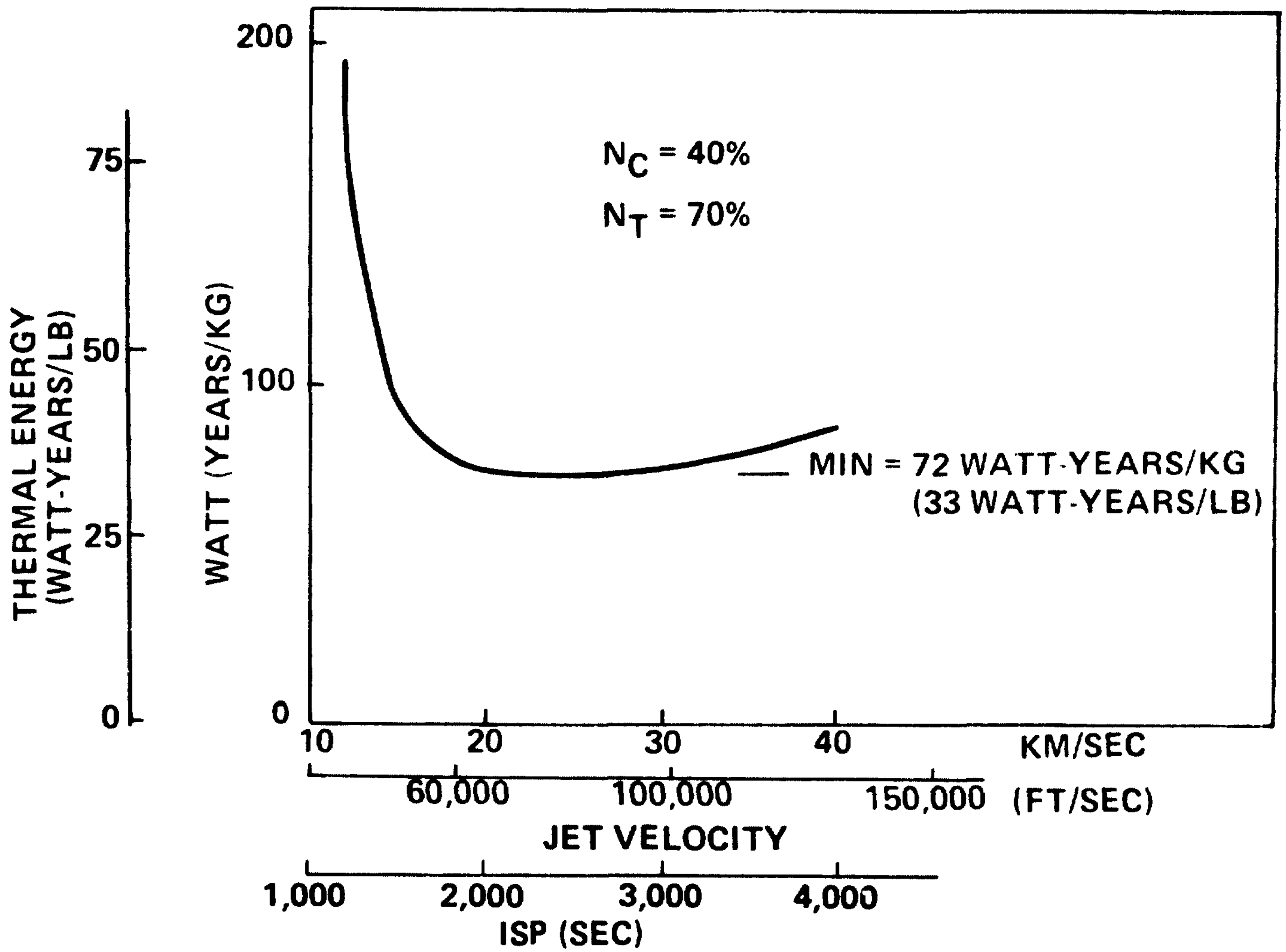

Figure 3-8. Energy Required for Solar System Escape