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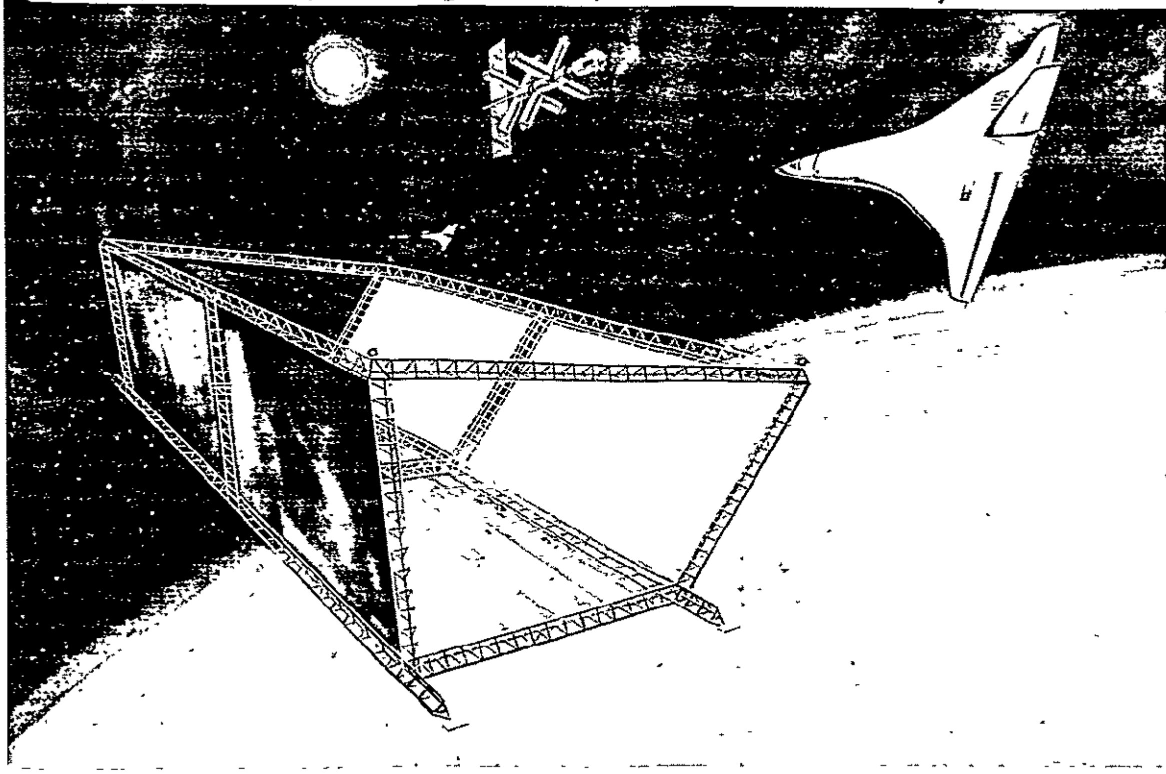

Satellite Power Systems (SPS) Concept Definition Study

FINAL REPORT (EXHIBIT C)

VOLUME IV

TRANSPORTATION ANALYSIS

Rockwell International

Satellite Systems Division
Space Systems Group

12214 Lakewood Boulevard
Downey, CA 90241

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FINAL REPORT (EXHIBIT C)
VOLUME IV

TRANSPORTATION ANALYSIS

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FOREWORD

This is Volume IV - *Transportation Analyses*, of the SPS Concept Definition Study final report as submitted by Rockwell International through the Satellite Systems Division. In addition to effort conducted in response to the NASA/MSFC Contract NAS8-32475, Exhibit C, dated March 28, 1978, company sponsored effort on a Horizontal Take-Off, Single-Stage-to-Orbit concept is included.

The SPS final report will provide the NASA with additional information on the selection of a viable SPS concept and will furnish a basis for subsequent technology advancement and verification activities. Other volumes of the final report are listed as follows:

Volume	Title
I	Executive Summary
II	Systems Engineering
III	Experimentation/Verification Element Definition
V	Special Emphasis Studies
VI	In-Depth Element Investigations
VII	Systems/Subsystems Requirements Data Book

The SPS Program Manager, G. M. Hanley, may be contacted on any of the technical or management aspects of this report. He may be reached at 213/594-3911, Seal Beach, California.

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1.0 INTRODUCTION

1.0 INTRODUCTION

The SPS transportation system, not unlike the SPS, presents a formidable challenge to our current concepts of space-oriented endeavors. Cost, more than ever, becomes the key denominator in transportation system selection. Methods of reducing transportation costs contribute significantly to the establishment of the SPS as a viable energy source option.

During previous phases of the SPS Concept Definition Study (Exhibits A and B), various transportation system elements were synthesized and evaluated on the basis of their potential to satisfy overall SPS transportation requirements and of their sensitivities, interfaces, and impact on the SPS. Study results led to the preliminary selection of preferred system concepts, as illustrated in Figure 1.0-1. However, the limited scope of the previous study effort precluded generation of sufficient substantiating data supportive of the SPS point design. The objective of this phase (Exhibit C) was to provide that data.

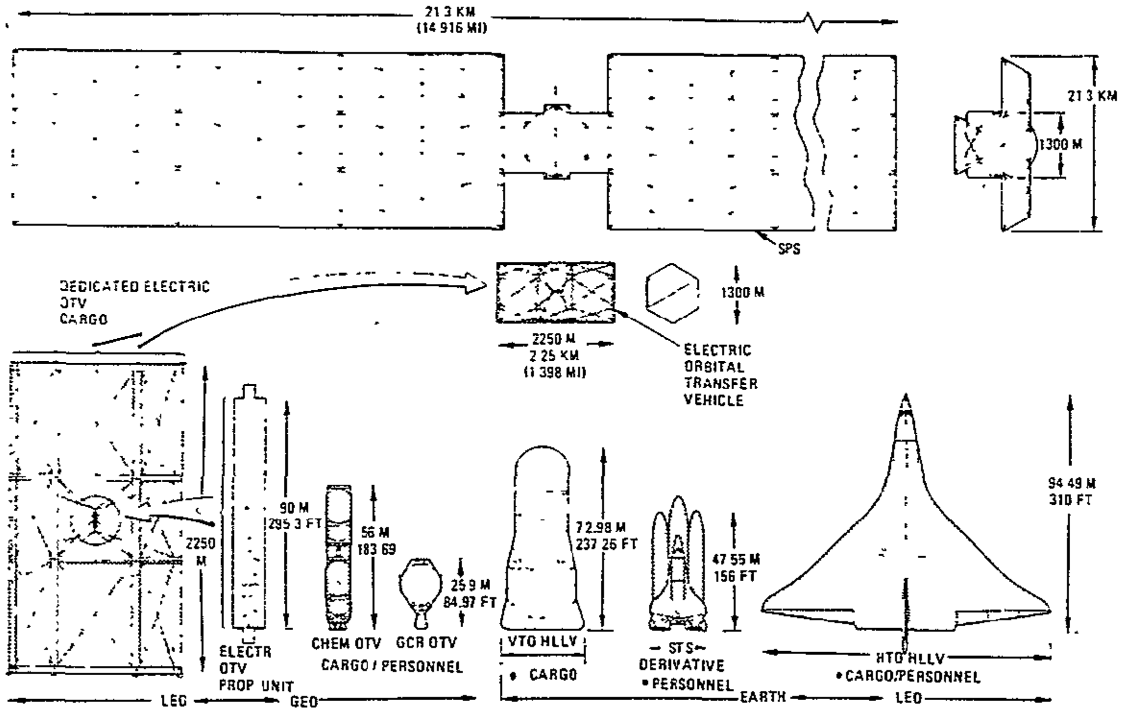

Figure 1.0-1. Transportation System Options—Vehicle Size Comparisons

Additional analyses and investigations have been conducted to further define transportation system concepts that will be needed for the developmental and operational phases of an SPS program. To accomplish these objectives, transportation systems such as Shuttle and its derivatives have been identified; new heavy-lift launch vehicle (HLLV) concepts, cargo and personnel orbital transfer vehicles (EOTV and POTV), and intra-orbit transfer vehicle (IOTV) concepts have been evaluated; and, to a limited degree, the program implications of their operations and costs were assessed. The results of these analyses have been integrated into other elements of the overall SPS concept definition studies.

Emphasis, in the area of HLLV analyses, was initially directed toward an update of the Rockwell winged, single-stage, air-breathing HLLV and in performing a comparative evaluation of that configuration with a two-stage version of that concept. Upon completion of the HTO-SSTO update, effort in this area was redirected toward the development of an alternate vertical launch/horizontal landing two-stage HLLV concept with a concomitant reduction of effort in the operations definition tasks. Configuration updates and additional data relative to the feasibility and cost of the cargo EOTV and POTV concepts were generated and requirements and concepts definition of an IOTV were pursued. Within each of these areas, supporting programmatic data (e.g., costs and schedule requirements) for the transportation system elements were developed.

SPS program and transportation system analyses continue to show that the prime element of transportation systems cost, and SPS program cost, is that of payload delivery to LEO or HLLV feasibility/cost.

2.0 TRANSPORTATION SYSTEM ELEMENTS

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As identified in previous study phases (Exhibits A and B), the SPS program will require a dedicated transportation system. In addition, because of the high launch rate requirements and environmental considerations, a dedicated launch facility for the vertical launch HLLV configurations is indicated.

The major elements of the SPS transportation system consist of the following:

- Heavy-Lift Launch Vehicle (HLLV)—SPS cargo to LEO
- Personnel Transfer Vehicle (PTV)—Personnel to LEO (Growth STS)
- Electric Orbit Transfer Vehicle (EOTV)—SPS cargo to GEO
- Personnel Orbit Transfer Vehicle (POTV)—Personnel from LEO to GEO
- Personnel Module (PM)—Personnel carrier from earth-LEO-GEO
- Intra-Orbit Transfer Vehicle (IOTV)—On-orbit transfer of cargo/personnel

Two basic SPS HLLV cargo delivery options were considered—a horizontal takeoff, single-stage-to-orbit(HTO/SSTO) HLLV (Figure 2.0-1) and a two-stage vertical takeoff horizontal landing (VTO/HL) HLLV (Figure 2.0-2). The latter

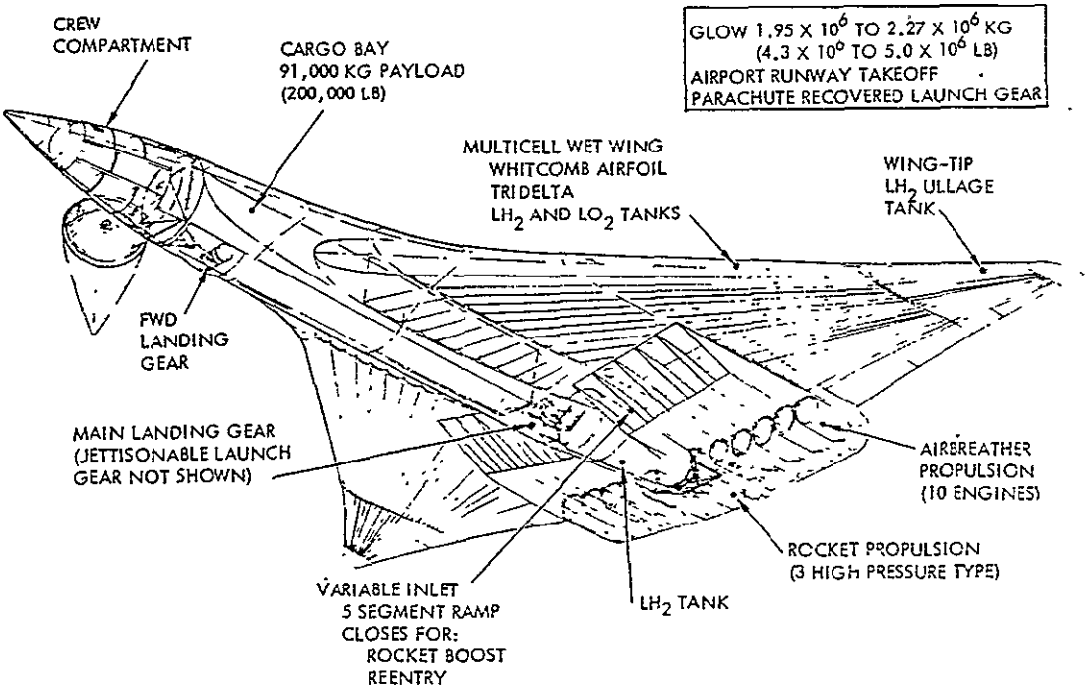


Figure 2.0-1. HTO/SSTO HLLV Concept

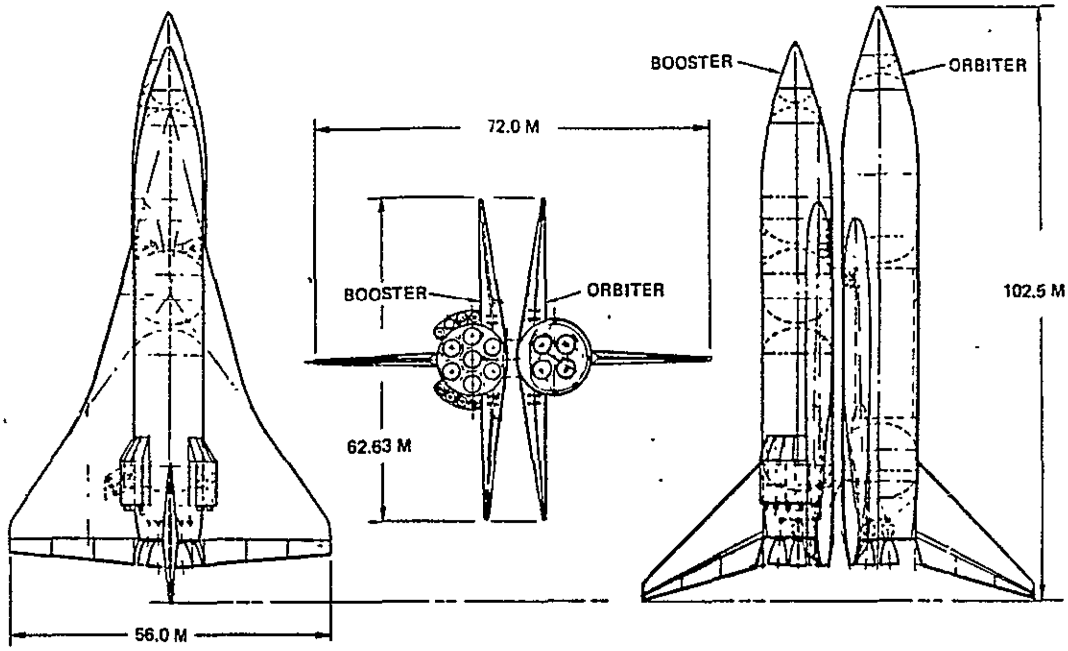


Figure 2.0-2. VTO/HL HLLV Concept

configuration option was established as the preferred or "baseline" concept for this study phase because of the uncertainty in technology readiness of the HTO/SSTO concept. A third, interim HLLV requirement was identified, to be employed during the initial SPS program development phase (Figure 2.0-3). This vehicle is designated as a Shuttle-derived or "Growth Shuttle" HLLV (STS-HLLV). This launch vehicle utilizes the same elements as the PLV (described below), except the orbiter is replaced with a payload module and an auxiliary recoverable engine module to provide a greater cargo capability.

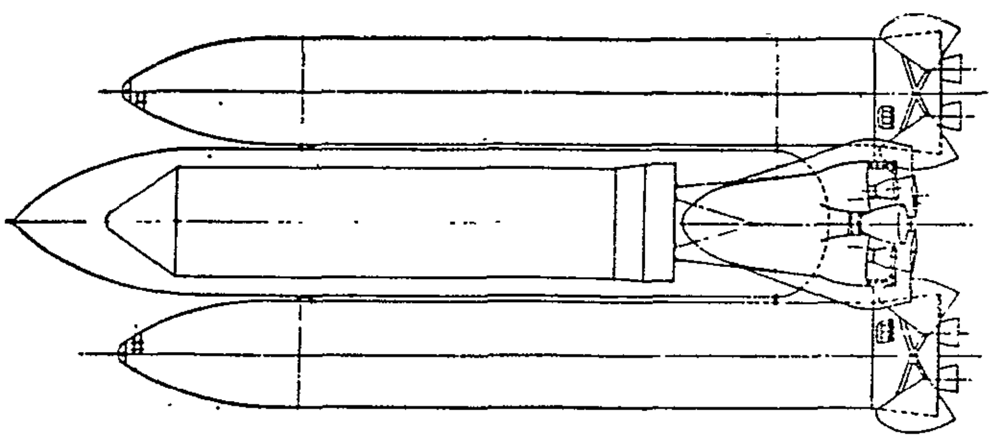


Figure 2.0-3. STS-HLLV Configuration

The Personnel Launch Vehicle (PLV) is used to transfer the SPS construction crew from earth to LEO. This launch vehicle is a modified Shuttle Transportation System (STS) configuration. The existing STS solid rocket boosters (SRB) are replaced with reusable liquid rocket boosters (LRB), thus affording a greater payload capability and lower overall operating cost, (Figure 2.0-4). The personnel module (described below) is designed to fit within the existing STS orbiter cargo bay. This vehicle will be utilized throughout the SPS program for the VTO/HL HLLV cargo delivery concept.

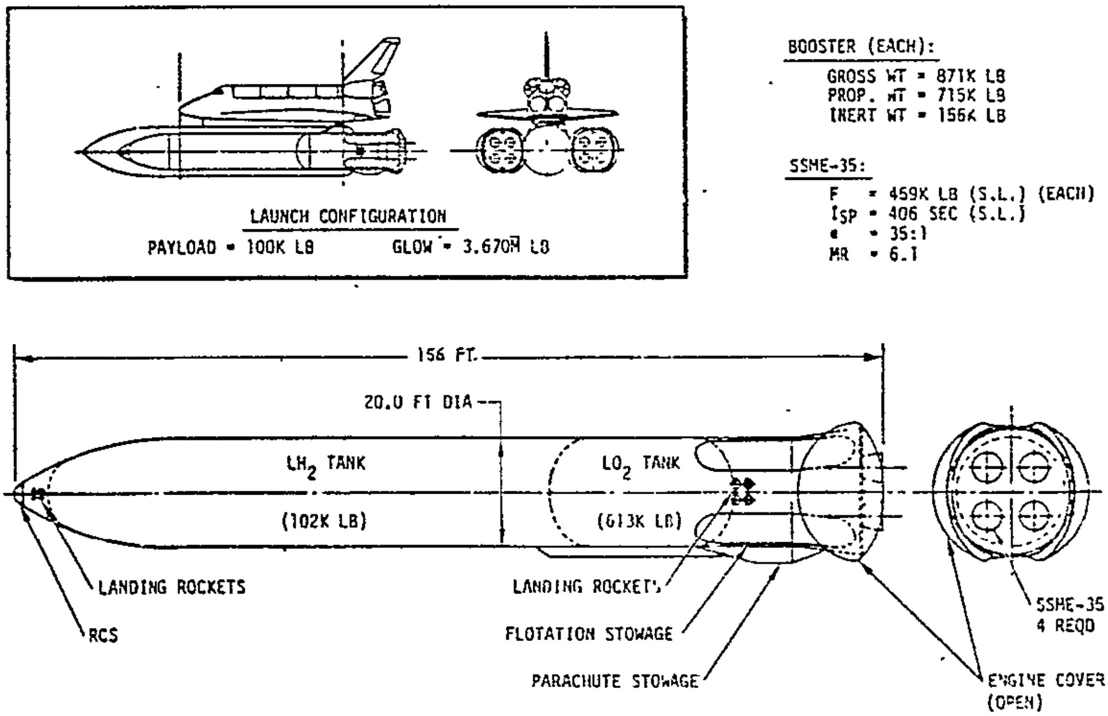


Figure 2.0-4. Growth Shuttle PLV

The Electric Orbital Transfer Vehicle (EOTV) is employed as the primary transportation element for SPS cargo from LEO to GEO. The vehicle configuration (Figure 2.0-5) defined to accomplish this mission phase utilizes the same power source and construction techniques as the SPS. The solar array consists of two "bays" of the SPS, electric argon ion engine arrays, and the requisite propellant storage and power conditioning equipment. The vehicle configuration, payload capability, and "trip time" have been established on the basis of overall SPS compatibility.

The Personnel Orbit Transfer Vehicle (POTV), as described herein, consists of that propulsive element required to transfer the Personnel Module (PM) and its crew/construction personnel from LEO to GEO. The mated configuration of POTV/PM is depicted in Figure 2.0-6. The POTV consists of a single, chemical (LOX/LH_2) rocket stage which is initially fueled in LEO and refueled in GEO for return to LEO. The POTV has been sized such that it is capable of fitting within the existing STS cargo bay and the growth STS payload delivery capability.

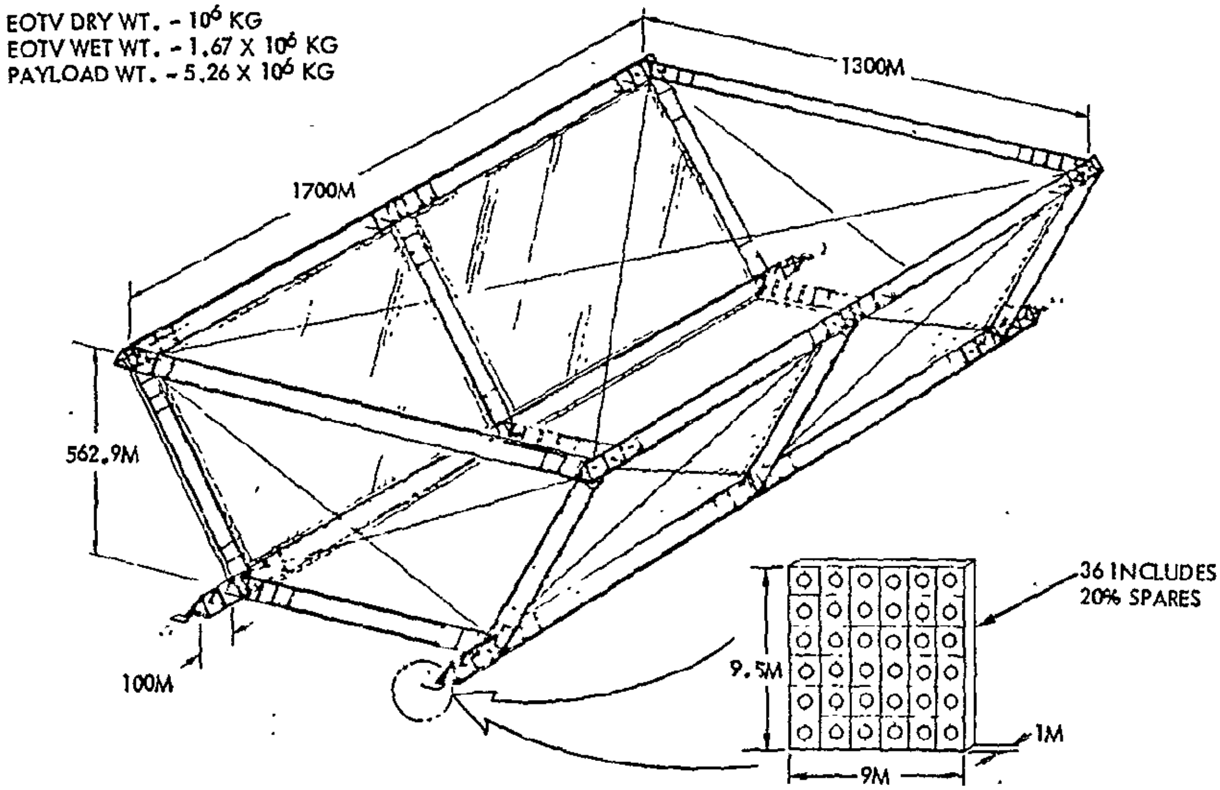
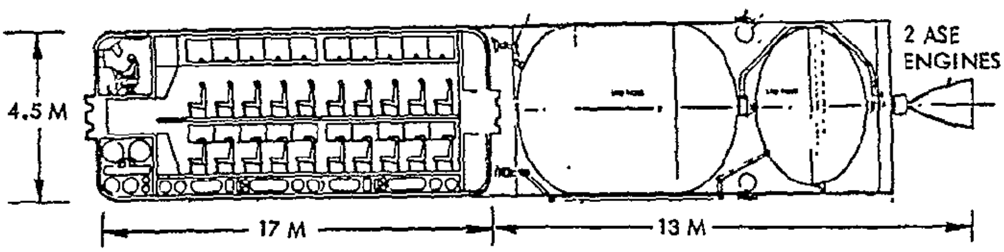


Figure 2.0-5. EOTV Configuration



• 60 MAN CREW MODULE	18,000 KG
• SINGLE STAGE OTV (GEO REFUELING)	36,000 KG
• BOTH ELEMENTS CAPABLE OF GROWTH STS LAUNCH	

Figure 2.0-6. POTV Configuration

The personnel module is designed to transport a 60-man construction crew from LEO to GEO to LEO (Figure 2.0-6). Primary considerations in sizing the PM were given to SPS construction crew demands and compatibility with the PLV concept. A considerable degree of latitude remains in the ultimate definition of a PM/POTV concept.

The intra-orbit transfer vehicle is defined in concept only. Because of the potential problems associated with docking and cargo transfer between the HLLV and EOTV in LEO and the EOTV and GEO construction base, a transfer vehicle capable of accomplishing this function is postulated. From cost and programmatic aspects of the overall SPS program, this element is depicted as a chemical rocket stage, manned or remotely operated.

In the following sections, each transportation system element will be discussed in more detail and the rationale for configuration selection presented. However, in order to maintain a continuity of data presentation, appendixes have been added to provide the substantiating technical analyses and trade study results where applicable.

3.0 TRANSPORTATION SYSTEM REQUIREMENTS

3.0 TRANSPORTATION SYSTEM REQUIREMENTS

As previously identified, the SPS will require a dedicated transportation system. In addition, because of the high launch rates and certain environmental considerations, it appears that a dedicated launch facility will also be required for SPS HLLV launches. Transportation system LEO operations are depicted in Figure 3.0-1. The SPS HLLV delivers cargo and propellants to LEO, which are transferred to a dedicated electric OTV (EOTV) by means of an intra-orbit transfer vehicle (IOTV) for subsequent transfer to GEO.

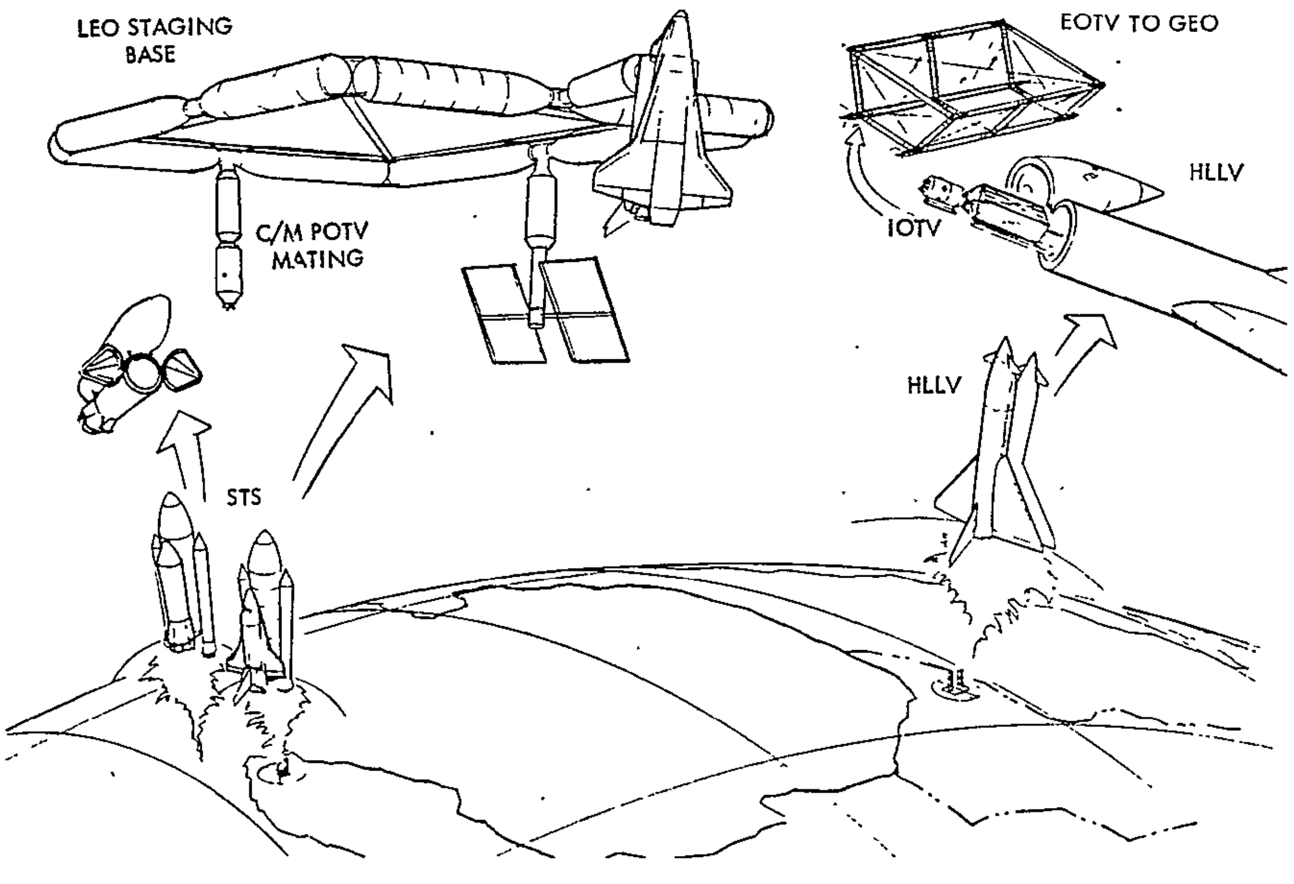


Figure 3.0-1. SPS LEO Transportation Operations

Space Shuttle transportation system derivatives (heavier payload capability) are employed for crew transfer from earth to LEO. The Shuttle-derived HLLV is employed early in the program for space base and precursor satellite construction and delivery of personnel orbit transfer vehicle (POTV) propellants. This element of the operational transportation system is phased out of the program with initiation of first satellite construction, or sooner.

Transportation system GEO operations are depicted in Figure 3.0-2. Upon arrival at GEO, the SPS construction cargo is transferred from the EOTV to the SPS construction base by IOTV. The POTV with crew module docks to the construction base to effect crew transfer and POTV refueling for return flight to LEO. Crew consumables and resupply propellants are transported to GEO by the EOTV.

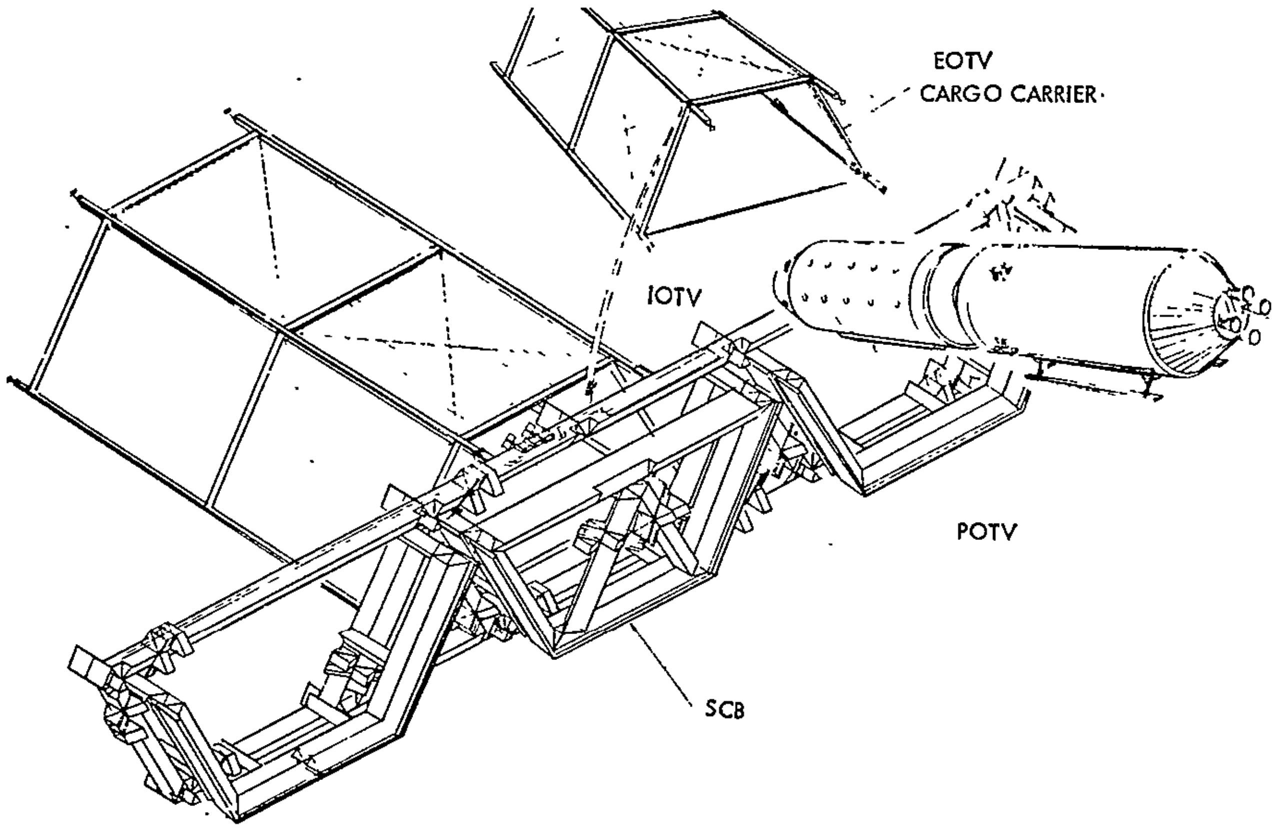


Figure 3.0-2. SPS GEO Transportation Operations

Transportation system requirements are dominated by the vast quantity of materials to be transported to LEO and GEO. Tables 3.0-1, 3.0-2, and 3.0-3 summarize the mass delivery requirements, and numbers of vehicle flights, for the baseline transportation elements. All mass figures include a 10% packaging factor. Table 3.0-1 summarizes transportation requirements for construction of the first satellite. Table 3.0-2 is a summary of requirements during the total satellite construction phase (i.e., the first 30 years). The average annual mass to LEO during this phase is in excess of 130 million kilograms with more than 750 HLLV launches per year. Table 3.0-3 presents a total program summary through retirement of the last satellite after 30 years of operation. Mass and flight requirements are separated between that required to construct the satellites and that required to operate and maintain the satellites. As indicated, the masses are nearly equal.

Table 3.0-1. TFU Transportation Requirements

	MASS x 10^6 KG		VEHICLE FLIGHTS					
			PLV	HLLV	POTV	EOTV	IOTV	
	LEO	GEO					LEO	GEO
SATELLITE CONST. MAINT. & PACKAGING	37.12	37.12	45	163.5	45	6.5	164	164
CREW CONSUMABLES & PKG.	0.98	0.94	-	4.3	-	0.2	4	4
POTV PROPELLANTS & PKG.	2.91	1.46	-	12.8	-	0.3	13	6
EOTV CONST., MAINT. & PKG.	7.20	-	15	31.7	-	-	32	-
EOTV PROPELLANTS & PKG.	4.79	-	-	21.1	-	-	21	-
IOTV PROPELLANTS & PKG.	0.13	0.06	-	0.6	-	-	1	-
							235	174
TOTAL	53.13	39.58	60	234.0	45	7.0	409	
			VEHICLE REQUIREMENTS					
TFU FLEET			2	5	4	6	4	
GROWTH SHUTTLE VEHICLES— PRECURSOR REQUIREMENTS: •LEO BASE •SPACE CONSTR. BASE •EOTV TEST VEHICLE			PERSONNEL (PLV) 72 FLIGHTS 1 VEHICLE			CARGO CARRIER/ENGINE MODULE AND LAUNCH VEH. 129 FLIGHTS 2 VEHICLES		

Table 3.0-2. SPS Program Transportation Requirements, 30-Year Construction Phase

	MASS x 10^6 KG		VEHICLE FLIGHTS					
			PLV	HLLV	POTV	EOTV	IOTV	
	LEO	GEO					LEO	GEO
SATELLITE CONST. & MAINT.	3,099.3	3,099.3	3187	13,653	3051	599.5	13,653	13,653
CREW CONSUMABLES	74.9	71.7	-	330	-	13.9	330	316
POTV PROPELLANTS	216.6	108.3	-	954	-	20.9	954	477
EOTV CONST. & MAINTENANCE	38.4	31.2	-	169	-	6.0	169	137
EOTV PROPELLANT	492.3	2.0	-	2,169	-	0.4	2,169	9
IOTV PROPELLANT	10.5	4.8	-	47	-	0.9	47	21
							17,322	14,613
TOTAL	3,932.0	3,317.3	3187	17,322	3051	642	31,935	
VEHICLE FLIGHT LIFE	-	-	100	300	100	20	200	
VEHICLE FLEET REQUIREMENTS	-	-	32	58	31	32	160	

Table 3.0-3. Total Transportation Requirements, 60-Year Program

	MASS x 10^6 KG		VEHICLE FLIGHTS					
			PLV	HLLV	POTV	EOTV	IOTV	
	LEO	GEO					LEO	GEO
SATELLITE								
CONSTRUCTION	2197.8	2197.8	1340	9682	1220	425.1	9682	9682
OPERATIONS & MAINTENANCE	1803.0	1803.0	3694	7943	3660	348.7	7943	7943
CREW CONSUMABLES								
CONSTRUCTION	31.5	28.7	-	139	-	5.6	139	126
OPERATIONS & MAINTENANCE	86.8	86.0	-	382	-	16.6	382	379
POTV PROPELLANTS								
CONSTRUCTION	82.7	41.4	-	364	-	8.0	364	182
OPERATIONS & MAINTENANCE	267.8	133.8	-	1180	-	25.9	1180	589
EOTV CONSTRUCTION								
CONSTRUCTION	28.2	24.2	-	124	-	4.7	124	107
OPERATIONS & MAINTENANCE	22.2	19.0	-	98	-	3.7	98	84
EOTV PROPELLANTS								
CONSTRUCTION	340.3	2.0	-	1499	-	0.4	1499	9
OPERATIONS & MAINTENANCE	304.0	-	-	1339	-	-	1339	-
IOTV PROPELLANTS								
CONSTRUCTION	7.2	3.3	-	32	-	0.6	32	15
OPERATIONS & MAINTENANCE	6.6	3.0	-	29	-	0.6	29	13
SUMMARY								
CONSTRUCTION	2687.7	2297.4	1340	11,840	1220	444	11,840	10,121
OPERATIONS & MAINTENANCE	2490.4	2044.8	3694	10,971	3660	396	10,971	9,008
TOTAL	5178.1	4342.2	5034	22,811	4880	840	22,811	19,129
VEHICLE FLEET								
CONSTRUCTION	-	-	14	39	12	22	110	
OPERATIONS & MAINTENANCE	-	-	37	37	37	20	100	
TOTAL	-	-	51	76	49	42	210	

4.0 HEAVY-LIFT LAUNCH VEHICLE

4.0 HEAVY-LIFT LAUNCH VEHICLE

4.0 HEAVY LIFT LAUNCH VEHICLE

Initial Heavy Lift Launch Vehicle (HLLV) studies were directed toward a horizontal takeoff single stage to orbit (HTO/SSTO) concept advanced by Rockwell during Exhibit A and B study phases. After providing an update of the HTO/SSTO, the reference launch vehicle configuration for the Exhibit C study phase was changed to a two stage vertical takeoff-horizontal landing (VTO/HL) configuration. This section of the report is directed toward the "Reference Vehicle" concept only. A summary of the HTO/SSTO effort conducted under a company sponsored program is included in Appendix A. An interim shuttle derived or "growth" shuttle HLLV configuration has been identified to satisfy early SPS precursor satellite construction requirements; and, because of it's similarity to the personnel launch vehicle (PLV), is discussed in that section of the report. In addition, the reference HLLV trade studies data are included in Appendix B along with the reference HLLV trajectory.

4.1 HLLV REQUIREMENTS/GROUND RULES

The primary driver in establishing HLLV requirements is the construction mass flow requirement (Section 3). Other factors include propellant cost/availability and environmental considerations. The basic ground rules and assumptions employed in vehicle sizing are summarized in Table 4.1-1.

Table 4.1-1. HLLV Sizing - Ground Rules/Assumptions

- TWO-STAGE VERTICAL TAKEOFF/HORIZONTAL LANDING (VTO/HL)
- FLY BACK CAPABILITY BOTH STAGES - ABES FIRST STAGE ONLY
- PARALLEL BURN WITH PROPELLANT CROSSFEED
- LOX/RP FIRST STAGE - LOX/LH_2 SECOND STAGE
- HI P_C GAS GENERATOR CYCLE ENGINE - FIRST STAGE [I_S (VAC) = 352 SEC.]
- HI P_C STAGED COMBUSTION ENGINE - SECOND STAGE [I_S (VAC) = 466 SEC.]
- STAGING VELOCITY - HEAT SINK BOOSTER COMPATIBLE
- CIRCA 1990 TECHNOLOGY BASE - BAC/MMC WEIGHT REDUCTION DATA
- ORBITAL PARAMETERS - 487 KM @ 31.6°
- PAYLOAD CAPABILITY - 227 x 10^3 KG UP/45 KG DOWN
- THRUST/WEIGHT - 1.30 LIFTOFF/3.0 MAX
- 15% WEIGHT GROWTH ALLOWANCE/0.75% ΔV MARGIN

The two stage VTO/HL HLLV concept with a payload capability of approximately 227,000 kg (500,000 lb) was adopted for a reference configuration. The payload capability was limited in order to maintain a "reasonable" vehicle size. Both stages have flyback capability to the launch site. The first stage only utilizes air breathing engines for return to launch site; the second stage is recovered in the same manner as the STS orbiter.

The launch vehicle utilizes a parallel burn mode with propellant crossfeed from the first stage tanks to the second stage engines. The first stage employs high chamber pressure gas generator cycle LOX/RP fueled engines with LH_2 cooling and the second stage employs a staged combustion engine similar to the space shuttle main engine (SSME) which is LOX/LH_2 fueled.

Although trade studies were conducted, a vehicle staging velocity compatible with a heat sink booster concept is desirable from an operations standpoint. Technology growth consistent with the 1990 time period was used to estimate weights and performance. The expected technology improvements are summarized in Table 4.1-2. Orbital parameters are consistent with SPS LEO base requirements and the thrust to weight limitations are selected to minimize engine size and for crew/passenger comfort. Growth margins of 15% in inert weight and 0.75% in propellant reserves were established. An STS scaling program was adapted for SPS HLLV sizing.

Table 4.1-2. Technology Advancement - Weight Reduction

BODY STRUCTURE	17%
WING STRUCTURE	15%
VERTICAL TAIL	18%
CANARD	12%
THERMAL PROTECTION SYSTEM	20%
AVIONICS	15%
ENVIRONMENTAL CONTROL	15%
REACTION CONTROL SYSTEM	15%
ROCKET ENGINES	
1st STAGE THRUST/WEIGHT = 120	
2nd STAGE THRUST/WEIGHT = 80	

4.2 HLLV CONFIGURATION

The reference HLLV configuration is shown in Figure 4.2-1 in the launch configuration. As illustrated, both stages have common body diameter, wing and vertical stabilizer; however, the overall length of the second stage (orbiter) is approximately 5 meters greater than the first stage (booster). The vehicle gross liftoff weight (GLOW) is 15,730,000 lb with a payload capability of 510,000 lb to the reference earth orbit. A summary weight statement is given in Table 4.2-1. The propellant weights indicated are total loaded propellant (i.e., not usable). The second stage weight (ULOW) includes the payload weight. During the booster ascent phase, the second stage LOX/LH_2 propellants are crossfed from the booster to achieve the parallel burn mode. Approximately 1.6 million pounds of propellant are crossfed from the booster to the orbiter during ascent.

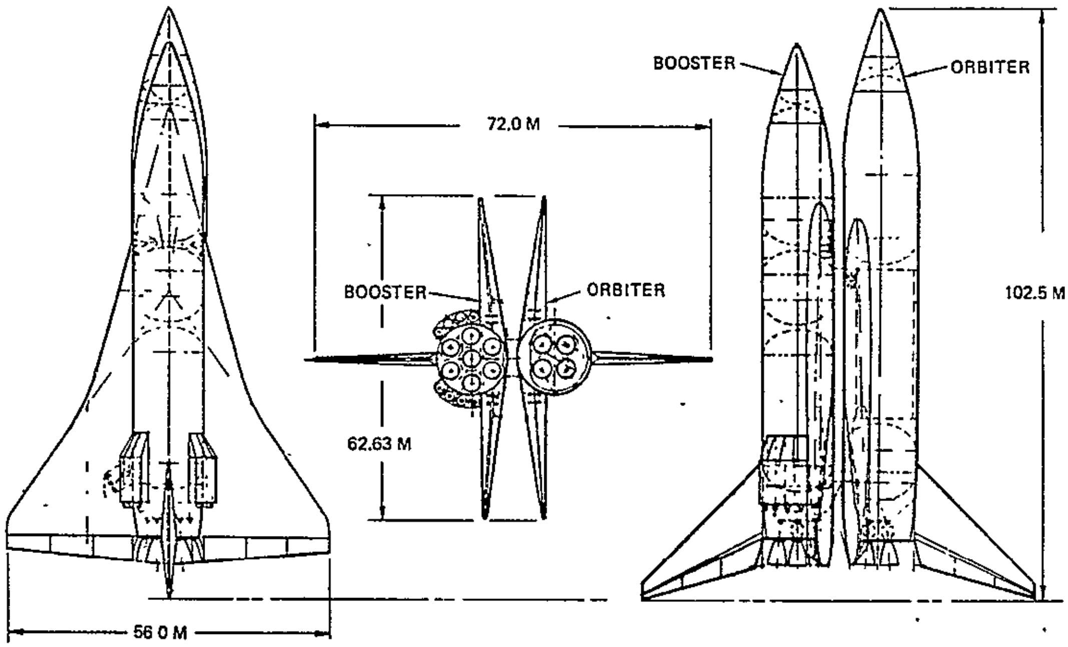


Figure 4.2-1. Reference HLLV Launch Configuration

Table 4.2-1. HLLV Mass Properties × 10^{-6}

	KG	LB
GLOW	7.14	15.73
BLOW	4.92	10.84
W_{P1}	4.49	9.89
ULOW	2.22	4.89
W_{P2}	1.66	3.65
PAYLOAD	0.23	0.51

4.2.1 HLLV FIRST STAGE (BOOSTER)

The HLLV booster is shown in the landing configuration in Figure 4.2-2. The vehicle is approximately 300 feet in length with a wing span of 184 feet and a maximum clearance height of 116 ft. The nominal body diameter is 40 feet. The vehicle has a dry weight of 1,045,500 lb. Seven high P_c gas generator driven LOX/RP engines are mounted in the aft fuselage with a nominal sea level thrust of 2.3 million pounds each. Eight turbojet engines are mounted on the upper portion of the aft fuselage with a nominal thrust of 20,000 lb each. A detailed weight statement is given in Table 4.2-2. The vehicle propellant weight summary is projected in Table 4.2-3.

4.2.2 HLLV SECOND STAGE (ORBITER)

The HLLV orbiter is depicted in Figure 4.2-3. The vehicle is approximately 317 feet in length with the same wing span, vertical height, and nominal body diameter as the booster. The orbiter employs four high P_c staged combustion LOX/LH_2 rocket engines with a nominal sea level thrust of 1.19 million lb each.

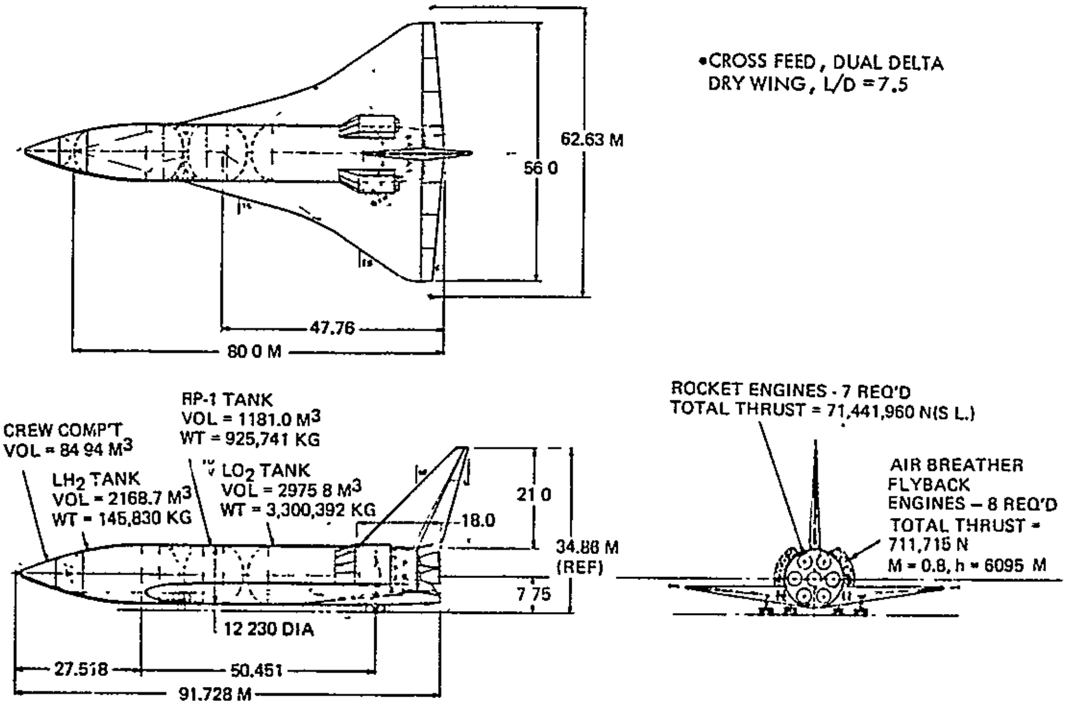


Figure 4.2-2. HLLV First Stage (Booster) - Landing Configuration

Table 4.2-2. HLLV Weight Statement
kg×10^{-3} (lb×10^{-3})

SUBSYSTEM	2ND STAGE	1ST STAGE
FUSELAGE	103.41 (227.98)	130.73 (288.22)
WING	39.20 (86.41)	78.17 (172.34)
VERTICAL TAIL	5.70 (12.57)	7.21 (15.89)
CANARD	1.39 (3.07)	2.21 (4.87)
TPS	52.59 (115.94)	-
CREW COMPARTMENT	12.70 (28.00)	**
AVIONICS	3.86 (8.50)	3.40 (7.50)
PERSONNEL	1.36 (3.00)	**
ENVIRONMENTAL	2.59 (5.70)	**
PRIME POWER	5.44 (12.00)	**
HYDRAULIC SYSTEM	3.86 (8.50)	**
ASCENT ENGINES	26.93 (59.38)	67.45 (148.70)
RCS SYSTEM	9.59 (21.15)	**
LANDING GEARS	18.38 (40.51)	**
PROPULSION SYSTEMS	*	44.99 (99.18)
ATTACH AND SEPARATION	-	4.59 (10.12)
APU	-	0.91 (2.00)
FLYBACK ENGINES	-	28.55 (62.95)
FLYBACK PROPULSION SYSTEM	-	18.39 (40.54)
SUBSYSTEMS	-	25.76 (56.80)
DRY WEIGHT	286.99 (632.71)	(909.12)
GROWTH MARGIN (15%)	43.05 (94.91)	(136.37)
TOTAL INERT WT.	330.04 (727.62)	(1045.49)

*INCLUDED IN FUSELAGE WEIGHT
**ITEMS INCLUDED IN SUBSYSTEMS

Table 4.2-3. HLLV Propellant Weight Summary × 10^{-6}

	FIRST STAGE		SECOND STAGE	
	LB	KG	LB	KG
USABLE	9.607	4.358	3.481	1.579
CROSSFEED	1.612	0.732	(1.612)	(0.731)
TOTAL BURNED	7.995	3.626	5.093	2.310
RESIDUALS	0.040	0.018	0.020	0.009
RESERVES	0.045	0.020	0.024	0.011
RCS	0.010	0.005	0.018	0.008
ON-ORBIT	-	-	0.095	0.043
BOIL-OFF	-	-	0.010	0.005
FLY-BACK	0.187	0.085	-	-
TOTAL LOADED	9.889	4.486	3.648	1.655

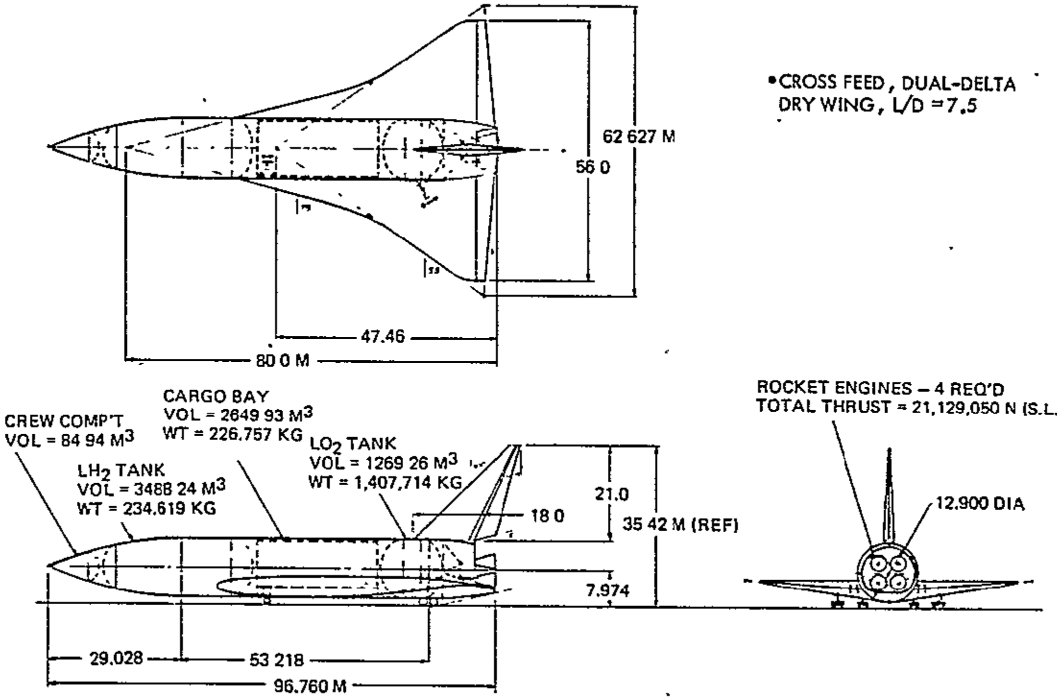


*Figure 4.2-3. HLLV Second Stage (Orbiter)
- Landing Configuration*

The cargo bay is located in the mid-fuselage in a manner similar to the STS orbiter and has a length of approximately 90 feet. The detailed weight statement and a propellant summary for the orbiter is included in Tables 4.2-2 and 4.2-3 respectively.

4.3 HLLV PERFORMANCE

The HLLV performance has been determined by using a modified STS scaling and trajectory program. The tabulated trajectory data for both nominal and abort conditions is contained in Appendix B. The vehicle can deliver a payload of approximately 231,000 kg to an orbital altitude of 487 km at an inclination of 31.6°. The engine performance parameters used in the analyses are given in Table 4.3-1.

Table 4.3-1. Engine Performance Parameters

ENGINE	SPECIFIC IMPULSE (SEC)		MIXTURE RATIO	THRUST/WEIGHT
	SEA LEVEL	VACUUM		
LOX/RP GG CYCLE	329.7	352.3	2.8:1	120
LOX/CH_4 GG CYCLE	336.9	361.3	3.5:1	120
LOX/LH_2 STAGED COMB.	337.0	466.7	6.0:1	80

The vehicle relative staging velocity is 2127 m/sec (6978 ft/sec) at an altitude of 55.15 km (181,000 ft) and a first stage burnout range of 88.7 km (48.5 nmi). The first stage flyback range is 387 km (211.8 nmi). For the reference HLLV configuration, all engine throttling to limit maximum dynamic pressure during the parallel burn mode is accomplished with the first or booster stage engines only (i.e., second stage engines operate at 100% rated thrust).

Summary vehicle characteristics are given in Tables 4.3-2 and 4.3-3. The computer CRT data are provided in Figure 4.3-1 through 4.3-35.

Table 4.3-2. Vehicle Characteristics (Nominal Mission)

STAGE	1	2	3
GROSS STAGE WEIGHT,(LB)	1572458.0	5043755.0	[illegible]
GROSS STAGE THRUST/WEIGHT	1.300	0.942	0.980
THRUST ACTUAL,(LB)	[illegible]	4750000.0	4750000.0
ISP VACUUM,(SEC)	370.891	400.700	400.700
STRUCTURE,(LB)	1045488.9	0.0	800665.0
PROPELLANT,(LB)	945000.0	224657.0	3400425.0
PERF. FRAC.,(ND)	0.6015	0.0440	0.7675
PROPELLANT FRAC.,(ND)	0.9004	1.0000	0.8007
BURNOUT TIME,(SEC)	154.092	170.700	543.320
BURNOUT VELOCITY,(FT/SEC)	7699.340	8394.405	23594.121
BURNOUT GAMMA,(DEGREES)	15.495	12.288	3.107
BURNOUT ALTITUDE,(FT)	175022.4	218297.4	317057.3
BURNOUT RANGE,(NM)	44.3	66.1	821.3
IDEAL VELOCITY,(FT/SEC)	10659.5	11294.0	25742.3

INJECTION VELOCITY,(FT/SEC)	0.0	FLYBACK RANGE(NM)	200.0
INJECTION PROPELLANT,(LB)	0.0	FLYBACK PROP(LBS)	104029.2
ON ORBIT DELTA-V,(FT/SEC)	1023.5		
ON ORBIT PROPELLANT,(LB)	95260.0		
ON ORBIT ISP,(SEC)	400.7		

THETA= 25.94 PITCH RATE= 0.00398 ATTEMPTS TO CONVERGE= 3

PAYLOAD,(LB) 550403.0

Table 4.3-3. Summary Weight Statement (Nominal Mission)

ORBITER WEIGHT BREAKDOWN		
DRY WEIGHT	72702.000	POUNDS
PERSONNEL	3000.000	POUNDS
RESIDUALS	2074.000	POUNDS
RESERVES	3300.000	POUNDS
IN-FLIGHT LOSSES	10439.000	POUNDS
ACPS PROPELLANT	10280.050	POUNDS
OMS PROPELLANT	35280.302	POUNDS
PAYLOAD	30403.000	POUNDS
BALLAST FOR CG CONTROL	0.0	POUNDS
OMS INSTALLATION KITS	0.0	POUNDS
PAYLOAD MODS	0.0	POUNDS
TOTAL END BOOST (ORBITER ONLY)	[illegible]372.0[illegible]	POUNDS
OMS BURNED DURING ASCENT	0.0	POUNDS
ACPS BURNED DURING ASCENT	0.0	POUNDS
EXTERNAL MAIN TANK		
TANK DRY WEIGHT	20400.000	POUNDS
RESIDUALS	17750.000	POUNDS
PROPELLANT BIAS	(2640.000)	POUNDS
PRESSURANT	(2123.200)	POUNDS
TANK AND LINES	(9320.000)	POUNDS
ENGINES	(5050.000)	POUNDS
FLIGHT PERFORMANCE RESERVE	20950.000	POUNDS
UNBURNED PROPELLANT (MAIN TANK)	0.0	POUNDS
TOTAL END BOOST (EXTERNAL TANK)	41300.000	POUNDS
USABLE PROPELLANT (EXTERNAL TANK)	[illegible]92055.000	POUNDS
FLYBACK PROPELLANT (FIRST STAGE)	134629.167	POUNDS
SOLID ROCKET MOTOR (FIRST STAGE)	[illegible]40540.00	POUNDS
SRM CASE WEIGHT(2)	1045420.017	POUNDS
SRM STRUCTURE & RCVY WEIGHT	0.0	POUNDS
SRM INERT STAGING WEIGHT	1045400.87	POUNDS
USABLE SRM PROPELLANT	7995000.00	POUNDS
TOTAL GROSS LIFT-OFF WEIGHT (GLOW)	[illegible]27450.0	POUNDS

