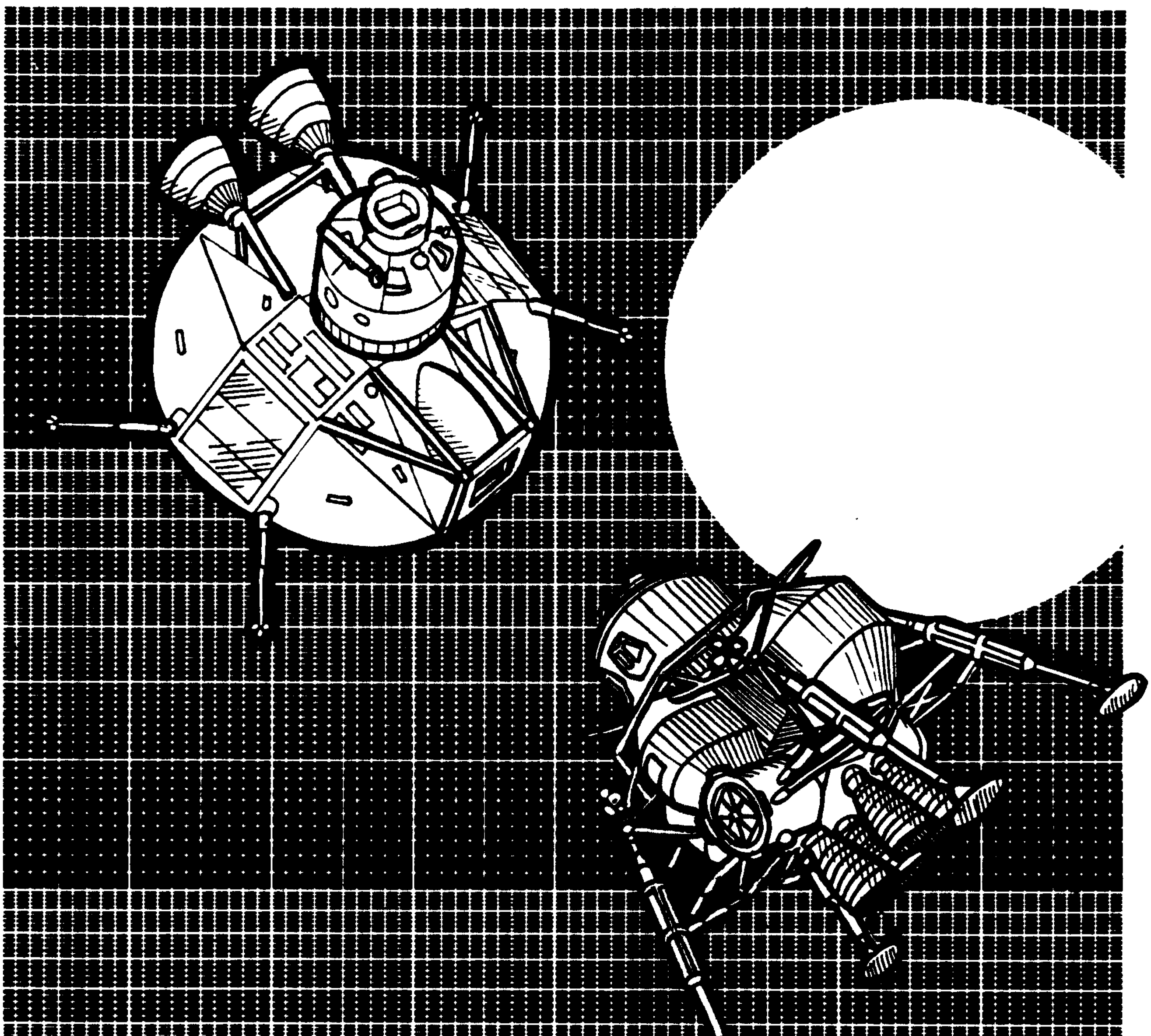
NASA CR-172061

Spacecraft Mass Estimation, Relationships, and Engine Data

LBSS



(NASA-CR-172061) SPACECRAFT MASS ESTIMATION, RELATIONSHIPS AND ENGINE DATA: TASK 1.1 OF THE LUNAR BASE SYSTEMS STUDY Report, Oct. 1987 - May 1988 (Eagle Engineering) 85 p CSCL 22B G3/16

N88-29836

Unclas 0142432

NASA Contract Number NAS 9-17878
EEI Report #87-171
April 6, 1988

EAGLE

SPACECRAFT MASS ESTIMATION RELATIONSHIPS AND ENGINE DATA

TASK 1.1 OF THE LUNAR BASE SYSTEMS STUDY

PREPARED UNDER CONTRACT TO THE
ADVANCED PROGRAMS OFFICE
AT
JOHNSON SPACE CENTER

BY

EAGLE ENGINEERING, INC.
HOUSTON, TEXAS

NASA CONTRACT NO. NAS 9-17878

EAGLE CONTRACT NO. TO-87-57

TASK 1.1 REPORT

EAGLE REPORT NO. 87-171

APRIL 6, 1988

FOREWORD

This report was prepared between October 1987 and May 1988 by Eagle Engineering, Inc. for the Advanced Programs Office of Johnson Space Center, a field center of the National Aeronautics and Space Administration. The objective is to present data for a variety of spacecraft and their subsystems, and to collect equations which are used for mass estimation and sizing of such vehicles.

Dr. J.W. Alred was the NASA technical monitor for the Advanced Space Transportation Study contract of which this task was a part. Mr. Andy Petro was the NASA task monitor for this particular task. Mr. W.B. Evans was the Eagle Project Manager and Mr. W.R. Stump was the Eagle Deputy Project Manager. Mr. C.C. Varner was the Eagle task manager for this task. Other participants providing valuable advice and information were Mr. G.R. Babb and Mr. P.G. Phillips. Mr. M. Stovall provided the cover.

TABLE OF CONTENTS

TABLE OF CONTENTS

(Continued)

List of Tables

List of Figures

List of Abbreviations

A-50	Aerozine 50 (N_2H_4 50%, UDMH 50%)
ACS, ACPS	Attitude Control Propulsion System
AIAA	American Institute of Aeronautics and Astronautics
AMTV	Advanced Mars Transport Vehicle
AOTV	Aerobraked Orbital Transfer Vehicle
ECLSS	Environmental Control and Life Support Systems
EVA	Extra-Vehicular Activity
FPR	Flight Performance Reserve
IRFNA	Inhibited Red Fuming Nitric Acid (HNO_3 14%, NO_2 2%, HF 7%)
Isp	Specific Impulse
JSC	Johnson Space Center
LEM	Lunar Excursion Module
LEO	Low Earth Orbit
LMDE	Lunar Module Descent Engine
LO_2	Liquid Oxygen
LOI	Lunar Orbit Injection
LH_2	Liquid Hydrogen
LSPI	Large Scale Programs Institute
MLI	Multi-Layered Insulation
MLV	Medium Lift Vehicle
MMH	Monomethyl Hydrazine (CH_3 - N_2H_3)
MPD	Magneto-Plasma Dynamic
MPS	Main Propulsion System
MSC	Manned Spaceflight Center (predecessor to JSC)

NEP	Nuclear Electric Propulsion
OMS	Orbital Maneuvering System
OTV	Orbital Transfer Vehicle
PAM	Payload Assist Module
RCS	Reaction Control System
SRB	Solid Rocket Booster
SSME	Space Shuttle Main Engine
STS	Space Transportation System (Space Shuttle)
TBD	To Be Determined
TEI	Trans-Earth Injection
TLI	Trans-Lunar Injection
TPF	Terminal Phase Finalization
TPI	Terminal Phase Initiation
UDMH	Unsymmetrical Dimethyl-Hydrazine $(CH_3)_2N_2H_2$

1.0 Introduction

This book contains a collection of scaling equations, weight statements, scaling factors, etc., useful to someone doing conceptual design of trans-lunar spacecraft. It provides rules of thumb and methods for calculating quantities of interest. Basic relationships for conventional--and several non-conventional--propulsion systems (nuclear and solar electric, and solar thermal) are included. The equations and other data have been taken from a number of sources and are not all consistent with each other in level of detail or method, but provide useful references for early estimation purposes.

Scaling equations are presented on two levels: overall vehicle sizing and sub-system sizing. The equations for overall vehicle sizing are quick and simple. They should be used when extreme accuracy is not a prerequisite. When higher fidelity is required, and time is not an overriding concern, the vehicle can be sized by sub-system, using the sub-system sizing equations and relationships.

Vehicle sub-systems can be broken down in any number of ways. To prevent confusion, a list of general sub-systems discussed throughout this book is presented here:

Propellant
Engines
Avionics
Structures
Aerobrakes and Heatshields
Environmental Control and Life Support
Crew
Power and Electrical
Landing and Docking
Propellant Tanks
Insulation and Thermal Protection
Attitude Control

The relationships and other numbers collected here are primarily for Orbital Transfer Vehicles (OTV's) operating between Low Earth Orbit (LEO) and Low Lunar Orbit (LLO), and for lunar surface landers/launchers.

2.0 Conventional Chemical Propulsion Systems

The first problem in sizing a spacecraft is to define the burnout mass (M_{bo}) of the vehicle accurately. The burnout mass is a combination of the inert mass, the payload mass, and the mass of any trapped fuel.

The inert mass for Orbital Transfer Vehicles (OTV's) and the various propulsive elements of the lunar transportation system can be estimated using the following equation.[26]

2.0-1) $$\text{Stage Inert Mass} = \frac{(C + B * Mp) + F * M_{PLA}}{(1 - F)}$$

Where :

Mp = Stage Propellant Capacity (in kg).

M_{PLA} = the maximum amount of payload that will be carried through the aerobraking maneuver (in kg).

F = the aerobrake mass fraction
= .15 estimated default value

and:

C =	2279 kg,	B = .04545	for cryogenic stages
C =	2352 kg,	B = .0228	for pump-fed storable stages
C =	2454 kg,	B = .04253	for pressure-fed storable stages

In this equation, "C" represents the mass of all OTV subsystems which are not dependent upon the propellant mass (M_p). These subsystems are known as invariant or constant mass systems. Data Communications, Power, Attitude Control, Structure, and even the Engines can be considered constant mass systems for OTV's. "B" is a mass factor which is used to estimate the mass of those systems which are dependent upon the propellant mass. These systems include Propellant Tanks and Thermal Protection.

The size of the aerobrake is dependent on the entry mass that must be aerobraked. F, which is the fraction of the entry mass that is aerobrake takes this effect into account.

$$\text{Aerobrake Mass} = F\ (\text{entry mass})$$

Due to the increased structural stiffness required for landers and launchers, equation 2.0-1 is not valid for vehicles operating to and from the lunar surface. The Large Scale Programs Institute (LSPI) has developed a similar equation for Oxygen/Hydrogen (cryogenic) lunar landers.

2.0-2) $Mi = B^* Mp + C$

where: Mi = Inert Mass

Mp = Propellant Mass

and: $B = 0.2$

$C = 1800$ <kg>

In this equation, "B," the mass factor for propellant dependent systems, is considerably larger than its counterpart in the OTV scaling equation (2.0-1). This is due to the higher strength requirements of landers and launchers.

The mass of invariant systems ("C") remains approximately the same as that of the OTV's. However, this theory is not universally accepted.[35] The argument has been made that for landers and launchers there are few systems which are invariant or constant mass systems. During time critical maneuvers, such as ascent and descent, the performance and load capabilities of a lunar lander or launcher must be maintained within fairly narrow tolerances. Therefore, sub-systems such as Engines, Structures, Attitude Control, and landing gear should be scaled with the gross or deorbit mass of the vehicle rather than be considered invariant. In this case, the equation for predicting the lander or launcher inert mass would be:

2.0-3) $$Mi = A^* Mg + B^* Mp + C$$

where: A = 0.0640

B = $0.0506^* F_D$

C = 390

and: Mi = Inert Mass <kg>

Mg = Gross Mass <kg>

Mp = Propellant Mass <kg>

where: F_D = Density Factor

The constants in this equation have been chosen using the Apollo Lunar Module as a guide.

The density factor (F_D) adjusts the "B" coefficient to account for different types of propellants. The volume effects of less dense cryogenic propellants are taken into account. The density factor is defined to be the ratio of the bulk density of Lunar Excursion Module (LEM) propellants to the bulk density of the propellants desired for use.

2.0-4) $$F_D = \frac{Db(l)}{Db(d)}$$

where: Db(l) = Bulk density of LEM propellants

Db(d) = Bulk density of desired propellants

For Example:

The LEM used a 1.6 mixture ratio of Nitrogen tetroxide (N_2O_4) oxidizer and Aerozine-50 fuel. From Table 2.2-5 (Page 25), Nitrogen tetroxide has a density of 89.52 <Lb/ft^3>, and Aerozine-50 is 56.10 <Lb/ft^3>. Using the bulk density equation at the bottom of the same table, it is found that the bulk density of the LEM propellants Db(1) is 72.83 <Lb/ft^3>. If a 6:1 mixture ratio of Liquid Oxygen to Liquid Hydrogen is desired, than the bulk density of the desired propellants Db(d) is 22.54 <Lb/ft^3>, and the density factor (F_D) is 3.23.

2.1 Propellant System

After the mission has been defined, an analysis will result in a set of maneuvers that the vehicle must complete. Each of these maneuvers will require a change in the vehicle's velocity (delta V). The propulsion system must supply this delta V through the expenditure of propellant.

2.1.1 Propellant Requirements

The first key parameter for finding the total propellant mass is the propellant's Specific Impulse (Isp). It is the best indicator of propulsion performance.

2.1-1) $Isp = T / R_m$

T = Total Engine Thrust

R_m = Rate of Propellant Mass Flow

The propulsion system has an Isp rating based on the engine design and the propellant mixture combination. In Section 2.2, Rocket Engine Performance, a list of Isp ratings can be found for various systems which are being used today or have been used in the past.

The propellant mass (M_p) can be related to the vehicle's Isp, burnout mass (M_{bo}), and required velocity change (delta V) using a modified form of Tsiolkovsky's Equation:

$$M_p = M_{bo} * (e^{([\text{Delta V}]/(g * Isp))} - 1) + M_u + M_v \quad (2.1\text{-}2)$$

g = Acceleration of Gravity at Earth's surface

M_v = Mass of the vented propellant (Boiloff)

M_u = Unused propellant

Calculating the mass of the vented propellant (Mv) is the topic of Section 2.1.4. Determining the amount of unused propellant (Mu) is discussed is Section 2.1.3. Note that unused propellant is also a component in the burnout mass.

Generally, the velocity changes (delta V's) are calculated using a simulation program which can determine the maneuvers and the optimal performance for the mission desired. Obtaining a detailed Maneuver Summary and Mission Plan is beyond the scope of this report. The reader interested in this subject is referred to references 39, 40, and 41.

2.1.2 Velocity Changes and Gravity Losses

The velocity changes (Delta-V's) required to transfer from one orbit to another are "impulsive" Delta-V's, which assume instantaneous velocity change. In practice, the velocity change must be performed by a rocket which takes a finite (sometimes rather long) time to finish the task. For a large Delta-V such as translunar injection (~ 10,500 ft/sec or 3.2 km/sec), even with a steady 1-g acceleration, Thrust to Weight (T/W)=1--over 5 minutes is required for the maneuver.

As the maneuver time increases, that is, as the thrust to weight ratio decreases, the maneuver becomes less efficient and the total Delta-V required increases. The differences between the real first burn Delta-V and the impulsive case are called "Gravity Losses" and are caused by the vehicle moving away from perigee during the burn. These "gravity losses" only become significant when the total engine burn time (maneuver time) becomes a significant percentage of the orbit period. This is a function of initial T/W, Delta V required, period of the initial orbit, and Isp.

For the Translunar injection (TLI) burn (3.2 km/sec) from Low Earth Orbit with a T/W of 1 (burn time 5 min), the g-losses are only 6 m/sec (20 fps). If T/W is .1 (50 min burn time) the losses increase to 360 m/sec (1200 fps), a substantial loss. This can be reduced dramatically by using a 2 burn option in which about half of the Delta V is delivered on the first burn. The engines are stopped and the spacecraft coasts around in the resultant ellipse and finishes the burn as the vehicle approaches and transits perigee. This intermediate ellipse has a period of about 4 hours. This technique reduces the g-losses for T/W = .1 from 360 m/sec to 75 m/sec (240 fps), an acceptable loss.

To find g-losses in the general case requires numerical integration through the burn and comparison with the impulsive case. However, for the lunar program where the Delta V is relatively constant and only a few Isp value level groups will be used, an analytical expression can be empirically derived. For TLI and an Isp. ~ 450 sec, the g losses are given by:

2.1-3) $\quad$ g-losses = 1635 m/sec / [1 - 9.86 T/W + 512 $(T/W)^2$] (from Ref. 26)

for a single burn and 1/3 to 1/4 of that for the two burn case. In this equation, T/W = initial thrust to weight, and it is assumed that the thrust remains constant throughout the burn (the weight changes). This means for the multi-stage cases that the stages are identical at least so far as thrust is concerned.

G-losses at the Moon (LOI and TEI) are not large for the lunar transport cases. The large stack weights (and large Delta V) are only at Earth departure while the actual thrust levels are a constant in the scenarios being considered.

