NASW-4435

IN 15-CR
73936
P- 275

CONCEPTUAL DESIGN OF

TWO-STAGE-TO-ORBIT

HYBRID LAUNCH VEHICLE

CASE WESTERN RESERVE UNIVERSITY

July 1, 1991

(NASA-CR-190006) CONCEPTUAL DESIGN OF TWO-STAGE-TO-ORBIT HYBRID LAUNCH VEHICLE (Case Western Reserve Univ.) 275 p CSCL 22B

N92-20666

Unclas

G3/15 0073936

ABSTRACT

The object of this design class was to design an earth-to-orbit vehicle to replace the present NASA space shuttle. The major motivations for designing a new vehicle were to reduce the cost of putting payloads into orbit and to design a vehicle that could better service the space station with a faster turn-around time. Another factor considered in the design was that near-term technology was to be used. Materials, engines, and other important technologies were to be realized in the next 10 to 15 years. The first concept put forth by NASA to meet these objectives was the NASP. The NASP is a single-stage earth-to-orbit air-breathing vehicle. This concept ran into problems with the air-breathing engine providing enough thrust in the upper atmosphere, among other things.

The solution of this design class is a two-stage-to-orbit vehicle. The first stage is air-breathing and the second stage is rocket-powered, similar to the space shuttle. The second stage is mounted on the top of the first stage in a piggy-back style. The vehicle takes off horizontally using only air-breathing engines, flies to Mach 6 at 100,000 feet, and launches the second stage towards its orbital path. The first stage, or booster, will weigh approximately 800,000 pounds and the second stage, or orbiter, will weigh approximately 300,000 pounds.

The major advantage of this design is the full recoverability of the first stage compared with the present solid rocket boosters that are only partially recoverable and used only a few times. This reduces the cost as well as provides a more reliable and more readily available design for servicing the space station. The booster can fly an orbiter up, turn around, land, refuel, and be ready to launch another orbiter in a matter of hours.

INTRODUCTION

There were several design concepts to choose from for improving the present space shuttle. The concepts ranged from two-stage vertical take-off vehicles similar to the space shuttle to three-stage concepts. The design class decided early upon a two-stage concept. The choices left were air-breathing or rocket for each stage. The design class decided on air-breathing engines for the first stage and rocket-powered engines for the second stage for several reasons. First, the concept was "new" or at least different from the present shuttle which was the object to replace. Second, the NASP already proved that using air-breathing engines in the upper atmosphere is not feasible in the next 10 to 15 years. Third, with air-breathing engines used in the first stage, the booster would basically be an airplane that is fully recoverable and has a relatively fast turn-around time.

After the choice of an air-breathing booster and a rocket orbiter was made, the next problem that was addressed was where to mount the orbiter on the booster. An interesting concept arose from the design group to mount the orbiter on the front of the booster in a conformal manner. One advantage of this concept perceived at the time was that the orbiter, rather than being "dead weight", could actually provide some lift from its wings during the initial ascent. Another advantage perceived was that the bottom surface of the orbiter could be used as part of the pressure recovery surface for the air-breathing engines on the booster. One last advantage that the group considered was in the area of aerodynamic heating. It was thought that if the orbiter was mounted on top of the booster that the small gap between them would cause an accelerated flow field that would lead to stagnation temperatures beyond the limit of the materials. This in turn would possibly lead to active cooling, something the design group wished to avoid for overall weight and complexity reasons. With the orbiter on the front, this accelerated flow field would be avoided and the top of the booster would be able to radiate heat to the atmosphere, reducing the need for active cooling.

The design group spent the first few weeks of the semester pursuing the front-mounted concept. Some of the proposed advantages were realized, but a few problems began to arise as well. The two major problems were in the stability of the entire vehicle while in flight and in actually taking off with this configuration. With the orbiter at the front, the center of gravity of the vehicle was too far forward which created stability problems. The vehicle would also have a rather awkward configuration in order for the wings of both the orbiter and booster to achieve the proper angle of attack required for take-off. After several weeks of trying to work with this concept, the problems began to outweigh the advantages and the idea was abandoned.

The second configuration that was pursued is the present one. The orbiter is mounted on the top of the booster. This design is much more stable, it is structurally easier on the booster, and the separation of the two stages is less complicated.

The remainder of this report focuses on the design of both stages. Each stage was divided into the following three areas: Aerodynamics, Propulsion, and Structure. Presented are the detailed reports on the design of each of these areas of the booster followed by the detailed reports on the design of the orbiter. Detailed calculations for each area can be found at the conclusion of each section.

CONCEPTUAL DESIGN OF TWO-STAGE-TO-ORBIT HYBRID LAUNCH VEHICLE

BOOSTER DESIGN

July 1, 1991

Aaron Durkee
Project Manager

AERODYNAMIC GROUP

Rick Siembor
Allan van de Wall
Richard van de Wall

AERODYNAMICS

CONFIGURATION DEVELOPMENT

Two configurations of the two-stage-to-orbit vehicle were proposed for further research. Of the two configurations, referenced as proposal number one and proposal number 2 in the appendix, proposal number 2 was chosen for the final conceptual design process. With the overall configuration chosen, an initial sketch could be made of the booster vehicle. In the appendix to this document, the total booster configuration development sketches can be found.

The initial booster sketch (appendix A) was completed early on in the conceptual design phase. The initial sketch incorporated a tailless double delta wing. The wing sweep was 75 degrees down to 45 degrees. The overall length and wingspan was determined to be 263 feet and 150 feet respectively. The purpose of this preliminary sketch was to determine to overall feasibility of a top mount orbiter configuration. No technical details were determined from this initial sketch.

The initial booster sketch was revised (appendix A) to obtain some preliminary aerodynamic calculations. The booster was configured with an initial wing loading of 100 at takeoff. At this point in the design, the gross takeoff weight was still determined to be one million pounds. The wing was still in a double delta configuration, however, the sweep of the main wing was now 60 degrees. The sweep of the wing was increased as far as possible up to the maximum defined for structural problems (ref. 1). The initial sweep of the double delta wing became a leading edge extension or chine with a sweep of 79 degrees. The leading edge extension was added in order to produce a pitch up moment for stability to help compensate for the rearward shift of the aerodynamic center as the vehicle passes from subsonic to supersonic flight. This increase in pitching moment, would decrease the requirement for wing elevons to produce a down force in order to maintain stability since a tailless delta was chosen. The overall lift of the wing would increase and the trim drag produced by the elevons would decrease. Also, the LEX would reduce the flow field through a series of shocks and could possibly establish the wing inside a Mach cone. The side of the fuselage of the airplane could resemble a 2-D ramp.

The aerodynamic configuration of a tailless delta was chosen because of the lower overall drag of this type of configuration at high Mach numbers. The fuselage of the vehicle had a circular crossection which was 50 feet in diameter in the middle of the airplane and tapered down to 40 feet in diameter at the rear. The nose of the airplane at this stage in the design was assumed to be a point. Heating, however, was being examined to determine the correct nose radius. The fineness ratio of the fuselage was determined to be 6. According to ref. 1, the fineness ratio for minimum supersonic drag is 14. Since the aircraft uses liquid hydrogen as a propellent, which is very low density, the fuselage diameter was increased from the minimum drag value in order to facilitate the required amount of hydrogen. The required amount of hydrogen was based on the existing data from the beta project (ref. 4) scaled down to the value for this design.

The booster was again revised to obtain more detail on the aerodynamic configuration (appendix A) and to accommodate an increased gross takeoff weight. The combined gross takeoff weight of the booster vehicle with the shuttle orbiter on top was increased from 1,000,000 lbs, to 1,300,000 lbs. This weight increase was done in order to accommodate the increase in shuttle gross liftoff weight of 300,000 lbs. This combined with the estimate for fuel needed of 400,000 lbs resulted in a combined payload of over 700,000 lbs. With an empty weight fraction of at least 0.43 from initial sizing estimates, the empty weight of the booster vehicle had to increase. In order to facilitate the larger weight, a larger wing was required. The wing loading was still assumed to be approximately 100. A wing loading of 100 is considered to be high for tailless delta configurations. However, the need to increase the wing area to a wing loading of less than 100 was not desired. At high speed, a large wing area is not needed due to the increase in dynamic pressure. The dynamic pressure is more effective in increasing the lift coefficient than the surface area of the wing, since the lift coefficient varies inversely as the square of the velocity. Therefore, a smaller lift coefficient would be needed at high speed than at low speed. The most critical link for the wing loading was at takeoff. Adequate takeoff performance was desired with no assistance from any auxiliary devices. Therefore, all moving tailplanes were added to the booster to give enough moment for takeoff rotation.

The all moving tailplanes were considered key to the design of the booster vehicle. The tailplanes were positioned on the tips of the wings. Between the wing tip and the tailplane, the vertical stabilizer with a hinged rudder was placed. It allowed a lower wing loading for takeoff than conventional tailless deltas and the flow field around the main wing will be less disturbed. The effective aspect ratio of the wing at low speeds would also increase due to the fact that the tailplanes would be producing negative lift with high pressure at the top and low pressure at the bottom. This would reduce the effect of the wingtip vorticies and therefore decrease induced drag and increase wing lift. It is apparent that this type of configuration would have to examined further as to the effectiveness of this tailplane.

The vertical stabilizer was placed between the wing and tailplane for two reasons. First, it removed the them from the centerline of the fuselage where the shuttle is positioned, thereby reducing the possibility of a shuttle separation accident. Secondly, it allowed the vertical stabilizer to be used as an endplate reducing induced drag. A all moving vertical stabilizer such as on the Lockheed SR-71 was not considered necessary as the engines were essentially of the centerline thrust type and engine out conditions were not determined to be critical.

The leading edge extension was increased to the nose of the airplane in a chine type fashion to provide more pitch-up at high speed and therefore reduce the dependency on the tailplanes at the higher Mach numbers. The inboard elevons, are envisioned to be used to provide roll and pitch control at high speed, and therefore, decreased any aileron reversal tendency. The nose radius of 1 ft was determined to give adequate cooling capacity without active cooling.

Fowler flaps were added to the booster to increase the surface area of the wing, decreasing takeoff speed and takeoff distance (appendix A). The horizontal all moving tailplanes were increased in size to create the large moment needed to rotate the aircraft around the main landing gear to achieve the required angle of attack for liftoff. The surface area of each tailplane was determined to be 630 square feet. This presented a structural problem of supporting the load at the tip of the wing. However, the maximum load that the tailplanes would see would be at the point of rotation. The sweep of the tailplane was 45 degrees. The tailplanes would not be used for control at high speeds, therefore, the wing took on an overall cranked arrow configuration resulting in better efficiency.

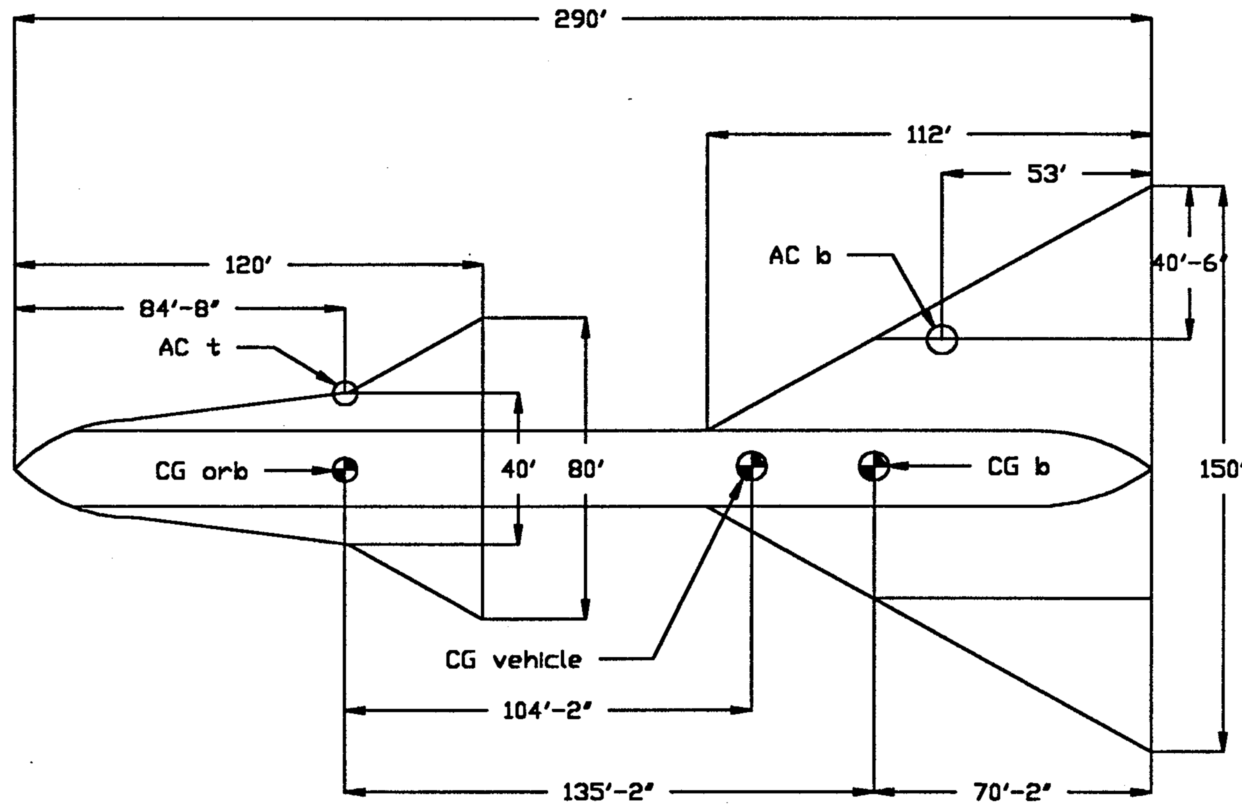
To increase the effectiveness of the tailplanes to a degree further than the variable incidence can, a hinged flap was added to the tailplane. The flap would move in the same direction as the tailplane itself, but with increased travel. This would also contribute to an increase in longitudinal stability.

After having determined that the aircraft's fuselage could hold the required amount of fuel, the fuselage sides were "coke- bottled" (ref. 1) to help reduce wave drag (appendix A). An extension was added to the rear of the aircraft to simulate an unknown nozzle design. The booster's length was increased, primarily in fuselage length, in order to accommodate the required fuel while not allowing the fuselage diameter to increase. The leading edge extensions were not allowed to extend to the nose of the aircraft. This was done to simplify structural considerations.

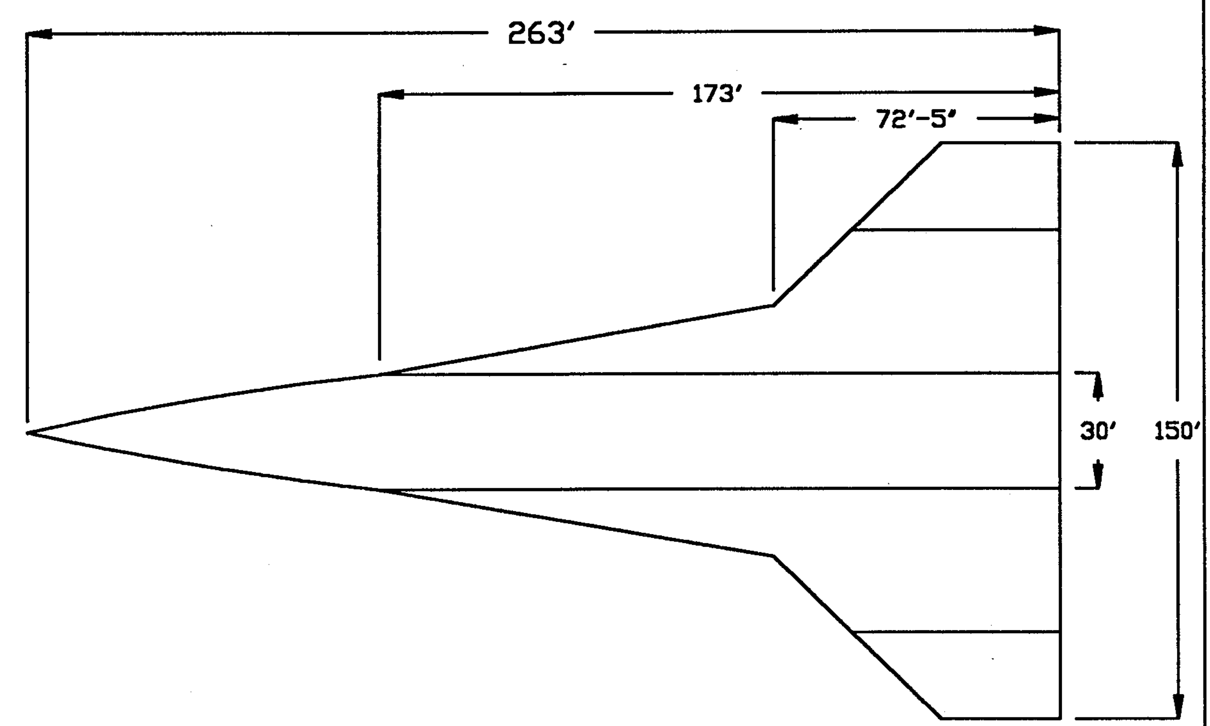
The final configuration for this design incorporated some fine tuning adjustments from the previous configuration. The length was again increased slightly to 303 feet to accommodate more fuel and increase the fineness ratio. The "coke-bottle" fuselage was increased to 70 feet of the total fuselage length. The trailing edge of the wing was swept at an angle of -5 degrees to increase the reference area of the wing slightly to 13320 square feet with minimal weight increase. The total wingspan remained at 218 feet. The LEX incorporated a 3 degree incidence angle for favorable high speed pitching moments. The wing has a 3 degree dihedral angle for stability considerations at separation.

