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A TWO STAGE LAUNCH VEHICLE for use as AN ADVANCED SPACE TRANSPORTATION SYSTEM FOR LOGISTICS SUPPORT OF THE SPACE STATION

A design project completed by students in the Department of Aerospace Engineering at Auburn University, Auburn, Alabama, under the sponsorship of USRA Advanced Space Design Program

Auburn University
Auburn, Alabama
June, 1987

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Dept. Aerospace Engineering
Auburn University, AL 36849
June 25, 1987

Universities Space Research Association
Suite 530, One Corporate Plaza
2525 Bay Area Blvd.
Houston, TX 77058

Attention: Mrs. Carol Hopf

Dear Mrs. Hopf:

Transmitted herein is the final report of the design project completed by students at Auburn University under USRA's University Advanced Design Program.

The report was originally written as three separate reports but has been combined herein to facilitate handling and review.

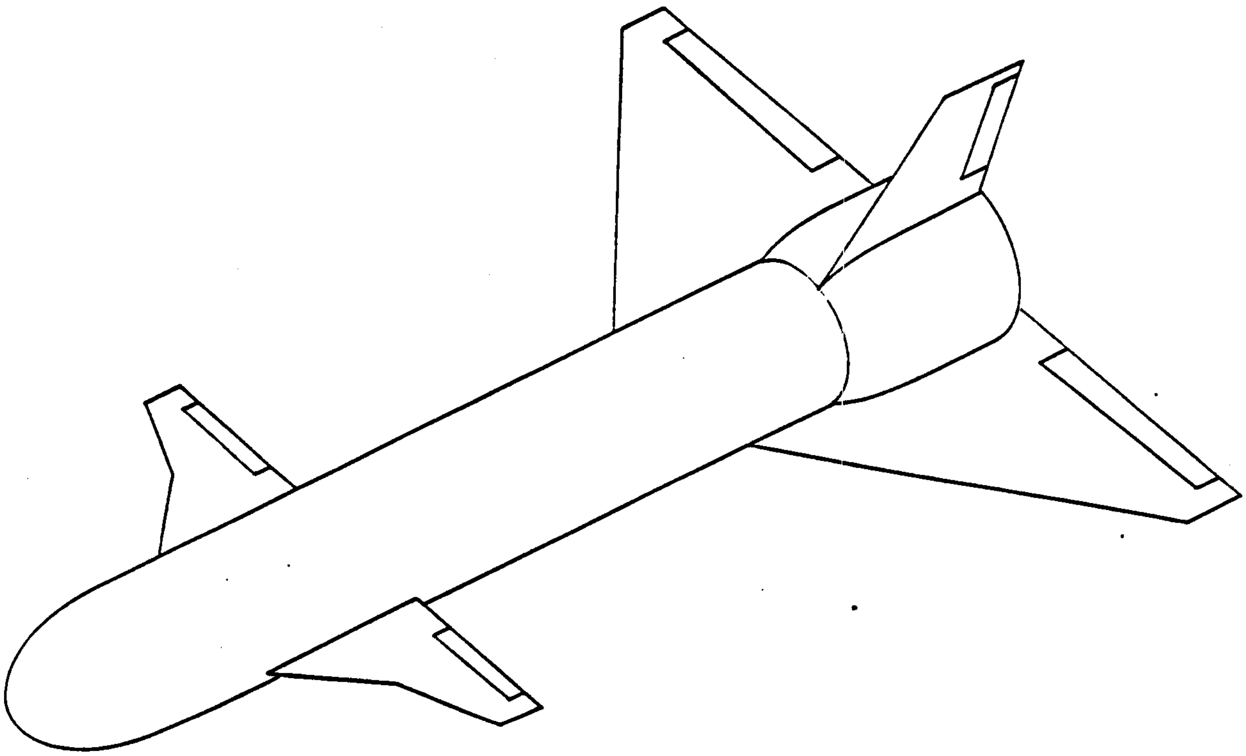
Two copies are being forwarded to you and one copy to Mr. Frank Swalley at Marshall Space Flight Center in Huntsville, Alalabama.

Sincerely,

James O. Nichols

James O. Nichols, PhD
Associate Professor

FULLY REUSABLE
WINGED FLYBACK BOOSTER



AUBURN UNIVERSITY
AUBURN, ALABAMA

AE 449 AEROSPACE DESIGN

Auburn University

Auburn, Alabama

Preliminary Design Specifications For A Winged Flyback Booster For The Advanced Space Transportation System

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ABSTRACT

This report describes the preliminary design specifications for an Advanced Space Transportation System consisting of a fully reusable flyback booster, an intermediate-orbit cargo vehicle, and a shuttle-type orbiter with an enlarged cargo bay. It provides a comprehensive overview of mission profile, aerodynamics, structural design, and cost analyses. These areas are related to the overall feasibility and usefulness of the proposed system.

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LIST OF SYMBOLS

Symbol	Definition	Units
a	Speed of Sound	ft/sec
A^*	Throat Area	ft
A_e	Exit Area	ft^2
A_i	injector area nozzle	ft^2
ALPHA	Angle of Attack	deg
A	Aspect Ratio	---
b	Wing Span	ft^2
B_C	Wing Span of Canard	ft
B_W	Wing Span of Wing	ft
BETA	Bank Angle	deg
C_1	Constant Given as a Function of Taper Ratio	---
C_2	Constant Given as a Function of Taper Ratio	---
C^*	Characteristic Velocity	ft/sec
C_D	Coefficient of Drag	---
$(C_{Di})_{WB}$	Induced Drag Coefficient of a Wing-Body Combination	---
$(C_{DL})_W$	Drag Coefficient Due to Lift of the Wing	---
$(C_{Do})_B$	Zero Lift Drag Coefficient of the Body	---
$(C_{Do})_N$	Zero Lift Drag Coefficient of the Wing	---
$(C_{Do})_{WB}$	Zero Lift Drag Coefficient of a Wing-Body Combination	---

Symbol	Definition	Units
$[C_{D(\alpha)}]_B$	Drag Coefficient Due to Angle of Attack	---
$\delta C_{D(MISC)}$	Miscellaneous Contributions to Drag Coefficient	---
Cd'	Discharge Coefficient	---
C_L	Coefficient of Lift	---
C_M	Coefficient of Moment	---
$C_{N\alpha}$	Normal Force Variation with Angle of Attack	---
C_P	Specific Heat Constant Pressure	$\frac{\text{ft-lbf}}{\text{lbm mole }^\circ\text{R}}$
C_T	Ideal Thrust Coefficient	---
C_V	Specific Heat Constant Volume	$\frac{\text{ft-lbf}}{\text{lbm mole }^\circ\text{R}}$
d	Tank Diameter	ft
d'	Diameter	ft
D	Drag	lbf
DM	Change in Mass	lbm
DT	Change in Time	sec
DV	Change in Velocity	ft/sec
E	Oswald's Efficiency Factor	---
F	Thrust	lbf
GCLAT	Geocentric Latitude	deg
g_c	Unit Conversion Factor	$\frac{\text{lbm-ft}}{\text{lbf sec}^2}$
g_o	Acceleration due to Gravity	$\frac{\text{ft}}{\text{sec}^2}$
h	Cylindrical Tank Height	ft

Symbol	Definition	Units
H	Total Tank Height	ft
I_{sp}	Specific Impulse	sec
Isp_{eq}	Equivelent Specific Impulse	sec
JP4	Hydrocarbon Fuel	---
K	Drag Due to Lift Factor	---
$K_{B(W)}$	Interference Factor Based on Exposed Wing	---
$K_{W(B)}$	Interference Factor Based on Body	---
L	Lift	lbf
l_n	Length of Nose	ft
l_f	Length of Nose & Forebody	
LH_2	Liquid Hydrogen	---
LOx	Liquid Oxygen	---
LONG	Longitute	deg
m	Mass Flow Rate	lbm/sec
MAC	Mean Aerodynamic Chord	ft
M_b	Total Mass at Burnout	lbm
MDOT	Total Mass Flow	lbm/sec
M'	Total Mass	lbm
M_o	Mass of Oxidizer	lbm
M_p	Mass of Propellant	lbm
M	Mach Number	---
M^*	Mach Number at Throat	---
M_e	Mach Number at Exit	---
P	Pressure	psi

Symbol	Definition	Units
P_a	Ambient Pressure	psi
P_e	Exit Pressure	psi
P_t	Total Pressure	psi
P	Pressure Drop	psf
q	Heat Rate	BTU/sec
Q	Volumetric Flow Rate	gal/min
Q_R	Heat of Reaction	$\frac{\text{ft-lbf}}{\text{lbm}}$
r	Tank Radius	ft
r'	Fuel Ratio	---
R	Gas Constant	$\frac{\text{ft-lbf}}{\text{lbm mole °R}}$
RANG	Range	ft
R_u	Universal Gas Constant	$\frac{\text{ft-lbm}}{\text{lbm mole °R}}$
S	Planform Wing Area	ft^2
S_B	Maximum Frontal Area	ft^2
S_F	Projected Frontal Area	ft^2
SFC	Specific Fuel Consumption	lbm/lbf hr
SG	Specific Gravity	
S_{REF}	Reference Area	ft^2
S_S	Surface Body Area	ft^2
S_W	Total Wing Area	ft^2
S_{WET}	Wetted Area	ft^2
S_π	Frontal Area	ft^2
t	Time	sec

Symbol	Definition	Units
t/c	Thickness Ratio	---
T	Thrust	lbf
T'	Temperature	°R
TB	Time to Staging	sec
T_e	Exit Temp	°R
T_t	Total Temp	°R
U_e	Exit Velocity	ft/sec
V	Velocity	ft/sec
V'	Volume	ft^3
Ṿ	Tank Volume	ft^3
W	Weight	lbm
W_F	Weight of Fuel	lbm
W%	Percent of Gross Weight	---
$X_{ac'}/C_{rc}$	Aerodynamic Center Location of the Body Nose and Forebody	---
$(X_{ac'}/C_{rc})_{W(B)}$	Aerodynamic Center Location of the Exposed Wing in the Presence of the Body	---
Y	Altitude	ft
Z	Empirical, Nonlinear Normal Force Correction Factor	---
α	Angle of Attack	deg
	Ratio of specific Heats	---
μ	Density	lbm/ft3
β	Mach Number Interference Factor	---

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INTRODUCTION

The support of the United States' future space station and further space ventures prompts the need for an advanced space transportation system which can transport cargo or passengers efficiently, safely, and economically. The existing Space Transportation System consists of three reusable shuttle orbiters fueled by an expendable external tank and boosted by two reusable solid rocket boosters. While this system is adequate for our present space needs, an increase in mission frequency, mission diversity, and payload quantity requires a more economical and flexible system with a greater payload capability.

The boosters used in the present system descend into the ocean after staging. The expense of the retrieval and refurbishment of these boosters, along with the cost of replacing the external tank, adds substantially to the cost of each mission. When the number of missions is increased in the near future, lowering these expenses will be of utmost importance. A winged booster capable of flying back under its own power could be used for a number of different missions, thus cutting down on the cost of each mission.

With the construction and operation of a space station, a larger cargo capability will be needed than is available on the present shuttle. The creation of a new, unmanned cargo vehicle would substantially increase the quantity of cargo that could be transported to the station in one mission. The ability of the system to perform such "shipping" missions also increases its

flexibility. Also, by designing a vehicle capable of transporting cargo to an intermediate orbit, "orbital warehouses" would be created that could store supplies for later use.

With increased space activities, a larger manned orbiter with a larger return payload capability will be needed. An enlarged shuttle-type vehicle with a 15' X 60' cargo bay would improve the United States' ability to perform passenger transport and repair missions. This vehicle could be boosted into orbit by the same flyback booster used to boost the cargo vehicle, thus doubling the role of the winged booster.

The Advanced Space Transportation System described above would more readily fulfill the needs of America's space program in the future than the existing system. The technology required to construct and operate this system is readily available today. Also, the savings incurred by this system, along with the increased compatibility with the needs of future space projects, would more than make up for the its cost.

GROUNDRULES FOR FLYBACK BOOSTER STUDY

1) Two vehicles are to be developed:

   * Manned shuttle II
   * Unmanned cargo vehicle

2) Both vehicle will be 2-stage vehicles utilizing the same fully reusable, winged, flyback booster.

3) The payload capability for each vehicle will be as follows:

   * Shuttle II - 40 Klb. to space station orbit (28.5° inclination / 270 n.m. altitude) with 40 Klb. return to earth
   * Cargo vehicle - 125-150 Klb. to 28.5° / 150 n.m.

4) The payload envelope (bay) for each vehicle is

   * Shuttle II - 15'D X 60'L
   * Cargo vehicle - 25'D X 90'L and 33'D X 100'L

5) Engine propellants:

   * 1st stage - liquid oxygen / hydrocarbon
   * 2nd stage - liquid oxygen / liquid hydrogen

6) Staging velocity - maximum of 7,000 fps

7) The manned shuttle II will see a maximum of 3 g's

8) The cargo vehicle will be ready for its first flight in 1998 therefore technology and design freeze approximately 1990

9) The manned shuttle II will be ready for its first flight in 2005; therefore technology and design freeze approximately 1997

TRAJECTORY ANALYSIS

The following analysis examines the complete trajectory of the flyback booster. In a general overview, the trajectory is broken into two phases. These phases include the trajectory to staging and the return trajectory.

Trajectory to Staging

The preliminary design of the Advanced Space Transportation System (ASTS) is largely contingent on data related to the system's trajectory to staging. Trajectory analysis allows the designer to generate various parameters essential for estimating the system's thrust, mass and structural requirements. The most important of these parameters are those associated with the data gathered at staging. This data consists of the mass, velocity, altitude and time to staging. From this data the required propulsion parameters can be determined. It is then possible to calculate the mass and volume of the propellant burned. Ultimately, the dimensions of the structure can be computed from the size requirements of the fuel and oxidizer tanks.

As in any preliminary design project, it is necessary for the designer to follow certain groundrules and to make various assumptions. The groundrules and assumptions allow the designer to generate data that can then be studied for validity. For the ASTS, the groundrules are listed on page 3

and the assumptions pertaining to the trajectory analysis are listed below:

*T/W=1.35

*3 G's acceleration limit

*Gravity turn into orbit

*Inertial and relative frames used

*Drag neglected during boost phase

*Parallel burn

*Crossfeed propellant from booster to 2nd stage

*Thrust of both stages is constant

*Booster characteristics

-LOx/JP4 propellant

-Isp=320 sec

-LOx/JP4 ratio=2.3

*2nd Stage characteristics

-LOx/LH2 propellant

-Isp=380 sec

-LOx/LH2 ratio=6.0

Some of the assumptions require further examination. The thrust to weight ratio was chosen to help maintain a G level below 3.0 due to the human element involved. The parallel burn allows the system to achieve the required thrust level without putting the entire thrust demand on the booster. The propellant crossfeed system eliminates excess structural weight which the second stage would otherwise have to carry into orbit. Finally, the hydrocarbon fuel JP4 was chosen

because it can be used in the liquid rocket engines as well as the air breathing engines.

The G turn is used to aid the second stage in achieving the required relative angle of 90 degrees for a circular orbit at a particular altitude. Figure 1 shows how the ASTS is tracked in a relative and inertial reference frame during its flight. Notice how the range and altitude are measured.

Coupled with the assumptions and groundrules, a variety of trajectory, propulsion and mass equations are incorporated into a BASIC program. Program 1, listed in Appendix B, is similar to the one used in the preliminary report, however the G turn is incorporated. A time increment of one second is used to determine all the parameters during the flight. The velocity restriction of 7000 feet per second at staging, together with the time increment, enables the designer to determine the time at which staging occurs.

Three variables are entered at the start of the program; the total mass of the ASTS, the altitude for the G turn to begin, and the liftoff thrust of the second stage. From this data the total thrust and the thrust of the booster is dictated. Dividing the thrust of each stage by its respective specific impulse yields the mass flow of each stage. In order to determine the mass flow for the ASTS, an equivalent specific impulse is calculated by,

$$Ispeq=T/(MDOT_1+MDOT_2) \qquad (1)$$

then the total mass flow is given by,

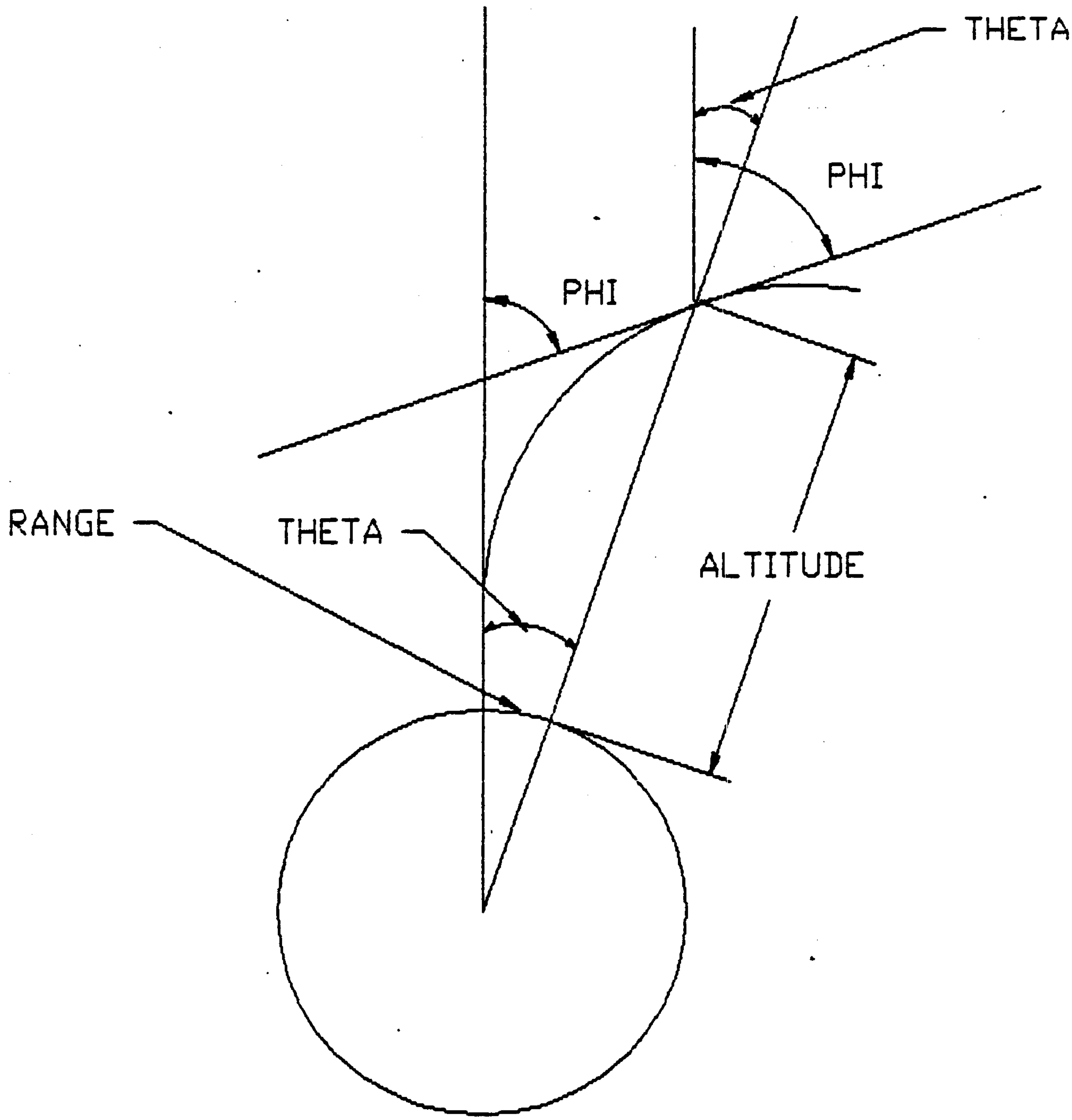


FIGURE 1. TRAJECTORY

$$MDOT=T/Ispeq \qquad (2)$$

In order to calculate the change in velocity during each time interval, the new total mass at each interval must first be calculated by,

$$DM=M-MDOT_1*DT-MDOT_2*DT \qquad (3)$$

Using this result, the change in velocity can be determined by,

$$DV=Ispeq*G*\log(M/DM)-G*DT \qquad (4)$$

where G is a function of altitude and the inertial angle θ. Once the change in velocity has been determined, the altitude can be found using,

$$Y=Y+.5*(2v+DV)*DT \qquad (5)$$

The range is given by the relationship,

$$RANG=RE*\theta \qquad (6)$$

where RE is the radius of the earth.

By using the above relations in a stepwise time interval loop, the parameters at staging (V=7000 fps) can be determined. From the time to staging parameter (TB) it is possible to find the total mass of propellant burned to staging by the equation,

$$Mp=MDOT*TB \qquad (7)$$

It follows that the total mass at staging is given by,

$$Mb=M-Mp \qquad (8)$$

It is also possible to examine the stages individually. The respective propellant masses are given as follows:

$$Mp_1=MDOT_1*TB \qquad (9)$$

$$Mp_2=MDOT_2*TB \quad (10)$$

where the subscripts 1 and 2 correspond to the stage number. The masses of the fuels and oxidizer are then calculated with the use of the equation,

$$Mp=MOx+Mf \quad (11)$$

together with the oxidizer to fuel ratios of 2.3 for the booster and 6.0 for the second stage. Putting these relations together yields:

$$Mf_1=Mp_1/3.3 \quad (12)$$

$$MOx_1=Mp_1-Mf_1 \quad (13)$$

$$Mf_2=Mp_2/7.0 \quad (14)$$

$$MOx_2=Mp_2-Mf_2 \quad (15)$$

It follows that the volumes of the propellants can be determined by dividing the various propellant masses by their respective densities.

Using Program 1 and the above relations the results listed in Table 1 were obtained for the booster with shuttle II configuration. A similar running of Program 1 for the booster with the cargo vehicle yielded the results in Table 2. It is evident that for both configurations, the structural weight of the booster remains a constant. Due to the extremely small density values encountered by the ASTS at staging, some problems for the return trajectory of the booster were created. This problem could probably be solved by incorporating a perigee injection trajectory. This is a problem to be examined in further detail in the future.

FYLBACK BOOSTER WITH SHUTTLE II

GENERAL INFORMATION

DENSITY OF OXIDIZER (O_2) = 71.2 lbm/ft
DENSITY OF SHUTTLE II FUEL (H_2) = 4.4 lbm/ft
DENSITY OF BOOSTER FUEL(JP4) = 47.299 lbm/ft

ISP OF SHUTTLE II = 380 sec
ISP OF BOOSTER = 320 sec
EQUIVILENT ISP = 325.1 sec

OXIDIZER TO FUEL RATIO FOR SHUTTLE II = 6.0
OXIDIZER TO FUEL RATIO FOR BOOSTER = 2.3

TOTAL MASS ON LAUNCH PAD = 7,400,000 lbf
TOTAL THRUST OF BOOSTER AND SHUTTLE II = 9,990,000 lbf
THRUST OF SHUTTLE II = 1,000,000 lbf
THRUST OF BOOSTER = 8,990,000 lbf

ALTITUDE OF G-TURN = 2,625 ft

STAGING DATA

TIME TO STAGING = 167 sec
VELOCITY AT STAGING = 7,076 ft/sec
ALTITUDE AT STAGING = 344,676 ft

MASS OF SHUTTLE II AT STAGING = 1,292,850 lbm
MASS OF BOOSTER AT STAGING
(STRUCTURE + RETURN FUEL) = 1,112,810 lbm

MASS FLOW OF SHUTTLE II = 2,623 lbm/sec
MASS FLOW OF BOOSTER = 28,094 lbm/sec

MASS OF BOOSTER PROPELLANT BURNED = 4,691,698 lbm
MASS OF SHUTTLE II PROPELLANT BURNED = 439,544 lbm
MASS OF BOOSTER OXIDIZER BURNED = 3,269,971 lbm
MASS OF BOOSTER FUEL BURNED = 1,421,727 lbm
MASS OF SHUTTLE II OXIDIZER BURNED = 376,752 lbm
MASS OF SHUTTLE II FUEL BURNED = 62,792 lbm

TABLE 1. Shuttle II Configuration

FYLBACK BOOSTER WITH CARGO VEHICLE

GENERAL INFORMATION

DENSITY OF OXIDIZER (O_2) = 71.2 lbm/ft
DENSITY OF CARGO VEHICLE FUEL (H_2) = 4.4 lbm/ft
DENSITY OF BOOSTER FUEL(JP4) = 47.299 lbm/ft

ISP OF CARGO VEHICLE = 380 sec
ISP OF BOOSTER = 320 sec
EQUIVILENT ISP = 327.7 sec

OXIDIZER TO FUEL RATIO FOR CARGO VEHICLE = 6.0
OXIDIZER TO FUEL RATIO FOR BOOSTER = 2.3

TOTAL MASS ON LAUNCH PAD = 7,571,000 lbf
TOTAL THRUST OF BOOSTER AND CARGO VEHICLE = 10,220,850 lbf
THRUST OF CARGO VEHICLE = 1,531,000 lbf
THRUST OF BOOSTER = 8,689,850 lbf

ALTITUDE OF G-TURN = 1,800 ft

STAGING DATA

TIME TO STAGING = 164 sec
VELOCITY AT STAGING = 6,987 ft/sec
ALTITUDE AT STAGING = 281,390 ft

MASS OF CARGO VEHICLE AT STAGING = 1,491,380 lbm
MASS OF BOOSTER AT STAGING
(STRUCTURE + RETURN FUEL) = 1,112,810 lbm

MASS FLOW OF CARGO VEHICLE = 4,029 lbm/sec
MASS FLOW OF BOOSTER = 27,156 lbm/sec

MASS OF BOOSTER PROPELLANT BURNED = 4,453,584 lbm
MASS OF CARGO VEHICLE PROPELLANT BURNED = 660,756 lbm
MASS OF BOOSTER OXIDIZER BURNED = 3,104,013 lbm
MASS OF BOOSTER FUEL BURNED = 1,349,571 lbm
MASS OF CARGO VEHICLE OXIDIZER BURNED = 566,362 lbm
MASS OF CARGO VEHICLE FUEL BURNED = 94,394 lbm

TABLE 2. Cargo vehicle Configuration

Notice that the staging parameters for both configurations are quite similar. This makes the next phase of the trajectory analysis much easier. The return trajectory of the booster will be examined for the shuttle configuration only. However, the results for both configurations are listed for comparision in Tables 4 and 5 on page 24 and 25. Again the results of this analysis would be similar for both configurations.

Return Trajectory

The return trajectory of the flyback booster was simulated using the fortran program DUAL, listed in Appendix B as Program # 2. The program was acquired by Auburn University from Marshall Space Flight Center. This very powerful program is outlined in the Users Manual, which accompanies this report.

As in the trajectory analysis to staging, certain groundrules and assumptions must be examined during this phase of the flight. The assumptions are as follows:

*95 percentile winds at 280 deg azimuth

*MFSC flyback booster aerodynamics

*Wing reference area=15500 sq ft

*Cruise engine start at 20000 ft

*High cruise altitude=10000 ft

*Low cruise altitude=1000 ft

*500 ft/min descent to 1000 ft

*End trajectory within 1 N.MI. of KSC

*Maximum angle of attack=70 deg

*Cruiseback angle of attack=6 deg

*Sea level thrust per air breathing engine=62500 lb

*Engine cantor=-6 deg

*Launch/Landing at KSC

The MSFC aerodynamics are used because the configuration of the booster is very similar to MSFC's booster. The wing reference area is determined such that the wing loading does not exceed 70 pounds per square foot. The maximum angle of attack is related to the angle at which the most drag is experienced by the booster during reentry. Finally, The cruiseback angle of attack is determined by the angle which creates the maximum lift to drag ratio.

The staging data which is of importance in the program DUAL is as follows:

*Booster weight (including fuel)=1,112,810 lbm

*GCLAT=28.46 deg   LONG=79.81 deg

*Relative velocity=7076 ft/sec

*Altitude=344676 ft

*Relative azimuth=93.91 deg

*Relative flight path angle=53.6 deg

*TB=167 sec

All of this data is input into the program DUAL through an input file called ZFLY.DAT which is listed in Appendix B. Once this data has been input, the program can run. In this

manner the entire trajectory from staging to the return to KSC can be examined.

The return trajectory is broken into eight phases by the program DUAL. These phases are outlined in Figure 2. The different phases after separation include coast to apogee, coast to reentry, altitude descent rate control, 45 degree bank turn, engine start, idle-engine descent, high cruise, letdown to low cruise, low cruise and arrival. During reentry, the altitude control phase begins when the dynamic pressure is sufficient to obtain a constant rate of descent of -100 feet per second. This is done by modulating the bank angle. The constant rate of descent down to low cruise is obtained by throttling back the cruise engines. Also included in this analysis is the capability of flying back with one engine out.

The program DUAL had to be run numerous times before acceptable results were obtained. The problems encountered included excessive G limits, inability to achieve high cruise, insufficient thrust levels, and high Mach number when engines are started. All of these problems were solved by adjusting the angles of attack, imposing a dynamic pressure limit, increasing the thrust level, and lowering the altitude when the engines are started. The results of the program are listed in Appendix B. Table 3 shows the values of various parameters during the eight phases of the return trajectory.

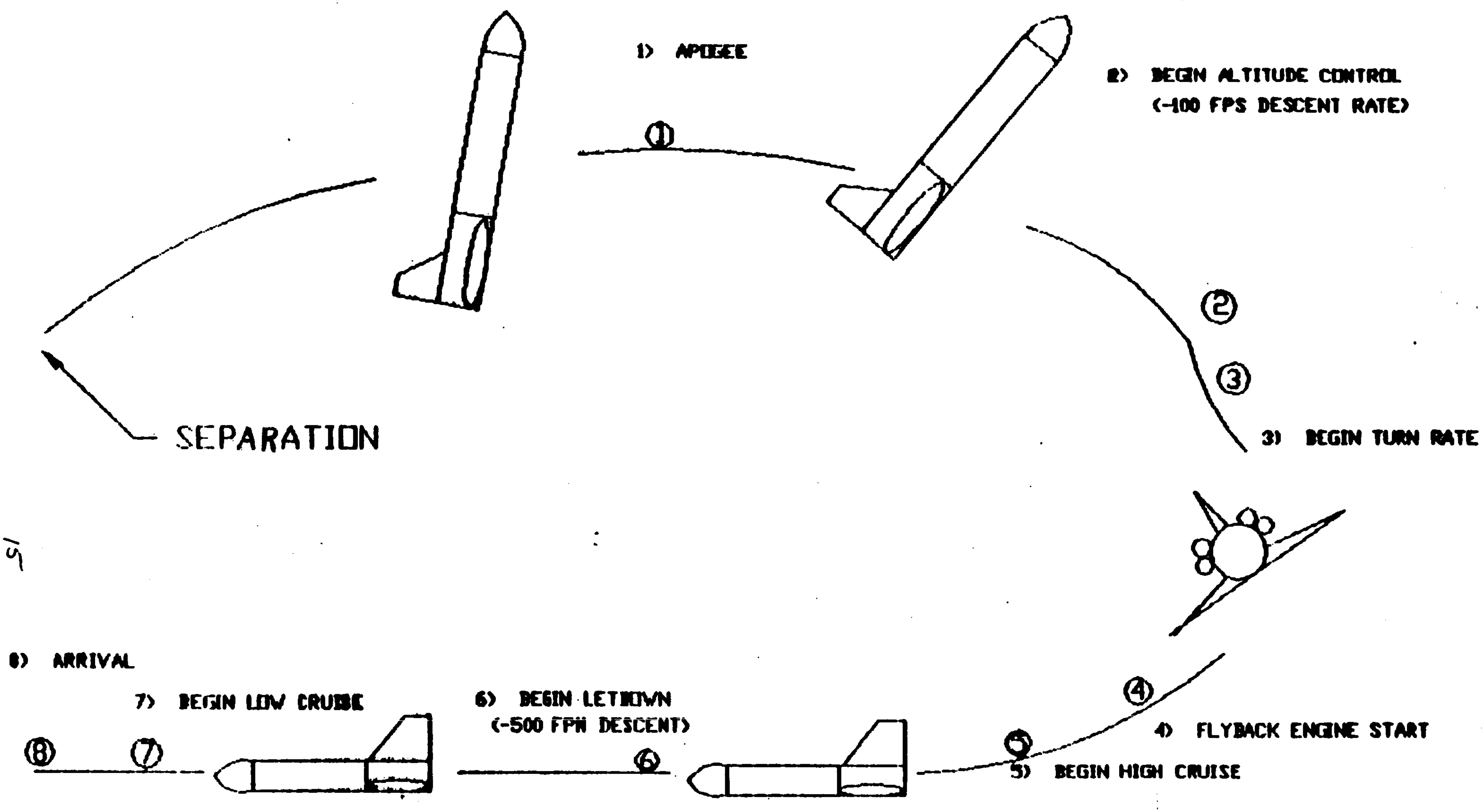


FIGURE 2. FLIGHT PROFILE

SHUTTLE II MISSION

	PHASE	t sec	h ft	V fps	M	α deg	β deg
	Separation	167	344,676	7076	7.0	0.0	0.0
1	Apogee	367	905,151	4019	1.47	70.0	0.0
2	Alt control	629	34,102	2683	2.69	0.0	0.0
3	begin turn	637	27,771	1799	1.75	0.0	28.7
4	engines on	667	20,049	797	0.75	6.0	45.0
5	high cruise	839	10,113	472	0.43	6.0	0.0
6	letdown	5889	5,526	420	0.38	6.0	0.0
7	low cruise	6483	1,060	391	0.35	6.0	0.0
8	arrival	6738	933	389	0.34	6.0	1.2

Table 3 : Return trajectory data for booster

It is important to examine the maximum values of some of the parameters throughout the flight. The parameters of interest are as follows:

Max QDOT	29.428	btu/ft²/s	TIME	613 sec
Max DYNP	3674	lb/ft²	TIME	621 sec
Max ACC	8.131	g's	TIME	603 sec
Max T	205934	lb_f	TIME	2419 sec
TOTAL FLIGHT TIME		1 hr 53 min 18 sec		

The time specificed for each parameter above is from launch to when the event occurs. It is evident that the dynamic pressure and the G level are both quite high. This is a result of the size of the booster and the high altitude at which staging occurs. The booster accelerates very quickly as it reenters the atmosphere and experiences large forces as it tries to deccelerate before entering the bank turn. As the designer becomes more familiar with the program DUAL these maximum values may be reduced by imposing additional limitations. Regarding the thrust requirements, the program was used to size the engines and determine the number of engines required for flight. Four 62500 pound engines are utilized. The engines reach a maximum throttle of 85 percent, and can therefore maintain an engine out capability if needed. The total fuel requirement for the four engines is 136866 pounds of JP4.

To achieve an overview of the entire flight, five graphs are included, which plot various parameters versus time.

Figure 3 shows altitude versus time. Notice that the booster reaches an altitude of 905151 feet at apogee. Figure 4 shows the mach number versus time. The mach number increases from apogee but then is decrease during the controlled rate of descent. Figure 5 shows the acceleration profile throughout the flight in terms of G's. Figure 6 is a plot of the dynamic pressure during the flight. Notice that it is a maximum during the controlled rate of descent and then decreases to nearly a constant. Finally, Figure 7 shows the thrust levels achieved versus time.

In conclusion, it should be evident that the booster can be flown back to its launch site, assuming that the structure can be built to withstand the large forces it will encounter during the critical controlled rate of descent. The booster will not be manned as originally planned, due to the high accelerations experienced after separation.

ALTITUDE VERSUS TIME

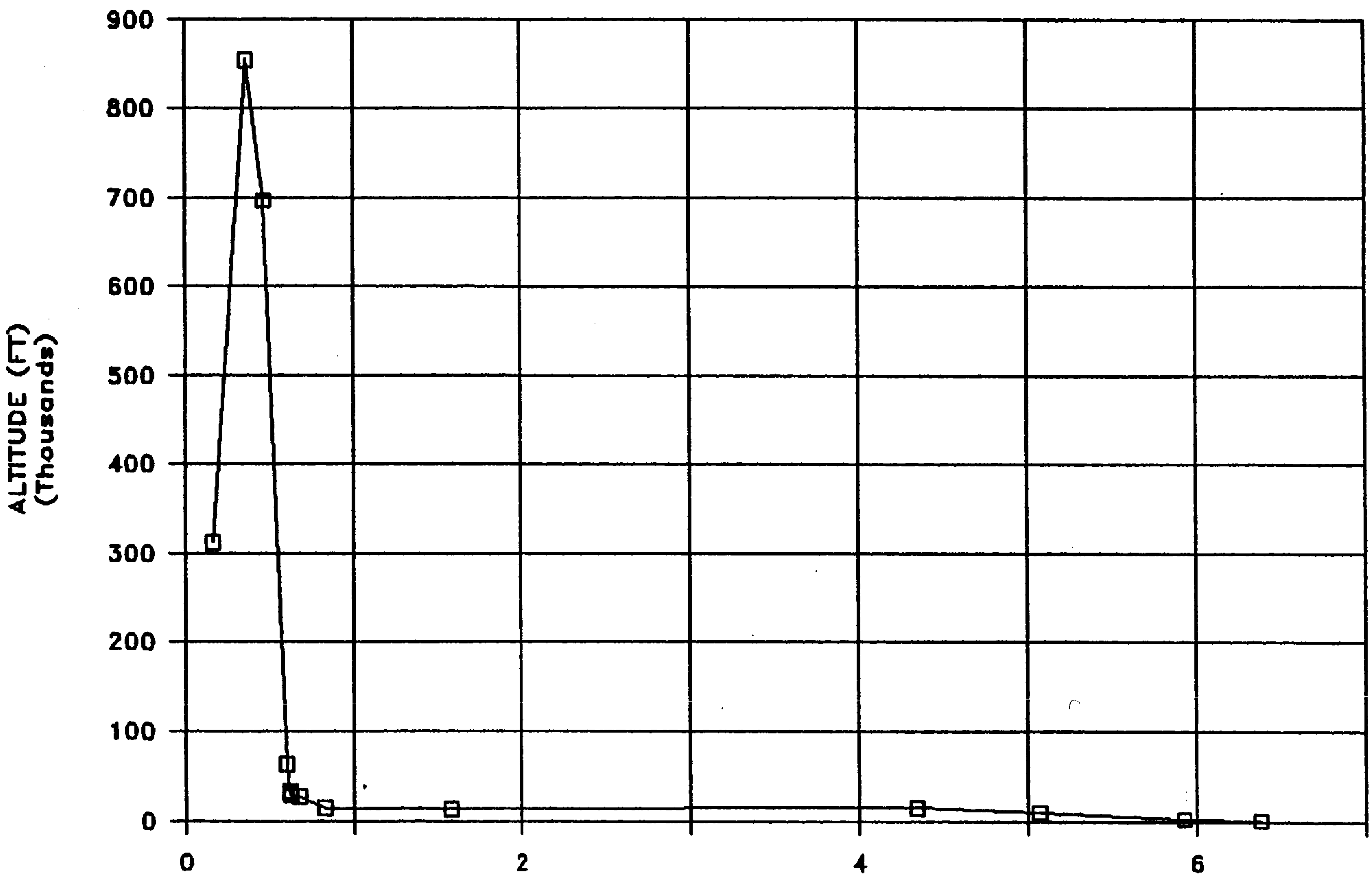


FIGURE 3 ALT vs t

MACH NUMBER VS TIME

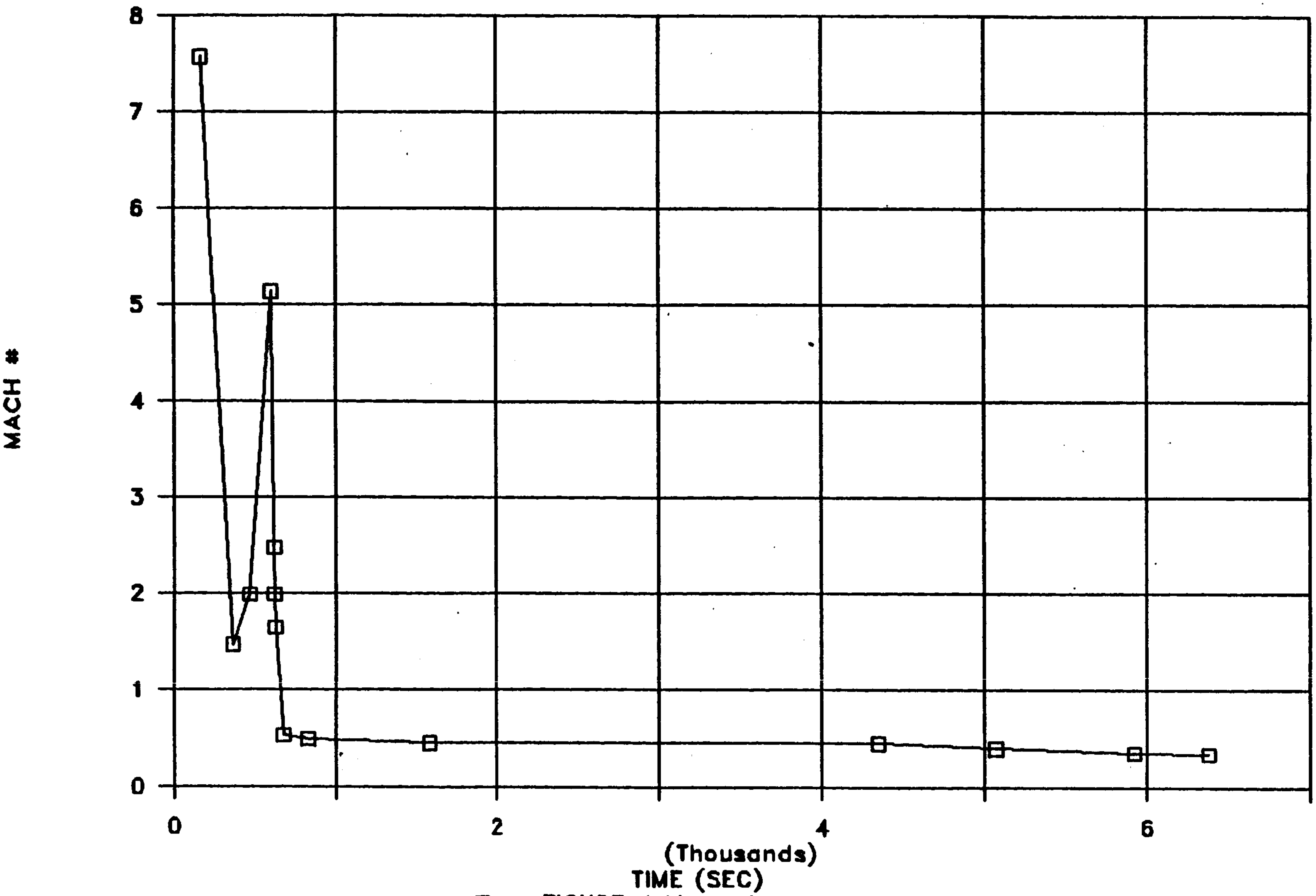
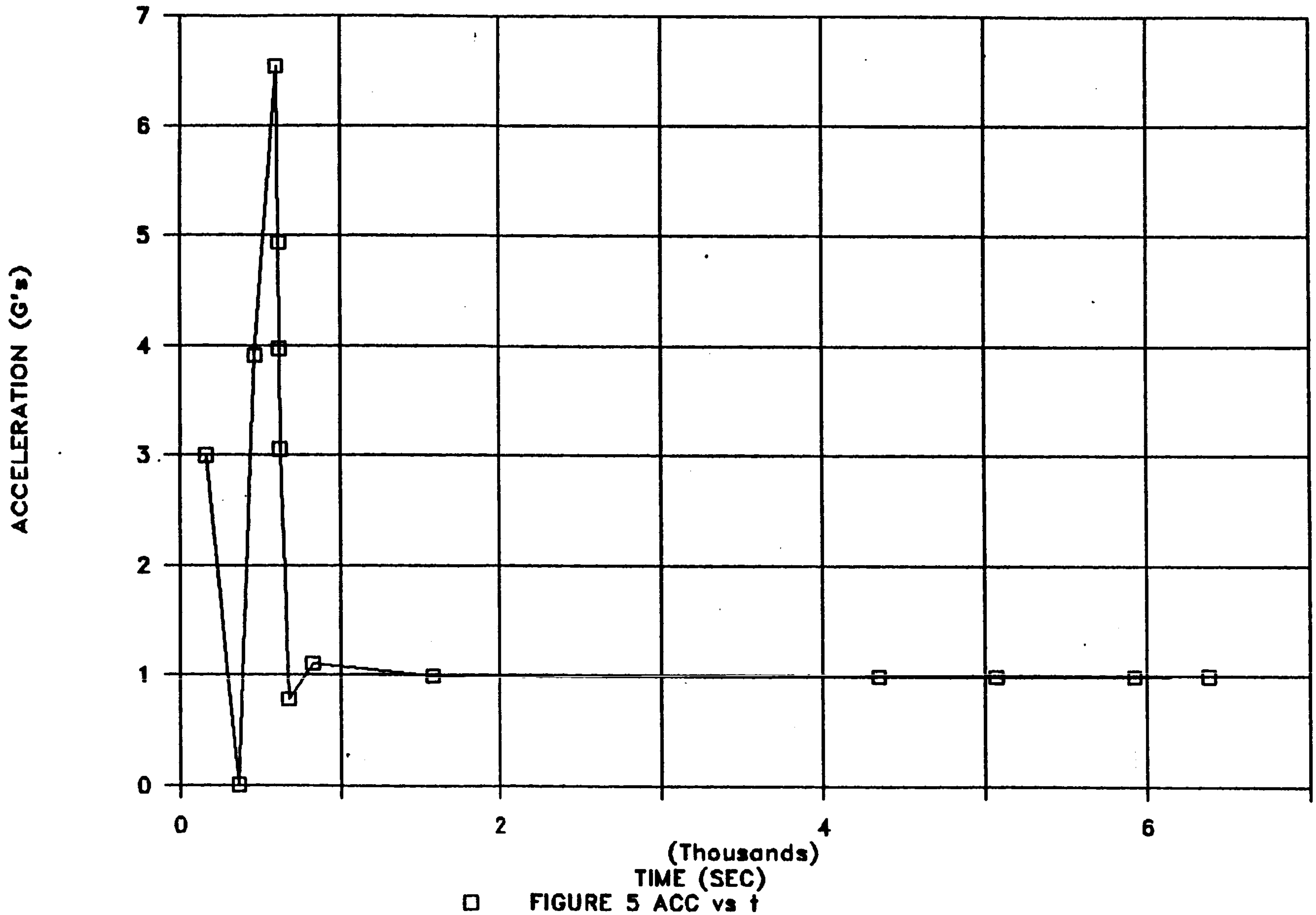
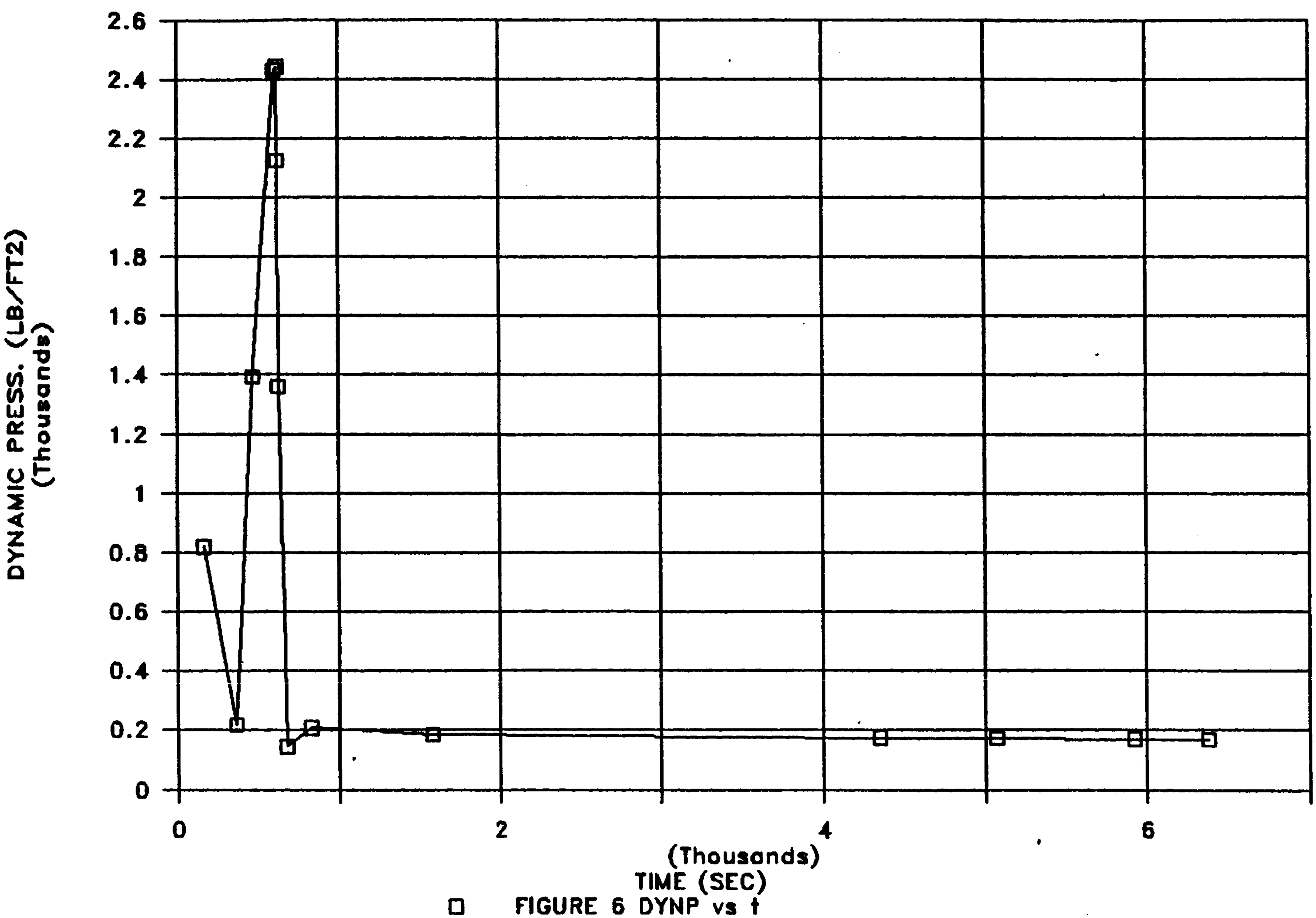


FIGURE 4 M# vs t

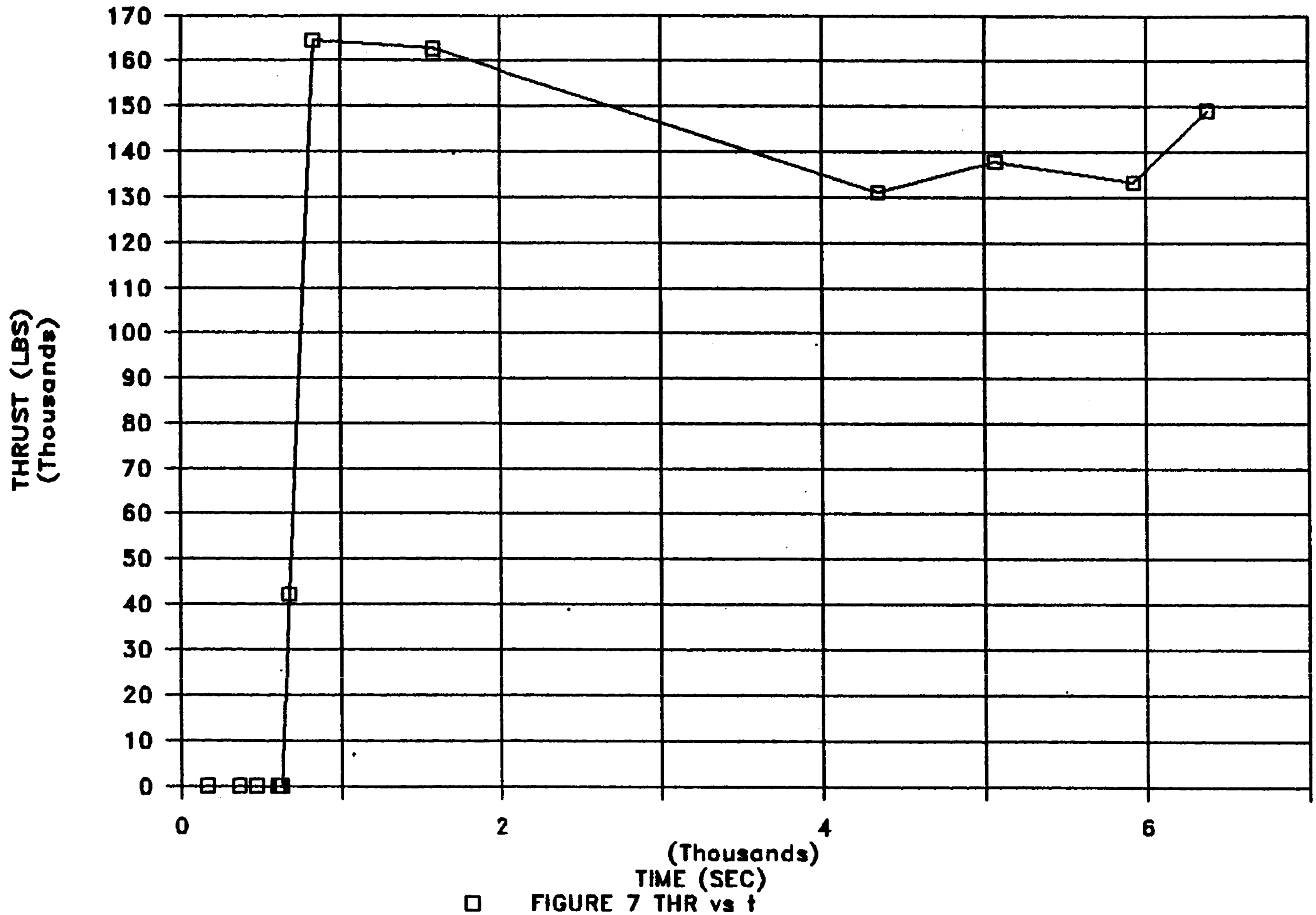
ACCELERATION VERSUS TIME



DYNAMIC PRESSURE VERSUS TIME



THRUST VERSUS TIME



SHUTTLE II MISSION

	PHASE	t sec	h ft	V fps	M	α deg	β deg
	Separation	167	344,676	7076	7.0	0.0	0.0
1	Apogee	367	905,151	4019	1.47	70.0	0.0
2	Alt control	629	34,102	2683	2.69	0.0	0.0
3	begin turn	637	27,771	1799	1.75	0.0	28.7
4	engines on	667	20,049	797	0.75	6.0	45.0
5	high cruise	839	10,113	472	0.43	6.0	0.0
6	letdown	5889	5,526	420	0.38	6.0	0.0
7	low cruise	6483	1,060	391	0.35	6.0	0.0
8	arrival	6738	933	389	0.34	6.0	1.2

CARGO VEHICLE MISSION

	PHASE	t sec	h ft	V fps	M	α deg	β deg
	Separation	164	281,390	6987	10.1	0.0	0.0
1	Apogee	360	823,557	3973	1.49	70.0	0.0
2	Alt control	608	33,848	2766	2.77	0.0	0.0
3	begin turn	618	25,703	1660	1.60	0.0	22.7
4	engines on	646	18,267	797	0.74	6.0	45.0
5	high cruise	800	10,055	467	0.42	6.0	0.0
6	letdown	5620	5,438	421	0.38	6.0	0.0
7	low cruise	6204	1,053	393	0.35	6.0	0.0
8	arrival	6454	932	391	0.34	6.0	1.2

Table 4 : Return trajectory data for booster for both missions

SHUTTLE II MISSION

PARAMETER	VALUE	UNITS	FLIGHT TIME OCCURRED
Max ACCL	8.131	g's	603 sec
Max QDOT	29.428	$btu/ft^2/s$	613 sec
Max STMP	2310	°F	613 sec
Max DYNP	3674	lb/ft^2	621 sec
Max T	205,934	lb_f	2419 sec

FLYBACK FUEL 136,866 lb_m

TOTAL FLIGHT TIME 1 hr 53 min 18 sec

CARGO VEHICLE MISSION

PARAMETER	VALUE	UNITS	FLIGHT TIME OCCURRED
Max ACCL	7.278	g's	594 sec
Max QDOT	29.113	$btu/ft^2/s$	592 sec
Max STMP	2303	°F	592 sec
Max DYNP	3764	lb/ft^2	602 sec
Max T	205,978	lb_f	2400 sec

FLYBACK FUEL 131,266 lb_m

TOTAL FLIGHT TIME 1 hr 47 min 34 sec

Table 5 : Return trajectory for booster for both missions

STRUCTURES

Due to the concept of the booster burning in parallel with the other two vehicles and having to carry the fuel and oxidizer for the same vehicles plus its own propellants during the ascent portion of the boost, the booster needed to be built around the propellant tanks. Since the booster and other vehicle would be using the same oxidizer this tank could be shared. The preliminary design dictated the use of a three tank booster containing LH_2, JP_4 and LO_2. From the given data, the following tank volumes were found:

1) $LH_2 = 21{,}454\ Ft^3$
2) $JP_4 = 32{,}952\ Ft^3$
3) $LO_2 = 51{,}551\ Ft^3$

The three tanks were designed to be cylindrical tanks with semi-prolate spheroid end caps. Originally spherical end caps were being used but by incorporating the semi-prolate spheroids the total tank height could be reduced thus decreasing the overall height of the booster. The volume of each tank is given by equation 1.

$$V=4/3\pi rb^2 + \pi r^2h \qquad (1)$$

Where r is the radius of the inside of the tank, b is the length of the semi-minor axis and h is the height of the cylindrical part of the tank. Knowing the volumes needed in each tank, equation one can be rearranged to give the overall tank height once a radius and semi-minor axis has been chosen. The total tank height H can be found using equation 2.

$$H = \frac{V - 4/3\pi rb^2}{\pi r^2} + 2b \qquad (2)$$

Equation 2 was used to find the heights of all three propellant tanks.

A radius of 18 feet was used so that the booster would be approximately the same diameter as the Shuttle and cargo vehicles. The height of the semi-minor axis was determined to be six feet. The tank heights were then calculated as follows: The height of the LH_2 tank is 31 feet, the height of the LO_2 tank is 60 feet and the height of the JP_4 is 42 feet. These tank heights were calculated using the maximum volumes of fuel used for any mission plus all flyback fuel needed.

The remaining heights of the booster sections were then determined. The three fuel tanks are placed with the LH_2 tank in the forward position, the LO_2 tank in the middle position and the JP_4 tank being in the rear position near the boosters main engines. The nosecone section of the booster will house the avionics needed to run all of the systems on the booster. This section will be 33 feet long. Second, the structural area behind the front tank will be used as attaching points for the canards, the air breathing engines and the mounts for the accompanying vehicle. Twenty feet of structure was needed to incorporate all of these attaching structures. Ten feet of

structure was placed between the LO_2 and JP_4 tanks for plumbing and insulation reasons. Forty eight feet of was need behind the JP_4 tank for the engine thrust structure, and aft attaching points for the vehicles. A diameter of 45 feet was need around the main engines therefor a firing was placed around the engines. The overall height of the booster came out to be 244 feet with a diameter of 36 feet and a firing of 45 feet in diameter placed around the engines and aft thrust structure. Figure 14 shows a three view drawing of the booster.

The booster system will be determined by trends and load analysis used on the X-15, the Saturn and the Space Shuttle programs. The ground rules for the analysis stated that no thermal protection will be used. The concept of a "Heat Sink" will be based upon the use of new high-temperature metals. Initial temperatures set for a warm windy day are as follows:

-50 Deg F Liquid Hydrogen Tank Wall
0 Deg F LOX Tank Wall
120 Deg F Other Surfaces

The maximum allowable temperatures are as follows:

350 Deg F Aluminum Lithium
800 Deg F Titanium (Load Structures)
1000 Deg F Titanium (Non-Load Structures)
1200 Deg F Inconel X

The basic concept of the heat sink booster is to allow the booster to act as a sink to store heat as it is generated

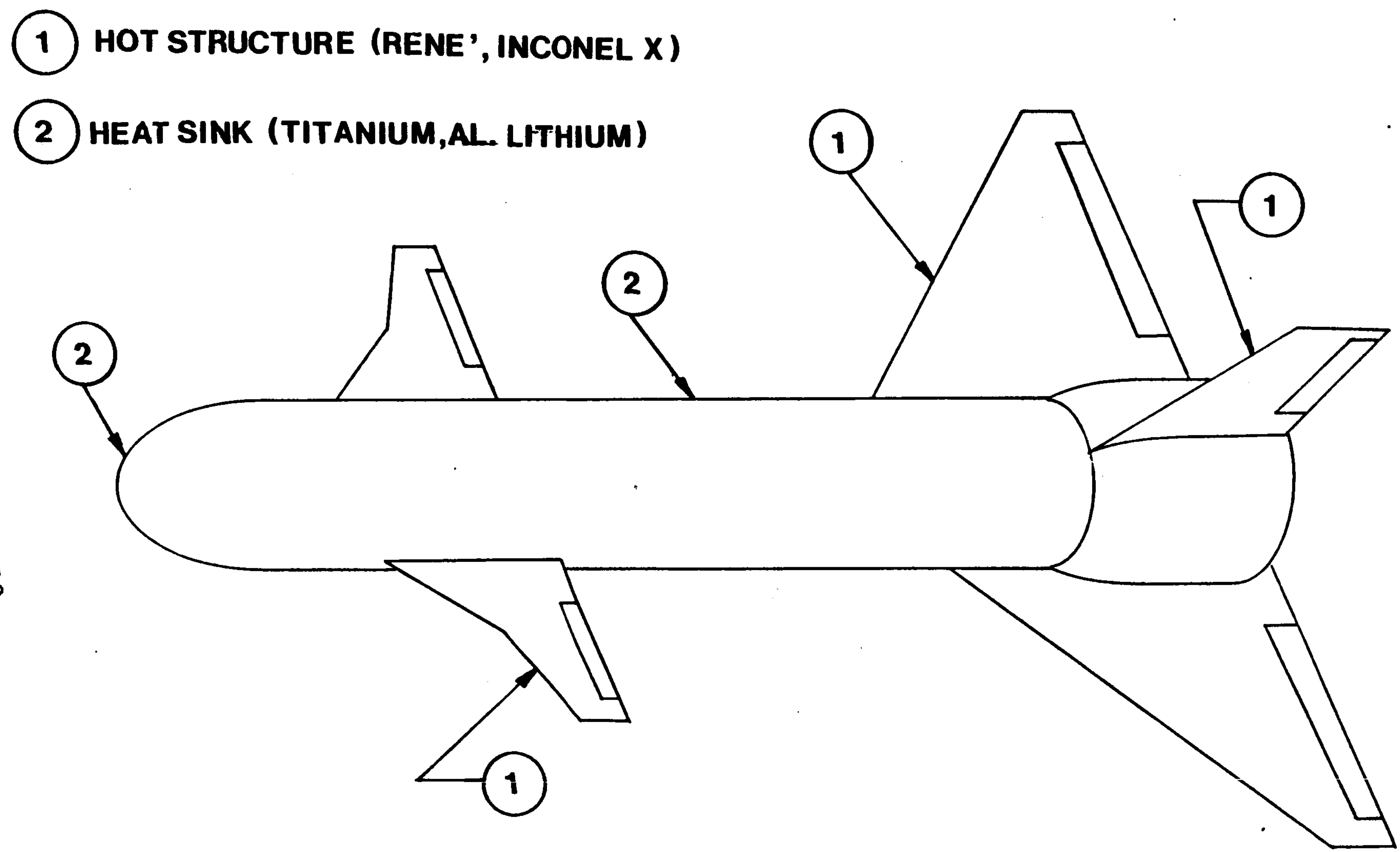


FIGURE 8. Material Locations

during ascent and reentry. The location of these materials on the booster are shown in figure 8.

The basic structure is a semi-monocoique design, in which the primary loads are carried in the external skin of the fuselage. The fuselage skin also forms the outer shell of the propellant tanks. Thus, it must withstand the stresses from the propellant weight as well as from internal tank pressurization.

The primary structure design will be aluminum skin and concentric stringer shells which are straight columns tied together by skin and frame. The tanks should be separated by a honeycomb construction with aluminum face sheets and thermal barriers. The nose will be constructed of an aluminum honeycomb structure capable of withstanding the high aerodynamic pressures of reentry. The shell has to be designed to withstand external pressures introduced by shockwaves. The propellant tank walls also have to be designed to resist the following loads due to supporting mass, drag loads, bending compressive loads, engine thrust and internal tank pressure loads. Assuming maximum shear loads and maximum bending moments to be at liftoff prerelease and at maximum q(alpha) where the dynamic pressure is greatest. The required wall thickness and stiffness can be established form these maximum loads.

For further studies, the wings, fin, and canards will be analyzed. However the fuselage, which makes up the majority of the booster surface and heat sink weight was analyzed in this study.

AERODYNAMICS AND DESIGN OF THE FLYBACK BOOSTER

Further changes in design required the recalculation of the aerodynamic and control characteristics. The main changes in the design were an increase in overall size, the addition of one rocket engine, and the addition of a faring over the rocket engines. The aerodynamic and control characteristics calculated at subsonic speeds include the location of the aerodynamic center, the static margin, CL, CD and CM versus alpha. At supersonic speeds CL and CD versus alpha at different Mach numbers were determined.

DESIGN

It is necessary for the booster vehicle to carry three propellant tanks, avionics equipment, mating structures, internal supports and other necessary components. Upon sizing the components and assembling them, the final booster length is 244 feet. The diameter of the booster is 36 feet. In addition, a faring with a diameter of 45 feet was placed around the rocket engines.

The final configuration is shown in figure 9. Some of the important design dimensions are:

Overall Length	= 244 feet
Wing Area	= 7,161 feet2
Canard Wing Area	= 625 feet2
Tail Area (wet)	= 1926 feet2
Cross Sectional Body Area	= 1017.9 feet2

Diameter of Body	= 36 feet
Wing Span (main)	= 175 feet
Wing Span (canard)	= 106 feet
Air Foil	= NASA 0012 64
Thickness ratio	= 12%
Sweep Angle (LE)	= 45
Aerodynamic Center	= 45.055% of the chord
Aspect Ratio	= 5.04
Roott Chord	= 85 feet
Tip Chord	= 8 feet
Taper Ratio	= .094
Mean Aerodynamic Chord	= 47.5 feet

SUBSONIC AERODYNAMICS

CL, CD, and CM versus alpha were calculated for this configuration. The aerodynamic center and static margin were also calculated. The method for finding the aerodynamic center is outlined in Appendix A. The aerodynamic center was calculated to be 45.055 percent of the wing's mean aerodynamic chord. The static margin on the pad, before staging, after staging, and at landing are listed in Table 6.

Table 6

Static Margin in Percent Mean Aerodynamic Chord

Fueled For	At Lift-off	At Staging	At Landing
Cargo Vehicle	1.025	0.7324	0.8714
Shuttle II	1.006	0.7324	0.8758

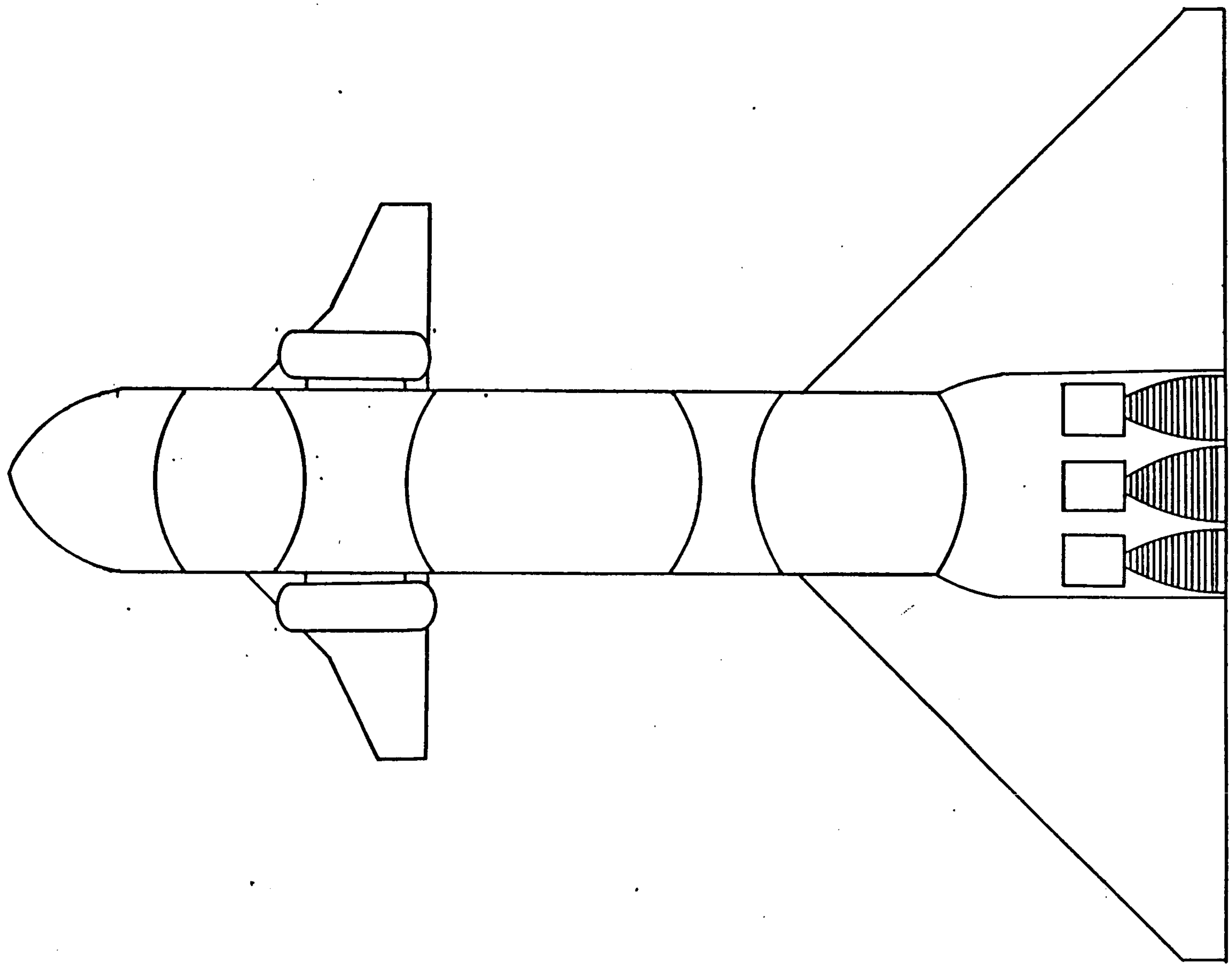


FIGURE 9. Top View

These values were calculated using the c.g. values found in the weights section.

The method of calculating pitching moment coefficient with angle of attack was determined using the Air Force's DATCOM manuals. The major equations used in this method are included in Appendix A. The values of CM for the six different configurations are plotted versus alpha in Figure 10. These plots have negative slopes and cross the x-axes at positive angles of attack; therefore the configurations are stable and trimable.

In order to find the subsonic lift and drag coefficients of the booster at various angles of attack, it was necessary to write a computer program (shown in Appendix B). The equations used in the program are referenced from Jan Roskam's Methods For Estimating Drag Polars of Subsonic Airplanes. (See Appendix A for the major equations used in the subsonic aerodynamics program.)

The calculations of CL and CD were made starting at a Mach number of 0.7 and an altitude of 35,000 feet. This is the point at which the jet engines are engaged. These values were calculated at angles of attack ranging from -5 degrees to +19 degrees in 2 degree increments.

The following assumptions were necessary in writing the program:

(1) Mach 0.7 was maintained from 35,000 feet (the approximate height were the returning booster

reaches mach 0.7) down to 15,000 feet (the altitude at which the booster will cruise to its destination).

(2) The weight change at this point in the trajectory is negligible.

(3) The density is based on the standard atmosphere.

(4) The wave drag coefficient is negligible at Mach 0.7.

(Additional assumptions are listed in Appendix A)

The results obtained for CL versus alpha and CD versus CL are listed in Table 7. These values are graphed in Figures 11. The maximum lift to drag ratio was found to be 5.211 at an angle of attack of 6.0 degrees. Therefore, the suggested angle of attack for the cruise back of the booster is 6.0 degrees.