2.1.3 Unusable Propellant

The unused propellant consists of Flight Performance Reserve (FPR) and Trapped Fuel. Based on empirical data gathered for numerous vehicles (Appendix A&B), the FPR and Trapped Fuel make up about 2.25% of the total propellant mass for the main propulsion system. Figure 2.1-1 shows a graphical depiction of the unusable propellant for those vehicles in Appendices A and B.

2.1.4 Insulation and Boiloff

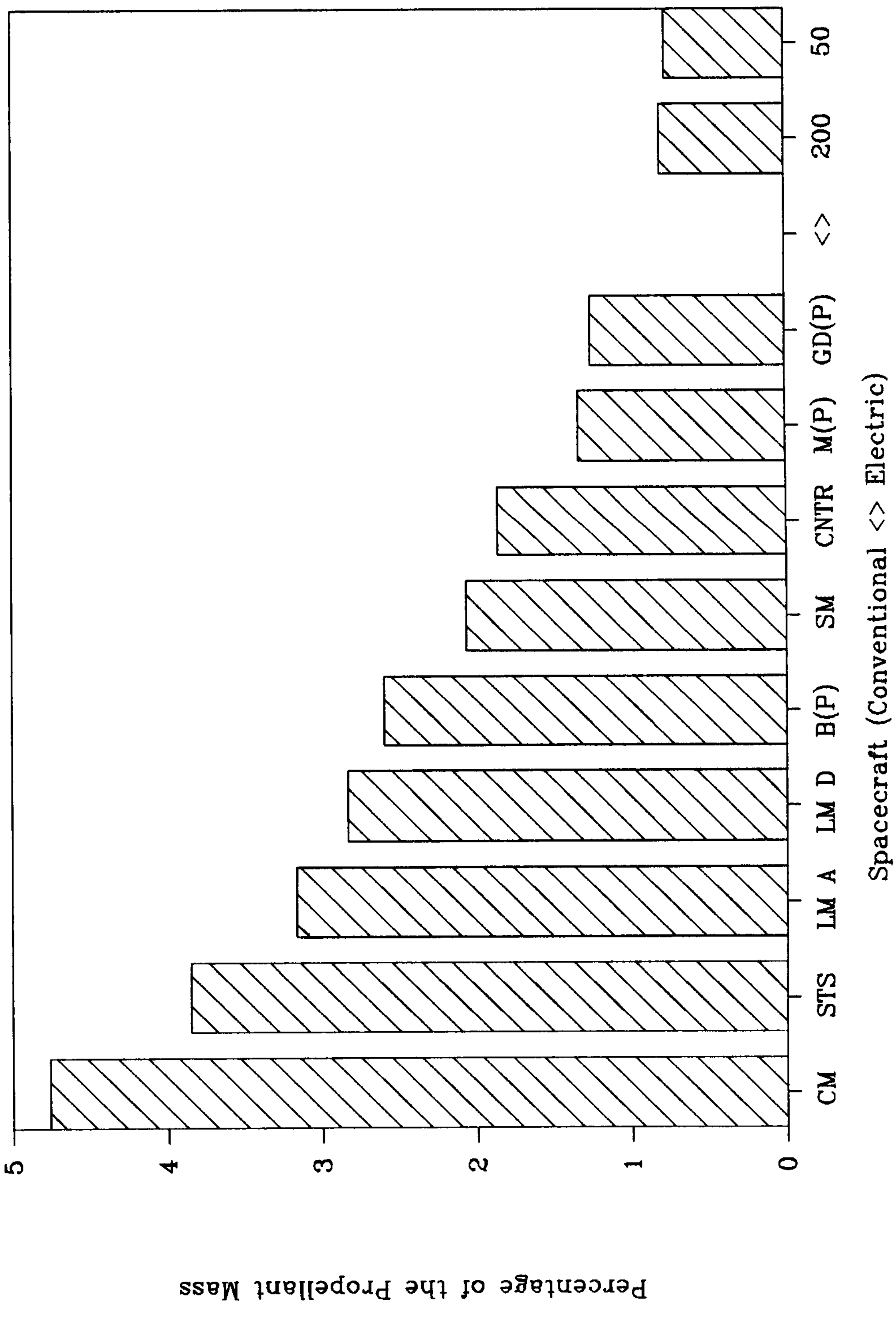
Spacecraft using liquid hydrogen as a fuel will have substantial propellant boiloff. As the propellant boils and returns to a gaseous state, it is vented to space. The venting is necessary to prevent an excessive internal pressure build-up.

The boiloff rate of propellant is dependent on many factors. The primary factors are:

- tank shape and size
- the operating pressure of the tank
- the thickness (X) and thermal conduction coefficient (K) for the tank and its insulation
- the absorptivity (a) and emissivity (e) of the tank surface coating
- the boiling point temperature (T_b), density (d), and heat of vaporization (H) of the propellant.

Figure 2.1—1

Trapped Fuel



The basic equation for a spherical, vented propellant tank of radius "r" is:

2.1-4) $R\% = F / (d * H * r) = RM_b / M_p$

Where:
- $R\%$ = Boiloff rate as a percentage of total mass <%/unit time>
- F = Heat Flux
- RM_b = Rate of propellant boiloff by mass <mass/unit time>
- M_p = Mass of propellant
- H = Heat of vaporization of propellant
- d = Propellant density

The heat flux (F), used in the above equation, is the rate at which heat passes through a unit area of tank wall. It is a function of the thickness, thermal conduction coefficient of the tank and its insulation, and the temperature difference (T) between the internal and external surfaces of the tank. It can be calculated using the following equation.

2.1-5) $F = K * T / X$

Where:
- T = $T_e - T_b$
- T_e = External Temperature of the Tank
- T_b = Internal (boiling) Temperature of the Tank
- X = Insulation Thickness
- K = Thermal Conduction Coefficient

The thermal conduction coefficient varies from 0.00003 Btu/(hr. ft. °F) for Super insulation to 130 Btu/(hr. ft. °F) for Aluminum[18]. Table 2.1-1 contains some of the properties of various types of insulations.

The temperature on the external surface of the tank (T_e) is affected by the tank surface properties, lighting conditions, and the existence of cooling systems such as vapor cooled shields. Assuming that the tank is not shaded and its entire surface area is exposed to solar radiation the external temperature without a cooling system is defined by the absorptivity and emissivity of the surface materials. The absorptivity to emissivity ratio can range from 0.1 for silvered teflon to 9 for copper or black nickel[20]. The external temperature of tanks with these surface materials is between -150°F and 800°F respectively when in near-Earth space.

The internal temperature of the tank is essentially the boiling point temperature (T_b) of the propellant. Table 2.1-2 shows some of the properties of cryogenic propellants.

The insulation thickness is obtained from the detailed design. For long duration Mars missions the optimal thickness for Multi-Layered Insulation (MLI) is between 2 and 4 inches for oxygen and hydrogen tanks[19]. The Centaur G upper stage vehicle has 1.5 inches of foam insulation on its hydrogen tank and none on the oxygen tank[12]. The optimal insulation thickness (X_{opt}) can be calculated from an equation obtained in reference 18.

2.1-6) $X_{opt} = (K * T * t / (M_f * d_i * H))^{\wedge}0.5$

Where:
- t = Time Exposed to Heating
- M_f = Spacecraft Mass Fraction
- d_i = Density of Insulation
- K = Thermal Conduction Coefficient
- T = External/Internal Temperature Difference
- H = Heat of Vaporization

This equation does not take into account the performance losses due to the differences in insulation mass. Therefore, it should not be used for long duration flights where the insulation mass would be a large percentage of the total vehicle mass.

Once the type and thickness of the insulation is known, the surface area (A) to be insulated can be determined from the tank size, and the weight of the insulation can be calculated.

2.1-7) $W_i = X * d_i * A$

Example:

The heat flux for the Centaur G hydrogen tank, with approximately 500 ft^2 of spherical surface area in near earth space, is 4 Btu/(hr*ft^2)[12]. Compare this to the Advanced Mars Transfer Vehicle of reference 19 which, with its vapor cooled shield, has a 0.12 (Btu/(hr* ft^2) heat flux for 3 inch Multi-Layered Insulation. Determine the propellant boiloff rate.

2.1-4) $R\% = F/(d*H*r)$

$F = 4 <btu/(hr* ft^2)>$	(Heat Flux - Given)
$H = 194.4 <Btu/Lb>$	(Heat of Vaporization of H_2 - Table 2.1-2)
$d = 4.37 <lb/ft^3>$	(Density of H_2 - Table 2.1-2)
$S = 500 <ft^2)$	(Surface Area - Given)

$r = \sqrt{S/(\pi * 4)}$ (Tank Radius)

$= 6.3 <ft>$

Table 2.1-1: Insulation Density and Thermal Conductivity

SUMMARY OF THERMAL CONDUCTIVITY VALUES

Material	Pressure	Temperature (°R)	Density ρ(lb/ft³)	Therm Cond K(Btu/hr ft°F)
Aluminum		Room	200	130.0
Magnesium (Alloys)		*	110	35.0 (to 100)
Stainless Steel		*	500	10.0
Mag Oxide		*	220	1.0
Solid Plastic Teflon, Nylon Polyethylene		*	100	0.15
Foam Plastic				
Polycel	1 Atmos	40 to 500	2	0.005
Styrofoam	1 Atmos	40 to 500	2	0.010
Santocel A	1 Atmos	160 to 520	6	0.014
	0.1 μ Hg	160 to 520	6	0.0012
Perlite	0.1 μ Hg	36 to 520	8	0.00072
Fiberglass AA	1 Atmos	140 to 535	15	0.0125
	10 μ Hg	140 to 535	15	0.00031
	10 μ Hg	140 to 525	4	0.0005
Super insulation:	1 Atmos	36 to 520	3-5	0.020
Multiple reflective layers SI-4 Alt. Al Foil Paper NRC-2 Crinkled Aluminized Mylar	0.1 μ Hg	36 to 520	3-5	0.000030

Table 2.1-2: Properties of Cryogenic Substances

PROPERTIES OF CRYOGENIC SUBSTANCES*

Substance	He	H_2	CH_4	NH_2	N_2	O_2	F_2
Density, 32°F 1 Atm lb/cu ft	0.01114	0.00561	0.0448	0.0481	0.0781	0.0892	0.106
Boiling Point, 1 Atm, °F	-452.0	-423.0	-258.7	-28.03	-320.4	-297.4	-306.5
Melting Point, 1 Atm, °F	26 Atm** -458.0	-434.6	-299.2	-107.9	-345.8	-361.1	-363.3
Vapor Density at BP, lb/cu ft	0.999	0.0830	0.1124	0.0556	0.288	0.296	-
Liquid Density at BP, lb/cu ft	7.803	4.37	26.47	42.58	50.19	71.29	94.4
Vapor Pr. Solid at MP, mm Hg		54	70	45.2	96.4	2.0	0.12
Heat of Vap. at BP, Btu/lb	10.3	194.4	248.4	588.6	85.7	91.6	73.7
Heat of Fusion at MP, Btu/lb	<1.8	25.2	26.1	151.2	11.0	5.9	5.8
Cp 59°F, 1 Atm, Btu/lb°F	-292°F** 1.25	3.39	0.528	0.523	0.248	0.218	0.180
Cp/Cv., 59° -68°F, 1 Atm	-292°F** 1.66	1.41	1.31	1.31	1.40	1.40	-
Critical Temperature, °F	-450.2	-399.8	-116.5	270.3	-232.8	-181.1	-200.2
Critical Pressure, PSIA	33.2	188.1	672.0	1638.7	492.3	730.3	808.3

*Above from table from Linde Company data card

**Pressure and Temperature conditions at which data is valid

Using Equation 2.1-4, it is possible to calculate the boil-off rate percentage and the rate of propellant boiloff.

$$R\% = 4/(194.4 * 4.37 * 6.3)$$
$$= 0.001$$
$$= 0.1\%/hr \qquad \text{(Boiloff Rate Percentage)}$$
$$RM_D = 0.001 * M_p = 0.001 * 4/3 * \pi * r^3 * d$$
$$= 0.001 * 4/3 * \pi * (6.3)^3 * 4.37$$
$$= 4.6 \ \langle Lb/hr \rangle \qquad \text{(Rate of Propellant Boiloff)}$$

The boiloff rate percentage (R%) for the Centaur G hydrogen tank is calculated to be 0.1% per hour or 4.6 lb per hour. This compares well with the 5 lb per hour rate known to exist for this vehicle[21].

2.1.5 Propellant Requirements for Rendezvous and Docking

The propellant required for a vehicle to rendezvous and dock with an object in space is primarily related to the vehicle's weight, propellant specific impulse (Isp), and the velocity change (delta V) that must be performed. The weight of the vehicle varies with spacecraft type, the Isp with propellant type; but the delta V should remain relatively constant.

The coelliptic rendezvous, developed during the Gemini Program, is the standard docking procedure. This three/four orbit technique requires four maneuvers - Phase Adjustment, Coelliptic, Terminal Phase Initiation (TPI), and Terminal Phase Finalization (TPF). In low earth orbit, this rendezvous technique ideally requires a total delta V of 55 ft/s for the phase adjustment maneuver, 55 ft/s for the coelliptic maneuver, and 70 ft/s during the terminal phase - total of 180 ft/s (55 m/sec).

The delta V's required for the phase adjustment and coelliptical maneuvers do not have much variation from one mission to another. However, typical Gemini missions required 1.5 to 2 times more fuel (or delta V) during the terminal phase was predicted by the ideal calculations. In fact, a docking accident on Gemini X quadrupled the ideal terminal phase fuel requirement. To make allowances for such accidents, or human error, the Apollo Lunar Excursion Module (LEM) was allotted fuel for a rendezvous and docking delta V of 500 ft/s (152 m/sec). This is 2.5 times the predicted delta V requirement for an ideal rendezvous and docking procedure.

The mass of the propellant used can be calculated from equation 2.0-1, where the mass of the vented propellant (M_v) is zero and the delta V is 500 ft/s.

2.2 Rocket Engine Performance, Mass, and Specific Impulse

The following tables provide the mass, thrust, Isp, and other data for a variety of historical, existing, and proposed rocket engines. The relationship between engine thrust and engine weight is graphically described in Figures 2.2-1 and 2.4-1. Figure 2.2-2 is a bar chart of the engine assembly mass for the vehicles in Appendix A. The vehicles are arranged in order of thrust level. Tables 2.2-5 and 2.2-6 provide data for a variety of propellant combinations as well as propellant properties.

Table 2.2-1: Rocket Engine Performance (LO_2/LH_2)

Liquid Oxygen / Liquid Hydrogen Engines

Application	Thrust (vac) lbf	Engine Mass lbm	Isp (vac) sec.	MR	Ref.
NASA M-1 Booster (proposed)	1,500,000	20,000	428	5	7
Booster (proposed)	588,400	7,775	446	7	7
Space Shuttle Main Engine (SSME)	470,000	7,000	453	6	1
Saturn IB & V Second Stage (J-2)	230,000	3,480	422	5.5	1,4,6
Centaur - Upper Stage Vehicle (RL-10)	15,000	292	444	5	1,4,6
Advanced Space Engine (proposed)	7,500	870	485	-	9
RCS (proposed)	1,250	31	427	4.5	7
Advanced OTV (proposed)	500	44	456	6	8
Space Propulsion/RCS (proposed)	25	3.8	400	3	7

Table 2.2-2: Rocket Engine Performance (LO_2/Hydrocarbon)

Application	Thrust (vac) lbf	Engine Mass lbm	Isp (vac) sec.	MR		Ref.
Saturn V S-1C Stage (F-1)	1,744,000	18,500	302	2	LO_2/RP-1	1,4
Booster (proposed)	735,300	5,335	346	2.4	LO_2/RJ-5	7
Titan I Booster Stage (AJ23-130)	344,400	2,704	289	2.25	LO_2/RP-1	7
Saturn IB Booster Stage (H-1)	236,000	2,003	295	2.23	LO_2/RP-1	23
Thor, Delta Rockets (MB-3)	195,100	2,080	289	2.15	LO_2/RP-1	23
Atlas Booster (MA-3A)	190,050	1,392	290	2.267	LO_2/RP-1	7
Titan I Second Stage (AJ23-131)	80,000	1,115	313	2.25	LO_2/RP-1	7
Atlas Sustainer (MA-3A)	85,200	1,019	309	2.27	LO_2/RP-1	3,6,23
Atlas Verniers (MA-3A)	1,003	51	239	1.8	LO_2/RP-1	1,2,6

Table 2.2-3: Rocket Engine Performance (Storable)

Application	Thrust (vac) lbf	Engine Mass lbm	Isp vac. sec.	MR		Ref.
Titan III-C Booster (AJ23-139)	520,860	3,841	302	1.915	N_2O_2/A-50	7
Titan III Upper Stage (AJ23-137)	100,000	1,258	312	1.8	N_2O_2/A-50	7
Apollo Service Propulsion System (AJ-10-137)	20,500	823	314	1.6	N_2O_4/A-50	7
Agena Propulsion System (Bell 8096)	16,000	320	293	2.57	IRFNA, UDMH	3,4
Apollo Lunar Module Descent Engine (LMDE)	9,750	415	303	1.6	N_2O_4/A-50	1,4
Transtage	8,150	211	303	2	N_2O_4/A-50	4
Shuttle OMS Engine (OMS)	6,000	297	316	1.65	N_2O_4/MMH	3,4
Apollo Lunar Module Ascent Engine (RS18)	3,500	230	306	1.6	N_2O_4/A-30	1,4,36
RCS (proposed)	1,600	12.8	280	1.6	N_2O_4/A-30	7