APPENDIX TO CONFIGURATION DEVELOPMENT

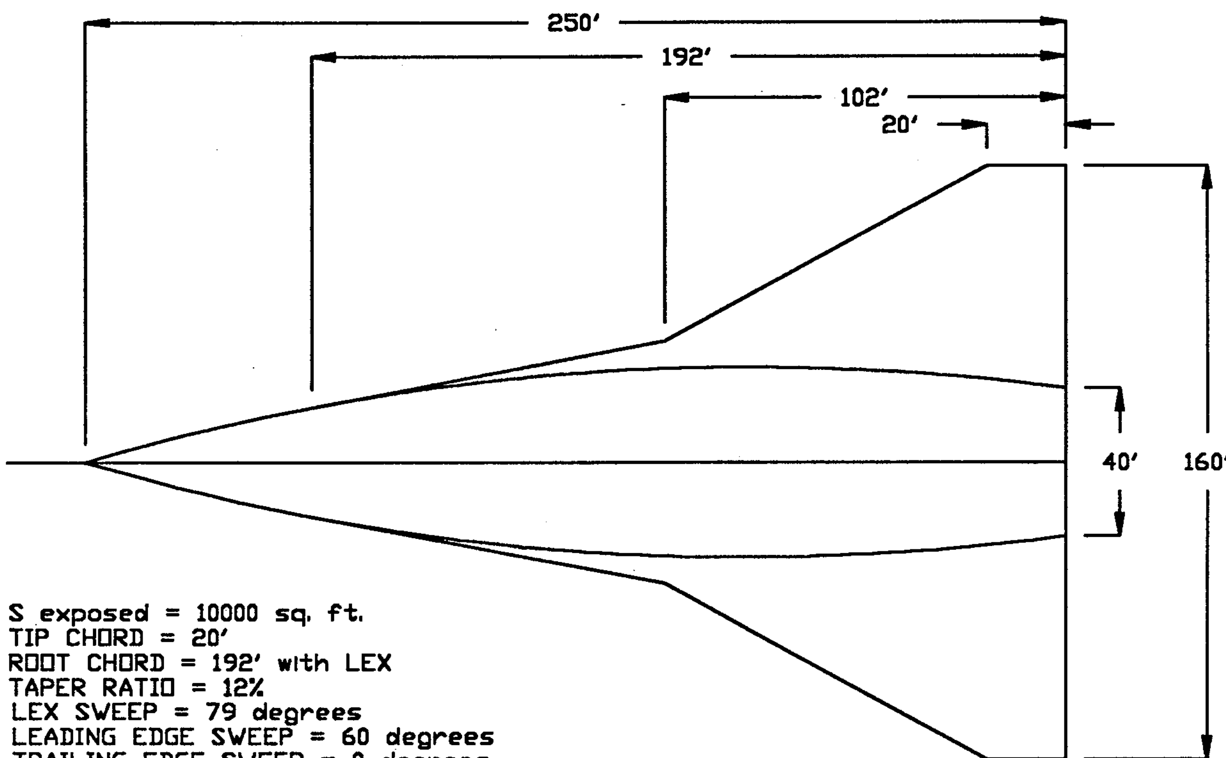
APPENDIX A



CONFIGURATION PROPOSAL NUMBER 1
ORBITER LOCATED ON FRONT OF BOOSTER VEHICLE

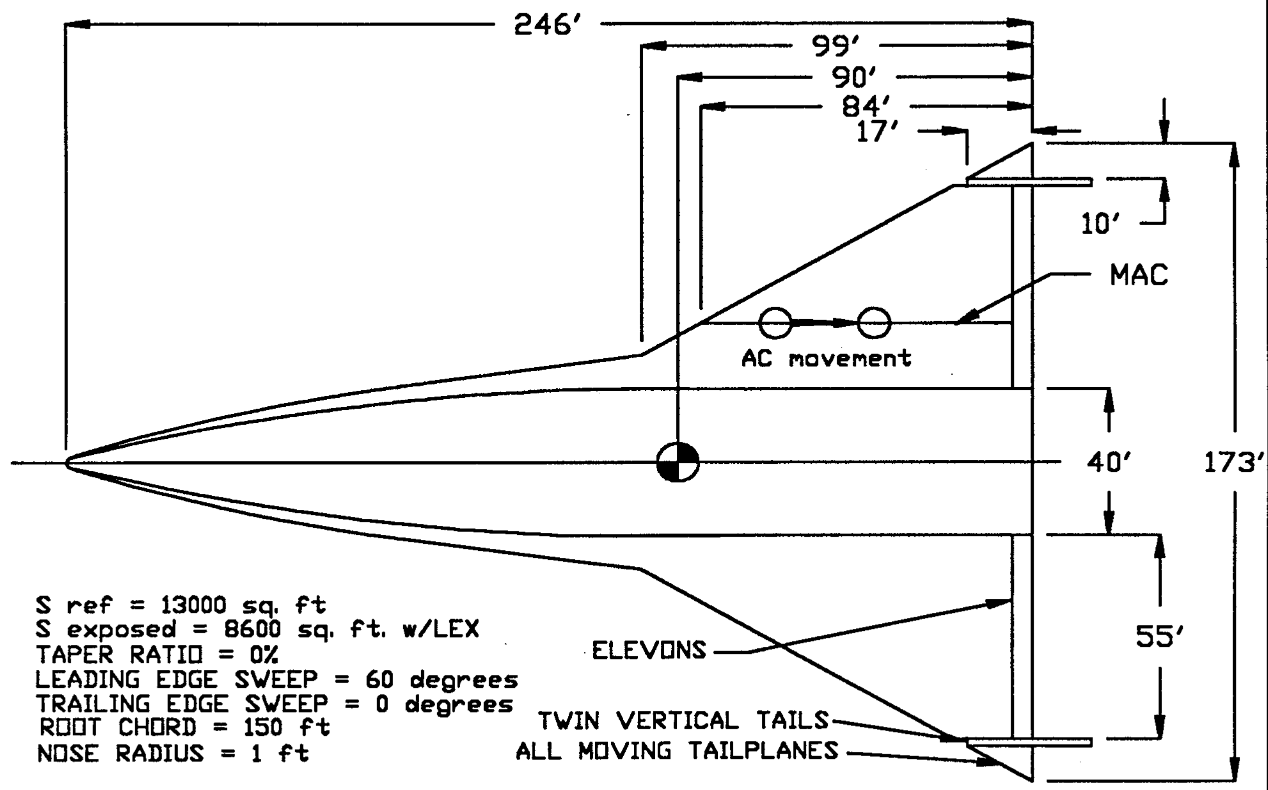


CONFIGURATION PROPOSAL NUMBER 2
ORBITER LOCATED ON TOP REAR OF BOOSTER VEHICLE

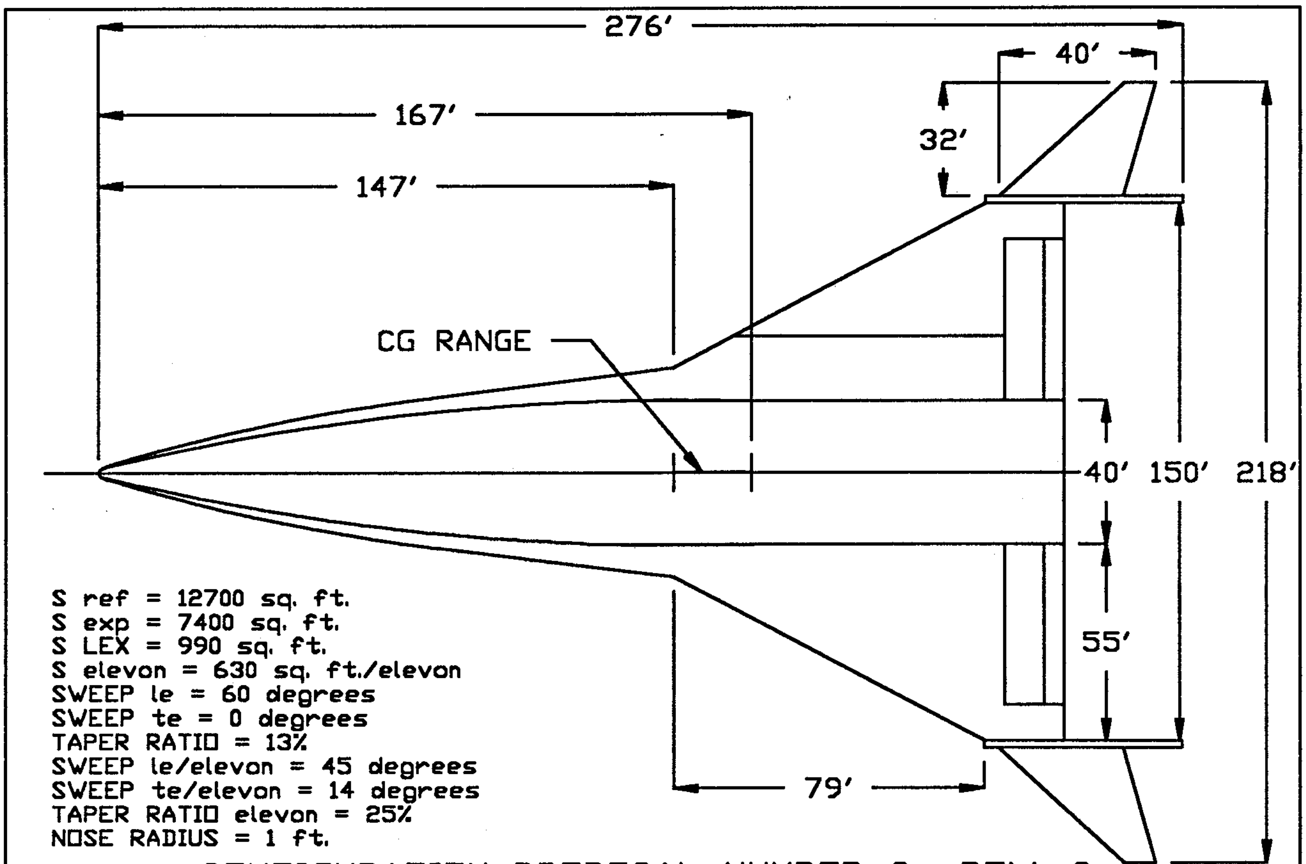


S exposed = 10000 sq. ft.
TIP CHORD = 20'
ROOT CHORD = 192' with LEX
TAPER RATIO = 12%
LEX SWEEP = 79 degrees
LEADING EDGE SWEEP = 60 degrees
TRAILING EDGE SWEEP = 0 degrees

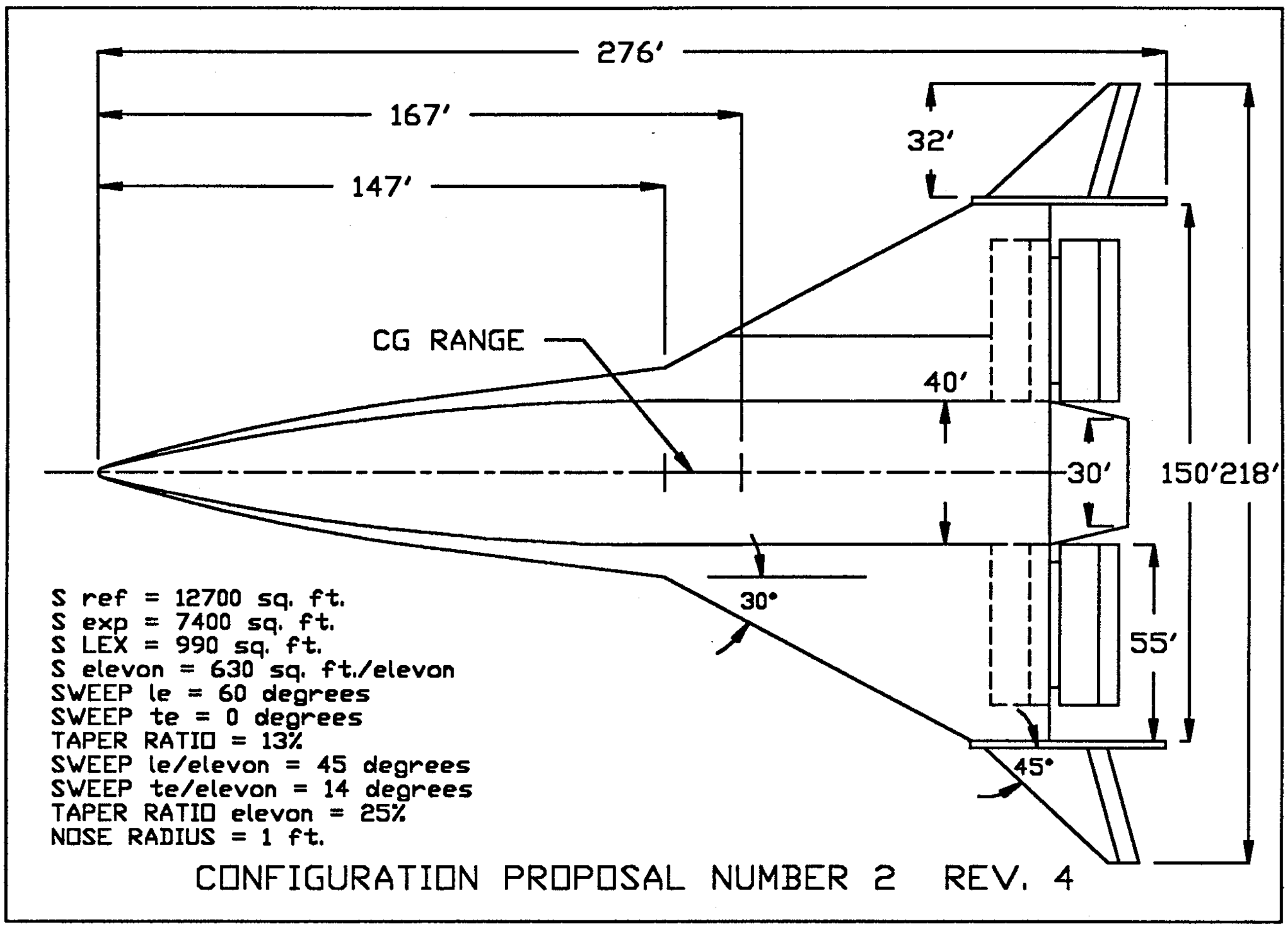
CONFIGURATION PROPOSAL NUMBER 2 REV. 1



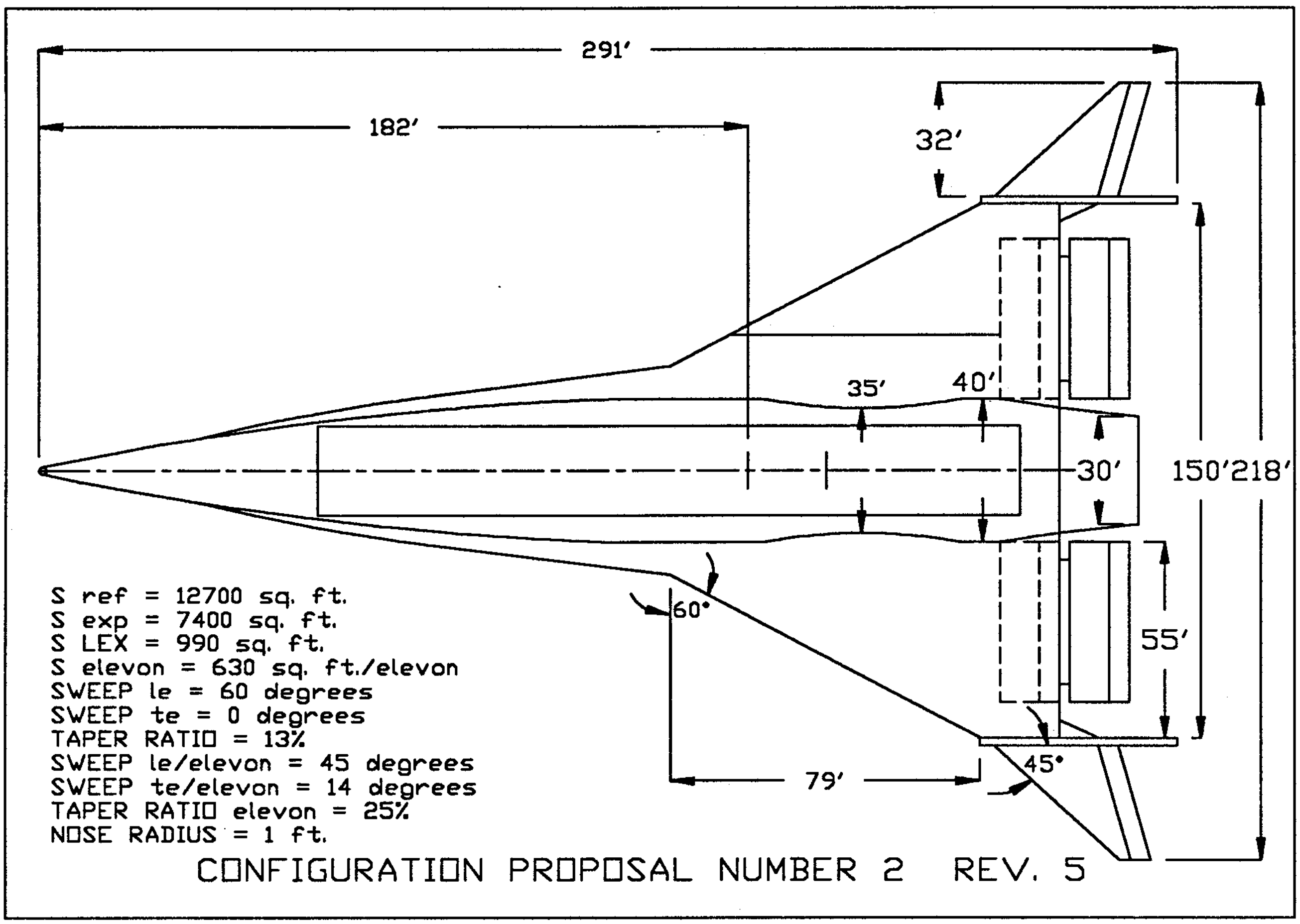
CONFIGURATION PROPOSAL NUMBER 2 REV. 2

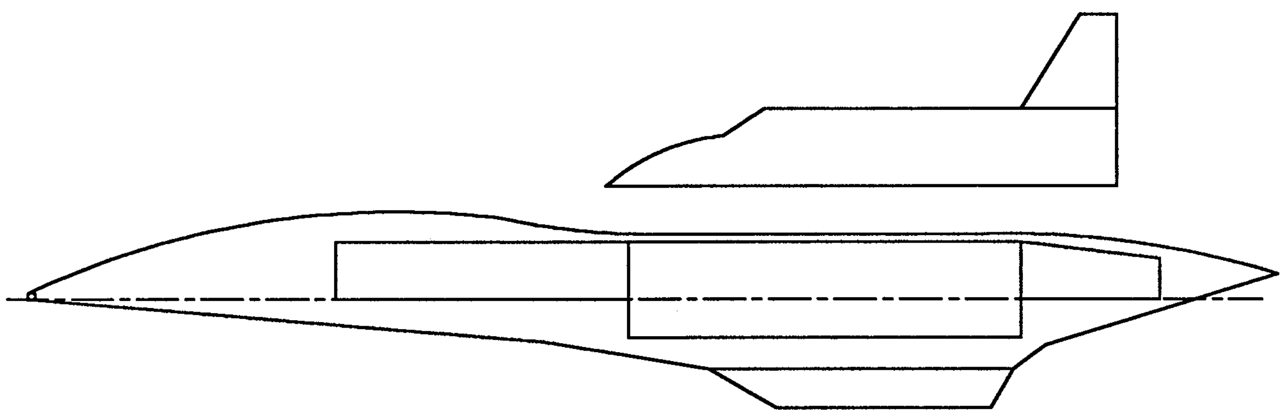


CONFIGURATION PROPOSAL NUMBER 2 REV. 3



276'
167'
147'
32'
CG RANGE
40'
30'
150'
218'
30°
55'
45°
S ref = 12700 sq. ft.
S exp = 7400 sq. ft.
S LEX = 990 sq. ft.
S elevon = 630 sq. ft./elevon
SWEEP le = 60 degrees
SWEEP te = 0 degrees
TAPER RATIO = 13%
SWEEP le/elevon = 45 degrees
SWEEP te/elevon = 14 degrees
TAPER RATIO elevon = 25%
NOSE RADIUS = 1 ft.
CONFIGURATION PROPOSAL NUMBER 2 REV. 4

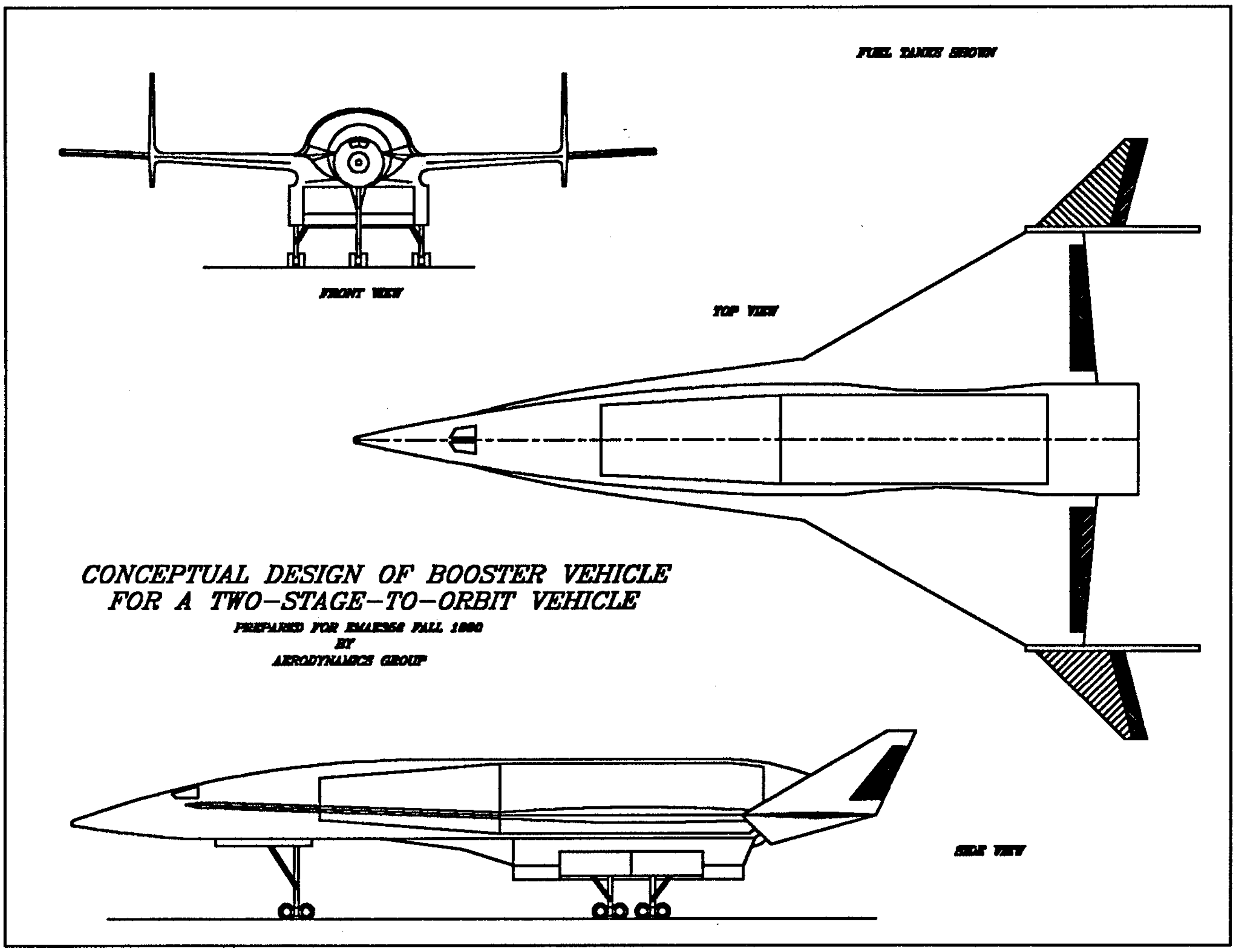

291'
182'
32'
35'
40'
30'
150'
218'
55'
60°
45°
79'
S ref = 12700 sq. ft.
S exp = 7400 sq. ft.
S LEX = 990 sq. ft.
S elevon = 630 sq. ft./elevon
SWEEP le = 60 degrees
SWEEP te = 0 degrees
TAPER RATIO = 13%
SWEEP le/elevon = 45 degrees
SWEEP te/elevon = 14 degrees
TAPER RATIO elevon = 25%
NOSE RADIUS = 1 ft.
CONFIGURATION PROPOSAL NUMBER 2 REV. 5