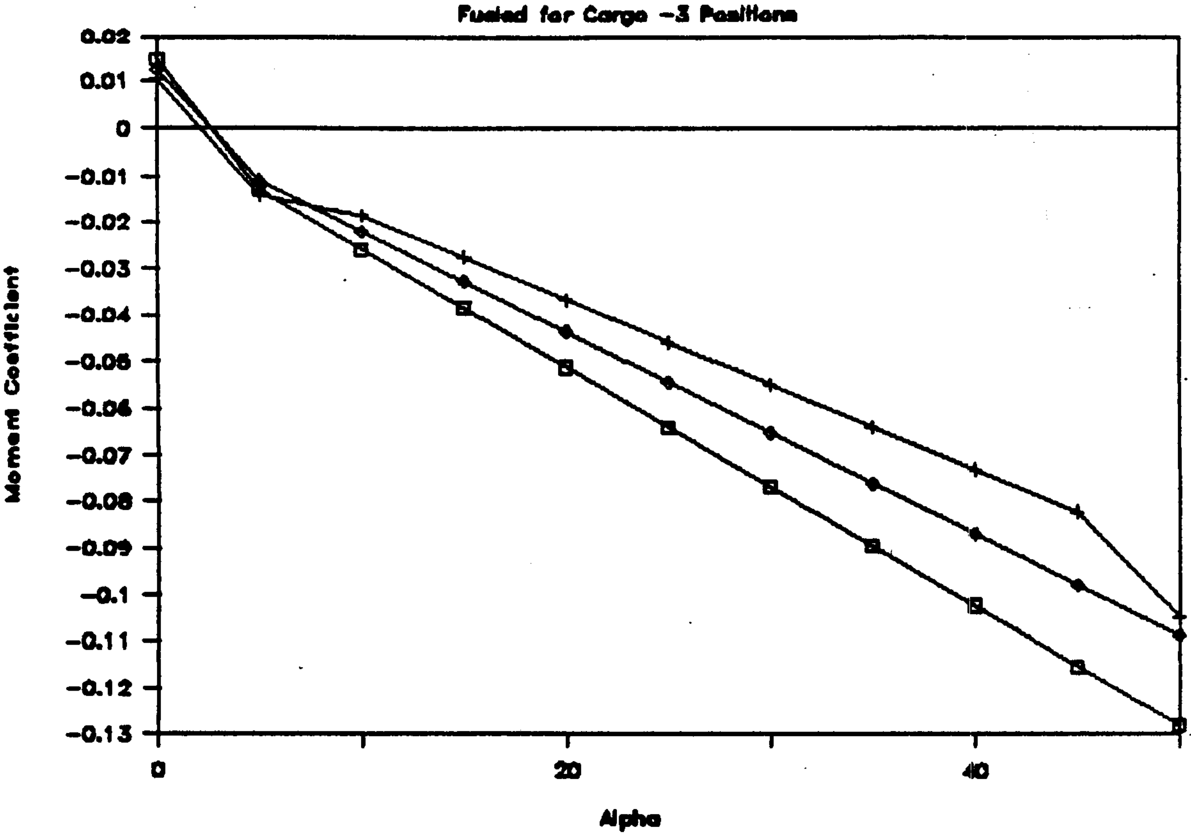
SUPERSONIC AERODYNAMICS

In the supersonic range, CL versus alpha and CD versus alpha were calculated using the Air Forces' DATCOM. The method used is outlined in Appendix A. CL versus alpha was found for a wing-body-tail configuration. A program was written to calculate CL versus Mach number. (For the program listing and calculations of constants for program, see Appendix B.) CL as a function of Mach number calculated using DATCOM is plotted in Figure 12.

FIGURE 10.

CM vs. Alpha

Fueled for Cargo -3 Positions



CM vs. Alpha

Fueled for Shuttle -3 Positions

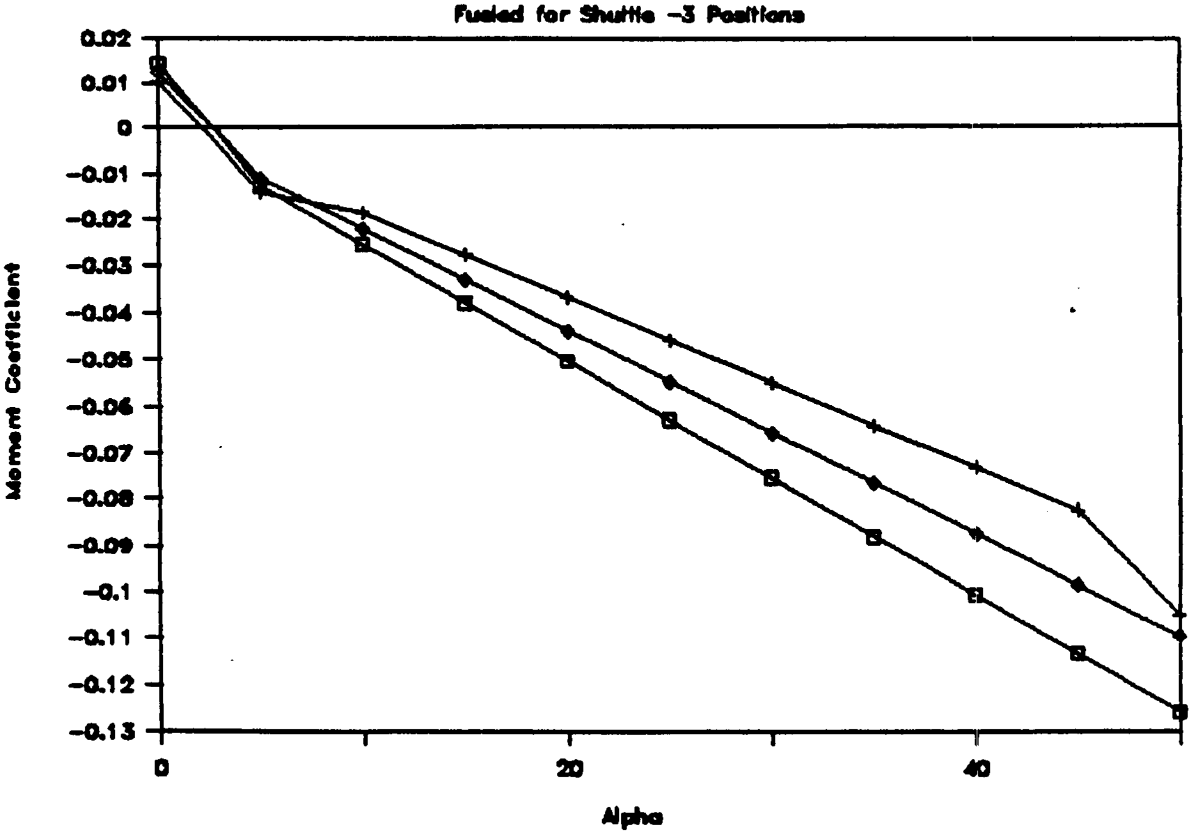
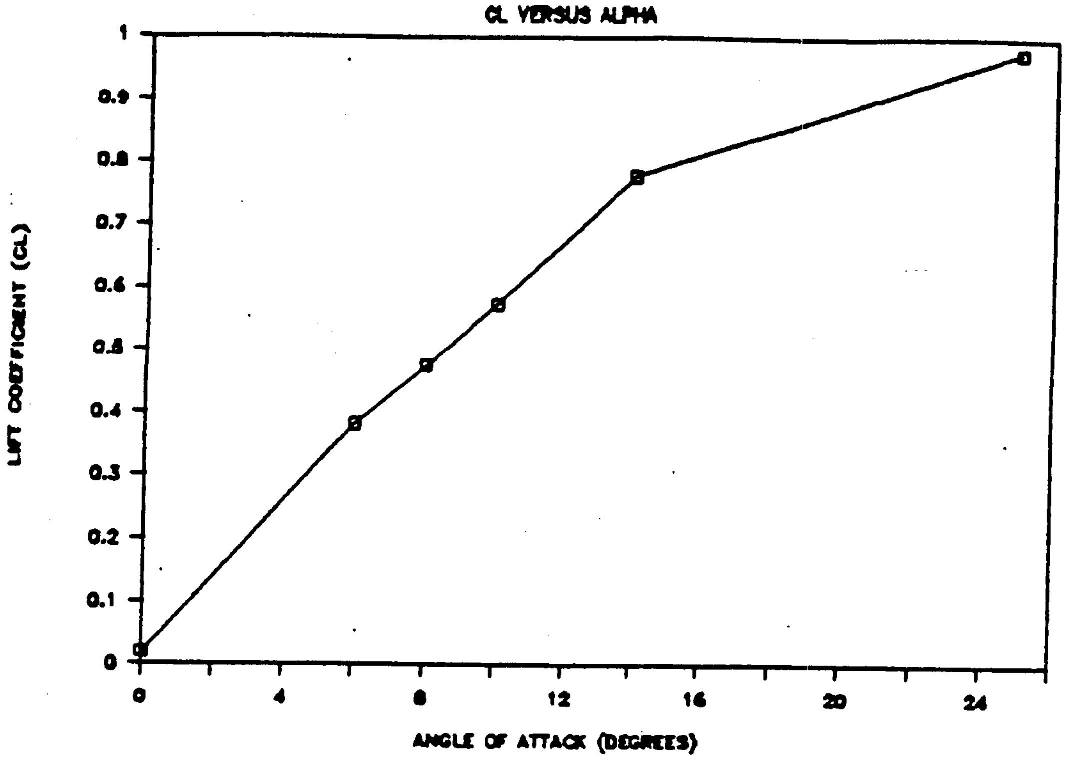


Table 7

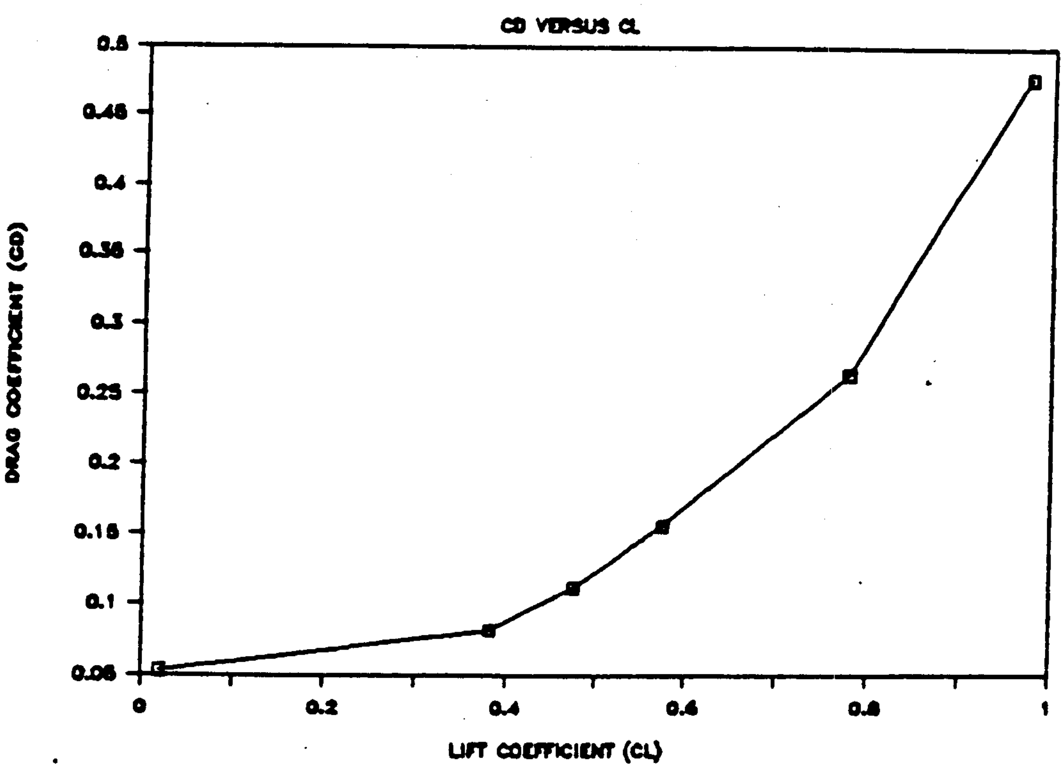
Subsonic CD and CL vs. Alpha

Alpha (deg)	CD	CL
-5.0	0.874	-1.688
-3.0	0.873	-1.674
-1.0	0.872	-1.686
1.0	0.873	1.686
3.0	0.873	1.687
5.0	0.875	1.689
7.0	0.877	1.691
9.0	0.881	1.695
11.0	0.886	1.700
13.0	0.893	1.707
15.0	0.901	1.715
17.0	0.912	1.725
19.0	0.924	1.737

FIGURE 11



CL VERSUS ALPHA
LIFT COEFFICIENT (CL)
ANGLE OF ATTACK (DEGREES)



CD VERSUS CL
DRAG COEFFICIENT (CD)
LIFT COEFFICIENT (CL)

CL VERSUS ALPHA

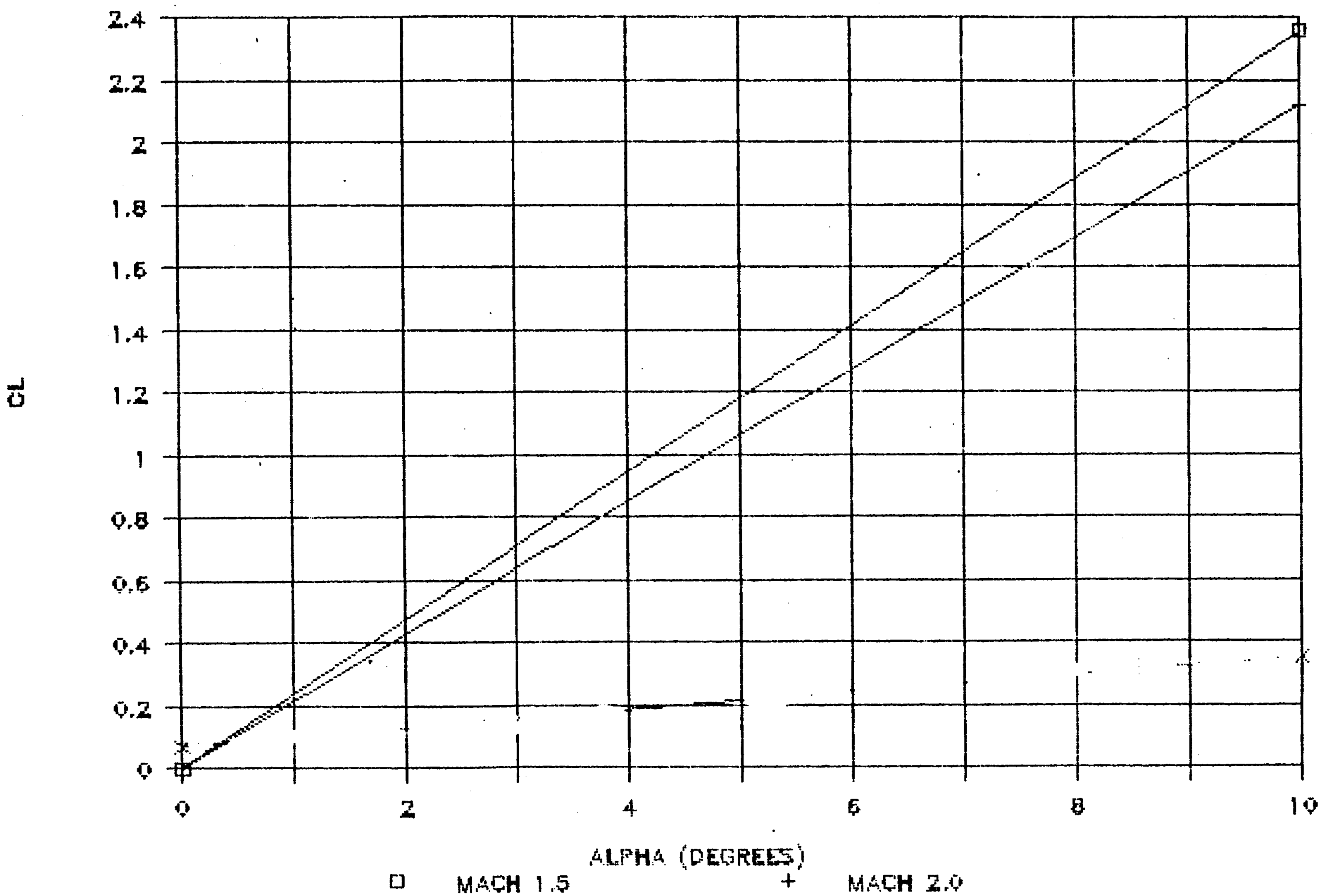


Figure 12

In the calculation of the supersonic zero lift drag coefficient, four different calculations of Mach number ranging from 1.5 to 3.0 in increments of 0.5 were performed while leaving all equations as functions of alpha. Calculations of CD were performed in a program found in Appendix B where alpha was varied from 10 to 60 . The data found using DATCOM is found in Table 8. Values of CL versus Alpha are plotted in Figure 13.

From the supersonic booster group's program, the maximum CL was found to be at an angle of attack of 50 degrees. Therefore, the suggested angle of attack for reentry is 50 degrees.

Table 8

Supersonic CD vs. Alpha

Mach # = 1.5		Mach # = 2	
CD	Alpha (deg)	CD	Alpha (deg)
.384	0.0	0.404	0.0
0.714	10.0	0.901	10.0
0.493	20.0	0.556	20.0
0.990	30.0	1.219	30.0
0.708	40.0	0.853	40.0
1.268	50.0	0.551	50.0
0.737	60.0	0.882	60.0

CONCLUSIONS

According to this analysis, the chosen booster configuration is feasible with today's technology. The values that were calculated for the moment coefficient and static margin indicate that the design is stable and trimable. The lift and drag coefficient plots have fairly standard slopes and the suggested angles for flight are feasible. More detailed study of this design would lead to further improvements in its flight characteristics.

CL VERSUS ALPHA (NASA)

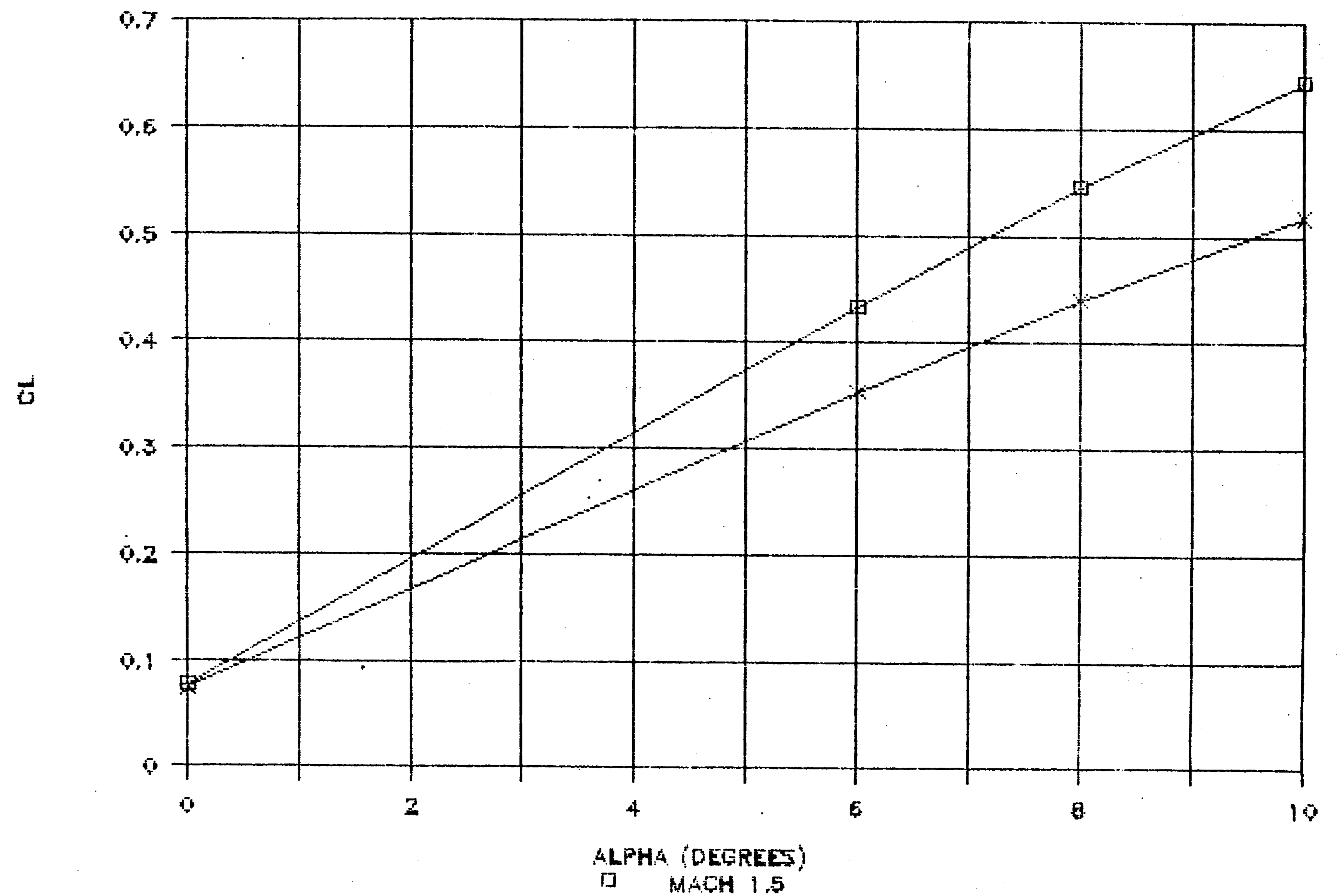


Figure 13

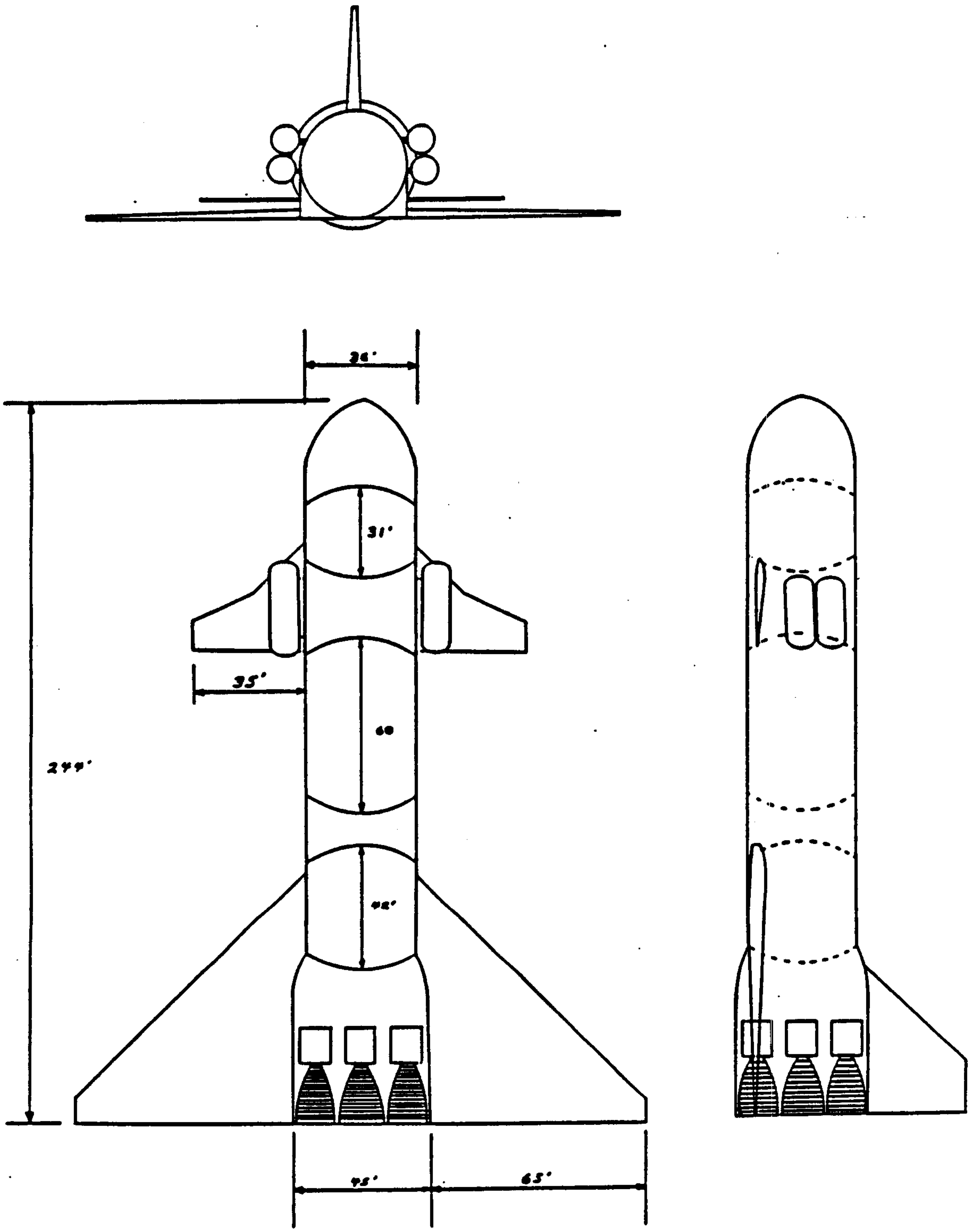


FIGURE 14. 3-Views

LIQUID ROCKET ANALYSIS

The principal objectives in the design of the rocket engines for the winged flyback booster are reliability and simplicity. With the forementioned in mind, along with the thrust requirement of approximately nine million pounds and the cost of developing an engine from the conceptional stage, the group decided to modify Rocketdyne's F-1 engine for use as the booster's launch to staging engines. The basic components of the F-1 engine are a tubular-wall thrust chamber, a direct-drive turbopump, a gas generator, and their controls. Figure #15, on page 45 gives a schematic representation of the F-1 engine. As depicted in Figure #15, the turbo pump is mounted directly on the thrust chamber. All other components are either mounted on these two assemblies or are in the plumbing system between them. We have determined that this layout is perfect for our application. One of the principal advantages of packaging the components in this manner is that the high-pressure propellant ducting does not need to be flexed as the engine is gimbled. Figure #15 on page 45 shows a simplified schematic of the F-1 engine.

An ideal rocket analysis approach was chosen to decide on specific modifications to be made on the ordinary engine. The choice of ideal rocket analysis was appropriate since it has become accepted practice to use ideal rocket parameters which are then modified by suitable correction factors.

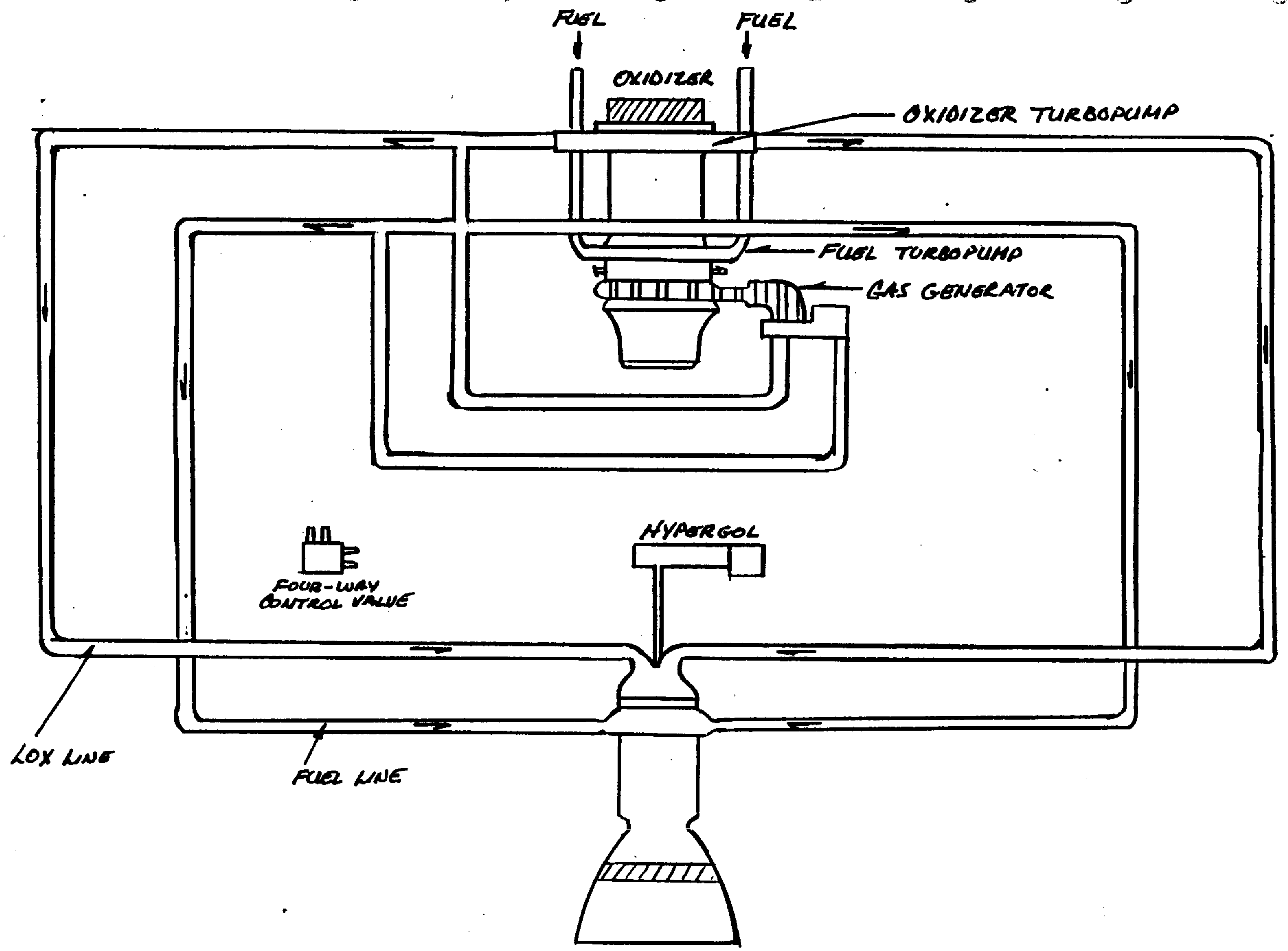


FIGURE 15. F-1 Engine Schematic Drawing

A list of assumptions for ideal rocket analysis along with a derivation of the necessary equations can be found in Appendix A. Furthermore, we have designed the engines to have an optimum thrust at both 30,000 and 60,000 feet. This is to be accompanied by the use of extending nozzles. Optimum thrust occurs when the exit pressure of the rocket nozzle equals the atmospheric pressure.

FUEL

The fuel selected for the booster is JP-4, and the oxidizer chosen is liquid oxygen. The reason for choosing this combination is two-fold. First, the ground rules obtained from NASA stipulated the use of liquid oxygen and hydrocarbon as the propellant for the first stage. Second, the choice of JP-4 allows both the booster engines and the air breathing engines to utilize the same fuel, thus saving weight in the form of a fuel tank and structural supports. In addition to the reasons stated above, JP-4, a noncryogenic fuel, has a good storability which means it can be stored in ordinary tanks over long periods of time and at various temperatures without decomposition or change of state. Based on information obtained from Jim Sanders, MSFC (NASA) Engineer, the propellant has a specific impulse (Isp) of 320 seconds at sea level. As shown in Figure #16, the vacuum specific impulse is a function of the percent weight of fuel in the propellant.

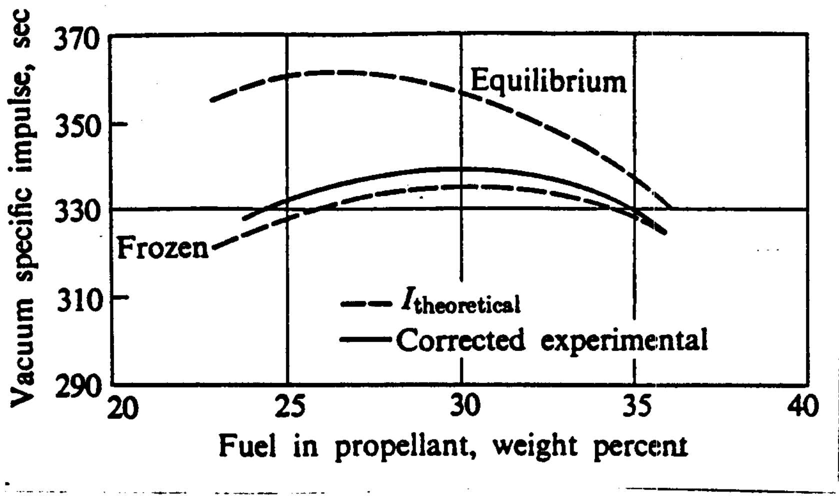


FIGURE # 16. Vacuum Isp of JP-4 and Oxygen Rocket

From this figure the best oxidizer to fuel ratio is 2.3 and the corresponding vacuum Isp is approximately 340 seconds. The sea level Isp is approximately 320 seconds, as stated by Jim Sanders. The ratio of specific heat is represented by gamma, , and is equal to 1.24. The propellant products have an average molecular weight of 22.0 lbm/lbm-mole, and the propellant itself has an adiabatic flame (theoretical combustion) temperature of 6230° R.

INJECTOR ANALYSIS

The injector sprays fuel and oxygen into the thrust chamber in a pattern to calculate the most efficient combustion. The injector face as shown in Figure #17 on the combustion side, contains the injector orifice pattern, determined by alternating fuel and oxygen rings.

The injector configuration can affect the local wall temperature in the combustion chamber and througout the rocket. The injector pattern must be controlled. Proper control will result in a layer of relatively cool gas near the wall. Poor injection will result in hot spots or possible burnout of the wall material.

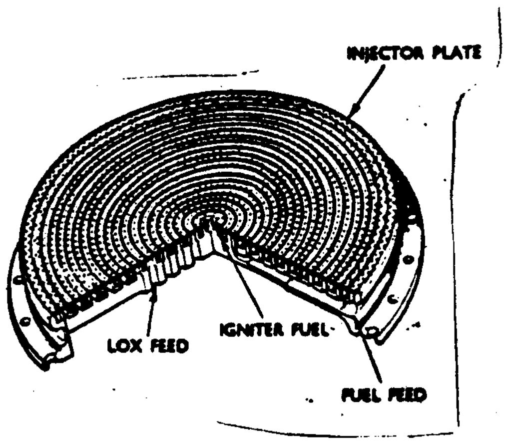


FIGURE # 17. Flat-Faced Injector

The mass flow rate of propellant and the pressure drop across the injector are related by the following equations:

$$\dot{m} = Cd*\rho_p *Ai\ (2*(\delta P/\mu)^{1/2} \qquad (1)$$

where ρ denotes the propellant density, Cd denotes the discharge coefficient, Ai denotes the injector nozzle area, and δP denotes the pressure drop across the injector nozzle.

The discharge coefficient, Cd, can vary in value depending on the type injector nozzle being used. Figure #18 on page 49 illustrates various types of injector nozzles and their respective discharge coefficients.

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Injector Discharge Coefficients

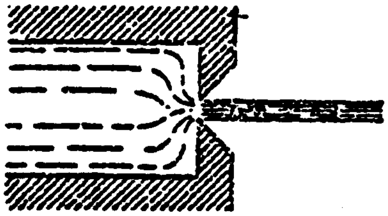
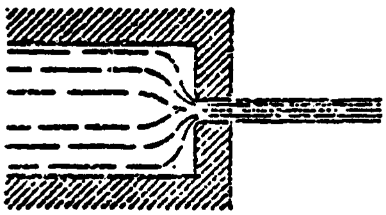
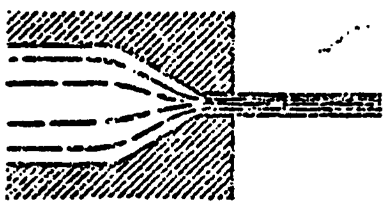
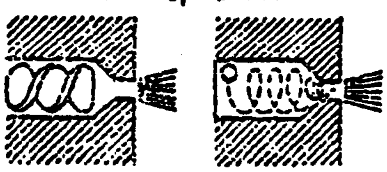
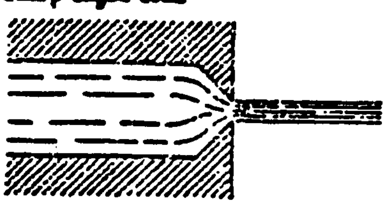
Orifice Type	*Diameter (inches)*	*Discharge Coefficient*
Sharp-edged orifice 	above 0.10 below 0.10	0.61 0.65 approx.
Short tube with rounded entrance $L/D > 3.0$ 	0.040 0.062 0.046 (with $L/D \sim 1.0$)	0.83 0.90 0.70
Short tube with conical entrance 	0.020 0.040 0.062 0.100 0.125	0.7 0.82 0.76 0.84–0.80 0.84–0.78
Short tube with spiral effect 	0.04–0.25	0.2 –0.55
Sharp-edged cone 	0.040 0.052	0.70–0.69 0.72

FIGURE 18. Injector Discharge Coefficients

In general, injector nozzles characterized by short tubes with round inlets have discharge coefficients in the range Of 0.97 to 0.99 and injector nozzle characterized by sharp cornered tubes have a discharge coefficient in the range of 0.6 to 0.8.

Typical pressure drops for most injectors are of the order of several hundred psi. A large pressure drop promotes good atomization and tends to control combustion instabilities that are associated with pressure oscillation in the combustion chamber and in the propellant supply system. A large pressure drop also induces high propellant inlet velocities. However, it should be noted, it is desirable to have a pressure drop that is consistent with good atomization and combustion stability.

The injector analysis performed on the modified Rocketdyne F-1 engine required three critical assumptions. The first assumption was that the total fuel area and total oxygen area were equal to one another. The second assumption was that the injector orifices were sharp edged orifices with a discharge coefficient of 0.61. The third assumption was to assume that the injector utilized a doublet impinging stream injection pattern with the angle of the oxygen stream equal to 20 . Based on the basic ideal rocket analysis results, a mass flow rate of 4685.7 lbm/sec was found along with a fuel to oxidizer ratio (r) of 2.3. The mass flow of oxygen ($\dot{m}_o$) and the mass flow rate of fuel ($\dot{m}_f$) was found by using the

following equations.

$$\dot{m}_o = (r/r+1)\ \dot{m} \qquad (2)$$

$$\dot{m}_f = (1/r+1)\ \dot{m} \qquad (3)$$

The result of equation (2) was $\dot{m}_o$= 3267.045 lbm/sec and the result of equation (3) was $\dot{m}_f$= 1420.455 lbm/sec. The same number of fuel orifices and oxidizer orifices were used as in the original F-1 engine. They are 3700 and 2600 respectively. The area of one fuel orifice was calculated to be 0.1698 inches squared and the area of one oxygen orifice was calculated to be 0.2147 inches squared. Referring to Figure noting that the areas are above 0.1 inch squared a discharge coefficient of 0.61 was chosen. Using equation (1) and rearranging terms, the pressure drop across the injector was found to be 82.2139 psia. In order to find the angle of the fuel flow, the following equation was used.

$$\dot{m}_f * Vf * \sin(\tau_f) = \dot{m}_o * Vo * \sin(\tau_o) \qquad (4)$$

In order to determine the fuel flow angle , an arbitrary angle of 20° was selected for the oxygen flow and an impingement angle of 0° (axial flow) was selected at the point in which the fuel and oxygen intersect. The velocity of the fuel and oxidizer was determined by the following equations:

$$Vo = Cp*(2*gc*\delta P/\rho_o)^{1/2} \qquad (5)$$

$$Vf = Cd*(2*gc*\delta P/\rho_f)^{1/2} \qquad (6)$$

The velocity of oxidizer was found to be 10.516 ft/sec and the velocity of fuel was found to be 12.903 ft/sec. The fuel angle was found to be 69.595 using equation (4).

GAS GENERATOR

The power necessary to drive the turbine is obtained from the gas generator, which generates gases by a chemical reaction of propellants similar to those in the thrust chamber. The generated gases have to be cooler than thrust chamber reaction gases, since excessive temperatures will cause failure in either the turbine nozzles or the turbine wheel. The gas generator burns a fuel-rich mixture of the same propellants used in the thrust chamber. It burns approximately two percent of the total propellants used in the engine. The gas generator used for the modified F-1 engine will be the same one used in the Rocketdyne F-1. It is partially spherical in shape and is approximately ten inches in diameter. The basic design of the F-1 gas generator incorporates a doubled wall combustion chamber, through which fuel flows to regeneratively cool the body.

TANK PRESSURIZATION AND PUMP ANALYSIS

The main objectives of tank pressurization is to keep the components so small and as lightweight as possible. On account of this, the pressures in the tank must be much lower than pressures required at the injectors. To perform the task of developing these large pressures, turbo-centrifugal pumps

are used to transport the propellants from their tanks to the injector. These pumps, one for the fuel and one for the oxidizer, provide a dynamic pump head (TDH) which is proportional to the pressure difference between the injectors and propellant tanks.

The propellants are supplied to their respective pumps through a single inlet, in line with the main shaft, and is discharged radially through dual outlets. The dual outlet design provides a balance of the centrifugal loads in the pump which, in turn, minimizes the pump diameter, see Figure #19 .

The pumps are powered by a turbine. The turbine is driven by hot fuel gases that are heated in a small gas generator, discussed in the forementioned section. The turbine gives approximately 55,000 brake horsepower to the oxidizer turbopump and about 25,000 brake horsepower to the fuel pump.

The turbine exhaust gases, which are gaseous fuel, are collected by a manifold and ducted through a nozzle skirt into the engine exhaust plume. Using a circumferential manifold the turbine exhaust gases introduce a cooler boundary layer to protect the wall of the nozzle. This idea allows for a comparatively light weight system with no need for turbine exhaust ducts and other attachments.

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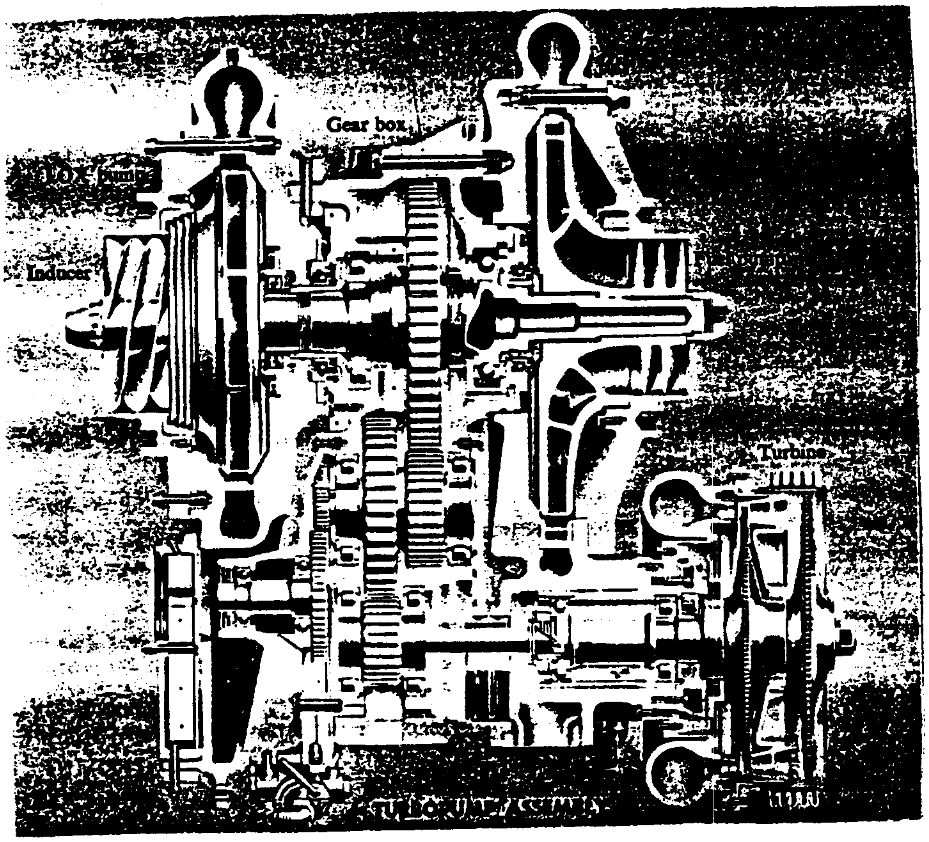


FIGURE # 19 Turbopump Assembly

Given specifications of an F-1 turbopump and turbine are given below:

$$Q = 24811 \text{ gal/min}$$

$$Q = 15741 \text{ gal/min}$$

$$P = 3729.9 \text{ psia}$$

$$BHP_{p} = 55000 \text{ hp}$$

$$BHP_{T} = 25000 \text{ hp}$$

To find the pressure of the propellant tanks:

$$TDH = \delta P*(2.31) = (BHP*\eta_{p}*3960)/Q*SG$$

where TDH is the dynamic pumphead, δP is the pressure change from tank to injector, and Q is the volumetric flow rate. For the oxidizer liquid oxygen, the density is 71.2 slugs/ft ; therefore, the specific gravity equals 1.141. Assuming a pump efficiency of 80%, δP = 2664.4 psia.

$$P_{tank\ oxidizer} = P_{injector} - \delta P = 1065.5 \text{ psia}$$

For the fuel JP-4, the density is 47.3 slugs/ft ; therefore, the specific gravity equals 0.758. Assuming the same efficiency, δP = 2873.5 psia.

$$P_{tank\ fuel} = P_{injector} - \delta P = 855.4 \text{ psia}$$

THRUST CHAMBER AND HEAT TRANSFER

In the drive to produce large, high-pressure engines, a major problem was a satisfactory means to cool the thrust chamber. Therefore, this section will discuss the configuration of the thrust chamber, the strength of the thrust chamber material, and how the heat transfer of our rocket was determined.

The thrust-chamber assembly consists of a tubular-wall, regeneratively cooled chamber with an uncooled extension, a double-inlet oxidizer dome, four integral fuel valves, and a flat-faced injector. The cooled portion of the thrust chamber has a 40 inch combustion chamber diameter, an 8 foot nozzle exit diameter, and is 11 feet in length. The chamber is designed for an uncooled extension which will be deployed at 45,000 feet.

Combustion chamber temperatures of rocket propellants typically are higher than the melting points of most common metals and alloys. However, at high temperatures the strength of most of these materials decline rapidly. This can be seen in the figure below.

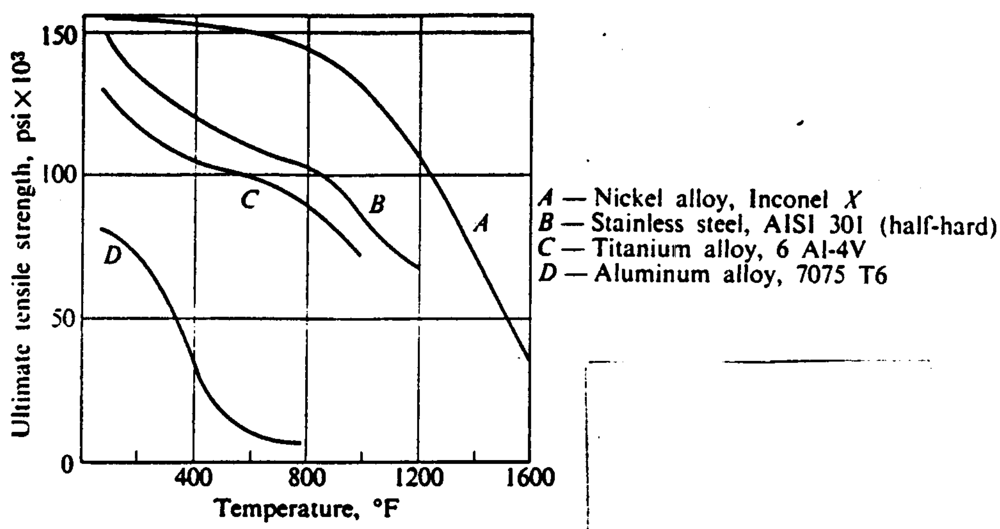


FIGURE # 20 Variation of Tensile Strength with Temperature

Since the wall thickness of the thrust chamber depends strongly on the stresses which it can support, it is desirable to use highly stressed materials. The wall should be cooled to a temperature considerably below its melting point, and below the propellant stagnation temperature.

After researching several different materials, the rocket propulsion group decided to use stainless steel as the material for the combustion chamber. Two major reasons for this choice are economically feasible and it is readily

available.

Since a liquid rocket is being used, it has already been stated that it will be regeneratively cooled. This means that the fuel or oxidizer will flow in tubular passages directly outside the chamber wall. These tubes reduce wall thickness and thermal resistance and, more importantly, increase the coolant velocity in the throat section.

For our liquid rocket, we selected liquid hydrogen as the coolant. After careful research of the thrust chamber wall material, stainless steel, it was decided to use a wall thickness of 0.125 inches.

In order to determine the heat transfer, it is first possible to find the total heat absorbed per second in the thrust chamber. The calculated value for this is:

$$q = 8164.8 \quad \text{BTU/sec} \tag{7}$$

Then the average thrust chamber area can be found by intergrating from the combustion chamber area to the exit area of the cooling section of the nozzle. The calculated average area is:

$$A = 4536.0 \quad \text{in}^2 \tag{8}$$

Now, by dividing equation (7) by equation (8) it is possible to find the average heat transfer of the combustion chamber. The calculated heat transfer is:

$$q/A = 1.8 \quad \text{BTU/in}^2\text{-sec} \tag{9}$$

From the calculated average value of the thrust chamber, it is easy to see that the heat transfer rate is high.

COMBUSTION INSTABILITIES

Rocket engines are sometimes subject to combustion instabilities in the form of large pressure oscillations within the chamber. Such instabilities can cause engine failure either through excess pressure, increased wall heat transfer, or a combination of the two. Because combusion instabilities cannot often be observed on smaller-scale models, the full scale model would have to be tested to detect any instabilities. Moreover, it is impossible to determine instabilities analitically and experimental test must be performed to insure that no instabilities exist; therefore, we cannot evaluate this area of the design at this stage of conceptual design.

NOZZLE ANALYSIS

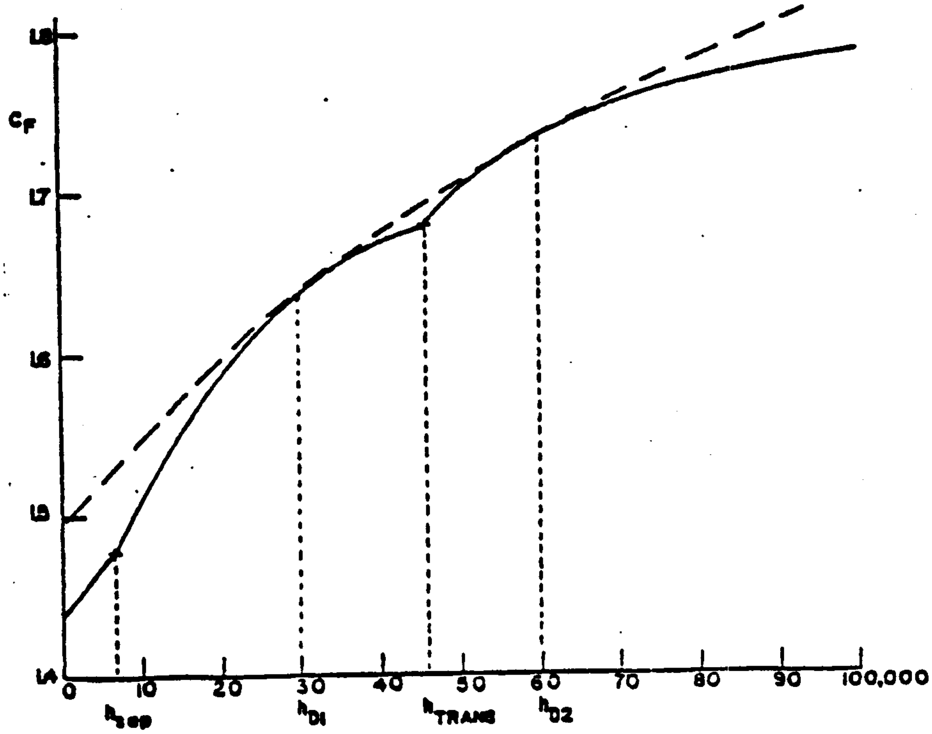
The design of a nozzle requires taking into account variations in velocity and pressure on surfaces normal to the streamlines. Improper shaping of the nozzle can result in shock formation and substantial performance loss. It has been found empirically that simple conical divergent nozzles provide best performance when their half-angles are betwwen 12 and 18 degrees. (10:193) Furthermore, the bell shaped nozzle permits additional advantages in reducing size and weight when compared with the standard 15 degree half-angle conical nozzle. Without any reduction in performance, the

bell shape also permits a 20 percent reduction in length. (10:194)

The F-1 engine currently features a bell-shaped nozzle. The proposed modified F-1 engine will feature a conical nozzle to satisfy the mission requirements. Through the use of the program in Appendix A, the area expansion ratios were found to be 55 and for altitudes of 30,000 and 60,000 feet respectively. As shown by Figure #21, on page , the thrust is to be optimized at 30,000 and 60,000 feet. The nozzles will be optimized for an altitude of 30,000 feet at the launch. Once the vehicle has reached approximately an altitude of 45,000 feet, the nozzle extension will move down to an area expansion ratio which gives optimum conditions at 60,000 feet. The nozzle will remain at this position until staging. Once the liquid rocket engines are shut down and staging is complete, the nozzles will be retracted to their initial area expansion ratio. In doing this the drag will be slightly reduced.

THRUST VECTORING

The thrust-vector control is achieved by gimballing the entire engine. The high-pressure fuel is used as the hydraulic actuating medium. Although this is an unconventional approach, it was the method used on the Saturn V's first stage. The use of the fuel has its drawbacks; however, it eliminates a separate hydraulic system.



Thrust coefficient vs altitude.

FIGURE # 21. Thrust Coefficient Versus Altitude

RESULTS

Parameters at Optimum Altitude

30,000 feet	**60,000 feet**
Pa=4.36 psi	Pa=1.04 psi
P_{tc}=3731.17 psi	P_{tc}=3731.17 psi
T_{tc}=6230°R	T_{tc}=6230°R
T_e=1800°R	T_e=1572°R
T*=5563°R	T*=5563°R
M_e=4.53	M_e=4.97
A*=2.01 ft^2	A*=2.01 ft^2
A_e=86.59 ft^2	A_e=145.37 ft^2
ϵ=43	ϵ=72

Nozzle Dimension

Half-Angle	16.22°
Length of Cooled Nozzle	11.00 ft
Length of Unextended Nozzle	15.30 ft
Length of Extended Nozzle	20.63 ft

CONCLUSION

The basic parameters of the modified Rocketdyne F-1 rocket engine have been determined. These parameters include the following: the temperature and pressure of the combusion chamber, the temperature, pressure and area of the throat, and the exit area of the rocket nozzle. The rocket features an expandable nozzle that is optimized at two different heights. These heights are 30,000 feet and 60,000 feet. The analysis of the engine also includes and injector analysis, turbopump analysis, and a heat transfer analysis based on data obtained from trajectory analysis. The decision has been made to use 6 F-1 engines to satisfy the booster's thrust requirement. The Rocketdyne F-1 engine will be used; however, there will be a few modifications made to the nozzle, injectors, and turbopump based on our data. The F-1 was chosen since it is economically feasible and readily available. A second reason the F-1 was chosen was because the research and development cost for a new rocket engine are astronomical and the development time is in excess of five or more years.

It is the recommendation of this group that Rocketdyne be awarded the contract to redesign the F-1 engine to meet the requirements specified by both the cargo and shuttle missions. It is estimated that the redesigning and testing of the F-1 engine will take approximately two years. The proposed cost of one of these engines in 1986 dollars is approximately 64.2 million dollars. It is also recommended that the modified

engines be tested at either Edward's Air Force Base or Rocketdyne's Santa Susana facility. Both facilities are capable of handling 1.5 million pound-force thrust levels.

AIR BREATHING JET PROPULSION

After the flyback booster reaches 7,000 feet per second, the rocket engines of the booster will shut down. At this time the manned Shuttle II or the unmanned cargo vehicle will separate from the flyback booster. The flyback booster will then return to a landing destination powered by turbofan engines. While the flyback booster is in this boosting stage, it will reach velocities that would destroy the turbofan engines. Therefore, it is necessary to protect the turbofans from damage with the use of inlet/outlet enclosures. These protecting enclosures, during reentry, will open when the booster's velocity has decelerated to subsonic speeds. At this point, the turbofans will fire up and power the booster to a predetermined landing destination.

ANALYSIS: The main information which will be found in this section of the report is: (1) the total thrust of, the weight of, and weight of fuel used by the engines, and (2) the design and placement of the specialized cowlings needed on the engines. To determine the this information, General Electric and NASA were consulted. See references 12 and 13.

The first step in this analysis was to determine which type of jet engine should be used. Since one of the specifications for this vehicle was to fly at low velocities

(Mach Numbers less than 1.0) during powered flight, a turbofan engine would be best suited for this particular case. This is due to the fact that turbofans have a high thrust to weight ratio, and at low specific fuel consumption at low velocities.

The next step in the analysis was to determine which bypass ratio was best suited for this case. NASA suggested a medium bypass ratio of approximately two. General Electric has in operation two engines with a bypass ratio of two, with one augmented with an afterburner for high thrust output. For comparison, further analysis was performed on three types of General Electric engines, with two different bypass ratios: (1) CF6-8032 (5 bypass), (2) F101 (2 bypass), (3) F101-augmented (2 bypass).

In determining the type of engine, the number of engines, and the amount of fuel use by the engines best suited for the flyback application, a program, obtained from NASA, was put into operation (program 2). This program considered a particular flyback vehicle from staging position and determined the flying characteristics of the return trajectory. To make the program fit this particular booster, engine performance and aerodynamic data were needed to be inputted into the program. The engine performance consisted of thrust, and specific fuel consumption, at different Mach Numbers and at different altitudes. Also, the program had a built in engine-out and ten minute go-around the landing field capability. NASA's program had the engines ignite at an

altitude of 35,000 feet. The vehicle would then begin a high cruise at 15,000 feet and let down to a low cruise of 10,000 feet before landing. The engines would run at a maximum time of 3,175 seconds (53.4 minutes).

In determining the performance of each engine in the analysis, information again obtained form NASA and General Electric was used. General Electric furnished the dimensions, sea level static thrust, and specific fuel consumption for each of the engines. NASA furnished the equations and data necessary to change the sea level thrust and specific fuel consumption for different altitudes and Mach Numbers. The following equation was used to convert the sea level thrust to different altitudes:

$$F_{ALT} = F_{SSL} \sqrt{\frac{T_{SSL}}{T_{ALT}}} \cdot \frac{P_{ALT}}{P_{SSL}}$$

To find the thrust at different Mach Numbers, data from NASA's generic engines was used (reference 13). The following ratio equation was applied:

$$\left(F_{X-MACH}\right)_{NEW} = \left(\frac{F_{\emptyset-MACH}}{F_{NEW-MACH}}\right)_{GENERIC} \cdot \left(F_{\emptyset-MACH}\right)_{NEW}$$

The above relation is an approximation which does have some error, yet, according to NASA, the error should be small. The specific fuel consumption also used a ratio assumption to determine its change due to Mach Number and altitude.

The next step in the analysis was to determine where the engines would be installed on the booster. Because of aerodynamic stability, the engines will be installed over the canards towards the front of the booster. Because, when the booster is powered flight, it is a flying empty fuel tank with 500,000 pounds of dead rocket engines on the tail end. The weight of the jet will help bring the center of gravity towards the nose of the booster resulting in a more stable flight.

One of the key points in the installation of these engines is the design of the specialized cowlings. The cowlings must accomplish two different tasks. It should enclose the engines completely during the high Mach Number boosting stage, yet it must be made streamline to reduce aerodynamic drag. Figure 22 illustrates one possible design that this report recommends. The engines will resemble this figure until an altitude of 35,000 feet. At this point, the welded seams on the blow-off caps will split. Then the cone shaped enclosures on the inlets and exits of the cowling will divide up into four different slices. Immediately after, each piece of the cone will retract quickly inside the inner sleeve in the cowling (Figure 23). The enclosure cones and the cowling should be made of the same material as the booster's outside skin, and sprayed with a carbon-carbon composite; since, the curvature of the cone is relatively blunt. This should protect the cones from the high temperatures during the

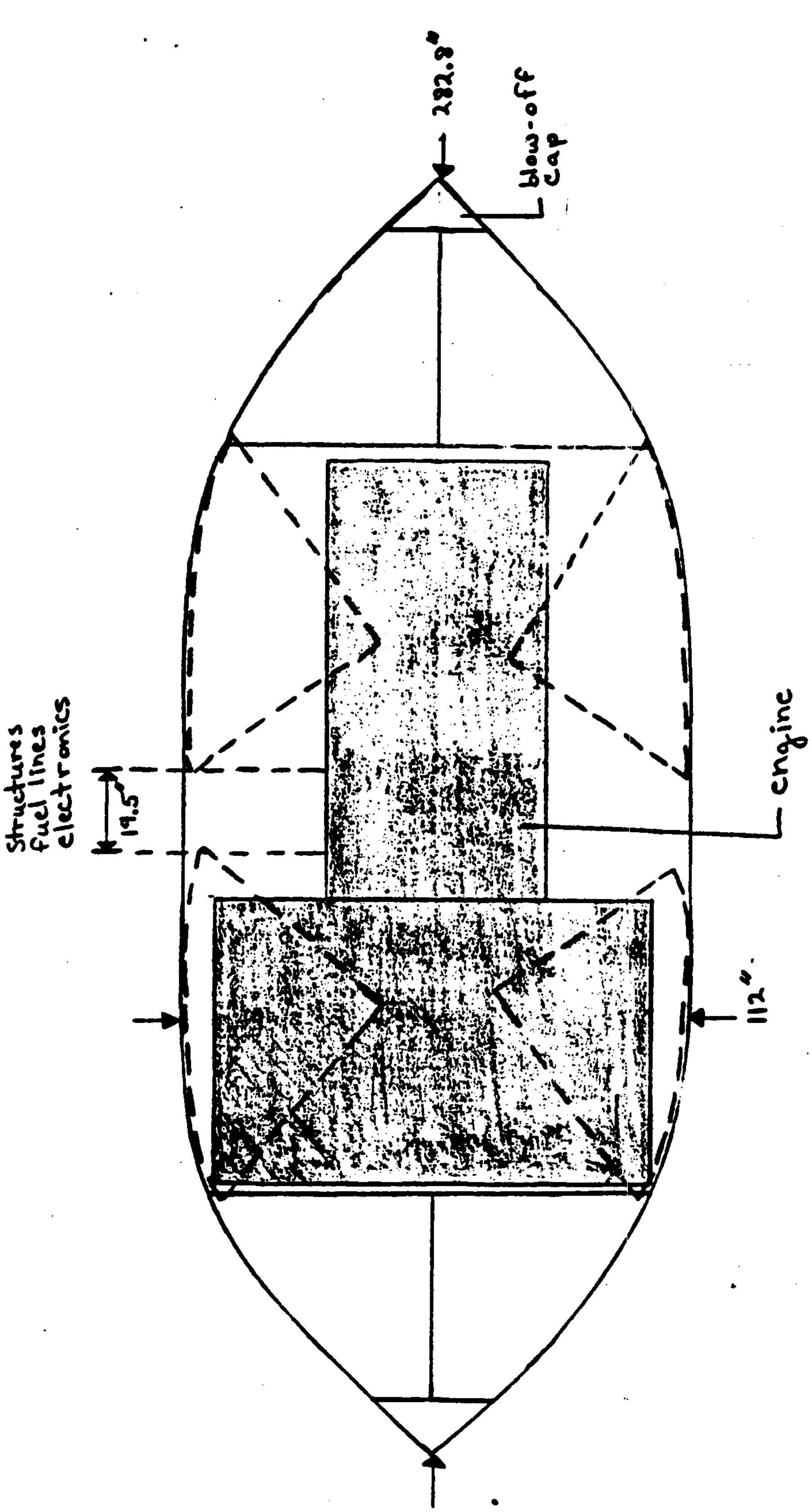


Figure 22: Enclosure cowling, closed

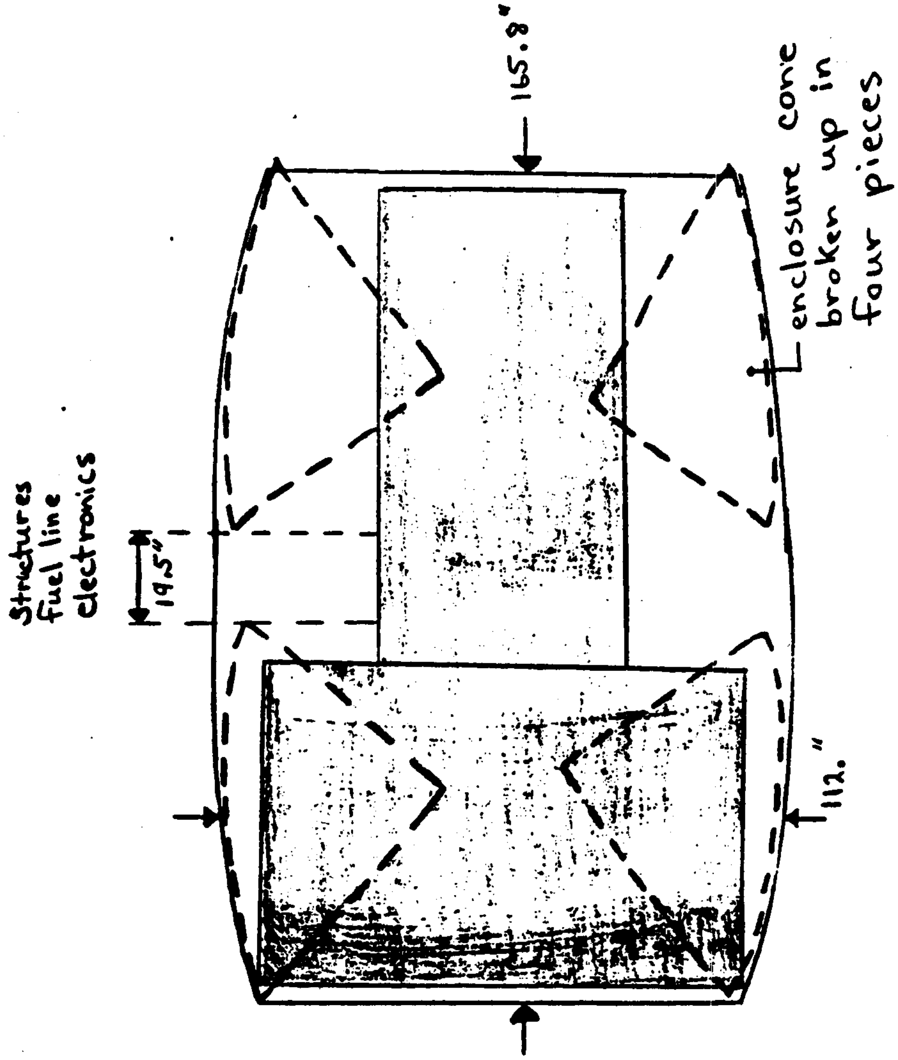


Figure 23: Enclosure cowling, open

boosting stage.

The most important part of this section's analysis is to decide on which engine to use on the booster. To make this decision, three important aspects should be viewed. The first aspect is the total weight of the engines and fuel. The second aspect is the aerodynamic drag. Since the aerodynamic drag is directly proportional to the frontal area, the following equations were used:

$$\text{Drag} \sim \text{Frontal Area} \quad \text{or} \quad D \sim S$$

$$S = \pi d^2/4$$

$$\text{Drag} = (0.50\ C_D \rho V^2)\, \pi d^2/4$$

The last aspect is the number of engines. The more engines used, the more money, structures, electronics, pumps, feed lines, and mechanical parts will be needed to supply the system. So, the above three factors will have to be weighed against each other on deciding on the type of engine.

In addition, the engines will be running on the same fuel (JP-4) as the main rocket engines. The JP-4 will be pumped from the booster's main tank. When installing the engines, the nozzles should be angled six degrees downward from the longitudinal axis for maximum performance; since, the booster will cruise during flyback at an angle of attack of six degrees.

RESULTS: After reading in the three different engines' thrust and specific fuel consumptions into NASA's program, the following data was outputed:

Engine	Engine Weight (lbs)	Weight of Fuel (lbs)	Total Weight (lbs)	Frontal Area (ft^2)	Number of Engines
CF6-80C2	32205	102587	134792	273.68	4
F101 w/ AB	31680	521132	552812	191.45	8
F101 w/o AB	46332	130449	176781	311.08	13

Table 9: Turbofan Engine Output Data

From these results, the CF6-80C2 is the best engine to use for the flyback booster. The CF6-80C2 is better than the F101 (without afterburner) in all categories, yet the augmented F101 seems to give the CF6-80C2 a challenge. The F101 with afterburner is lighter in engine weight by 525 lbs., and also has a smaller frontal area by 82 square feet. However, the CF6-80C2 uses 80% less fuel, (418,545 lbs. less) which makes this the deciding factor in the determination of the engine type. After taking all the categories into account, the CF6-80C2 is the better selection.

Below is the data on the CF6-80C2 furnished by General Electric (Figure 24).

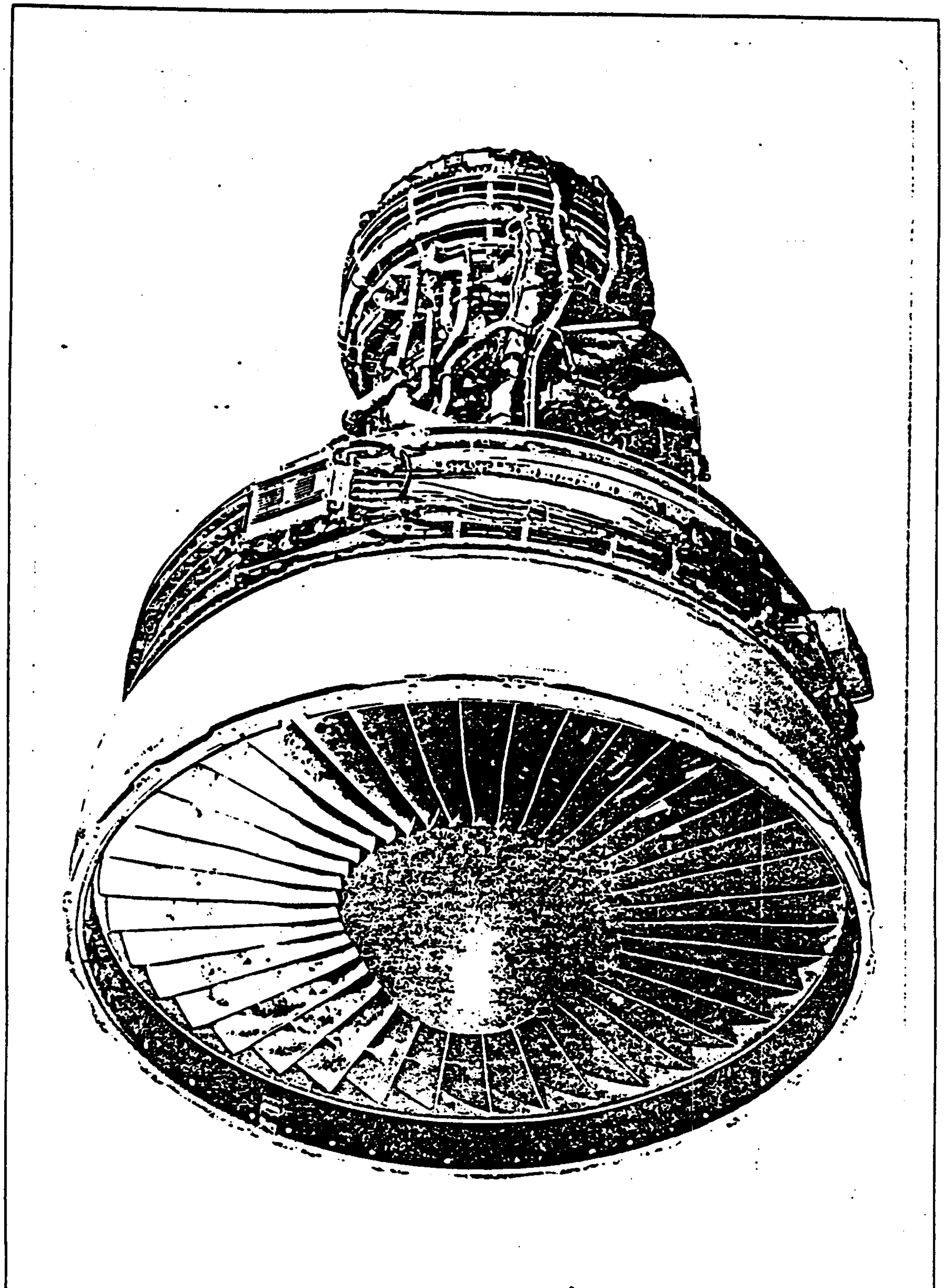


Figure 24: CF6-80C2 Propulsion system.