Table 2.2-4: Rocket Engine Performance (Solid)

Application	Thrust (Max) lbf	Total Mass lbm	Mass Fraction	Isp sec.	Ref.
Space Shuttle SRB	2,590,000	1,256,400	0.8835	287	24
Stage 1 Inertial Upper Stage (IUS)	56,000	24,000	0.8958	279	24,37
Perigee Kick Motor (Star 75)	54,594	17,783	0.9299	288	25
Space Shuttle PAM (Star 48)	16,860	4,659	0.946	288	25
Upper Stage Motor (Star 37)	12,298	2,532	0.929	290	25
Altair III (Star 20)	7,060	663	0.908	287	25
Orbit Insertion Motor (Star 13)	1,000	78	0.869	273	25
Apogee Kick Motor (Star 5)	499	4.4	0.87	289	25

Figure 2.2-1: Rocket Engine Weights

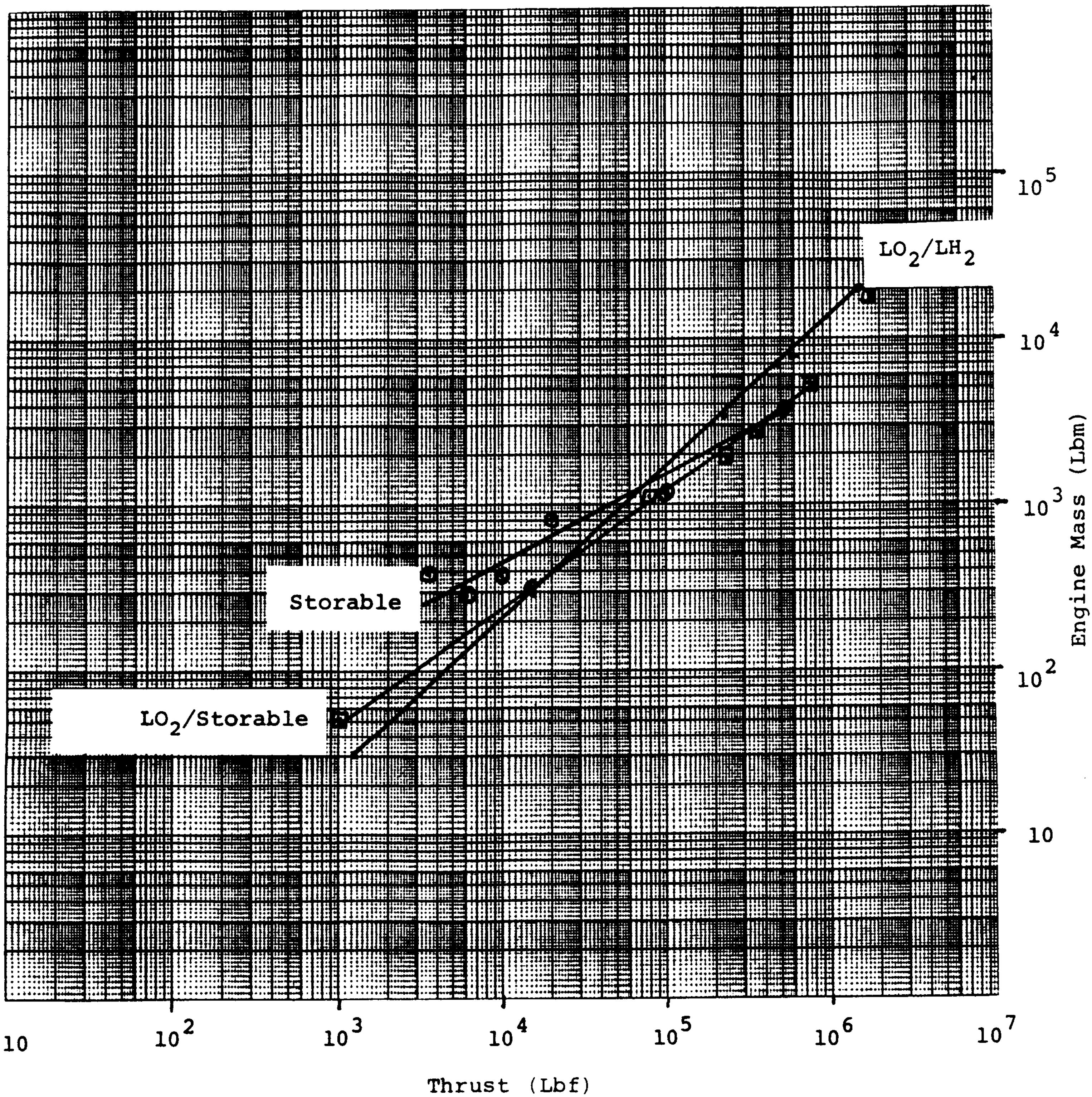


Figure 2.2-2

Space & Lunar Engine Systems

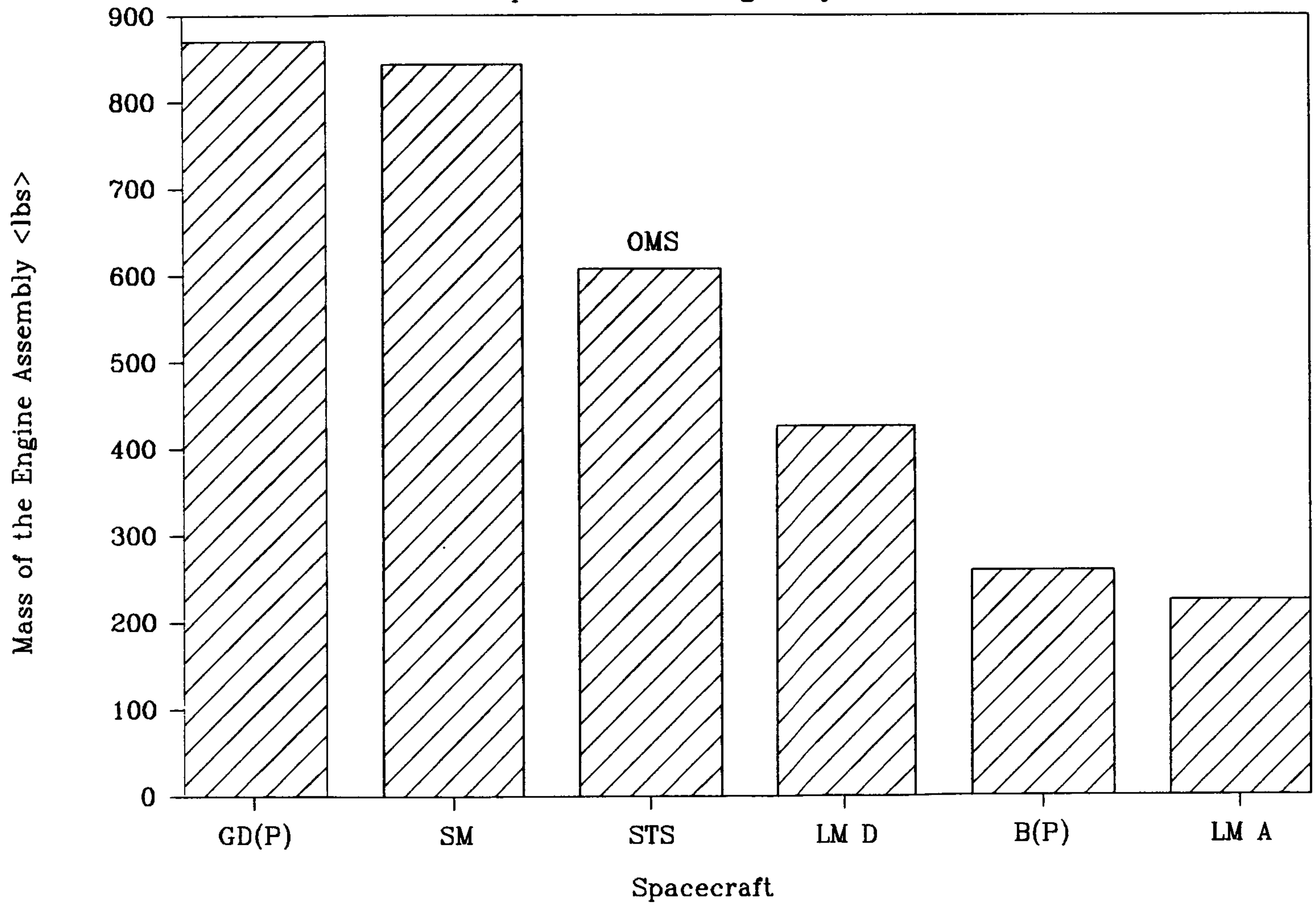


Table 2.2-5: Propellant Properties

PROPELLANT	FORMULA OR COMPOSITION	FREEZING POINT, °F	BOILING POINT, °F	DENSITY lb/ft^3	AT TEMP. °F
OXIDIZERS					
Chlorine Pentafluoride	ClF_5	-153.40	7.30	110.88	77
Inhibited Red Fuming Nitric Acid (IRFNA, Type III)	HNO_3 + 14% NO_2 2% H_2O + 0.7% HF	~ -86.00	~140.00	96.83	77
Liquid Oxygen	O_2	-361.83	-297.35	71.21	-297.35
Nitrogen Tetroxide	N_2O_2	11.75	70.40	89.52	77
Fluorine	F_2	-363.60	-306.90	94.27	-306.90
Tetranitromethane	$C(NO_2)_4$	56.50	258.50	101.80	77
FUELS					
AeroZINE - 50 (A-50)	50% N_2H_4 + 50% UDMH	22.00	158.00	56.10	77
Aniline	C_6H_5-NH_2	21.20	363.90	63.51	77
Hydrazine	N_2H_4	34.75	237.60	62.66	77
JP-X	40% UDMH + 60% JP-4	-71.00	~211.00	48.50	77
Liquid Hydrogen	H_2	-434.84	-423.21	4.419	-423.21
Methane	CH_4	-296.80	-259.20	26.48	-259.20
Monomethylhydrazine (MMH)	CH_3-N_2H_3	-62.30	189.80	54.32	77
RJ-5	$C_{14}H_{18.4}$	-65.00	470.00	66.80	77
RP-1	$(CH_2)_X$	< -50.00	~422.00	49.70	77
Unsymmetrical Dimethyl-Hydrazine (UDMH)	$(CH_3)_2N_2H_2$	-70.94	144.20	49.00	77

$$\mu_B = \frac{1 + MR}{\left\{\frac{MR}{\mu_O} + \frac{1}{\mu_F}\right\}} \qquad \frac{V_O}{V_F} = \frac{MR^*\mu_F}{\mu_O}$$

Where:

μ_B = BULK DENSITY (lb/ft^3)

MR = MIXTURE RATIO (lb_O/lb_F)

μ_O = DENSITY OXIDIZER (lb/ft^3)

μ_F = DENSITY FUEL (lb/ft^3)

$\frac{V_O}{V_F}$ = PROPELLANT VOLUME RATIO

Obtained from Reference 7.

Table 2.2-6 Propellant Performance

Propellant[a]		P_c = 1000 psia, optimum sea level expansion, shifting equilibrium					P_c = 100 psia, ϵ_e = 40, shifting equilibrium				
Oxidizer	Fuel	I_s, sec	r	ρ_p, lb/ft^3	T_c, °R	c*, ft/sec	I_{vac}, sec	r	ρ_p, lb/ft^3	T_c, °R	c*, ft/sec
O_2	H_2	388	4.00	17.7	5330	7950	454	4.50	19.0	5610	7840
	CH_4	310	3.15	50.7	6350	6120	364	3.25	51.0	5840	5940
	C_2H_2	330	1.60	53.5	7460	6590	384	1.65	53.8	6690	6390
	C_2H_4	312	2.45	55.2	6850	6100	365	2.45	55.2	6160	5960
	C_2H_6	307	2.95	55.9	6480	6050	362	2.95	55.9	5930	5900
	C_3H_8	305	2.75	56.8	6520	6020	359	2.82	57.0	5960	5860
	C_4H_8	305	2.55	57.6	6600	6000	359	2.55	57.6	6080	5830
	C_5H_{12}	302	2.90	58.7	6500	5960	356	2.90	58.7	5980	5790
	RP-1	299	2.55	63.5	6610	5890	351	2.60	63.7	6030	5730
	N_2H_4	313	0.90	66.5	6120	6220	367	0.95	66.6	5680	6060
	MMH	311	1.40	63.2	6390	6150	365	1.43	63.4	5880	5980
	UDMH	310	1.67	60.9	6490	6120	364	1.67	60.9	5940	5960
	50% N_2H_4 - 50% UDMH	312	1.29	63.8	6340	6170	366	1.32	63.9	5840	6000
	NH_3	295	1.41	52.1	5580	5880	347	1.41	5211	5260	5790
	Be	255	1.40	84.4	9730	4840	311	1.50	83.8	8190	4890
	53% Be - 47% H_2	454	0.93	15.5	5330	9100	535	0.93	15.5	5040	9080
F_2	H_2	410	8.00	28.9	7120	8370	473	10.0	33.1	6910	8020
	CH_4	345	4.35	63.8	7620	6750	408	4.45	64.1	7050	6650

[a] Propellant densities and heats of formation used are those given in the propellant property tables.

ORIGINAL PAGE IS
OF POOR QUALITY

Table 2.2-6 (cont.)

Propellant[a]		P_c = 1000 psia, optimum sea level expansion, shifting equilibrium					P_c = 100 psia, ϵ_e = 40, shifting equilibrium				
Oxidizer	Fuel	I_s, sec	r	ρ_p, lb/ft^3	T_c, °R	c*, ft/sec	I_{vac}, sec	r	ρ_p, lb/ft^3	T_c, °R	c*, ft/sec
F_2	C_2H_4	329	2.50	64.0	7630	6370	392	2.50	64.0	7110	6290
	C_3H_8	334	3.25	68.6	7540	6480	395	3.30	68.8	7030	6390
	RP-1	318	2.56	75.3	7450	6180	377	2.56	75.3	6970	6090
	B_2H_6	371	5.45	68.2	8980	7370	431	5.50	68.4	7920	7110
	B_5H_9	360	4.60	75.0	9120	7130	417	4.60	75.0	8000	6860
	N_2H_4	364	2.25	81.6	8420	7280	422	2.30	81.8	7580	7070
	MMH	347	2.38	77.6	8200	6840	408	2.43	77.8	7400	6650
	UDMH	339	2.45	74.4	7770	6610	400	2.45	74.4	7030	6440
	50% N_2H_4 - 50% UDMH	351	2.40	78.5	8160	6940	412	2.40	78.5	7350	6730
	NH_3	360	3.27	69.8	8240	7210	417	3.30	69.9	7450	7000
	Li	380	2.54	62.2	10180	7590	441	2.54	62.2	8920	7300
	39% Li - 61% H_2	438	1.07	12.9	4130	8660	504	1.07	12.9	3550	8410
N_2O_4	N_2H_4	292	1.33	75.6	5870	5860	342	1.36	75.8	5500	5740
	MMH	288	2.17	74.4	6110	5730	338	2.26	74.8	5660	5570
	UDMH	287	2.60	72.8	6190	5680	336	2.70	73.2	5720	5520
	50% N_2H_4 - 50% UDMH	289	2.00	74.6	6050	5740	339	2.05	74.9	5620	5600
	NH_3	273	2.00	61.4	5260	5490	318	2.00	61.4	5020	5460
H_2O_2 (100%)	B_2H_6	333	1.84	49.7	4880	6460	391	1.85	49.7	4660	6280
	B_5H_9	313	2.25	63.9	5450	6100	370	2.25	63.9	5230	6030
	N_2H_4	288	2.27	79.4	5270	5770	338	2.32	79.6	5010	5700
OF_2	H_2	402	5.75	23.5	6190	8200	464	6.00	24.2	5910	8000

[a] Propellant densities and heats of formation used are those given in the propellant property tables.

2.3 Aerobrakes and Heatshields

The fraction of the entry mass that is aerobrake or heat shield is a key number for scaling computations. The size of the aerobrake/heatshield is also of interest. Aerobrake and heatshield types, roughly in order of weight, include ablative, metallic, ceramic or other tile, cloth, and ballute. The following table lists a few types and numbers of interest.

Vehicle	Aerobrake Type	Entry Mass <Klbm>	Aero-brake Mass <Klbm>	Area <ft^2>	Aerobrake/ Entry Mass	Veh.Entry Mass/Area <lbm/ft^2>
Apollo CM	Ablative Heat Shield	9.00	2.62	160	.29	56
STS Orbiter	Tile Heat Shield	139	27.7	5000	.20	28
GD SBOTV (proposed)	Fabric Aerobrake	6.4 (GEO round trip)	1.5	1,256	.23	5.1
Boeing GBOTV (proposed)	Ballute Aerobrake	10.5 (GEO round trip)	.93	854	.089	12
JSC SBOTV (proposed)	Tile Aerobrake	28.6 (lunar round trip)	4.3	1,256	.15	23
H.Davis OTV (proposed)	Aerobrake				.10	

Aerobrakes vary from 9 to 29% of the entry mass in the numbers noted. Some optimists will consider 2 %. 15% is a number commonly used. Heat shields, such as those used on the Shuttle and the Apollo Command Module, are generally heavier--20 and 30% of the entry mass respectively.