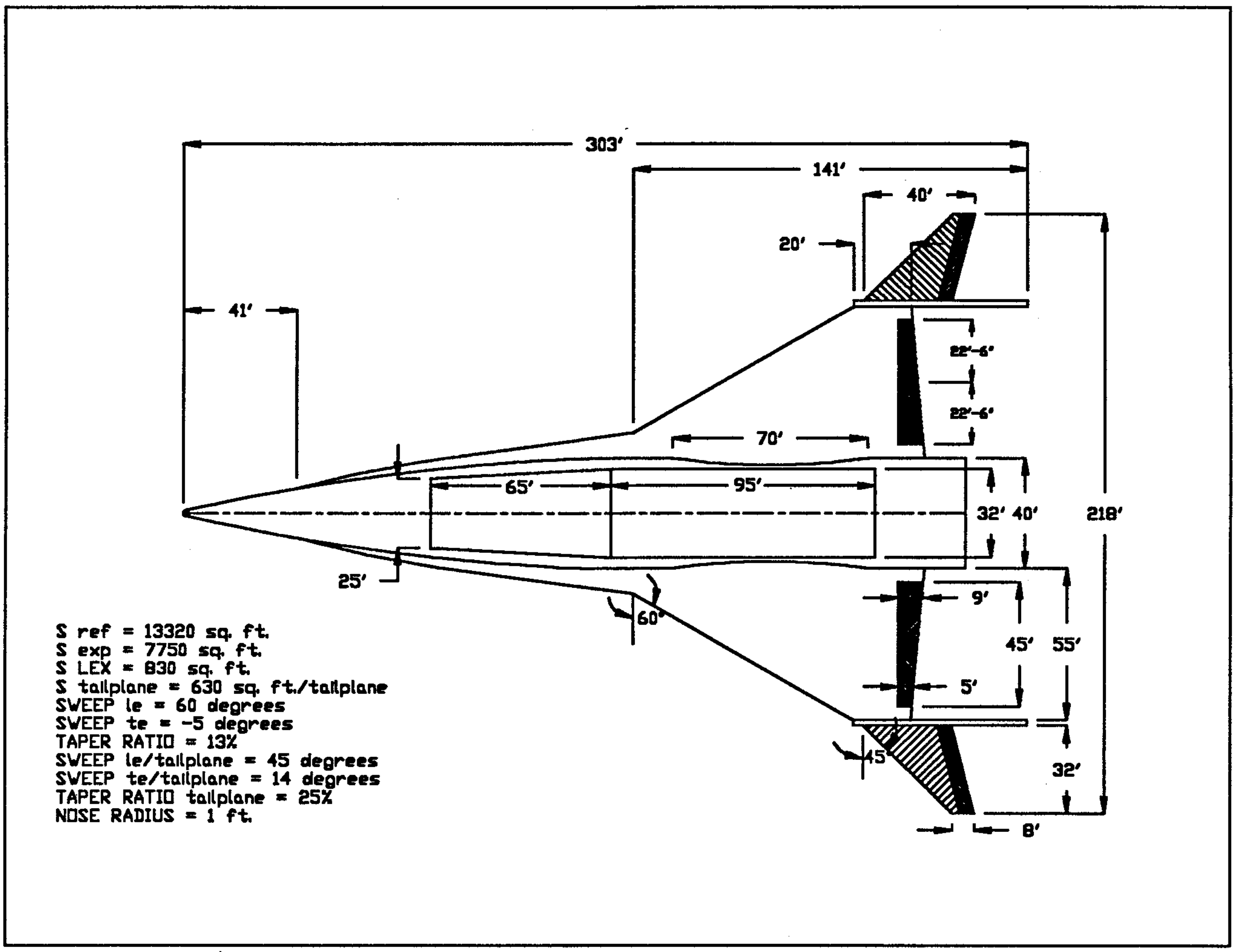


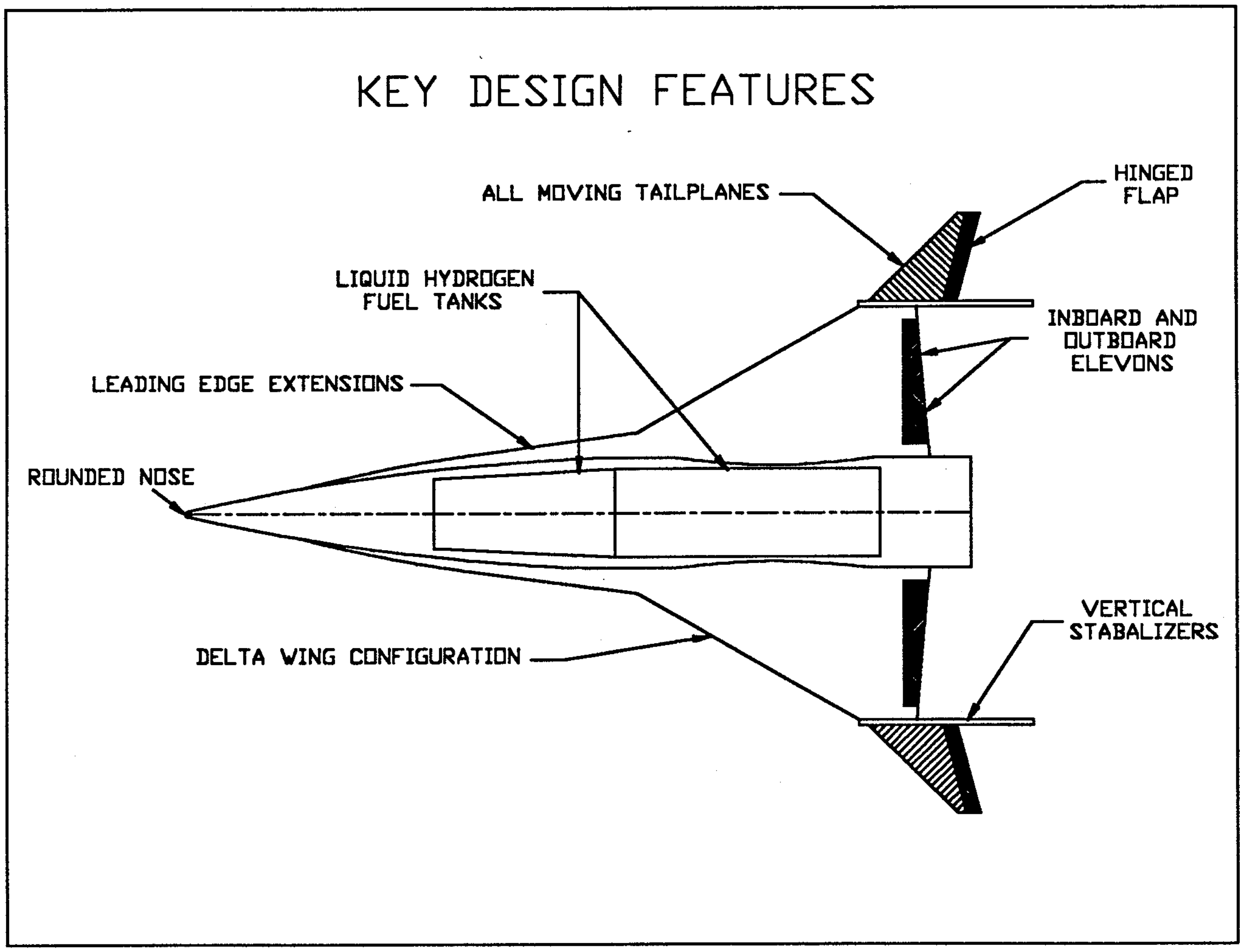
CONFIGURATION PROPOSAL NUMBER 2

INITIAL SIDE VIEW SKETCH

FINAL DESIGN PROPOSAL FOR BOOSTER VEHICLE


CONCEPTUAL DESIGN OF BOOSTER VEHICLE
FOR A TWO-STAGE-TO-ORBIT VEHICLE


303'
141'
40'
20'
41'
22'-6'
22'-6'
70'
65'
95'
32'
40'
218'
25'
60°
9'
45'
55'
5'
45°
32'
8'
S ref = 13320 sq. ft.
S exp = 7750 sq. ft.
S LEX = 830 sq. ft.
S tailplane = 630 sq. ft./tailplane
SWEEP le = 60 degrees
SWEEP te = -5 degrees
TAPER RATIO = 13%
SWEEP le/tailplane = 45 degrees
SWEEP te/tailplane = 14 degrees
TAPER RATIO tailplane = 25%
NOSE RADIUS = 1 ft.



KEY DESIGN FEATURES
ALL MOVING TAILPLANES
HINGED FLAP
LIQUID HYDROGEN FUEL TANKS
INBOARD AND OUTBOARD ELEVONS
LEADING EDGE EXTENSIONS
ROUNDED NOSE
VERTICAL STABALIZERS
DELTA WING CONFIGURATION

AIRFOIL DESIGN

The goals set for the airfoil were:

1) A lift/drag ratio greater then 2
2) Maximum thickness/chord ratio less than 5 percent
3) Low wave drag

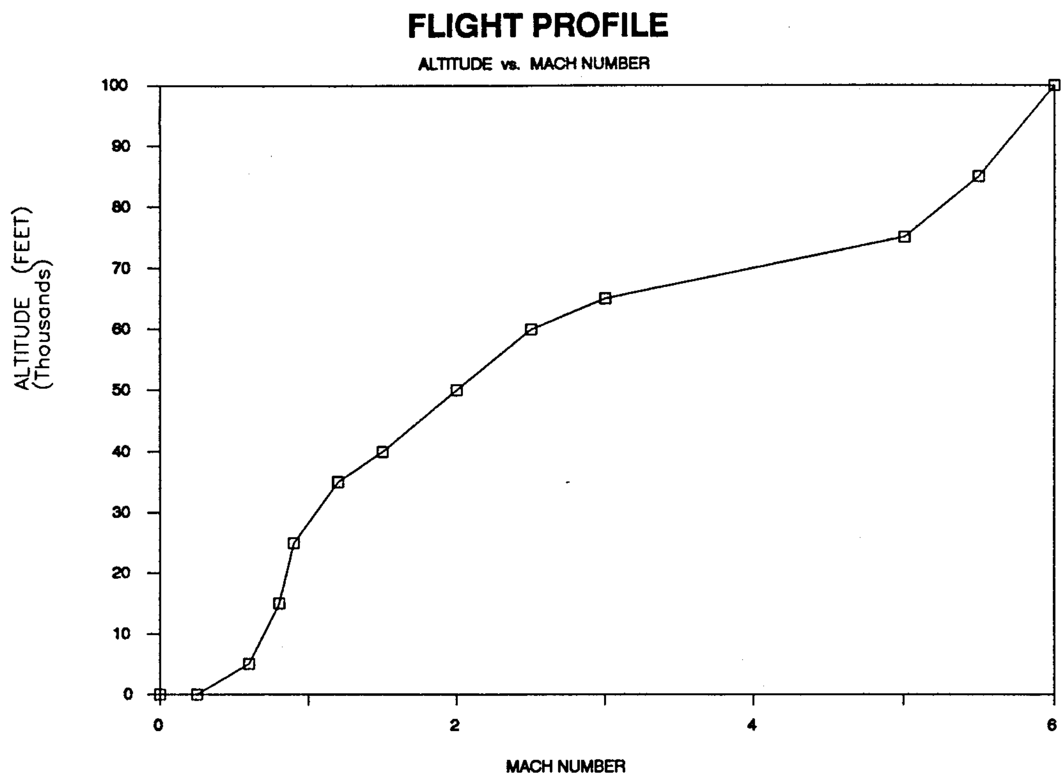
The airfoil was designed with a maximum thickness at the midchord point to reduce wave drag. This airfoil would try to have shock formation as far aft along the airfoil as possible. The leading edge, however, to maintain temperatures below the limits of the materials, could not be sharp. The desire not to use active cooling dictated the use of a blunt leading edge. This blunt leading edge reduce the lift to drag ratios by approximately 50 percent. The top of the airfoil was designed with a slight reflex. This reflex would help achieve a positive moment for the delta wing configuration.

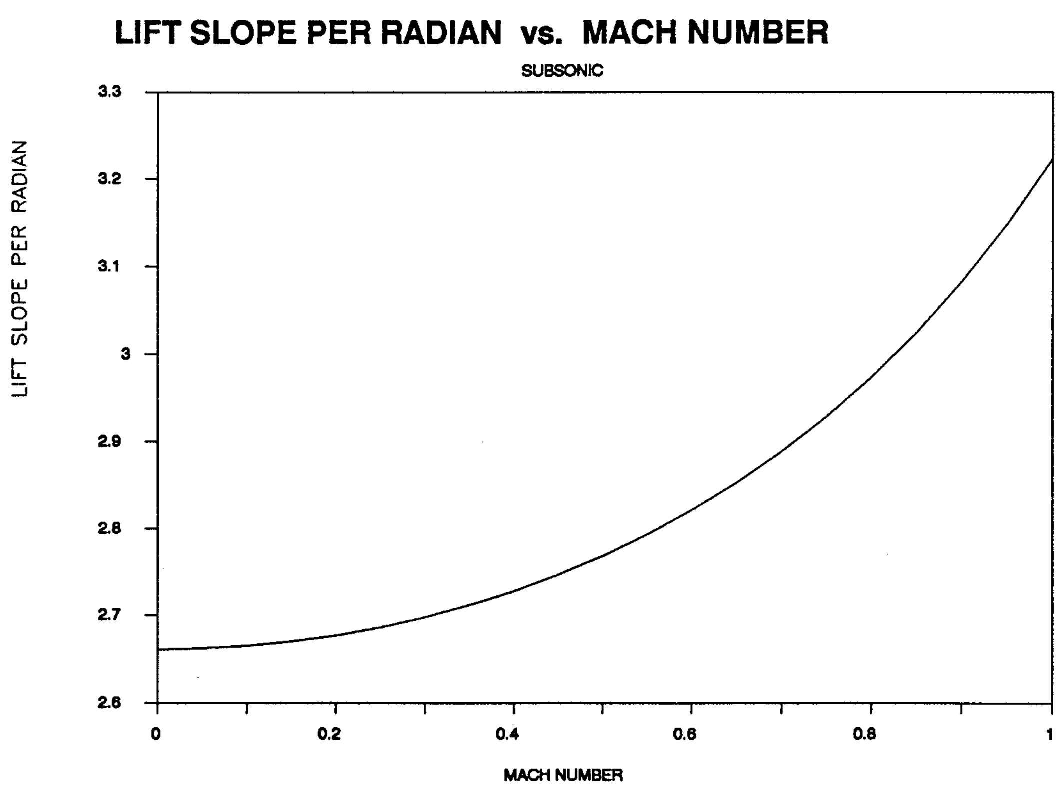
The airfoil chosen was proposal number two. This airfoil section was essentially identical to proposal number one, except for the blunt leading edge for cooling purposes. The airfoil designed did meet the specifications for this stage in the conceptual design. Refer to the appendix for airfoil specifications.

The lift slope verses Mach number and drag coefficient of drag verses Mach number plots were obtained using the methods described in ref. 1. The subsonic lift curve was determined using a correlation between lift slope and the aspect ratio, Mach number, fuselage size, and effective wing sweep. Details of this calculation are in the appendix to this section. The drag coefficient verses Mach number was determined using ref. 7 with an angle of attack of approximately 2 degrees. The supersonic lift slope and drag coefficient verses Mach number was determined using the linearized flow program written for this design project. The transonic lift slope and drag coefficient was determined by a hand fit line drawn though the data points. The drag transonically was determined to be approximately 850,000 lbs. This value was in good agreement with the various NASA references for a vehicle of this type.

APPENDIX TO BOOSTER AIRFOIL DESIGN

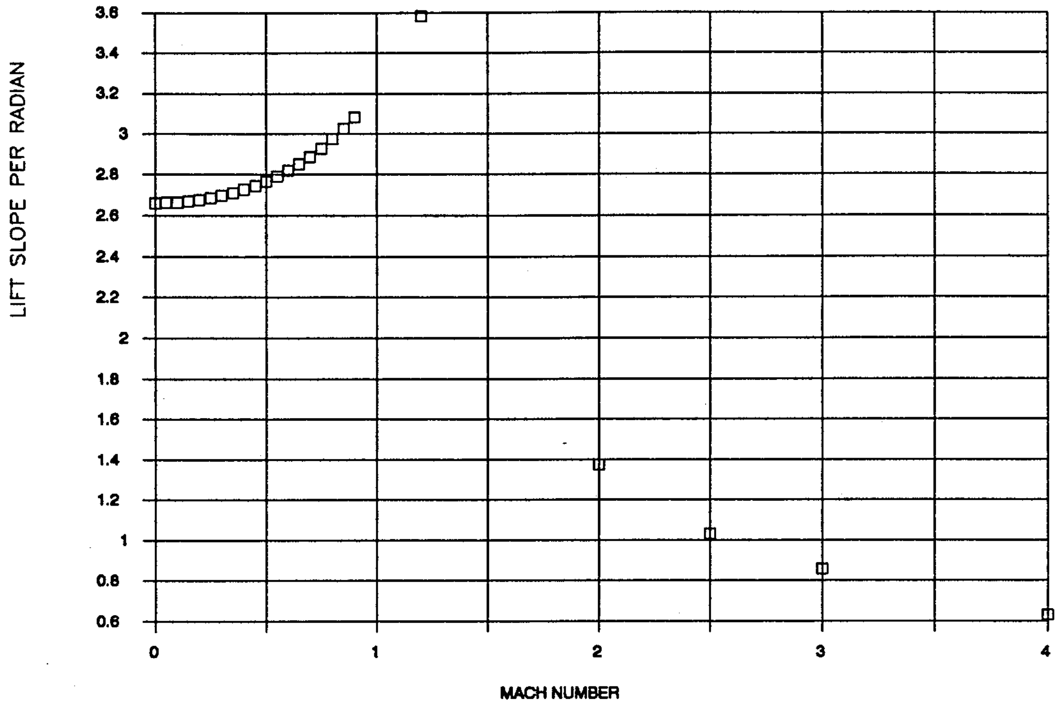
APPENDIX B


FLIGHT PROFILE
ALTITUDE vs. MACH NUMBER
ALTITUDE (FEET)
(Thousands)
100
90
80
70
60
50
40
30
20
10
0
0
2
4
6
MACH NUMBER


LIFT SLOPE PER RADIAN vs. MACH NUMBER
SUBSONIC
LIFT SLOPE PER RADIAN
3.3
3.2
3.1
3
2.9
2.8
2.7
2.6
0
0.2
0.4
0.6
0.8
1
MACH NUMBER

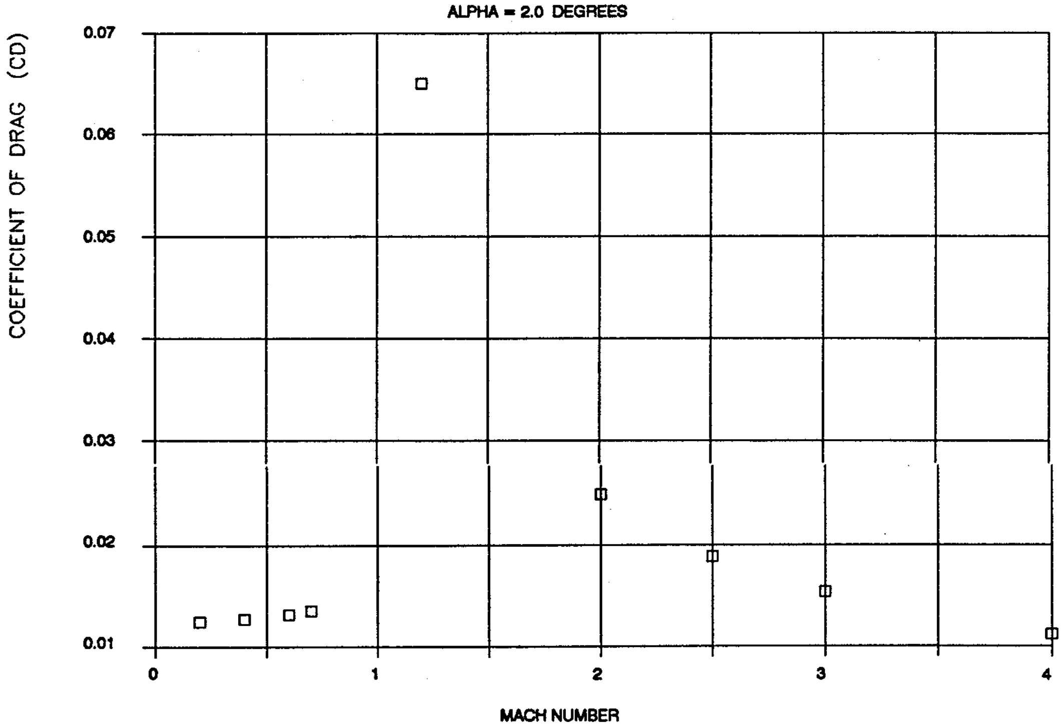
LIFT SLOPE PER RADIAN vs. MACH NUMBER

TRANSONIC REGION



CD vs. MACH NUMBER

ALPHA = 2.0 DEGREES



DETERMINATION OF LIFT SLOPE SUBSONICLY

From Eq 12.6 Ref 1, The lift slope is

$$C_L(\text{alpha}) = \frac{2*pi*A*(S_{exp}/S_{ref})*F}{2+(4+A^2B^2/n^2(1+Tan^2(L_{max,t})/B^2))^{1/2}}$$

where

B^2	=	$1 - M^2$
n	=	0.95 (estimated)
F	=	Fuselage lift factor = $1.07(1 + d/b)^2$
L	=	Sweep of wing at chord location of maximum thickness

subsituting values yields

$$C_L(\text{alpha}) = \frac{16}{2 + (16.1 - 7.31M^2)^{1/2}}$$

DETERMINATION OF FRICTIONAL DRAG

$$Re = \frac{rho*V*l}{mu}$$ where "l " is a characteristic length

Assuming turbulent flow over the entire aircraft, from Eq 12.27, Ref 1,

$$C_f = 0.455/((\log_{10}Re)^{2.58}(1+0.144M^2)^{0.65})$$

Using wetted area for the fuselage from Eq 7.12, Ref 1

$$S_{wet} = 29{,}000 \text{ square feet}$$

From Eq 7.10, Ref 1, for S_{wet} of wing with t/c < 0.05

$$S_{wet} = 15{,}523 \text{ square feet}$$

$$S_{wet,elevon} = 2{,}524 \text{ square feet}$$

$$S_{wet,lex} = 1{,}740 \text{ square feet}$$

$$S_{wet,rudder} = 4{,}500 \text{ square feet}$$

Find C_f at several data points along the flight path

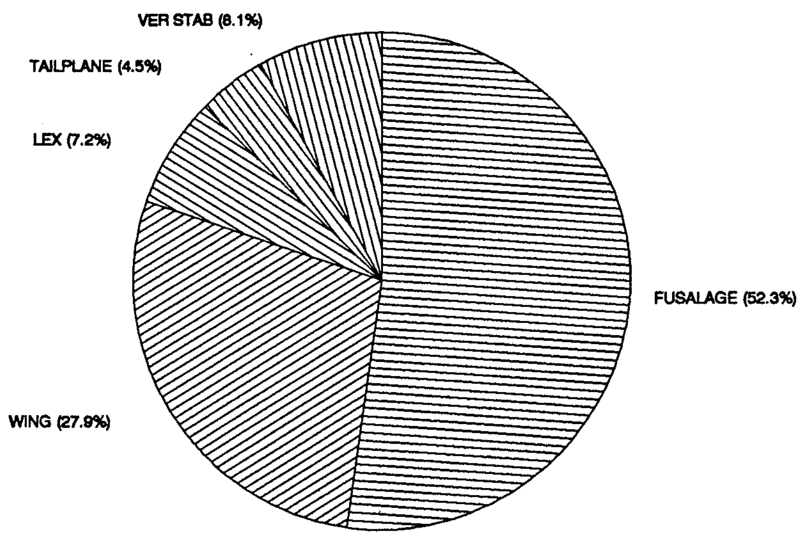
$$S_{total} = 53{,}300 \text{ square feet}$$

POINTS:

MACH	ALTITUDE
1.2	35,000 ft
2.0	50,000
2.5	60,000
3.0	65,000
4.0	70,000
5.0	75,000
5.5	85,000
6.0	100,000

SURFACE AREA BREAKDOWN

BOOSTER VEHICLE



DETERMINATION OF OSWALD EFFICIENCY FACTOR

From Eq 12.50, Ref 1, for a swept-wing aircraft:

$$e = 4.61(1 - 0.045A^{0.68})(\cos(\text{sweep}_{LE}))^{0.15} - 3.1$$

where

A = Effective aspect ratio

Sweep_{LE} = Leading edge sweep = 60

From Eq 12.10, Ref1, for endplate wings:

$$A_{eff} = A(1 + 1.9h/b) = 2.57$$

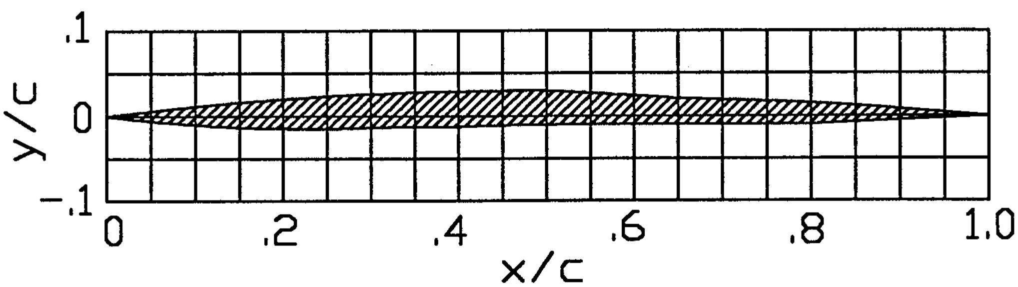
Therefore, e becomes...