Bypass Ratio	5.09
Fan Tip Diameter	93 in.
Length	160.9 in.
Weight	** 8,946 lbs. **
Thrust	** 62,500 lbs. **
Specific Fuel Cnspt.	0.40
Thrust/Weight	7.763
Fuel	JP-4
Max. Cowling Dia.	112 in.
Frontal Area	9,852 in.^2

(** 10% change due to technological advancements)

Some of the data above was changed from the 1986 data due to future technological advancements. These advancements were suggested through Ms. Mary Pryor at General Electric. See Appendix C for calculations.

The booster will require four CF6-80C2 turbofan engines. If all engines fire up, each engine will run at a maximum of 85% power. If one engine fails, the remaining three will run at over 95% power for 30 minutes. After 30 minutes, the engines will throttle down to under 90% power for the rest of the flight.

In table 10, a summary of the thrust performance is presented. This data was inputted along with the specific fuel consumption data, in table 11, into NASA's program. The

ALTITUDE (FEET)	M=0.0	M=0.2	M=0.4	M=0.6	M=0.8
0	62500	55000	51877	50627	50000
1524	59395	53738	53034	53738	56459
3048	56459	51447	51447	52310	56459
4572	53569	49743	50702	53569	57395
6096	50637	45137	46234	48435	52839
10668	43551	38967	40116	42409	44701

TABLE 10: Thrust at Different Mach Numbers and Altitudes of the CF6-80C2.

calculations for both tables is presented in Appendix C.

ALTITUDE (feet)	S.F.C. (lbm/lbf-hr)
0	.3321
1524	.3985
3084	.4151
4572	.4317
6096	.4317
10668	.4317

Table 11: Specific Fuel Consumptions at Different Altitudes

CONCLUSION: With weight, cost, drag, electronics, and other data taking into account, the CF6-80C2 is the obvious choice of the engines analyzed.

In the designing of the enclosure cowlings, testing will have to be an integral part in the design. This is mainly due to the high complexity of the opening device. Also, the four piece cone may under go flutter as the blow-off cap releases its tension. The design of the enclosure cowling may also have some drawbacks in weight, yet they are completely reusable. On the other hand, if the cones were blown off completely, they might risk damaging the wing, fuselage, or vertical tail on their trajectory towards the ground. Because of the above reasons, enclosure cones which retract or just open would be prime answer to the problem of enclosing turbofan engines during supersonic flight.

WEIGHTS

When a major project like the flyback booster is attempted, a beginning point is needed. In most cases the starting point for a technical project begins with weights. Weights set the parameter each project engineer must work under and also are used when estimating the cost of the entire project.

For the flyback booster, a weight specialist has two main objectives. First, a complete breakdown of component weights is needed. Weights are assigned to various components of the booster and that information is passed to the various engineers working on the project. Once a component is assigned a certain weight, the engineer must complete his project under that weight. If the engineer finds it impossible to stay under the assigned weight, the weight specialist must recalculate all numbers and assign new weights. The second objective is to calculate the cutoff weight. This is essential for the performance engineer.

A computer program was obtained from MSFC called Ascent Vehicle Interactive Design (AVID) and was used to accomplish both objectives. AVID is preliminary design software used in establishing performance capabilities over a wide range of aerodynamic designs.

Before AVID can be used, a preliminary vehicle must be constructed, and its dimensions must be fed into AVID.

Information on engine type, propellants used, material constants, and modes of operation must also be fed to AVID. After this information has been entered, the different weights of the booster can be calculated.

To calculate the different weights certain formulas are implemented. Some of the formulas are shown below.

The fuselage weight is calculated by:

$$W(fuse)=A(5)*SB(wet)^{1.033}*(RSTR)$$

where A(5) is the fuselage weight constant
SB(wet) is the body wetted area

The thrust structure weight is calculated by:

$$W(thrust)=A(6)*(TH1*eng+TH2*eng)^{1.033}*(RSTR)$$

where A(6) is the thrust weight constant
TH1 is the mode 1 engine vacuum thrust
eng is the number of engines
TH2 is the mode 2 engine vacuum thrust

The landing gear weight is calculated by:

$$W(lng)=A(12)*W(lnd)^{A(13)}*(RSTR)$$

where A(12) and A(13) are landing gear constants

The reaction control system weight is calculated by:

$$W(rcs)=A(16)*W(ent)*(RSS)$$

where A(16) is the rcs weight constant
W(ent) is the vehicle entry weight
RSS is the subsystem weight reduction.

AVID can calculate the weights for thirty components of the flyback booster. See Table 12 for a list of the components and their weights.

Center Of Gravity Calculations

As in all flight vehicles it is imperative that the vehicle's center of gravity (c.g.) location be within a certain range along the vehicle's longitudinal axis. On the flyback booster this range must be located such that the vehicle is controllable at all times during the trajectory. In order to keep the booster's c.g. within this range, the location of each item is carefully considered.

The following method is used to find the c.g. along the longitudinal axis.

1. Determine a datum or reference from which the c.g. of each component will be measured. The tip of the booster's nosecone is used for the datum.
2. List each component and its weight.
3. Measure each component's c.g. location from the datum to obtain the lever arm.
4. Multiply each component's weight by its lever arm length to obtain the moment arm.
5. Sum up all the weights to obtain total weights and moments to obtain total moment.
6. Divide the total moment by the total weight to obtain the c.g. location as measured from the datum.

These calculations are shown in the following table for the booster with all propellants removed. The location of the c.g. is shown in Figure 25 on page 81.

COMPONENT	WEIGHT (LBS)	DISTANCE (FEET)	MOMENT (FOOT-LBS/1000)
Nose Structure with Avionics	8131.0	20.0	162.2
Canard	4000.0	74.0	296.0
Structure 1	95700.0	29.0	2775.3
Structure 2	220000.0	74.0	16280.0
Structure 3	156000.0	149.0	23244.0
Structure 4	190000.0	206.0	39140.0
Nose Wheel	1513.0	74.0	112.0
Wings	24292.0	220.0	5344.0
Tail	3378.0	229.0	773.6
Thrust Structure & Therm. Protect.	18488.0	144.0	2662.3
LH2 Tank	12010.0	48.5	582.5
LO2 Tank	28764.0	114.0	3279.0
JP4 Tank	17880.0	175.0	3129.0
Main Gear	7533.0	204.0	1536.7
React. Cont. Sys.	2000.0	20.0	40.0
Air Breath. Eng.	32064.0	74.0	2372.7
Rocket Engines	121200.0	232.8	28215.4
Engine Gimbling	2100.0	214.0	449.4
Vehicle Mounts			
Fore	7000.0	74.0	518.0
Aft	7000.0	206.0	1442.0
Engine Plumbing	8344.0	175.0	1460.2
Hydraulic Syst. & Control Surfaces	5547.0	175.0	970.7
Rocket Engine Ferring	3000.0	224.0	672.0

TOTAL WEIGHT = 975944 lbs
CENTER OF GRAVITY LOCATION = 138.8 feet

TABLE 12. List of Structural Components and Weight

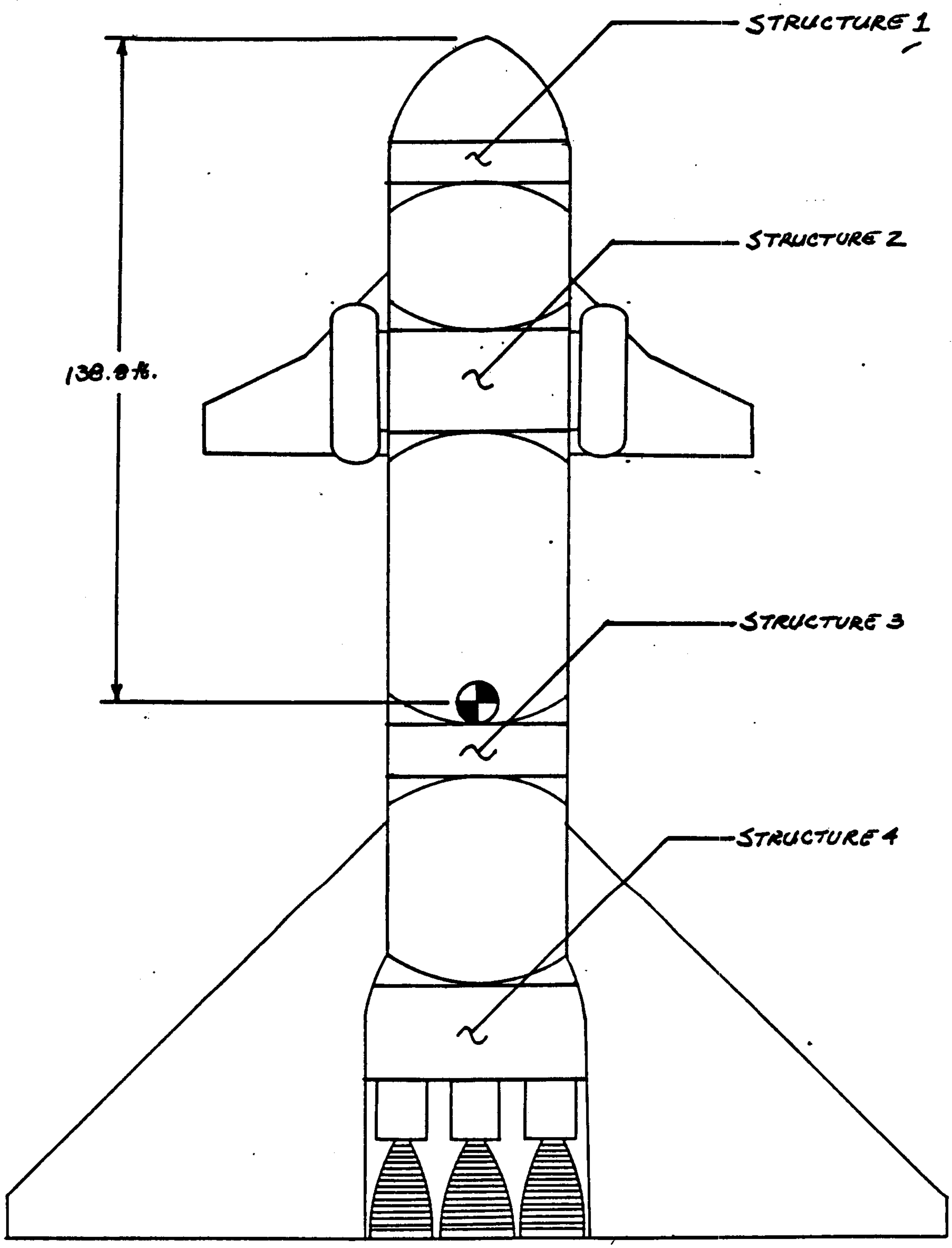


FIGURE 25. C.G. LOCATION WITH TANKS EMPTY

In addition the c.g. was calculated at lift-off, separation, and at landing. These calculations were necessary in order to see how the propellant weights would affect the c.g. at various times during the trajectory. Since the booster is used for two different missions, to boost the shuttle vehicle and to boost the cargo vehicle, it is necessary to find the c.g.'s at various times in the two missions. These c.g. calculations are shown on the following table.

	CARGO VEHICLE	SHUTTLE VEHICLE
AT LIFT-OFF		
weight	6227150 lbs	6224052 lbs
c.g.	131.69 feet	132.61 feet
BEFORE/AFTER SEPARATION		
weight	1112810 lbs	1112810 lbs
c.g.	145.61 feet	145.61 feet
AT LANDING		
weight	981544 lbs	975944 lbs
c.g.	139.01 feet	138.80 feet

TABLE 13. Weights and C.G. At Lift-off, Separation, and Landing

C-2

COST ESTIMATION ANALYSIS

This cost model contains normalized Cost Estimating Relationships (CERs) for the Flyback Booster Vehicle System. The cost model was developed for the Engineering Cost Group (PP03) of NASA's Marshall Space Flight Center (MSFC) by PRC system services, a Planning Research Corporation Company, located in Huntsville, Alabama, and is applied to the Flyback Booster/Shuttle Orbiter (FBSO) project of Auburn University's Aerospace Engineering, AE449 FBSO group.

This model provides the FBSO group with the CERs required to prepare the estimates of the Design, Development, Test and Evaluation (DDT & E) and Flight Hardware (FH) manufacturing costs of the Flyback Booster Vehicle System (FBVS). To this end CERs have been provided using data from historical operations of previous Launch Vehicle , Aircraft, Spacecraft, Boosters, Liquid Rocket Motors and Propellant Tank data. These data were used to augment launch vehicle data in several of the subsystem CERs. All data used in this model were extracted from MSFC's REDSTAR (REsource Data And Retrieval) data base system in Huntsville, Alabama.

The recommended CERs contained under each individual subsystem area are the result of regression combinations of X and Y values for the entire data base. To formulate the recommended CERs the data recorded consisted of two parameters, cost and weight, which were graphically presented

on log-log format. From these plots the CERs are determined as a result of curve fitting and determining the mathematical equation for each curve with supporting statistical data. If in the case due to the limited data available, slopes of 0.5 for D & D and 0.7 for FH & A were assumed.

With all the required CERs determined, the Launch Vehicle Cost Model (LVCM) was generated and is shown in Figure 26. The LVCM is arranged in spreadsheet form using the program "LOTUS 123". At this point it is necessary to analyze the basic structure of the LVCM, in order to gain insight on the fundamental operation.

The LVCM structure allows the user to develop cost estimates at the subsystem and system-levels, separated between Non-recurring Design, Development, Test and Evaluation (DDT & E) and Recurring Flight Hardware (FH) unit cost. It serves to categorize those hardware components, subassemblies, and assemblies into generic hardware groups or subsystems. The primary function of the cost model structure is to establish a functional commonality of both system and subsystem-level elements.

Application of the model is based on the building-block approach. A sequence of calculations, utilizing the CERs, are made by the user for each subsystem and system-level element. The known performance characteristic (independent variable or CER parameter) is introduced as the input into the worksheet. All independent variables (CER input) are subsystem dry weight

LAUNCH VEHICLE COST MODEL
STAGE ONE
COST ESTIMATE
DATE: 22-Mar-87
INFLATION : 1.229 FROM 82$ TO 86$
WT RESERVE: 1.23

COST ELEMENT	CER'S INDEP. VAR. DDT&E	TFU	COMPLEXITY FACTOR DDT&E	TFU	COST (86 $ MILLIONS) DDT&E	TFU
------------	----	----	---	----	----	----
STRUCTURES/TPS	734354.00	734354.00	1.00	1.00	1112.42	1104.14
THERMAL CONTROL						
ENVIRONMENTAL CONTROL	3358.00	3358.00	1.00	1.00	17.32	1.68
BASE HEAT SHIELD	18488.00	18488.00	1.00	1.00	44.15	12.10
WING/TAIL/LEADING EDGE	31670.00	31670.00	1.00	1.00	478.30	86.92
LANDING GEAR	9046.00	9046.00	1.00	1.00	63.68	15.95
AVIONICS	8131.00	8131.00	1.00	1.00	1171.69	119.98
ELECTRICAL POWER	10639.00	10639.00	1.00	1.00	439.15	113.02
PROPULSION (LESS ENGINES)	13444.00	13444.00	1.00	1.00	262.40	34.13
SEPARATION PROVISIONS	2134.00	2134.00	1.00	1.00	41.51	16.49
SURFACE FLIGHT CONTROLS	2129.00	2129.00	1.00	1.00	185.33	50.78
AUXILARY POWER UNIT	6387.00	6387.00	1.00	1.00	252.51	81.36
HYDRAULICS	5547.00	5547.00	1.00	1.00	112.64	28.06
SUBTOTAL	815686.00	815686.00			3326.70	1453.78
STRUC. TOOLING	1104.14		1.00		1631.19	
SYS. TST. HRDW. & ASSEMBLY	1453.78		1.00		1817.65	
SYS. TST. OPS.	1817.65		1.00		520.45	
SUBTOTAL					7295.99	1453.78
GSE	7295.99		1.00		875.05	
SUBTOTAL					8171.04	1453.78
SE&I	8171.04	1453.78	1.00	1.00	663.66	94.64
SUBTOTAL					8834.70	1548.42
PROGRAM MANAGEMENT	8834.70	1548.42	1.00	1.00	233.96	41.12
SUBTOTAL					9068.66	1589.53
ENGINES (CV=W*Tv*ISP*Pc)	21706454000.00	130238720000.00	1.00	1.00	91.21	0.21 CE
SUBTOTAL					9159.87	1589.74
FEE (14%)					1282.38	222.56
PROG. SUPPORT (3% DEV., 2% PROD.)					313.27	36.25
SUBTOTAL					10755.52	1848.55
COST CONTINGENCY (15%)					1613.33	277.28
TOTAL					12368.84	2125.84

FIGURE 26. Launch Vehicle Cost Model

with the exception of rocket engines which require a composite variable of engine dry weight X vacuum thrust X Isp X chamber pressure. The subsystem dry weights are the result of weight estimation performed on the FBSO, which are also documented in this report. The resulting output of each subsystem element is performed and a cost estimation of DD & E and Theoretical First Unit (TFU) cost is calculated. This is presented in the far right hand column of the LVCM worksheet. The following documentations are a simplified break down of all the subsystems used in determination of the cost estimation of the FBVS. Each subsystem includes all the historical data utilized to determine the recommended CERs and includes a basic break down and description of what each subsystem consists of:

(1) STRUCTURES/THERMAL PROTECTION SYSTEM:

All CERs are the result of historical structures data from the S-IC, S-II, S-IVB and Centaur boosters, Shuttle Orbiter, External Tank and IUS. All visible tooling cost have been removed from the structures/TPS cost to give a more accurate Design and Development Cost (D & DC).

This element includes the forward, mid, aft fuselage and crew module, frame structure, intertank, thrust structure, tank support structure and combined LOX and fuel tanks. Thermal control related to these elements is included in the weight and cost.

(2) THERMAL CONTROL:

The CERs are the result of historical data from S-IC, S-II and SO. This subsystem covers those thermal control items not in the other subsystem. Due to limited data available, all slopes have been assumed to be 0.5 for D&D and 0.7 for FH & A.

This subsystem is broken into two parts:

Base Heat Shield: This element protects the thrust structure, propellant tanks and other elements from excessive heat and gases given off by the engines during their burn.

Environmental Control: This element provides temperature, humidity, and hazardous gases control to the forward and aft mounted equipment containers.

(3) WING/TAIL/LEADING EDGE:

These CERs are from historical data from the C-5A, C-141, KC-135, C-130A aircraft and Shuttle Orbiter.

this element provides the FBVS with aerodynamic lift and stabilization functions. Major elements included are wings, tail and leading edges. This system consists of the wing box structure, leading and trailing edge structure and leading and trailing edge control surfaces.

(4) LANDING GEAR:

The CERs are from historical data from the C-5A, C-141, KC-135, C-130A aircraft and Shuttle Orbiter.

This subsystem provides the launch vehicle with the

capability to land. Major elements included are;

* Landing Gear
* Supports
* Landing Gear Hydraulics

(5) AVIONICS:

Historical data from numerous space programs were used to develop these CERs.

This subsystem serves as the control element of the FBVS. Its functions include guidance, navigation and control; display and control of vehicle functions and status; sensing to monitor vehicle operations; and communications and data handling to assure proper operation of the vehicle subsystem during its mission. The Avionics Subsystem is composed of four main elements;

* Guidance, Navigation, and Control
* Display and Controls
* Instrumentation
* Communications and Data Handling

(6) ELECTRICAL POWER:

Historical data from S-IC, S-II, S-IVB, Centaur-D, IUS and Shuttle Orbiter were used to develop these CERs.

This element includes the power source; electrical power air-conditioning, distribution, and control equipment. The subsystem provides the power, from batteries or fuel cells

through power distribution system, to operate the pressurization, propellant management, engine controls, staging and separation systems, controls for hydraulics systems, telemetry, display and controls, data handling, and communications systems.

(7) PROPULSION SUBSYSTEM (LESS ENGINES):

These CERs are develop from historical data from Centaur-D, S-IC, S-II, S-IVB, External Tank and Shuttle Orbiter.

This subsystem provides the launch vehicle rocket engines with fuel. Major elements included are;

* Propellant Feed System
* Recirculation System
* Propellant Management System
* Pressurization System
* Plumbing, Valves and Lines
* Fuel Fill, Drain and Vent System

(8) SEPARATION SUBSYSTEM:

Historical data from the Shuttle Orbiter -ET Separation System, S-II Separation System and the Shuttle SRB were used to develop these CERs.

The subsystem provides a means of separating the shuttle orbiter or from the FBVS as their requirement for the mission ends.

(9) SURFACE FLIGHT CONTROLS:

Historical data from the Shuttle Orbiter surface flight controls and actuators were used to develop these CERs.

This subsystem provides the flight control surfaces and the mechanisms required for proper control of the FBVS during atmospheric flight.

(10) AUXILARY POWER UNIT:

Historical data from the Shuttle Orbiter auxiliary power unit were used to develop these CERs.

This subsystem provides the power to drive the hydraulic pumps which provides the power for the TVC and other hydraulic power mechanisms.

(11) HYDRAULIC SYSTEM:

The required CERs are developed from historical data from the Shuttle Orbiter hydraulic system. Due to the limited data available assumed slopes of 0.5 for D & D and 0.7 for FH & A were performed.

This subsystem consists of components required for generation, control distribution, monitoring, and use of hydraulics. The hydraulic power operates the aerosurface controls (aileron, eleron, rudder/speed brake, and body flaps); the FBVS main engine control valves, lockup and unlock of landing gear; operate main wheel brakes and provide nose

wheel steering.

At this point it is necessary to recognize that all estimated cost of each subsystem are based upon certain assumptions and ground rules established prior to the initial study. This is mainly attributed to the amount of progress obtained so far in the design of the FBSO project and not all needed information is available at this time. For a list of all ground rules and assumptions made in determining the LVCM, see Appendix A . It is anticipated that, as the Space Transportation System (STS) progresses, these data will become available to support a more accurate cost estimation.

After all assumptions were established the LVCM needed to be updated to represent current dollars, since all the recommended CERs were determined using $82. This was accomplished by introducing an inflation factor of 1.229. Upon using this value all DDT & E and TFU cost estimation were automatically updated to $86. This factor was found using the NASA R&D Inflation Indices provided by MSFC.

Another item of importance of the LVCM is a factor titled, Complexity Factor (CF). These CFs have a direct multiplicative effect upon the cost and determine the complexity of the structure to that of normal historical experience. It has been determined from previous cost models from NASA's MSFC, the CFs were typically found to be 0.75 < 1.0 <1.25. Since at this point in time little is known on the

complexity of the structure, a CF of 1.0 was used.

Once the basic procedures are defined for the cost model, the application of the CERs to obtain both DD & E and TFU cost is straight forward. First the analyst obtains both D&D and FH&A subtotals by exercising the subsystem-level CERs contained in the LVCM. Using these subtotals as a starting point, the analyst then estimates the total DDT & E and TFU cost by varying the performance characteristics (subsystem dry weights). Subsequently, this produces an accurate representation of the final cost estimation of the FBSO project.

CONCLUSIONS

The results of each of the subgroup's studies have been tated. From the trajectory analysis, it is evident that the ?TS achieves a substantial altitude of nearly 52 nautical les before staging occurs. This high altitude tended to ate some difficulties in the return trajectory analysis. booster coasts to an apogee altitude of nearly 138 cal miles. Due to low density values at such high ides, the booster experiences little aerodynamic drag and ates rapidly during the ballistic reentry. As a result acceleration high G levels are experienced when the ed rate of descent is imposed. As mentioned earlier, lem might be simplified by using a perigee injection and should be examined in further detail in the espite the minor problems, the booster's return t was successfully modeled by the program DUAL. The b was found to be capable of boosting either of the two second stages to a desired orbit and returning to the launch site for a horizontal landing.

From the structures group it is apparent that the shapes and dimensions of the booster's propellant tanks can easily be determined. From the tank sizes the dimensions of the booster's fuselage were dictated. The booster was calculated to be 193 feet in length with a diameter of 32 feet. The materials for the booster's structure together with the

allowable temperature extremes are also determined by the structures group. The booster is capable of completing the entire trajectory without exceeding the temperature extremes calculated.

The performance analysis and configuration design of the booster is conducted by the aerodynamics group. The booster is very similar in design to the one created by NASA. Due this fact, the aerodynamic data used by NASA is also used in this analysis. The major difference in the design is the mass of the booster. The aerodynamics group gives a detailed report on the performance characteristics of the booster. It is apparent that the booster is capable of producing the required lift necessary for the return trajectory, despite its large mass of 880,110 pounds. An attempt is made to calculate the required performance coefficients without the use of NASA's values. This will be an area for future work.

The propulsion groups successfully calculated the thrust requirements during both phases of flight, and found present engines to fulfill these requirements. Studies were conducted on the design and operating characteristics of the engines. Both the rocket engines and the air breathing engines are capable of propelling the booster as dictated by the trajectory analysis.

The final analysis deals with the weights and costs of the booster. Both areas of study are conducted with the use of NASA programs. The booster was broken into its various

components and each component examined for its value of weight. The weights were then totaled and found to match the 880100 pounds dictated by the trajectory analysis. The cost analysis shows that the booster, despite its high cost, is finacially feasible.

In an overview, the booster designers have generated the necessary parameters, equations and programs to provide the preliminary analysis of the flyback booster's potential for design and development. This advanced booster design is found to be very practical for the missions presently being designed in the space field. The booster is larger than expected, but the technology presently exists for it to be design to withstand the forces which result from its large mass. The results of this paper prove promising for future analysis of an Advanced Space Transportation System.

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APPENDIX A
(Derivations and Assumptions)

Aerodynamic Theory and Equations

I.) Method for Finding Aerodynamic Center

The aerodynamic center location of a wing-body configuration is given as follows:

Xac'/Cre=(Xac'/Cre)N+(Xac'/Cre)W(B)*CL W(B)+(Xac'/Cre)B(W)*CL B(W

CL N + CL W(B) + CL B(W)

where

CL N = (CN)B * (*d**2)/(4*Sw)

CL W(B) = KW(B) * (CL)e * (Se/Sw)

CL B(W) = KB(W) * (CL)e * (Se/Sw)

Here KW(B) and KB(W) are interference factors obtained from Figure (27). The subscript W(B) refers to the exposed wing in the presence of the body and B(W) refers to the body in the presence of the wing.

The static margin, (hn - h), is used as indication to the stability of an aircraft, where;

hn is defined as the non-dimensional distance from neutral point or aerodynamic center to some reference point.

h is defined as the non-dimensional distance from center of gravity to the same reference point.

If (hn - h) is positive the plane is statically stable.

II>) Method for Finding Moment Coefficient

The following equation is used to find the pitching-moment coefficient based on the product of the planform area and root chord:

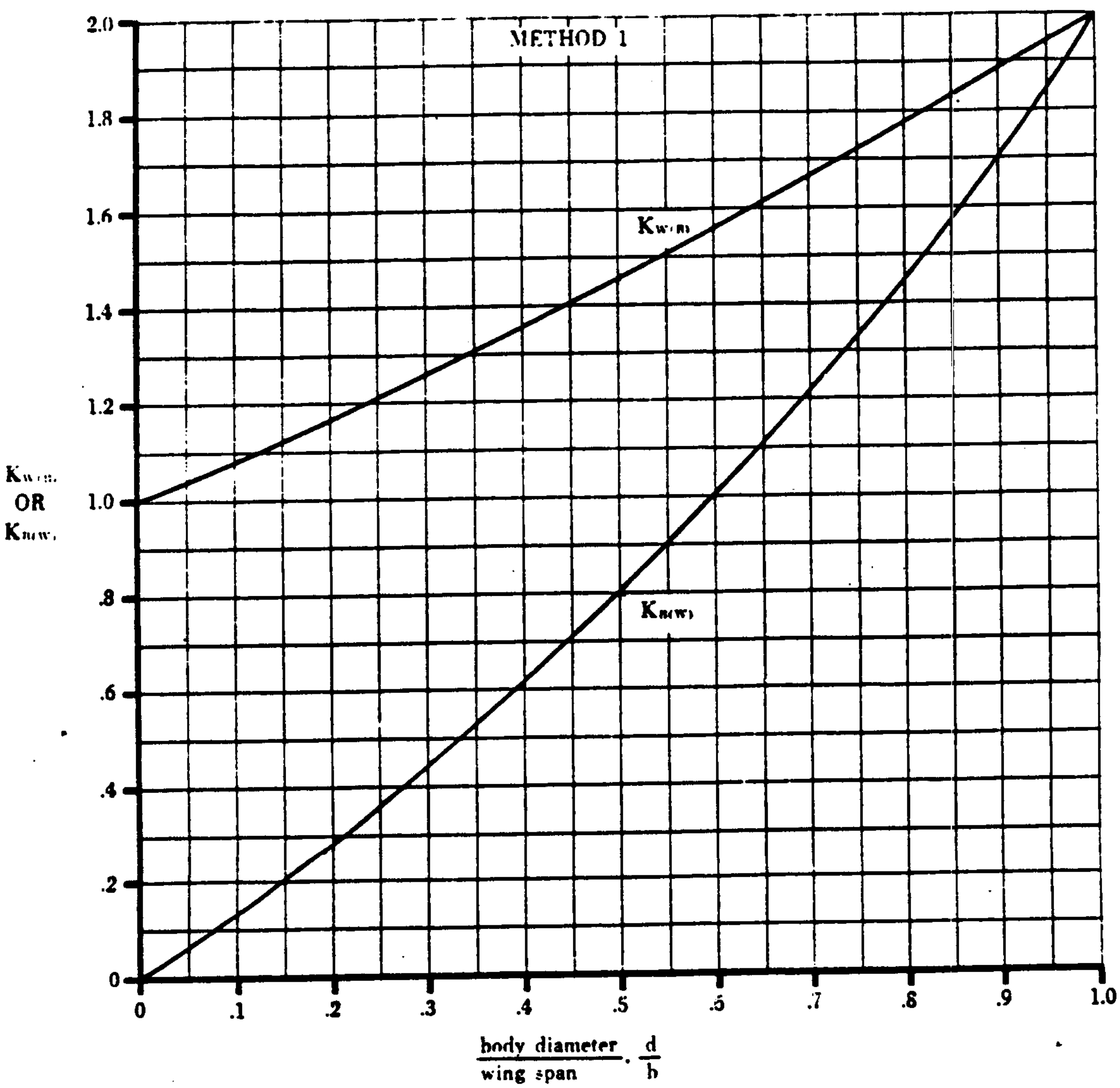

FIGURE 27. Interference Factors

$$Cm = CmNO + (Xm/Cr - Xcp/Cr) * CN' \qquad (6)$$

where

CmNO is the zero normal force pitching moment coefficient. For symmetrical configurations, as in our case, CmNO = 0.0

Xm/Cr is the distance from the nose of the configuration to the desired moment reference center measured in root chords

Xcp/Cr is the distance from the nose of the configuration to the center of pressure

CN' is the normal force variation with angle of attack

III.) Method for Calculating Subsonic Characteristics

Due to the numerous equations necessary to write the program, only the most important equations will be mentioned here. The drag coefficient at a given angle of attack is given by:

$$CD = (CDO)wb + (CDi)wb + (CD)misc \qquad (1)$$

where (CD)misc takes into account drag contributors. The

parasite drag coefficient for the wing body combination is found from:

$$(CDo)wb = (CDo)w + (CDo)b * Sb / Sref \qquad (2)$$

The induced drag coefficient for the wing body combination is found from:

$$(CDi)wb = (CDi)w + [CD(\ )]b * Sb / Sref \qquad (3)$$

After calculating CD, CL was found by rearranging the equation:

$$CD = CDob + CL2 / (\ pi * A * E\ ) \qquad (4)$$

to get:

$$CL = sqrt\ [\ (\ CD - CDowb\ ) * pi * A * E \qquad (5)$$

IV.) Method for Calculating Supersonic Characteristics

The main equation for this calculation is given by;

$$
\begin{aligned}
CL \ = &\ (CL\ )'e\ [KN + KW(B) + KB(W)]'Se'/S' \\
&+ (CL\ )''e\ [KW(B) + KB(W)]''\frac{q''}{q}\frac{S''}{S'}\frac{Se''}{S''} \\
&+ (CL\ )W''(v)
\end{aligned}
$$

where

CL is the lift-curve slope of the wing-body tail

combination.

$(C_{L_\alpha})_e$ is the lift-curve slopes of the exposed forward and aft surfaces.

$\frac{q''}{q}$ is the average dynamic-pressure ratio acting on the aft surface.

S is the projected wing area.

S_e is the exposed wing area.

(') denotes the canard.

('') denotes the wing.

The equation of supersonic zero lift drag coefficient used was:

$$C_D = (C_{D_o})_{WB} + (C_{DL})_{WB}$$

where

C_D is the drag coefficient at supersonic speeds

$(C_{D_o})_{WB}$ is the zero lift drag of the wing body configuration

$(C_{DL})_{WB}$ is the drag due to lift of the wing body configuration

LIQUID ROCKET ANALYSIS
ASSUMPTIONS & DERIVATION OF EQUATIONS

The use of ideal rocket analysis takes into consideration the following assumptions for the thrust chamber:

1. The working fluid (propellant products) is a perfect gas of constant composition.
2. The chemical reaction is equivalent to a constant pressure heating process.
3. The expansion process is steady, one-dimensional,and isentropic.
4. Chemical equilibrium is established within the combustion chamber, and does not shift in the nozzle.

With these assumptions, equations for the analysis of the chamber can be developed. Through the application of the energy equation:

$$Q + W = (h+V^2/2+gz)\rho V\ dA \qquad (1)$$

in combination with the heating process, the following equation is obtained.

$$Tt_2=Tt_1 + Qr/Cp \qquad (2)$$

This equation can be further developed assuming an isentropic nozzle expansion.

$$Ue^2 = 2\ Cp\ Tt_2\ [1-(Pe/Pt)^{\tau-1/\tau}] \qquad (3)$$

Assuming that the nozzle exit pressure is equal to the

atmospheric pressure at sea level, the exit velocity can now be related to the specific impulse by:

$$Ue = Isp*g_o \qquad (4)$$

thus, an expression is obtained for the total chamber pressure.

$$Pt_2 = Pe/(1 - Ue^2 \ / \ 2 \ g_c \ Cp \ Tt_2)^{\tau/\tau-1} \qquad (5)$$

From the calculation of the total chamber pressure, the exit Mach number can be evaluated by:

$$Pt_2 \ / \ Pe = \{1 + [(\tau - 1)/\ 2] \ Me^2 \ \}^{\tau/\tau-1} \qquad (6)$$

solving for Me, the following relation is found.

$$Me = \{(2/\ \tau -1) \ [(Pt_2 \ /Pe)^{\tau-1/\tau} \ -1]\}^{1/2} \qquad (7)$$

After calculating the exit Mach number, the nozzle areas at the throat and exit can be determined using the law of conservation of mass.

$$m = p \ V \ A = constant \qquad (8)$$

assuming: $p = P \ / \ R \ T$ and $V = a \ M$

$$m = (\ P \ / \ R \ T \ ) \ a \ M \ A \qquad (9)$$

Putting both static condition pressures and temperatures into stagnation quantities using these equations:

$$P = Pt_2 \ \{1 + [(\tau - 1)/\ 2 \ ]M^2\}^{\tau/\tau-1} \qquad (10)$$

$$T = Tt_2 \ \{1 + [(\tau - 1)/\ 2 \ ]M \ \} \qquad (11)$$

the mass flow rate can be written now in terms of stagnation conditions.

$$m = (Pt \ / \ R \ Tt) \ a \ M \ A \ \{1 + [(\tau - 1)/\ 2 \ ]M^2\}^{1/\tau-1} \qquad (12)$$

Using the definition of the speed of sound

$$a = (\tau\ g_c\ \ R\ T)^{1/2} \qquad (13)$$

$$a = \tau\ g_c\ \ R\ Tt\ \{1 + [(\tau - 1)/\ 2]M^2\}^{1/2} \qquad (14)$$

the mass flow rate can be written as:

$$m = Pt\ A\ (\tau\ g_c/R\ Tt)^{1/2}\ \{1+[(\tau-1)/2]M^2\}^{\tau+1/2(\tau-1)} \quad (15)$$

Realizing the mass flow rate is:

$$m = (T\ /\ Isp)\ (\ g_c\ /\ g_o\ ) \qquad (16)$$

the area can be evaluated in terms of the mass flow rate.

$$A = m\ /(Pt_2\ M)*(R\ Tt_2/\tau\ g_c)\ \{1+[(\tau\ -1)/\ 2]M^2\} \qquad (17)$$

After further algebraic manipulations, and using throat qualities of M = 1 and A* = A, the throat area is equal to the following expression.

$$A^* = (m/Pt_2)\ [R\ Tt_2/g_c\ \tau\ (2/\tau+1)^{\tau+1/\tau-1}]^{1/2} \qquad (18)$$

Now using the ratio of mass flow rates at the throat and the exit an equation results which relates the exit area to the exit Mach number and A*

$$Ae = (A^*/Me)\ [(2/\tau+1)\ \{1+[(\tau\ -\ 1)/2]Me^2\}]^{\tau+1/2(\tau-1)} \qquad (19)$$

From these areas the diameters of the throat and exit can be calculated.

$$d = 2\ (A/\pi)^{1/2} \qquad (20)$$

Thrust chamber characteristics can be written in terms of the characteristic velocity, C* , and the thrust coefficient, CT.

$$C^* = (Pt_2\ A^*\ )/(m) \qquad (21)$$

and

$$CT = T/(Pt\ A^*\ ) \qquad (22)$$

GROUND RULES AND ASSUMPTIONS

FOR COST ESTIMATION

* All cost and equations are shown in 1986 dollars (in millions).
* The complexity of the structure has the same complexity of normal historical experience.
* Flight Hardware and Assembly costs are included in the Flight Hardware cost at the subsystem level.
* Systems Test Hardware and Assembly costs, Structural Tooling and Systems Test operations are not addressed in the cost model at this time.
* Slopes for the FBSO subsystems were assumed by utilizing data from Aerospace Corporation SRM cost model and PRC developed CERs from other cost models.
* The inflation factor is based on NASA Comptroller Indices, date January 1986.
* An historical cost data utilized in this model are contained within the REDSTAR Data Base System.
* No institutional cost are included.
* Contractor (Prime) fee and/ or profit is excluded.

APPENDIX B

Programs, Input & Output Files

PROGRAM I

TRAJECTORY TO STAGING

ORIGINAL PAGE IS
OF POOR QUALITY

```
ALT=0
DT=0.1
RANG=0
ISP1=320
ISP2=380
INPUT"ENTER INITIAL MASS";M1
M=M1
INPUT"ENTER ALTITUDE FOR G-TURN";AGT1
AGT=AGT1
TH=1.35*M
INPUT"ENTER LIFTOFF THRUST (CARGO ENGINES)";TH21
TH2=TH21
TH1=TH-TH2
MDOT1=TH1/ISP1
MDOT2=TH2/ISP2
ISP=TH/(MDOT1+MDOT2)
MDOT=TH/ISP
MC1=0
MC2=0
Y=0
X=0
G0=32.2
R0=2.092E+07
V0=0
V=V0
T=0
PRINT
PRINT"TIME         V            RANG          ALT          G'S         MASS         THRUST"
PRINT
Z$="###.#        ####.#       ######        ######       #.##        ######       ......"
M2=M-MDOT1*DT-MDOT2*DT
G=G0*(R0/(R0+ALT))^2
DV=(DT*G0*LOG(M/M2))-G*DT
AG1=DV/G0/DT
REM IF AG1>3 THEN 400
CHY=0.5*(V0+V0+DV)*DT
IF Y+CHY>AGT THEN 380
Y=Y+CHY
ALT=Y
V=V0+DV
AG=AG1
GOTO 460
DV=3*G0
M2=M/EXP((DV+G*DT)/ISP/G0)
A=DV/G0
V=V0+DV
Y=Y+0.5*(V0+V)*DT
ALT=Y
T=T+DT
DM=M-M2
MDT=DM/DT
TH=MDT*ISP
MC1=MC1+TH1/ISP1*DT
MC2=MC2+(TH-TH1)/ISP2*DT
MC=MC1+MC2
PRINT USING Z$;T,V,RANG,ALT,AG,M,TH
REM
M=M2
V0=V
GOTO 280
PRINT
PRINT"TIME         V            RANG          ALT          G'S         MASS         THRUST"
```

ORIGINAL PAGE IS
OF POOR QUALITY

```
20      M=M+MDOT1*DT+MDOT2*DT
30      DPSI=(PI/180)*0.1
0       PSI0=PI/120
0       PSI=PSI0
0       GAM=1
0       Z0=SIN(PSI0)/(1+COS(PSI0))
0       PRINT"TIME      V       RANG        ALT        PSI    GAM    THRUST    MASS    G'S"
0       F$="###  #####   #######  #######    ##.#    ##.#  #######  ######   #.##"
0       PRINT
0       N=TH/M
0       PRINT USING F$;T,V,RANG,ALT,PSI*180/PI,GAM,TH,M,AG
0       C=V0/(Z0^(N-1))/(1+Z0^2)
0       PSI=PSI+DPSI
0       Z=SIN(PSI)/(1+COS(PSI))
0       V1=C*(Z^(N-1))*(1+Z^2)
0       IF V1>7000 THEN 1390
0       V=V1
0       DT=C/G*Z^(N-1)*(1/(N-1)+Z^2/(N+1))-C/G*Z0^(N-1)*(1/(N-1)+Z0^2/(N+1))
0       DX=0.5*(V0*SIN(PSI0)+V*SIN(PSI))*DT
0       DY=0.5*(V0*COS(PSI0)+V*COS(PSI))*DT
0       AG=(V-V0)/DT/30
0       IF AG>3 THEN 1060
0       X=X+DX
5       Y=Y+DY
0       THETA=ATN(X/(R0+Y))
0       GAMMA=PSI-THETA
0       GAM=GAMMA*180/PI
0       ALT=(Y+R0)/COS(THETA)-R0
0       RANG=R0*THETA
0       T=T+DT
0       PSI0=PSI
0       Z0=Z
0       V0=V
0       IF ALT<100000 THEN 1020
0       ISP1=290
0       ISP2=462
00      MDOT1=TH1/ISP1
10      MDOT2=TH2/ISP2
20      M=M-MDOT1*DT-MDOT2*DT
30      IF M<0 THEN 1390
40      G=G0*(R0/(R0+ALT))^2
50      GOTO 720
60      N=3*G0/G*((1-Z0^2)/(1+Z0^2))
70      C=V0/Z0^(N-1)/(1+Z0^2)
80      V=C*Z^(N-1)*(1+Z^2)
90      D1=C/G*Z^(N-1)*(1/(N-1)+Z^2/(N+1))-C/G*Z0^(N-1)*(1/(N-1)+Z0^2/(N+1))
00      DX=0.5*(V0*SIN(PSI0)+V*SIN(PSI))*DT
10      DY=0.5*(V0*COS(PSI0)+V*COS(PSI))*DT
20      X=X+DX
30      RANG=R0*THETA
40      Y=Y+DY
50      ALT=(Y+R0)/COS(THETA)-R0
60      T=T+DT
70      PSI0=PSI
80      Z0=Z
90      V0=V
00      TH=N*M
10      TH1=TH-TH2
20      MDOT1=TH1/ISP1
230     MDOT2=TH2/ISP2
240     M=M-MDOT1*DT-MDOT2*DT
250     MC1=MC1+MDOT1*DT
260     MC2=MC2+MDOT2*DT
270     MC=MC1+MC2
280     IF M<0 THEN 1390
```

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```
300     PSI=PSI+DPSI
310     THETA=ATN(X/(RO+Y))
320     GAMMA=PSI-THETA
330     GAM=GAMMA*180/PI
350     PRINT USING F$;T,V,RANG,ALT,PSI,GAM,TH,M,AG
360     IF V>7000 THEN 1390
370     Z=SIN(PSI)/(1+COS(PSI))
380     GOTO 1060
390     PRINT
391     PSI=PSI-DPSI
392     PRINT V1
400     PRINT"TIME      V      RANG    ALT    PSI    GAM    THRUST    MASS    G'S"
420     PRINT"------------------------------------------------------------------------"
421     PRINT
422     PRINT"                    S E P A R A T I O N                    "
423     PRINT
424     PRINT"------------------------------------------------------------------------"
430     PRINT"time      v      rang    alt    psi    gam2    thrust    [illegible]    g's"
[illegible] PRINT
[illegible] F$="### ####.# ####### ####### ###.# ###.# ####### ####### # .##"
[illegible] M=M-102000 (1,112,810)          15%
[illegible] ISP=ISP2
[illegible] TH=TH2
[illegible] MDOT=TH/ISP
[illegible] V0=V
[illegible] PSI0=PSI
[illegible] GAM2=PI/2-+ THETA
[illegible] GAMM=GAM2*180./PI
[illegible] Z0=SIN(PSI0)/(1+COS(PSI0))
[illegible] N=TH/M
[illegible] PRINT USING F$;T,V,RANG,ALT,PSI*180./PI,GAMM,TH,M,AG
[illegible] C=V0/(Z0^(N-1))/(1+Z0^2)
[illegible] PSI=PSI+DPSI
[illegible] THETA=ATN(X/(RO+Y))
[illegible] GAM2=PI/2+THETA
[illegible] GAMM=GAM2*180./PI
[illegible] IF PSI>GAM2 THEN 2080
[illegible] Z=SIN(PSI)/(1+COS(PSI))
[illegible] V=C*Z^(N-1)*(1+Z^2)
[illegible] IF V>25000 THEN 2080
[illegible] DT=C/G*Z^(N-1)*(1/(N-1)+Z^2/(N+1))-C/G*Z0^(N-1)*(1/(N-1)+Z0^2/(N+1))
[illegible] DX=.5*(V0*SIN(PSI0)+V*SIN(PSI))*DT
[illegible] DY=.5*(V0*COS(PSI0)+V*COS(PSI))*DT
[illegible] AG=(V-V0)/DT/G0
[illegible] X=X+DX
[illegible] Y=Y+DY
[illegible] THETA=ATN(X/(RO+Y))
[illegible] RANG=RO*THETA
[illegible] ALT=(Y+RO)/COS(THETA)-RO
[illegible] T=T+DT
[illegible] PSI0=PSI
[illegible] Z0=Z
[illegible] V0=V
[illegible] M=M-MDOT*DT
[illegible] IF M<0 THEN 2080
[illegible] G=G0*(RO/(RO+ALT))^2
[illegible] GOTO 1520
[illegible] REM
[illegible] PRINT
[illegible] PRINT"INITIAL MASS        :";M1
[illegible] PRINT"ALT FOR G-TURN      :";AGT1        111
[illegible] PRINT"THRUST (CARGO VEHICLE) :";TH21
[illegible] PRINT"FINAL VELOCITY :";V
[illegible] PRINT"FINAL MASS :";M
[illegible] PRINT"FINAL PSI   :";PSI*180/PI
```

```
115    PRINT"FINAL GAMM :";GAMM
116    INPUT"TRY AGAIN Y=1 N=2";TRY
117    IF TRY<2 THEN 15
```

```
      PROGRAM DUAL
                                              26 AUG 1986
      IMPLICIT DOUBLE PRECISION (A-H,O-Z)
      INCLUDE 'DUAL.CMN'
      CHARACTER*1 IDUM
      CHARACTER*8 HNM(5)
      CHARACTER*12 FNM,GNM
      DATA HNM/'PLOT1   ','PLOT2   ','PLOT3   ','PLOT4   ','PLOT5   '/

      WRITE(IOUT,200)
      READ (IIN,210) FNM
      WRITE(IOUT,220)
      READ(IIN,210) GNM
      OPEN(UNIT=ICR,FILE=FNM,STATUS='OLD')
      OPEN(UNIT=IPR,FILE=GNM,STATUS='UNKNOWN')
      OPEN(UNIT=IMT,FILE='SCRATCH',STATUS='SCRATCH',FORM='UNFORMATTED')
      WRITE(IOUT,230)
      READ(IIN,210) DATE

      CALL EDATA
      NPLOT=0

   10 DO 20 I=1,96                      ! INITIALIZE INTEGRATION ARRAY
   20 A(I)=ZERO
      REWIND IMT

      READ TITLE                                                 CARD 1
      READ OPTIONS                                               CARD 2
      POSITION AND VELOCITY                                      CARD 3
      FLY ALT,ENG ON ALT,GUIDE CHECKS                            CARD 4
      ALP,ALPC,ALPHA PROFILE                                     CARD 5
      ALIM,ALPHADOT PROFILE                                      CARD 6
      BET,BETC,BETT,BETM,BETL,BETX                               CARD 7
      BLIM,BETADOT PROFILE                                       CARD 8
      THRUST PROFILE,ENGM,ENGF                                   CARD 9
      WEIGHTS-EMPTY,FUEL,ON ORBIT,RESERVE  RATES-APU,VENT        CARD 10
      ACC LIMIT,AERO PERTRB,FLY THRST PERTRB,SITE 8              CARD 11
      TIME,DT,FLY DT,PRINT TIME,DTP,FLY DTP                      CARD 12

      READ(ICR,210,END=50) TITLE
      WRITE(IOUT,215) TITLE
      READ(ICR,100) NABT,NATM,NFLY,NLIM,NLND,NPRN,
     1NPLT,NPUT,NQAL,NTRY,NTYP,NVNT,NWIN,NMET

      READ (ICR,101) FLAT,FLON,R,V,GAM,HDNG,
     1ALTH,ALTE,RDTN,TCON,TSTG,VTST,
     2ALF,ALFD,BETA,BETAD,THRF,
     3WGTE,WGTF,WGTO,WGTR,FMDA,FMDV,
     4ACCL,DCD,DCL,DTH,XLLA(8),XLLO(8),
     5T,DT,DTF,TP,DTP,DTPF

      IF (NABT.EQ.2) FMDV=ZERO          ! CHECK CROSS COUPLING
      IF (NLND.LE.0) NLND=1
      IF (FMDA.GT.ZERO) FMDA=-FMDA
      IF (FMDV.GT.ZERO) FMDV=-FMDV
      IF (FMDV.EQ.ZERO) NVNT=1


      NPG=1                             ! RESET CASE PAGE NUMBER
      CALL HEADER                       ! PRINT TITLE AND CASE DATE
      WRITE(IPR,217) FNM,GNM
      IF(NPLOT.GE.5) NPLT=1
```

```
      JPR=IPR
      DO 25 J=1,2
      WRITE (JPR,102)NABT,NATM,NFLY,NLIM,NLND,NPRN,        ! OPTION TABLE
     1NPLT,NPUT,NQAL,NTRY,NTYP,NVNT,NWIN,NMET
   25 JPR=IOUT
      WRITE(IOUT,140)
      READ (IIN ,210) IDUM
      WRITE(IOUT,215) TITLE

      WRITE (IPR,103) FLAT,FLON,R,V,GAM,HDNG,              ! PRINT INPUT
     1ALTH,ALTE,RDTN,TCON,TSTG,VTST,
     2ALF,ALFD,BETA,BETAD,THRF
      WRITE (IPR,104) WGTE,WGTF,WGTO,WGTR,FMDA,FMDV,
     1ACCL,DCD,DCL,DTH,XLLA(8),XLLO(8),
     2T,DT,DTF,TP,DTP,DTPF

      CALL HEADER
      WRITE (IPR,105)

      TBEGIN=T                            ! SAVE INITIAL TIME
      CALL TYPE
      CALL RDWIND
      CALL INITIAL
      CALL FUN
      IF(NPRN.EQ.3) CALL HEADER
      CALL DRIVER

      IF(NPLT.EQ.1.AND.NPRN.EQ.2) GO TO 40
      IF(NPLT.EQ.2) THEN
      NPLOT=NPLOT+1
      OPEN(UNIT=IFL,FILE=HNM(NPLOT),STATUS='UNKNOWN')
      WRITE(IFL,'(A60,A12)') TITLE,DATE
      IPTS=23
      WRITE(IFL,'(I6)') IPTS
      WRITE(IFL,'(10A8)') 'TIME-SEC','LONG-D  ','GCLAT-D ','ALT-KFT ',
     1        'ALPHA-D ','BETA-D  ','ACCT G''S','DYNP-PSF','Q-BTU/S ',
     2        'QSUM-BTU','MACH    ','VREL-KT ','THRUST-P','WEIGHT-P',
     3        'FUEL-P/S','RANGE-NM','FPA-D   ','AZM-D   ','ACCX G''S',
     4        'WPOS G''S','ACCT I  ','THR-%   ','Q*ALPHA '
      END IF

      REWIND IMT
      NBLKS=15
   30 READ (IMT,END=40) T,ACC,ACCX,ACCZ,DACZ,ALP,ALT,BET,CD,CL,CRNG,
     1CTMP,DRTG,DYNP,FLAT,FLON,GAM,GAMR,HDNG,NHDP,QALF,QDOT,QTOT,
     2RDOT,RHO,V,VMACH,VRL,WGT,WDOT,XDRGM,XLFTM,XTHRM,THRTL,ACCI

      IF(NPRN.EQ.2) GO TO 34
      IF(NBLKS.GE.10) THEN
      NBLKS =0
      CALL HEADER
      WRITE (IPR,106)
      END IF

      NBLKS=NBLKS+1
      WRITE (IPR,107) T,ALT,V,GAM,DYNP,QDOT,ALP,FLAT,DRTG,ACCZ
      WRITE (IPR,108) NHDP,RDOT,VRL,GAMR,VMACH,QTOT,BET,FLON,HDNG,QALF
      WRITE (IPR,109) RHO,XTHRM,WGT,WDOT,CD,CL

        PLOTTING

   34 IF(NPLT.EQ.2) THEN
      ALT=ALT/1000.
      VRL=VRL/1.68781
      GPOS=-ACCZ
      WRITE(IFL,'(5E16.7)')T,FLON,FLAT,ALT,ALP,BET,ACC,DYNP,QDOT,
     *QTOT,VMACH,VRL,XTHRM,WGT,WDOT,DRTG,GAMR,HDNG,ACCX,GPOS,ACCI,THRTL,
```

```
*QALF
 END IF

 GO TO 30
40 CALL HEADER
 DYNPM=DYNPM/47.88025         ! LBS/SQFT
 QALFM=CNV*QALFM/47.88025     ! DEGS*LBS/SQFT
 FUEL=WGTE-WGT

 JPR=IPR
 WNEG=-ACZPM
 WPOS=-ACZNM
 DO 45 K=1,2
 WRITE (JPR,110) QDOTM,TQDOT,CTPM,TCTPM,STPM,TSTPM,DYNPM,TDYNPM,
1QALFM,TQALF,ACCTM,TACCT,ACCXM,TACCX,ACCYM,TACCY,
2ACCZR,TACCR,WNEG,TACZP,WPOS,TACZN

 IF(NFLY.NE.1) WRITE(JPR,130) FUEL

 JPR=IOUT
45 CONTINUE

 IF(NPLT.EQ.2) CLOSE (IFL)
 GO TO 10
50 STOP
100 FORMAT(20I2)
101 FORMAT (6E13.8)
102 FORMAT (//52X,'OPTION TABLE'//
 130X,8H  NABT =,I4,4X,'MAIN ENGINES OFF / MAIN ENGINES ON'//
 230X,8H  NATM =,I4,4X,'PRA63 ATMOS / STD US62 ATMOS'//
 330X,8H  NFLY =,I4,4X,'NO FLYBACK / FLYBACK / FLYBACK & G.S.'//
 430X,8H  NLIM =,I4,4X,'NO G LIMIT / EXP LIMIT / FIXED LIMIT'//
 530X,8H  NLND =,I4,4X,'CAPE KEN / HOUSTON / EDWARDS / HICKAM'
 6        /,46X,'NAS CECIL / CHERRY POINT / AZORES / INPUT'//
 730X,8H  NPRN =,I4,4X,'FULL PRINT / STATE ONLY / TABLE ONLY'//
 830X,8H  NPLT =,I4,4X,'NO PLOT FILE / WRITE DATA TO PLOT FILE'//
 930X,8H  NPUT =,I4,4X,'INERTIAL / INERTIAL AND ENC / RELATIVE'//
 *30X,8H  NQAL =,I4,4X,'NO LIMIT / Q ALPHA LIMIT'//
 130X,8H  NTRY =,I4,4X,'BEGIN AT THIS SEGMENT'//
 230X,8H  NTYP =,I4,4X,'BOOSTER / ORBITER'//
 430X,8H  NVNT =,I4,4X,'NO VENT / VENT'//
 530X,8H  NWIN =,I4,4X,'NO WIND / 95% INPUT / 95% IN+HEAD'//
 630X,8H  NMET =,I4,4X,'INPUT ENGLISH / METRIC')
103 FORMAT (/,5X,'INPUT VALUES'/
 16H   LATF12.4,6H   LONF12.4,6H ALT/RF12.2,
 26H   VVVF12.3,6H   GAMF12.3,6HEN/HDNF12.3/
 36H  ALTHF12.2,6H  ALTEF12.2,6H  RDTNF12.2,
 46H  TCONF12.3,6H  TSTGF12.3,6H  VTSTF12.3/
 56H   ALPF12.3,6H  ALPCF12.3,4(6X,F12.3)/
 66H  ALIMF12.3,5(6X,F12.3)/
 76H   BETF12.3,6H  BETCF12.3,6H  BETTF12.3,
 86H  BETMF12.3,6H  BETLF12.3,6H  BETXF12.3/
 96H  BLIMF12.3,5(6X,F12.3)/
 *6H THFACF12.3,3(6X,F12.3),6H  ENGMF12.3,6H  ENGFF12.3)
104 FORMAT (
 16H  WGTEF12.2,6H  WGTFF12.2,6H  WGTOF12.2,
 26H  WGTRF12.2,6H  FMDAF12.2,6H  FMDVF12.2/
 36H  ACCLF12.2,6H   DCDF12.2,6H   DCLF12.2,
 46H   DTHF12.2,6H  LAT8F12.2,6H  LON8F12.2/
 56H     TF12.2,6H    DTF12.2,6H   DTFF12.2,
 66H    TPF12.2,6H   DTPF12.2,6H  DTPFF12.2)
105 FORMAT(15X,'OUTPUT UNITS'//
 1 10X,'TIME',16X,'SEC'/
 2 10X,'ALTITUDE',12X,'FEET'/
 3 10X,'VELOCITY',12X,'FEET/SEC'/
```

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     4 10X,'ANGLES',14X,'DEGS'/
     5 10X,'RATES',15X,'FEET/SEC'/
     6 10X,'RANGES',14X,'NAUT MILES'/
     7 10X,'WEIGHT',14X,'LBS'/
     8 10X,'THRUST',14X,'LBSF'/
     9 10X,'HEAT IN',13X,'BTU/FT**2'/
     * 10X,'HEAT RATE',11X,'BTU/FT**2/SEC'/
     1 10X,'PRESSURE',12X,'LBS/FT*2'/
     2 10X,'ACCELERATIONS',7X,'Gs'/
     3 10X,'Q ALPHA',13X,'DEGS*LBS/FT*2')
 106 FORMAT(
     * //,40X,'ENGLISH UNITS'
     1 /,31X,'TIME',2X,'ALTITUDE',3X,'VNRT GAMNRT',3X,'DYNP',3X,
     2 'QDOT',2X,'ALPHA',4X,'LAT',3X,'DRTG',3X,'ACCZ'/
     3 41X,'RDOT',3X,'VREL GAMREL',3X,'MACH',3X,'QTOT',3X,'BETA',4X,
     4 'LON',3X,'HDNG',3X,'QALF'/
     5 42X,'RHO',8X,'THRUST',11X,'WGT',10X,'WDOT',5X,'CD',5X,'CL')
 107 FORMAT ( /25X,F10.3,F10.0,F7.0,F7.2,4F7.1,F7.0,F7.2)
 108 FORMAT (33X,A2,3X,2F7.0,2F7.2,F7.0,2F7.1,F7.0,F7.0)
 109 FORMAT(35X,E10.3,2(5X,F9.0),5X,F9.2,2F7.3)
 110 FORMAT(//15X,'MAXIMUM VALUES'//
     *18X,10HMAX QDOT  ,F7.3,16H BTU/FT2/S TIME ,F8.1/
     118X,10HMAX CTMP  ,F7.0,16H DEGS F    TIME ,F8.1/
     218X,10HMAX STMP  ,F7.0,16H DEGS F    TIME ,F8.1/
     318X,10HMAX DYNP  ,F7.0,16H LB/FT2    TIME ,F8.1/
     418X,10HMAX QALF  ,F7.0,16H D*LB/FT2  TIME ,F8.1/
     518X,10HMAX ACCT  ,F7.3,16H GRAVS     TIME ,F8.1/
     618X,10HMAX ACCX  ,F7.3,16H GRAVS     TIME ,F8.1/
     718X,10HMAX ACCY  ,F7.3,16H GRAVS     TIME ,F8.1/
     818X,10HMIN ACZR  ,F7.3,16H GRAVS     TIME ,F8.1/
     918X,10HWING -G   ,F7.3,16H GRAVS     TIME ,F8.1/
     *18X,10HWING +G   ,F7.3,16H GRAVS     TIME ,F8.1)
 115 FORMAT(6X,'TIME',7X,'VRL',7X,'FPA',2X,'ALTITUDE',
     14X,'ATTACK',6X,'BANK',6X,'DENSITY'/
     25X,'(SEC)',2X,'(FT/SEC)',5X,'(DEG)',6X,'(FT)',
     35X,'(DEG)',5X,'(DEG)',6X,'(SLUGS)')
 120 FORMAT(F10.3,F10.3,F10.4,F10.2,2F10.3,E13.5)
 130 FORMAT(//15X,'FLYBACK FUEL',F10.2//)
 140 FORMAT(//10X,'HIT RETURN TO CONTINUE')
 200 FORMAT(//10X,'INPUT FILE     ',$)
 205 FORMAT(1X,A60)
 210 FORMAT(A)
 215 FORMAT(/,5X,A60/)
 217 FORMAT(15X,'INPUT FROM ',A12,5X,'OUTPUT TO ',A12)
 220 FORMAT(  10X,'OUTPUT FILE    ',$)
 230 FORMAT(  10X,'DATE           ',$)
     END
     SUBROUTINE DRIVER
     IMPLICIT DOUBLE PRECISION (A-H,O-Z)
     INCLUDE 'DUAL.CMN'
     EXTERNAL FUN

     ILINE='BEGIN ENTRY '
     NHD=2
     NHDP=2HBE

     IF(NABT.EQ.2) GO TO 3
     IF(NVNT.EQ.1) GO TO 6
     TT=-WGTR/FMDV                      ! VENT CONTROL
     TV=T+TT
     IF(TT.GT.ZERO) GO TO 4
   3 IF(FMDV.NE.ZERO) THEN
     NHD=2
     NHDP=2HVN
     ILINE='END VENT   '
     FMDV=ZERO
```

```
      END IF
      NVNT=1
      GO TO 6
    4 IF(TV-T.GT.DT) GO TO 6
      NVNT=3
      FMDV=(TV-T)*FMDV/DT

    6 IF(NABT.EQ.2) GO TO 30          ! CONTROL LOOP
   10 CALL GUIDE
      CALL FUN
      CALL TSTMAX
      IF(NEND.EQ.2) GO TO 40
   20 CALL PRINT
      TF=T+DT
      IF(TF.GT.TP) TF=TP
      CALL RUN(A,T,TF,FUN,7)
      IF(ALT.LT.10.) THEN
      ILINE='CRASHED      '
      NHD=2
      NHDP=2HZZ
      GO TO 40
      END IF
      GO TO (6,4,3),NVNT

   30 GO TO (31,32,34,36),NSTG        ! THRUST CONTROL
   31 NSTG=2
      ILINE='THRUST EVENT'
      NHD=2
      NHDP=2HTE
      WTST=WGTE+WGTR
   32 IF(TSTG.GT.T) GO TO 34
      ENGM=ENGM/TWO
      NSTG=3
      NHD=2
      NHDP=2HTE
   34 FISP=FISPL+BEX*DELP
      THRST=(THRSL+AEX*DELP)*THFAC*ENGM
      FMDM=-THRST/FISP
      WGT=FMASS*GO
      IF (WGT+(FMDA+FMDM)*DT.GT.WTST) GO TO 39
      NSTG=4
      FMDM=(WTST-WGT)/DT-FMDA
      THRST=-FISP*FMDM
      GO TO 39
   36 NABT=1
      FMDM=ZERO
      THRST=ZERO
      IF(WGTR.LE.ZERO) GO TO 38

      FMASS=FMASS-WGTR/GO        ! DROP PERFORMANCE RESERVE
      ILINE='WGT DROP    '
      NHD=2
      NHDP=2HWD
      GO TO 39
   38 ILINE='THRUST EVENT'
      NHD=2
      NHDP=2HTE
   39 XTHRM=THRST/FMASS
      GO TO 10
   40 TP=T
      CALL PRINT
      RETURN
      END
      SUBROUTINE INITIAL
      IMPLICIT DOUBLE PRECISION (A-H,O-Z)
      INCLUDE 'DUAL.CMN'
```

```
      TACCR=T
      TACCT=T
      TACCX=T
      TACCY=T
      TACZN=T
      TACZP=T
      TCTPM=T
      TDYNPM=T
      TQDOT=T
      TSTPM=T

      ACCTM=ZERO
      ACCXM=ZERO
      ACCYM=ZERO
      ACCZR=ACCL
      ACZPM=ZERO
      ACZNM= ONE
      CTMP=ZERO
      CTPM=ZERO
      DACZ=ACCL
      DYNPM=ZERO
      QALFM=ZERO
      QDOTM=ZERO
      STPM=ZERO

      NBET=1
      NBLP=4
      NEND=1
      NSTG=1
      NTF=1
      NTR=1

      IF(NMET.EQ.1) THEN                   ! CONVERT ENGLISH TO METRIC
      R=R*FTM
      V=V*FTM
      ALTH=ALTH*FTM
      ALTE=ALTE*FTM
      RDTN=RDTN*FTM
      END IF

      DO 2 I=1,6                           ! CONVERT CONTROL ANGLES
      ALF(I)=ALF(I)/CNV
      ALFD(I)=ALFD(I)/CNV
      BETA(I)=BETA(I)/CNV
    2 BETAD(I)=BETAD(I)/CNV

      ALP=ALF(1)                           ! SET INITIAL CONTROL ANGLES
      SALP=SIN(ALP)
      CALP=COS(ALP)
      BET=BETA(1)
      SBET=SIN(BET)
      CBET=COS(BET)
      DALP=TWO/CNV

      XLAT=XLLA(NLND)/CNV                  ! LANDING SITE
      XLON=XLLO(NLND)/CNV
      SLLAT=SIN(XLAT)
      CLLAT=COS(XLAT)
      RLN=RE/SQRT(1.+GF*SLLAT*SLLAT)
      XL(3)=RLN*SLLAT

      FLAT=FLAT/CNV                        ! INITIAL LATITUDE
      SLAT=SIN(FLAT)
      CLAT=COS(FLAT)

      ALON=FLON/CNV+T*OMEGA                ! LONGITUDE ADJUSTED
```

```
      ALON=PLON/CNVT+OMEGA              ! LONGITUDE ADJUSTED
      SLON=SIN(ALON)
      CLON=COS(ALON)

      GAM=GAM/CNV                       ! FLIGHT PATH ANGLE
      SGAM=SIN(GAM)
      CGAM=COS(GAM)

      HDNG=HDNG/CNV                     ! HEADING(AZIMUTH)
      SHDN=SIN(HDNG)
      CHDN=COS(HDNG)

      NPUT= 2, HEADING COMPUTED ASCENDING FROM INPUT INCLINATION
           -2, HEADING COMPUTED DESCENDING
      IF(ABS(NPUT).EQ.2) THEN
      SHDN=CHDN/CLAT
      CHDN=SQRT(ONE-SHDN*SHDN)
      IF(NPUT.LT.ZERO) CHDN=-CHDN
      END IF

      REI=RE*SQRT(1.-GF*SLAT*SLAT)      ! OBLATE EARTH RADIUS

      IF(R.LT.REI) THEN                 ! ALTITUDE INPUT
      IF(NMET.EQ.2) R=R*1000.           ! KM
      R=REI+R
      END IF

      X(1)=R*CLON*CLAT                  ! INITIAL POSITION
      X(2)=R*SLON*CLAT
      X(3)=R*SLAT

      XDA(1)=-X(2)*OMEGA                ! ATMOSPHERE VELOCITY VECTOR
      XDA(2)= X(1)*OMEGA
      XDA(3)= ZERO

      CALL COMPASS
      DO 4 I=1,3
    4 XD(I)=CHDN*AA(I)+SHDN*BB(I)
      CALL VUNIT(XD,XD)
      DO 6 I=1,3
      XD(I)=(CGAM*XD(I)+SGAM*CC(I))*V
    6 XDW(I)=ZERO                       ! NO WIND
      DIR=ZERO
      CDIR=ONE
      SDIR=ZERO
      VWIN=ZERO

      ALT=R-REI
      IF(NWIN.NE.1) CALL WIND
      IF(NPUT.EQ.3) THEN                ! INPUT WAS RELATIVE VELOCITY
      DO 8 I=1,3
    8 XD(I)=XD(I)+XDA(I)+XDW(I)
      END IF

      DO 10 I=1,3
   10 XDR(I)=XD(I)-XDA(I)-XDW(I)

      A(1)=X(1)                         ! INITIAL INTEGRATION VARIABLES S
      A(4)=X(2)
      A(7)=X(3)
      A(2)=XD(1)
      A(5)=XD(2)
      A(8)=XD(3)

      XS(1)=X(1)                        ! SAVE INITIAL POSITION
      XS(2)=X(2)
      XS(3)=X(3)
```

```
      XS(3)=X(3)

      CALL VCROSS(XN,XDR,X)            ! ENTRY PLANE NORMAL
      CALL VUNIT(XN,XN)

      TOTAL WEIGHT = BASIC WGT + ASCENT FUEL + ORBIT FUEL + RESERVE
      WGT=WGTE+WGTF+WGTO+WGTR
      FMASS=WGT/GO
      FMD=ZERO
      FMDF=ZERO
      FMDM=ZERO

      SFC=ZERO
      THRST=ZERO
      XTHRM=ZERO
      RETURN
      END
```