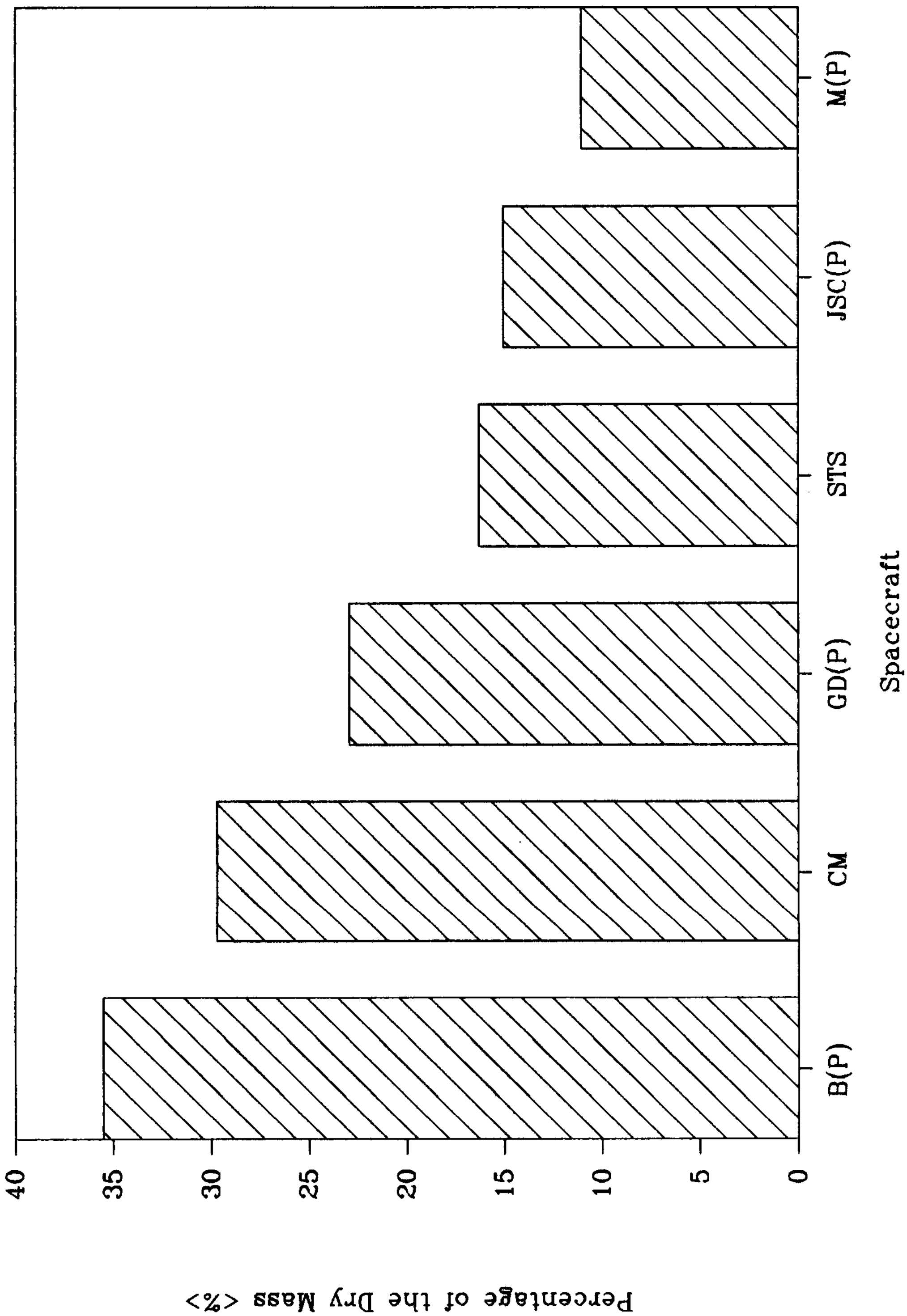


Figure 2.3-1
Heatshields and Aerobrakes

2.4 Propellant Tankage

The propellant tankage mass can be obtained from Figure 2.4-1 if the propellant volume is known. The propellant volume can be calculated from the density (Figure 2.2-3) and the propellant weight obtained from the equations of section 2.1. Figure 2.4-2 is a bar chart of some of the vehicles in Appendix A&B.

In order to use Figure 2.4-1, the following procedure should be employed:

1. Determine the volume of the propellant (V_B) using the equations at the bottom of the chart.

2. Choose the type of propellant system from the legend (1 to 5).

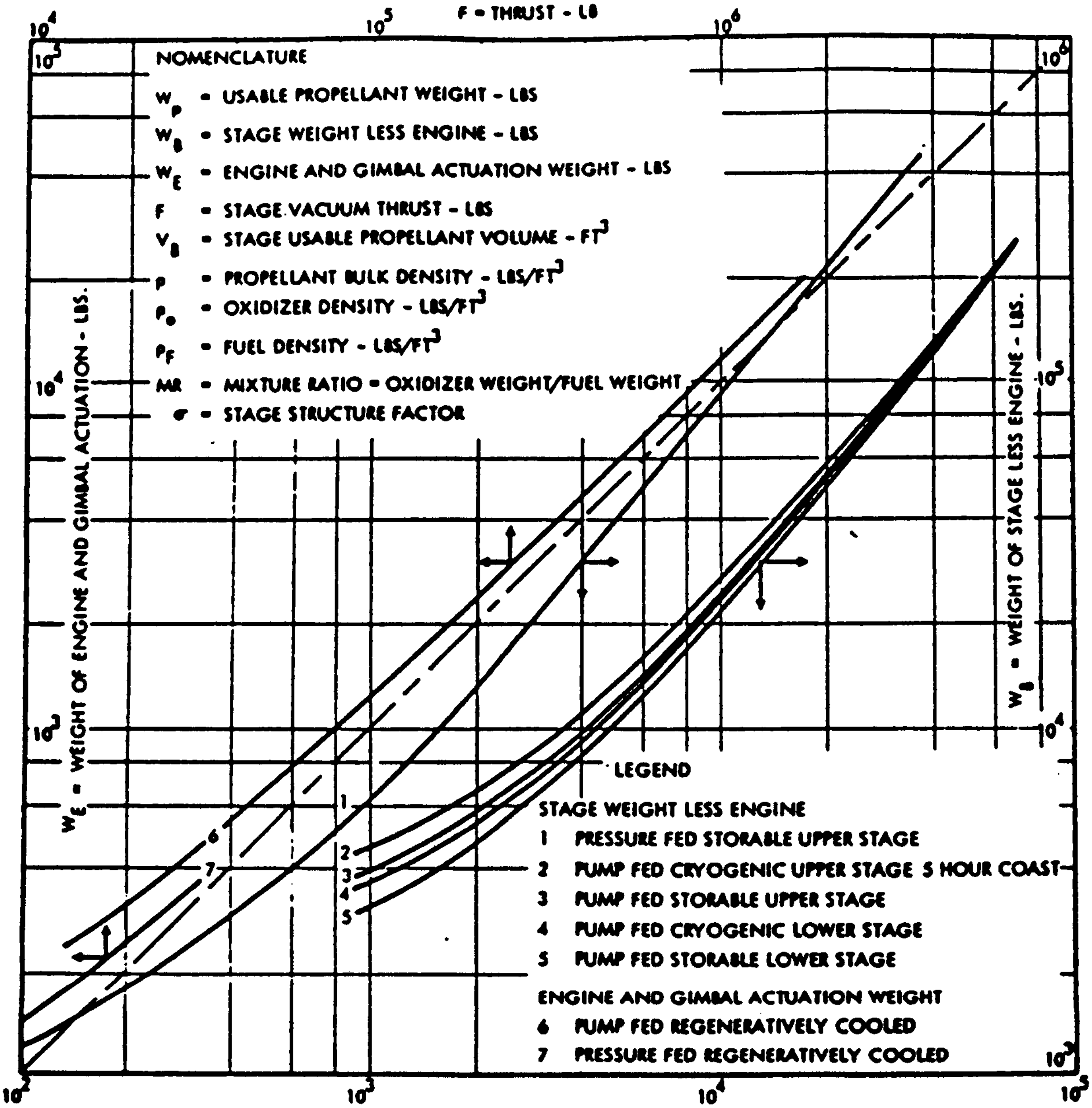
3. Read the stage weight without engines by moving vertically along the propellant volume line to the appropriate propellant system graph, then read the stage weight from the right side of the chart.

4. Choose the desired thrust level.

5. Choose the type of Engine system from the legend (6 or 7).

6. Read down and to the left to obtain the weight of the engine and actuation system.

Figure 2.4-1: Weight of Tanks and Rocket Engines



$$V_B = \frac{W_P}{\rho} = \frac{W_P}{(1+MR)}\left(\frac{MR}{\rho_o} + \frac{1}{\rho_F}\right) = \text{STAGE USABLE PROPELLANT VOLUME} = FT^3$$

PROCEDURE

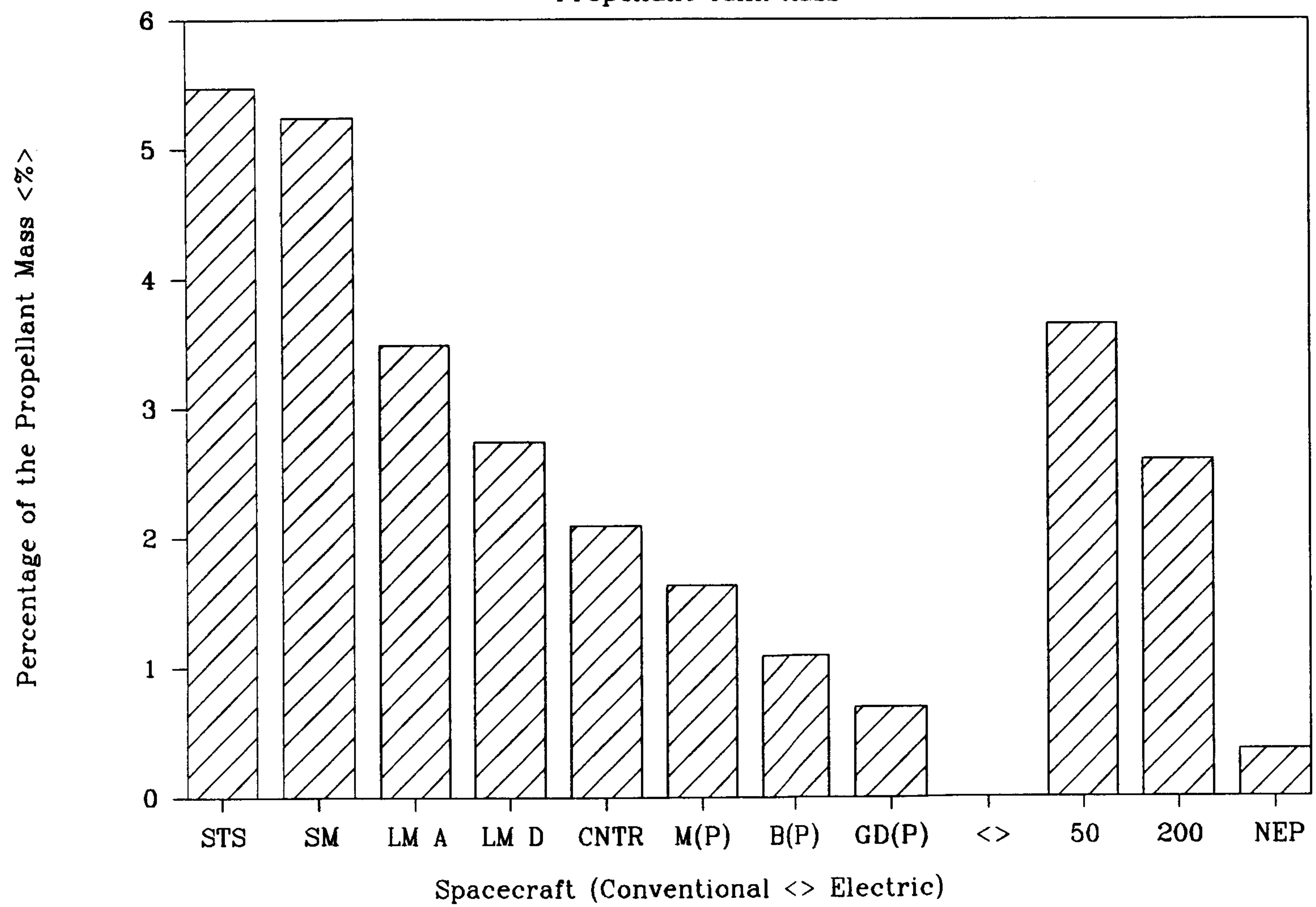
1. DETERMINE V_B AND READ W_B FROM GRAPH
2. AT DESIRED THRUST LEVEL READ W_E FROM GRAPH (IF REQUIRED USE MULTIPLE ENGINES TO OBTAIN HIGHER THRUST THAN SHOWN ON GRAPH)
3. $\sigma = \dfrac{W_B + W_E}{W_B + W_E + W_P}$

*Obtained from Reference 18

ORIGINAL PAGE IS OF POOR QUALITY

Figure 2.4-2

Propellant Tank Mass



3.0 Electric Propulsion Systems

Electric propulsion systems are more integrated with the trajectory than impulsive systems. As the thrust to weight goes down, typically the delta V and trip time go up. The key parameters of interest include:

delta V	- Velocity change (meters/sec)
Isp	- Specific Impulse (kgm-sec/kgf or lbm-sec/lbf)
E_t	- Thruster Efficiency
Psp	- Specific power for the power source (W/Kg)
P	- Power available (W)

Electric propulsion systems use electricity from an external power source to accelerate propellants and create thrust. These systems are generally characterized by high specific impulse and low thrust levels. There are three type of electric system: electrothermal, electromagnetic, and electrostatic.

Electrothermal systems utilize electricity to heat propellants and then accelerate them in a nozzle. These systems are thermodynamic rockets and are not considered in this document.

Electromagnetic systems utilize an electrically generated magnetic field to accelerate the propellants. These systems include magneto-plasma dynamic (MPD) systems, pulsed plasma thrusters, rail guns, and mass drivers. Only MPD systems are discussed in this chapter.

In MPD thrusters, a high current is passed through a gas which is heated and turned into a plasma. This plasma is accelerated by the magnetic field created by the current flow.

Electrostatic (ion thruster systems) use the electric charge of ions created in a chamber. The ions are attracted to one end of the chamber by a high voltage grid called an accelerator.

The components of an electric system are the power source, power converter, power conditioner, and the thruster. Power source and conversion systems are usually taken together. The systems of note are solar photovoltaic, solar dynamic, nuclear thermoelectric or thermionic, and heat engines. The power conditioning system is used to supply the appropriate electrical energy to the thruster. MPD systems require high current and relatively low DC voltage. Ion systems require high voltage but lower current. A more extensive discussion of power systems is contained in Section 4.8.

Electric propulsion systems are heavily dependent upon the mission plan and the trajectory design. Delta V and thrust level are closely related for low thrust electric propulsion systems. As the thrust goes down the Delta V can go up dramatically. The thrust/weight controls the Delta V. The total power, (specific power) and Isp control the thrust/weight.

The overall system can be sized beginning with the power system or the thruster system. Thus a power system may be chosen and a thruster system will be sized to match it; or a thruster system may be chosen and the power system will be sized to match thrusters. An optimum power/thruster system may be found for each mission.

The following steps describe a method of iterating through a design beginning with the power system:

- Select the desired type of thruster system. MPD and Ion are the two proposed for consideration here. The MPD has lower Isp and can handle higher power levels. The ion thruster has a high Isp but can not handle high powers without numerous thrusters. .

- Select the Isp for the system. The different types of thrusters can be designed to function over a range of Isps. (Section 3.1 & 3.2). Isp and power level can become variables in an optimization scheme. For a first cut, however, the Isp is simply chosen from experience.

- Select the power system type, power level, and specific power. The total power available, specific power, and the percentage of power available for the thruster system are the most important parameters. In short, a lot of power from a little mass is desired. The chief problem with electric propulsion to date is that power in sufficient quantity has not been available.

$$M_p = P/P_{spp}$$

Where:	M_p	=	Power System Mass <kg>
	P	=	Power Level <W>
	P_{spp}	=	Power System Specific Power <W/kg>

- Since some of this power may be lost or used by other vehicle subsystems, find the power to the thruster system, P_t. Thruster system power is a percentage (N) of the total power.

$$P_t = N \% * P$$

- Calculate the mass of the thruster and power conditioning system:

$$M_T \quad = \quad P_t/P_{spt}$$

Where:

M_T	=	Thruster System Mass <kg>
P_t	=	Thruster System Power <W>
P_{spt}	=	Thruster System Specific Power <W/kg>

- Thrust power may be calculated as follows:

$$\text{Thrust Power} = P_t * E_t * E_{pc}$$

Where:

E_t = Thruster Efficiency

E_{pc} = Power Conditioner Efficiency

- The thrust is:

$$\text{Thrust (T)} = 2 * \text{Thrust Power} / (\text{Isp} * g)$$

Where: g = Acceleration of Gravity (9.81 m/s^2)

- Total system dry mass, less payload, is the sum of the Power system mass and the Thruster system mass.

$$M_{Propulsion} = M_T + M_p$$

To begin with the thruster system:

- Select the desired type of thruster system.
- Select the Isp for the system.
- Determine Thrust.
- Calculate Thrust Power.

$$\text{Thrust Power} = \text{Thrust} * \text{Isp} * g/2$$

- Calculate required power to the thruster system:

$$Pt = \text{Thrust Power}/(E_t * E_{pc})$$

- Calculate the mass of the thruster and power conditioning system:

$$M_T = Pt/P_{spt}$$

- Since some of this power may be used by other vehicle subsystems, find the total power system power level:

$$P = P_t/N\%$$

- Select the power system type, power level, and specific power.