$$e = 4.61(1 - 0.045(2.57)^{0.68})(\cos 60)^{0.15} - 3.1$$
$$= 0.70$$

AIRFOIL DATA

AIRFOIL SECTION

PROPOSAL NUMBER 1



SPECIFICATIONS:

MAXIMUM THICKNESS/CHORD RATIO - 0.04 @ x/c = 0.5
LEADING EDGE RADIUS - SHARP
POSITIVE CAMBER

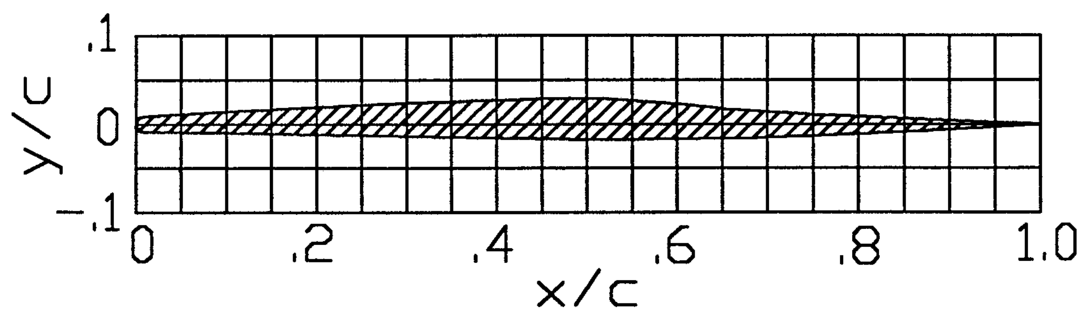
AIRFOIL DATA
PROPOSAL NUMBER 1

NUMBER OF POINT PER SURFACE = 35

POINT	X POINT	Y POINT UPPER	Y POINT LOWER
1	0.0	0.0	0.0
2	0.025	0.003	0.0032
3	0.05	0.0059	0.0060
4	0.075	0.0086	0.0084
5	0.1	0.0112	0.0104
6	0.125	0.0136	0.0121
7	0.15	0.0158	0.0134
8	0.175	0.0179	0.0143
9	0.2	0.0198	0.0148
10	0.225	0.0215	0.0150
11	0.25	0.0231	0.0145
12	0.275	0.0245	0.0142
13	0.3	0.0258	0.0135
14	0.325	0.0269	0.0132
15	0.35	0.0278	0.0129
16	0.375	0.0286	0.0125
17	0.4	0.0292	0.0121
18	0.425	0.0296	0.0116
19	0.45	0.0299	0.0111
20	0.475	0.0300	0.0106
21	0.5	0.0300	0.0100
22	0.525	0.0285	0.0097
23	0.55	0.0269	0.0097
24	0.575	0.0254	0.0097
25	0.6	0.0254	0.0097
26	0.625	0.0238	0.0097
27	0.65	0.0222	0.0097
28	0.675	0.0208	0.0097
29	0.7	0.0194	0.0097
30	0.8	0.0165	0.0097
31	0.825	0.0151	0.0097
32	0.85	0.0138	0.0085
33	0.875	0.0124	0.0073
34	0.9	0.0104	0.0061
35	1.0	0.0	0.0

AIRFOIL SECTION

PROPOSAL NUMBER 2



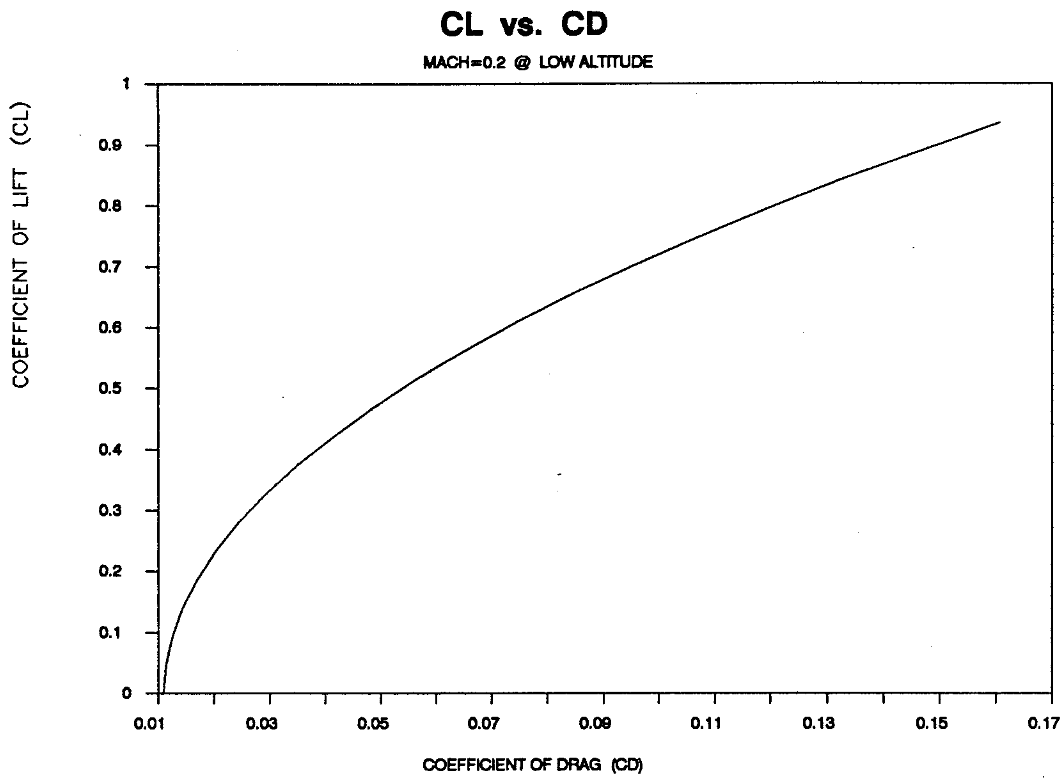
SPECIFICATIONS:

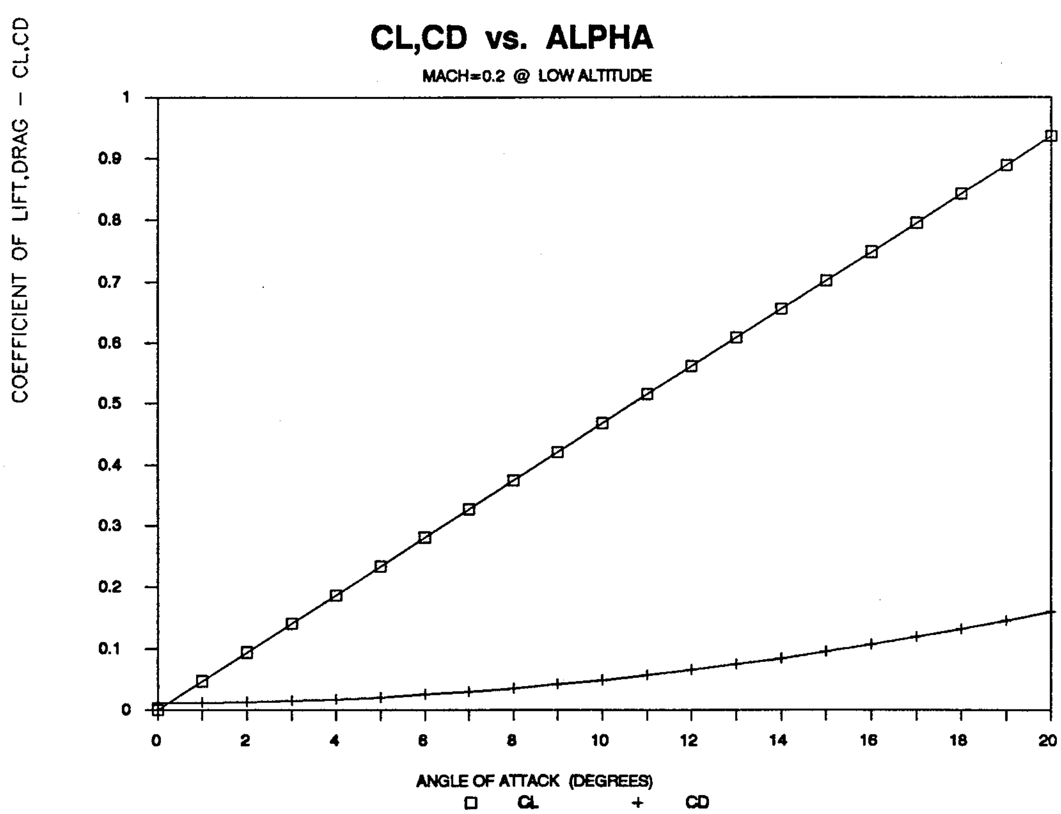
MAXIMUM THICKNESS/CHORD RATIO - 0.04 @ x/c = 0.5
LEADING EDGE RADIUS/CHORD RATIO - 0.0083
POSITIVE CAMBER

AIRFOIL DATA
PROPOSAL NUMBER 2

NUMBER OF POINT PER SURFACE = 16

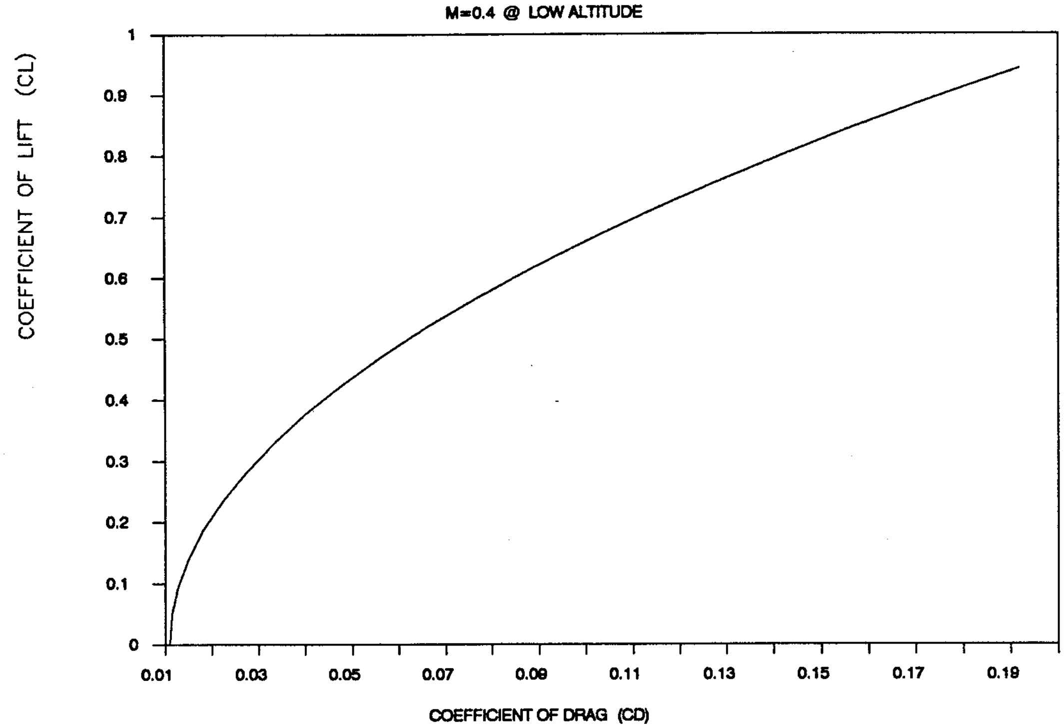
POINT	X POINT	Y POINT UPPER	Y POINT LOWER
1	0.0	0.0	0.0
2	0.003125	0.0065	0.0065
3	0.00625	0.0080	0.0080
4	0.0125	0.0086	0.0084
5	0.025	0.0094	0.0086
6	0.02625	0.0102	0.0088
7	0.0275	0.0110	0.0091
8	0.2	0.0194	0.0119
9	0.35	0.0256	0.0147
10	0.45	0.0285	0.0166
11	0.50	0.0296	0.0175
12	0.55	0.0268	0.0174
13	0.60	0.0228	0.0169
14	0.75	0.0105	0.0134
15	0.80	0.0084	0.0114
16	1.0	0.0	0.0


CL vs. CD
MACH=0.2 @ LOW ALTITUDE
COEFFICIENT OF LIFT (CL)
COEFFICIENT OF DRAG (CD)
1
0.9
0.8
0.7
0.6
0.5
0.4
0.3
0.2
0.1
0
0.01
0.03
0.05
0.07
0.09
0.11
0.13
0.15
0.17


CL,CD vs. ALPHA
MACH=0.2 @ LOW ALTITUDE
COEFFICIENT OF LIFT,DRAG – CL,CD
ANGLE OF ATTACK (DEGREES)
CL
CD
1
0.9
0.8
0.7
0.6
0.5
0.4
0.3
0.2
0.1
0
0
2
4
6
8
10
12
14
16
18
20