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OF POOR QUALITY

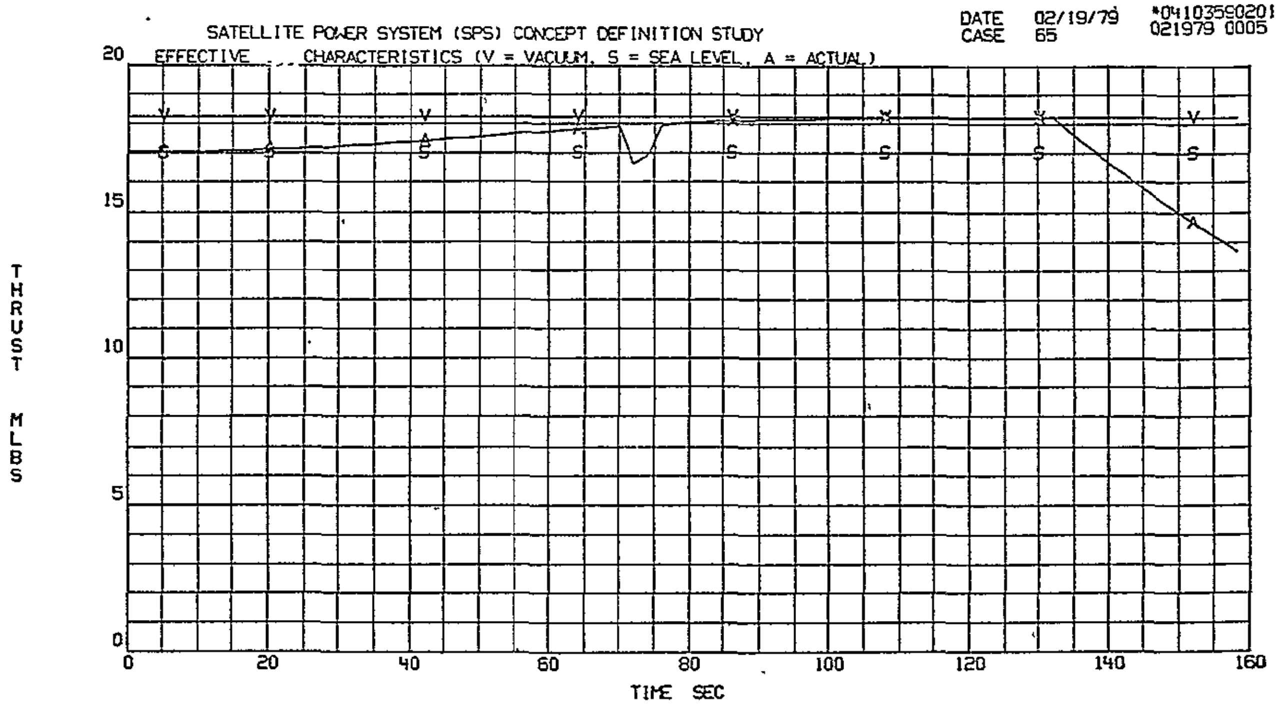

Figure 4.3-1. First Stage Thrust vs Time

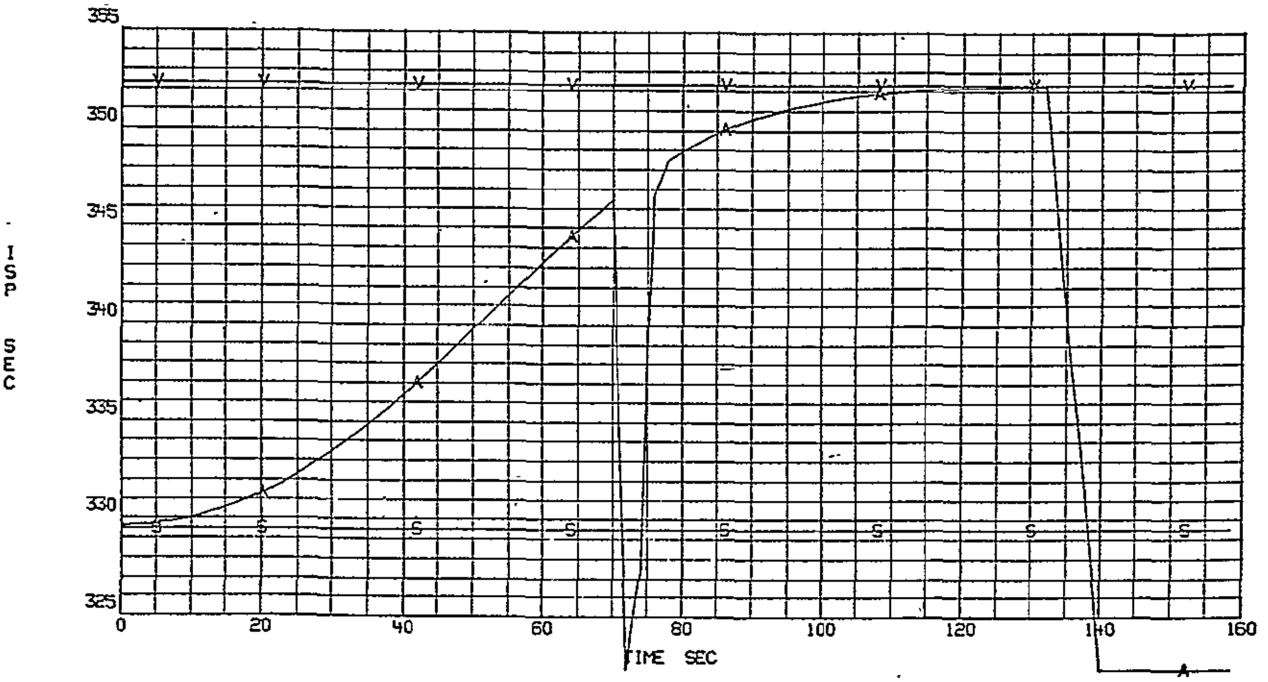

Figure 4.3-2. First Stage Specific Impulse vs Time

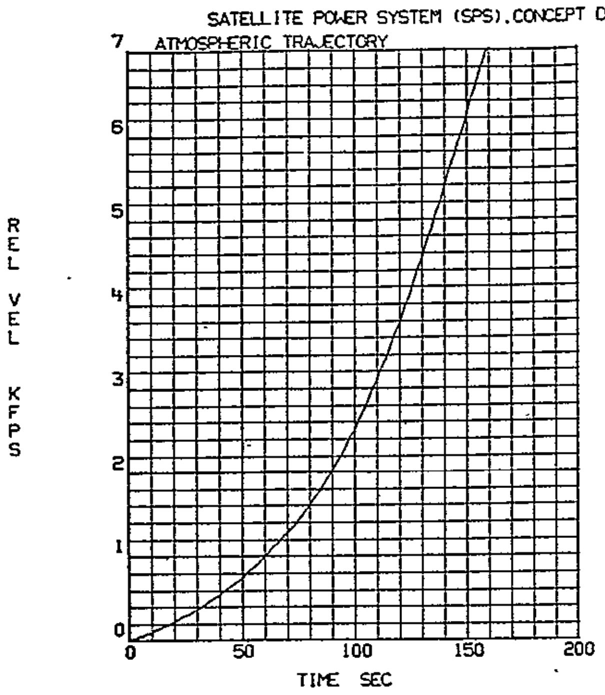

Figure 4.3-3. First Stage Relative Velocity vs Time

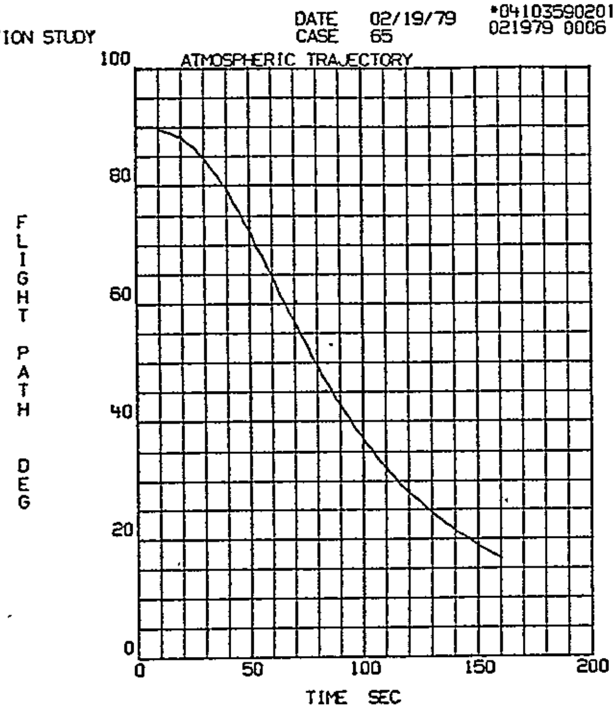

Figure 4.3-4. First Stage Flight Path Angle vs Time

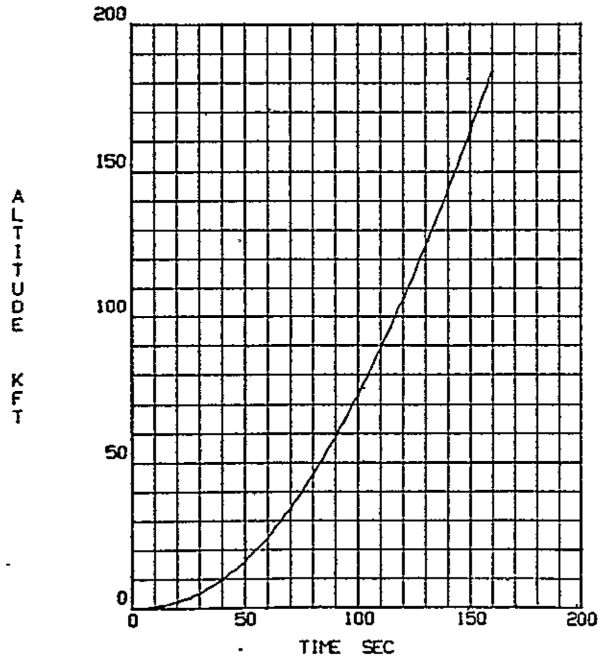

Figure 4.3-5. First Stage Altitude vs Time

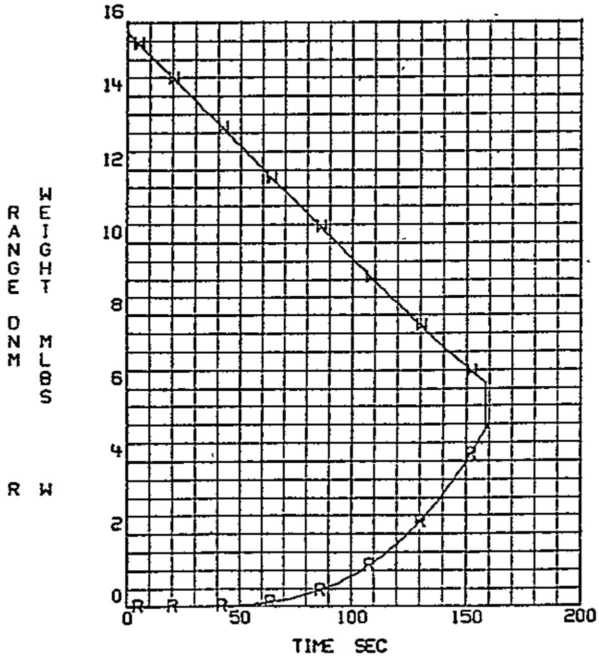

Figure 4.3-6. First Stage Weight and Range vs Time

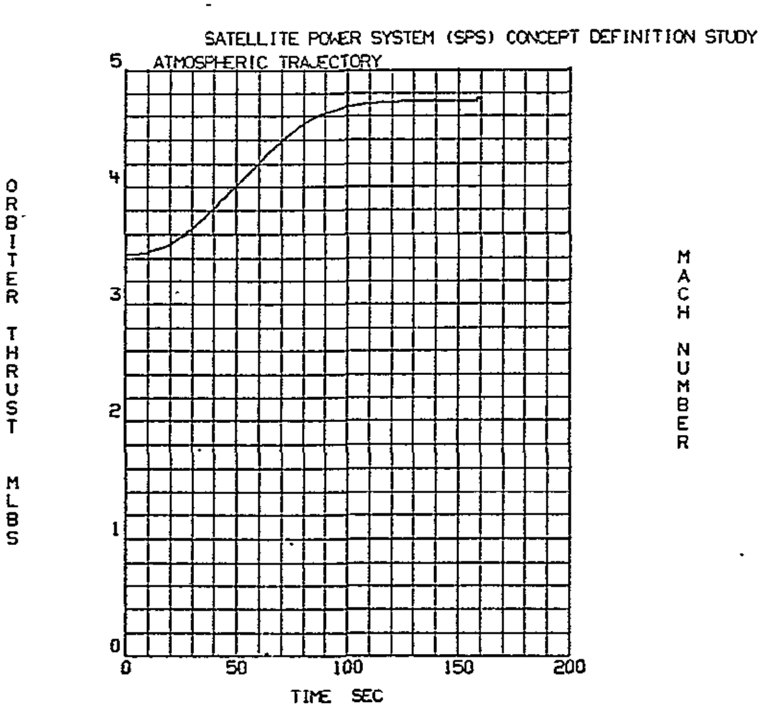

Figure 4.3-7. Second Stage Thrust vs Time

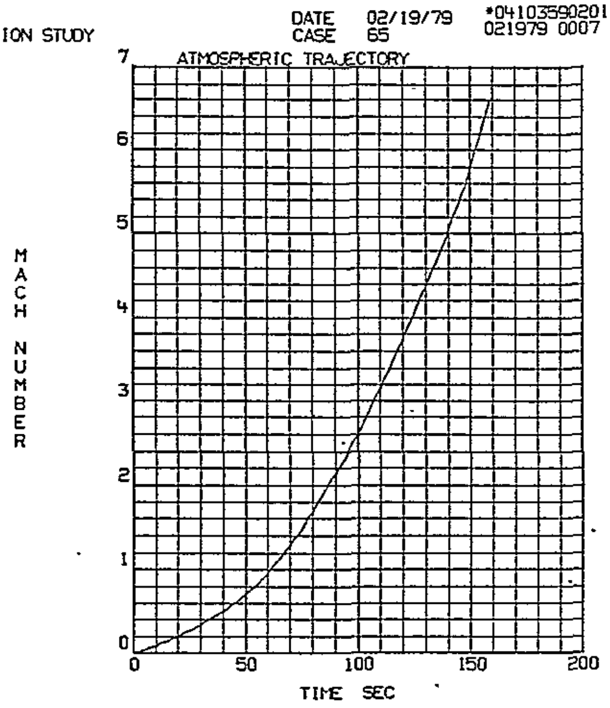

Figure 4.3-8. Mach Number vs Time

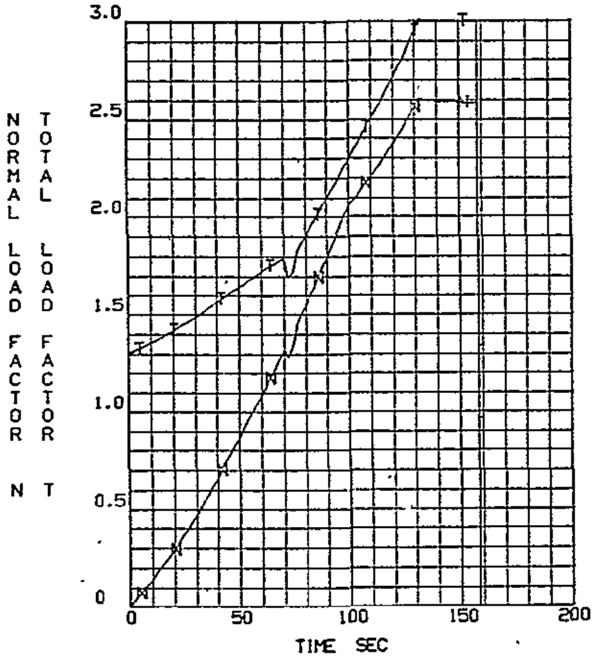

Figure 4.3-9. Normal and Total Load Factor vs Time

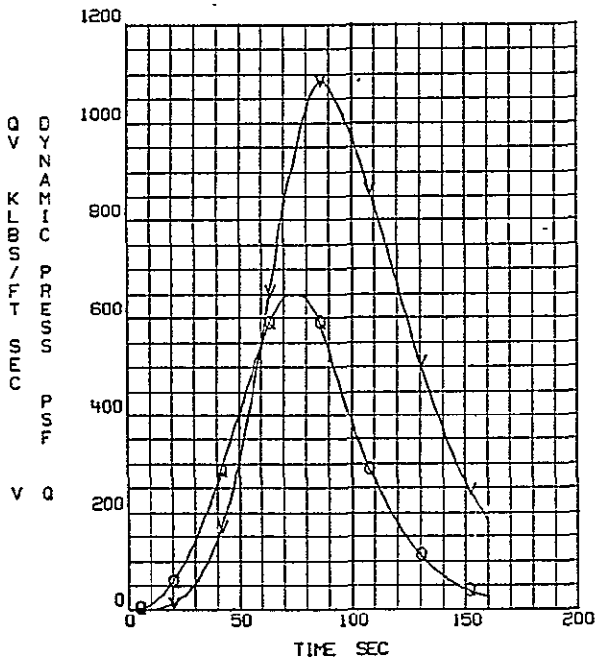

Figure 4.3-10. Q and QV vs Time

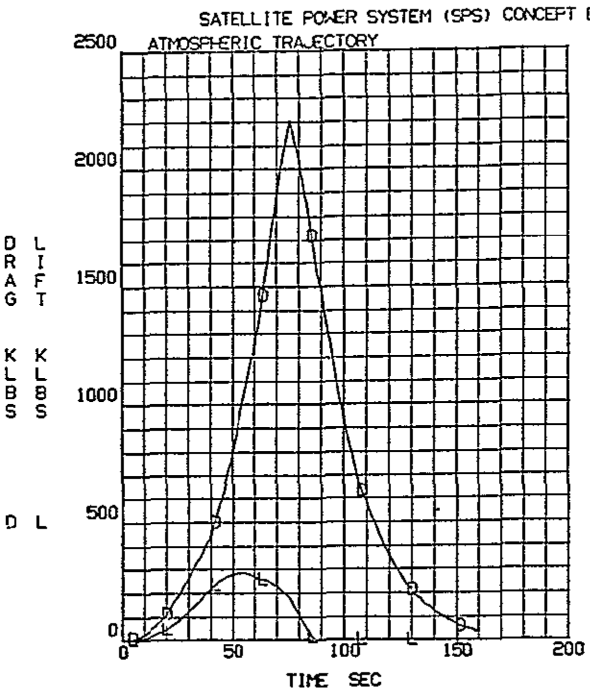

Figure 4.3-11. Lift and Drag vs Time

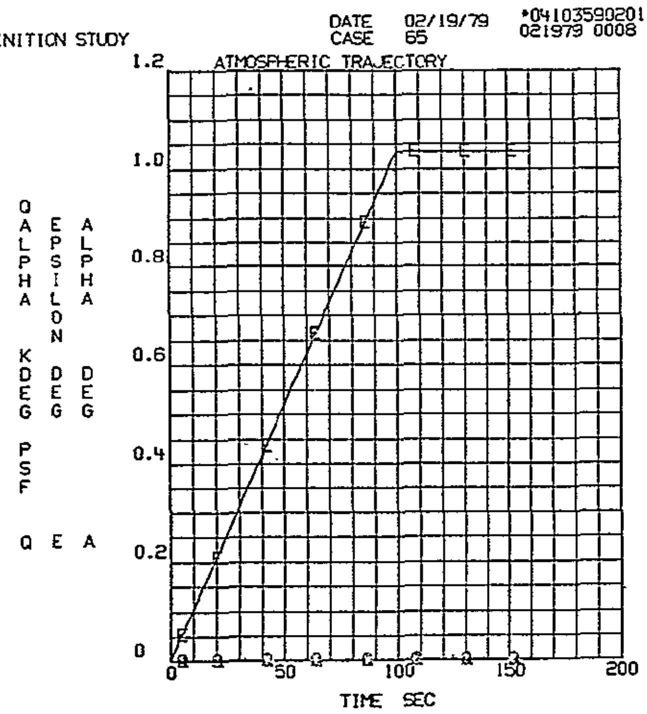

Figure 4.3-12. α, ε and αQ vs Time

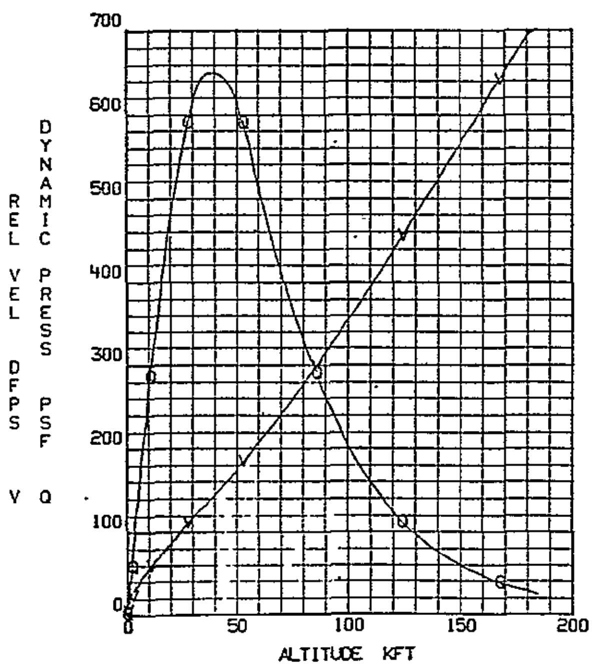

Figure 4.3-13. Relative Velocity and Q vs Altitude

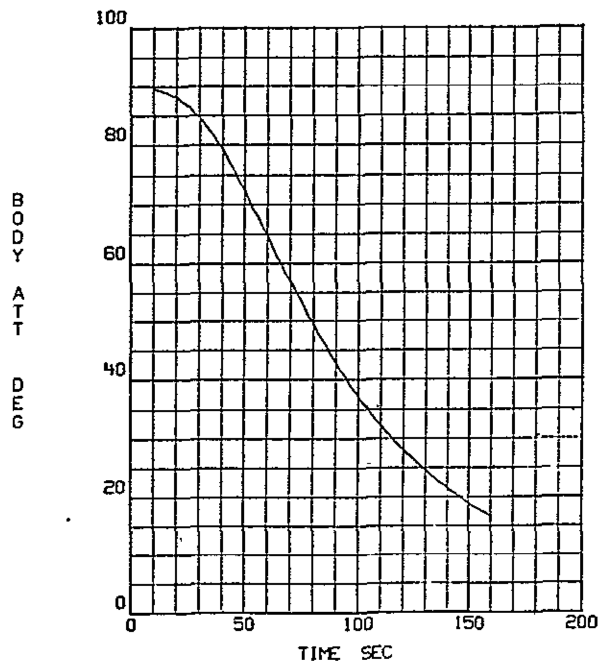

Figure 4.3-14. Body Attitude vs Time

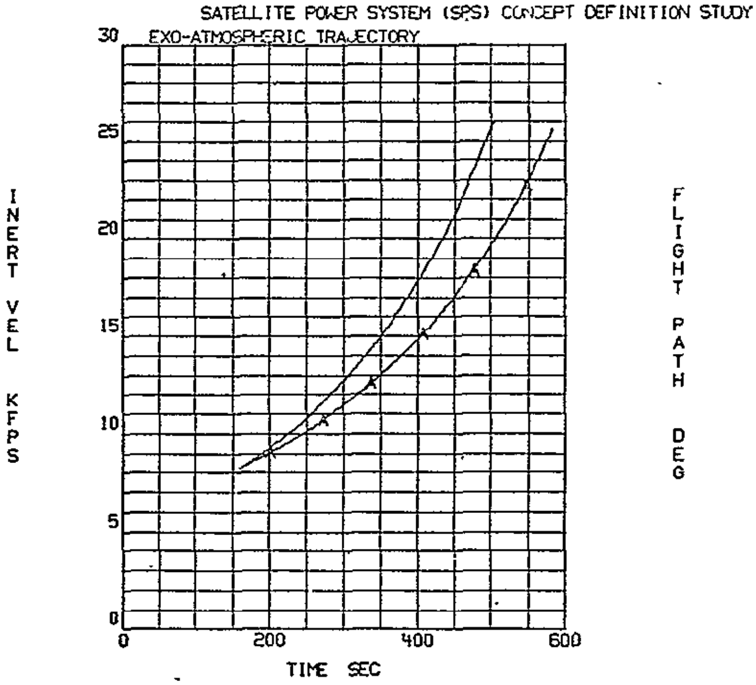

Figure 4.3-15. Inertial Velocity vs Time

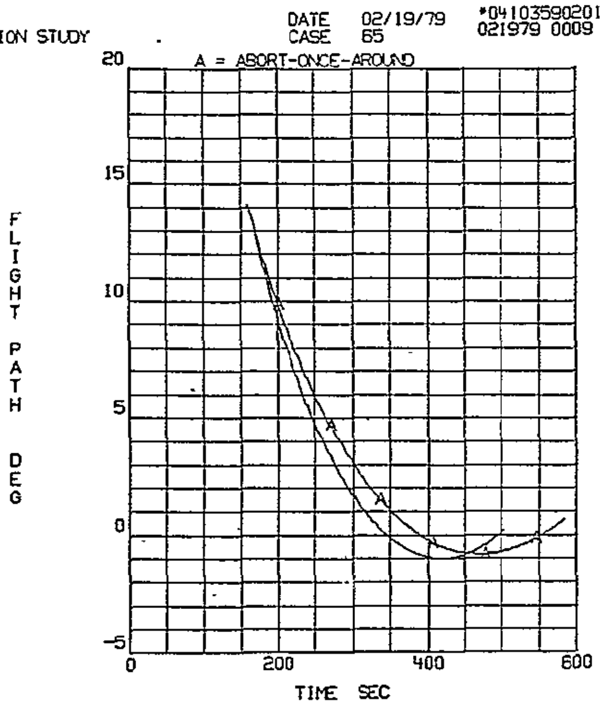

Figure 4.3-16. Flight Path Angle vs Time

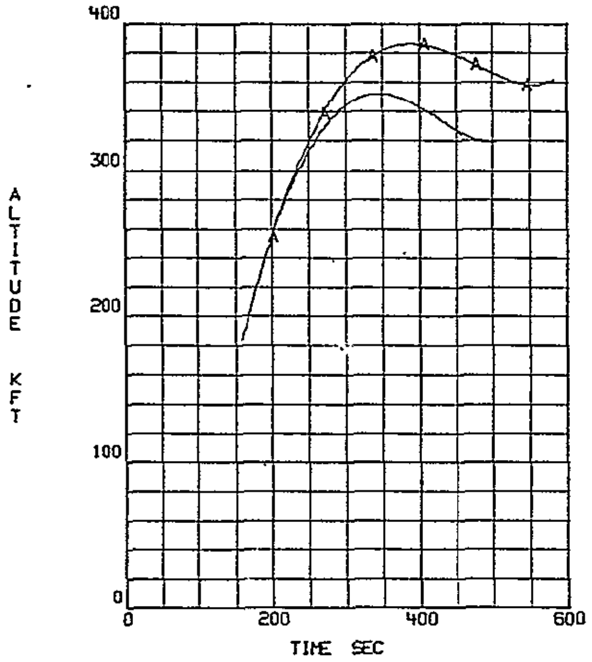

Figure 4.3-17. Altitude vs Time

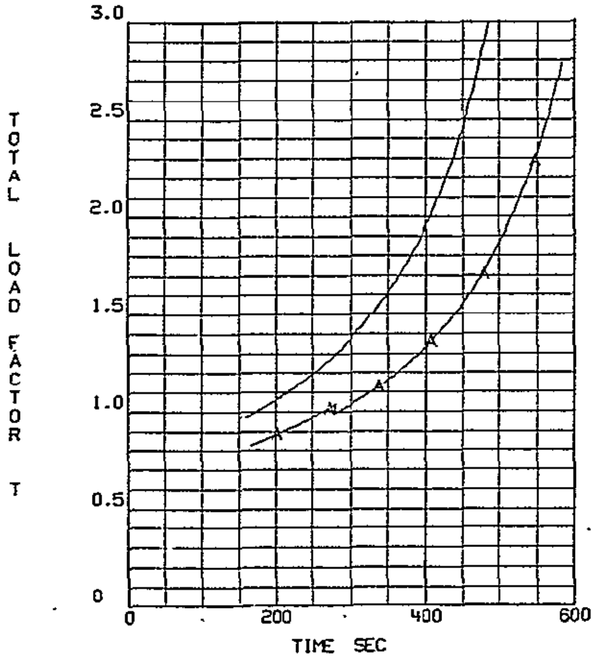

Figure 4.3-18. Total Load Factor vs Time

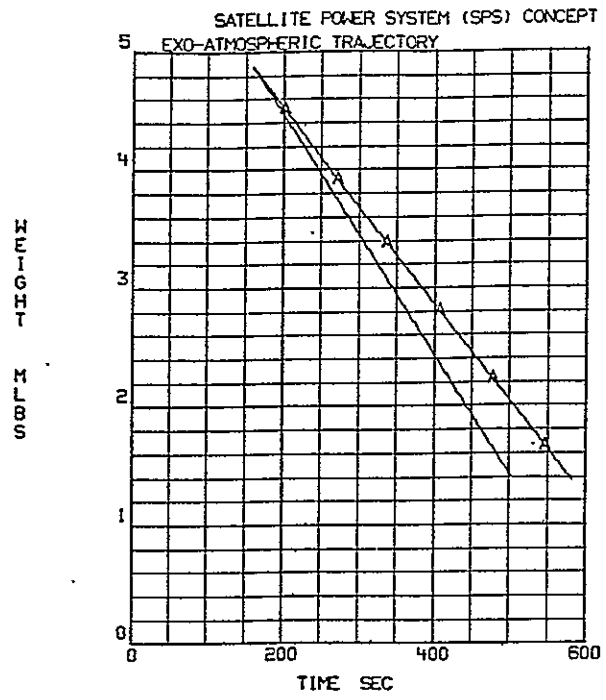

Figure 4.3-19. Weight vs Time

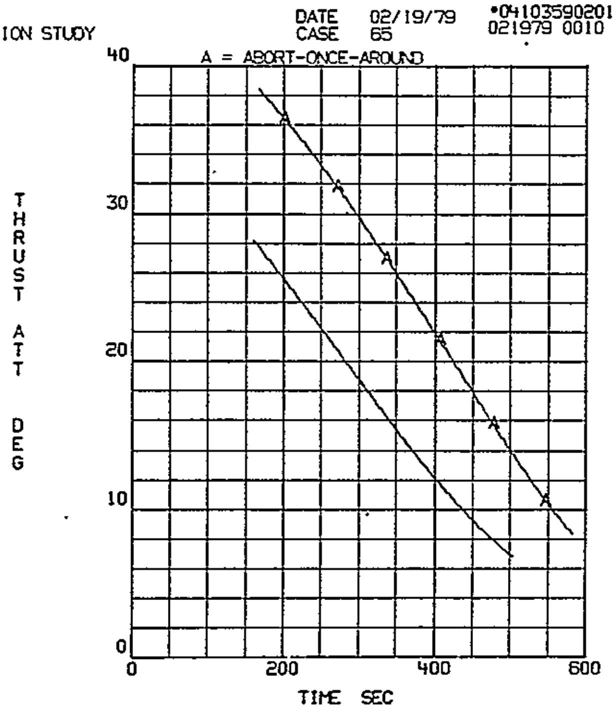

Figure 4.3-20. Thrust Attitude vs Time

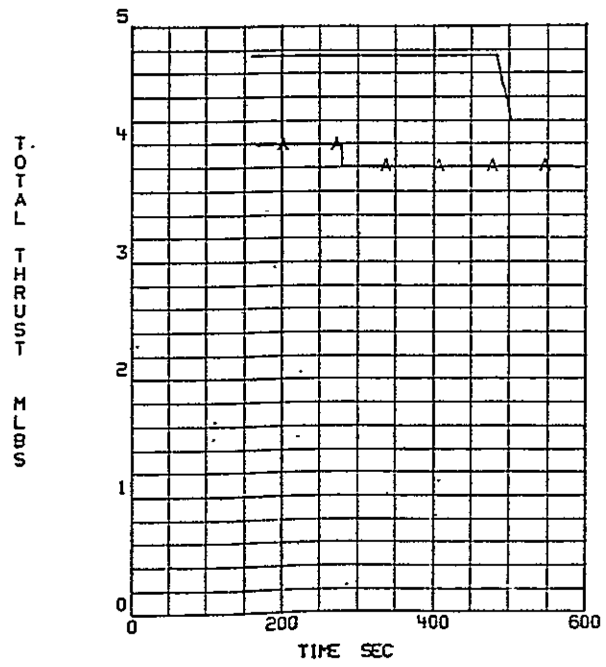

Figure 4.3-21. Total Thrust vs Time

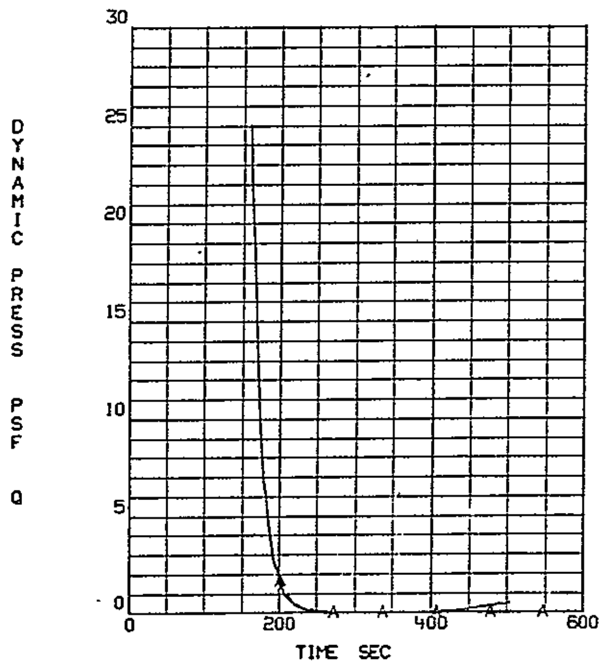

Figure 4.3-22. Dynamic Pressure vs Time

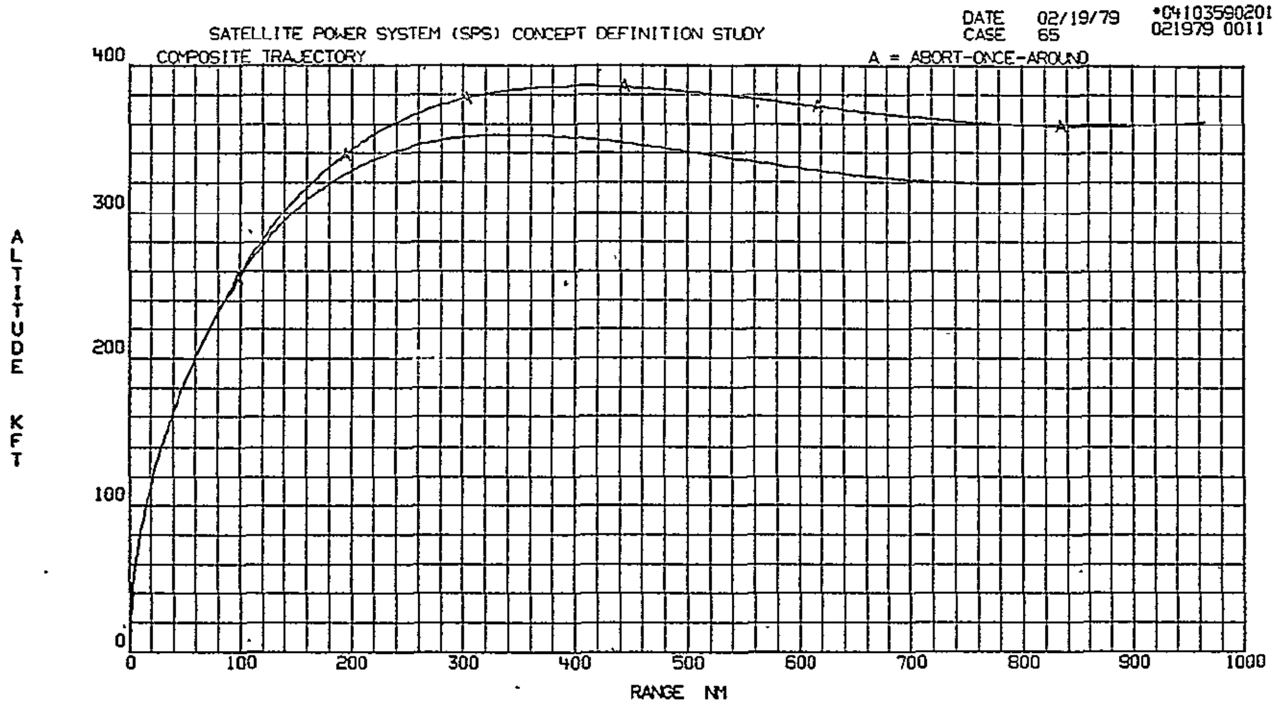

Figure 4.3-23. Altitude vs Range

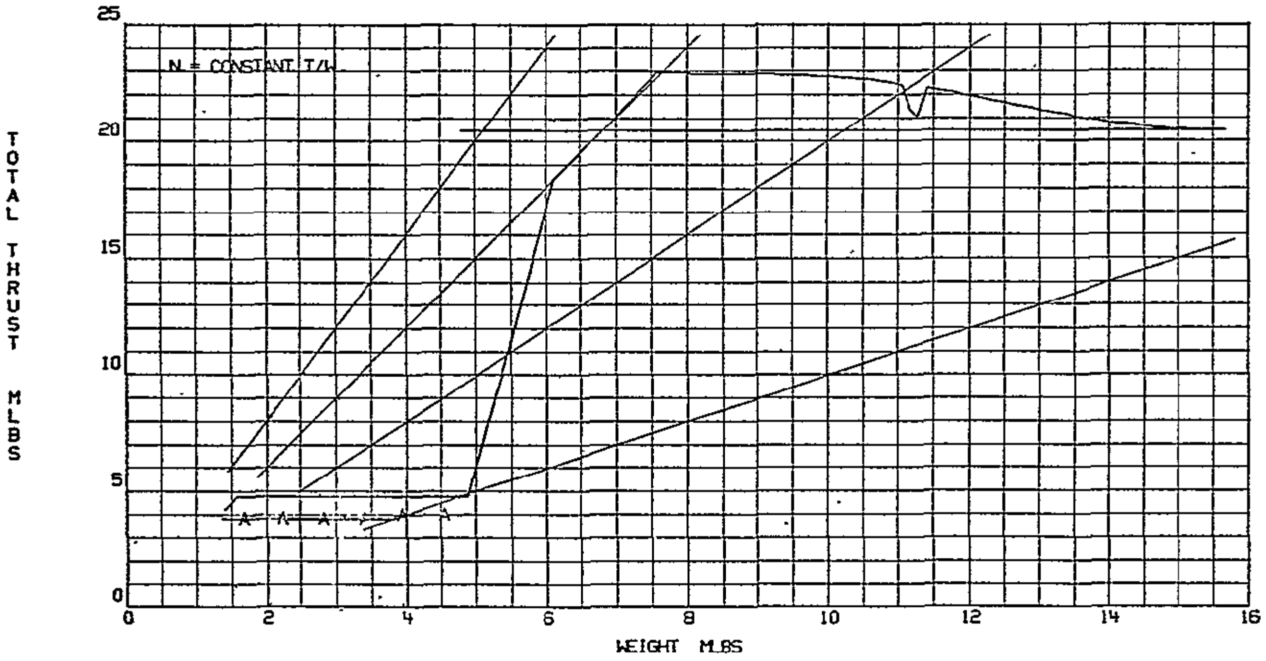

Figure 4.3-24. Total Thrust vs Weight

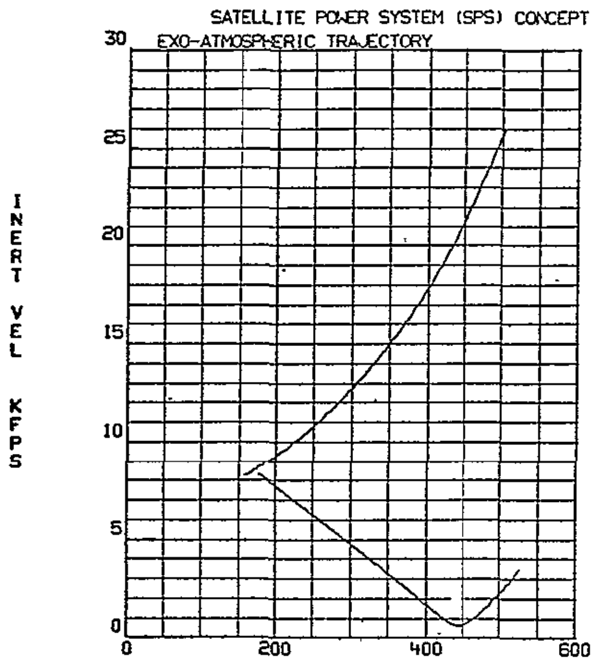

Figure 4.3-25. Inertial Velocity vs Time

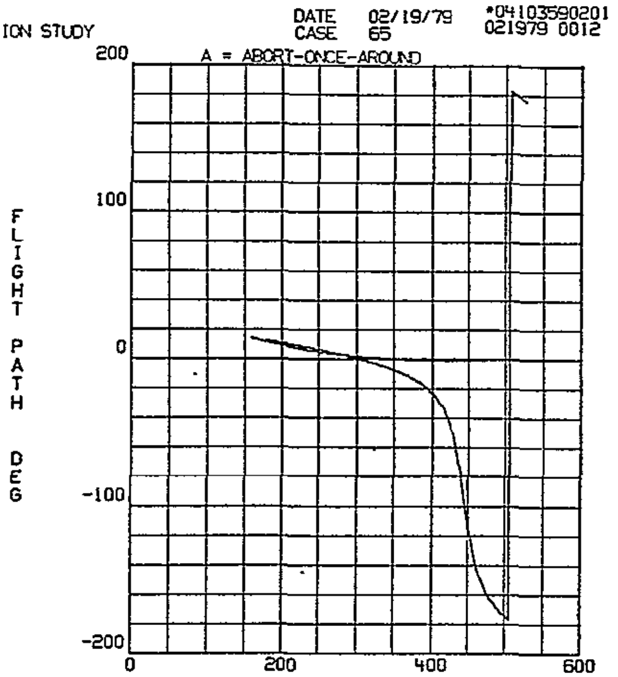

Figure 4.3-26. Flight Path Angle vs Time

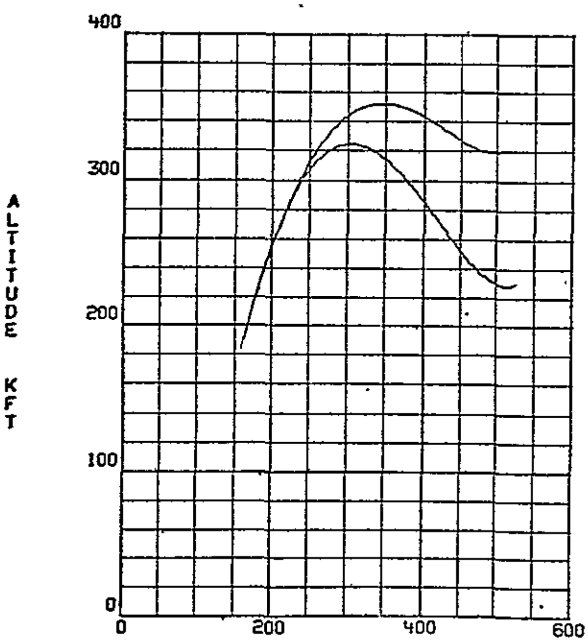

Figure 4.3-27. Altitude vs Time

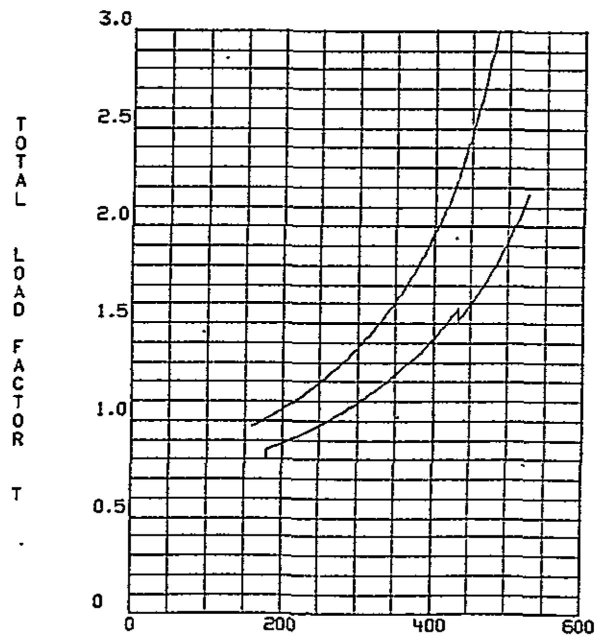

Figure 4.3-28. Total Load Factor vs Time

SATELLITE POWER SYSTEM (SPS) CONCEPT DEFINITION STUDY

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CASE 65 021979 0013

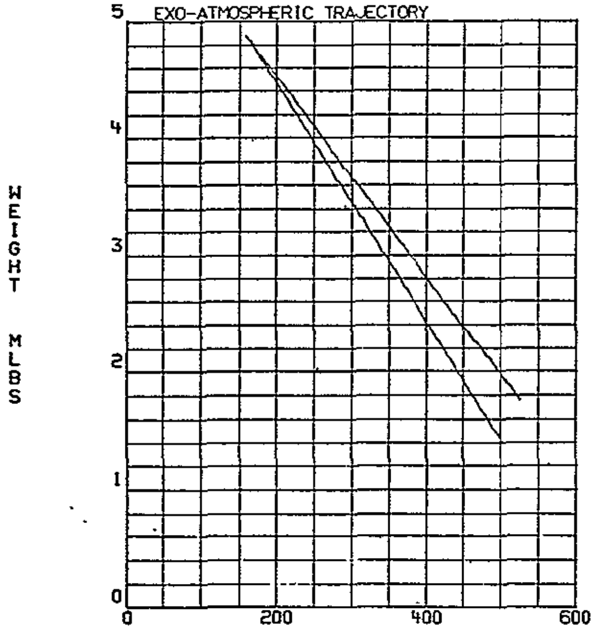


Figure 4.3-29. Weight vs Time

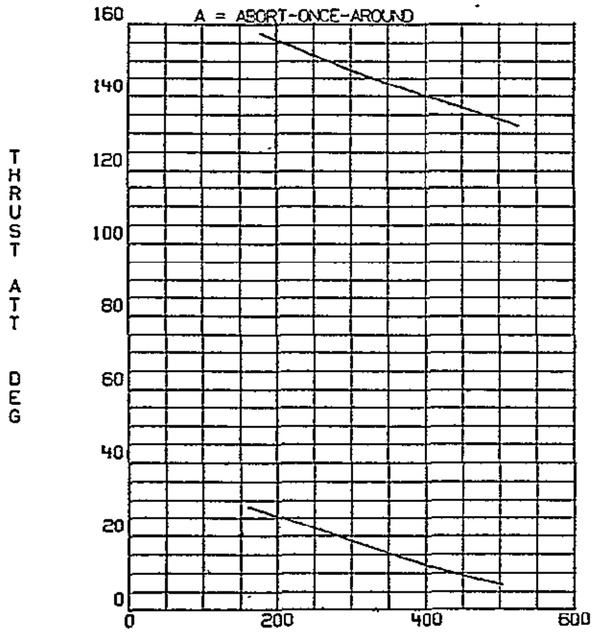


Figure 4.3-30. Thrust Attitude vs Time

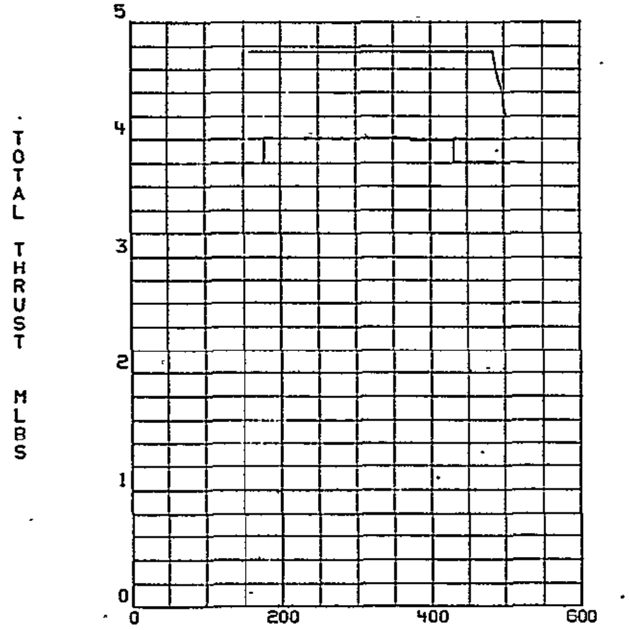


Figure 4.3-31. Total Thrust vs Time

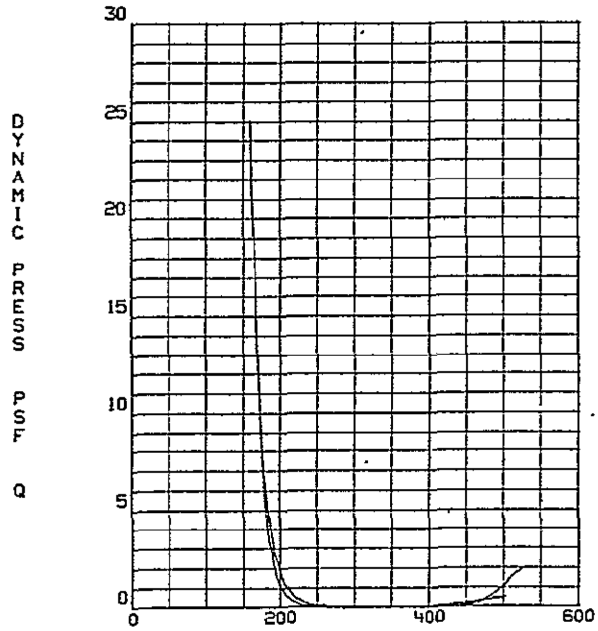


Figure 4.3-32. Dynamic Pressure vs Time

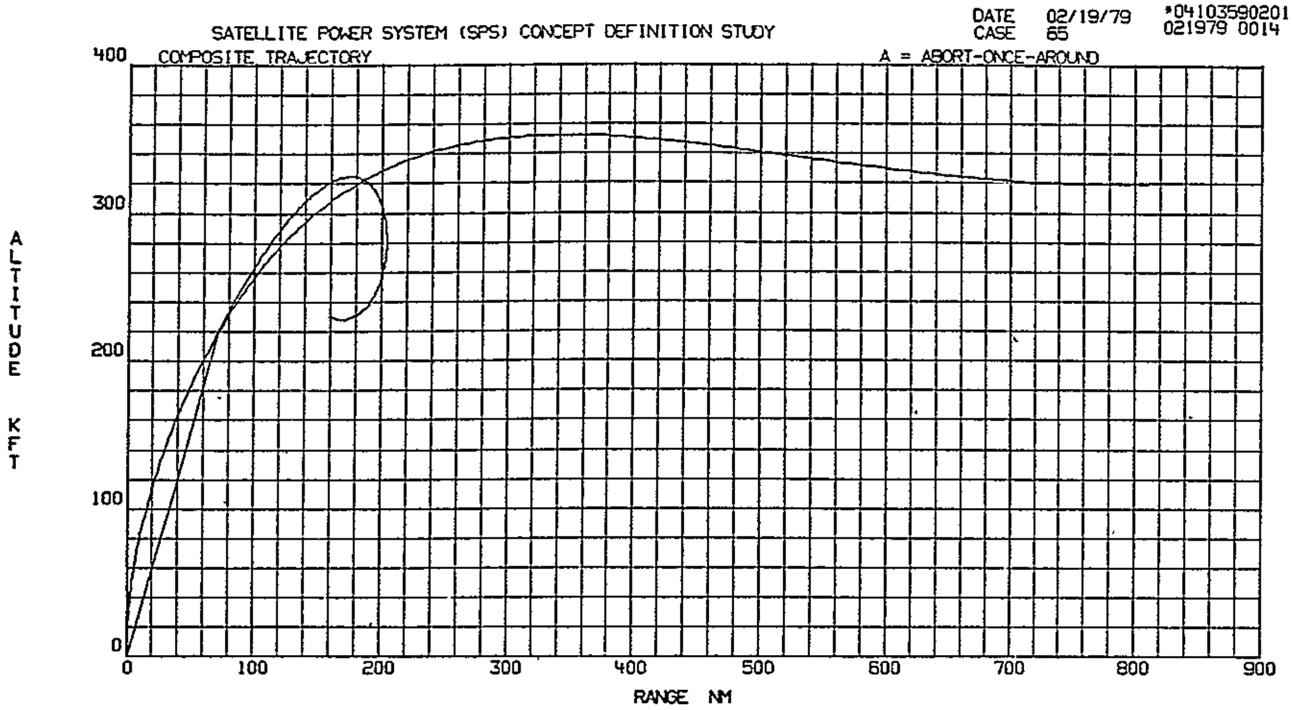


Figure 4.3-33. Altitude vs Range

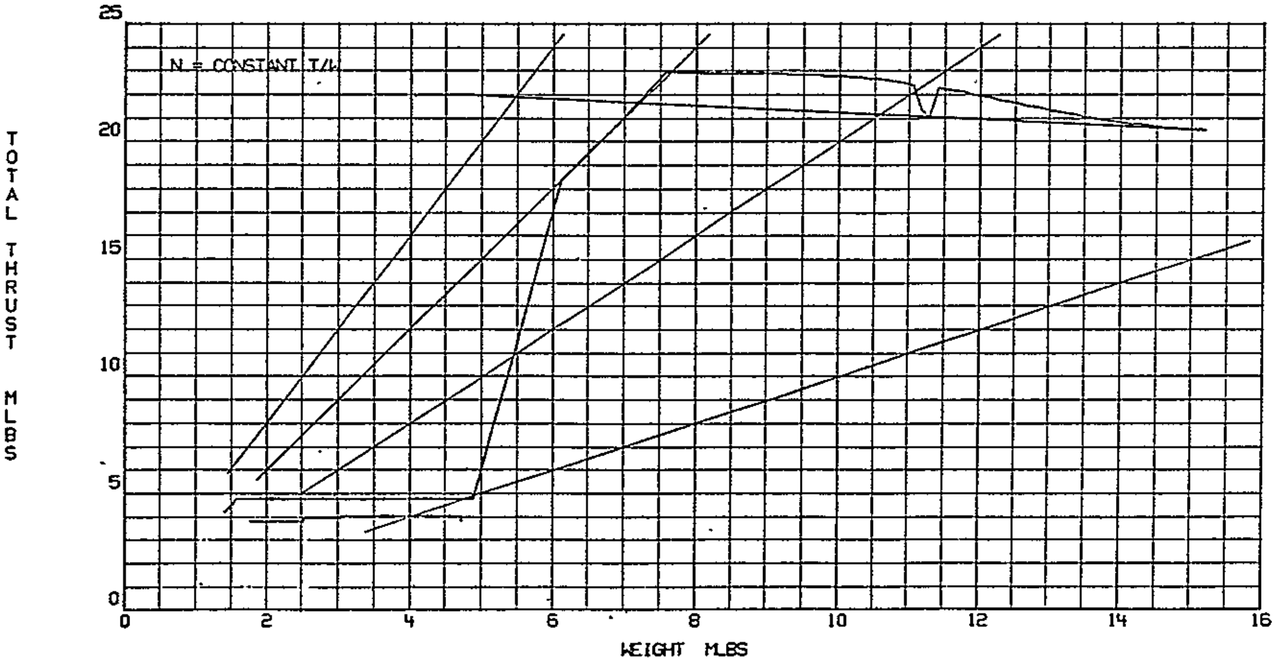


Figure 4.3-34. Total Thrust vs Weight

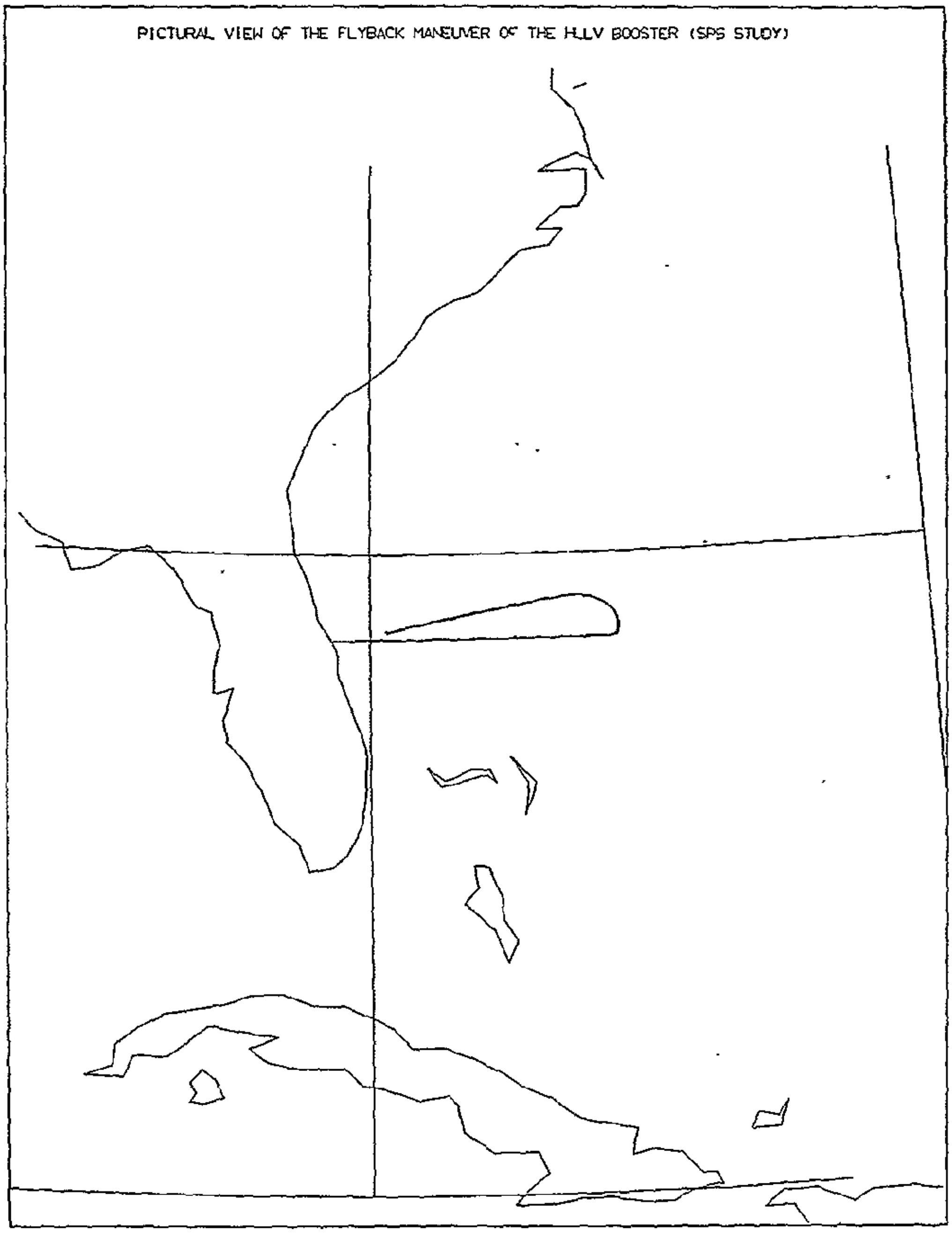


Figure 4.3-35. First Stage Flyback Trajectory

4.4 TRADE STUDY OPTIONS

The trade study options data are given in Appendix B. The several trade options evaluated included the following:

- First and Second Stage Engine Throttling
- First Stage Propellant Weight Sensitivity
- Second Stage Propellant Weight Sensitivity
- Lift-off Thrust-to-Weight Sensitivity
- Alternate First Stage Propellants (LOX/CH_4 and LOX/LH_2)

With the exception of the engine throttling trades, all trajectories assumed 100% throttling by the first stage engines (i.e., second stage engines operate at maximum thrust throughout the parallel burn ascent phase) in order to stay within maximum allowable load factor and dynamic pressure, 3 g and 650 psf respectively.

The engine throttling study shows little effect on vehicle payload capability when doing 100% of the throttling with either stage. All intermediate options (i.e., partial throttling of both stages) shows a degradation in payload capability.

The first stage propellant weight sensitivity analyses show an improvement in glow/payload weight ratio (smaller) as first stage propellant weight is increased, however, the staging velocity exceeds the capability of a heat sink booster. The second stage propellant weight sensitivity indicates an opposite effect to the first stage data.

By combining the effects of throttling of second stage only and increasing first stage propellant weight could result in a 10-15% improvement over the reference HLLV configuration.

The alternate propellant trades, LOX/CH_4 and LOX/LH_2, show 7% and 37% increased performance over the reference HLLV configuration. The LOX/LH_2 configuration, however, becomes extremely large (volume) and less cost effective because of handling and propellant costs. The LOX/CH_4 booster appears to be a viable option.

5.0 LEO-TO-GEO TRANSPORTATION, EOTV

5.0 LEO-TO-GEO TRANSPORTATION, EOTV

5.0 LEO-TO-GEO TRANSPORTATION - EOTV

It was previously shown that a chemical orbital transfer vehicle requires a prohibitive propellant mass to place the SPS mass in GEO because of the limited available specific impulse of chemical systems. An electric argon ion orbital transfer system was therefore selected as a baseline for SPS cargo transfer from LEO-to-GEO. This study phase was directed toward better definition and a degree of optimization of the EOTV concept. Detailed electric thruster analyses and parametric scaling data are included in Appendix C.

5.1 ELECTRIC ORBITAL TRANSFER VEHICLE CONCEPT

The electric OTV concept, Figure 5.1-1 is based upon a rigid design which can accommodate two "standard" solar blanket areas of 600 meters by 750 meters from the MSFC/Rockwell baseline satellite concept. The commonality of the structural configuration and construction processes with the satellite design is noted. Since the thrust levels will be very low (as compared to chemical stages), the engines and power processing units are mounted in four arrays at the lower corners of the structure/solar array. Each array contains 36 thrusters, however, only sixty-four thrusters are capable of firing simultaneously. The additional thrusters provide redundancy when one or more arrays cannot be operated due to potential plume impingement on the solar array. Up to 16 thrusters, utilizing stored electrical power are used for attitude hold only during periods of occultation. The attitude determination system is the same as the SPS, mounted in 6 locations as indicated. Payload attach platforms are located so that loading/unloading operations can be conducted from "outside" the light weight structure.

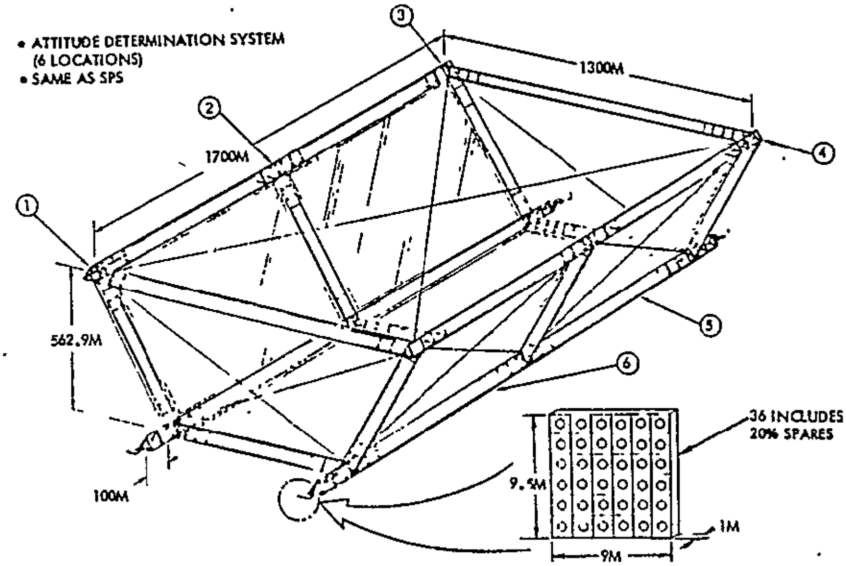


Figure 5.1-1. EOTV Configuration

5.1.1 EOTV SIZING ASSUMPTIONS

A list of primary assumptions used in EOTV sizing are summarized in Table 5.1-1. The orbital parameters are consistent with SPS requirements and the delta "V" requirement was taken from previous SEP and EOTV trajectory calculations. A 0.75% delta "V" margin is included in the figure given.

Table 5.1-1. EOTV Sizing Assumptions

- LEO ALTITUDE - 487 KM @ 31.6° INCLINATION
- SOLAR INERTIAL ORIENTATION
- LAUNCH ANY TIME OF YEAR
- 5700 M/SEC ΔV REQUIREMENT
- SOLAR INERTIAL ATTITUDE HOLD ONLY DURING OCCULTATION PERIODS
- 50° PLUME CLEARANCE
- NUMBER OF THRUSTERS - MINIMIZE
- 20% SPARE THRUSTERS - FAILURES/THRUST DIFFERENTIAL
- PERFORMANCE LOSSES DURING THRUSTING - 5%
- ACS POWER REQUIREMENT - MAXIMUM OCCULTATION PERIOD
- ACS PROPELLANT REQUIREMENTS - 100% DUTY CYCLE
- 25% WEIGHT GROWTH ALLOWANCE

During occultation periods, attitude hold only is required (i.e., thrusting for orbital change is not required).

Since it is currently anticipated that thruster grid changes will be required after each mission, a minimum number of thrusters are desired to minimize operational requirements.

An excess of thrusters are included in each array to provide for potential failures and primarily to permit higher thrust from active arrays when thrusting is limited or precluded from a specific array due to potential thruster exhaust impingement on the solar array or to provide thrust differential as required for thrust vector/attitude control. A 5% specific impulse penalty was also applied to compensate for thrust cosine losses due to thrust vector/attitude control.

An all-electric thruster system was selected for attitude control during occultation periods. The power storage system was sized to accommodate maximum gravity gradient torques and occultation periods. A very conservative duty cycle of 100% was assumed for establishing ACS propellant requirements. A 25% weight growth margin was applied as in the case of the SPS.

5.1.2 EOTV SIZING APPROACH

The key criteria in sizing the EOTV are given in Table 5.1-2. As stated previously the EOTV power source utilizes the same construction approach as the basic SPS. Structural bays and solar blanket sizes are consistent with those of the SPS.

Table 5.1-2. EOTV Sizing Approach

- SAME CONSTRUCTION/CONFIGURATION AS SPS
- PAYLOAD CAPABILITY $\gtrsim 4\times10^6$ KG UP/10% DOWN
- SELF-ANNEALING SOLAR CELLS (GaAlAs)
- TRIP TIME LEO-TO-GEO $\simeq$ 120 DAYS
  GEO-TO-LEO $\lesssim$ 30 DAYS
- END-OF-LIFE PERFORMANCE CRITERIA - 15% DEGRADATION
- SAME CRITERIA USED FOR SI EOTV CONFIGURATION

The payload capability of 4×10^{-6} kilograms is consistent with previous study results which indicated minimum transportation costs based on 8 to 12 EOTV flights and LEO-to-GEO trip times between 100 and 130 days (see Trade Studies). A 10% down payload capability is provided in order to return payload packaging materials.

The GaAlAs cells are assumed to be self-annealing of electron damage occurring during transit through the Van Allen belt. A lifetime degradation in performance of 15% is consistent with basic SPS criteria. This end-of-life performance was conservatively used in all performance calculations.

The issue of silicon cell annealing was not addressed. However, the same assumptions used for the GaAlAs system were applied to the silicon cell configuration (see Trade Studies).

5.1.3 EOTV SIZING LOGIC

The logic employed in sizing the EOTV and thruster selection are summarized in Table 5.1-3.

Table 5.1-3. EOTV Sizing Logic

- SOLAR ARRAY CONFIGURATION - AVAILABLE POWER
- GRID OPERATING TEMPERATURE - MAXIMUM TOTAL VOLTAGE
- GRID VOLTAGE (PLASMA LIMITED) - SPECIFIC IMPULSE
- *NUMBER OF THRUSTERS - BEAM CURRENT/DIAMETER/THRUST
- TRIP TIME - PROPELLANT WEIGHT/PAYLOAD WEIGHT

*CONSISTENT WITH ACS THRUST REQUIREMENTS

Having adopted a basic solar array configuration, the available power is thus established. The solar array consisting of two SPS bays has a total power output of 335.5 megawatts. Line losses of 6% and an end-of-life cell degradation of 15% were assumed which yields a net power to the thruster arrays of 268.1 megawatts. The thruster array losses were determined to be negligible. The power storage system was also sized on the same basis as for the SPS, 200 kilowatt-hours per kilogram weight.

The practical upper operating temperature limit of 1900°K for molybdenum thruster grids fixes the maximum absolute operating voltage of the thrusters at 8300 volts (see Appendix C).

The solar array voltages must be as high as possible to reduce wiring weight penalties, yet, power loss by current leaking through the surrounding plasma must be at an acceptable level. There is no significant flight test data available on plasma-current leakage. [Planned experiments aboard the SPHINX satellite (February 1974) were lost due to a launch failure.] K. L. Kennerud in 1974 predicted plasma power loss based on analysis and plasma-chamber experiments, Figure 5.1-2. The plasma loss from a 90 percent insulated array is plotted in the figure as a function of altitude with voltage as a parameter. At 500 km altitude and very large arrays and high efficiency cells, it may be possible to utilize 2000 volts.

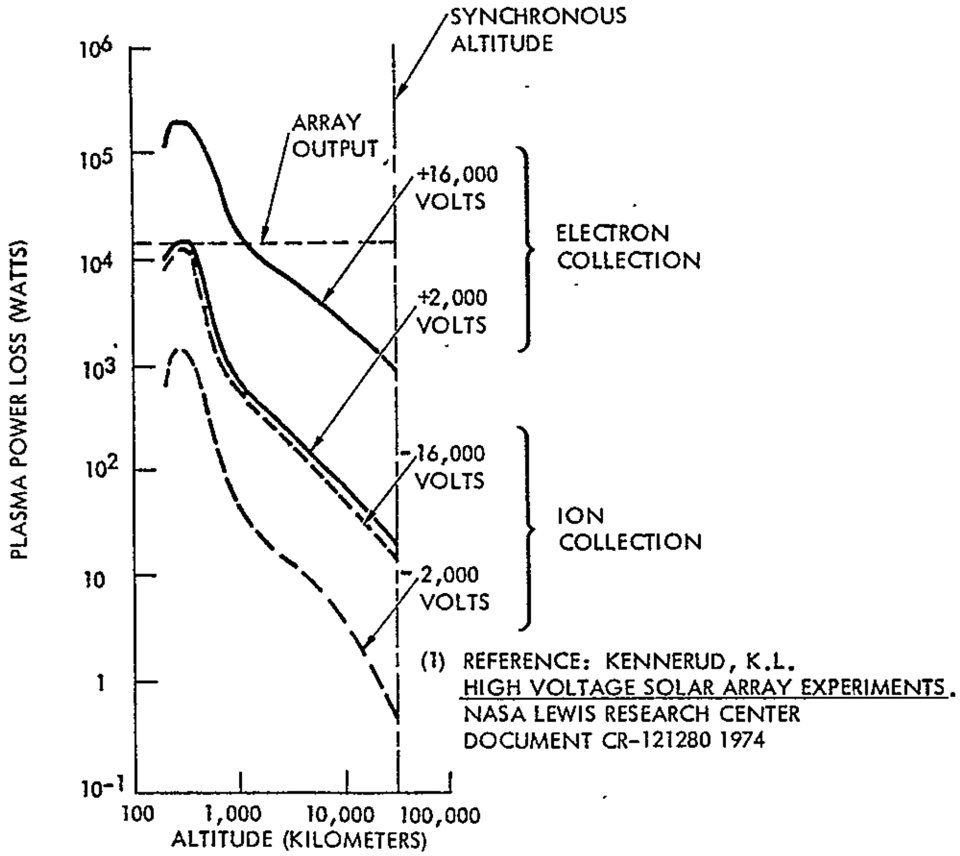


Figure 5.1-2. Plasma Power Losses from a 15 kW Solar Array with 90% Insulating Surface

An upper limit of +2000 volts was therefore assumed in order to preclude the possibility of arcing due to LEO plasma effects. A specific trade of conductor insulation requirements as a function of positive voltage is indicated. The screen grid voltage establishes propellant specific impulse at 8221 sec. The number of thrusters selected establishes the remaining thruster parameters.

(The number of thrusters should be selected such that the individual thrust is consistent with attitude control thrust requirements in order to preclude the need for dedicated ACS thrusters.) Thruster characteristics are summarized in Table 5.1-4.

Table 5.1-4. EOTV Thruster Characteristics

- MAXIMUM OPERATING TEMPERATURE - 1900° K
- TOTAL VOLTAGE - 8300 VOLTS
- GRID VOLTAGE - 2000 VOLTS MAXIMUM
- BEAM CURRENT - 1887 AMP
- SPECIFIC IMPULSE - 8213 SEC
- THRUSTER DIAMETER ~ 76 CM
- THRUST/THRUSTER - 69.7 NEWTON
- NUMBER OF THRUSTERS - 144 (INCLUDES 25% SPARES)
- MAXIMUM OF 64 THRUSTERS OPERABLE SIMULTANEOUSLY

By establishing trip time (see Trade Studies), the maximum quantity of propellant which can be consumed during transit is established; which in turn fixes maximum payload capability.

5.1.4 EOTV WEIGHT/PERFORMANCE SUMMARY

Based upon the assumptions, approach and logic described above, the EOTV weights and performance are essentially established. The selected EOTV weight and performance summary is given in Table 5.1-5, and the configuration is shown in Figure 5.1-3.

Table 5.1-5. EOTV Weight/Performance Summary (kg)

SOLAR ARRAY		588,196
CELLS/STRUCTURE	299,756	
POWER CONDITIONING	288,440	
THRUSTER ARRAY (4)		96,685
THRUSTERS/STRUCTURE	10,979	
CONDUCTORS	4,607	
BEAMS/GIMBALS	2,256	
PROPELLANT TANKS	78,843	
ATTITUDE CONTROL SYSTEM		186,872
POWER SUPPLY	184,882	
SYSTEM COMPONENTS	274	
PROPELLANT TANKS	1,716	
EOTV INERT WEIGHT		871,753
25% GROWTH		217,938
TOTAL INERT WEIGHT		1,089,691
PROPELLANT WEIGHT		666,660
TRANSFER PROPELLANT	655,219	
ACS PROPELLANT	11,441	
EOTV LOADED WEIGHT		1,756,351
PAYLOAD WEIGHT		5,171,318
LEO DEPARTURE WEIGHT		6,927,669
PROPELLANT COST DELIVERED ($/KG P/L)		4.72

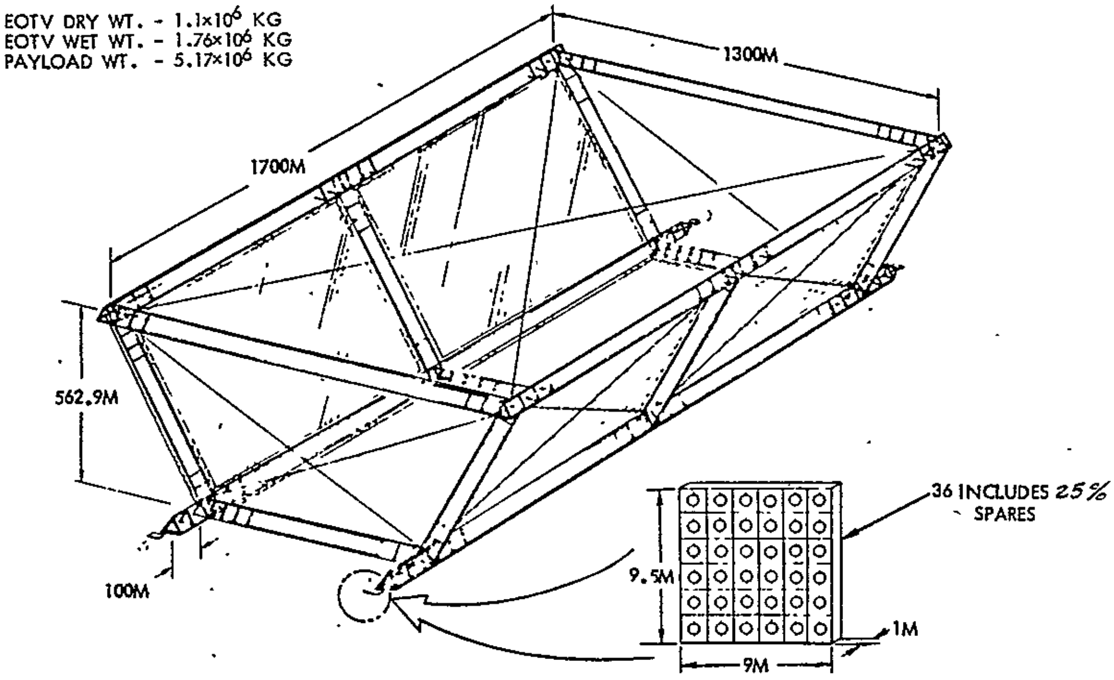


Figure 5.1-3. Selected EOTV Configuration

The solar array weights are consistent with baseline SPS weights criteria. The thruster array weights are dictated by the size/performance of the individual thruster whose performance is fixed by available power and voltage/temperature limitations.

The major element of attitude control system weight, (the power supply) is based on the same sizing criteria as the SPS battery system.

The transfer propellant weight of 666,660 kg is the maximum that can be consumed by the thrusters during the assumed transit time of 120 days up (100 days thrusting) and the resultant return trip time of approximately 30 days (22 days thrusting).

The EOTV dry weight (including growth) is approximately 1.09×10^6 kg and has a payload delivery capability to GEO of 5.17×10^6 kg with a 10% return payload capability to LEO.

The estimated cost of $4.72/kg-payload reflects propellant costs only (delivered to LEO).

5.2 ELECTRIC ORBITAL TRANSFER VEHICLE TRADE STUDIES

Several trade studies were conducted with the objective of achieving a ear cost-optimum EOTV configuration. In addition, parametric sizing data ere generated for thrusters, thruster arrays, conductors, and overall EOTV izing. These data are contained in Appendix C. The results of selected rade studies are summarized herein.

.2.1 SOLAR ARRAY VOLTAGE, GRID TEMPERATURE, NUMBERS OF THRUSTERS

The effects of lowering the total solar array voltage from the baseline f 8300 volts to 5500 volts was evaluated and the results were found to be egligible. The thruster diameter increased to 120 cm and the grid tempera-ure was lowered to 1500°K. Although the thruster array weight increased pproximately 2.5 times the total impact on EOTV inert weight is negligible. n addition the added array weight could be offset by a reduction in conductor nsulation weight. A lower total voltage would appear to be advantageous only f the power conditioning weight would be effected significantly which present ata indicates would not be the case.

Similarly, the number of thrusters in the baseline was reduced by 50%, hus doubling the unit beam current and thrust. The thruster diameter in-reases to 108 cm with no significant change in thruster array weight. The igher thrust appears to be disadvantageous from the standpoint of ACS re-uirements (i.e., dedicated lower thrust units might be required to satisfy inimum ACS demands).

Three EOTV configurations reflecting changes of the type described and lso trip time are summarized in Table 5.2-1. As may be seen the relative ropellant costs between configuration 11A and 11B show an increase with a ecrease in trip time from the baseline. Configuration 12 also shows an in-rease in cost with increased numbers of thrusters with lower accelerating oltage. Although configuration 11A appears to be more efficient than the aseline, it is noted that only 10% spare thrusters and a 15% weight growth as allowed in these configurations. When these corrections are made, all hree configurations exceed the baseline selection.

.2.2 POWER DISTRIBUTION AND CONTROL WEIGHT

A simplified block diagram, Figure 5.2-1, illustrates the EOTV power dis-ribution interface for the solar photovoltaic concept. The distribution sub-ystem consists of interties, main feeders, summing bus, tie bar, switch gears, nd dc/dc converters. The solar arrays feed the load buses with a direct nergy transfer. Provisions are included to switch power from any bus to any hruster location. The basic voltages supplied are +2000 V dc and -6300 V dc. ndividual power supplies will be included as required at the thrusters to upply other voltages.

Figure 5.2-2 shows the power distribution and control weight comparisons or several EOTV configurations studied. A solar array voltage output of 080 V dc was selected as the upper limit for power generation to stay within olerable plasma power losses for low earth orbit operations. The lowest weight

Table 5.2-1. EOTV Configuration Trades

CONFIGURATION	11A	11B	12
THRUSTER DATA			
ACCELERATING VOLTAGE, V	2000	2000	1268
SPECIFIC IMPULSE, SEC	8213	8213	6540
DIAMETER, CM	127	127	127
GRID SET TEMP., °K	1300	1300	1300
NO. (INCLUDING 10% SPARES)	116	116	180
TRIP TIME, DAYS			
LEO-GEO	100	80	100
GEO-LEO	22.3	20	20.9
PROPELLANT, KG	(659,739)	(540,766)	(1,009,000)
LEO-GEO	532,444	425,952	824,636
GEO-LEO	118,712	107,186	171,930
ACS	8,583	7,628	12,434
EOTV WEIGHTS, KG			
SOLAR ARRAY & COND.	588,196	588,196	588,196
THRUSTER ARRAY	112,586	96,469	200,386
POWER SUPPLY	60,413	67,029	54,524
TOTAL DRY WT. (INCL. 15% GROWTH)	875,374	864,448	969,578
*PAYLOAD WT., KG	5,456,250	4,186,384	6,758,069
**PROPELLANT COST (DELIVERED) ($/KG PAYLOAD)	4.51	4.81	5.57

*Based on 10% down payload capability.
**Rockwell reference configuration—$4.72

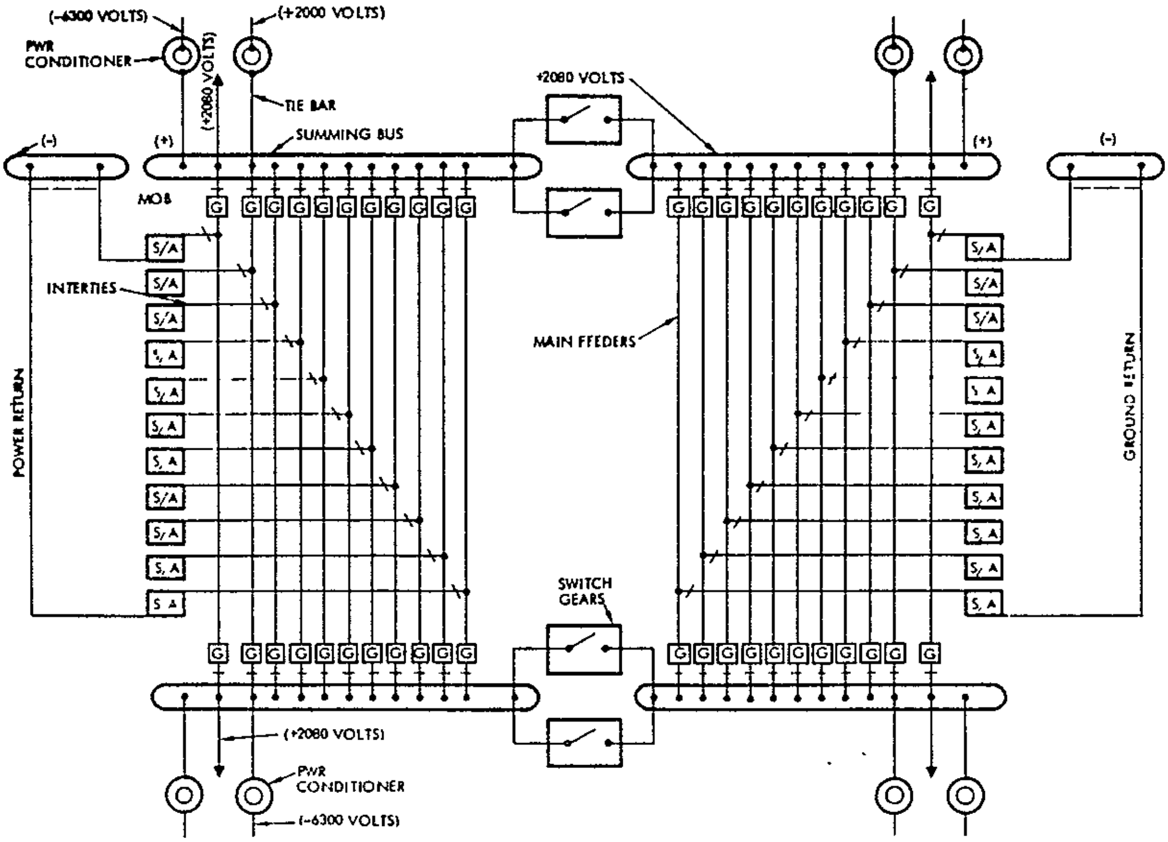
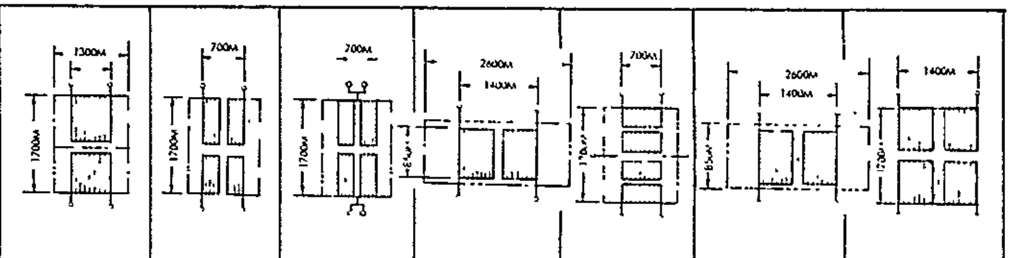


Figure 5.2-1. EOTV Power Distribution Simplified Block Diagram

EOTV CONFIGURATION							
CELL MAT'L	GaAs →						SILICON
CR	2 →						1
TRANS. VOLTAGE	+2080V →					-6300V	-6300V
PANEL CONFIG.	SPLIT 2 PANELS	SPLIT 4 PANELS	SPLIT 4 PANELS	SPLIT 2 PANELS	SPLIT 4 PANELS	SPLIT 2 PANELS	SPLIT 4 PANELS
WEIGHTS (10^6 KG)							
INTERTIES	221,940	67,260	177,550	177,550	177,500	19,540	19,540
MAIN FEEDERS	144,520	119,230	57,810	57,810	57,810	22,850	83,740
SUMMING BUS	177,550	44,390	177,550	177,550	55,490	68,800	68,800
TIE BARS	24,660	24,660	24,660	24,660	24,660	8,140	8,140
SW GEARS	2,290	2,290	2,290	2,290	2,290	9,460	7,310
POWER CONDIT.	-	-	-	-	-	75,490	75,490
INSUL.	4,400	4,400	4,400	4,400	4,400	4,400	16,150
SEC. STRUCT.	57,540	26,220	44,200	44,430	3,220	20,870	27,920
TOTAL	632,900	288,440	486,180	488,690	354,420	229,550	307,090

NOTE CORRECTION FACTORS

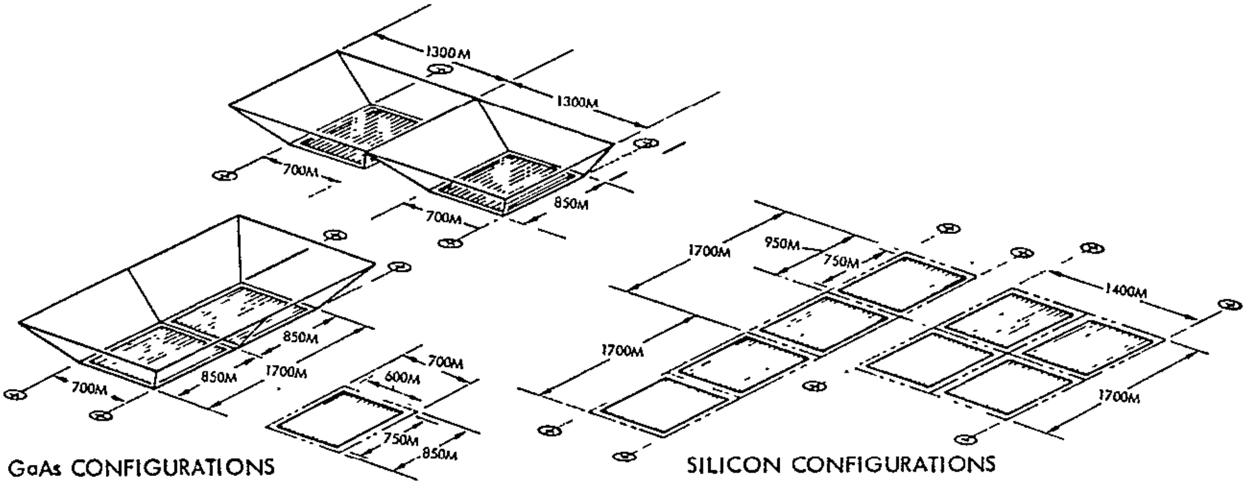
[illegible]

Figure 5.2-2. EOTV Power Distribution and Control Weight Comparisons

concept results in a power distribution subsystem weight of 288,440 kg. This configuration is a direct energy transfer to the engines. This weight was calculated at a distribution (line loss) efficiency of 94% (i.e., 6% line loss). The weight calculations ranged up to 632,900 kg dependent upon specific configuration details. A negative voltage system was compared to show impact of higher voltage. A negative 6300 volts was selected for this purpose since this is the second voltage requirement of the EOTV thruster system. This concept requires power conditioning at the thrusters to provide the +2000 volt inputs required. The silicon system was compared for the lowest weight approach and results in a weight penalty of ~33% (307,090 kg vs 229,550 kg). The +2080 volt concept is the recommended approach since it does not require major power conditioning (i.e., direct power transfer) and the -6300 volt system is susceptable to arcing problems in the plasma environment.

5.2.3 GALLIUM ARSENIDE VERSUS SILICON SOLAR CELLS

A comparison was made of the EOTV requirements using GaAs and silicon solar cells. The configurations used in the comparison are shown in Figure 5.2-3 with a tabulation of solar array parameters and values. The silicon solar array weights are 725,904 kg compared to 263,511 kg driven by higher specific weight (.426 kg/m^2 vs .252 kg/m^2) and requirement for large area (1,704,242 m^2 vs 886,950 m^2). The impact of reflector weight on the GaAs configuration is negligible.



PARAMETER	GaAs	SILICON
SOLAR INPUT	1319.5 W/M^2	1319.5 W/M^2
ENERGY ONTO CELLS	2414.7 (CR ≈ 1.83)	1319.5 (CR – 1)
η (T)	424.98 (17.6%)	221.17 (16.74%)
DESIGN FACTOR	278.24 (.89)	196.85 (.89)
POWER OUTPUT (ARRAY)(1)	335.48 MEGAWATTS	335.48 MEGAWATTS
AREA REQM'T	886,950 M^2	1,704,242 M^2
ARRAY AREA	900,000 M^2	1,800,000 M^2
ARRAY WEIGHT (KG)	223,511 (.252 KG/M^2)	725,904 (.426 KG/M^2)
REFLECTOR AREA	2,210,000 M^2	-
REFLECTOR WEIGHT	40,000 KG	-
SUBTOTAL	263,511 KG	725,904 KG

NOTE:
(1) NO SPACE DEGRADATION ALLOWANCES

Figure 5.2-3. EOTV Solar Array Comparisons (GaAs versus Si Solar Cells)

Estimated weights and performance for two representative EOTV configurations are given in Table 5.2-2. The increased solar array weight for the silicon solar cell configuration results in a 14% reduction in payload capability and a longer return trip time. Because of these factors and the unknowns in annealing of the silicon cells in space, the gallium arsenide approach is more desirable.

5.2.4 ATTITUDE CONTROL SYSTEM

The selection of an "all-electric" propulsion system was based on prior studies which indicated a prohibitive propellant requirement for chemical thrusters, even when used in the ACS mode only.

The Rockwell EOTV concept utilizes attitude hold only during the shadowed period of orbit. Electric thrusters powered by storage batteries are used for ACS during this period. Worst case ACS requirements during Earth shadow periods were evaluated in order to determine battery power and thruster requirements; the objective being to minimize ACS requirements.

Thruster redundancy in each thruster array was also considered to preclude thruster exhaust impingement on the solar array.

Table 5.2-2. GaAlAs and Silicon Powered EOTV Weight Comparison (kg)

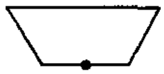
ELEMENT	GaAlAs	SILICON
SOLAR ARRAY	493,056	1,032,991
THRUSTER ARRAY	104,046	113,355
ATTITUDE CONTROL SYSTEM	50,471	50,576
EOTV INERT WEIGHT	647,573	1,196,922
GROWTH - 25%	161,893	299,231
TOTAL EOTV INERT WT.	809,466	1,496,153
DELTA V PROPELLANT	540,420	593,170
ACS PROPELLANT	6,874	7,471
TOTAL EOTV LOADED WT.	1,356,760	2,096,794
PAYLOAD WEIGHT	5,310,568	4,570,534
LEO DEPARTURE WT.	6,667,328	6,667,328
TRIP TIME (UP/DOWN)	120/16	120/28

EOTV dry and loaded inertia data, Table 5.2-3, were generated for two payload stowage options. These data were generated for comparison with MSFC data and for ACS thruster requirement determination for the reference EOTV configuration described earlier.

Table 5.2-3. Preliminary Moments of Inertia

• EOTV REFERENCE CONFIGURATION

	MOMENTS OF INERTIA, KG-M^2 X 10''		
	I_X	I_Y	I_Z
INERT EOTV WITHOUT PAYLOAD & PROPELLANT	3.0	.51	3.5
EOTV FULLY LOADED			
• PAYLOAD CONCENTRATED ON EACH SIDE AT $\ell/2$	6.94	4.43	11.37
• PAYLOAD DISTRIBUTED ABOUT C.M.	6.96	1.21	8.14



The approach to sizing ACS power requirements was to integrate the overall thruster requirements over the earth shadow period rather than taking maximum values which lead to ultra conservative design requirements, Figure 5.2-4.

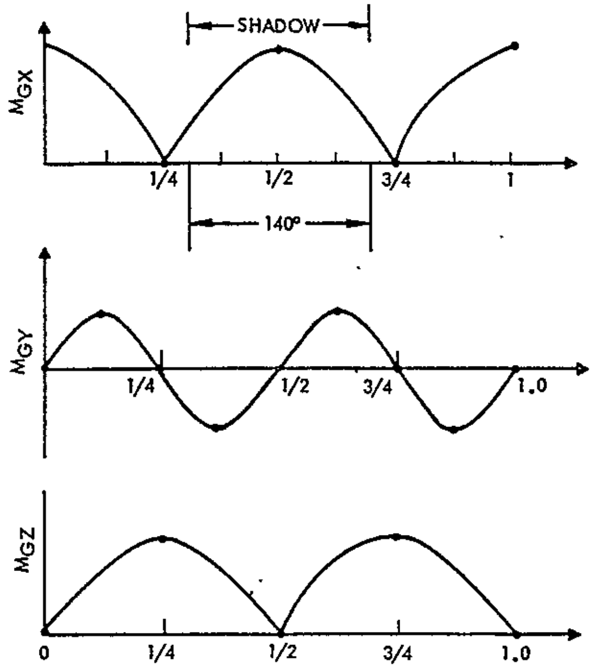


Figure 5.2-4. Typical Gravity Gradient Torque Curves

Based upon average gravity gradient torques, the number of thrusters required were determined for two vehicle orientations, three beta angles, and two payload locations. The calculated thruster requirements are summarized in Table 5.2-4.

*Table 5.2-4. Thruster Requirements in Shadow**

● LONG AXIS INITIALLY POP

	AVERAGE NO. THRUSTERS	
BETA (DEG)	PAYLOAD DISTRIBUTED ABOUT C.M.	PAYLOAD CONCENTRATED ON EACH SIDE AT L/2
10	8.6	23.0
30	16.2	19.9
45	18.2	17.7

● LONG AXIS INITIALLY IN ORBIT PLANE

10	15.2	15.6
30	16.0	20.9
45	19.9	23.3

*BASED ON 487 KM ALTITUDE
AVERAGE SHADOW PERIOD 36.7 MIN.

Although the number of thrusters required to satisfy all ACS requirements are greater than previously estimated (i.e., 16 in lieu of 4, nominal), other options are available to further reduce ACS requirements. These include EOTV configuration changes, off-set solar pointing, attitude maneuvers to lower gravity gradient torque during shadow periods, etc.

Potential methods of reducing thruster requirements by configuration changes are illustrated in Figure 5.2-5. Many other configuration options also exist.

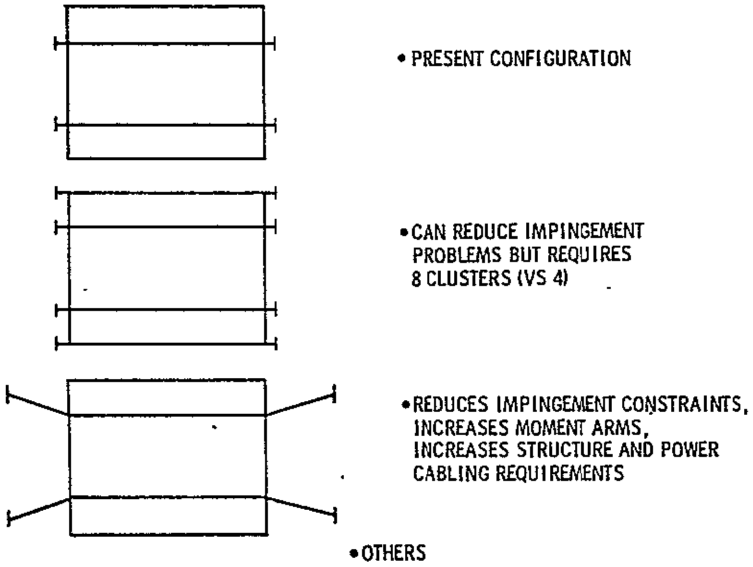


Figure 5.2-5. Alternative Thruster Configurations

Another method of providing reduced ACS thruster requirements is to roll the vehicle relative to the solar inertial axis. Although some loss in solar blanket efficiency might occur, the reduction in numbers of thrusters may off-set those losses. The effect on solar blanket efficiency with off-set pointing is shown in Figure 5.2-6.

Although alternate configurations are recommended for future evaluation, the current concepts are adequate for this phase of program definition. Table 5.2-5 summarizes the current ACS trade study results.

5.2.5 TRIP-TIME OPTIMIZATION ANALYSIS

An analysis was performed to define an approach for comparing EOTV's having differing LEO-to-GEO trip times on a $/kg-of-payload basis. Although the number of EOTV variables assessed are limited, the basic study result is believed to be valid. Later studies might include variations and refinements on any major parameter (i.e., electric engine size, thrust level and specific impulses). (EOTV and COTV are used synonymously in this section of the report.)

The basic equations used are presented in Table 5.2-6 to give the reader sufficient data to check succeeding calculations if desired. Note that the ΔV of 4508 m/sec is applicable to an equatorial departure orbit at 300 nautical miles. For departures from inclined orbits, the Edelbaum equations are suggested. The calculation of initial EOTV mass in LEO, M_i, was modified slightly to account for ACS propellant use.

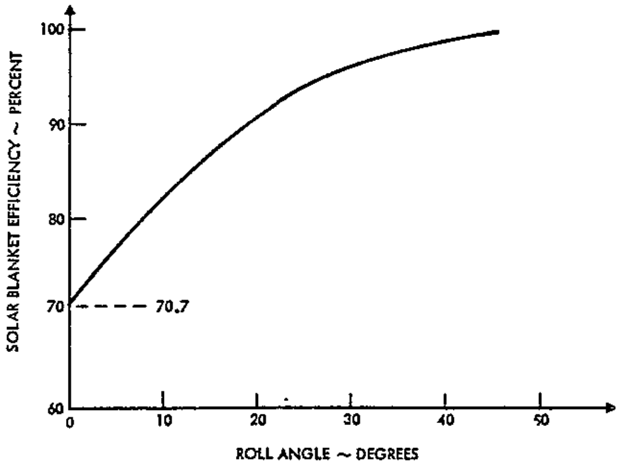


Figure 5.2-6. Partial Solar Pointing

Table 5.2-5. ACS Trade Study Results

- LONG AXIS INITIALLY POP WITH PAYLOAD DISTRIBUTED ABOUT C.M. IS THE PREFERRED ORIENTATION
- FOR ATTITUDE HOLD IN SHADOW PERIOD, THE AVERAGE NUMBER OF THRUSTERS IS 8.6 FOR LOW β AND 18.2 FOR WORST-CASE β.
- PRESENT THRUSTER CONFIGURATION OF FOUR CLUSTERS REQUIRES 36 THRUSTERS PER CORNER INCLUDING 20% SPARING; COSINE LOSSES IN VERTICAL PLANE DUE TO 15° PLUME CONSTRAINT (APPROX. WORST CASE COSINE LOSS = 12%)
- PARTIAL SOLAR POINTING ATTRACTIVE FOR HIGH β ORBITS
- CONSTRAIN MISSION TO REDUCE MAXIMUM β (AND CONTROL REQUIREMENTS) APPEARS FEASIBLE; REQUIRES FURTHER MISSION ANALYSIS TO DEFINE MAXIMUM β
- INVESTIGATE ALTERNATIVE THRUSTER CLUSTERING CONFIGURATIONS

By "freezing" the electric EOTV size and non-propulsive subsystems, trip time variations are introduced by varying the payload to change the thrust-to-weight relationships. From computer data, the following LEO-to-GEO trip times and thruster burn times were established.

LEO-TO-GEO TRANSFER

Total Trip Times (Days)	Thruster Burn Times (Days)
30	20.8
60	47.0
90	73.2
120	99.4
150	125.7
180	151.8

Table 5.2-6. Basic Equations Used in Analysis

THRUSTER PROPELLANT FLOW RATE

$$\dot{m} = \frac{T}{g I_{sp}}$$

$$\dot{m} = \frac{13.02}{(9.8065)(13,000)}$$

$$\dot{m} = 10.213 \times 10^{-5}$$

ELECTRIC COTV GROSS WEIGHT IN LEO

M_p = MASS OF PROPELLANT (LEO-TO-GEO)

M_f = MASS REMAINING IN GEO AFTER EXPENDING PROPELLANT M_p

M_i = INITIAL COTV MASS IN LEO

$$M_p = M_f\left(e^{\frac{\Delta V}{g I_{sp}}} - 1\right) \quad \text{WHERE } \Delta V = 4,508 \text{ m/sec (NO PLANE CHANGE)}$$

$$M_p = 0.03606\ M_f$$

$$M_i = M_p + M_f = 28.73\ M_p$$

With these data, one can compute the LEO-to-GEO argon propellant requirements and multiply by 0.2 to estimate tankage and line masses needed to calculate GEO-to-LEO propulsive requirements. The return trip-time results which correlate with the above LEO-to-GEO transfers are as follows:

GEO-TO-LEO TRANSFER

Total Trip Times (Days)	Thruster Burn Times (Days)
21.1	14.0
21.3	14.2
21.6	14.4
21.8	14.6
22.2	14.9
22.4	15.1

The payload mass capabilities for the various EOTV trip times are summarized in Table 5.2-7.

Minor adjustments were made to the gross weights (i.e., from ~10,000 to ~20,000 kg) to account for expended ACS propellants during the transfers. The weight growth margins are reflected in the propellant mass calculations since they had been added to the non-variable EOTV masses.

The assumptions affecting EOTV trip-time cost are summarized in Table 5.2-8. The numbers shown for each assumption are not "hard" in the sense of being fully justifiable and the reader is encouraged to introduce his own where discrepancies may appear. The EOTV operations cost variable is introduced to account for the slightly higher degree of activity at the LEO base for the shorter trip time concepts, and is not to be taken as the cost of LEO base operations. EOTV turnaround times were based on total trip times plus assumed delays per trip and loading/unloading operations times.

Table 5.2-7. Sizing the EOTV - Payload Mass Capabilities

NON-VARIABLE COTV MASSES (KG)	
STRUCTURES AND SUPPORTS	252,000
SOLAR BLANKETS	226,800
REFLECTORS	25,200
THRUSTER MODULES	32,400
ROTARY JOINT	6,540
PWR DISTRIB. & CONTROL	46,500
IMS	11,400
ACS HARDWARE (ALL)	10,800
ACS PROPELLANT - LEO	10,800
	622,440
+30% GROWTH MARGIN	186,730
	809,170

	LEO-TO-GEO TRIP TIMES					
TRIP-TIME VARIABLE MASSES (KG)	30 DAYS	60 DAYS	90 DAYS	120 DAYS	150 DAYS	180 DAYS
LEO-TO-GEO ARGON PROPELLANT	42,210	95,390	148,560	201,740	255,110	308,080
GEO-TO-LEO ARGON PROPELLANT	28,460	28,880	29,300	29,720	30,140	30,560
ARGON TANKAGE/LINES	14,130	24,860	35,570	46,290	57,050	67,730
ACS FLIGHT PROPELLANT	5,400	10,800	16,200	21,600	27,000	32,400
SUBTOTAL	90,200	159,930	229,630	299,350	369,300	438,770
NON-VARIABLE COTV MASS	809,170	809,170	809,170	809,170	809,170	809,170
ELECTRIC COTV MASS	899,370	969,100	1,038,800	1,108,520	1,178,470	1,247,940
GW IN LEO	1,221,740	2,751,620	4,281,230	5,811,110	7,346,460	8,870,310
PAYLOAD CAPABILITY	322,370	1,782,520	3,242,430	4,702,590	6,167,990	7,622,370

Table 5.2-8. Assumptions Affecting EOTV Trip-Time Cost Comparisons

HLLV PAYLOAD COSTS TO LEO = \$30/KG PAYLOAD
HLLV PAYLOAD INTEGRATION PENALTY OF 10%
HLLV ADDITIONAL PAYLOAD INTEGRATION PENALTY OF 20% FOR PROPELLANT CONTAINMENT
EOTV RESUPPLY PROPELLANT COSTS AVERAGE \$1/KG
EOTV THRUSTER GRIDS REPLACED AFTER 4,000 HOURS BURN TIME
EOTV THRUSTER GRIDS WEIGH 4 KG/GRID AND COST \$500/GRID
EOTV "LIFE" IS DEFINED AS 100% REPLACEABLE AND IS BASED ON EOTV FLIGHT TIMES USING 360-DAY YEARS
EOTV OPERATIONS COST VARIABLE IS \$200,000 FOR EACH FLIGHT TURNAROUND
EOTV INITIAL ON-ORBIT COST IS $\$150\times10^6$
SATELLITE INVESTMENT AT $\$5\times10^9$
DISCOUNT RATE IS 7.5%
EOTV TURNAROUND TIMES AS LISTED:

LEO-TO-GEO TRIP TIMES	TURNAROUND TIMES
30 DAYS	57.6 DAYS
60 DAYS	94.1 DAYS
90 DAYS	130 6 DAYS
120 DAYS	160.8 DAYS
150 DAYS	203 9 DAYS
180 DAYS	240.4 DAYS

An example calculation is shown in Figure 5.2-7 for the 180-day LEO-to-GEO trip time case with its up payload capability of 7,622,370 kg to demonstrate how costs are apportioned on a \$/kg payload basis. The results for all LEO-to-GEO trip-time cases are also presented and summed. Note that no apportionment has yet been made for the initial/replacement cost of the vehicle. This will be considered in the material to follow.