- Calculate the mass of the power subsystem:

$$MP = P/P_{spp}$$

- Total system mass is the sum of the Power system mass and the Thruster system mass:

$$M_{Propulsion} = M_T + M_P$$

3.1 MPD Thrusters

Specific Power of MPD thrusters is estimated to be 1.19 kw/kg[34], 30 kw/kg[33].

Isp ranges and propellant selections:

CURRENT SYSTEMS (Demonstrated):	
Argon	1,000 sec - 3,000 sec
Nitrogen	2,000 sec - 4,000 sec
Hydrogen	2,000 sec - 4,000 sec
Ammonia	2,000 sec - 4,000 sec
PROJECTED SYSTEMS:	
Various Propellants	1,000 sec - 20,000 sec
	(20,000 sec upper limit is unsubstantiated)

Efficiency:

Power Conditioning	98 %
Thruster System:	
ARGON	30 % demonstrated
	50 % - 65 % theoretical limit
NITROGEN	40 % demonstrated
HYDROGEN AND AMMONIA	30 % - 50 % demonstrated
PROJECTED SYSTEMS	50 % - 60 % maximum

Note that the wasted thruster power must be radiated away. More detailed designs require thruster radiator sizing.

3.2 Ion Thrusters

Specific Power:

Mercury (projected)	1.9 kw/kg
Xenon (projected)	7.5 kw/kg
Mercury (current)	.125 kw/kg

Isp ranges and propellant selections:

Mercury	2,000 - 5,000 sec
Xenon	2,000 - 5,000 sec

Efficiency 50 % - 70 %

References: 33, & 34

3.3 Solar Thermal Propulsion

The Air Force has proposed the design of a solar thermal rocket with an Isp of 870 s. This rocket has solar reflectors which can provide 1.6 MWt of power with an ideal thrust of 84 lbf[16]. A mass statement is unavailable at this time.

4.0 General Subsystems

These subsystems are applicable to any spacecraft whether it is conventional chemical, nuclear electric, or solar thermal in nature.

4.1 Structures

The mass of the structure is a larger percentage of the vehicle mass for landers than for orbiters. Orbital vehicles such as the Centaur, the Apollo Command and Service Modules (CM&SM), the LEM Ascent Module, and the GD AOTV require that 8% of the total wet mass be structure. This 8% does not include tank mass.

Lander vehicles (STS, LEM Descent Module, CM) require about 45% of their landing mass to be structure. They also require landing gear. Typically, an additional 2% of the landing mass is allotted for lunar landing gear, while 4% is allotted for Earth landing gear. Figure 4.1-1 shows the percentage of the burnout mass that is landing gear for some of the vehicles in Appendices A and B.

4.2 Attitude Control

With the exception of the STS, the vehicles in Appendix A all have an attitude control system which is approximately 3% of the total spacecraft wet mass. Half of the mass of the ACS is propellant, the other half is systems such as tanks and thrusters. This is shown graphically in Figure 4.2-1 and 4.2-2.

Figure 4.1-1

Landing Gear and Docking Systems

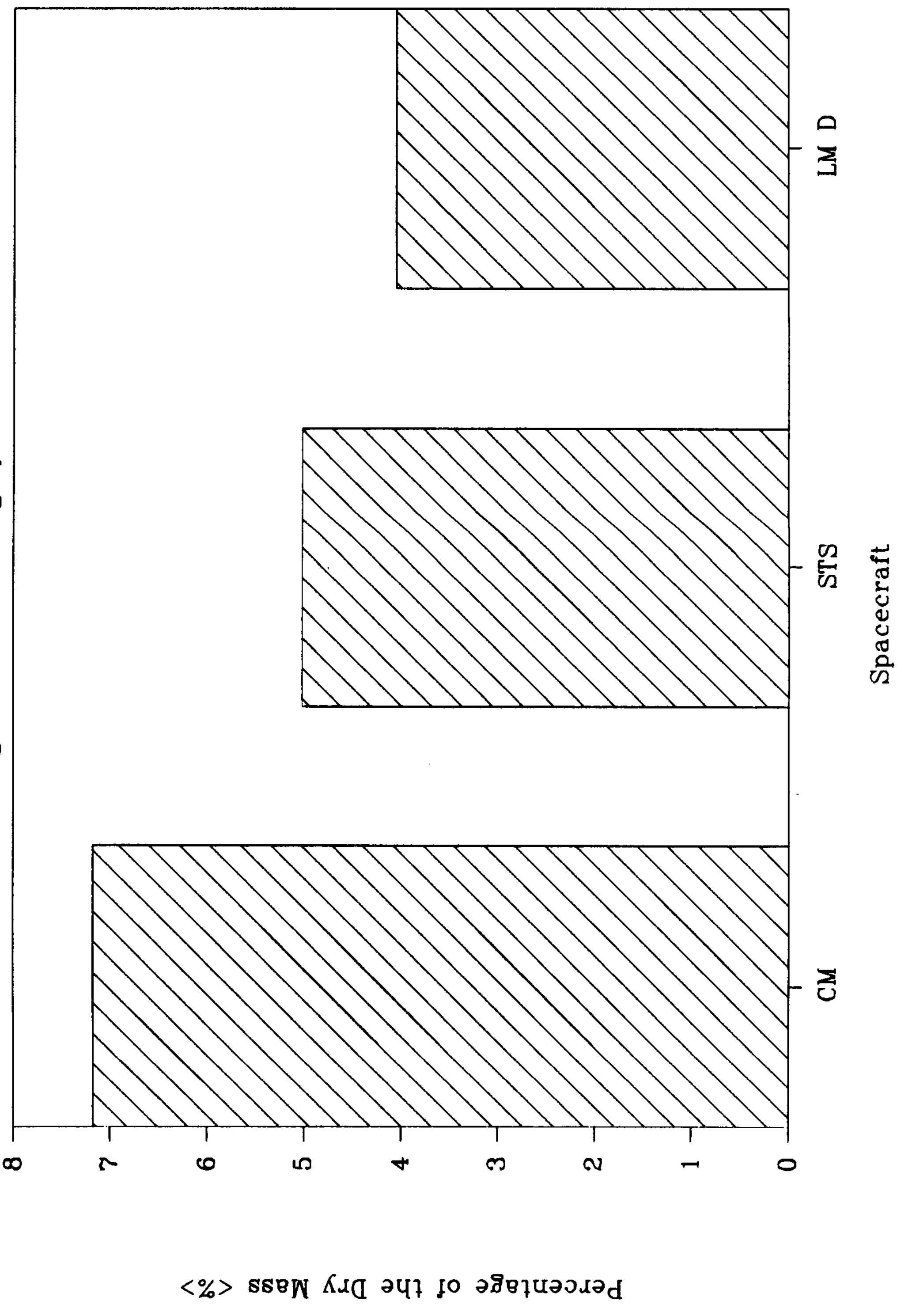


Figure 4.2-1

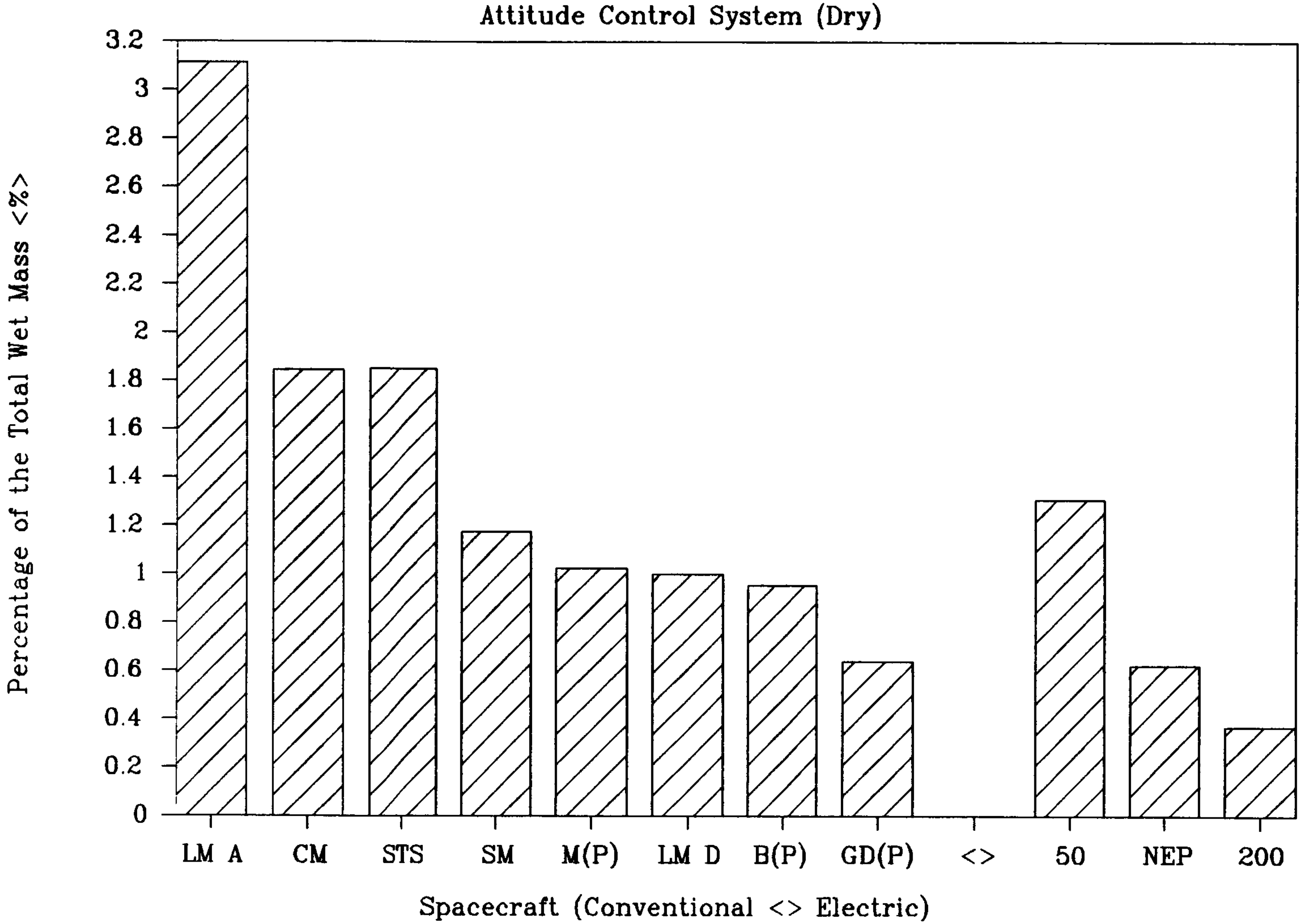
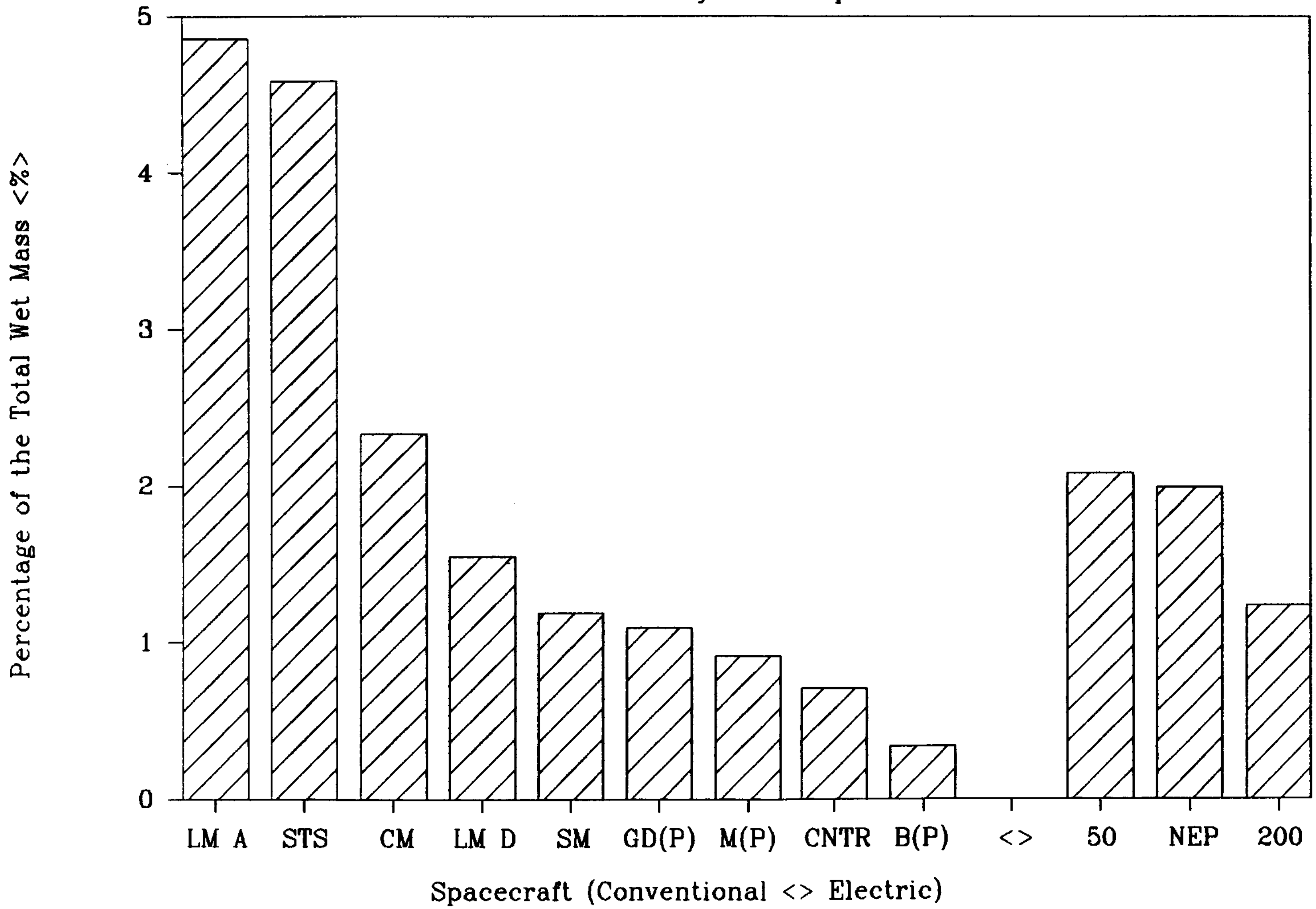

Attitude Control System (Dry)
Percentage of the Total Wet Mass <%>
3.2
3
2.8
2.6
2.4
2.2
2
1.8
1.6
1.4
1.2
1
0.8
0.6
0.4
0.2
0
LM A
CM
STS
SM
M(P)
LM D
B(P)
GD(P)
<>
50
NEP
200
Spacecraft (Conventional <> Electric)

Figure 4.2−2

Attitude Control System Propellant



4.3 Crew Module

The crew module is that section of the spacecraft which holds the astronauts. It may contain numerous other subsystems if they are integral to its design. Spacecraft sections which can be detached or separated without endangering the crew are not part of the crew module.

The mass of an OTV style crew module has been predicted to be 5.5 m. tons for a four man capability[26]. Lunar lander vehicles are 3.25 m. tons for four crewmen. Table 4.3-1 gives some data on known vehicles[26].

Table 4.3-1 Manned Spacecraft.

Spacecraft	Mass (lbm)	Crew	Longest Stay (days)	Habitable Volume (ft^3)
Mercury	2,400 (at recovery)	1	< 1	55
Gemini	5,900 (with Heatshield)	2	14 (2 crew)	80
Apollo LEM Ascent	4,000 (w/o Engine)	2	8 (2 crew)	159
Apollo Command Module	6,400 (w/o Heatshield)	3	14 (1-3)	208
Soyuz	13,500 (total)	3	18 (2 crew)	363
Salyut	41,675	3	237 (2 crew)	3,500
Mir	44,000	6	326 (2 crew)	4,600
Space Shuttle Orbiter	190,000	7	10 (6 crew)	2,625
Skylab	203,000	3	84 (3 crew)	12,800
Mir Complex	235,000	6	-	18,000
Space Station (2 Modules)	278,000	8	-	8,000

4.4 ECLSS and Consumables, Open and Partially Closed and Crew Provisions

Tables 4.4-1 and 4.4-2 describe the environmental control and life support requirements of spacecraft.