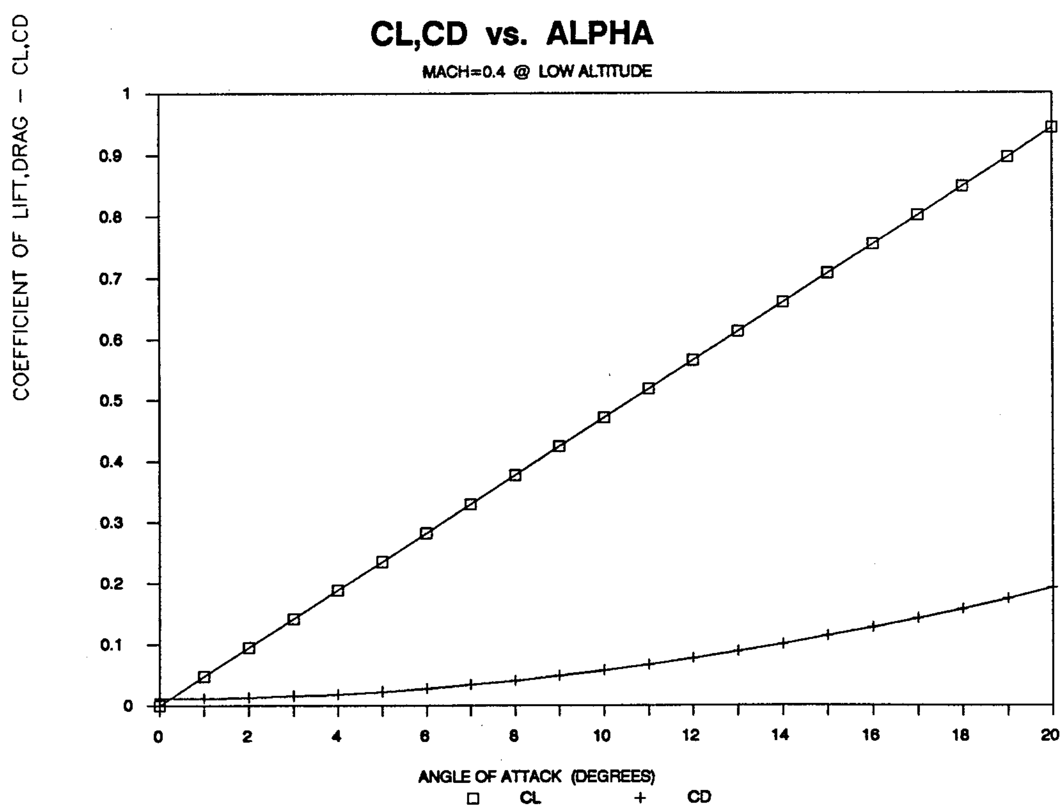
CL vs. CD

M=0.4 @ LOW ALTITUDE



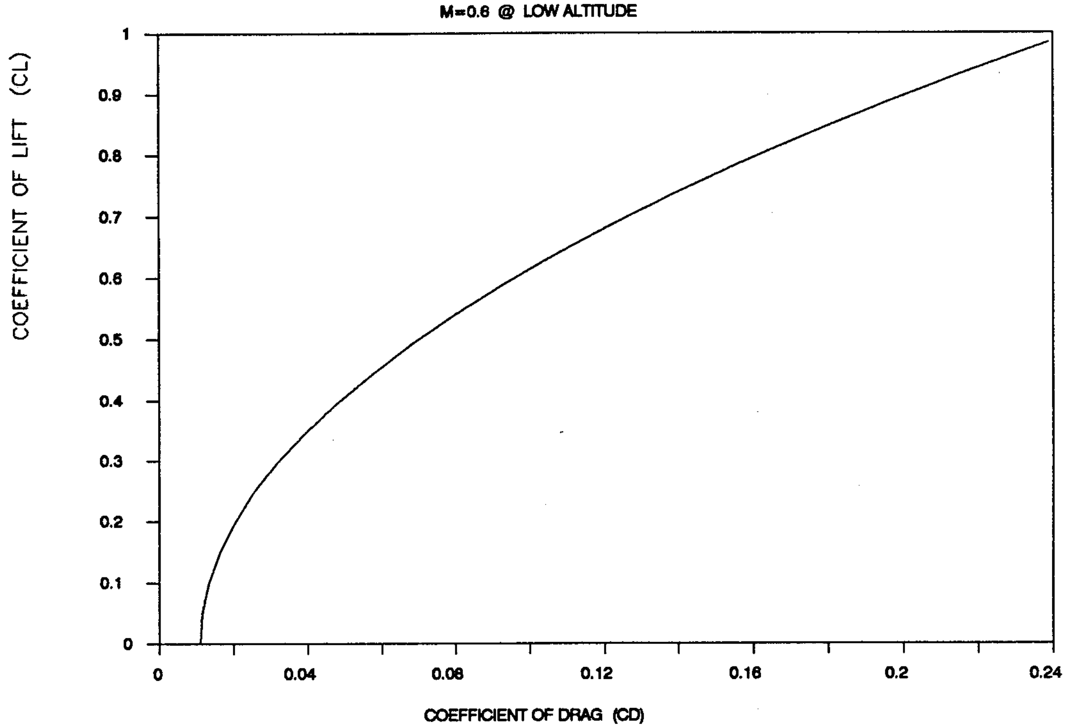
CL,CD vs. ALPHA

MACH=0.4 @ LOW ALTITUDE



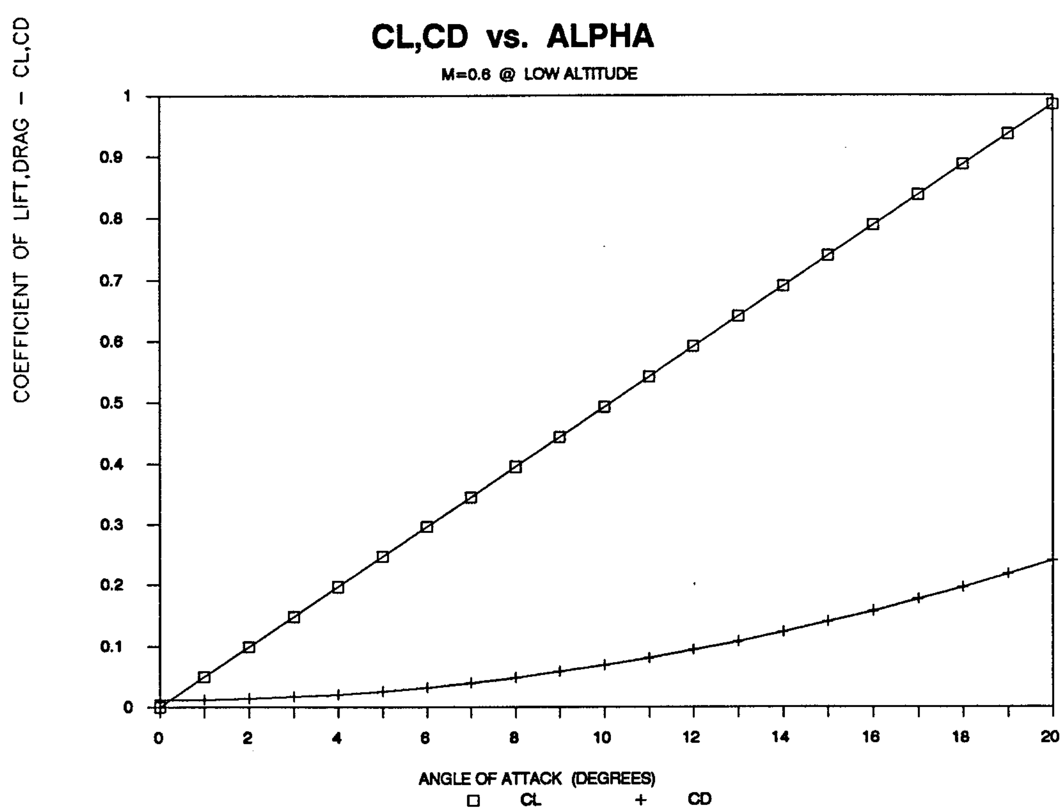
CL vs. CD

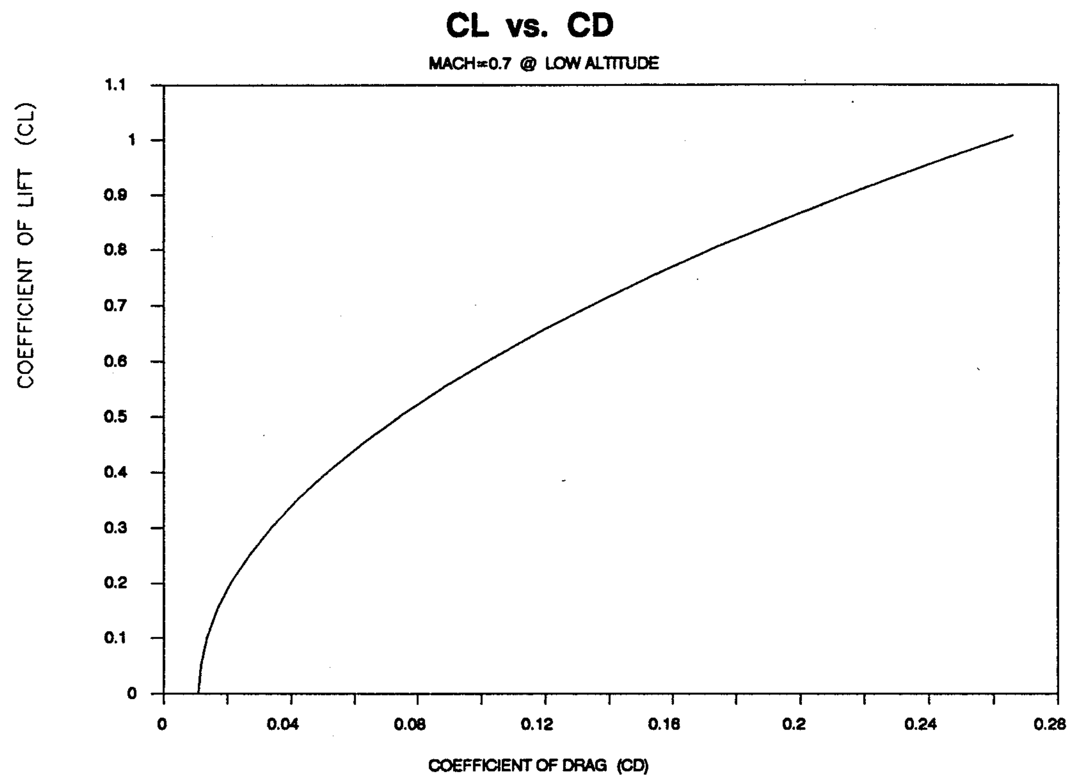
M=0.6 @ LOW ALTITUDE

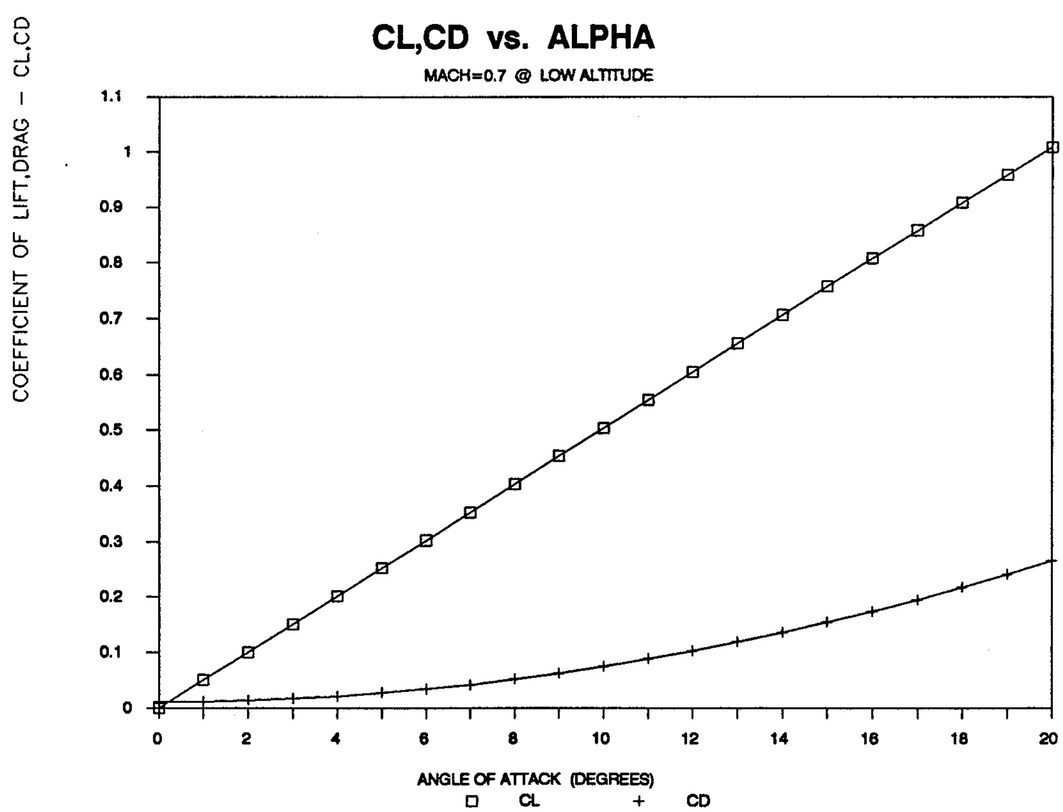


CL,CD vs. ALPHA

M=0.6 @ LOW ALTITUDE




CL vs. CD
MACH=0.7 @ LOW ALTITUDE
COEFFICIENT OF LIFT (CL)
COEFFICIENT OF DRAG (CD)
1.1
1
0.9
0.8
0.7
0.6
0.5
0.4
0.3
0.2
0.1
0
0
0.04
0.08
0.12
0.16
0.2
0.24
0.28


CL,CD vs. ALPHA
MACH=0.7 @ LOW ALTITUDE
COEFFICIENT OF LIFT,DRAG – CL,CD
ANGLE OF ATTACK (DEGREES)
CL
CD
1.1
1
0.9
0.8
0.7
0.6
0.5
0.4
0.3
0.2
0.1
0
0
2
4
6
8
10
12
14
16
18
20

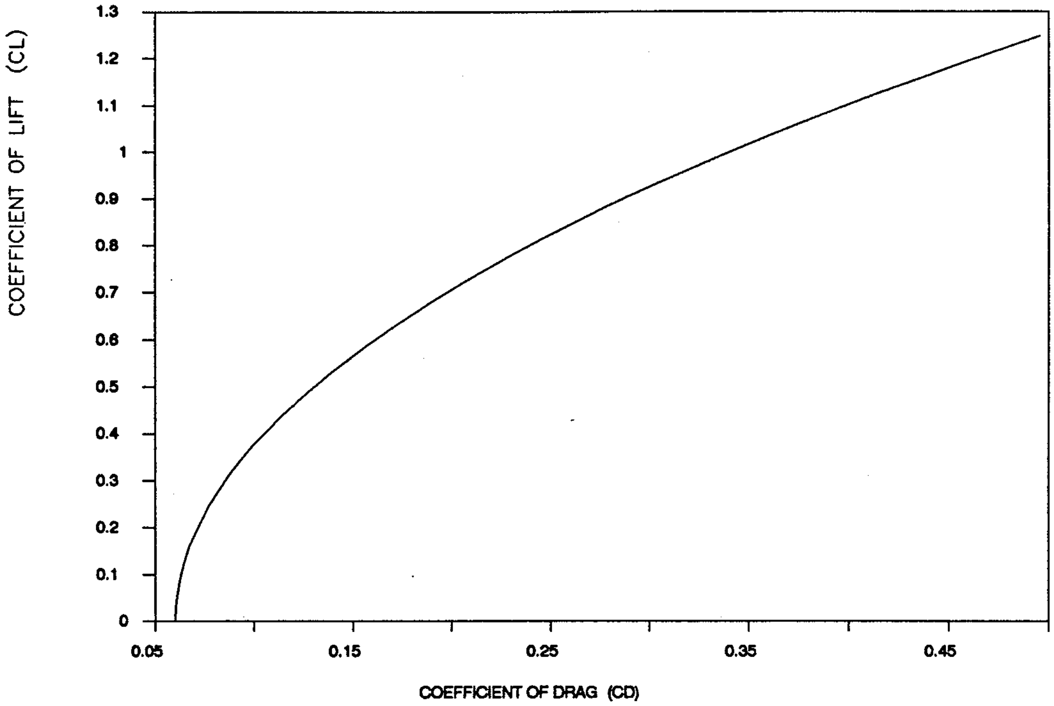
AIRFOIL DATA

MACH=1.2 AT 35000 FT

ALPHA deg	Cl	Cd	L/D	LIFT (pounds)	DRAG (pounds)
0	0	0.0602	0	0.002888	404140
0.5	0.0312	0.0605	0.5158	209130	405970
1	0.0624	0.0613	1.0179	418250	411440
1.5	0.0936	0.0627	1.4937	627380	420570
2	0.1248	0.0646	1.9328	836510	433340
2.5	0.156	0.067	2.3276	1045600	449770
3	0.1872	0.07	2.6737	1254800	469840
3.5	0.2184	0.0735	2.9692	1463900	493570
4	0.2496	0.0776	3.2148	1673000	520940
4.5	0.2808	0.0823	3.4132	1882100	551970
5	0.3119	0.0874	3.5681	2091300	586640
5.5	0.3431	0.0931	3.684	2300400	624970
6	0.3743	0.0994	3.7658	2509500	666940
6.5	0.4055	0.1062	3.8182	2718600	712570
7	0.4367	0.1136	3.8458	2927800	761840
7.5	0.4679	0.1215	3.8526	3136900	814760
8	0.4991	0.1299	3.8425	3346000	871340
8.5	0.5303	0.1389	3.8185	3555200	931560
9	0.5615	0.1484	3.7836	3764300	995440
9.5	0.5927	0.1585	3.74	3973400	1063000
10	0.6239	0.1691	3.6896	4182500	1134100
10.5	0.6551	0.1803	3.6342	4391700	1209000
11	0.6863	0.192	3.5751	4600800	1287400
11.5	0.7175	0.2042	3.5134	4809900	1369600
12	0.7487	0.217	3.45	5019000	1455300
12.5	0.7799	0.2303	3.3857	5228200	1544800
13	0.8111	0.2442	3.3209	5437300	1637800
13.5	0.8423	0.2587	3.2563	5646400	1734600
14	0.8735	0.2736	3.1921	5855600	1834900
14.5	0.9047	0.2891	3.1287	6064700	1938900
15	0.9358	0.3052	3.0662	6273800	2046600
15.5	0.967	0.3218	3.005	6482900	2157900
16	0.9982	0.339	2.945	6692100	2272900
16.5	1.0294	0.3567	2.8863	6901200	2391500
17	1.0606	0.3749	2.8291	7110300	2513800
17.5	1.0918	0.3937	2.7734	7319400	2639700
18	1.123	0.413	2.7191	7528600	2769300
18.5	1.1542	0.4329	2.6663	7737700	2902500
19	1.1854	0.4533	2.6151	7946800	3039400
19.5	1.2166	0.4743	2.5653	8156000	3179900
20	1.2478	0.4958	2.5169	8365100	3324100

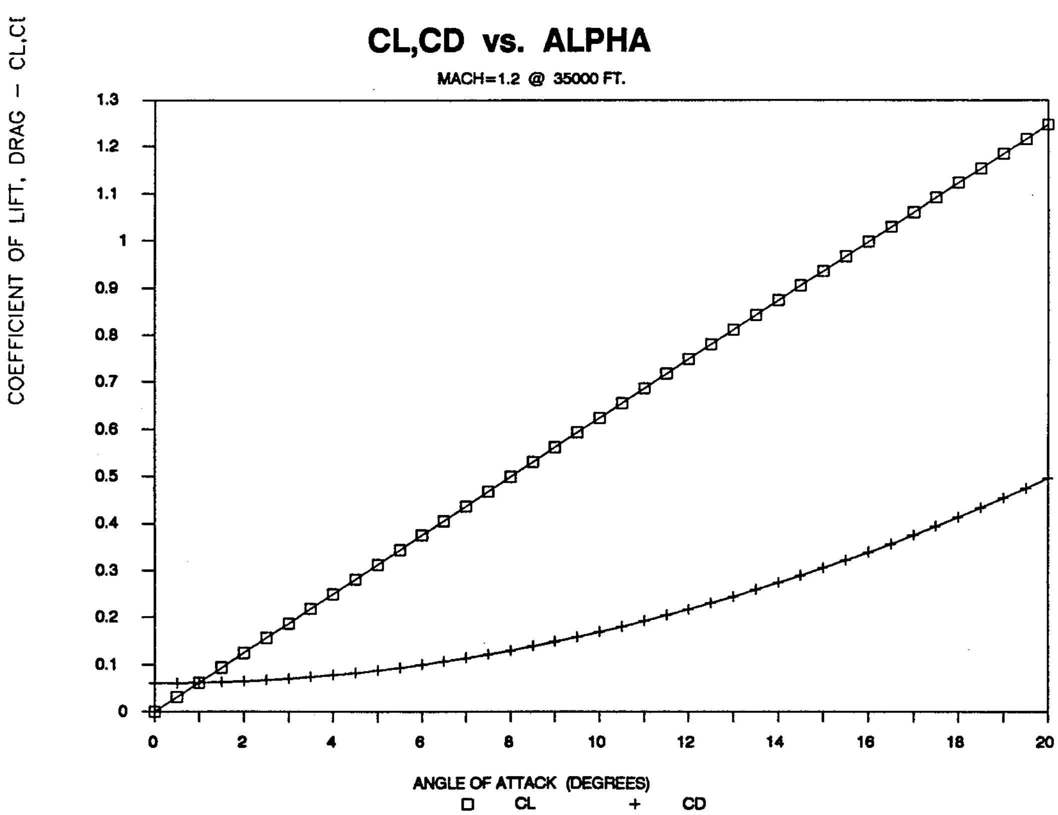
CL vs. CD

MACH=1.2 @ 35000 FT.



CL,CD vs. ALPHA

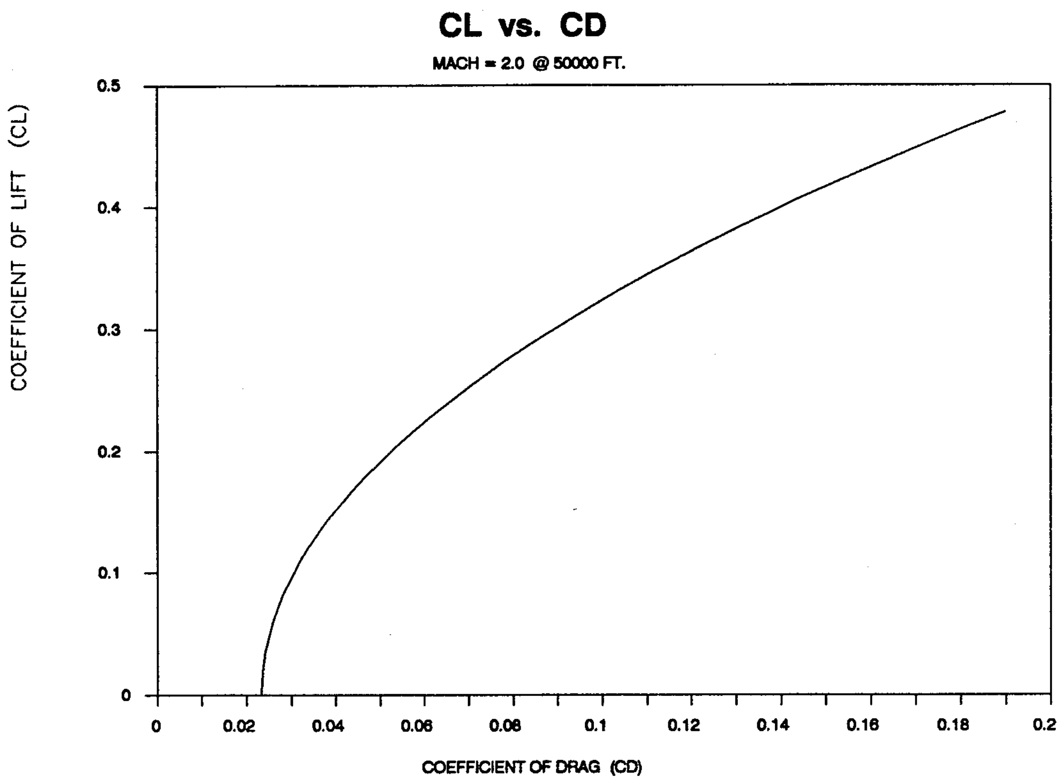
MACH=1.2 @ 35000 FT.

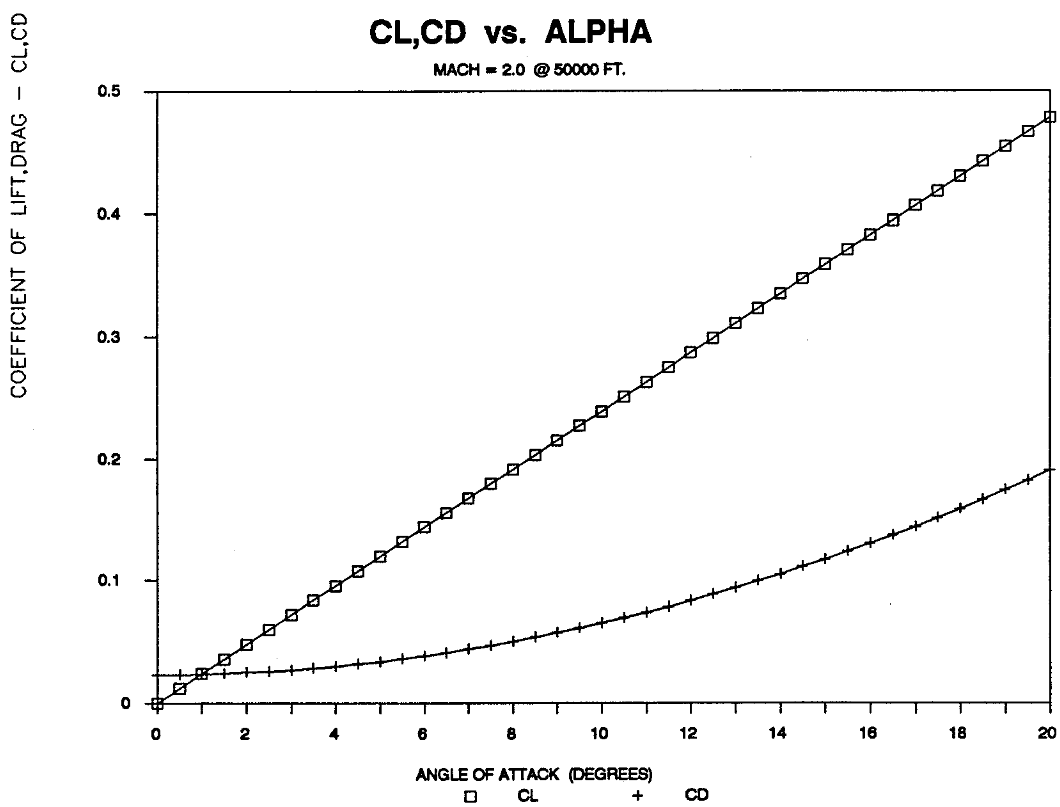


AIRFOIL DATA

MACH=2.0 AT 50000 FT

ALPHA deg	Cl	Cd	L/D	LIFT (pounds)	DRAG (pounds)
0	0	0.0233	0	0.002460	212190
0.5	0.0119	0.0234	0.5107	108540	213140
1	0.0239	0.0237	1.008	217080	215980
1.5	0.0358	0.0242	1.4795	325620	220710
2	0.0478	0.025	1.915	434160	227350
2.5	0.0597	0.0259	2.307	542700	235870
3	0.0717	0.027	2.6509	651240	246290
3.5	0.0836	0.0284	2.9451	759780	258600
4	0.0956	0.03	3.1902	868320	272810
4.5	0.1075	0.0317	3.3885	976860	288910
5	0.1195	0.0337	3.5438	1085400	306910
5.5	0.1314	0.0359	3.6604	1193900	326800
6	0.1434	0.0383	3.7432	1302500	348580
6.5	0.1553	0.0409	3.7967	1411000	372260
7	0.1673	0.0437	3.8255	1519600	397840
7.5	0.1792	0.0467	3.8337	1628100	425310
8	0.1911	0.05	3.8248	1736600	454670
8.5	0.2031	0.0534	3.8021	1845200	485930
9	0.215	0.0571	3.7683	1953700	519080
9.5	0.227	0.0609	3.7259	2062300	554120
10	0.2389	0.065	3.6766	2170800	591070
10.5	0.2509	0.0693	3.6222	2279300	629900
11	0.2628	0.0737	3.564	2387900	670630
11.5	0.2748	0.0784	3.5031	2496400	713250
12	0.2867	0.0833	3.4405	2605000	757770
12.5	0.2987	0.0884	3.3769	2713500	804180
13	0.3106	0.0938	3.3128	2822000	852490
13.5	0.3226	0.0993	3.2487	2930600	902690
14	0.3345	0.105	3.1851	3039100	954780
14.5	0.3465	0.111	3.1222	3147700	1008800
15	0.3584	0.1171	3.0602	3256200	1064700
15.5	0.3703	0.1235	2.9994	3364700	1122400
16	0.3823	0.13	2.9398	3473300	1182100
16.5	0.3942	0.1368	2.8815	3581800	1243700
17	0.4062	0.1438	2.8246	3690400	1307100
17.5	0.4181	0.151	2.7691	3798900	1372500
18	0.4301	0.1584	2.7152	3907400	1439700
18.5	0.442	0.166	2.6626	4016000	1508900
19	0.454	0.1738	2.6116	4124500	1579900
19.5	0.4659	0.1819	2.562	4233100	1652900
20	0.4779	0.1901	2.5139	4341600	1727700

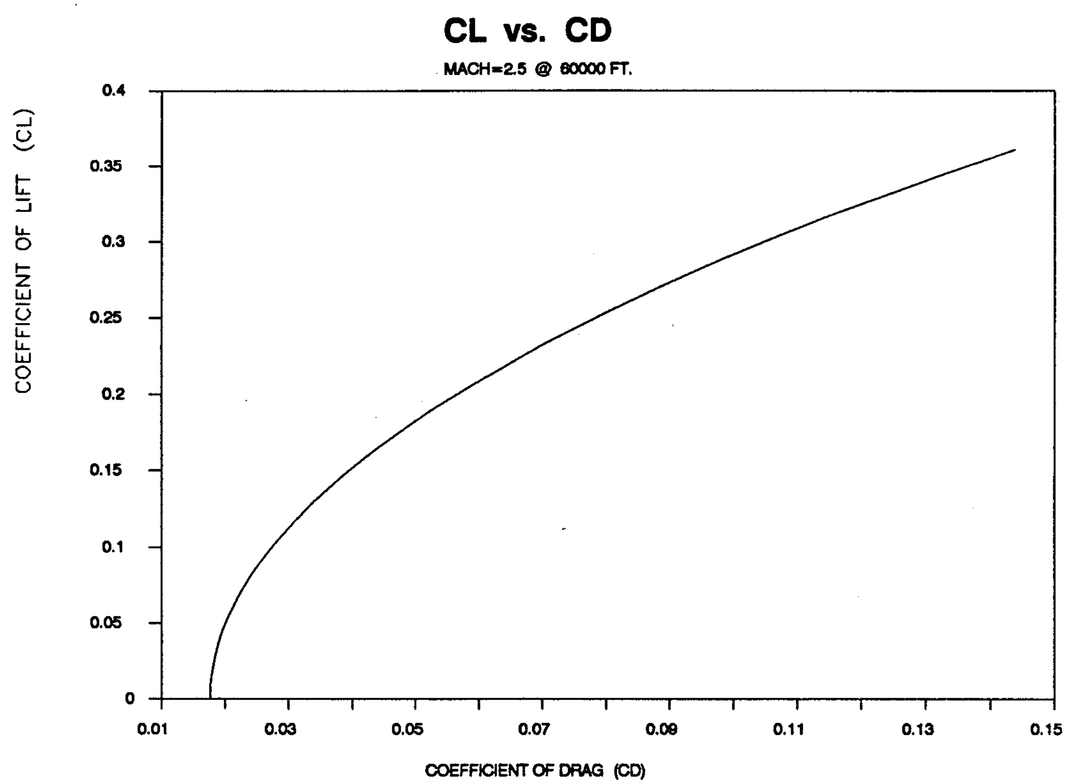

CL vs. CD
MACH = 2.0 @ 50000 FT.
COEFFICIENT OF LIFT (CL)
0.5
0.4
0.3
0.2
0.1
0
0
0.02
0.04
0.06
0.08
0.1
0.12
0.14
0.16
0.18
0.2
COEFFICIENT OF DRAG (CD)