EXAMPLE CALCULATION 180-DAY LEO-TO-GEO TRIP TIME CASE - PAYLOAD = 7,622,370

RESUPPLY:

HLLV OPERATIONS COSTS

- ALL PROPELLANTS (385,080 KG) × 1.1 (PAYLOAD INTEGRATION) × 1.2 (CONTAINMENT) × $30/KG (LAUNCH TO LEO) = $15,249,170
- GRID MASS REPLACEMENTS (4 KG/GRID × 270 GRIDS × 1.3 GROWTH) × (166.9 BURN DAYS × 24 HRS/DAY ÷ 4,000 HRS)× 1.1 (P/L) × $30/KG = 46,400

$15,295,570

= $2.007/KG PL

MATERIALS/PROPELLANT COSTS

- PROPELLANT MASS (385,080) × $1/KG = $385,080
- THRUSTER MODULE REPLACEMENT GRIDS = 135,190

$520,270

= $0.068/KG PL

SPACE OPERATIONS:

TURNAROUND COSTS

- AT $200,000 PER FLIGHT, DIVIDED BY PAYLOAD = $0.026/KG PL

ALL TRIP-TIME CASES

	LEO-TO-GEO TRIP TIMES					
	30 DAYS	60 DAYS	90 DAYS	120 DAYS	150 DAYS	180 DAYS
RESUPPLY - HLLV OPERATIONS	$11.099	$3.322	$2.550	$2.255	$2.101	$2.007
- MATERIALS/PROP.	$ 0.367	$0.111	$0.086	$0.076	$0.071	$0.068
SPACE OPERATIONS	$ 0.620	$0.112	$0.062	$0.043	$0.032	$0.026
TOTALS	$12.086	$3.545	$2.698	$2.374	$2.204	$2.101

Figure 5.2-7. Apportioned Resupply and Operations Cost/kg of EOTV Payload

The definition of vehicle "life" was stated in the assumptions as requiring 100% replaceability. An example is given here assuming that vehicle life is limited to 5 years of flight time. For the 180-day LEO-to-GEO trip-time case, 5 years times 360 days/year divided by 202.4 flight days per trip yields an average vehicle life of 8.8933 flights. From this data, program buys can be computed and are shown in Figure 5.2-8. Also from the data provided, fleet size calculations can be made for each trip-time case. Note that a 10-year "life" would halve the program buy requirements but would not alter the fleet size demands.

The investment streams for capital purchase of the EOTV's is developed from consideration of average vehicle cost, fleet size, total program buy, and vehicle life. For this analysis it was assumed that the average vehicle cost - in place - would be $\$150 \times 10^6$ regardless of the total numbers purchased. The example shown in Figure 5.2-9 is for a 5-year vehicle "life" and assumes that the initial fleet production investment was begun six years prior to the first SPS IOC date. All LEO-to-GEO trip-time cases are shown except the 30-day case which is now recognized as not cost-effective. If the last purchase of 10-year life point was plotted for the 60-day trip-time, it would appear at $9.15 $\overline{B}$ on the ordinate and 18.728 years on the abcissa, but the initial fleet complement investment point would remain unchanged.

EXAMPLE CALCULATION FOR 180-DAY LEO-TO-GEO TRIP TIME

- LIFE OF VEHICLE IS 8.8933 FLIGHTS

  DURING THE VEHICLE LIFE, IT WILL TRANSPORT 8.8933 x 7,622,370 KG = 67,788,020 KG. THE PROGRAM REQUIREMENTS ARE 120 SATELLITES AT 40 x 10^6 KG EACH DIVIDED BY 67,788,020 KG YIELDS THE REQUIRED PROGRAM BUY OF 71 VEHICLES

- ASSUMING THAT A SINGLE SATELLITE MASS OF 40 x 10^6 KG MUST BE DELIVERED DURING A 90-DAY INCREMENT, THEN THE FLEET SIZE REQUIREMENT IS 90 DAYS DIVIDED BY TURNAROUND TIME OF 240 DAYS TIMES THE PAYLOAD = 2,858,390. THIS IS THE EQUIVALENT PAYLOAD DELIVERED BY ONE VEHICLE OVER 90 DAYS. SINCE 40 x 10^6 KG IS REQUIRED, THEN DIVIDE BY THE EQUIVALENT PAYLOAD TO GIVE A FLEET SIZE OF 14 VEHICLES.

RESULTS

		ELECTRIC COTV LEO-TO-GEO TRIP TIMES					
		30 DAYS	60 DAYS	90 DAYS	120 DAYS	150 DAYS	180 DAYS
FLEET SIZES	CALCULATION	79.412	23.462	17.902	15.793	11.692	11 017
	ROUNDED	80	24	18	16	15	14
PROGRAM BUY	CALCULATION	422.703	121.626	91 783	80.110	74.449	70.809
	ROUNDED	123	122	92	81	75	71

Figure 5.2-8. Electric EOTV Fleet Sizes and Program Buys

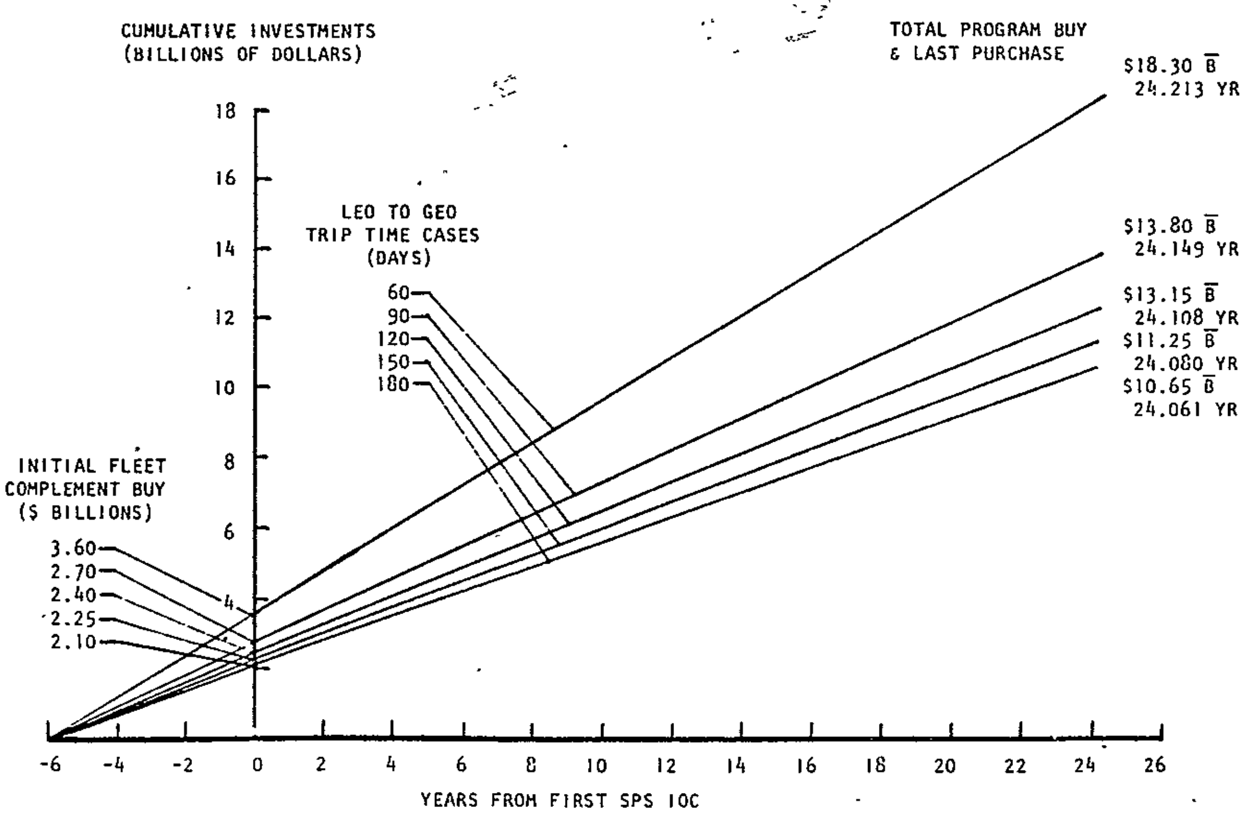


Figure 5.2-9. EOTV Capital Investment Streams

The time-value of money impact on cost comparisons is discussed in Figure 5.2-10 and expressed for all trip-time cases in terms of \$/kg of EOTV payload. The investment dollars were subtracted from the 180-day trip time case and only the Δ differences are tabulated.

THE TIME-VALUE OF MONEY MUST BE CONSIDERED IN THE COST COMPARISONS OF THE ELECTRIC COTV ALTERNATIVES.

(1) SATELLITE CAPITAL INVESTMENT

LEO-TO-GEO TRANSFER TIMES SHOULD BE CONSIDERED AS PERIODS OF TIME DURING WHICH THE INTEREST ON A CAPITAL INVESTMENT (E.G., THE SATELLITE VALUED AT APPROXIMATELY \$5 BILLION) IS LOST. FOR EXAMPLE, THE "INTEREST LOST" FOR A 180-DAY PERIOD AT A 7.5% DISCOUNT RATE IS APPROXIMATELY \$184.1 MILLION. APPORTIONED ON A SATELLITE MASS BASIS EQUATES TO \$4.603/KG.

(2) COTV CAPITAL INVESTMENT

FROM THE PREVIOUS CHART IT IS TO BE NOTED THAT THE SHORTER TRIP-TIME CASES NOT ONLY REQUIRE HIGHER INITIAL INVESTMENTS, BUT ALSO THE INVESTMENT STREAM IS HIGHER. AGAIN, USING A 7.5% DISCOUNT RATE, FUTURE VALUE COMPUTATIONS WERE MADE FOR EACH INVESTMENT STREAM AND THE DIFFERENCES IN \$/KG PAYLOAD (AGAINST THE LOWER COST CASE—E.G., THE 180-DAY TRIP-TIME CASE) WERE ESTABLISHED.

	LEO-TO-GEO TRIP TIMES					
	30 DAYS	60 DAYS	90 DAYS	120 DAYS	150 DAYS	180 DAYS
INTEREST LOST (\$/KG)	0.755	1.516	2.280	3.050	3.824	4.603
COTV INVESTMENT Δ'S (\$/KG)	40.128	5.877	2.403	1.158	0.192	-

Figure 5.2-10. Time-Value of Money Impact on Cost Comparisons

Cost in terms of \$/kg of EOTV payload for resupply, operations, "lost" interest, and investment Δ's were summed and plotted for each of the LEO-to-GEO trip time cases, Figure 5.2-11. The results are presented for EOTV lifetimes of 5, 10 and 15 years illustrating the shift in minimum cost ranges toward the shorter LEO-to-GEO trip-times. These results are encouraging from the standpoint of long-duration transfer palatability. Within reasonable bound and for the performance values and cost assumptions presented, the physical size of the electric EOTV vehicle can be changed without appreciably altering these results.

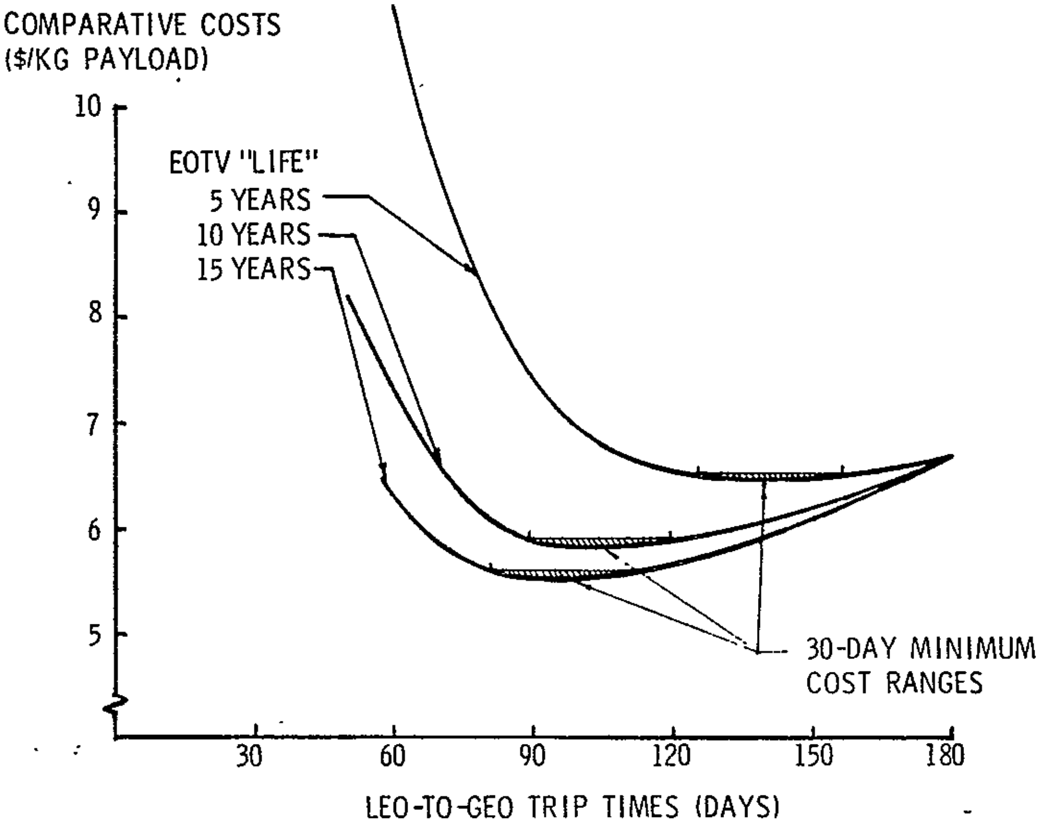


Figure 5.2-11. Electric EOTV Cost Comparisons

6.0 ON-ORBIT MOBILITY SYSTEMS

6.0 ON-ORBIT MOBILITY SYSTEMS

On-orbit mobility systems have been synthesized in terms of application and concept only. On-orbit elements considered here are powered by a chemical (LOX/LH_2) propulsion system. At least three distinct applications have been identified; (1) the need to transfer cargo from the HLLV to the EOTV in LEO and from the EOTV to the SPS construction base in GEO; (2) the need to move materials about the SPS construction base; and (3) the probable need to move men or materials between operational SPS's. Clearly the POTV, used for transfer of personnel from LEO to GEO and return, is too large to satisfy the on-orbit mobility systems requirements. A "free-flyer" teleoperator concept would appear to be a logical solution to the problem. A propulsive element was synthesized to satisfy the cargo transfer application from HLLV-EOTV-SPS base in order to quantify potential on-orbit propellant requirements. This transportation element has been designated intra-orbit transfer vehicle (IOTV).

Sizing of the IOTV was based on a minimum safe separation distance between EOTV and the SPS base of 10 km. It was also assumed that a reasonable transfer time would be in the order of two hours (round trip), which equates to a ΔV requirement on the order of 3 to 5 m/sec. A single advanced space engine (ASE) is employed with a specific impulse of 473 sec (see Section 7.2 for complete engine description). The pertinent IOTV parameters are summarized in Table 6.0-1.

Table 6.0-1. IOTV Weight Summary

SUBSYSTEM	WEIGHT (kg)
ENGINE (1 ASE)	245
PROPELLANT TANKS	15
STRUCTURE AND LINES	15
DOCKING RING	100
ATTITUDE CONTROL	50
OTHER	100
SUBTOTAL	525
GROWTH (10%)	53
TOTAL INERT	578
PROPELLANT	300
TOTAL LOADED	878

7.0 PERSONNEL TRANSFER SYSTEMS

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The personnel transfer systems consist of three basic elements: a personnel launch vehicle (PLV) to transfer construction personnel within an independent personnel module (PM) from earth to LEO; a personnel orbital transfer vehicle (POTV), a single chemical propulsive stage to transfer the PM from LEO to GEO; and the PM, a self-contained crew/personnel module containing all the necessary guidance, navigation, communication, and life support systems for construction crew transfer from earth to LEO.

7.1 PERSONNEL LAUNCH VEHICLE (PLV)

The PLV is a derivative or growth version of the currently defined Space Shuttle Transportation System (STS). The configuration selected as a baseline for SPS studies is representative of various growth options evaluated in Rockwell-funded studies and NASA contracts, NAS8-32015 and NAS8-32395.

The current STS configuration is depicted in Figure 7.1-1, and the growth version (PLV) is shown in Figure 7.1-2. As indicated in the figures, the growth

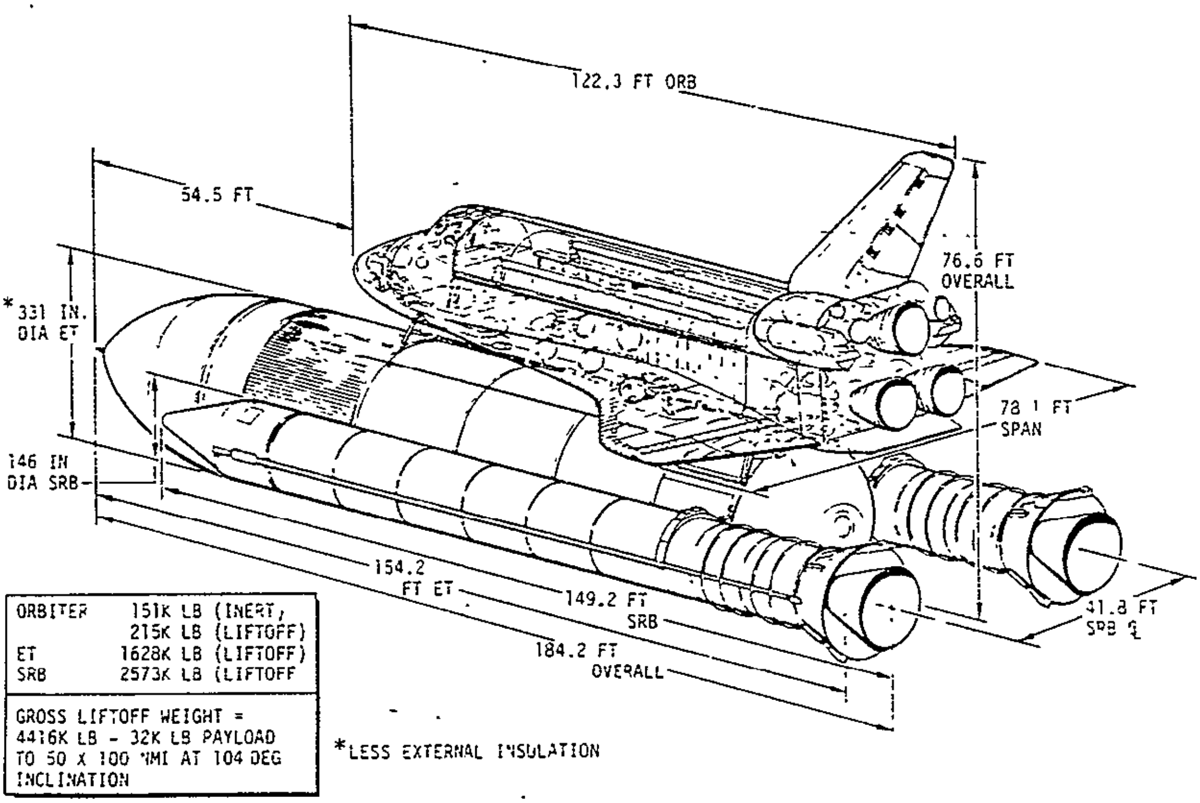


Figure 7.1-1. Baseline Space Shuttle Vehicle

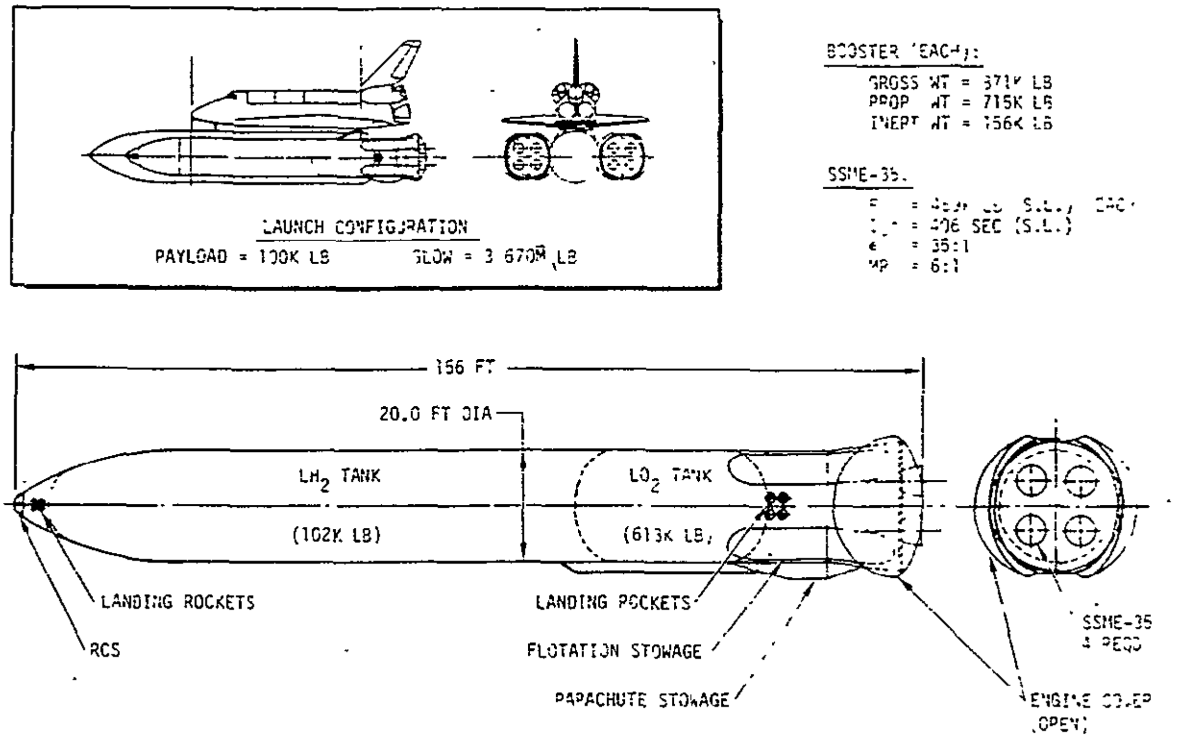


Figure 7.1-2. LO_2/LH_2 SSME Integral Twin Ballistic Booster

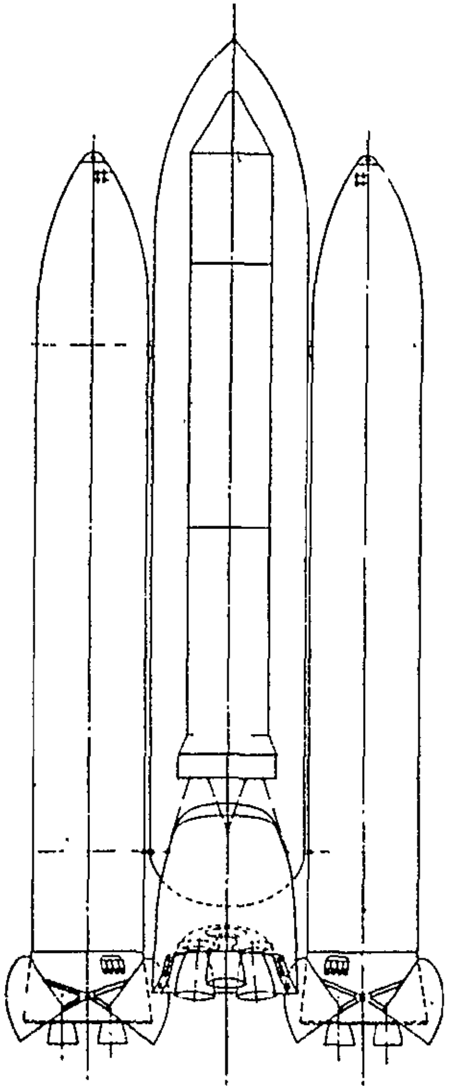
version or PLV is achieved by replacing the existing solid rocket boosters (SRB) with a pair of liquid rocket boosters (LRB). The existing orbiter and external tank are used in their current configuration. The added performance afforded by the LRB increases the orbiter payload capability to the reference STS orbit by approximately 54%, or a total payload capability of 45,350 kg (100,000 lb).

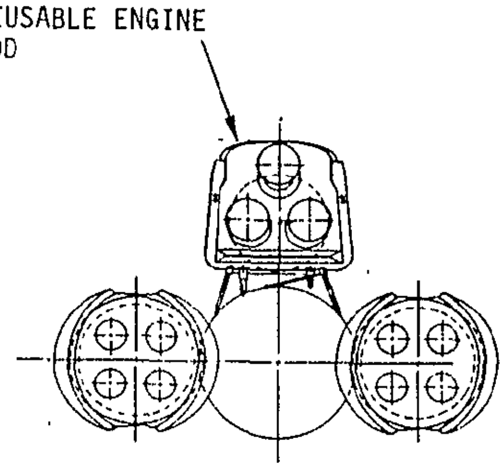
The STS-derived heavy lift launch vehicle (STS-HLLV), employed in the precursor phase of SPS, is derived by replacing the STS orbiter on the PLV with a payload module and a reusable propulsion and avionics module (PAM) to provide the required orbiter functions. The PAM may be recovered ballistically or, preferably, as a down payload for the PLV. These modifications yield an STS-HLLV with a payload capability of approximately 100,000 kg (Figure 7.1-3).

7.1.1 LIQUID ROCKET BOOSTER (LRB)

The LRB illustrated in Figure 7.1-2 has a gross weight of 395,000 kg, made up of 324,000 kg of propellant (278,000 kg of LO_2 and 46,000 kg of LH_2), and 71,000 kg of inert weight. The overall length of the LRB is 47.55 meters with a nominal diameter of 6.1 meters. Four Space Shuttle main engine (SSME) derivatives are employed with a gross thrust of 412.7 newtons (sea level), providing a liftoff thrust-to-weight ratio of 1.335.

Unique design features of the LRB, as compared to an expendable liquid booster system, are presented in Table 7.1-1. The necessity to preclude ice damage to the orbiter requires the LH_2 tank to be located forward since the insulation system, which must be internal to avoid water impact damage, is not compatible with LO_2. In addition, the thickness of insulation required on the LH_2 tank is about two times that required to maintain propellant quality.





LIFTOFF WEIGHTS (10^3 kg)	
PAYLOAD	100.0
EXTENAL TANK	738.3
LRB (2)	790.0
REUSABLE POD	13.7
TOTAL	1642.0

Figure 7.1-3. STS HLLV Configuration

Table 7.1-1. Shuttle LRB Unique Design Features

ORBITER ICE DAMAGE AVOIDANCE	• LH_2 TANK FWD, INSULATED TO PRECLUDE ICE
ENTRY PROVISIONS	• RCS TO ORIENT BOOSTER • CLAMSHELL COVERS FOR ENGINE PROTECTION • HEAT SINK STRUCTURE
WATER LANDING PROVISIONS	• PARACHUTES & RETRO-SUSTAINER ROCKETS • INTERNAL LH_2 TANK INSULATION • RCS FOR WAVE ALIGNMENT • REINFORCED STRUCTURE • AVIONICS TO CONTROL LANDING
WATER PROTECTION PROVISIONS	• CLAMSHELL COVER FOR ENGINE PROTECTION • SEALED STRUCTURE • FLOTATION BAGS FOR ORIENTATION
RECOVERY PROVISIONS	• RADIO BEACON AND LIGHTS • HANDLING HARDPOINTS

Other unique features are the provisions required for entry, water landing, water protection, and recovery. In addition to these supplementary provisions, the structure (unlike that of an expendable system) must act as a heat sink for reentry heat loads, be reinforced to absorb landing loads, and be sealed to prevent sea water contamination.

The basic structure consists of the propellant tank assembly and an engine compartment. The tank assembly is made up of the LH_2 tank and the LO_2 tank, with a common bulkhead similar to the Saturn S-II separating the propellants. The engine compartment comprises a skirt section, thrust structure, launch support structure, heat shield, and movable covers that protect the engines during atmospheric reentry and water recovery. The locations of the landing rockets, the APU, avionics packages, parachutes, the flotation bag, and RCS system are indicated in Figure 7.1-2.

The structural design of a recoverable LRB is governed by five basic load conditions: water impact, high-Q boost, internal tank pressures, prelaunch loads, and maximum thrust.

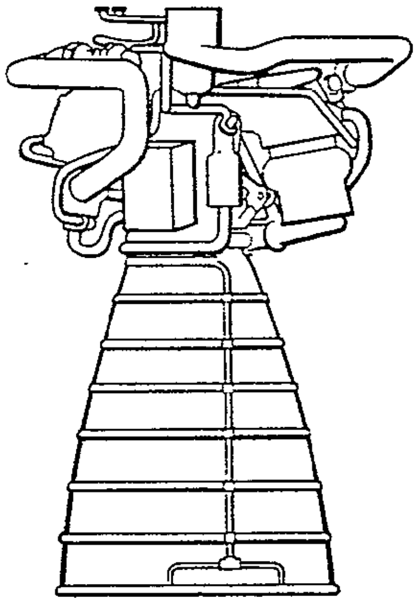
The nose cap primary structure and tank frames are designed to withstand loads due to initial water impact and subsequent water penetration with resultant slap-down loads being reacted by the tank ring frames. Launch maximum aerodynamic pressures (high-Q) loads influence the structural design of the main frames, forward portions of the LH_2 tank, and engine thrust structure. The LH_2 and LO_2 tank walls and domes are structurally sized for maximum internal tank pressures. Equivalent tank wall thickness due to internal pressure exceeds those required by other load conditions. The maximum body bending moment occurs at the aft end of the booster. The design of the aft skirt and frames is governed by prelaunch loads when the boosters are loaded and free-standing on the launch pad. The ET attachments thrust structure are designed by maximum thrust loads at launch.

There are four structural attachments between the ET and each booster. The three aft attachments take lateral shears and bending moments, and the forward attachment takes lateral shears and thrust loads. This four-point interface is statically determinate, so that structural loads are not induced by deformations in the adjacent body. This interface arrangement is the same as that for the baseline Shuttle.

The electrical interface between the booster and ET is accomplished by external cables mounted on one of the aft struts. They are separated at pull-away connectors when the strut is cut. The increased number of wires required for the LRB may increase the number of cables and connectors.

7.1.2 LIQUID ROCKET BOOSTER ENGINE (SSME-35)

The LRB utilizes a derivative of the Space Shuttle main engine (SSME). The only difference between the LRB engines and the SSME is in nozzle expansion ration, 35 in lieu of 77.5 to 1. The SSME-35 and its characteristics are depicted in Figure 7.1-4.



THRUST, LBF	459,000 (S.L.) 503,000 (VAC.)
EXPANSION AREA RATIO	35.1
CHAMBER PRESSURE, PSIA	3230
MIXTURE RATIO	6.0:1
SPECIFIC IMPULSE, SECONDS	406 (S.L.) 445 (VAC.)
ENGINE WEIGHT, LBF	6340
SERVICE LIFE, HOURS	7.5
STARTS	55
ENVELOPE: LENGTH, INCHES	146
DIAMETER, INCHES	
POWERHEAD	105
NOZZLE EXIT	63

Figure 7.1-4. Liquid Rocket Booster Main Engine (SSME-35)

7.1.3 LIQUID ROCKET BOOSTER RECOVERY CONCEPT

After the boosters separate from the orbiter-ET, the engine covers close and the reaction control system (RCS) fires to pitch the boosters over and align them for reentry (Figure 7.1-5). The drogue and then the main chutes deploy to slow descent. Retro motors are fired to minimize landing velocity. Upon splashdown, the chutes release and flotation bags inflate at the aft end to hold the engine area out of the water.

The booster will be commanded by the recovery vessel to start depressurizing (one propellant at a time) upon landing. The recovery vessel will pick up chutes during booster depressurization. After the booster is depressurized, the aft end of the ship is aligned to the booster, the aft gate is lowered, and the compartment is flooded (<30 minutes). A craft is then launched to attach tow lines to the booster, which is then pulled into the ship. The booster is positioned over contour supports or lifted in a crane cradle, rear gate is closed, and the compartment is pumped dry. The booster undergoes washdown and inspection as the ship returns to port. Utilizing this system, a booster can be retrieved and returned to port in 20 to 24 hours maximum (a function of distance and sea state). Booster recovery will be accomplished in waves up to eight feet. The booster recovery system is shown in Figure 7.1-6.

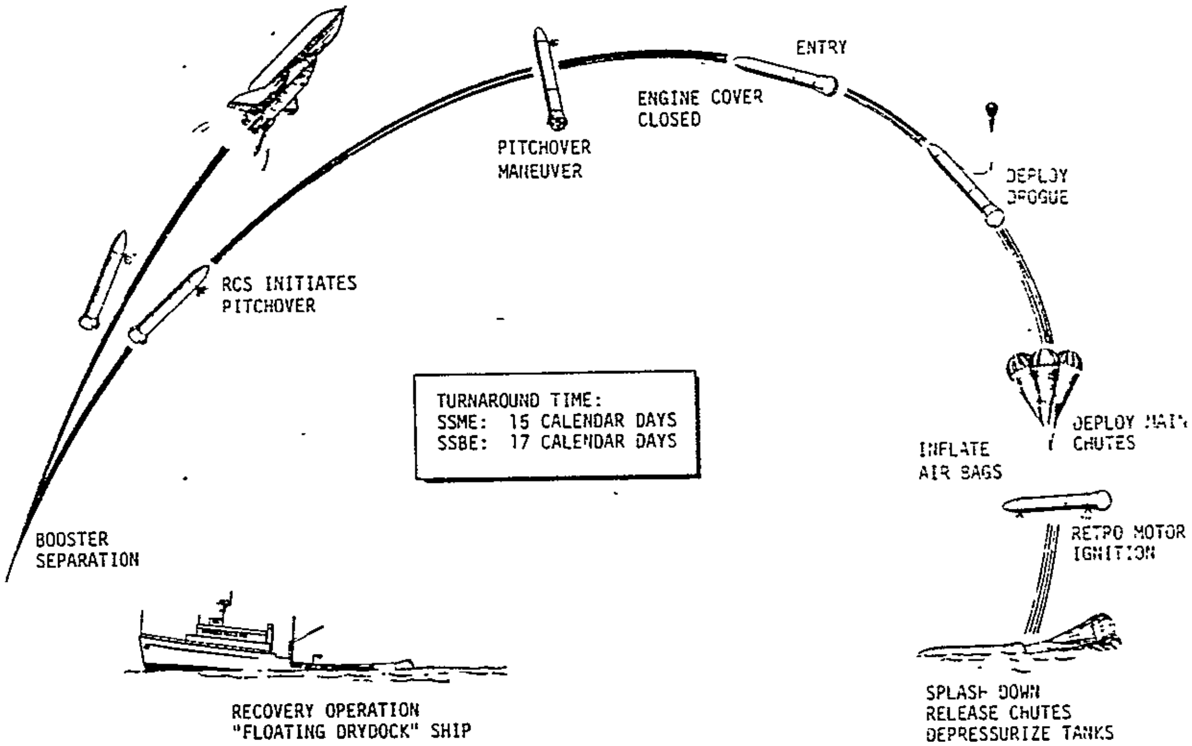
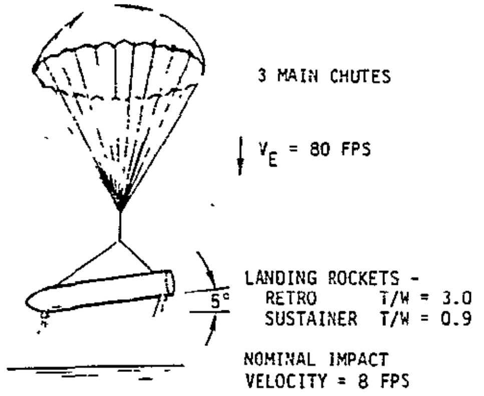


Figure 7.1-5. Integral Booster Recovery Concept



SYSTEM ERRORS

ERROR SOURCE	VALUE
CHUTE VARIATIONS	±4.7 FPS
AIR DENSITY	+3.47 FPS -2.37 FPS
THRUST	±1%
WEIGHT	±2615 LB
ALTIMETER	±2 FT
SIGNAL TIME	±4 FT

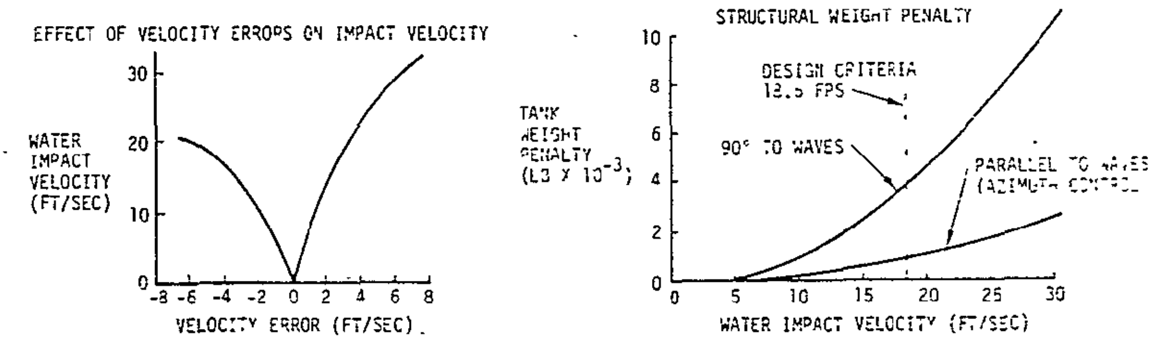


Figure 7.1-6. Booster Recovery System

7.2 PERSONNEL ORBITAL TRANSFER VEHICLE (POTV)

As stated previously, the POTV is the propulsive element used to transfer the personnel module (PM) from LEO to GEO and return. In previous scenarios, the POTV reference concept used two common stage LO_2/LH_2 propulsive elements. The first stage provided an initial delta-V and returned to LEO. The second stage provided the remaining delta-V required for PM ascent to GEO and the requisite delta-V for return of the PM to LEO.

The alternate concept described herein uses a single stage to transport the PM and its crew and passengers to GEO (Figure 7.2-1). After initial delivery of the POTV to LEO by the STS or SPS-HLLV, the propulsive stage is subsequently refueled in LEO (at the LEO station) with sufficient propellants to execute the transfer of the PM to GEO. At GEO, the stage is refueled for a return trip of crew and passengers to LEO. The HLLV delivers crew consumables and POTV propellants to LEO and the EOTV delivers the same items required in GEO. The PM with crew/personnel is delivered to LEO by the PLV.

Although significant propellant savings occur with this approach, as compared to the reference concept, the percentage of total mass is small when compared with satellite construction mass. However, the major impact is realized in the smaller propulsive stage size and the overall reduction in orbital operations requirements.

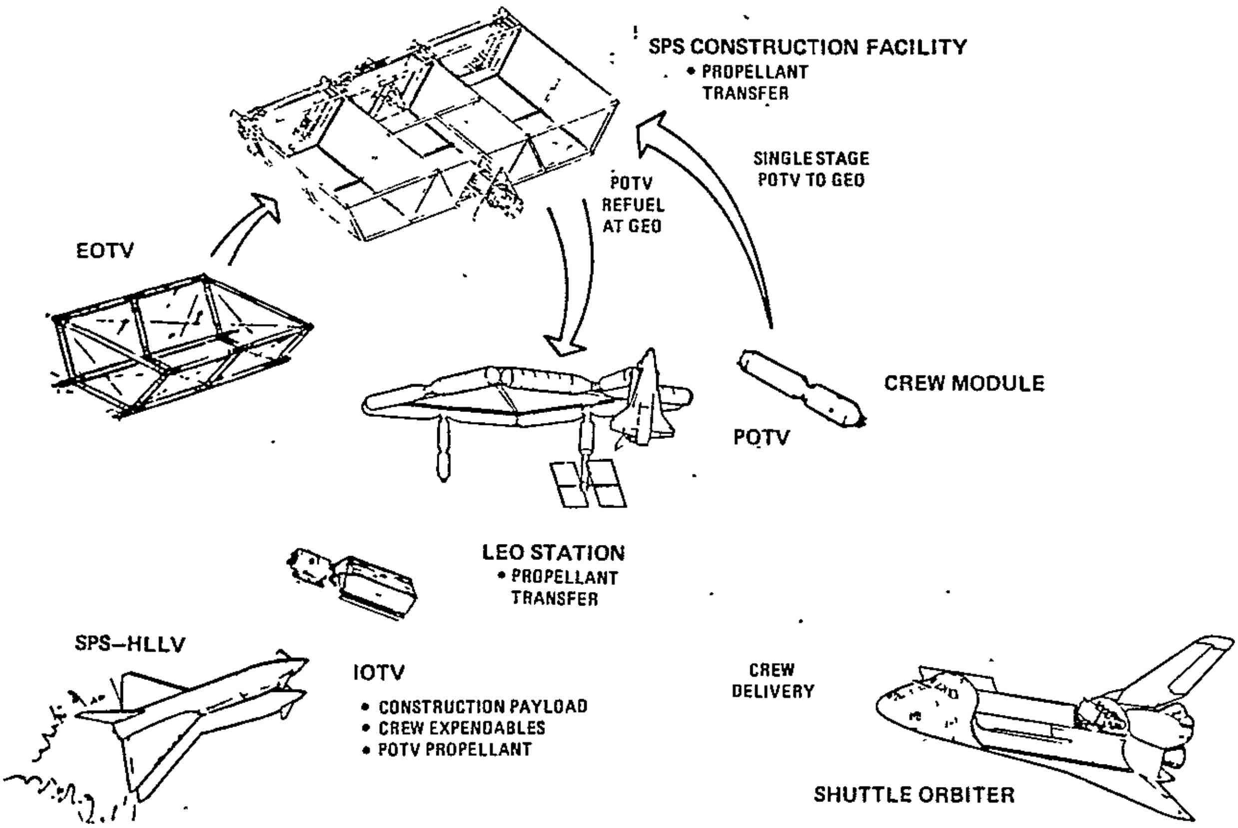
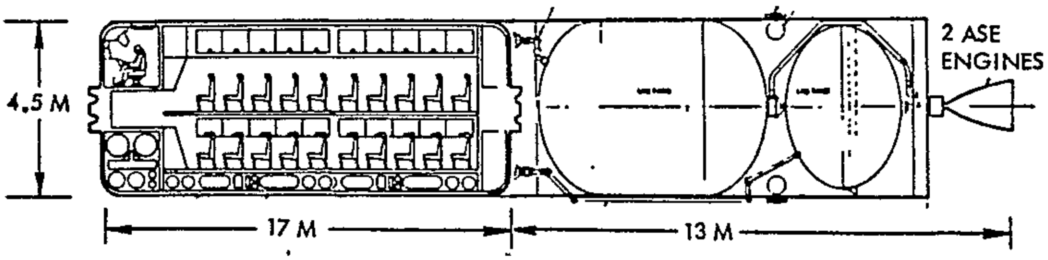


Figure 7.2-1. POTV Operations Scenario

7.2.1 PERSONNEL ORBITAL TRANSFER VEHICLE CONFIGURATION

The recommended POTV configuration is shown in Figure 7.2-2 in the mated configuration with the PM. Either element is capable of delivery from earth to LEO in the PLV; however, subsequent propellant requirements for the POTV will be delivered to LEO by the HLLV because of the lesser $/kg payload cost.

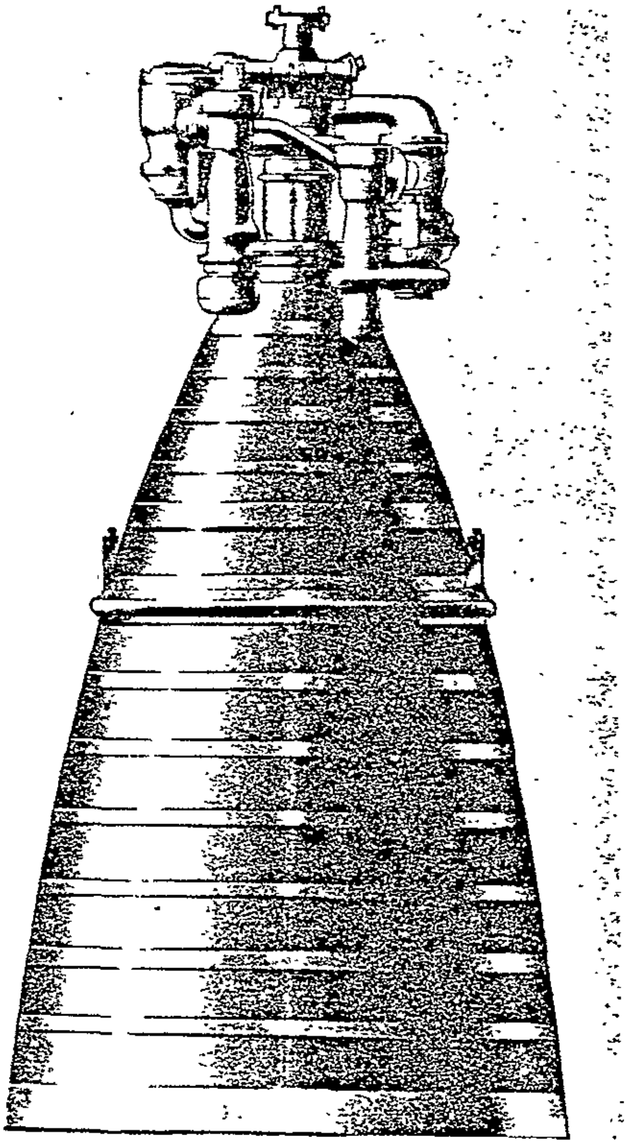


• 60 MAN CREW MODULE	18,000 KG
• SINGLE STAGE OTV (GEO REFUELING)	36,000 KG
• BOTH ELEMENTS CAPABLE OF GROWTH STS LAUNCH	

Figure 7.2-2. Recommended POTV Configuration

Individual propellant tanks are indicated for the LO_2 and LH_2 in this configuration because of uncertainties at this time in specific attitude control requirements. With further study, it may be advantageous to provide a common bulkhead tank as in the case of the Saturn-II, and locate the ACS at the mating station of the POTV and PM, or in the aft engine compartments—space permitting.

The POTV utilizes two advanced space engines (ASE), which are similar in operation to the Space Shuttle main engine (SSME). The engine is of high performance with a staged combustion cycle capable of idle-mode operation. The engine employs autogenous pressurization and low inlet NPSH operation. A two-position nozzle is used to minimize packaging length requirements. The ASE and pertinent parameters are shown in Figure 7.2-3. A current engine weight statement is given in Table 7.2-1.



THRUST (LB)	20,000
CHAMBER PRESSURE (PSIA)	2000
EXPANSION RATIO	400
MIXTURE RATIO	6.0
SPECIFIC IMPULSE (SEC)	473.0
DIAMETER (IN.)	48.5
LENGTH (IN.)	
NOZZLE RETRACTED	50.5
NOZZLE EXTENDED	94.0

Figure 7.2-3. Advanced Space Engine

Table 7.2-1. Current ASE Engine Weight

Fuel boost and main pumps	74.5
Oxidizer boost and main pumps	89.8
Preburner	12.4
Ducting	25.0
Combustion chamber assembly	62.8
Regen. cooled nozzle ($\varepsilon = 175:1$)	58.4
Extendable nozzle and actuators ($\varepsilon = 400:1$)	122.0
Ignition system	6.1
Controls, valves, and actuators	74.0
Heat exchanger	14.0
Total (lb)*	539.0

*Based on major component current measured weights.

Since the POTV concept utilizes an on-orbit maintenance/refueling approach, an on-board system capable of identifying/correcting potential subsystem problems in order to minimize/eliminate on-orbit checkout operations is postulated.

The recommended POTV configuration has a loaded weight of 36,000 kg and an inert weight of 3750 kg. A weight summary is presented in Table 7.2-2.

Although the current POTV configuration provides a suitable concept for identifying and developing other SPS programmatic issues, further trade studies are indicated such as tank configuration and ACS location(s). Also, future studies might be directed toward the evolution of a configuration that would be compatible with potential near-term STS OTV development requirements.

Table 7.2-2. POTV Weight Summary

Subsystem	Weight (kg)
Tank (5)	1,620
Structures and lines	702
Docking ring	100
Engine (2)	490
Attitude control	235
Other	262
Subtotal	3,409
Growth (10%)	341
Total inert	3,750
Propellant	32,750
Total loaded	36,000

7.2.2 PERSONNEL MODULE (PM)

In Volume III, a construction sequence has been developed which requires a crew rotation every 90 days for crew complements in multiples of 60. The PM was synthesized on this basis. A limitation on PM size was established to assure compatibility with the PLV cargo bay dimensions and payload weight capacity (i.e., 4.5 m × 17 m and 45,000 kg).

The PM shown in Figure 7.2-2 is based on parametric scaling data developed in previous studies. It is assumed that a command station is required to monitor and control POTV/PM functions during the flight. This function is provided in the forward section of the PM as shown. Spacing and layout of the PM is comparable to current commercial airline practice. Seating is provided on the basis of one meter, front to rear, and a width of 0.72 meter. PM mass was established on the basis of 110 kg/man (including personal effects) and approximately 190 kg/man for module mass. The PM design has provisions for 60 passengers and two flight crew members.

Several POTV/PM options were evaluated (Figure 7.2-4 and Table 7.2-3). All options utilize a single-stage propulsive element which is fueled in LEO and refueled in GEO for the return trip. The various options considered transfer of both crew and consumables as well as crew only. Transfer of consumables by EOTV was determined to be more cost effective. Another potential option, which is yet to be evaluated, is a 30-man crew module and integral single-stage capable of storage within the PLV cargo bay.

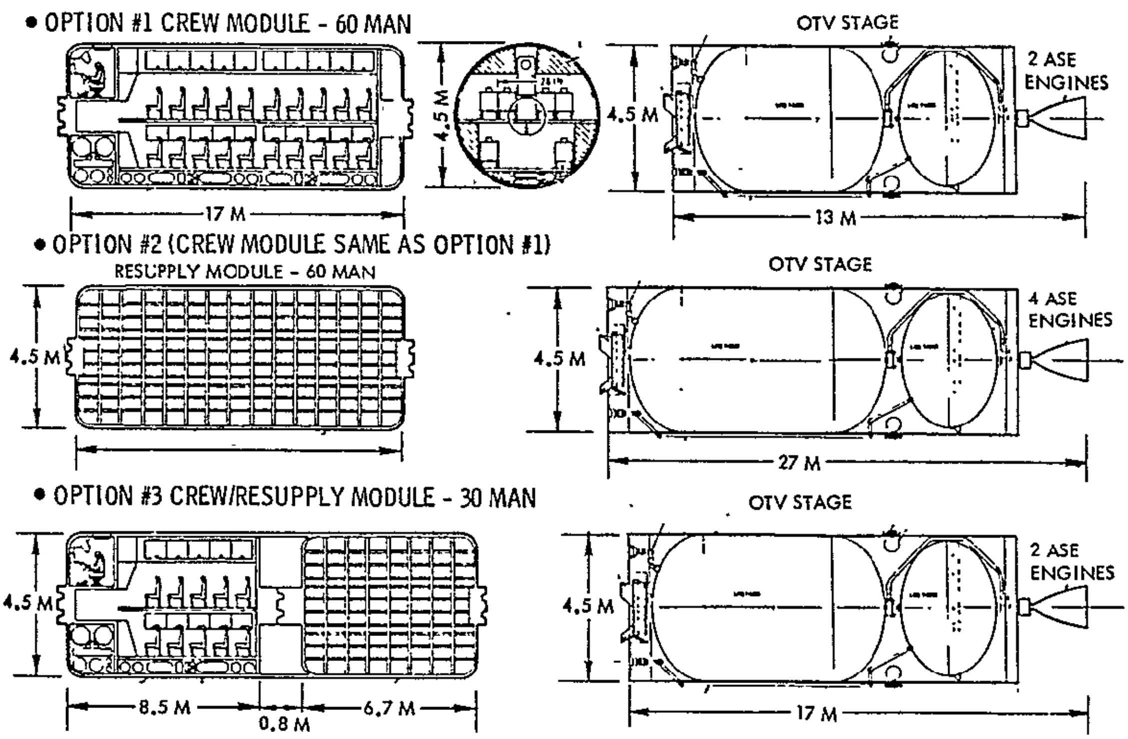


Figure 7.2-4. POTV/PM Configuration Options

Table 7.2-3. POTV/PM Options—Element Mass

	kg
60-man crew module	18,000
60-man resupply module	26,000
Integrated 30-man crew/resupply module	22,000
Option 1 OTV	36,000
Option 2 OTV	87,000
Option 3 OTV	44,000

8.0 COST AND PROGRAMMATICS

8.0 COST AND PROGRAMMATICS

8.0 COST AND PROGRAMMATICS

A summary of transportation costs and schedules are presented. More detailed data and costing assumptions are included in Volume II, Part 2.

Table 8.0-1 presents a summary of the SPS program development cost. The transportation system elements (WBS 1.3) account for approximately 42 percent of the total program development cost. In Table 8.0-2 it may be seen that the PLV and STS-derived HLLV (WBS 1.3.3) contribute almost 26 percent to the transportation development costs.

Table 8.0-3 presents a summary of SPS program average cost, where the transportation cost is approximately 15 percent of that average cost. The PLV and STS-derived HLLV accounts for approximately 22.5% of that cost (Table 8.0-4).

The amortized HLLV cost/kg to LEO can be obtained by multiplying Column 1 (Investment per Satellite) by the number of satellites (60), and adding the product of Column 4 (Total Operation) and the number of satellites (60) and the number of satellite years (30); then divide that quantity by the product of total number of HLLV flights from Table 3.0-3 (22,811) and the HLLV payload (0.231×10^6 kg).

$$\frac{(C_1\times60) + (C_4\times60\times30)}{N\times PL} = \text{HLLV } \$/\text{kg}$$

The results of that calculation yields a payload cost to LEO of \$62/kg (\$28/lb).

SPS transportation schedules are presented in Figures 8.0-1 and 8.0-2. The schedules show the need for major technology development programs commitment in CY 1981, and a commitment for full-scale development of transportation elements by 1990 in order to meet an IOC date at the end of CY 2000.

Table 8.0-1. Satellite Power System (SPS) Program Development Cost

WBS #	DESCRIPTION	DEVELOPMENT DDT&E	TFU	TOTAL
1	SATELLITE POWER SYSTEM (SPS) PROGRAM	33401.762	51103.242	84505.000
1.1	SATELLITE SYSTEM	7933.570	7950.922	15884.492
1.2	SPACE CONSTRUCTION & SUPPORT	7331.180	8602.523	15933.703
1.3	TRANSPORTATION	12468.816	22866.199	35335.016
1.4	GROUND RECEIVING STATION	115.699	3618.727	3734.427
1.5	MANAGEMENT AND INTEGRATION	1392.463	2151.918	3544.382
1.6	MASS CONTINGENCY	4160.031	5912.945	10072.977

Table 8.0-2. *Satellite Power System (SPS) Transportation Systems Development Cost*

WBS #	DESCRIPTION	DDT&E	DEVELOPMENT TFU	TOTAL
1.3	TRANSPORTATION	10748.816	19671.199	30420.016
1.3.1	SPS-HEAVY LIFT LAUNCH VEHICLE(HLLV)	8600.000	9530.492	18130.492
1.3.1.1	SPS-HLLV FLEET	8600.000	8950.176	17550.176
1.3.1.2	SPS-HLLV OPERATIONS	0.0	580.320	580.320
1.3.2	CARGO ORBITAL TRANSFER VEHICLE(COTV)	31.818	3625.720	3657.538
1.3.2.1	COTV VEHICLES	31.818	3621.310	3653.128
1.3.2.1.1	PRIMARY STRUCTURE	3.930	9.267	13.197
1.3.2.1.2	SECONDARY STRUCTURE	4.582	2478.750	2483.332
1.3.2.1.3	CONCENTRATOR	1.685	15.818	17.503
1.3.2.1.4	SOLAR BLANKET	7.664	338.117	345.781
1.3.2.1.5	SWITCHGEAR AND CONVERTERS	2.054	8.760	10.814
1.3.2.1.6	CONDUCTORS AND INSULATION	2.205	8.584	10.789
1.3.2.1.7	ACS HARDWARE	9.697	762.015	771.712
1.3.2.1.8	INFO. MGMT. AND CONTROL	0.0	0.0	0.0
1.3.2.2	COTV OPERATIONS	0.0	4.410	4.410
1.3.3	PERSONNEL LAUNCH VEHICLE(PLV)	1549.000	6251.230	7800.230
1.3.3.1	STS-PLV FLEET	1549.000	3908.082	5457.082
1.3.3.1.1	STS-PLV ORBITER	0.0	1682.531	1682.531
1.3.3.1.2	STS-PLV EXTERNAL TANK	0.0	606.205	606.205
1.3.3.1.3	STS-PLV LIQ. ROCKET BOOSTER	1304.000	873.985	2177.985
1.3.3.1.4	STS CARGO CARRIER AND EM	245.000	745.362	990.362
1.3.3.2	PLV & STS-HLLV OPERATIONS	0.0	2343.150	2343.150
1.3.3.2.1	PLV OPERATIONS	0.0	1214.400	1214.400
1.3.3.2.2	STS HLLV CARGO OPERATIONS	0.0	1128.750	1128.750
1.3.4	PERSONNEL ORBITAL TRANS VEHICLE	350.000	56.282	406.282
1.3.4.1	POTV-FLEET	350.000	54.764	404.764
1.3.4.2	POTV-OPERATIONS	0.0	1.518	1.518
1.3.5	PERSONNEL MODULE(PM)	118.000	201.910	319.910
1.3.5.1	PM FLEET	118.000	198.610	316.610
1.3.5.2	PM OPERATIONS	0.0	3.300	3.300
1.3.6	INTRAORBITAL TRANSFER VEHICLE(IOTV)	100.000	5.567	105.567
1.3.6.1	IOTV FLEET	100.000	5.476	105.476
1.3.6.2	IOTV OPERATIONS	0.0	0.091	0.091

Satellite Systems Division
Space Systems Group
Rockwell International

Table 8.0-3. Satellite Power System (SPS) Program Average Cost

WBS #	DESCRIPTION	INV PER SAT	** OPS COST PER SAT PER YEAR ** RCI	O&M	TOTAL OPS	TOTAL
1	SATELLITE POWER SYSTEM (SPS) PROG	13877.668	451.531	193.713	645.244	14522.910
1.1	SATELLITE SYSTEM	5325.422	205.265	0.705	205.970	5531.391
1.2	SPACE CONSTRUCTION & SUPPORT	1148.332	51.428	11.274	62.701	1211.033
1.3	TRANSPORTATION	1949.004	119.343	80.869	200.212	2149.216
1.4	GROUND RECEIVING STATION	3590.822	0.275	78.377	78.652	3669.474
1.5	MANAGEMENT AND INTEGRATION	600.679	18.815	8.561	27.377	628.055
1.6	MASS CONTINGENCY	1263.413	56.405	13.927	70.332	1333.745

Satellite Systems Division
Space Systems Group
Rockwell International

Table 8.0-4. Satellite Power System (SPS) Transportation System Average Cost

WBS #	DESCRIPTION	INV PER SAT	** OPS COST PER SAT PER YEAR ** RCI	O&M	TOTAL OPS	TOTAL
1.3	TRANSPORTATION	1895.754	115.794	79.094	194.888	2090.641
1.3.1	SPS-HEAVY LIFT LAUNCH VEHICLE(HLLV)	1256.406	99.642	39.372	139.014	1395.420
1.3.1.1	SPS-HLLV FLEET	767.020	99.642	24.256	123.898	890.917
1.3.1.2	SPS-HLLV OPERATIONS	489.387	0.0	15.116	15.116	504.502
1.3.2	CARGO ORBITAL TRANSFER VEHICLE(COTV)	210.343	1.957	6.371	8.328	218.671
1.3.2.1	COTV VEHICLES	205.681	1.957	6.233	8.190	213.871
1.3.2.1.1	PRIMARY STRUCTURE	0.566	0.005	0.017	0.023	0.589
1.3.2.1.2	SECONDARY STRUCTURE	142.934	1.364	4.331	5.696	148.630
1.3.2.1.3	CONCENTRATOR	0.914	0.009	0.028	0.036	0.951
1.3.2.1.4	SOLAR BLANKET	20.077	0.192	0.608	0.800	20.878
1.3.2.1.5	SWITCHGEAR AND CONVERTERS	0.465	0.001	0.014	0.016	0.481
1.3.2.1.6	CONDUCTORS AND INSULATION	0.525	0.002	0.016	0.017	0.542
1.3.2.1.7	ACS HARDWARE	40.199	0.384	1.218	1.602	41.801
1.3.2.1.8	INFO. MGMT. AND CONTROL	0.0	0.0	0.0	0.0	0.0
1.3.2.2	COTV OPERATIONS	4.662	0.0	0.139	0.139	4.801
1.3.3	PERSONNEL LAUNCH VEHICLE(PLV)	423.752	12.995	32.927	45.922	469.674
1.3.3.1	STS-PLV FLEET	188.433	12.995	14.047	27.042	215.474
1.3.3.1.1	STS-PLV ORBITER	100.340	5.797	8.250	14.047	114.387
1.3.3.1.2	STS-PLV EXTERNAL TANK	41.679	0.0	3.330	3.330	45.010
1.3.3.1.3	STS-PLV LIQ. ROCKET BOOSTER	33.991	7.198	2.466	9.664	43.655
1.3.3.1.4	STS CARGO CARRIER AND EM	12.423	0.0	0.0	0.0	12.423
1.3.3.2	PLV & STS-HLLV OPERATIONS	235.319	0.0	18.880	18.880	254.200
1.3.3.2.1	PLV OPERATIONS	216.507	0.0	18.880	18.880	235.387
1.3.3.2.2	STS HLLV CARGO OPERATIONS	18.813	0.0	0.0	0.0	18.813
1.3.4	PERSONNEL ORBITAL TRANS VEHICLE	2.488	0.736	0.254	0.990	3.478
1.3.4.1	POTV-FLEET	1.802	0.736	0.185	0.921	2.723
1.3.4.2	POTV-OPERATIONS	0.686	0.0	0.069	0.069	0.755
1.3.5	PERSONNEL MODULE(PM)	1.294	0.199	0.126	0.324	1.618
1.3.5.1	PM FLEET	0.746	0.199	0.075	0.273	1.019
1.3.5.2	PM OPERATIONS	0.548	0.0	0.051	0.051	0.599
1.3.6	INTRAORBITAL TRANSFER VEHICLE(IOTV)	1.471	0.265	0.045	0.310	1.780
1.3.6.1	IOTV FLEET	1.389	0.265	0.042	0.307	1.697
1.3.6.2	IOTV OPERATIONS	0.061	0.0	0.002	0.002	0.064

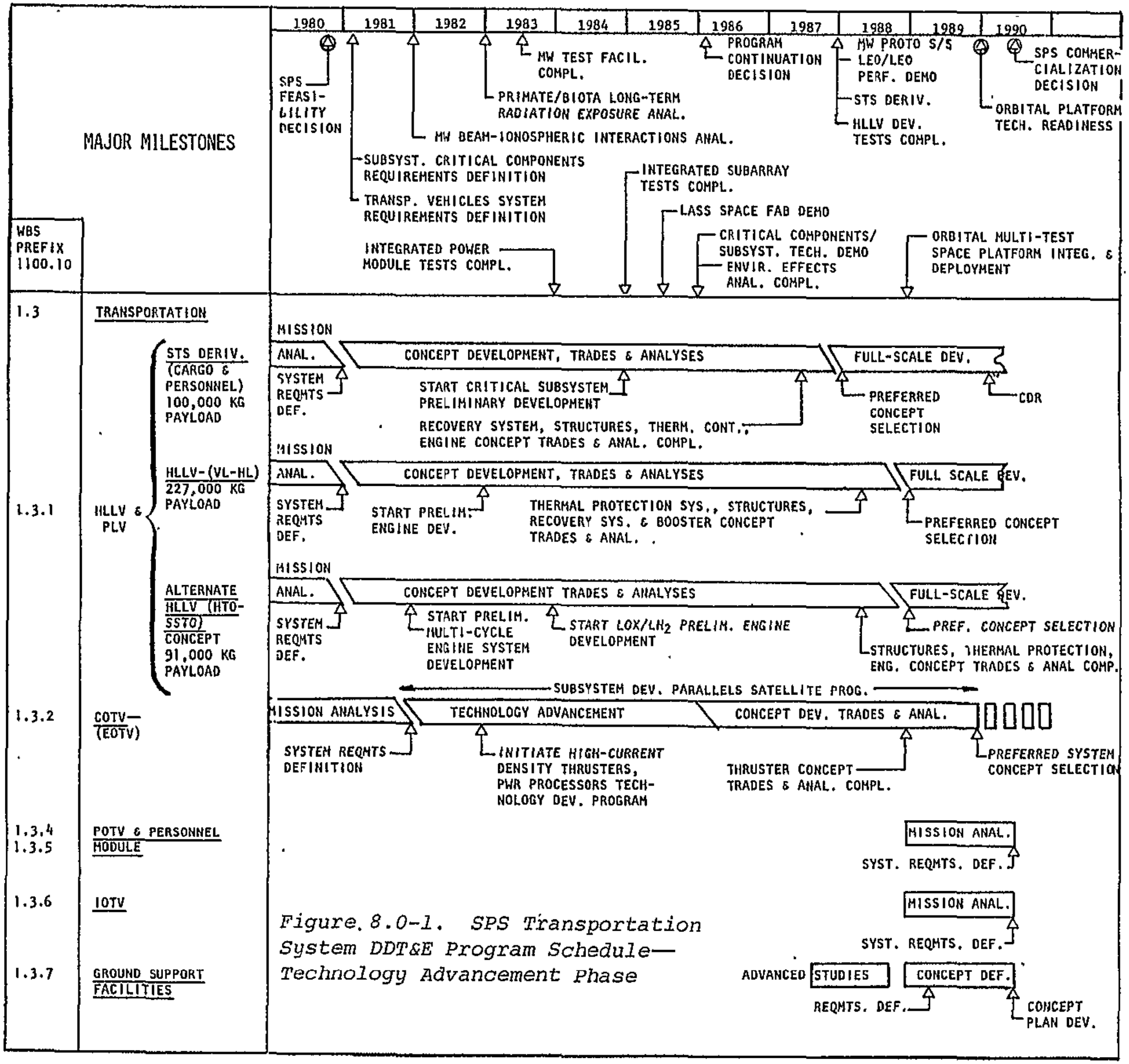


Figure 8.0-1. SPS Transportation System DDT&E Program Schedule— Technology Advancement Phase

APPENDIX A. HORIZONTAL TAKEOFF—SINGLE STAGE TO ORBIT TECHNICAL SUMMARY

APPENDIX A

HORIZONTAL TAKEOFF - SINGLE STAGE TO ORBIT TECHNICAL SUMMARY

APPENDIX A

HORIZONTAL TAKEOFF - SINGLE STAGE TO ORBIT TECHNICAL SUMMARY

A.0 INTRODUCTION

Evolving Satellite Power System (SPS) program concepts envision the assembly and operation of sixty solar-powered satellites in synchronous equatorial orbit over a period of thirty years. With each satellite weighing approximately 35 million kiolgrams, economic feasibility of the SPS is strongly dependent upon low-cost transportation of SPS elements. The rate of delivery of SPS elements alone to LEO for this projected program is 70 million kilograms per year. This translates into 770 flights per year or 2.1 flights per day using a fleet of vehicles, each delivering a cargo of 91,000 kilograms.

The magnitude and sustained nature of this advanced space transportation program concept require long-term routine operations somewhat analogous to commercial airline/airfreight operations. Vertical-takeoff, heavy lift launch vehicles (e.g., 400,000 kg payload) can reduce the launch rate to 175 or more flights per year. However, requirements such as water recovery of stages with subsequent refurbishment, stacking, launch pad usage, and short turnaround schedules introduce severe problems for routine operations. Studies performed previously showed that substantial operational advantages are offered by an advanced horizontal takeoff, single-stage-to-orbit (HTO-SSTO) aerospace vehicle concept. Further analysis of this concept was needed to provide a promising alternative to vertical launch heavy lift launch vehicle approaches for LEO logistics support of the SPS.

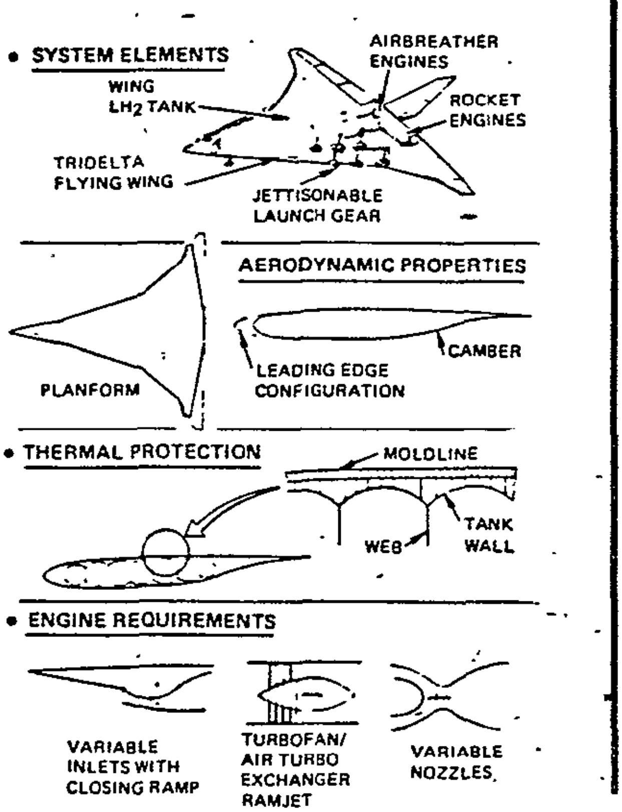
The technical problems requiring investigation were of two types: (a) the need for further development of the vehicle system concept including a multi-cell wet wing containing cryogenic propellants in a blended wing-body configuration; and (b) technology issues, particularly the technical feasibility and performance potential of an advanced hybrid airbreathing engine system, and technical assessment of a flight mode involving horizontal takeoff, long range cruise, subsequent insertion into an equatorial orbit and return via aeromaneuver to the higher-latitude take-off site.

The general objective of this study was to improve system definition and to advance subsystem technologies for a horizontal takeoff, single-stage-to-orbit vehicle which can provide economical, routine earth-to-LEO transportation in support of the Satellite Power Systems program. Specific objectives were:

1. To improve the design definition and technical and operational features of the HTO-SSTO vehicle concept primarily using existing aerodynamic, aerothermal, structural, thermal protection, airbreather and rocket propulsion, flight mechanics and operations technology integrated into a total systems design.

2. To identify disciplines and subsystems in which the application of advanced technology would produce the greatest increase in system performance, and to advance technologies in specific areas.

The primary elements of the HTO-SSTO study and the related technology issues are summarized in Figure A-1. Technical briefings and study progress briefings were given to NASA Headquarters, MSFC, JSC and LaRC, and to USAF/SAMSO. A code showing the general level of technical assurance of the study data as being suitable for feasibility confirmation is placed adjacent to technology items. A filled square, ■ , indicates a high degree of confidence in analytical methods and results. A half-filled square, ◪ , indicates data requiring further technical analyses. The hollow square, □ , relates to technology issues not analyzed or which will require detailed in-depth analysis to produce data suitable for feasibility confirmation.



TECHNOLOGY ISSUES (TO PRE PHASE A TECHNOLOGY LEVEL)

VEHICLE AND SYSTEMS INTEGRATION
PRELIMINARY LAYOUTS OF STRUCTURE
◪ WET WING ■ CARGO BAY ■ CREW COMPARTMENT
MAJOR SUBSYSTEM INSTALLATION
◪ AIRBREATHER ■ ROCKET ENGINES ◪ GEAR
ENVIRONMENTAL PROTECTION SYSTEM
■ TILE ■ METALLIC TPS ■ INSULATION

REFINE AERODYNAMIC PROPERTIES
FOR WAVEDRAG REDUCTION
□ LEADING EDGE CONFIGURATION
■ SPAN THICKNESS DISTRIBUTION
CENTER OF PRESSURE CONTROL
■ TRIDELTA MIX FOR PLAN FORM
□ ENGINE INLET INTERACTIONS

DETERMINE THERMAL PROTECTION SYSTEM
■ AEROTHERMAL ANALYSIS
■ TEMPERATURE PROFILES
■ CANDIDATE TPS SYSTEMS
■ TILE ■ METALLIC

DETERMINE ENGINE REQUIREMENTS
◪ AIRBREATHER ■ ROCKETS
■ INLETS □ NOZZLES
□ MULTIPLE CYCLE ANALYSIS ■ IDENTIFIED
□ AIRBREATHER PERFORMANCE

STRUCTURE ANALYSIS
◪ WING ■ BODY TANK ■ CREW COMPARTMENT
□ CARGO BAY
□ AIRBREATHER ■ ROCKET THRUST STRUCTURE

MASS PROPERTIES ANALYSIS
□ MASS □ CENTER OF GRAVITY

TRAJECTORY ANALYSIS
□ AIRBREATHER ■ ROCKET
■ ENERGY METHOD ■ LaRC POST PROGRAM

Figure A-1. Study Summary -- Advanced Transportation System for SPS

The combined systems design/performance and technology development studies produced a number of significant results.

1. Demonstrated, with end-to-end simulation, the ability of the vehicle to take off from KSC, cruise to the equatorial plane, insert into a 300 nmi equatorial orbit with 151,000-pound payload, and then to re-enter and return to the launch site; also to deliver a 196,000-pound payload with a due-East launch.

2. Devised a modified airbreathing engine cycle for operation in turbofan, air-turbo-exchanger and ramjet modes to provide an effective match with takeoff, cruise and acceleration requirements.

3. Showed that the HTO-SSTO lower surface temperatures during reentry are several hundred degrees lower than the STS orbiter lower surface temperatures because of a lower wing loading. As a result, an advanced titanium aluminide system shows promise of being lighter than the RSI tile for this application.

This study was funded primarily by Rockwell IR&D funds and a summary only is contained herein.

A.1 OPERATIONAL FEATURES

The HTO-SSTO concept adapts existing and advanced commercial and/or military air transport system concepts, operations methods, maintenance procedures, and cargo handling equipment to include a space-related environment. The principal operational objective is to provide economic, reliable transportation of large quantities of material between earth and LEO at high flight frequencies with routine logistics operations and minimal environmental impact. An associated operational objective was to reduce the number of operations required to transport material and equipment from their place of manufacture on earth to low earth orbit.

Operations features derived in the study are as follows:

- Single orbit up/down to/from the same launch site (at any launch azimuth subject to payload/launch azimuth match)
- Capable of obtaining 300 nmi equatorial orbit when launched from KSC
- Takeoff and land on 8,000 to 14,000-foot runways (launch velocity $\approx$ 225 knots; landing velocity $\leq$ 115 knots)
- Simultaneous multiple launch capability
- Total system recovery including the takeoff gear which is jettisoned and recovered at the launch site
- Aerodynamic flight capability from payload manufacturing site to launch site, addition of launch gear and fueling, and launch into earth orbit

- Amenable to alternative launch/landing sites

- Incorporates Air Force (C-5A Galaxy) and commercial (747 cargo) payload handling, including railroad, truck, and cargo-ship containerization concepts, modified to meet space environment requirements

- Swing-nose loading/unloading, permitting normal aircraft loading-door facility concept application

- Propulsion system service using existing support equipment on runway aprons or near service hangars

- In-flight refueling options (option not included in reference vehicle data)

A.2 DESIGN FEATURES

The HTO-SSTO utilizes a tri-delta flying wing concept, consisting of a multi-cell pressure vessel of tapered, intersecting cones. The tri-delta planform (blended fuselage-wing) and a Whitcomb airfoil section offer an efficient aerodynamic shape from a performance standpoint and high propellant volumetric efficiency. The outer panels of the wing and vent system lines in the wing's leading edge provide the gaseous ullage space for LH_2 fuel. LH_2 and LO_2 tanks are located in each wing near the vehicle, c.g., and extend from the root rib to the wing tip LH_2 ullage tank (Figure A-2). Approximately 20% of the volume of the vertical stabilizer is utilized as part of the gaseous ullage volume of the integral wing-mounted LO_2 tanks. In the aft end of the vehicle, three uprated high-P_c rocket engines (thrust = 3.2×10^6 lb) are attached with a double-cone thrust structure to a two-cell LH_2 tank.

Most of the cargo bay side walls are provided by the root-rib bulkhead of the LH_2 wing tank. The cargo bay floor is designed similar to the C5-A military transport aircraft. This permits the use of MATS and Airlog cargo loading and retention systems. The top of the cargo bay is a mold-line extension of the wing upper contours, wherein the frame inner caps are arched to resist pressure at minimum weight. The forward end of the cargo bay has a circular seal/docking provision to the forebody. Cargo is deployed in orbit by swinging the forebody to 90 or more degrees about a vertical axis at the side of the seal, and transferring cargo from the bay into space or to in-space receivers on telescoping rails.

The forebody is an RM-10 ogive of revolution with an aft dome closure. The ogive is divided horizontally into two levels. The upper level provides seating for crew and passengers, as well as the flight deck. The lower compartment contains electronic, life support, power (fuel cell), and other subsystems including spare life support and emergency recovery equipment.

Ten high-bypass, supersonic-turbofan/airturbo-exchanger/ramjet engines with a combined static thrust of 1.4×10^6 lb are mounted under the wing. The inlets are variable area retractable ramps that also close and fair the bottom into a smooth surface during rocket powered flight and for high angle-of-attach ballistic re-entry.

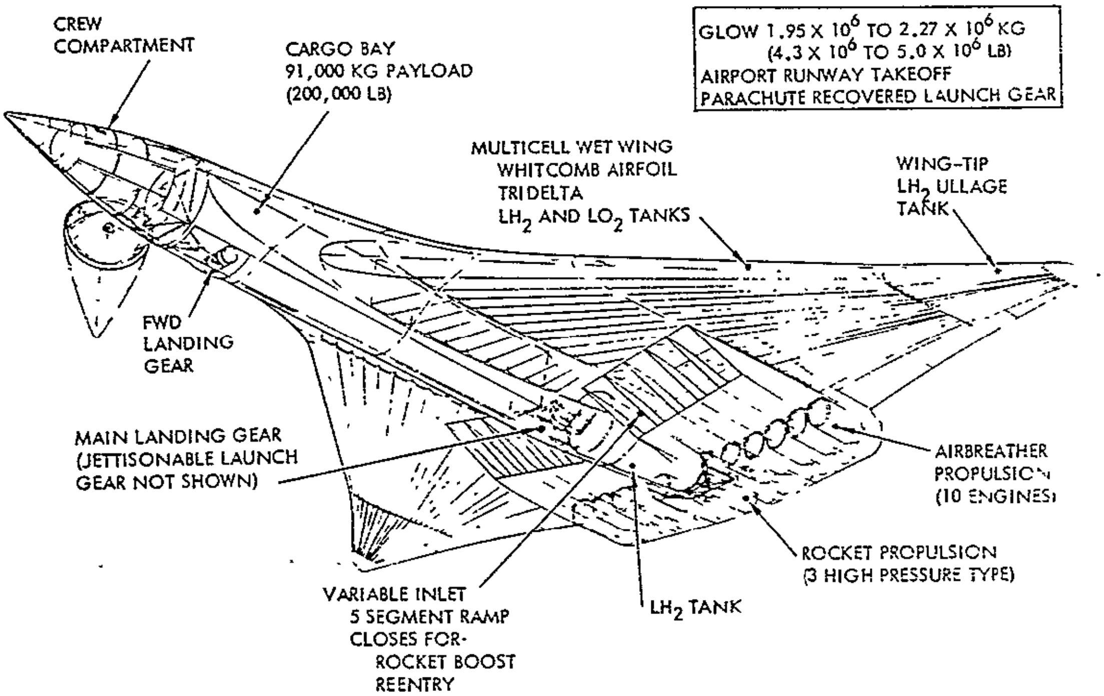


Figure A-2. HTO-SSTO Design Features

Figure A-3 shows an inboard profile of the vehicle, illustrating the details of body construction, crew compartment, cargo bay length, LH_2 tank configuration, and location of the rocket engines at rear of fuselage. The hinging and rotation of the nose section for loading and unloading the payloads are illustrated, with indication of view angle from the rear of the nose section during these operations. The multiple landing gear concept shows the position of the nose gear bogie, the jettisonable takeoff gear, and the main landing gear for powered landing.

Figure A-4 presents front and rear views of the vehicle showing the blended wing, engine inlet ducts, landing gear arrangement, and vertical stabilizer. Also shown are typical sections through the vehicle at:

- The hinge line section (B-B) aft of the crew compartment and forward of the nose gear. Cross-sectional dimensions of the cargo bay are indicated.

- The 40% chord line fuselage section (C-C) illustrating the wing and fuselage construction and the profile of the wing/fuselage fairing.

- The main landing gear station (D-D) illustrating the gear retraction geometry, the relationship of the gear to the engine air inlet ducts and the wing construction and profile to the fuselage shape.