INPUT FILE FOR PROGRAM 2

```
FLYBACK#5, BOOSTER, WO 260K, SLT 103.5K
1 1 2 3 1 1 2 3 2 1 1 1 2 1
28.46          -79.87          312418.00   6976.00        53.60           93.98
10000.         35000.          -100.       10.            1000.           2.5
0.             70.             70.         70.            70.             6.
5.             5.              5.          5.             .53             .43
0.             0.              1.          1.             45.             45.
5.             5.              5.          5.             5.              5.
0.             0.              0.          0.             0.              4.
380110.
3.             0.              0.          0.
161.           2.              5.          [illegible]    [illegible]     50.
MSFC AERO #1
12500.,1.,0.,0.,0.,0.,[illegible]
.5,1292.,20000.,.6,[illegible]
-6.333,1000.,.72,5.,200.,.92,
10,10,
0.,6.,8.,10.,15.,20.,30.,40.,50.,60.,
0.,.5,.8,1.0,1.2,1.46,1.95,3.40,4.96,5.6,
.056, .056, .052, .088, .175, .143, .133, .11, .10, .170,
.071, .071, .056, .185, .226, .207, .171, .15, .14, .18,
.094, .094, .121, .237, .261, .244, .215, .17, .15, .186,
.132, .132, .163, .285, .310, .268, .241, .20, .17, .197,
.234, .234, .280, .417, .410, .365, .320, .27, .23, .239,
.234, .234, .60, .75, .70, .60, .55, .446, .394, .118,
.234, .234, .84, 1.05, .98, .84, .72, .623, .559, .562,
.234, .234, .84, 1.80, 1.68, 1.44, 1.32, 1.072, .986, .959,
.234, .234, .84, 1.80, 2.43, 2.03, 1.91, 1.55, 1.40, 1.461,
.234, .234, .84, 1.80, 2.43, 2.49, 2.28, 1.85, 1.70, 1.992,
.025, .025, .018, .084, .056, .073, .073, .05, .04, .006
.370, .370, .338, .547, .476, .434, .354, .26, .22, .06,
.493, .493, .458, .693, .616, .546, .441, .32, .27, .083,
.604, .604, .561, .815, .749, .644, .519, .38, .32, .114,
.807, .807, .764, 1.050, .983, .840, .672, .49, .42, .215,
.807, .807, 1.06, 1.35, 1.30, 1.15, .950, .691, .589, .480,
.807, .807, 1.68, 1.70, 1.64, 1.45, 1.195, .869, .761, .626,
.807, .807, 1.68, 2.27, 2.19, 1.94, 1.599, 1.163, 1.055, .883,
.807, .807, 1.68, 2.27, 2.54, 2.25, 1.86, 1.35, 1.20, 1.025,
.807, .807, 1.68, 2.27, 2.54, 2.43, 2.01, 1.55, 1.30, .993,
GENERIC(M)
6,5,
0.,1524.,3048.,4572.,6096.,10668.,
0.,.2,.4,.6,.8,
62500., 55000., 51877., 50627., 50000.,
59395., 53738., 53034., 53738., 53738.,
56459., 51477., 51477., 52310., 56459.,
53569., 49743., 50702., 53569., 57395.,
50637., 45137., 46234., 48435., 52939.,
43551., 38987., 40116., 42409., 44701.,
.3321, .3321, .3321, .3321, .3321,
.3985, .3985, .3985, .3985, .3985,
.4151, .4151, .4151, .4151, .4151,
.4317, .4317, .4317, .4317, .4317,
.4317, .4317, .4317, .4317, .4317,
.4317, .4317, .4317, .4317, .4317
280.,280.,280.,280.,280.,280.,280.,280.,~~280~~,280.,
280.,280.,
EB
$
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OUTPUT" FILE FOR PROGRAM Z

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PROGRAM DUAL PD34/DAURO/544-0546 PAGE 1

FLYBACK#5, BOOSTER, WO 260K, SLT 103.5K
INPUT FROM zfly.dat . OUTPUT TO chris.dat

OPTION TABLE

NABT =	1	MAIN ENGINES OFF / MAIN ENGINES ON
NATM =	1	PRA63 ATMOS / STD US62 ATMOS
NFLY =	2	NO FLYBACK / FLYBACK / FLYBACK & G.S.
NLIM =	3	NO G LIMIT / EXP LIMIT / FIXED LIMIT
NLND =	1	CAPE KEN / HOUSTON / EDWARDS / HICKAM NAS CECIL / CHERRY POINT / AZORES / INPUT
NPRN =	1	FULL PRINT / STATE ONLY / TABLE ONLY
NPLT =	2	NO PLOT FILE / WRITE DATA TO PLOT FILE
NPUT =	3	INERTIAL / INERTIAL AND ENC / RELATIVE
NGAL =	2	NO LIMIT / G ALPHA LIMIT
NTRY =	1	BEGIN AT THIS SEGMENT
NTYP =	1	BOOSTER / ORBITER
NVNT =	1	NO VENT / VENT
NWIN =	2	NO WIND / 95% INPUT / 95% IN+HEAD
NMET =	1	INPUT ENGLISH / METRIC

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INPUT VALUES

LAT	28.4600	LON	-79.8700	ALT/R	312418.00	VV	6976.000	GAM	53.600	EN/HDN	93.980
ALTH	15000.00	ALTE	35000.00	RDTN	-100.00	TCON	10.000	TSTG	1000.000	VTST	2.500
ALP	0.000	ALPC	70.000		70.000		70.000		70.000		6.000
ALIM	5.000		5.000		5.000		5.000		0.430		0.430
BET	0.000	BETC	0.000	BETT	1.000	BETM	1.000	BETL	45.000	BETX	45.000
BLIM	5.000		5.000		5.000		5.000		5.000		5.000
THFAC	0.000		0.000		0.000		0.000	ENOM	0.000	ENOF	4.000
WOTE	880110.00	WOTF	0.00	WOTO	0.00	WOTR	0.00	FMDA	0.00	FMDV	0.00
ACCL	3.00	DCD	0.00	DCL	0.00	DTH	0.00	LATS	0.00	LONG	0.00
T	161.00	DT	2.00	DTF	5.00	TP	161.00	DTP	300.00	DTPF	50.00

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PROGRAM DUAL PD34/DAURO/544-0546 PAGE 2

FLYBACK#5, BOOSTER, WO 260K, SLT 103.5K
OUTPUT UNITS

TIME	SEC
ALTITUDE	FEET
VELOCITY	FEET/SEC
ANGLES	DEGS
RATES	FEET/SEC

RANGES	NAUT MILES
WEIGHT	LBS
THRUST	LBSF
HEAT IN	BTU/FT**2
HEAT RATE	BTU/FT**2/SEC
PRESSURE	LBS/FT*2
ACCELERATIONS	Gs
Q ALPHA	DEGS*LBS/FT*2

1

FLYBACK#5, BOOSTER, WO 260K, SLT 103.5K

MSFC AERO #1

VEHICLE CHARACTERISTICS ENGLISH UNITS

AREA	12500.00	RDNS	1.00	GALFM	1295.00	BASE	200000.00	ISPL	389.1000	THRSL	69000.00
CTMA	2000.0000	CTMD	-0.0200	ACCM	3.0000	ACK	0.5000	ISPV	429.1000	THRSV	76093.00
RDTL	-8.33	ALTL	1000.00	DTHRL	0.950	RDTA	-5.00	ALTA	500.00	DTHRA	0.920

CD AERO TABLE

MACH / ALPHA	0.00	0.50	0.80	1.00	1.20	1.46	1.95	3.48	4.96	5.60
0.00	0.056	0.056	0.052	0.088	0.175	0.143	0.135	0.110	0.100	0.170
6.00	0.071	0.071	0.086	0.185	0.226	0.207	0.191	0.150	0.140	0.180
8.00	0.094	0.094	0.121	0.237	0.261	0.244	0.215	0.170	0.150	0.186
10.00	0.132	0.132	0.168	0.285	0.310	0.268	0.241	0.200	0.170	0.197
14.00	0.234	0.234	0.280	0.417	0.410	0.365	0.328	0.270	0.230	0.239
25.00	0.234	0.234	0.600	0.750	0.700	0.600	0.550	0.446	0.394	0.418
30.00	0.234	0.234	0.840	1.050	0.980	0.840	0.770	0.623	0.559	0.562
40.00	0.234	0.234	0.840	1.800	1.680	1.440	1.320	1.072	0.986	0.959
50.00	0.234	0.234	0.840	1.800	2.430	2.080	1.910	1.550	1.400	1.461
60.00	0.234	0.234	0.840	1.800	2.430	2.490	2.280	1.850	1.700	1.992

CL AERO TABLE

MACH / ALPHA	0.00	0.50	0.80	1.00	1.20	1.46	1.95	3.48	4.96	5.60
0.00	0.025	0.025	0.018	0.084	0.056	0.073	0.073	0.050	0.040	0.006
6.00	0.370	0.370	0.388	0.547	0.476	0.434	0.354	0.260	0.220	0.060
8.00	0.493	0.493	0.468	0.693	0.616	0.546	0.441	0.320	0.270	0.083
10.00	0.604	0.604	0.561	0.815	0.749	0.644	0.518	0.380	0.320	0.114
14.00	0.807	0.807	0.764	1.050	0.983	0.840	0.672	0.490	0.420	0.215
25.00	0.807	0.807	1.060	1.350	1.300	1.150	0.950	0.691	0.589	0.480
30.00	0.807	0.807	1.680	1.700	1.640	1.450	1.195	0.869	0.761	0.626
40.00	0.807	0.807	1.680	2.270	2.190	1.940	1.599	1.163	1.055	0.883
50.00	0.807	0.807	1.680	2.270	2.540	2.250	1.860	1.350	1.200	1.025
60.00	0.807	0.807	1.680	2.270	2.540	2.430	2.010	1.460	1.300	0.993

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FLYBACK#5, BOOSTER, WO 260K, SLT 103.5K

GENERIC(M) CANT -6.00

THRUST TABLE

MACH / ALTITUDE	0.00	0.20	0.40	0.60	0.80
0.	62500.	55000.	51877.	50627.	50000.

465.	59395.	53738.	53034.	53738.	53738.
929.	56459.	51477.	51477.	52310.	56459.
1394.	53569.	49743.	50702.	53569.	57395.
1858.	50637.	45137.	46234.	48435.	52839.
3252.	43551.	38967.	40116.	42409.	44701.

SFC TABLE

MACH / ALTITUDE	0.00	0.20	0.40	0.60	0.80
0.	0.3321	0.3321	0.3321	0.3321	0.3321
465.	0.3985	0.3985	0.3985	0.3985	0.3985
929.	0.4151	0.4151	0.4151	0.4151	0.4151
1394.	0.4317	0.4317	0.4317	0.4317	0.4317
1858.	0.4317	0.4317	0.4317	0.4317	0.4317
3252.	0.4317	0.4317	0.4317	0.4317	0.4317

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WIND TABLE OPTION 2

ALT (FT)	VEL (F/S)	DIR (DEGS)
3281.	62.	280.
19685.	144.	280.
36089.	240.	280.
39370.	246.	280.
42651.	243.	290.
65617.	66.	280.
75459.	66.	280.
131234.	256.	280.
164042.	312.	280.
190289.	367.	280.
196850.	367.	280.
246063.	98.	280.

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BEGIN ENTRY

*XDRG	1.5839	*YDRG	0.9855	*ZDRG	-0.2492	DRGG	0.1919	CD	0.0710	LON	-80.5861
*XTHR	-1.5826	*YTHR	-0.9847	*ZTHR	0.2490	THRM	149092.16	THFAC	70.2465	WDOT	-15.42
ACCX	0.1044	ACCY	0.0000	ACCZ	-0.9946	GDOT	0.0299	GTOT	1088.	WOT	777522.49
ACPL	3.0000	ACNL	-1.0000	DACZ	2.0054	GALF	1008.79	CTMP	0.00	STMP	0.00
DRTG	0.742	BRNG	277.179	HDNR	278.670	CRNG	229.628	DRNG	1312.862	HDNI	87.3242

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PROGRAM DUAL PD34/DAURO/544-0546 PAGE 49

FLYBACK#5, BOOSTER, WO 260K, SLT 103.5K

ENGLISH UNITS

TIME	ALTITUDE RDOT RHO	VNRT VREL	GAMNRT GAMREL THRUST	DYNP MACH	GDOT GTOT WOT	ALPHA BETA	LAT LON WDOT	DRTG HDNG CD	ACCZ GALF CL
161.000	312418.	7927.	45.10	0.1	0.3	0.0	28.5	39.	0.00
BE	5615.	6976.	53.60	7.57	0.	0.0	-79.9	94.	0.
	0.237E-08		0.		880110.		0.00	0.170	0.006
357.000	854459.	5457.	-0.37	0.0	0.0	70.0	28.3	168.	0.00
AP	-36.	3967.	-0.51	1.47	2.	0.0	-77.4	96.	0.
	0.164E-12		0.		880110.		0.00	2.484	2.417
461.000	696870.	6262.	-28.62	0.0	0.0	70.0	28.1	235.	0.00
	-2999.	5014.	-36.74	1.99	3.	0.0	-76.2	97.	0.
	0.496E-12		0.		880110.		0.00	2.268	1.994
597.000	63329.	6081.	-28.81	2433.9	26.4	4.0	27.9	327.	-4.71
TM	-2931.	4883.	-36.89	5.14	479.	0.0	-74.5	99.	9736.
	0.204E-03		0.		880110.		0.00	0.141	0.127
613.000	33540.	3960.	-14.08	2442.7	6.1	0.0	27.9	336.	-2.26
BR	-963.	2472.	-22.93	2.47	766.	0.0	-74.3	99.	0.
	0.799E-03		0.		880110.		0.00	0.126	0.065
615.000	31782.	3730.	-12.34	2126.1	4.6	0.0	27.9	336.	-2.08
	-797.	2237.	-20.86	2.22	778.	10.0	-74.3	99.	0.
	0.849E-03		0.		880110.		0.00	0.131	0.069
617.000	30327.	3522.	-10.84	1834.9	3.5	0.0	27.9	337.	-1.88
	-662.	2028.	-19.06	1.99	787.	20.0	-74.3	100.	0.
	0.893E-03		0.		880110.		0.00	0.134	0.072
619.000	29107.	3336.	-9.67	1579.2	2.6	0.0	27.9	338.	-1.64
	-560.	1843.	-17.69	1.80	794.	27.1	-74.3	102.	0.
	0.930E-03		0.		880110.		0.00	0.137	0.073
621.000	28065.	3172.	-8.75	1361.0	2.0	0.0	27.9	338.	-1.41
ER	-483.	1682.	-16.67	1.64	799.	18.9	-74.2	103.	0.
	0.962E-03		0.		880110.		0.00	0.140	0.073
623.000	27159.	3027.	-8.09	1177.7	1.5	0.0	27.9	339.	-1.20
ED	-426.	1541.	-16.04	1.49	803.	28.9	-74.2	104.	0.
	0.992E-03		178804.		880110.		-21.44	0.142	0.073

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PROGRAM DUAL PD34/DAURO/544-0546 PAGE 50

FLYBACK#5, BOOSTER, WO 260K, SLT 103.5K

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ENGLISH UNITS

TIME	ALTITUDE / RDOT / RHO	VNRT / VREL	GAMNRT / GAMREL / THRUST	DYNP / MACH	GDOT / GTOT / WOT	ALPHA / BETA	LAT / LON / WDOT	DRTG / HDNG / CD	ACCZ / GALF / CL
673.000	27556.	1180.	-6.29	146.7	0.1	6.0	27.8	343.	-0.78
	-129.	547.	-13.65	0.53	822.	45.0	-74.2	232.	880.
	0.979E-03		42222.		879655.		-5.06	0.073	0.372
723.000	20393.	955.	2.78	188.6	0.1	6.0	27.8	339.	-1.00
	46.	553.	4.80	0.52	827.	1.4	-74.2	279.	1132.
	0.123E-02		41995.		879393.		-5.04	0.072	0.371
773.000	16685.	944.	-6.51	216.7	0.1	6.0	27.8	336.	-1.16
	-107.	559.	-11.04	0.52	830.	0.0	-74.3	279.	1300.
	0.139E-02		41960.		879144.		-5.03	0.072	0.371
803.000	15137.	987.	-2.23	176.5	0.1	6.0	27.8	334.	-0.92
.. HC	-38.	492.	-4.47	0.45	832.	0.0	-74.3	279.	1059.
	0.146E-02		162952.		878994.		-19.54	0.071	0.370
823.000	14364.	945.	-1.17	209.0	0.1	6.0	27.8	333.	-1.10
	-19.	529.	-2.10	0.49	833.	0.0	-74.4	279.	1254.
	0.149E-02		164459.		878601.		-19.72	0.071	0.370
873.000	14720.	976.	-1.14	184.2	0.1	6.0	27.8	330.	-0 97
	-19.	500.	-2.23	0.46	836.	0.0	-74.4	279.	1105.
	0.147E-02		163232.		877621.		-19.57	0.071	0.370
923.000	14514.	976.	0.32	184.4	0.1	6.0	27.8	327.	-0.97
	5.	498.	0.62	0.46	839.	1.1	-74.5	279.	1106.
	0.148E-02		163160.		876640.		-19.57	0.071	0.370
973.000	14254.	967.	0.07	191.6	0.1	6.0	27.8	324.	-1.01
	1.	506.	0.14	0.47	841.	1.1	-74.5	279.	1149.
	0.150E-02		163461.		875660.		-19.60	0.071	0.370
1023.000	14182.	969.	-0.33	190.0	0.1	6.0	27.8	320.	-1.00
	-6.	503.	-0.64	0.46	844.	0.0	-74.6	279.	1140.
	0.150E-02		163342.		874681.		-19.59	0.071	0.370
1073.000	14101.	972.	-0.18	187.7	0.1	6.0	27.9	317.	-0.99
	-3.	500.	-0.36	0.46	847.	0.0	-74.7	279.	1126.
	0.150E-02		163182.		873703.		-19.57	0.071	0.370

PROGRAM DUAL PD34/DAURO/544-0546 PAGE 51

FLYBACK#5, BOOSTER, WO 260K, SLT 103.5K

ENGLISH UNITS

TIME	ALTITUDE / RDOT / RHO	VNRT / VREL	GAMNRT / GAMREL / THRUST	DYNP / MACH	GDOT / GTOT / WOT	ALPHA / BETA	LAT / LON / WDOT	DRTG / HDNG / CD	ACCZ / GALF / CL
1123.000	13971.	972.	-0.08	188.2	0.1	6.0	27.9	314.	-0.99
	-1.	499.	-0.15	0.46	850.	0.0	-74.7	279.	1129.
	0.151E-02		163159.		872724.		-19.57	0.071	0.370

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1173.000	13855.	971.	-0.13	188.5	0.1	6.0	27.9	311.	-1.00
	-2.	499.	-0.25	0.46	852.	0.0	-74.8	279.	1131.
	0.152E-02		163131.		871746.		-19.56	0.071	0.370
1223.000	13758.	972.	-0.15	188.0	0.1	6.0	27.9	308.	-0.99
	-3.	497.	-0.29	0.46	855.	0.0	-74.8	279.	1128.
	0.152E-02		163065.		870768.		-19.55	0.071	0.370
1273.000	13659.	973.	-0.12	187.7	0.1	6.0	27.9	305.	-0.99
	-2.	496.	-0.24	0.46	858.	0.0	-74.9	279.	1126.
	0.152E-02		163007.		869790.		-19.55	0.071	0.370
1323.000	13560.	973.	-0.11	187.6	0.1	6.0	27.9	301.	-0.99
	-2.	495.	-0.23	0.45	860.	0.0	-75.0	279.	1125.
	0.153E-02		162962.		868813.		-19.54	0.071	0.370
1373.000	13467.	973.	-0.12	187.4	0.1	6.0	27.9	298.	-0.99
	-2.	494.	-0.23	0.45	863.	0.0	-75.0	279.	1124.
	0.153E-02		162915.		867836.		-19.54	0.071	0.370
1423.000	13379.	973.	-0.11	187.2	0.1	6.0	27.9	295.	-0.99
	-2.	493.	-0.22	0.45	866.	0.0	-75.1	279.	1123.
	0.154E-02		162867.		866859.		-19.53	0.071	0.370
1473.000	13294.	974.	-0.11	186.9	0.1	6.0	27.9	292.	-0.99
	-2.	492.	-0.21	0.45	868.	0.0	-75.1	279.	1122.
	0.154E-02		162822.		865883.		-19.53	0.071	0.370
1523.000	13213.	974.	-0.10	186.7	0.1	6.0	27.9	289.	-0.99
	-2.	491.	-0.20	0.45	871.	0.0	-75.2	279.	1120.
	0.155E-02		162779.		864907.		-19.52	0.071	0.370
1573.000	13136.	974.	-0.10	186.5	0.1	6.0	27.9	286.	-0.99
	-2.	491.	-0.19	0.45	874.	0.0	-75.2	279.	1119.
	0.155E-02		162737.		863931.		-19.51	0.071	0.370

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FLYBACK#5, BOOSTER, WO 260K, SLT 103.5K

ENGLISH UNITS

TIME	ALTITUDE RDOT RHO	VNRT VREL	GAMNRT GAMREL THRUST	DYNP MACH	QDOT QTOT WGT	ALPHA BETA	LAT LON WDOT	DRTG HDNG CD	ACCZ GALF CL
1623.000	13063.	975.	-0.09	186.3	0.1	6.0	27.9	283.	-0.99
	-2.	490.	-0.18	0.45	876.	0.0	-75.3	278.	1118.
	0.155E-02		162697.		862955.		-19.51	0.071	0.370
1673.000	12994.	975.	-0.09	186.1	0.1	6.0	27.9	280.	-0.99
	-1.	489.	-0.17	0.45	879.	0.0	-75.4	278.	1117.
	0.156E-02		162659.		861980.		-19.51	0.071	0.370
1723.000	12929.	975.	-0.08	185.9	0.1	6.0	27.9	276.	-0.99
	-1.	488.	-0.16	0.45	881.	0.0	-75.4	278.	1115.
	0.156E-02		162622.		861005.		-19.50	0.071	0.370
1773.000	12868.	975.	-0.08	185.7	0.1	6.0	28.0	273.	-0.99
	-1.	487.	-0.15	0.45	884.	0.0	-75.5	278.	1114.

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	0.156E-02		162587.		860030.		-19.50	0.071	0.370
1823.000	12811.	976.	-0.07	185.5	0.1	6.0	28.0	270.	-0.99
	-1.	487.	-0.15	0.45	887.	0.0	-75.5	278.	1113.
	0.157E-02		162553.		859055.		-19.49	0.071	0.370
1873.000	12758.	976.	-0.07	185.3	0.1	6.0	28.0	267.	-0.99
	-1.	486.	-0.14	0.44	889.	0.0	-75.6	278.	1112.
	0.157E-02		162522.		858081.		-19.49	0.071	0.370
1923.000	12710.	976.	-0.06	185.1	0.1	6.0	28.0	264.	-0.99
	-1.	485.	-0.13	0.44	892.	1.0	-75.6	278.	1110.
	0.157E-02		162492.		857106.		-19.49	0.071	0.370
1973.000	12666.	977.	-0.06	184.8	0.1	6.0	28.0	261.	-0.99
	-1.	485.	-0.12	0.44	894.	1.1	-75.7	278.	1109.
	0.157E-02		162463.		856132.		-19.48	0.071	0.370
2023.000	12626.	977.	-0.05	184.6	0.1	6.0	28.0	258.	-0.99
	-1.	484.	-0.11	0.44	897.	0.0	-75.8	278.	1108.
	0.157E-02		162436.		855158.		-19.48	0.071	0.370
2073.000	12591.	977.	-0.05	184.4	0.1	6.0	28.0	255.	-0.99
	-1.	484.	-0.10	0.44	899.	0.0	-75.8	278.	1107.
	0.158E-02		162411.		854184.		-19.48	0.071	0.370

FLYBACK#5, BOOSTER, WO 260K, SLT 103.5K

ENGLISH UNITS

TIME	ALTITUDE RDOT RHO	VNRT VREL	GAMNRT GAMREL THRUST	DYNP MACH	GDOT GTOT WGT	ALPHA BETA	LAT LON WDOT	DRTG HDNG CD	ACCZ GALF CL
2123.000	12560.	977.	-0.04	184.2	0.1	6.0	28.0	252.	-0.99
	-1.	483.	-0.09	0.44	902.	0.0	-75.9	278.	1105.
	0.158E-02		162388.		853210.		-19.47	0.071	0.370
2173.000	12533.	978.	-0.04	184.0	0.1	6.0	28.0	249.	-0.99
	-1.	483.	-0.07	0.44	904.	0.0	-75.9	278.	1104.
	0.158E-02		162366.		852237.		-19.47	0.071	0.370
2223.000	12511.	978.	-0.03	183.8	0.1	6.0	28.0	246.	-0.99
	-1.	482.	-0.06	0.44	907.	0.0	-76.0	278.	1103.
	0.158E-02		162346.		851263.		-19.47	0.071	0.370
2273.000	12494.	978.	-0.03	183.6	0.0	6.0	28.0	243.	-0.99
	0.	482.	-0.05	0.44	909.	0.0	-76.1	278.	1102.
	0.158E-02		162328.		850290.		-19.47	0.071	0.370
2323.000	12481.	978.	-0.02	183.4	0.0	6.0	28.0	239.	-0.99
	0.	481.	-0.04	0.44	912.	0.0	-76.1	278.	1100.
	0.158E-02		162312.		849317.		-19.46	0.071	0.370
2373.000	12473.	978.	-0.01	183.2	0.0	6.0	28.0	236.	-0.99
	0.	481.	-0.03	0.44	914.	0.0	-76.2	278.	1099.
	0.158E-02		162297.		848344.		-19.46	0.071	0.370

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2423.000	12469.	978.	-0.01	183.0	0.0	6.0	28.0	233.	-0.99
	0.	481.	-0.02	0.44	917.	0.0	-76.2	278.	1098.
	0.158E-02		162284.		847370.		-19.46	0.071	0.370
2473.000	12471.	979.	0.00	182.8	0.0	6.0	28.1	230.	-0.99
	0.	481.	-0.01	0.44	919.	1.0	-76.3	278.	1097.
	0.158E-02		162273.		846397.		-19.46	0.071	0.370
2523.000	12477.	979.	0.00	182.5	0.0	6.0	28.1	227.	-0.99
	0.	480.	0.00	0.44	922.	0.0	-76.3	278.	1095.
	0.158E-02		162264.		845425.		-19.46	0.071	0.370
2573.000	12488.	979.	0.01	182.3	0.0	6.0	28.1	224.	-0.99
	0.	480.	0.02	0.44	924.	0.0	-76.4	278.	1094.
	0.158E-02		162257.		844452.		-19.46	0.071	0.370

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FLYBACK#5, BOOSTER, WO 260K, SLT 103.5K

ENGLISH UNITS

TIME	ALTITUDE RDOT RHO	VNRT VREL	GAMNRT GAMREL THRUST	DYNP MACH	GDOT GTOT WGT	ALPHA BETA	LAT LON WDOT	DRTG HDNG CD	ACCZ GALF CL
2623.000	12504.	979.	0.01	182.1	0.0	6.0	28.1	221.	-0.99
	0.	480.	0.03	0.44	927.	0.0	-76.5	278.	1093.
	0.158E-02		162251.		843479.		-19.46	0.071	0.370
2673.000	12525.	979.	0.02	181.9	0.0	6.0	28.1	218.	-0.99
	0.	480.	0.04	0.44	929.	0.0	-76.5	278.	1091.
	0.158E-02		162248.		842506.		-19.46	0.071	0.370
2723.000	12551.	979.	0.03	181.7	0.0	6.0	28.1	215.	-0.99
	0.	480.	0.05	0.44	932.	0.0	-76.6	278.	1090.
	0.158E-02		162246.		841533.		-19.46	0.071	0.370
2773.000	12583.	979.	0.03	181.5	0.0	6.0	28.1	212.	-0.99
	1.	480.	0.07	0.44	934.	0.0	-76.6	278.	1089.
	0.158E-02		162246.		840560.		-19.46	0.071	0.370
2823.000	12619.	979.	0.04	181.3	0.0	6.0	28.1	209.	-0.99
	1.	480.	0.08	0.44	937.	0.0	-76.7	278.	1088.
	0.158E-02		162248.		839588.		-19.46	0.071	0.370
2873.000	12661.	979.	0.04	181.1	0.0	6.0	28.1	206.	-0.99
	1.	480.	0.09	0.44	939.	0.0	-76.7	278.	1086.
	0.157E-02		162252.		838615.		-19.46	0.071	0.370
2923.000	12708.	979.	0.05	180.9	0.0	6.0	28.1	203.	-0.99
	1.	480.	0.10	0.44	941.	1.0	-76.8	278.	1085.
	0.157E-02		162258.		837642.		-19.46	0.071	0.370
2973.000	12761.	980.	0.06	180.6	0.0	6.0	28.1	200.	-0.99
	1.	480.	0.12	0.44	944.	0.0	-76.9	278.	1084.
	0.157E-02		162267.		836669.		-19.46	0.071	0.370
3023.000	12819.	980.	0.06	180.4	0.0	6.0	28.1	197.	-0.99
	1.	480.	0.13	0.44	946.	0.0	-76.9	278.	1083.

	0.157E-02		162277.		835696.		-19.46	0.071	0.370
3073.000	12883.	980.	0.07	180.2	0.0	6.0	28.1	194.	-0.99
	1.	480.	0.14	0.44	949.	0.0	-77.0	278.	1081.
	0.156E-02		162290.		834723.		-19.46	0.071	0.370

ENGLISH UNITS

TIME	ALTITUDE	VNRT	GAMNRT	DYNP	GDOT	ALPHA	LAT	DRTG	ACCZ
	RDOT	VREL	GAMREL	MACH	GTOT	BETA	LON	HDNG	GALF
	RHO		THRUST		WOT		WDOT	CD	CL
3123.000	12952.	979.	0.08	180.0	0.0	6.0	28.1	191.	-0.99
	1.	481.	0.16	0.44	951.	0.0	-77.0	278.	1080.
	0.156E-02		162304.		833750.		-19.46	0.071	0.370
3173.000	13028.	979.	0.08	179.8	0.0	6.0	28.1	187.	-0.99
	1.	481.	0.17	0.44	954.	0.0	-77.1	278.	1079.
	0.156E-02		162321.		832777.		-19.46	0.071	0.370
3223.000	13109.	979.	0.09	179.6	0.0	6.0	28.1	184.	-0.99
	2.	481.	0.19	0.44	956.	0.0	-77.1	278.	1078.
	0.155E-02		162340.		831803.		-19.47	0.071	0.370
3273.000	13196.	979.	0.10	179.4	0.0	6.0	28.2	181.	-0.99
	2.	482.	0.20	0.44	959.	0.0	-77.2	278.	1076.
	0.155E-02		162361.		830830.		-19.47	0.071	0.370
3323.000	13289.	979.	0.11	179.2	0.0	6.0	28.2	178.	-0.99
	2.	482.	0.21	0.44	961.	0.0	-77.3	278.	1075.
	0.154E-02		162385.		829856.		-19.47	0.071	0.370
3373.000	13389.	979.	0.11	179.0	0.0	6.0	28.2	175.	-0.99
	2.	482.	0.23	0.44	964.	1.0	-77.3	278.	1074.
	0.154E-02		162411.		828883.		-19.48	0.071	0.370
3423.000	13494.	979.	0.12	178.7	0.0	6.0	28.2	172.	-0.99
	2.	483.	0.24	0.44	966.	0.0	-77.4	278.	1072.
	0.153E-02		162440.		827909.		-19.48	0.071	0.370
3473.000	13606.	979.	0.13	178.5	0.0	6.0	28.2	169.	-0.99
	2.	484.	0.26	0.44	969.	0.0	-77.4	278.	1071.
	0.153E-02		162471.		826935.		-19.48	0.071	0.370
3523.000	13724.	979.	0.13	178.3	0.0	6.0	28.2	166.	-0.99
	2.	484.	0.27	0.44	971.	0.0	-77.5	278.	1070.
	0.152E-02		162504.		825961.		-19.49	0.071	0.370
3573.000	13846.	979.	0.13	178.0	0.0	6.0	28.2	163.	-0.99
	2.	485.	0.25	0.45	974.	0.0	-77.5	278.	1068.
	0.152E-02		161916.		824987.		-19.42	0.071	0.370

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ENGLISH UNITS

TIME	ALTITUDE RDOT RHO	VNRT VREL	GAMNRT GAMREL THRUST	DYNP MACH	GDOT GTOT WGT	ALPHA BETA	LAT LON WDOT	DRTG HDNG CD	ACCZ GALF CL
3623.000	13951.	979.	0.11	177.9	0.0	6.0	28.2	160.	-0.99
	2.	485.	0.22	0.45	976.	0.0	-77.6	278.	1067.
	0.151E-02		161191.		824019.		-19.33	0.071	0.370
3673.000	14041.	979.	0.09	177.7	0.0	6.0	28.2	157.	-0.99
	2.	486.	0.18	0.45	979.	0.0	-77.6	278.	1066.
	0.151E-02		160539.		823054.		-19.25	0.071	0.370
3723.000	14120.	979.	0.08	177.4	0.0	6.0	28.2	154.	-0.99
	1.	486.	0.16	0.45	981.	0.0	-77.7	278.	1065.
	0.150E-02		159950.		822093.		-19.18	0.071	0.370
3773.000	14188.	978.	0.07	177.2	0.1	6.0	28.2	151.	-0.99
	1.	486.	0.13	0.45	984.	0.0	-77.8	277.	1063.
	0.150E-02		159418.		821135.		-19.12	0.071	0.370
3823.000	14246.	979.	0.06	177.0	0.1	6.0	28.2	148.	-0.99
	1.	486.	0.11	0.45	986.	1.0	-77.8	278.	1062.
	0.150E-02		158935.		820181.		-19.06	0.071	0.370
3873.000	14296.	979.	0.05	176.8	0.1	6.0	28.2	145.	-0.99
	1.	486.	0.09	0.45	989.	0.0	-77.9	278.	1061.
	0.149E-02		158495.		819229.		-19.01	0.071	0.370
3923.000	14340.	979.	0.04	176.6	0.0	6.0	28.2	142.	-0.99
	1.	486.	0.08	0.45	991.	0.0	-77.9	278.	1060.
	0.149E-02		158090.		818280.		-18.96	0.071	0.370
3973.000	14377.	979.	0.03	176.4	0.0	6.0	28.2	139.	-0.99
	1.	487.	0.07	0.45	994.	0.0	-78.0	278.	1059.
	0.149E-02		157717.		817333.		-18.91	0.071	0.370
4023.000	14409.	979.	0.03	176.2	0.0	6.0	28.2	136.	-0.99
	0.	486.	0.06	0.45	996.	0.0	-78.0	278.	1057.
	0.149E-02		157371.		816388.		-18.87	0.071	0.370
4073.000	14437.	979.	0.02	176.0	0.0	6.0	28.3	133.	-0.99
	0.	486.	0.05	0.45	998.	0.0	-78.1	277.	1056.
	0.149E-02		157048.		815446.		-18.83	0.071	0.370

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FLYBACK#5, BOOSTER, WO 260K, SLT 103.5K

ENGLISH UNITS

TIME	ALTITUDE RDOT RHO	VNRT VREL	GAMNRT GAMREL THRUST	DYNP MACH	GDOT GTOT WGT	ALPHA BETA	LAT LON WDOT	DRTG HDNG CD	ACCZ GALF CL
4123.000	14461.	979.	0.02	175.8	0.0	6.0	28.3	130.	-0.99
	0.	486.	0.04	0.45	1001.	0.0	-78.2	277.	1055.
	0.149E-02		156745.		814505.		-18.80	0.071	0.370

4173.000	14482.	979.	0.02	175.6	0.0	6.0	28.3	127.	-0.99
	0.	486.	0.03	0.45	1003.	0.0	-78.2	277.	1054.
	0.149E-02		156459.		813566.		-18.76	0.071	0.370
4223.000	14500.	979.	0.01	175.4	0.0	6.0	28.3	124.	-0.99
	0.	486.	0.03	0.45	1006.	0.0	-78.3	277.	1052.
	0.149E-02		156189.		812628.		-18.73	0.071	0.370
4273.000	14515.	980.	0.01	175.2	0.0	6.0	28.3	121.	-0.99
	0.	486.	0.02	0.45	1008.	1.0	-78.3	277.	1051.
	0.148E-02		155931.		811693.		-18.70	0.071	0.370
4323.000	14529.	980.	0.01	175.0	0.0	6.0	28.3	118.	-0.99
	0.	486.	0.01	0.45	1011.	0.0	-78.4	278.	1050.
	0.148E-02		155685.		810758.		-18.67	0.071	0.370
4348.000	14535.	980.	0.01	174.9	0.0	6.0	28.3	116.	-1.00
BL	0.	486.	0.01	0.45	1012.	0.0	-78.4	277.	1049.
	0.148E-02		131004.		810292.		-15.71	0.071	0.370
4373.000	14337.	981.	-0.96	174.5	0.0	6.0	28.3	115.	-0.99
	-16.	484.	-1.94	0.44	1013.	0.5	-78.4	277.	1047.
	0.149E-02		147152.		809879.		-17.65	0.071	0.370
4423.000	14131.	987.	-0.37	170.1	0.0	6.0	28.3	112.	-0.97
	-6.	476.	-0.76	0.44	1016.	0.0	-78.5	277.	1020.
	0.150E-02		141749.		809004.		-17.00	0.071	0.370
4473.000	13700.	983.	-0.07	174.0	0.0	6.0	28.3	109.	-0.99
	-1.	478.	-0.14	0.44	1018.	0.0	-78.6	277.	1044.
	0.152E-02		141829.		808145.		-17.01	0.071	0.370
4523.000	13280.	980.	-0.41	176.5	0.0	6.0	28.3	106.	-1.01
	-7.	478.	-0.84	0.44	1021.	1.0	-78.6	277.	1059.
	0.154E-02		142152.		807298.		-17.05	0.071	0.370

PROGRAM DUAL PD34/DAURO/544-0546 PAGE 58

FLYBACK#5, BOOSTER, WO 260K, SLT 103.5K

ENGLISH UNITS

TIME	ALTITUDE RDOT RHO	VNRT VREL	GAMNRT GAMREL THRUST	DYNP MACH *	GDOT GTOT WGT	ALPHA BETA	LAT LON WDOT	DRTG HDNG CD	ACCZ GALF CL
4573.000	12928.	984.	-0.64	174.7	0.0	6.0	28.3	103.	-1.00
	-11.	473.	-1.34	0.43	1023.	0.0	-78.7	277.	1048.
	0.156E-02		140827.		806455.		-16.89	0.071	0.370
4623.000	12558.	987.	-0.53	172.9	0.0	6.0	28.3	100.	-0.99
	-9.	468.	-1.12	0.43	1025.	0.0	-78.7	277.	1037.
	0.158E-02		139650.		805613.		-16.75	0.071	0.370
4673.000	12143.	987.	-0.40	173.2	0.0	6.0	28.3	97.	-0.99
	-7.	465.	-0.86	0.43	1028.	0.0	-78.8	277.	1039.
	0.160E-02		139502.		804773.		-16.73	0.071	0.370
4723.000	11722.	987.	-0.43	173.9	0.0	6.0	28.3	94.	-1.00
	-7.	463.	-0.91	0.42	1030.	0.0	-78.8	277.	1043.

TIME	ALTITUDE / RDOT / RHO	VNRT / VREL	GAMNRT / GAMREL / THRUST	DYNP / MACH	GDOT / GTOT / WGT	ALPHA / BETA	LAT / LON / WDOT	DRTG / HDNG / CD	ACCZ / GALF / CL
	0.162E-02		139622.		803936.		-16.74	0.071	0.370
4773.000	11316.	988.	-0.50	173.8	0.0	6.0	28.3	91.	-1.00
	-9.	460.	-1.07	0.42	1032.	0.0	-78.9	277.	1043.
	0.164E-02		139401.		803101.		-16.72	0.071	0.370
4823.000	10917.	990.	-0.51	173.2	0.0	6.0	28.3	88.	-0.99
	-9.	457.	-1.10	0.42	1034.	0.0	-78.9	277.	1039.
	0.166E-02		138991.		802267.		-16.67	0.071	0.370
4873.000	10513.	991.	-0.48	172.8	0.0	6.0	28.3	85.	-0.99
	-8.	453.	-1.05	0.41	1036.	0.0	-79.0	277.	1037.
	0.168E-02		138682.		801434.		-16.63	0.071	0.370
4923.000	10103.	992.	-0.47	172.8	0.0	6.0	28.3	82.	-0.99
	-8.	450.	-1.02	0.41	1038.	0.0	-79.1	277.	1037.
	0.170E-02		138497.		800603.		-16.61	0.071	0.370
4973.000	9693.	992.	-0.47	172.7	0.0	6.0	28.4	79.	-1.00
	-8.	447.	-1.04	0.41	1041.	0.0	-79.1	277.	1036.
	0.173E-02		138321.		799773.		-16.59	0.071	0.370
5023.000	9286.	993.	-0.48	172.6	0.0	6.0	28.4	76.	-1.00
	-8.	444.	-1.07	0.40	1043.	1.0	-79.2	277.	1035.
	0.175E-02		138099.		798944.		-16.56	0.071	0.370

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FLYBACK89, BOOSTER, WO 260K, SLT 103.5K

ENGLISH UNITS

TIME	ALTITUDE / RDOT / RHO	VNRT / VREL	GAMNRT / GAMREL / THRUST	DYNP / MACH	GDOT / GTOT / WGT	ALPHA / BETA	LAT / LON / WDOT	DRTG / HDNG / CD	ACCZ / GALF / CL
5073.000	8880.	994.	-0.48	172.4	0.0	6.0	28.4	73.	-1.00
	-8.	441.	-1.08	0.40	1045.	1.0	-79.2	277.	1034.
	0.177E-02		137853.		798117.		-16.53	0.071	0.370
5123.000	8473.	995.	-0.48	172.1	0.0	6.0	28.4	70.	-1.00
	-8.	438.	-1.08	0.40	1047.	0.0	-79.3	277.	1033.
	0.179E-02		137615.		797291.		-16.50	0.071	0.370
5173.000	8065.	996.	-0.47	172.0	0.0	6.0	28.4	68.	-1.00
	-8.	435.	-1.08	0.39	1049.	0.0	-79.3	277.	1032.
	0.181E-02		137389.		796467.		-16.48	0.071	0.370
5223.000	7657.	996.	-0.47	171.8	0.0	6.0	28.4	65.	-1.00
	-8.	432.	-1.09	0.39	1050.	0.0	-79.4	277.	1031.
	0.184E-02		137169.		795643.		-16.45	0.071	0.370
5273.000	7250.	997.	-0.47	171.7	0.0	6.0	28.4	62.	-1.00
	-8.	430.	-1.10	0.39	1052.	0.0	-79.4	277.	1030.
	0.186E-02		136947.		794822.		-16.42	0.071	0.370
5323.000	6842.	998.	-0.47	171.5	0.0	6.0	28.4	59.	-1.00
	-8.	427.	-1.11	0.38	1054.	0.0	-79.5	277.	1029.
	0.188E-02		136721.		794001.		-16.40	0.071	0.370

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5373.000	6435.	999.	-0.47	171.3	0.0	6.0	28.4	56.	-1.00
	-8.	424.	-1.12	0.38	1056.	0.0	-79.5	277.	1028.
	0.191E-02		136495.		793182.		-16.37	0.071	0.370
5423.000	6027.	999.	-0.47	171.1	0.0	6.0	28.4	53.	-1.00
	-8.	421.	-1.12	0.38	1058.	0.0	-79.6	277.	1027.
	0.193E-02		136270.		792364.		-16.34	0.071	0.370
5473.000	5620.	1000.	-0.47	171.0	0.0	6.0	28.4	50.	-1.00
	-8.	418.	-1.13	0.37	1060.	0.0	-79.7	277.	1026.
	0.195E-02		136045.		791548.		-16.31	0.071	0.370
5523.000	5212.	1001.	-0.47	170.8	0.0	6.0	28.4	48.	-1.00
	-8.	416.	-1.14	0.37	1061.	0.0	-79.7	277.	1025.
	0.198E-02		135821.		790733.		-16.29	0.071	0.370

FLYBACK#5, BOOSTER, WO 260K, SLT 103.5K

ENGLISH UNITS

TIME	ALTITUDE RDOT RHO	VNRT VREL	GAMNRT GAMREL THRUST	DYNP MACH	GDOT GTOT WGT	ALPHA BETA	LAT LON WDOT	DRTG HDNG CD	ACCZ GALF CL
5573.000	4805.	1001.	-0.47	170.6	0.0	6.0	28.4	45.	-1.00
	-8.	413.	-1.15	0.37	1063.	0.0	-79.8	277.	1024.
	0.200E-02		135598.		789919.		-16.26	0.071	0.370
5623.000	4397.	1002.	-0.47	170.5	0.0	6.0	28.4	42.	-1.00
	-8.	410.	-1.15	0.37	1065.	0.0	-79.8	277.	1023.
	0.203E-02		135375.		789108.		-16.16	0.071	0.370
5673.000	3990.	1002.	-0.47	170.3	0.0	6.0	28.4	39.	-1.00
	-8.	407.	-1.16	0.36	1067.	0.0	-79.9	277.	1022.
	0.205E-02		135154.		788304.		-15.97	0.071	0.370
5723.000	3582.	1003.	-0.47	170.1	0.0	6.0	28.4	36.	-1.00
	-8.	405.	-1.17	0.36	1068.	1.0	-79.9	277.	1021.
	0.208E-02		134935.		787510.		-15.78	0.071	0.370
5773.000	3176.	1004.	-0.45	170.2	0.0	6.0	28.4	34.	-1.00
	-8.	403.	-1.12	0.36	1070.	0.0	-80.0	277.	1021.
	0.210E-02		134657.		786726.		-15.58	0.071	0.370
5823.000	2787.	1006.	-0.46	169.9	0.0	6.0	28.4	31.	-1.00
	-8.	400.	-1.17	0.36	1072.	0.0	-80.0	277.	1020.
	0.212E-02		134050.		785953.		-15.35	0.071	0.370
5873.000	2389.	1008.	-0.47	169.7	0.0	6.0	28.4	28.	-1.00
	-8.	397.	-1.19	0.35	1073.	0.0	-80.1	277.	1018.
	0.215E-02		133608.		785191.		-15.14	0.071	0.370
5923.000	1987.	1011.	-0.47	169.5	0.0	6.0	28.5	25.	-1.00
	-8.	395.	-1.20	0.35	1075.	0.0	-80.1	277.	1017.
	0.217E-02		133268.		784439.		-14.94	0.071	0.370
5973.000	1583.	1013.	-0.47	169.3	0.0	6.0	28.5	22.	-1.00
	-8.	392.	-1.21	0.35	1076.	0.0	-80.2	277.	1016.

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            0.220E-02        132987.          783696.            -14.74  0.071   0.370

 6023.000       1177.   1015.   -0.47  169.2     0.0     6.0    28.5     20.   -1.00
                  -8.    390.   -1.21   0.34   1078.     0.0   -80.2    277.   1015.
            0.222E-02        132744.          782972.            -14.14  0.071   0.370
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FLYBACK#5, BOOSTER, WO 260K, SLT 103.5K

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                ENGLISH UNITS
     TIME  ALTITUDE    VNRT  GAMNRT   DYNP    GDOT   ALPHA     LAT   DRTG    ACCZ
               RDOT    VREL  GAMREL   MACH    GTOT    BETA     LON   HDNG    GALF
                RHO          THRUST            WGT            WDOT     CD      CL

 6037.000      1063.   1016.   -0.46  169.1     0.0     6.0    28.5     19.   -0.99
       LC        -8.    389.   -1.21   0.34   1078.     0.0   -80.2    277.   1015.
           0.223E-02        146882.          782775.            -15.44  0.071   0.370

 6087.000      1013.   1018.   -0.25  168.1     0.0     6.0    28.5     16.   -0.99
                 -4.    388.   -0.66   0.34   1080.     0.0   -80.3    277.   1008.
           0.223E-02        147867.          782003.            -15.45  0.071   0.370

 6137.000       979.   1018.   -0.12  167.8     0.0     6.0    28.5     14.   -0.99
                 -2.    387.   -0.31   0.34   1081.     0.0   -80.3    277.   1007.
           0.224E-02        148487.          781230.            -15.46  0.071   0.370

 6187.000       956.   1018.   -0.05  167.9     0.0     6.0    28.5     11.   -0.99
                 -1.    387.   -0.12   0.34   1083.     0.0   -80.4    277.   1007.
           0.224E-02        148853.          780457.            -15.45  0.071   0.370

 6237.000       942.   1018.   -0.01  168.0     0.0     6.0    28.5      8.   -0.99
                  0.    387.   -0.03   0.34   1084.     0.0   -80.4    277.   1008.
           0.224E-02        149039.          779684.            -15.45  0.071   0.370

 6287.000       933.   1018.    0.00  168.0     0.0     6.0    28.5      6.   -0.99
                  0.    387.    0.00   0.34   1086.     0.0   -80.5    277.   1008.
           0.224E-02        149100.          778911.            -15.44  0.071   0.370

 6337.000       928.   1019.    0.00  168.0     0.0     6.0    28.5      3.   -0.99
                  0.    387.    0.01   0.34   1087.     0.0   -80.5    278.   1008.
           0.224E-02        149076.          778139.            -15.43  0.071   0.370

 6377.000       923.   1019.    0.00  168.1     0.0     6.0    28.5      1.   -0.99
       AR         0.    388.   -0.01   0.34   1088.     1.4   -80.6    279.   1009.
           0.224E-02        149092.          777522.            -15.42  0.071   0.370
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PROGRAM DUAL PD34/DAURO/544-0546 PAGE 62

FLYBACK#5, BOOSTER, WO 260K, SLT 103.5K

MAXIMUM VALUES

```
MAX GDOT    26.955 BTU/FT2/S TIME    595.0
MAX CTMP     2784.  DEGS F   TIME    603.0
MAX STMP     2250.  DEGS F   TIME    595.0
MAX DYNP     3452.  LB/FT2   TIME    605.0
MAX GALF    17636.  D*LB/FT2 TIME    587.0
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MAX ACCT	7.715	GRAVS	TIME	585.0
MAX ACCX	-5.282	GRAVS	TIME	605.0
MAX ACCY	0.000	GRAVS	TIME	581.0
MIN ACZR	-4.652	GRAVS	TIME	585.0
WING -G	0.000	GRAVS	TIME	161.0
WING +G	7.652	GRAVS	TIME	585.0

FLYBACK FUEL 102587.51

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```
C                       THIS PROGRAM IS DESIGNED TO COMPUTE CL AND CD AT VARIOUS
C                       ALTITUDES AFTER THE CRAFT HAS BEEN SLOWED TO MACH 0.7 !!
C
C
C
C
C
C
C
C      CD = COEFFICIENT OF DRAG
C      CD2 = SECOND METHOD FOR FINDING DRAG
C      CD3 = CD2 CORRECTED FOR ERRORS MENTIONED IN ASSUMPTION 13.
C      CDO = ZERO LIFT DRAG COEFFICIENT
C      CL = COEFFICIENT OF LIFT
C      CFW = FLAT PLATE SKIN FRICTION COEFFICIENT OF THE WING
C      CFB - TURBULENT FLAT PLATE SKIN FRICTION COEFFICIENT OF THE BODY
C      (CDF)W - SKIN FRICTION DRAG COEFFICIENT OF THE WING
C      CDPB - PRESSURE DRAG OF THE BODY
C      CDBB - BASE DRAG COEFFICIENT BASED ON THE MAXIMUM
C             FRONTAL BODY AREA
C      CDOB - IS THE ZERO LIFT DRAG COEFFICIENT OF THE BODY
C      CD WB - THE ZERO LIFT DRAG COEFFICIENT OF THE WING & THE BODY
C      CDL - COEFFICIENT OF DRAG DUE TO LIFT
C      CDAB - COEFFICIENT OF DRAG OF THE BODY DUE TO ANGLE OF ATTACK
C      CDIWB - INDUCED DRAG COEFFICIENT OF WING BODY COMBINATION
C      CDFN - SKIN DRAG COEFFICIENT OF THE NACELLES
C      CDPN = PRESSURE DRAG COEFFICINT DUE TO NACELLES
C      CDIN - INTERFERENCE DRAG COEFFICIENT OF THE NACELLES
C      BETA = (1 - M SQUARED) ** 0.5
C      A - ASPECT RATIO
C      AL - ANGLE OF ATTACK (IN RADIANS)
C      B - WING SPAN
C      CL = COEFICIENT OF LIFT
C      S = WING AREA
C      E = OSWALD EFFICIENCY FACTOR
C      E2 = 2nd METHOD FOR FINDING OSW. EFFIC. FAC.
C      F = EQUIVALENT PARASITE AREA
C      SREF = WING PLANFORM OR REFERENCE AREA
C      SS = BODY WETTED AREA
C      SB = BODY FRONTAL AREA
C      SNWET = WETTED AREA OF THE ENGINE NACELLES
C      RHO = AIR DENSITY
C              NOTE  -  RHO DEPENDS ON ALT WHERE M=0.7 IS BEGUN; IN ADDITION,
C                    ITS CHANGE, IE- RHO(1) TO RHO(2), DEPENDS UPON THE SHIP'S
C                    RATE OF DESCENT !!
C      V = CRAFT VELOCITY
C      DP = DYNAMIC PRESSURE
C      L = LIFT (=WEIGHT)
C      MU = VISCOSITY
C      LL = AIRFOIL THICKNESS LOCATION PARAMETER
C      RE = REYNOLDS NUMBER BASED ON THE WING CORD
C      REB = REYNOLDS NUMBER BASED ON THE BODY LENGTH
C      REN = REYNOLDS NUMBER BASED ON THE ENGINE NACELLES
C
C      ASSUMPTIONS :
C              1. WEIGHT CHANGE AT THIS POINT IS NEGLIGIBLE.
C              2. THE CRAFT IS FLYING THROUGH AN IDEAL STANDARD ATMOSPHERE
C              3. EACH ITERATION IS A 500 FOOT INCREMENT (30 SECOND INTERVAL
C              4. SPEED REMAINS AT MACH = 0.7
C              5. WAVE DRAG COEFFICIENT IS NEGLIGIBLE AT M=0.7
C              6. THICKNESS RATIO (t/c) IS 0.12 AND IT IS LOCATED @ 0.5C
C              7. BODY LENGTH USED 192.7 FEET
C              8. SHARP LEADING EDGE (ie - R1 LER = 0)
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C       [illegible]
C       [illegible] 10. EFFECTS OF THE VERTICAL TAIL ARE NEGLIGIBLE ON DRAG
C       [illegible] AND LANDING GEAR IS NEGLIGIBLE.
C       [illegible] THE ENGINES ARE [illegible] FEET FROM THE BODY
C       [illegible] CD3 IS USED TO INCLUDE MISCELLANEOUS CAUSES OF DRAG
C                SUCH AS CANARDS, INTERFERENCE BETWEEN THE NACELLES
C                AND SKIN ROUGHNESS.
C
      DIMENSION RO(50),ALT(50),CCD(50),VEL(50),T(50),MU(50),
     & RE(50),CFW(50),CDFW(50),CDOW(50),REB(50),CFB(50),CDFB(50),
     & CDPB(50),CDB2(50),CDOB(50),CDOWB(50),CLAW(50)
      REAL LAM,L,M,MU,LL,LODMAX,LOD
      PI=3.141592654
      LODMAX=0.0
C
      DO 1000 J=14,16,2
C
            AALT=35500
            PP=J
            P=PP-12.0
            AL=P*PI/180.0
            BAL=AL*180.0/PI
      WRITE (6,5)BAL
    5 FORMAT('1',T40,'ALPHA = ',F5.1,/)
      WRITE(6,10)
   10 FORMAT(T5,'ALT [FT]',T17,'VEL [FT]',T30,'DENS [SLUGS/FT**3]',
     & T52,'DYN PRESS [LBS/FT**2]',T77,'CD',T87,
     &  'CL',T96,'ATM TEMP [R]',/)
C
      DO 500 I=1,41
C
      M=0.7
      BETA=(1.0-M**2)**0.5
      KAPPA=1.0
      ALT(I)=AALT-500.0
      T(I)=518.69-(0.00356)*(ALT(I))
      MU(I)=1.458E-6*T(I)**1.5/(T(I)+110.4)
      RO(I)=0.0023769*(T(I)/518.69)**5.26669
      VEL(I)=M*(1.4*1617.0*T(I))**0.5
      RE(I)=RO(I)*VEL(I)*47.3329/MU(I)
      REN=RO(I)*VEL(I)*30.0/MU(I)
      REB(I)=RO(I)*VEL(I)*192.7/MU(I)
      CFW(I)=-0.0014*(LOG10(RE(I))-9.0237)
      CFB(I)=-0.001945*(LOG10(REB(I))-8.419)
      CFN=-0.00197*LOG10(REN)+0.01613
      PI=3.141592654
      A=3.0511
      B=154.0
      L=880110
      S=7773.0+625.0
      SB=PI*(18.0**2)
      SS=2*PI*18.0*192.7
      SP=6937.2
      SNWET=6785.85
C     S=10545.0+1525.0
      LL=1.2
      DP=0.5*RO(I)*(VEL(I)**2)*S
      CL=L/DP
      E=1.0/(1.0/0.85+1.0/1.4+1.0/0.05)
      F=28.0
      CLAW(I)=(2*PI*A)/(2.0+SQRT((A*BETA/KAPPA)**2*
     & (1.0+1.0/BETA**2)+4.0))
      E2=1.1*CLAW(I)/(A*PI)
      CDL=CL**2/(PI*A*E2)
      SREF=7773.0+625.0
      S=10545.0+1525.0
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      CDO=F/SREF
      [illegible]=CDOI(CLX[illegible]PI*A*E[illegible]
      [illegible]=DP/S
      R=CL/CD
      CDFW(I)=CFW(I)*(1.0+LL*0.12)*2.0*7773.0/10545.0
      CDOW(I)=CDFW(I)
      CDFB(I)=CFB(I)*SS/SB
      CDPB(I)=CFB(I)*(60.0/(192.7*36.0)**3+0.0025*(192.7/36.0))*SS/SB
      CDB2(I)=0.029/(CDFB(I)**0.5)
      CDOB(I)=CDFB(I)+CDPB(I)+CDB2(I)
      CDOWB(I)=CDOW(I)+CDOB(I)*SB/SREF
      CDAB=2*AL**2+0.62*1.2*SP/SB*AL**3
      CDIWB=CDL+CDAB*SB/SREF
      CDFN=CFN*SNWET/(PI*36.0*6.0)
      CDPN=CFN*(60.0/(30.0/6.0)**3+0.0025*(30.0/6.0))
     & *SNWET/(PI*36.0*6.0)
      CDIN=0.085*6.0
C
      CD2=CDOWB(I)+CDIWB+CDIN+CDPN+CDFN
      CD3=1.05*CD2
C
      CL2=CLAW(I)*AL
      L=CL2/CD3
      CL3=L*COS(AL)/(0.5*RO(I)*VEL(I)**2*SREF)
      D=CL3/CD3
      ALPHA=AL*180.0/PI
      CL4=SORT((CD3-CDOWB(I))*PI*A*E2)
      IF (AL .LE. 0.0) CL4=-CL4
      LOD=CL4/CD3
      IF (LOD .GT. LODMAX) ALF=ALPHA
      IF (LOD .GT. LODMAX) LODMAX=LOD
      AALT=ALT(I)
      WRITE(6,250)ALT(I),VEL(I),RO(I),DP,CD3,CL4,T(I)
  250 FORMAT(T6,F7.1,T19,F6.2,T34,F10.8,T58,F6.2,T74,F6.3,
     & T85,F6.3,T100,F5.1)
  500 CONTINUE
 1000 CONTINUE
      WRITE(6,120)LODMAX,ALF
  120 FORMAT(///,T30,'(L/D)max = ',F7.3,T58,'AT ALPHA = ',
     & F5.2,' [DEG]',//)
      STOP
      END
$
```

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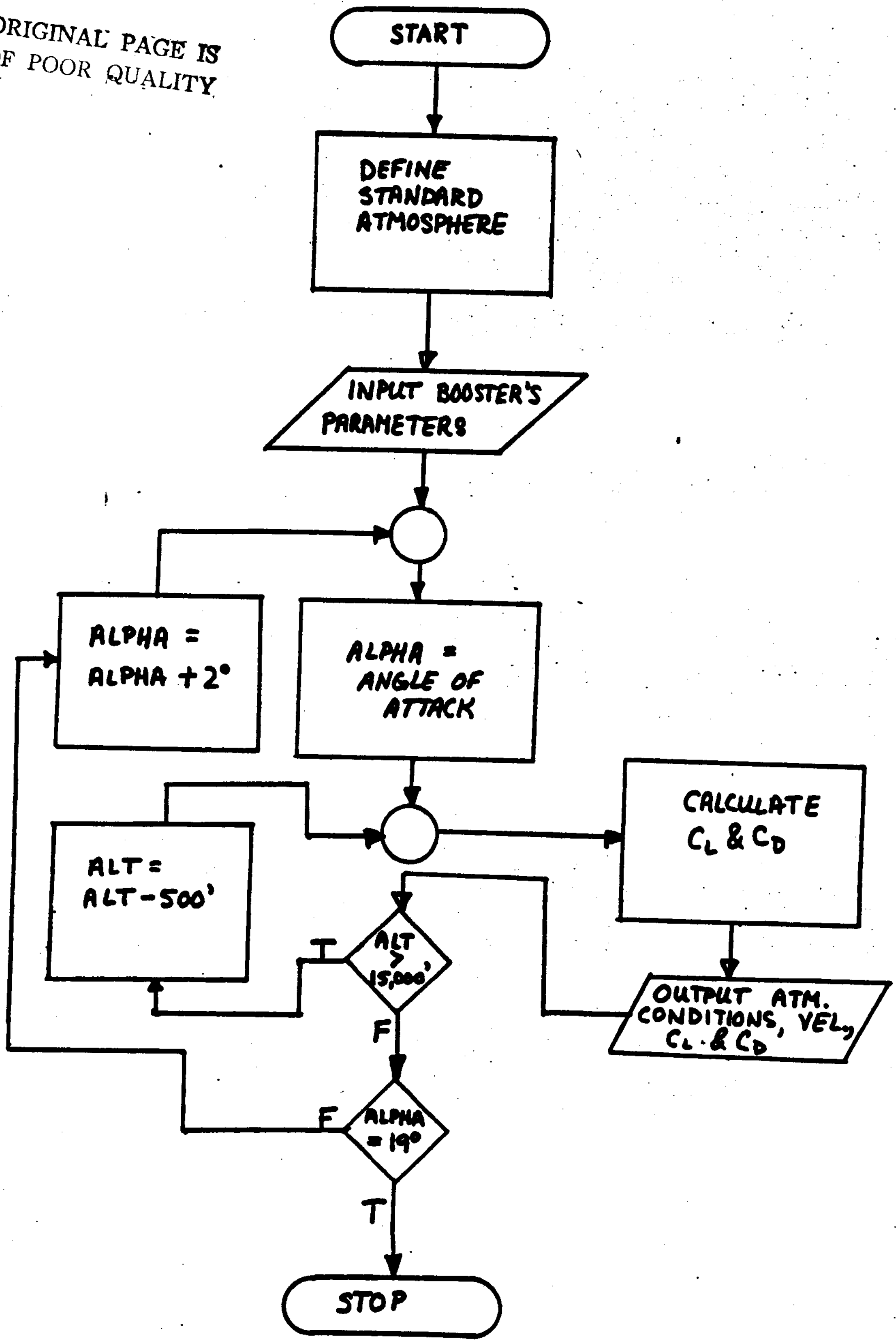
PROGRAM 3 FLOWCHART



SAMPLE OUTPUT FROM PROGRAM 3

ALPHA = 4.0

ALT [FT]	VEL [FT]	DENS [SLUGS/FT**3]	DYN PRESS [LBS/FT**2]	CD	CL	ATM TEMP [R]
35000.0	661.17	0.00055928	122.24	0.874	1.688	394.1
34500.0	662.66	0.00057271	125.75	0.861	1.675	395.9
34000.0	664.15	0.00058641	129.33	0.849	1.663	397.6
33500.0	665.64	0.00060036	133.00	0.837	1.651	399.4
33000.0	667.12	0.00061459	136.76	0.827	1.640	401.2
32500.0	668.60	0.00062909	140.61	0.816	1.629	403.0
32000.0	670.07	0.00064386	144.55	0.807	1.620	404.8
31500.0	671.54	0.00065891	148.57	0.798	1.610	406.6
31000.0	673.01	0.00067425	152.70	0.789	1.601	408.3
30500.0	674.48	0.00068987	156.92	0.781	1.593	410.1
30000.0	675.94	0.00070579	161.24	0.773	1.584	411.9
29500.0	677.40	0.00072200	165.65	0.766	1.577	413.7
29000.0	678.85	0.00073852	170.17	0.759	1.569	415.5
28500.0	680.31	0.00075533	174.79	0.753	1.562	417.2
28000.0	681.76	0.00077246	179.52	0.747	1.556	419.0
27500.0	683.20	0.00078990	184.35	0.741	1.550	420.8
27000.0	684.65	0.00080766	189.29	0.735	1.544	422.6
26500.0	686.09	0.00082574	194.34	0.730	1.538	424.4
26000.0	687.52	0.00084414	199.51	0.725	1.533	426.1
25500.0	688.96	0.00086288	204.79	0.720	1.527	427.9
25000.0	690.39	0.00088195	210.19	0.716	1.523	429.7
24500.0	691.82	0.00090137	215.70	0.712	1.518	431.5
24000.0	693.24	0.00092112	221.34	0.708	1.513	433.3
23500.0	694.67	0.00094123	227.10	0.704	1.509	435.0
23000.0	696.09	0.00096169	232.99	0.700	1.505	436.8
22500.0	697.50	0.00098251	239.00	0.697	1.501	438.6
22000.0	698.92	0.00100369	245.14	0.694	1.498	440.4
21500.0	700.33	0.00102525	251.42	0.690	1.494	442.1
21000.0	701.74	0.00104717	257.83	0.687	1.491	443.9
20500.0	703.14	0.00106947	264.38	0.685	1.488	445.7
20000.0	704.54	0.00109216	271.07	0.682	1.485	447.5
19500.0	705.94	0.00111524	277.89	0.679	1.482	449.3
19000.0	707.34	0.00113871	284.87	0.677	1.479	451.1
18500.0	708.74	0.00116257	291.98	0.675	1.477	452.8
18000.0	710.13	0.00118684	299.25	0.672	1.474	454.6
17500.0	711.52	0.00121152	306.67	0.670	1.472	456.4
17000.0	712.90	0.00123662	314.24	0.668	1.469	458.2
16500.0	714.29	0.00126213	321.97	0.666	1.467	460.0
16000.0	715.67	0.00128807	329.86	0.664	1.465	461.7
15500.0	717.05	0.00131444	337.91	0.663	1.463	463.5
15000.0	718.42	0.00134124	346.13	0.661	1.461	465.3

(L/D)max = 2.212 AT ALPHA = 2.00 [DEG]

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```
10 REM                  Chuck Harnden and Mike O'connor
20 REM                  CALCULATION OF CLALPH AT DIFFERENT MACH NUMBERS
30 REM                                PROGRAM 3
40 REM
50 REM  CLALPHW = LIFT CURVE SLOPE OF WING
60 REM  CLALPHC = LIFT CURVE SLOPE OF WING
70 REM  KNWB, KNBW, AND KNC ARE WING BODY INTERFERENCE FACTORS
71 REM  SEC = EXPOSED WING AREA OF CANARD
72 REM  SC =  PROJECTED WING AREA OF CANARD
73 REM  SEW = EXPOSED WING AREA OF WING
74 REM  SW  = PROJECTED WING AREA OF WING
75 REM  QRAT = AVERAGE DYNAMIC-PRESSURE RATIO ACTING ON THE AFT SURFACE
76 REM  BC = WING SPAN OF CANARD
77 REM  BW = WING SPAN OF WING
78 REM  DC = DIAMETER OF FUSELAGE AT CANARD
79 REM  DW = DIAMETER OF FUSELAGE AT WING
80 REM  AW = CROSS SECTIONAL AREA OF FUSELAGE
100 CLALPHW = 3.364
110 CLALPHC = 3.3315
120 SEW     = 7161
130 SW      = 10545
135 PI = 3.1452
140 SEC     = 625
150 SC      = 1849
160 KWBC    = .96
170 KWBW    = .99
180 KNC     = .0845
190 QRAT     = .9
200 IVW     = -3
210 BC      = 86
220 BW      = 190
230 DC      = 36
240 DW      = 36
250 AW      = 3.42
260 INPUT "KBWC=";KBWC
265 IF KBWC=0 THEN 430
270 INPUT "KBWW=";KBWW
280 INPUT "Mach Number =";M
300 INPUT "BVW";BVW
310 A=CLALPHC*(SEC/SC)*CLALPHW*QRAT*KWBC*IVW*(BW/2-DW/2)
320 B = (2*PI*AW*(BVW/2-DC/2))
330 CLALWCV = A/B/57.293
340 C =(CLALPHC/57.296)*(KNC+KWBC+KBWC)*SEC/SC
350 D =(CLALPHW/57.296)*(KWBW+KBWW)*QRAT*SW*SEW/(SC*SW)
360 E = CLALWCV
370 CLALPH = C+D+E
380 PRINT "Mach number =";M
390 LPRINT "Mach number =";M
400 PRINT "CLalpha = ";CLALPH
410 LPRINT "CLalpha =";CLALPH
420 GOTO 260
430 END
```

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Mach number = 1.5
CLalpha = .2361887
Mach number = 2
CLalpha = .2126911
Mach number = 2.5
CLalpha = .206774
Mach number = 3
CLalpha = .202602

```
10 REM                              CHUCK HARNDEN AND MIKE O'CONNOR
20 REM                                        PROGRAM 4
30 REM        CALCULATION OF DRAG COEF. AT AS A FUNCTON OF ALPHA AND MACH NU
40 REM
50 REM  CDO = DRAD COEFFICIENT AT ZERO LIFT
60 REM  CDOWB = ZERO LIFT DRAG COEFFICIENT OF WING BODY COMBINATION
70 REM  CDOP  = ZERO LIFT DRAG COEFFICIENT OF THE EXPOSED TAIL PANELS
80 REM  CDOW  = ZERO LIFT DRAG COEFFICIENT OF THE WING
90 REM  CDOB = ZERO LIFT DRAG COEFFICIENT OF THE BODY
100 REM SB/SREF = THE RATIO OF BODY BASE AREA TO THE REFERENCE AREA
110 SWETE=14322
115 SREF =10545
120 LAMALE=.7854
130 TOC=.12
140 SWE=7161
150 SS=19079.52
160 SB=1017.87
170 INPUT"MACH#";M
175 IF M=3.5 THEN 300
180 INPUT"CFW";CFW
190 INPUT"CDLE";CDLE
200 INPUT"CFB";CFB
203 INPUT"Alpha";ALPHA
205 CDOB=CFB*SS/SB+.125
210 CDWW=CDLE+16/3*(TOC)^2*SWE/SREF
215 CDFW=CFW*SWETE/SREF
220 CDOW=CDFW+CDWW
230 CDLWB=.07039+.6791*ALPHA^2+.00081*ALPHA^4
240 CDOWB=CDOW+CDOB*SB/SREF
250 CD=CDOWB+CDLWB
260 PRINT "Mach number =;M
261 LPRINT "Mach number =;M
270 PRINT "Alpha =";ALPHA
271 LPRINT "Alpha =";ALPHA
280 PRINT "CD = ";CD
281 LPRINT "CD = ";CD
290 IF ALPHA=1.0472 THEN 170
293 GOTO 203
300 END
```

Mach number =;M 1.5
Alpha = 0
CD = .3837323
Mach number =;M 2.0
Alpha = .17453
CD = .404419
Mach number =;M 2.5
Alpha = .34907
CD = .4664926
Mach number =;M 3.0
Alpha = .5236
CD = .5699733
Mach number =;M 1.5
Alpha = .69813 2
CD = .7149083
Mach number =;M 2.0
Alpha = .87266
CD = .9013608
Mach number =;M 2.5
Alpha = 1.0472
CD = 1.129426
Mach number =;M 3.0
Alpha = 0
CD = .4728455
Mach number =;M ~~3~~1.5
Alpha = .17453
CD = .4935321
Mach number =;M ~~4.5~~ 2.0
Alpha = .34907
CD = .5556057
Mach number =;M 2.5
Alpha = .5236
CD = .6590864
Mach number =;M 3.
Alpha = .69813
CD = .8040214
Mach number =;M 1.5
Alpha = .87266
CD = .9904739
Mach number =;M 2.
Alpha = 1.0472
CD = 1.21854
Mach number =;M 2.5
Alpha = 0
CD = .5218952
Mach number =;M 3.
Alpha = .34907
CD = .6046555
Mach number =;M 1.5
Alpha = .5236
CD = .7081361
Mach number =;M 2.
Alpha = .69813
CD = .8530711
Mach number =;M 2.5
Alpha = 1.0472
CD = 1.267589
Mach number =;M 3.
Alpha = .87266
CD = 1.039524
Mach number =;M 1.5
Alpha = 1.0472
CD = 1.267589
Mach number =;M 2.