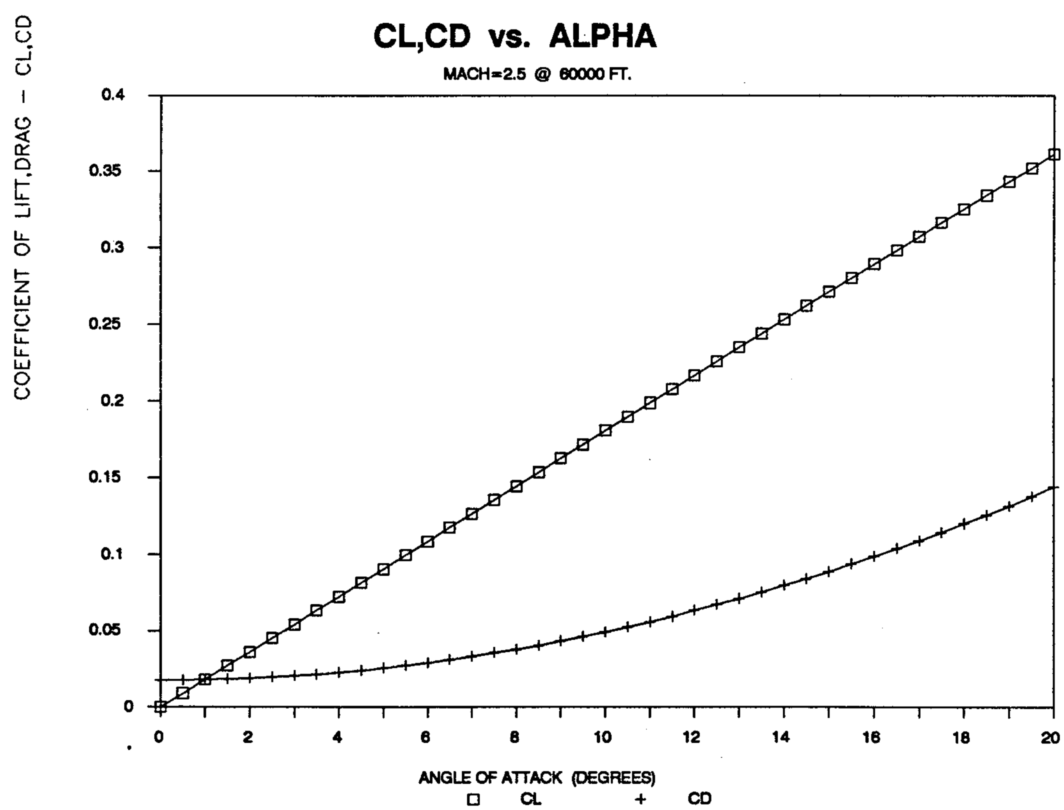

CL,CD vs. ALPHA
MACH = 2.0 @ 50000 FT.
COEFFICIENT OF LIFT,DRAG – CL,CD
0.5
0.4
0.3
0.2
0.1
0
0
2
4
6
8
10
12
14
16
18
20
ANGLE OF ATTACK (DEGREES)
CL
CD

AIRFOIL DATA

MACH=2.5 AT 60000 FT

ALPHA deg	Cl	Cd	L/D	LIFT (pounds)	DRAG (pounds)
0	0	0.0177	0	0.001146	155990
0.5	0.009	0.0177	0.509	79468	156680
1	0.0181	0.018	1.0045	158940	158760
1.5	0.0271	0.0184	1.4745	238400	162230
2	0.0361	0.0189	1.9087	317870	167090
2.5	0.0452	0.0196	2.2997	397340	173330
3	0.0542	0.0205	2.6429	476800	180960
3.5	0.0632	0.0215	2.9367	556270	189970
4	0.0722	0.0227	3.1815	635740	200370
4.5	0.0813	0.024	3.3798	715210	212160
5	0.0903	0.0255	3.5352	794670	225340
5.5	0.0993	0.0272	3.6521	874140	239900
6	0.1084	0.029	3.7352	953610	255850
6.5	0.1174	0.031	3.7892	1033100	273190
7	0.1264	0.0331	3.8184	1112500	291910
7.5	0.1355	0.0354	3.827	1192000	312020
8	0.1445	0.0378	3.8186	1271500	333520
8.5	0.1535	0.0404	3.7963	1350900	356410
9	0.1626	0.0432	3.763	1430400	380680
9.5	0.1716	0.0461	3.7208	1509900	406340
10	0.1806	0.0492	3.6719	1589400	433380
10.5	0.1896	0.0524	3.6179	1668800	461820
11	0.1987	0.0558	3.56	1748300	491640
11.5	0.2077	0.0594	3.4995	1827800	522840
12	0.2167	0.0631	3.4371	1907200	555440
12.5	0.2258	0.0669	3.3737	1986700	589420
13	0.2348	0.0709	3.3099	2066200	624780
13.5	0.2438	0.0751	3.2461	2145600	661540
14	0.2529	0.0795	3.1826	2225100	699680
14.5	0.2619	0.0839	3.1199	2304600	739210
15	0.2709	0.0886	3.0581	2384000	780130
15.5	0.28	0.0934	2.9974	2463500	822430
16	0.289	0.0984	2.9379	2543000	866120
16.5	0.298	0.1035	2.8797	2622400	911190
17	0.307	0.1088	2.823	2701900	957660
17.5	0.3161	0.1142	2.7676	2781400	1005500
18	0.3251	0.1198	2.7138	2860800	1054700
18.5	0.3341	0.1256	2.6613	2940300	1105400
19	0.3432	0.1315	2.6104	3019800	1157400
19.5	0.3522	0.1375	2.5609	3099200	1210800
20	0.3612	0.1438	2.5128	3178700	1265600

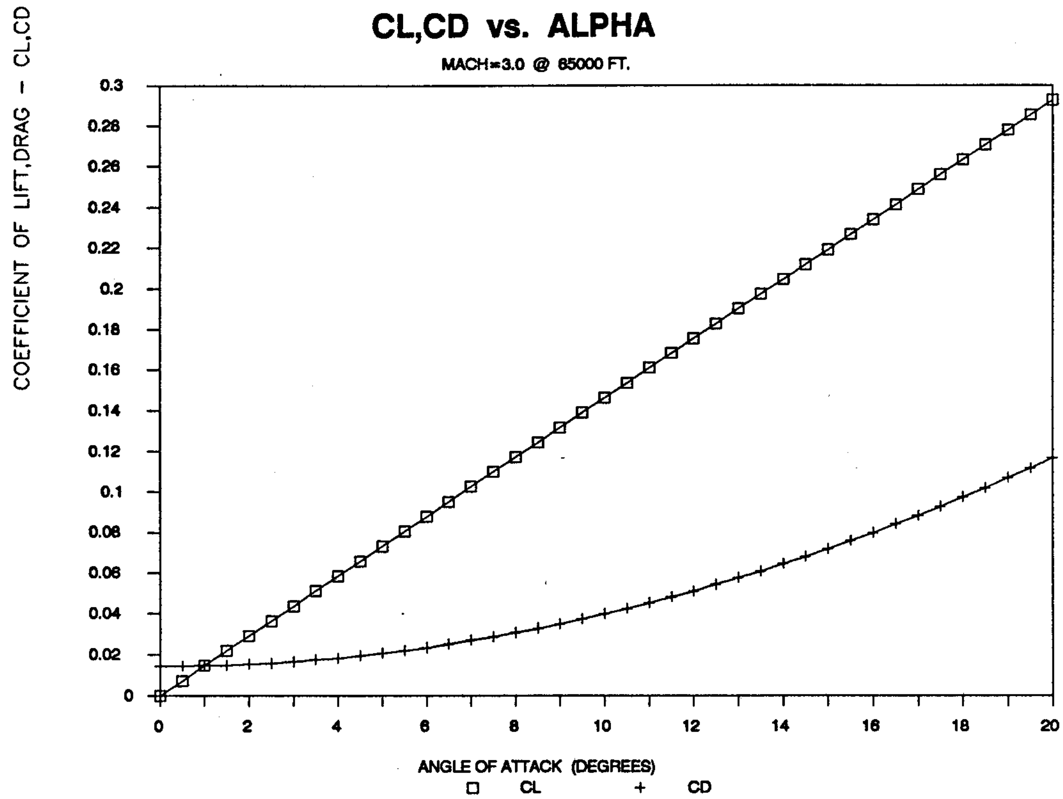

CL vs. CD
MACH=2.5 @ 60000 FT.
COEFFICIENT OF LIFT (CL)
0.4
0.35
0.3
0.25
0.2
0.15
0.1
0.05
0
0.01
0.03
0.05
0.07
0.09
0.11
0.13
0.15
COEFFICIENT OF DRAG (CD)

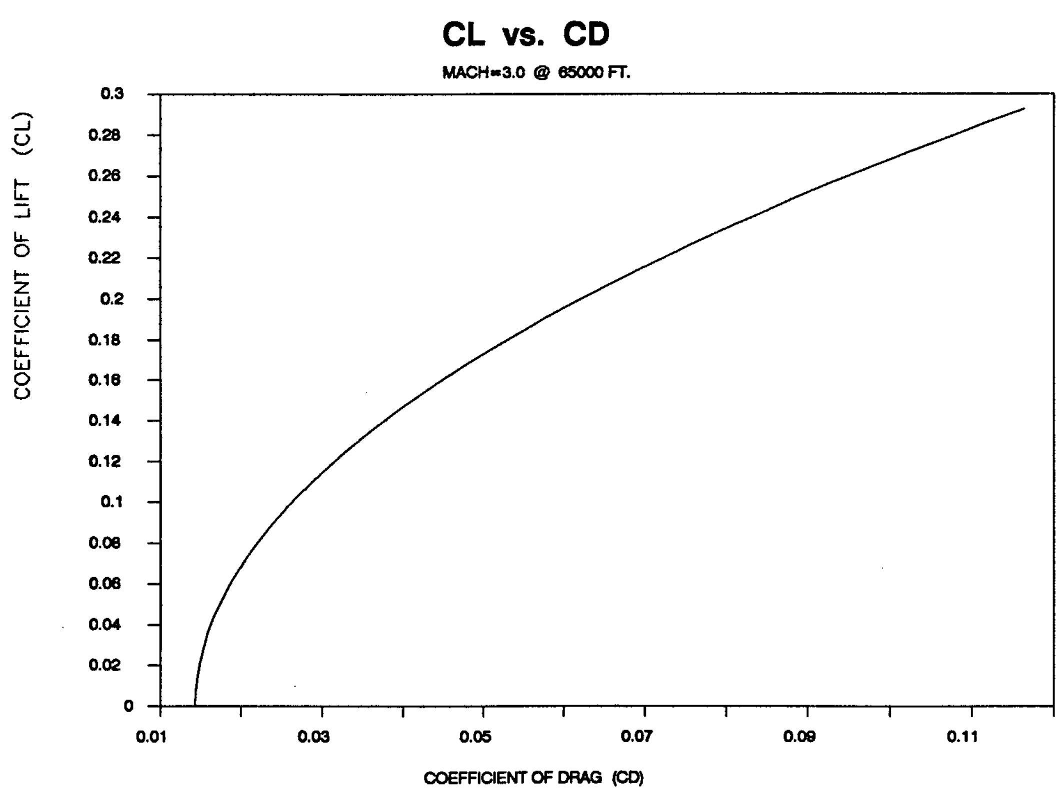

CL,CD vs. ALPHA
MACH=2.5 @ 60000 FT.
COEFFICIENT OF LIFT,DRAG – CL,CD
0.4
0.35
0.3
0.25
0.2
0.15
0.1
0.05
0
0
2
4
6
8
10
12
14
16
18
20
ANGLE OF ATTACK (DEGREES)
□ CL
+ CD

AIRFOIL DATA

MACH=3.0 AT 65000 FT

ALPHA deg	Cl	Cd	L/D	LIFT (pounds)	DRAG (pounds)
0	0	0.0143	0	0.001849	150130
0.5	0.0073	0.0144	0.508	76310	150800
1	0.0146	0.0146	1.0026	152620	152800
1.5	0.0219	0.0149	1.4717	228930	156130
2	0.0293	0.0154	1.9052	305240	160790
2.5	0.0366	0.0159	2.2956	381550	166780
3	0.0439	0.0166	2.6384	457860	174110
3.5	0.0512	0.0175	2.9319	534170	182770
4	0.0585	0.0184	3.1766	610480	192750
4.5	0.0658	0.0195	3.3749	686790	204070
5	0.0732	0.0207	3.5304	763100	216730
5.5	0.0805	0.0221	3.6474	839410	230710
6	0.0878	0.0235	3.7307	915720	246030
6.5	0.0951	0.0251	3.7849	992030	262680
7	0.1024	0.0269	3.8144	1068300	280660
7.5	0.1097	0.0287	3.8232	1144600	299970
8	0.1171	0.0307	3.815	1221000	320610
8.5	0.1244	0.0328	3.793	1297300	342590
9	0.1317	0.035	3.7599	1373600	365900
9.5	0.139	0.0374	3.718	1449900	390540
10	0.1463	0.0399	3.6693	1526200	416510
10.5	0.1536	0.0425	3.6155	1602500	443810
11	0.1609	0.0452	3.5578	1678800	472440
11.5	0.1683	0.0481	3.4974	1755100	502410
12	0.1756	0.0511	3.4352	1831400	533710
12.5	0.1829	0.0542	3.372	1907700	566340
13	0.1902	0.0575	3.3083	1984100	600300
13.5	0.1975	0.0609	3.2446	2060400	635600
14	0.2048	0.0644	3.1812	2136700	672220
14.5	0.2122	0.068	3.1186	2213000	710180
15	0.2195	0.0718	3.0569	2289300	749470
15.5	0.2268	0.0757	2.9963	2365600	790090
16	0.2341	0.0797	2.9369	2441900	832050
16.5	0.2414	0.0839	2.8788	2518200	875330
17	0.2487	0.0881	2.8221	2594500	919950
17.5	0.2561	0.0925	2.7668	2670800	965900
18	0.2634	0.0971	2.713	2747200	1013200
18.5	0.2707	0.1017	2.6606	2823500	1061800
19	0.278	0.1065	2.6097	2899800	1111700
19.5	0.2853	0.1114	2.5602	2976100	1163000
20	0.2926	0.1165	2.5122	3052400	1215600

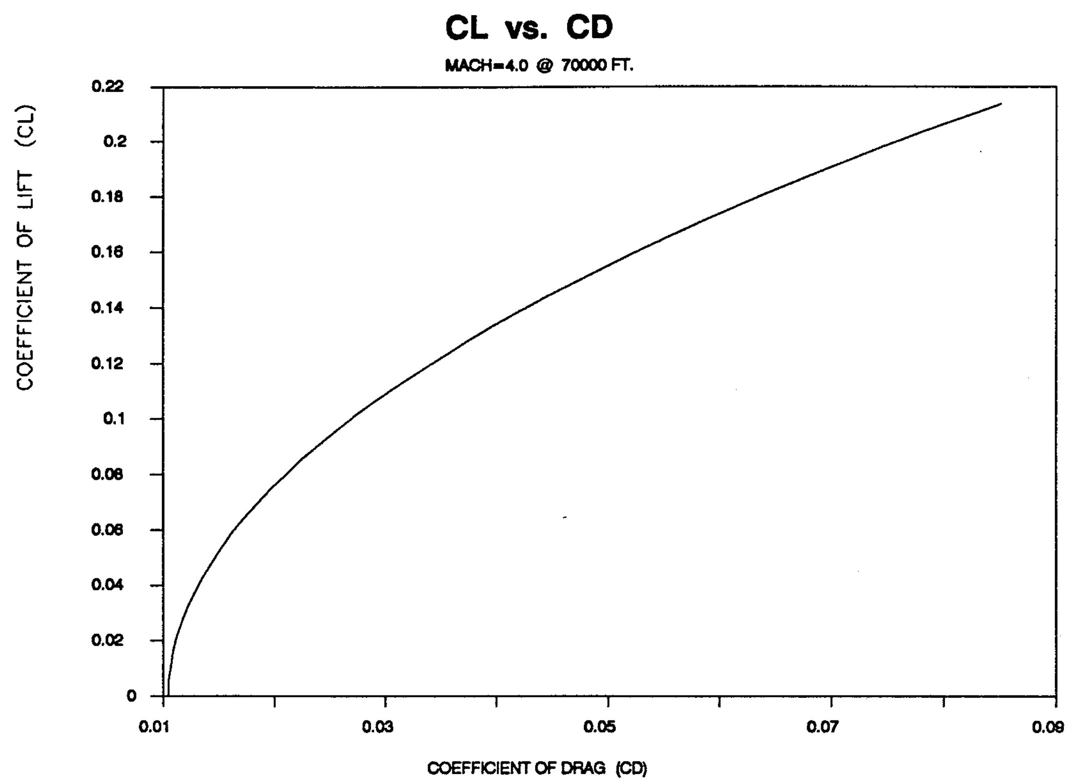

CL,CD vs. ALPHA
MACH=3.0 @ 65000 FT.
COEFFICIENT OF LIFT,DRAG - CL,CD
ANGLE OF ATTACK (DEGREES)
CL
CD

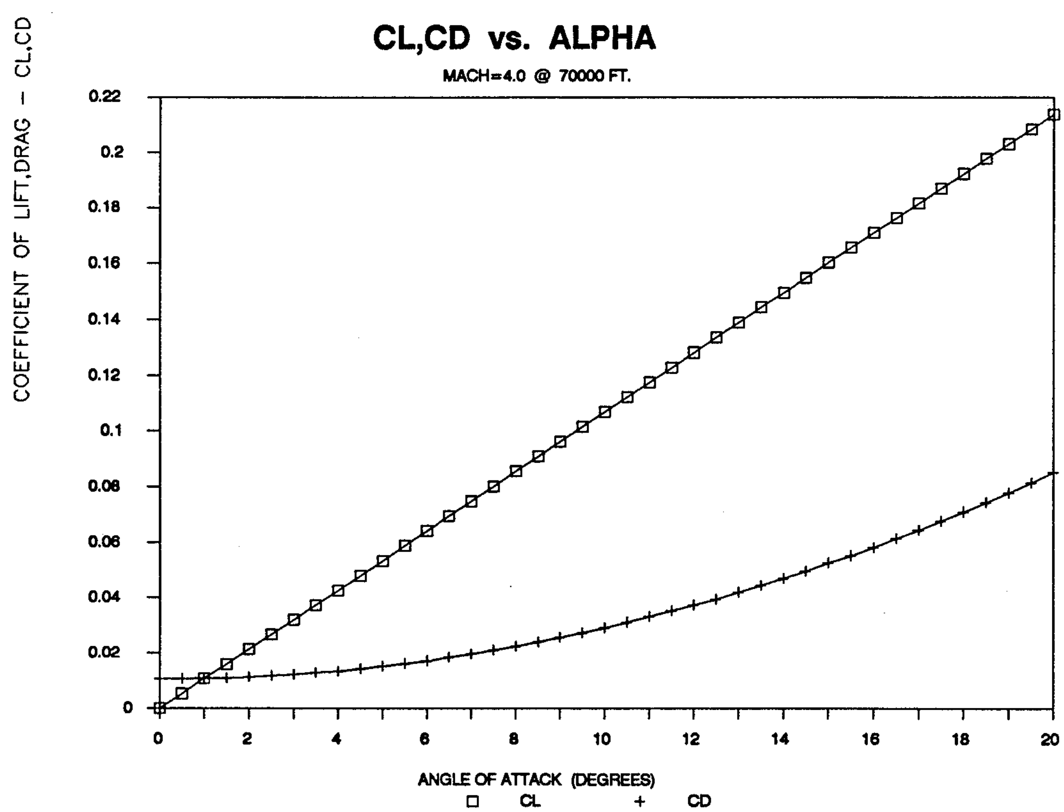

CL vs. CD
MACH=3.0 @ 65000 FT.
COEFFICIENT OF LIFT (CL)
COEFFICIENT OF DRAG (CD)