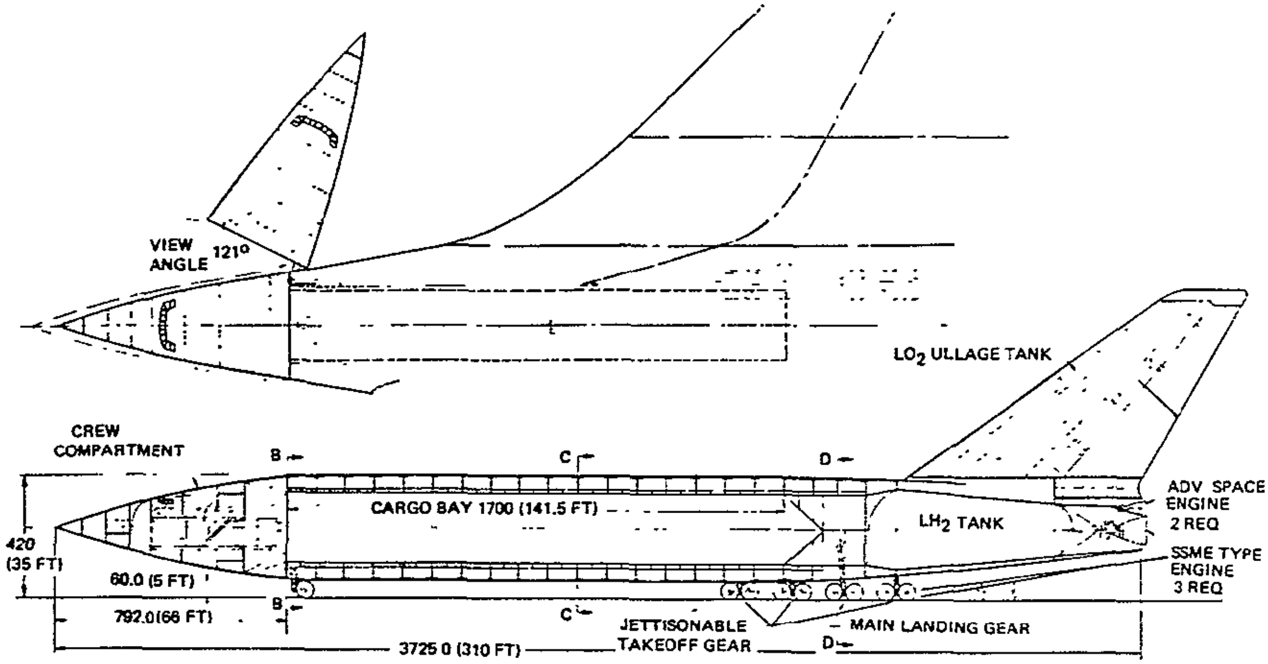


Figure A-3. HTO-SSTO Inboard Profile

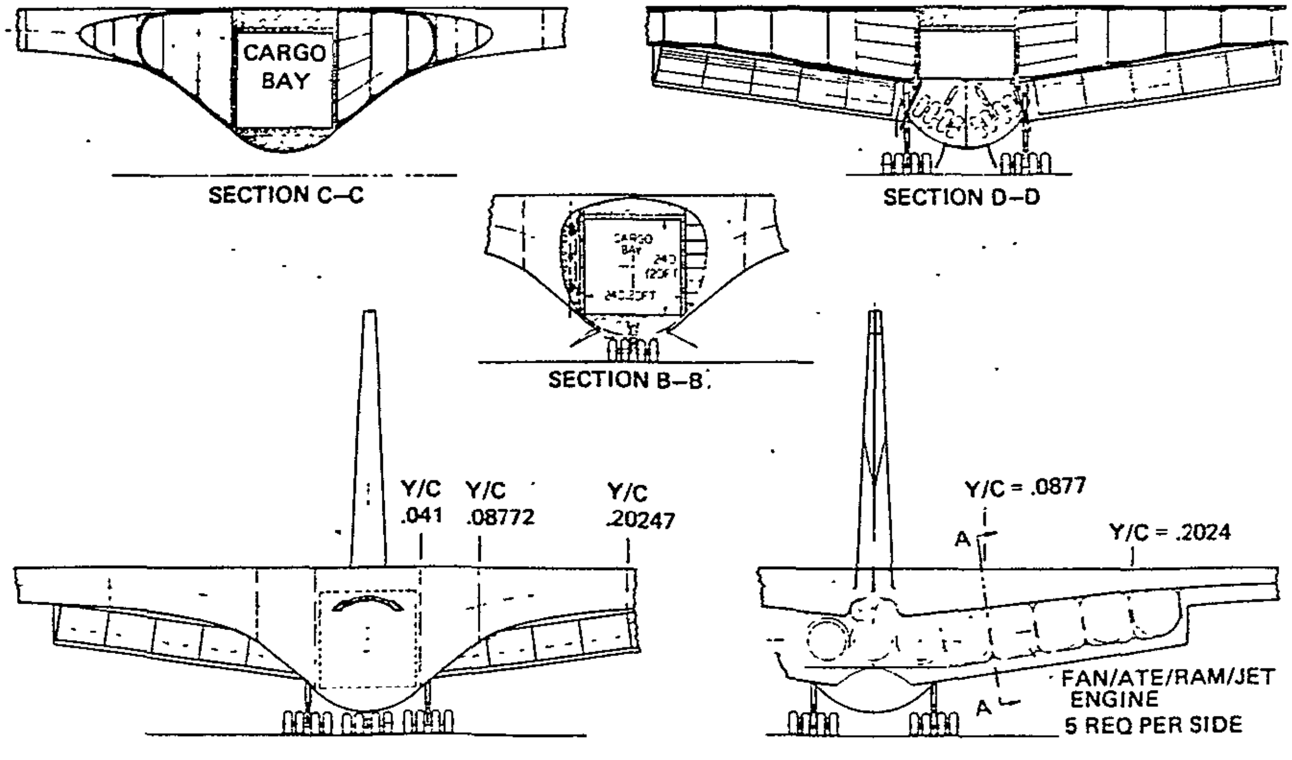


Figure A-4. Vehicle Section Results

Figure A-5 presents details of the basic multi-cell structure of the wing. The upper portion illustrates the application of "Shuttle-type" RSI tile thermal protection system (TPS). The lower portion shows a potential utilization of a "metallic" TPS.

The wing is an integrated structural system consisting of an inner multi-cell pressure vessel, a foam-filled structural core, an inner facing sheet, a perforated structural honeycomb core, and an outer facing sheet. The inner multi-cell pressure vessel arched shell and webs are configured to resist pressure. The pressure vessel and the two facing sheets, which are structurally interconnected with phenolic-impregnated, glass fiber, honeycomb core, resist wing spanwise and chordwise bending moments. Cell webs react winglift shear forces. Torsion is reacted by the pressure vessel and the two facing sheets as a multi-box wing structure.

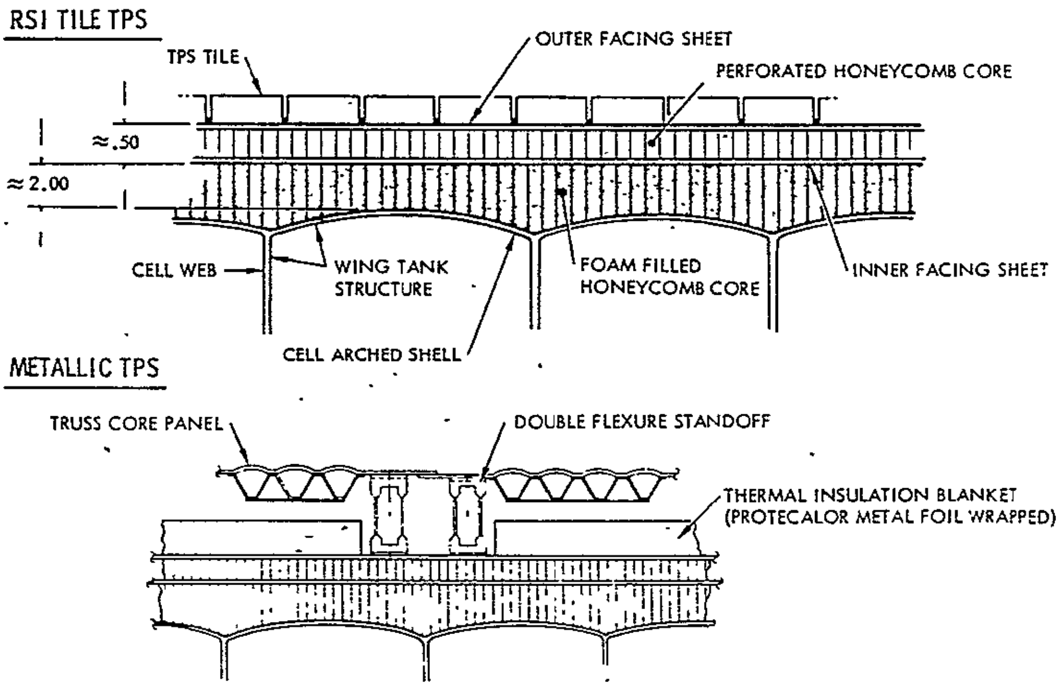


Figure A-5. Wing Construction Detail with Candidate TPS Configurations

The outer honeycomb core is perforated and partitioned to provide a controlled passage, purge and gas leak detection system function in addition to the function of structural interconnect of the inner and outer facing sheets. The construction of the wing structure utilizes the "Inflation Assembly Technique" developed by Rockwell for the Saturn II booster common bulkhead.

A.3 MULTI-CYCLE AIRBREATHER ENGINE SYSTEM

Takeoff and climb to 100,000 ft altitude and 5,800 fps is by airbreather propulsion. Parallel burn of airbreather and rocket propulsion occurs between 5,800 to 7,200 fps. Rocket power is then employed from 7,200 fps to orbit.

The multi-cycle airbreathing engine system, Figure A-6 is derived from the General Electric CJ805 aircraft engine, the Pratt and Whitney SWAT 201 supersonic wrap-around turbofan/ramjet engine, the Aerojet Air Turborocket, Marquardt variable plug-nozzle, ramjet engine technology, and Rocketdyne tubular-cooled, high-P_C rocket engine technology.

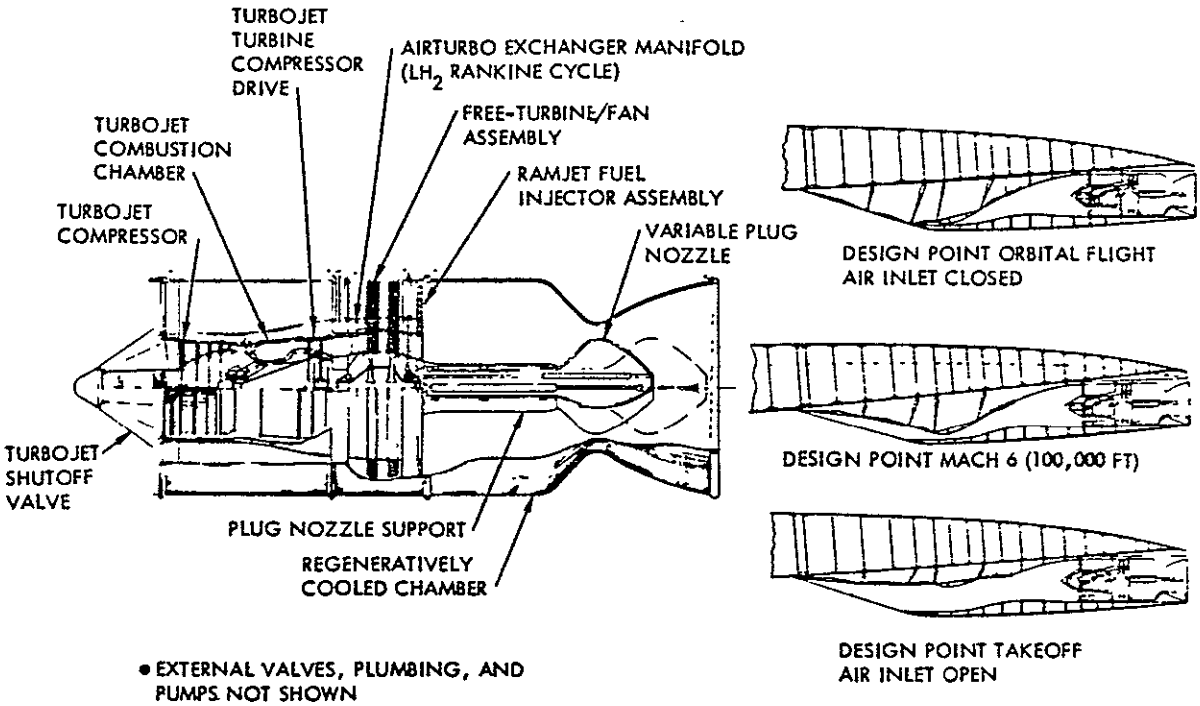


Figure A-6. Multi-Cycle Airbreathing Engine and Inlet, Turbofan/Air Turboexchanger/Ramjet

The multi-mode power cycles include: an aft-fan, turbofan cycle, a LH_2, regenerative Rankine, air-turboexchanger cycle; and a ramjet cycle that can also be used as a full flow (turbojet core and fan bypass flow) thrust-augmented turbofan cycle. These four thermal cycles may receive fuel in any combination permitting high engine performance over a flight profile from sea level takeoff to Mach 6 at 100,000 ft altitude.

The engine air inlet and duct system is based on a five-ramp variable inlet system with actuators to provide ramp movement from fully closed (upper RH figure) for rocket-powered and re-entry flight, to fully open (lower RH figure) for takeoff operation.

The inlet area was determined by the engine airflow required at the Mach 6 design point. The configuration required 1.4×10^6 pounds thrust at the Mach 6 condition and at least 1.2×10^6 pounds for takeoff. This resulted in an inlet area of approximately 1200 ft^2 or 120 ft^2/engine for a 10-engine configuration. In order to provide pressure recovery with minimum spillage drag over the wide range of Mach numbers, a variable multi-ramp inlet is required. Inlet pressure recovery efficiency vs. velocity is plotted on Figure A-7. Higher recoveries are possible for the HTO vehicle than for military aircraft which must operate during more violent maneuvers. However, the pressure recovery must still provide a margin which prevents inlet instability and possible engine flameout from expulsion of the normal shock during transients.

Estimated engine thrust (total of 10 engines) versus velocity is given in Figure A-8. Initially, a constant thrust of 1.4 million pounds of thrust was assumed for the Rockwell modified Rutowski energy method trajectory analysis (dashed curve of Figure A-8). A tentative airbreather engine performance map was estimated from engine data sources previously described. Subsequent analyses produced the engine thrust versus Mach number estimate shown by the upper solid curve of Figure A-8.

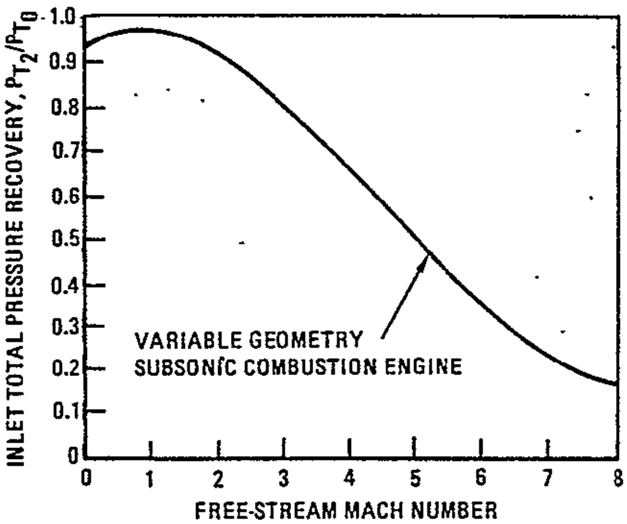


Figure A-7. Air Induction System Performance

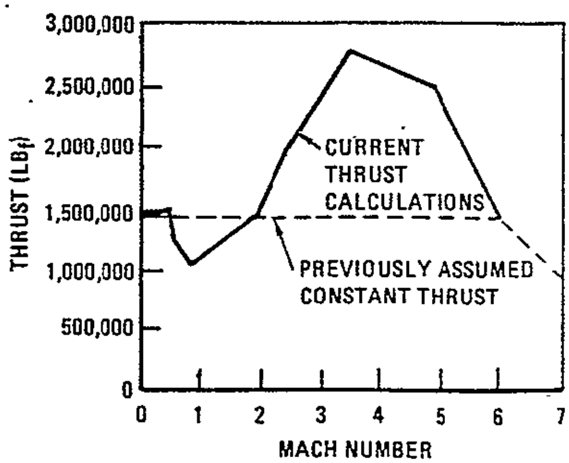


Figure A-8. Airbreather Thrust Versus Mach Number

Major engine companies were contacted to obtain assistance in advanced cycle analysis and to obtain the results of any studies which investigated this operating regime. Data from a Pratt and Whitney report (Reference 1) on an advanced hydrogen burning engine, the SWAT 201 turbofan ramjet, were evaluated and scaled up to the size required. However, this engine, which uses a bypass valve to close off the engine core above Mach 3.1 and operates the afterburner as a ramjet at higher speeds, did not provide a good match of thrust requirements over the required operating range. Also because of the high compression-ratio design, the engine thrust-to-weight ratio (T/W) was in the range of 4.5 to 5.5 for an installed system. Single-stage-to-orbit launch vehicle analysis showed that a T/W of at least 8 would be necessary to meet the vehicle payload requirements. From Aerojet, (Reference 2) data were obtained on an air turborocket concept which provides a potential for meeting the required T/W values while providing a better match of thrust required at takeoff, transonic and supersonic conditions. A modification of this cycle was devised by Rockwell to best match the SSTO requirements. This engine operates as an augmented turbofan for takeoff, a turbofan for high-efficiency cruise, an augmented turbofan for acceleration, and as a ramjet above Mach 3.

The engine components include a rotary vane assembly to close off the compressor-turbine assembly at higher Mach numbers. The use of LH_2 fuel permits the use of a Rankine-cycle air turboexchanger concept to provide power for the bypass fan. This allows elimination of approximately one-half of the normal turbofan compressor stages normally needed for fan drive. Heating of the LH_2 in outer walls and nozzle plug of tubular construction, in addition to providing fan drive power, permits stoichiometric combustion in the augmentor/ramjet by cooling of exposed surfaces. The 5500-degree combustion temperature provides high cycle efficiency. During ramjet mode operation, the fan is allowed to windmill and is cooled by flow of LH_2 through the fan guide vanes.

The scope of this study did not permit a detailed evaluation of engine components to provide further, more accurate calculation of the performance capability of this engine concept. Engine manufacturers are best equipped to further refine the design and provide real data on concept feasibility and system weight.

For preliminary estimation of airbreathing propulsion system size requirement, a computer program was developed for the Hewlett Packard computer. A flow diagram of this program is shown in Figure A-9.

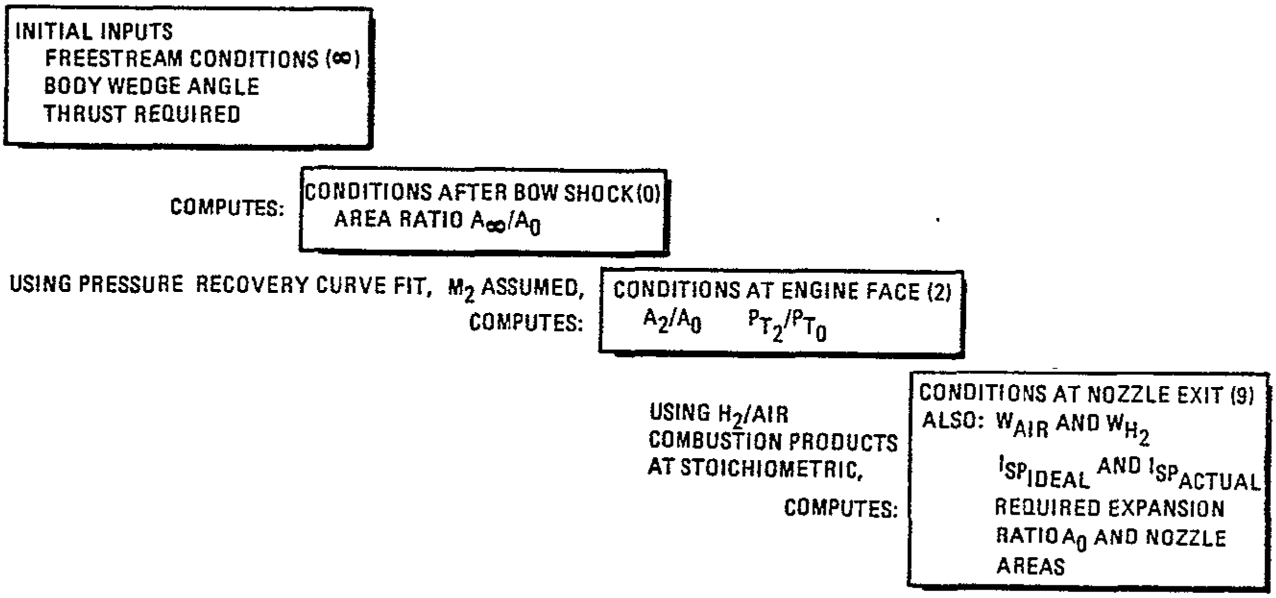


Figure A-9. Computer Program Flow Diagram for Airbreather Propulsion System Sizing

A computer program which has the capability of computing performance of mixed-cycle engines including JP and LH_2 fuel, as well as the air turboexchanger cycle was obtained from the Los Angeles Division of Rockwell (Reference 3). This program was developed under NASA contract in 1966 and is currently used by LAD for calculation of JP-fueled turbojet and turbofan engine data for advanced aircraft.

In order to maximize the payload boosted to orbit, an optimization technique is required to define the proper engine sequencing over the flight trajectory.

A.4 AERODYNAMIC CHARACTERISTICS

The selected wing shape is a supercritical Whitcomb airfoil with a relatively blunt leading edge, flat upper surfaces and cambered trailing edges. The trailing-edge camber and the tri-delta shape minimize translation of the center of pressure throughout the flight Mach number regime. The blunt leading edge offers good subsonic characteristics, but produces relatively high supersonic wave drag; therefore, further shape and refinements are required. The wing has a spanwise thickness distribution of 10 percent at the root, 6 percent near midspan, and 5 percent at the tip, providing a large interior volume for storage of fuel.

Aerodynamic coefficients (C_L, C_D, C.P.) were calculated using the Flexible Unified Distributed Panel program FA-475, which was developed by the LAD Aerodynamic group. Because the governing equation is linear, singular behavior of the linear equation and nonlinearity near M = 1.0 preclude the transonic solutions. Also, the hypersonic solution cannot be calculated with this theory due to the introduction of nonlinear terms. However, aerodynamic coefficients computed at M_∞ = 5.0 can be frozen and can be used for hypersonic application. Viscous drag due to the skin friction is not computed by this program. This effect was added in a separate analysis. The resulting aerodynamic coefficients are plotted versus flight Mach number in Figure A-10.

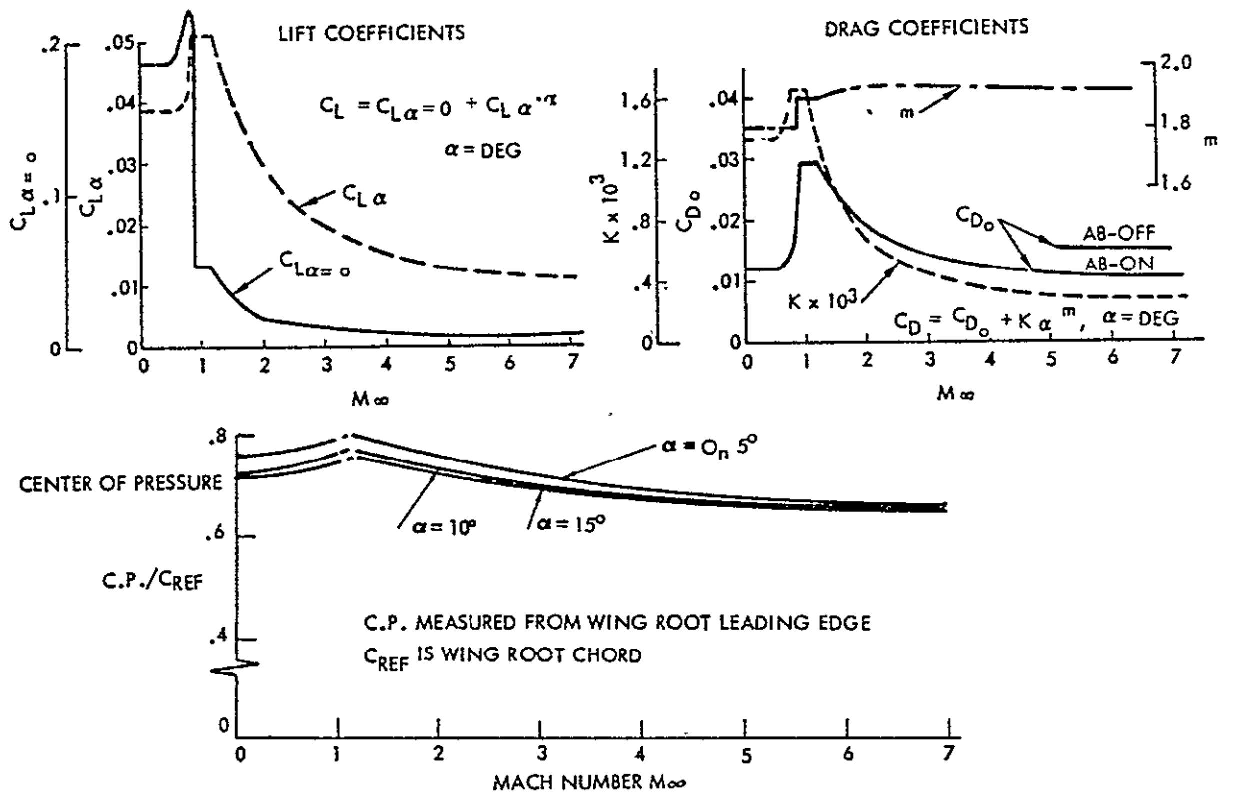


Figure A-10. Aerodynamic Coefficients

Maximum lift/drag and corresponding lift coefficients and angle of attack versus Mach number are given in Figure A-11.

- Subsonic: $(L/D)_{max} \simeq 16.0$ at $\alpha \simeq 1.0$, $C_L \simeq 0.22$
- Supersonic: $(L/D)_{max}$ from 5.4 to 4.0 at $4.5° \leq \alpha \leq 6.2°$
- Hypersonic: For airbreather-OFF, rocket only $(L/D)_{max} \simeq 3.4$

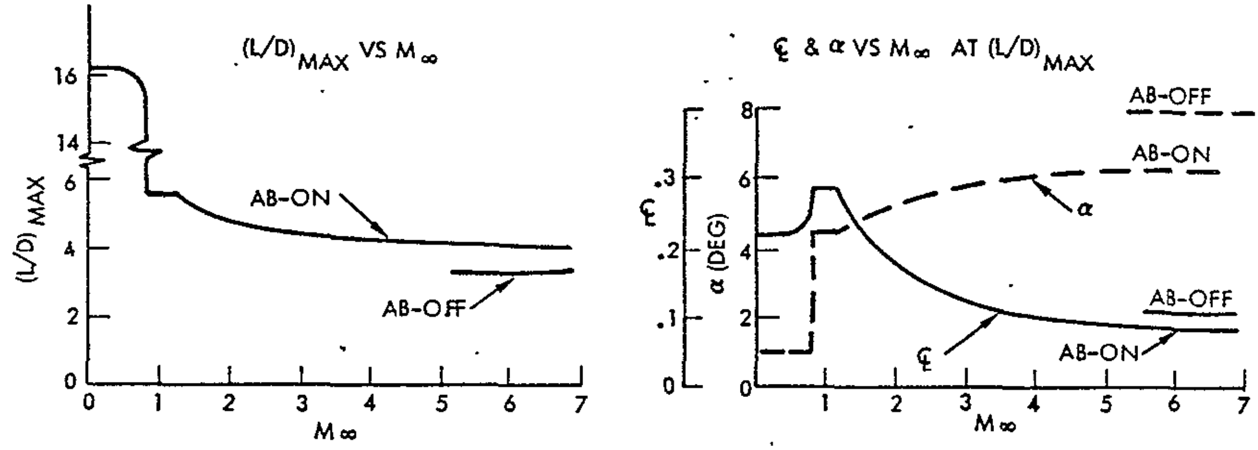


Figure A-11. Maximum Lift/Drag

The wing bending moments are based on the following data:

- Differential pressure distributions computed by the Unified Distributed Panel Program
- X = 10°
- 2 g loading on wing
- GLOW = 4×10^6 lb

Lift force (L_F) and bending moment (BM) at the wing root for the above conditions are shown in the following tabulation.

M_∞	$L_F \times 10^{-6}$ lb	BM $\times 10^{-6}$ ft-lb
0.5	4.0	318
0.8	4.0	322
1.2	3.94	334
2.0	3.87	278
3.0	3.8	251
5.0	3.0	185

A.5 FLIGHT MECHANICS

The majority of the ascent performance analysis for the SSTO vehicle concept was accomplished using a recently developed lifting ascent program based on a modified Rutowski Energy Method (Ikawa Method). This technique accurately estimated payload and propellant performance; however, it did not provide a bona fide integrated time history of trajectory state from liftoff to orbit insertion. A second computer program, the Two-Dimensional Trajectory Program (TDTP), was then used to compute the ascent trajectory timeline.

In order to do an end-to-end simulation of the SSTO (i.e., airbreather horizontal takeoff, climb, cruise, turn, airbreather ascent, rocket ascent, coast, and final orbit insertion) with flight optimization including aerodynamic effects, Rockwell acquired the Langley POST computer program (program to optimize simulated trajectories, developed by Martin-Marietta). POST was installed on the CDC system at Rockwell and several launch cases were executed.

The SSTO uses aircraft-type flight from airport takeoff to approximately Mach 6, with a parallel burn transition of airbreather and rocket engines from Mach 6 to 7.2, and rocket-only burn from Mach 7.2 to orbit. Figure A-12 illustrates a nominal trajectory from KSC to 300-nmi earth equatorial orbit. Prime elements of the trajectory are:

- Runway takeoff under high-pass turbofan/airturbo exchanger (ATE)/ ramjet power, with the ramjets acting as supercharged afterburners
- Jettison and parachute recovery of launch gear
- Climb to optimum cruise altitude with turbofan power
- Cruise at optimum altitude, Mach number, and direction vector to earth's equatorial plane, using turbofan power
- Execute a large-radius turn into the equatorial plane with turbofan power
- Climb subsonically at optimum climb angle and velocity to an optimum altitude, using high bypass turbofan/ATE/ramjet (supercharged afterburner) power
- Perform an optimum pitch-over into a nearly constant-energy (shallow γ-angle) dive if necessary, and accelerate through the transonic region to approximately Mach 1.2, using turbofan/ramjet (supercharged afterburner) power
- Execute a long-radius optimum pitch-up to an optimum supersonic climb flight path, using turbofan/ATE/ramjet power
- Climb to approximately 29 km (95 kft) altitude, and 1900 m/s (6200 fps) velocity, at optimum flight path angle and velocity, using proportional fuel-flow throttling from turbofan/ATE/ramjet, or full ramjet, as required to maximize total energy acquired per unit mass of fuel consumed as function of velocity and altitude

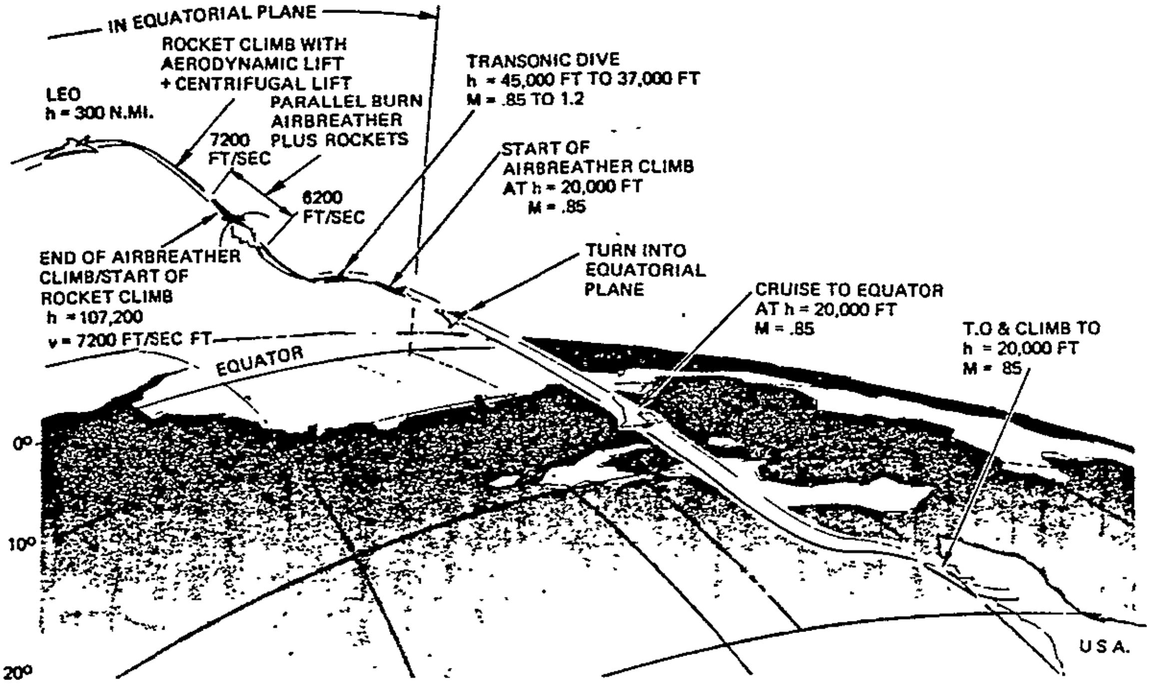


Figure A-12. SSTO Trajectory

- Ignite rocket engines to full required thrust level at 6200 fps and parallel burn to 7200 fps
- Shut down airbreather engines while closing airbreather inlet ramps
- Continue rocket power at full thrust
- Insert into an equatorial elliptical orbit 91×556 km (50×300 nmi) along an optimum lift/drag/thrust flight profile
- Shut down rocket engines and execute a Hohmann transfer to 556 km (300 nmi)
- Circularize Hohmann transfer

The re-entry trajectory is characterized by low gamma (flight path angle) high alpha (angle of attack) similar to Shuttle. The main re-entry trajectory elements are:

- Perform delta velocity (ΔV) maneuver and insert into an equatorial elliptical orbit 91×556 km (50×300 nmi)
- Perform a low-gamma, high-alpha deceleration to approximately Mach 6.0
- Reduce alpha to maximum lift/drag (L/D) for high-velocity glide and cross-range maneuvers to subsonic velocity (approximately Mach 0.85)

- Open inlets and start airbreather engines as required

- Perform powered flight to landing field, land on runway, and taxi to dock

Flyback fuel requirements include approximately 300 nmi subsonic cruise and two landing approach maneuvers (first approach waveoff with flyaround for second approach).

Typical I_{sp} characteristics of AB/rocket engine system are:

- Subsonic range - Linear reduction of I_{sp} from 9700 to 4000 sec at 1200 fps

- Supersonic range - Reduction of I_{sp} from 4000 sec at 1200 fps to 3500 sec at ≈5600 fps (AB)

- Rocket - I_{sp} = 455 sec

The airbreather cruise mode, which results in an economical orbit plane change from the launch site to the equatorial orbit, was analyzed. The estimated fuel requirements to cruise 1000 statute miles down-range for alternate propulsion modes are given below.

V (ft/sec)	Altitude (k-ft)	Δt (sec)	ΔW_F (lb)	Engine
800	20	6600	72,000	Turbofan Jet
6000	85	880	386,000	Ramjet

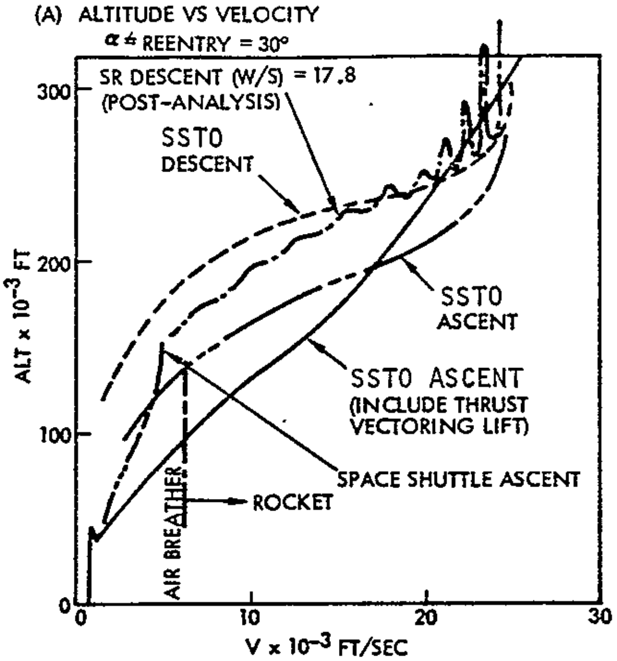
Although subsonic cruise takes a longer time (110 minutes), the amount of fuel consumed is substantially less when the orbital plane change is accomplished with subsonic cruise at maximum L/D.

A transition maneuver from high-lift configuration to $(L/D)_{max}$ configuration is performed shortly after liftoff (beginning at 3000 ft altitude). The maximum angle of attack of 13 degrees is reduced gradually to 1 degree for subsonic $(L/D)_{max}$ climb configuration.

Velocity and angle of attack vs flight time indicate the time required to reach 300 nmi orbit (not including subsonic cruise leg) varies from 1800 to 2300 sec, depending upon $(W/S)_O$, (T/W), and engine operational mode.

Variation in load factor, altitude, and dynamic pressure with respect to velocity and time during supersonic ascent show a maximum load acceleration less than 2.3 g. Maximum dynamic pressure is 940 psf, which is within load limits. From takeoff to burnout, the ascent profile is quite shallow - with flight path angle ranging between -0.7 and 4.5 degrees.

Ascent and descent trajectories of the SSTO and the Space Shuttle missions are compared in Figure A-13. Because the performance of airbreathing engines and aerodynamic lifting of winged vehicle depend on the high dynamic pressure,



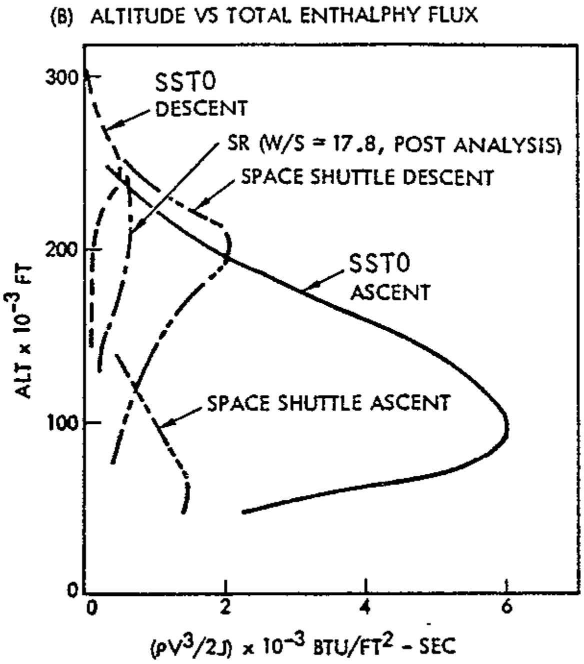


Figure A-13. Ascent and Descent Trajectory Comparisons

the SSTO flies at much lower altitude during the powered climb than the vertical ascent trajectory of the Space Shuttle for a given flight velocity. Light wing loading of the SSTO contributes to the rapid deceleration during deorbit.

The total enthalpy flux histories which indicate the severity of expected aerodynamic heating are shown in Figure A-13. As expected, the aerodynamic heating of ascent trajectory may design the SSTO TPS requirement. The maximum total enthalpy flux of 6000 Btu/ft^2-sec is estimated near the end of airbreather power climb trajectory. Except in the vicinity of vehicle nose, wing leading edge, or structural protuberances, where interference heating may exist, most of the ascent heating is from the frictional flow heating on the relatively smooth flat surface.

The descent heating is mainly produced by the compressive flow on the vehicle windward surface during the high-angle-of-attack re-entry, and is expected to be considerably lower than the Space Shuttle re-entry heating.

Weight in orbit is summarized in Table A-1. The data entries identified by an asterisk are revised reference vehicle data resulting from Rockwell and NASA/MSFC data exchange in May 1978. Calculations reflect additional fuel reserves, performance losses and a 10-percent growth factor. Inert weight in orbit was increased from 694,510 lb to 775,800 lb and airbreather engine thrust of 1.4×10^6 lb constant was revised to reflect increase in airbreather thrust potential shown in Figure A-8.

Table A-1. SSTO Weight in Orbit Summary

ORBIT	GLOW $W_0 \times 10^6$ LB	ROCKET I_{SP} = 455. SEC (SHUTTLE VALUES) ENERGY METHOD W_f (LB)	PAYLOAD (LB)	POST ANALYSIS W_f (LB)	PAYLOAD (LB)	ROCKET I_{SP} = 468 SEC (LaRC VALUES) ENERGY METHOD W_f (LB)	PAYLOAD (LB)
EQUATORIAL	4.31	787,400.	*92,890				
ORBIT	4 31 (P B)	801,700	107,190	790 000	95,490.	832,800	138 290.
CRUISE	4 62 (P.B)	845,800.	151,290				
FROM KSC	5 00 (P B)	895,300.	200,790.				
INCLINED	4 31	864,500	169,990.			897,000	202 490.
ORBIT	4.31 (P B)	882,600	188,090.	849,000.	154,490	917,300	222,790
KSC	4 62 (P B)	925,100	230,590				
DUE EAST	*5 00 (PB)			*972,400	*196,580		

- DATA FOR 300 N MI ORBITAL INSERTION
- REFERENCE WING AREA (S_{REF}) = 40,900 SQ FT
- WEIGHT IN ORBIT (EXCLUDING PAYLOAD) = 694,510 LB * = 775,800 LB
- LAUNCH FROM KSC
- PB = PARALLEL BURN
- AIRBREATHER
  - THRUST = 1.4×10^6 LB
  - I_{SP} - VARIABLE
  - VELOCITY = $0 \leq V \leq 6200$ FT/SEC
- ROCKET
  - THRUST = 3.2×10^6 LB
  - I_{SP} = SEE CHART
  - VELOCITY = $6200 \leq V \leq V_{ORBIT}$ FT/SEC

A.6 AERODYNAMIC AND STRUCTURAL HEATING

Preliminary aerodynamic heating evaluation of the SSTO configuration was performed for several wing spanwise stations and the fuselage centerline.

For the wing lower surfaces, heating rates were computed including the chordwise variation of local flow properties. Effects of leading edge shock and angle of attack were included in the local flow property evaluation. Leading edge stagnation heating rates were based on the flow conditions normal to the leading edge neglecting cross-flow effects. All computations were performed using ideal gas thermodynamic properties.

Wing upper-surface heating rates were computed using free-stream flow properties, i.e., neglecting chordwise variations of flow properties. Heating rates were computed for several prescribed wall temperatures as well as the reradiation equilibrium wall temperature condition. Transition from laminar to turbulent flow was taken into account in the computations. Wing/body and inlet interference heating effects were not included in this preliminary analysis. The analysis was limited to the ascent trajectory, since the descent trajectory is thermodynamically less severe.

These parametrically generated aerodynamic heating rate data were used for thermal analysis of the various candidate insulation systems. Radiation equilibrium temperatures for emissivity, $\varepsilon = 0.85$, are based on:

- Leading edge stagnation heating rates peak at M = 16.4, alt = 196,000 ft
- Upper wing surface uniform static pressure assumed, temperatures peak at M = 6.4, alt = 86,500 ft
- Lower wing surface heating rates and temperatures peak at M = 7.9, alt = 116,000 ft
- Local flow property variation, angle of attack, and leading-edge shock effects are included
- Inlet interference effects were not included

Isotherms of the peak surface temperatures for upper and lower surfaces (excluding engine inlet interference effects) for the SSTO and Orbiter are shown in Figure A-14. Leading edge and upper wing surface temperatures have similar profiles. The SSTO lower-surface temperatures are from 400°F to 600°F lower than the orbiter due to lower re-entry wing loading (23 versus 67 psf).

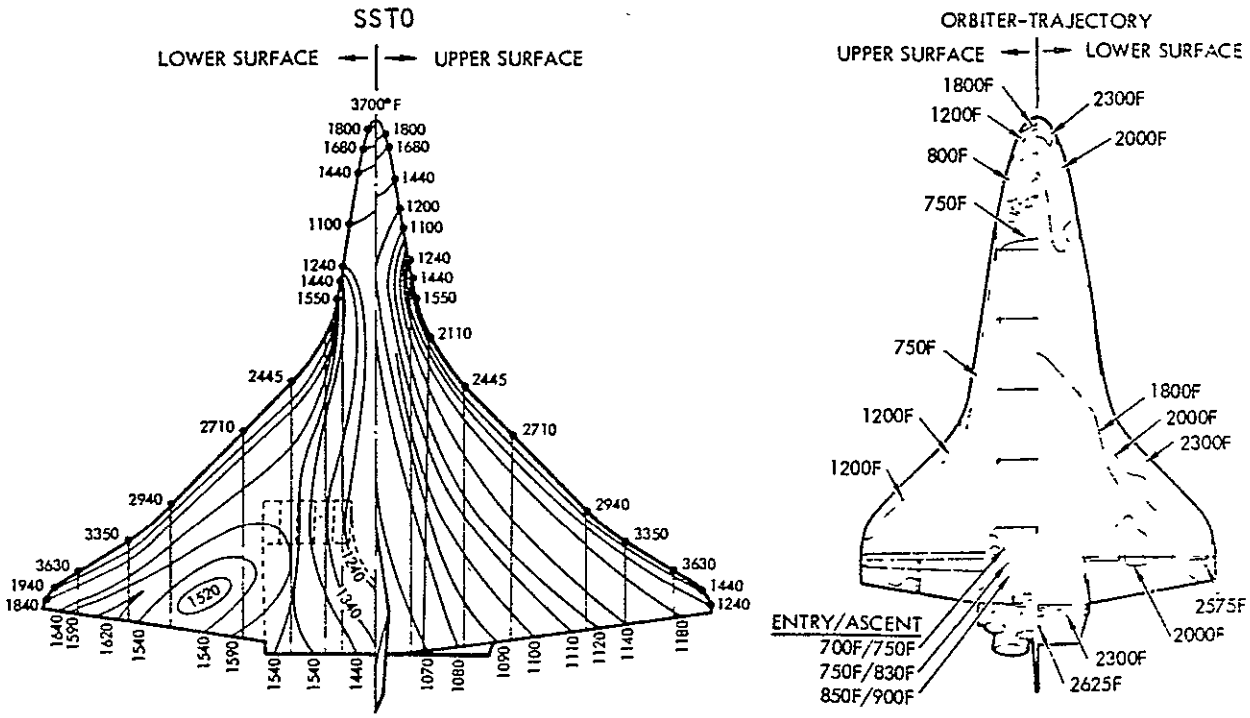


Figure A-14. Isotherms of Peak Surface Temperatures During Ascent

Structural heating analyses include: (a) typical variations of heat leak rate (BTU/ft^2-hr) and total heat flux (BTU/ft^2) as a function of HRSI tile thickness for typical LH_2 upper and lower wing tank surface locations; (b) variation of bondline temperatures versus tile maximum temperature to thickness ratio for RSI tile insulation, including bondline temperatures for the dry, wingtip ullage tank, the wetted lower surface of the LH_2 tank, and the dry upper surface of the LH_2 tank; and (c) typical thermal response as a function of launch trajectory exposure time of the insulation system.

Figure A-15 shows HRSI tile thickness profiles for bondline temperatures of 350°F. Preliminary data indicate that the titanium aluminide system described in the TPS section of this report may be lighter than the RSI tile for the SSTO TPS system due to the low average temperature (1000°F to 1600°F) profiles occurring over 80 and 85 percent of the vehicle exterior surface.

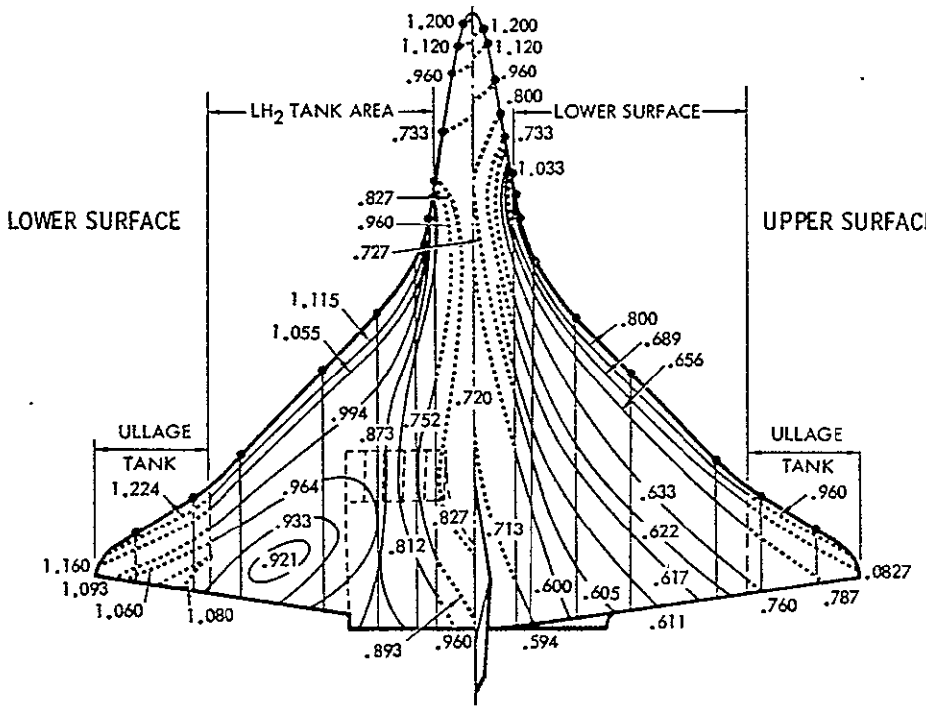


Figure A-15. HRSI Tile Thickness Contours for 350°F Bondline Temperature

A.7 THERMAL PROTECTION SYSTEM

Ceramic coated RSI tile, used on Shuttle, and metallic truss core sandwich structure, developed for the B-1 bomber, were investigated as potential thermal protection systems for the SSTO, Figure A-5.

The radiative surface panel consists of a truss core sandwich structure fabricated by superplastic/diffusion bonding process. For temperatures up to 1500/1600°F, the concept utilizes an alloy based on the titanium-aluminum systems which show promise for high-temperature applications currently being developed by the Air Force. For temperatures higher than 1500/1600°F, it is anticipated that an alloy will be available from the dispersion-strengthened superalloys currently being developed for use in gas turbine engines. Flexible supports are designed to accommodate longitudinal thermal expansion while retaining sufficient stiffness to transmit surface pressure loads to the primary structure. Also prominent in metallic TPS designs are expansion joints which must absorb longitudinal thermal growth of the radiative surface, and simultaneuously prevent the ingress of hot boundary layer gases to the panel interior.

The insulation consists of flexible thermal blankets, often encapsulated in foil material to prevent moisture absorption. The insulation protects the primary load-carrying structure from the high external temperature.

During the past two years, Rockwell and Pratt and Whitney Aircraft have participated in an Air Force Materials Laboratory sponsored program, F33615-75-C-1167, directed toward the exploitation of Ti_3Al base alloy systems. The titanium aluminide intermetallic compounds based on the compositions Ti_3Al (α_2) and TiAl (γ) which form the binary Ti-Al alloys have been shown to have attractive elevated-temperature strength and high modulus/density ratios.

Titanium hardware of complex configurations have been developed, utilizing a process which combines superplastic forming and diffusion bonding (SPF/DB). This Rockwell proprietary process has profound implications for titanium fabrication technology, per se. In addition, the unprecedented low-cost hardware it generates promises to revolutionize the design of airframe structure. The versatile nature of the process may be shown by the nature of the complex deep-drawn structure and sandwich structure with various core configurations which have been fabricated. This manufacturing method and the design freedom it affords offer a solution to the high cost of aircraft structure. Manufacturing feasibility and cost and weight savings potential of these processes have been established through both IR&D efforts at Rockwell and Air Force contracts. These structures may be used for engine cowling, landing gear doors, etc., in addition to providing major TPS components.

Unit masses of the SSTO TPS concept, state-of-the-art TPS hardware and advanced thermal-structural designs are compared with the unit mass of the orbiter RSI in Figure A-16. The unit mass of the RSI includes the tiles, the strain isolator pad, and bonding material. The hashed region shown for the RSI mass is indicative of insulation thickness variations necessary to maintain mold line over the bottom surface of the orbiter. The RSI is required to prevent the primary structure temperature from exceeding 350°F. The unit masses of the metallic TPS are plotted at their corresponding maximum use temperatures. The advanced designs are seen to be competitive with the directly bonded RSI.

A.8 STRUCTURAL ANALYSIS

The multi-cell wing tanks provide a structure which is capable of sustaining pressure while, at the same time, reacting aerodynamic loads. The tanks are sized based on ullage pressures of 32-34 psia (LH_2) and 22-22 psia (LOX). Maximum wing bending occurs at about Mach 1.2. The LH_2 and LOX wing tanks are the major load path for reacting these loads. The wing also supports the air-breather engine system.

The primary wing attachment is to the cargo bay structure. The cargo bay aft section, in turn, is connected to the LH_2 tank. The LH_2 interconnects the cargo bay, aft portions of the wing, the vertical surface, and the rocket engine thrust structure.

An ultimate factor of safety of 1.50 was used in the analysis. The prime driver in the structural sizing of the multi-cell wing tanks is the bending moment resulting from air loads at Mach 1.2. The net bending moment on the

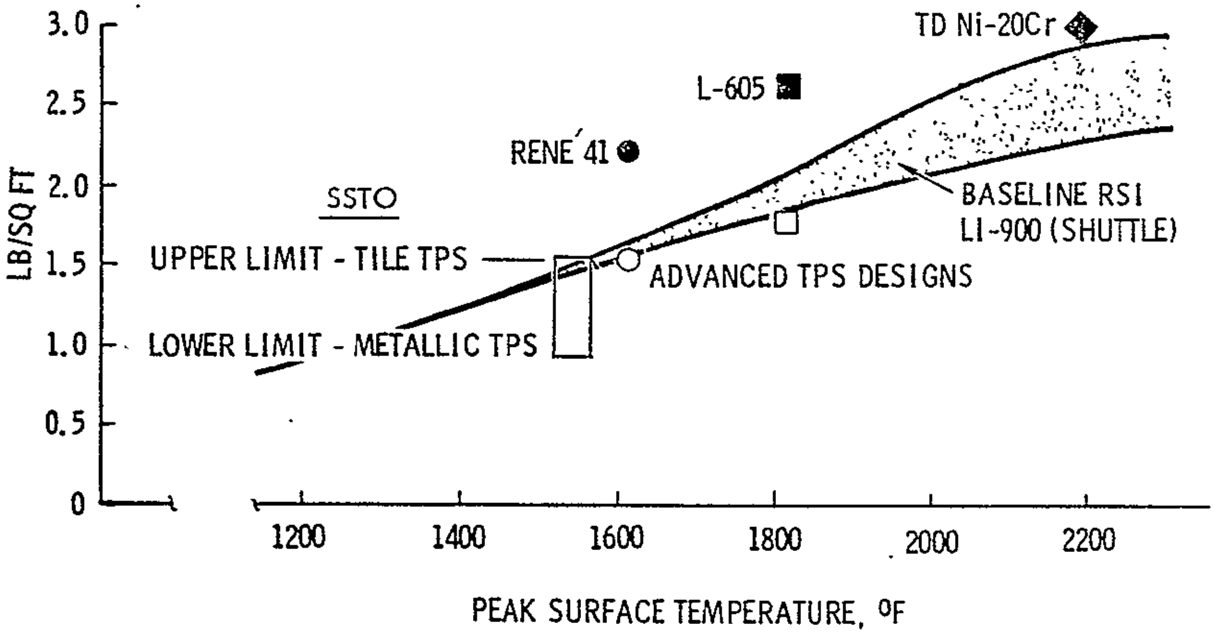
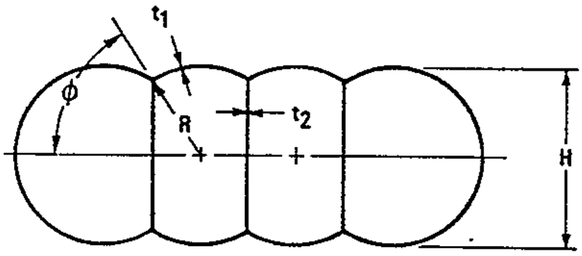


Figure A-16. Unit Mass of TPS Designs

wing is the difference between the lift moment and the relieving moment due to LOX remaining in the wing. Trades were performed to determine the structural wing weights required to sustain these bending moments plus internal pressure. An intermediate location was chosen for LOX propellant where lift moment ~2 times relieving moment. Locating LOX outboard results in a lower net flight bending moment, but the critical design condition then becomes prelaunch under full propellant loading. To sustain this prelaunch bending moment, the wing weight would be in excess of 200,000 lb.

The wing LH_2 tank was designed to sustain the loads from both internal pressure and wing bending. Al 2219-T87 was chosen for the tank material on the basis of high strength at cryogenic temperatures, fracture toughness, and weldability. Loads resulting from wing bending moments are dominant in determining membrane thickness, which is based on a maximum tank ullage pressure of 34 psia, and an ultimate factor of safety of 1.50. Figure A-17 shows material thickness versus wing station due to pressure and wing bending. The column showing bending only relates to wing-bending contribution, not an unpressurized wing design.

The fuselage LH_2 tank is the primary load path for reacting total vehicle mass inertias during the maximum acceleration condition (3.0 g). Approximately 27 percent of the propellant remains at that time. The tank has a twin-cone "Siamese" configuration which is required in order to fit in the fuselage at maximum propellant volume. The forward end of the tank is cylindrical, while the aft end is closed out with a double modified ellipsoidal shell. The bulkheads react the internal pressures while the sidewall carries pressure and axial compression loads. The bulkheads are monocoque construction while the sidewall is an integral skin-stringer with ring frames construction. Tank



STA* (FT)	H_{NOM} (IN.)	PRESSURE REQUIREMENT $t_1 = t_2$** (IN.)	BENDING ONLY t_1 (IN.)	BENDING + PRESSURE t_1 (IN.)
10.9	240	0.066	0.021	0.087
23.0	146	0.040	0.076	0.116
54.0	110	0.031	0.092	0.123
107.0	48	0.014	0.120	0.134

*DISTANCE FROM VEHICLE ℄
**FOR Ø = 60 DEG ONLY

Figure A-17. Material Thickness Versus Wing Station

configuration and bulkhead membrane and sidewall "smeared" thickness requirements to sustain the internal pressure and axial compression loads have been determined. The structural design of all cryo tanks is based on cryogenic temperature material properties and allowables.

A.9 MASS PROPERTIES

SSTO mass properties are dominated by the tri-delta wing structure, the thermal protection system and the airbreather and rocket propulsion system. The initial reference vehicle data, shown in Table A-2, were generated by Rockwell during the period of December 1977 - January 1978. These data were reviewed by NASA MSFC/LaRC during February and March 1978, resulting in two extremes of mass estimates. A reassessment by Rockwell during May produced the final reference vehicle data. The data presented in this report are considered to be reasonably achievable targets. The technology items coded on Figure A-1 require study in greater depth and degree of sophistication to confirm SSTO mass property data.

Table A-2. SSTO Weight Summary

ITEM DESCRIPTION	ROCKWELL	MSFC		ROCKWELL
	INITIAL REFERENCE VEHICLE	NORMAL TECHNOLOGY	ACCELER TECHNOLOGY	FINAL REFERENCE VEHICLE
AIRFRAME, AEROSURFACES, TANKS AND TPS	367,000	458,000	249,000	370,000
LANDING GEAR	27,700	53 000	39,000	27,700
ROCKET PROPULSION	63,700	40,000	40,000	71,700
AIRBREATHER PROPULSION	148,000	200,000	148,000	140,000
RCS PROPULSION	4 000	16,000	11,000	10,000
OMS PROPULSION	1,200	9,000	7,000	5,000
OTHER SYSTEMS	35,500	41,000	22 000	37,800
SUBTOTAL	647,100	817,000	516,000	662,200
10% GROWTH		81,700	51,600	66 220
TOTAL INERT WEIGHT (DRY WEIGHT)	647,100	898,700	567,600	728,420
USEFUL LOAD (FLUIDS, RESERVES, ETC)	47,400	—	—	47,400
INERT WEIGHT & USEFUL LOAD	694,500	—	—	775 820
PAYLOAD WEIGHT	107,200			196,580
ORBITAL INSERTION WEIGHT	801,700	—	—	972,400
PROPELLANT ASCENT	3 438,080			4,027,600
GLOW (POST JETTISON LAUNCH GEAR)	4,239,780	—	—	5,000,000

300 NMI EQUATORIAL ORBIT
NOTE- THIS VEHICLE HAS 51,000 CU FT EXCESS PROPELLANT TANK VOLUME SEE WEIGHT IN ORBIT SUMMARY

300 NMI 28.5° INCLINED ORBIT

REFERENCES

1. *Estimated Performance of a Mach 8.0 Hydrogen Fueled Turbofan Ramjet*, Pratt and Whitney Aircraft Report STFRV-230A (January 1965)

2. *Air-Turborocket Application Study*, Aerojet General Corporation (December 1964)

3. *Final Report and Users Manual for the Hypersonic Airbreathing Propulsion Computer Program, NASA Contract NAS2-2985*, North American Aviation Reports NA66-479 and NA66-530 (May 1966)

4. *Airbreathing Engine/Rocket Trajectory Optimization Study*, Virgil K. Smith, University of Alabama (August 1978)

5. *Feasibility Study of Reusable Aerodynamic Space Vehicle*, SAMSO-TR-76-223, Boeing Aerospace Company (November 1976)

APPENDIX B. HLLV REFERENCE VEHICLE TRAJECTORY AND TRADE STUDY DATA

APPENDIX B

HLLV REFERENCE VEHICLE TRAJECTORY AND TRADE STUDY DATA

APPENDIX B

HLLV REFERENCE VEHICLE TRAJECTORY AND TRADE STUDY DATA

B.0 INTRODUCTION

The reference heavy lift launch vehicle trajectory data and a summary of the various trade studies performed are contained in this appendix. The several trade options include:

- First and Second Stage Engine Throttling
- First Stage Propellant Weight Sensitivity
- Second Stage Propellant Weight Sensitivity
- Lift-off Thrust-to-Weight Sensitivity
- Alternate First Stage Propellants (LOX/CH_4 and LOX/LH_2)

With the exception of the engine throttling trades, all trajectories assumed 100% throttling by the first stage engines (i.e., second stage engines operate at maximum thrust throughout the parallel burn ascent phase) in order to stay within maximum allowable load factor and dynamic pressure, 3 g and 650 psf respectively.

The engine throttling study shows little effect on vehicle payload capability when doing 100% of the throttling with either stage. All intermediate options (i.e., partial throttling of both stages) shows a degradation in payload capability.

The first stage propellant weight sensitivity analyses show an improvement in glow/payload weight ratio (smaller) as first stage propellant weight is increased, however, the staging velocity exceeds the capability of a heat sink booster. The second stage propellant weight sensitivity indicates an opposite effect to the first stage data.

By combining the effects of throttling of second stage only and increasing first stage propellant weight could result in a 10-15% improvement over the reference HLLV configuration.

The alternate propellant trades, LOX/CH_4 and LOX/LH_2, show 7% and 37% increased performance over the reference HLLV configuration. The LOX/LH_2 configuration, however, becomes extremely large (volume) and less cost effective because of handling and propellant costs. The LOX/CH_4 booster appears to be a viable option.

B.1 HLLV REFERENCE VEHICLE TRAJECTORY

This section contains the tabulated reference vehicle characteristics and trajectory data. The nominal and abort modes [once around and second stage return to launch site (RTLS)] data are included. Because an adaptation of the space shuttle transportation system scaling program was used, certain vehicle parameters are listed under headings of "External Tank" and "Solid Rocket Booster."

The first two pages of the tabulated data list the pertinent ground rules and assumptions employed in making the computer run. In the list of "Vehicle Characteristics" (third page), the structure weight given refers to the booster total inert weight plus residuals and reserves but exclusive of flyback propellant. The propellant value given is the total usable ascent propellant loaded in the first stage (i.e., includes that propellant crossfeed to the second stage during first stage burn).

In the summary weight statement (fourth page), the "Orbiter" and "External Tank" listings refer to second stage weights. The "External Tank" values apply to main propulsion residuals and reserves. The total usable propellant (External Tank) is the total propellant burned in the second stage (i.e., propellant loaded plus crossfeed from first stage). The usable SRM propellant listing is the total propellant burned through the first stage engines. To determine the amount of crossfeed propellant, the usable SRM propellant may be subtracted from the total propellant loaded in the second stage which is given under Vehicle Characteristics, third page of data.

CRT plots of significant HLLV parameters are included following the tabulated data.

The reference vehicle has a gross liftoff weight of 7,135,492 kg (15,731,068 lb) and a payload capacity of 231,195 kg (509,653 lb).

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GENERAL ASCENT TRAJECTORY AND SIZING PROGRAM BY R.L.POWELL

DATE - 01/15/79                              TIME - 18:18:27

SATELLITE POWER SYSTEM (SPS) CONCEPT DEFINITION STUDY
TWO-STAGE VERTICAL TAKE-OFF HORIZONTAL LANDING HLLV CONCEPT
BOTH STAGES HAVE FLYBACK CAPABILITY TO LAUNCH SITE (KSC)
FIRST STAGE HAS AIRBREATHER FLYBACK AND LANDING CAPABILITY
FLYBACK PROPELLANT HAS A SPECIFIC FUEL CONSUMPTION OF 3500 SEC
SECOND STAGE USES THE ABORT-ONCE-AROUND FLYBACK MODE (AOA)
FIRST STAGE HAS LOX/RP/LH2 TRIPROPELLANT SYSTEM
  WITH H2 COOLED HIGH PC ENGINES (VACUUM ISP = 352.3 SEC)
SECOND STAGE USES LOX/LH2 PROPELLANT WITH VACUUM ISP  466.7 SEC
THE DESIGN PAYLOAD SHALL BE 500 KLB INTO A CIRCULAR ORBIT OF
  270 N. MILES AND AN INERTIAL INCLINATION OF 31.6 DEGREES
ASCENT SHAPED TO THE NOMINAL ASCENT MISSION
MECO CONDITIONS ARE TO A THEORETICAL ORBIT OF 169.22 N.MILES
  BY 50.42 N. MILES (COASTS TO APOGEE OF 160 N.MILES)
ON-ORBIT DELTA VELOCITY REQUIREMENT OF 1110 FEET/SECOND
RCS SYSTEM SIZED FOR A DELTA VELOCITY REQMT OF 220 FEET/SECOND
THE VEHICLE SIZED FOR A THRUST/WEIGHT RATIO AT LIFT-OFF OF 1.30
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MAXIMUM AXIAL LOAD FACTOR DURING ASCENT IS 3.0 G'S

TRAJECTORY HAS A MAXIMUM AERO PRESSURE OF 650 LBS/FT2

MAXIMUM AERO PRESSURE AT STAGING LIMITED TO 25 LBS/FT2

DIRECT ENTRY FROM 270 N.MILES ASSUMMED (DELTA V = 415 FT/SEC)

FLIGHT PERFORMANCE RESERVE = 0.75% TOTAL CHAC ASCENT VELOCITY

WEIGHT SCALING PER ROCKWELL IR AND D HLLV STUDIES

A WEIGHT GROWTH ALLOWANCE OF 15% IS ASSUMMED FOR BOTH STAGES

FIRST STAGE BURNS 7995060 POUNDS OF ASCENT PROPELLANT

SECOND STAGE (ORBITER) ENGINES BURN 5092633 LBS OF PROPELLANT

SECOND STAGE DRY WEIGHT WITHOUT PAYLOAD EQUALS 727620 LBS

SECOND STAGE THRUST LEVEL @ STAGING EQUALS 4750000 LBS

SECOND STAGE ASSUMES 4 ENGINES FOR ASCENT WITH 1 OUT FOR ABORT

SECOND STAGE EPL THRUST LEVEL FOR ABORT IS 112 % FULL POWER

SECOND STAGE OVERALL BOOSTER MASS FRACTION = 0.8489 W/O MARGIN

SECOND STAGE WEIGHT BREAKDOWN :

RESIDUAL WEIGHT = 2070 POUNDS

RESERVES WEIGHT = 3300 POUNDS

RCS PROP WEIGHT = 18280 POUNDS

BURN-OUT ALTITUDE AT SECOND STAGE THRUST TERMINATION = 50 N. MILES

ADVANCED TECHNOLOGY WILL BE COMPATABLE WITH THE YEARS 1990'S ON

THIS RUN IS MADE WITH A CONSTANT KICK ANGLE - LOX/RP-1 BASELINE

VEHICLE CHARACTERISTICS (NOMINAL MISSION) CASE 65

STAGE	1	2	3
GROSS STAGE WEIGHT,(LB)	15731068.0	4891645.0	4817477.0
GROSS STAGE THRUST/WEIGHT	1.300	0.971	0.986
THRUST ACTUAL,(LB)	20450352.0	4750000.0	4750000.0
ISP VACUUM,(SEC)	370.886	466.700	466.700
STRUCTURE,(LB)	1045488.9	0.0	806009.0
PROPELLANT,(LB)	9607069.0	74168.0	3406460.0
PERF. FRAC.,(NU)	0.6107	0.0152	0.7071
PROPELLANT FRAC.,(NUB)	0.9019	1.0000	0.8087
BURNOUT TIME,(SEC)	158.387	165.674	502.194
BURNOUT VELOCITY,(FT/SEC)	8238.750	8407.051	25954.109
BURNOUT GAMMA,(DEGREES)	14.396	13.338	0.187
BURNOUT ALTITUDE,(FT)	180948.6	195447.2	319657.5
BURNOUT RANGE,(NM)	48.5	56.6	809.7
IDEAL VELOCITY,(FT/SEC)	10960.3	11189.7	29628.0

INJECTION VELOCITY,(FT/SEC)	0.0	FLYBACK RANGE(NM)	211.9
INJECTION PROPELLANT,(LB)	0.0	FLYBACK PROP(LBS)	186864.9
ON ORBIT DELTA-V,(FT/SEC)	1083.5		
ON ORBIT PROPELLANT,(LB)	95354.1		
ON ORBIT ISP,(SEC)	466.7		

THETA= 28.14 PITCH RATE= 0.00192 ATTEMPTS TO CONVERGE= 3

PAYLOAD,(LB) 509653.0

SUMMARY WEIGHT STATEMENT (NOMINAL MISSION) CASE 65

ORBITER WEIGHT BREAKDOWN		
DRY WEIGHT	727620.000	POUNDS
PERSONNEL	3000.000	POUNDS
RESIDUALS	2070.000	POUNDS
RESERVES	3300.000	POUNDS
IN-FLIGHT LOSSES	10439.000	POUNDS
ACPS PROPELLANT	18280.000	POUNDS
OMS PROPELLANT	95354.125	POUNDS
PAYLOAD	509653.000	POUNDS
BALLAST FOR CG CONTROL	0.0	POUNDS
OMS INSTALLATION KITS	0.0	POUNDS
PAYLOAD MODS	0.0	POUNDS
TOTAL END BOOST (ORBITER ONLY)	1369716.00	POUNDS
OMS BURNED DURING ASCENT	0.0	POUNDS
ACPS BURNED DURING ASCENT	0.0	POUNDS
EXTERNAL MAIN TANK		
TANK DRY WEIGHT	2640.000	POUNDS
RESIDUALS	17730.000	POUNDS
PROPELLANT BIAS	(2640.000)	POUNDS
PRESSURANT	(2120.000)	POUNDS
TANK AND LINES	(9320.000)	POUNDS
ENGINES	(3650.000)	POUNDS
FLIGHT PERFORMANCE RESERVE	20930.000	POUNDS
UNBURNED PROPELLANT (MAIN TANK)	0.0	POUNDS
TOTAL END BOOST (EXTERNAL TANK)	41300.000	POUNDS
USABLE PROPELLANT (EXTERNAL TANK)	5092633.00	POUNDS
FLYBACK PROPELLANT (FIRST STAGE)	186864.937	POUNDS
SOLID ROCKET MOTOR (FIRST STAGE)	9040548.00	POUNDS
SRM CASE WEIGHT(2)	1045488.87	POUNDS
SRM STRUCTURE & RCVY WEIGHT	0.0	POUNDS
SRM INERT STAGING WEIGHT	1045488.87	POUNDS
USABLE SRM PROPELLANT	7995060.00	POUNDS
TOTAL GROSS LIFT-OFF WEIGHT (GLOW)	15731068.0	POUNDS

TIME	VREL	ALT	GAMMA	QBAR	LOAD FACTOR
W	VDOT	GDT	VGRAV	VDRG	THRUST
ALPHA	MACH	LIFT	RANGE	DRAG	THROTTLE
THRUST 1	VAC THRUST 1	THROTTLE 1	THRUST 2	VAC THRUST 2	THROTTLE 2
0.0	0.0	0.183000E+03	0.900000E+02	0.0	0.130101E+01
0.157311E+08	0.974443E+01	0.0	0.0	0.0	0.204662E+08
0.0	0.0	0.0	0.0	0.0	0.100000E+01
0.170419E+08	0.182167E+08	0.100000E+01	0.342432E+07	0.473192E+07	0.100000E+01
0.100000E+01	0.982707E+01	0.187900E+03	0.900000E+02	0.110324E+00	0.130616E+01
0.156692E+08	0.991005E+01	0.0	0.321948E+02	0.155268E-03	0.204666E+08
0.0	0.864182E-02	0.941394E+02	0.0	0.226589E+03	0.100000E+01
0.170421E+08	0.182167E+08	0.100000E+01	0.342454E+07	0.473192E+07	0.100000E+01
0.199999E+01	0.198209E+02	0.202710E+03	0.900000E+02	0.448634E+00	0.131137E+01
0.156073E+08	0.100779E+02	0.0	0.643897E+02	0.125196E-02	0.204679E+08
0.0	0.174313E-01	0.382819E+03	0.0	0.916862E+03	0.100000E+01
0.170427E+08	0.182167E+08	0.100000E+01	0.342522E+07	0.473192E+07	0.100000E+01
0.299999E+01	0.299836E+02	0.227598E+03	0.900000E+02	0.102594E+01	0.131665E+01
0.155454E+08	0.102480E+02	0.0	0.965844E+02	0.427170E-02	0.204701E+08
0.0	0.263713E-01	0.875430E+03	0.0	0.208667E+04	0.100000E+01
0.170437E+08	0.182167E+08	0.100000E+01	0.342635E+07	0.473192E+07	0.100000E+01
0.399998E+01	0.403174E+02	0.262734E+03	0.900000E+02	0.185321E+01	0.132200E+01
0.154836E+08	0.104202E+02	0.0	0.128779E+03	0.102408E-01	0.204731E+08
0.0	0.354650E-01	0.158134E+04	0.0	0.375194E+04	0.100000E+01
0.170452E+08	0.182167E+08	0.100000E+01	0.342795E+07	0.473192E+07	0.100000E+01
0.499998E+01	0.508246E+02	0.308289E+03	0.900000E+02	0.294136E+01	0.132742E+01
0.154217E+08	0.105947E+02	0.0	0.160974E+03	0.202311E-01	0.204771E+08
0.0	0.447154E-01	0.250987E+04	0.0	0.592866E+04	0.100000E+01
0.170470E+08	0.182167E+08	0.100000E+01	0.343002E+07	0.473192E+07	0.100000E+01

TIME	VREL	ALT	GAMMA	QBAR	LOAD FACTOR
W	VDOT	GDT	VGRAV	VDRG	THRUST
ALPHA	MACH	LIFT	RANGE	DRAG	THROTTLE
THRUST 1	VAC THRUST 1	THROTTLE 1	THRUST 2	VAC THRUST 2	THROTTLE 2
0.599997E+01	0.615073E+02	0.364439E+03	0.900000E+02	0.430121E+01	0.133291E+01
0.153598E+08	0.107713E+02	0.0	0.193168E+03	0.353610E-01	0.204819E+08
0.0	0.541258E-01	0.367022E+04	0.0	0.863285E+04	0.100000E+01
0.170493E+08	0.182167E+08	0.100000E+01	0.343257E+07	0.473192E+07	0.100000E+01
0.699997E+01	0.723679E+02	0.431361E+03	0.900000E+02	0.594342E+01	0.133846E+01
0.152979E+08	0.109503E+02	0.0	0.225362E+03	0.567957E-01	0.204877E+08
0.0	0.636992E-01	0.507151E+04	0.0	0.118805E+05	0.100000E+01
0.170521E+08	0.182167E+08	0.100000E+01	0.343560E+07	0.473192E+07	0.100000E+01
0.799996E+01	0.834085E+02	0.509233E+03	0.900000E+02	0.787844E+01	0.134409E+01
0.152361E+08	0.111314E+02	0.0	0.257556E+03	0.857485E-01	0.204943E+08
0.0	0.734391E-01	0.672267E+04	0.0	0.156876E+05	0.100000E+01
0.170552E+08	0.182167E+08	0.100000E+01	0.343912E+07	0.473192E+07	0.100000E+01
0.899995E+01	0.946314E+02	0.598236E+03	0.900000E+02	0.101165E+02	0.134978E+01
0.151742E+08	0.113149E+02	0.0	0.289748E+03	0.123481E+00	0.205020E+08
0.0	0.833487E-01	0.863243E+04	0.0	0.200699E+05	0.100000E+01
0.170588E+08	0.182167E+08	0.100000E+01	0.344314E+07	0.473192E+07	0.100000E+01
0.999995E+01	0.106039E+03	0.698555E+03	0.900000E+02	0.126676E+02	0.135555E+01
0.151123E+08	0.115007E+02	0.0	0.321941E+03	0.171304E+00	0.205105E+08
0.0	0.934315E-01	0.108093E+05	0.0	0.250430E+05	0.100000E+01
0.170629E+08	0.182167E+08	0.100000E+01	0.344766E+07	0.473192E+07	0.100000E+01
0.100000E+02	0.106040E+03	0.698560E+03	0.896315E+02	0.126678E+02	0.135555E+01
0.151123E+08	0.115013E+02	-0.742977E-01	0.321942E+03	0.171307E+00	0.205105E+08
0.0	0.934319E-01	0.108094E+05	0.0	0.250433E+05	0.100000E+01
0.170629E+08	0.182167E+08	0.100000E+01	0.344766E+07	0.473192E+07	0.100000E+01

TIME	VREL	ALT	GAMMA	QBAR	LOAD FACTOR
W	VDOT	GDT	VGRAV	VDRG	THRUST
ALPHA	MACH	LIFT	RANGE	DRAG	THROTTLE
THRUST 1	VAC THRUST 1	THROTTLE 1	THRUST 2	VAC THRUST 2	THROTTLE 2
0.120000E+02	0.129420E+03	0.933917E+03	0.894573E+02	0.187478E+02	0.136729E+01
0.149886E+08	0.118806E+02	-0.100452E+00	0.386326E+03	0.302954E+00	0.205305E+08
0.0	0.114133E+00	0.159975E+05	0.353495E-03	0.368251E+05	0.100000E+01
0.170724E+08	0.182167E+08	0.100000E+01	0.345820E+07	0.473192E+07	0.100000E+01
0.140000E+02	0.153569E+03	0.121679E+04	0.892266E+02	0.261905E+02	0.137931E+01
0.148648E+08	0.122699E+02	-0.130656E+00	0.450705E+03	0.491951E+00	0.205544E+08
0.0	0.135570E+00	0.223484E+05	0.958813E-03	0.511527E+05	0.100000E+01
0.170836E+08	0.182167E+08	0.100000E+01	0.347078E+07	0.473192E+07	0.100000E+01
0.160000E+02	0.178507E+03	0.154872E+04	0.889321E+02	0.350603E+02	0.139162E+01
0.147411E+08	0.126696E+02	-0.164246E+00	0.515079E+03	0.750575E+00	0.205822E+08
0.0	0.157774E+00	0.299170E+05	0.194111E-02	0.681403E+05	0.100000E+01
0.170968E+08	0.182167E+08	0.100000E+01	0.348541E+07	0.473192E+07	0.100000E+01
0.180000E+02	0.204256E+03	0.193129E+04	0.885675E+02	0.454118E+02	0.140421E+01
0.146173E+08	0.130804E+02	-0.200568E+00	0.579444E+03	0.109190E+01	0.206139E+08
0.0	0.180776E+00	0.387499E+05	0.346320E-02	0.878946E+05	0.100000E+01
0.171118E+08	0.182166E+08	0.100000E+01	0.350211E+07	0.473192E+07	0.100000E+01
0.199999E+02	0.230838E+03	0.236610E+04	0.881282E+02	0.572882E+02	0.141710E+01
0.144936E+08	0.135034E+02	-0.238996E+00	0.643794E+03	0.152977E+01	0.206495E+08
0.0	0.204610E+00	0.488840E+05	0.572963E-02	0.110512E+06	0.100000E+01
0.171286E+08	0.182166E+08	0.100000E+01	0.352086E+07	0.473192E+07	0.100000E+01
0.219999E+02	0.258279E+03	0.285476E+04	0.876104E+02	0.707194E+02	0.143027E+01
0.143698E+08	0.139394E+02	-0.278948E+00	0.708124E+03	0.207881E+01	0.206890E+08
0.0	0.229311E+00	0.603448E+05	0.899030E-02	0.136077E+06	0.100000E+01
0.171473E+08	0.182166E+08	0.100000E+01	0.354166E+07	0.473192E+07	0.100000E+01

TIME	VREL	ALT	GAMMA	QBAR	LOAD FACTOR
W	VDOT	GDT	VGRAV	VDRG	THRUST
ALPHA	MACH	LIFT	RANGE	DRAG	THROTTLE
THRUST 1	VAC THRUST 1	THROTTLE 1	THRUST 2	VAC THRUST 2	THROTTLE 2
0.239999E+02	0.286606E+03	0.339893E+04	0.870116E+02	0.857205E+02	0.144372E+01
0.142461E+08	0.143898E+02	-0.319875E+00	0.772425E+03	0.275436E+01	0.207323E+08
0.0	0.254919E+00	0.731452E+05	0.135434E-01	0.164662E+06	0.100000E+01
0.171678E+08	0.182166E+08	0.100000E+01	0.356448E+07	0.473192E+07	0.100000E+01
0.259998E+02	0.315850E+03	0.400025E+04	0.863305E+02	0.102291E+03	0.145746E+01
0.141223E+08	0.148558E+02	-0.361273E+00	0.836684E+03	0.357247E+01	0.207793E+08
0.0	0.281474E+00	0.872845E+05	0.197378E-01	0.196321E+06	0.100000E+01
0.171901E+08	0.182166E+08	0.100000E+01	0.358926E+07	0.473192E+07	0.100000E+01
0.279998E+02	0.346042E+03	0.466035E+04	0.855665E+02	0.120412E+03	0.147148E+01
0.139986E+08	0.153392E+02	-0.402676E+00	0.900886E+03	0.454980E+01	0.208300E+08
0.0	0.309023E+00	0.102747E+06	0.279755E-01	0.231094E+06	0.100000E+01
0.172141E+08	0.182166E+08	0.100000E+01	0.361597E+07	0.473192E+07	0.100000E+01
0.299998E+02	0.377226E+03	0.538087E+04	0.847201E+02	0.140052E+03	0.148612E+01
0.138748E+08	0.158526E+02	-0.443649E+00	0.965015E+03	0.569766E+01	0.208842E+08
0.0	0.337622E+00	0.119506E+06	0.387124E-01	0.264198E+06	0.100000E+01
0.172397E+08	0.182166E+08	0.100000E+01	0.364453E+07	0.473192E+07	0.100000E+01
0.319998E+02	0.409474E+03	0.616346E+04	0.837925E+02	0.161184E+03	0.150141E+01
0.137511E+08	0.163989E+02	-0.483767E+00	0.102905E+04	0.700059E+01	0.209418E+08
0.0	0.367358E+00	0.137538E+06	0.524606E-01	0.295400E+06	0.100000E+01
0.172670E+08	0.182166E+08	0.100000E+01	0.367486E+07	0.473192E+07	0.100000E+01
0.339997E+02	0.442840E+03	0.700977E+04	0.827859E+02	0.183748E+03	0.151704E+01
0.136273E+08	0.169701E+02	-0.522656E+00	0.109295E+04	0.846756E+01	0.210025E+08
0.0	0.398302E+00	0.156792E+06	0.697886E-01	0.328968E+06	0.100000E+01
0.172957E+08	0.182166E+08	0.100000E+01	0.370685E+07	0.473192E+07	0.100000E+01