Table 4.4-1: ECLSS Average Design Loads*

Metabolic O_2	0.83 kg/man day
Leakage Air	2.27 kg/day total
EVA O_2	0.55 kg/8 hr EVA per man
EVA CO_2	0.67 kg/8 hr EVA per man
Metabolic CO_2	1.00 kg/man day
Drink H_2O	1.86 kg/man day
Food preparation H_2O	0.72 kg/man day
Metabolic H_2O production	0.35 kg/man day
Clothing wash H_2O	12.74 kg/man day
Handwash H_2O	1.81 kg/man day
Shower H_2O	3.63 kg/man day
EVA H_2O	4.39 kg/8kg hr EVA per man
Perspiration and respiration H_2O	1.82 kg/man day
Urinal flush H_2O	.049 kg/man day
Urine H_2O	1.50 kg/man day
Food solids	0.73 kg/man day
Food H_2O	0.45 kg/man day
Food packaging	0.45 kg/man day
Urine solids	0.06 kg/man day
Fecal solids	0.03 kg/man day
Sweat solids	0.02 kg/man day
EVA Wastewater	0.91 kg/k8 hr EVA per man
Charcoal required	0.06 kg/man day
Metabolic sensible heat	2.05 kW-hr/man day
Hygiene Latent H2O	0.44 kg/man day
Food preparation latent H_2O	0.03 kg/man day
Wash H_2O solids	0.44 percent
Shower/hand wash H_2O solids	0.12 percent
Airlock gas loss	0.60 kg/use
Trash	0.82 kg/man day
Trash volume	0.0028 m^3/man day

*Taken from Reference 27

Table 4.4-2: Open and Partially Closed Loop ECLSS Consumables Usage

Oxygen	
Metabolic O_2	-0.83 kg/man/day
Leakage O_2	-0.50 kg/day total leakage
Nitrogen	
Leakage N_2	-1.77 kg/day total leakage
Water	
Potable drinking water	-1.86 kg/man/day
Food prep water	-0.72 kg/man/day
Hygiene water	-5.44 kg/man/day
Clothing wash water	- 12.47 kg/man/day
Urinal flush water	-0.49 kg/man/day
Housekeeping water	- .45 kg/man/day
Subtotal	- 21.43 kg/man/day
Food	
Food solids	-0.73 kg/man/day
Food H_2O	-0.45 kg/man/day
Food Packaging	-0.45 kg/man/day
Subtotal	-1.63 kg/man/day
Total Open Loop - Oxygen, Nitrogen, Water and food	= 23.89 kg/man/day +2.27 kg/day Leakage
for a crew of 6 for 100 days (open loop)	Total = 14.5 Metric Tons
for a crew of 6 for 100 days (water loop 90% closed)	Total = 4.1 Metric Tons

Figure 4.4-1 depicts the mass of the ECLSS for the vehicles of Appendix A.

There are many other items which must be included in order to support the crew. Spacesuits, tools, and equipment for External Vehicular Activity (EVA) are often referred to as Crew Provisions. Figure 4.4-2 gives some idea of the mass comparison for these items on vehicles of the past and present.

Figure 4.4—1
Environmental Control & Life Support

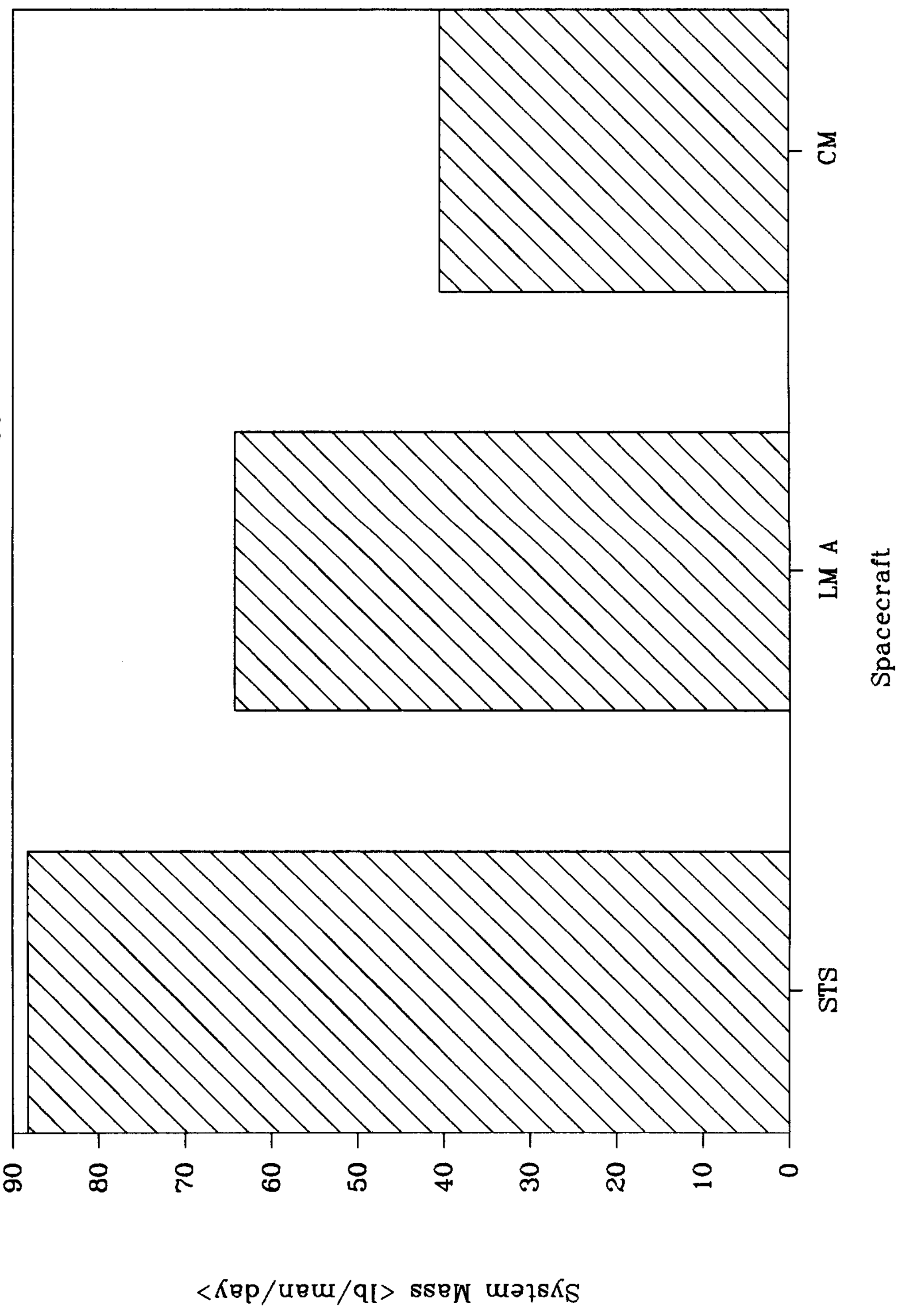
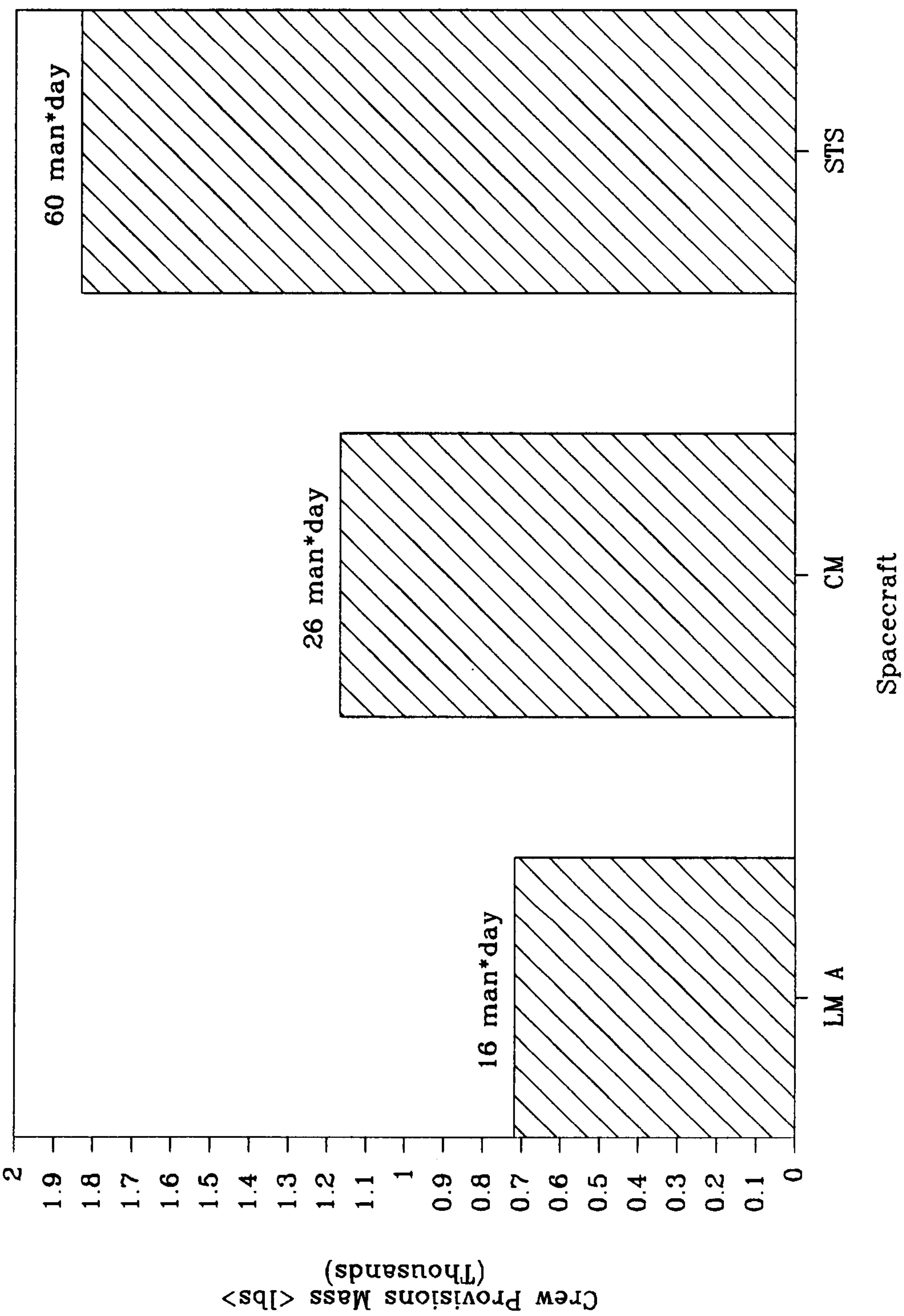


Figure 4.4—2
Crew Provisions



4.5 Docking Fixtures

The mass required for docking fixtures varies with the requirements. A few numbers are noted here:

Item	Reference	Mass, lbm
Universal Docking Adapter	28	250
Shuttle docking adapter (L shaped tunnel from shuttle airlock to Space Station)	29	2,750
Space Station Node mechanism (docking adapter plus hatch)	30	470
Apollo-Soyuz Docking Module/Airlock (total weight including experiments, stowage and fluids)	6	7,390

4.6 Contingency Factors

Aerospace vehicles always grow in size from conceptual design to flight hardware. This is because it is rare for designers to think of everything in conceptual design and also strive for maximum performance. It is therefore customary to add a contingency factor to the dry mass. Numbers from 10 to 30% are commonly used depending on the designer's confidence in the original estimate. Vehicles with existing analogs can be more accurately estimated.

4.7 Avionics

The avionics subsystem provides the spacecraft with the data required for guidance, communications, and flight operations. Sensors, electronics, and computers generally fit into the category of avionics.

The mass of the avionics subsystem is typically 3% to 4% of the total spacecraft weight. However, as Figure 4.7-1 demonstrates, this subsystem may vary between 1 and 10%.

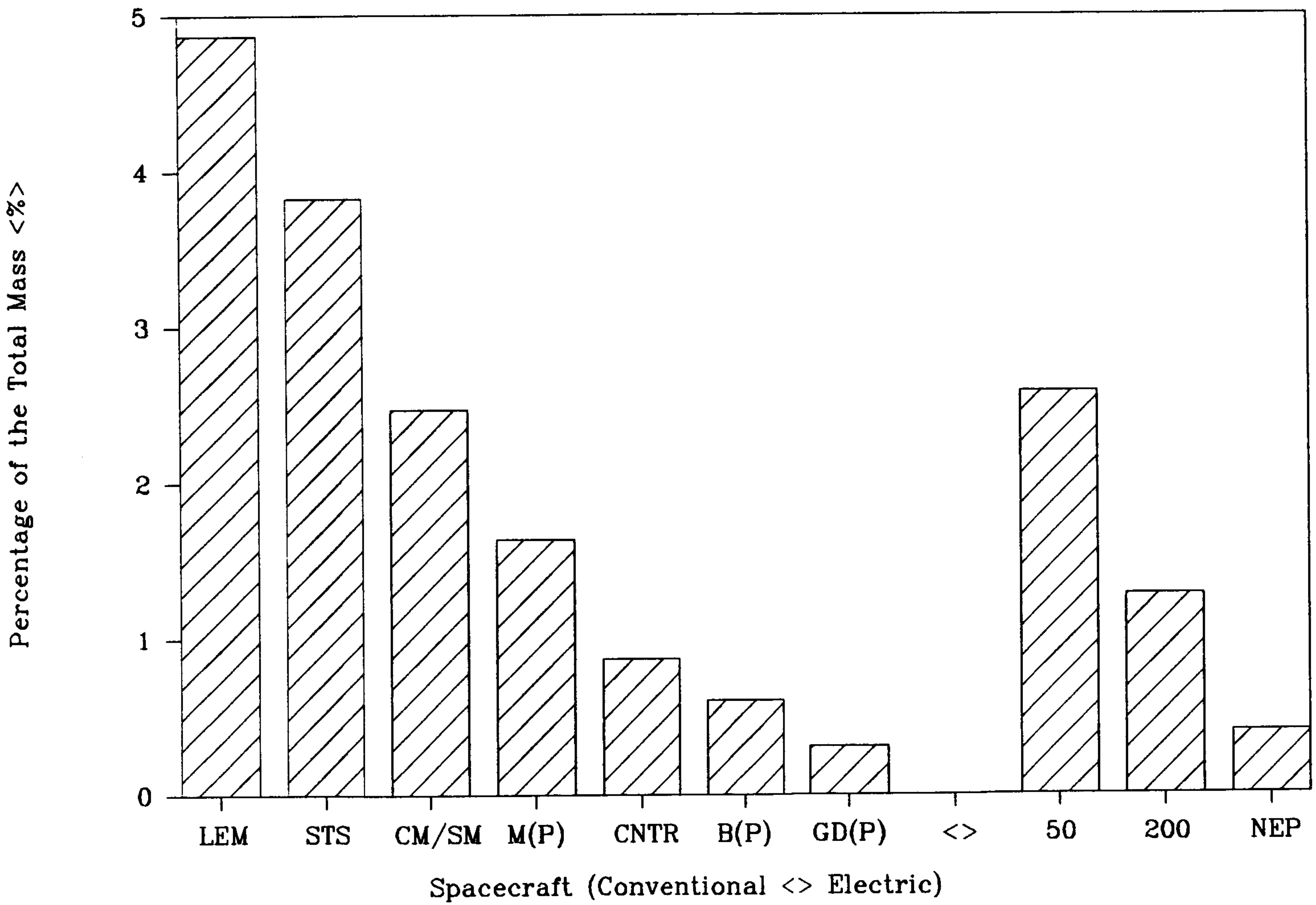
4.8 Power and Electrical

The power and electrical subsystem makes up approximately 5% of the total wet mass of unmanned vehicles like the Centaur and the GD AOTV. For manned vehicles, this subsystem is about 10% of the total mass of the vehicle. Figure 4.8-1 shows the mass fraction of the Power and Electrical subsystem for many of the vehicles in Appendices A and B.

There are several types of power sources which can be used for long duration spaceflight. Solar power systems include solar photovoltaic and solar dynamic. Nuclear power sources can be thermoelectric, thermionic, or heat engines (dynamic).