AIRFOIL DATA

MACH=4.0 AT 70000 FT

ALPHA deg	Cl	Cd	L/D	LIFT (pounds)	DRAG (pounds)
0	0	0.0105	0	0.001156	147210
0.5	0.0053	0.0105	0.507	74660	147860
1	0.0107	0.0107	1.0008	149320	149810
1.5	0.016	0.0109	1.4691	223980	153070
2	0.0214	0.0112	1.9019	298640	157630
2.5	0.0267	0.0117	2.2918	373300	163490
3	0.0321	0.0122	2.6342	447960	170660
3.5	0.0374	0.0128	2.9274	522620	179130
4	0.0427	0.0135	3.172	597280	188900
4.5	0.0481	0.0143	3.3703	671940	199980
5	0.0534	0.0152	3.5258	746600	212360
5.5	0.0588	0.0161	3.643	821260	226040
6	0.0641	0.0172	3.7265	895920	241030
6.5	0.0695	0.0184	3.7809	970580	257320
7	0.0748	0.0196	3.8106	1045200	274910
7.5	0.0801	0.021	3.8197	1119900	293800
8	0.0855	0.0224	3.8117	1194600	314000
8.5	0.0908	0.024	3.7899	1269200	335500
9	0.0962	0.0256	3.757	1343900	358300
9.5	0.1015	0.0273	3.7154	1418500	382410
10	0.1069	0.0291	3.6669	1493200	407820
10.5	0.1122	0.0311	3.6132	1567900	434530
11	0.1175	0.0331	3.5557	1642500	462550
11.5	0.1229	0.0352	3.4955	1717200	491870
12	0.1282	0.0373	3.4334	1791800	522490
12.5	0.1336	0.0396	3.3703	1866500	554410
13	0.1389	0.042	3.3067	1941200	587640
13.5	0.1443	0.0445	3.2431	2015800	622170
14	0.1496	0.047	3.1799	2090500	658010
14.5	0.1549	0.0497	3.1174	2165100	695140
15	0.1603	0.0525	3.0558	2239800	733580
15.5	0.1656	0.0553	2.9952	2314500	773330
16	0.171	0.0582	2.9359	2389100	814370
16.5	0.1763	0.0613	2.8779	2463800	856720
17	0.1817	0.0644	2.8212	2538400	900370
17.5	0.187	0.0676	2.766	2613100	945330
18	0.1923	0.0709	2.7122	2687800	991590
18.5	0.1977	0.0743	2.6599	2762400	1039200
19	0.203	0.0778	2.609	2837100	1088000
19.5	0.2084	0.0814	2.5596	2911700	1138200
20	0.2137	0.0851	2.5116	2986400	1189700

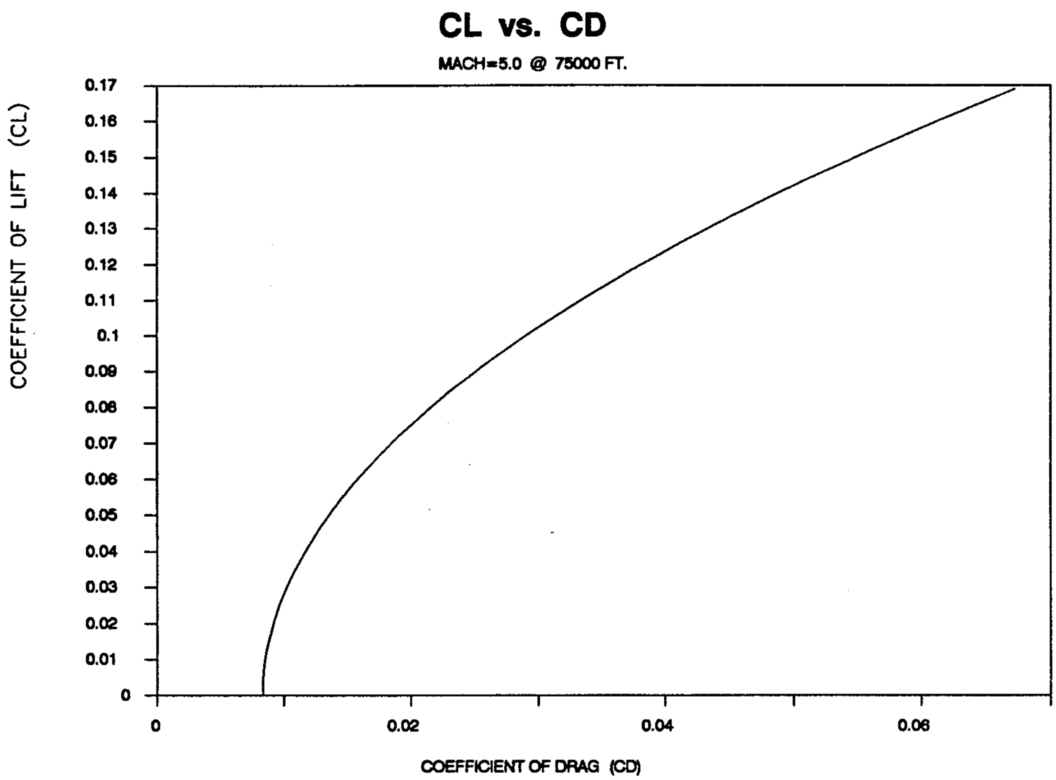

CL vs. CD
MACH=4.0 @ 70000 FT.
COEFFICIENT OF LIFT (CL)
0.22
0.2
0.18
0.16
0.14
0.12
0.1
0.08
0.06
0.04
0.02
0
0.01
0.03
0.05
0.07
0.09
COEFFICIENT OF DRAG (CD)

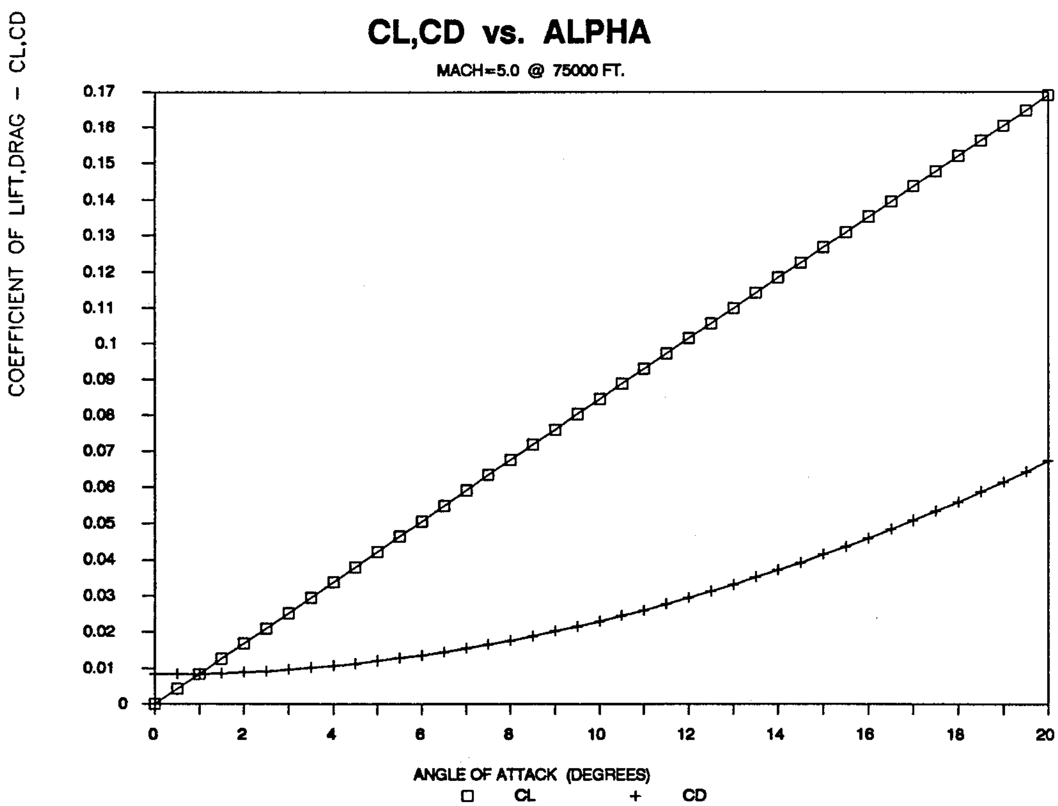

CL,CD vs. ALPHA
MACH=4.0 @ 70000 FT.
COEFFICIENT OF LIFT,DRAG – CL,CD
0.22
0.2
0.18
0.16
0.14
0.12
0.1
0.08
0.06
0.04
0.02
0
0
2
4
6
8
10
12
14
16
18
20
ANGLE OF ATTACK (DEGREES)
CL
CD

AIRFOIL DATA

MACH=5.0 AT 75000 FT

ALPHA deg	Cl	Cd	L/D	LIFT (pounds)	DRAG (pounds)
0	0	0.0083	0	0.000748	150310
0.5	0.0042	0.0083	0.5068	76196	150980
1	0.0084	0.0084	1.0003	152390	152970
1.5	0.0127	0.0086	1.4685	228590	156300
2	0.0169	0.0089	1.9011	304780	160950
2.5	0.0211	0.0092	2.2909	380980	166930
3	0.0253	0.0096	2.6332	457170	174250
3.5	0.0296	0.0101	2.9264	533370	182890
4	0.0338	0.0107	3.1709	609570	192870
4.5	0.038	0.0113	3.3692	685760	204170
5	0.0422	0.012	3.5247	761960	216800
5.5	0.0465	0.0128	3.642	838160	230770
6	0.0507	0.0136	3.7255	914350	246060
6.5	0.0549	0.0145	3.7799	990550	262680
7	0.0591	0.0155	3.8097	1066700	280640
7.5	0.0634	0.0166	3.8188	1142900	299920
8	0.0676	0.0177	3.8109	1219100	320530
8.5	0.0718	0.0189	3.7892	1295300	342480
9	0.076	0.0202	3.7564	1371500	365750
9.5	0.0803	0.0216	3.7148	1447700	390350
10	0.0845	0.023	3.6663	1523900	416280
10.5	0.0887	0.0246	3.6127	1600100	443550
11	0.0929	0.0261	3.5552	1676300	472140
11.5	0.0971	0.0278	3.495	1752500	502060
12	0.1014	0.0295	3.433	1828700	533310
12.5	0.1056	0.0313	3.3699	1904900	565890
13	0.1098	0.0332	3.3064	1981100	599810
13.5	0.114	0.0352	3.2428	2057300	635050
14	0.1183	0.0372	3.1796	2133500	671620
14.5	0.1225	0.0393	3.1171	2209700	709520
15	0.1267	0.0415	3.0555	2285900	748750
15.5	0.1309	0.0437	2.995	2362100	789310
16	0.1352	0.046	2.9356	2438300	831200
16.5	0.1394	0.0484	2.8776	2514500	874420
17	0.1436	0.0509	2.821	2590700	918970
17.5	0.1478	0.0535	2.7658	2666900	964850
18	0.1521	0.0561	2.712	2743000	1012100
18.5	0.1563	0.0588	2.6597	2819200	1060600
19	0.1605	0.0615	2.6089	2895400	1110500
19.5	0.1647	0.0644	2.5595	2971600	1161700
20	0.169	0.0673	2.5115	3047800	1214200

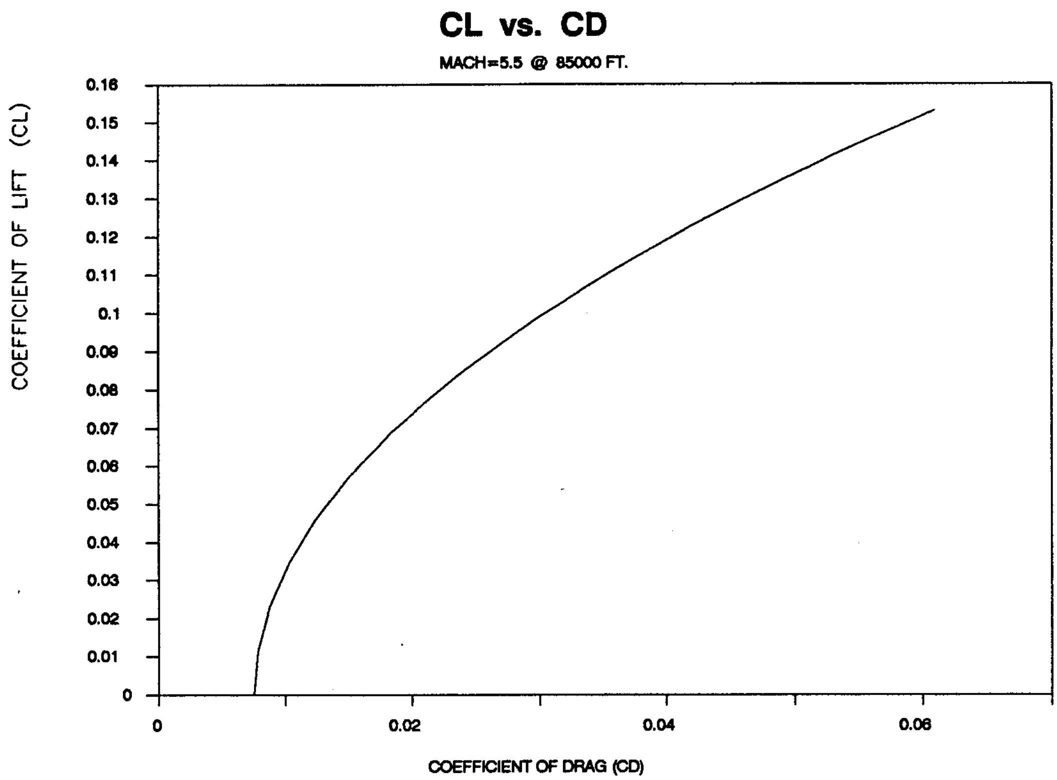

CL vs. CD
MACH=5.0 @ 75000 FT.
COEFFICIENT OF LIFT (CL)
0.17
0.16
0.15
0.14
0.13
0.12
0.11
0.1
0.09
0.08
0.07
0.06
0.05
0.04
0.03
0.02
0.01
0
0
0.02
0.04
0.06
COEFFICIENT OF DRAG (CD)

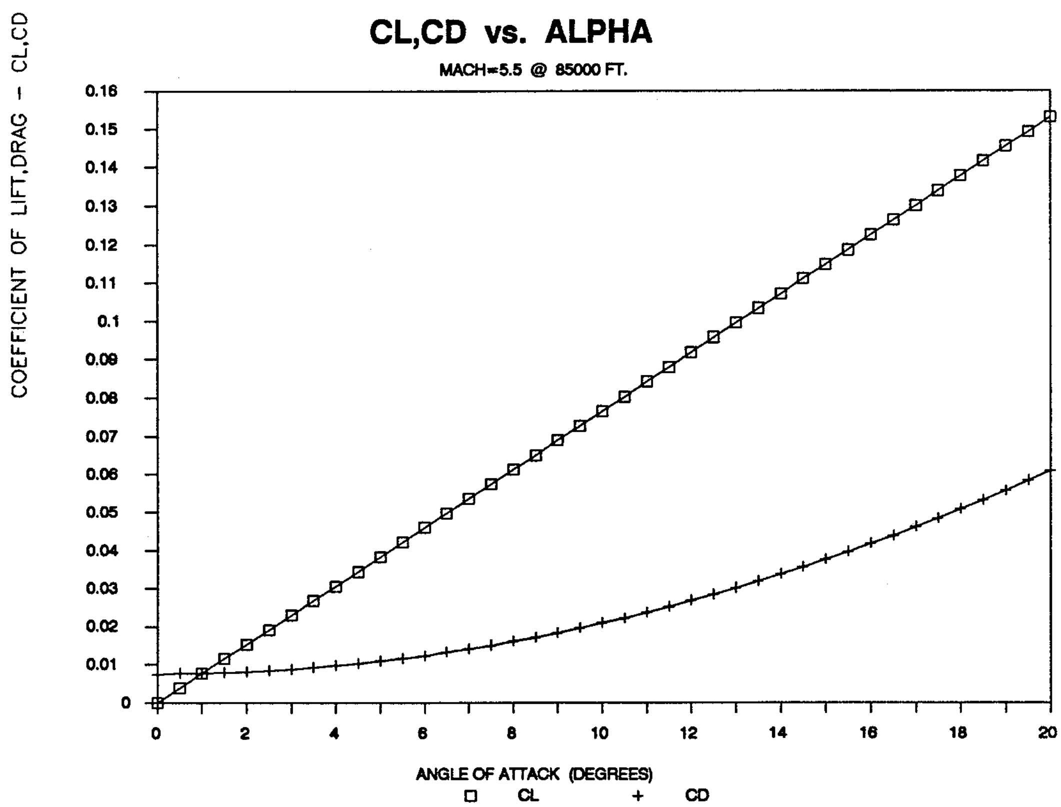

CL,CD vs. ALPHA
MACH=5.0 @ 75000 FT.
COEFFICIENT OF LIFT,DRAG - CL,CD
0.17
0.16
0.15
0.14
0.13
0.12
0.11
0.1
0.09
0.08
0.07
0.06
0.05
0.04
0.03
0.02
0.01
0
0
2
4
6
8
10
12
14
16
18
20
ANGLE OF ATTACK (DEGREES)
CL
CD

AIRFOIL DATA

MACH=5.5 AT 85000 FT

ALPHA deg	Cl	Cd	L/D	LIFT (pounds)	DRAG (pounds)
0	0	0.0075	0	0.000927	96509
0.5	0.0038	0.0076	0.5063	48917	96936
1	0.0077	0.0077	0.9994	97834	98217
1.5	0.0115	0.0078	1.4671	146750	100350
2	0.0153	0.0081	1.8994	195670	103340
2.5	0.0191	0.0084	2.2889	244590	107180
3	0.023	0.0087	2.6311	293500	111880
3.5	0.0268	0.0092	2.9241	342420	117430
4	0.0306	0.0097	3.1686	391340	123830
4.5	0.0344	0.0102	3.3668	440250	131090
5	0.0383	0.0109	3.5224	489170	139200
5.5	0.0421	0.0116	3.6397	538090	148160
6	0.0459	0.0123	3.7233	587010	157980
6.5	0.0497	0.0132	3.7779	635920	168650
7	0.0536	0.0141	3.8078	684840	180180
7.5	0.0574	0.015	3.817	733760	192560
8	0.0612	0.0161	3.8093	782670	205790
8.5	0.065	0.0172	3.7876	831590	219880
9	0.0689	0.0183	3.7549	880510	234820
9.5	0.0727	0.0196	3.7134	929420	250610
10	0.0765	0.0209	3.6651	978340	267260
10.5	0.0803	0.0222	3.6115	1027300	284760
11	0.0842	0.0237	3.5541	1076200	303120
11.5	0.088	0.0252	3.494	1125100	322330
12	0.0918	0.0268	3.4321	1174000	342390
12.5	0.0957	0.0284	3.3691	1222900	363310
13	0.0995	0.0301	3.3056	1271800	385080
13.5	0.1033	0.0319	3.2421	1320800	407710
14	0.1071	0.0337	3.1789	1369700	431180
14.5	0.111	0.0356	3.1165	1418600	455520
15	0.1148	0.0376	3.0549	1467500	480700
15.5	0.1186	0.0396	2.9944	1516400	506740
16	0.1224	0.0417	2.9351	1565300	533640
16.5	0.1263	0.0439	2.8772	1614300	561380
17	0.1301	0.0461	2.8206	1663200	589980
17.5	0.1339	0.0484	2.7654	1712100	619440
18	0.1377	0.0508	2.7117	1761000	649750
18.5	0.1416	0.0532	2.6594	1809900	680910
19	0.1454	0.0557	2.6085	1858800	712930
19.5	0.1492	0.0583	2.5591	1907800	745800
20	0.153	0.0609	2.5112	1956700	779520

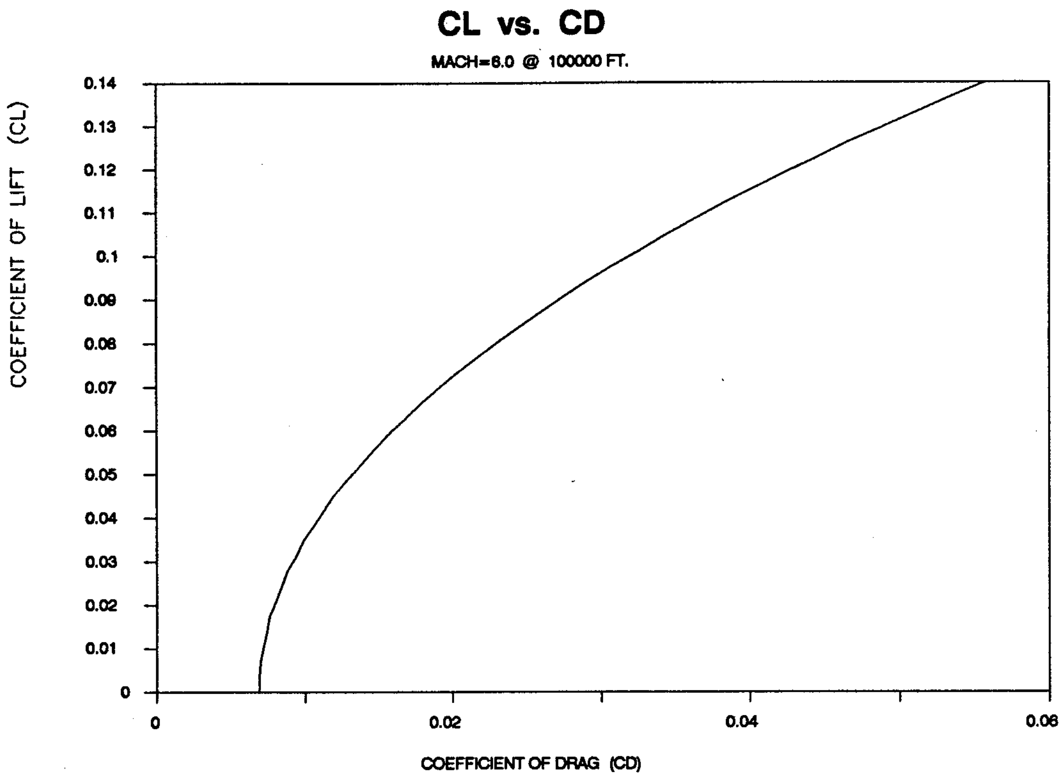

CL vs. CD
MACH=5.5 @ 85000 FT.
COEFFICIENT OF LIFT (CL)
COEFFICIENT OF DRAG (CD)
0.16
0.15
0.14
0.13
0.12
0.11
0.1
0.09
0.08
0.07
0.06
0.05
0.04
0.03
0.02
0.01
0
0
0.02
0.04
0.06