TIME	VREL	ALT	GAMMA	QBAR	LOAD FACTOR
W	VDOT	GDT	VGRAV	VDRG	THRUST
ALPHA	MACH	LIFT	RANGE	DRAG	THROTTLE
THRUST 1	VAC THRUST 1	THROTTLE 1	THRUST 2	VAC THRUST 2	THROTTLE 2
0.359997E+02	0.477374E+03	0.792137E+04	0.817030E+02	0.207661E+03	0.153300E+01
0.135036E+08	0.175681E+02	-0.559987E+00	0.115671E+04	0.101127E+02	0.210662E+08
0.0	0.430531E+00	0.177197E+06	0.913211E-01	0.364857E+06	0.100000E+01
0.173258E+08	0.182166E+08	0.100000E+01	0.374040E+07	0.473192E+07	0.100000E+01
0.379997E+02	0.513133E+03	0.889982E+04	0.805473E+02	0.232826E+03	0.154928E+01
0.133798E+08	0.181949E+02	-0.595465E+00	0.122028E+04	0.119502E+02	0.211326E+08
0.0	0.464132E+00	0.198671E+06	0.117738E+00	0.403004E+06	0.100000E+01
0.173572E+08	0.182166E+08	0.100000E+01	0.377539E+07	0.473192E+07	0.100000E+01
0.399996E+02	0.550176E+03	0.994655E+04	0.793227E+02	0.259129E+03	0.156590E+01
0.132561E+08	0.188524E+02	-0.628830E+00	0.128362E+04	0.139944E+02	0.212015E+08
0.0	0.499201E+00	0.221114E+06	0.149774E+00	0.443330E+06	0.100000E+01
0.173898E+08	0.182166E+08	0.100000E+01	0.381166E+07	0.473192E+07	0.100000E+01
0.419996E+02	0.588531E+03	0.110629E+05	0.780356E+02	0.286405E+03	0.158170E+01
0.131323E+08	0.195056E+02	-0.657884E+00	0.134669E+04	0.162944E+02	0.212725E+08
0.0	0.535818E+00	0.236397E+06	0.188211E+00	0.500546E+06	0.100000E+01
0.174234E+08	0.182166E+08	0.100000E+01	0.384906E+07	0.473192E+07	0.100000E+01
0.439996E+02	0.628216E+03	0.122499E+05	0.766933E+02	0.314461E+03	0.159749E+01
0.130086E+08	0.201816E+02	-0.684006E+00	0.140945E+04	0.189126E+02	0.213453E+08
0.0	0.574062E+00	0.250185E+06	0.233866E+00	0.563649E+06	0.100000E+01
0.174579E+08	0.182166E+08	0.100000E+01	0.388743E+07	0.473192E+07	0.100000E+01
0.459995E+02	0.669275E+03	0.135086E+05	0.753018E+02	0.343115E+03	0.161325E+01
0.128848E+08	0.208809E+02	-0.707110E+00	0.147185E+04	0.218839E+02	0.214196E+08
0.0	0.614051E+00	0.262292E+06	0.287597E+00	0.632537E+06	0.100000E+01
0.174931E+08	0.182166E+08	0.100000E+01	0.392659E+07	0.473192E+07	0.100000E+01

TIME W ALPHA THRUST 1	VREL VDOT MACH VAC THRUST 1	ALT GDT LIFT THROTTLE 1	GAMMA VGRAV RANGE THRUST 2	QBAR VDRG DRAG VAC THRUST 2	LOAD FACTOR THRUST THROTTLE THROTTLE 2
0.479995E+02	0.711756E+03	0.148396E+05	0.738672E+02	0.372167E+03	0.162895E+01
0.127611E+08	0.216031E+02	-0.727110E+00	0.153384E+04	0.252444E+02	0.214951E+08
0.0	0.655916E+00	0.272362E+06	0.350294E+00	0.707378E+06	0.100000E+01
0.175288E+08	0.182166E+08	0.100000E+01	0.396635E+07	0.473192E+07	0.100000E+01
0.499995E+02	0.755704E+03	0.162437E+05	0.723957E+02	0.401407E+03	0.164452E+01
0.126373E+08	0.223474E+02	-0.743961E+00	0.159537E+04	0.290324E+02	0.215714E+08
0.0	0.699804E+00	0.280035E+06	0.422877E+00	0.788309E+06	0.100000E+01
0.175649E+08	0.182166E+08	0.100000E+01	0.400652E+07	0.473192E+07	0.100000E+01
0.519995E+02	0.801161E+03	0.177210E+05	0.708937E+02	0.430604E+03	0.165995E+01
0.125136E+08	0.231126E+02	-0.757658E+00	0.165638E+04	0.332876E+02	0.216481E+08
0.0	0.745873E+00	0.284948E+06	0.506293E+00	0.875421E+06	0.100000E+01
0.176012E+08	0.182166E+08	0.100000E+01	0.404691E+07	0.473192E+07	0.100000E+01
0.539994E+02	0.848168E+03	0.192718E+05	0.693674E+02	0.459514E+03	0.167518E+01
0.123898E+08	0.238974E+02	-0.768229E+00	0.171683E+04	0.380514E+02	0.217248E+08
0.0	0.794296E+00	0.286743E+06	0.601507E+00	0.968748E+06	0.100000E+01
0.176375E+08	0.182166E+08	0.100000E+01	0.408733E+07	0.473192E+07	0.100000E+01
0.559994E+02	0.896764E+03	0.208960E+05	0.678231E+02	0.487868E+03	0.169018E+01
0.122661E+08	0.247003E+02	-0.775735E+00	0.177666E+04	0.433663E+02	0.218012E+08
0.0	0.845252E+00	0.285068E+06	0.709499E+00	0.106825E+07	0.100000E+01
0.176736E+08	0.182166E+08	0.100000E+01	0.412758E+07	0.473192E+07	0.100000E+01
0.579994E+02	0.947024E+03	0.225933E+05	0.662667E+02	0.515419E+03	0.170633E+01
0.121423E+08	0.255642E+02	-0.780230E+00	0.183583E+04	0.492330E+02	0.218769E+08
0.0	0.898970E+00	0.279595E+06	0.831257E+00	0.115696E+07	0.100000E+01
0.177095E+08	0.182166E+08	0.100000E+01	0.416747E+07	0.473192E+07	0.100000E+01

TIME W ALPHA THRUST 1	VREL VDOT MACH VAC THRUST 1	ALT GDT LIFT THROTTLE 1	GAMMA VGRAV RANGE THRUST 2	QBAR VDRG DRAG VAC THRUST 2	LOAD FACTOR THRUST THROTTLE THROTTLE 2
0.599993E+02	0.999034E+03	0.243635E+05	0.647043E+02	0.541900E+03	0.172238E+01
0.120186E+08	0.264481E+02	-0.781796E+00	0.189428E+04	0.556420E+02	0.219516E+08
0.0	0.955692E+00	0.270013E+06	0.967785E+00	0.124981E+07	0.100000E+01
0.177448E+08	0.182166E+08	0.100000E+01	0.420682E+07	0.473192E+07	0.100000E+01
0.619993E+02	0.105285E+04	0.262061E+05	0.631414E+02	0.566972E+03	0.173877E+01
0.118948E+08	0.273648E+02	-0.781096E+00	0.195197E+04	0.626114E+02	0.220249E+08
0.0	0.101563E+01	0.259646E+06	0.112009E+01	0.134117E+07	0.100000E+01
0.177795E+08	0.182166E+08	0.100000E+01	0.424545E+07	0.473192E+07	0.100000E+01
0.639993E+02	0.110844E+04	0.281203E+05	0.615805E+02	0.590182E+03	0.175296E+01
0.117711E+08	0.282312E+02	-0.779493E+00	0.200886E+04	0.702285E+02	0.220966E+08
0.0	0.107890E+01	0.256421E+06	0.128917E+01	0.146094E+07	0.100000E+01
0.178134E+08	0.182166E+08	0.100000E+01	0.428319E+07	0.473192E+07	0.100000E+01
0.659992E+02	0.116577E+04	0.301051E+05	0.600250E+02	0.611013E+03	0.176658E+01
0.116473E+08	0.290989E+02	-0.775725E+00	0.206492E+04	0.786063E+02	0.221662E+08
0.0	0.114559E+01	0.250355E+06	0.147604E+01	0.158874E+07	0.100000E+01
0.178464E+08	0.182166E+08	0.100000E+01	0.431987E+07	0.473192E+07	0.100000E+01
0.679992E+02	0.122481E+04	0.321591E+05	0.584790E+02	0.628941E+03	0.177889E+01
0.115236E+08	0.299422E+02	-0.769983E+00	0.212009E+04	0.878297E+02	0.222335E+08
0.0	0.121574E+01	0.241331E+06	0.168168E+01	0.173264E+07	0.100000E+01
0.178782E+08	0.182165E+08	0.100000E+01	0.435533E+07	0.473192E+07	0.100000E+01
0.699992E+02	0.128555E+04	0.342808E+05	0.569464E+02	0.643434E+03	0.179095E+01
0.113998E+08	0.307930E+02	-0.762463E+00	0.217435E+04	0.979710E+02	0.222982E+08
0.0	0.128936E+01	0.229318E+06	0.190705E+01	0.187991E+07	0.100193E+01
0.179088E+08	0.182165E+08	0.100000E+01	0.438940E+07	0.473192E+07	0.100000E+01

TIME W ALPHA THRUST 1	VREL VDOT MACH VAC THRUST 1	ALT GDT LIFT THROTTLE 1	GAMMA VGRAV RANGE THRUST 2	QBAR VDRG DRAG VAC THRUST 2	LOAD FACTOR THRUST THROTTLE THROTTLE 2
0.719991E+02	0.134345E+04	0.364653E+05	0.554257E+02	0.650106E+03	0.168962E+01
0.112807E+08	0.280111E+02	-0.758029E+00	0.222766E+04	0.109015E+03	0.210707E+08
0.0	0.136220E+01	0.214127E+06	0.215255E+01	0.200880E+07	0.942336E+00
0.166487E+08	0.182165E+08	0.928121E+00	0.442197E+07	0.473192E+07	0.100000E+01
0.739991E+02	0.140060E+04	0.387034E+05	0.539161E+02	0.650144E+03	0.172531E+01
0.111642E+08	0.296520E+02	-0.750991E+00	0.228000E+04	0.120853E+03	0.213834E+08
0.0	0.143524E+01	0.196524E+06	0.241838E+01	0.212002E+07	0.953832E+00
0.169307E+08	0.182165E+08	0.942390E+00	0.445271E+07	0.473192E+07	0.100000E+01
0.759991E+02	0.146364E+04	0.409948E+05	0.524254E+02	0.649544E+03	0.182883E+01
0.110436E+08	0.334864E+02	-0.738961E+00	0.233134E+04	0.133433E+03	0.224046E+08
0.0	0.151504E+01	0.175299E+06	0.270591E+01	0.220600E+07	0.996951E+00
0.179231E+08	0.182165E+08	0.996191E+00	0.448155E+07	0.473192E+07	0.100000E+01
0.779991E+02	0.153245E+04	0.433451E+05	0.509621E+02	0.647164E+03	0.186723E+01
0.109200E+08	0.352330E+02	-0.724228E+00	0.238167E+04	0.146150E+03	0.225243E+08
0.0	0.160106E+01	0.143675E+06	0.301691E+01	0.213270E+07	0.100000E+01
0.180158E+08	0.182165E+08	0.100000E+01	0.450850E+07	0.473192E+07	0.100000E+01
0.799990E+02	0.160452E+04	0.457560E+05	0.495295E+02	0.640127E+03	0.190133E+01
0.107962E+08	0.368460E+02	-0.708146E+00	0.243095E+04	0.158529E+03	0.225718E+08
0.0	0.169001E+01	0.110428E+06	0.335264E+01	0.204336E+07	0.100000E+01
0.180383E+08	0.182165E+08	0.100000E+01	0.453353E+07	0.473192E+07	0.100000E+01
0.819990E+02	0.167988E+04	0.482276E+05	0.481303E+02	0.628140E+03	0.193732E+01
0.106725E+08	0.385223E+02	-0.690945E+00	0.247921E+04	0.170469E+03	0.226156E+08
0.0	0.178099E+01	0.765559E+05	0.371423E+01	0.193863E+07	0.100000E+01
0.180590E+08	0.182165E+08	0.100000E+01	0.455660E+07	0.473192E+07	0.100000E+01

TIME W ALPHA THRUST 1	VREL VDOT MACH VAC THRUST 1	ALT GDT LIFT THROTTLE 1	GAMMA VGRAV RANGE THRUST 2	QBAR VDRG DRAG VAC THRUST 2	LOAD FACTOR THRUST THROTTLE THROTTLE 2
0.839990E+02	0.175864E+04	0.507598E+05	0.467663E+02	0.611210E+03	0.197465E+01
0.105487E+08	0.402418E+02	-0.672920E+00	0.252641E+04	0.181878E+03	0.226556E+08
0.0	0.187309E+01	0.431671E+05	0.410287E+01	0.182508E+07	0.100000E+01
0.180779E+08	0.182165E+08	0.100000E+01	0.457768E+07	0.473192E+07	0.100000E+01
0.859989E+02	0.184086E+04	0.533527E+05	0.454390E+02	0.589563E+03	0.201246E+01
0.104250E+08	0.419753E+02	-0.654367E+00	0.257258E+04	0.192733E+03	0.226919E+08
0.0	0.196523E+01	0.114087E+05	0.451974E+01	0.171202E+07	0.100000E+01
0.180951E+08	0.182165E+08	0.100000E+01	0.459680E+07	0.473192E+07	0.100000E+01
0.879989E+02	0.192656E+04	0.560062E+05	0.441479E+02	0.563735E+03	0.205088E+01
0.103012E+08	0.437270E+02	-0.637203E+00	0.261771E+04	0.203005E+03	0.227245E+08
0.0	0.205639E+01	0.0	0.496605E+01	0.159802E+07	0.100000E+01
0.181105E+08	0.182165E+08	0.100000E+01	0.461396E+07	0.473192E+07	0.100000E+01
0.899989E+02	0.201579E+04	0.587201E+05	0.428900E+02	0.531466E+03	0.209065E+01
0.101775E+08	0.455201E+02	-0.620639E+00	0.266180E+04	0.212675E+03	0.227533E+08
0.0	0.214031E+01	0.0	0.544300E+01	0.147595E+07	0.100000E+01
0.181242E+08	0.182165E+08	0.100000E+01	0.462915E+07	0.473192E+07	0.100000E+01
0.919988E+02	0.210862E+04	0.614938E+05	0.416656E+02	0.500748E+03	0.213042E+01
0.100537E+08	0.473098E+02	-0.603717E+00	0.270487E+04	0.221693E+03	0.227787E+08
0.0	0.222761E+01	0.0	0.595184E+01	0.136012E+07	0.100000E+01
0.181362E+08	0.182165E+08	0.100000E+01	0.464251E+07	0.473192E+07	0.100000E+01
0.939988E+02	0.220503E+04	0.643269E+05	0.404752E+02	0.470700E+03	0.217046E+01
0.992999E+07	0.491031E+02	-0.586577E+00	0.274691E+04	0.230090E+03	0.228010E+08
0.0	0.231716E+01	0.0	0.649382E+01	0.124848E+07	0.100000E+01
0.181467E+08	0.182165E+08	0.100000E+01	0.465426E+07	0.473192E+07	0.100000E+01

TIME	VREL	ALT	GAMMA	QBAR	LOAD FACTOR
W	VDOT	GDT	VGRAV	VDRG	THRUST
ALPHA	MACH	LIFT	RANGE	DRAG	THROTTLE
THRUST 1	VAC THRUST 1	THROTTLE 1	THRUST 2	VAC THRUST 2	THROTTLE 2
0.959988E+02	0.230504E+04	0.672188E+05	0.393193E+02	0.441500E+03	0.221076E+01
0.980624E+07	0.508987E+02	-0.569336E+00	0.278795E+04	0.237879E+03	0.228205E+08
0.0	0.240923E+01	0.0	0.707016E+01	0.114144E+07	0.100000E+01
0.181560E+08	0.182165E+08	0.100000E+01	0.466457E+07	0.473192E+07	0.100000E+01
0.979987E+02	0.240863E+04	0.701687E+05	0.381979E+02	0.413293E+03	0.225131E+01
0.968249E+07	0.526953E+02	-0.552097E+00	0.282800E+04	0.245077E+03	0.228377E+08
0.0	0.250411E+01	0.0	0.768209E+01	0.103953E+07	0.100000E+01
0.181641E+08	0.182165E+08	0.100000E+01	0.467360E+07	0.473192E+07	0.100000E+01
0.999987E+02	0.251580E+04	0.731761E+05	0.371108E+02	0.386181E+03	0.229174E+01
0.955874E+07	0.544801E+02	-0.534949E+00	0.286706E+04	0.251716E+03	0.228526E+08
0.0	0.260209E+01	0.0	0.833082E+01	0.946632E+06	0.100000E+01
0.181712E+08	0.182165E+08	0.100000E+01	0.468148E+07	0.473192E+07	0.100000E+01
0.101999E+03	0.262655E+04	0.762400E+05	0.360573E+02	0.360243E+03	0.233247E+01
0.943499E+07	0.562669E+02	-0.518586E+00	0.290516E+04	0.257830E+03	0.228657E+08
0.0	0.270343E+01	0.0	0.901755E+01	0.858951E+06	0.100000E+01
0.181773E+08	0.182165E+08	0.100000E+01	0.468835E+07	0.473192E+07	0.100000E+01
0.103999E+03	0.274087E+04	0.793595E+05	0.350363E+02	0.335521E+03	0.237354E+01
0.931124E+07	0.580559E+02	-0.502424E+00	0.294232E+04	0.263440E+03	0.228770E+08
0.0	0.280840E+01	0.0	0.974350E+01	0.776479E+06	0.100000E+01
0.181827E+08	0.182165E+08	0.100000E+01	0.469432E+07	0.473192E+07	0.100000E+01
0.105999E+03	0.285878E+04	0.825326E+05	0.340474E+02	0.312029E+03	0.241500E+01
0.918750E+07	0.598481E+02	-0.486510E+00	0.297854E+04	0.268570E+03	0.228869E+08
0.0	0.291717E+01	0.0	0.105099E+02	0.699143E+06	0.100000E+01
0.181874E+08	0.182165E+08	0.100000E+01	0.469951E+07	0.473192E+07	0.100000E+01

TIME W ALPHA THRUST 1	VREL VDOT MACH VAC THRUST 1	ALT GDT LIFT THROTTLE 1	GAMMA VGRAV RANGE THRUST 2	QBAR VDRG DRAG VAC THRUST 2	LOAD FACTOR THRUST THROTTLE THROTTLE 2
0.107999E+03	0.298026E+04	0.857612E+05	0.330900E+02	0.289746E+03	0.245644E+01
0.906375E+07	0.616303E+02	-0.470886E+00	0.301386E+04	0.273249E+03	0.228954E+08
0.0	0.302983E+01	0.0	0.113178E+02	0.630902E+06	0.100000E+01
0.181914E+08	0.182165E+08	0.100000E+01	0.470400E+07	0.473192E+07	0.100000E+01
0.109999E+03	0.310527E+04	0.890414E+05	0.321636E+02	0.268615E+03	0.249715E+01
0.894000E+07	0.633791E+02	-0.455594E+00	0.304829E+04	0.277569E+03	0.229028E+08
0.0	0.314623E+01	0.0	0.121686E+02	0.578312E+06	0.100000E+01
0.181949E+08	0.182165E+08	0.100000E+01	0.470790E+07	0.473192E+07	0.100000E+01
0.111999E+03	0.323379E+04	0.923729E+05	0.312674E+02	0.248632E+03	0.253852E+01
0.881625E+07	0.651388E+02	-0.440660E+00	0.308184E+04	0.281579E+03	0.229092E+08
0.0	0.326501E+01	0.0	0.130633E+02	0.529027E+06	0.100000E+01
0.181979E+08	0.182165E+08	0.100000E+01	0.471128E+07	0.473192E+07	0.100000E+01
0.113999E+03	0.336584E+04	0.957546E+05	0.304007E+02	0.228891E+03	0.258080E+01
0.869250E+07	0.669176E+02	-0.426096E+00	0.311455E+04	0.285290E+03	0.229148E+08
0.0	0.338173E+01	0.0	0.140031E+02	0.481233E+06	0.100000E+01
0.182006E+08	0.182165E+08	0.100000E+01	0.471421E+07	0.473192E+07	0.100000E+01
0.115998E+03	0.350146E+04	0.991854E+05	0.295627E+02	0.210341E+03	0.262382E+01
0.856876E+07	0.687097E+02	-0.411911E+00	0.314643E+04	0.288709E+03	0.229196E+08
0.0	0.350046E+01	0.0	0.149891E+02	0.436760E+06	0.100000E+01
0.182028E+08	0.182165E+08	0.100000E+01	0.471673E+07	0.473192E+07	0.100000E+01
0.117998E+03	0.364069E+04	0.102664E+06	0.287527E+02	0.192954E+03	0.266764E+01
0.844501E+07	0.705172E+02	-0.398112E+00	0.317750E+04	0.291855E+03	0.229237E+08
0.0	0.362103E+01	0.0	0.160226E+02	0.395493E+06	0.100000E+01
0.182048E+08	0.182165E+08	0.100000E+01	0.471890E+07	0.473192E+07	0.100000E+01

TIME W ALPHA THRUST 1	VREL VDOT MACH VAC THRUST 1	ALT GDT LIFT THROTTLE 1	GAMMA VGRAV RANGE THRUST 2	QBAR VDRG DRAG VAC THRUST 2	LOAD FACTOR THRUST THROTTLE THROTTLE 2
0.119998E+03	0.378355E+04	0.106190E+06	0.279699E+02	0.176702E+03	0.271232E+01
0.832126E+07	0.723420E+02	-0.384699E+00	0.320778E+04	0.294742E+03	0.229272E+08
0.0	0.374331E+01	0.0	0.171046E+02	0.357322E+06	0.100000E+01
0.182065E+08	0.182165E+08	0.100000E+01	0.472076E+07	0.473192E+07	0.100000E+01
0.121998E+03	0.393007E+04	0.109762E+06	0.272136E+02	0.161557E+03	0.275793E+01
0.819751E+07	0.741861E+02	-0.371672E+00	0.323729E+04	0.297387E+03	0.229303E+08
0.0	0.386722E+01	0.0	0.182364E+02	0.322128E+06	0.100000E+01
0.182079E+08	0.182165E+08	0.100000E+01	0.472236E+07	0.473192E+07	0.100000E+01
0.123998E+03	0.408030E+04	0.113579E+06	0.264829E+02	0.147490E+03	0.280449E+01
0.807376E+07	0.760502E+02	-0.359027E+00	0.326607E+04	0.299806E+03	0.229329E+08
0.0	0.399268E+01	0.0	0.194192E+02	0.290137E+06	0.100000E+01
0.182091E+08	0.182165E+08	0.100000E+01	0.472374E+07	0.473192E+07	0.100000E+01
0.125998E+03	0.423429E+04	0.117039E+06	0.257772E+02	0.134467E+03	0.285204E+01
0.795002E+07	0.779361E+02	-0.346762E+00	0.329411E+04	0.302019E+03	0.229351E+08
0.0	0.411970E+01	0.0	0.206541E+02	0.261348E+06	0.100000E+01
0.182102E+08	0.182165E+08	0.100000E+01	0.472491E+07	0.473192E+07	0.100000E+01
0.127998E+03	0.439206E+04	0.120744E+06	0.250956E+02	0.122452E+03	0.290073E+01
0.782627E+07	0.798485E+02	-0.334865E+00	0.332146E+04	0.304042E+03	0.229370E+08
0.0	0.424833E+01	0.0	0.219423E+02	0.235104E+06	0.100000E+01
0.182111E+08	0.182165E+08	0.100000E+01	0.472592E+07	0.473192E+07	0.100000E+01
0.129998E+03	0.455369E+04	0.124490E+06	0.244374E+02	0.111405E+03	0.295065E+01
0.770252E+07	0.817900E+02	-0.323332E+00	0.334811E+04	0.305889E+03	0.229386E+08
0.0	0.437871E+01	0.0	0.232850E+02	0.211239E+06	0.100000E+01
0.182119E+08	0.182165E+08	0.100000E+01	0.472678E+07	0.473192E+07	0.100000E+01

TIME	VREL	ALT	GAMMA	QBAR	LOAD FACTOR
W	VDOT	GDT	VGRAV	VDRG	THRUST
ALPHA	MACH	LIFT	RANGE	DRAG	THROTTLE
THRUST 1	VAC THRUST 1	THROTTLE 1	THRUST 2	VAC THRUST 2	THROTTLE 2
0.131998E+03	0.471924E+04	0.128279E+06	0.238019E+02	0.101281E+03	0.300186E+01
0.757877E+07	0.837635E+02	-0.312152E+00	0.337411E+04	0.307576E+03	0.229400E+08
0.0	0.451109E+01	0.0	0.246835E+02	0.189613E+06	0.100000E+01
0.182125E+08	0.182165E+08	0.100000E+01	0.472752E+07	0.473192E+07	0.100000E+01
0.133998E+03	0.488700E+04	0.132108E+06	0.231880E+02	0.919675E+02	0.300018E+01
0.745618E+07	0.840257E+02	-0.301806E+00	0.339946E+04	0.309113E+03	0.225399E+08
0.0	0.464411E+01	0.0	0.261386E+02	0.169978E+06	0.982505E+00
0.178117E+08	0.182165E+08	0.977963E+00	0.472815E+07	0.473192E+07	0.100000E+01
0.135998E+03	0.505536E+04	0.135974E+06	0.225943E+02	0.833892E+02	0.300015E+01
0.733580E+07	0.843318E+02	-0.291976E+00	0.342419E+04	0.310513E+03	0.221606E+08
0.0	0.477667E+01	0.0	0.276507E+02	0.152154E+06	0.965930E+00
0.174319E+08	0.182165E+08	0.957084E+00	0.472869E+07	0.473192E+07	0.100000E+01
0.137998E+03	0.522432E+04	0.139875E+06	0.220198E+02	0.755351E+02	0.300011E+01
0.721755E+07	0.846291E+02	-0.282631E+00	0.344831E+04	0.311786E+03	0.217895E+08
0.0	0.490938E+01	0.0	0.292200E+02	0.136066E+06	0.949718E+00
0.170604E+08	0.182165E+08	0.936662E+00	0.472915E+07	0.473192E+07	0.100000E+01
0.139998E+03	0.539386E+04	0.143808E+06	0.214634E+02	0.683791E+02	0.300008E+01
0.710138E+07	0.849181E+02	-0.273736E+00	0.347184E+04	0.312943E+03	0.214263E+08
0.0	0.504293E+01	0.0	0.308466E+02	0.121613E+06	0.933856E+00
0.166968E+08	0.182165E+08	0.916681E+00	0.472954E+07	0.473192E+07	0.100000E+01
0.141998E+03	0.556398E+04	0.147769E+06	0.209245E+02	0.618854E+02	0.300004E+01
0.698727E+07	0.851991E+02	-0.265255E+00	0.349480E+04	0.313992E+03	0.210704E+08
0.0	0.517820E+01	0.0	0.325308E+02	0.108280E+06	0.918317E+00
0.163405E+08	0.182165E+08	0.897106E+00	0.472988E+07	0.473192E+07	0.100000E+01

TIME	VREL	ALT	GAMMA	QBAR	LOAD FACTOR
W	VDOT	GDT	VGRAV	VDRG	THRUST
ALPHA	MACH	LIFT	RANGE	DRAG	THROTTLE
THRUST 1	VAC THRUST 1	THROTTLE 1	THRUST 2	VAC THRUST 2	THROTTLE 2
0.143998E+03	0.573464E+04	0.151755E+06	0.204021E+02	0.560110E+02	0.300000E+01
0.687515E+07	0.854723E+02	-0.257161E+00	0.351719E+04	0.314940E+03	0.207216E+08
0.0	0.531622E+01	0.0	0.342728E+02	0.961661E+05	0.903098E+00
0.159915E+08	0.182165E+08	0.877934E+00	0.473017E+07	0.473192E+07	0.100000E+01
0.145998E+03	0.590586E+04	0.155764E+06	0.198955E+02	0.507089E+02	0.299997E+01
0.676500E+07	0.857380E+02	-0.249426E+00	0.353905E+04	0.315795E+03	0.203801E+08
0.0	0.545822E+01	0.0	0.360727E+02	0.853122E+05	0.888193E+00
0.156497E+08	0.182165E+08	0.859157E+00	0.473042E+07	0.473192E+07	0.100000E+01
0.147998E+03	0.607759E+04	0.159793E+06	0.194041E+02	0.459299E+02	0.299993E+01
0.665677E+07	0.859964E+02	-0.242026E+00	0.356038E+04	0.316565E+03	0.200454E+08
0.0	0.560566E+01	0.0	0.379307E+02	0.755918E+05	0.873594E+00
0.153148E+08	0.182165E+08	0.840765E+00	0.473063E+07	0.473192E+07	0.100000E+01
0.149998E+03	0.624983E+04	0.163840E+06	0.189272E+02	0.418585E+02	0.299990E+01
0.655042E+07	0.862480E+02	-0.234938E+00	0.358119E+04	0.317260E+03	0.197178E+08
0.0	0.577557E+01	0.0	0.398469E+02	0.671934E+05	0.859301E+00
0.149870E+08	0.182165E+08	0.822760E+00	0.473081E+07	0.473192E+07	0.100000E+01
0.151998E+03	0.642257E+04	0.167901E+06	0.184641E+02	0.381742E+02	0.299986E+01
0.644591E+07	0.864928E+02	-0.228141E+00	0.360151E+04	0.317887E+03	0.193965E+08
0.0	0.595628E+01	0.0	0.418215E+02	0.596192E+05	0.845288E+00
0.146655E+08	0.182165E+08	0.805106E+00	0.473097E+07	0.473192E+07	0.100000E+01
0.153998E+03	0.659579E+04	0.171975E+06	0.180144E+02	0.347555E+02	0.299983E+01
0.634321E+07	0.867311E+02	-0.221619E+00	0.362134E+04	0.318451E+03	0.190812E+08
0.0	0.614309E+01	0.0	0.438547E+02	0.526742E+05	0.831540E+00
0.143501E+08	0.182165E+08	0.787786E+00	0.473111E+07	0.473192E+07	0.100000E+01

TIME W ALPHA THRUST 1	VREL VDOT MACH VAC THRUST 1	ALT GDT LIFT THROTTLE 1	GAMMA VGRAV RANGE THRUST 2	QBAR VDRG DRAG VAC THRUST 2	LOAD FACTOR THRUST THROTTLE THROTTLE 2
0.155998E+03	0.676948E+04	0.176059E+06	0.175774E+02	0.315847E+02	0.299980E+01
0.624228E+07	0.869631E+02	-0.215353E+00	0.364069E+04	0.318956E+03	0.187719E+08
0.0	0.633600E+01	0.0	0.459464E+02	0.463149E+05	0.818052E+00
0.140407E+08	0.182165E+08	0.770795E+00	0.473123E+07	0.473192E+07	0.100000E+01
0.157998E+03	0.694363E+04	0.180152E+06	0.171528E+02	0.286459E+02	0.299976E+01
0.614310E+07	0.871891E+02	-0.209328E+00	0.365959E+04	0.319407E+03	0.184684E+08
0.0	0.653499E+01	0.0	0.480969E+02	0.405020E+05	0.804817E+00
0.137370E+08	0.182165E+08	0.754121E+00	0.473133E+07	0.473192E+07	0.100000E+01
0.158387E+03	0.697757E+04	0.180949E+06	0.170715E+02	0.280998E+02	0.300000E+01
0.612400E+07	0.872401E+02	-0.208181E+00	0.366321E+04	0.319488E+03	0.184114E+08
0.0	0.657441E+01	0.0	0.485205E+02	0.394314E+05	0.802334E+00
0.136801E+08	0.182165E+08	0.750993E+00	0.473135E+07	0.473192E+07	0.100000E+01
0.158387E+03	0.697757E+04	0.180949E+06	0.170715E+02	0.280998E+02	0.962865E+00
0.489165E+07	0.217080E+02	-0.217895E+00	0.0	0.0	0.474942E+07
0.0	0.657441E+01	0.0	0.485205E+02	0.394314E+05	0.100000E+01
0.0	0.0	0.0	0.0	0.0	0.0
0.160063E+03	0.701423E+04	0.184355E+06	0.167072E+02	0.250765E+02	0.967318E+00
0.487459E+07	0.220463E+02	-0.216876E+00	0.154049E+02	0.406158E+00	0.474950E+07
0.0	0.664260E+01	0.0	0.503541E+02	0.342198E+05	0.100000E+01
0.0	0.0	0.0	0.0	0.0	0.0

EXO-ATMOSPHERIC TRAJECTORY

TIME W ALPHA	V(R) V(I) CD	GAM(R) GAM(I)	ALT THETA(R)	RANGE QBAR	T/W VDRAG
0.160063E+03	0.701423E+04	0.167072E+02	0.184355E+06	0.503541E+02	0.974441E+00
0.487459E+07	0.827763E+04	0.140993E+02	0.281440E+02	0.250765E+02	0.0
-0.643683E+01	0.185358E+00				
0.162063E+03	0.705782E+04	0.163678E+02	0.188356E+06	0.525551E+02	0.978527E+00
0.485423E+07	0.832330E+04	0.138250E+02	0.280162E+02	0.219180E+02	0.740648E+00
-0.664836E+01	0.186050E+00				
0.164063E+03	0.710198E+04	0.160327E+02	0.192305E+06	0.547734E+02	0.982647E+00
0.483388E+07	0.836947E+04	0.135539E+02	0.278881E+02	0.191725E+02	0.135302E+01
-0.685535E+01	0.186749E+00				
0.165674E+03	0.713798E+04	0.157659E+02	0.195447E+06	0.565728E+02	0.985992E+00
0.481748E+07	0.840705E+04	0.133378E+02	0.277847E+02	0.172214E+02	0.176351E+01
-0.701884E+01	0.187316E+00				
0.165674E+03	0.713798E+04	0.157659E+02	0.195447E+06	0.565728E+02	0.985993E+00
0.481748E+07	0.840705E+04	0.133378E+02	0.277847E+02	0.172214E+02	0.176351E+01
-0.701884E+01	0.187316E+00				
0.167674E+03	0.718319E+04	0.154385E+02	0.199298E+06	0.588223E+02	0.990177E+00
0.479712E+07	0.845416E+04	0.130725E+02	0.276561E+02	0.150815E+02	0.218261E+01
-0.721762E+01	0.188023E+00				
0.169674E+03	0.722896E+04	0.151154E+02	0.203094E+06	0.610891E+02	0.994396E+00
0.477677E+07	0.850178E+04	0.128104E+02	0.275273E+02	0.132132E+02	0.251237E+01
-0.741191E+01	0.188734E+00				

TIME W ALPHA	V(R) V(I) CD	GAM(R) GAM(I)	ALT THETA(R)	RANGE QBAR	T/W VDRAG
0.171674E+03	0.727530E+04	0.147965E+02	0.206836E+06	0.633733E+02	0.998651E+00
0.475641E+07	0.854991E+04	0.125514E+02	0.273983E+02	0.115803E+02	0.276334E+01
-0.760172E+01	0.189447E+00				
0.173674E+03	0.732220E+04	0.144819E+02	0.210523E+06	0.656756E+02	0.100294E+01
0.473605E+07	0.859855E+04	0.122955E+02	0.272690E+02	0.101519E+02	0.294471E+01
-0.778705E+01	0.190159E+00				
0.175674E+03	0.736967E+04	0.141715E+02	0.214157E+06	0.679951E+02	0.100727E+01
0.471570E+07	0.864770E+04	0.120428E+02	0.271394E+02	0.890115E+01	0.306452E+01
-0.796793E+01	0.190872E+00				
0.177674E+03	0.741769E+04	0.138652E+02	0.217737E+06	0.703330E+02	0.101164E+01
0.469534E+07	0.869736E+04	0.117932E+02	0.270096E+02	0.780586E+01	0.312983E+01
-0.814436E+01	0.191583E+00				
0.179674E+03	0.746628E+04	0.135631E+02	0.221264E+06	0.726888E+02	0.101604E+01
0.467499E+07	0.874752E+04	0.115467E+02	0.268795E+02	0.684603E+01	0.314684E+01
-0.831641E+01	0.192291E+00				
0.181674E+03	0.751542E+04	0.132651E+02	0.224738E+06	0.750624E+02	0.102048E+01
0.465463E+07	0.879820E+04	0.113033E+02	0.267493E+02	0.600475E+01	0.312102E+01
-0.848412E+01	0.192995E+00				
0.183674E+03	0.756511E+04	0.129713E+02	0.228159E+06	0.774545E+02	0.102496E+01
0.463427E+07	0.884938E+04	0.110630E+02	0.266187E+02	0.526751E+01	0.305718E+01
-0.864743E+01	0.193694E+00				

TIME W ALPHA	V(R) V(I) CD	GAM(R) GAM(I)	ALT THETA(R)	RANGE QBAR	T/W VDRAG
0.185674E+03	0.761537E+04	0.126814E+02	0.231529E+06	0.798641E+02	0.102949E+01
0.461392E+07	0.890108E+04	0.108256E+02	0.264879E+02	0.462105E+01	0.295958E+01
-0.880652E+01	0.194389E+00				
0.187674E+03	0.766618E+04	0.123957E+02	0.234845E+06	0.822923E+02	0.103405E+01
0.459356E+07	0.895328E+04	0.105913E+02	0.263569E+02	0.405471E+01	0.283197E+01
-0.896128E+01	0.195076E+00				
0.189674E+03	0.771754E+04	0.121139E+02	0.238110E+06	0.847392E+02	0.103865E+01
0.457321E+07	0.900599E+04	0.103601E+02	0.262257E+02	0.355853E+01	0.269887E+01
-0.911176E+01	0.241397E+00				
0.191674E+03	0.776945E+04	0.118362E+02	0.241323E+06	0.872044E+02	0.104329E+01
0.455285E+07	0.905921E+04	0.101318E+02	0.260942E+02	0.312380E+01	0.258125E+01
-0.925805E+01	0.292179E+00				
0.193674E+03	0.782192E+04	0.115624E+02	0.244484E+06	0.896884E+02	0.104798E+01
0.453249E+07	0.911293E+04	0.990645E+01	0.259626E+02	0.274314E+01	0.247481E+01
-0.940014E+01	0.345263E+00				
0.195674E+03	0.787492E+04	0.112926E+02	0.247593E+06	0.921916E+02	0.105270E+01
0.451214E+07	0.916716E+04	0.968412E+01	0.258307E+02	0.240990E+01	0.237502E+01
-0.953805E+01	0.400719E+00				
0.197674E+03	0.792847E+04	0.110267E+02	0.250649E+06	0.947140E+02	0.105747E+01
0.449178E+07	0.922189E+04	0.946475E+01	0.256985E+02	0.211832E+01	0.227815E+01
-0.967182E+01	0.458606E+00				

TIME W ALPHA	V(R) V(I) CD	GAM(R) GAM(I)	ALT THETA(R)	RANGE QBAR	T/W VDRAG
0.199674E+03	0.798256E+04	0.107647E+02	0.253655E+06	0.972553E+02	0.106228E+01
0.447143E+07	0.927712E+04	0.924829E+01	0.255662E+02	0.186309E+01	0.218115E+01
-0.980149E+01	0.519017E+00				
0.201674E+03	0.803721E+04	0.105065E+02	0.256612E+06	0.998153E+02	0.106714E+01
0.445107E+07	0.933287E+04	0.903468E+01	0.254337E+02	0.163971E+01	0.208154E+01
-0.992717E+01	0.582021E+00				
0.203674E+03	0.809239E+04	0.102521E+02	0.259517E+06	0.102395E+03	0.107204E+01
0.443071E+07	0.938913E+04	0.882399E+01	0.253009E+02	0.144432E+01	0.197735E+01
-0.100488E+02	0.647529E+00				
0.205674E+03	0.814812E+04	0.100016E+02	0.262371E+06	0.104994E+03	0.107699E+01
0.441036E+07	0.944589E+04	0.861617E+01	0.251680E+02	0.127345E+01	0.186698E+01
-0.101664E+02	0.715523E+00				
0.207674E+03	0.820439E+04	0.975482E+01	0.265174E+06	0.107614E+03	0.108198E+01
0.439000E+07	0.950315E+04	0.841123E+01	0.250348E+02	0.112404E+01	0.174924E+01
-0.102800E+02	0.786202E+00				
0.209674E+03	0.826121E+04	0.951171E+01	0.267928E+06	0.110252E+03	0.108702E+01
0.436965E+07	0.956093E+04	0.820905E+01	0.249015E+02	0.993251E+00	0.161859E+01
-0.103897E+02	0.825107E+00				
0.211674E+03	0.831856E+04	0.927232E+01	0.270633E+06	0.112910E+03	0.109211E+01
0.434929E+07	0.961921E+04	0.800969E+01	0.247679E+02	0.878837E+00	0.146617E+01
-0.104956E+02	0.828781E+00				

TIME W ALPHA	V(R) V(I) CD	GAM(R) GAM(I)	ALT THETA(R)	RANGE QBAR	T/W VDRAG
0.213674E+03	0.837646E+04	0.903661E+01	0.273287E+06	0.115589E+03	0.109724E+01
0.432893E+07	0.967800E+04	0.781314E+01	0.246342E+02	0.774895E+00	0.128915E+01
-0.105976E+02	0.832317E+00				
0.215674E+03	0.843490E+04	0.880449E+01	0.275892E+06	0.118287E+03	0.110243E+01
0.430853E+07	0.973730E+04	0.761932E+01	0.245003E+02	0.678955E+00	0.108917E+01
-0.106958E+02	0.835721E+00				
0.217674E+03	0.849388E+04	0.857595E+01	0.278449E+06	0.121006E+03	0.110766E+01
0.428822E+07	0.979711E+04	0.742822E+01	0.243662E+02	0.596544E+00	0.868314E+00
-0.107902E+02	0.838991E+00				
0.219674E+03	0.855341E+04	0.835095E+01	0.280958E+06	0.123744E+03	0.111294E+01
0.426787E+07	0.985745E+04	0.723983E+01	0.242319E+02	0.525599E+00	0.629110E+00
-0.108809E+02	0.842130E+00				
0.221674E+03	0.861347E+04	0.812949E+01	0.283418E+06	0.126503E+03	0.111827E+01
0.424751E+07	0.991829E+04	0.705415E+01	0.240974E+02	0.464381E+00	0.373732E+00
-0.109679E+02	0.845139E+00				
0.223674E+03	0.867408E+04	0.791151E+01	0.285830E+06	0.129283E+03	0.112366E+01
0.422715E+07	0.997964E+04	0.687114E+01	0.239628E+02	0.411425E+00	0.104036E+00
-0.110513E+02	0.848020E+00				
0.225674E+03	0.873523E+04	0.769701E+01	0.288192E+06	0.132084E+03	0.112909E+01
0.420680E+07	0.100415E+05	0.669081E+01	0.238280E+02	0.365541E+00	-0.178386E+00
-0.111310E+02	0.850772E+00				

TIME W ALPHA	V(R) V(I) CD	GAM(R) GAM(I)	ALT THETA(R)	RANGE QBAR	T/W VDRAG
0.227674E+03	0.879692E+04	0.748592E+01	0.290508E+06	0.134905E+03	0.113458E+01
0.418644E+07	0.101039E+05	0.651309E+01	0.236931E+02	0.325646E+00	-0.472172E+00
-0.112071E+02	0.853401E+00				
0.229674E+03	0.885916E+04	0.727824E+01	0.292775E+06	0.137748E+03	0.114013E+01
0.416609E+07	0.101668E+05	0.633801E+01	0.235579E+02	0.290901E+00	-0.776153E+00
-0.112797E+02	0.855905E+00				
0.231674E+03	0.892193E+04	0.707389E+01	0.294996E+06	0.140611E+03	0.114572E+01
0.414573E+07	0.102302E+05	0.616550E+01	0.234227E+02	0.260558E+00	-0.108932E+01
-0.113488E+02	0.858288E+00				
0.233674E+03	0.898524E+04	0.687290E+01	0.297168E+06	0.143496E+03	0.115137E+01
0.412538E+07	0.102941E+05	0.599558E+01	0.232873E+02	0.231916E+00	-0.141107E+01
-0.114144E+02	0.860548E+00				
0.235674E+03	0.904910E+04	0.667516E+01	0.299295E+06	0.146402E+03	0.115708E+01
0.410502E+07	0.103586E+05	0.582819E+01	0.231517E+02	0.207001E+00	-0.174086E+01
-0.114766E+02	0.862690E+00				
0.237674E+03	0.911352E+04	0.648069E+01	0.301374E+06	0.149330E+03	0.116285E+01
0.408467E+07	0.104236E+05	0.566334E+01	0.230160E+02	0.185534E+00	-0.207790E+01
-0.115353E+02	0.864714E+00				
0.239674E+03	0.917847E+04	0.628946E+01	0.303408E+06	0.152280E+03	0.116867E+01
0.406431E+07	0.104891E+05	0.550160E+01	0.228802E+02	0.166963E+00	-0.242152E+01
-0.115908E+02	0.866621E+00				

TIME	V(R)	GAM(R)	ALT	RANGE	T/W
W	V(I)	GAM(I)	THETA(R)	QBAR	VDRAG
ALPHA	CD				
0.241674E+03	0.924396E+04	0.610144E+01	0.305393E+06	0.155252E+03	0.117455E+01
0.404396E+07	0.105551E+05	0.534118E+01	0.227443E+02	0.150845E+00	-0.277112E+01
-0.116428E+02	0.868413E+00				
0.243674E+03	0.931001E+04	0.591653E+01	0.307335E+06	0.158245E+03	0.118049E+01
0.402360E+07	0.106216E+05	0.518379E+01	0.226082E+02	0.136787E+00	-0.312621E+01
-0.116917E+02	0.870092E+00				
0.245674E+03	0.937661E+04	0.573478E+01	0.309230E+06	0.161261E+03	0.118649E+01
0.400325E+07	0.106887E+05	0.502886E+01	0.224720E+02	0.124501E+00	-0.348637E+01
-0.117372E+02	0.871657E+00				
0.247674E+03	0.944376E+04	0.555610E+01	0.311080E+06	0.164299E+03	0.119256E+01
0.398289E+07	0.107563E+05	0.487635E+01	0.223357E+02	0.113719E+00	-0.385126E+01
-0.117796E+02	0.873114E+00				
0.249674E+03	0.951146E+04	0.538047E+01	0.312886E+06	0.167359E+03	0.119868E+01
0.396254E+07	0.108245E+05	0.472623E+01	0.221993E+02	0.104228E+00	-0.422057E+01
-0.118188E+02	0.874461E+00				
0.251674E+03	0.957973E+04	0.520788E+01	0.314646E+06	0.170441E+03	0.120487E+01
0.394218E+07	0.108932E+05	0.457851E+01	0.220627E+02	0.958523E-01	-0.459404E+01
-0.118548E+02	0.875699E+00				
0.253674E+03	0.964854E+04	0.503825E+01	0.316363E+06	0.173547E+03	0.121112E+01
0.392183E+07	0.109624E+05	0.443312E+01	0.219261E+02	0.884350E-01	-0.497146E+01
-0.118876E+02	0.876832E+00				

TIME W ALPHA	V(R) V(I) CD	GAM(R) GAM(I)	ALT THETA(R)	RANGE QBAR	T/W VDRAG
0.255674E+03	0.971791E+04	0.487162E+01	0.318035E+06	0.176675E+03	0.121744E+01
0.390147E+07	0.110322E+05	0.429012E+01	0.217894E+02	0.818561E-01	-0.535264E+01
-0.119178E+02	0.877860E+00				
0.257674E+03	0.978784E+04	0.470791E+01	0.319662E+06	0.179826E+03	0.122382E+01
0.388112E+07	0.111025E+05	0.414942E+01	0.216526E+02	0.760021E-01	-0.573741E+01
-0.119447E+02	0.878782E+00				
0.259674E+03	0.985834E+04	0.454710E+01	0.321247E+06	0.183001E+03	0.123027E+01
0.386076E+07	0.111733E+05	0.401102E+01	0.215157E+02	0.707773E-01	-0.612565E+01
-0.119686E+02	0.879604E+00				
0.261674E+03	0.992940E+04	0.438917E+01	0.322787E+06	0.186199E+03	0.123679E+01
0.384041E+07	0.112447E+05	0.387492E+01	0.213787E+02	0.661094E-01	-0.651725E+01
-0.119895E+02	0.880323E+00				
0.263674E+03	0.100010E+05	0.423408E+01	0.324285E+06	0.189420E+03	0.124338E+01
0.382005E+07	0.113167E+05	0.374110E+01	0.212417E+02	0.619270E-01	-0.691209E+01
-0.120076E+02	0.880942E+00				
0.265674E+03	0.100732E+05	0.408177E+01	0.325740E+06	0.192665E+03	0.125004E+01
0.379970E+07	0.113892E+05	0.360948E+01	0.211046E+02	0.581691E-01	-0.731011E+01
-0.120228E+02	0.881463E+00				
0.267674E+03	0.101460E+05	0.393226E+01	0.327152E+06	0.195934E+03	0.125677E+01
0.377934E+07	0.114622E+05	0.348011E+01	0.209674E+02	0.547908E-01	-0.771122E+01
-0.120352E+02	0.881887E+00				

TIME	V(R)	GAM(R)	ALT	RANGE	T/W.
W	V(I)	GAM(I)	THETA(R)	QBAR	VDRAG
ALPHA	CD				
0.269674E+03	0.102193E+05	0.378553E+01	0.328521E+06	0.199227E+03	0.126358E+01
0.375899E+07	0.115358E+05	0.335298E+01	0.208302E+02	0.516858E-01	-0.811538E+01
-0.120447E+02	0.882212E+00				
0.271674E+03	0.102932E+05	0.364153E+01	0.329848E+06	0.202544E+03	0.127046E+01
0.373863E+07	0.116100E+05	0.322804E+01	0.206929E+02	0.487697E-01	-0.852256E+01
-0.120514E+02	0.882444E+00				
0.273674E+03	0.103677E+05	0.350017E+01	0.331134E+06	0.205885E+03	0.127741E+01
0.371828E+07	0.116848E+05	0.310524E+01	0.205556E+02	0.461508E-01	-0.893274E+01
-0.120555E+02	0.882583E+00				
0.275674E+03	0.104428E+05	0.336149E+01	0.332379E+06	0.209251E+03	0.128444E+01
0.369792E+07	0.117601E+05	0.298459E+01	0.204183E+02	0.437985E-01	-0.934587E+01
-0.120568E+02	0.882629E+00				
0.277674E+03	0.105184E+05	0.322546E+01	0.333582E+06	0.212642E+03	0.129155E+01
0.367757E+07	0.118360E+05	0.286609E+01	0.202809E+02	0.416822E-01	-0.976192E+01
-0.120555E+02	0.882583E+00				
0.279674E+03	0.105946E+05	0.309204E+01	0.334743E+06	0.216057E+03	0.129873E+01
0.365721E+07	0.119124E+05	0.274971E+01	0.201435E+02	0.397761E-01	-0.101809E+02
-0.120515E+02	0.882447E+00				
0.281674E+03	0.106715E+05	0.296121E+01	0.335865E+06	0.219498E+03	0.130600E+01
0.363686E+07	0.119895E+05	0.263544E+01	0.200061E+02	0.380558E-01	-0.106027E+02
-0.120449E+02	0.882221E+00				

TIME W ALPHA	V(R) V(I) CD	GAM(R) GAM(I)	ALT THETA(R)	RANGE QBAR	T/W VDRAG
0.283674E+03	0.107489E+05	0.283288E+01	0.336947E+06	0.222963E+03	0.131335E+01
0.361650E+07	0.120671E+05	0.252321E+01	0.198687E+02	0.365013E-01	-0.110273E+02
-0.120358E+02	0.881909E+00				
0.285674E+03	0.108269E+05	0.270709E+01	0.337989E+06	0.226454E+03	0.132078E+01
0.359615E+07	0.121453E+05	0.241305E+01	0.197313E+02	0.350964E-01	-0.114549E+02
-0.120242E+02	0.881510E+00				
0.287674E+03	0.109055E+05	0.258382E+01	0.338991E+06	0.229971E+03	0.132830E+01
0.357579E+07	0.122241E+05	0.230496E+01	0.195939E+02	0.338283E-01	-0.118852E+02
-0.120101E+02	0.881026E+00				
0.289674E+03	0.109843E+05	0.246301E+01	0.339953E+06	0.233513E+03	0.133590E+01
0.355544E+07	0.123035E+05	0.219888E+01	0.194565E+02	0.326828E-01	-0.123184E+02
-0.119935E+02	0.880456E+00				
0.291674E+03	0.110646E+05	0.234464E+01	0.340877E+06	0.237081E+03	0.134359E+01
0.353508E+07	0.123835E+05	0.209482E+01	0.193191E+02	0.316477E-01	-0.127545E+02
-0.119744E+02	0.879803E+00				
0.293674E+03	0.111450E+05	0.222869E+01	0.341762E+06	0.240675E+03	0.135137E+01
0.351473E+07	0.124641E+05	0.199274E+01	0.191817E+02	0.307136E-01	-0.131934E+02
-0.119530E+02	0.879069E+00				
0.295674E+03	0.112261E+05	0.211516E+01	0.342609E+06	0.244295E+03	0.135924E+01
0.349437E+07	0.125453E+05	0.189266E+01	0.190444E+02	0.298729E-01	-0.136352E+02
-0.119292E+02	0.878253E+00				

TIME W ALPHA	V(R) V(I) CD	GAM(R) GAM(I)	ALT THETA(R)	RANGE QBAR	T/W VDRAG
0.297674E+03	0.113078E+05	0.200398E+01	0.343418E+06	0.247942E+03	0.136720E+01
0.347402E+07	0.126271E+05	0.179453E+01	0.189071E+02	0.291163E-01	-0.140798E+02
-0.119031E+02	0.877356E+00				
0.299674E+03	0.113901E+05	0.189516E+01	0.344186E+06	0.251616E+03	0.137526E+01
0.345366E+07	0.127095E+05	0.169835E+01	0.187698E+02	0.284362E-01	-0.145273E+02
-0.118747E+02	0.876380E+00				
0.301674E+03	0.114730E+05	0.178864E+01	0.344922E+06	0.255316E+03	0.138341E+01
0.343331E+07	0.127926E+05	0.160410E+01	0.186326E+02	0.278316E-01	-0.149778E+02
-0.118440E+02	0.875327E+00				
0.303674E+03	0.115566E+05	0.168444E+01	0.345619E+06	0.259043E+03	0.139166E+01
0.341295E+07	0.128762E+05	0.151176E+01	0.184955E+02	0.272921E-01	-0.154311E+02
-0.118111E+02	0.874195E+00				
0.305674E+03	0.116408E+05	0.158247E+01	0.346281E+06	0.262797E+03	0.140001E+01
0.339260E+07	0.129606E+05	0.142130E+01	0.183584E+02	0.268140E-01	-0.158874E+02
-0.117760E+02	0.872989E+00				
0.307674E+03	0.117257E+05	0.148280E+01	0.346904E+06	0.266580E+03	0.140846E+01
0.337224E+07	0.130455E+05	0.133275E+01	0.182214E+02	0.263960E-01	-0.163466E+02
-0.117386E+02	0.871707E+00				
0.309674E+03	0.118112E+05	0.138533E+01	0.347492E+06	0.270389E+03	0.141701E+01
0.335189E+07	0.131311E+05	0.124606E+01	0.180845E+02	0.260321E-01	-0.168088E+02
-0.116992E+02	0.870351E+00				

TIME W ALPHA	V(R) V(I) CD	GAM(R) GAM(I)	ALT THETA(R)	RANGE QBAR	T/W VDRAG
0.311674E+03	0.118973E+05	0.129008E+01	0.348045E+06	0.274227E+03	0.142567E+01
0.333153E+07	0.132173E+05	0.116122E+01	0.179477E+02	0.257196E-01	-0.172740E+02
-0.116576E+02	0.868922E+00				
0.313674E+03	0.119841E+05	0.119701E+01	0.348562E+06	0.278092E+03	0.143443E+01
0.331118E+07	0.133042E+05	0.107823E+01	0.178110E+02	0.254568E-01	-0.177423E+02
-0.116140E+02	0.867420E+00				
0.315674E+03	0.120716E+05	0.110610E+01	0.349045E+06	0.281986E+03	0.144330E+01
0.329082E+07	0.133918E+05	0.997054E+00	0.176744E+02	0.252409E-01	-0.182136E+02
-0.115683E+02	0.865847E+00				
0.317674E+03	0.121598E+05	0.101735E+01	0.349493E+06	0.285908E+03	0.145228E+01
0.327047E+07	0.134800E+05	0.917707E+00	0.175379E+02	0.250699E-01	-0.186880E+02
-0.115205E+02	0.864204E+00				
0.319674E+03	0.122486E+05	0.930718E+00	0.349907E+06	0.289859E+03	0.146137E+01
0.325011E+07	0.135688E+05	0.840151E+00	0.174015E+02	0.249418E-01	-0.191655E+02
-0.114708E+02	0.862492E+00				
0.321674E+03	0.123381E+05	0.846160E+00	0.350287E+06	0.293839E+03	0.147058E+01
0.322976E+07	0.136584E+05	0.764359E+00	0.172653E+02	0.248548E-01	-0.196461E+02
-0.114191E+02	0.860711E+00				
0.323674E+03	0.124283E+05	0.763678E+00	0.350635E+06	0.297847E+03	0.147991E+01
0.320941E+07	0.137486E+05	0.690334E+00	0.171292E+02	0.248083E-01	-0.201299E+02
-0.113655E+02	0.858863E+00				

TIME W ALPHA	V(R) V(I) CD	GAM(R) GAM(I)	ALT THETA(R)	RANGE QBAR	T/W VDRAG
0.325674E+03	0.125192E+05	0.683275E+00	0.350949E+06	0.301885E+03	0.148935E+01
0.318905E+07	0.138396E+05	0.618083E+00	0.169932E+02	0.248017E-01	-0.206169E+02
-0.113099E+02	0.856947E+00				
0.327674E+03	0.126107E+05	0.604905E+00	0.351231E+06	0.305953E+03	0.149892E+01
0.316870E+07	0.139312E+05	0.547568E+00	0.168574E+02	0.248338E-01	-0.211071E+02
-0.112525E+02	0.854967E+00				
0.329674E+03	0.127030E+05	0.528555E+00	0.351481E+06	0.310050E+03	0.150861E+01
0.314834E+07	0.140235E+05	0.478785E+00	0.167218E+02	0.249038E-01	-0.216006E+02
-0.111932E+02	0.852921E+00				
0.331674E+03	0.127960E+05	0.454199E+00	0.351700E+06	0.314177E+03	0.151842E+01
0.312799E+07	0.141165E+05	0.411712E+00	0.165863E+02	0.250120E-01	-0.220973E+02
-0.111321E+02	0.850812E+00				
0.333674E+03	0.128898E+05	0.381856E+00	0.351896E+06	0.318335E+03	0.152837E+01
0.310763E+07	0.142103E+05	0.346372E+00	0.164510E+02	0.251588E-01	-0.225974E+02
-0.110692E+02	0.848638E+00				
0.335674E+03	0.129842E+05	0.311454E+00	0.352042E+06	0.322523E+03	0.153844E+01
0.308728E+07	0.143048E+05	0.282702E+00	0.163160E+02	0.253425E-01	-0.231009E+02
-0.110045E+02	0.846403E+00				
0.337674E+03	0.130794E+05	0.243028E+00	0.352167E+06	0.326742E+03	0.154865E+01
0.306692E+07	0.143999E+05	0.220741E+00	0.161811E+02	0.255656E-01	-0.236078E+02
-0.109380E+02	0.844106E+00				

TIME W ALPHA	V(R) V(I) CD	GAM(R) GAM(I)	ALT THETA(R)	RANGE QBAR	T/W VDRAG
0.339674E+03	0.131753E+05	0.176520E+00	0.352263E+06	0.330991E+03	0.155900E+01
0.304657E+07	0.144959E+05	0.160439E+00	0.160464E+02	0.258263E-01	-0.241181E+02
-0.108699E+02	0.841749E+00				
0.341674E+03	0.132720E+05	0.111919E+00	0.352328E+06	0.335272E+03	0.156948E+01
0.302622E+07	0.145926E+05	0.101790E+00	0.159120E+02	0.261263E-01	-0.246319E+02
-0.108000E+02	0.839332E+00				
0.343674E+03	0.133694E+05	0.492218E-01	0.352365E+06	0.339584E+03	0.158011E+01
0.300586E+07	0.146900E+05	0.447969E-01	0.157777E+02	0.264655E-01	-0.251492E+02
-0.107285E+02	0.836855E+00				
0.345674E+03	0.134676E+05	-0.115903E-01	0.352373E+06	0.343929E+03	0.159088E+01
0.298551E+07	0.147882E+05	-0.105553E-01	0.156438E+02	0.268457E-01	-0.256700E+02
-0.106554E+02	0.834320E+00				
0.347674E+03	0.135666E+05	-0.705376E-01	0.352353E+06	0.348305E+03	0.160180E+01
0.296515E+07	0.148871E+05	-0.642805E-01	0.155100E+02	0.272676E-01	-0.261945E+02
-0.105806E+02	0.831728E+00				
0.349674E+03	0.136663E+05	-0.127638E+00	0.352305E+06	0.352713E+03	0.161287E+01
0.294480E+07	0.149869E+05	-0.116391E+00	0.153766E+02	0.277319E-01	-0.267225E+02
-0.105042E+02	0.829079E+00				
0.351674E+03	0.137668E+05	-0.182890E+00	0.352229E+06	0.357153E+03	0.162409E+01
0.292445E+07	0.150874E+05	-0.166883E+00	0.152434E+02	0.282413E-01	-0.272542E+02
-0.104262E+02	0.826374E+00				

TIME	V(R)	GAM(R)	ALT	RANGE	T/W
W	V(I)	GAM(I)	THETA(R)	QBAR	VDRAG
ALPHA	CD				
0.353674E+03	0.138682E+05	-0.236349E+00	0.352128E+06	0.361626E+03	0.163548E+01
0.290409E+07	0.151887E+05	-0.215800E+00	0.151104E+02	0.287945E-01	-0.277897E+02
-0.103468E+02	0.823615E+00				
0.355674E+03	0.139703E+05	-0.287979E+00	0.351999E+06	0.366133E+03	0.164702E+01
0.288374E+07	0.152909E+05	-0.263109E+00	0.149778E+02	0.293969E-01	-0.283288E+02
-0.102657E+02	0.820800E+00				
0.357674E+03	0.140733E+05	-0.337833E+00	0.351846E+06	0.370672E+03	0.165872E+01
0.286338E+07	0.153938E+05	-0.308853E+00	0.148454E+02	0.300477E-01	-0.288718E+02
-0.101833E+02	0.817933E+00				
0.359674E+03	0.141771E+05	-0.385902E+00	0.351666E+06	0.375245E+03	0.167060E+01
0.284303E+07	0.154976E+05	-0.353021E+00	0.147134E+02	0.307505E-01	-0.294185E+02
-0.100993E+02	0.815013E+00				
0.361674E+03	0.142817E+05	-0.432228E+00	0.351463E+06	0.379851E+03	0.168264E+01
0.282268E+07	0.156022E+05	-0.395646E+00	0.145817E+02	0.315055E-01	-0.299692E+02
-0.100139E+02	0.812040E+00				
0.363674E+03	0.143871E+05	-0.476783E+00	0.351234E+06	0.384492E+03	0.169486E+01
0.280232E+07	0.157076E+05	-0.436702E+00	0.144503E+02	0.323179E-01	-0.305237E+02
-0.992704E+01	0.809016E+00				
0.365674E+03	0.144935E+05	-0.519612E+00	0.350982E+06	0.389167E+03	0.170726E+01
0.278197E+07	0.158139E+05	-0.476224E+00	0.143192E+02	0.331891E-01	-0.310822E+02
-0.983881E+01	0.805942E+00				

TIME W ALPHA	V(R) V(I) CD	GAM(R) GAM(I)	ALT THETA(R)	RANGE QBAR	T/W VDRAG
0.367674E+03	0.146006E+05	-0.560728E+00	0.350708E+06	0.393876E+03	0.171984E+01
0.276161E+07	0.159211E+05	-0.514223E+00	0.141885E+02	0.341215E-01	-0.316447E+02
-0.974921E+01	0.802818E+00				
0.369674E+03	0.147087E+05	-0.600126E+00	0.350410E+06	0.398620E+03	0.173261E+01
0.274126E+07	0.160291E+05	-0.550690E+00	0.140581E+02	0.351189E-01	-0.322112E+02
-0.965825E+01	0.799644E+00				
0.371674E+03	0.148177E+05	-0.637812E+00	0.350090E+06	0.403400E+03	0.174557E+01
0.272091E+07	0.161380E+05	-0.585627E+00	0.139281E+02	0.361854E-01	-0.327817E+02
-0.956595E+01	0.796422E+00				
0.373674E+03	0.149275E+05	-0.673819E+00	0.349749E+06	0.408215E+03	0.175872E+01
0.270055E+07	0.162478E+05	-0.619061E+00	0.137985E+02	0.373239E-01	-0.333564E+02
-0.947236E+01	0.793152E+00				
0.375674E+03	0.150382E+05	-0.708144E+00	0.349386E+06	0.413066E+03	0.177208E+01
0.268020E+07	0.163585E+05	-0.650988E+00	0.136693E+02	0.385385E-01	-0.339352E+02
-0.937748E+01	0.789836E+00				
0.377674E+03	0.151499E+05	-0.740813E+00	0.349004E+06	0.417952E+03	0.178563E+01
0.265984E+07	0.164702E+05	-0.681426E+00	0.135405E+02	0.398333E-01	-0.345183E+02
-0.928134E+01	0.786472E+00				
0.379674E+03	0.152625E+05	-0.771828E+00	0.348602E+06	0.422875E+03	0.179940E+01
0.263949E+07	0.165828E+05	-0.710376E+00	0.134121E+02	0.412128E-01	-0.351055E+02
-0.918396E+01	0.783064E+00				

TIME / W / ALPHA	V(R) / V(I) / CD	GAM(R) / GAM(I)	ALT / THETA(R)	RANGE / QBAR	T/W / VDRAG
0.381674E+03	0.153761E+05	-0.801204E+00	0.348180E+06	0.427835E+03	0.181338E+01
0.261914E+07	0.166963E+05	-0.737848E+00	0.132842E+02	0.426812E-01	-0.356970E+02
-0.908536E+01	0.779610E+00				
0.383674E+03	0.154906E+05	-0.828955E+00	0.347741E+06	0.432832E+03	0.182758E+01
0.259878E+07	0.168108E+05	-0.763853E+00	0.131566E+02	0.442447E-01	-0.362929E+02
-0.898557E+01	0.776113E+00				
0.385674E+03	0.156061E+05	-0.855078E+00	0.347284E+06	0.437865E+03	0.184201E+01
0.257843E+07	0.169263E+05	-0.788383E+00	0.130295E+02	0.459079E-01	-0.368931E+02
-0.888460E+01	0.772572E+00				
0.387674E+03	0.157226E+05	-0.879603E+00	0.346809E+06	0.442937E+03	0.185666E+01
0.255807E+07	0.170427E+05	-0.811466E+00	0.129024E+02	0.476770E-01	-0.374977E+02
-0.878249E+01	0.768987E+00				
0.389674E+03	0.158401E+05	-0.902499E+00	0.346317E+06	0.448047E+03	0.187155E+01
0.253772E+07	0.171601E+05	-0.833069E+00	0.127767E+02	0.495602E-01	-0.381067E+02
-0.867922E+01	0.765361E+00				
0.391674E+03	0.159586E+05	-0.923803E+00	0.345810E+06	0.453195E+03	0.188668E+01
0.251737E+07	0.172786E+05	-0.853224E+00	0.126510E+02	0.515626E-01	-0.387201E+02
-0.857485E+01	0.761693E+00				
0.393674E+03	0.160781E+05	-0.943542E+00	0.345288E+06	0.458381E+03	0.190206E+01
0.249701E+07	0.173981E+05	-0.871952E+00	0.125259E+02	0.536892E-01	-0.393381E+02
-0.846940E+01	0.757984E+00				

TIME W ALPHA	V(R) V(I) CD	GAM(R) GAM(I)	ALT THETA(R)	RANGE QBAR	T/W VDRAG
0.395674E+03	0.161987E+05	-0.961692E+00	0.344751E+06	0.463606E+03	0.191769E+01
0.247666E+07	0.175186E+05	-0.889229E+00	0.124012E+02	0.559502E-01	-0.399606E+02
-0.836286E+01	0.754234E+00				
0.397674E+03	0.163203E+05	-0.978271E+00	0.344199E+06	0.468871E+03	0.193358E+01
0.245630E+07	0.176402E+05	-0.905068E+00	0.122770E+02	0.583537E-01	-0.405876E+02
-0.825528E+01	0.750445E+00				
0.399674E+03	0.164430E+05	-0.993294E+00	0.343636E+06	0.474175E+03	0.194974E+01
0.243595E+07	0.177629E+05	-0.919483E+00	0.121534E+02	0.609059E-01	-0.412193E+02
-0.814667E+01	0.746617E+00				
0.401674E+03	0.165668E+05	-0.100677E+01	0.343059E+06	0.479519E+03	0.196616E+01
0.241560E+07	0.178866E+05	-0.932474E+00	0.120303E+02	0.636166E-01	-0.418556E+02
-0.803704E+01	0.742751E+00				
0.403674E+03	0.166917E+05	-0.101869E+01	0.342471E+06	0.484904E+03	0.198287E+01
0.239524E+07	0.180115E+05	-0.944043E+00	0.119077E+02	0.664960E-01	-0.424966E+02
-0.792643E+01	0.738847E+00				
0.405674E+03	0.168177E+05	-0.102907E+01	0.341872E+06	0.490329E+03	0.199986E+01
0.237489E+07	0.181375E+05	-0.954190E+00	0.117858E+02	0.695534E-01	-0.431422E+02
-0.781484E+01	0.734906E+00				
0.407674E+03	0.169449E+05	-0.103793E+01	0.341263E+06	0.495796E+03	0.201715E+01
0.235453E+07	0.182646E+05	-0.962930E+00	0.116644E+02	0.727987E-01	-0.437925E+02
-0.770230E+01	0.730928E+00				

TIME W ALPHA	V(R) V(I) CD	GAM(R) GAM(I)	ALT THETA(R)	RANGE QBAR	T/W VDRAG
0.409674E+03	0.170732E+05	-0.104526E+01	0.340644E+06	0.501304E+03	0.203474E+01
0.233418E+07	0.183929E+05	-0.970255E+00	0.115436E+02	0.762427E-01	-0.444476E+02
-0.758882E+01	0.726915E+00				
0.411674E+03	0.172027E+05	-0.105105E+01	0.340017E+06	0.506854E+03	0.205263E+01
0.231383E+07	0.185223E+05	-0.976167E+00	0.114234E+02	0.798982E-01	-0.451074E+02
-0.747442E+01	0.722866E+00				
0.413674E+03	0.173334E+05	-0.105534E+01	0.339381E+06	0.512446E+03	0.207085E+01
0.229347E+07	0.186530E+05	-0.980678E+00	0.113038E+02	0.837749E-01	-0.457720E+02
-0.735913E+01	0.718782E+00				
0.415674E+03	0.174653E+05	-0.105812E+01	0.338739E+06	0.518082E+03	0.208939E+01
0.227312E+07	0.167849E+05	-0.983782E+00	0.111848E+02	0.878868E-01	-0.464414E+02
-0.724296E+01	0.714665E+00				
0.417674E+03	0.175985E+05	-0.105939E+01	0.338090E+06	0.523760E+03	0.210826E+01
0.225276E+07	0.189180E+05	-0.985494E+00	0.110665E+02	0.922445E-01	-0.471156E+02
-0.712594E+01	0.710514E+00				
0.419674E+03	0.177329E+05	-0.105916E+01	0.337437E+06	0.529482E+03	0.212748E+01
0.223241E+07	0.190523E+05	-0.985802E+00	0.109489E+02	0.968620E-01	-0.477946E+02
-0.700806E+01	0.706330E+00				
0.421674E+03	0.178686E+05	-0.105744E+01	0.336778E+06	0.535247E+03	0.214705E+01
0.221206E+07	0.191880E+05	-0.984717E+00	0.108319E+02	0.101752E+00	-0.484784E+02
-0.688937E+01	0.702114E+00				

TIME W ALPHA	V(R) V(I) CD	GAM(R) GAM(I)	ALT THETA(R)	RANGE QBAR	T/W VDRAG
0.423674E+03	0.180056E+05	-0.105422E+01	0.336117E+06	0.541057E+03	0.216699E+01
0.219170E+07	0.193249E+05	-0.982244E+00	0.107156E+02	0.106925E+00	-0.491670E+02
-0.676987E+01	0.697366E+00				
0.425674E+03	0.181439E+05	-0.104953E+01	0.335453E+06	0.546912E+03	0.218730E+01
0.217135E+07	0.194632E+05	-0.978380E+00	0.106001E+02	0.112400E+00	-0.498605E+02
-0.664958E+01	0.693587E+00				
0.427674E+03	0.182835E+05	-0.104335E+01	0.334787E+06	0.552812E+03	0.220799E+01
0.215100E+07	0.196028E+05	-0.973129E+00	0.104852E+02	0.118189E+00	-0.505587E+02
-0.652853E+01	0.689278E+00				
0.429674E+03	0.184246E+05	-0.103570E+01	0.334120E+06	0.558758E+03	0.222908E+01
0.213064E+07	0.197438E+05	-0.966494E+00	0.103710E+02	0.124302E+00	-0.512618E+02
-0.640672E+01	0.684939E+00				
0.431674E+03	0.185670E+05	-0.102658E+01	0.333454E+06	0.564749E+03	0.225058E+01
0.211029E+07	0.198862E+05	-0.958478E+00	0.102576E+02	0.130755E+00	-0.519697E+02
-0.628419E+01	0.680571E+00				
0.433674E+03	0.187108E+05	-0.101599E+01	0.332790E+06	0.570787E+03	0.227250E+01
0.208994E+07	0.200300E+05	-0.949070E+00	0.101449E+02	0.137562E+00	-0.526824E+02
-0.616093E+01	0.676174E+00				
0.435674E+03	0.188561E+05	-0.100393E+01	0.332127E+06	0.576872E+03	0.229485E+01
0.206958E+07	0.201753E+05	-0.938286E+00	0.100330E+02	0.144733E+00	-0.533999E+02
-0.603697E+01	0.671749E+00				

TIME W ALPHA	V(R) V(I) CD	GAM(R) GAM(I)	ALT THETA(R)	RANGE QBAR	T/W VDRAG
0.437674E+03	0.190029E+05	-0.990401E+00	0.331468E+06	0.583005E+03	0.231764E+01
0.204923E+07	0.203220E+05	-0.926109E+00	0.992193E+01	0.152284E+00	-0.541222E+02
-0.591233E+01	0.667296E+00				
0.439674E+03	0.191511E+05	-0.975398E+00	0.330813E+06	0.589186E+03	0.234088E+01
0.202888E+07	0.204702E+05	-0.912540E+00	0.981162E+01	0.160224E+00	-0.548493E+02
-0.578702E+01	0.662816E+00				
0.441674E+03	0.193009E+05	-0.958939E+00	0.330164E+06	0.595415E+03	0.236460E+01
0.200852E+07	0.206200E+05	-0.897593E+00	0.970212E+01	0.168562E+00	-0.555810E+02
-0.566106E+01	0.658309E+00				
0.443674E+03	0.194523E+05	-0.941022E+00	0.329521E+06	0.601693E+03	0.238881E+01
0.198817E+07	0.207713E+05	-0.881262E+00	0.959344E+01	0.177305E+00	-0.563176E+02
-0.553447E+01	0.653777E+00				
0.445674E+03	0.196052E+05	-0.921645E+00	0.328886E+06	0.608021E+03	0.241351E+01
0.196782E+07	0.209242E+05	-0.863544E+00	0.948561E+01	0.186462E+00	-0.570588E+02
-0.540726E+01	0.649220E+00				
0.447674E+03	0.197598E+05	-0.900804E+00	0.328259E+06	0.614399E+03	0.243874E+01
0.194746E+07	0.210787E+05	-0.844436E+00	0.937864E+01	0.196037E+00	-0.578047E+02
-0.527944E+01	0.644637E+00				
0.449674E+03	0.199160E+05	-0.878510E+00	0.327643E+06	0.620827E+03	0.246449E+01
0.192711E+07	0.212349E+05	-0.823943E+00	0.927253E+01	0.205766E+00	-0.585553E+02
-0.515104E+01	0.640031E+00				

TIME W ALPHA	V(R) V(I) CD	GAM(R) GAM(I)	ALT THETA(R)	RANGE QBAR	T/W VDRAG
0.451674E+03	0.200739E+05	-0.854752E+00	0.327038E+06	0.627307E+03	0.249080E+01
0.190676E+07	0.213928E+05	-0.802053E+00	0.916732E+01	0.215769E+00	-0.593108E+02
-0.502207E+01	0.635401E+00				
0.453674E+03	0.202335E+05	-0.829544E+00	0.326445E+06	0.633838E+03	0.251767E+01
0.188641E+07	0.215524E+05	-0.778780E+00	0.906301E+01	0.226141E+00	-0.600710E+02
-0.489255E+01	0.630748E+00				
0.455674E+03	0.203949E+05	-0.802871E+00	0.325866E+06	0.640421E+03	0.254513E+01
0.186605E+07	0.217137E+05	-0.754105E+00	0.895962E+01	0.236874E+00	-0.608362E+02
-0.476249E+01	0.626071E+00				
0.457674E+03	0.205581E+05	-0.774738E+00	0.325302E+06	0.647057E+03	0.257319E+01
0.184570E+07	0.218769E+05	-0.728033E+00	0.885717E+01	0.247953E+00	-0.616062E+02
-0.463190E+01	0.621374E+00				
0.459674E+03	0.207231E+05	-0.745144E+00	0.324754E+06	0.653747E+03	0.260188E+01
0.182535E+07	0.220418E+05	-0.700561E+00	0.875567E+01	0.259356E+00	-0.623811E+02
-0.450081E+01	0.616654E+00				
0.461674E+03	0.208899E+05	-0.714095E+00	0.324223E+06	0.660491E+03	0.263122E+01
0.180499E+07	0.222087E+05	-0.671692E+00	0.865513E+01	0.271067E+00	-0.631610E+02
-0.436923E+01	0.611914E+00				
0.463674E+03	0.210587E+05	-0.681586E+00	0.323713E+06	0.667289E+03	0.266122E+01
0.178464E+07	0.223774E+05	-0.641418E+00	0.855558E+01	0.283048E+00	-0.639459E+02
-0.423717E+01	0.607153E+00				

TIME	V(R)	GAM(R)	ALT	RANGE	T/W
W	V(I)	GAM(I)	THETA(R)	QBAR	VDRAG
ALPHA	CD				
0.465674E+03	0.212295E+05	-0.647612E+00	0.323222E+06	0.674142E+03	0.269192E+01
0.176429E+07	0.225482E+05	-0.609736E+00	0.845703E+01	0.295276E+00	-0.647360E+02
-0.410464E+01	0.602372E+00				
0.467674E+03	0.214022E+05	-0.612170E+00	0.322753E+06	0.681052E+03	0.272334E+01
0.174393E+07	0.227209E+05	-0.576640E+00	0.835950E+01	0.307709E+00	-0.655313E+02
-0.397167E+01	0.597572E+00				
0.469674E+03	0.215770E+05	-0.575265E+00	0.322307E+06	0.688017E+03	0.275549E+01
0.172358E+07	0.228957E+05	-0.542132E+00	0.826301E+01	0.320295E+00	-0.663319E+02
-0.383827E+01	0.592753E+00				
0.471674E+03	0.217539E+05	-0.536880E+00	0.321886E+06	0.695040E+03	0.278842E+01
0.170323E+07	0.230725E+05	-0.506196E+00	0.816757E+01	0.332999E+00	-0.671380E+02
-0.370445E+01	0.587916E+00				
0.473674E+03	0.219329E+05	-0.497024E+00	0.321492E+06	0.702121E+03	0.282214E+01
0.168288E+07	0.232515E+05	-0.468837E+00	0.807320E+01	0.345748E+00	-0.679499E+02
-0.357023E+01	0.583061E+00				
0.475674E+03	0.221141E+05	-0.455639E+00	0.321125E+06	0.709260E+03	0.285668E+01
0.166252E+07	0.234327E+05	-0.430046E+00	0.797993E+01	0.358483E+00	-0.687677E+02
-0.343561E+01	0.578189E+00				
0.477674E+03	0.222976E+05	-0.412868E+00	0.320789E+06	0.716459E+03	0.289209E+01
0.164217E+07	0.236161E+05	-0.389816E+00	0.788776E+01	0.371129E+00	-0.695916E+02
-0.330062E+01	0.573299E+00				

TIME W ALPHA	V(R) V(I) CD	GAM(R) GAM(I)	ALT THETA(R)	RANGE QBAR	T/W VDRAG
0.479674E+03	0.224833E+05	-0.368570E+00	0.320483E+06	0.723718E+03	0.292838E+01
0.162182E+07	0.238019E+05	-0.348152E+00	0.779671E+01	0.383620E+00	-0.704220E+02
-0.316528E+01	0.568394E+00				
0.481674E+03	0.226714E+05	-0.322781E+00	0.320210E+06	0.731037E+03	0.296559E+01
0.160146E+07	0.239900E+05	-0.305040E+00	0.770681E+01	0.395850E+00	-0.712593E+02
-0.302959E+01	0.563473E+00				
0.483437E+03	0.228392E+05	-0.281175E+00	0.319999E+06	0.737540E+03	0.299919E+01
0.158352E+07	0.241578E+05	-0.265829E+00	0.762853E+01	0.406355E+00	-0.720032E+02
-0.290970E+01	0.559123E+00				
0.485437E+03	0.230307E+05	-0.232983E+00	0.319793E+06	0.744976E+03	0.299919E+01
0.156329E+07	0.243493E+05	-0.220366E+00	0.754063E+01	0.417829E+00	-0.728543E+02
-0.277381E+01	0.554188E+00				
0.487437E+03	0.232223E+05	-0.184148E+00	0.319625E+06	0.752473E+03	0.299919E+01
0.154332E+07	0.245408E+05	-0.174254E+00	0.745431E+01	0.428694E+00	-0.737134E+02
-0.263846E+01	0.549271E+00				
0.489437E+03	0.234138E+05	-0.134632E+00	0.319495E+06	0.760033E+03	0.299919E+01
0.152361E+07	0.247323E+05	-0.127455E+00	0.736899E+01	0.438876E+00	-0.745809E+02
-0.250362E+01	0.544369E+00				
0.491437E+03	0.236053E+05	-0.844453E-01	0.319405E+06	0.767655E+03	0.299919E+01
0.150415E+07	0.249239E+05	-0.799780E-01	0.728487E+01	0.448268E+00	-0.754571E+02
-0.236931E+01	0.539483E+00				

TIME W ALPHA	V(R) V(I) CD	GAM(R) GAM(I)	ALT THETA(R)	RANGE QBAR	T/W VDRAG
0.493437E+03	0.237969E+05	-0.335773E-01	0.319356E+06	0.775339E+03	0.299919E+01
0.148493E+07	0.251154E+05	-0.318146E-01	0.720194E+01	0.456779E+00	-0.763427E+02
-0.223552E+01	0.534613E+00				
0.495437E+03	0.239884E+05	0.179788E-01	0.319350E+06	0.783085E+03	0.299919E+01
0.146596E+07	0.253069E+05	0.170422E-01	0.712023E+01	0.464323E+00	-0.772382E+02
-0.210225E+01	0.529759E+00				
0.497437E+03	0.241800E+05	0.702566E-01	0.319386E+06	0.790893E+03	0.299919E+01
0.144723E+07	0.254985E+05	0.666237E-01	0.703974E+01	0.470824E+00	-0.781442E+02
-0.196948E+01	0.524920E+00				
0.499437E+03	0.243715E+05	0.123234E+00	0.319469E+06	0.798763E+03	0.299919E+01
0.142875E+07	0.256900E+05	0.116909E+00	0.696046E+01	0.476186E+00	-0.790612E+02
-0.183723E+01	0.520098E+00				
0.501437E+03	0.245630E+05	0.176948E+00	0.319597E+06	0.806696E+03	0.299919E+01
0.141049E+07	0.258816E+05	0.167933E+00	0.688242E+01	0.480356E+00	-0.799901E+02
-0.170547E+01	0.515291E+00				
0.502194E+03	0.246356E+05	0.197472E+00	0.319658E+06	0.809716E+03	0.299919E+01
0.140364E+07	0.259541E+05	0.187440E+00	0.685319E+01	0.481602E+00	-0.803450E+02
-0.165571E+01	0.513475E+00				