Figure 4.7–1

Avionics



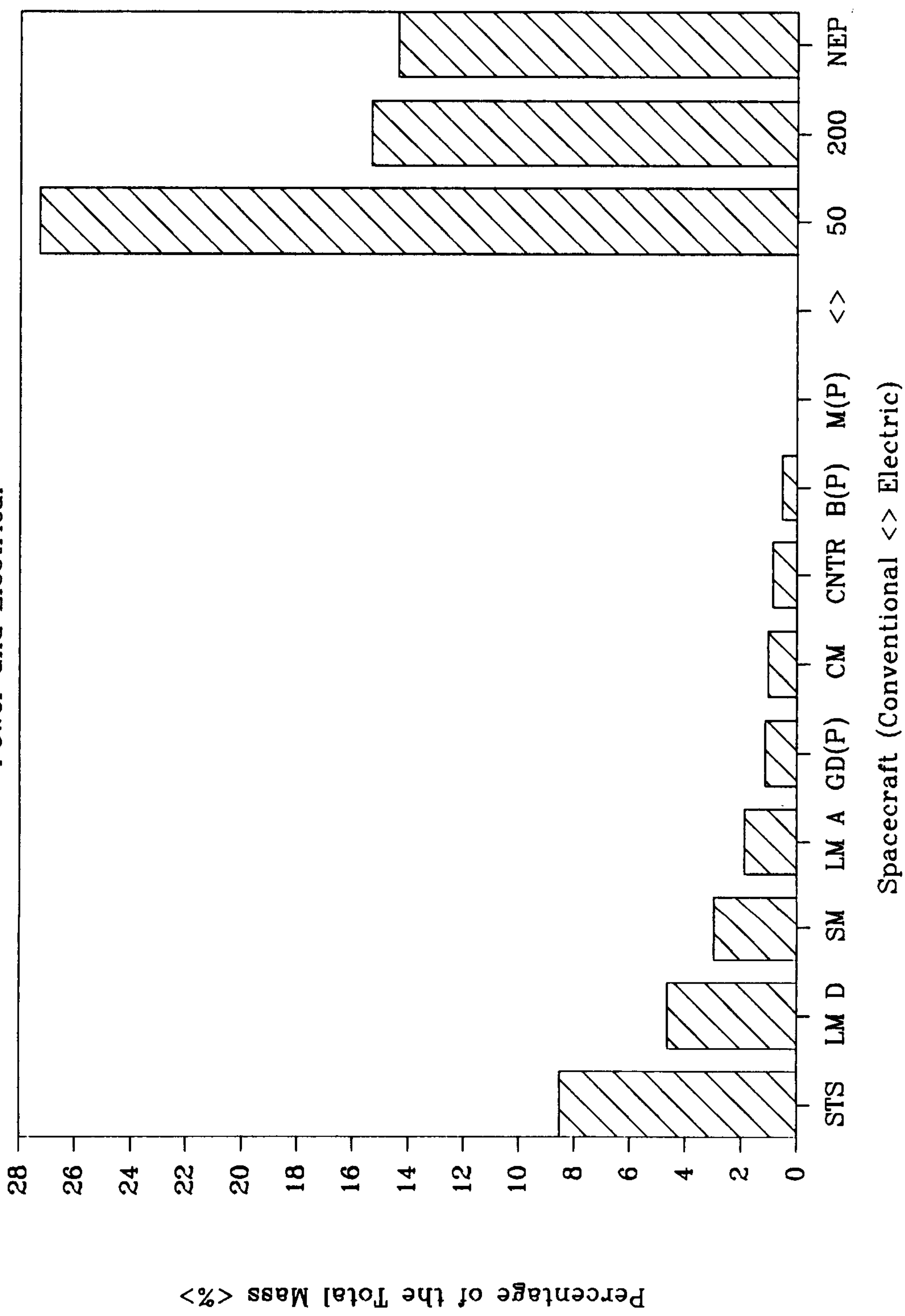


Figure 4.8-1
Power and Electrical

4.8.1 Solar Photovoltaic

Three types of solar arrays are considered here--Silicon (Si), Gallium Aluminum Arsenide (GaAlAs), and Indium Phosphide (InP). The silicon solar arrays have a power conversion efficiency of 9%. This which degrades by 25% after 10 years exposure to space radiation. The silicon arrays have a specific power of 8.8 <W/kg>. The Gallium Aluminum Arsenide arrays have a power conversion efficiency of 17%, which degrades by 12 to 15% after 10 years, and have a specific power of 35 <W/kg>. Indium Phosphide arrays are new. They are slightly more efficient (20 to 25%) than the Gallium arrays, which gives them slightly higher specific power. But more importantly, their power output does not degrade over time due to radiation damage.

It should be noted that the power degradation discussed up to this point is known as radiation degradation. It is power lost due to long term exposure to solar radiation. There are other forms of degradation which can affect a solar array. Micrometeoroids and local space contamination can damage and reduce the efficiency of solar arrays. In some cases, these are more destructive than radiation, and should be seriously considered when designing and sizing solar arrays.

Nominally the specific power of current solar arrays is 22 <W/kg>. This varies based on the mounting and orientation of the solar array. Figure 4.8-2 shows the power to mass relationship for solar arrays for different mountings and orientations.

The 32 kWs of output power provided by the Skylab solar arrays is the largest space-based solar array power source to date. Larger systems, such as the Solar Electric Propulsion (SEP) Array (See Figure 4.8-3), have been proposed.

Figure 4.8-2: Solar Array Mass and Power

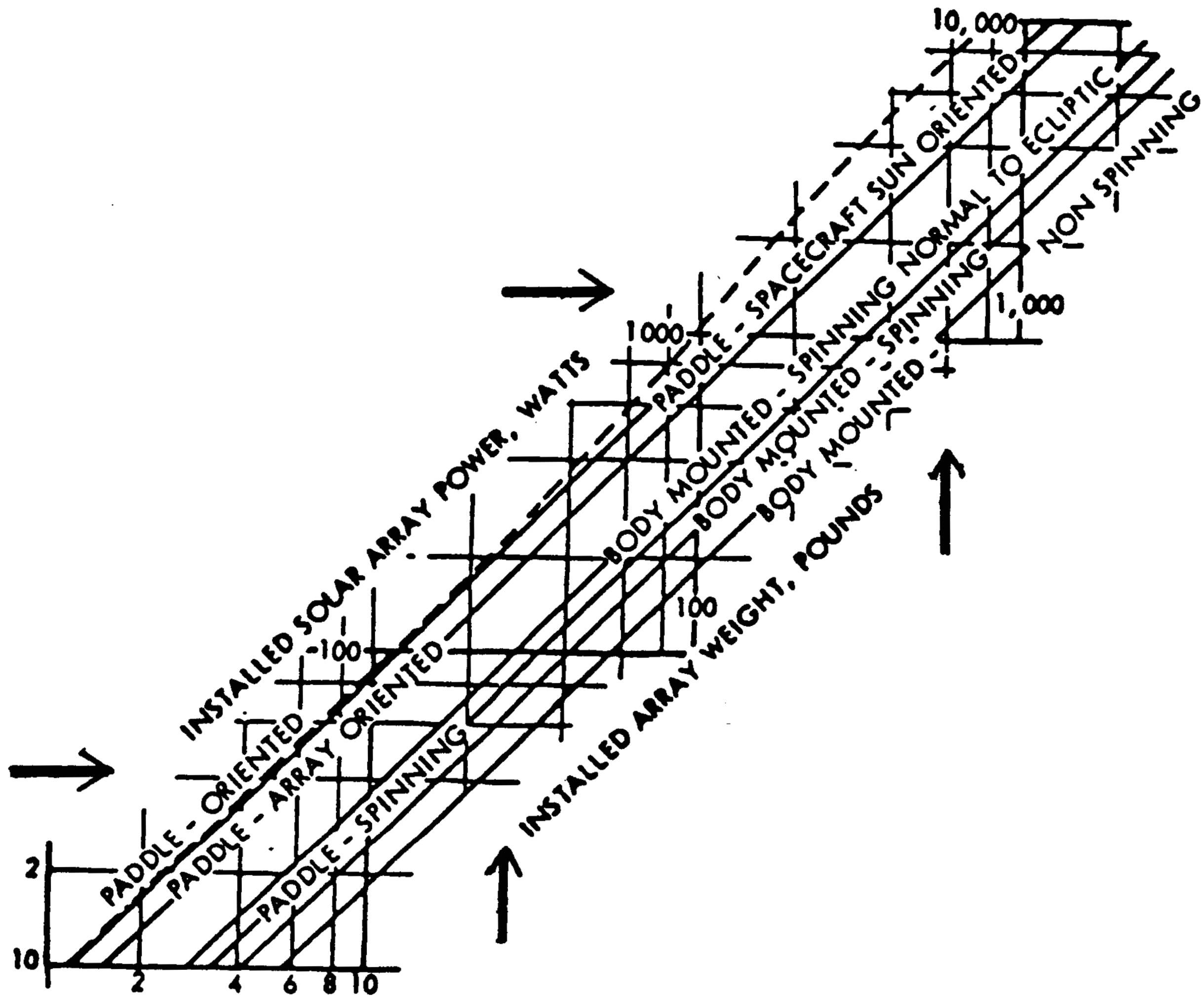
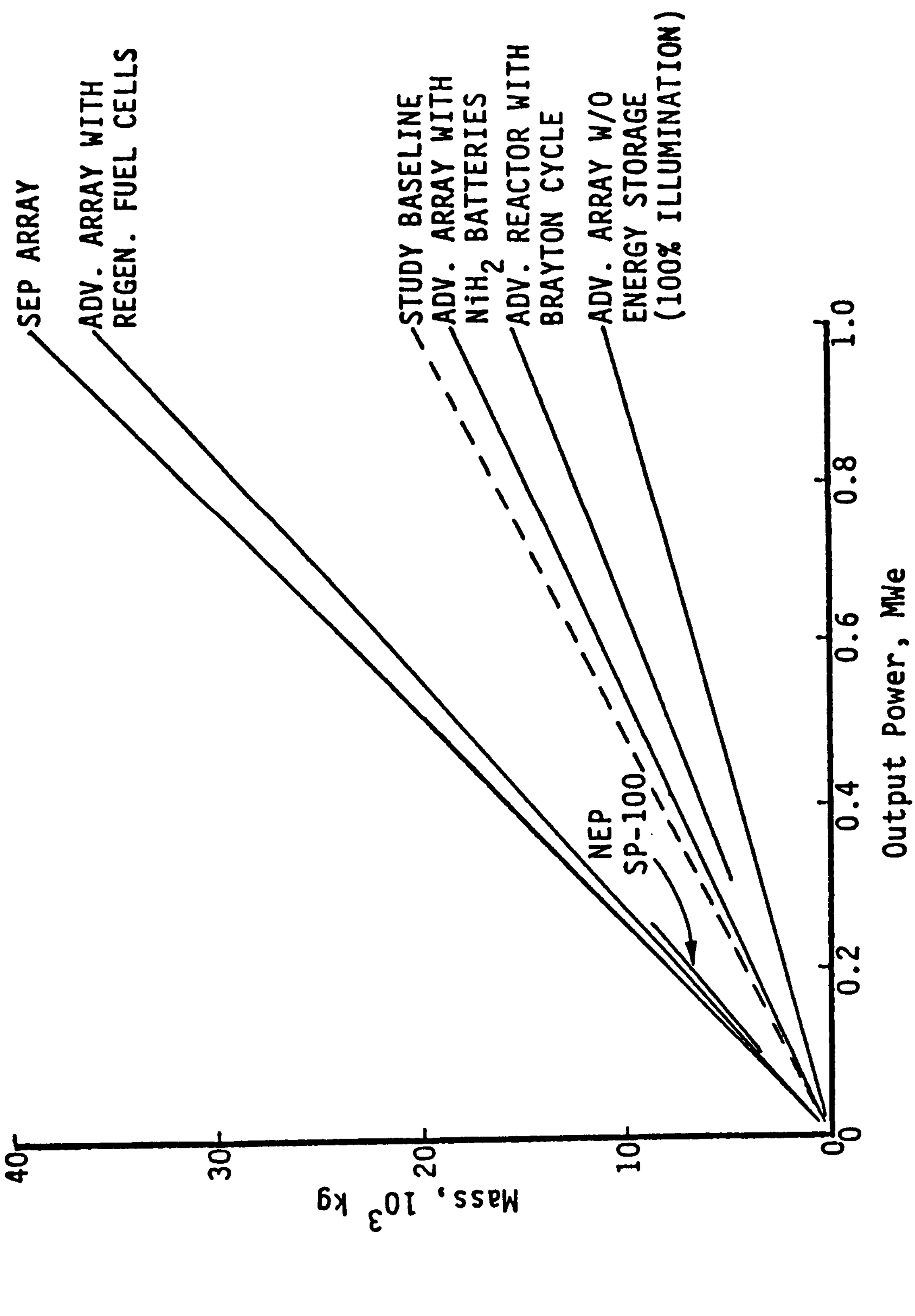


Figure 4.8-3: Power System Masses*



*Obtained from Reference 17

4.8.2 Nuclear Thermoelectric/Thermionic

The following equations are valid for nuclear thermoelectric power systems from 0.3 megawatts (MW) through 10 MW. Specific Power (Psp) is given in kw/kg or MW/MTon. Power is expressed in MW. These reactors have efficiencies ranging from 5 percent at 0.3 MW to 11 percent at 10 MW. Radiators are sized at 900°K. Shielding is a shadow shield.

Instrument: $Psp = 0.02 + 0.097*P - 0.0129*P^2 + 0.00081*P^3$

Man-rated: $Psp= 0.0049 + 0.048*P - 0.0043*P^2 + 0.00024*P^3$

Where: P = Power <MW>

Psp = Specific Power <MW/Mton>

Source Data Table

Power Level (MW)	Reactor (MT)	Radiator Converter (MT)	Power (MT)	Shield inst/man (MT)	Total inst/man (MT)	Psp inst/man (MW/MT)
0.3	1.6	2.0	1.7	0.9/10.9	6.2/16.2	0.048/0.019
1.0	2.4	2.9	2.3	1.9/12.7	9.5/20.3	0.105/0.049
3.0	3.9	3.6	3.7	2.6/14.5	13.8/25.7	0.217/0.117
10.0	7.4	4.3	4.8	3.2/17.4	19.7/33.9	0.508/0.295

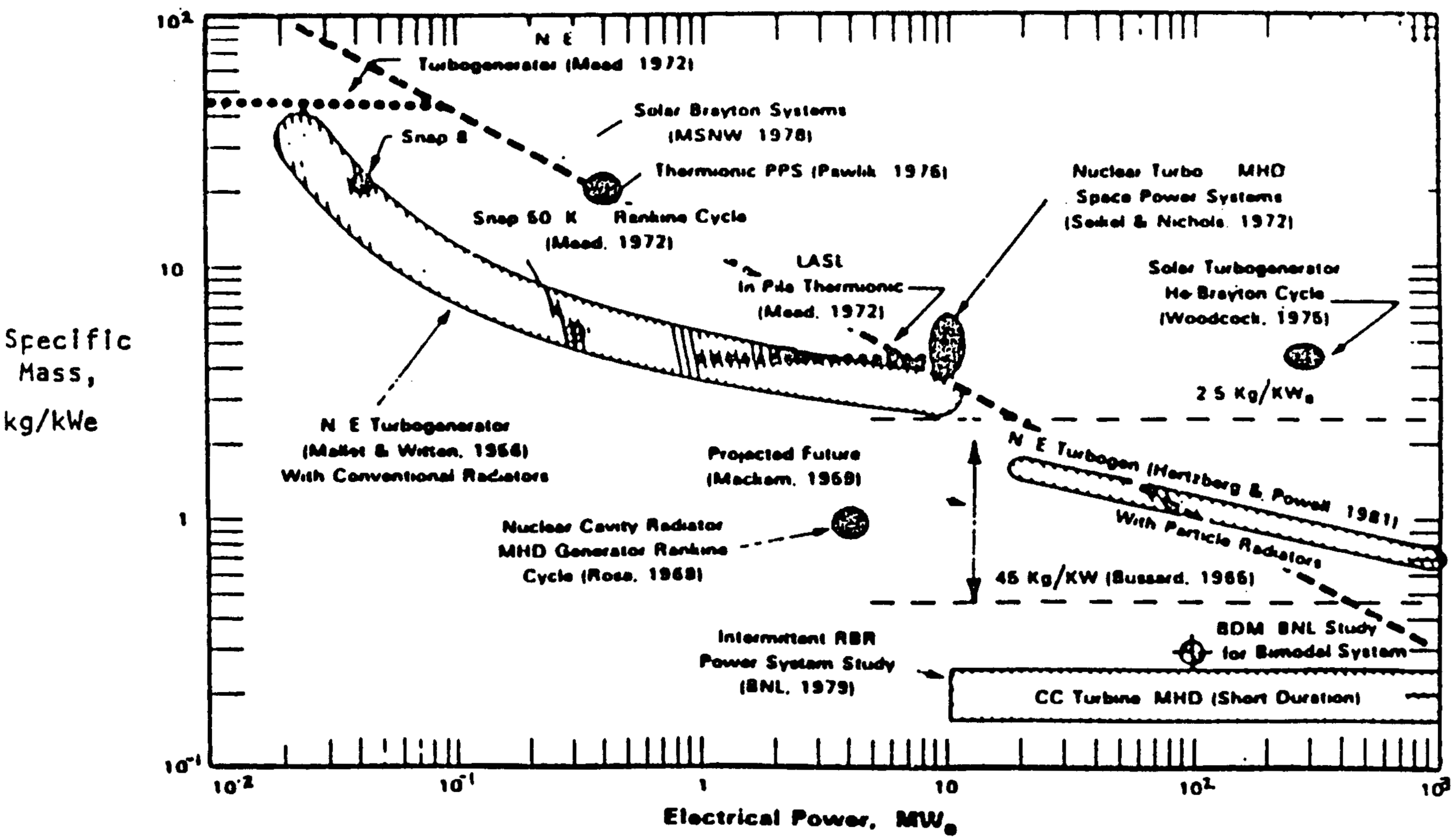
Data is from Don Carlson via Paul Keaton, both of Los Alamos National Laboratory.

From graph 4.8-3, the SP-100 nuclear electric power system has a power output of between 100 and 250 kw for systems of mass 3000 and 9000 kg respectively. This system has an average specific power of 0.030 <MW/MTon>. The equation above (Instrument Specific Power) compares well with this data at the 100 kw power level.

4.8.3 Heat Engine

Figure 4.8-4 gives the specific mass of many electric (nuclear and non-nuclear) closed cycle dynamic heat engines. The dashed line represents the best-fit linear relationship of specific mass to electrical power.

4.8-4: Space Electric Closed Cycle Power Systems*



*Graph obtain from Reference 15.

4.8.4 Nuclear-Safe Orbits

A significant issue for nuclear powered spacecraft is the decay time of the assembly and departure orbit. There is some speculation that a nuclear reactor of significant size would not be permitted to park in LEO. If the vehicles are required to park in higher orbits with long decay times, significant cost and performance penalties will result, mainly the need for an additional transport shuttle.

Previous studies have used 500 nm circular orbits, but the actual politically permissible altitude has yet to be determined.