CD = .55123
Mach number =;M 2.5
Alpha = .17453
CD = .5719166
Mach number =;M 3.
Alpha = .34907
CD = .6339903
Mach number =;M 1.5
Alpha = .5236
CD = .7374709
Mach number =;M 2
Alpha = .69813
CD = .8824059
Mach number =;M 2.5
Alpha = .87266
CD = 1.068858
Mach number =;M 3
Alpha = 1.0472
CD = 1.296924

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```
C                              PROGRAM # 5

C        PROGRAM FOR FLYBACK BOOSTER LIQUID ROCKET ANALYSIS

         PI=3.14159
         XISP=320.0
         TBOOS=1500000.0
         TTOT2=6230.0
         GAMMA=1.24
         XMBAR=22.0
         GC=32.2
         GE=32.2
         RLITT=2.3
         PA=1.04
         RBAR=1545.0
         DOTMB=(GC*TBOOS)/(XISP*GE)
         R=RBAR/XMBAR
         CP=(GAMMA*R)/(GAMMA-1.0)
         CV=CP/GAMMA
         UE=XISP*GE
         PE=PA
C        PTOT2=PE/((1.0-((UE**2)/(2.0*GC*CP*TTOT2)))**(GAMMA/
C     *  (GAMMA-1.0)))
         XME=SQRT((2.0/(GAMMA-1.0))*(((PTOT2/PE)**((GAMMA-1.0)/GAMMA))-1.0))
         TE=TTOT2-((UE**2)/(2.*CP*GC))
         A=SQRT(GAMMA*GC*R*TE)
         TSTAR=TTOT2/(1.+((GAMMA-1.)/2.))
         ATHOA=SQRT(GAMMA*GC*R*TSTAR)
         USTAR=ATHOA
         QR=(UE**2)/(2.*GC*(1.-(TE/TTOT2)))
         QRBAR=QR*XMBAR
         GAMM1=SQRT(GAMMA*(2.0/(GAMMA+1.0))**((GAMMA+1.0)/(GAMMA-1.0)))
         WRITE(6,799)TBOOS,XMBAR,RLITT
  799    FORMAT('0',5X,'Thrust=',F9.1,1X,'lbf',5X,'M=',F4.1,17X,'r=',
      *  F3.1)
         WRITE(6,800)R,CP,CV
  800    FORMAT('0',5X,'R=',F6.3,17X,'Cp=',F7.3,13X,'Cv=',F7.3)
         WRITE(6,801)XISP,UE,XME
  801    FORMAT('0',5X,'Isp=',F5.1,1X,'sec',12X,'Ue=',F9.3,1X,'ft/sec',
      *  4X,'Me=',F5.3)
         WRITE(6,802)DOTMB,PA,PE
  802    FORMAT('0',5X,'m=',F9.3,1X,'lbm/sec',6X,'Pa=',F5.2,1X,'psi',
      *  11X,'Pe=',F4.1,1X,'psi')
         WRITE(6,803)PTOT2,TTOT2,TE
  803    FORMAT('0',5X,'Pt=',F9.3,1X,'psi',9X,'Tt=',F6.1,1X,'R',
      *  12X,'Te=',F8.3,1X,'R')
         WRITE(6,804)TSTAR,USTAR,QR
  804    FORMAT('0',5X,'T =',F8.3,1X,'R',12X,'U =',F8.3,1X,'ft/sec',
      *  5X,'Qr=',F12.3)
         DOTM=DOTMB
```

```
      ASTAR=(DOTM/GAMM1*SQRT((R*TTOT2)/GC))/(PTOT2*144.0)
      ASTAR=1.55
      PTOT2=(DOTM/GAMM1*SQRT((R*TTOT2)/GC))/(ASTAR*144)
      DSTAR=SQRT(ASTAR*4.0/PI)
      GAMM2=(GAMMA+1.0)/(2.0*(GAMMA-1.0))
      AEXIT=ASTAR/XME*((2./(GAMMA+1.))*(1.+((GAMMA-1.)/2.)*XME**2))**GAMM2
      DEXIT=SQRT(AEXIT*4.0/PI)
      N=5
      WRITE(6,809)N
809   FORMAT('0',5X,'Number of engines ',I2)
      WRITE(6,810)ASTAR,DSTAR
810   FORMAT(7X,'A =',F7.3,1X,'sq ft ',5X,'D =',F6.3,1X,'ft')
      WRITE(6,811)AEXIT,DEXIT
811   FORMAT(7X,'Ae=',F7.3,1X,'sq ft ',5X,'De=',F6.3,1X,'ft')
      STOP
      END

 $
```

APPENDIX C

Sample Calculations

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Calculations

Aerodynamic Center

$$\frac{d}{b} = \frac{36 ft}{190 ft} = 0.1895$$

From Fig( ) $K_{B(W)} = .275$
$K_{W(B)} = 1.16$

$$(C_{N\alpha})_B = 2 \text{ rad}^{-1}$$

$$(C_{L\alpha})_e = 8 \tan^{-1}\left(\frac{\pi A}{16 + \pi A/(1+2\lambda \tan \Lambda_{LE})}\right)$$

$$= 8 \tan^{-1}\left(\frac{\pi(3.423)}{16 + \frac{\pi(3.423)}{(1+2(.1037)\tan 45^\circ)}}\right) = 3.364 \text{ rad}^{-1}$$

$$C_{L\alpha_N} = (C_{N\alpha})_B \frac{\pi d^2}{4 S_W}$$

$$= 2 \text{ rad}^{-1}\left(\frac{\pi(36 ft)^2}{4(10,545 ft^2)}\right) = 0.19305 \text{ rad}^{-1}$$

$$C_{L\alpha_{W(B)}} = K_{W(B)} (C_{L\alpha})_e \frac{S_e}{S_W}$$

$$= (1.16)(3.364 \text{ rad}^{-1}) \frac{7161 ft^2}{10,545 ft^2} = 2.650 \text{ rad}^{-1}$$

$$C_{L\alpha_{B(W)}} = K_{B(W)} (C_{L\alpha})_e \frac{S_e}{S_W}$$

$$= (0.275)(3.364 \text{ rad}^{-1}) \frac{7161 ft^2}{10,545 ft^2} = 0.62823 \text{ rad}^{-1}$$

$$\left(\frac{X_{ac}'}{C_{re}}\right)_{W(B)} \qquad \frac{\beta}{\tan \Lambda_{LE}} = \frac{.714}{1} = .714$$

$$A \tan \Lambda_{LE} = (3.423)(1) = 3.423$$

From Figures( ) and ( ) the interpolated value of

$$\left(\frac{X_{ac}'}{C_{re}}\right)_{W(B)} = 0.55$$

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$\left(\frac{X_{ac}'}{C_{re}}\right)_{B(W)}$ $\quad \frac{1}{4} A(1+\lambda)\tan\Lambda_{LE}$

$$= \frac{1}{4}(3.423)(1+.1039)\tan 45° = 0.9446$$

From Fig (12) $\left(\frac{X_{ac}'}{C_{re}}\right)_{B(W)} = 0.48$

$\left(\frac{X_{ac}'}{C_{re}}\right)_N$

$$(f)_{eq} = (f)_{nose} + 1.6(f)_{forebody}$$

where (f) represents fineness ratios

$$(f)_{nose} = \frac{l_n}{d} = \frac{25}{36} = 0.6944$$

$$(f)_{forebody} = \frac{l_F}{d} = \frac{91.7}{36} = 2.5472$$

$$(f)_{eq} = 0.6944 + 2.5472 = 3.2416$$

$$l_{eq} = (f)_{eq}\, d = (3.2416)(36\text{ ft}) = 116.698\text{ ft.}$$

$\left(\frac{X_{ac}}{l_{eq}}\right)_N = -0.666$ using ellipsoid nose from Figure (12)

$$\left(\frac{X_{ac}'}{C_{re}}\right)_N = \left(\frac{X_{ac}}{l_{eq}}\right)\left(\frac{l_{eq}}{C_{re}}\right) = (-0.666)\left(\frac{116.698}{77}\right) = -1.0103$$

Finally:

$$\frac{X_{ac}'}{C_r} = \frac{(-1.0103)(.19305\text{ rad}^{-1}) + (0.55)(2.65\text{ rad}^{-1}) + (0.48)(.62823\text{ rad}^{-1})}{(.19305) + (2.65) + (.62823)}$$

$$\frac{X_{ac}'}{C_r} = 0.45055$$

$$X_{ac}' = 34.69\text{ ft from } C_r$$

Determining constants for wing-body tail lift curve

$C_{L\alpha})_e$ from 4.1.3.2

$$C_{L\alpha})_e = \frac{8\tan^{-1}\pi A}{16+\pi A/(1+2\lambda\tan\Lambda_{LE})}$$

$$A = AR = \frac{b^2}{S} = \frac{140^2}{10545} = 3.423$$

$$\lambda = \frac{C_T}{C_R} = \frac{8}{77} = .1039$$

$$\Lambda_{LE} = 45°$$

$$C_{L\alpha})_e = \frac{8\tan^{-1}\pi(3.423)}{16+\pi 3.423/(1+2(.1039)\tan 45°}$$

$$C_{L\alpha})''_W = 3.364$$

$$C_{L\alpha})_c = 3.315$$

$[K_N + K_{W(B)} + K_{B(W)}]^c$ are found from 4.3.1.2

$[K_{W(B)} + K_{B(W)}]_W$ are found from 4.3.1.2

Wing	M	1.5	2.0	2.5	3
$\beta = \sqrt{M^2-1}$	β	1.118	1.732	2.291	2.828
$\frac{\beta d}{C_{r_w}}$		.52	.81	1.07	1.32
$\beta\cot\Lambda_{le}$		1.68	2.44	3.23	3.99
$K_{BW}[\beta(C_{L\alpha})_e(\lambda_e+1)(\frac{b}{d}-1)]$		3.5	2.5	2.1	1.75
K_{BW}		.07	.026	.017	.012

Canard

M	1.5	2.0	2.5	3
β	1.118	1.732	2.291	2.828
$k_{B(W)}$	1.6	.92	.80	.68

$K_{W(B)c} \longrightarrow \frac{b}{d} = .419 \xrightarrow[\text{graph}]{\text{from}} K_{W(B)} = .96$

$K_{W(B)w} \longrightarrow \frac{b}{d} = .19 \xrightarrow[\text{graph}]{\text{from}} K_{W(B)} = .99$

$$\frac{b''}{2} - \frac{d''}{2} = \frac{190}{2} - \frac{36}{2} = 77 \text{ feet}$$

$$K_n = \frac{(C_{L\alpha})_w S_{wref}}{(C_{L\alpha})_w S_e}$$

$$K_n = \frac{2(1017.9)}{3.364(7161)} = .0845$$

M	1.5	2.0	2.5	3.0
β	1.118	1.732	2.291	2.828
$\beta A_e'$	3.35	5.2	6.87	8.48
$\dfrac{\frac{b_v'}{2} - \frac{d'}{2}}{\frac{b'}{2} - \frac{d'}{2}}$	.75	.725	.7	.675

$\frac{b_v'}{2} - \frac{d'}{2} = 25(.75 \quad .725 \quad .7 \quad .675)$

$\frac{b_v'}{2} =$	36.75	36.125	35.5	34.875
$\frac{b_v''/2}{b''/2} =$	.193	.190	.186	.184

$I_{vw}'(w'') = -.4$

Liquid Rocket Analysis

Sample Calculations

Injector Analysis

Mass Flow of Oxygen: $\dot{m}_o = \frac{r}{r+1}\dot{m} = \frac{2.3}{3.3}(4687.5) = 3267.045 \frac{lbm}{sec}$

Mass Flow of Fuel: $\dot{m}_f = \frac{1}{r+1}\dot{m} = \frac{1}{3.3}(4687.5) = 1420.455 \frac{lbm}{sec}$

Area of one fuel orifice: $\frac{\text{Area of total fuel}}{\#\text{ of fuel orifices}}$

$$\frac{418.879 \text{ in}^2}{3700} = 0.1698 \text{ in}^2$$

Pressure Drop Across the Injector:

$$\Delta P = \left(\frac{\dot{m}_o}{C_{o_o} \rho_o A_o}\right)^2 \frac{\rho_o}{2g_c}$$

$$\Delta P = \left(\frac{3267.045}{0.61\,(71.2)(4.3633)}\right)^2 \frac{71.2}{2(32.174)}$$

$$\Delta P = 2.2837 \text{ psi}$$

Velocity out of one injector orifice:

$$V_f = C_{o_f}\sqrt{\frac{2g_c\,\Delta P}{\rho}}$$

$$V_f = 0.61\sqrt{\frac{2(32.174)(328.852)}{47.299}}$$

$$V_f = 12.903 \text{ ft/sec}$$

Angle of Deflection for the fuel:

$$\gamma_f = \sin^{-1}\left[\frac{\dot{m}_o V_o \sin\gamma_o}{\dot{m}_f V_f}\right]$$

$$\gamma_f = \sin^{-1}\left[\frac{3267.045\,(10.516)(\sin 30)}{1420.455\,(12.903)}\right]$$

$$\gamma_f = 69.595^\circ$$

HEAT TRANSFER

CALCULATION OF q:

$$q = C_p \dot{m} \Delta T = C_p \dot{m} (T_{out} - T_{in})_c$$

$$C_p = \frac{326.841}{550} = 0.59425 \frac{Btu}{lb\text{-mole}\cdot{}^\circ R}$$

$$C_p = \frac{C_p}{M} = \frac{0.59425}{22} = 0.02701 \frac{Btu}{lbm\text{-}{}^\circ R}$$

$T_{in} = 167^\circ R$

$T_{out} = -427^\circ F = 33^\circ R$

$$q = (.02701)(4687.5)(167-33)$$

$$q = 16.965 \frac{Btu}{sec}$$

Calculations: Propulsions - Air Breathing Engines

1). Thrust: change from sea level to a higher altitude.

equation: $$(F_{altitude}) = F_{\substack{sea\\level}} \cdot \sqrt{\frac{T_{sea\ level}}{T_{altitude}}}\ \frac{P_{altitude}}{P_{sea\ level}}$$

$$\left(F_{10{,}668\ ft.}\right)_{CF6\text{-}80C2} = (62{,}500\ lb_f.)\sqrt{\frac{518.69\,°R}{418.0\,°R}}\left(\frac{1420.\ lb_f/in^2}{2116.2\ lb_f/in^2}\right)$$

$$\left(F_{10{,}668\ Ft.}\right)_{CF6\text{-}80C2} = 43{,}550.5\ lb._f$$

2). Thrust: change from static conditions to a velocity (mach #).

equation: $$\left(F_{x\text{-}mach\,\#}\right)_{\substack{new\\engine}} = \left(\frac{F_{zero\ mach\,\#}}{F_{new\ mach\,\#}}\right)_{\substack{NASA's\\generic\\engine}} \cdot \left(F_{zero\ mach\#}\right)_{\substack{new\\engine}}$$

$$\left(F_{mach\ .2}\right)_{CF6\text{-}80E2} = \left(\frac{17{,}250\ lb_f}{15{,}180\ lb_f}\right)_{\substack{NASA's\\generic\\engine}} \cdot \left(62{,}500\ lb_f\right)$$

$$\left(F_{mach\ .2}\right)_{CF6\text{-}80C2} = 55{,}000\ lb_f$$

3). Specific Fuel Consumption: change from sea level conditions to a higher altitude.

equation: $$\left(SFC_{\substack{sea\\level}}\right)_{\substack{new\\engine}} = \left(\frac{(SFC_{sea\ level})}{(SFC_{altitude})}\right)_{\substack{NASA's\\generic\\engine}} \cdot \left(SFC_{\substack{sea\\level}}\right)_{\substack{new\\engine}}$$

$$(SFC)_{CF6\text{-}80C2} = \left(\frac{.5}{.6}\right)_{\substack{NSA's\\generic\\engine}} \cdot (.3321)_{CF6\text{-}80C2}$$

$$\left(SFC_{1524\ ft.}\right)_{CF6\text{-}90E2} = .3985\ \frac{lb_m}{lb_f\,hr.}$$

4). Frontal Area : total frontal area of all engines with cowlings.

equation: $$(S_\pi)_{engine} = (\#\text{ of engines})(\pi)\left(\frac{d}{2}\right)^2_{engine\ cowling}$$

$$(S_\pi)_{CF6\text{-}80C2} = (4)(\pi)\left(\frac{112\ in.}{2}\right)^2_{CF6\text{-}80C2}$$

$$(S_\pi)_{CF6\text{-}80C2} = 39{,}409.9\ in.^2 = 273.68\ ft.^2$$

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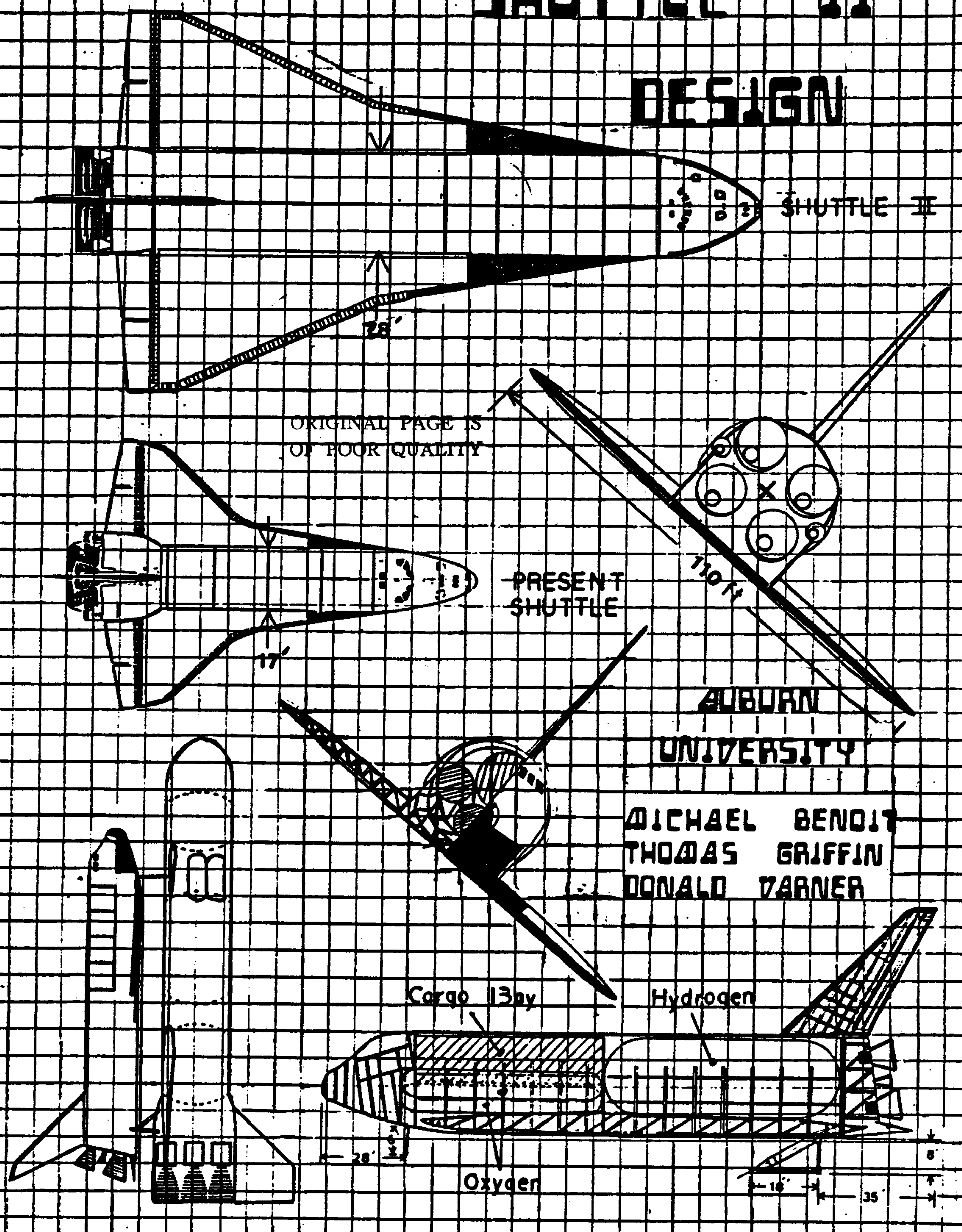
SPACE SHUTTLE II DESIGN



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FINAL DESIGN REPORT

FOR

SPACE SHUTTLE II

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Submitted by: Michael J. Benoit
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Date Submitted: 22 May 1987

ABSTRACT

Work was completed this quarter on the design of the Shuttle II orbiter. Included in this final report is a brief explanation and continuation of the preliminary and intermediate design phases. During the final phase of the project, the trajectory analysis was finalized. Also, all A.S.T.S. groups agreed upon respective masses with regard to the center of gravity locations and vehicle mating. Overall c.g. locations for assorted weight conditions were determined for Shuttle II. Nonintegral versus integral internal tank structures, one-piece wing planform, and differing internal structural makeup were investigated. An overall cost analysis was also performed to complete the Shuttle II project.

TABLE OF CONTENTS

LIST OF SYMBOLS

dv/dt= Acceleration of vehicle....................ft/sec^2

b= Wing span......................................ft

g= Gravity attraction at any point................ft/sec^2

g_e= Gravity on earth's surface...................ft/sec^2

h1= Altitude at booster separation................N.mi.

h2= Altitude of Shuttle II orbit..................N.mi.

Isp= Specific Impulse.............................sec

M= Mass...lbm

$\dot{M}$= Mass flow rate..................................lbm/sec

M_i= Initial mass...................................lbm

M_f= Final mass.....................................lbm

M_s= Mass of Shuttle II.............................lbm

M_c= Mass of Cargo..................................lbm

M_{H_2}= Mass of hydrogen...............................lbm

M_{O_2}= Mass of oxygen.................................lbm

MR= Mass ratio....................................(unitless)

S= Wing area......................................ft^2

T= Thrust...lbf

V=Volume..ft^3

V_{H_2}= Volume of hydrogen.............................ft^3

V_{O_2}= Volume of oxygen...............................ft^3

LIST OF FIGURES

INTRODUCTION

The information contained within this report deals with a design effort which was carried out on a new, improved Space Shuttle Orbiter, known as Shuttle II. This project was undertaken as part of a design study consisting of preliminary, intermediate, and final design stages. These stages are now reviewed.

I. Review of Preliminary Design Stage.

Design was begun fall quarter 1986. Initial design specifications set forth by NASA for the Shuttle II called for:

* Two-stage vehicle utilizing a fully reusable, winged, flyback booster.

* Payload capable of 40,000 lb. transported to space station orbit (28.5 degree inclination at 270 NM altitude).

* 40,000 lb. payload return capability.

* Cargo bay having dimensions of 15 ft. diameter X 60 ft. length.

* Engine propellant (second stage) of liquid oxygen and liquid hydrogen.

* Maximum velocity of 7000 fps. for staging.

* Maximum acceleration of 3 G's during ascent into orbit.

* Projected first flight in 2005; therefore, technology and design freeze approximately 1997.

Design work was coordinated with the Booster group. Using a gravity turn analysis, it was determined that the

Shuttle II should have an initial mass of one million pounds. A Shuttle II vehicle weight was then calculated using a mass ratio of 4.5.

It was subsequently found that the calculated final velocity was not great enough to achieve the specified in-orbit altitude. Because of this, thrust and velocity were varied with values for velocity and mass ratio being tabulated in an unsuccessful attempt to achieve the correct final velocity.

By taking into account the determined initial mass of one million pounds, the Shuttle II was tentatively dimensioned with increased wing and fuselage areas. A computer program was written which utilized the assumed mass ratio of 4.5 to calculate the mass of the Shuttle II, mass of the fuel, and the corresponding fuel volume.

II. Review of Intermediate Design Phase.

Work on the Shuttle II design continued into the intermediate phase winter quarter 1987. Trajectory was finalized for the ASTS by use of the program S2TRAJ.BAS. Through this program, it was found that Shuttle II achieved its desired trajectory and orbital requirements. Briefly investigated also was a secondary trajectory into orbit utilizing a technique similar to a Hohmann transfer. (Appendices A,E)

Also written were computer programs which calculated

masses and volumes of the propellants as well as possible lengths and diameters of the propellant tanks. Corresponding empty and full tank masses were also calculated. (Appendices B,C)

A propulsion system for the Shuttle II was selected based on such criteria as: number of engines required, engine c.g. locations, and required firewall thicknesses. An Orbital Maneuvering System (OMS) and a Reaction Control System (RCS) were also selected.

Finally, structural design was undertaken to determine such considerations as: selection of construction materials with respect to vehicle structural loads, material cost comparisons, and overall Shuttle II configuration.

III. Review of Final Design Phase

Work on the final design phase began spring quarter 1987. All ASTS groups agreed upon respective masses with regard to launch by use of the program S2TRAJ.BAS. Weight and center of gravity locations were calculated by use of the program ACCEPT.FOR. A cost analysis on production of the Shuttle II was also performed. Nonintegral versus integral internal fuel tank structures were investigated with a decision being made to use nonintegral structures. Alternatives were considered such as different placement of cargo bay and fuel tanks to facilitate a more advantageous c.g. location.

THEORY

A. Trajectory

Initial design specifications call for the Shuttle II to achieve a circular orbit at a 270 NM altitude. To reach this orbit, a direct powered ascent was chosen. Important parameters of the trajectory such as time of flight, velocity, range, altitude, thrust, and G-forces were continuously calculated at time intervals from launch until attainment of desired orbit. These values were calculated using an iterative-type program, S2TRAJ.BAS. S2TRAJ executed these calculations with input of initial mass of the Shuttle II, engine thrust, and the altitude at which the gravity turn was initiated. (Fig. 1A, 1B)

A secondary method was considered which used a powered ascent up to some elliptical orbit with a perigee near 60 nautical miles and an apogee of 270 NM. Upon reaching this apogee, an impulse is applied using the OMS to adjust the elliptical orbit into a circular orbit. The maximum velocity change required for this maneuver would be 376 fps. To derive at this value, necessary velocities for orbits with perigees from 55 NM to 110 NM and apogees from 220 NM to 270 NM were calculated. This value was compiled by the program ORB shown in Appendix E. As can be seen from the section in this report concerning propulsion, this velocity change is within the capability of the OMS. It is hoped that this method will

reduce the fuel required to attain orbit and in turn, the total mass required for the shuttle system.

The program mentioned, which determined the actual trajectory, can also be used to determine the elliptical orbit for this secondary method. In order to determine if the burnout data supplied by this program actually attains orbit, the program ASTRO can be used (see Appendix F). This program takes burnout data and calculates the orbital elements. The burnout data can either be in the 3-dimensional celestial system mentioned previously, or in the 2-dimensional system used by the trajectory program. Pertinent results of importance for this program are as follows:

Ha(NM)	Hp(NM)	a(NM)	e	Vave(fps)	Vdiff
355.	185.	3710.	.0229	24723	285
350.	190.	3710.	.0216	24739	268
345.	195.	3710.	.0202	24755	252
340.	200.	3710.	.0189	24772	235

TABLE 1

Data found in this table conforms to the necessary orbital requirements as set by NASA. These necessary requirements set by NASA for the circular orbit phase are as follows:

a	3710 nautical miles
e	0 (circular orbit)
Ω, Omega	261 degrees (longitude of ascending node)
i	28.5 degrees (latitude of Cape Kennedy)
θ, Theta	0 to 360 degrees (circular orbit)
ω, omega	0 to 360 degrees (circular orbit)

However, because of the uncertainty involved in this procedure and the deduced assumptions, it was decided to proceed with the direct powered ascent trajectory. It should also be stressed that a Hohmann transfer be considered and analyzed for the Shuttle II's final orbit achievement.

B. Propulsion

From trajectory analysis, it was found that Shuttle II will be required to supply one million pounds of thrust at liftoff. The engine chosen for Shuttle II was the STME 481 rocket engine. (Fig. 2) A minimum of three of these engines will be necessary to produce the required liftoff thrust. The Shuttle II will be equipped with four of these engines to provide an engine-out successful mission capability, and also to increase the lifespan and reliability of the engines since they will not be required to operate at their maximum capability.

The examination of the efficiency of the Shuttle's engines becomes a pertinent matter in mission planning and cost analysis. Each engine is capable of producing 396,492 pounds of thrust. The flight configuration with four engines, three engines, and two engines (an impossible configuration as shown below), have been tabulated as follows:

Number of Engines	Thrust Each Produces	% of Max Thrust
4	250,000.33	63
3	333,333.33	84
2	500,000.00	126

TABLE 2.

As can be seen by the four and the three engine cases, the percent of the maximum thrust necessary to complete the mission are 63 percent and 84 percent, respectively. The percent difference between these two cases is 25 percent. The overall advantage yielded by the three engine versus four engine case is that more missions can be performed per engine in the four engine case. This can be seen in that the 25 percent difference between the operating thrusts of the two cases yields that for every four three-engine missions, five four-engine missions can be performed. Another important difference, as mentioned above, is the safety factor. The engine-out possibility for the three-engine mission would be fatal to the mission as can be seen from the two-engine data in Table 2. However, the engine-out possibility for the four-engine case would allow, with 16 percent reserve thrust, the success of the mission. Therefore, with the discussed advantages and the comparison of data in Table 2, it is intuitively obvious that the four-engine case be implemented.

The engine configuration decided upon utilizes a rectangular engine placement. (Fig. 3) The dimensions of this rectangle are approximately 35 ft. in the vertical and 32.5 ft. in the horizontal direction.

The maneuverability of the Shuttle II will be assisted by this rectangular engine configuration together with engine gimbaling. The engines will be gimbaled 10.5 degrees for pitch control and 8.5 degrees for yaw control. The limiting

area that the end of each nozzle can traverse is indicated by an ellipse with eccentricity of 0.3843 with reference to the nozzle as the center of the ellipse.

Investigated also was an Orbital Maneuvering System (OMS) for the Shuttle II. The OMS for the current orbiter consists of two 6000 pounds force engines. This system carries enough fuel to produce a total velocity change of 1000 fps. A normal shuttle mission requires a velocity change of less than 400 fps., allowing 600 fps. as needed for any abnormal circumstances. On performing a mass analysis for Shuttle II it was decided to keep the OMS at two 6000 lbf. engines with an increased fuel capacity. It should be noted that with the increased mass of Shuttle II maneuver time will be increased. (Fig.4)

The Reaction Control System (RCS) is utilized for altitude correction of Shuttle II. After investigation, it was recommended that the existing RCS in use on the present orbiter be utilized for Shuttle II as well.

C. Aerodynamics

The aerodynamic considerations for the Shuttle II are based upon its reentry requirements. These requirements dictate that Shuttle II be able to fly in the speed range of Mach 25 at reentry to Mach 0.28 at touchdown. This range demands hypersonic, supersonic, transonic, and subsonic flight configurations.

The hypersonic flight regime, with a speed range of Mach 5 and higher, is the least explored regime of the four. The accumulated data retrieved from the vehicles that have flown in this regime is limited.

The supersonic regime (with Mach numbers from 5 to 1.2), the transonic regime (with Mach numbers from 1.2 to 0.8), and the subsonic regime (with Mach numbers below 0.8) are all speed regimes which have been thoroughly explored for many years by many flight vehicles. Because of this, data concerning flight configuration and parameters is easily obtained. However, it should be noted that the shape of Shuttle II will be heavily dependent on the characteristics of the reentry (hypersonic) regime.

In consideration of the effects that the hypersonic regime had on Shuttle II's aerodynamics, it is important to note that the key point concentrated upon in this regime was overall shape. (Fig. 5) A breakdown of Shuttle II's key components and why they were shaped and sized as they were is listed below:

* Nose Cap - Contoured for hypersonic trim, performance, and heating

* Fuselage - Sized by payload requirements

* Double Delta Wing - Sized by 188 kt. landing design speed

* Vertical Tail - Sized by subsonic stability

* Flared Rudder - Speed Brake - Rudder sized by crosswind landing

* OMS Pod - Sized by OMS required fuel tankage

* Aft Fuselage - Sized by Shuttle II main engines

* Body Flap - Sized to protect main engines from reentry heating

* Full Span Elevons/Ailerons - Sized by hypersonic trim; pitchdown maneuver

It is important to note that the shape of the Shuttle II wing was determined through reference to previous NASA studies on shuttle concepts. (Ref. 2) The planform shape decided upon was that of an initial sweepback angle of 79 degrees with an outboard sweepback of 63 degrees. (Fig. 6)

It is also important to note that although the main component shapes were a function of the hypersonic reentry phase, that the subsonic and hypersonic regimes dictate the stability and control analysis of Shuttle II.

An important component of the stability and control analysis is the correct determination of overall center of gravity location. A breakdown of all major Shuttle II components by individual c.g. location is necessary in order that the overall c.g. location may be determined.

To analyze the major components, a total weight breakdown of all major structural sections of the existing orbiter (ie. wing group, tail group, etc.) was obtained from the Marshall Space Flight Center in Huntsville, Alabama. (Appendix G)

By considering only the major components, an estimated weight analysis was determined by a ratio method. This method included comparison of the existing orbiter mass-to-area ratios to Shuttle II mass (unknown)-to-area (known) ratios. From this comparison, approximate component and section masses were found. (Fig. 7)

A similar method was utilized in the calculation of the approximate c.g. locations with reference to an origin located at the Shuttle II nose. By utilizing geometric configuration comparisons, c.g. locations relative to the nose are represented with a rectangular coordinate system, XYZ.

Two extreme conditions were considered for calculation of the c.g. locations. The first was that of launch with a 40,000 pound payload, full fuel complement, and all provisions and support equipment. The other condition was that of landing with no cargo, all fuel expended, and minimum amount of provisions and support equipment to contribute to weight. (Figs. 8,9)

Also considered was a third condition of landing with a 40,000 pound payload, all fuel expended, and minimum provisions and support equipment. (Fig. 10) Figure 11 illustrates the reference coordinate system and the resulting

c.g. shifts taking into account the above three conditions.

In conclusion, it is important to realize that although the Shuttle II design was based upon all four flight regimes, each regime in itself was a contributing factor to the success of the overall design configuration. The hypersonic regime dictated the shape and size, while the sub-hypersonic regimes dictated the location and size of components which in turn affect the stability and control of Shuttle II.

D. Materials and Structures

In selecting the materials for the airframe of the Shuttle II, the conventional tradeoff studies considering the weight and cost of production had to accurately reflect the compatibility requirements demanded of the structure by the external Thermal Protection System (TPS). The very low strength, brittle TPS tiles, when bonded to Shuttle II's skin, demand that when exposed to 115 percent of maximum expected loads during ascent and 100 percent during atmospheric reentry, that the skin not buckle. This skin-buckling requirement makes aluminum and titanium equivalent in terms of weight criteria since both materials have a ratio of compression modulus to density approximately equal to 10(10E7) inches. Since flight loads can accumulate high stresses, Shuttle II's structure favored the high-strength material such as titanium. Another consideration for selection of the structural material involved the thermal effects at atmospheric exit and entry at elevated temperatures. After considering such factors as TPS compatibility, heat capacitance, and strength as set forth by the buckling requirements, titanium proved to be the best rated material for the primary structure in terms of limiting total weight. However, the cost of titanium is approximately 300 percent greater than that of aluminum. Therefore, in terms of economic practicality, the selection of aluminum was made as the primary structural material for Shuttle II resulting from

these cost and production risk comparisons.

A major area of the structure where aluminum is not the primary structural material is the main engine thrust structure. The thrust structure supports one million pounds of main engine thrust and distributes the vertical stabilizer loads onto the fuselage. Here, the possibility of truss compression/buckling was much too great for the aluminum structural supports.

From Reference (15), NASA studies in minimum weight structural concepts were observed. These studies identified tradeoff allowances for the orbiter structure. Some of these studies also proved beneficial to Shuttle II's structure. For Shuttle II, a space frame concept for the thrust structure as an alternative to the competing plate girder concept together with an aft carrythrough spar with the bulkhead instead of a floating carrythrough was found to allgviate much unneeded weight. Through these weight and cost studies, NASA engineers selected the composite material systems used in the payload bay doors, the OMS pod external shell, and thrust structure. However, the graphite-epoxy composite used in the payload bay doors was found to fail due to moisture being absorbed by diffusion, accompanied by degradation at temperatures above 250 degrees Fahrenheit. For Shuttle II's structure, it was decided to us aluminum-lithium honeycomb panels in place of the graphite-epoxy composite. Figure 12 displays the primary structure with the corresponding materials used.

Also considered was the interrelation of the structure and aerodynamic characteristics. Emphasis was placed on such concerns as the size and mass of each fuel tank and its effects on the structural weight. Another concern was the required increase in wing area and thus an increase in total material and weight of Shuttle II.

Also investigated were the advantages and disadvantages of nonintegral as compared to integral internal fuel tank structures. The results found are as follows:

I. Nonintegral Internal Fuel Tanks

Advantages

1. Allows expansion space due to changes in temperature.

2. Reduced tank masses due to thinner shells made possible by insulation of tanks with reliable, light-weight material.

3. Allows space for conventional airplane-type spar/stringer structural configuration beneath tanks for support of wing/fuselage junction.

Disadvantages

1. Propellant tanks develop exterior moisture. This necessitates adequate plumbing and drainage vents to remove moisture from tank section.

2. Required truss support systems result in increased structural mass.

3. Enlarged fuselage diameter also results in increased structural mass.

II. Integral Internal Fuel Tanks

Advantages

1. Less structural materials required for truss support.

2. Reduced fuselage diameter results in less structural mass of fuselage.

3. No moisture removal system will be necessary because moisture will form on exterior of fuselage.

Disadvantages

1. No available space for spar/stringer structural supports of wing/fuselage junction.

2. Increased mass of tanks due to corresponding increase in shell thickness of each tank necessitated by thermal protection requirements.

3. Additional fuselage structural supports necessary to accomodate the increased tank masses.

By comparing the advantages and disadvantages associated with these two concepts, it was determined that the nonintegral internal fuel tanks proved to be the most advantageous even though a moisture removal system is necessary.

After aerodynamics had determined the best location within the fuselage to balance the center of gravity, the fuel tanks had to be positioned so as not to alter this location. The tanks will be made of 2219 Aluminum alloy with a skin thickness of 0.03166 ft. The four fuel tanks give a total empty mass of 32,301 pounds. The diameter and length of the three oxygen tanks are 15 ft. and 80 ft., respectively; while

those for the hydrogen tank are 30 ft. and 80 ft. (Figs. 13,14) Using these dimensions as design criteria, it was found that Shuttle II will definitely increase over the present orbiter in not only material masses but also in structural masses.

Accompanying this mass increase due to placement of fuel tanks within the fuselage, it was necessary to increase the wing area in order to supply the required lift. Again, this increase in structure demands additional material, and thus, more weight.

Additionally considered was the wing/fuselage junction. It was decided to utilize a conventional airplane-type of design of spars and stringers continuously connected to form a single wing planform which the fuselage would then be set upon. (Figs. 15,16)

It should be noted that at this point the Shuttle II structure consists of a significant amount of unused internal volume for which a suitable purpose has not yet been found. With the advancement of new technology and the discovery of new composites, the additional mass resulting from this unused volume may be diminished, thereby enabling more favorable performance characteristics.

Discussion/Summary

Final design for the Shuttle II was primarily concerned with finalization of trajectory, agreement between all ASTS groups on respective masses, weight and c.g. analysis, and weight and cost considerations with respect to materials and structures.

The use of the program S2TRAJ enabled the finalization of the Shuttle II trajectory. Knowledge of this precise trajectory into orbit provided important values such as: initial mass (M), mass at separation from booster, Shuttle II liftoff thrust, orbital inclination angles (ϕ, γ), and final velocity and altitude. Use of S2TRAJ by the other ASTS groups (Booster and Cargo Vehicle) enabled agreement on such issues as duration and magnitude of thrust during launch and correct vehicle mating locations with respect to weights and c.g. locations. The values obtained through the trajectory analysis dictated important design considerations such as: total amount of fuel required, overall configuration size, and required structural materials.

Research into the desired propulsion unit for Shuttle II had to satisfy certain specifications, among them being the capability of successful completion of mission after one engine loss. Therefore, a total of four main engines for Shuttle II was decided upon. Slight redesign to the Orbital Maneuvering System in the form of increased fuel capacity was

decided upon for maneuvering of the enlarged configuration. This decision for added fuel rather than an additional engine is also cost effective in terms of final weight. One disadvantage to this choice is the increased time required for each maneuver.

The design phase concerning aerodynamics for Shuttle II was primarily concerned with the reentry portion of the mission profile. The main consideration was the shaping and sizing of the major Shuttle II components with respect to the particular speed regime through which the vehicle is traveling. Another major consideration was that of stability and control analysis. This analysis depends heavily upon correct determination of the overall vehicle c.g. location which was found using the program ACCEPT.FOR.

The main guideline followed in the selection of materials for Shuttle II was a comparison to the existing orbiter and modern material technology. Taken into consideration were compatibility requirements, thermal effects, and costs. Comparisons between aluminum and titanium were investigated, with aluminum being chosen as the primary structural material. Selection of composite materials was not made in favor of aluminum honeycomb panels for certain areas of Shuttle II such as: payload bay doors and the OMS external pod shell. Also investigated were nonintegral versus integral internal fuel tank structures with the decision being made to utilize nonintegral tanks. A cost analysis was performed for Shuttle

II which indicated that an approximate total cost of development and production would be 6.558 billion dollars. (Fig. 17)

Finally, it should be noted that this design proposal (Fig. 18) is a major change over the existing orbiter. A change in not only dimensions, (Figs. 19,20), but also in concept to a degree. The Shuttle II will hopefully prove to be a much more adaptable, multi-role oriented component of the Advanced Space Transportation System than the present orbiter was ever envisioned to be. In conclusion, it is hoped to one day see the Shuttle II and Booster (Fig. 21) lifting off together bound for another successful mission in space.

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FIGURE 1A: GRAPHICAL REPRESENTATION OF TRAJECTORY

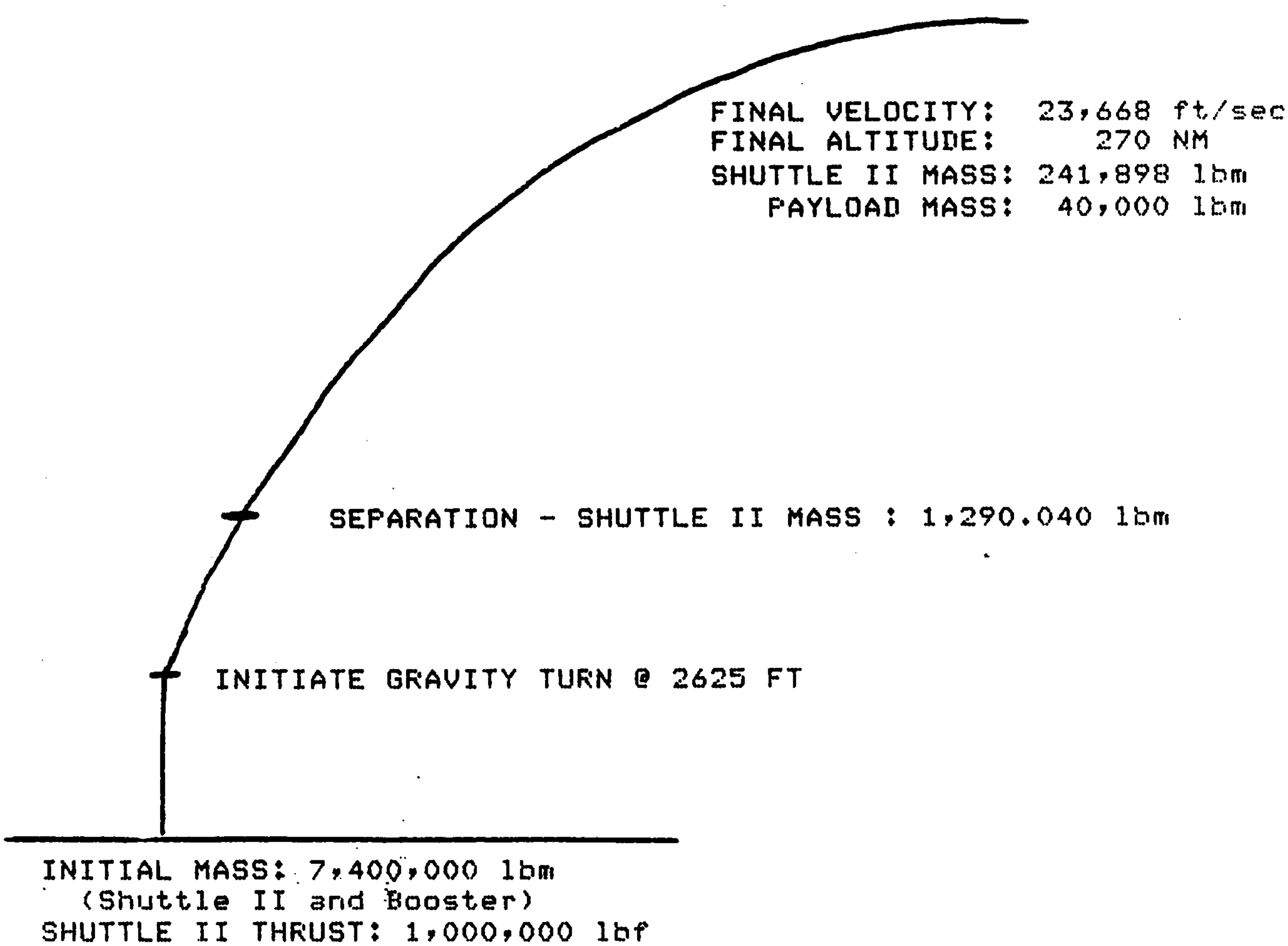


FIGURE 1B: TABULATED DATA

PROGRAM 1:

	INITIAL MASS	7,400,000.00	LBM.
(INPUT)	G-TURN ALTITUDE	2,625.00	FT.
	SHUTTLE II LIFTOFF THRUST	1,000,000.00	LBF.
(RESULTS)	SHUTTLE MASS AT SEPARATION	1,290,040.00	LBM.
	FINAL ALTITUDE	1,643,340.00	FT.
		270.00	NM
	FINAL VELOCITY	23,668.00	FT./S.
	PSI	103.80	DEGREES
	GAMMA 2	103.80	DEGREES
	SHUTTLE MASS IN ORBIT	241,898.00	LBM.

PROGRAM 2:

	SHUTTLE MASS IN ORBIT	241,898.00	LBM.
(INPUT)	MASS RATIO	4.59	
	MASS OF SHUTTLE FUEL BURNED	1,008,142.00	LBM.
	MASS OF OXYGEN	864,121.90	LBM.
(RESULTS)	MASS OF HYDROGEN	144,020.11	LBM.
	MASS OF PAYLOAD	40,000.00	LBM.
	VOLUME OF HYDROGEN TANK	32,510.23	FT.^3
	VOLUME OF OXYGEN TANKS (3)	12,099.16	FT.^3

PROGRAM 3:

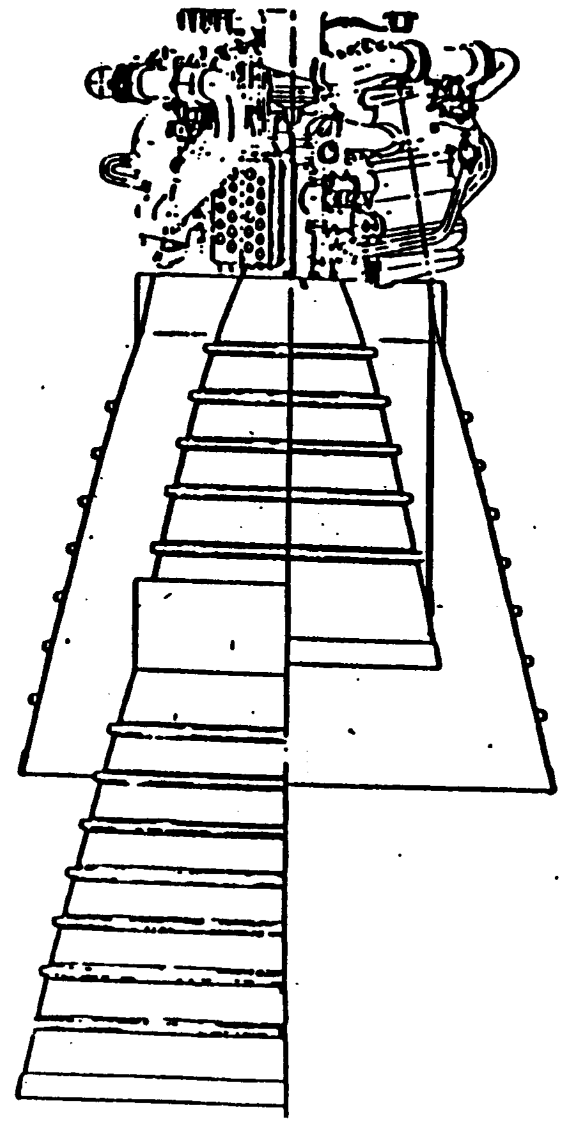
	VOLUME OF HYDROGEN / ONE TANK	32,510.23	FT.^3
(INPUT)	MAX. DIAMETER OF TANK	30.00	FT.
	MAX. LENGTH OF TANK	80.00	FT.
(RESULTS)	DIAMETER OF TANK	25.07	FT.
	LENGTH OF TANK	74.42	FT.
	VOLUME OF OXYGEN (EACH OF THREE TANKS)	4,033.05	FT.^3
(INPUT)	MAX. DIAMETER OF TANK	15.00	FT.
	MAX. LENGTH OF TANK	80.00	FT.
(RESULTS)	DIAMETER OF TANK	9.11	FT.
	LENGTH OF TANK	65.41	FT.

PROGRAM 4:

	WALL THICKNESS	0.03166	FT.
	METAL DENSITY	176.256	LBM/FT^3
(INPUT)	VOLUME OF HYDROGEN	32,510.230	FT.^3
	INSIDE TANK DIAMETER	25.035	FT.
	MASS OF HYDROGEN	144,020.000	LBM.
(RESULT)	MASS OF EMPTY TANK	16,338.000	LBM.
	VOLUME OF OXYGEN PER TANK	4,033.050	FT.^3
(INPUT)	INSIDE TANK DIAMETER	9.075	FT.
	MASS OF OXYGEN PER TANK	288,040.000	LBM.
(RESULT)	MASS OF EMPTY TANKS	5,3210.00	LBM.

FIGURE 2 **SPACE TRANSPORTATION MAIN ENGINE (STME)**

	STME 481
● PROPELLANTS	LOX/LH2
● NOZZLE AREA RATIO (STOWED/EXTENDED)	55/150
● VACUUM THRUST (LBF)	468K/481K
● VACUUM ISP (SEC)	449/461
● CHAMBER PRESSURE (PSIA)	3006
● MIXTURE RATIO (O/F)	6.0
● LENGTH (IN)	139/219
● NOZZLE EXIT DIAMETER (IN)	76.2/126.3
● ENGINE INSTALLED WT (LBM)	7142
● SEA LEVEL THRUST (LBF) (STOWED)	397K
● SEA LEVEL ISP (SEC)	380.4
● FLOWRATE (LB/SEC)	1043.4



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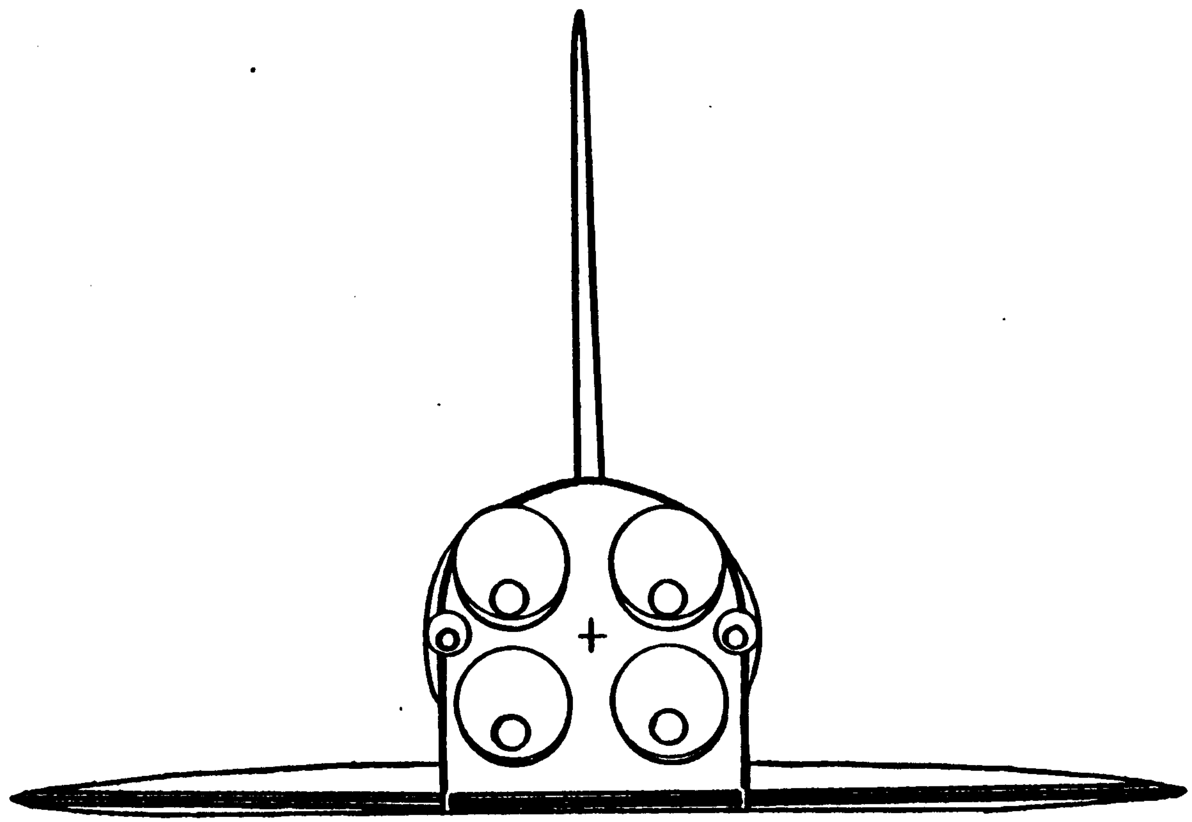

FIGURE 3 SQUARE POSITIONING OF ENGINES

FIGURE 4

OMS MASS ANALYSIS

	SHUTTLE I		SHUTTLE II	
MASS SHUTTLE	155,000	LBM	241,898	LBM
MASS PAYLOAD	65,000	LBM	40,000	LBM
TOTAL MASS	220,000	LBM	281,898	LBM
VELOCITY CHANGE	1,000	FT/S	1,000	FT/S
MASS/VELOCITY	220	LBM S/FT	282	LBM S/FT
SCALE FACTOR	220/220 = 1		282/220 = 1.28	
PROPELLANT				
N2OH	14,866	LBM	19,028	LBM
MMH	9,010	LBM	11,353	LBM
TOTAL	23,876	LBM	30,561	LBM

CHANGE IN PROPELLANT MASS
6,685 LBM

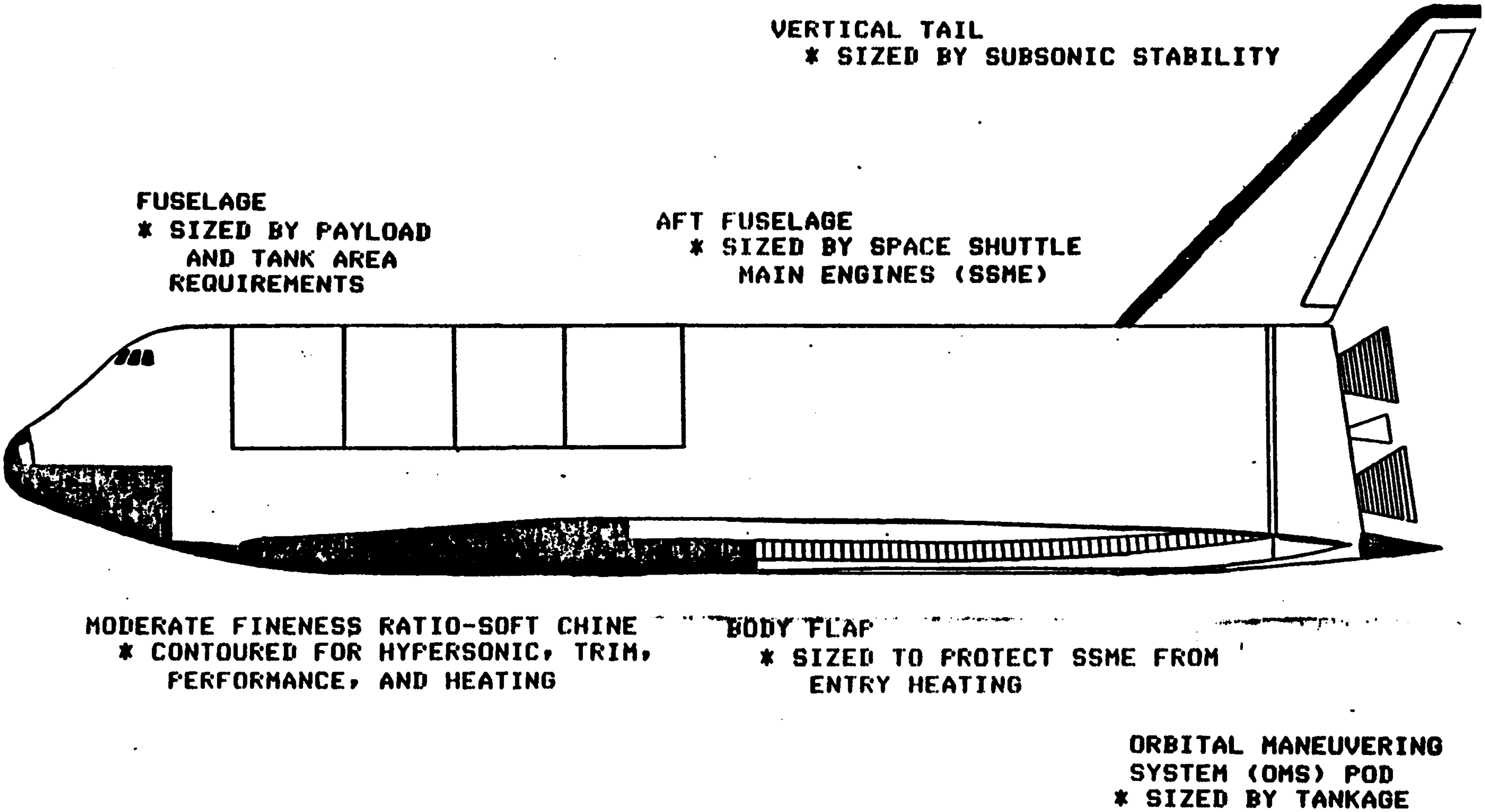

Figure 5 AERODYNAMIC SIZING CRITERIA

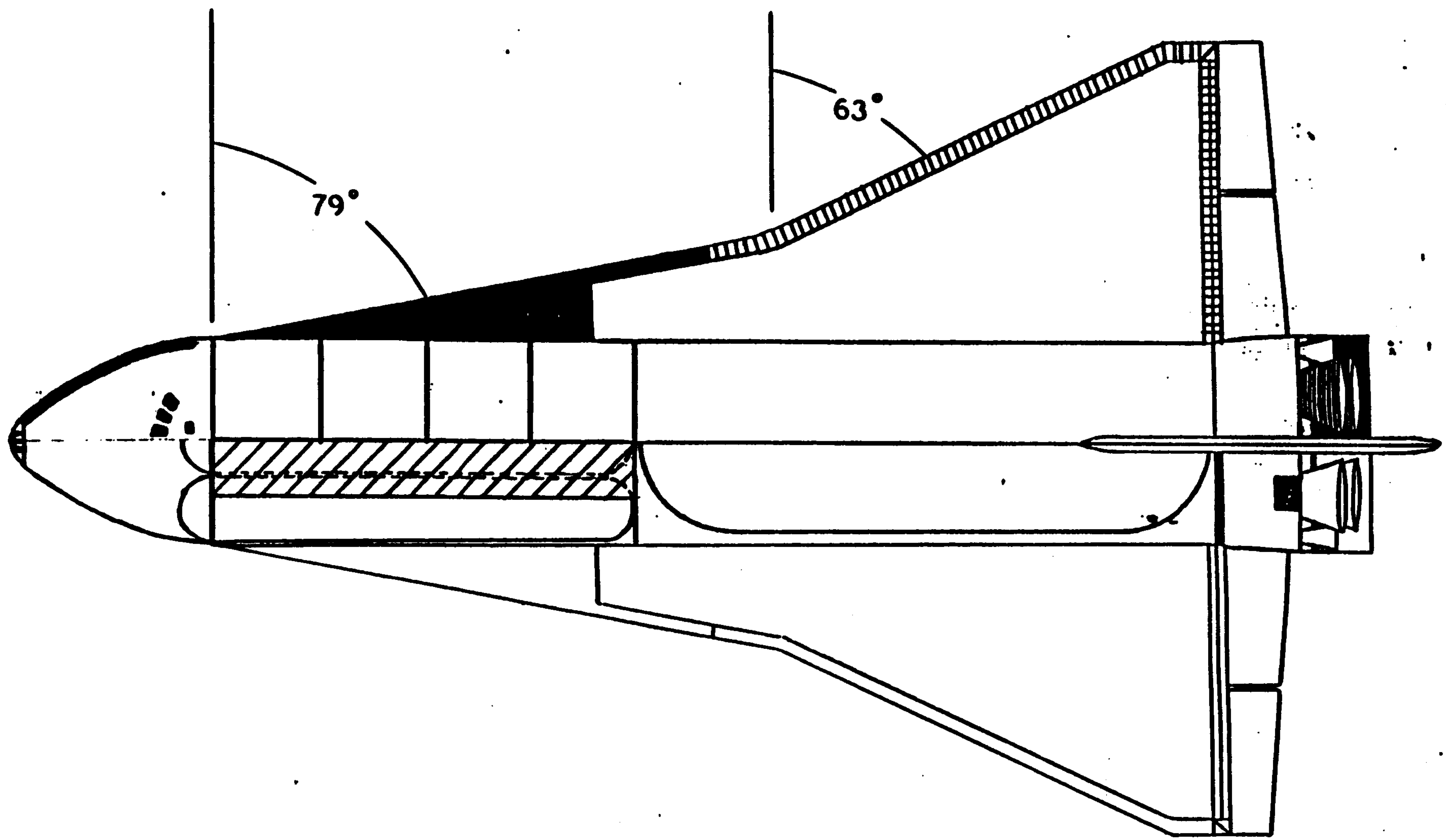

Figure 6 CUT-AWAY VIEW SHOWING PROPELLANT TANKS AND CARGO

FIGURE 7

WEIGHT ANALYSIS OF MAJOR SHUTTLE COMPONENTS

DESCRIPTION OF COMPONENT	MASS
CARGO	40000.00
OXYGEN TANK # 1 + FUEL	5210.00 + 288040.00
OXYGEN TANK # 2 + FUEL	5210.00 + 288040.00
OXYGEN TANK # 3 + FUEL	5210.00 + 288040.00
HYDROGEN TANK + FUEL	16338.00 + 144020.00
STME ENGINE # 1	7140.00
STME ENGINE # 2	7140.00
STME ENGINE # 3	7140.00
STME ENGINE # 4	7140.00
OMS ENGINE # 1	1050.00
OMS ENGINE # 2	1050.00
REACTION CONTROL SYSTEM	1404.00
WING GROUP	15950.00
BASIC STRUCTURE	37455.00
TAIL GROUP	7828.00
THRUST STRUCTURE	3874.00
BODY FLAP	939.00
INDUCED ENVIRONMENT	30230.00
FRONT LANDING GEAR	2245.00
RIGHT LANDING GEAR	2600.00
LEFT LANDING GEAR	2600.00
FUEL CELLS	276.00
REACTANT DEWARS	498.00

REACTANTS	655.00
BATTERIES	675.00
ELEC. CONV. AND DIST.	8476.00
SURFACE CONTROLS	3932.00
AVIONICS	4956.00
ENVIRONMENTAL CONTROLS	4625.00
PERSONNEL PROVISIONS	794.00
PERSONNEL	2873.00
RESIDUAL FLUIDS	15227.00
RCS PROPELLANT	2923.00
OMS PROPELLANT	28237.00

FIGURE 8 : CASE #1, MAXIMUM LOAD

CENTER OF GRAVITY LOCATION CHART WITH DESCRIPTION OF COMPONENT AND ITS MASS.

DESCRIPTION OF COMPONENT	X-COORD	Y-COORD	Z-COORD	MASS
CARGO	61.00	0.00	10.00	40000.00
OXYGEN TANK # 1 W/FUEL	59.00	-9.50	2.00	293250.00
OXYGEN TANK # 2 W/FUEL	59.00	0.00	3.00	293250.00
OXYGEN TANK # 3 W/FUEL	59.00	9.50	2.00	293250.00
HYDROGEN TANK W/FUEL	130.00	0.00	5.00	160358.00
STME ENGINE # 1	172.00	-8.00	- 2.00	7140.00
STME ENGINE # 2	172.00	8.00	- 2.00	7140.00
STME ENGINE # 3	174.00	8.00	4.00	7140.00
STME ENGINE # 4	174.00	-8.00	4.00	7140.00
OMS ENGINE # 1	173.00	13.00	4.00	1050.00
OMS ENGINE # 2	173.00	-13.00	4.00	1050.00
REACTION CONTROL SYSTEM	165.00	0.00	9.00	1404.00
WING GROUP	131.00	0.00	-12.00	15950.00
BASIC STRUCTURE	94.00	0.00	- 5.00	37455.00
TAIL GROUP	169.00	0.00	34.00	7828.00
THRUST STRUCTURE	165.50	0.00	6.00	3874.00
BODY FLAP	183.00	0.00	-10.00	939.00
INDUCED ENVIRONMENT	18.00	0.00	5.00	30230.00
FRONT LANDING GEAR	21.00	0.00	-5.00	2245.00
RIGHT LANDING GEAR	131.00	24.00	-10.00	2600.00
LEFT LANDING GEAR	131.00	-24.00	-10.00	2600.00
FUEL CELLS	22.00	8.00	-5.00	276.00
REACTANT DEWARS	21.00	-8.00	-5.00	498.00

REACTANTS	23.00	-3.00	-6.00	655.00
BATTERIES	24.00	4.00	-5.00	675.00
ELEC. CONV. AND DIST.	125.00	-1.00	-8.00	8476.00
SURFACE CONTROLS	172.00	3.00	-10.00	3932.00
AVIONICS	15.00	0.00	5.00	4956.00
ENVIRONMENTAL CONTROLS	18.00	-5.00	10.00	4625.00
PERSONNEL PROVISIONS	18.00	7.00	2.00	794.00
PERSONNEL	18.00	0.00	9.00	2873.00
RESIDUAL FLUIDS	18.00	-6.00	7.00	15227.00
RCS PROPELLANT	165.00	8.00	5.00	2923.00
OMS PROPELLANT	168.00	-6.00	5.00	28237.00

THE COORDINATES WITH RESPECT TO THE SHUTTLE NOSE, OF THE SHUTTLE OVERALL C.G. ARE AS FOLLOWS:

X C.G. = 75.39
Y C.G. = -0.20
Z C.G. = 2.74
TOTAL MASS = 1290040.00

FIGURE 9 : CASE #2, CARGO W/NO FUEL

****

CENTER OF GRAVITY LOCATION CHART WITH DESCRIPTION OF COMPONENT AND ITS MASS.

****

DESCRIPTION OF COMPONENT	X-COORD	Y-COORD	Z-COORD	MASS
CARGO	61.00	0.00	10.00	40000.00
OXYGEN TANK # 1	59.00	-9.50	2.00	5210.00
OXYGEN TANK # 2	59.00	0.00	3.00	5210.00
OXYGEN TANK # 3	59.00	9.50	2.00	5210.00
HYDROGEN TANK	130.00	0.00	5.00	16338.00
STME ENGINE # 1	172.00	-8.00	- 2.00	7140.00
STME ENGINE # 2	172.00	8.00	- 2.00	7140.00
STME ENGINE # 3	174.00	8.00	4.00	7140.00
STME ENGINE # 4	174.00	-8.00	4.00	7140.00
OMS ENGINE # 1	173.00	13.00	4.00	1050.00
OMS ENGINE # 2	173.00	-13.00	4.00	1050.00
REACTION CONTROL SYSTEM	165.00	0.00	9.00	1404.00
WING GROUP	131.00	0.00	-12.00	15950.00
BASIC STRUCTURE	94.00	0.00	-5.00	37455.00
TAIL GROUP	169.00	0.00	34.00	7828.00
THRUST STRUCTURE	165.50	0.00	6.00	3874.00
BODY FLAP	183.00	0.00	-10.00	939.00
INDUCED ENVIRONMENT	18.00	0.00	5.00	30230.00
FRONT LANDING GEAR	21.00	0.00	-5.00	2245.00
RIGHT LANDING GEAR	131.00	24.00	-10.00	2600.00
LEFT LANDING GEAR	131.00	-24.00	-10.00	2600.00
FUEL CELLS	22.00	8.00	-5.00	276.00
REACTANT DEWARS	21.00	-8.00	-5.00	498.00

REACTANTS	23.00	-3.00	-6.00	655.00
BATTERIES	24.00	4.00	-5.00	675.00
ELEC. CONV. AND DIST.	125.00	-1.00	-8.00	8476.00
SURFACE CONTROLS	172.00	3.00	-10.00	3932.00
AVIONICS	15.00	0.00	5.00	4956.00
ENVIRONMENTAL CONTROLS	18.00	-5.00	10.00	4625.00
PERSONNEL PROVISIONS	18.00	7.00	2.00	0.00
PERSONNEL	18.00	0.00	9.00	2873.00
RESIDUAL FLUIDS	18.00	-6.00	7.00	15227.00
RCS PROPELLANT	165.00	8.00	5.00	0.00
OMS PROPELLANT	168.00	-6.00	5.00	0.00

THE COORDINATES WITH RESPECT TO THE SHUTTLE NOSE, OF THE SHUTTLE OVERALL C.G. ARE AS FOLLOWS:

X C.G. = 89.25
Y C.G. = -0.45
Z C.G. = 2.57
TOTAL MASS = 249946.00

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FIGURE 10 : CASE #3, MINIMUM LOAD

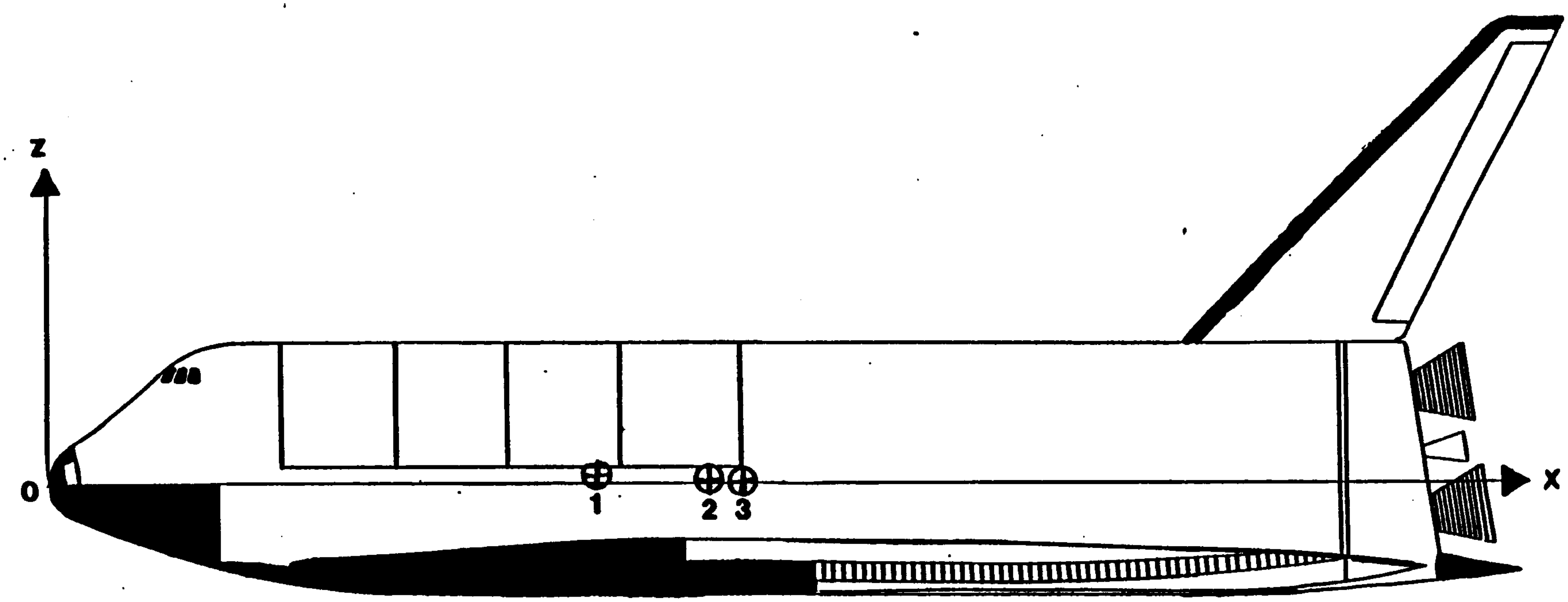
CENTER OF GRAVITY LOCATION CHART WITH DESCRIPTION OF COMPONENT AND ITS MASS.

DESCRIPTION OF COMPONENT	X-COORD	Y-COORD	Z-COORD	MASS
CARGO	61.00	0.00	10.00	0.00
OXYGEN TANK # 1 WO/FUEL	59.00	-9.50	2.00	5210.00
OXYGEN TANK # 2 WO/FUEL	59.00	0.00	3.00	5210.00
OXYGEN TANK # 3 WO/FUEL	59.00	9.50	2.00	5210.00
HYDROGEN TANK WO/FUEL	130.00	0.00	5.00	16338.00
STME ENGINE # 1	172.00	-8.00	-2.00	7140.00
STME ENGINE # 2	172.00	8.00	-2.00	7140.00
STME ENGINE # 3	174.00	8.00	4.00	7140.00
STME ENGINE # 4	174.00	-8.00	4.00	7140.00
OMS ENGINE # 1	173.00	13.00	4.00	1050.00
OMS ENGINE # 2	173.00	-13.00	4.00	1050.00
REACTION CONTROL SYSTEM	165.00	0.00	9.00	1404.00
WING GROUP	131.00	0.00	-12.00	15950.00
BASIC STRUCTURE	94.00	0.00	-5.00	37455.00
TAIL GROUP	169.00	0.00	34.00	7828.00
THRUST STRUCTURE	165.50	0.00	6.00	3874.00
BODY FLAP	183.00	0.00	-10.00	939.00
INDUCED ENVIRONMENT	18.00	0.00	5.00	30230.00
FRONT LANDING GEAR	21.00	0.00	-5.00	2245.00
RIGHT LANDING GEAR	131.00	24.00	-10.00	2600.00
LEFT LANDING GEAR	131.00	-24.00	-10.00	2600.00
FUEL CELLS	22.00	8.00	-5.00	276.00
REACTANT DEWARS	21.00	-8.00	-5.00	498.00

REACTANTS	23.00	-3.00	-6.00	655.00
BATTERIES	24.00	4.00	-5.00	675.00
ELEC. CONV. AND DIST.	125.00	-1.00	-8.00	8476.00
SURFACE CONTROLS	172.00	3.00	-10.00	3932.00
AVIONICS	15.00	0.00	5.00	4956.00
ENVIRONMENTAL CONTROLS	18.00	-5.00	10.00	4625.00
PERSONNEL PROVISIONS	18.00	7.00	2.00	0.00
PERSONNEL	18.00	0.00	9.00	2873.00
RESIDUAL FLUIDS	18.00	-6.00	7.00	15227.00
RCS PROPELLANT	165.00	8.00	5.00	0.00
OMS PROPELLANT	168.00	-6.00	5.00	0.00

THE COORDINATES WITH RESPECT TO THE SHUTTLE NOSE,
OF THE SHUTTLE OVERALL C.G. ARE AS FOLLOWS:

X C.G. = 94.63
Y C.G. = -0.53
Z C.G. = 1.15
TOTAL MASS = 209946.00



CASE ##	CONDITION: CARGO	CONDITION: FUEL	CONDITION: PROVISIONS	CG LOCATION (FT): X	CG LOCATION (FT): Y	CG LOCATION (FT): Z	TOTAL MASS (lbs)
1	YES	YES	YES	75.39	-0.20	2.74	1290040.00
2	YES	NO	NO	89.25	-0.45	2.57	249946.00
3	NO	NO	NO	94.63	-0.53	1.15	209946.00

FIGURE 11 : CENTER OF GRAVITIES UNDER SPECIFIC LOADING CONDITIONS IN THE XYZ- COORDINATE SYSTEM

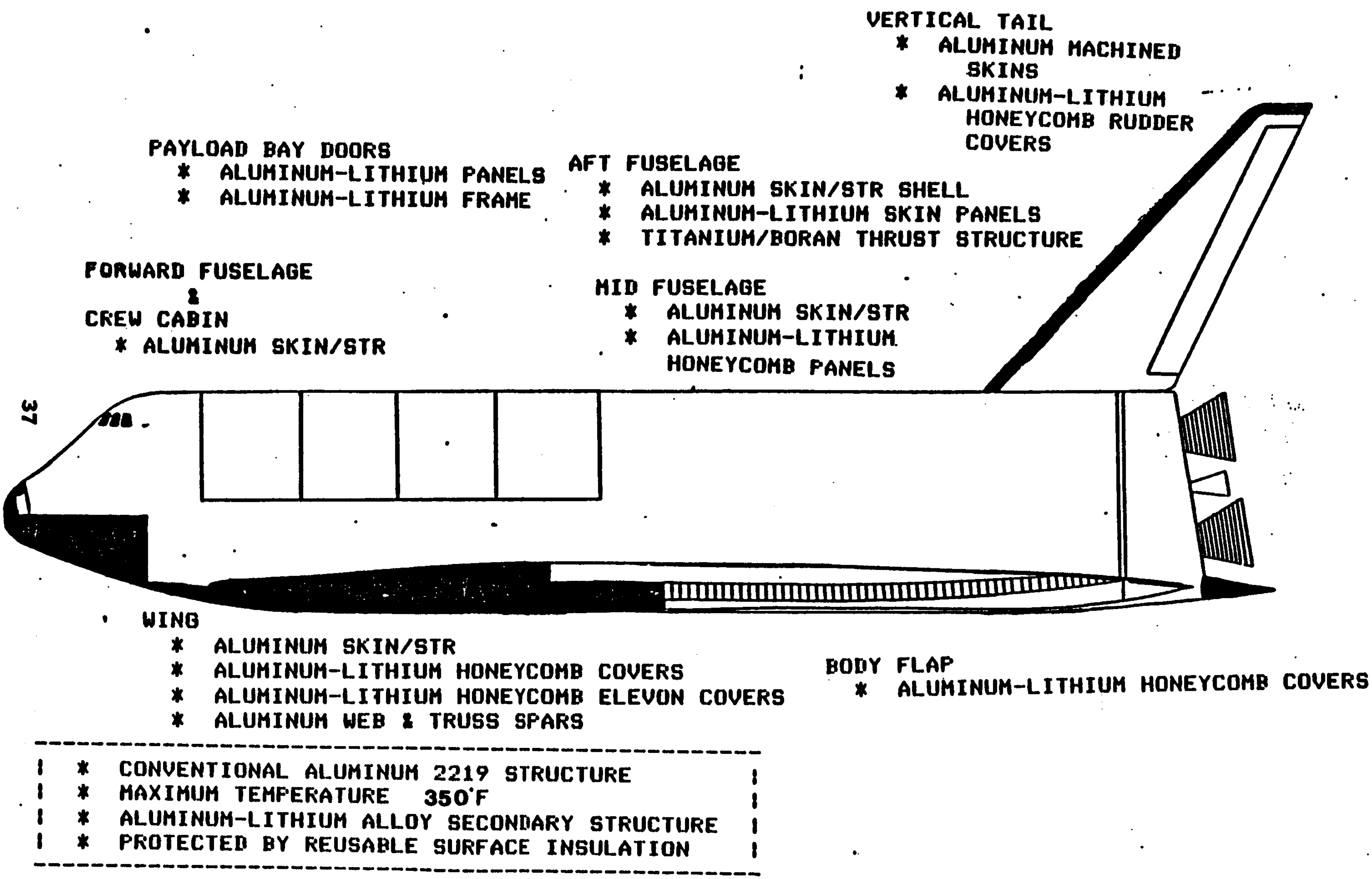


Figure 12 STRUCTURAL MATERIAL COMPOSITION

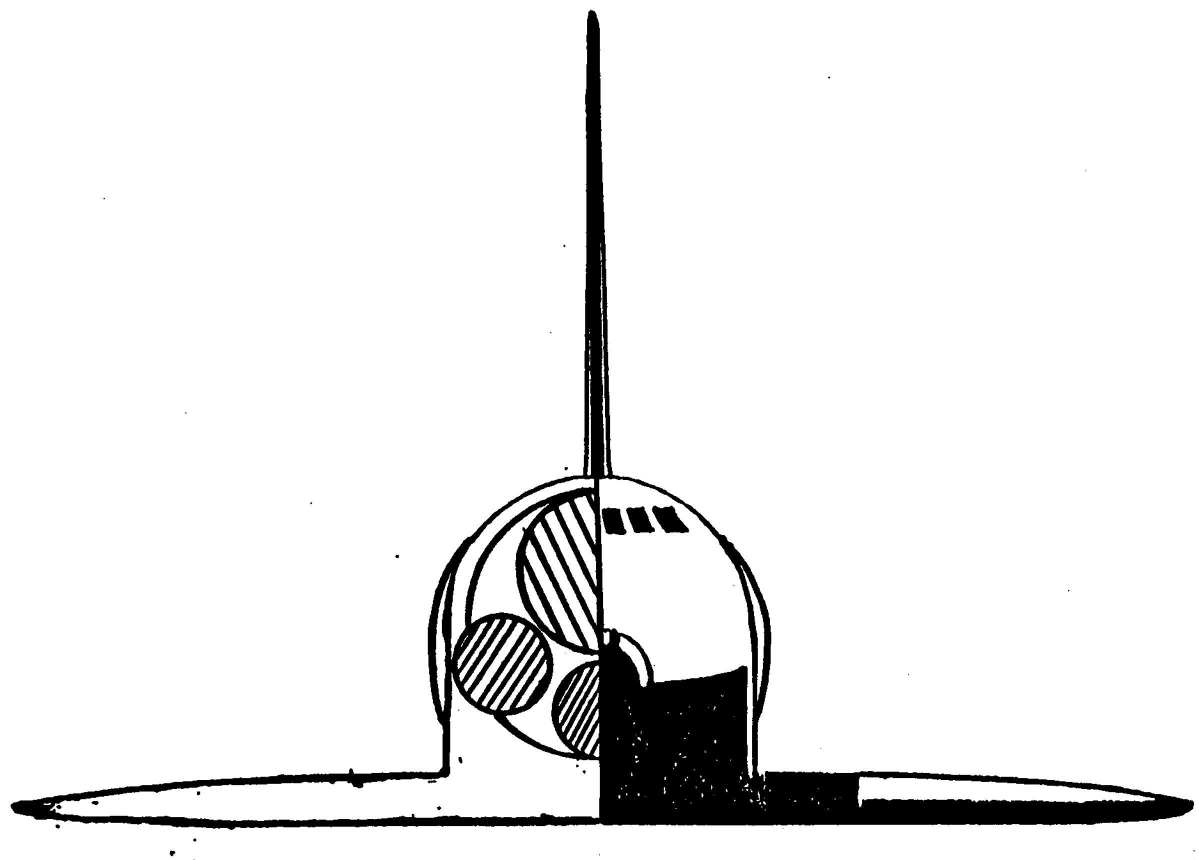


Figure 13 FRONTAL CUT-AWAY VIEW SHOWING PROPELLANT TANKS AND CARGO

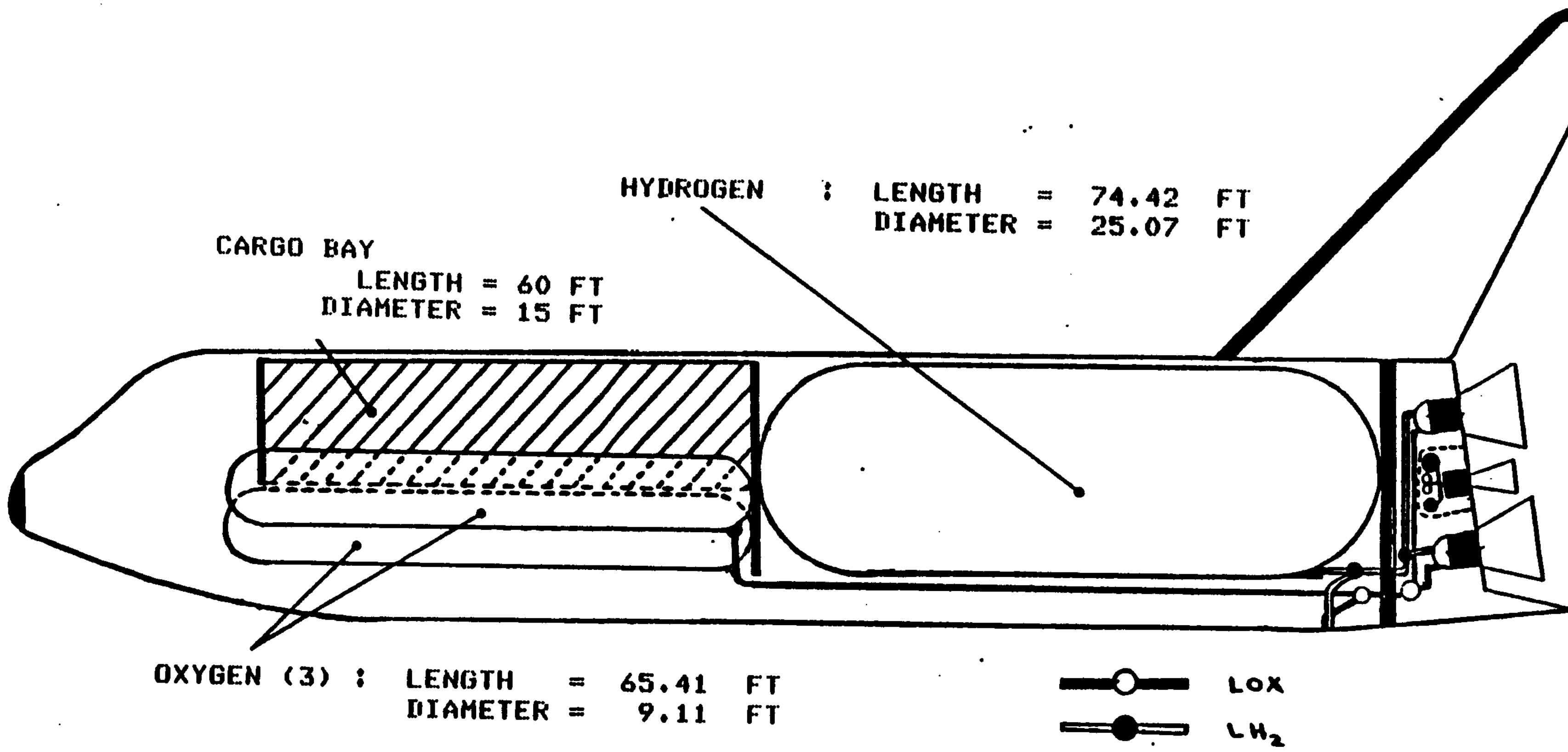


Figure 14 CUT-AWAY VIEW SHOWING PROPELLANT TANKS AND PLUMBING

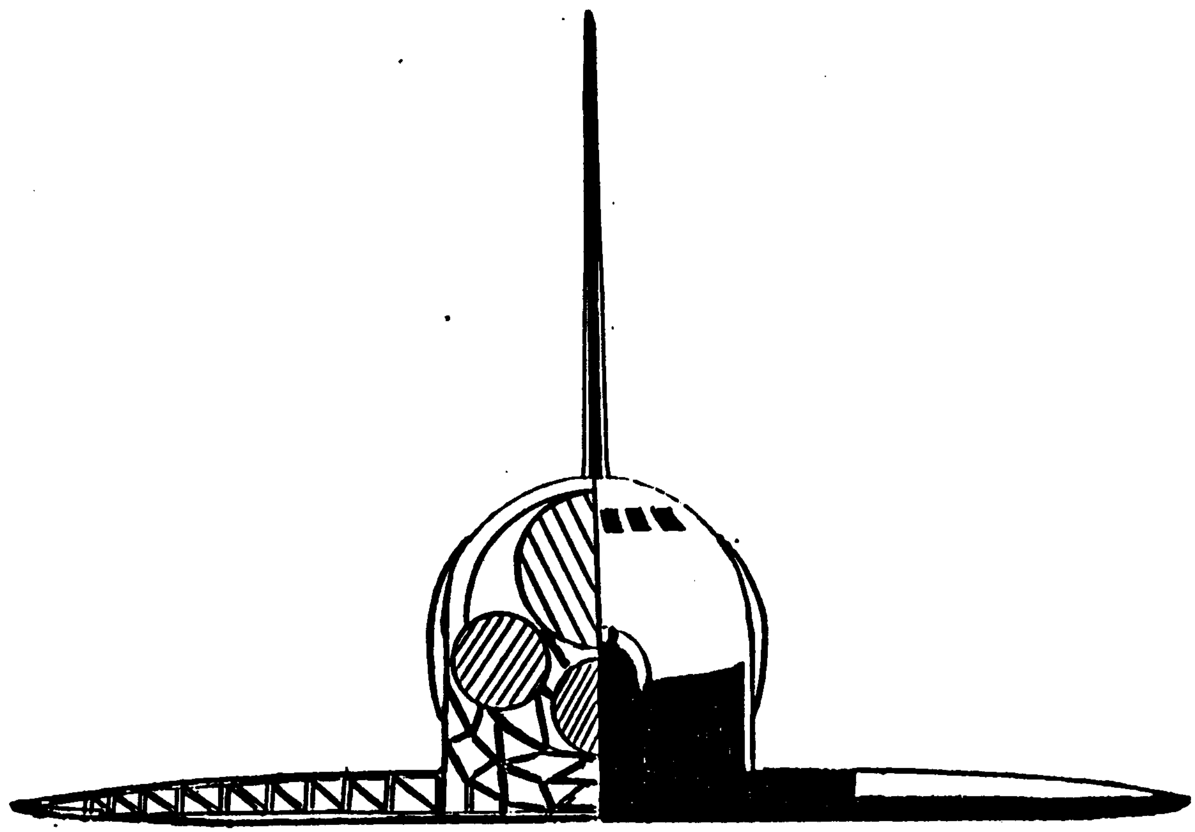


FIGURE 15 FRONT CUT-AWAY VIEW SHOWING STRUCTURES

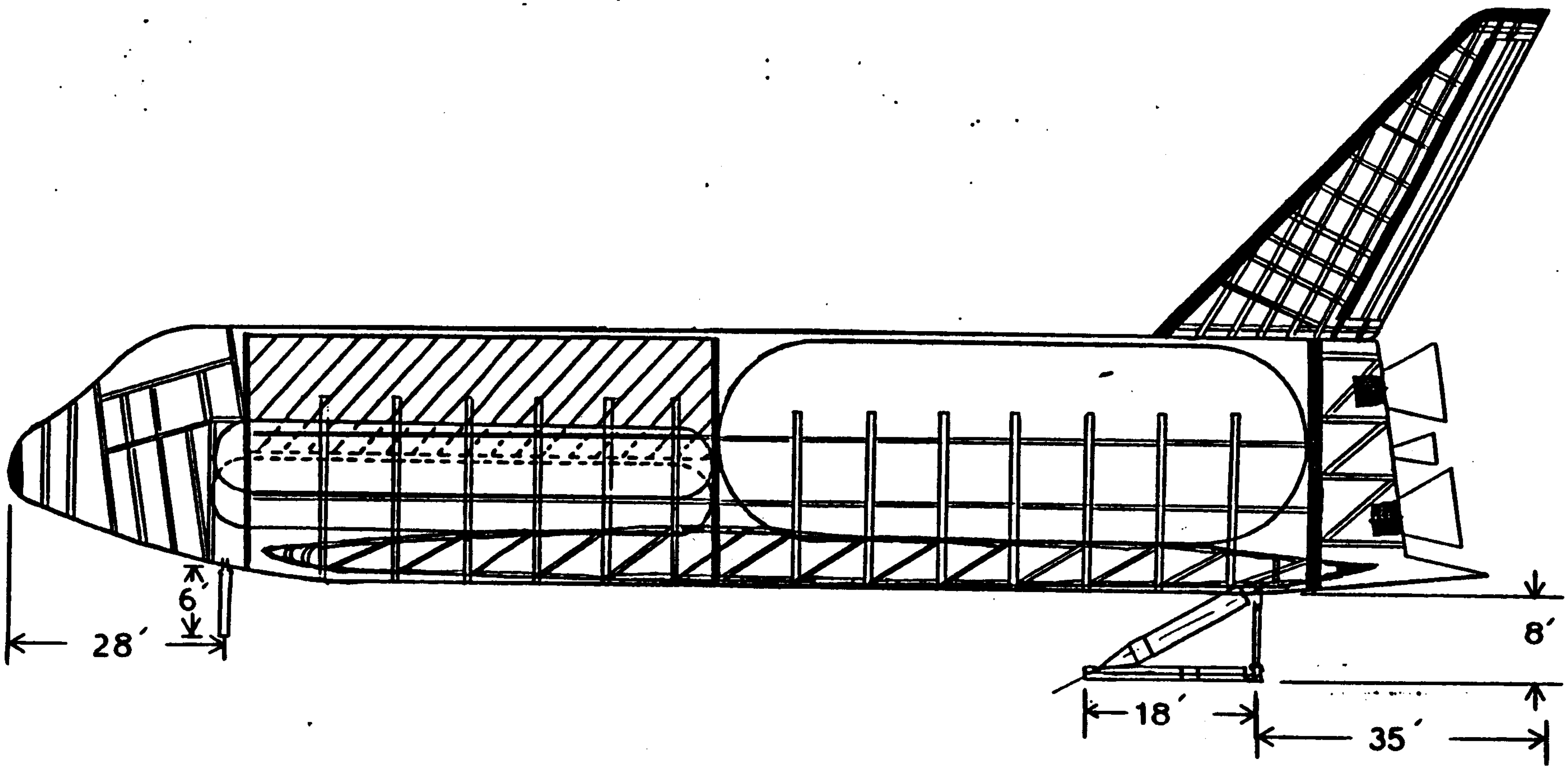


Figure 16 SIDE CUTAWAY VIEW OF STRUCTURES

FIGURE 17 :

LAUNCH VEHICLE COST MODEL
STAGE ONE
COST ESTIMATE

DATE: 23-Mar-87
INFLATION : 1.229 FROM 82$ TO 86$
WT RESERVE: 1.23

COST ELEMENT	CER'S INDEP. VAR. DOT&E	TFU	COMPLEXITY FACTOR DOT&E	TFU	COST (86 $ MILLIONS) DOT&E	TFU
STRUCTURES/TPS	73297.00	73297.00	1.00	1.00	650.84	196.43
THERMAL CONTROL						
ENVIRONMENTAL CONTROL	4625.00	4625.00	1.00	1.00	20.33	2.10
BASE HEAT SHIELD	939.00	939.00	1.00	1.00	9.95	1.50
WING/TAIL/LEADING EDGE	23743.00	23743.00	1.00	1.00	368.16	66.95
LANDING GEAR	7445.00	7445.00	1.00	1.00	56.32	14.11
AVIONICS	4956.00	4956.00	1.00	1.00	914.76	84.84
ELECTRICAL POWER	9151.00	9151.00	1.00	1.00	408.16	99.47
PROPULSION (LESS ENGINES)	276.00	276.00	1.00	1.00	37.60	1.88
SEPARATION PROVISIONS	1153.00	1153.00	1.00	1.00	29.97	9.86
SURFACE FLIGHT CONTROLS	3932.00	3932.00	1.00	1.00	251.86	78.02
AUXILARY POWER UNIT	0.00	0.00	1.00	1.00	0.00	0.00
HYDRAULICS	1404.00	1404.00	1.00	1.00	56.67	10.72
SUBTOTAL	124156.00	124156.00			2428.53	465.39
STRUC. TOOLING	196.43		1.00		426.04	
SYS. TST. HRDW. & ASSEMBLY	465.39		1.00		603.48	
SYS. TST. OPS.	603.48		1.00		222.57	
SUBTOTAL					3680.62	465.39
GSE	3680.62		1.00		565.05	
SUBTOTAL					4245.66	465.39
SE&I	4245.66	465.39	1.00	1.00	410.14	37.31
SUBTOTAL					4655.81	502.70
PROGRAM MANAGEMENT	4655.81	502.70	1.00	1.00	157.75	17.84
SUBTOTAL					4813.56	520.55
ENGINES (CV=W*Tv*ISP*Pc)	391254695.50	1565018782.00	1.00	1.00	42.94	0.04 CE
SUBTOTAL					4856.50	520.59
FEE (14%)					679.91	72.88
PROG. SUPPORT (3% DEV., 2% PROD.)					166.09	11.87
SUBTOTAL					5702.50	605.34
COST CONTINGENCY (15%)					855.37	90.80
TOTAL					6557.87	696.14

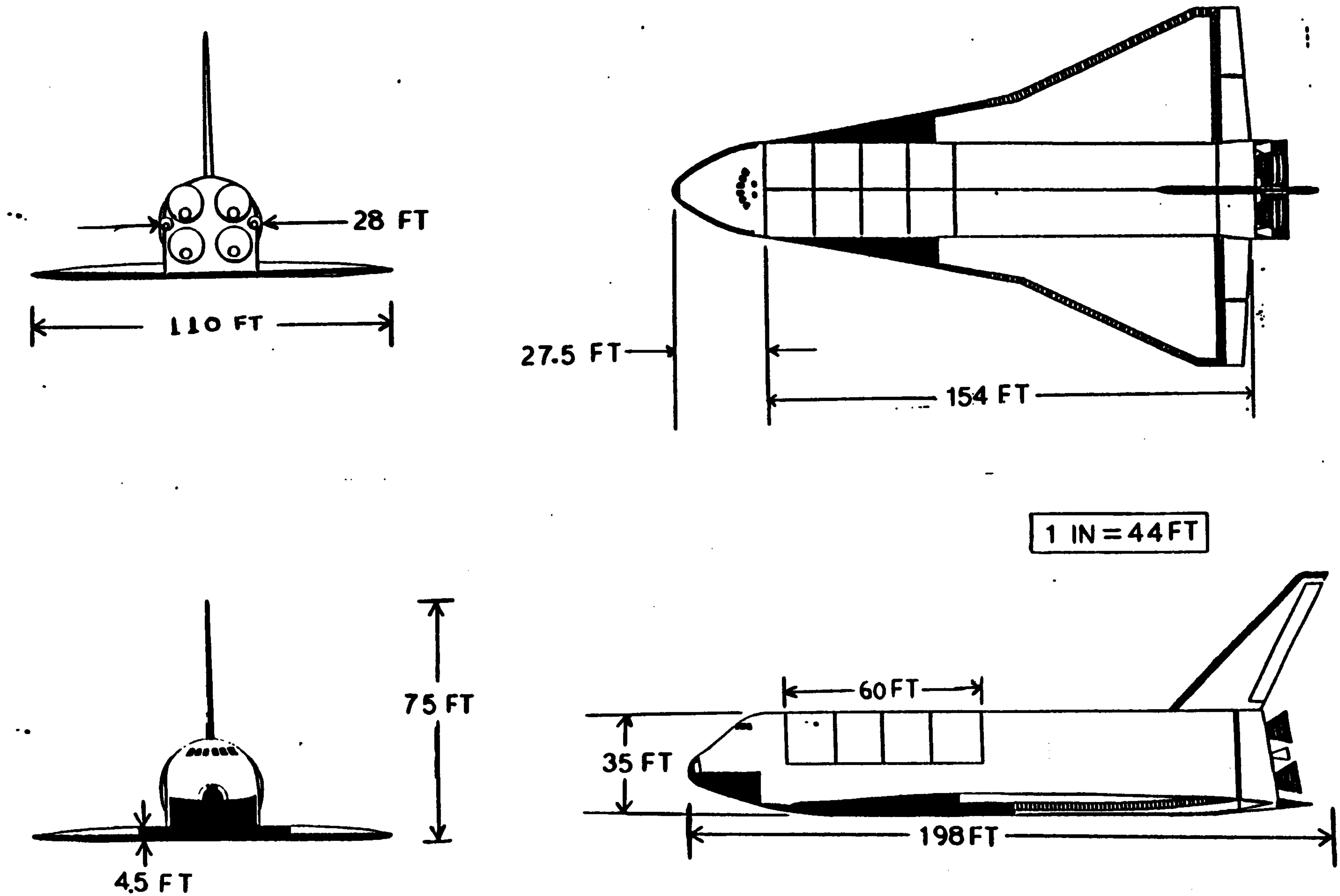


Figure 18   3-D VIEW OF SHUTTLE II

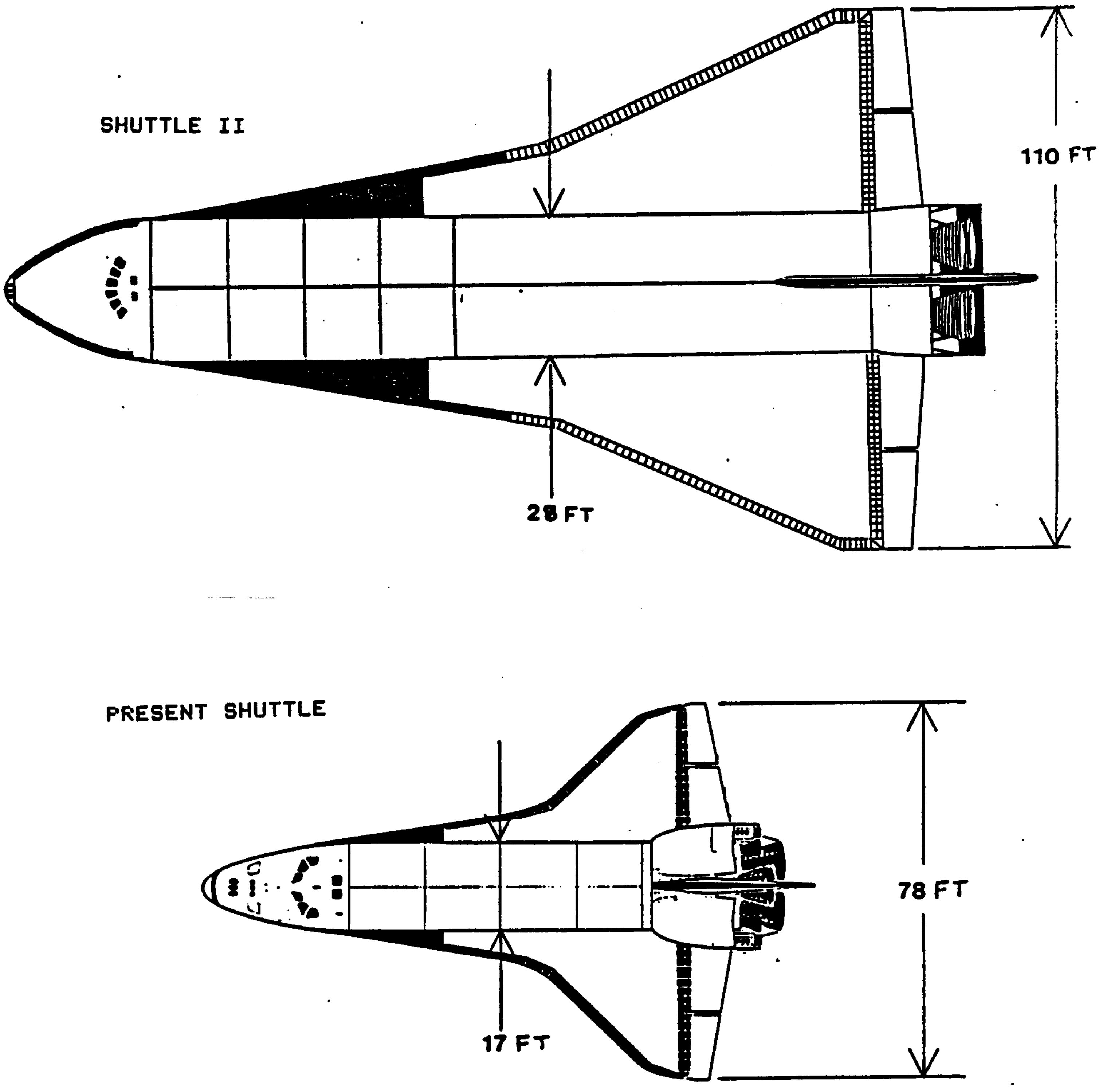


FIGURE 19: DIMENSIONAL COMPARISON BETWEEN EXISTING SHUTTLE AND SHUTTLE II

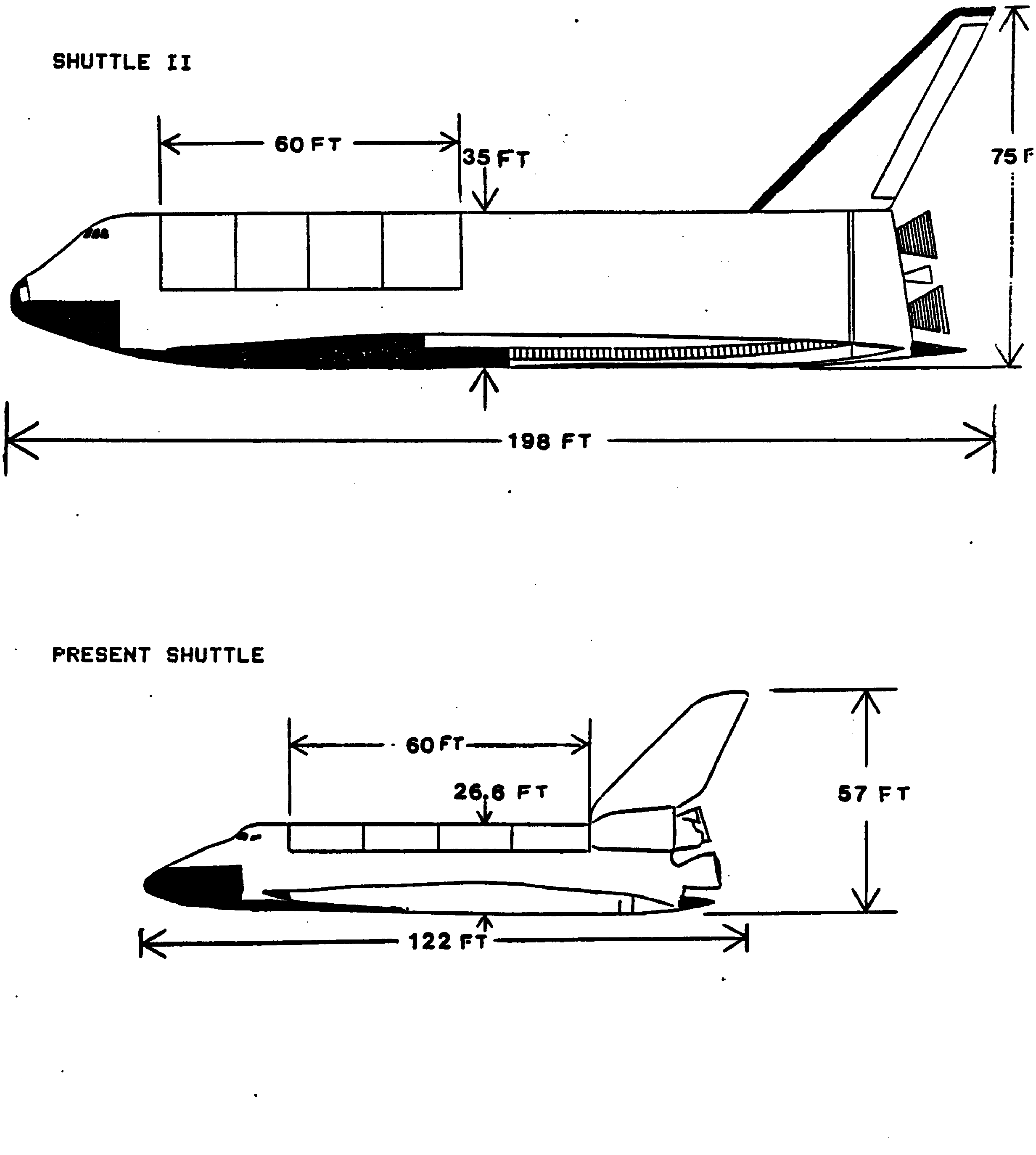


FIGURE 20: DIMENSIONAL COMPARISON BETWEEN EXISTING SHUTTLE AND SHUTTLE II

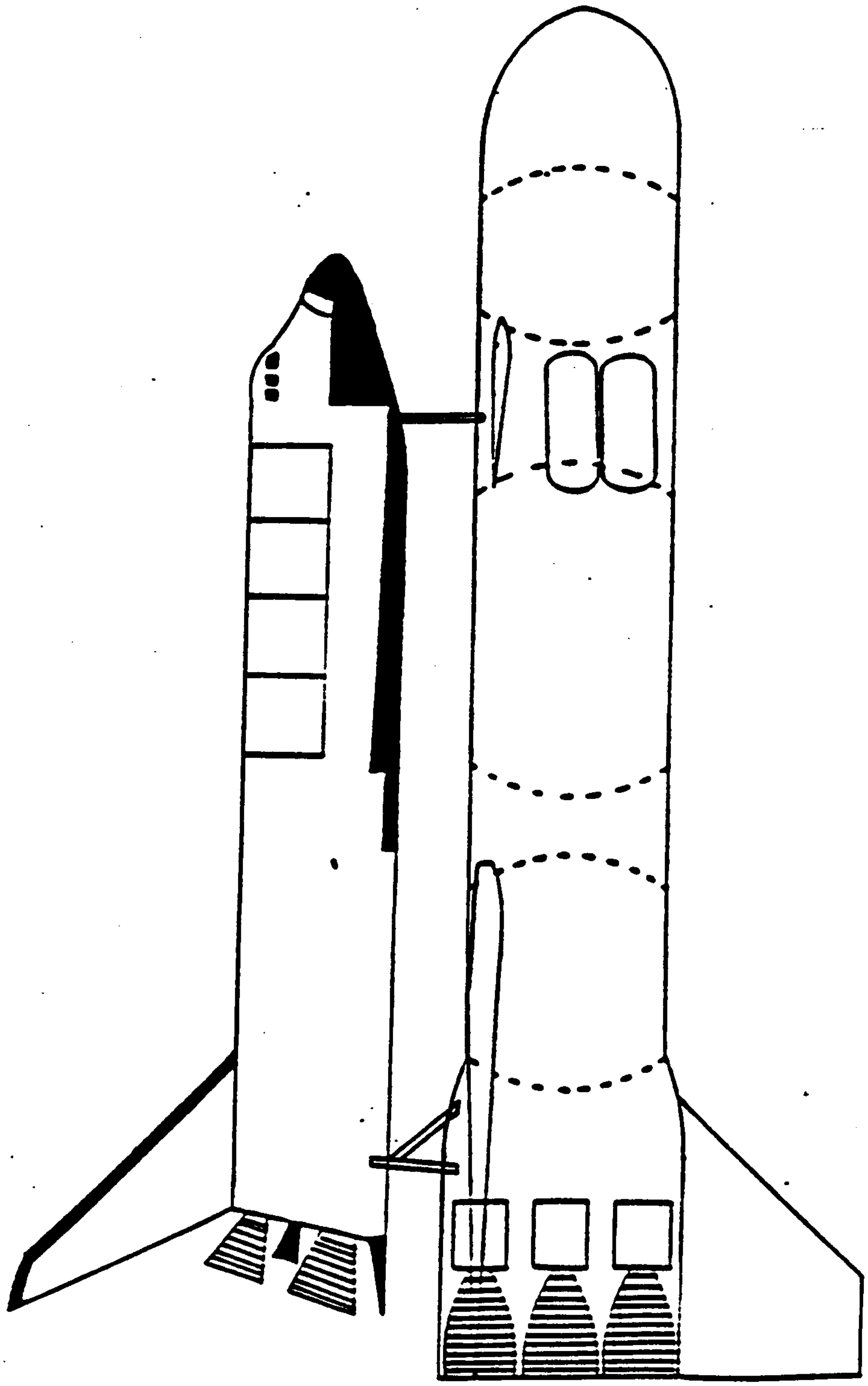


FIGURE 21 : FLYBACK BOOSTER AND SHUTTLE II MATING CONFIGURATION

APPENDIX A
PROGRAM 1 WITH FLOWCHART