ORBITER ABORT DATA
VEHICLE CHARACTERISTICS

CASE 65

STAGE	1	2
GROSS STAGE WEIGHT,(LB)	4817477.0	3838478.0
GROSS STAGE THRUST/WEIGHT	0.828	0.994
THRUST ACTUAL,(LB)	3990000.0	3815000.0
ISP VACUUM,(SEC)	466.700	466.700
STRUCTURE,(LB)	0.0	796009.0
PROPELLANT,(LB)	978998.6	2470349.0
PERF. FRAC.,(NU)	0.2032	0.6436
PROPELLANT FRAC.,(NUB)	1.0000	0.7563
BURNOUT TIME,(SEC)	280.185	582.390
BURNOUT VELOCITY,(FT/SEC)	10940.555	25580.176
BURNOUT GAMMA,(DEGREES)	4.104	0.650
BURNOUT ALTITUDE,(FT)	347293.4	362190.9
BURNOUT RANGE,(NM)	208.2	966.2
IDEAL VELOCITY,(FT/SEC)	14600.9	30091.5

ON-ORBIT PROPELLANT USED,(LB) 42891.0
OMS-ORBIT 95354.1 OMS-ASCENT 0.0
ON ORBIT PROPELLANT AVAIL,(LB) 95354.1
DELTA ON ORBIT PROPELLANT,(LB) 52463.1

ON-ORBIT MISSION PROP REQ'D,(LB) 25965.9

THETA= 38.47 PITCH RATE= 0.00226 ATTEMPTS TO CONVERGE= 0

SUMMARY WEIGHT STATEMENT (ABORT MODE) CASE 65

ORBITER WEIGHT BREAKDOWN		
DRY WEIGHT	727620.000	POUNDS
PERSONNEL	3000.000	POUNDS
RESIDUALS	2070.000	POUNDS
RESERVES	3300.000	POUNDS
IN-FLIGHT LOSSES	10439.000	POUNDS
ACPS PROPELLANT	8280.000	POUNDS
OMS PROPELLANT	52463.125	POUNDS
PAYLOAD	509653.000	POUNDS
BALLAST FOR CG CONTROL	0.0	POUNDS
OMS INSTALLATION KITS	0.0	POUNDS
PAYLOAD MODS	0.0	POUNDS
TOTAL END BOOST (ORBITER ONLY)	1316825.00	POUNDS
OMS BURNED DURING ASCENT	42891.000	POUNDS
ACPS BURNED DURING ASCENT	10000.000	POUNDS
EXTERNAL MAIN TANK		
TANK DRY WEIGHT	2640.000	POUNDS
RESIDUALS	17730.000	POUNDS
PROPELLANT BIAS	(2640.000)	POUNDS
PRESSURANT	(2120.000)	POUNDS
TANK AND LINES	(9320.000)	POUNDS
ENGINES	(3650.000)	POUNDS
FLIGHT PERFORMANCE RESERVE	20930.000	POUNDS
UNBURNED PROPELLANT (MAIN TANK)	0.0	POUNDS
TOTAL END BOOST (EXTERNAL TANK)	41300.000	POUNDS
USABLE PROPELLANT (EXTERNAL TANK)	5092633.00	POUNDS
FLYBACK PROPELLANT (FIRST STAGE)	186864.937	POUNDS
SOLID ROCKET MOTOR (FIRST STAGE)	9040548.00	POUNDS
SRM CASE WEIGHT(2)	1045488.87	POUNDS
SRM STRUCTURE & RCVY WEIGHT	0.0	POUNDS
SRM INERT STAGING WEIGHT	1045488.87	POUNDS
USABLE SRM PROPELLANT	7995060.00	POUNDS
TOTAL GROSS LIFT-OFF WEIGHT (GLOW)	15731068.0	POUNDS

ORBITER WEIGHT BREAKDOWN		
DRY WEIGHT	727620.000	POUNDS
PERSONNEL	3000.000	POUNDS
RESIDUALS	2070.000	POUNDS
RESERVES	3300.000	POUNDS
IN-FLIGHT LOSSES	10439.000	POUNDS
ACPS PROPELLANT	7530.000	POUNDS
OMS PROPELLANT	0.0	POUNDS
PAYLOAD	509656.187	POUNDS
BALLAST FOR CG CONTROL	0.0	POUNDS
OMS INSTALLATION KITS	0.0	POUNDS
PAYLOAD MODS	0.0	POUNDS
TOTAL END BOOST (ORBITER ONLY)	1263615.00	POUNDS
OMS BURNED DURING ASCENT	95354.125	POUNDS
ACPS BURNED DURING ASCENT	10750.000	POUNDS
EXTERNAL MAIN TANK		
TANK DRY WEIGHT	2640.000	POUNDS
RESIDUALS	17730.000	POUNDS
PROPELLANT BIAS	(2640.000)	POUNDS
PRESSURANT	(2120.000)	POUNDS
TANK AND LINES	(9320.000)	POUNDS
ENGINES	(3650.000)	POUNDS
FLIGHT PERFORMANCE RESERVE	11837.000	POUNDS
UNBURNED PROPELLANT (MAIN TANK)	454129.812	POUNDS
TOTAL END BOOST (EXTERNAL TANK)	486336.812	POUNDS
USABLE PROPELLANT (EXTERNAL TANK)	4647596.00	POUNDS
FLYBACK PROPELLANT (FIRST STAGE)	186864.937	POUNDS
SOLID ROCKET MOTOR (FIRST STAGE)	9040548.00	POUNDS
SRM CASE WEIGHT(2)	1045488.87	POUNDS
SRM STRUCTURE & RCVY WEIGHT	0.0	POUNDS
SRM INERT STAGING WEIGHT	1045488.87	POUNDS
USABLE SRM PROPELLANT	7995060.00	POUNDS
TOTAL GROSS LIFT-OFF WEIGHT (GLOW)	15731068.0	POUNDS

VEHICLE CHARACTERISTICS (RTLS MODE) CASE 65

STAGE	1	2	3	4	5
GROSS STAGE WEIGHT,(LB)	4817477.0	4711475.0	4711475.0	3046419.0	2526652.0
GROSS STAGE THRUST/WEIGHT	0.792	0.810	0.852	1.310	1.510
THRUST ACTUAL,(LB)	3815000.0	3815000.0	4015000.0	3990000.0	3815000.0
ISP VACUUM,(SEC)	466.700	466.700	466.592	466.700	466.700
STRUCTURE,(LB)	0.0	0.0	0.0	0.0	786166.0
PROPELLANT,(LB)	106001.7	0.0	1665056.0	519766.9	776699.2
PERF. FRAC.,(NU)	0.0220	0.0	0.3534	0.1706	0.3074
PROPELLANT FRAC.,(NUB)	1.0000	0.0	1.0000	1.0000	0.4970
BURNOUT TIME,(SEC)	178.640	178.640	372.140	432.936	526.708
BURNOUT VELOCITY,(FT/SEC)	8335.742	8335.738	2572.291	751.336	3476.763
BURNOUT GAMMA,(DEGREES)	12.690	12.690	-12.711	-61.328	175.868
BURNOUT ALTITUDE,(FT)	219979.1	219968.0	302894.7	263349.4	230004.1
BURNOUT RANGE,(NM)	71.1	71.1	200.5	201.9	159.4
IDEAL VELOCITY,(FT/SEC)	11254.6	11254.6	17800.4	20609.4	26124.7

THETA=157.64 PITCH RATE= 0.00232 ATTEMPTS TO CONVERGE= 4

UNBURNED MAIN PROPELLANT,(LB) 454129.8

PAYLOAD,(LB) 509656.2

PROPELLANT SUMMARY FOR THE ABORT MODES FOR CASE 65

ASCENT TRAJECTORY SHAPED TO THE NOMINAL MISSION MODE UP TO 165.674 SECONDS

UNBURNED MAIN PROPELLANT IN THE ABORT MODE = 0.0 POUNDS

EXCESS ON-ORBIT PROPELLANT IN THE ABORT MODE = 26497.250 POUNDS

UNBURNED MAIN PROPELLANT IN THE RTLS MODE = 454129.812 POUNDS

EXCESS ON-ORBIT PROPELLANT IN THE RTLS MODE = 0.0 POUNDS

MINUS SIGN INDICATES PROPELLANT SHORTAGE IN BURN MODE INDICATED

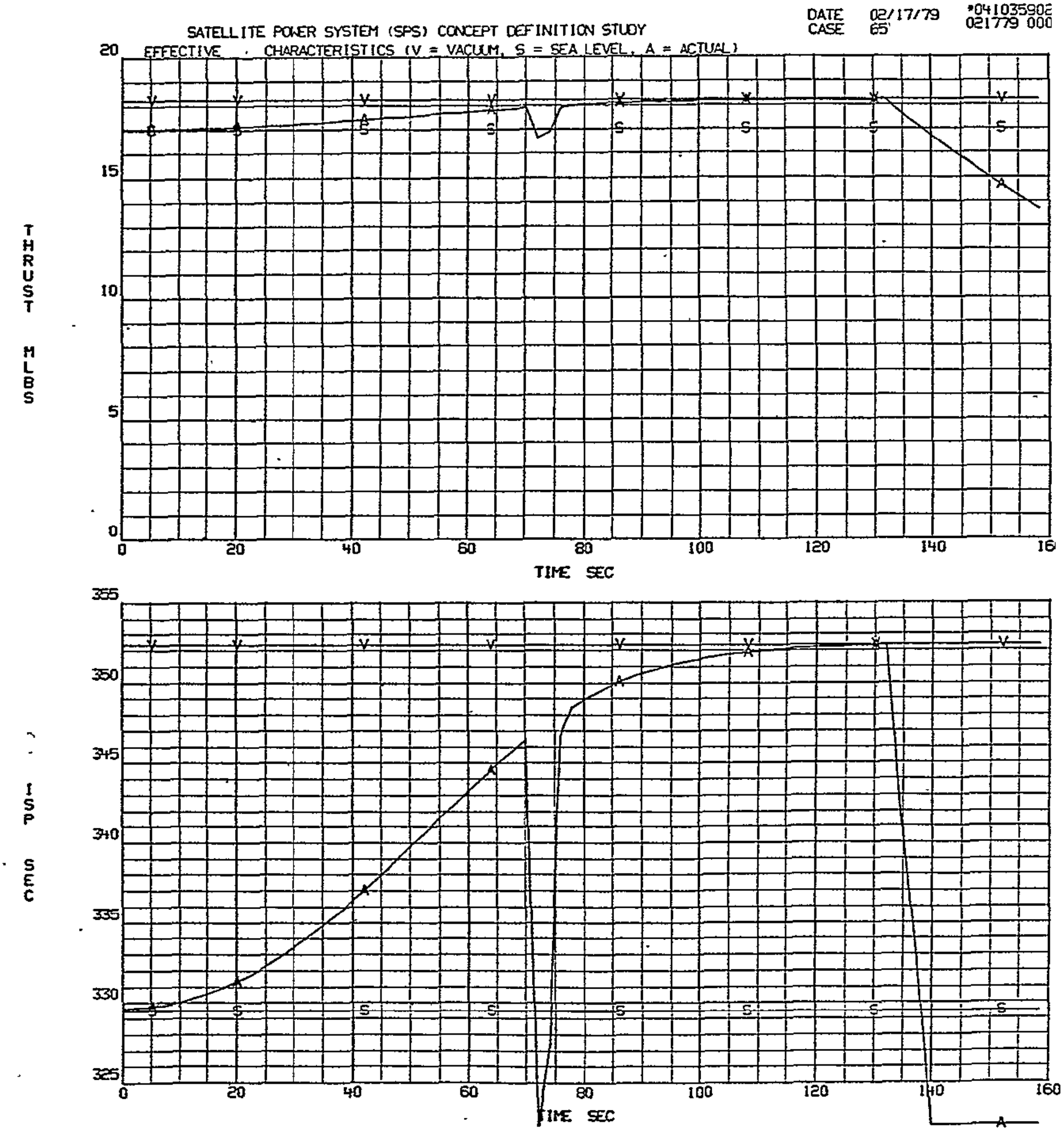
SHUTTLE SYSTEM NET PAYLOAD WITHOUT OMS KITS = 509653.000 POUNDS

MAIN PROPELLANT BURNED TO AOA/RTLS ABORT TIME= 1686177.00 POUNDS

SHUTTLE GROSS LIFT-OFF WEIGHT (GLOW) = 15731068.0 POUNDS

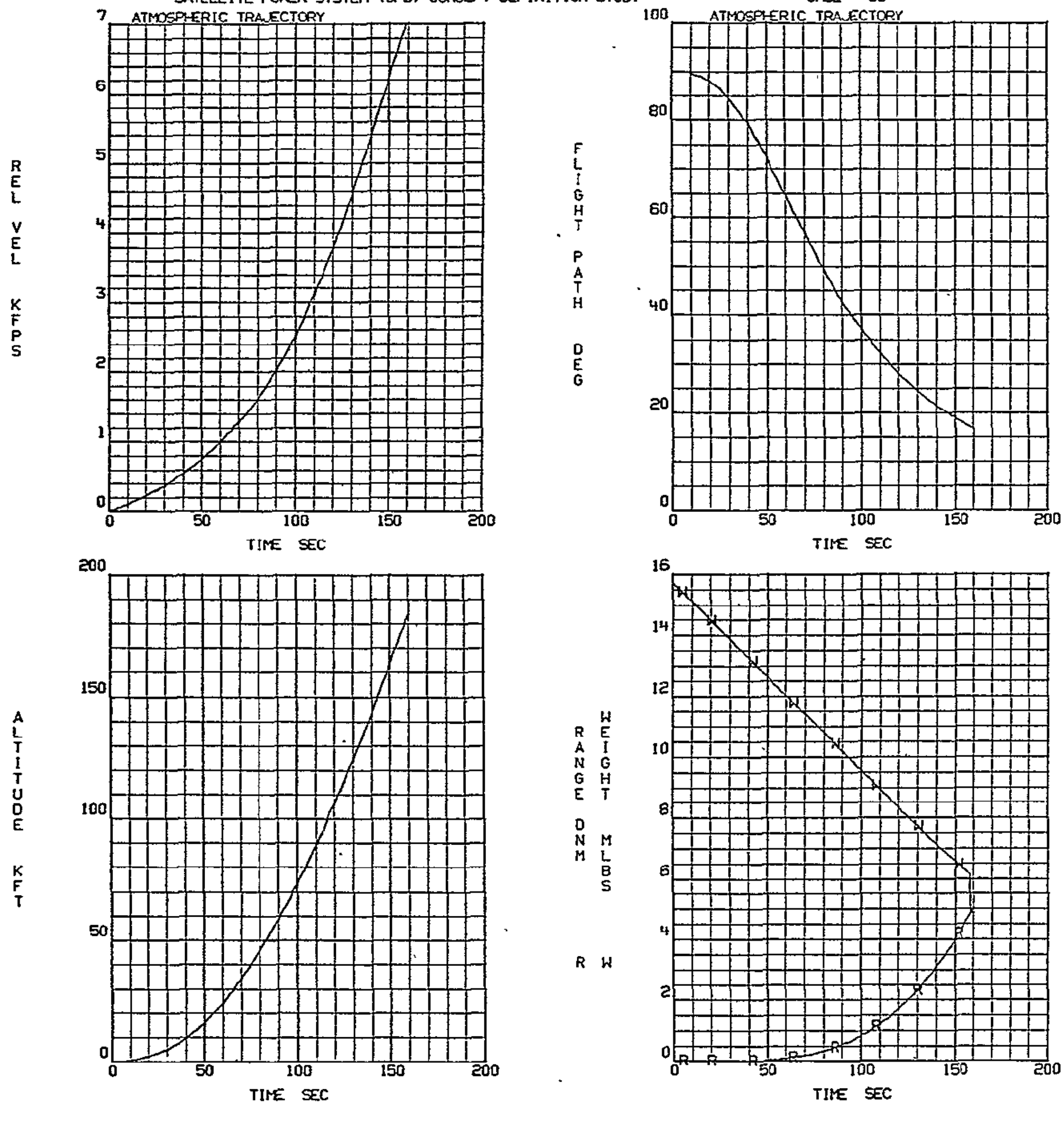
PROPELLANT CROSS FEED FROM FIRST - SECOND STAGE= 1612009.00 POUNDS

SECOND STAGE PROPELLANT CAPACITY - CROSS FEED = 3480624.00 POUNDS


DATE 02/17/79
CASE 65
*041035902
021779 000
SATELLITE POWER SYSTEM (SPS) CONCEPT DEFINITION STUDY
EFFECTIVE . CHARACTERISTICS (V = VACUUM, S = SEA LEVEL, A = ACTUAL)
THRUST MLBS
TIME SEC
ISP SEC
TIME SEC

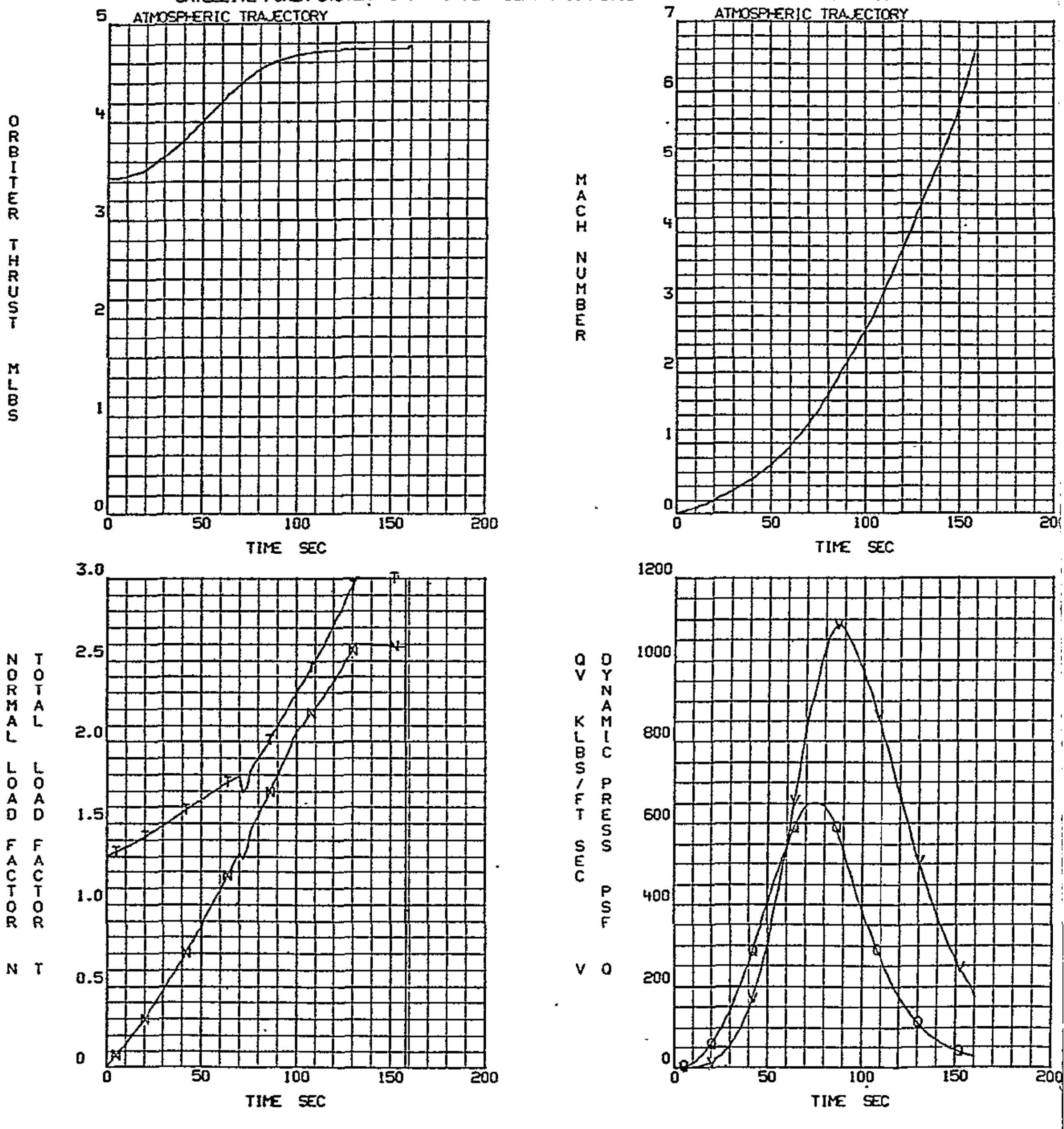
DATE 02/17/79 *04103590201
CASE 65 021779 0006

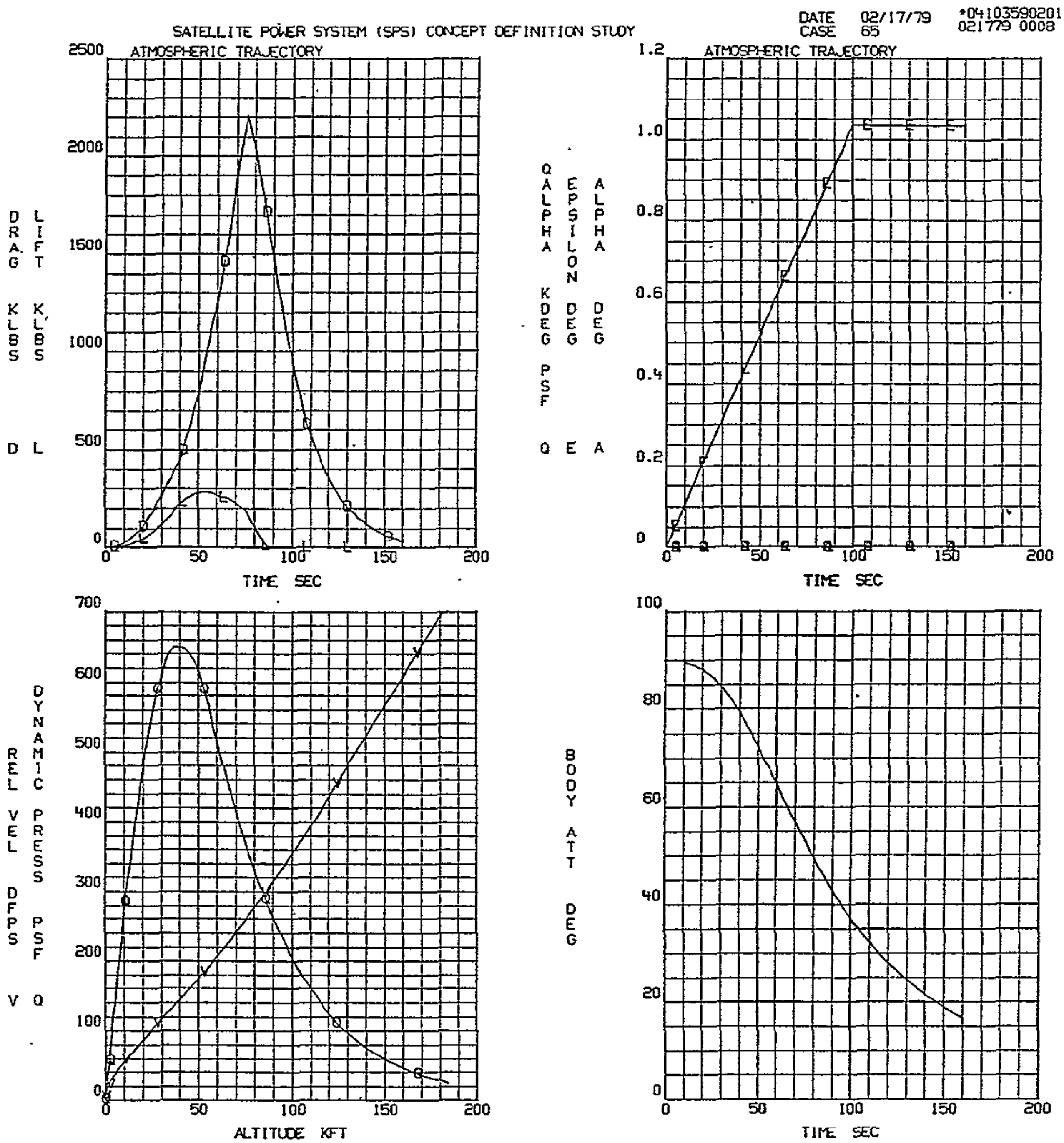
SATELLITE POWER SYSTEM (SPS) CONCEPT DEFINITION STUDY



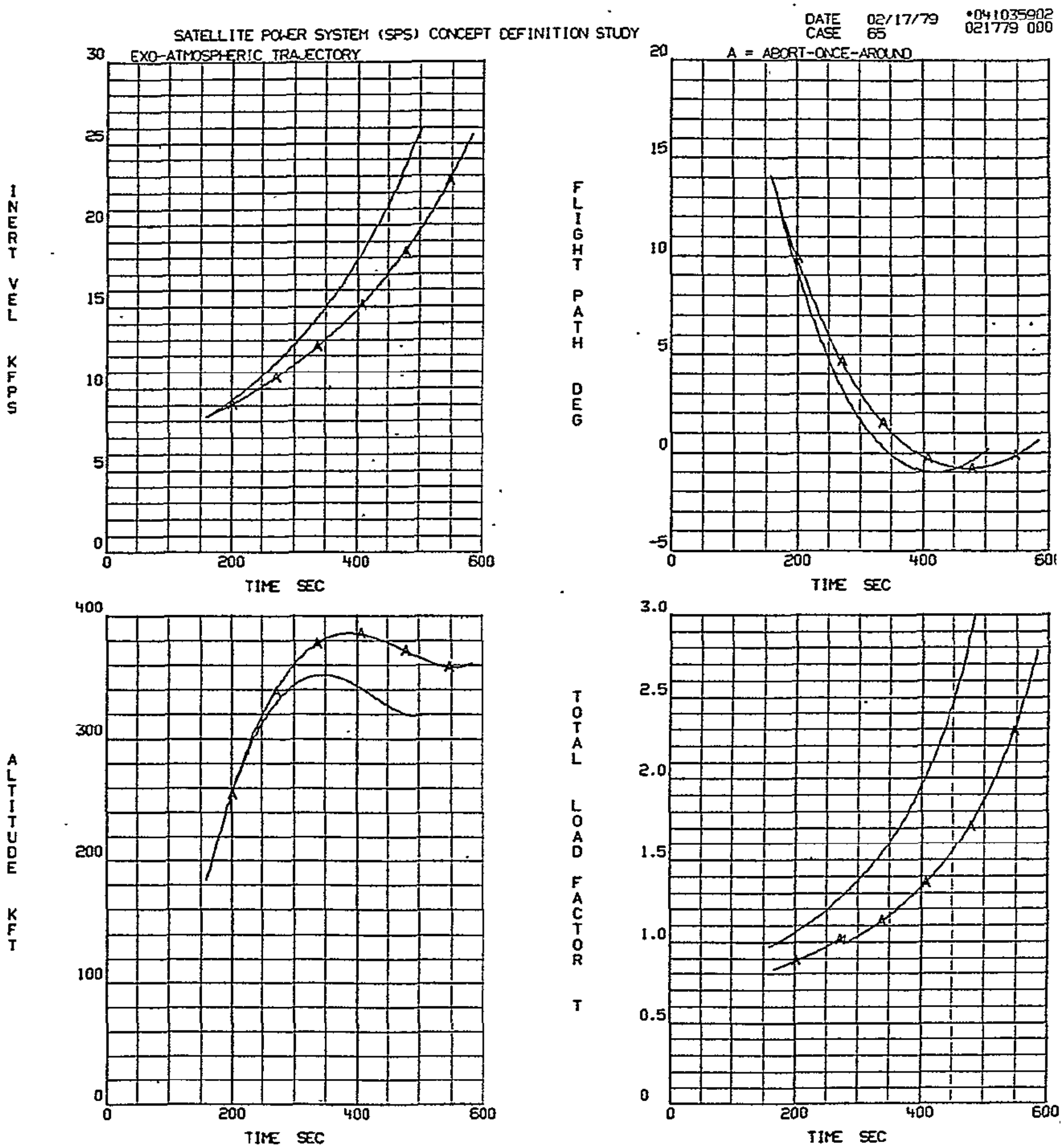
SATELLITE POWER SYSTEM (SPS) CONCEPT DEFINITION STUDY

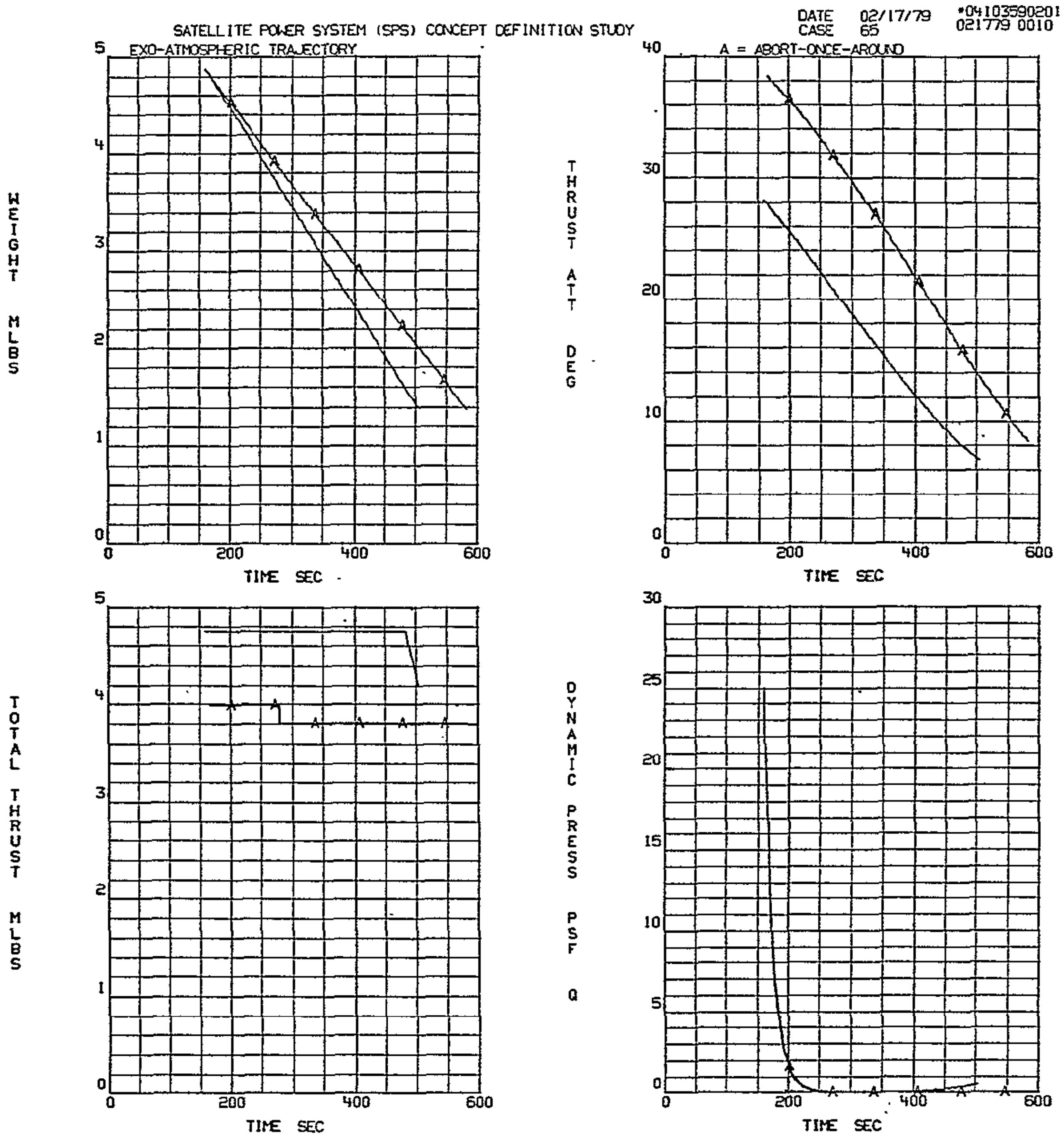
DATE 02/17/79 *041035902
CASE 65 021779 000



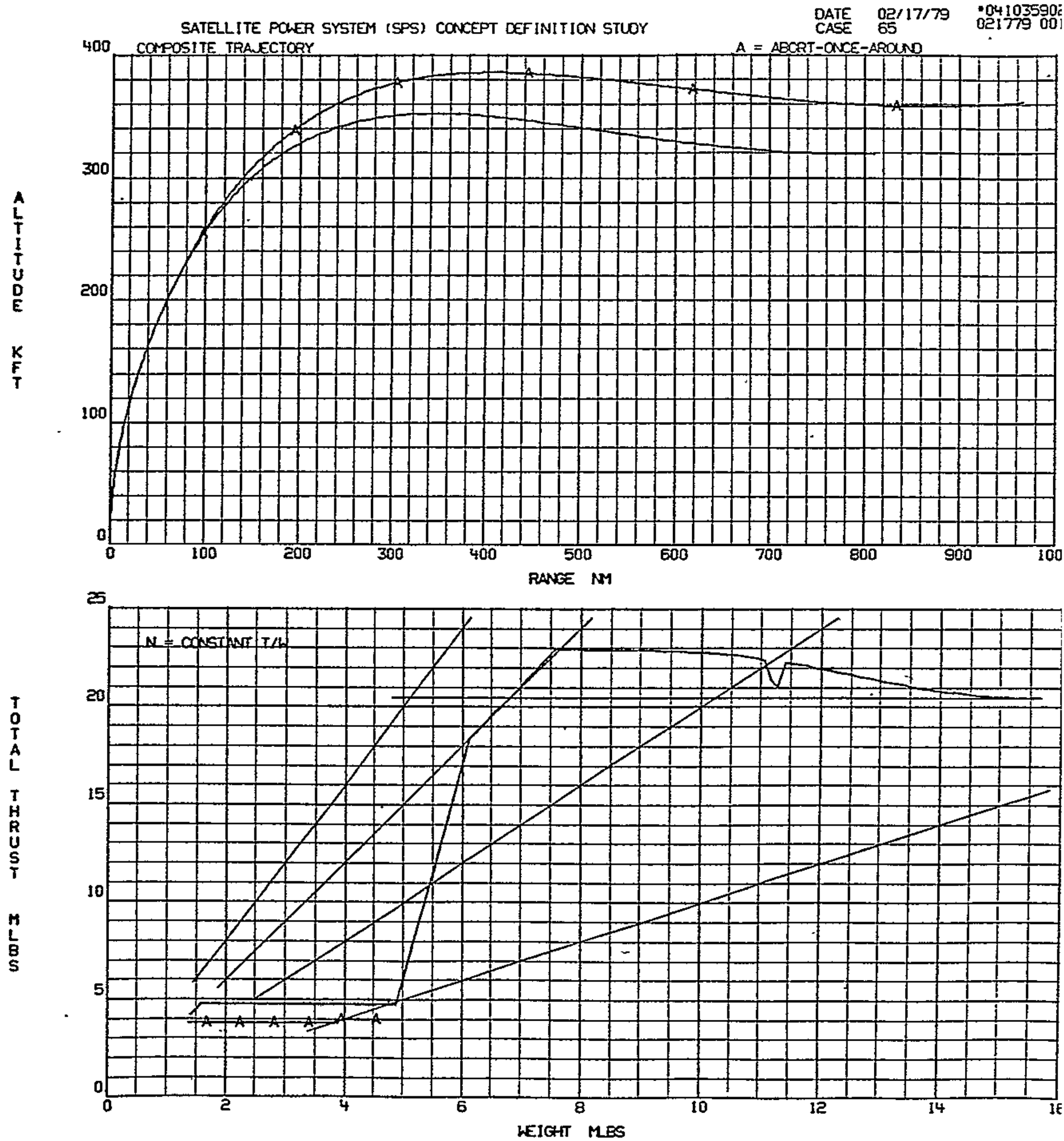


SATELLITE POWER SYSTEM (SPS) CONCEPT DEFINITION STUDY
DATE 02/17/79
CASE 65
*04103590201
021779 0008
ATMOSPHERIC TRAJECTORY
DRAG KLBS D
LIFT KLBS L
QALPHA KDEG PSF Q
EPSILON DEG E
ALPHA DEG A
TIME SEC
REL VEL DFPS V
DYNAMIC PRESS PSF Q
ALTITUDE KFT
BODY ATT DEG

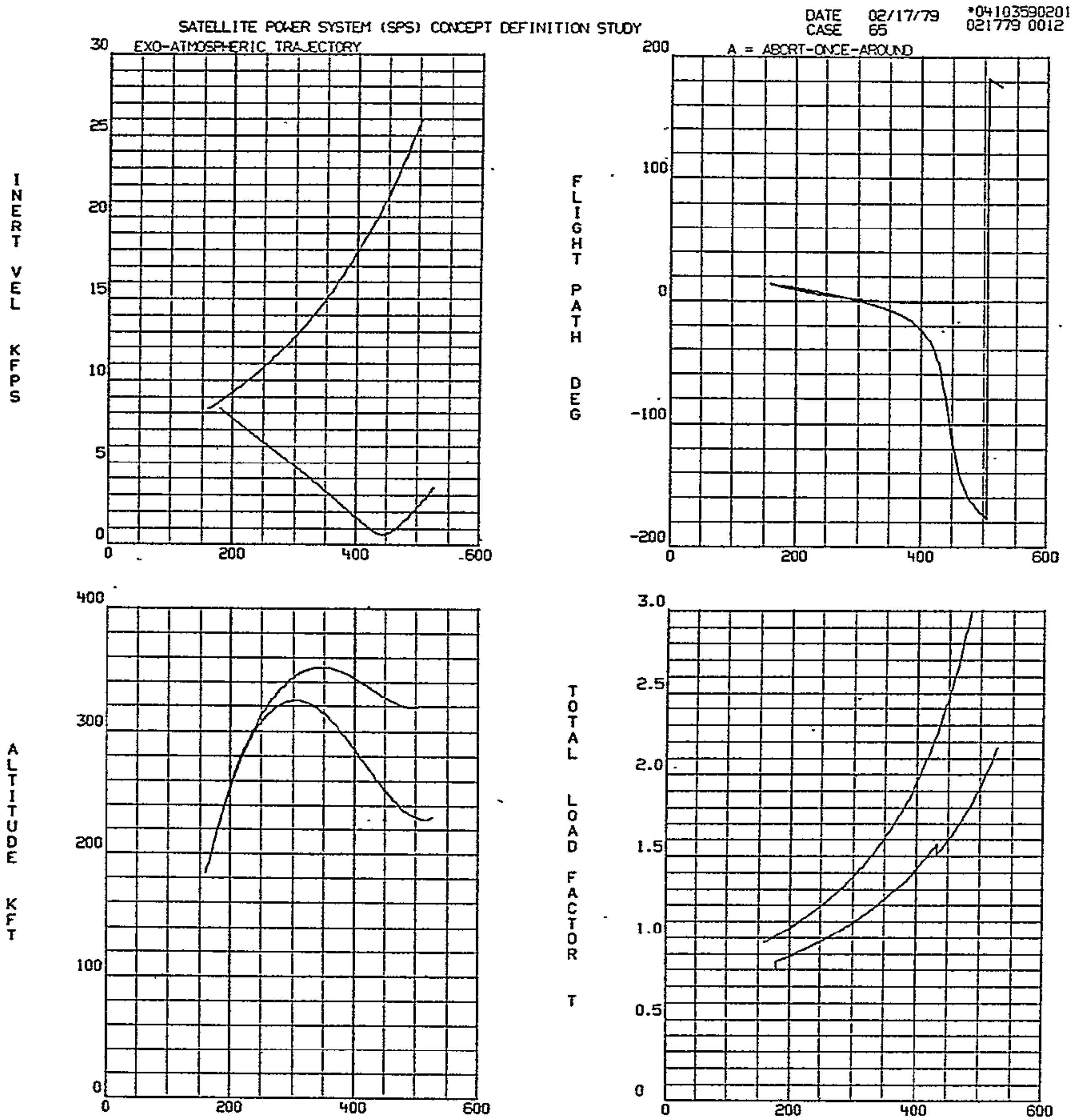

SATELLITE POWER SYSTEM (SPS) CONCEPT DEFINITION STUDY
DATE 02/17/79
CASE 65
*041035902
021779 000
EXO-ATMOSPHERIC TRAJECTORY
A = ABORT-ONCE-AROUND
INERT VEL KFPS
FLIGHT PATH DEG
ALTITUDE KFT
TOTAL LOAD FACTOR T
TIME SEC
TIME SEC
TIME SEC
TIME SEC

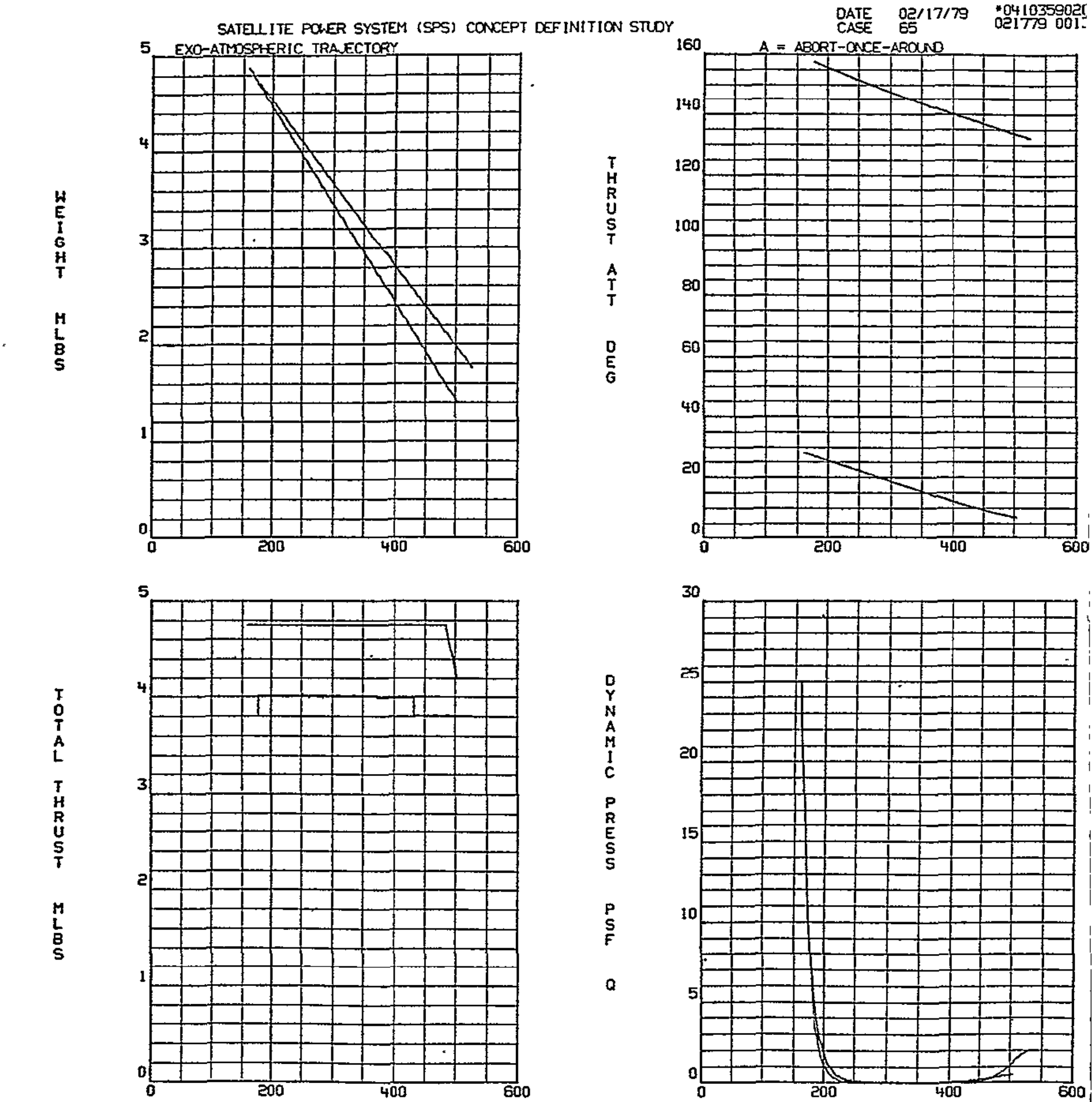


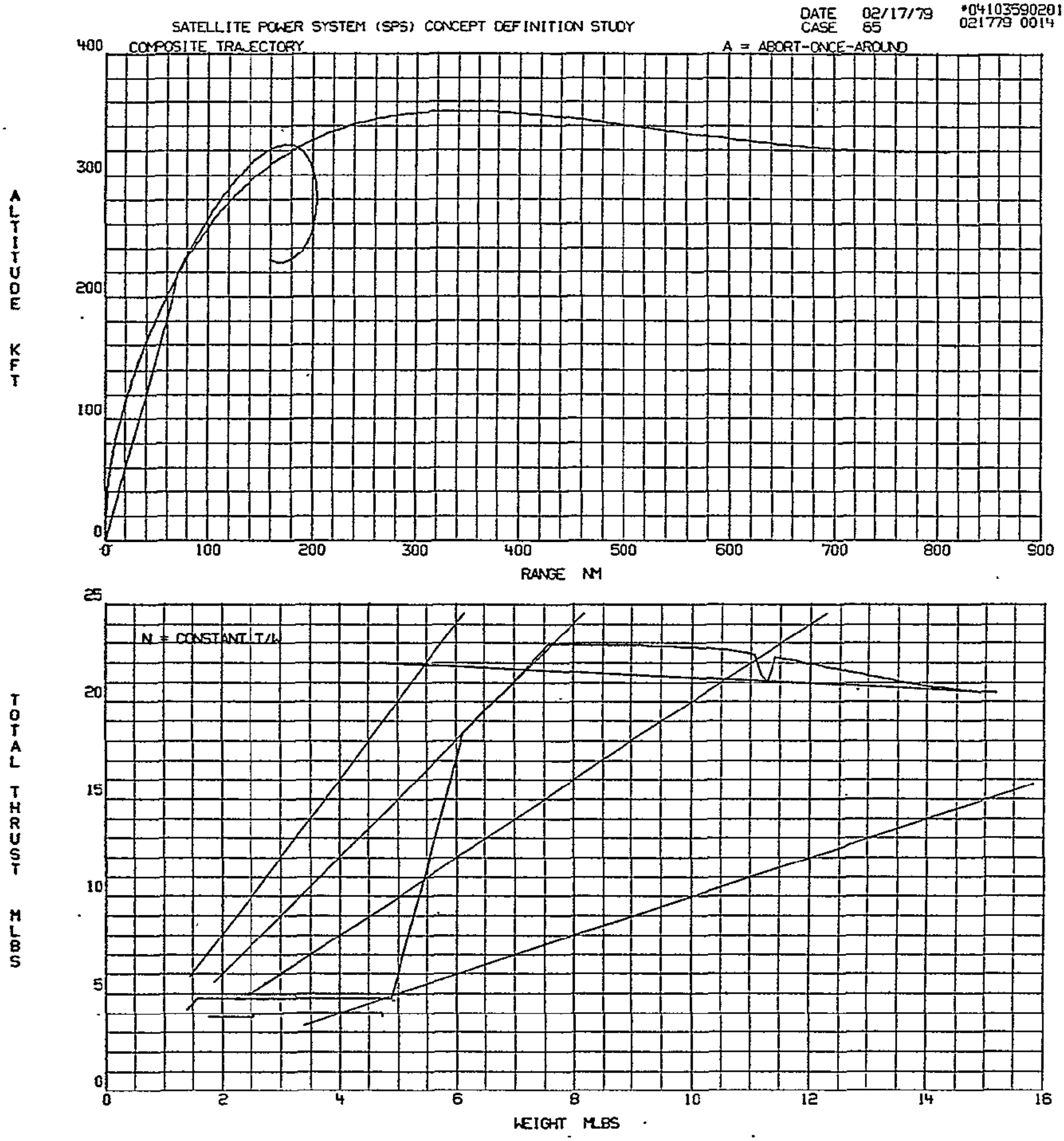
DATE 02/17/79
CASE 65
*04103590201
021779 0010
SATELLITE POWER SYSTEM (SPS) CONCEPT DEFINITION STUDY
EXO-ATMOSPHERIC TRAJECTORY
A = ABORT-ONCE-AROUND
WEIGHT M LBS
THRUST ATT DEG
TOTAL THRUST M LBS
DYNAMIC PRESS PSF Q
TIME SEC



DATE 02/17/79
CASE 65
*04103590
021779 00
SATELLITE POWER SYSTEM (SPS) CONCEPT DEFINITION STUDY
COMPOSITE TRAJECTORY
A = ABORT-ONCE-AROUND
ALTITUDE KFT
RANGE NM
N = CONSTANT T/W
TOTAL THRUST MLBS
WEIGHT MLBS


SATELLITE POWER SYSTEM (SPS) CONCEPT DEFINITION STUDY
DATE 02/17/79
CASE 65
*04103590201
021779 0012
EXO-ATMOSPHERIC TRAJECTORY
A = ABORT-ONCE-AROUND
INERT VEL KFPS
FLIGHT PATH DEG
ALTITUDE KFT
TOTAL LOAD FACTOR T
30
25
20
15
10
5
0
200
100
0
-100
-200
400
300
200
100
0
3.0
2.5
2.0
1.5
1.0
0.5
0
200
400
600


SATELLITE POWER SYSTEM (SPS) CONCEPT DEFINITION STUDY
DATE 02/17/79
CASE 65
EXO-ATMOSPHERIC TRAJECTORY
A = ABORT-ONCE-AROUND
WEIGHT MLBS
THRUST ATT DEG
TOTAL THRUST MLBS
DYNAMIC PRESS PSF Q



SATELLITE POWER SYSTEM (SPS) CONCEPT DEFINITION STUDY
DATE 02/17/79
CASE 65
*04103590201
021779 0014
COMPOSITE TRAJECTORY
A = ABORT-ONCE-AROUND
ALTITUDE KFT
RANGE NM
N = CONSTANT T/W
TOTAL THRUST MLBS
WEIGHT MLBS

B.2 HLLV THROTTLING STUDY

This section contains the results of variations in throttling percentage between first and second stage engines to stay within the maximum load factor and dynamic pressure constraints, 3 g and 650 PSF respectively. The propellant weight consumed by the first and second stage during ascent was held constant and the amount of crossfeed propellant from the first to second stage was allowed to vary accordingly (i.e., the second stage propellant loaded weight was allowed to vary). An assessment was made as to the effects on payload, staging velocity and gross liftoff weight (GLOW). A summary of the results are tabulated in Table B.2-1 and vehicle characteristics are included in the tabulated sheets for each case. (Refer to Section B.1 for reference vehicle characteristics.)

Table B.2-1. Engine Throttle Trade Summary

CASE NO.	1ST STAGE THROTTLE %	STAGING VELOCITY (FT/SEC)	PAYLOAD ($LB \times 10^3$)	2ND STAGE PROP. LOADED $LB \times 10^6$	GLOW $LB \times 10^6$	GLOW/PAYLOAD
REF. CONFIG.	100	6978	509.7	3.481	15.73	30.87
85	86	6893	505.9	3.509	15.73	31.10
65	68	6887	499.6	3.543	15.72	31.46
45	50	6808	499.5	3.574	15.72	31.73
66	0	6646	508.4	3.631	15.73	30.92

As may be seen from Table B.2-1, a 2.8% decrease in payload is realized when the throttle level of the first stage is reduced from 100% to 50% with a similar decrease in staging velocity. However, when throttling 100% with the second stage, essentially the same payload capability as afforded by the reference configuration was achieved at a significantly lower staging velocity (Case 66).

VEHICLE CHARACTERISTICS (NOMINAL MISSION) CASE 85

STAGE	1	2	3
GROSS STAGE WEIGHT,(LB)	15733913.0	4920108.0	4842005.0
GROSS STAGE THRUST/WEIGHT	1.300	0.965	0.981
THRUST ACTUAL,(LB)	20454048.0	4750000.0	4750000.0
ISP VACUUM,(SEC)	370.883	466.700	466.700
STRUCTURE,(LB)	1045488.9	0.0	809575.0
PROPELLANT,(LB)	9578332.0	78103.0	3431252.0
PERF. FRAC.,(NU)	0.6088	0.0159	0.7086
PROPELLANT FRAC.,(NUB)	0.9016	1.0000	0.8091
BURNOUT TIME,(SEC)	157.588	165.261	504.240
BURNOUT VELOCITY,(FT/SEC)	8149.641	8323.281	25954.121
BURNOUT GAMMA,(DEGREES)	15.057	13.955	0.187
BURNOUT ALTITUDE,(FT)	182132.3	197947.2	319657.5
BURNOUT RANGE,(NM)	47.3	55.7	810.9
IDEAL VELOCITY,(FT/SEC)	10888.8	11129.1	29646.4

INJECTION VELOCITY,(FT/SEC)	0.0	FLYBACK RANGE(NM)	216.4
INJECTION PROPELLANT,(LB)	0.0	FLYBACK PROP(LBS)	189983.5
ON ORBIT DELTA-V,(FT/SEC)	1083.5		
ON ORBIT PROPELLANT,(LB)	95325.3		
ON ORBIT ISP,(SEC)	466.7		

THETA= 27.39 PITCH RATE= 0.00182 ATTEMPTS TO CONVERGE= 3

PAYLOAD,(LB) 505852.0

SUMMARY WEIGHT STATEMENT (NOMINAL MISSION) CASE 85

ORBITER WEIGHT BREAKDOWN		
DRY WEIGHT	731000.000	POUNDS
PERSONNEL	3000.000	POUNDS
RESIDUALS	2070.000	POUNDS
RESERVES	3300.000	POUNDS
IN-FLIGHT LOSSES	10508.000	POUNDS
ACPS PROPELLANT	18280.000	POUNDS
OMS PROPELLANT	95325.312	POUNDS
PAYLOAD	505852.000	POUNDS
BALLAST FOR CG CONTROL	0.0	POUNDS
OMS INSTALLATION KITS	0.0	POUNDS
PAYLOAD MODS	0.0	POUNDS
TOTAL END BOOST (ORBITER ONLY)	1369335.00	POUNDS
OMS BURNED DURING ASCENT	0.0	POUNDS
ACPS BURNED DURING ASCENT	0.0	POUNDS
EXTERNAL MAIN TANK		
TANK DRY WEIGHT	2640.000	POUNDS
RESIDUALS	17847.000	POUNDS
PROPELLANT BIAS	(2640.000)	POUNDS
PRESSURANT	(2120.000)	POUNDS
TANK AND LINES	(9437.000)	POUNDS
ENGINES	(3650.000)	POUNDS
FLIGHT PERFORMANCE RESERVE	20930.000	POUNDS
UNBURNED PROPELLANT (MAIN TANK)	0.0	POUNDS
TOTAL END BOOST (EXTERNAL TANK)	41417.000	POUNDS
USABLE PROPELLANT (EXTERNAL TANK)	5092633.00	POUNDS
FLYBACK PROPELLANT (FIRST STAGE)	189983.500	POUNDS
SOLID ROCKET MOTOR (FIRST STAGE)	9040548.00	POUNDS
SRM CASE WEIGHT(2)	1045488.87	POUNDS
SRM STRUCTURE & RCVY WEIGHT	0.0	POUNDS
SRM INERT STAGING WEIGHT	1045488.87	POUNDS
USABLE SRM PROPELLANT	7995060.00	POUNDS
TOTAL GROSS LIFT-OFF WEIGHT (GLOW)	15733913.0	POUNDS

VEHICLE CHARACTERISTICS (NOMINAL MISSION) CASE 65

STAGE	1	2	3
GROSS STAGE WEIGHT,(LB)	15719436.0	4952269.0	4873984.0
GROSS STAGE THRUST/WEIGHT	1.300	0.959	0.975
THRUST ACTUAL,(LB)	20435232.0	4750000.0	4750000.0
ISP VACUUM,(SEC)	370.900	466.700	466.700
STRUCTURE,(LB)	1045488.9	0.0	814780.0
PROPELLANT,(LB)	9545080.0	78285.0	3464330.0
PERF. FRAC.,(NU)	0.6072	0.0158	0.7108
PROPELLANT FRAC.,(NUB)	0.9013	1.0000	0.8096
BURNOUT TIME,(SEC)	157.086	164.777	506.964
BURNOUT VELOCITY,(FT/SEC)	8152.324	8331.051	25954.117
BURNOUT GAMMA,(DEGREES)	13.752	12.493	0.187
BURNOUT ALTITUDE,(FT)	173511.4	188242.0	319656.5
BURNOUT RANGE,(NM)	47.6	56.0	817.0
IDEAL VELOCITY,(FT/SEC)	10826.0	11065.3	29693.2

INJECTION VELOCITY,(FT/SEC) 0.0 FLYBACK RANGE(NM) 196.8
INJECTION PROPELLANT,(LB) 0.0 FLYBACK PROP(LBS) 176597.2

ON ORBIT DELTA-V,(FT/SEC) 1083.5
ON ORBIT PROPELLANT,(LB) 95236.8
ON ORBIT ISP,(SEC) 466.7

THETA= 31.41 PITCH RATE= 0.00220 ATTEMPTS TO CONVERGE= 3

PAYLOAD,(LB) 499637.0

SUMMARY WEIGHT STATEMENT (NOMINAL MISSION) CASE 65

ORBITER WEIGHT BREAKDOWN		
DRY WEIGHT	735930.000	POUNDS
PERSONNEL	3000.000	POUNDS
RESIDUALS	2070.000	POUNDS
RESERVES	3300.000	POUNDS
IN-FLIGHT LOSSES	10610.000	POUNDS
ACPS PROPELLANT	18280.000	POUNDS
OMS PROPELLANT	95236.812	POUNDS
PAYLOAD	499637.000	POUNDS
BALLAST FOR CG CONTROL	0.0	POUNDS
OMS INSTALLATION KITS	0.0	POUNDS
PAYLOAD MODS	0.0	POUNDS
TOTAL END BOOST (ORBITER ONLY)	1368063.00	POUNDS
OMS BURNED DURING ASCENT	0.0	POUNDS
ACPS BURNED DURING ASCENT	0.0	POUNDS
EXTERNAL MAIN TANK		
TANK DRY WEIGHT	2640.000	POUNDS
RESIDUALS	18020.000	POUNDS
PROPELLANT BIAS	(2640.000)	POUNDS
PRESSURANT	(2120.000)	POUNDS
TANK AND LINES	(9610.000)	POUNDS
ENGINES	(3650.000)	POUNDS
FLIGHT PERFORMANCE RESERVE	20930.000	POUNDS
UNBURNED PROPELLANT (MAIN TANK)	0.0	POUNDS
TOTAL END BOOST (EXTERNAL TANK)	41590.000	POUNDS
USABLE PROPELLANT (EXTERNAL TANK)	5092633.00	POUNDS ✓
FLYBACK PROPELLANT (FIRST STAGE)	176597.250	POUNDS
SOLID ROCKET MOTOR (FIRST STAGE)	9040548.00	POUNDS
SRM CASE WEIGHT(2)	1045488.87	POUNDS
SRM STRUCTURE & RCVY WEIGHT	0.0	POUNDS
SRM INERT STAGING WEIGHT	1045488.87	POUNDS
USABLE SRM PROPELLANT	7995060.00	POUNDS ✓
TOTAL GROSS LIFT-OFF WEIGHT (GLOW)	15719436.0	POUNDS

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VEHICLE CHARACTERISTICS (NOMINAL MISSION) CASE 45

STAGE	1	2	3
GROSS STAGE WEIGHT,(LB)	15720730.0	4983925.0	4902416.0
GROSS STAGE THRUST/WEIGHT	1.300	0.953	0.969
THRUST ACTUAL,(LB)	20436912.0	4750000.0	4750000.0
ISP VACUUM,(SEC)	370.898	466.700	466.700
STRUCTURE,(LB)	1045488.9	0.0	819097.0
PROPELLANT,(LB)	9513551.0	81509.0	3492634.0
PERF. FRAC.,(NU)	0.6052	0.0164	0.7124
PROPELLANT FRAC.,(NUB)	0.9010	1.0000	0.8100
BURNOUT TIME,(SEC)	156.267	164.275	509.259
BURNOUT VELOCITY,(FT/SEC)	8070.586	8252.945	25954.113
BURNOUT GAMMA,(DEGREES)	14.168	12.875	0.187
BURNOUT ALTITUDE,(FT)	173832.4	189042.4	319656.9
BURNOUT RANGE,(NM)	46.6	55.2	819.0
IDEAL VELOCITY,(FT/SEC)	10750.5	10998.1	29712.0

INJECTION VELOCITY,(FT/SEC) 0.0 FLYBACK RANGE(NM) 198.5

INJECTION PROPELLANT,(LB) 0.0 FLYBACK PROP(LBS) 177764.2

ON ORBIT DELTA-V,(FT/SEC) 1083.5

ON ORBIT PROPELLANT,(LB) 95235.7

ON ORBIT ISP,(SEC) 466.7

THETA= 31.24 PITCH RATE= 0.00215 ATTEMPTS TO CONVERGE= 3

PAYLOAD,(LB) 495449.0

SUMMARY WEIGHT STATEMENT (NOMINAL MISSION) CASE 45

ORBITER WEIGHT BREAKDOWN		
DRY WEIGHT	740019.000	POUNDS
PERSONNEL	3000.000	POUNDS
RESIDUALS	2070.000	POUNDS
RESERVES	3300.000	POUNDS
IN-FLIGHT LOSSES	10695.000	POUNDS
ACPS PROPELLANT	18280.000	POUNDS
OMS PROPELLANT	95235.750	POUNDS
PAYLOAD	495449.000	POUNDS
BALLAST FOR CG CONTROL	0.0	POUNDS
OMS INSTALLATION KITS	0.0	POUNDS
PAYLOAD MODS	0.0	POUNDS
TOTAL END BOOST (ORBITER ONLY)	1368048.00	POUNDS
OMS BURNED DURING ASCENT	0.0	POUNDS
ACPS BURNED DURING ASCENT	0.0	POUNDS
EXTERNAL MAIN TANK		
TANK DRY WEIGHT	2640.000	POUNDS
RESIDUALS	18163.000	POUNDS
PROPELLANT BIAS	(2640.000)	POUNDS
PRESSURANT	(2120.000)	POUNDS
TANK AND LINES	(9753.000)	POUNDS
ENGINES	(3650.000)	POUNDS
FLIGHT PERFORMANCE RESERVE	20930.000	POUNDS
UNBURNED PROPELLANT (MAIN TANK)	0.0	POUNDS
TOTAL END BOOST (EXTERNAL TANK)	41733.000	POUNDS
USABLE PROPELLANT (EXTERNAL TANK)	5092633.00	POUNDS
FLYBACK PROPELLANT (FIRST STAGE)	177764.187	POUNDS
SOLID ROCKET MOTOR (FIRST STAGE)	9040548.00	POUNDS
SRM CASE WEIGHT(2)	1045488.87	POUNDS
SRM STRUCTURE & RCVY WEIGHT	0.0	POUNDS
SRM INERT STAGING WEIGHT	1045488.87	POUNDS
USABLE SRM PROPELLANT	7995060.00	POUNDS
TOTAL GROSS LIFT-OFF WEIGHT (GLOW)	15720730.0	POUNDS

VEHICLE CHARACTERISTICS (NOMINAL MISSION) CASE 66

STAGE	1	2	3
GROSS STAGE WEIGHT,(LB)	15727522.0	5040779.0	4816132.0
GROSS STAGE THRUST/WEIGHT	1.300	0.942	0.986
THRUST ACTUAL,(LB)	20445744.0	4750000.0	4750000.0
ISP VACUUM,(SEC)	370.890	466.700	466.700
STRUCTURE,(LB)	1045488.9	0.0	806009.0
PROPELLANT,(LB)	9456590.0	224647.0	3406483.0
PERF. FRAC.,(NU)	0.6013	0.0446	0.7675
PROPELLANT FRAC.,(NUB)	0.9004	1.0000	0.8087
BURNOUT TIME,(SEC)	154.692	176.764	513.331
BURNOUT VELOCITY,(FT/SEC)	7899.039	8394.184	25954.113
BURNOUT GAMMA,(DEGREES)	15.410	12.291	0.187
BURNOUT ALTITUDE,(FT)	175636.7	218325.6	319657.4
BURNOUT RANGE,(NM)	44.3	60.1	821.3
IDEAL VELOCITY,(FT/SEC)	10609.5	11293.9	29742.6

INJECTION VELOCITY,(FT/SEC) 0.0 FLYBACK RANGE(NM) 208.7
INJECTION PROPELLANT,(LB) 0.0 FLYBACK PROP(LBS) 184663.7

ON ORBIT DELTA-V,(FT/SEC) 1083.5
ON ORBIT PROPELLANT,(LB) 95257.7
ON ORBIT ISP,(SEC) 466.7

THETA= 29.93 PITCH RATE= 0.00197 ATTEMPTS TO CONVERGE= 3

PAYLOAD,(LB) 508362.0

SUMMARY WEIGHT STATEMENT (NOMINAL MISSION) CASE 66

ORBITER WEIGHT BREAKDOWN		
DRY WEIGHT	727620.000	POUNDS
PERSONNEL	5000.000	POUNDS
RESIDUALS	2070.000	POUNDS
RESERVES	3300.000	POUNDS
IN-FLIGHT LOSSES	16439.000	POUNDS
ACPS PROPELLANT	18280.000	POUNDS
OMS PROPELLANT	95257.750	POUNDS
PAYLOAD	506382.000	POUNDS
BALLAST FOR CG CONTROL	0.0	POUNDS
OMS INSTALLATION KITS	0.0	POUNDS
PAYLOAD MODS	0.0	POUNDS
TOTAL END BOOST (ORBITER ONLY)	1368348.00	POUNDS
OMS BURNED DURING ASCENT	0.0	POUNDS
ACPS BURNED DURING ASCENT	0.0	POUNDS
EXTERNAL MAIN TANK		
TANK DRY WEIGHT	2640.000	POUNDS
RESIDUALS	17730.000	POUNDS
PROPELLANT BIAS	(2640.000)	POUNDS
PRESSURANT	(2120.000)	POUNDS
TANK AND LINES	(9320.000)	POUNDS
ENGINES	(3650.000)	POUNDS
FLIGHT PERFORMANCE RESERVE	20930.000	POUNDS
UNBURNED PROPELLANT (MAIN TANK)	0.0	POUNDS
TOTAL END BOOST (EXTERNAL TANK)	41300.000	POUNDS
USABLE PROPELLANT (EXTERNAL TANK)	5092633.00	POUNDS
FLYBACK PROPELLANT (FIRST STAGE)	184663.687	POUNDS
SOLID ROCKET MOTOR (FIRST STAGE)	9040548.00	POUNDS
SRM CASE WEIGHT(2)	1045488.87	POUNDS
SRM STRUCTURE & RCVY WEIGHT	0.0	POUNDS
SRM INERT STAGING WEIGHT	1045488.67	POUNDS
USABLE SRM PROPELLANT	7995060.00	POUNDS
[illegible]	[illegible]	[illegible]

B.3 FIRST STAGE PROPELLANT LOADING STUDY

An analysis of the effects of varying first stage propellant loading was performed. The results are summarized in Table B.3-1 and specific vehicle characteristics are included in the attached data sheets. As expected, the payload capability increases as the first stage propellant mass is increased. The ratio of glow/payload weights is also improved. However, the staging velocity also increases significantly. In this trade study the first stage inert weight was not penalized for the additional TPS required at the higher staging velocities. By including that delta weight the glow/payload ratio would not be as favorable. By combining the results of this study with the throttling trade results, however, a payload increase may be achieved without the significant increase in staging velocity.

Table B.3-1. First Stage Propellant Trade Summary

CASE	1ST STAGE PROP. ($LB \times 10^6$)	GLOW ($LB \times 10^6$)	PAYLOAD ($LB \times 10^3$)	STAGING VELOCITY (FT/SEC)	GLOW/PAYLOAD
REFERENCE	7.995	15.731	509.7	6978	30.87
21	8.495	16.328	551.6	7281	29.60
22	8.995	16.921	589.0	7573	28.73
23	9.495	17.514	624.9	7852	28.03
24	9.995	18.108	659.3	8114	27.46

GENERAL ASCENT TRAJECTORY AND SIZING PROGRAM BY R.L.POWELL

DATE - 01/18/79 TIME - 16:50: 0

SATELLITE POWER SYSTEM (SPS) CONCEPT DEFINITION STUDY

TWO-STAGE VERTICAL TAKE-OFF HORIZONTAL LANDING HLLV CONCEPT

BOTH STAGES HAVE FLYBACK CAPABILITY TO LAUNCH SITE (KSC)

FIRST STAGE HAS AIRBREATHER FLYBACK AND LANDING CAPABILITY

FLYBACK PROPELLANT HAS A SPECIFIC FUEL CONSUMPTION OF 3500 SEC

SECOND STAGE USES THE ABORT-ONCE-AROUND FLYBACK MODE (AOA)

FIRST STAGE HAS LOX/RP/LH2 TRIPROPELLANT SYSTEM

WITH H2 COOLED HIGH PC ENGINES (VACUUM ISP = 352.3 SEC)

SECOND STAGE USES LOX/LH2 PROPELLANT WITH VACUUM ISP 466.7 SEC

THE DESIGN PAYLOAD SHALL BE 500 KLB INTO A CIRCULAR ORBIT OF

270 N. MILES AND AN INERTIAL INCLINATION OF 31.6 DEGREES

ASCENT SHAPED TO THE NOMINAL ASCENT MISSION

MECO CONDITIONS ARE TO A THEORETICAL ORBIT OF 169.22 N.MILES

BY 50.42 N. MILES (COASTS TO APOGEE OF 160 N.MILES)

ON-ORBIT DELTA VELOCITY REQUIREMENT OF 1110 FEET/SECOND

RCS SYSTEM SIZED FOR A DELTA VELOCITY REQMT OF 220 FEET/SECOND

THE VEHICLE SIZED FOR A THRUST/WEIGHT RATIO AT LIFT-OFF OF 1.30

ORIGINAL PAGE IS OF POOR QUALITY

MAXIMUM AXIAL LOAD FACTOR DURING ASCENT IS 3.0 G'S

TRAJECTORY HAS A MAXIMUM AERO PRESSURE OF 650 LBS/FT2

MAXIMUM AERO PRESSURE AT STAGING LIMITED TO 25 LBS/FT2

DIRECT ENTRY FROM 270 N.MILES ASSUMMED (DELTA V = 415 FT/SEC)

FLIGHT PERFORMANCE RESERVE = 0.75% TOTAL CHAC ASCENT VELOCITY

WEIGHT SCALING PER ROCKWELL IR AND D HLLV STUDIES

A WEIGHT GROWTH ALLOWANCE OF 15% IS ASSUMMED FOR BOTH STAGES

FIRST STAGE BURNS 8495060 POUNDS OF ASCENT PROPELLANT

SECOND STAGE (ORBITER) ENGINES BURN 5092633 LBS OF PROPELLANT

SECOND STAGE DRY WEIGHT WITHOUT PAYLOAD EQUALS 716827 LBS

SECOND STAGE THRUST LEVEL @ STAGING EQUALS 4750000 LBS

SECOND STAGE OVERALL BOOSTER MASS FRACTION = 0.8469 W/O MARGIN

SECOND STAGE WEIGHT BREAKDOWN :

RESIDUAL WEIGHT = 2070 POUNDS

RESERVES WEIGHT = 3300 POUNDS

FPR WEIGHT = 28306 POUNDS

RCS PROP WEIGHT = 17766 POUNDS

BURN-OUT ALTITUDE AT SECOND STAGE THRUST TERMINATION = 50 N. MILES

ADVANCED TECHNOLOGY WILL BE COMPATABLE WITH THE YEARS 1995 & ON

ASCENT HLLV SIZING RUNS MADE BY R.L.POWELL (EXT 3703 SEAL BEACH)

ORIGINAL PAGE IS
OF POOR QUALITY

VEHICLE CHARACTERISTICS (NOMINAL MISSION) CASE 21

STAGE	1	2	3
GROSS STAGE WEIGHT,(LB)	16327675.0	4851939.0	4538432.0
GROSS STAGE THRUST/WEIGHT	1.3[illegible]	0.971	1.047
THRUST ACTUAL,(LB)	21225930.0	4750000.0	4750000.0
ISP VACUUM,(SEC)	370.200	466.700	466.700
STRUCTURE,(LB)	1094325.0	0.0	795765.0
PROPELLANT,(LB)	10141553.0	353507.0	3093223.0
PERF. FRAC.,(NO)	0.6211	0.0723	0.6816
PROPELLANT FRAC.,(NOD)	0.9026	1.0000	0.7954
BURNOUT TIME,(SEC)	161.775	196.508	501.867
BURNOUT VELOCITY,(FT/SEC)	6548.090	9418.051	25954.074
BURNOUT GAMMA,(DEGREES)	15.611	9.444	0.187
BURNOUT ALTITUDE,(FT)	186268.0	247911.3	519655.7
BURNOUT RANGE,(NM)	52.5	94.8	818.0
IDEAL VELOCITY,(FT/SEC)	11271.6	12597.8	29580.6

INJECTION VELOCITY,(FT/SEC) 0.0 FLYBACK RANGE(NM) 221.5
INJECTION PROPELLANT,(LB) 0.0 FLYBACK PROP(LBS) 199858.0

ON ORBIT DELTA-V,(FT/SEC) 1084.0
ON ORBIT PROPELLANT,(LB) 97833.6
ON ORBIT ISP,(SEC) 466.7

THETA= 27.59 PITCH RATE= 0.00190 ATTEMPTS TO CONVERGE= 3

PAYLOAD,(LB) 551830.0

SUMMARY WEIGHT STATEMENT (NOMINAL MISSION) CASE 21

ORBITER WEIGHT BREAKDOWN			
DRY WEIGHT		718827.000	POUNDS
PERSONNEL		3000.000	POUNDS
RESIDUALS		2070.000	POUNDS
RESERVES		3300.000	POUNDS
IN-FLIGHT LOSSES		10312.000	POUNDS
ACPS PROPELLANT		17766.000	POUNDS
OMS PROPELLANT		97833.812	POUNDS
PAYLOAD		551610.000	POUNDS
BALLAST FOR CG CONTROL		0.0	POUNDS
OMS INSTALLATION KITS		0.0	POUNDS
PAYLOAD MODS		0.0	POUNDS
TOTAL END BOOST (ORBITER ONLY)		1404718.00	POUNDS
OMS BURNED DURING ASCENT		0.0	POUNDS
ACPS BURNED DURING ASCENT		0.0	POUNDS
EXTERNAL MAIN TANK			
TANK DRY WEIGHT		2640.000	POUNDS
RESIDUALS		17512.000	POUNDS
PROPELLANT BIAS	(2566.000)		POUNDS
PRESSURANT	(2059.000)		POUNDS
TANK AND LINES	(9341.000)		POUNDS
ENGINES	(3546.000)		POUNDS
FLIGHT PERFORMANCE RESERVE		20338.000	POUNDS
UNBURNED PROPELLANT (MAIN TANK)		0.0	POUNDS
TOTAL END BOOST (EXTERNAL TANK)		40490.000	POUNDS
USABLE PROPELLANT (EXTERNAL TANK)		5093225.00	POUNDS
FLYBACK PROPELLANT (FIRST STAGE)		199658.000	POUNDS
SOLID ROCKET MOTOR (FIRST STAGE)		9589385.00	POUNDS
SRM CASE WEIGHT(2)		1094325.00	POUNDS
SRM STRUCTURE & RCVY WEIGHT		0.0	POUNDS
SRM INERT STAGING WEIGHT		1094325.00	POUNDS
USABLE SRM PROPELLANT		8495060.00	POUNDS
TOTAL GROSS LIFT-OFF WEIGHT (GLOW)		16327675.0	POUNDS

PROPELLANT SUMMARY FOR THE ABORT MODES FOR CASE 21

ASCENT TRAJECTORY SHAPED TO THE NOMINAL MISSION MODE UP TO 196.506 SECONDS

UNBURNED MAIN PROPELLANT IN THE ABORT MODE = 0.0 POUNDS

EXCESS ON-ORBIT PROPELLANT IN THE ABORT MODE = 38491.750 POUNDS

UNBURNED MAIN PROPELLANT IN THE RTLS MODE = 111300.000 POUNDS

EXCESS ON-ORBIT PROPELLANT IN THE RTLS MODE = 0.0 POUNDS

MINUS SIGN INDICATES PROPELLANT SHORTAGE IN BURN MODE INDICATED

SHUTTLE SYSTEM NET PAYLOAD WITHOUT OMS KITS = 551610.000 POUNDS

MAIN PROPELLANT BURNED TO AOA/RTLS ABORT TIME= 2000000.00 POUNDS

SHUTTLE GROSS LIFT-OFF WEIGHT (GLOW) = 16327675.0 POUNDS

PROPELLANT CROSS FEED FROM FIRST - SECOND STAGE= 1646493.00 POUNDS

SECOND STAGE PROPELLANT CAPACITY - CROSS FEED = 3446732.00 POUNDS

VEHICLE CHARACTERISTICS (NOMINAL MISSION) CASE 22

STAGE	1	2	3
GROSS STAGE WEIGHT,(LB)	16921312.0	4892320.0	4571699.0
GROSS STAGE THRUST/WEIGHT	1.300	0.971	1.039
THRUST ACTUAL,(LB)	21997064.0	4750000.0	4750000.0
ISP VACUUM,(SEC)	369.582	466.700	466.700
STRUCTURE,(LB)	1139450.0	0.0	789092.0
PROPELLANT,(LB)	10674439.0	320621.0	3093419.0
PERF. FRAC.,(ND)	0.6308	0.0655	0.6766
PROPELLANT FRAC.,(ND)	0.9035	1.0000	0.7968
BURNOUT TIME,(SEC)	165.027	196.589	501.662
BURNOUT VELOCITY,(FT/SEC)	6845.020	9657.465	25954.070
BURNOUT GAMMA,(DEGREES)	12.875	5.249	0.187
BURNOUT ALTITUDE,(FT)	191097.8	246438.6	319655.7
BURNOUT RANGE,(NM)	56.5	96.3	826.0
IDEAL VELOCITY,(FT/SEC)	11568.7	12586.5	29539.3

INJECTION VELOCITY,(FT/SEC)	0.0	FLYBACK RANGE(NM)	233.5
INJECTION PROPELLANT,(LB)	0.0	FLYBACK PROP(LBS)	215102.1
ON ORBIT DELTA-V,(FT/SEC)	1004.5		
ON ORBIT PROPELLANT,(LB)	100211.6		
ON ORBIT ISP,(SEC)	466.7		

THETA= 27.18   PITCH RATE= 0.00189   ATTEMPTS TO CONVERGE= 3

PAYLOAD,(LB) 366976.0

SUMMARY WEIGHT STATEMENT (NOMINAL MISSION) CASE 22

ORBITER WEIGHT BREAKDOWN		
DRY WEIGHT	712630.000	POUNDS
PERSONNEL	3000.000	POUNDS
RESIDUALS	2070.000	POUNDS
RESERVES	3300.000	POUNDS
IN-FLIGHT LOSSES	10206.000	POUNDS
ACPS PROPELLANT	17584.000	POUNDS
OMS PROPELLANT	100211.562	POUNDS
PAYLOAD	568976.000	POUNDS
BALLAST FOR CG CONTROL	0.0	POUNDS
OMS INSTALLATION KITS	0.0	POUNDS
PAYLOAD MODS	0.0	POUNDS
TOTAL END BOOST (ORBITER ONLY)	1438177.00	POUNDS
OMS BURNED DURING ASCENT	0.0	POUNDS
ACPS BURNED DURING ASCENT	0.0	POUNDS
EXTERNAL MAIN TANK		
TANK DRY WEIGHT	2640.000	POUNDS
RESIDUALS	17332.000	POUNDS
PROPELLANT BIAS	(2539.000)	POUNDS
PRESSURANT	(2038.000)	POUNDS
TANK AND LINES	(9245.000)	POUNDS
ENGINES	(3510.000)	POUNDS
FLIGHT PERFORMANCE RESERVE	20130.000	POUNDS
UNBURNED PROPELLANT (MAIN TANK)	0.0	POUNDS
TOTAL END BOOST (EXTERNAL TANK)	40102.000	POUNDS
USABLE PROPELLANT (EXTERNAL TANK)	5093433.00	POUNDS
FLYBACK PROPELLANT (FIRST STAGE)	215102.002	POUNDS
SOLID ROCKET MOTOR (FIRST STAGE)	10134510.0	POUNDS
SRM CASE WEIGHT(2)	1139450.00	POUNDS
SRM STRUCTURE & RCVY WEIGHT	0.0	POUNDS
SRM INERT STAGING WEIGHT	1139450.00	POUNDS
USABLE SRM PROPELLANT	8995060.00	POUNDS
TOTAL GROSS LIFT-OFF WEIGHT (GLOW)	16921512.0	POUNDS

PROPELLANT SUMMARY FOR THE ABORT MODES FOR CASE 22

ASCENT TRAJECTORY SHAPED TO THE NOMINAL MISSION MODE UP TO 196.569 SECONDS

UNBURNED MAIN PROPELLANT IN THE ABORT MODE = 0.0 POUNDS

EXCESS ON-ORBIT PROPELLANT IN THE ABORT MODE = 40304.937 POUNDS

UNBURNED MAIN PROPELLANT IN THE RTLS MODE = 44002.000 POUNDS

EXCESS ON-ORBIT PROPELLANT IN THE RTLS MODE = 0.0 POUNDS

MINUS SIGN INDICATES PROPELLANT SHORTAGE IN BURN MODE INDICATED

SHUTTLE SYSTEM NET PAYLOAD WITHOUT OMS KITS = 588976.000 POUNDS

MAIN PROPELLANT TURNED TO AOA/RTLS ABORT TIME= 2000000.00 POUNDS

SHUTTLE GROSS LIFT-OFF WEIGHT (GLOW) = 16921312.0 POUNDS

PROPELLANT CROSS FEED FROM FIRST - SECOND STAGE= 1879379.00 POUNDS

SECOND STAGE PROPELLANT CAPACITY - CROSS FEED = 3414054.00 POUNDS

VEHICLE CHARACTERISTICS (NOMINAL MISSION)				CASE 23
STAGE	1	2	3	
GROSS STAGE WEIGHT,(LB)	17514446.0	4893323.0	4705601.0	
GROSS STAGE THRUST/WEIGHT	1.300	0.971	1.009	
THRUST ACTUAL,(LB)	22768752.0	4750000.0	4750000.0	
ISP VACUUM,(SEC)	368.998	466.700	466.700	
STRUCTURE,(LB)	1163803.0	0.0	782819.0	
PROPELLANT,(LB)	11205559.0	187722.6	3195405.0	
PERF. FRAC.,(NU)	0.6398	0.0384	0.6791	
PROPELLANT FRAC.,(NUB)	0.9644	1.0000	0.8032	
BURNOUT TIME,(SEC)	168.065	186.509	581.417	
BURNOUT VELOCITY,(FT/SEC)	9128.727	9583.203	25954.059	
BURNOUT GAMMA,(DEGREES)	12.161	10.070	0.187	
BURNOUT ALTITUDE,(FT)	195238.7	228393.1	319656.9	
BURNOUT RANGE,(NM)	60.4	84.1	633.7	
IDEAL VELOCITY,(FT/SEC)	11649.7	12437.1	29502.5	
INJECTION VELOCITY,(FT/SEC)	0.0	FLYBACK RANGE(NM)		247.3
INJECTION PROPELLANT,(LB)	0.0	FLYBACK PROP(LBS)		231762.4
ON ORBIT DELTA-V,(FT/SEC)	1665.0			
ON ORBIT PROPELLANT,(LB)	102505.9			
ON ORBIT ISP,(SEC)	466.7			

THETA= 26.57 PITCH RATE= 0.00190 ATTEMPTS TO CONVERGE= 3

PAYLOAD,(LB) [illegible]

SUMMARY WEIGHT STATEMENT (NOMINAL MISSION) CASE 25

ORBITER WEIGHT BREAKDOWN		
DRY WEIGHT	707192.000	POUNDS
PERSONNEL	3000.000	POUNDS
RESIDUALS	2070.000	POUNDS
RESERVES	3300.000	POUNDS
IN-FLIGHT LOSSES	10107.000	POUNDS
ACPS PROPELLANT	17413.000	POUNDS
OMS PROPELLANT	102505.875	POUNDS
PAYLOAD	624871.000	POUNDS
BALLAST FOR CG CONTROL	0.0	POUNDS
OMS INSTALLATION KITS	0.0	POUNDS
PAYLOAD MODS	0.0	POUNDS
TOTAL END BOOST (ORBITER ONLY)	1470458.00	POUNDS
OMS BURNED DURING ASCENT	0.0	POUNDS
ACPS BURNED DURING ASCENT	0.0	POUNDS
EXTERNAL MAIN TANK		
TANK DRY WEIGHT	2640.000	POUNDS
RESIDUALS	17163.000	POUNDS
PROPELLANT BIAS	(2514.000)	POUNDS
PRESSURANT	(2018.000)	POUNDS
TANK AND LINES	(9155.000)	POUNDS
ENGINES	(3476.000)	POUNDS
FLIGHT PERFORMANCE RESERVE	19934.000	POUNDS
UNBURNED PROPELLANT (MAIN TANK)	0.0	POUNDS
TOTAL END BOOST (EXTERNAL TANK)	39737.000	POUNDS
USABLE PROPELLANT (EXTERNAL TANK)	5093629.00	POUNDS
FLYBACK PROPELLANT (FIRST STAGE)	231762.375	POUNDS
SOLID ROCKET MOTOR (FIRST STAGE)	10678863.0	POUNDS
SRM CASE WEIGHT(2)	1183863.00	POUNDS
SRM STRUCTURE & RCVY WEIGHT	0.0	POUNDS
SRM INERT STAGING WEIGHT	1183863.00	POUNDS
USABLE SRM PROPELLANT	9495000.00	POUNDS
TOTAL GROSS LIFT-OFF WEIGHT (GLOW)	17514448.0	POUNDS

PROPELLANT SUMMARY FOR THE ABORT MODES FOR CASE 23

ASCENT TRAJECTORY SHAPED TO THE NOMINAL MISSION MODE UP TO 186.509 SECONDS

UNBURNED MAIN PROPELLANT IN THE ABORT MODE = 0.0 POUNDS

EXCESS ON-ORBIT PROPELLANT IN THE ABORT MODE = 38675.937 POUNDS

UNBURNED MAIN PROPELLANT IN THE RTLS MODE = 71138.000 POUNDS

EXCESS ON-ORBIT PROPELLANT IN THE RTLS MODE = 0.0 POUNDS

MINUS SIGN INDICATES PROPELLANT SHORTAGE IN BURN MODE INDICATED

SHUTTLE SYSTEM NET PAYLOAD WITHOUT OMS KITS = 624871.000 POUNDS

MAIN PROPELLANT BURNED TO AOA/RTLS ABORT TIME= 1898221.00 POUNDS

SHUTTLE GROSS LIFT-OFF WEIGHT (GLOW) = 17514448.0 POUNDS

PROPELLANT CROSS FEED FROM FIRST - SECOND STAGE= 1718499.00 POUNDS

SECOND STAGE PROPELLANT CAPACITY - CROSS FEED = 3385130.00 POUNDS

VEHICLE CHARACTERISTICS (NOMINAL MISSION) CASE 24

STAGE	1	2	3
GROSS STAGE WEIGHT,(LB)	18108288.0	4895076.0	4736525.0
GROSS STAGE THRUST/WEIGHT	1.300	0.970	1.003
THRUST ACTUAL,(LB)	23540736.0	4750000.0	4750000.0
ISP VACUUM,(SEC)	368.451	466.700	466.700
STRUCTURE,(LB)	1228251.0	0.0	776910.0
PROPELLANT,(LB)	11734730.0	158551.0	3195585.0
PERF. FRAC.,(NU)	0.6480	0.0324	0.6747
PROPELLANT FRAC.,(NUB)	0.9052	1.0000	0.8044
BURNOUT TIME,(SEC)	170.931	186.509	501.306
BURNOUT VELOCITY,(FT/SEC)	9394.344	5779.437	25954.066
BURNOUT GAMMA,(DEGREES)	11.551	9.853	0.187
BURNOUT ALTITUDE,(FT)	199211.2	226864.7	319655.9
BURNOUT RANGE,(NM)	64.2	84.8	641.1
IDEAL VELOCITY,(FT/SEC)	12113.8	12608.6	29469.2

INJECTION VELOCITY,(FT/SEC)	0.0	FLYBACK RANGE(NM)	261.9
INJECTION PROPELLANT,(LB)	0.0	FLYBACK PROP(LBS)	250230.1
ON ORBIT DELTA-V,(FT/SEC)	1085.5		
ON ORBIT PROPELLANT,(LB)	104714.9		
ON ORBIT ISP,(SEC)	466.7		

THETA= 26.71 PITCH RATE= 0.00141 ATTEMPTS TO CONVERGE= 3

PAYLOAD,(LB) 659315.0

SUMMARY WEIGHT STATEMENT (NOMINAL MISSION) CASE 24

ORBITER WEIGHT BREAKDOWN		
DRY WEIGHT	701880.000	POUNDS
PERSONNEL	3000.000	POUNDS
RESIDUALS	2070.000	POUNDS
RESERVES	3360.000	POUNDS
IN-FLIGHT LOSSES	10013.000	POUNDS
ACPS PROPELLANT	17252.000	POUNDS
OMS PROPELLANT	104714.937	POUNDS
PAYLOAD	659315.000	POUNDS
BALLAST FOR CG CONTROL	0.0	POUNDS
OMS INSTALLATION KITS	0.0	POUNDS
PAYLOAD MODS	0.0	POUNDS
TOTAL END BOOST (ORBITER ONLY)	1501544.00	POUNDS
OMS BURNED DURING ASCENT	0.0	POUNDS
ACPS BURNED DURING ASCENT	0.0	POUNDS
EXTERNAL MAIN TANK		
TANK DRY WEIGHT	2640.000	POUNDS
RESIDUALS	17065.000	POUNDS
PROPELLANT BIAS	(2492.000)	POUNDS
PRESSURANT	(2000.000)	POUNDS
TANK AND LINES	(9070.000)	POUNDS
ENGINES	(5443.000)	POUNDS
FLIGHT PERFORMANCE RESERVE	19750.000	POUNDS
UNBURNED PROPELLANT (MAIN TANK)	0.0	POUNDS
TOTAL END BOOST (EXTERNAL TANK)	39395.000	POUNDS
USABLE PROPELLANT (EXTERNAL TANK)	5093613.00	POUNDS
FLYBACK PROPELLANT (FIRST STAGE)	250230.125	POUNDS
SOLID ROCKET MOTOR (FIRST STAGE)	11223311.0	POUNDS
SRM CASE WEIGHT(2)	1228251.0	POUNDS
SRM STRUCTURE & RCVY WEIGHT	0.0	POUNDS
SRM INERT STAGING WEIGHT	1228251.00	POUNDS
USABLE SRM PROPELLANT	9995660.00	POUNDS
TOTAL GROSS LIFT-OFF WEIGHT (GLOW)	18106288.0	POUNDS

PROPELLANT SUMMARY FOR THE ABORT MODES FOR CASE 24

ASCENT TRAJECTORY SHAPED TO THE NOMINAL MISSION MODE UP TO 186.509 SECONDS

UNBURNED MAIN PROPELLANT IN THE ABORT MODE = 0.0 POUNDS

EXCESS ON-ORBIT PROPELLANT IN THE ABORT MODE = 40249.937 POUNDS

UNBURNED MAIN PROPELLANT IN THE RTLS MODE = 13047.000 POUNDS

EXCESS ON-ORBIT PROPELLANT IN THE RTLS MODE = 0.0 POUNDS

MINUS SIGN INDICATES PROPELLANT SHORTAGE IN BURN MODE INDICATED

SHUTTLE SYSTEM NET PAYLOAD WITHOUT OMS KITS = 659315.000 POUNDS

MAIN PROPELLANT BURNED TO AOA/RTLS ABORT TIME= 1898221.00 POUNDS

SHUTTLE GROSS LIFT-OFF WEIGHT (GLOW) = 18108288.0 POUNDS

PROPELLANT CROSS FEED FROM FIRST - SECOND STAGE= 1739670.00 POUNDS

SECOND STAGE PROPELLANT CAPACITY - CROSS FEED = 3354143.00 POUNDS

B.4 SECOND STAGE PROPELLANT WEIGHT ANALYSES

The second stage propellant weights were varied in a similar manner as the first stage (B.3). Vehicle characteristic data sheets for the various cases are included in this section and the results are summarized in Table B.4-1. The results of this analysis, as might be expected, are just the opposite of those presented in the previous section for the first stage weight variation. As second stage propellant weight is increased the payload weight increases but the staging velocity decreases and the glow/payload weight ratio becomes worse. Also, when the throttling function is shifted to the second stage, the penalties become worse rather than showing an improvement as in the case of first stage propellant weight increases.