REFERENCES

1. John W. R. Taylor, Jane's All The World's Aircraft: 1969-70, Jane's Yearbooks - McGraw Hill Book Company, New York, NY, 1971.

2. John W. R. Taylor, Jane's All The World's Aircraft: 1971-72, Jane's Yearbooks - Mcgraw Hill Book Company, New York, NY, 1973.

3. John W. R. Taylor, Jane's All The World's Aircraft: 1984-85, Jane's Publishing Inc., New York, NY, 1984.

4. Warren Brasher, "Propulsion, Maneuvering, and Attitude Control Systems", JSC Advanced Programs Office Presentation Charts, September 9, 1987.

5. Apollo Spacecraft News Reference, North American Rockwell - Space Division, Downey, Calif., 1969.

6. Kenneth Gatland, The Illustrated Encyclopedia of Space Technology, Crown Publishers, Inc., New York, NY, 1981.

7. J. R. Rowe, Liquid Rocket Engines, Aerojet Liquid Rocket Company, Sacramento California, 1975.

8. L. R. Redd, "Advanced LO_2/LH_2 Orbit Transfer Vehicle Study", Martin Marietta Corp., October, 1983.

9. Orbital Transfer Vehicle Concept Definition and System Analysis Study, Final Report, General Dynamics Space Systems Division - Advanced Space Programs, Rep. GDSS-SP-86-011, December 1986.

10. Dana Andrews, "Summary Weight Statement Aerobraked Oxygen Tanker - Intercompany Memo", Boeing, Seattle, Wash., April, 1981.

11. Skene, Wells, Levine, & Nelson, "Apollo Weight and Balance Report", NASA MSC Internal Memo, 10 August, 1962.

12. Centaur G Technical Description: A High-Performance Upper Stage For Use in the Space Transportation System, General Dynamics Convair Division Report CGTD-3, February 1983.

13. LEM Mass Property Report, Grumman, Report LED-490-19, (Contract NAS 9-1100), Jan. 14, 1963.

14. Paul Phillips, "NEP Freighter: A Point Design", Case For Mars Paper, Paper No. 11.10, July 1987.

15. J. P. Layton, Advanced Propulsion Systems, AIAA Professional Study Seminar Series: Spacecraft Propulsion Systems, RCA Astro-Electronics, May 21, 1984.

16. William B. Scott, "USAF Pursues Development of Solar-Powered Rocket", Aviation Week & Space Technology, October 19, 1987.

17. MPD Thruster Definition Study, Eagle Engineering Inc. & Martin Marietta Denver Aerospace, April, 1984.

18. TRW Space Data, TRW Space Systems Group, Redondo Beach, California, 1967.

19. Norman Brown, "Mars Transit Vehicle Thermal Protection System: Issues, Options, and Trades, Manned Mars Missions: Working Group Papers, Volume II, Los Alamos National Laboratories, 1986.

20. J. R. French & M. D. Griffin, Spacecraft Systems Design and Engineering.

21. Ken Kroll, Johnson Space Center Propulsion Section, personal communication.

22. R. E. Thomas, Orbiter Detailed Weight Statement, Rockwell, Report SD75-SH-0116-138 (Contract # NAS9-14000), February 2, 1987.

23. Summary of Rocketdyne Medium and Heavy Launch Vehicle Main Engines (LOX/Kerosene), Rockwell International - Rocketdyne Division, Document BC 86-113, 1986.

24. John W. R. Taylor, Jane's All The World's Aircraft: 1985-86, Jane's Yearbooks-McGraw Hill Book Company, New York, NY, 1971.

25. Aerospace Group Space Rocket Motor Catalog, Morton Thiokol, Ogden, Utah, 1986.

26. Impact of Lunar and Planetary Missions on the Space Station, Final Report, Eagle Engineering, Inc., Report No. 84-85D (Contract NAS 9 - 17176), November 1984.

27. "Manned Mars Missions, Working Group Papers", NASA MOO2, June 1986 - Manned Mars Mission Environmental Control and Life Support Subsystem Preliminary Report, Vive Hueter.

28. Personal communication from Dr. Dana Andrews, Boeing, Seattle, Jan., 1983.

29. Space Industries preliminary design.

30. Space Station Projects Requirements Document, JSC 31000, Rev. C, March 6, 1987.

31. Orbital Transfer Vehicle, Concept Definition and Systems Analysis Study, Final Report. Phase I, Vol. I (Rev. A). Executive Summary, 1986, Boeing D180-29108-1, NAS8-36107, NASA MSFC.

32. Roberts, Barney B., OTV Weight Estimates, JSC Memorandum ET4/8401-11L.

33. Manned Mars Missions, Working Group Papers, Vol. I, NASA M002, June 1986, Nuclear Electric Propulsion, Paul W. Keaton and David J. Tubb.

34. Personal Communication with Paul Keaton, September 18, 1985.

35. Lunar Lander Conceptual Design, Eagle Engineering, Inc., for NASA-JSC Advanced Projects Office, "Lunar Base Systems Study Task 2.2", March 23, 1988.

36. Propulsion and RCS Subsystem Study Guide Lunar Module LM-3, NASA Contract NAS 9-1100, by Grumman, February, 1968.

37. Mass Properties Status Report (IUS), Boeing Co., Contract F04701-78-C-0040, Code Ident. No. 81205, May 28, 1980.

38. Shuttle Systems Weight & Performance, NASA-JSC-09095-74, Houston, TX, August 24, 1983, Status Report.

39. Bate, Mueller, and White, Fundamentals of Astrodynamics, Dover Publications, Inc., New York, NY, 1971.

40. M.H. Kaplan, Modern Spacecraft Dynamics and Control, John Wiley & Sons, New York, NY, 1976.

41. Victor Szebehely, Theory of Orbits, Academic Press, Inc., New York, NY, 1967.

APPENDIX A

2.2 Mass Breakdowns for Selected Conventional Propulsion Designs

The following weight statements can be used to help determine subsystem masses. Some vehicles described here are proposals only and the numbers should be viewed with caution. Others are flight hardware.

If the fuel and oxidizer propellant tanks cannot be separated then the total mass of both tanks is recorded as the "Fuel Tank(s)" mass, and a "-" is recorded for the mass of the Oxidizer Tank(s).

All masses are in pounds.

Spacecraft:	Aerobraked Oxygen Tanker	
Date of Operation:	Proposal 1986	
Manufacturer:	Boeing	

	lbm
Propulsion (1,280)	
Fuel Tank(s)	400
Oxidizer Tank(s)	620
Pressurization System	-
Engine Assembly	260
Avionics	610
Structure	1,175
Thermal Protection and Aerobrake	2,615
Crew	0
Power and Electrical	540
Landing / Docking	0
Attitude Control System (ACS)	180
Contingency	960
Dry Mass	7,360
Propellants (93,595)	
Unusable (Main Propulsion System - MPS)	2,410
Unusable (Attitude Control System - ACS)	20
Vented Propellant	-
Usable (MPS - includes reserves)	90,845
Usable (ACS - includes reserves)	320
Total Mass	100,955

Note: 2% FPR, 10% ACS Margin

Reference: 10

Spacecraft:	Single Task Set Orbital Transfer Vehicle
Date of Operation:	Proposal 1986
Manufacturer:	General Dynamics

	lbm
Propulsion (1171)	
Fuel Tank (s)	292
Oxidizer Tank (s)	-
Pressurization	9
Engine Assembly	870
Avionics	150
Structure	2,732
Thermal/Heatshield/Aerobrake	1,466
Crew	0
Power & Electrical	555
Landing/Docking	0
Attitude Control System (ACS)	308
Contingency	0
Dry Mass	6,382
Propellants (41,956)	
Unusable (Main Propulsion System - MPS)	408
Unusable (Attitude Control Propulsion System - ACS)	121
Vented Propellant	-
Usable (MPS - incl. FPR + 176)	41,019
Usable (ACPS)	408
Total Mass	48,338

Reference: 9

Spacecraft:	Advanced LO_2/LH_2 Orbit Transfer Vehicle (proposal 1983)
Date of Operation:	Proposal 1983
Manufacturer:	Martin Marietta Corporation, Pratt & Whitney

	lbm
Propulsion (1450)	
Fuel Tank (s)	352
Oxidizer Tank (s)	308
Pressurization	746
Engine Assembly	44
Avionics	736
Structure	1,025
Thermal/Heatshield/Aerobrake	508
Crew	0
Power & Electrical	-
Landing/Docking	0
Attitude Control System (ACS)	459
Contingency	418
Dry Mass	4,596
Propellants (40,343)	
Unusable (Main Propulsion System - MPS)	474
Unusable (Attitude Control System - ACS)	66
Vented Propellant	1,003
Usable (MPS - includes reserves)	38,454
Usable (ACPS - includes reserves)	346
Total Mass	44,939

Notes: 2% FPR, 10% ACS Margin

Reference: 8

Spacecraft:	Centaur G	
Date of Operation:	1962 (Atlas)	
Manufacturer:	General Dynamics	
		lbm
Propulsion (2,166)		
Fuel Tank (s)		626
Oxidizer Tank (s)		-
Pressurization		-
Engine Assembly		1,540
Avionics		314
Structure		3,370
Thermal/Heatshield/Aerobrake		0
Environmental Control		-
Crew		0
Power & Electrical		313
Landing/Docking		0
Attitude Control System (ACS)		-
Contingency		0
Dry Mass		6,163
Propellants (29,916)		
Unusable (Main Propulsion System)		557
Unusable (Attitude Control System - ACS)		-
Vented Propellant		40
Usable (MPS - includes reserves)		29065
Usable (ACPS - includes reserves)		254
Total Mass		36,079

Note: Centaur G can deploy a 10,288 lb payload into a geosynchronous orbit at 0° inclination. The Space Shuttle delivers the Centaur to a nominal parking orbit 150 nm circular with an inclination of 28.5°. The performance is based on deployment occurring within 8 hours after liftoff.

Reference: 12

Spacecraft:	Space Shuttle Orbiter
Date of Operation:	November, 1982
Manufacturer:	Rockwell Corporation

	lbm
Propulsion (3,042)	
Fuel Tank (s)	832
Oxidizer Tank	842
Pressurization	760
Engine Assembly	608
Avionics	6,505
Structure	67,427
Thermal/Heatshield/Aerobrake	27,722
Environmental Control	5,298
Crew	1,833
Power & Electrical	14,522
Landing/Docking	8,544
Attitude Control System (ACS)	3,142
Contingency	0
Dry Mass	138,035
Propellants (30,594)	
Unusable (Trapped Main Propulsion System)	800
Unusable (Attitude Control System - Trapped ACS)	378
Vented Propellant	0
Usable (MPS - includes reserves)	22,000
Usable (ACPS - includes reserves)	7,416
Total Mass	168,629

Note: The structural weight includes the Hydraulic conversion and surface controls. The SSME's (20,884 lbs) and their support equipment (10,354 lbs) are not included in this weight statement. The main propulsion system referenced in this table is the Orbital Maneuvering System (OMS).

Reference: 22

Spacecraft:	Apollo Service Module
Date of Operation:	May, 1964
Manufacturer:	North American Aviation Inc.

	lbm
Propulsion (3495)	
Fuel Tank (s)	2,443
Oxidizer Tank (s)	-
Pressurization	209
Engine Assembly	843
Avionics	181
Structure	3,133
Thermal/Heatshield/Aerobrake	176
Environmental Control	601
Crew	0
Power & Electrical	1,680
Landing/Docking	0
Attitude Control System (ACS)	662
Contingency	0
Dry Mass	9,928
Propellants (46,572)	
Unusable (Main Propulsion System - MPS)	900
Unusable (Attitude Control System - ACS)	61
Vented Propellant	0
Usable (MPS - includes reserves)	45,000
Usable (ACS - includes reserves)	611
Total Mass	56,500

Reference: 11

Spacecraft:	Apollo Lunar Excursion Descent Module	
Date of Operation:	January, 1968	
Manufacturer:	Grumman	

	lbm
Propulsion (1,140)	
Fuel Tank (s)	239
Oxidizer Tank (s)	239
Pressurization	200
Engine Assembly	462
Avionics	1,152
Structure (incl. Ascent Propulsion System & Propellant)	7,737
Thermal/Heatshield/Aerobrake	-
Environmental Control	695
Crew (incl. Astronauts)	717
Power & Electrical	1,448
Landing/Docking	560
Attitude Control System (ACS)	311
Contingency (Parking Orbit, Tank Failure, & Checkout RCS)	48
Dry Mass	13,808
Propellants (17,445)	
Unusable (Main Propulsion System - MPS)	455
Unusable (Attitude Control System - ACS)	39
Vented Propellant	0
Usable (MPS - includes reserves)	16,505
Usable (ACS - includes reserves)	446
Total Mass	31,253

Note: This mass statement includes the Lunar Ascent Module.

Reference: 13

Spacecraft:	Apollo Lunar Excursion Ascent Module
Date of Operation:	January, 1968
Manufacturer:	Grumman

	lbm
Propulsion	515
Fuel Tank (s)	92
Oxidizer Tank (s)	92
Pressurization	105
Engine Assembly	226
Avionics	856
Structure	1,158
Thermal/Heatshield/Aerobrake	0
Environmental Control	334
Crew (incl. Astronauts)	717
Power & Electrical	774
Landing/Docking	0
Attitude Control System (ACS)	311
Contingency (Parking Orbit, Tank Failure, & Checkout RCS)	48
Dry Mass	4,713
Propellants (5,275)	
Unusable (Main Propulsion System - MPS)	128
Unusable (Attitude Control System - ACS)	39
Vented Propellant	0
Usable (MPS - includes reserves)	4,662
Usable (ACS - includes reserves)	446
Total Mass	9,988

Reference: 13

Spacecraft:	Apollo Command Module	
Date of Operation:	November, 1967	
Manufacturer:	North American Aviation Inc.	

	lbm
Propulsion	0
Fuel Tank (s)	0
Oxidizer Tank (s)	0
Pressurization	0
Engine Assembly	0
Avionics	1,434
Structure	1,655
Thermal/Heatshield/Aerobrake	2,615
Environmental Control	448
Crew	1,166
Power & Electrical	675
Landing/Docking	631
Attitude Control System (ACS)	166
Contingency	0
Dry Mass	8,790
Propellant (210)	
Unusable (Main Propulsion System - MPS)	0
Unusable (Attitude Control System - ACS)	10
Vented Propellant	0
Usable (MPS - includes reserves)	0
Usable (ACS - includes reserves)	200
Total Mass	9,000

Reference: 11

APPENDIX B

Mass Breakdowns for Selected Electric Propulsion Designs

The following weight statements can be used as reference points for calculating subsystem mass for electric propulsion vehicles.

All masses are in pounds.

Spacecraft:	Nuclear Electronic Propulsion (NEP) Freighter	
Date of Operation:	Proposal 1985	
Manufacturer:	Eagle Engineering	

	lbm
Propulsion (15,675)	
Propellant Tank(s)	1,001
Pressurization	-
Engine Assembly	14,674
Power and Electrical (50,715)	
Reactor/Power Conversion	11,025
Shielding	6,615
Radiator	33,075
Avionics	1,424
Structure	6,414
Thermal Protection and Aerobrake	0
Environmental Control	0
Crew	0
Landing/Docking	0
Attitude Control System (ACS)	2,185
Contingency	0
Dry Mass	76,413
Propellants (275,913)	
Unusable (Main Propulsion System - MPS)	-
Unusable (Attitude Control System - ACS)	-
Vented Propellant	-
Usable (MPS)	268,897
Usable (ACS)	7,016
Total Mass	352,326

Note: The Freighter carried 400,000 lbs of payload.

Reference: 14

Spacecraft: 200 KWt Electric Orbital Transfer Vehicle

Date of Operation: Proposal 1984

Manufacturer: Martin Marietta

	lbm
Propulsion (5382)	
Fuel Tanks	1,228
Oxidizer Tank	-
Pressurization	-
Engine Assembly	4,154
Power and Electrical (10,610)	
Reactor/Power Conversion	6,461
Shielding	198
Radiator	3,951
Avionics	887
Structure	3,688
Thermal/Heatshield/Aerobrake	0
Environmental Control	0
Crew	0
Landing	0
Attitude Control System (ACS)	255
Contingency	1,091
Dry Mass	21,913
Propellants (47,001)	
Unusable (Main Propulsion System - MPS)	381
Unusable (Attitude Control System - ACS)	-
Vented Propellant	202
Usable (MPS)	45,764
Usable (ACS)	856
Total Mass	69,116

Note: Payload carries is 44,000 lbs.

Reference: 17

Spacecraft:	50 KWt Electric Orbital Transfer Vehicle
Date of Operation:	Proposal 1984
Manufacturer:	Martin Marietta

	lbm
Propulsion (578)	
Fuel Tanks	301
Oxidizer Tank	-
Pressurization	-
Engine Assembly	277
Power and Electrical (3,948)	
Reactor/Power Conversion	3,948
Shielding	-
Radiator	-
Avionics	374
Structure	825
Thermal/Heatshield/Aerobrake	-
Environmental Control	-
Crew	-
Landing	-
Attitude Control System	189
Contingency	293
Dry Mass	6,207
Propellants (8,247)	
Unusable (Main Propulsion System - MPS)	64
Unusable (Attitude Control System - ACS)	-
Vented Propellant	101
Usable (MPS)	7,781
Usable (ACS)	301
Total Mass	14,454

Note: Payload carried is 11,000 lbs.

Reference: 17