```
5     REM                    PROGRAM 1
10    REM
15    ALT=0
20    DT=0.1
30    RANG=0
40    ISP1=320
50    ISP2=380
60    INPUT"ENTER INITIAL MASS";M1
65    M=M1
70    INPUT"ENTER ALTITUDE FOR G-TURN";AGT1
75    AGT=AGT1
80    TH=1.35*M
90    INPUT"ENTER LIFTOFF THRUST (SHUTTLE II ENGINES)";TH21
95    TH2=TH21
100   TH1=TH-TH2
110   MDOT1=TH1/ISP1
120   MDOT2=TH2/ISP2
130   ISP=TH/(MDOT1+MDOT2)
140   MDOT=TH/ISP
150   AMC1=0
160   MC2=0
170   Y=0
180   X=0
190   G0=32.2
200   R0=2.092E+07
210   V0=0
220   V=V0
230   T=0
240   PRINT
250   PRINT"TIME      V         RANG        ALT        G'S      MASS       THRUST"
260   PRINT
270   F$="###.#     #####.#   ######     ######     #.##     #######     #######"
280   M2=M-MDOT1*DT-MDOT2*DT
290   G=G0*(R0/(R0+ALT))^2
300   DV=ISP*G0*LOG(M/M2)-G*DT
310   AG1=DV/G0/DT
320   IF AG1>3 THEN 400
330   CHY=0.5*(V0+V0+DV)*DT
340   IF Y+CHY>AGT THEN 580
350   Y=Y+CHY
360   ALT=Y
370   V=V0+DV
380   AG=AG1
390   GOTO 460
400   DV=3*G0
410   M2=M/EXP((DV+G*DT)/ISP/G0)
420   A=DV/G0
430   V=V0+DV
440   Y=Y+0.5*(V0+V)*DT
450   ALT=Y
460   T=T+DT
470   DM=M-M2
480   MDT=DM/DT
490   TH=MDT*ISP
500   MC1=MC1+TH1/ISP*DT
510   MC2=MC2+(TH-TH1)/ISP2*DT
520   MC=MC1+MC2
```

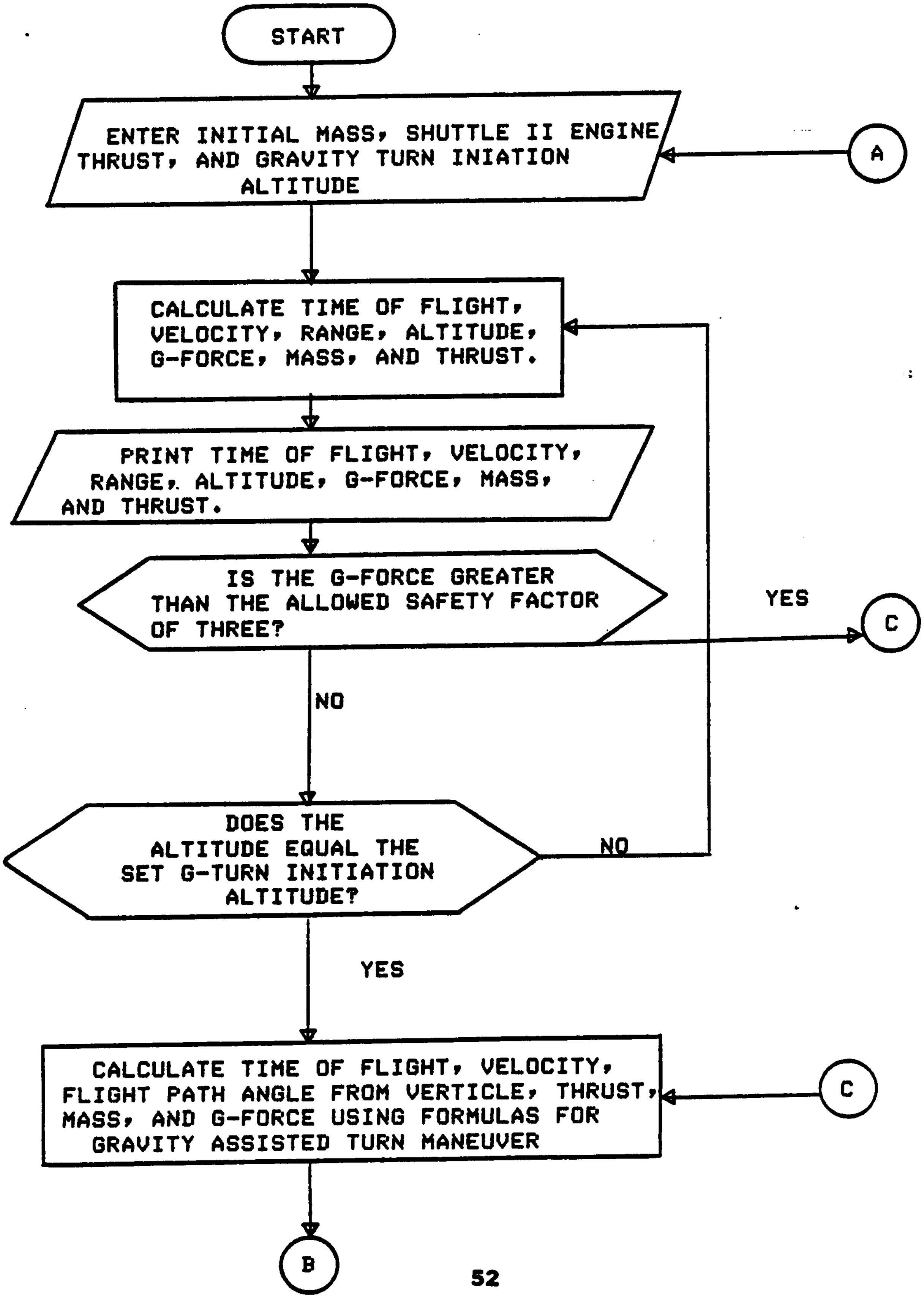
```
530     PRINT USING F$;T,V,RANG,ALT,AG,M,TH
540     REM
550     M=M2
560     VO=V
570     GOTO 280
580     PRINT
590     PRINT"TIME      V       RANG      ALT       G'S       MASS      THRUST"
600     PRINT"-------------------------------------------------------------------"
620     M=M+MDOT1*DT+MDOT2*DT
630     DPSI=(PI/180)*0.1
640     PSIO=PI/120
650     PSI=PSIO
660     GAM=1
670     ZO=SIN(PSIO)/(1+COS(PSIO))
680     PRINT"TIME      V      RANG       ALT       PSI    GAM    THRUST    MASS    G'S"
700     F$="###  #####   #######  #######    ##.#    ##.#  #######  #######   #.##"
710     PRINT
720     N=TH/M
730     PRINT USING F$;T,V,RANG,ALT,PSI*180/PI,GAM,TH,M,AG
740     C=VO/(ZO^(N-1))/(1+ZO^2)
750     PSI=PSI+DPSI
760     Z=SIN(PSI)/(1+COS(PSI))
770     V1=C*(Z^(N-1))*(1+Z^2)
780     IF V1>7000 THEN 1390
785     V=V1
790     DT= C/G*Z^(N-1)*(1/(N-1)+Z^2/(N+1))-C/G*ZO^(N-1)*(1/(N-1)+ZO^2/(N+1))
800     DX=0.5*(VO*SIN(PSIO)+V*SIN(PSI))*DT
810     DY=0.5*(VO*COS(PSIO)+V*COS(PSI))*DT
820     AG=(V-VO)/DT/GO
830     IF AG>3 THEN 1060
840     X=X+DX
845     Y=Y+DY
850     THETA=ATN(X/(RO+Y))
860     GAMMA=PSI-THETA
870     GAM=GAMMA*180/PI
880     ALT=(Y+RO)/COS(THETA)-RO
900     RANG=RO*THETA
930     T=T+DT
940     PSIO=PSI
950     ZO=Z
960     VO=V
970     IF ALT<100000 THEN 1020
980     ISP1=340
990     ISP2=462
1000    MDOT1=TH1/ISP1
1010    MDOT2=TH2/ISP2
1020    M=M-MDOT1*DT-MDOT2*DT
1030    IF M<0 THEN 1390
1040    G=GO*(RO/(RO+ALT))^2
1050    GOTO 720
1060    N=3*GO/G+((1-ZO^2)/(1+ZO^2))
1070    C=VO/ZO^(N-1)/(1+ZO^2)
1080    V=C*Z^(N-1)*(1+Z^2)
1090    D1=C/G*Z^(N-1)*(1/(N-1)+Z^2/(N+1))-C/G*ZO^(N-1)*(1/(N-1)+ZO^2/(N+1))
1100    DX=0.5*(VO*SIN(PSIO)+V*SIN(PSI))*DT
1110    DY=0.5*(VO*COS(PSIO)+V*COS(PSI))*DT
1120    X=X+DX
1130    RANG=RO*THETA
```

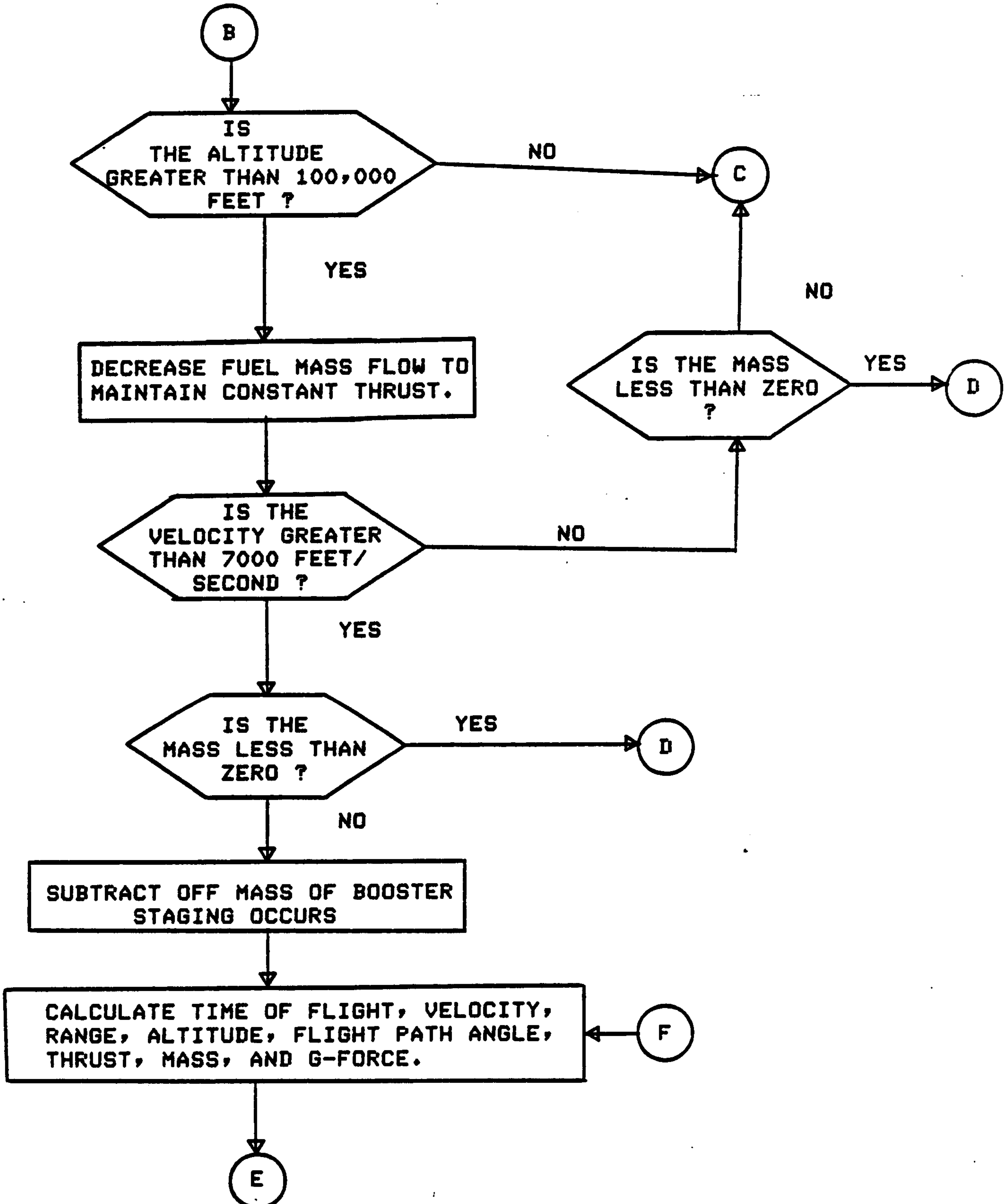
```
1140  Y=Y+DY
1150  ALT=(Y+RO)/COS(THETA)-RO
1160  T=T+DT
1170  PSIO=PSI
1180  ZO=Z
1190  VO=V
1200  TH=N*M
1210  TH1=TH-TH2
1220  MDOT1=TH1/ISP1
1230  MDOT2=TH2/ISP2
1240  M=M-MDOT1*DT-MDOT2*DT
1250  MC1=MC1+MDOT1*DT
1260  MC2=MC2+MDOT2*DT
1270  MC=MC1+MC2
1280  IF M<0 THEN 1390
1290  G=GO*(RO/(RO+ALT))^2
1300  PSI=PSI+DPSI
1310  THETA=ATN(X/(RO+Y))
1320  GAMMA=PSI-THETA
1330  GAM=GAMMA*180/PI
1350  PRINT USING F$;T,V,RANG,ALT,PSI,GAM,TH,M,AG
1360  IF V>7000 THEN 1390
1370  Z=SIN(PSI)/(1+COS(PSI))
1380  GOTO 1060
1390  PRINT
1391  PSI=PSI-DPSI
1392  PRINT V1
1400  PRINT"TIME     V     RANG    ALT     PSI    GAM    THRUST    MASS     G'S"
1420  PRINT"------------------------------------------------------------------------"
1421  PRINT
1422  PRINT"                        S E P A R A T I O N              "
1423  PRINT
1424  PRINT"------------------------------------------------------------------------"
1430  PRINT"time     v     rang    alt     psi    gam2   thrust     mass      g's"
1431  PRINT
1432  F$="###  #####  ####### ####### ###.# ###.#  #######  #######  #.##"
1435  M=(1-.15*M1/M)*M
1440  ISP=ISP2
1450  TH=TH2
1460  MDOT=TH/ISP
1470  VO=V
1480  PSIO=PSI
1490  GAM2=PI/2 + THETA
1500  GAMM=GAM2*180./PI
1510  ZO=SIN(PSIO)/(1+COS(PSIO))
1520  N=TH/M
1530  PRINT USING F$;T,V,RANG,ALT,PSI*180./PI,GAMM,TH,M,AG
1540  C=VO/(ZO^(N-1))/(1+ZO^2)
1550  PSI=PSI+DPSI
1560  THETA=ATN(X/(RO+Y))
1570  GAM2=PI/2+THETA
1580  GAMM=GAM2*180./PI
1590  IF PSI>GAM2 THEN 2080
1600  Z=SIN(PSI)/(1+COS(PSI))
1610  V=C*(Z^(N-1))*(1+Z^2)
1620  IF V>25000 THEN 2080
1630  DT=C/G*Z^(N-1)*(1/(N-1)+Z^2/(N+1))-C/G*ZO^(N-1)*(1/(N-1)+ZO^2/(N+1))
1640  DX=.5*(VO*SIN(PSIO)+V*SIN(PSI))*DT
```

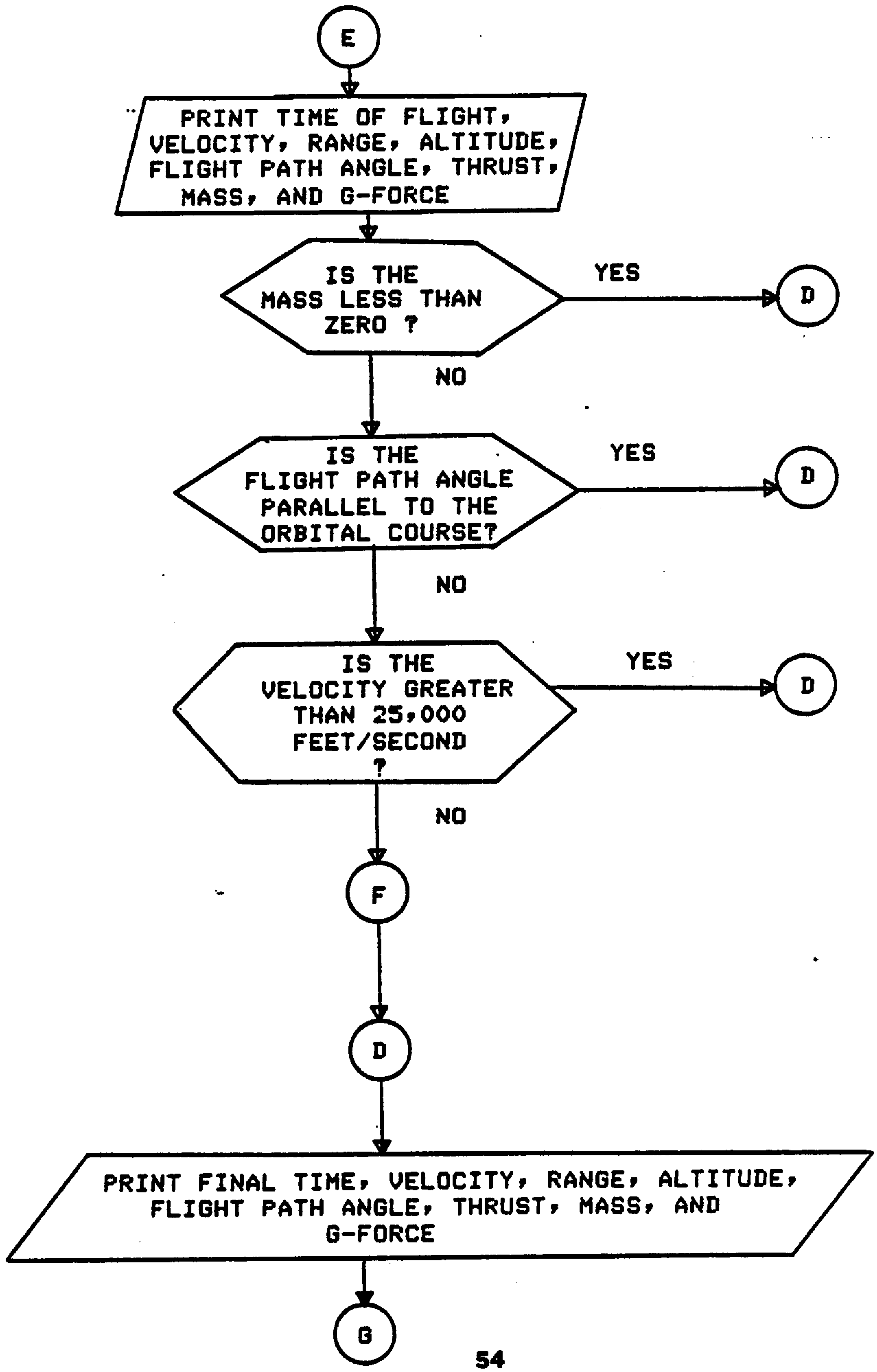
```
1650  DY=.5*(V0*COS(PSI0)+V*COS(PSI))*DT
1660  AG=(V-V0)/DT/G0
1680  X=X+DX
1700  Y=Y+DY
1701  THETA=ATN(X/(R0+Y))
1702  RANG=R0*THETA
1710  ALT=(Y+R0)/COS(THETA)-R0
1720  T=T+DT
1730  PSI0=PSI
1740  Z0=Z
1750  V0=V
1760  M=M-MDOT*DT
1770  IF M<0 THEN 2080
1780  G=G0*(R0/(R0+ALT))^2
1790  GOTO 1520
1800  N=3*G0/G+((1-Z0^2)/(1+Z0^2))
1810  C=V0/Z0^(N-1)/(1+Z0^2)
1820  V=C*Z^(N-1)*(1+Z^2)
1830  DT=C/G*Z^(N-1)*(1/(N-1)+Z^2/(N+1))-C/G*Z0^(N-1)*(1/(N-1)+Z0^2/(N+1))
1840  DX=.5*(V0*SIN(PSI0)+V*SIN(PSI))*DT
1850  DY=.5*(V0*COS(PSI0)+V*COS(PSI))*DT
1860  X=X+DX
1880  Y=Y+DY
1881  THETA=ATN(X/(R0+Y))
1882  RANG=R0*THETA
1890  ALT=(Y+R0)/COS(THETA)-R0
1900  T=T+DT
1910  PSI0=PSI
1920  Z0=Z
1930  V0=V
1940  TH=N*M
1950  MDOT=TH/ISP
1960  M=M-MDOT*DT
1970  IF M<0 THEN 2080
1980  G=G0*(R0/(R0+ALT))^2
1990  AG=3
2000  PSI=PSI+DPSI
2010  THETA=ATN(X/(R0+Y))
2020  GAM2=PI/2+THETA
2025  GAMM=GAM2*180./PI
2040  PRINT USING F$;T,V,RANG,ALT,PSI*180./PI,GAMM,TH,M,AG
2045  IF PSI>GAM2 THEN 2080
2050  IF V>24000 THEN 2080
2060  Z=SIN(PSI)/(1+COS(PSI))
2070  GOTO 1800
2080  REM
2081  PRINT
2090  PRINT"INITIAL MASS      :";M1
2100  PRINT"ALT FOR G-TURN    :";AGT1
2110  PRINT"THRUST ( SHUTTLE II ) :";TH21
2111  PRINT"FINAL VELOCITY :";V
2112  PRINT"FINAL MASS :";M
2113  PRINT"FINAL PSI  :";PSI*180/PI
2114  PRINT"FINAL ALTITUDE :";ALT/6080
2115  PRINT"FINAL GAMM :";GAMM
2116  INPUT"TRY AGAIN Y=1 N=2";TRY
2117  IF TRY<2 THEN 15
```

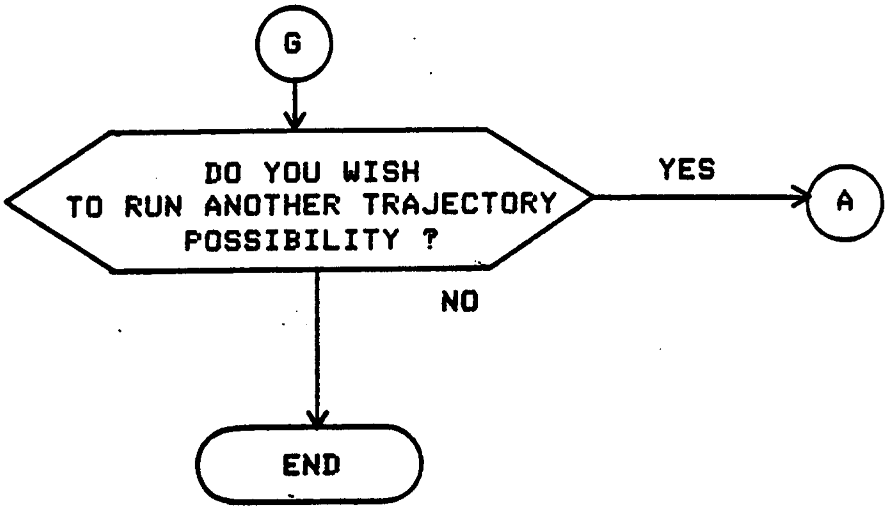
FLOWCHART FOR TRAJECTORY ANALYSIS

PROGRAM "S2TRAJ"




B
IS THE ALTITUDE GREATER THAN 100,000 FEET ?
NO
C
YES
NO
DECREASE FUEL MASS FLOW TO MAINTAIN CONSTANT THRUST.
IS THE MASS LESS THAN ZERO ?
YES
D
IS THE VELOCITY GREATER THAN 7000 FEET/ SECOND ?
NO
YES
IS THE MASS LESS THAN ZERO ?
YES
D
NO
SUBTRACT OFF MASS OF BOOSTER STAGING OCCURS
CALCULATE TIME OF FLIGHT, VELOCITY, RANGE, ALTITUDE, FLIGHT PATH ANGLE, THRUST, MASS, AND G-FORCE.
F
E

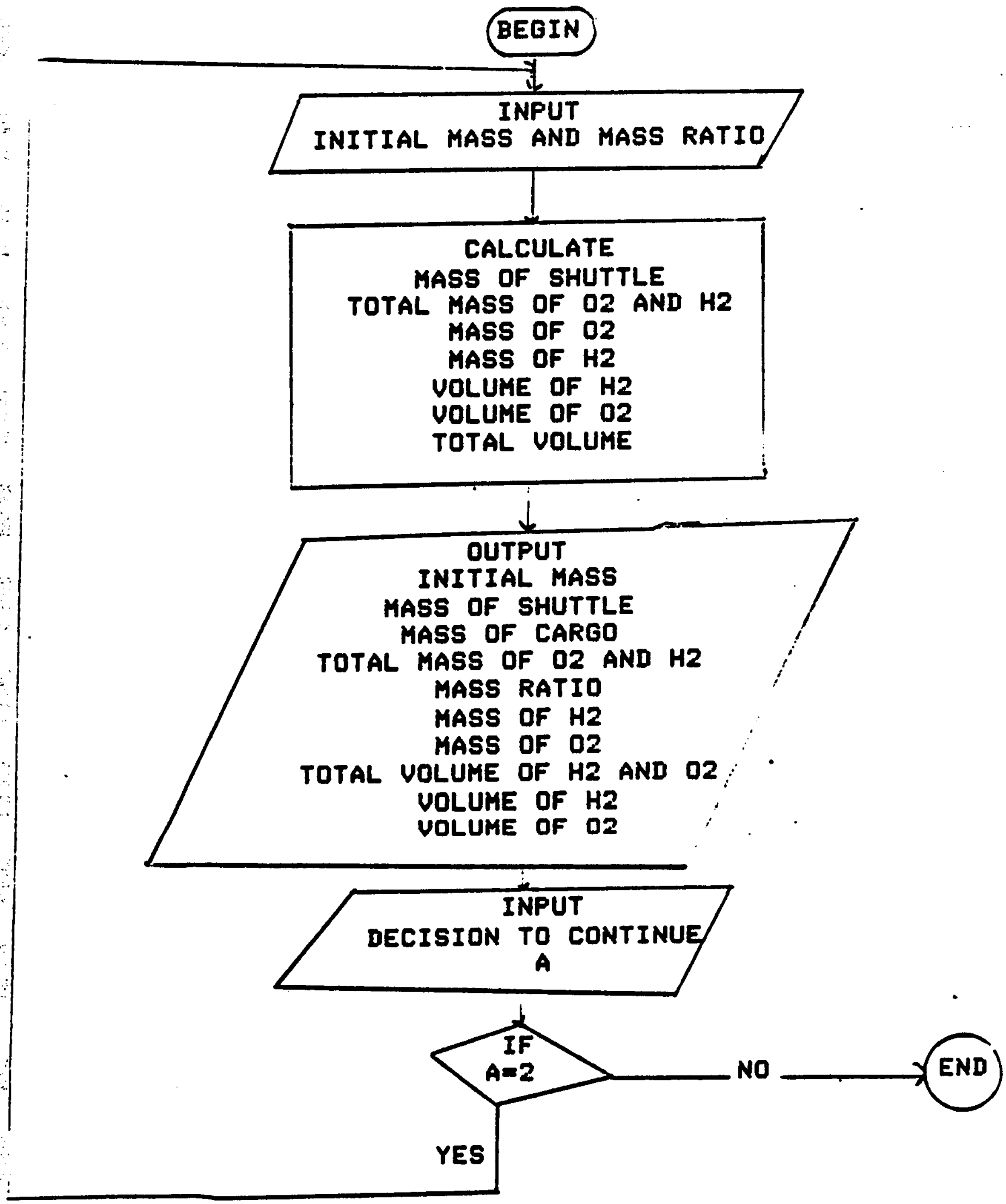

E
PRINT TIME OF FLIGHT, VELOCITY, RANGE, ALTITUDE, FLIGHT PATH ANGLE, THRUST, MASS, AND G-FORCE
IS THE MASS LESS THAN ZERO ?
YES
D
NO
IS THE FLIGHT PATH ANGLE PARALLEL TO THE ORBITAL COURSE?
YES
D
NO
IS THE VELOCITY GREATER THAN 25,000 FEET/SECOND ?
YES
D
NO
F
D
PRINT FINAL TIME, VELOCITY, RANGE, ALTITUDE, FLIGHT PATH ANGLE, THRUST, MASS, AND G-FORCE
G


G
DO YOU WISH
TO RUN ANOTHER TRAJECTORY
POSSIBILITY ?
YES
A
NO
END

APPENDIX B
PROGRAM 2 WITH FLOWCHART

```
C                              PROGRAM 2
C
C      -------------------------------------------------------
C      THIS PROGRAM CALCULATES PERTINENT MASSES AND VOLUMES
C      OF CERTAIN COMPONENTS OF THE SPACE SHUTTLE II PROGRAM.
C      -------------------------------------------------------
C
       REAL MR,MI,MSH,MC,MO2H2,MH2,MO2
       MC=40000.
C
       200  PRINT *,'INPUT INITIAL MASS AND MASS RATIO.'
       READ *,MI,MR
C
       MSH=(MI/MR)-MC
       MO2H2=(MC+MSH)*(MR-1.)
C
C      THE MASS RATIO OF O/H IS 6.0
C
       MH2=MO2H2/7.0
       MO2=6.0*MH2
C
C      THE DENSITY OF O IS 71.42 LBM/FT
C      THE DENSITY OF H IS  4.43 LBM/FT
C
       VH2=MH2/4.43
       VO2=MO2/71.42
       VT=VH2+VO2
C.
       PRINT *,'MASS INITIALLY                 MI    =',MI,'POUNDS MASS'
       PRINT *,'MASS OF SHUTTLE                MSH   =',MSH,'POUNDS MASS'
       PRINT *,'MASS OF CARGO                  MC    =',MC,'POUNDS MASS'
       PRINT *,'MASS OF O2 AND H2              MO2H2 =',MO2H2,'POUNDS MASS'
       PRINT *,'MASS RATIO                     MR    =',MR,'POUNDS MASS'
       PRINT *,'MASS OF H2                     MH2   =',MH2,'POUNDS MASS'
       PRINT *,'MASS OF O2                     MO2   =,',MO2,'POUNDS MASS'
       PRINT *,'  '
       PRINT *,'VOLUME OF O2 AND H2            VO2H2 =',VT,'CUBIC FEET'
       PRINT *,'VOLUME OF H2                   VH2   =',VH2,'CUBIC FEET'
       PRINT *,'VOLUME OF O2                   VO2   =',VO2,'CUBIC FEET'
       PRINT *,'   '
       PRINT *,'   '
       PRINT *,'WANT TO CALCULATE NEW VALUES?'
       PRINT *,' 1=YES   2=NO'
       READ *,A
       IF(A.EQ.2)GOTO 100
       PRINT *,'   '
       PRINT *,'   '
       PRINT *,'   '
       GOTO 200
  100  STOP
       END
```

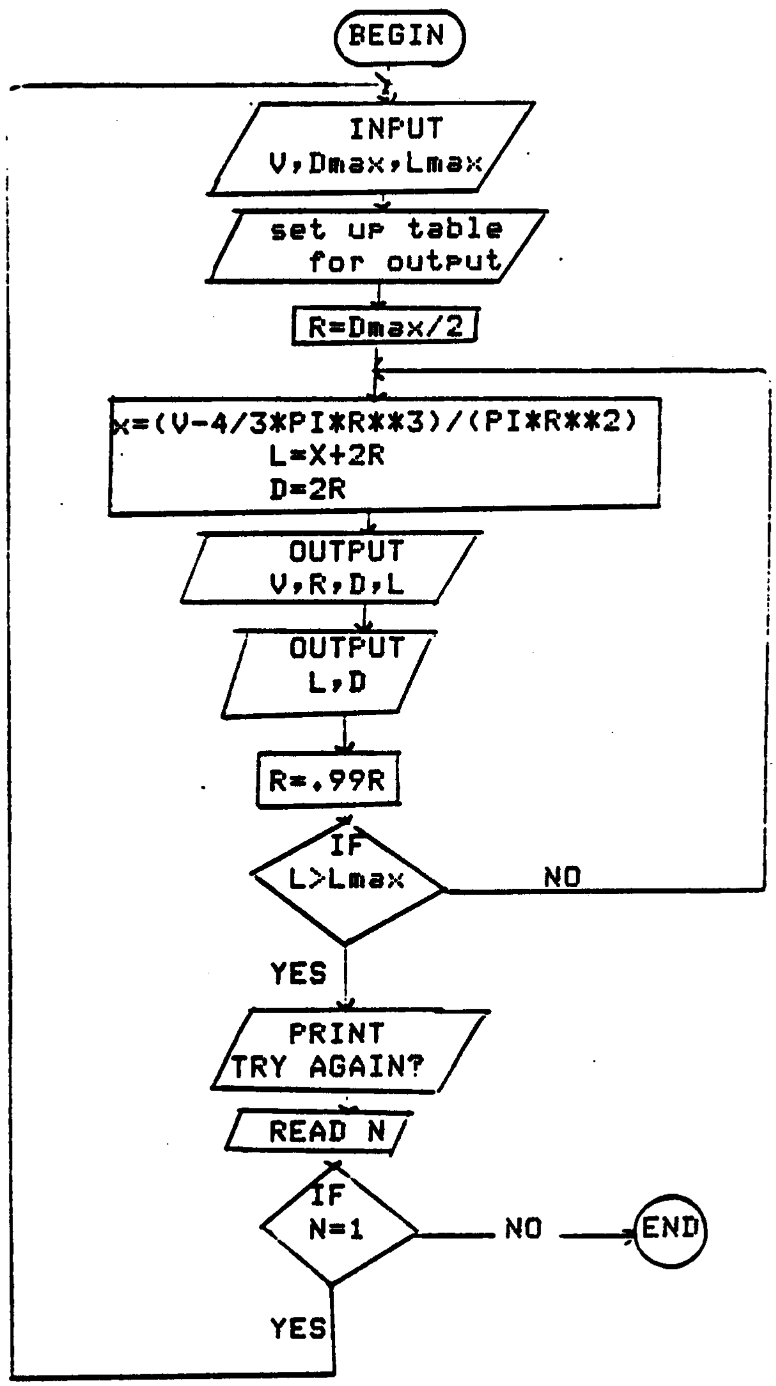
FLOW CHART FOR PROGRAM 2, MR.FOR



APPENDIX C
PROGRAM 3 WITH FLOWCHART