Table B.4-1. Second Stage Propellant Weight Study Summary

CASE	SECOND STAGE PROP. WEIGHT ($LB \times 10^6$)	STAGING VELOCITY (FT/SEC)	PAYLOAD ($LB \times 10^3$)	GLOW ($LB \times 10^6$)	GLOW/PAYLOAD
REFERENCE	5.093	6978	509.7	15.731	30.87
30	5.570	6608	519.6	16.310	31.39
31	6.068	6238	521.1	16.918	32.46
32	6.565	5851	515.2	17.540	34.05

GENERAL ASCENT TRAJECTORY AND SIZING PROGRAM BY R.L.POWELL

DATE - 01/19/79   TIME - 17:57:20

SATELLITE POWER SYSTEM (SPS) CONCEPT DEFINITION STUDY

TWO-STAGE VERTICAL TAKE-OFF HORIZONTAL LANDING HLLV CONCEPT

BOTH STAGES HAVE FLYBACK CAPABILITY TO LAUNCH SITE (KSC)

FIRST STAGE HAS AIRBREATHER FLYBACK AND LANDING CAPABILITY

FLYBACK PROPELLANT HAS A SPECIFIC FUEL CONSUMPTION OF 3500 SEC

SECOND STAGE USES THE ABORT-ONCE-AROUND FLYBACK MODE (AOA)

FIRST STAGE HAS LOX/RP/LH2 TRIPROPELLANT SYSTEM

WITH H2 COOLED HIGH PC ENGINES (VACUUM ISP = 352.3 SEC)

SECOND STAGE USES LOX/LH2 PROPELLANT WITH VACUUM ISP 466.7 SEC

THE DESIGN PAYLOAD SHALL BE 500 KLB INTO A CIRCULAR ORBIT OF

270 N. MILES AND AN INERTIAL INCLINATION OF 31.6 DEGREES

ASCENT SHAPED TO THE NOMINAL ASCENT MISSION

MECO CONDITIONS ARE TO A THEORETICAL ORBIT OF 169.22 N.MILES

BY 50.42 N. MILES (COASTS TO APOGEE OF 160 N.MILES)

ON-ORBIT DELTA VELOCITY REQUIREMENT OF 1110 FEET/SECOND

RCS SYSTEM SIZED FOR A DELTA VELOCITY REQMT OF 220 FEET/SECOND

THE VEHICLE SIZED FOR A THRUST/WEIGHT RATIO AT LIFT-OFF OF 1.30

MAXIMUM AXIAL LOAD FACTOR DURING ASCENT IS 3.0 G'S

TRAJECTORY HAS A MAXIMUM AERO PRESSURE OF 650 LBS/FT2

MAXIMUM AERO PRESSURE AT STAGING LIMITED TO 25 LBS/FT2

DIRECT ENTRY FROM 270 N.MILES ASSUMMED (DELTA V = 415 FT/SEC)

PFIGHT PERFORMANCE RESERVE = 0.75% TOTAL CHAC ASCENT VELOCITY

WEIGHT SCALING PER ROCKWELL IR AND D HLLV STUDIES

A WEIGHT GROWTH ALLOWANCE OF 15% IS ASSUMMED FOR BOTH STAGES

SECOND STAGE (ORBITER) ENGINES BURN 5592633 LBS OF PROPELLANT

SECOND STAGE DRY WEIGHT WITHOUT PAYLOAD EQUALS 792904 LBS

SECOND STAGE THRUST LEVEL @ STAGING EQUALS 5212010 LBS

SECOND STAGE OVERALL BOOSTER MASS FRACTION = 0.8489 W/O MARGIN

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SECOND STAGE WEIGHT BREAKDOWN :

RESERVES WEIGHT = 3300 POUNDS

RESIDUAL WEIGHT = 2070 POUNDS

RCS PROP WEIGHT = 19806 POUNDS

FPR PROP WEIGHT = 22673 POUNDS

BURN-OUT ALTITUDE AT SECOND STAGE THRUST TERMINATION = 50 N. MILES

ADVANCED TECHNOLOGY WILL BE COMPATABLE WITH THE YEARS 1990 & ON

ASCENT HLLV SIZING RUNS MADE BY R.L.POWELL (EXT 3703 SEAL BEACH)

VEHICLE CHARACTERISTICS (NOMINAL MISSION) CASE 30

STAGE	1	2	3
GROSS STAGE WEIGHT,(LB)	16310355.0	5352967.0	5118293.0
GROSS STAGE THRUST/WEIGHT	1.300	0.974	1.018
THRUST ACTUAL,(LB)	21203424.0	5212010.0	5212010.0
ISP VACUUM,(SEC)	371.934	466.700	466.700
STRUCTURE,(LB)	1063207.0	0.0	877407.0
PROPELLANT,(LB)	9710386.0	234674.0	3619955.0
PERF. FRAC.,(NU)	0.5954	0.0438	0.7073
PROPELLANT FRAC.,(NUB)	0.9013	1.0000	0.8049
BURNOUT TIME,(SEC)	153.598	174.612	501.149
BURNOUT VELOCITY,(FT/SEC)	7862.922	8359.094	25954.102
BURNOUT GAMMA,(DEGREES)	15.246	12.193	0.187
BURNOUT ALTITUDE,(FT)	172889.7	212930.6	319656.2
BURNOUT RANGE,(NM)	43.9	66.5	798.0
IDEAL VELOCITY,(FT/SEC)	10527.5	11200.7	29646.8

INJECTION VELOCITY,(FT/SEC)	0.0	FLYBACK RANGE(NM)	204.3
INJECTION PROPELLANT,(LB)	0.0	FLYBACK PROP(LBS)	183794.9
ON ORBIT DELTA-V,(FT/SEC)	1665.0		
ON ORBIT PROPELLANT,(LB)	101324.1		
ON ORBIT ISP,(SEC)	466.7		

THETA= 29.18 PITCH RATE= 0.00200 ATTEMPTS TO CONVERGE= 3

PAYLOAD,(LB) 519606.0

SUMMARY WEIGHT STATEMENT (NOMINAL MISSION) CASE 30

ORBITER WEIGHT BREAKDOWN		
DRY WEIGHT	792904.000	POUNDS
PERSONNEL	3000.000	POUNDS
RESIDUALS	2070.000	POUNDS
RESERVES	3300.000	POUNDS
IN-FLIGHT LOSSES	11496.000	POUNDS
ACPS PROPELLANT	19806.000	POUNDS
OMS PROPELLANT	101324.125	POUNDS
PAYLOAD	519606.000	POUNDS
BALLAST FOR CG CONTROL	0.0	POUNDS
OMS INSTALLATION KITS	0.0	POUNDS
PAYLOAD MODS	0.0	POUNDS
TOTAL END BOOST (ORBITER ONLY)	1453506.00	POUNDS
OMS BURNED DURING ASCENT	0.0	POUNDS
ACPS BURNED DURING ASCENT	0.0	POUNDS
EXTERNAL MAIN TANK		
TANK DRY WEIGHT	2640.000	POUNDS
RESIDUALS	19518.000	POUNDS
PROPELLANT BIAS	(2860.000)	POUNDS
PRESSURANT	(2295.000)	POUNDS
TANK AND LINES	(10410.000)	POUNDS
ENGINES	(3953.000)	POUNDS
FLIGHT PERFORMANCE RESERVE	22673.000	POUNDS
UNBURNED PROPELLANT (MAIN TANK)	0.0	POUNDS
TOTAL END BOOST (EXTERNAL TANK)	44831.000	POUNDS
USABLE PROPELLANT (EXTERNAL TANK)	5569960.00	POUNDS
FLYBACK PROPELLANT (FIRST STAGE)	183794.875	POUNDS
SOLID ROCKET MOTOR (FIRST STAGE)	9058267.00	POUNDS
SRM CASE WEIGHT(2)	1063207.00	POUNDS
SRM STRUCTURE & RCVY WEIGHT	0.0	POUNDS
SRM INERT STAGING WEIGHT	1063207.00	POUNDS
USABLE SRM PROPELLANT	7995060.00	POUNDS
TOTAL GROSS LIFT-OFF WEIGHT (GLOW)	16310355.0	POUNDS

C-3

PROPELLANT SUMMARY FOR THE ABORT MODES FOR CASE 30

ASCENT TRAJECTORY SHAPED TO THE NOMINAL MISSION MODE UP TO 174.612 SECONDS

UNBURNED MAIN PROPELLANT IN THE ABORT MODE = 0.0 POUNDS

EXCESS ON-ORBIT PROPELLANT IN THE ABORT MODE = -7920.250 POUNDS

UNBURNED MAIN PROPELLANT IN THE RTLS MODE = 361288.250 POUNDS

EXCESS ON-ORBIT PROPELLANT IN THE RTLS MODE = 0.0 POUNDS

MINUS SIGN INDICATES PROPELLANT SHORTAGE IN BURN MODE INDICATED

SHUTTLE SYSTEM NET PAYLOAD WITHOUT OMS KITS = 519606.000 POUNDS

MAIN PROPELLANT BURNED TO AOA/RTLS ABORT TIME= 1950000.00 POUNDS

SHUTTLE GROSS LIFT-OFF WEIGHT (GLOW) = 16310355.0 POUNDS

PROPELLANT CROSS FEED FROM FIRST - SECOND STAGE= 1715326.00 POUNDS

SECOND STAGE PROPELLANT CAPACITY - CROSS FEED = 3854634.00 POUNDS

VEHICLE CHARACTERISTICS (NOMINAL MISSION) CASE 31

STAGE	1	2	3
GROSS STAGE WEIGHT,(LB)	16917712.0	5823346.0	5053036.0
GROSS STAGE THRUST/WEIGHT	1.300	0.981	1.130
THRUST ACTUAL,(LB)	21992992.0	5710110.0	5710110.0
ISP VACUUM,(SEC)	373.099	466.700	466.700
STRUCTURE,(LB)	1076520.0	0.0	957032.0
PROPELLANT,(LB)	9824750.0	770310.0	3467687.0
PERF. FRAC.,(NU)	0.5807	0.1323	0.6863
PROPELLANT FRAC.,(NUB)	0.9012	1.0000	0.7837
BURNOUT TIME,(SEC)	149.543	212.502	499.109
BURNOUT VELOCITY,(FT/SEC)	7480.551	9126.133	25954.066
BURNOUT GAMMA,(DEGREES)	16.710	8.200	0.187
BURNOUT ALTITUDE,(FT)	168079.7	275562.5	319656.9
BURNOUT RANGE,(NM)	39.5	108.8	782.8
IDEAL VELOCITY,(FT/SEC)	10130.3	12260.9	29666.8

INJECTION VELOCITY,(FT/SEC)	0.0	FLYBACK RANGE(NM)	215.2
INJECTION PROPELLANT,(LB)	0.0	FLYBACK PROP(LBS)	193095.7
ON ORBIT DELTA-V,(FT/SEC)	1086.3		
ON ORBIT PROPELLANT,(LB)	107222.5		
ON ORBIT ISP,(SEC)	466.7		

THETA= 28.63 PITCH RATE= 0.00193 ATTEMPTS TO CONVERGE= 3

PAYLOAD,(LB) [illegible]

SUMMARY WEIGHT STATEMENT (NOMINAL MISSION) CASE 31

ORBITER WEIGHT BREAKDOWN		
DRY WEIGHT	865186.000	POUNDS
PERSONNEL	3000.000	POUNDS
RESIDUALS	2070.000	POUNDS
RESERVES	3300.000	POUNDS
IN-FLIGHT LOSSES	12644.000	POUNDS
ACPS PROPELLANT	21784.000	POUNDS
OMS PROPELLANT	107222.500	POUNDS
PAYLOAD	521094.000	POUNDS
BALLAST FOR CG CONTROL	0.0	POUNDS
OMS INSTALLATION KITS	0.0	POUNDS
PAYLOAD MODS	0.0	POUNDS
TOTAL END BOOST (ORBITER ONLY)	1536300.00	POUNDS
OMS BURNED DURING ASCENT	0.0	POUNDS
ACPS BURNED DURING ASCENT	0.0	POUNDS
EXTERNAL MAIN TANK		
TANK DRY WEIGHT	2640.000	POUNDS
RESIDUALS	21471.000	POUNDS
PROPELLANT BIAS	(3146.000)	POUNDS
PRESSURANT	(2524.000)	POUNDS
TANK AND LINES	(11453.000)	POUNDS
ENGINES	(4348.000)	POUNDS
FLIGHT PERFORMANCE RESERVE	24937.000	POUNDS
UNBURNED PROPELLANT (MAIN TANK)	0.0	POUNDS
TOTAL END BOOST (EXTERNAL TANK)	49048.000	POUNDS
USABLE PROPELLANT (EXTERNAL TANK)	6067696.00	POUNDS
FLYBACK PROPELLANT (FIRST STAGE)	193095.750	POUNDS
SOLID ROCKET MOTOR (FIRST STAGE)	9071580.00	POUNDS
SRM CASE WEIGHT(2)	1076520.00	POUNDS
SRM STRUCTURE & RCVY WEIGHT	0.0	POUNDS
SRM INERT STAGING WEIGHT	1076520.00	POUNDS
USABLE SRM PROPELLANT	7995060.00	POUNDS
TOTAL GROSS LIFT-OFF WEIGHT (GLOW)	16917712.0	POUNDS

PROPELLANT SUMMARY FOR THE ABORT MODES FOR CASE 31

ASCENT TRAJECTORY SHAPED TO THE NOMINAL MISSION MODE UP TO 212.502 SECONDS

UNBURNED MAIN PROPELLANT IN THE ABORT MODE = 0.0 POUNDS

EXCESS ON-ORBIT PROPELLANT IN THE ABORT MODE = -22702.500 POUNDS

UNBURNED MAIN PROPELLANT IN THE RTLS MODE = 10886.750 POUNDS

EXCESS ON-ORBIT PROPELLANT IN THE RTLS MODE = 0.0 POUNDS

MINUS SIGN INDICATES PROPELLANT SHORTAGE IN BURN MODE INDICATED

SHUTTLE SYSTEM NET PAYLOAD WITHOUT OMS KITS = 521094.000 POUNDS

MAIN PROPELLANT BURNED TO AOA/RTLS ABORT TIME= 2600000.00 POUNDS

SHUTTLE GROSS LIFT-OFF WEIGHT (GLOW) = 16917/12.0 POUNDS

PROPELLANT CROSS FEED FROM FIRST - SECOND STAGE= 1829690.00 POUNDS

SECOND STAGE PROPELLANT CAPACITY - CROSS FEED = 4238006.00 POUNDS

VEHICLE CHARACTERISTICS (NOMINAL MISSION) CASE 32

STAGE	1	2	3
GROSS STAGE WEIGHT,(LB)	17540464.0	6299794.0	5431321.0
GROSS STAGE THRUST/WEIGHT	1.300	0.985	1.143
THRUST ACTUAL,(LB)	22802560.0	6208210.0	6208210.0
ISP VACUUM,(SEC)	374.122	466.700	466.700
STRUCTURE,(LB)	1090057.0	0.0	1038091.0
PROPELLANT,(LB)	9926587.0	868473.0	3765389.0
PERF. FRAC.,(NU)	0.5659	0.1379	0.6933
PROPELLANT FRAC.,(NUB)	0.9011	1.0000	0.7839
BURNOUT TIME,(SEC)	145.202	210.489	497.677
BURNOUT VELOCITY,(FT/SEC)	7073.633	8770.648	25954.086
BURNOUT GAMMA,(DEGREES)	18.946	8.947	0.187
BURNOUT ALTITUDE,(FT)	165000.7	283603.5	319655.5
BURNOUT RANGE,(NM)	34.8	102.2	767.9
IDEAL VELOCITY,(FT/SEC)	9731.0	11958.4	29703.8

INJECTION VELOCITY,(FT/SEC)	0.0	FLYBACK RANGE(NM)	255.7
INJECTION PROPELLANT,(LB)	0.0	FLYBACK PROP(LBS)	224025.2

ON ORBIT DELTA-V,(FT/SEC)	1087.5
ON ORBIT PROPELLANT,(LB)	112659.8
ON ORBIT ISP,(SEC)	466.7

THETA= 27.09 PITCH RATE= 0.00177 ATTEMPTS TO CONVERGE= 3

PAYLOAD,(LB) 515181.0

SUMMARY WEIGHT STATEMENT (NOMINAL MISSION) CASE 32

ORBITER WEIGHT BREAKDOWN		
DRY WEIGHT	938763.000	POUNDS
PERSONNEL	3000.000	POUNDS
RESIDUALS	2070.000	POUNDS
RESERVES	3300.000	POUNDS
IN-FLIGHT LOSSES	13814.000	POUNDS
ACPS PROPELLANT	23800.000	POUNDS
OMS PROPELLANT	112659.812	POUNDS
PAYLOAD	515181.000	POUNDS
BALLAST FOR CG CONTROL	0.0	POUNDS
OMS INSTALLATION KITS	0.0	POUNDS
PAYLOAD MODS	0.0	POUNDS
TOTAL END BOOST (ORBITER ONLY)	1612587.00	POUNDS
OMS BURNED DURING ASCENT	0.0	POUNDS
ACPS BURNED DURING ASCENT	0.0	POUNDS
EXTERNAL MAIN TANK		
TANK DRY WEIGHT	2640.000	POUNDS
RESIDUALS	23458.000	POUNDS
PROPELLANT BIAS	(3437.000)	POUNDS
PRESSURANT	(2758.000)	POUNDS
TANK AND LINES	(12513.000)	POUNDS
ENGINES	(4750.000)	POUNDS
FLIGHT PERFORMANCE RESERVE	27246.000	POUNDS
UNBURNED PROPELLANT (MAIN TANK)	0.0	POUNDS
TOTAL END BOOST (EXTERNAL TANK)	53344.000	POUNDS
USABLE PROPELLANT (EXTERNAL TANK)	6565387.00	POUNDS
FLYBACK PROPELLANT (FIRST STAGE)	224025.187	POUNDS
SOLID ROCKET MOTOR (FIRST STAGE)	9085117.00	POUNDS
SRM CASE WEIGHT(2)	1090057.00	POUNDS
SRM STRUCTURE & RCVY WEIGHT	0.0	POUNDS
SRM INERT STAGING WEIGHT	1090057.00	POUNDS
USABLE SRM PROPELLANT	7995060.00	POUNDS
TOTAL GROSS LIFT-OFF WEIGHT [illegible]	[illegible]	POUNDS

PROPELLANT SUMMARY FOR THE ABORT MODES FOR CASE 32

ASCENT TRAJECTORY SHAPED TO THE NOMINAL MISSION MODE UP TO 210.489 SECONDS

UNBURNED MAIN PROPELLANT IN THE ABORT MODE = 0.0 POUNDS

EXCESS ON-ORBIT PROPELLANT IN THE ABORT MODE = -72984.562 POUNDS

UNBURNED MAIN PROPELLANT IN THE RTLS MODE = 6693.000 POUNDS

EXCESS ON-ORBIT PROPELLANT IN THE RTLS MODE = 0.0 POUNDS

MINUS SIGN INDICATES PROPELLANT SHORTAGE IN BURN MODE INDICATED

SHUTTLE SYSTEM NET PAYLOAD WITHOUT OMS KITS = 515181.000 POUNDS

MAIN PROPELLANT BURNED TO AOA/RTLS ABORT TIME= 2800000.00 POUNDS

SHUTTLE GROSS LIFT-OFF WEIGHT (GLOW) = 17540464.0 POUNDS

PROPELLANT CROSS FEED FROM FIRST - SECOND STAGE= 1931527.00 POUNDS

SECOND STAGE PROPELLANT CAPACITY - CROSS FEED = 4633860.00 POUNDS

B.5 LIFTOFF THRUST-TO-WEIGHT

The liftoff thrust-to-weight (T/W) was reduced from the reference value of 1.30 to 1.25 in order to assess the effects. This variation in T/W resulted in approximately 1% reduction in payload capability without an appreciable change in staging velocity. The glow was also reduced slightly. The major effect was a shift of approximately 70,000 lb of second stage stored propellant over to the first stage crossfeed tanks. This shift in propellant weight should bring both vehicles within the same volumetric envelope. Selected vehicle parameters are compared with the reference HLLV configuration in Table B.5-1 and vehicle characteristics are given in the attached computer data sheets.

Table B.5-1. Comparison of Liftoff T/W of 1.25 with Reference HLLV

	THRUST/WEIGHT	
	1.3 (REF)	1.25
GLOW ($LB\times10^6$)	15.731	15.697
PAYLOAD ($LB\times10^3$)	509.7	503.9
GLOW/PAYLOAD	30.87	31.15
STAGING VELOCITY (FT/SEC)	6978	7000
FIRST STAGE PROPELLANT - LOADED ($LB\times10^6$)	9.607	9.679
SECOND STAGE PROPELLANT - LOADED ($LB\times10^6$)	3.481	3.410

The lower thrust-to-weight system would be of advantage only if the impact on engine size is of sufficient magnitude to warrant paying the small penalty in payload capability.

GENERAL ASCENT TRAJECTORY AND SIZING PROGRAM BY R.L.POWELL

DATE - 01/17/79 TIME - 21:31:36

SATELLITE POWER SYSTEM (SPS) CONCEPT DEFINITION STUDY

TWO-STAGE VERTICAL TAKE-OFF HORIZONTAL LANDING HLLV CONCEPT

BOTH STAGES HAVE FLYBACK CAPABILITY TO LAUNCH SITE (KSC)

FIRST STAGE HAS AIRBREATHER FLYBACK AND LANDING CAPABILITY

FLYBACK PROPELLANT HAS A SPECIFIC FUEL CONSUMPTION OF 3500 SEC

SECOND STAGE USES THE ABORT-ONCE-AROUND FLYBACK MODE (AOA)

FIRST STAGE HAS LOX/RP/LH2 TRIPROPELLANT SYSTEM

WITH H2 COOLED HIGH PC ENGINES (VACUUM ISP = 352.3 SEC)

SECOND STAGE USES LOX/LH2 PROPELLANT WITH VACUUM ISP 466.7 SEC

THE DESIGN PAYLOAD SHALL BE 500 KLB INTO A CIRCULAR ORBIT OF

270 N. MILES AND AN INERTIAL INCLINATION OF 31.6 DEGREES

ASCENT SHAPED TO THE NOMINAL ASCENT MISSION

MECO CONDITIONS ARE TO A THEORETICAL ORBIT OF 169.22 N.MILES

BY 50.42 N. MILES (COASTS TO APOGEE OF 160 N.MILES)

ON-ORBIT DELTA VELOCITY REQUIREMENT OF 1110 FEET/SECOND

RCS SYSTEM SIZED FOR A DELTA VELOCITY REQMT OF 220 FEET/SECOND

THE VEHICLE SIZED FOR A THRUST/WEIGHT RATIO AT LIFT-OFF OF 1.25

MAXIMUM AXIAL LOAD FACTOR DURING ASCENT IS 3.0 G'S

TRAJECTORY HAS A MAXIMUM AERO PRESSURE OF 650 LBS/FT2

MAXIMUM AERO PRESSURE AT STAGING LIMITED TO 25 LBS/FT2

DIRECT ENTRY FROM 270 N.MILES ASSUMMED (DELTA V = 415 FT/SEC)

PFIGHT PERFORMANCE RESERVE = 0.75% TOTAL CHAC ASCENT VELOCITY

WEIGHT SCALING PER ROCKWELL IR AND D HLLV STUDIES

A WEIGHT GROWTH ALLOWANCE OF 15% IS ASSUMMED FOR BOTH STAGES

FIRST STAGE BURNS 7995060 POUNDS OF ASCENT PROPELLANT

SECOND STAGE (ORBITER) ENGINES BURN 5092633 LBS OF PROPELLANT

SECOND STAGE DRY WEIGHT WITHOUT PAYLOAD EQUALS 713154 LBS

SECOND STAGE THRUST LEVEL @ STAGING EQUALS 4730000 LBS

SECOND STAGE ASSUMES 4 ENGINES FOR ASCENT WITH 1 OUT FOR ABORT

SECOND STAGE EPL THRUST LEVEL FOR ABORT IS 112 % FULL POWER

SECOND STAGE OVERALL BOOSTER MASS FRACTION = 0.8329

SECOND STAGE WEIGHT BREAKDOWN :

RESIDUAL WEIGHT = 2070 POUNDS

RESERVES WEIGHT = 3300 POUNDS

FPR WEIGHT = 20141 POUNDS

RCS WEIGHT = 17594 POUNDS

BURN-OUT ALTITUDE AT SECOND STAGE THRUST TERMINATION = 50 N. MILES

ADVANCED TECHNOLOGY WILL BE COMPATABLE WITH THE YEARS 1990 & ON

VEHICLE CHARACTERISTICS (NOMINAL MISSION) CASE 25

STAGE	1	2	3
GROSS STAGE WEIGHT,(LB)	15696635.0	4796839.0	4794255.0
GROSS STAGE THRUST/WEIGHT	1.250	0.990	0.991
THRUST ACTUAL,(LB)	19620768.0	4750000.0	4750000.0
ISP VACUUM,(SEC)	371.672	466.700	466.700
STRUCTURE,(LB)	1040199.7	0.0	789453.0
PROPELLANT,(LB)	9678653.0	2584.0	3407204.0
PERF. FRAC.,(NU)	0.6166	0.0005	0.7107
PROPELLANT FRAC.,(NUB)	0.9030	1.0000	0.8119
BURNOUT TIME,(SEC)	165.421	165.675	502.543
BURNOUT VELOCITY,(FT/SEC)	8267.918	8274.047	25954.113
BURNOUT GAMMA,(DEGREES)	13.522	13.477	0.187
BURNOUT ALTITUDE,(FT)	180447.9	180938.1	319657.8
BURNOUT RANGE,(NM)	49.5	49.8	798.1
IDEAL VELOCITY,(FT/SEC)	11149.8	11157.9	29780.8

INJECTION VELOCITY,(FT/SEC)	0.0	FLYBACK RANGE(NM)	204.1
INJECTION PROPELLANT,(LB)	0.0	FLYBACK PROP(LBS)	180942.2
ON ORBIT DELTA-V,(FT/SEC)	1082.7		
ON ORBIT PROPELLANT,(LB)	93697.7		
ON ORBIT ISP,(SEC)	466.7		

THETA= 29.10 PITCH RATE= 0.00205 ATTEMPTS TO CONVERGE= 3

PAYLOAD,(LB) 503900.0

SUMMARY WEIGHT STATEMENT (NOMINAL MISSION) CASE 25

ORBITER WEIGHT BREAKDOWN		
DRY WEIGHT	713154.000	POUNDS
PERSONNEL	3000.000	POUNDS
RESIDUALS	2070.000	POUNDS
RESERVES	3300.000	POUNDS
IN-FLIGHT LOSSES	10212.000	POUNDS
ACPS PROPELLANT	17594.000	POUNDS
OMS PROPELLANT	93697.687	POUNDS
PAYLOAD	503900.000	POUNDS
BALLAST FOR CG CONTROL	0.0	POUNDS
OMS INSTALLATION KITS	0.0	POUNDS
PAYLOAD MODS	0.0	POUNDS
TOTAL END BOOST (ORBITER ONLY)	1346927.00	POUNDS
OMS BURNED DURING ASCENT	0.0	POUNDS
ACPS BURNED DURING ASCENT	0.0	POUNDS
EXTERNAL MAIN TANK		
TANK DRY WEIGHT	2640.000	POUNDS
RESIDUALS	17342.000	POUNDS
PROPELLANT BIAS	(2540.000)	POUNDS
PRESSURANT	(2040.000)	POUNDS
TANK AND LINES	(9250.000)	POUNDS
ENGINES	(3512.000)	POUNDS
FLIGHT PERFORMANCE RESERVE	20141.000	POUNDS
UNBURNED PROPELLANT (MAIN TANK)	0.0	POUNDS
TOTAL END BOOST (EXTERNAL TANK)	40123.000	POUNDS
USABLE PROPELLANT (EXTERNAL TANK)	5093422.00	POUNDS
FLYBACK PROPELLANT (FIRST STAGE)	180942.250	POUNDS
SOLID ROCKET MOTOR (FIRST STAGE)	9035259.00	POUNDS
SRM CASE WEIGHT(2)	1040199.75	POUNDS
SRM STRUCTURE & RCVY WEIGHT	0.0	POUNDS
SRM INERT STAGING WEIGHT	1040199.75	POUNDS
USABLE SRM PROPELLANT	7995060.00	POUNDS
TOTAL GROSS LIFT-OFF WEIGHT (GLOW)	15696635.0	POUNDS

ORBITER ABORT DATA

VEHICLE CHARACTERISTICS

CASE 25

STAGE	1	2
GROSS STAGE WEIGHT,(LB)	4794255.0	379420 7.0
GROSS STAGE THRUST/WEIGHT	0.832	1.005
THRUST ACTUAL,(LB)	3990000.0	3815000.0
ISP VACUUM,(SEC)	466.700	466.700
STRUCTURE,(LB)	0.0	779453.0
PROPELLANT,(LB)	1000047.9	2451088.0
PERF. FRAC.,(NU)	0.2086	0.6460
PROPELLANT FRAC.,(NUB)	1.0000	0.7587
BURNOUT TIME,(SEC)	282.647	582.496
BURNOUT VELOCITY,(FT/SEC)	10859.383	25586.543
BURNOUT GAMMA,(DEGREES)	4.174	0.650
BURNOUT ALTITUDE,(FT)	335653.9	362187.6
BURNOUT RANGE,(NM)	202.6	951.8
IDEAL VELOCITY,(FT/SEC)	14670.7	30264.1
ON-ORBIT PROPELLANT USED,(LB)	43890.0	

OMS-ORBIT 93697.7 OMS-ASCENT 0.0

ON ORBIT PROPELLANT AVAIL,(LB) 93697.7

DELTA ON ORBIT PROPELLANT,(LB) 49807.7

ON-ORBIT MISSION PROP REQ'D,(LB) 25520.6

THETA= 39.55 PITCH RATE= 0.00236 ATTEMPTS TO CONVERGE= 0

SUMMARY WEIGHT STATEMENT (ABORT MODE) CASE 25

ORBITER WEIGHT BREAKDOWN		
DRY WEIGHT	713154.000	POUNDS
PERSONNEL	3000.000	POUNDS
RESIDUALS	2070.000	POUNDS
RESERVES	3300.000	POUNDS
IN-FLIGHT LOSSES	10212.000	POUNDS
ACPS PROPELLANT	7594.000	POUNDS
OMS PROPELLANT	49807.687	POUNDS
PAYLOAD	503900.000	POUNDS
BALLAST FOR CG CONTROL	0.0	POUNDS
OMS INSTALLATION KITS	0.0	POUNDS
PAYLOAD MODS	0.0	POUNDS
TOTAL END BOOST (ORBITER ONLY)	1293037.00	POUNDS
OMS BURNED DURING ASCENT	43890.000	POUNDS
ACPS BURNED DURING ASCENT	10000.000	POUNDS
EXTERNAL MAIN TANK		
TANK DRY WEIGHT	2640.000	POUNDS
RESIDUALS	17342.000	POUNDS
PROPELLANT BIAS	(2540.000)	POUNDS
PRESSURANT	(2040.000)	POUNDS
TANK AND LINES	(9250.000)	POUNDS
ENGINES	(3512.000)	POUNDS
FLIGHT PERFORMANCE RESERVE	20141.000	POUNDS
UNBURNED PROPELLANT (MAIN TANK)	0.0	POUNDS
TOTAL END BOOST (EXTERNAL TANK)	40123.000	POUNDS
USABLE PROPELLANT (EXTERNAL TANK)	5093422.00	POUNDS
FLYBACK PROPELLANT (FIRST STAGE)	180942.250	POUNDS
SOLID ROCKET MOTOR (FIRST STAGE)	9035259.00	POUNDS
SRM CASE WEIGHT(2)	1040199.75	POUNDS
SRM STRUCTURE & RCVY WEIGHT	0.0	POUNDS
SRM INERT STAGING WEIGHT	1040199.75	POUNDS
USABLE SRM PROPELLANT	7995060.00	POUNDS
TOTAL GROSS LIFT-OFF WEIGHT (GLOW)	15696635.0	POUNDS

VEHICLE CHARACTERISTICS (RTLS MODE) CASE 25

STAGE	1	2	3	4	5
GROSS STAGE WEIGHT,(LB)	4794255.0	4690199.0	4690199.0	3025143.0	2543142.0
GROSS STAGE THRUST/WEIGHT	0.796	0.813	0.856	1.319	1.500
THRUST ACTUAL,(LB)	3815000.0	3815000.0	4015000.0	3990000.0	3815000.0
ISP VACUUM,(SEC)	466.700	466.700	466.592	466.700	466.700
STRUCTURE,(LB)	0.0	0.0	0.0	0.0	770399.0
PROPELLANT,(LB)	104055.4	0.0	1665056.0	482000.2	757731.1
PERF. FRAC.,(NU)	0.0217	0.0	0.3550	0.1593	0.2980
PROPELLANT FRAC.,(NUB)	1.0000	0.0	1.0000	1.0000	0.4959
BURNOUT TIME,(SEC)	178.403	178.403	371.903	428.281	519.492
BURNOUT VELOCITY,(FT/SEC)	8184.465	8184.465	2421.007	702.479	3304.023
BURNOUT GAMMA,(DEGREES)	12.836	12.836	-12.228	-57.180	175.809
BURNOUT ALTITUDE,(FT)	204908.4	204895.1	291505.2	258602.7	229997.7
BURNOUT RANGE,(NM)	63.8	63.8	188.7	189.4	149.3
IDEAL VELOCITY,(FT/SEC)	11224.3	11224.3	17807.4	20413.5	25725.3

THETA=156.66 PITCH RATE= 0.00228 ATTEMPTS TO CONVERGE= 4

UNBURNED MAIN PROPELLANT,(LB) 511152.9

PAYLOAD,(LB) 503858.1

SUMMARY WEIGHT STATEMENT (RTLS MODE) CASE 25

ORBITER WEIGHT BREAKDOWN		
DRY WEIGHT	713154.000	POUNDS
PERSONNEL	3000.000	POUNDS
RESIDUALS	2070.000	POUNDS
RESERVES	3300.000	POUNDS
IN-FLIGHT LOSSES	10212.000	POUNDS
ACPS PROPELLANT	6844.000	POUNDS
OMS PROPELLANT	0.0	POUNDS
PAYLOAD	503858.125	POUNDS
BALLAST FOR CG CONTROL	0.0	POUNDS
OMS INSTALLATION KITS	0.0	POUNDS
PAYLOAD MODS	0.0	POUNDS
TOTAL END BOOST (ORBITER ONLY)	1242438.00	POUNDS
OMS BURNED DURING ASCENT	93697.687	POUNDS
ACPS BURNED DURING ASCENT	10750.000	POUNDS
EXTERNAL MAIN TANK		
TANK DRY WEIGHT	2640.000	POUNDS
RESIDUALS	17342.000	POUNDS
PROPELLANT BIAS	(2540.000)	POUNDS
PRESSURANT	(2040.000)	POUNDS
TANK AND LINES	(9250.000)	POUNDS
ENGINES	(3512.000)	POUNDS
FLIGHT PERFORMANCE RESERVE	11837.000	POUNDS
UNBURNED PROPELLANT (MAIN TANK)	511152.875	POUNDS
TOTAL END BOOST (EXTERNAL TANK)	542971.875	POUNDS
USABLE PROPELLANT (EXTERNAL TANK)	4590573.00	POUNDS
FLYBACK PROPELLANT (FIRST STAGE)	180942.250	POUNDS
SOLID ROCKET MOTOR (FIRST STAGE)	9035259.00	POUNDS
SRM CASE WEIGHT(2)	1040199.75	POUNDS
SRM STRUCTURE & RCVY WEIGHT	0.0	POUNDS
SRM INERT STAGING WEIGHT	1040199.75	POUNDS
USABLE SRM PROPELLANT	7995060.00	POUNDS
TOTAL GROSS LIFT-OFF WEIGHT (GLOW)	15696635.0	POUNDS

PROPELLANT SUMMARY FOR THE ABORT MODES FOR CASE 25

ASCENT TRAJECTORY SHAPED TO THE NOMINAL MISSION MODE UP TO 165.675 SECONDS

UNBURNED MAIN PROPELLANT IN THE ABORT MODE = 0.0 POUNDS

EXCESS ON-ORBIT PROPELLANT IN THE ABORT MODE = 24287.062 POUNDS

UNBURNED MAIN PROPELLANT IN THE RTLS MODE = 511152.875 POUNDS

EXCESS ON-ORBIT PROPELLANT IN THE RTLS MODE = 0.0 POUNDS

MINUS SIGN INDICATES PROPELLANT SHORTAGE IN BURN MODE INDICATED

SHUTTLE SYSTEM NET PAYLOAD WITHOUT OMS KITS = 503900.000 POUNDS

MAIN PROPELLANT BURNED TO AOA/RTLS ABORT TIME= 1686177.00 POUNDS

SHUTTLE GROSS LIFT-OFF WEIGHT (GLOW) = 15696635.0 POUNDS

PROPELLANT CROSS FEED FROM FIRST - SECOND STAGE= 1683593.00 POUNDS

SECOND STAGE PROPELLANT CAPACITY - CROSS FEED = 3409829.00 POUNDS

B.6 ALTERNATE FIRST STAGE PROPELLANTS

A performance comparison was made of the reference configuration using LOX/RP with alternate propellant systems of LOX/CH_4 (Methane) and LOX/LH_2. The comparative vehicle characteristics are tabulated in the attached computer data sheets and selected parameters are compared in Table B.6-1. Although the LOX/LH_2 configuration affords significant gains in payload capability, the considerably higher cost of LOX/LH_2 and the larger vehicle volume requirements result in a less cost effective configuration than the baseline. The increase in performance (~6%) afforded by the methane system is significant and contingent upon cost/availability in the quantities required for SPS, is the preferred propellant system.

Table B.6-1. Alternate Propellant Concepts

VEHICLE WEIGHT (KG×10^6)	FIRST STAGE PROPELLANT		
	LOX/RP	LOX/CH_4	LOX/LH_2
GLOW	7.135	7.151	7.532
BLOW	4.831	4.849	5.109
W_{P1}	4.359	4.372	4.385
ULOW	2.177	2.196	2.260
W_{P2}	1.579	1.564	1.552
PAYLOAD	0.231	0.245	0.318
GLOW/PAYLOAD	30.87	29.18	23.70

GENERAL ASCENT TRAJECTORY AND SIZING PROGRAM BY R.L.POWELL

DATE - 01/17/79 TIME - 21:58:24

SATELLITE POWER SYSTEM (SPS) CONCEPT DEFINITION STUDY

TWO-STAGE VERTICAL TAKE-OFF HORIZONTAL LANDING HLLV CONCEPT

BOTH STAGES HAVE FLYBACK CAPABILITY TO LAUNCH SITE (KSC)

FIRST STAGE HAS AIRBREATHER FLYBACK AND LANDING CAPABILITY

FLYBACK PROPELLANT HAS A SPECIFIC FUEL CONSUMPTION OF 3500 SEC

SECOND STAGE USES THE ABORT-ONCE-AROUND FLYBACK MODE (AOA)

FIRST STAGE HAS LOX/METHANE/LH2 TRIPROPELLANT SYSTEM

WITH H2 COOLED HIGH PC ENGINES (VACUUM ISP = 3361.3SEC)

SECOND STAGE USES LOX/LH2 PROPELLANT WITH VACUUM ISP 466.7 SEC

THE DESIGN PAYLOAD SHALL BE 500 KLB INTO A CIRCULAR ORBIT OF

270 N. MILES AND AN INERTIAL INCLINATION OF 31.6 DEGREES

ASCENT SHAPED TO THE NOMINAL ASCENT MISSION

MECO CONDITIONS ARE TO A THEORETICAL ORBIT OF 169.22 N.MILES

BY 50.42 N. MILES (COASTS TO APOGEE OF 160 N.MILES)

ON-ORBIT DELTA VELOCITY REQUIREMENT OF 1110 FEET/SECOND

RCS SYSTEM SIZED FOR A DELTA VELOCITY REQMT OF 220 FEET/SECOND

THE VEHICLE SIZED FOR A THRUST/WEIGHT RATIO AT LIFT-OFF OF 1.30

MAXIMUM AXIAL LOAD FACTOR DURING ASCENT IS 3.0 G'S

TRAJECTORY HAS A MAXIMUM AERO PRESSURE OF 650 LBS/FT2

MAXIMUM AERO PRESSURE AT STAGING LIMITED TO 25 LBS/FT2

DIRECT ENTRY FROM 270 N.MILES ASSUMMED (DELTA V = 415 FT/SEC)

PFIGHT PERFORMANCE RESERVE = 0.75% TOTAL CHAC ASCENT VELOCITY

WEIGHT SCALING PER ROCKWELL IR AND D HLLV STUDIES

A WEIGHT GROWTH ALLOWANCE OF 15% IS ASSUMMED FOR BOTH STAGES

FIRST STAGE BURNS 7995060 POUNDS OF ASCENT PROPELLANT

SECOND STAGE (ORBITER) ENGINES BURN 5092633 LBS OF PROPELLANT

SECOND STAGE DRY WEIGHT WITHOUT PAYLOAD EQUALS 719503.LBS

SECOND STAGE ASSUMES 4 ENGINES FOR ASCENT WITH 1 OUT FOR ABORT

B-110 SECOND STAGE EPL THRUST LEVEL FOR ABORT IS 112 % FULL POWER

SECOND STAGE OVERALL BOOSTER MASS FRACTION = 0.8489 W/O MARGIN

SECOND STAGE WEIGHT BREAKDOWN :

RESIDUAL WEIGHT = 2070 POUNDS

RESERVES WEIGHT = 3300 POUNDS

RCS PROP WEIGHT = 17787 POUNDS

FPR WEIGHT = 20362 POUNDS

BURN-OUT ALTITUDE AT SECOND STAGE THRUST TERMINATION = 50 N. MILES

ADVANCED TECHNOLOGY WILL BE COMPATABLE WITH THE YEARS 1990 & ON

ASCENT HLLV SIZING RUNS MADE BY R.L.POWELL (EXT 3703 SEAL BEACH)

VEHICLE CHARACTERISTICS (NOMINAL MISSION) CASE 26

STAGE	1	2	3
GROSS STAGE WEIGHT,(LB)	15765263.0	4882263.0	4776883.0
GROSS STAGE THRUST/WEIGHT	1.300	0.973	0.994
THRUST ACTUAL,(LB)	20494800.0	4750000.0	4750000.0
ISP VACUUM,(SEC)	378.691	466.700	466.700
STRUCTURE,(LB)	1051005.0	0.0	797077.0
PROPELLANT,(LB)	9639680.0	105380.0	3342640.0
PERF. FRAC.,(NU)	0.6115	0.0216	0.6998
PROPELLANT FRAC.,(NUB)	0.9017	1.0000	0.8075
BURNOUT TIME,(SEC)	161.591	171.945	501.922
BURNOUT VELOCITY,(FT/SEC)	8472.344	8715.793	25954.094
BURNOUT GAMMA,(DEGREES)	13.737	12.388	0.187
BURNOUT ALTITUDE,(FT)	185572.9	205651.7	319657.5
BURNOUT RANGE,(NM)	51.7	63.6	814.8
IDEAL VELOCITY,(FT/SEC)	11213.8	11541.4	29607.5

INJECTION VELOCITY,(FT/SEC)	0.0	FLYBACK RANGE(NM)	218.8
INJECTION PROPELLANT,(LB)	0.0	FLYBACK PROP(LBS)	192314.9
ON ORBIT DELTA-V,(FT/SEC)	1083.8		
ON ORBIT PROPELLANT,(LB)	97008.6		
ON ORBIT ISP,(SEC)	466.7		

THETA= 27.73 PITCH RATE= 0.00190 ATTEMPTS TO CONVERGE= 3

PAYLOAD,(LB) 540157.0

SUMMARY WEIGHT STATEMENT (NOMINAL MISSION) CASE 26

ORBITER WEIGHT BREAKDOWN		
DRY WEIGHT	719503.000	POUNDS
PERSONNEL	3000.000	POUNDS
RESIDUALS	2070.000	POUNDS
RESERVES	3300.000	POUNDS
IN-FLIGHT LOSSES	10324.000	POUNDS
ACPS PROPELLANT	17787.000	POUNDS
OMS PROPELLANT	97008.562	POUNDS
PAYLOAD	540157.000	POUNDS
BALLAST FOR CG CONTROL	0.0	POUNDS
OMS INSTALLATION KITS	0.0	POUNDS
PAYLOAD MODS	0.0	POUNDS
TOTAL END BOOST (ORBITER ONLY)	1393149.00	POUNDS
OMS BURNED DURING ASCENT	0.0	POUNDS
ACPS BURNED DURING ASCENT	0.0	POUNDS
EXTERNAL MAIN TANK		
TANK DRY WEIGHT	2640.000	POUNDS
RESIDUALS	17523.000	POUNDS
PROPELLANT BIAS	(2560.000)	POUNDS
PRESSURANT	(2061.000)	POUNDS
TANK AND LINES	(9352.000)	POUNDS
ENGINES	(3550.000)	POUNDS
FLIGHT PERFORMANCE RESERVE	20930.000	POUNDS
UNBURNED PROPELLANT (MAIN TANK)	0.0	POUNDS
TOTAL END BOOST (EXTERNAL TANK)	41093.000	POUNDS
USABLE PROPELLANT (EXTERNAL TANK)	5092633.00	POUNDS
FLYBACK PROPELLANT (FIRST STAGE)	192314.875	POUNDS
SOLID ROCKET MOTOR (FIRST STAGE)	9046065.00	POUNDS
SRM CASE WEIGHT(2)	1051005.00	POUNDS
SRM STRUCTURE & RCVY WEIGHT	0.0	POUNDS
SRM INERT STAGING WEIGHT	1051005.00	POUNDS
USABLE SRM PROPELLANT	7995060.00	POUNDS
TOTAL GROSS LIFT-OFF WEIGHT (GLOW)	15765263.0	POUNDS

PROPELLANT SUMMARY FOR THE ABORT MODES FOR CASE 26

ASCENT TRAJECTORY SHAPED TO THE NOMINAL MISSION MODE UP TO 171.945 SECONDS

UNBURNED MAIN PROPELLANT IN THE ABORT MODE = 0.0 POUNDS

EXCESS ON-ORBIT PROPELLANT IN THE ABORT MODE = 30091.312 POUNDS

UNBURNED MAIN PROPELLANT IN THE RTLS MODE = 349875.625 POUNDS

EXCESS ON-ORBIT PROPELLANT IN THE RTLS MODE = 0.0 POUNDS

MINUS SIGN INDICATES PROPELLANT SHORTAGE IN BURN MODE INDICATED

SHUTTLE SYSTEM NET PAYLOAD WITHOUT OMS KITS = 540157.000 POUNDS

MAIN PROPELLANT BURNED TO AOA/RTLS ABORT TIME= 1750000.00 POUNDS

SHUTTLE GROSS LIFT-OFF WEIGHT (GLOW) = 15765263.0 POUNDS

PROPELLANT CROSS FEED FROM FIRST - SECOND STAGE= 1644620.00 POUNDS

SECOND STAGE PROPELLANT CAPACITY - CROSS FEED = 3448013.00 POUNDS

GENERAL ASCENT TRAJECTORY AND SIZING PROGRAM BY R.L.POWELL

DATE - 01/19/79 TIME - 17:56:54

SATELLITE POWER SYSTEM (SPS) CONCEPT DEFINITION STUDY

TWO-STAGE VERTICAL TAKE-OFF HORIZONTAL LANDING HLLV CONCEPT

BOTH STAGES HAVE FLYBACK CAPABILITY TO LAUNCH SITE (KSC)

FIRST STAGE HAS AIRBREATHER FLYBACK AND LANDING CAPABILITY

FLYBACK PROPELLANT HAS A SPECIFIC FUEL CONSUMPTION OF 3500 SEC

SECOND STAGE USES THE ABORT-ONCE-AROUND FLYBACK MODE (AOA)

FIRST STAGE HAS LOX/RP/LH2 TRIPROPELLANT SYSTEM

WITH H2 COOLED HIGH PC ENGINES (VACUUM ISP = 352.3 SEC)

SECOND STAGE USES LOX/LH2 PROPELLANT WITH VACUUM ISP 466.7 SEC

THE DESIGN PAYLOAD SHALL BE 500 KLB INTO A CIRCULAR ORBIT OF

270 N. MILES AND AN INERTIAL INCLINATION OF 31.6 DEGREES

ASCENT SHAPED TO THE NOMINAL ASCENT MISSION

MECO CONDITIONS ARE TO A THEORETICAL ORBIT OF 169.22 N.MILES

BY 50.42 N. MILES (COASTS TO APOGEE OF 160 N.MILES)

ON-ORBIT DELTA VELOCITY REQUIREMENT OF 1110 FEET/SECOND

RCS SYSTEM SIZED FOR A DELTA VELOCITY REQMT OF 220 FEET/SECOND

THE VEHICLE SIZED FOR A THRUST/WEIGHT RATIO AT LIFT-OFF OF 1.30

MAXIMUM AXIAL LOAD FACTOR DURING ASCENT IS 3.0 G'S

TRAJECTORY HAS A MAXIMUM AERO PRESSURE OF 650 LBS/FT2

MAXIMUM AERO PRESSURE AT STAGING LIMITED TO 25 LBS/FT2

DIRECT ENTRY FROM 270 N.MILES ASSUMMED (DELTA V = 415 FT/SEC)

PFIGHT PERFORMANCE RESERVE = 0.75% TOTAL CHAC ASCENT VELOCITY

WEIGHT SCALING PER ROCKWELL IR AND D HLLV STUDIES

A WEIGHT GROWTH ALLOWANCE OF 15% IS ASSUMMED FOR BOTH STAGES

FIRST STAGE BURNS 7995060 POUNDS OF ASCENT PROPELLANT

SECOND STAGE (ORBITER) ENGINES BURN 5092633 LBS OF PROPELLANT

SECOND STAGE DRY WEIGHT WITHOUT PAYLOAD EQUALS 715166 LBS

SECOND STAGE THRUST LEVEL @ STAGING EQUALS 4750000 LBS

SECOND STAGE ASSUMES 4 ENGINES FOR ASCENT WITH 1 OUT FOR ABORT

SECOND STAGE EPL THRUST LEVEL FOR ABORT IS 112 % FULL POWER

SECOND STAGE OVERALL BOOSTER MASS FRACTION = 0.8489 W/O MARGIN

SECOND STAGE WEIGHT BREAKDOWN :

RESIDUAL WEIGHT = 2070 POUNDS

RESERVES WEIGHT = 3300 POUNDS

FPR WEIGHT = 20202.POUNDS

RCS PROP WEIGHT = 17648 POUNDS

BURN-OUT ALTITUDE AT SECOND STAGE THRUST TERMINATION = 50 N. MILES

ADVANCED TECHNOLOGY WILL BE COMPATABLE WITH THE YEARS 1990 & ON

VEHICLE CHARACTERISTICS (NOMINAL MISSION) CASE 35

STAGE	1	2	3
GROSS STAGE WEIGHT,(LB)	16604204.0	5021797.0	4894494.0
GROSS STAGE THRUST/WEIGHT	1.300	0.946	0.970
THRUST ACTUAL,(LB)	21585424.0	4750000.0	4750000.0
ISP VACUUM,(SEC)	466.500	466.700	466.700
STRUCTURE,(LB)	1596503.0	0.0	791663.0
PROPELLANT,(LB)	9667757.0	127303.0	3293366.0
PERF. FRAC.,(NU)	0.5822	0.0254	0.6729
PROPELLANT FRAC.,(NUB)	0.8583	1.0000	0.8062
BURNOUT TIME,(SEC)	164.350	176.858	501.196
BURNOUT VELOCITY,(FT/SEC)	9592.059	9888.875	25954.094
BURNOUT GAMMA,(DEGREES)	11.793	10.415	0.187
BURNOUT ALTITUDE,(FT)	195481.4	218899.2	319657.2
BURNOUT RANGE,(NM)	65.2	82.0	864.2
IDEAL VELOCITY,(FT/SEC)	12154.0	12539.5	29318.1

INJECTION VELOCITY,(FT/SEC)	0.0	FLYBACK RANGE(NM)	271.6
INJECTION PROPELLANT,(LB)	0.0	FLYBACK PROP(LBS)	318146.2
ON ORBIT DELTA-V,(FT/SEC)	1086.9		
ON ORBIT PROPELLANT,(LB)	108996.7		
ON ORBIT ISP,(SEC)	466.7		

THETA= 26.21 PITCH RATE= 0.00183 ATTEMPTS TO CONVERGE= 3

PAYLOAD,(LB) 700468.0

SUMMARY WEIGHT STATEMENT (NOMINAL MISSION) CASE 35

ORBITER WEIGHT BREAKDOWN		
DRY WEIGHT	715166.000	POUNDS
PERSONNEL	3000.000	POUNDS
RESIDUALS	2070.000	POUNDS
RESERVES	3300.000	POUNDS
IN-FLIGHT LOSSES	10243.000	POUNDS
ACPS PROPELLANT	17648.000	POUNDS
OMS PROPELLANT	108996.687	POUNDS
PAYLOAD	700468.000	POUNDS
BALLAST FOR CG CONTROL	0.0	POUNDS
OMS INSTALLATION KITS	0.0	POUNDS
PAYLOAD MODS	0.0	POUNDS
TOTAL END BOOST (ORBITER ONLY)	1560891.00	POUNDS
OMS BURNED DURING ASCENT	0.0	POUNDS
ACPS BURNED DURING ASCENT	0.0	POUNDS
EXTERNAL MAIN TANK		
TANK DRY WEIGHT	2640.000	POUNDS
RESIDUALS	17394.000	POUNDS
PROPELLANT BIAS	(2548.000)	POUNDS
PRESSURANT	(2045.000)	POUNDS
TANK AND LINES	(9279.000)	POUNDS
ENGINES	(3522.000)	POUNDS
FLIGHT PERFORMANCE RESERVE	20202.000	POUNDS
UNBURNED PROPELLANT (MAIN TANK)	0.0	POUNDS
TOTAL END BOOST (EXTERNAL TANK)	40236.000	POUNDS
USABLE PROPELLANT (EXTERNAL TANK)	5093361.00	POUNDS
FLYBACK PROPELLANT (FIRST STAGE)	318146.187	POUNDS
SOLID ROCKET MOTOR (FIRST STAGE)	9591563.00	POUNDS
SRM CASE WEIGHT(2)	1596503.00	POUNDS
SRM STRUCTURE & RCVY WEIGHT	0.0	POUNDS
SRM INERT STAGING WEIGHT	1596503.00	POUNDS
USABLE SRM PROPELLANT	7995060.00	POUNDS
TOTAL GROSS LIFT-OFF WEIGHT (GLOW)	16604204.0	POUNDS

PROPELLANT SUMMARY FOR THE ABORT MODES FOR CASE 35

ASCENT TRAJECTORY SHAPED TO THE NOMINAL MISSION MODE UP TO 176.858 SECONDS

UNBURNED MAIN PROPELLANT IN THE ABORT MODE = 0.0 POUNDS

EXCESS ON-ORBIT PROPELLANT IN THE ABORT MODE = 40335.250 POUNDS

UNBURNED MAIN PROPELLANT IN THE RTLS MODE = -31336.000 POUNDS

EXCESS ON-ORBIT PROPELLANT IN THE RTLS MODE = 0.0 POUNDS

MINUS SIGN INDICATES PROPELLANT SHORTAGE IN BURN MODE INDICATED

SHUTTLE SYSTEM NET PAYLOAD WITHOUT OMS KITS = 700468.000 POUNDS

MAIN PROPELLANT BURNED TO AOA/RTLS ABORT TIME= 1800000.00 POUNDS

SHUTTLE GROSS LIFT-OFF WEIGHT (GLOW) = 16604204.0 POUNDS

PROPELLANT CROSS FEED FROM FIRST - SECOND STAGE= 1672697.00 POUNDS

SECOND STAGE PROPELLANT CAPACITY - CROSS FEED = 3420664.00 POUNDS

APPENDIX C. ELECTRIC ORBITAL TRANSFER VEHICLE SIZING

APPENDIX C

ELECTRIC ORBITAL TRANSFER VEHICLE SIZING

APPENDIX C.

ELECTRICAL ORBITAL TRANSFER VEHICLE SIZING

C.0 INTRODUCTION

The data contained herein relates to preliminary sizing of large electric orbital transfer vehicles (EOTV) capable of delivering payloads from LEO to GEO of the order of 5×10^6 kg and return payloads (payload packaging) of 10% of the LEO to GEO payload. Total trip times are of the order of 2700 hours.

The benefits to be derived from employing large electron bombardment ion thruster systems using argon propellant have been discussed in References 1, 2, and 3. Maximum useful thruster size (diameter) for single grid systems have been estimated in Reference 3 where it was shown that thruster system cost is relatively insensitive to thruster size. A grid set span to gap ratio of 600 is considered a practical limit. In this study, the span to gap ratio problem is alleviated by assuming multiple, concentric grid sets up to three as required. Five grid sets have been tested in the laboratory at NASA Lewis Research Center (LRC). Sovey (Reference 3), with the help of Child's law, has determined an empirical expression for the ability of a grid set to extract the maximum ion current (per hole) for minimum total accelerating voltage (Perveance limit). Beyers and Rawlin (Reference 1) have projected the performance of 100 cm diameter thrusters based on identified constraints such as perveance and temperature. They indicate that thrusters might operate at temperatures as high as 1900 K. However, they used a conservative temperature of 973 K (where the grids begin to glow) in their own work. Since molybdenum grids have survived temperatures of 1900 K for several hundred thousand hours without significant creep (References 4 and 5), 1900 K was taken as the upper temperature limit in this study.

The EOTV sizing philosophy used in this study is in harmony with the philosophy found implicitly in References 1 and 3. That is, since thruster system cost is relatively insensitive to component size, a considerable cost savings can be achieved by operating at high thrust levels with a small number of large diameter thrusters. This is in lieu of a large number of small thrusters which impose a severe burden on orbital labor with respect to both construction and refurbishment. The lengths of electrical conductors and propellant lines can be many kilometers for small diameter thrusters. Further, the reduction in the number of components associated with large diameter thrusters implies an increase in system reliability.

The grid sets are more subject to failure than other thruster components because of bombardment by singly and doubly charged ions. It is therefore assumed that the grid sets will be refurbished after each round trip. When large payloads are returned it may be necessary to refurbish or replace grid sets more often, i.e., after each payload transfer. The grid set lifetime as a function of beam current (operating temperature) is not known for the operational time period under consideration. There is currently at least a decade to improve thruster state-of-the-art. The data presented will therefore reflect what is believed to be the technology of the next decade.

The choice of argon as the working fluid is based upon its great abundance and environmental suitability. Argon is currently obtained as a by-product in air reduction processes. The one billion kilograms of argon produced annually are largely discarded thus affording a readily available and low cost propellant.

C.1 STUDY GUIDELINES

The following ground rules and assumptions were employed for the EOTV study:

- The LEO parking orbit is at 500 km altitude and 31.6 degree inclination.
- Transfer time from LEO to GEO will be 120 days of which 20 days is in the Earth's shadow.
- The vehicles will either return empty or with ten percent of the up payload.
- Ten percent of the payload mass is packaging.
- The propellant utilization efficiency is 0.82.
- The steady state loss in thrust because of ion beam divergence is five percent. $\lambda_D = 0.95$.
- The thrust vector steering loss is five percent. $\gamma_S = 0.95$.
- Gallium aluminum arsenide solar cells are used with an assumed self annealing capability at 125°C. It is assumed that all electron damage due to radiation is annealed out and only proton damage results in degradation to the cell. Those losses are assumed as follows:

  4% non-annealable loss due to proton damage over 10 year life
  6% plasma loss when operating in LEO
  5% loss due to pointing errors
  6% loss in line due to voltage drop

  21% total loss in system efficiency

- Electric power is provided by two SPS panels with a blanket area of 900,000 m^2. Solar reflectors are employed with a concentration ratio of 2.
- A plane change with optimum steering to the equatorial plane is assumed with a velocity increment of 5688 m/s.

- A propellant reserve of 0.75 percent is assumed effectively increasing ΔV to 5730 m/s.
- Attitude hold only is employed during periods of Earth shadowing. Ion thrusters powered by storage batteries provide the required thrust.
- Advanced storage batteries are used that yield 200 watt-hours/kg of electrical energy.

C.2 ESTIMATING RELATIONSHIPS

The necessary formulas for estimating electric thruster system parameters and payload masses are presented herein. An attempt is made to ensure that the estimating relationships are self-consistent, realistic for the second decade, and that power and energy are conserved. Each formula is discussed, referenced when required, and derived when presented for the first time, or when additional clarity is justified.

An objective of this study is to take advantage of economies of scale. This coupled with the desire to have larger thrusters and fewer components leads to high grid set temperatures. Grid temperature was therefore a driving independent variable in this study, and ranged from 1900 K down to 1000 K. For each temperature selected, three maximized dependent variables are automatically defined, i.e., total extraction voltage (V_T), maximum thruster diameter (d), and maximum beam current (J_B).

C.2.1 Total Extraction Voltage - V_T (Volts)

Referring to Figure C-1, V_T is the potential difference between the anode and the accelerator grid. The total extraction voltage is limited by the allowable grid-set temperature, and for the maximum thruster parameters considered here, it is uniquely related to operating temperature. That is,

$$V_T = 0.012307T^{1.7778} \qquad (1)$$

independent of thruster diameter. Equation (1) is derived from work by Sovey (Reference 3) who found that the average measured temperature of the grid-set corresponded to a model grid with an emissivity of 0.4, that absorbed 25 percent of the discharge power. The discharge chamber loss ε_I was taken to be 200 for argon.

C.2.2 Net Accelerating Voltage - V_N (Volts)

Once again referring to Figure C-1, V_N, is the positive part of V_T, responsible for imparting the initial momentum to the ionized argon.

For convenience the ratio R is used to relate V_N and V_T, i.e.,

$$R = V_N/V_T \qquad (2)$$

Thrusters have been operated with values of R ranging from 0.2 to 0.9.

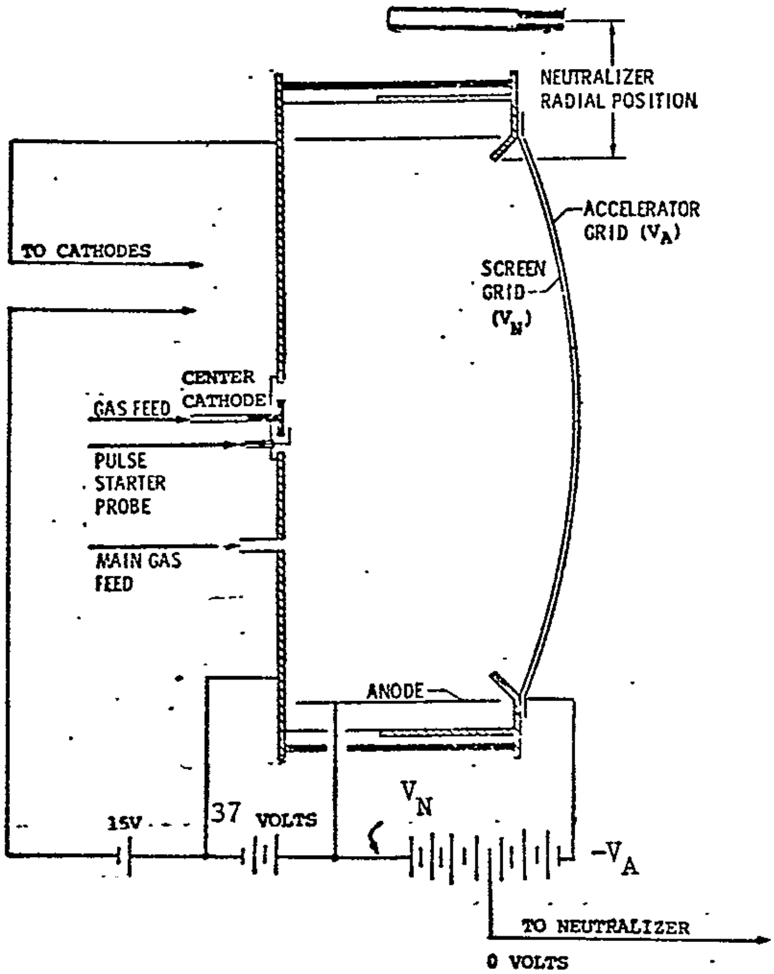

Figure C-1. Argon Ion Thruster Module (not to scale), Modified from Reference 1

C.2.3 Propellant Utilization Efficiency - η_u

The electric ion bombardment thruster operates by accelerating argon, or other suitable ions, to high speeds by subjecting them to a suitable potential difference. In the thrusters considered here, argon gas is first introduced into the thrust chamber and ionized by a voltage of about 40 volts which is high enough to ionize argon atoms with a single impact. The first ionization potential is 15.755 electron volts. Argon atoms that are initially excited but not ionized, may occasionally become doubly ionized (requiring 43.38 ev). Doubly ionized argon atoms are apt to bombard the grid structure, causing damage (sputtering) and penalyzing thrust and specific impulse.

In addition, some of the propellant remains un-ionized and is exhausted at low speed as a diffusing hot gas. It is necessary therefore to introduce a penalty, η_u, on both thrust and specific impulse that can be determined by measurement. The parameter η_u is called the propellant utilization efficiency.

By making two reasonable assumptions, one can acquire a feeling for propellant utilization. First, assume that all singly charged argon ions are accelerated to identical speeds, v, by the net potential difference V_N. Second, assume that the fraction of doubly charged ions is small compared to the fraction of singly charged ions. Then from conservation of momentum

$$\sum_{i=1}^{k} v_i m_i = v \sum_{i=1}^{k} m_i = \bar{v}\, m_p$$

where $v = v_1 = v_2 = - - - = v_k$ = ion speed,

$\bar{v}$ = mean speed of all exhaust materials,

and m_p = mass of exhausted material (ions and neutrals).

The propellant utilization efficiency is then defined by

$$0.8 \lesssim \eta_u = \frac{\bar{v}}{v} = \frac{\sum_{i=1}^{k} m_i}{m_p} \lesssim 0.9 \qquad (3)$$

where the limits on η_u apply to ionized argon.

C.2.4 Specific Impulse - I_{sp} (seconds)

Actual specific impulse can be defined by

$$I_{sp} = \frac{\bar{v}}{g} \qquad (4)$$

where $g = 9.807\ m/s^2$ the mean acceleration of gravity. This can also be expressed in terms of electric parameters. If ions are accelerated through a potential difference V_N one can write (summing i from 1 to k)

$$\tfrac{1}{2} \Sigma\, m_i v_i{}^2 = \tfrac{1}{2} v^2\, k\, m = \Sigma\, q_i\, V_N \qquad (5)$$

where q_i is the charge on each ion of mass m. Solving Eq. (5) for v^2 yields

$$v = \frac{2V_N \Sigma q_i}{km} = \frac{2V_N\ (kq)}{km} = 2V_N\ (q/m)$$

$$= \bar{v}^2/\eta_u{}^2 = g^2 I_{sp}{}^2/\eta_u{}^2$$

and

$$I_{sp} = (\eta_u/g)\sqrt{2V_N(q/m)}\,. \qquad (6)$$

The ratio of charge to mass for argon is

$$q/m = 2.4162\times10^{6} \text{ C/kg}, \tag{7}$$

and

$$\eta_u = 0.82.$$

After substituting the numerical values from Eq. (7) into Eq. (6) one obtains

$$I_{sp} = 223.96\ \eta_u\ V_N^{0.5} \tag{8}$$

$$= 183.65\ V_N^{0.5} \text{ seconds},$$

and conversely

$$V_N = 1.994\ I_{sp}^{2}/(\eta_u^{2}\times10^{5}) \tag{9}$$

$$= 2.9655\ I_{sp}^{2}\ 10^{-5} \text{ volts}.$$

Specific impulse as a function of voltage ratio and grid temperature is depicted in Figure C-2.

Ideal or "electrical" specific impulse is obtained by setting η_u equal to unity. The specific impulse used herein is as defined in Eq. (6). It is based on conservation of energy and momentum and yields either a maximum ion speed (η_u=1) or a mean propellant exhaust speed. The fact that the beam may be diverging and producing a useless component of thrust will be considered later by introducing a thrust efficiency term, γ_t. Thrust is a measurable quantity and, in particular, the useful thrust along the thruster axis can be determined.

Estimated thrust vector steering losses (γ_s) will also be introduced at the same time. With this approach there is no pseudo modification of maximum or mean propellant exhaust speeds or of specific impulse. The modification comes in the total propellant mass for rate ($\dot{m}_p$); part of it diverges and does no useful work. This is taken into account empirically and avoids giving the impression of an improvement in specific impulse.

Factors which enter into beam divergence include: (1) electric field intensity divergence; (2) mutual repulsions of singly and doubly charged ions; (3) the applied magnetic field; and (4) the discharge power that creates the ions. The discharge may be ten percent or more of the total power provided.

C.2.5 Maximum Thruster Diameter - D_b (cm)

An expression for the maximum useful beam diameter, D_b, which is tantamount to the maximum useful thruster diameter, d, was presented in Reference 2:

$$d = 1.5\times10^{-8}\ I_{sp}^{2}\ m/\eta_u^{2}R \tag{10}$$

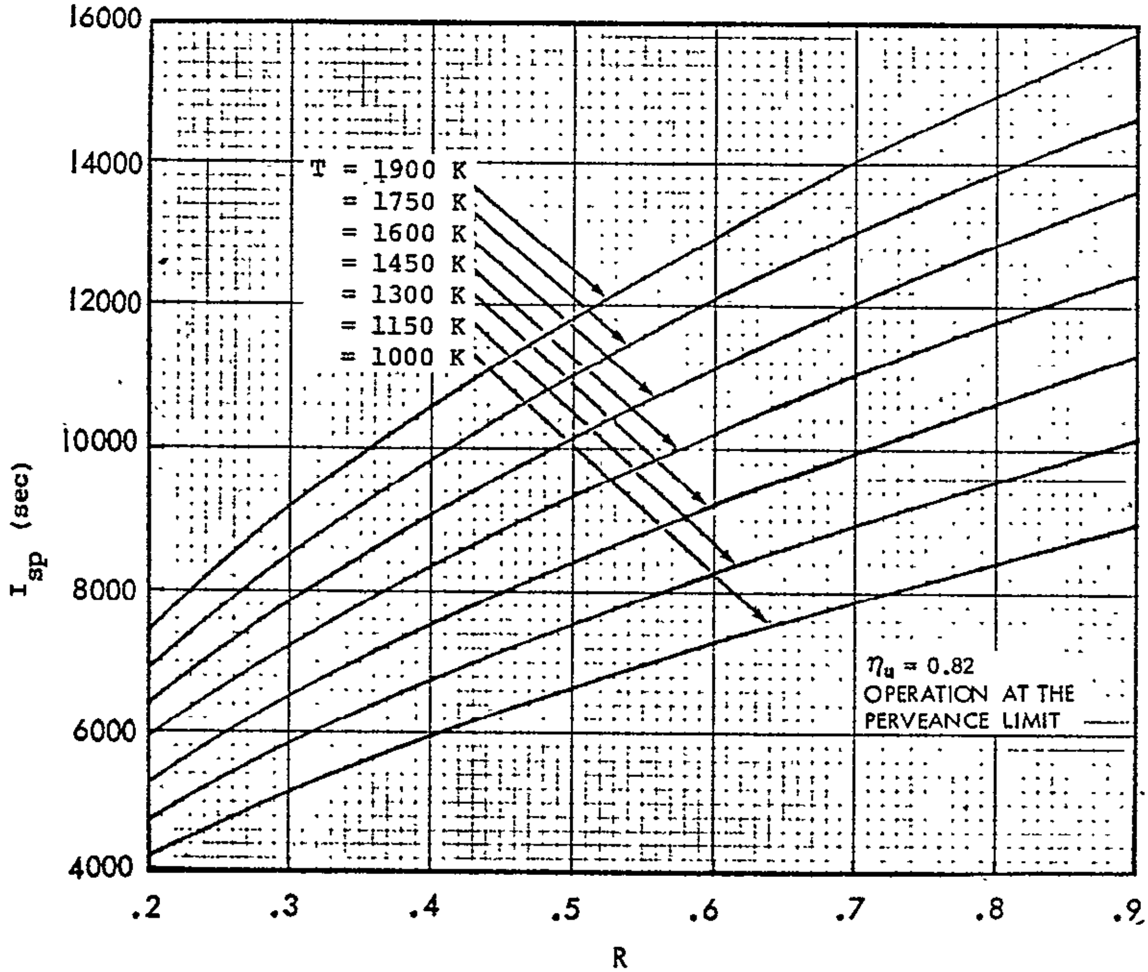

Figure C-2. Specific impulse as a function of voltage ratio, R, for operation at temperatures indicated.

where m = 39.948, the molecular weight of argon. Taking this value for m, with the help of Eqs. (8) and (1), and using 0.82 for η_u yields

$$d = 8.9117\times10^{-7}\ I_{sp}{}^2/R, \qquad (11)$$

$$= 3.0051\times10^{-2}\ V_T\ (\text{cm})$$

The straight dashed line in Figure C-3 is a plot of V_T versus maximum thruster diameter based on Reference 2. The maximum operating temperature corresponding to V_T is shown as a solid line which is almost linear over the range of V_T (5100 to 8300 volts).

C.2.6 Maximum Beam Current – J_B (Amperes)

The accelerator system, consisting of a screen grid and an accelerator grid (Figure C-1), imposes a basic limitation on the obtainable beam current density because of the "perveance" limit. The perveance limit in effect determines the point where any increase in the total accelerating voltage, V_T, results in high voltage breakdown.

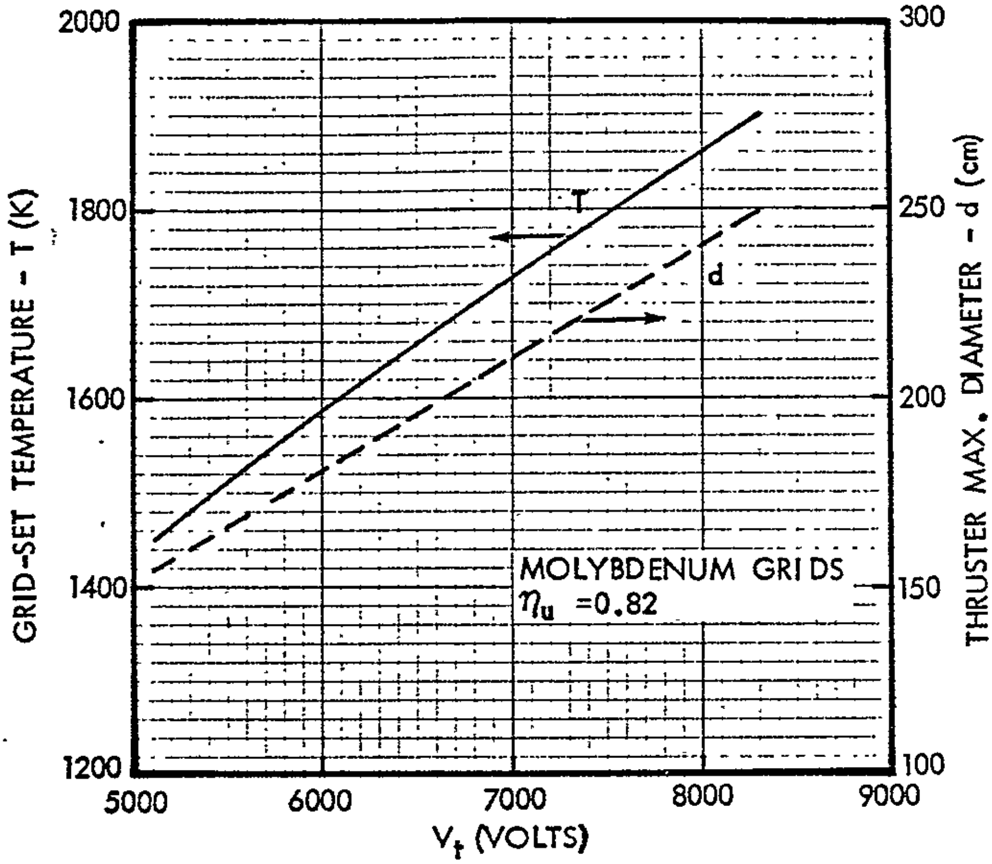

Figure C-3. Total extraction voltage versus selected grid-set operating temperatures, based on Eq. (1), and thruster diameter, based on Eq. (11).

Sovey (Reference 3) has determined an empirical relationship for argon thrusters which yields the maximum practical ion current, J_B, for dished grid systems, operating near the minimum gap (0.06 ± 0.008 cm). This is given by

$$J_B = 4.97\ d^2\ V_T^{2.25} \times 10^{-10} \tag{12}$$

where J_B = beam current (amps),

and d = maximum thruster diameter (cm).

The maximum value for V_T has already been given by Eq. (1) where the selected operating temperature, T, is the independent variable. In terms of T, the maximum beam current becomes

$$J_B = 2.5072\ d^2 T^4\ 10^{-14} \tag{13}$$

C.2.7 Beam Electrical Power - P_B (Watts)

The beam electrical power is given by

$$P_B = J_B V_T R \tag{14}$$

$$= J_B V_N$$

The beam power is controlled by the mass flow rate of argon entering the thrust chamber. The discharge power, P_d, which is the power expended in ionizing the incoming argon gas, is necessary in order to have an ion beam but is not part of the beam power. A plot of thruster module power as a function of extraction voltage ratio, R, for operating under conditions of maximum beam power and thruster size (as determined by the perveance limit, a grid-set span to gap ratio of 600) for various operating temperatures is shown in Figure C-4.

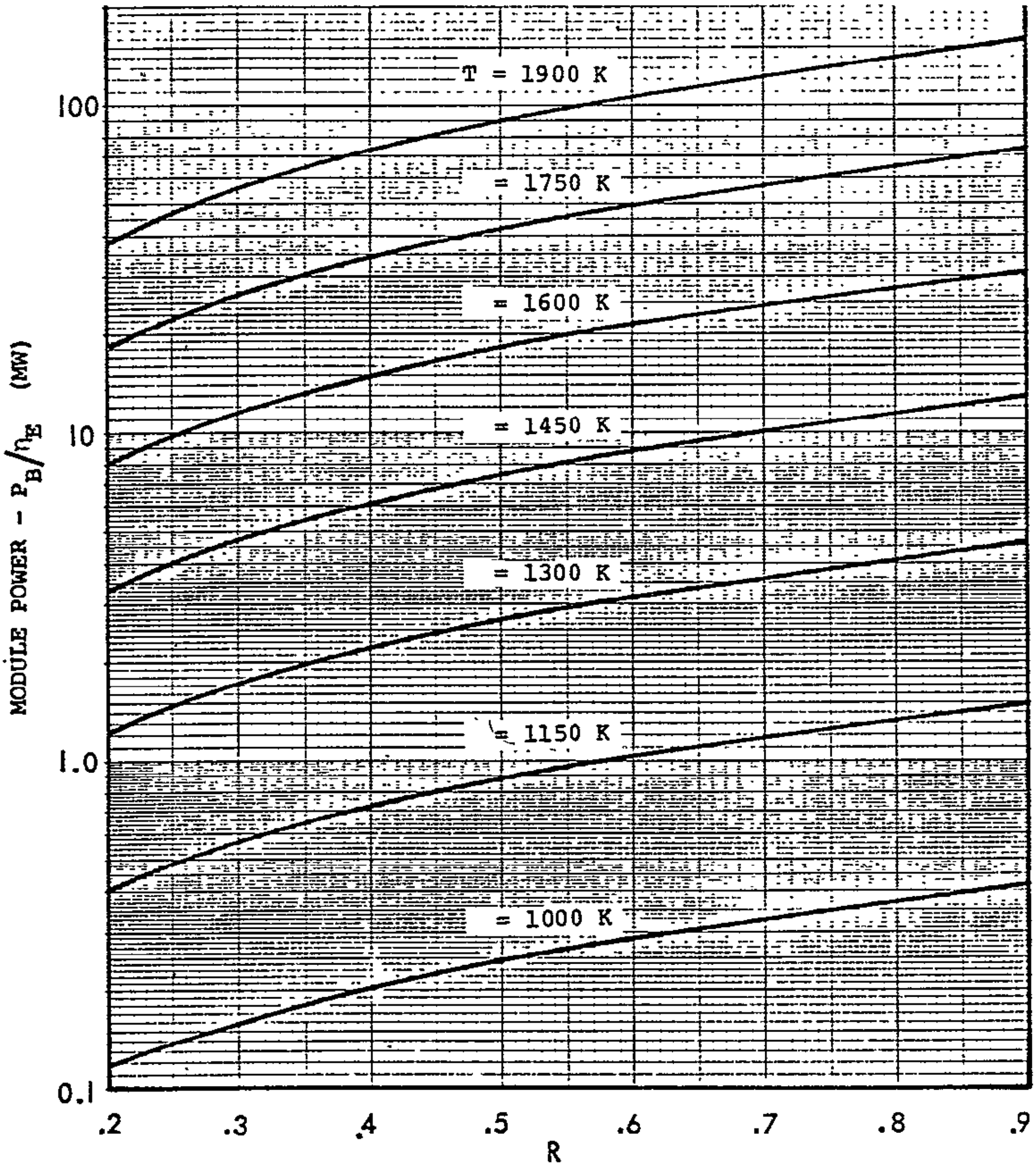

Figure C-4. Thruster Module power as a function of extraction voltage ratio, R.

C.2.8 Thruster Module Electrical Efficiency - η_e

The electrical power efficiency, η_e, of a thruster module in achieving a beam power, P_B, is given by

$$\eta_e = \frac{P_B}{P_B + P_{GS} + P_D\ P_N} \tag{15}$$

where P_{GS} = Grid set loss*,
= $0.0025\ J_B V_N$, (an empirical value)

P_D = Discharge power loss*,
= $200\ J_B$,

and P_N = Beam neutralization loss*,
= 300 Watts (assumed constant).

In terms of voltages and currents

$$\eta_e = \frac{J_B V_N}{J_B V_N + 0.0025\ J_B V_T + 200\ J_B + 300} \tag{16}$$

$$= \frac{R V_N}{R V_N + 200\ R + 0.0025\ V_N + 300\ R/J_B}$$

$$= \frac{1}{\left(\frac{R + 0.0025}{R}\right) + \frac{200}{V_N} + \frac{300}{P_B}}$$

For the large, high power thrusters considered in this study the efficiency may be approximated by

$$\eta_e = V_N/(V_N+200)$$

within 0.6% at the extremes. When the beam power is small (i.e., $\leq$ 300 W) Eqs. (15) and (16) should be used.

A plot of thruster electric efficiency versus R is presented in Figure C-5 for six values of I_{sp}. A temperature of 1900 K was considered the maximum allowable for extended operation of molybdenum grids. This is indicated by the dashed line in Figure C-5. Operation in the shaded area is not permitted. At these higher temperatures it is assumed that the grids would be replaced periodically.

In Figure C-6 the electrical efficiency is plotted against R for various selected operating temperatures. The efficiency increases with grid-set temperature, and at a given temperature, also increases with R.

*Based on conversations with V. K. Rawlin, NASA, LRC

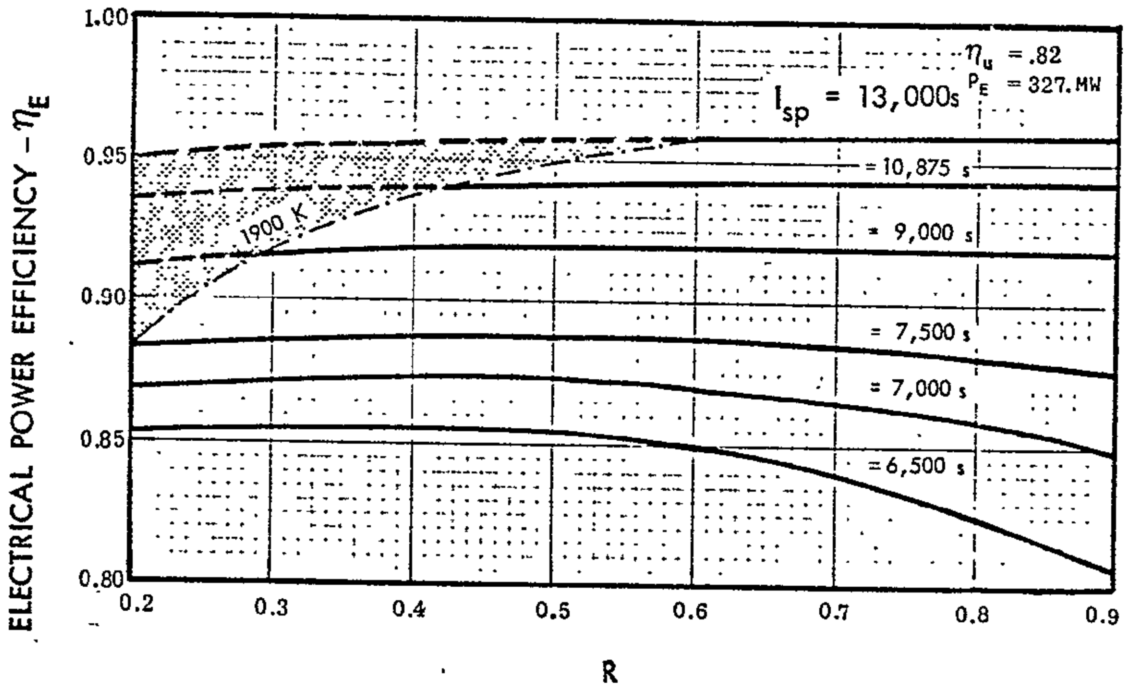

Figure C-5. Electrical power efficiency as a function of extraction voltage ratio, R.

Knowing the electrical efficiency, one can determine the required input power per thruster, P_{TH}, for operation at maximum beam power (i.e., maximum thrust). This is given by

$$P_{TH} = P_B/\eta_E \approx P_B \left(1 + \frac{200}{V_N}\right) \tag{17}$$

However, Eq. (17) does not include electrical power losses or conductor mass penalties attributable to the power input lines distributed within a thruster array. This is the subject of the next section. Such penalties can be serious when the number of thrusters becomes large. Figure C-7 indicates the number of thrusters required for a total array input power of 268.1 MW as a function of extraction voltage ratio and grid-set temperature.

C.2.9 Thruster Performance

Electric and Mechanic Power. The ion energy, E, from Eq. (5) is

$$E = kmv = kq\ V_N$$

$$= Mv\ = Q\ V_N$$

where M = total mass of k ions

Q = total charge of k ions.

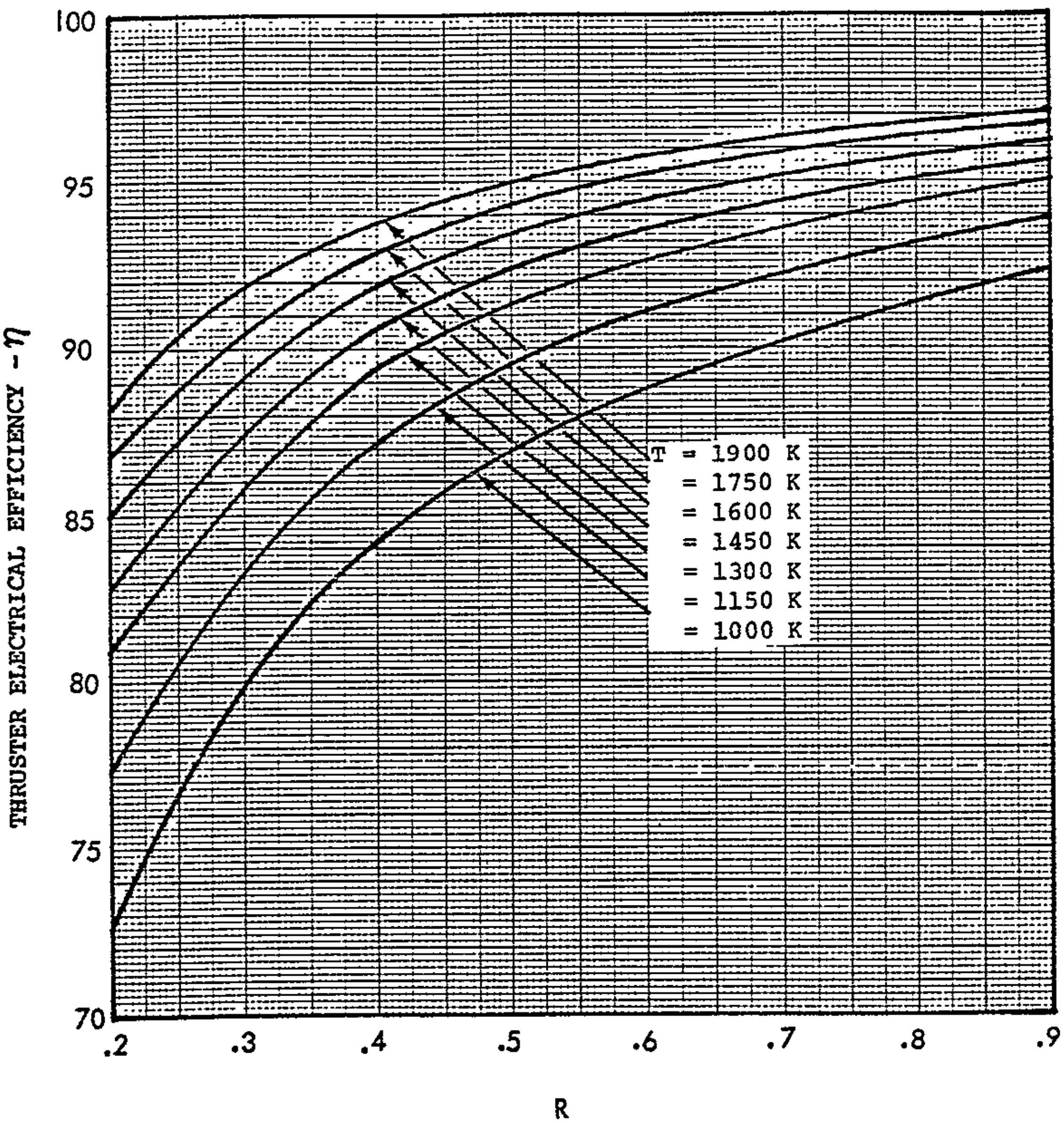

Figure C-6. Thruster electrical efficiency as a function of extraction voltage ratio, R

Power is the rate of change of energy with respect to time. Thus

$$\text{Power} = \tfrac{1}{2}\,\dot{M}v^2 = \dot{Q}\,\bar{V}_N \text{ (Watts)} \tag{18}$$

But, differentiating Eq. (3) with respect to time yields

$$\dot{m}_p\,\bar{v}/v = \dot{M} \tag{19}$$

Now eliminating $\dot{M}$ from Eq. (18) by using Eq. (19) gives

$$\tfrac{1}{2}\,\dot{m}_p\,\bar{v}v = J_B V_N \tag{20}$$

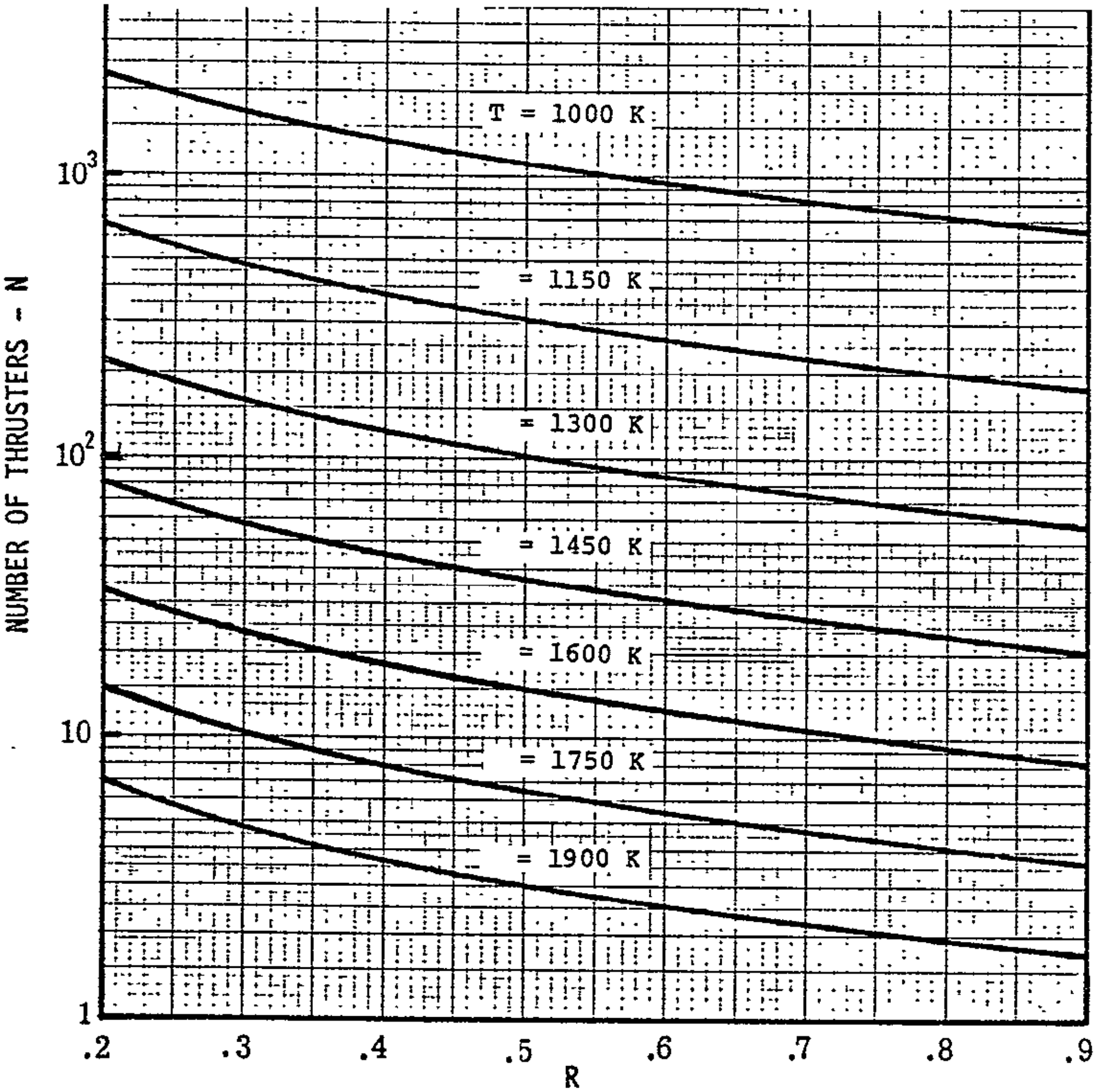

Figure C-7. Number of thrusters for a fixed array input power of 268.1 MW as a function of extraction voltage ratio and grid-set temperature.

where the beam current J_B is used for $\dot{Q}$. Now with the help of Eq. (3) and (4) v and $\bar{v}$ can be eliminated to give

$$\frac{1}{2}\,\dot{m}_p\,\bar{v}^2/\eta_u = \frac{1}{2}\,\dot{m}_p g^2 I_{sp}{}^2/\eta_u$$

$$= J_B V_N = P_B$$

The propellant flow rate is therefore

$$\dot{m}_p = 2\,J_B V_B \eta_u/\left(g\ I_{sp}\right)^2,\ \text{(kg/s)} \tag{21}$$

or for N thrusters each with beam power P_B

$$\dot{m}_p = 2\ N\ P_B \eta_u / \left(g\ I_{sp}\right)^2\ \text{(kg/s)}. \tag{22}$$

Clearly, the mechanical power, P_m, is equal to the electric power P_E, and is

$$P_m = \tfrac{1}{2}\ \dot{m}_p\ g^2\ I_{sp}{}^2/\eta_u \tag{23}$$

Thrust. Thrust is the rate of change of momentum with respect to time. Since the propellant exhaust speed is constant, the thrust, F, is derived from the mass flow rate. Thus

$$F = \dot{m}_p \bar{v} \gamma = m_p g I_{sp} \gamma \tag{24}$$

where $\gamma = \gamma_D \gamma_S \approx 0.9025.$

As defined here γ is the thrust utilization efficiency which accounts for thrust losses caused by beam divergence (γ_D) and the thrust vector steering (γ_S). According to V. K. Rawlin of NASA, LRC, grid compensation techniques should be able to maintain γ_D at 0.95 or more.

Equation (24) can be expressed in terms of beam power by employing Equation (22).

$$F = 2NP_B \eta_u \gamma / gI_{sp} \tag{25}$$

C.2.10 The Rocket Equation

Consider an EOTV with initial mass m_i, final mass (at burnout) m_f and a required velocity increment ΔV.

The total propellant expended in time Δt is

$$m_p = \dot{m}_p \Delta t \tag{26}$$

Gravity losses for low thrust flights between LEO and GEO are assumed to be small. The thrust acting on the EOTV is given by

$$F = \dot{m}_p \bar{v} \gamma = \left(m_i - \dot{m}_p t\right) \dot{V}_s \tag{27}$$

where t = time, or thrust duration,

$\dot{V}_S$ = vehicle acceleration,

and m_i = vehicle initial mass (t=0).

The acceleration of the spacecraft at any time, t, from Eq. (27) is

$$\dot{V}_s = \dot{m}_p \bar{v} \gamma / \left(\dot{m}_i = m_p t\right) \tag{28}$$

Now substituting $W = m_i - \dot{m}_p t$,

and $dW = \dot{m}_p dt$,

in Eq. (28) and integrating yields

$$\Delta V_x = \int_0^{\Delta t} \dot{V}_s dt = -\bar{v}\gamma \int_{m_i}^{m_i - \dot{m}_p \Delta t} \frac{dW}{W}\text{, or} \tag{29}$$

$$\Delta v = \Delta V_s = gI_{sp}\gamma \ell n \left[m_i / \left(m_i - m_p\right)\right].$$

With the help of exponentials, Eq. (29) can be written

$$m_f = m_i e^{\Delta v/gI_{sp}\gamma}\text{,} \quad \text{where} \tag{30}$$

$$m_i = m_p + m_f\text{,} \quad \text{and} \tag{31}$$

$$m_p = m_f\left(e^{\Delta v/gI_{sp}\gamma} - 1\right)\text{,} \quad \text{or}$$

$$m_p = m_i\left(1 - e^{-\Delta v/gI_{sp}\gamma}\right). \tag{32}$$

C.2.11 Attitude Control Propellant

Some of the electric thrusters are used for attitude control while in the Earth's shadow. (Batteries are used to provide the required power). The maximum control thrust requirement occurs in LEO where the gravitational torques are highest. Control requirements become quite small in GEO. In this analysis, the average control thrust was taken to be 400 N, which is believed to be conservative.

The control propellant mass was estimated by taking appropriate fractions of the total propellant consumed during the daylight thrusting period. Thus, for a 120 day trip time and 100 days of thrusting time the shadow period is close to 20 days, which gives a factor of 0.2. The propellant mass is further reduced by the ratio of control thrust (400 N) to total thrust (F). Thus, the control propellant mass, m_{pc}, is given by

$$m_{pc} = \left(\frac{\dot{m}_p \Delta t}{5}\right)\left(\frac{400}{F}\right) \quad (33)$$

$$= 17280\ \dot{m}_p \Delta t(\text{days}) \times \left(\frac{400}{\dot{m}_p g I_{sp} \gamma}\right)$$

$$= 780{,}945\ \Delta t(\text{days})/I_{sp}$$

C.2.12 Thruster Array Properties

Total Distributed Conductor Length. Figure C-8 represents an upper quadrant of a rectangular array of thrusters. The array is fed from a junction at the center labeled P_o. We shall consider only this quadrant and calculate the total mass and total power loss of the power distribution wiring between the thrusters in the quadrant and the terminals in the junction box.

Each of the N thrusters is connected by a pair of conductors that run horizontally along the width L_w of the array, and then vertically along the height, L_h. This is illustrated for the kth thruster. The thruster diameter, d, and the number of thrusters, determine the array dimensions. The separation distance between thrusters, or between a peripheral thruster and the adjacent edge of the array structure, is half the thruster diameter, i.e., d/2. Thus, the vertical distance ℓ_k to the kth thruster is

$$\ell_k = d\left[1 + 1.5\ (k-1)\right] = \frac{d}{2}\ (3\ K-1) \quad (34)$$

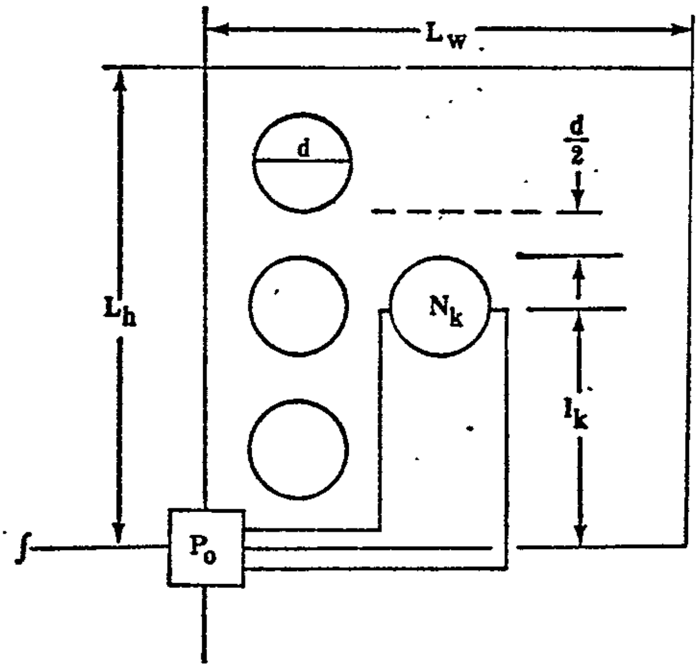

Figure C-8. Schematic representing one quadrant of a rectangular array of thrusters

If there are N_h thrusters in each column the cumulative length of N_h wires (one way) is given by the sum

$$\sum_{k=1}^{N_h} \ell_k = dN_h \left(1 + 3\ N_h\right)/4 \tag{35}$$

Since each thruster requires two wires the total vertical wire length per column becomes

$$L_v = dN_h \left(1 + 3\ N_h\right)/2 \tag{36}$$

Since there are N_w columns, the total length of vertical wiring is

$$L_{vt} = dN_h N_w \left(1 + 3\ N_h\right)/2 \tag{37}$$

There is also a horizontal component of wire, the total length, L_{ht}, of which is given by a similar type formula,

$$L_{ht} = dN_h N_w \left(1 + 3\ N_w\right)/2 \tag{38}$$

If Equations (37) and (38) are added together the total required two-way wire length, ℓ_t, is obtained by

$$\ell_t = dN_h N_w \left[1 + 1.5 \left(N_h + N_w\right)\right] . \tag{39}$$

For a square array

$$N_h = N_w = \sqrt{N} \tag{40}$$

and $$\ell_t = dN \left[1 + 3\sqrt{N}\right] ,$$

where N is the number of thrusters.

Array conductor length as a function of extraction voltage ratio for several operating temperatures is presented in Figure C-9 for an array input power of 268.1 MW.

Distributed Conductor Size, Mass, and Power Loss. Transmission of electric power from the array input junction to each thruster is critical to the array sizing problem, not only with respect to mass, length, power loss and cost, but also with respect to orbital labor, ease of construction, and refurbishment. It is desirable to have conductors that radiate heat efficiently, but are not of excessive area so that the insulation is subject to numerous pin holes from micrometeoroid impacts. Each such opening is a potential site for plasma discharge losses when at low orbital altitude. Restrictions were therefore applied to the size and shape of the conductors.

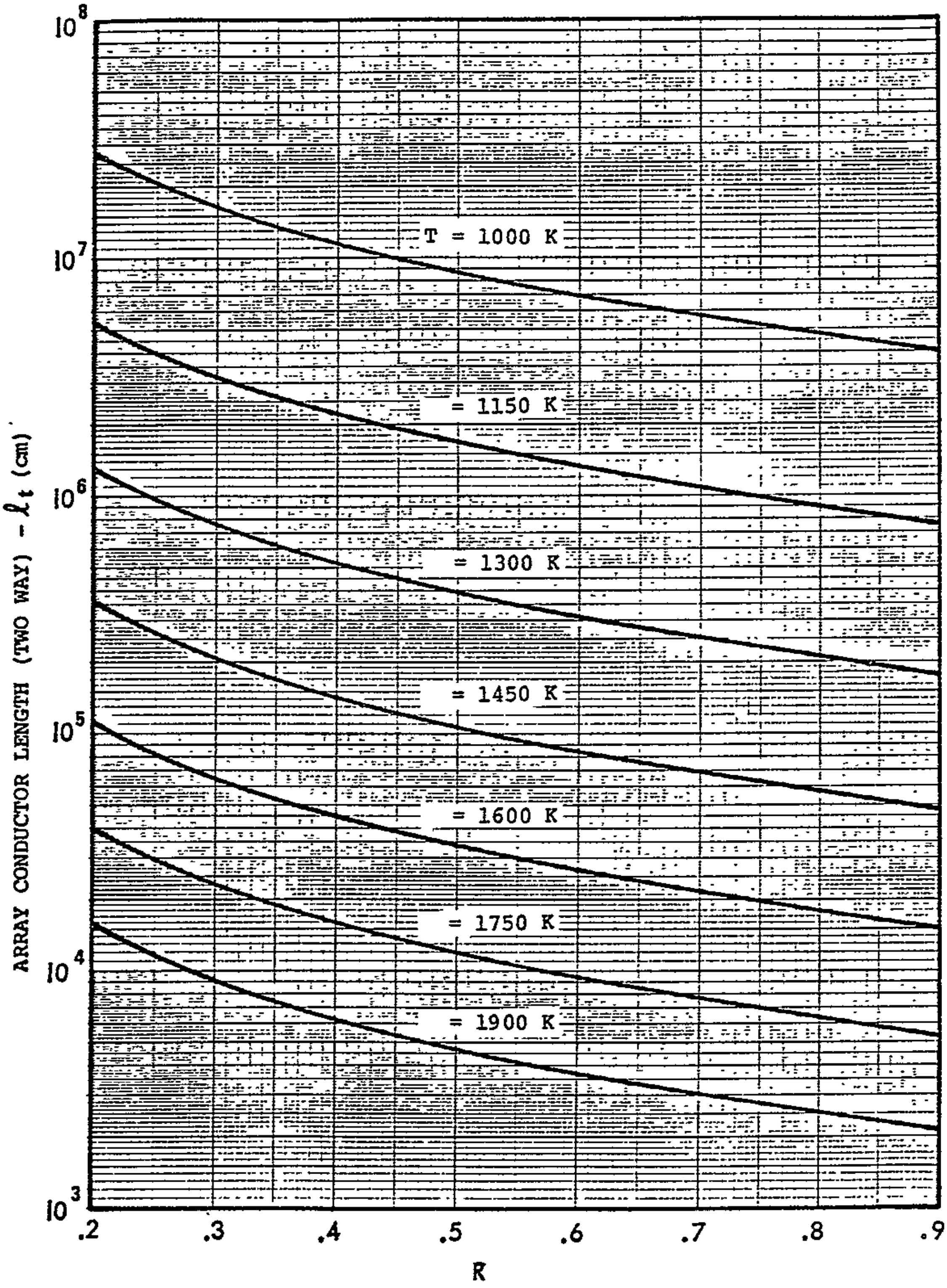

Figure C-9. Electrical Conductors (feeders) length for an array of thrusters, operating at the indicated grid-set temperatures, as a function of extraction voltage ratio, R.

In a point design there are good reasons why cylindrical conductors might be preferred. For example, the conductor area exposed to meteor streams could be reduced by an order of magnitude. This is important with regard to the Kapton insulation which could deteriorate prematurely both thermally and electrically. Small "pinholes" can yield significant plasma discharge losses in LEO (Reference 6). The reduction in conductor area permits an associated increase in the Kapton mass density. Further, there is the possibility of heating the argon by piping it through the cylindrical conductors. This also tends to keep the conductors cooler and therefore yields more available electric power. However, time did not permit a completion of this analysis. For purposes of this parametric study the conductors are assumed to be rectangular and shaded at all times.

A conducting strip with a width/thickness (m/n) ratio of 20 can be a reasonably good thermal radiator, and still retain structural integrity. A lower limit of 0.038 cm (15 mils) was placed on thickness. Strips of this size can be handled during construction or repair phases without excessive difficulties.

The power dissipated in a flat conductor is lost mostly by radiated heat. A layer of Kapton).00254 cm thick (one mil) was used to improve the radiation efficiency and also for insulation to help prevent plasma discharges. Kapton has an emissivity, ε, of approximately 0.68 which is an improvement on aluminum (0.05 to 0.11).

The maximum allowable wire temperature from electric power loss heating was assumed to be 373.16 K (100°C). A summary of the assumed conductor characteristics is given below:

$T \leq 373.16$ K maximum conductor temperature,

$m = 20\ n$ width of conductor,

$A = mn = 0.05\ m^2$ cross section,

$n > 0.0381$ cm (15 mils) in thickness,

$\rho = 2.70\ g/cm^3$ density,

and for the electrical resistivity

$$\gamma_E = 2.828\times10^{-6}\ [1+0.0039\ (T-293.16)]\ \text{ohm-cm}$$

$$= 3.7103\times10^{-6}\ \text{ohm-cm at } 373.16\ K$$

The thermal power radiated is given by

$$P_H = 2\ell m\varepsilon\sigma\ T^4 + 2\ell n\varepsilon\sigma T^4, \qquad (41)$$

$$= 2\ell\varepsilon\sigma\ T^4\ (m + n)$$

where $\sigma = 5.66961\times10^{-12}\ W/cm^2/K^4$, The Stephan-Boltzman constant.

The thermal power radiated, P_H, is balanced by the electrical power P_ℓ lost, or dissipated, in the conductor. The power lost in a conductor of length ℓ, with a voltage drop ΔV and current I is

$$P_\ell = \Delta VI = I^2\left(\gamma_E \ell/A\right) = 20I^2 \left(\gamma_E \ell/m^2\right) \tag{42}$$

Equating the rhs's of Equations (41) and (42) yields

$$mn\ (m + n) = I^2\gamma_E/(2\varepsilon\sigma T^4) = 0.0525\ m^3, \tag{43}$$

and $$m = 9.5238\ I^2\gamma_E/\varepsilon\sigma T^4,$$

$$= 6.986\times10^6\ I^2[1+0.0039(T-293.16)]/T^4 \tag{44}$$

At the upper temperature limit (373.16 K)

$$m^3 = 4.72696\times10^{-4}\ I^2,\ cm^3 \tag{45}$$

and $$m = 7.78982\times10^{-2}\ I^{2/3},\ cm$$

The total conductor mass M_c, of length ℓ_t, which includes a 10 percent penalty for structural support is given by

$$M_c = 1.1\ \rho A\Omega_t = 1.1\ \rho mn\ell_t \tag{46}$$

$$= 1.485\times10^{-4}\ m^2\ \ell_t,\ kg$$

the total power lost in the array wiring of length ℓ_t is

$$P_{\ell t} = \frac{5.656\times10^{-5}\ [1+0.0039(T-293.16)]\ell_t^{\ 2}}{m^2} \tag{47}$$

$$= 7.42067\times10^{-5}\ell_t I^2/m^2,\ \text{Watts at } 373.16\ K$$

Equations (45 through (47) can be used to size the array conductors once the current I is known.

Solar Panel Bussbar Power. The required power for the thruster array from the solar panels is

$$P_o = N\left(P_o' + P_{TH}\right) \tag{48}$$

$$= N\left[I^2\gamma_E\ell/(mn) + J_B V_N/\eta_E\right]$$

where P_o' = conductor panel loss per thruster,

N = number of thrusters,

and $\ell = \ell_t/N$, average two-way conductor length from junction box to each thruster.

The net voltage drop, V_o, in the distributed wiring and thruster array is assumed to be

$$V_o = I\gamma_E \ell/(mn) + V_N \tag{49}$$

where conservation of current requires that

$$I = J_B/\eta_E \tag{50}$$

Equation (48) can therefore be written

$$P_o = NI\ [V_N + I\gamma_E \ell/(mn)]\ . \tag{51}$$

The bussbar current for the entire array is therefore

$$I_o = NJ_B/\eta_E\ . \tag{52}$$

Application to Electric Thruster Arrays. It is desired that the voltage V_N at each thruster be fixed, for any given specific impulse, I_{sp}. In order to keep the voltage, V_N, at each thruster identical it will be assumed that the thrusters are connected in parallel, each with a properly designed "fuse" in case of a short circuit. The power losses, P_ℓ in the distributed conductors are assumed to be identical for each thruster. In order to make a fair comparison of required wire mass and sizes the conductor width m is determined initially from Equation (45) under conditions where the current per thruster is at a maximum and therefore m is at a maximum. This occurs, assuming fixed total available power, when the array size is at a minimum (R = 0.9), and the grid-set temperature, and therefore V_T, are at the highest values to be considered [see Eqs. (1) and (2)].

Equation (47) is then used to determine total conductor power loss. This power loss $P_{\ell t}$, is fixed thereafter in order to have a fair basis of comparison. Thus, as R is increased, m can be determined from the relation

$$m = 8.6143\times10^{-3}\ I\ \sqrt{\ell_t/P_{\ell t}}\ ,\ \text{cm} \tag{53}$$

which then leads to conductor mass.

Conductor masses are shown in Figure C-10. The increases in conductor mass are phenomenal with decreases in R and/or T.

For subsequent point design studies it was found beneficial to keep the ratio of $P\ell_t/M_c$ comparable to M_{pld}/P_o where M_{pld} is the mass of the payload. In other words up to a point it pays to increase the array conductor mass, and thereby reduce the array electrical power loss. This increases thrust

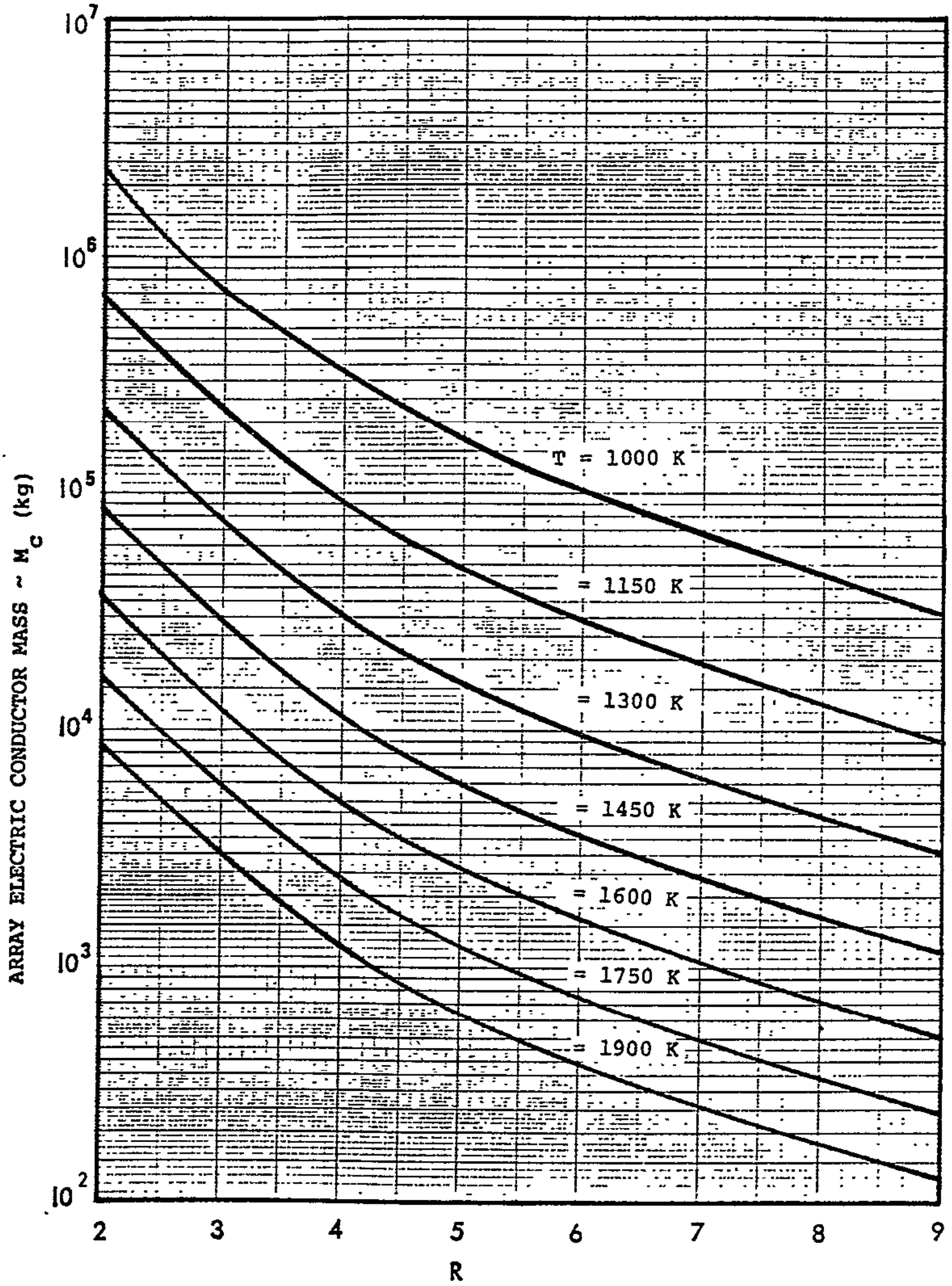

Figure C-10. Electrical conductor mass of length ℓ_t required to feed N thrusters as a function of grid set temperature and extraction voltage ratio, R.

and may yield an increase in payload that exceeds the increase in conductor mass. Also, it enables operation at much lower wire temperature which reduces resistivity. Thus, from Eqs. (46) and (47), and the relation

$$P_{\ell t}/M_c = M_{pld}/P_o$$

it follows that

$$m = 0.78559\ [1+0.0039\ (T-293.16)]^{\frac{1}{4}} I^{\frac{1}{2}} \left[\frac{P_o}{M_{pld}}\right]^{\frac{1}{4}} . \tag{54}$$

Thruster and Supporting Structure Mass. Referring to Figure C-8, the height of the array is L_h and the width L_w. In terms of thruster diameter, d, the array height and width is given by

$$L_h = 1.5\ N_h\ d,$$

and

$$L_w = 1.5\ N_w\ d.$$

Also

$$N = N_h N_w,$$

where N_h and N_w are the respective number of thrusters along the height and width, and N the total number of thrusters. The total thruster module mass is given by

$$M_{th} = 120\ N_h N_w\ d^2,\ \text{kg} \tag{55}$$

where d is in meters.

The structure mass can be taken to be ten percent of the total thruster mass. The total mass of thrusters and structure M_{sth} is therefore

$$M_{sth} = 132\ N_h N_w\ d^2,\ \text{kg}, \tag{56}$$

Thruster array mass as a function of grid-set temperature and extraction voltage ratio are presented in Figure C-11.

Battery Mass. During periods of darkness when the EOTV is eclipsed by Earth, a fraction of the thrusters are operated on batteries to accomplish attitude control. The required battery capacity is determined by the longest duration of darkness, t_D, about 30 minutes. There is ample time between eclipses for the batteries to recharge. If F_c is the required control thrust and E_B is the watt-hours/kg capability of the batteries then the battery mass, m_B, is

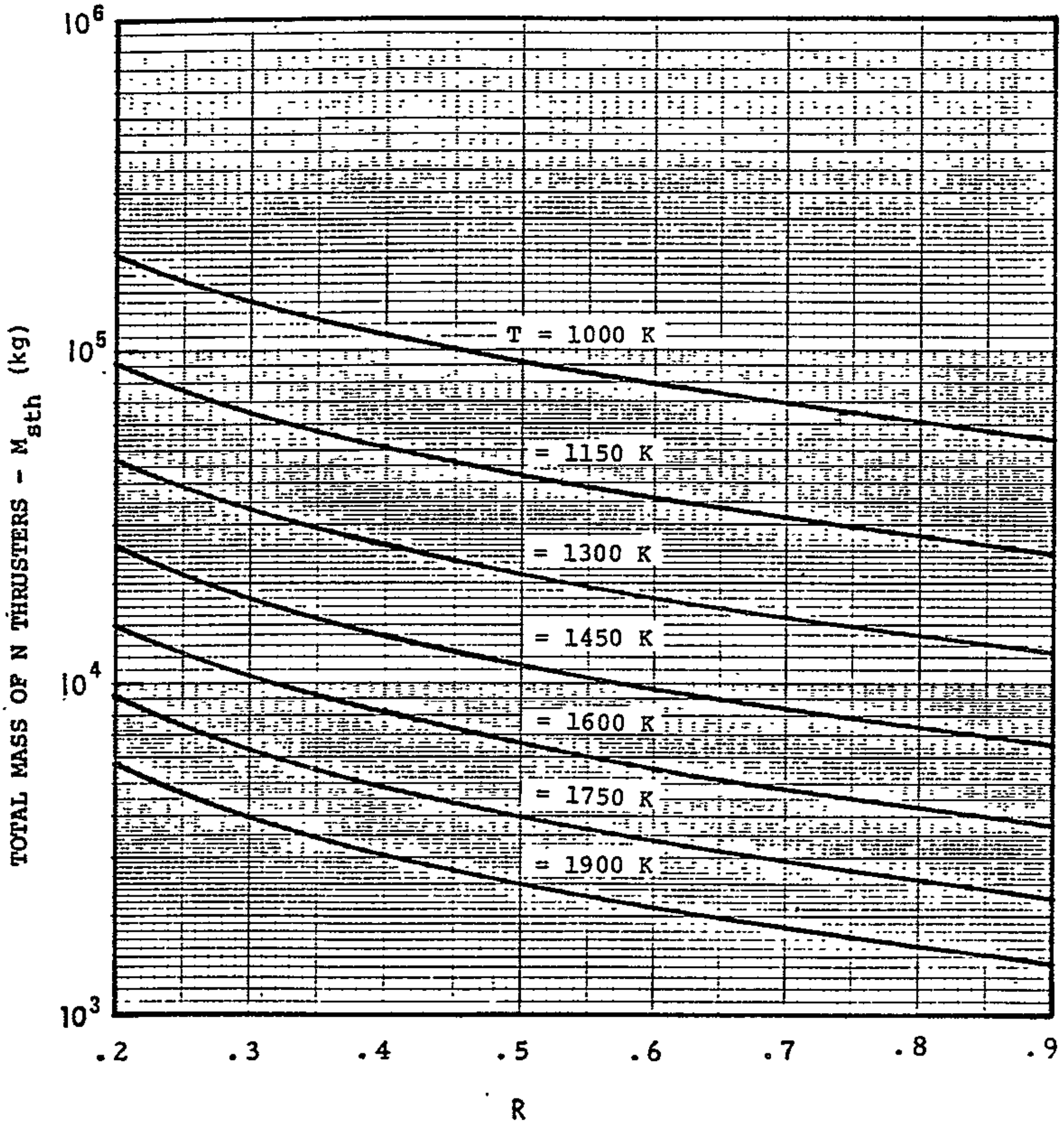

Figure C-11. Mass of N thrusters including supporting structure, as a function of grid-set temperature and extraction voltage ratio, R.

$$m_B = \left(\frac{F_c}{F}\right)\left(\frac{t_d P_o}{E_B}\right) \tag{57}$$

$$= \left(\frac{F_c \quad t_d}{2\gamma\eta_u NP_B/gI_{sp}}\right)\left(\frac{NP_B/\eta_E}{E_B}\right)$$

$$= \frac{gI_{sp} t_d F_c}{2\gamma\eta_u \eta_E E_B}$$

Adding ten percent for structure, yields

$$m_B = \frac{5.39385\ I_{sp} t_d F_c}{\gamma\eta_u \eta_E E_B}\ . \tag{58}$$

For the parametric study the following values were assumed:

F_c = 1000 N

t_D = 0.5 hours,

and E_B = 200 Watt-hours/kg.

Equation (58) can therefore be written

$$m_B = 18.22\ I_{sp}/\eta_E,\ \text{kg} \tag{59}$$

or in terms of V_N

$$m_B = 3346 \times \left(\frac{V_{N+200}}{V_N}\right)\ . \tag{60}$$

C.3 PARAMETRIC EOTV SIZING

Figures C-12 through C-20 present some of the results of the parametric study which, in effect, are estimates of thruster and spacecraft parameters as a function of grid-set temperature and extraction voltage ratio. The temperatures ranged from 1000 K to 1900 K. All of the figures have captions that should be self-explanatory.

The electric power was assumed to be constant at the thruster array junction box. The total power available, after subtracting the various losses such as 15 percent solar array degradation, and 6 percent line loss, etc., at the junction box was 268.1 mW. Initial power from two SPS bay solar arrays was 335.5 mW. The power available per thruster array for four arrays is 67.025 mW.

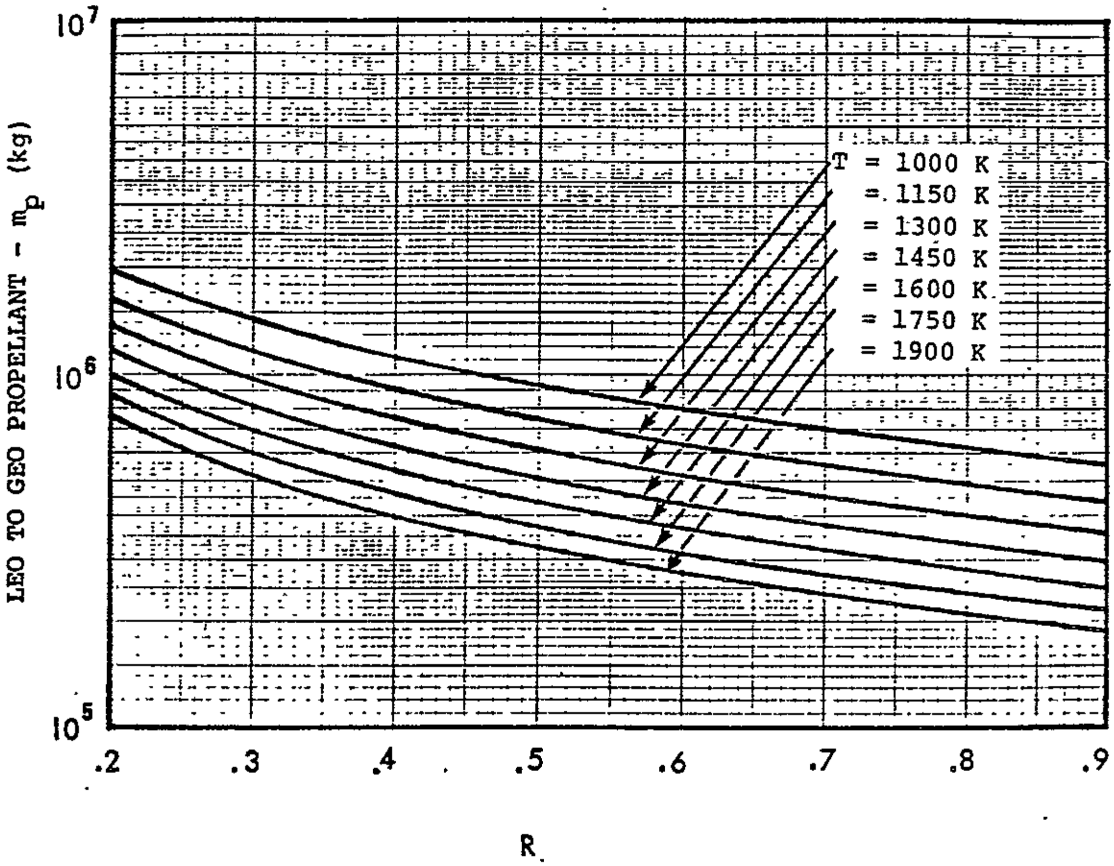

Figure C-12. Propellant expended by the electric OTV in transporting payloads between LEO and GEO for the indicated temperatures as a function of extraction voltage ratio, R.

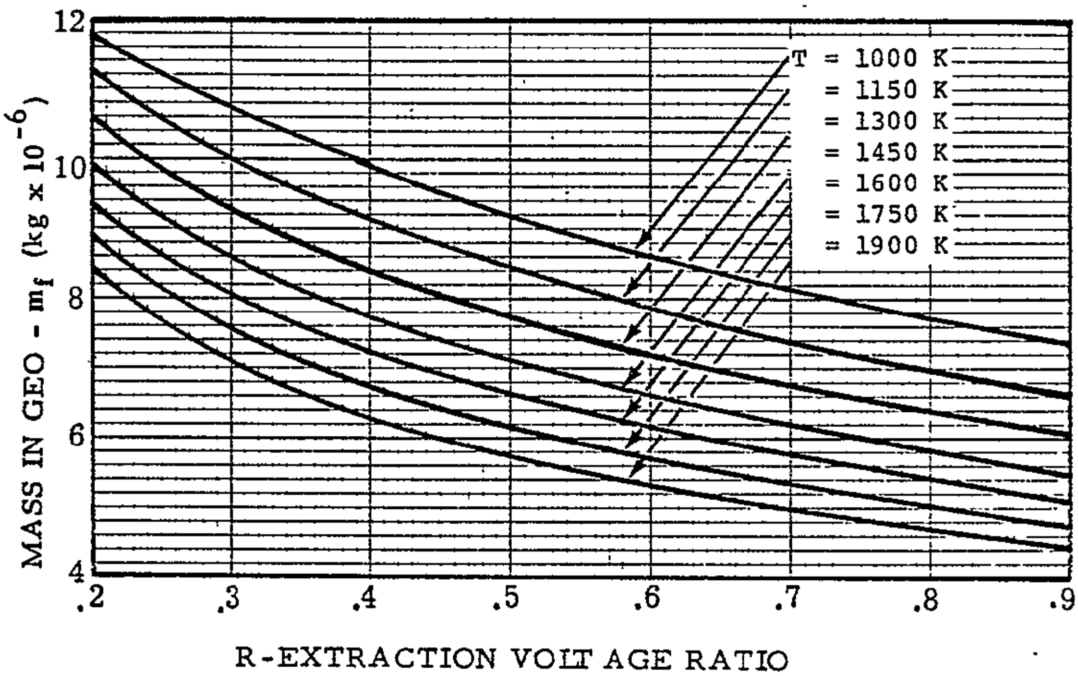

Figure C-13. Final mass, m_f, remaining upon arrival in GEO after expending a mass of propellant, m_p as a function of R for the indicated grid-set temperatures.

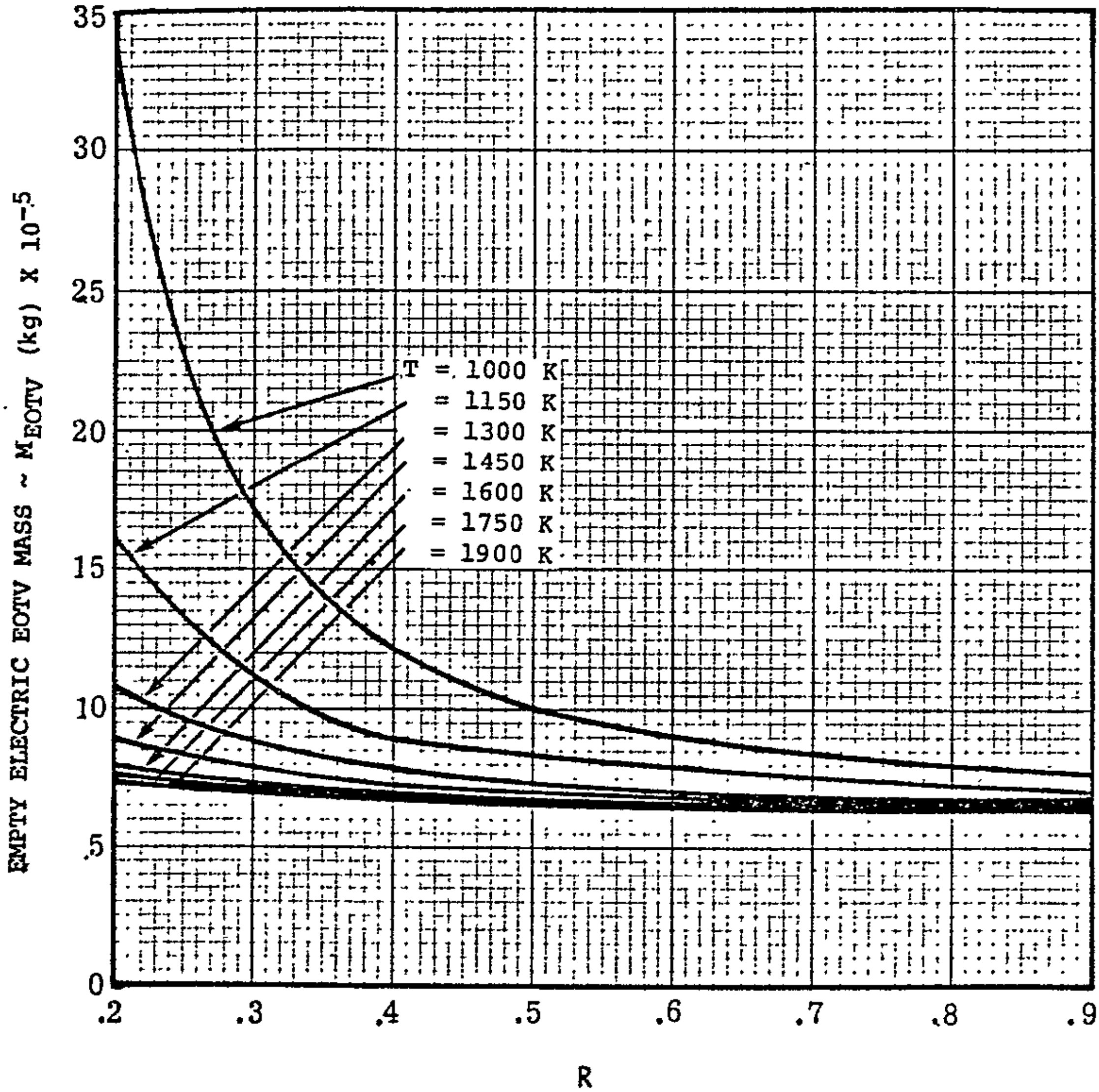

Figure C-14. Empty EOTV mass as a function of R for the indicated grid-set temperatures. (Return propellant lines and tanks not included.)

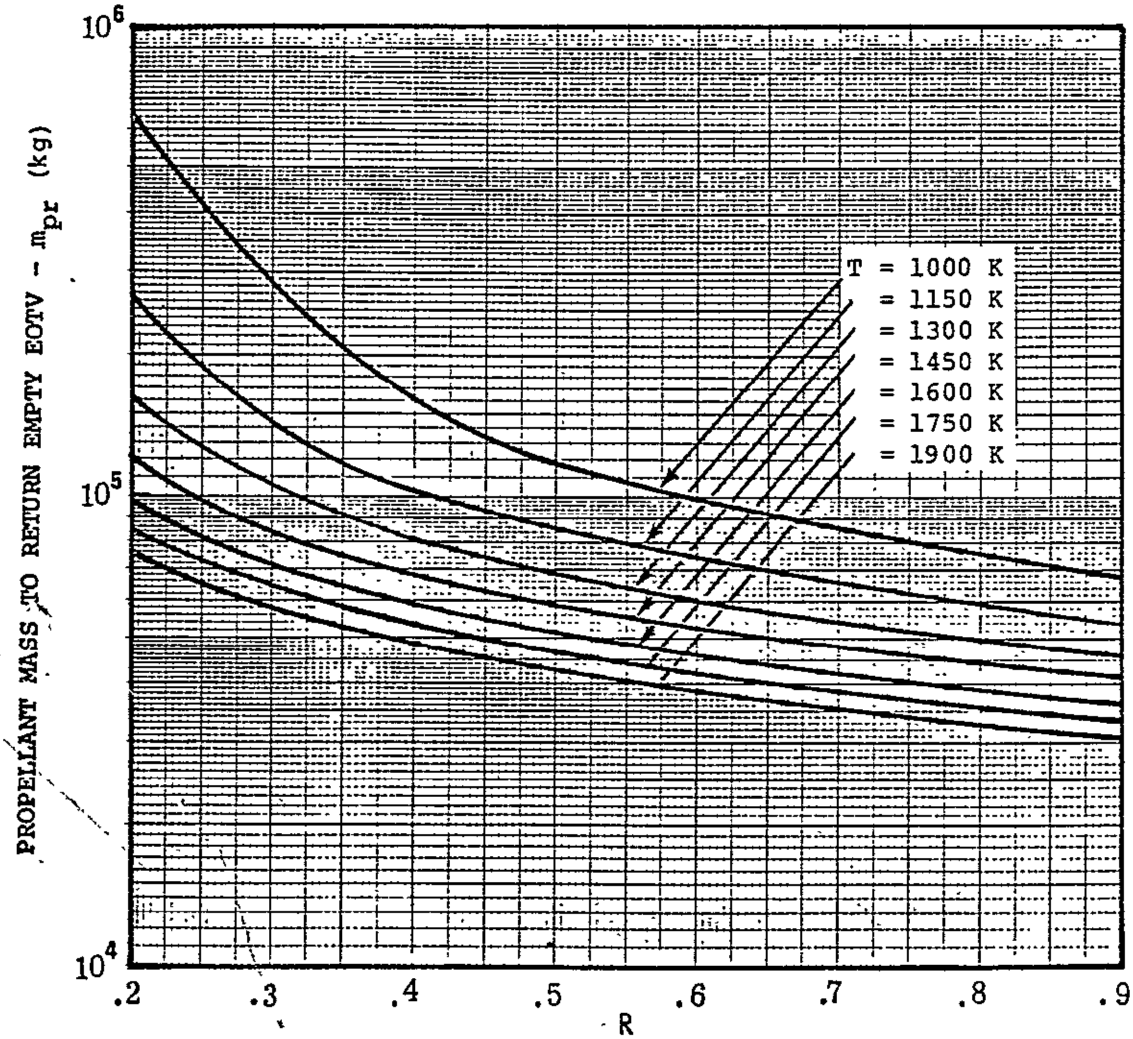

Figure C-15. Propellant required to return the empty EOTV from GEO to LEO. (15% growth margin included.)

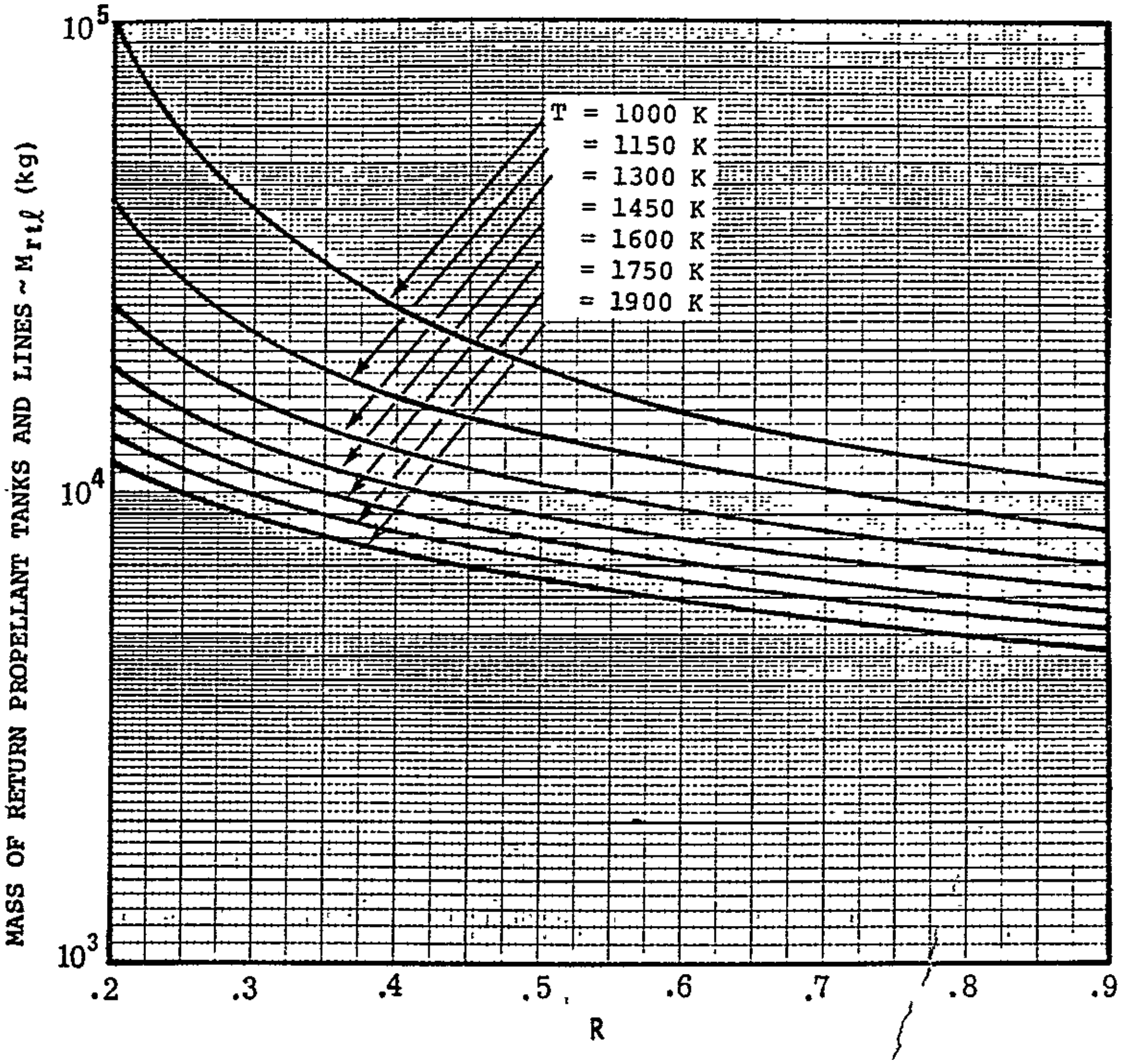

Figure C-16. Mass of return propellant tanks and lines as a function of R.

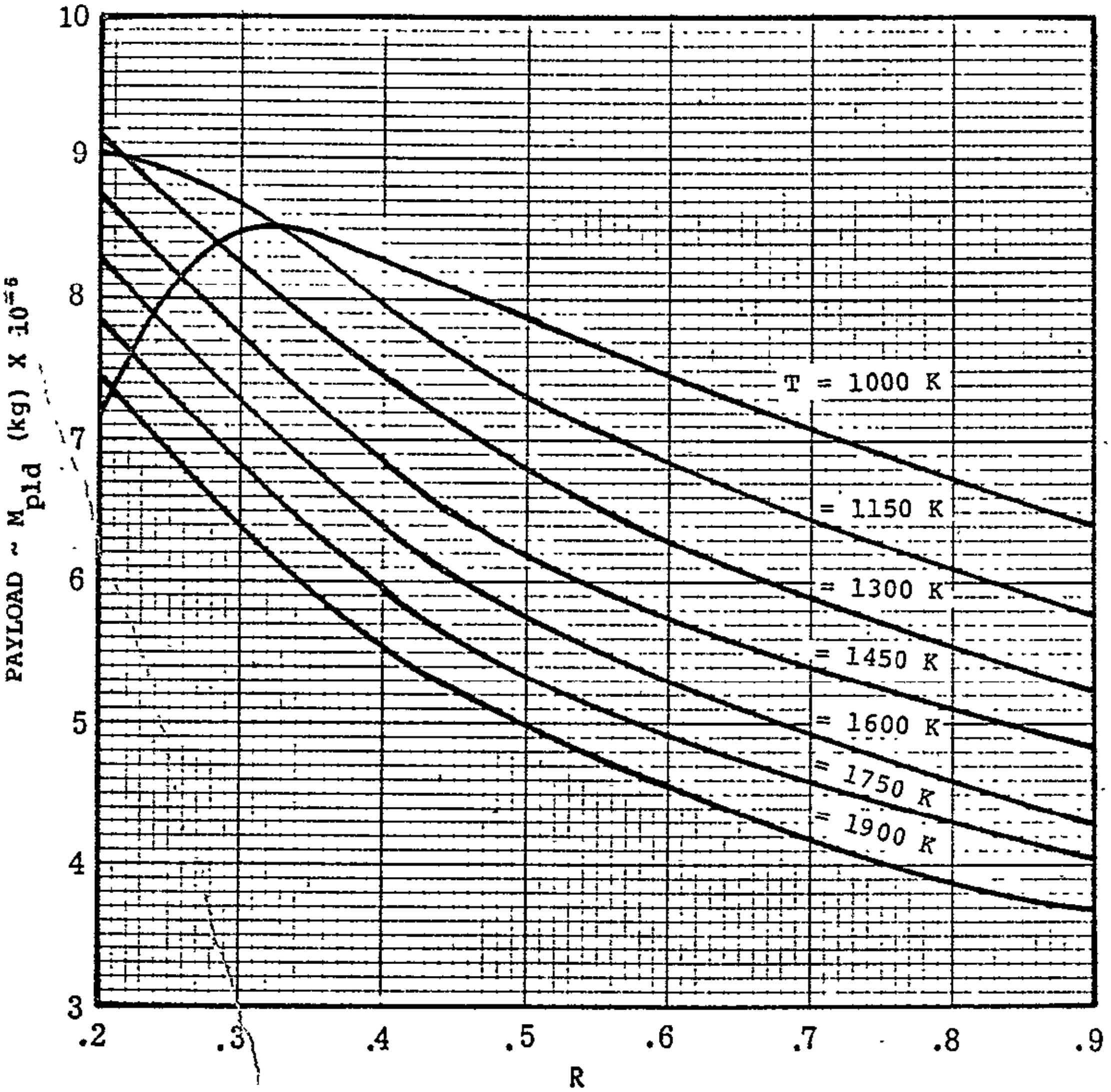

Figure C-17. Payload delivered to GEO with EOTV returning without payload to LEO.

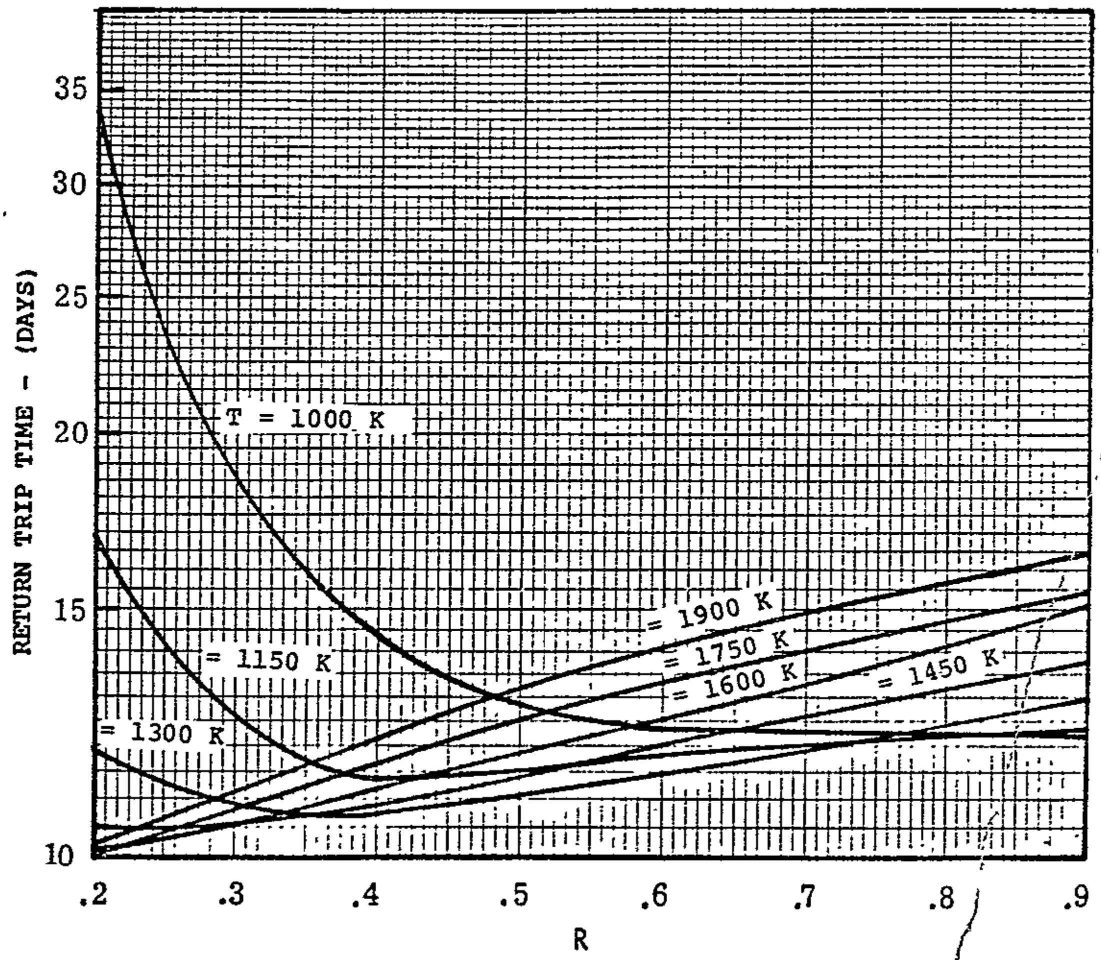

Figure C-18. Electric OTV return trip time from GEO to LEO without payload.

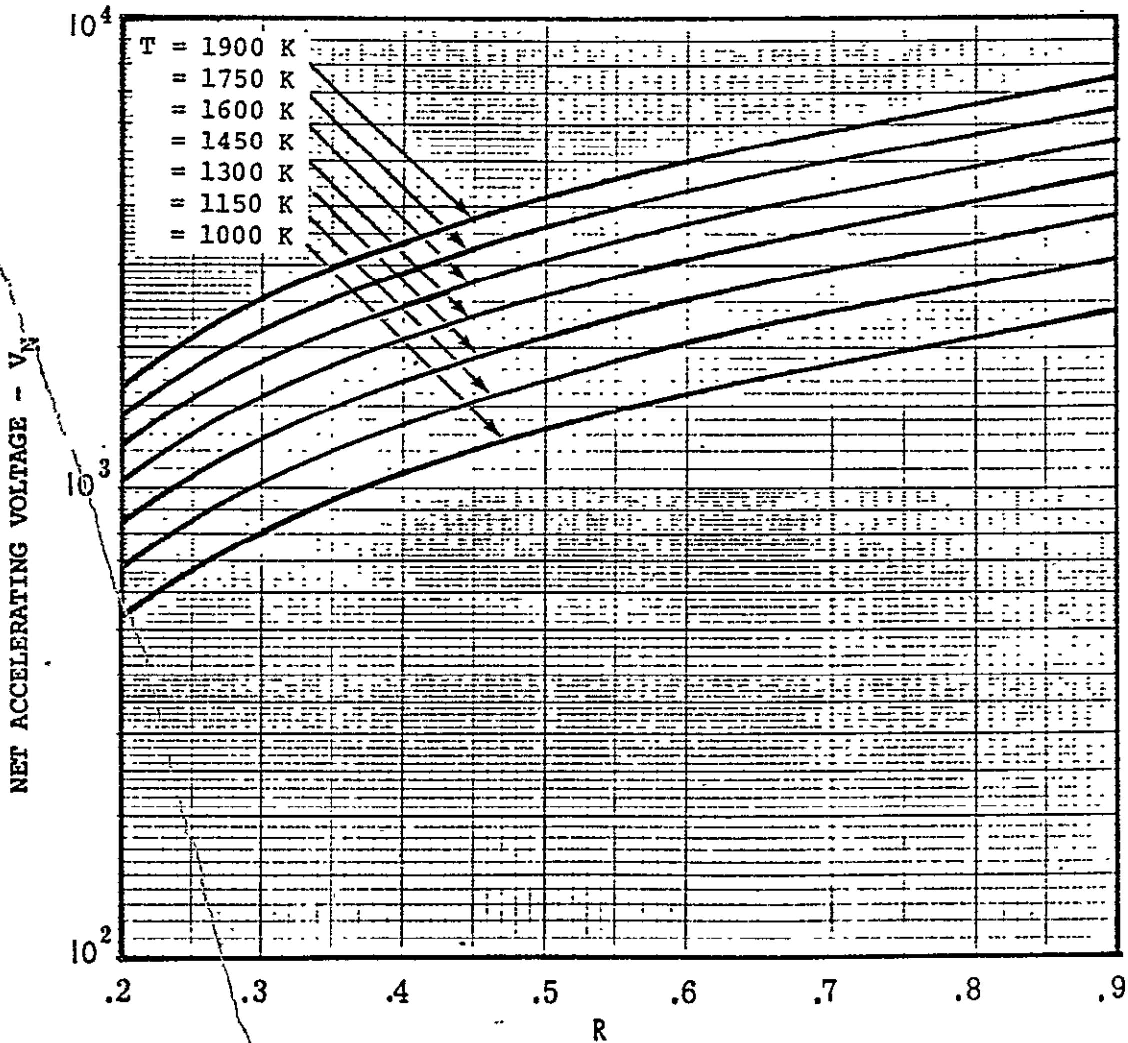

Figure C-19. Net accelerating voltage for the indicated grid-set temperatures as a function of the extraction voltage, ratio, R.

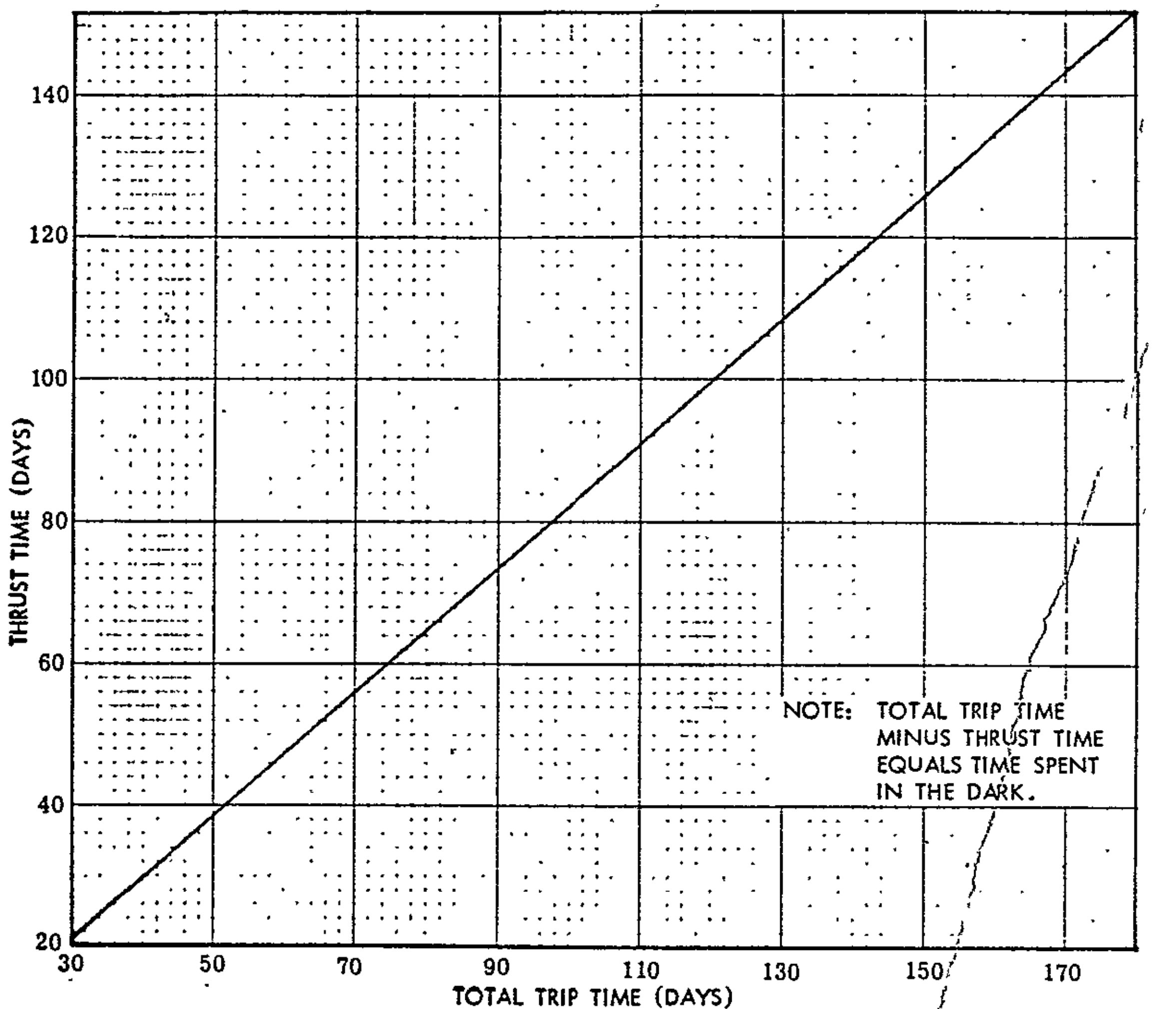

Figure C-20. Estimated thrust duration versus total trip time for optimum thrust vector steering.

The various EOTV fixed masses (kg) were:

Solar Array		588,196
cells/structure	299,756	
power conditioning	288,410	
Thruster Arrays (4)		2,256
beam/gimbals	2,256	
Attitude Control System		1,000
system components	274	
		590,726 kg

An interesting result was deduced from the supporting calculations for Figure C-17. The payloads delivered to GEO increase as the grid-set temperature decreases, down to about 1300 K. At 1150 K the payload falls below the 1300 K curve, as R approaches 0.2, because of excessive electrical conductor mass. At 1000 K, and at R = 0.2, the payload drops almost two million kilograms more but peaking at R = 0.32. Presumably, as the temperature is lowered this peak would occur at increasing values of R.

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