```
C                                         PROGRAM 3
C
C      ------------------------------------------------------------------
C      THIS PROGRAM TAKES A GIVEN VOLUME OF FLUID, ALONG WITH VARIABLES
C      FOR MAXIMUM DIAMETER AND LENGTH, AND FINDS DIMENSIONS FOR ALL
C      POSSIBLE TANK SIZES THAT COULD ACCOMIDATE THE VOLUME OF FUEL.
C      ------------------------------------------------------------------
C
       REAL L
       OPEN(UNIT=1,FILE='TWOLOX.DAT',STATUS='OLD')
       PI=3.141592654
 101   WRITE(6,102)
 102   FORMAT(/////)
       PRINT *,'INPUT VOLUME (FT.^3),MAX.DIAMETER (FT.),MAX.LENGTH (FT.)'
       READ *,V,DMAX,LMAX
       WRITE(6,103)
 103   FORMAT('   VOLUME     RADIUS     DIAMETER     LENGTH')
       WRITE(6,104)
 104   FORMAT('   ------     ------     --------     ------')
       R=DMAX/2.
  99   X=(V-(4./3.)*PI*R**3)/(PI*R**2)
       L=X+2*R
       D=2*R
       WRITE(6,105)V,R,D,L
 105   FORMAT(1X,F9.2,F11.4,F11.4,F12.4)
       WRITE(1,*)L,D
       R=.99*R
       IF(L.GE.LMAX)GOTO 100
       GOTO 99
 100   PRINT *,'                '
       PRINT *,'                '
       PRINT *,' TRY AGAIN??'
       PRINT *,' YES=1     NO=2 '
       READ *,N
       IF(N.EQ.1)GOTO 101
       CLOSE UNIT=1
       STOP
       END
```

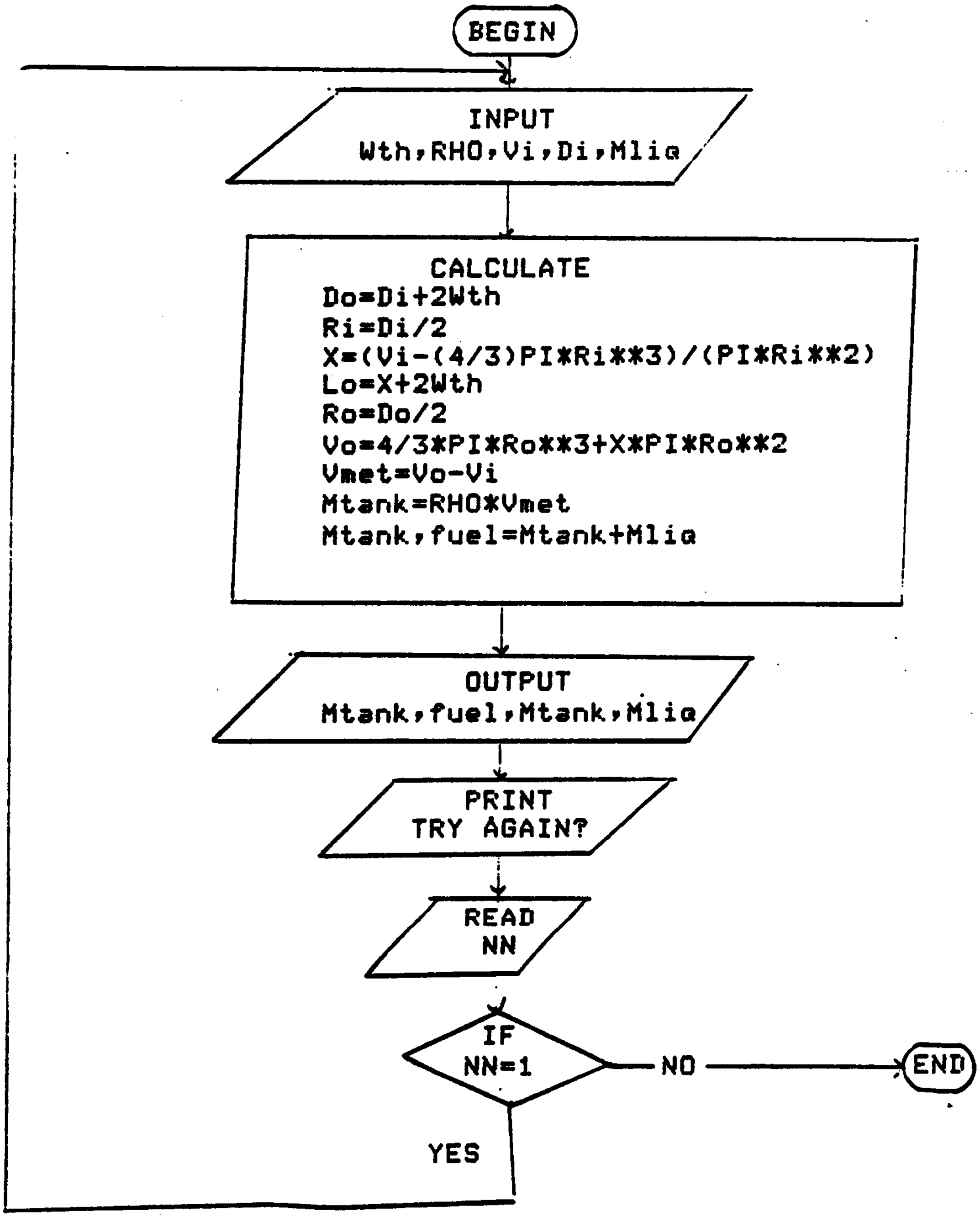
FLOW CHART FOR VOL.FOR



APPENDIX D
PROGRAM 4 WITH FLOWCHART

```
C                              PROGRAM 4
C
C     ---------------------------------------------------------------
C     THIS PROGRAM CALCULATES THE WEIGHT OF THE FUEL TANKS GIVEN THE WALL
C     THICKNESS, THE DENSITY OF THE METAL, THE VOLUME OF FLUID,THE INSIDE
C     DIAMETER, AND THE FUEL MASS.
C     ---------------------------------------------------------------
C
C
      REAL LO,MASSTANK,MTANKFUEL,MLIQUID
C
 10   PRINT *,'ENTER WALL THICKNESS (FT.),AND METAL DENSITY(LBM/FT^3)?'
      READ *,WTH,RHO
      PRINT *,'ENTER VOLUME OF FLUID(FT.^3), AND INSIDE DIAMETER(FT.)?'
      READ *,VI,DI
      PRINT *,'ENTER FUEL MASS ?'
      READ *,MLIQUID
      PI=3.141592654
      DO=DI+2.*WTH
      RI=DI/2.
      X=(VI-(4./3.)*PI*(RI)**3)/(PI*RI**2)
      LO=X+2.*WTH
      RO=DO/2.
      VO=(4./3.)*PI*RO**3+X*PI*RO**2
      VMETAL=VO-VI
      MASSTANK=RHO*VMETAL
      MTANKFUEL=MASSTANK+MLIQUID
C
C
      WRITE(6,100)
 100  FORMAT(' TANK AND FUEL MASS     TANK MASS    FUEL MASS')
      WRITE(6,99)
 99   FORMAT(' ------------------     ---------    ---------')
      WRITE(6,98)MTANKFUEL,MASSTANK,MLIQUID
 98   FORMAT(F13.2,12X,F8.2,4X,F9.2)
      WRITE(6,97)
 97   FORMAT(//////)
      PRINT *,'          '
      PRINT *,'          '
      PRINT *,' TRY AGAIN ?    YES=1'
      READ *,NN
      IF(NN.EQ.1)GOTO 10
      STOP
      END
```

FLOW CHART FOR METAL.FOR



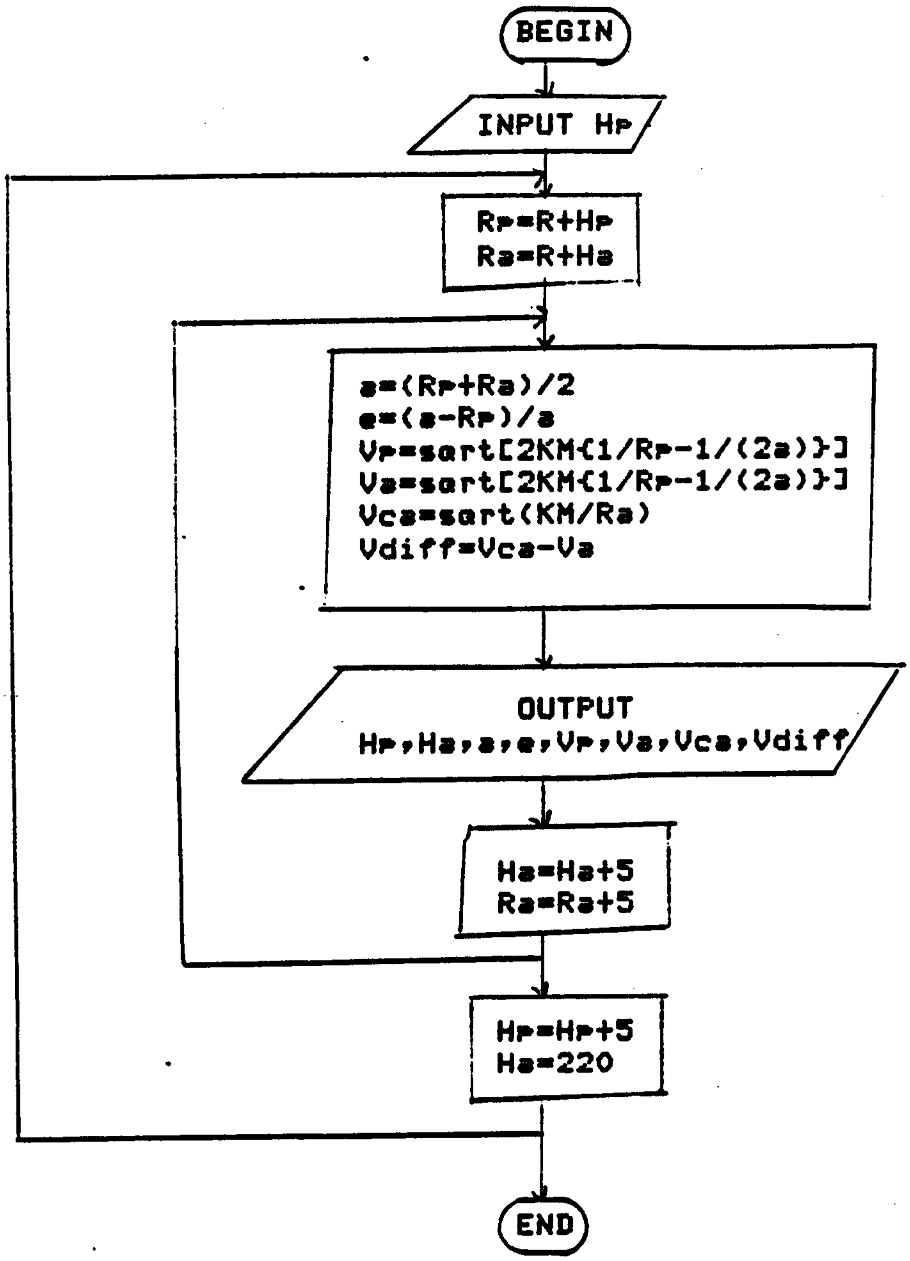
APPENDIX E
PROGRAM 5 WITH FLOWCHART

ORBITAL VELOCITY PROGRAM

THIS PROGRAM IS DESIGNED TO CALCULATE THE VELOCITIES
REQUIRED FOR VARIOUS ELLIPTICAL ORBITS

```
      REAL KM
      R=3440.0
      HA=220.
      HP=55.
      NNN=1
      KM=62746.8
      WRITE(5,001)
001   FORMAT(//,9X,'Ha',3X,'Hp',5x,'a',7x,'e',5x,'Vp',5x,'Va',5x,
     @'Vca',7x,'Vdiff')
      DO 105 K=1,12
      RP=R+HP
      RA=R+HA
      DO 100 I=1,11
      A=(RP+RA)/2.
      E=(A-RP)/A
      VP=SQRT(2.*KM*(1./RP-1./(2.*A)))*6080.
      VA=SQRT(2.*KM*(1./RA-1./(2.*A)))*6080.
      VCA=SQRT(KM/RA)*6080.
      VDIFF=VCA-VA
      NNN=NNN+1
      WRITE(5,220)HP,HA,A,E,VP,VA,VCA,VDIFF
220   FORMAT(/,5X,2F6.1,F8.1,F7.4,3F8.0,F7.0)
      HA=HA+5.
      RA=RA+5.
      IF(NNN.GT.28)THEN
            WRITE(5,221)
221         FORMAT(//////)
            NNN=0
      END IF
100   CONTINUE
      HP=HP+5.
      HA=220.
105   CONTINUE
      END
```

FLOW CHART FOR ORB.FOR



APPENDIX F
PROGRAM 6 WITH FLOWCHART

```
                           ASTRO.FOR

               THIS PROGRAM IS DESIGNED TO CALCULATE THE
               ORBITAL ELEMENTS GIVEN THE BURNOUT DATA
               OR THE BURNOUT VECTORS GIVEN THE ORBITAL
               ELEMENTS


               MAIN PROGRAM
                       CONTROLS THE TYPE OF DATA TO BE OBTAINED

     REAL I,K,M
     COMMON/DATA/X,Y,Z,XDOT,YDOT,ZDOT,A,ECCEN,THETA,OMEGA,I,W
050  WRITE(6,100)
100  FORMAT(5X,'IF YOU WISH TO FIND BURNOUT DATA TYPE 1',/,5X,
    @'IF YOU WISH TO FIND THE ORBITAL ELEMENTS TYPE 2',/,5X,
    @'INPUT DATA MUST BE IN UNITS OF RAD, NM AND NM/SEC')
     READ(5,*)J
     IF(J.EQ.1)THEN
        CALL ORBIT
     ELSE
        CALL BURN
     END IF
     WRITE(6,110)
110  FORMAT(//,5X,'IF YOU WISH TO CONTINUE TYPE 1')
     READ(5,*)K
     IF(K.EQ.1)GO TO 050
     END
     SUBROUTINE ORBIT

               SUBROUTINE ORBIT
                       CONTROL INPUT OF ORBITAL ELEMENTS AND
                       OUTPUT OF BURNOUT DATA


     REAL I
     COMMON/DATA/X,Y,Z,XDOT,YDOT,ZDOT,A,ECCEN,THETA,OMEGA,I,W
     WRITE(6,300)
300  FORMAT(/,5X,'INPUT OF ORBITAL ELEMENTS',/,5X,'A=')
     READ(5,*)A
     WRITE(6,305)
305  FORMAT(/,5X,'ECCENTRICITY=')
     READ(5,*)ECCEN
     WRITE(6,310)
310  FORMAT(/,5X,'THETA=')
     READ(5,*)THETA
     WRITE(6,315)
315  FORMAT(/,5X,'OMEGA=')
     READ(5,*)OMEGA
     WRITE(6,320)
320  FORMAT(/,5X,'I=')
     READ(5,*)I
     WRITE(6,325)
325  FORMAT(/,5X,'W=')
     READ(5,*)W
```

```
      CALL BURNOUT
      WRITE(6,330)X,XDOT,Y,YDOT,Z,ZDOT
330   FORMAT(//,9X,'RADIAL COMPONENTS',5X,'VELOCITY COMPONENTS',
     @//,2X,'XDIR',6X,F10.2,15X,F7.3,//,2X,'YDIR',6X,F10.2,15X,F7.3,
     @//,2X,'ZDIR',6X,F10.2,15X,F7.3)
      RETURN
      END
      SUBROUTINE BURN

                SUBROUTINE BURN
                        CONTROLS INPUT OF BURNOUT DATA AND
                        OUTPUT OF ORBITAL ELEMENTS


      REAL I
      COMMON/DATA/X,Y,Z,XDOT,YDOT,ZDOT,A,ECCEN,THETA,OMEGA,I,W
      WRITE(6,200)
200   FORMAT(/,5X,'IF THE COMPONENTS OF THE BURNOUT DATA ARE'
     @'NOT KNOWN TYPE 1, ELSE TYPE 2.')
      READ(5,*)J
      IF(J.EQ.1)THEN
              PRINT *,'INPUT THE ALTITUDE'
              READ(5,*)R
              PRINT *,'INPUT THE VELOCITY MAGNITUDE'
              READ(5,*)V
              PRINT *,'INPUT THE ANGLE PSI'
              READ(5,*)PSI
              X=(3440.+R)*COS(PSI)
              Y=(3440.+R)*SIN(PSI)*COS(-.4974)
              Z=(3440.+R)*SIN(PSI)*SIN(-.4974)
              XDOT=-1.*V*SIN(PSI)
              YDOT=V*COS(PSI)*COS(-.4974)
              ZDOT=-1.*V*COS(PSI)*SIN(-.4974)
      ELSE
      WRITE(6,400)
400   FORMAT(/,5X,'INPUT OF THE BURNOUT DATA IN COMPONENT FORM',//,
     @5X,'X=')
      READ(5,*)X
      WRITE(6,405)
405   FORMAT(/,5X,'Y=')
      READ(5,*)Y
      WRITE(6,410)
410   FORMAT(/,5X,'Z=')
      READ(5,*)Z
      WRITE(6,415)
415   FORMAT(/,5X,'XDOT=')
      READ(5,*)XDOT
      WRITE(6,420)
420   FORMAT(/,5X,'YDOT=')
      READ(5,*)YDOT
      WRITE(6,425)
425   FORMAT(/,5X,'ZDOT=')
      READ(5,*)ZDOT
      END IF
      CALL ORB_ELEM
      WRITE(6,440)A,ECCEN,THETA,OMEGA,I,W
```

```
0   FORMAT(//,5X,'A=',F12.3,//,5X,'E=',F6.4,//,5X,'THETA=',
   @F6.4,//,5X,'OMEGA=',F6.4,//,5X,'I=',F6.4,//,5X,'W=',F7.4)
99  RETURN
    END
    SUBROUTINE ORB_ELEM

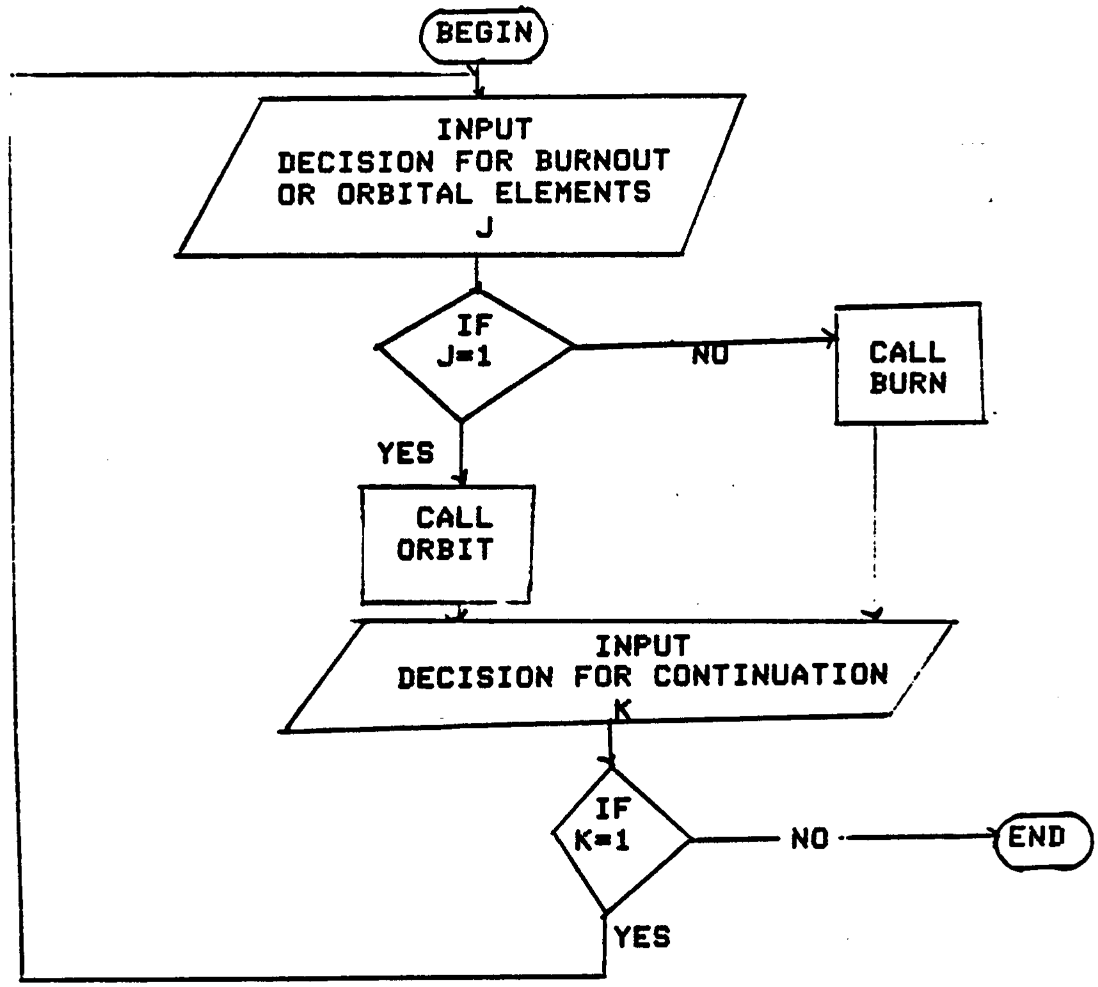
              SUBROUTINE ORB_ELEM
                      CALCULATES THE ORBITAL ELEMENTS USING
                      THE BURNOUT DATA


    REAL I,KM
    COMMON/DATA/X,Y,Z,XDOT,YDOT,ZDOT,A,ECCEN,THETA,OMEGA,I,W
    KM=62746.8
    V=SQRT(XDOT**2+YDOT**2+ZDOT**2)
    R=SQRT(X**2+Y**2+Z**2)
    CONST=R*V**2/KM
    A=R/(2-CONST)
    H=SQRT((Y*ZDOT-YDOT*Z)**2+(XDOT*Z-X*ZDOT)**2+(X*YDOT-XDOT*Y)**2)
    ALPHA=ASIN((X*XDOT+Y*YDOT+Z*ZDOT)/(R*V))
    ECCEN=SQRT((CONST-1.)**2*COS(ALPHA)**2+SIN(ALPHA)**2)
500 THETA=ACOS((H**2/KM/R-1.)/ECCEN)
    OMEGA=ATAN((Y*ZDOT-YDOT*Z)/(X*ZDOT-XDOT*Z))
    I=ACOS((X*YDOT-Y*XDOT)/H)
    W=ATAN((-1.*X*SIN(OMEGA)*COS(I)+Y*COS(OMEGA)*COS(I)+Z*SIN(I))/
   @(X*COS(OMEGA)+Y*SIN(OMEGA)))-THETA
00  RETURN
    END
    SUBROUTINE BURNOUT

              SUBROUTINE BURNOUT
                      CALCULATES THE BURNOUT DATA FROM THE
                      ORBITAL ELEMENTS


    REAL I,N,L1,M1,N1,L2,M2,N2,KM
    COMMON/DATA/X,Y,Z,XDOT,YDOT,ZDOT,A,ECCEN,THETA,OMEGA,AI,AW
    KM=62746.8
    EL=A*(1.-ECCEN**2)
    R=EL/(1.+ECCEN*COS(THETA))
    B=A*SQRT(1.-ECCEN**2)
    N=SQRT(KM/A**3)
    V=SQRT(KM*(2./R-1./A))
    E=ACOS(1./ECCEN-R/(A*ECCEN))
    L1=COS(OMEGA)*COS(AW)-SIN(OMEGA)*SIN(AW)*COS(AI)
    M1=SIN(OMEGA)*COS(AW)+COS(OMEGA)*SIN(AW)*COS(AI)
    N1=SIN(AW)*SIN(AI)
    L2=-1.*COS(OMEGA)*SIN(AW)-SIN(OMEGA)*COS(AW)*COS(AI)
    M2=-1.*SIN(OMEGA)*SIN(AW)+COS(OMEGA)*COS(AW)*COS(AI)
    N2=COS(AW)*SIN(AI)
    X=A*L1*COS(E)+B*L2*SIN(E)-A*ECCEN*L1
    Y=A*M1*COS(E)+B*M2*SIN(E)-A*ECCEN*M1
    Z=A*N1*COS(E)+B*N2*SIN(E)-A*ECCEN*N1
    CONST=N*A/R
    XDOT=CONST*(B*L2*COS(E)-A*L1*SIN(E))
    YDOT=CONST*(B*M2*COS(E)-A*M1*SIN(E))
    ZDOT=CONST*(B*N2*COS(E)-A*N1*SIN(E))
    RETURN
    END
```

FLOW CHART FOR ASTRO.FOR
MAIN PROGRAM



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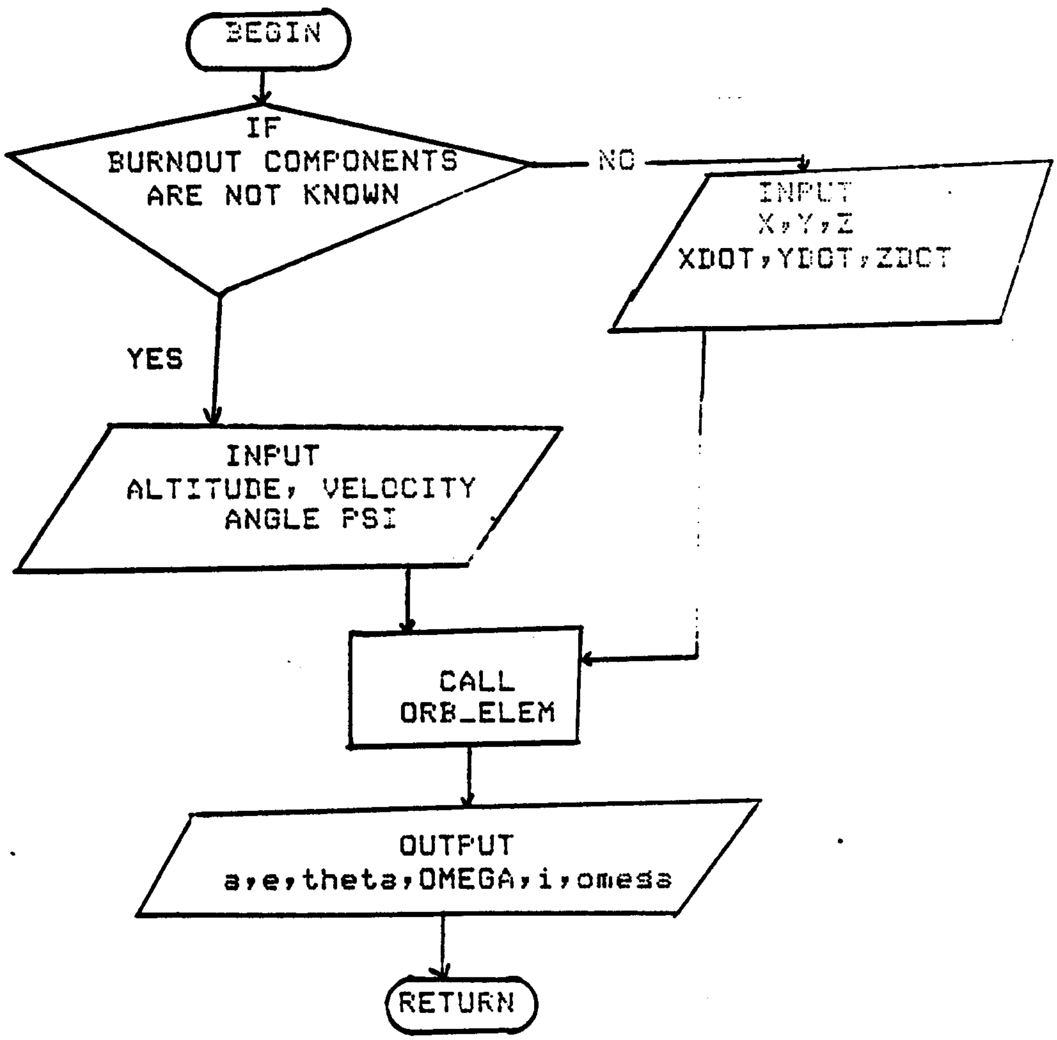
FLOW CHART FOR ASTRO.FOR
SUBROUTINE BURN



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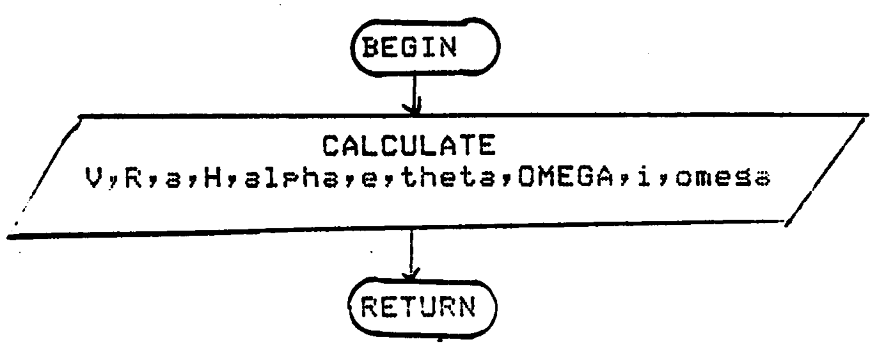
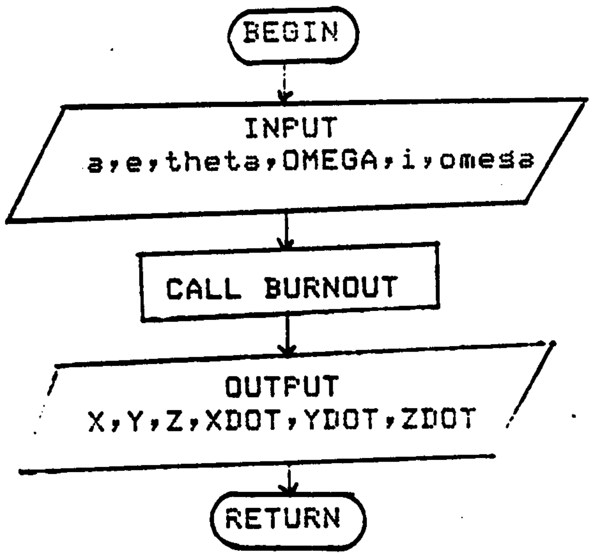
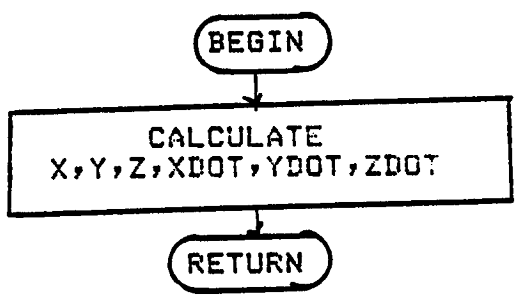
SUBROUTINE ORB_ELEM



FLOW CHART FOR ASTRO.FOR
SUBROUTINE ORBIT



FLOW CHART FOR ASTRO.FOR
SUBROUTINE BURNOUT



APPENDIX G - REFERENCE MATERIAL
RECEIVED FROM MARSHALL SPACE FLIGHT CENTER

1. MASS REPORT
2. DESIGN DATA
3. GROUP WEIGHT STATEMENT

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****
MASS REPORT
****

AEROSHELL GRAPHITE-POLYIMIDE SSTO 50% LOX-RP/TOTAL

INITIAL GUESS VERSION 0.1

1.0 WING GROUP			4334. kg	
2.0 TAIL GROUP			1234. kg	
3.0 BODY GROUP			38405. kg	
BASIC STRUCTURE		15566. kg		
THRUST STRUCTURE		4349. kg		
RP-1 TANK		1896. kg		
LOX TANK		8500. kg		
LH2 TANK		7680. kg		
BODY FLAP		414. kg		
4.0 INDUCED ENVIRONMENT			13712. kg	
5.0 LANDING GEAR			3377. kg	
6.0 PROPULSION			18200. kg	
7.0 PROPULSION, RCS			1265. kg	
8.0 PROPULSION, OMS			637. kg	
9.0 PRIME POWER			954. kg	
9.1 FUEL CELLS		125. kg		
9.2 REACTANT DEWARS		226. kg		
9.3 REACTANTS		297. kg		
9.4 BATTERIES		306. kg		
10.0 ELEC CONV AND DISTR			4568. kg	
11.0 CONTROLS			2485. kg	
11.1 HYDRAULICS		0. kg		
11.2 SURFACE CONTROLS		2485. kg		
13.0 AVIONICS			2248. kg	
14.0 ENVIRONMENTAL CONTROL			2000. kg	
15.0 PERSONNEL PROVISIONS			368. kg	
16.0 MARGIN			7587. kg	
DRY WEIGHT		101445. kg		(0.000)
17.0 PERSONNEL			1303. kg	
18.0 RESIDUAL FLUIDS			6907. kg	
LANDED WEIGHT W/O CARGO		109655. kg		(0.000)
20.0 CARGO (RETURNED)			15420. kg	
LANDED WEIGHT		125075. kg		(0.000)
ENTRY WEIGHT		125075. kg		(0.000)
23.0 ACPS PROPELLANT			14134. kg	
RCS	1326. kg			
OMS	12808. kg			
24.0 CARGO DELIVERED			0. kg	
25.0 ASCENT RESERVES			1275. kg	
26.0 INFLIGHT LOSSES			1031. kg	
27.0 ASCENT PROPELLANT			1062542. kg	
RP-1 150263. kg	LH2 68074. kg	LOX 844205. kg		
GROSS LIFT OFF WEIGHT		1204056. kg		(0.000)

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***************
* DESIGN DATA *
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                     INITIAL GUESS  VERSION 0.1

STRUCTURAL MASS REDUCTION            0.15
SUBSYSTEM MASS REDUCTION             0.15

BODY LENGTH                         40.87 m
BODY STR. WETTED AREA             1546.66 sqm
VERTICAL TAIL AREA                  25.50 sqm
THEORETICAL WING AREA              458.47 sqm
WING SPAN                           36.06 m
STRUCTURAL SPAN                     21.70 m
MAX ROOT THICKNESS                   1.17 m
CONTROL SURFACE AREA               167.31 sqm
BODY FLAP AREA                      32.54 sqm


BODY VOLUME                       3624.70 cum
FREE WING VOLUME                     0.00 cum
UTILIZED WING VOLUME                 0.00 cum
TANK EFFICIENCY                    0.5450
BASE AREA                            0.00 sqm
ENGINE DEPARTMENT LENGTH             0.00 m
FIXED VOLUME                         0.00 cum


                                 MODE 1               MODE 2

NUMBER OF ENGINES                0.9965               0.6020
SEA LEVEL THRUST            11473776.00 N         7640185.00 N
VACUUM THRUST               12615004.00 N         9061978.00 N
ENGINE MASS/VACUUM THRUST    0.00083600           0.00152730

HC PROPELLANT FRACTION           0.1414
LH                               0.0641
OX                               0.7945
TANK OXIDIZER VOLUME           773.06 cum
OXIDIZER DENSITY              1140.50 kg/cum
HYDROCARBON FUEL VOLUME        104.01 cum
HYDROCARBON DENSITY            800.00 kg/cum
HYDROGEN FUEL VOLUME          1000.44 cum
HYDROGEN DENSITY                70.50 kg/cum
PERCENT ULLAGE                 0.0425

T/W                             1.300
MASS RATIO                     0.5084
MODE 2 S.L. THRUST/TOTAL       0.3000
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GROUP WEIGHT STATEMENT ORBITER 103 (INERT) 2 AUG 75

FUNCTION	FUNCTION DESCRIPTION	WEIGHT (LB)	X-CG (IN)	Y-CG (IN)	Z-CG (IN)
1.0 0 0	WING GROUP	15684.0	1227.8	0.0	298.3
1.1 0 0	BASIC STRUCTURE	9757.1	1250.4	0.0	300.2
1.1 B 0	OUTER PANEL INTERM SECTION	2256.4	1107.4	0.0	305.8
1.1 C 0	OUTER PANEL TORQUE BOX	6235.6	1281.1	0.0	298.6
1.1 D 0	OUTER PANEL LEADING EDGE	129.9	1159.0	0.0	300.0
1.1 E 0	OUTER PANEL TRAILING EDGE	1135.2	1376.9	0.0	297.3
1.2 0 0	SECONDARY STRUCTURE	3188.9	1005.5	0.0	297.8
1.2 A 0	STRUCTURE	2517.3	977.8	0.0	295.9
1.2 B 0	OPERATING MECH & CONTROLS	350.0	1112.9	0.0	306.0
1.2 D 0	OPERATING MECH-ACTUATOR LOCKS	193.6	1101.1	0.0	286.6
1.2 F 0	MISC PROV & SUPPT	128.0	1111.6	0.0	328.9
1.3 0 0	CONTROL SURFACES	2738.0	1403.9	0.0	292.1
1.3 B 0	ELEVON INBOARD SURFACE	1334.2	1423.6	0.0	286.1
1.3 C 0	ELEVON INBOARD SUPPORT MECHANISM	310.6	1358.0	0.0	288.4
1.3 D 0	ELEVON OUTBOARD SURFACE	722.4	1412.0	0.0	300.9
1.3 E 0	ELEVON OUTBOARD SUPPORT MECH	370.8	1355.9	0.0	299.6
2.0 0 0	TAIL GROUP	2862.0	1509.3	0.1	633.6
2.1 0 0	BASIC STRUCTURE-VERTICAL FIN	1777.5	1470.3	0.2	616.1
2.1 C 0	OUTER PANEL TORQUE BOX	1523.0	1469.9	0.2	621.2
2.1 D 0	OUTER PANEL LEADING EDGE	95.6	1421.6	0.0	631.6
2.1 E 0	OUTER PANEL TRAILNG EDGE	158.9	1503.9	0.0	558.3
2.2 0 0	SECONDARY STRUCTURE-DOORS,MISC	188.6	1519.1	0.0	639.9
2.2 A 0	STRUCTURE	58.6	1487.6	0.0	601.9
2.2 E 0	OPER MECH SUPPORTS	130.0	1533.3	0.0	657.0
2.3 0 0	CONTROL SURFACES	895.9	1584.5	-0.0	667.0
2.3 B 0	RUDDER/SPEED BRAKE UPPER SURFACE	376.3	1617.4	-0.0	716.1
2.3 C 0	RUDDER/SPEED BRAKE UPR SUPT MECH	55.1	1606.6	0.0	729.6
2.3 D 0	RUDDER/SPEED BRAKE LOWER SURFACE	393.4	1556.6	-0.0	617.6
2.3 F 0	RUDDEP/SPEED BRAKE LWR SUPT MECH	71.1	1547.4	0.1	632.3
3.0 0 0	BODY GROUP	43180.0	998.4	-0.7	375.2
3.1 0 0	BASIC STRUCTURE	31034.0	995.0	-0.1	351.2
3.1 A 0	FORWARD FUSELAGE	4014.6	456.4	0.0	362.2
3.1 B 0	CREW MODULE	4038.1	501.2	-3.4	388.1

PAGE 1

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GROUP WEIGHT STATEMENT ORBITER 103 (INERT) 2 AUG 75

FUNCTION	FUNCTION DESCRIPTION	WEIGHT (LB)	X-CG (IN)	Y-CG (IN)	Z-CG (IN)
3.1 C 0	MID FUSELAGE	11275.5	1008.4	-0.7	327.2
3.1 D 0	AFT FUSELAGE - BODY	7257.0	1381.6	0.0	356.0
3.1 E 0	AFT FUSELAGE THRUST STRUCTURE	3831.0	1416.0	4.5	360.5
3.1 F 0	FWD RCS MODULE	617.8	324.8	0.0	361.5
3.2 0 0	SECONDARY STRUCTURE	3896.1	980.0	0.8	400.8
3.2 A 0	FORWARD FUSELAGE	501.0	477.4	0.0	453.5
3.2 B 0	CREW MODULE	1556.1	513.0	2.1	433.6
3.2 C 0	MID FUSELAGE	259.0	974.4	0.0	354.8
3.2 D 0	AFT FUSELAGE - BODY	1135.0	1493.9	0.0	386.5
3.2 E 0	AFT FUSELAGE - BODY FLAP	430.0	1563.0	0.0	287.0
3.2 F 0	FWD RCS MODULE	15.0	329.0	0.0	368.0
3.3 0 0	SECONDARY STRUCT DOORS & MISC	3619.9	1171.5	-7.8	434.7
3.3 A 0	FWD FUSELAGE	368.6	378.3	0.6	357.3
3.3 B 0	FWD RCS MODULE	22.0	354.0	0.0	399.0
3.3 C 0	CREW MODULE	568.8	525.6	-43.1	377.0
3.3 D 0	OMS/RCS POD	2269.0	1453.4	0.0	482.6
3.3 E 0	MID FUSELAGE	65.5	793.2	-61.1	361.6
3.3 F 0	AFT FUSELAGE	326.0	1364.1	0.0	306.1
3.4 0 0	SEC STRUCT PAYLOAD BAY DOORS	4630.0	934.9	-0.0	467.8
3.4 A 0	STRUCTURE-TULSA	2970.0	947.0	-1.0	468.7
3.4 B 0	DOOR ACTUATION-PAYLOAD BAY	206.0	905.6	0.0	406.8
3.4 C 0	DOOR LATCH	722.0	945.0	3.7	487.1
3.4 D 0	RADIATOR HINGE	240.0	762.1	0.0	458.4
3.4 E 0	MISCELLANEOUS	492.0	943.8	0.0	464.6
4.0 0 0	INDUCED ENVIRONMENTAL PROTECT	20002.0	1097.5	0.0	352.1
4.1 0 0	TPS (EXTERNAL) WING	8076.0	1218.7	0.0	309.7
4.1 A 0	WING LEADING EDGE RCC	2695.0	1141.7	0.0	309.9
4.1 B 0	FIXED SURFACE (WING)	4108.0	1205.3	0.0	320.0
4.1 C 0	ELEVON-INBOARD (WING)	751.0	1430.0	0.0	272.1
4.1 D 0	ELEVON-OUTBOARD (WING)	522.0	1418.0	0.0	282.5
4.2 0 0	TPS (EXTERNAL) TAIL	919.0	1528.1	0.0	661.0
4.2 A 0	FIN	787.0	1516.9	0.0	659.8
4.2 B 0	RUDDER/SPEED BRAKE LOWER	65.0	1565.3	0.0	620.0
4.2 C 0	RUDDER/SPEED BRAKE UPPER	67.0	1623.2	0.0	715.0

FUNCTION	FUNCTION DESCRIPTION	WEIGHT (LB)	X-CG (IN)	Y-CG (IN)	Z-CG (IN)
4.3 0 0	TPS (EXTERNAL) BODY	6635.0	996.8	0.0	339.6
4.3 A 0	FORWARD (BODY)	1794.0	376.6	0.0	339.1
4.3 B 0	MIDBODY	1771.0	933.0	0.0	294.5
4.3 C 0	MIDBODY-PAYLOAD BAY DOORS	578.0	987.1	0.0	465.8
4.3 D 0	AFT (BODY)	1042.0	1456.5	0.0	311.5
4.3 E 0	BODY FLAP	850.0	1554.8	0.0	281.8
4.3 F 0	OMS/RCS PODS	600.0	1460.1	0.0	402.8
4.4 0 0	THERMAL CONTROL SYS (INTERNAL)	3413.0	935.6	0.0	387.4
4.4 A 0	WING	75.0	1336.0	0.0	355.0
4.4 B 0	TAIL	24.0	1451.8	0.0	525.3
4.4 C 0	BODY-FORWARD	960.0	453.5	0.3	372.4
4.4 D 0	BODY-MID DOORS	384.0	1030.3	0.0	452.3
4.4 E 0	BODY-MID OTHER	1113.0	905.6	0.0	337.4
4.4 F 0	BODY-AFT	857.0	1422.7	-0.3	439.1
4.5 0 0	PVD-PURGE AND VENT SYSTEM	874.0	957.5	0.0	374.7
4.5 A 0	WING	86.0	1021.3	0.0	319.7
4.5 B 0	TAIL	9.0	1467.0	0.0	640.0
4.5 C 0	BODY-FORWARD	251.0	573.1	0.1	361.8
4.5 D 0	BODY-MID	325.0	945.9	0.0	380.5
4.5 E 0	BODY-AFT	141.0	1393.3	0.0	384.1
4.5 F 0	BODY-OMS RCS	62.0	1420.8	0.0	413.4
4.6 0 0	PVD-WINDOW COND SYS-HODS	85.0	715.2	-0.0	354.1
4.6 A 0	WING	5.0	1109.0	0.0	300.0
4.6 B 0	TAIL	1.0	1467.0	0.0	640.0
4.6 C 0	BODY-FORWARD	51.0	405.3	-0.1	334.9
4.6 D 0	BODY-MID	12.0	940.0	0.0	350.0
4.6 E 0	BODY-AFT	12.0	1357.2	0.0	396.7
4.6 F 0	BODY-OMS/RCS	4.0	1385.0	0.0	480.0
5.[illegible] 0	LANDING & AUXILIARY SYSTEMS	7603.0	989.4	-9.3	313.7
5.1 0 0	ALIGHTING GEAR STRUCTURE	5159.0	979.8	0.0	304.4
5.1 A 0	MAIN ROLLING GEAR	1696.0	1074.0	0.0	305.0
5.1 B 0	MAIN GEAR STRUCTURE	2637.0	1121.9	0.0	300.4
5.1 C 0	NOSE ROLLING GEAR	189.0	304.0	0.0	323.0
5.1 D 0	NOSE GEAR STRUCTURE	637.0	341.5	0.0	313.9

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GROUP WEIGHT STATEMENT ORBITER 103 (INERT) 2 AUG 75

FUNCTION	FUNCTION DESCRIPTION	WEIGHT (LB)	X-CG (IN)	Y-CG (IN)	Z-CG (IN)
5.2 0 0	ALIGHTING GEAR CONTROLS	1024.0	896.7	-1.0	315.9
5.2 A 0	MAIN RETRACT	501.3	1136.9	0.4	312.5
5.2 B 0	BRAKE OPERATION	219.1	1076.4	0.1	313.1
5.2 D 0	NOSE RETRACT	261.8	373.0	-5.1	323.8
5.2 E 0	NOSE STEERING	41.8	353.8	1.8	321.3
5.3 0 0	AUXILIARY SYSTEMS	1500.0	1085.6	-46.8	344.1
5.3 C 0	AUX SYSTEMS - SEPARATION	740.0	1238.0	-0.6	250.8
5.3 E 0	AUX SYSTEMS PAYLOAD HANDLING	760.0	937.3	-91.8	435.0
6.0 0 0	PROPULSION - ASCENT	28054.0	1465.6	1.6	375.7
6.1 0 0	ENGINE ACCESSORIES	2297.0	1470.7	10.4	375.5
6.1 A 0	SSME INTERFACE	60.0	1449.7	0.0	375.3
6.1 B 0	HEAT SHIELD	339.0	1515.3	0.0	373.3
6.1 C 0	GIMBAL SYSTEM	1818.0	1464.8	12.8	374.8
6.1 D 0	PURGE SYSTEM	23.0	1466.8	-25.2	399.1
6.1 E 0	ENGINE HYDRAULIC SUPPLY	57.0	1427.7	22.0	400.2
6.2 0 0	PROPELLANT SYSTEM	6419.0	1387.4	2.2	351.8
6.2 B 0	PNEUMATIC SYSTEM	574.0	1420.7	0.2	304.0
6.2 C 0	E/T PRESSURIZATION SYSTEM	323.7	1414.1	33.2	376.0
6.2 D 0	PROPELLANT SYSTEM-FUEL	2613.8	1377.2	-22.0	368.1
6.2 E 0	PROPELLANT SYSTEM - OXIDIZER	2796.5	1385.7	20.3	344.5
6.2 F 0	POGO SUPPRESSION	111.0	1418.5	36.5	329.3
6.3 0 0	ENGINE SSME	19338.0	1491.0	0.3	383.7
6.3 A 0	SSME	19338.0	1491.0	0.3	383.7
7.0 0 0	PROPULSION - RCS	2657.0	920.4	0.0	426.9
7.1 0 0	REACTION CONTROL SYS (FWD)	1260.0	333.8	0.0	373.1
7.1 A 0	RCS FWD MODULE ENGINE INST	705.0	338.3	0.0	381.4
7.1 B 0	RCS FWD PRESSURIZATION SYSTEM	156.0	313.7	0.0	358.6
7.1 C 0	RCS FWD PROPELLANT SYSTEM FUEL	199.0	333.9	-20.4	364.1
7.1 D 0	RCS FWD PROPELLANT SYSTEM OXID	200.0	333.9	20.4	364.0
7.2 0 0	REACTION CONTROL SYS (AFT)	1397.0	1449.5	0.0	475.5
7.2 A 0	RCS AFT POD ENGINE INSTALLATION	468.0	1537.3	0.0	462.0
7.2 B 0	RCS AFT PRESSURIZATION SYSTEM	301.0	1397.2	0.0	487.8
7.2 C 0	RCS AFT PROPELLANT SYSTEM FUEL	314.0	1408.8	0.0	485.1

GROUP WEIGHT STATEMENT ORBITER 103 (INERT) 2 AUG 75

FUNCTION	FUNCTION DESCRIPTION	WEIGHT (LB)	X-CG (IN)	Y-CG (IN)	Z-CG (IN)
7.2 D 0	RCS AFT PROPELLANT SYSTEM OXID	314.0	1409.6	0.0	474.2
8.[redacted]	PROPULSION -OMS	2902.0	1454.7	0.0	478.5
8.1 0 0	ORBIT MANEUVER SYSTEM	2902.0	1454.7	0.0	478.5
8.1 B 0	ENGINE INSTL	522.0	1520.0	0.0	493.0
8.1 C 0	PRESSURIZATION SYS	810.0	1461.4	0.0	465.1
8.1 D 0	OMS FUEL SYSTEM	785.0	1429.5	-0.1	496.9
8.1 E 0	OMS OXIDIZER SYSTEM	785.0	1429.4	0.1	464.4
9.[redacted]	PRIME POWER	2929.0	898.2	2.3	334.7
9.1 0 0	AUXILIARY POWER SYSTEM	743.0	1352.7	-12.2	422.9
9.1 A 0	APU POWER SYSTEM	321.0	1320.9	-7.0	425.0
9.1 B 0	APU FUEL SYSTEM	307.0	1394.8	-20.9	401.0
9.1 C 0	APU LUBE OIL COOLANT LOOP	65.0	1332.6	-5.7	476.5
9.1 D 0	APU EXHAUST SYSTEM	50.0	1323.9	-0.7	474.2
9.2 0 0	ELECTRICAL POWER SYSTEM	2186.0	743.7	7.3	304.7
9.2 A 0	POWER GEN SYS FUEL CELL	897.0	653.0	21.0	305.7
9.2 C 0	PRSD OXYGEN	664.0	755.8	-1.9	304.3
9.2 D 0	PRSD HYDROGEN	625.0	861.1	-2.8	303.6
10.0 0 0	ELECTRICAL CONVERSION & DISTR	6942.0	857.4	5.7	352.7
10.1 0 0	ELECT POWER CONVERSION & DISTR	2863.0	740.5	10.4	352.2
10.1 A 0	CONVERSION EQUIPMENT	223.0	455.0	17.7	385.7
10.1 B 0	CONTROL UNITS	949.0	880.3	8.5	341.7
10.1 C 0	DISTRIBUTION EQUIPMENT	533.0	784.7	11.1	340.6
10.1 D 0	LIGHT SYSTEM	137.0	668.8	-7.4	404.7
10.1 E 0	ELECTRICAL SUPPORTS	276.0	632.5	15.1	348.4
10.1 F 0	ELECTRICAL INSTALLATION	745.0	669.6	11.8	355.4
10.2 0 0	CIRCUITRY	4079.0	939.3	2.5	353.0
10.2 A 0	ELECTRICAL CABLING	2633.8	939.6	3.1	363.3
10.2 B 0	ELECTRICAL CONNECTORS	347.1	963.0	2.9	352.3
10.2 C 0	CABLING INSTALLATION & SUPPORT	1098.1	931.3	1.0	328.6
11.0 0 0	HYDRAULIC CONVERSION & DISTR	1840.0	1338.5	0.2	437.2
11.1 0 0	HYDR POWER CONVERSION & DISTR	1840.0	1338.5	0.2	437.2

FUNCTION	FUNCTION DESCRIPTION	WEIGHT (LB)	X-CG (IN)	Y-CG (IN)	Z-CG (IN)
11.1 A 0	HYDR POWER SUPPLY EQUIPMENT	150.0	1316.9	3.6	438.3
11.1 B 0	HYDR DISTRIBUTION & CONTROL	1153.0	1326.6	-0.3	416.3
11.1 C 0	HYDR TEMPERATURE CONTROL SYSTEM	537.0	1370.3	0.4	481.8
12.0 0 0	SURFACE CONTROLS	2758.0	1443.7	3.4	449.0
12.1 0 0	SURFACE CONTROL SYSTEMS	2758.0	1443.7	3.4	449.0
12.1 A 0	COCKPIT CONTROLS	65.0	468.3	-1.8	425.2
12.1 B 0	SYSTEM ACTUATION-BODY FLAP	446.8	1524.8	16.2	286.6
12.1 C 0	SYSTEM ACTUATION-ELEVON	1031.0	1362.8	0.6	298.1
12.1 D 0	SYSTEM ACTUAT-RUDDER SPD,BRAKE	1215.2	1534.7	1.2	638.1
13.0 0 0	AVIONICS	5755.0	609.6	11.5	395.6
13.1 0 0	GUIDANCE NAVIGATION+CONTRL SYS	934.0	742.2	-4.9	373.8
13.1 A 0	UNITS	765.0	804.5	-6.2	368.6
13.1 B 0	GUIDANCE NAVIG & CONTROL INSTALL	169.0	460.2	0.7	397.2
13.2 0 0	COMMUNICATION & TRACKING SYS	1196.0	533.9	33.1	383.7
13.2 A 0	UNITS	566.0	534.4	26.7	374.8
13.2 B 0	CIRCUITRY	197.0	503.5	12.3	336.5
13.2 C 0	ANTENNA	274.0	535.7	42.3	407.2
13.2 D 0	INSTALLATION	159.0	566.8	65.4	433.0
13.3 0 0	DISPLAYS & CONTROLS SYSTEM	1725.0	505.9	4.0	450.7
13.3 A 0	UNITS	1283.0	504.8	4.9	449.5
13.3 B 0	CIRCUITRY	85.0	512.8	0.0	456.5
13.3 D 0	DISPLAYS & CONTROLS INSTALLATION	357.0	508.2	1.8	453.8
13.4 0 0	INSTRUMENTATION SYSTEM	461.0	772.2	15.2	387.0
13.4 A 0	UNITS	410.0	765.6	15.4	387.5
13.4 B 0	CIRCUITRY	22.0	583.0	0.0	355.0
13.4 D 0	OPERATIONAL FLIGHT INSTR INSTALL	29.0	1010.0	19.0	404.8
13.5 0 0	DATA PROCESSING & SOFTWARE	1439.0	658.7	12.1	356.4
13.5 A 0	UNITS	1434.0	659.3	12.0	356.2
13.5 D 0	INSTALLATION	5.0	474.0	37.0	423.0
14.0 0 0	ENVIRONMENTAL CONTROL	5060.0	780.5	5.7	368.5
14.1 0 0	CABIN & PERSONNEL SYSTEM	2281.5	605.7	13.7	338.6
14.1 A 0	CABIN PRESSURE CONTROL	502.5	668.7	30.4	330.2

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FUNCTION	FUNCTION DESCRIPTION	WEIGHT (LB)	X-CG (IN)	Y-CG (IN)	Z-CG (IN)
14.1 B 0	CABIN ATMOSPHERE REVITALIZATION	600.1	490.4	16.4	344.9
14.1 C 0	HEAT TRANSPORT WATER LOOP	453.7	577.7	-0.0	322.3
14.1 D 0	EQUIP ENVIR CONTROL LOOP	725.2	675.1	8.4	349.3
14.2 0 0	EQUIPMENT ENVIR & HT TRANSPORT	2732.5	930.2	-0.7	393.9
14.2 B 0	HEAT TRANSPORT FREON LOOP	2622.7	912.1	-2.3	392.0
14.2 C 0	AMMONIA SYSTEM	109.8	1361.8	38.7	439.1
14.3 0 0	CABIN AND PERSONNEL SYSTEM	46.0	555.3	-10.5	348.8
14.3 A 0	AIRLOCK SUPPORT SYSTEM	46.0	555.3	-10.5	348.8
15.0 0 0	PERSONNEL PROVISION	1100.0	508.2	-15.7	362.3
15.1 0 0	FIXED LIFE SUPPORT SYSTEM	730.0	500.6	-21.7	330.5
15.1 A 0	FOOD MANAGEMENT SYSTEMS	161.0	469.0	-67.0	367.0
15.1 B 0	WASTE MANAGEMENT SYSTEM	325.5	515.2	-21.2	318.0
15.1 C 0	WATER MANAGEMENT SYSTEM	210.5	505.6	8.2	310.0
15.1 D 0	FIRE DETECTION SYSTEM	33.0	479.5	2.5	407.5
15.2 0 0	PERSONNEL ACCOMMODATIONS	300.0	522.2	0.0	432.7
15.2 A 0	COMMANDER STATION	96.0	501.0	22.0	437.0
15.2 B 0	PILOT STATION	96.0	501.0	-22.0	437.0
15.2 C 0	PAYLOAD MONITOR STATION	54.0	560.0	-32.0	425.0
15.2 D 0	MISSION MONITOR STATION	54.0	560.0	32.0	425.0
15.3 0 0	FURNISHINGS & EQUIPMENT	70.0	527.1	-20.3	391.9
15.3 A 0	FURNISHINGS	70.0	527.1	-20.3	391.9
18.0 0 0	PAYLOAD PROVISIONS	585.0	1158.8	25.0	418.3
18.1 0 0	FIXED SCAR ITEMS	135.0	1441.4	21.8	496.1
18.1 B 0	MID FUSELAGE	5.0	1200.0	0.0	350.0
18.1 C 0	AFT FUSELAGE	64.0	1394.5	48.4	357.7
18.1 E 0	VERTICAL TAIL	66.0	1505.1	-2.3	641.4
18.2 0 0	REMOVABLE PAYLOAD PROVISIONS	450.0	1074.0	26.0	395.0
18.2 B 0	MID FUSELAGE	450.0	1074.0	26.0	395.0
19.0 0 0	MARGIN (103)	1332.0	1063.0	1.0	367.0
19.1 0 0	MARGIN	1332.0	1063.0	1.0	367.0
19.1 A 0	MARGIN	1332.0	1063.0	1.0	367.0

GROUP WEIGHT STATEMENT ORBITER 103 (INERT) 2 AUG 75

FUNCTION	FUNCTION DESCRIPTION	WEIGHT (LB)	X-CG (IN)	Y-CG (IN)	Z-CG (IN)
	INERT WEIGHT (INCL SSME)	151325.0	1117.0	0.6	369.7

AE 449 AEROSPACE DESIGN

Auburn University

Auburn, Alabama

FINAL REPORT

for

INTERMEDIATE-ORBIT CARGO VEHICLE

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DATE SUBMITTED: 17 June 1987

ABSTRACT

This report describes progress made to improve initial design considerations for the Intermediate-Orbit Cargo Vehicle. It provides a comprehensive overview of mission profile, structural design and cost analyses, and relates these to the overall feasibility and usefulness of the proposed system.

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INTRODUCTION

The decision to initiate design of a new cargo vehicle was based on predictions of imminent near-space equipment needs to support the space station. As the US presses on towards the 21st century, many new aerospace projects will require payload capabilities beyond that of present launch vehicles. By devising a vehicle capable of transporting cargo to an intermediate orbit and in essence "parking" it in that orbit, the US will achieve the capability of having "orbital warehouses" where material and supplies for orbital operations can be stored.

The mission requirements established for this vehicle are shown in Figure 1. They call for an unmanned vehicle that will place a payload of 150,000 lbs in a 150 nautical mile orbit. The payload bay size must be 25 ft by 90 ft or 33 ft by 100 ft. The staging velocity must be a maximum of 7000 ft/s. The fly-back-booster which will launch the cargo vehicle into orbit must see a maximum of 3 g's. The proposed configuration is shown in Figure 2.

After separation from the fly-back-booster, the restriction that acceleration be held to within 3 g's for human tolerance is removed since the cargo vehicle is unmanned. The vehicle will sustain a maximum of 5 g's. All preliminary specifications were based on data obtained from the Saturn and Apollo projects. Subsequently, as a result of continuing research and follow-on design considerations, the accuracy of design specifications has increased drastically. Research on the cargo vehicle includes precise trajectory analysis, estimates of structural sizes and weights, fuel tank sizing, and estimated costs of production.

As stated earlier, the cargo vcehicle's mission requires that it be placed in orbit at an altitude of 150 nautical miles. After it has achieved

the required orbit, the nose cone will detach from the cargo section and will burn up as it re-enters the atmosphere. The cargo will then be removed through the open end of the payload bay. An orbital tranfer vehicle (OTV) will attach to a docking mechanism (Figure 3) similar to the one used on the Saturn Command Module. Docking will be achieved by maneuvering the OTV close enough to the multiple docking adaptor (MDA) located on the first payload frame's forward spider beam so that the probe on the OTV engages with a drogue on the multiple docking adaptor. When the probe comes into contact with the drogue it will be guided into a socket at the bottom of the drogue. Three capture latches in the probe head will secure the OTV to the first payload frame. The OTV will withdraw the first payload frame from the payload bay (Figure 4), towing the remaining frames forward until the second frame is in place in the space origunally occupied by the first frame. The OTV will then separate the first payload frame from the second payload frame and proceed to the space station with the first payload frame in tow. After transporting the first payload frame to the space station, the OTV will return to the cargo vehicle and engage the MDA on the second patload frame's forward spider beam. The OTV will remove the second payload frame, tow it to the space station, and then return to the cargo vehicle. It will repeat this off-loading procedure until all the cargo is secured at the space station. After cargo off-loading is completed, explosive bolts will separate the upper and lower sections, each of which will be equiped with an MDA, for transportation to the space station. This will save the fuel tanks, flight instruments, engines, and empty payload bay for future use.

Although it would be more economical to tow the entire cargo vehicle, cargo included, to the space station in one trip, considerations for the capabilities of the OTV dictate that the components be transferred individually.

TRAJECTORY

A direct ascent trajectory will be used to place the vehicle in the parking orbit. In this approach, the engines burn continuously from lift-off to orbital insertion. When burnout occurs, the vehicle is at circular satellite speed and the angle between the orbital direction and the velocity vector of the vehicle is zero. When using this approach, angle and velocity are critical. If the angle or velocity is incorrect, the vehicle will not attain the proper orbit. Therefore, extra fuel space has been made available so that the engines can be used for orbital corrections. Reaction controls located in the nose cone will be used to pitsh the vehicle so that errors in the ascent trajectory can be corrected by re-firing the engines for short periods of time.

Using a TRAJECTORY program, the trajectory data were calculated. For brevity, a summary is given here and important events are illustrated in Figure 5. The program and the flow chart are shown in Appendix A.

Vehicle Mass - 383,247.3

Payload Mass - 150,000 lbs

Propellant Mass - 1,108,133 lbs

Structural Mass - 233,247.3 lbs

Initial Mass
at Separation - 1,491,380.3 lbs

G-Turn Altitude - 1800 ft

Thrust - Cargo - 1,531,100 lbs

- Total at
Lift-off - 10,221,000 lbs

Separation Velocity - 7080 ft/s

Separation Altitude - 46.92 nmi

Separation Range - 38.87 nmi downrange

Burnout Velocity - 24,352 ft/s

Burnout Altitude - 159.4 nmi

Burnout Range - 720 nmi downrange

Refering to Figure 5, point A denotes the commencement of the gravity assisted turn. Point B denotes the point of booster separation and point C denotes the point of orbital insertion.

The next step was to investigate the advantages of the conventional burn-coast-burn trajectory. In this approach, the vehicle's engines burn at a very high rate up to a certain point. This maneuver places the vehicle into an ascent ellipse with apogee at the desired orbital altitude. When the vehicle reaches the apogee of the ascent ellipse, the engines are again fired to accelerate the vehicle to circular orbital speed and to align the vehicle with the direction of the orbit. It was initially presumed that lift-off mass might be reduced significantly through use of this approach. However, subsequent analysis indicated that use of the burn-coast-burn trajectory would not result in a significant reduction of lift-off mass.

STRUCTURES

Continuing development of the cargo vehicle led to extensive research in the area of vehicle structures. Originally, structural weights were assumed and structural materials were not even considered. In addition, two cargo bays were initially included in the design. However, because of design re-specification, only one cargo bay is used in the final design. Specific calculated structural sizes and weights of the lower and upper sections are listed below and shown in Figure 6.

LOWER SECTION

Aluminum 2014 Tb Alloy

Overall Size - 89.0 ft (height)
- 34.0 ft (diameter)

Fuel Tanks - 67.0 ft (height)
- 33.0 ft (diameter)

Engine Compartment
and Engine Cones - 19.0 ft (height)
- 34.0 ft (diameter)

Instrument Unit and
Separation Wall - 3.0 ft (height)
- 34.0 ft (diameter)

Overall Weight - 109,913 lbs (Est. 113,368.78 lbs)

Fuel Tanks (Dry) - 36,500 lbs

Skin - 29,845 lbs at .25 in (thickness)

Instrument Unit - 6,000 lbs

Thrust Structure - 7,000 lbs

Engines (4) - 28,568 lbs

Other - 2,000 lbs (Pipes, Insulation, Wiring, Bulkheads)

UPPER SECTION

Aluminum 2014 Tb Alloy

Overall Size - 101.0 ft (height)
- 34.0 ft (diameter)

Skin - 101.0 ft (height)
- 34.0 ft (diameter)

Payload Structure - 100.0 ft (height)
- 33.0 ft (diameter)

Spider Beams (2) - 1.0 ft (height)
- 33.0 ft (diameter)

Overall Weight - 123,334.0 lbs

Skin - 47,065.0 lbs at .375 in (thickness)

Payload Structure - 37,715.0 lbs

Spider Beams (2) - 4,000.0 lbs

Payload Containers - 9,127.0 lbs

Cone - 12,000.0 lbs

Interstage - 6,000.0 lbs

Restraints - 7,424.0 lbs

TOTAL STRUCTURAL SIZE AND WEIGHT

Total Structural Mass - 233,247.0 lbs

Total Structural Height - 223.0 ft

Lower Stage - 82.0 ft

Payload Stage - 101.0 ft

Interstage - 3.0 ft

Cone - 30.0 ft

Engine Cones - 7.0 ft

The structure of the cargo vehicle can be divided into three main categories: the lower section, the interstage, and the upper or payload section.

LOWER SECTION

The lower section ,as shown in Figure 7, is composed of an instrument unit, a tail unit, a propellant tank unit, and an engine unit.

The instrument unit (Figure 8) is an unpressurized, cylindrical, load-supporting structure. It will be 34.0 ft in diameter and 3.0 ft high. It will weigh approximately 6,000 lbs.

The instrument unit will be constructed in three segments to facilitate shipping and packaging. The structure consists of inner and outer stage skins, a honeycomb core, and forward and aft innerstage rings. Splice plates connect the three segments. After assembly of the instrument unit, a door designed to act as a load-carrying member during flight provides access to the electronic equipment inside the structure.

The instrument mounting panels, which are used for mounting various components of systems housed in the instrument unit, attach to welded brackets around the inner periphery of the instrument unit.

The instrument unit houses electrical and mechanical equipment. This equipment is used for guidance and control and monitors the vehicle's flight performance from lift-off to orbit. The instrument unit houses several systems such as guidance and control, measuring and telemetry, instrument unit command, tracking, and electrical systems.

The tail unit (Figure 9) is composed mainly of the thrust structure. This structure provides a mounting point for the four engines and the instrument unit. Engine thrust is transmitted through the thrust structure to the instrument unit and then up through the propellant tank system. The

forward end of the tail unit is enclosed by fire walls and the aft end is enclosed by heat shield panels.

The thrust structure consists of the barrel assembly and thrust supports. The instrument unit fire wall panels are also housed in the thrust structure. The bottom of the barrel assembly will contain the heat shields for the four engines.

The heat shield panels seen in Figure 9 enclose the aft end of the tail unit assembly to form an engine compartment that protects the engines from the heat of engine exhaust. Each panel is a layer of corrosion-resistant steel honeycomb sandwiched between two layers of corrosion-resistant steel sheet. A plaster-type thermal insulation is trowelled into the cellular openings of the honeycomb.

The flame shield closes the area between the engines. The flame shield is made of corrosion-resistant steel. All parts of the flame shield that come in contact with the engines are insulated.

Flame curtains will be installed between the heat shield panels and the four engines' thrust chambers. This will prevent engine exhaust flame from entering the engine compartment. The curtain will be made of silicon rubber insulating material sandwiched between a double thickness of fiberglass cloth tape on the side closest to the engines.

The propellant tank unit (Figure 7) will house the propellant tanks. These propellant tanks will incorporate the principle of the common bulkhead, which comprises the top half of the liquid oxygen tank and the bottom of the liquid hydrogen tank.

The common bulkhead causes problems because the H_2 is so much colder than the O_2; therefore, the bulkhead will be well insulated between the two ellipsoid sphere halves forming it. The insulation will be a honeycombed

phenolic material that will be fitted between the base of the LH_2 tank and the top of the LOX tank. It is estimated that the common bulkhead will save about 12.0 ft in length and 12,250.0 lbs in weight. For information on tank sizing see Appendix B.

The propellant pipes for the H_2 will run through the center of the O_2 tank and instrument unit to the engines. They will be insulated. The pipes for the O_2 tank will also run through the center of the instrument unit.

Fins will not be used on the cargo vehicle. It was determined that during the initial boost phase the combined booster and cargo vehicle could be stabilized by the use of the combined thrust vectoring capabilities and the lifting surfaces of the booster. With 10.5 degrees total pitch and 8.5 degrees horizontal yaw, the resultant force can be made to counteract the moment produced by the adverse difference in the center of pressure and the center of gravity.

After separation from the booster, the atmospheric conditions at 47 nautical miles altitude are such that the aerodynamic effects are very minute. Therefore, stability must be maintained using non-aerodynamic effects (i.e. thrust vectoring).

The engine unit (Figure 7) consists of the propulsion system. The lower stage will be powered by a cluster of four engines. Each engine develops 395,000 lbs thrust for a total thrust of 1,580,000 lbs for the whole rocket (see Figure 10). The four engines will be gimballed for thrust vectoring. In addition, each engine will have an extendable nozzle for added thrust after separation.

INTERSTAGE

The interstage forms a structural interface between the lower section and the upper or payload section. It is a cylindrical, skin-stringer-type aluminum structure. It is 3.0 ft high and 34.0 ft in diameter. It will weigh approximately 6,000.0 lbs.

UPPER OR PAYLOAD SECTION

The payload section (Figure 11) will be of aluminum skin-stringer-type construction. It will be 101.0 ft tall and have an outside diameter of 34.0 ft. It will be constructed in two parts, an outside shell and an internal frame which will hold the payload.

The outside shell will have an aluminum skin, 8 horseshoe beam longitudinal members, and nine rings. The 8 horseshoe beam members and skin will provide most of the structural integrity of the system. They will also provide support for the payload frame tracks. The nine rings will provide support for the system. The 8 horseshoe beams will be supported by aluminum rings on both ends. The aft ring will be attached to the interstage. The outside shell will be 101.0 ft long and have an inside diameter of 33.5 ft (see Figure 10).

The payload frames (Figure 12) will each have 8 I-beam type longitudinal members and will feature circumferential aluminum bands placed at regular intervals to provide additional structural integrity. The fore and aft ends of each payload frame will be capped with spider beam supports. The aft spider beam of the rearmost payload frame will help transfer weight and thrust between the interstage and the payload bay. Each payload frame will have an inside diameter of 33.0 ft and an outside diameter of 33.25 ft.

The 8 horseshoe beams of the outside shell will act as tracks for

movement of the payload frame I-beam members during cargo off-loading. The tracks and I-beams will be coated with Teflon and sprayed with silicon to provide smooth, virtually frictionless movement.

The nose-cone will be made of aluminum skin-stringer-type construction. It will serve as an aerodynamic fairing and will also house the vehicle's reaction controls. The nose-cone will be 34.0 ft in diameter at the base and 30.0 ft high. It will have the shape of a tangential ogive.

FUEL SECTION

Knowing that the total mass of the propellant is 1,108,133 lbs and that the mixture ratio is 1 part oxygen to 6 parts hydrogen, the masses of the fuel and oxidizer were found. The mass of the oxygen was determined to be approximately 949,828 lbs and the mass of the hydrogen was calculated to be approximately 158,304 lbs. The density of the oxygen will be 71.2 lbm/ft^3 and the density of the hydrogen will be 4.43 lbm/ft^3. By dividing the mass of each propellant component by its particular density, the volume necessary to accommodate the required amount of propellant can be determined.

$$V_{O_2} = \frac{949{,}828 \text{ lbs}}{71.2 \text{ lbm/ft}^3} = 13{,}340.28 \text{ ft}^3$$

$$V_{H_2} = \frac{158{,}304 \text{ lbs}}{4.43 \text{ lbm/ft}^3} = 35{,}734.698 \text{ ft}^3$$

The oxygen will be held in an ellipsoid shaped tank with radius equal to 16.5 ft and height equal to 23.4 ft. For details on the tank sizing see Appendix B.

The hydrogen will be held in a cylindrical tank joined in tandem with the oxygen tank (Figure 7). The upper end of the cylinder will be closed with a curved cap equal in volume to half that of the oxygen tank. The bottom end of the cylinder will be joined to the oxygen tank, which will intrude half its volume upon the cylinder, reducing the cylinder's volume by an amount equal to half the volume of the oxygen tank. The net effect of the upper cylinder cap in tandem connection with the oxygen tank will be to provide space for a volume of hydrogen equal to that available in the standard, unenhanced cylinder. To determine the total weight of the fuel tanks, the heights of the

ellipsoid and the enhanced cylinder were combined. As stated previously, the ellipsoid height was determined to be 23.4 ft. The height of the cylinder was calculated from the cylindrical volume equation (Appendix B) and was found to be 41.78 ft. Thus the total fuel section height will be 65.18 ft, rounded up to 67.0 ft to accommodate orbital maneuvering purposes.

COST ANALYSIS

The costs for the vehicle were determined using the Launch Vehicle Cost Model program at the Marshall Space Flight Center. A compilation of all data used in the program and the results are presented in Figure 13.

The structural weight used in the program was 187,288 lbs and includes the vehicle skin, fuel tanks, internal supports, pipes, insulation,bulkhead, wiring, frame, forward and aft skirts, tunnels, interstage, and nose fairing. The base heat shield protects the thrust structure, propellant tanks, and other stage elements from excessive heat and gases given off by the engines during their burn. A weight of 1,000 lbs was used for this component in the program. The avionics, whose weight was estimated to be 250.0 lbs, will be composed of the following components: (1) Guidance, Navigation , and Control, and (2) Communications and Data Handling. The electrical power equipment weight used in the program was 500.0 lbs. This weight will be comprised of batteries of fuel cells, a power distribution system, relays, inverters, power transformers, cables, and wiring. The propulsion components other than the engines were estimated to weigh a total of 1500.0 lbs and will include a propellant feed system, recirculation system, propellant management system, pressurization system, plumbing, valves, lines, and gimballing bearings. The total cost for the vehicle elements other than engines is $1.12473 billion during the design, development, test, and evaluation (DDT&E) stage and $417.21 million for the total first unit (TFU).

The structural tooling, systems test hardware and assembly, and systems test operations will cost an estimated $2.60864 billion for DDT&E. Ground support equipment (GSE) during DDT&E will have an estimated cost of $453.44 million, while systems engineering and integration is estimated at $322.55

million during DDT&E and $34.12 million for the TFU. The estimated cost for the program management is $129.65 million during DDT&E and $16.47 million for the TFU.

The engines and incidental costs are the final considerations. The engine parameter used in the program was calculated by dividing the product of weight, vacuum thrust, specific impulse, and chamber pressure by 10,000,000 (this parameter is known as the "composite variable"). Use of this parameter in the program yielded estimated costs of $44.1 million during DDT&E and $40,000 for the TFU. The incidental costs include a contractor's fee of $498.17 million during DDT&E and $65.5 million for the TFU; program support costs of $121.7 million during DDT&E and $10.67 million for the TFU; and cost contingencies of $626.74 million during DDT&E and $81.6 million for the TFU.

A simplicity factor of 1 was used in the cost analysis program since our vehicle was designed based on previous Saturn project data. Also, surface flight controls and hydraulics were neglected due to the elimination of aerodynamic fins from the final design.

CONCLUSIONS

The goal of the ongoing research described in this report is to perfect the design and production of an unmanned, intermediate-orbit cargo vehicle for use with fly back booster. Recent work has led to significant enhancements in the projected capabilities of the vehicle, including an improved mission profile, a cost-efficient development and production plan, and an innovative approach to cargo storage and transferral.

The overall design of this cargo vehicle meets and exceeds all required standards set by NASA. It can be seen that aerospace projects of this nature will lead to the advancement of the US space program in the near future.

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GROUND RULES FOR CARGO VEHICLE

-UNMANNED

-PLACE 150,000 lb PAYLOAD IN 150 NAUTICAL MILE ORBIT

-PAYLOAD BAY: 25'D X 90'L OR 33'D X 100'L

-PROPELLANT: LIQUID OXYGEN/LIQUID HYDROGEN

-FIRST FLIGHT IN 1998; DESIGN FREEZE IN 1990

Figure 1. Mission Requirements

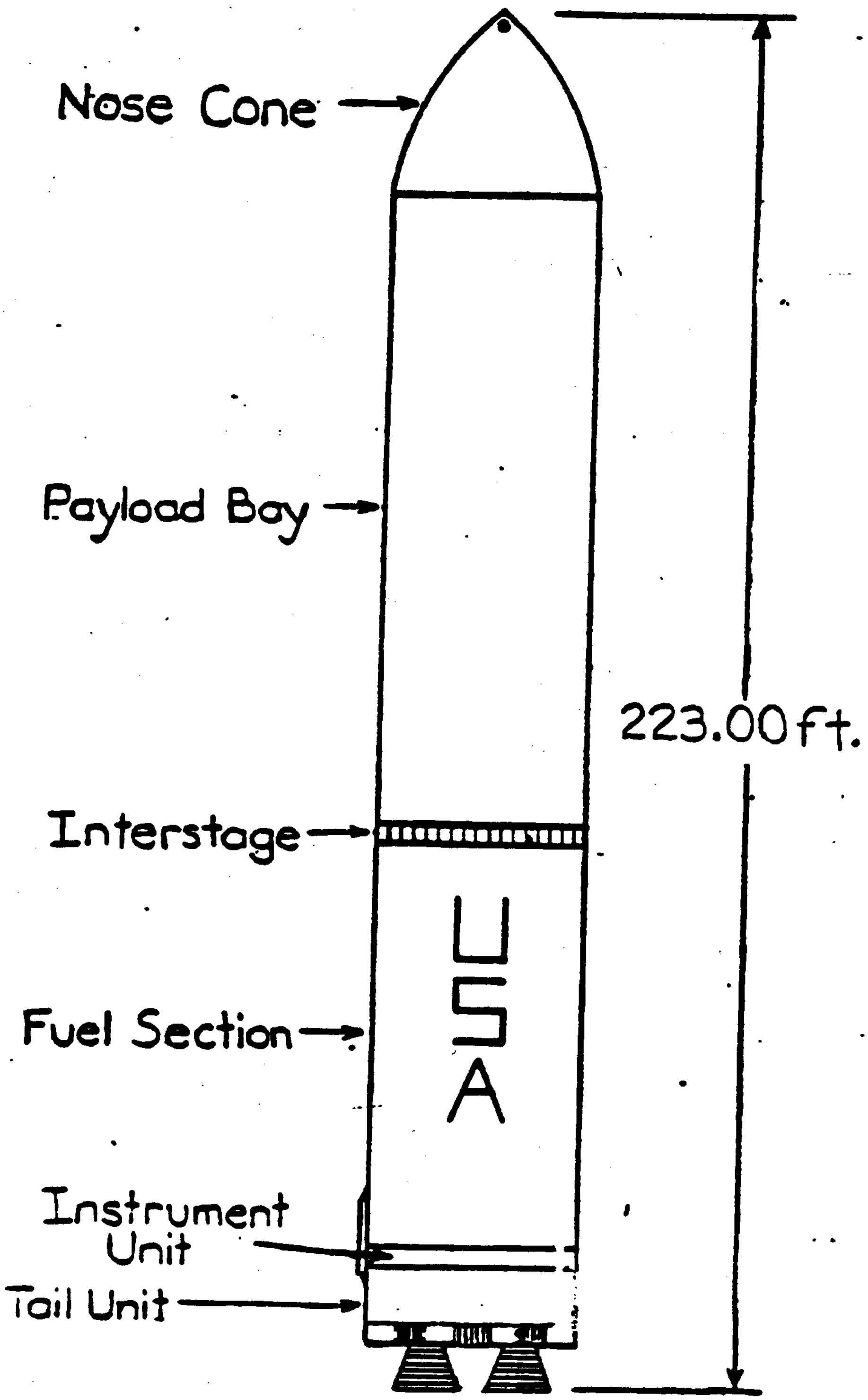
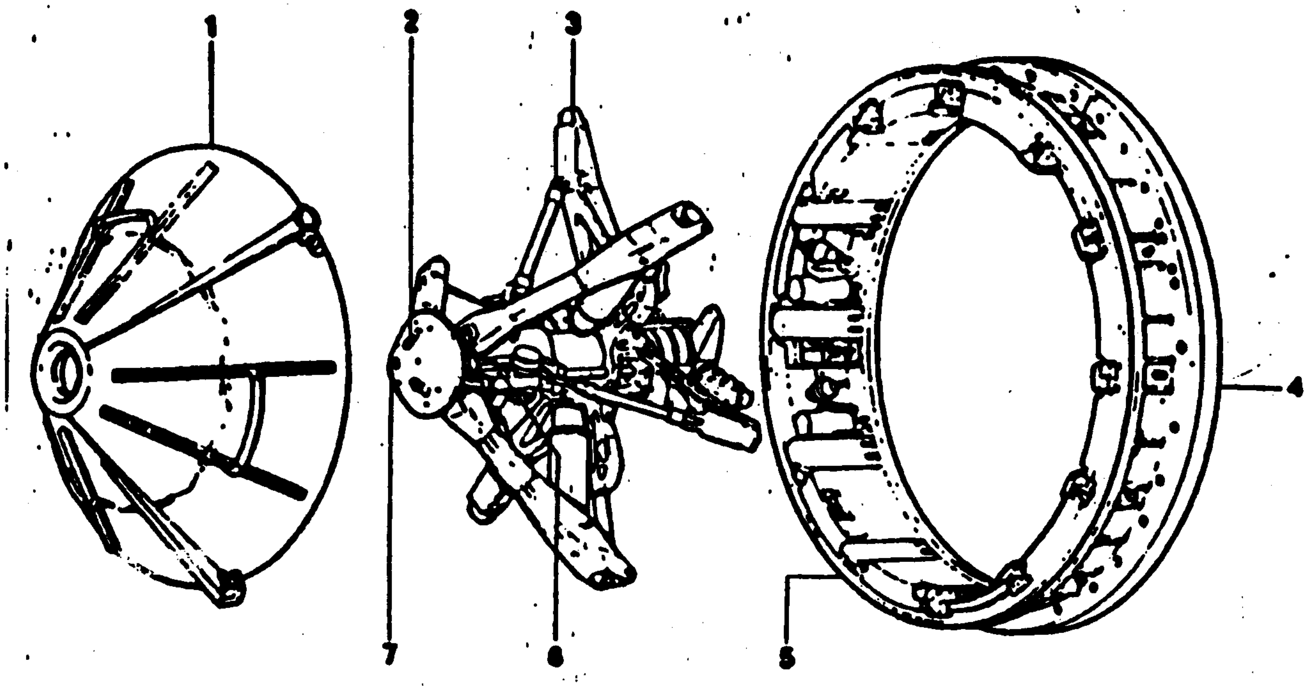


Figure 2. Unmanned Cargo Vehicle



Docking Unit
1 Drogue assembly
2 Capture latches
3 Probe assembly
4 Docking ring
5 Latch assemblies
6 Self-locking extension latch
7 Capture latch release handle

Figure 3. Docking Mechanism

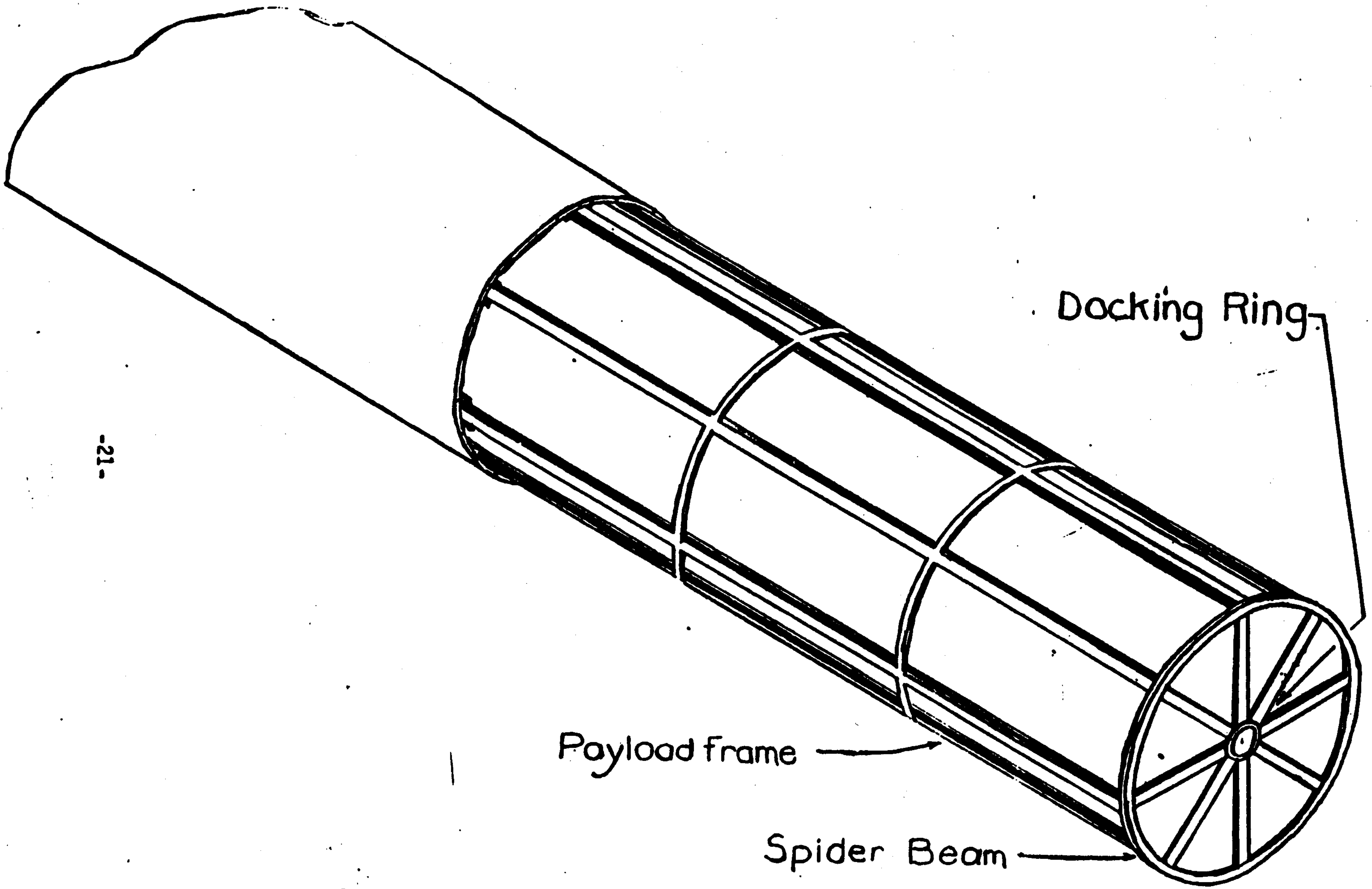


Fig. 4. Extended Payload Frame

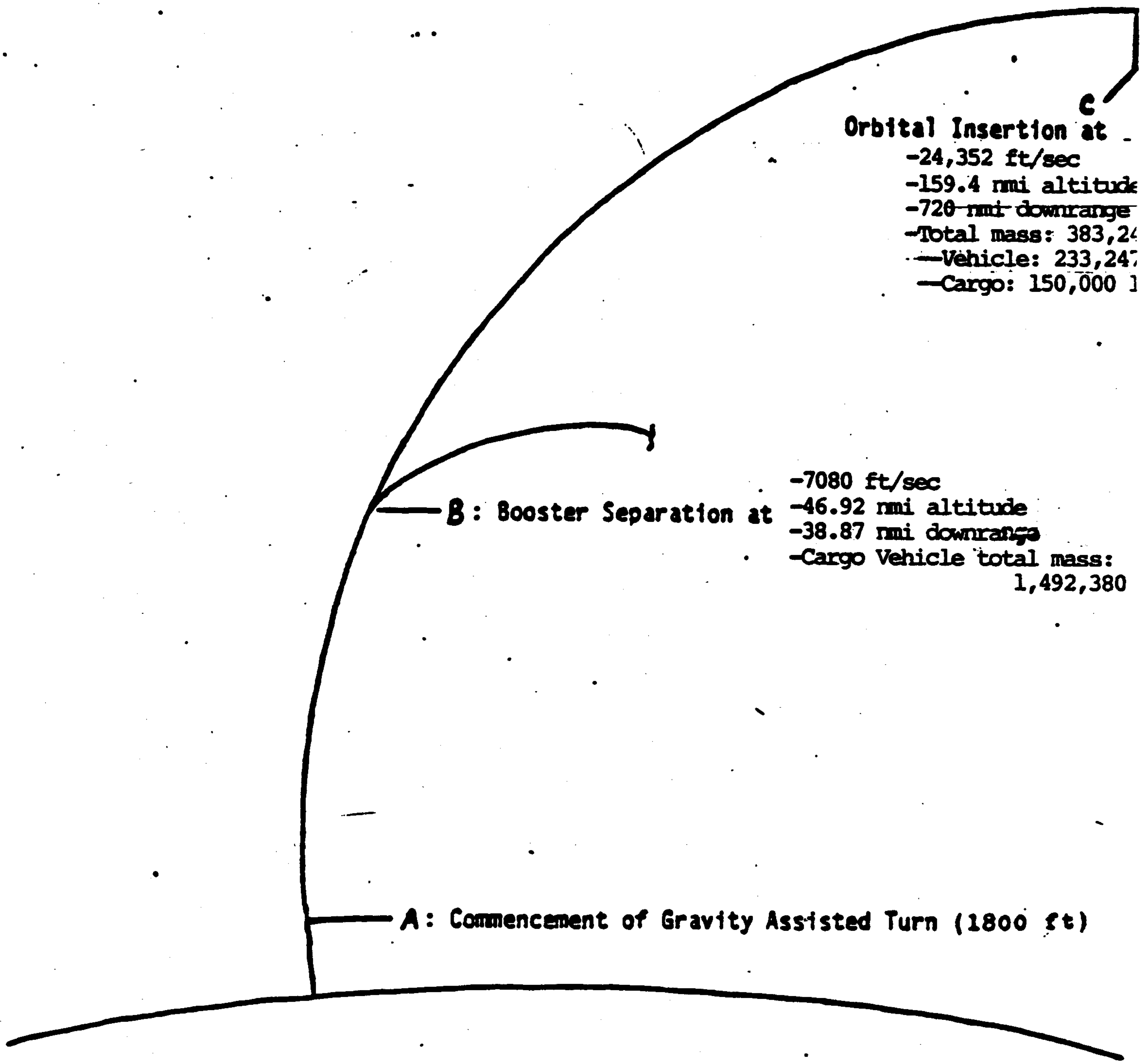


FIGURE 5. Graphical Representation of Trajectory

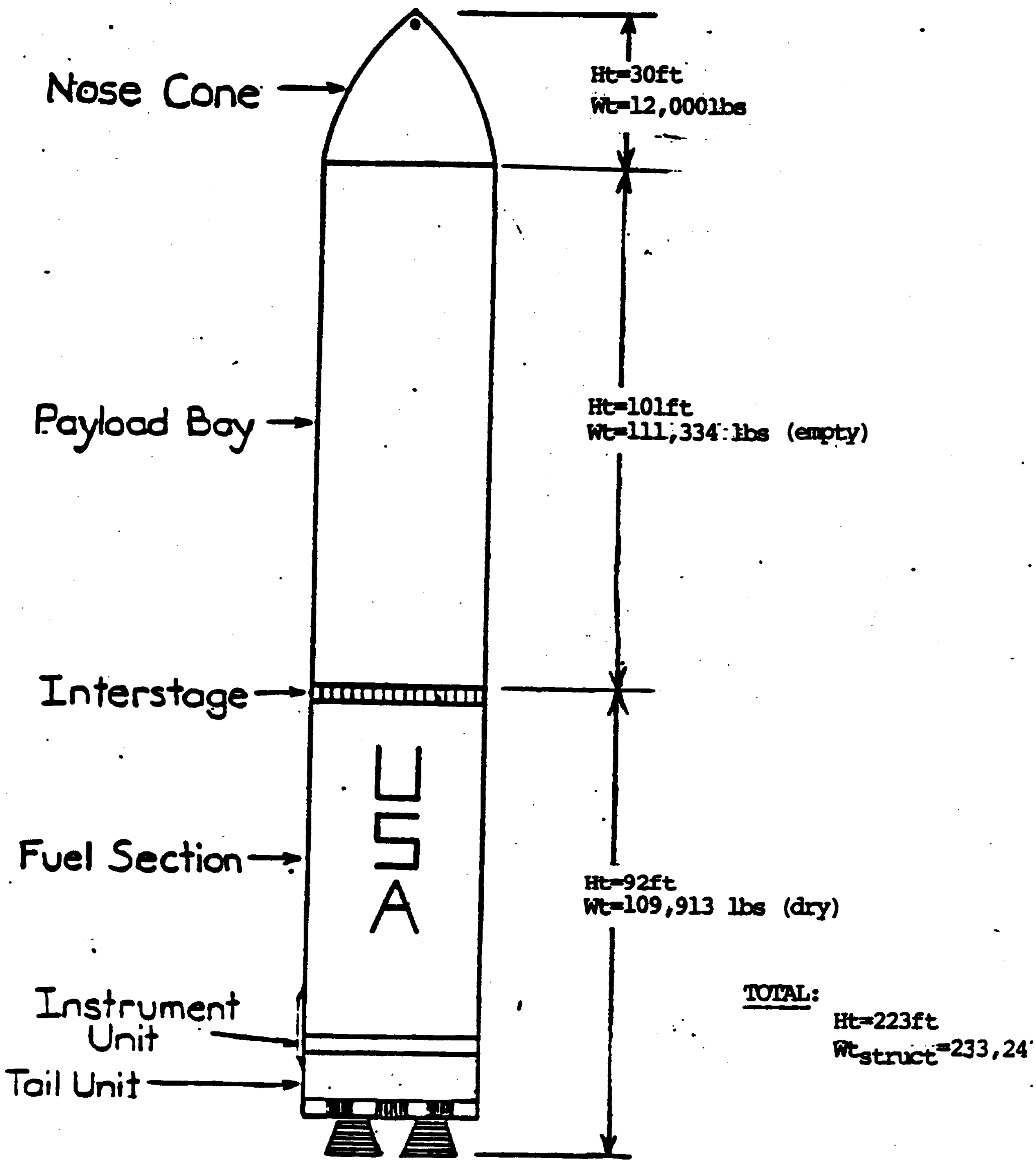


Figure 6. Unmanned Cargo Vehicle

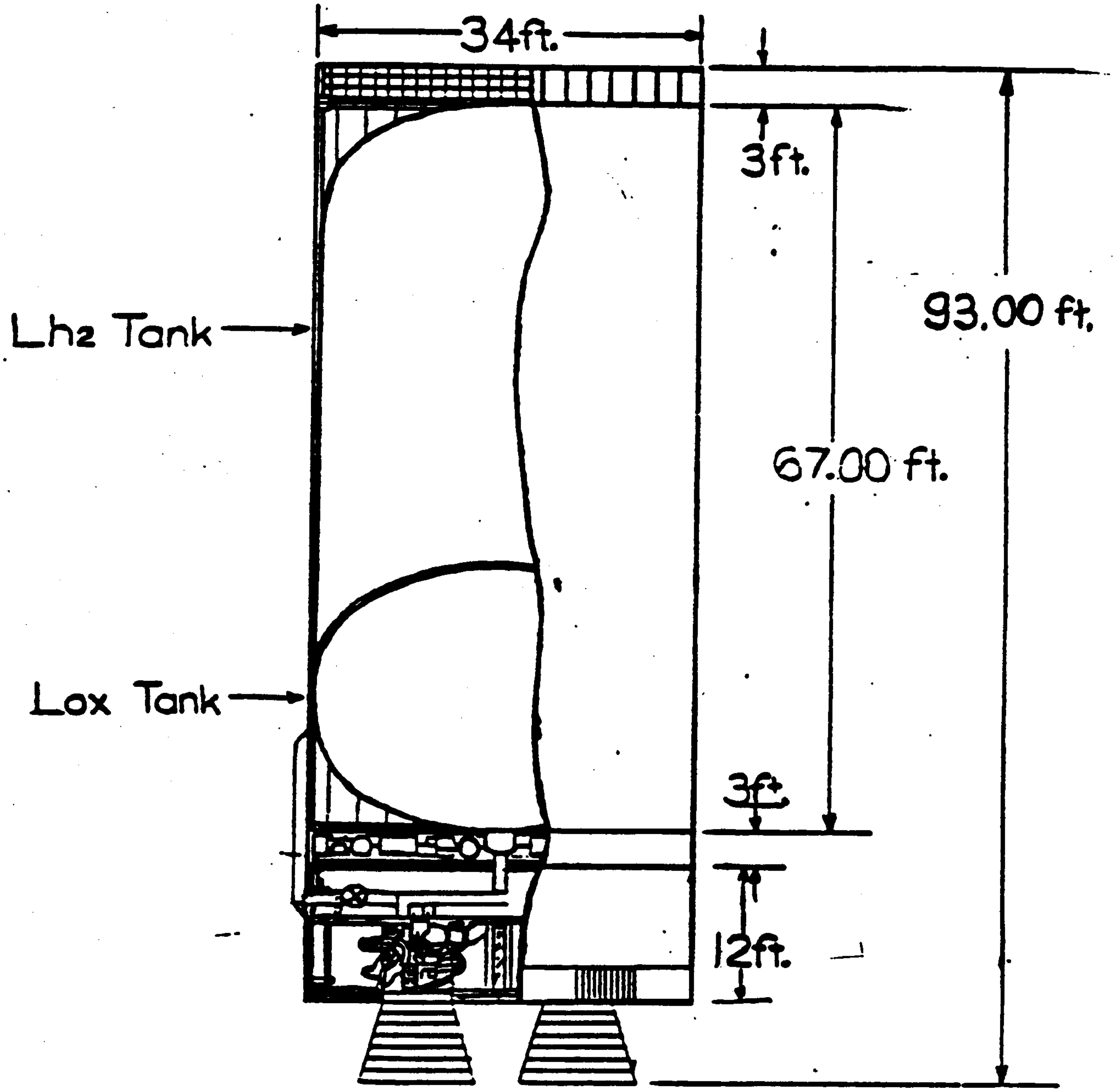


Fig. 7. Cutaway of Lower Section and Interstage

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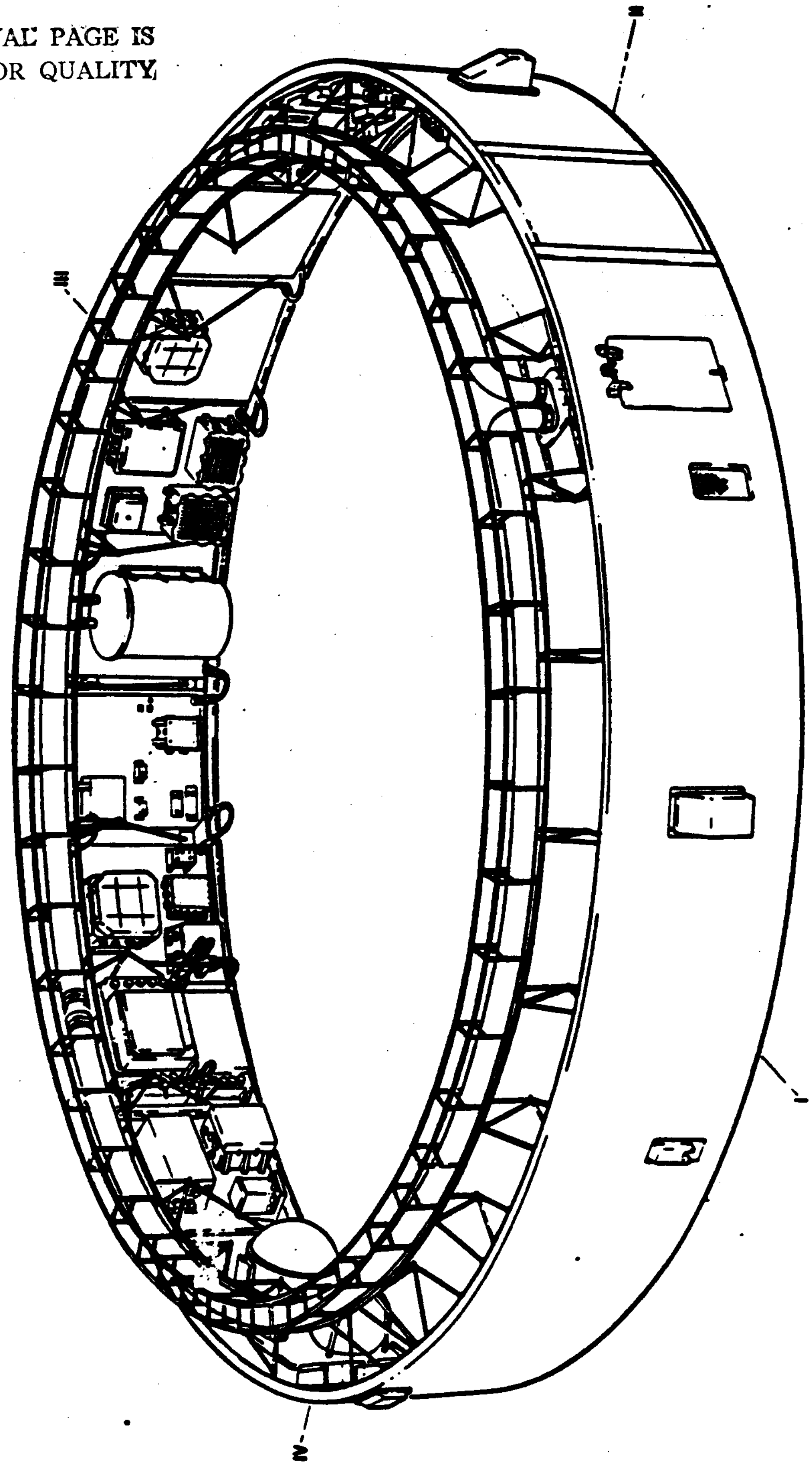


FIGURE 8. INSTRUMENT UNIT.

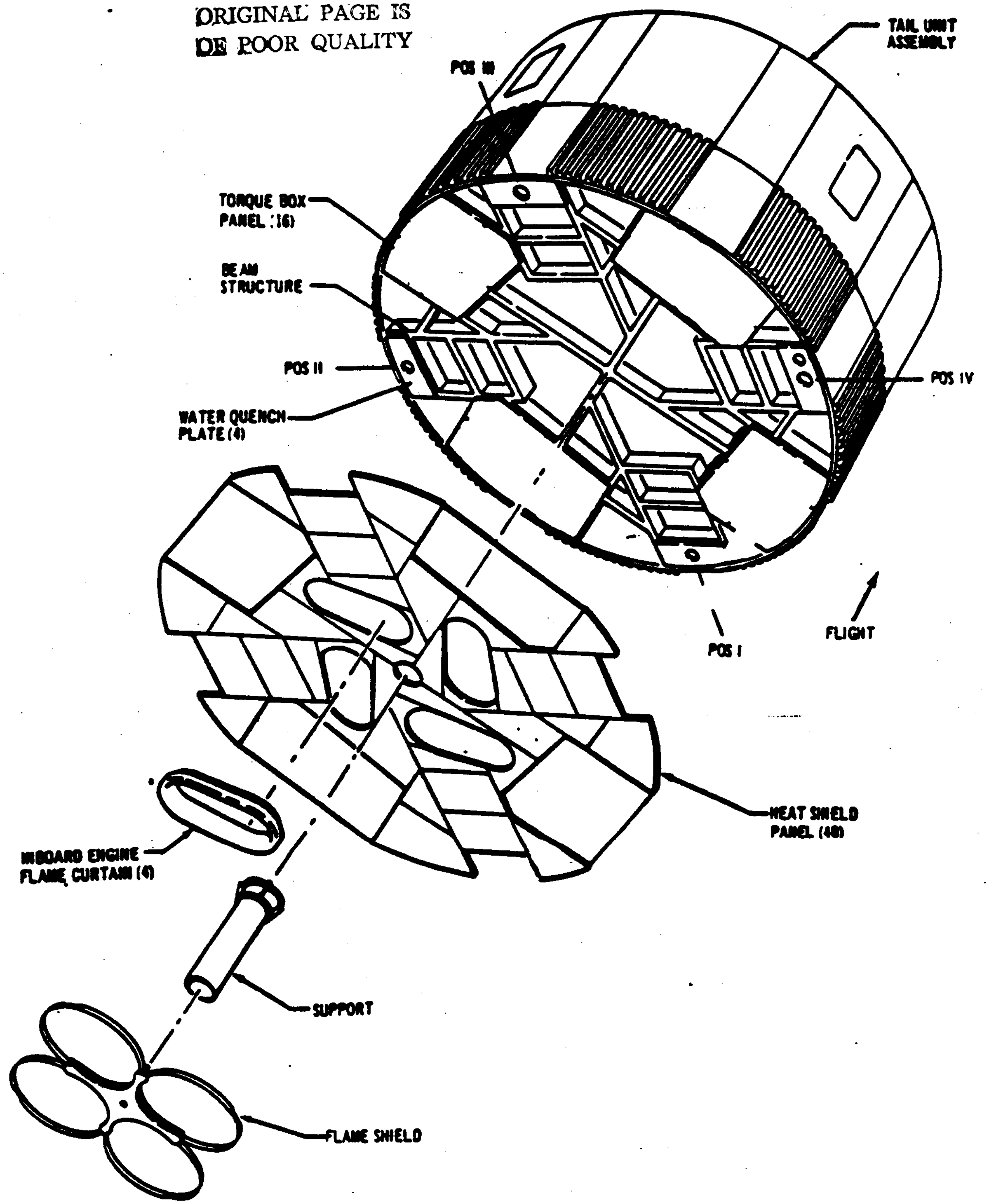


FIGURE 9. FLAME AND HEAT PROTECTION

SPACE TRANSPORTATION MAIN ENGINE (STME)

	STME
● PROPELLANTS	LOX/LH2
● NOZZLE AREA RATIO (STOWED/EXTENDED)	55/150
● VACUUM THRUST (LBF)	468K/481K
● VACUUM ISP (SEC)	449/461
● CHAMBER PRESSURE (PSIA)	3006
● MIXTURE RATIO (O/F)	6.0
● LENGTH (IN)	139/219
● NOZZLE EXIT DIAMETER (IN)	76.2/126.3
● ENGINE INSTALLED WT (LBM)	7142
● SEA LEVEL THRUST (LBF) (STOWED)	397K
● SEA LEVEL ISP (SEC)	380.4
● FLOWRATE (LB/SEC)	1043.4

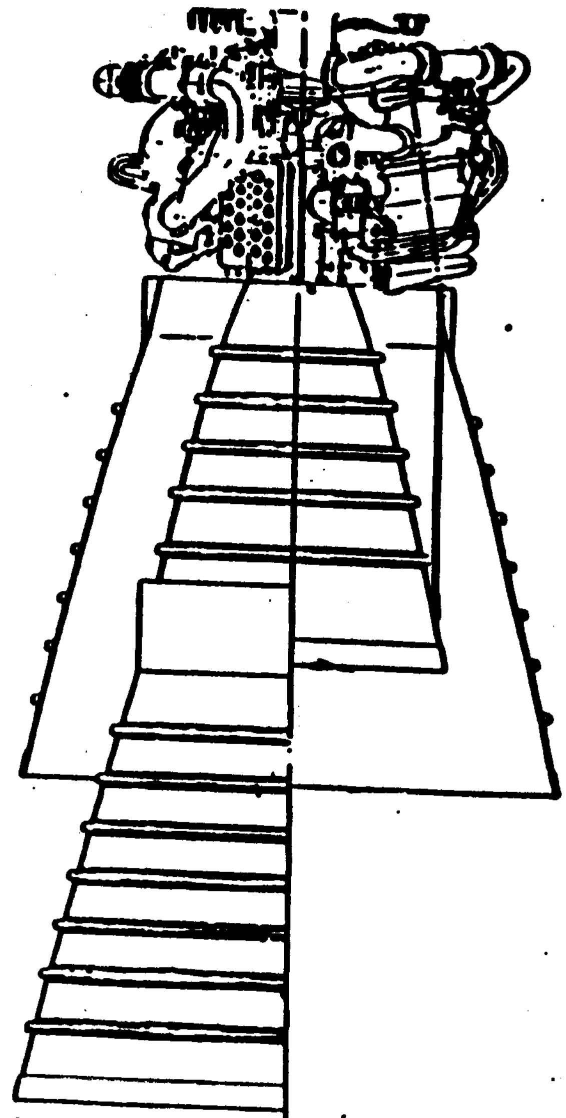
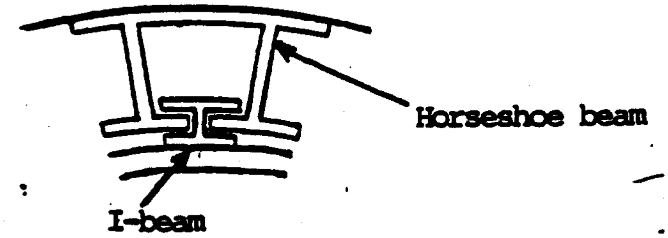


Figure 10. Space Transportation System Main Engine



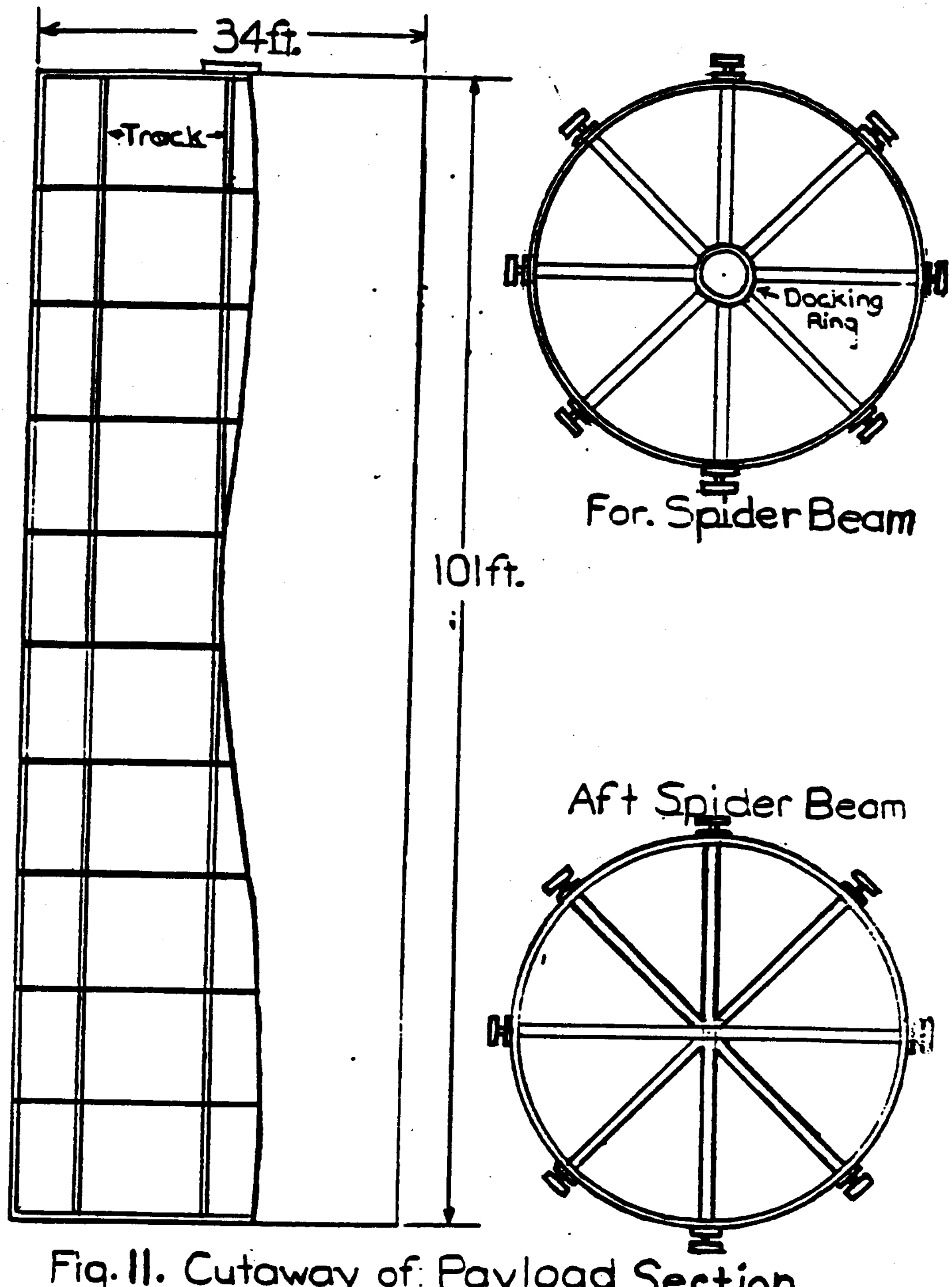


Fig. 11. Cutaway of Payload Section
With Payload Frame

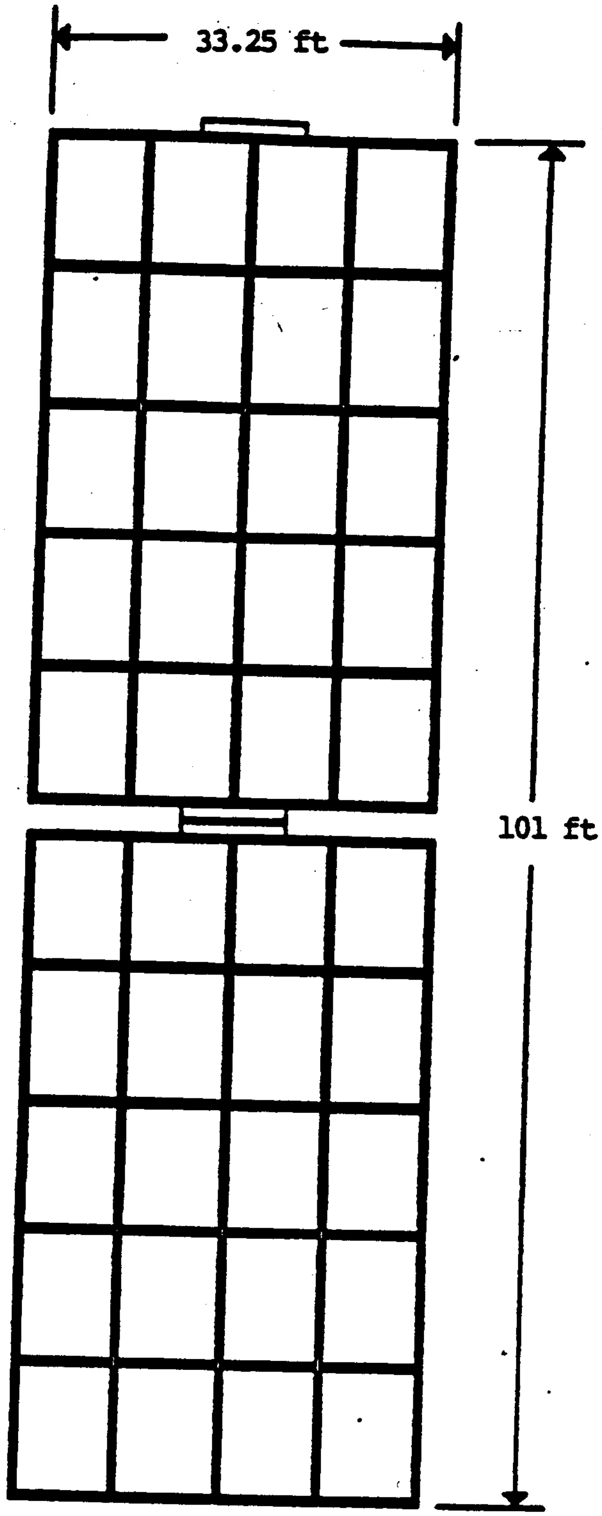


Figure 12. Two payload frames in docked configuration.

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APPENDIX A

TRAJECTORY ANALYSIS DATA

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TRAJECTORY ANALYSIS PROGRAM

"CTRAJ"

This is a general purpose trajectory analysis program utilizing the gravity of the Earth to achieve the proper attitude for insertion into a low Earth orbit. This maneuver is called a gravity assisted turn maneuver. Since this program was used to model the trajectory for an unmanned cargo vehicle, the forces due to high acceleration were of secondary importance, and not limited. The cargo vehicle was used in conjunction with a reuseable flyback booster which returns to a landing strip after staging from the cargo vehicle. The program is set up so that staging does not occur until the total configuration is accelerated to 7000 feet per second. As another limit the program ceases execution if all of the mass is burned as fuel for obvious reasons.

```
15    ALT=0
20    DT=0.1
30    RANG=0
40    ISP1=320
50    ISP2=380
60    INPUT"ENTER INITIAL MASS";M1
65    M=M1
70    INPUT"ENTER ALTITUDE FOR G-TURN";AGT1
75    AGT=AGT1
80    TH=1.35*M
90    INPUT"ENTER LIFTOFF THRUST (CARGO ENGINES)";TH21
95    TH2=TH21
100   TH1=TH-TH2
110   MDOT1=TH1/ISP1
120   MDOT2=TH2/ISP2
130   ISP=TH/(MDOT1+MDOT2)
140   MDOT=TH/ISP
150   MC1=0
160   MC2=0
170   Y=0
180   X=0
190   G0=32.2
200   R0=2.092E+07
210   V0=0
220   V=V0
230   T=0
240   PRINT
250   PRINT"TIME       V         RANG        ALT        G'S      MASS       THRUST
260   PRINT
270   F$="###.#     #####.#    ######     ######     #.##    #######    ######
280   M2=M-MDOT1*DT-MDOT2*DT
290   G=G0*(R0/(R0+ALT))^2
```

```
300  DV=ISP*GO*LOG(M/M2)-G*DT
310  AG1=DV/GO/DT
320  REM IF AG1>3 THEN 400
330  CHY=0.5*(VO+VO+DV)*DT
340  IF Y+CHY>AGT THEN 580
350  Y=Y+CHY
360  ALT=Y
370  V=VO+DV
380  AG=AG1
390  GOTO 460
400  DV=3*GO
410  M2=M/EXP((DV+G*DT)/ISP/GO)
420  A=DV/GO
430  V=VO+DV
440  Y=Y+0.5*(VO+V)*DT
450  ALT=Y
460  T=T+DT
470  DM=M-M2
480  MDT=DM/DT
490  TH=MDT*ISP
500  MC1=MC1+TH1/ISP*DT
510  MC2=MC2+(TH-TH1)/ISP2*DT
520  MC=MC1+MC2
530  PRINT USING F$;T,V,RANG,ALT,AG,M,TH
540  REM
550  M=M2
560  VO=V
570  GOTO 280
580  PRINT
590  PRINT"TIME      V        RANG      ALT       G'S       MASS      THRUST"
600  PRINT"-----------------------------------------------------------------"
620  M=M+MDOT1*DT+MDOT2*DT
630  DPSI=(PI/180)*0.1
640  PSIO=PI/120
650  PSI=PSIO
660  GAM=1
670  ZO=SIN(PSIO)/(1+COS(PSIO))
680  PRINT"TIME      V       RANG       ALT      PSI   GAM    THRUST    MASS    G'S
700  F$="###  #####   #######  #######    ##.#    ##.#  #######  #######  #.#
710  PRINT
720  N=TH/M
730  PRINT USING F$;T,V,RANG,ALT,PSI*180/PI,GAM,TH,M,AG
740  C=VO/(ZO^(N-1))/(1+ZO^2)
750  PSI=PSI+DPSI
760  Z=SIN(PSI)/(1+COS(PSI))
770  V1=C*(Z^(N-1))*(1+Z^2)
780  IF V1>7000 THEN 1390
785  V=V1
790  DT= C/G*Z^(N-1)*(1/(N-1)+Z^2/(N+1))-C/G*ZO^(N-1)*(1/(N-1)+ZO^2/(N+1))
800  DX=0.5*(VO*SIN(PSIO)+V*SIN(PSI))*DT
810  DY=0.5*(VO*COS(PSIO)+V*COS(PSI))*DT
820  AG=(V-VO)/DT/GO
830  IF AG>3 THEN 1060
```


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```
840     X=X+DX
845     Y=Y+DY
850     THETA=ATN(X/(RO+Y))
860     GAMMA=PSI-THETA
870     GAM=GAMMA*180/PI
880     ALT=(Y+RO)/COS(THETA)-RO
900     RANG=RO*THETA
930     T=T+DT
940     PSIO=PSI
950     ZO=Z
960     VO=V
970     IF ALT<100000 THEN 1020
980     ISP1=340
990     ISP2=462
1000    MDOT1=TH1/ISP1
1010    MDOT2=TH2/ISP2
1020    M=M-MDOT1*DT-MDOT2*DT
1030    IF M<0 THEN 1390
1040    G=GO*(RO/(RO+ALT))^2
1050    GOTO 720
1060    N=3*GO/G+((1-ZO^2)/(1+ZO^2))
1070    C=VO/ZO^(N-1)/(1+ZO^2)
1080    V=C*Z^(N-1)*(1+Z^2)
1090    D1=C/G*Z^(N-1)*(1/(N-1)+Z^2/(N+1))-C/G*ZO^(N-1)*(1/(N-1)+ZO^2/(N+1))
1100    DX=0.5*(VO*SIN(PSIO)+V*SIN(PSI))*DT
1110    DY=0.5*(VO*COS(PSIO)+V*COS(PSI))*DT
1120    X=X+DX
1130    RANG=RO*THETA
1140    Y=Y+DY
1150    ALT=(Y+RO)/COS(THETA)-RO
1160    T=T+DT
1170    PSIO=PSI
1180    ZO=Z
1190    VO=V
1200    TH=N*M
1210    TH1=TH-TH2
1220    MDOT1=TH1/ISP1
1230    MDOT2=TH2/ISP2
1240    M=M-MDOT1*DT-MDOT2*DT
1250    MC1=MC1+MDOT1*DT
1260    MC2=MC2+MDOT2*DT
1270    MC=MC1+MC2
1280    IF M<0 THEN 1390
1290    G=GO*(RO/(RO+ALT))^2
1300    PSI=PSI+DPSI
1310    THETA=ATN(X/(RO+Y))
1320    GAMMA=PSI-THETA
1330    GAM=GAMMA*180/PI
1350    PRINT USING F$;T,V,RANG,ALT,PSI,GAM,TH,M,AG
1360    IF V>7000 THEN 1390
1370    Z=SIN(PSI)/(1+COS(PSI))
1380    GOTO 1060
1390    PRINT
```

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```
1391    PSI=PSI-DPSI
1392    PRINT V1
1400    PRINT"TIME     V     RANG    ALT    PSI    GAM    THRUST    MASS    G'S"
1420    PRINT"------------------------------------------------------------------"
1421    PRINT
1422    PRINT"                   S E P A R A T I O N             "
1423    PRINT
1424    PRINT"------------------------------------------------------------------"
1430    PRINT"time     v     rang    alt    psi    gam2    thrust    mass     g's"
1431    PRINT
1432    F$="###  #####  ####### ####### ###.# ###.#  #######  #######  #.##"
1435    M=M-1020000
1440    ISP=ISP2
1450    TH=TH2
1460    MDOT=TH/ISP
1470    VO=V
1480    PSIO=PSI
1490    GAM2=PI/2 + THETA
1500    GAMM=GAM2*180./PI
1510    ZO=SIN(PSIO)/(1+COS(PSIO))
1520    N=TH/M
1530    PRINT USING F$;T,V,RANG,ALT,PSI*180./PI,GAMM,TH,M,AG
1540    C=VO/(ZO^(N-1))/(1+ZO^2)
1550    PSI=PSI+DPSI
1560    THETA=ATN(X/(RO+Y))
1570    GAM2=PI/2+THETA
1580    GAMM=GAM2*180./PI
1590    IF PSI>GAM2 THEN 2080
1600    Z=SIN(PSI)/(1+COS(PSI))
1610    V=C*(Z^(N-1))*(1+Z^2)
1620    IF V>25000 THEN 2080
1630    DT=C/G*Z^(N-1)*(1/(N-1)+Z^2/(N+1))-C/G*ZO^(N-1)*(1/(N-1)+ZO^2/(N+1))
1640    DX=.5*(VO*SIN(PSIO)+V*SIN(PSI))*DT
1650    DY=.5*(VO*COS(PSIO)+V*COS(PSI))*DT
1660    AG=(V-VO)/DT/GO
1680    X=X+DX
1700    Y=Y+DY
1701    THETA=ATN(X/(RO+Y))
1702    RANG=RO*THETA
1710    ALT=(Y+RO)/COS(THETA)-RO
1720    T=T+DT
1730    PSIO=PSI
1740    ZO=Z
1750    VO=V
1760    M=M-MDOT*DT
1770    IF M<0 THEN 2080
1780    G=GO*(RO/(RO+ALT))^2
1790    GOTO 1520
2080    REM
2081    PRINT
2090    PRINT"INITIAL MASS       :";M1
2100    PRINT"ALT FOR G-TURN     :";AGT1
2110    PRINT"THRUST (CARGO VEHICLE) :";TH21
```

```
2111    PRINT"FINAL VELOCITY :";V
2112    PRINT"FINAL MASS :";M
2113    PRINT"FINAL PSI  :";PSI*180/PI
2114    PRINT"FINAL ALTITUDE :";ALT/6080
2115    PRINT"FINAL GAMM :";GAMM
2116    INPUT"TRY AGAIN Y=1 N=2";TRY
2117    IF TRY<2 THEN 15
```

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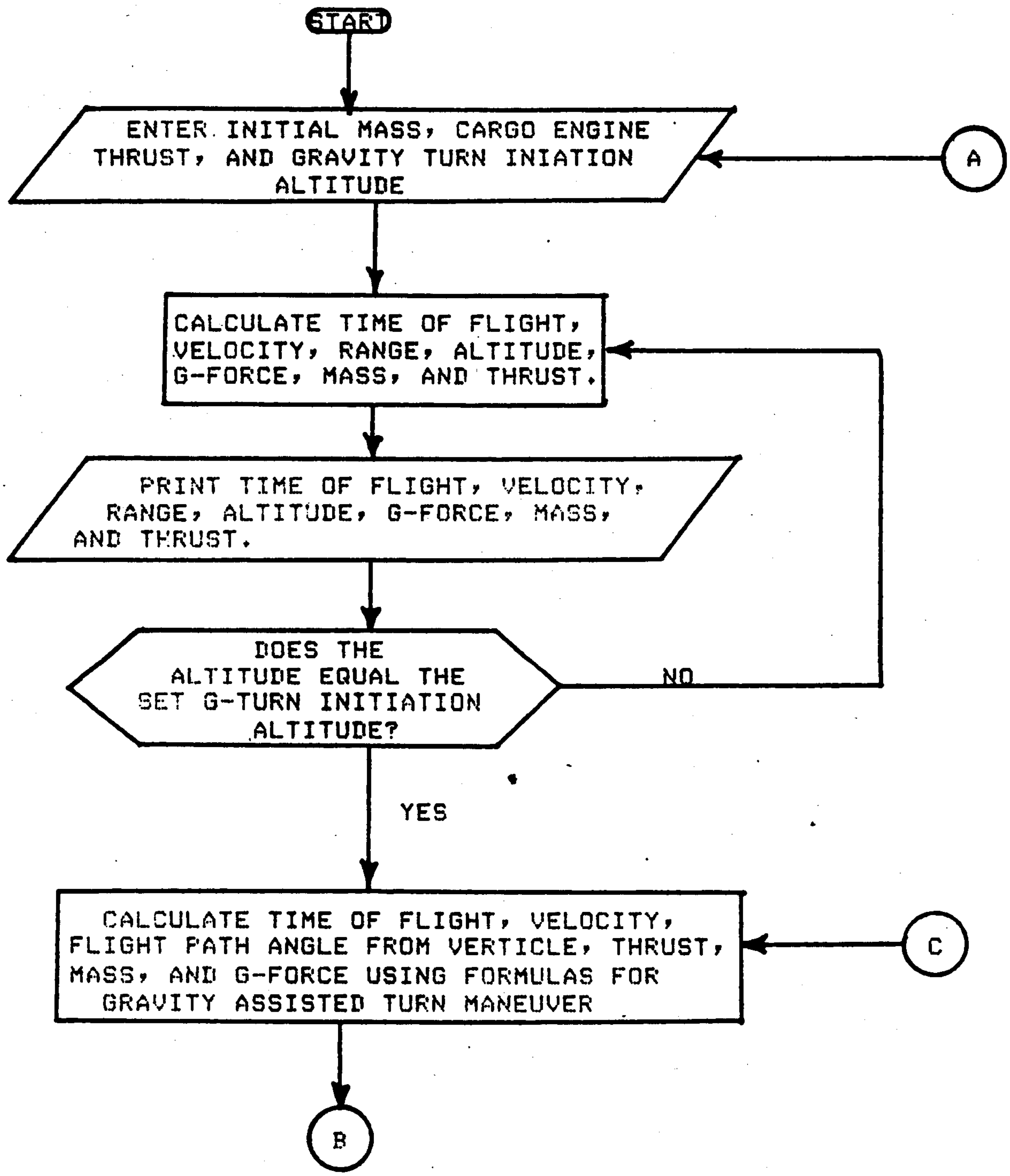
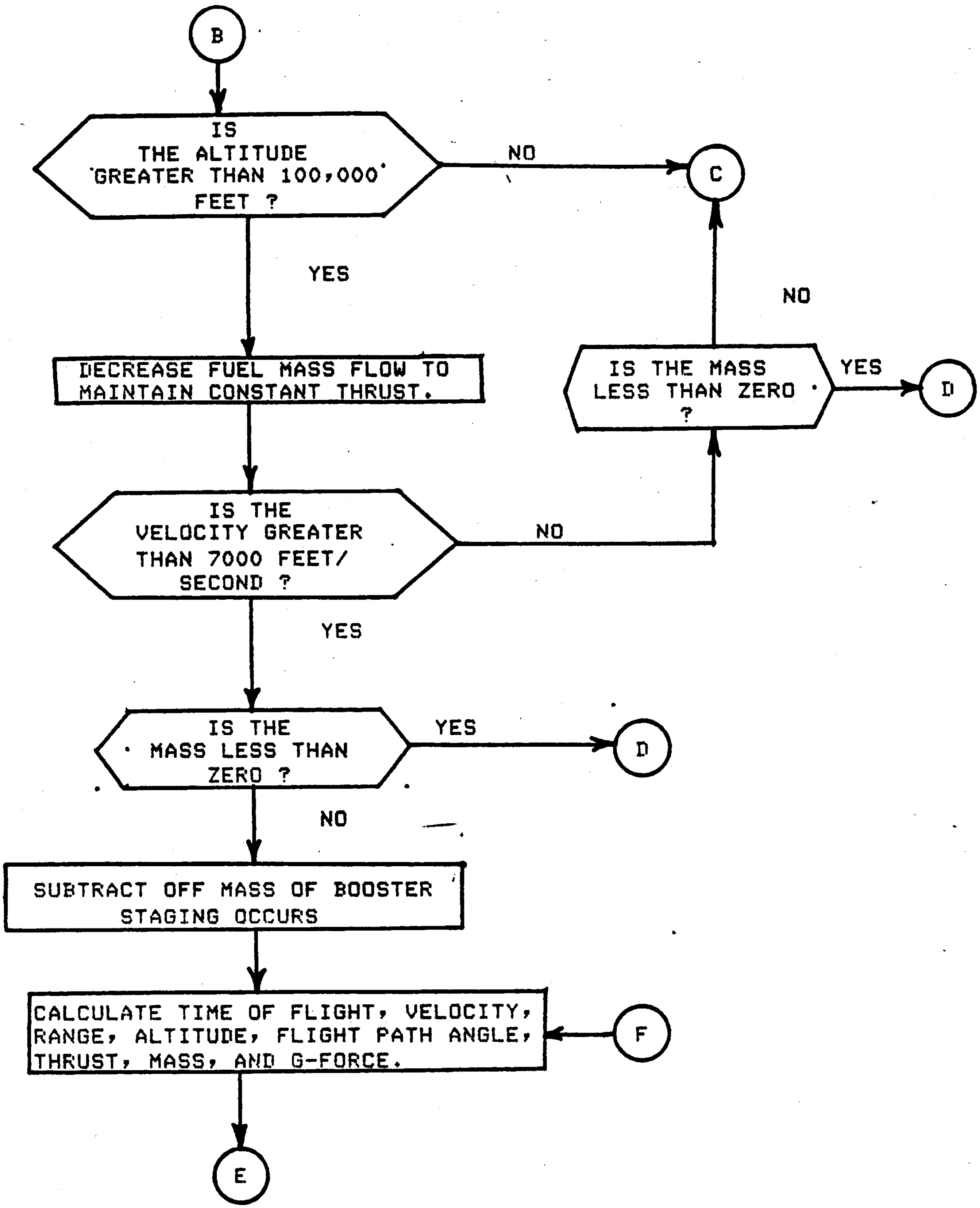
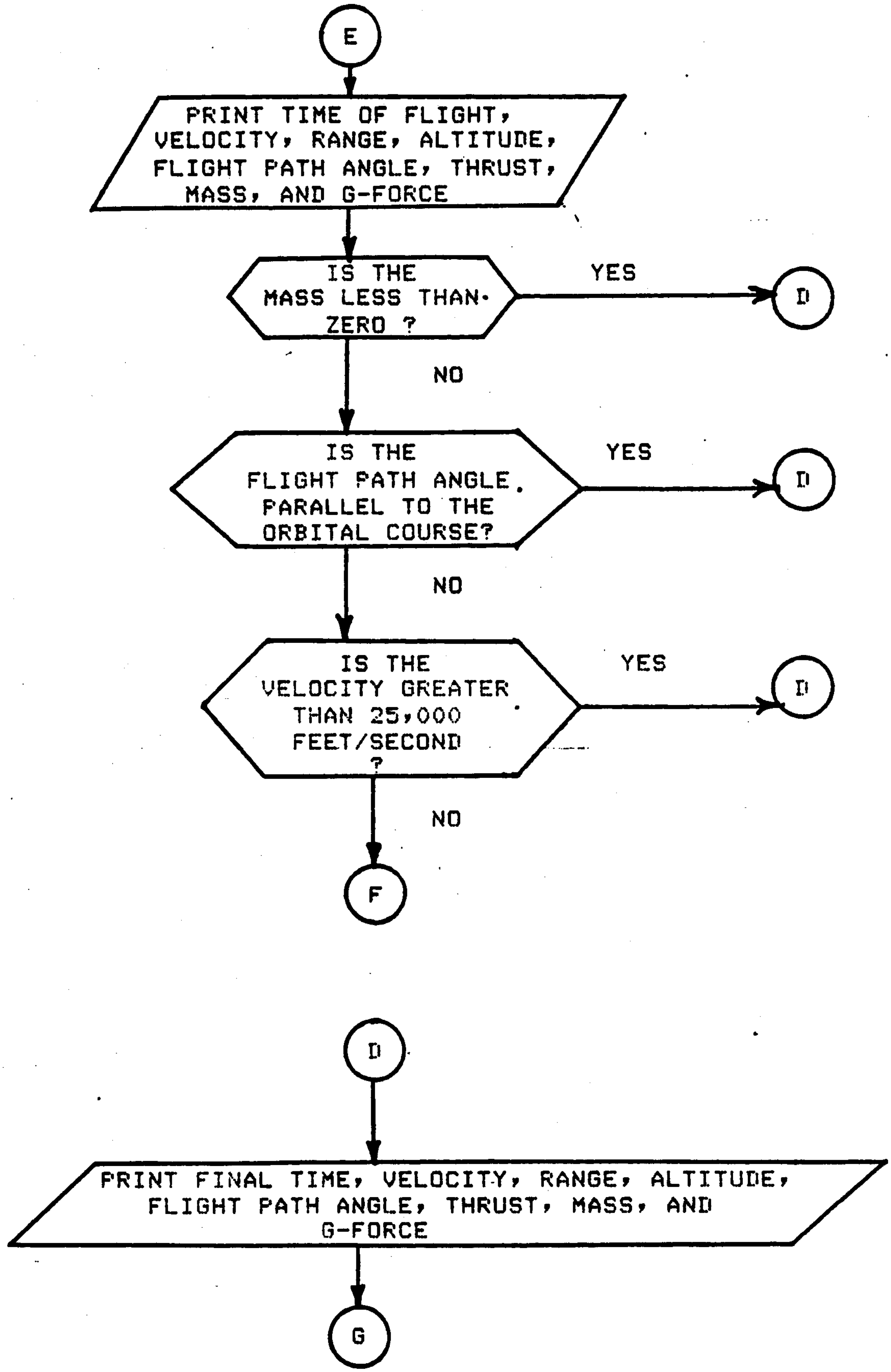
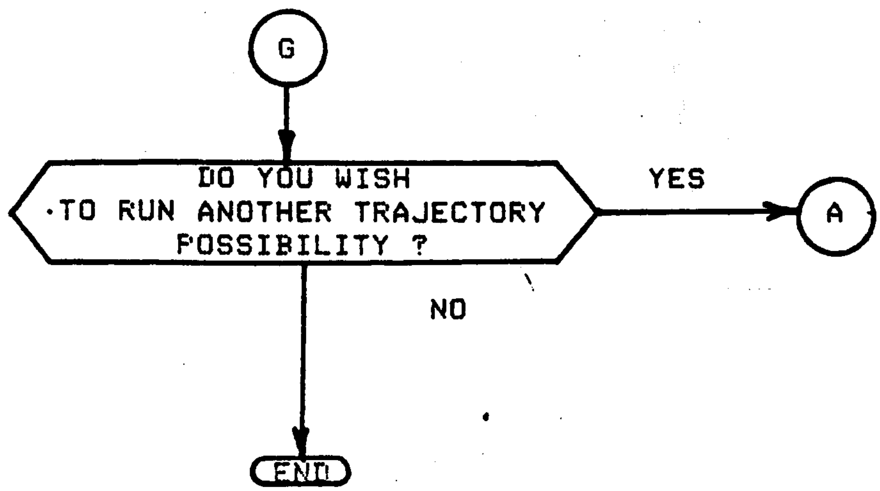
FLOWCHART FOR TRAJECTORY ANALYSIS

PROGRAM "CTRAJ"




B
IS THE ALTITUDE GREATER THAN 100,000 FEET ?
NO
C
YES
NO
DECREASE FUEL MASS FLOW TO MAINTAIN CONSTANT THRUST.
IS THE MASS LESS THAN ZERO ?
YES
D
IS THE VELOCITY GREATER THAN 7000 FEET/ SECOND ?
NO
YES
IS THE MASS LESS THAN ZERO ?
YES
D
NO
SUBTRACT OFF MASS OF BOOSTER STAGING OCCURS
CALCULATE TIME OF FLIGHT, VELOCITY, RANGE, ALTITUDE, FLIGHT PATH ANGLE, THRUST, MASS, AND G-FORCE.
F
E


E
PRINT TIME OF FLIGHT, VELOCITY, RANGE, ALTITUDE, FLIGHT PATH ANGLE, THRUST, MASS, AND G-FORCE
IS THE MASS LESS THAN ZERO ?
YES
D
NO
IS THE FLIGHT PATH ANGLE PARALLEL TO THE ORBITAL COURSE?
YES
D
NO
IS THE VELOCITY GREATER THAN 25,000 FEET/SECOND ?
YES
D
NO
F
D
PRINT FINAL TIME, VELOCITY, RANGE, ALTITUDE, FLIGHT PATH ANGLE, THRUST, MASS, AND G-FORCE
G


G
DO YOU WISH
TO RUN ANOTHER TRAJECTORY
POSSIBILITY ?
YES
A
NO
END

APPENDIX B

FUEL TANK DATA CALCULATIONS

CALCULATIONS FOR FUEL TANK DATA

$M_{fuel} = 1,108,133$ lbs

$M_{payload} = 150,000$ lbs

$M_{structure} = 233,247$ lbs

$M_{O_2} + M_{H_2} = M_{fuel}$

$M_{O_2}/M_{H_2} = 6$

$M_{H_2} = (1,108,133)/(7) = 158,304.71$ lbs $\rightarrow M_{O_2} = 949,828.29$ lbs

$\rho_{O_2} = 71.2$ lbs/ft^3 $\quad\quad \rho_{H_2} = 4.43$ lbs/ft^3

$V_{H_2} = M_{H_2}/\ O_2 = 35,724.698\ ft^3$

$V_{O_2} = 13,340.285\ ft^3$

OXYGEN TANK: Ellipsoid (similar to two saucers placed convex to one another)

$V = \frac{4}{3}\pi abc = \frac{4}{3}\pi a^2 b$ because $a = c$

$b = 11.70\ ft \longrightarrow h = 2b = 23.40\ ft$

HYDROGEN TANK: Cylinder with ellipsoid convex upper cap and ellipsoid concave base.

$V = \pi R^2 h = \pi a^2 h$

$h = 41.78\ ft$

TOTAL HEIGHT OF TWO TANKS IN TANDEM:

$h_{total} = 23.40\ ft + 41.78\ ft = 65.18\ ft$

APPENDIX C

C.G. LOCATIONS

C.G. Location (Dry)

Component	Weight (lb)	Location (ft)
Engines	28568	5
Engine Compartment	15250	6
Instrument Unit	6000	13.5
Fuel Section	60095	37.59
Interstage	6000	81.68
Upper Stage	105334	133.68
Payload	150000	133.68
Nose Cone	12000	194.18

Total C.G. Location (Dry) = 104.71 feet

C.G. Location (Wet)

Engines	28568	5
Engine Compartment	15250	6
Instrument Unit	6000	13.5
Liquid Oxygen	949828	26.7
Fuel Section	60095	37.59
Liquid Hydrogen	158304	59.29
Interstage	6000	81.68
Upper Stage	105334	133.68
Payload	150000	133.68
Nose Cone	12000	194.18

Total C.G. Location (Wet) = 50.21 feet