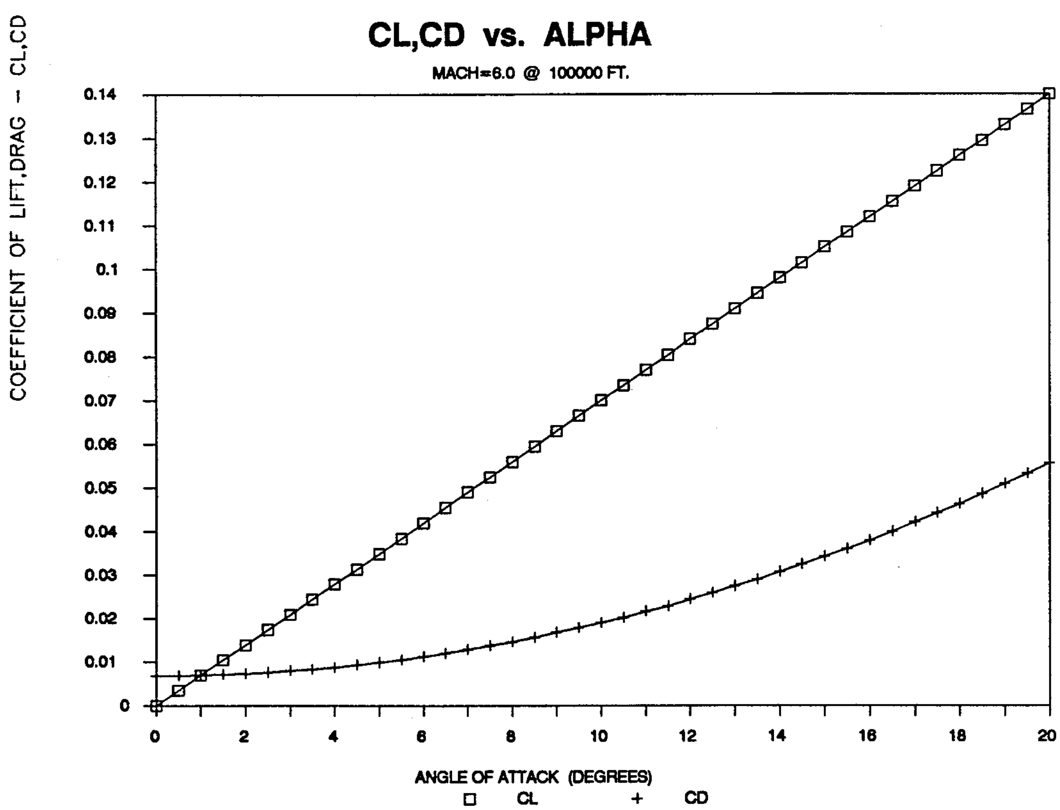

CL,CD vs. ALPHA
MACH=5.5 @ 85000 FT.
COEFFICIENT OF LIFT,DRAG – CL,CD
ANGLE OF ATTACK (DEGREES)
□ CL
+ CD
0.16
0.15
0.14
0.13
0.12
0.11
0.1
0.09
0.08
0.07
0.06
0.05
0.04
0.03
0.02
0.01
0
0
2
4
6
8
10
12
14
16
18
20

AIRFOIL DATA

MACH=6.0 AT 100000 FT

ALPHA deg	Cl	Cd	L/D	LIFT (pounds)	DRAG (pounds)
0	0	0.0069	0	0.000632	53608
0.5	0.0035	0.0069	0.5057	27108	53845
1	0.007	0.007	0.9983	54217	54554
1.5	0.0105	0.0072	1.4655	81325	55737
2	0.014	0.0074	1.8973	108430	57393
2.5	0.0175	0.0076	2.2865	135540	59522
3	0.021	0.008	2.6284	162650	62125
3.5	0.0245	0.0084	2.9213	189760	65200
4	0.028	0.0088	3.1657	216870	68748
4.5	0.0315	0.0094	3.3639	243980	72770
5	0.035	0.0099	3.5196	271080	77265
5.5	0.0385	0.0106	3.637	298190	82233
6	0.042	0.0113	3.7207	325300	87674
6.5	0.0455	0.012	3.7754	352410	93588
7	0.049	0.0129	3.8054	379520	99975
7.5	0.0525	0.0138	3.8148	406630	106840
8	0.056	0.0147	3.8072	433740	114170
8.5	0.0595	0.0157	3.7857	460840	121980
9	0.063	0.0168	3.7531	487950	130260
9.5	0.0665	0.0179	3.7117	515060	139010
10	0.07	0.0191	3.6635	542170	148230
10.5	0.0735	0.0203	3.6101	569280	157930
11	0.0769	0.0217	3.5528	596390	168110
11.5	0.0804	0.023	3.4928	623490	178750
12	0.0839	0.0245	3.431	650600	189870
12.5	0.0874	0.026	3.368	677710	201460
13	0.0909	0.0275	3.3046	704820	213530
13.5	0.0944	0.0291	3.2412	731930	226060
14	0.0979	0.0308	3.1781	759040	239080
14.5	0.1014	0.0326	3.1157	786150	252560
15	0.1049	0.0344	3.0542	813250	266520
15.5	0.1084	0.0362	2.9938	840360	280950
16	0.1119	0.0381	2.9345	867470	295850
16.5	0.1154	0.0401	2.8766	894580	311230
17	0.1189	0.0422	2.82	921690	327080
17.5	0.1224	0.0443	2.7649	948800	343400
18	0.1259	0.0464	2.7112	975900	360200
18.5	0.1294	0.0487	2.6589	1003000	377470
19	0.1329	0.051	2.6081	1030100	395210
19.5	0.1364	0.0533	2.5588	1057200	413420
20	0.1399	0.0557	2.5108	1084300	432110


CL vs. CD
MACH=6.0 @ 100000 FT.
COEFFICIENT OF LIFT (CL)
0.14
0.13
0.12
0.11
0.1
0.09
0.08
0.07
0.06
0.05
0.04
0.03
0.02
0.01
0
0
0.02
0.04
0.06
COEFFICIENT OF DRAG (CD)


CL,CD vs. ALPHA
MACH=6.0 @ 100000 FT.
COEFFICIENT OF LIFT,DRAG - CL,CD
0.14
0.13
0.12
0.11
0.1
0.09
0.08
0.07
0.06
0.05
0.04
0.03
0.02
0.01
0
0
2
4
6
8
10
12
14
16
18
20
ANGLE OF ATTACK (DEGREES)
CL
CD

COMPUTATIONAL FLUID DYNAMICS METHOD

INTRODUCTION

Various methods exist for determination of lift and drag characteristics for an airfoil in subsonic and supersonic flow. Linearized flow was chosen to model the flowfield around a particular airfoil section. Because of the complexity of the flowfield around a wing in both subsonic and supersonic flow, the linearized flow technique was applied to particular airfoil sections of the wing. Linearized flow cannot be applied when the Mach number is close to one due to the formation of shock waves which the flow technique neglects, and therefore other methods were used to calculate lift and drag at Mach numbers below 1.2.

GOALS

The goals for the computational fluid dynamics program were as follows:

1) The ability to compute total lift and drag on the wing of the booster at Mach numbers greater than 1.2.
2) The ability to yield reasonable results for a conceptual design.
3) A small amount of variation from the program written for this project and the model algorithm in ref. 3, pp. 391-401.

DISCUSSION

The program developed for use with project required the airfoil section to be entered in. Since the linearized flow technique depends only on the inclination of the surface of the airfoil to the incident flow velocity, the size of the airfoil does not enter into the calculations for lift and drag coefficients. However, to determine the actual lift and drag, the actual dimensions of the wing must be entered. The program requires as input an x distance along the chord as a fraction of the total chord length. The height of the top and bottom of the airfoil is also entered as fractions of the chord length. The airfoil section does not have to be symmetrical as the program requires both top and bottom values of airfoil height. The data can either be entered from an input file called "IDATA" or entered manually point by point. The airfoil sections were drawn in Autocad v10.0 and the mathematically exact points entered into a data file to be inputed into the program.

The program also requires as input the freestream static conditions. These conditions are:

MACH NUMBER
STATIC PRESSURE
STATIC TEMPERATURE
RATIO OF SPECIFIC HEATS
UNIVERSAL GAS CONSTANT
VISCOSITY

The program calculates the total lift and drag on the wing by summing up the section lifts and drags by using similar airfoil sections. It calculates the lift and drag on an airfoil section of unit depth starting at the exposed root chord. It then uses a linear model to determine the new airfoil parameters. The planform of the wing must be entered to determine the lift and drag. The planform parameters required are:

TOTAL WING REFERENCE AREA
TOTAL WING EXPOSED AREA
MEAN AERODYNAMIC CHORD LENGTH
EXPOSED WING ROOT CHORD LENGTH
WING TIP CHORD LENGTH
EXPOSED WING HALF SPAN
TOTAL WING TWIST

A linear wing twist model is also incorporated if a wing twist is desired.

The program also calculates chordwise pressures. It then averages the chordwise pressures and calculates the spanwise pressure distribution. If desired, either actual pressure of pressure coefficient data can be saved to a file.

The program first calculates the data for an airfoil section at a particular angle of attack. It then varies the chordwise dimensions and calculates the total lift on the wing. The angle of attack is then varied and the data is again calculated at the particular angle of attack. The angle of attack is varied from an initial to a final user inputed value in increments of 0.5 degrees.

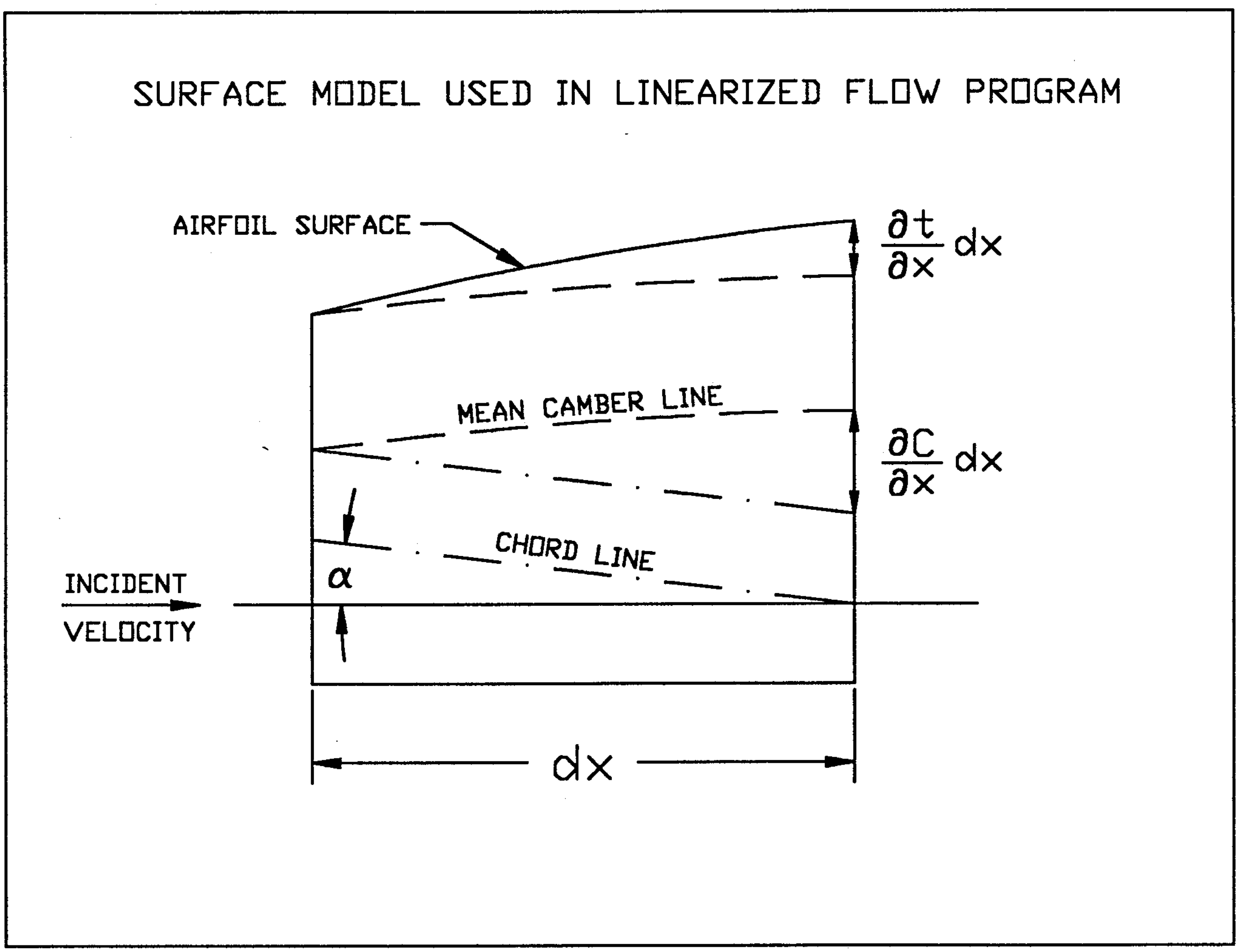
The process of airfoil selection was determined by designing some various airfoil section according to some qualitative parameters. The airfoil section was then entered into the program. The planform data was entered so that only one airfoil section of unit depth was evaluated. The data from this was compared to other airfoil sections of unit depth. The optimum airfoil was then selected. This unit depth comparison was done due to the large number of calculations involved in the program and the large number of times the computer had to write to disk. The program was written in Fortran and run on a Zenith 386 computer running at 20 MHZ. Refer to the appendix for specifications on linearized flow model.

EVALUATION OF PROGRAM

The program was run using the airfoil and conditions from the program in ref. 3. The data from the program for the drag coefficient was found to have good agreement to the program in ref. 3. The program also gave good results for lift coefficient, however, the program did overestimate the lift coefficient of the model by approximately 10 percent. In determining lift and drag coefficients for the actual airfoil chosen for the booster, the lift to drag ratio seemed to be quite low. The maximum lift to drag ratio was approximately 50 percent of the value in ref. 5, especially at higher altitudes and Mach numbers.

CONCLUSIONS

The linearized flow program did meet the design goals for this conceptual design project. However, there is a question of the accuracy of this program at higher altitudes and Mach numbers. A further investigation of the linearized flow technique must be done in order to determine its accuracy in this booster design project.


SURFACE MODEL USED IN LINEARIZED FLOW PROGRAM
AIRFOIL SURFACE
$\frac{\partial t}{\partial x} dx$
MEAN CAMBER LINE
$\frac{\partial C}{\partial x} dx$
CHORD LINE
INCIDENT
VELOCITY
α
dx

```
 program aero
   real GAMMA,PRESS,COEFF1,COEFF2,COEFF3,ALPHA,DCDX,DTDXU,DTDXL,DRAG,
+    LIFT,CD,CL,LD,DX,X(100),YU(100),YL(100),TWIST,ROOT,TAPER,
+    TIP,CPU,CPL,PU,PL,MACH,CHORD,MCHORD,VIS,DEN,C,R
  double precision D1,D2
   integer N,I,J,PNUM,INIA,MAXA,B,SURF,SWET,SPAN,O
   open(1,file='idata')
   open(2,file='odata.dat')
   open(3,file='cpress.dat')
   open(4,file='spress.dat')
   write(*,*) ' LINEARIZED FLOW PROGRAM'
   write(*,*) ' Please note that the first and nth points must have'
   write(*,*) ' both upper and lower y values of zero. Units of the'
   write(*,*) ' x and y scales must be identical.  They are'
   write(*,*) ' normalized with the chord.  Consistent units'
   write(*,*) ' should be used.'
   write(*,*) ' Number of points per surface up to 100.'
   write(*,*) ' '
   write(*,*) ' ENTER FREESTREAM MACH NUMBER: '
   read(*,*) MACH
   write(*,*) ' ENTER FREESTREAM STATIC PRESSURE: '
   read(*,*) PRESS
   write(*,*) ' ENTER GAMMA: '
   read(*,*) GAMMA
   write(*,*) ' ENTER FREESTREAM STATIC DENSITY: '
   read(*,*) DEN
   write(*,*) ' ENTER VISCOSITY OF THE AIR: '
   read(*,*) VIS
   write(*,*) ' ENTER SPEED OF SOUND AT ALTITUDE: '
   read(*,*) C
   write(*,*) ' ENTER INITIAL ANGLE OF ATTACK (DEGREES): '
   read(*,*) INIA
   write(*,*) ' ENTER MAX ANGLE OF ATTACK (DEGREES): '
   read(*,*) MAXA
   write(*,*) ' THE TOTAL LIFT AND DRAG IS CALCULATED BY USING'
   write(*,*) ' SIMILAR AIRFOIL SECTIONS'
   write(*,*) ' '
   write(*,*) ' ENTER TOTAL WING REFERENCE AREA: '
   read(*,*) SURF
   write(*,*) ' ENTER TOTAL WING EXPOSED AREA: '
   read(*,*) SWET
   SWET=2.003*SWET
   write(*,*) ' ENTER MEAN AERODYNAMIC CHORD LENGTH: '
   read(*,*) MCHORD
   R=(DEN*MACH*C*MCHORD)/VIS
   write(*,*) ' ENTER EXPOSED WING ROOT CHORD LENGTH: '
   read(*,*) ROOT
```

```
      write(*,*) ' ENTER WING TIP CHORD: '
      read(*,*) TIP
      write(*,*) ' ENTER WING EXPOSED HALF SPAN: '
      read(*,*) SPAN
      TAPER=TIP/ROOT
      write(*,*) ' ENTER TOTAL WING TWIST FROM ROOT TO TIP (DEGREES)'
      write(*,*) ' POSITIVE TWIST DECREASES ANGLE OF ATTACK FROM'
      write(*,*) ' WING ROOT TO WING TIP: '
      read(*,*) TWIST
      TWIST=TWIST/180.0*3.14159
      write(*,*) ' '
      write(*,*) ' ENTER 1 IF YOU WOULD LIKE PRESS COEFF DATA SAVED TO'
      write(*,*) ' A FILE, OTHERWISE DATA WILL BE IN PRESSURE: '
      read(*,*) O
      write(*,*) ' WOULD YOU LIKE TO ENTER AIRFOIL DATA FROM A FILE? '
      write(*,*) ' ENTER  (1)  FOR YES.  FILE=IDATA: '
      read(*,*) RESP
      if (RESP .ne. 1) then
        goto 1
      endif
      read(1,*) N
      do 10 I=1,N
        read(1,*) PNUM,X(I),YU(I),YL(I)
        YL(I)=-YL(I)
10    continue
      goto 20
****
**** input surfaces
****
1      write(*,*) ' ENTER NUMBER OF POINTS PER SURFACE: '
      read(*,*) N
      do 15 I=1,N
        write(*,*) ' POINT NUMBER: ',I
        write(*,*) ' '
        write(*,*) ' ENTER X LOCATION IN FRACTIONS OF CHORD LENGTH: '
        read(*,*) X(I)
        write(*,*) ' ENTER UPPER SURFACE Y LOCATION (Y/CHORD): '
        read(*,*) YU(I)
        write(*,*) ' ENTER LOWER SURFACE Y LOCATION (Y/CHORD): '
        read(*,*) YL(I)
        YL(I)=-YL(I)
15    continue
*****
20    COEFF1=GAMMA*PRESS*MACH**2./(MACH**2.-1.)**.5
      COEFF2=(0.5*GAMMA*PRESS*MACH**2.)
      COEFF3=2./((MACH**2.-1.)**0.5)
****
```

```
      CHORD=ROOT
      do 300 J=INIA,(MAXA*2)
       write(4,*) ' '
       ALPHA=J*1./2.*(3.14159/180.)
       DRAG=0.
       LIFT=0.
       CL=0.
       CD=0.
       CPAU=0.
       CPAL=0.
       PAU=0.
       PAL=0.
       do 200 B=SPAN,0,-1
        CHORD=((ROOT-TIP)/(SPAN*1.))*B*1.+TIP
        ALPHA=ALPHA+((TWIST/(SPAN*1.))*B*1.-TWIST)
        do 100 I=2,N
         DX=(X(I)-X(I-1))
         DCDX=((YU(I)-YL(I))/2.-(YU(I-1)-YL(I-1))/2.)/DX
         DTDXU=((YU(I)-YU(I-1))/DX)-DCDX
         DTDXL=((YL(I)-YL(I-1))/DX)-DCDX
         D1=(-ALPHA+DTDXU+DCDX)*(-ALPHA+DTDXU+DCDX)
         D2=(-ALPHA+DTDXL+DCDX)*(-ALPHA+DTDXL+DCDX)
         DRAG=DRAG+((D1+D2)*COEFF1*DX*CHORD)
         CPU=COEFF3*(-ALPHA+DTDXU+DCDX)
         CPL=-COEFF3*(-ALPHA+DTDXL+DCDX)
         PU=CPU*COEFF2+PRESS
         PL=CPL*COEFF2+PRESS
         CPAU=CPAU+CPU
         CPAL=CPAL+CPL
         PAU=PAU+PU
         PAL=PAL+PL
         if (O .eq. 1) then
          write(3,51) CPU,CPL
         else
          write(3,51) PU,PL
         endif
100      continue
        CPAU=CPAU/N
        CPAL=CPAL/N
        PAU=PAU/N
        PAL=PAL/N
        write(3,*) ' '
        if (O .eq. 1) then
         write(4,51) CPAU,CPAL
        else
         write(4,51) PAU,PAL
        endif
```

```
      LIFT=LIFT+(2.*ALPHA*CHORD*2.*COEFF2)/(MACH**2.-1.)**0.5
      CPAU=0.
      CPAL=0.
      PAU=0.
      PAL=0.
200    continue
     CL=(2*LIFT)/(COEFF2*SURF)
     CD=(2*DRAG)/(COEFF2*SURF)
     CF=0.455/((log10(R))**2.58*(1.0+0.144*MACH**2.)**0.65)
     CD=(CF*SWET)/SURF+CD
     if (CD .eq. 0.) then
      CD=10000.
     endif
     LD=LIFT/DRAG
     ALPHA=J/2.
     write(3,*) ' '
     write(2,50) ALPHA,CL,CD,LD,LIFT,DRAG
300  continue
50   format(F4.1,3(5X,F7.4),2(5X,E14.5))
51   format(F12.4,5x,F12.4)
    PRINT *,' DONE!!! '
    stop
    end
```

STABILITY ANALYSIS

VEHICLE STABILITY AND CONTROL

The booster/orbiter combination during the assent does not lend itself to a simple static stability analysis due to the fact that the vehicle must fly through Mach numbers 0-6. Many factors unique to this vehicle, which do not enter with more conventional vehicles, must be considered. The goals set for the stability of the vehicle:

1) Positive static stability (longitudinal, directional, lateral)
2) Positive dynamic stability
3) Reduced static margin (low trim drag)
4) Minimum effect on booster and orbiter stability after separation
5) Ability to trim out adverse effects (compensate for failures)

A variety of unique and conventional features are designed into the vehicle to allow these goal to be met.

LONGITUDINAL STATIC STABILITY

The following features were considered in the static longitudinal stability analysis.

- lift produced by the forward body at supersonic speeds due to the expansion above the nose and the inlet precompression below the nose.
- lift produced by the LEX or chine. As with the forward body, the moment produced is more significant than the actual lift.
- lift produced by the orbiter riding on the back of the booster. Since the lift is produced aft of the overall CG it contributes a pitch down moment.
- drag produced by the orbiter riding on the back of the booster. Since the drag is produced above the overall CG it contributes a pitch up moment.
- inlet and engine thrust produced. These produce a helpful pitch up moment.
- vertical thrust force produced by the nozzle. Since the entire lower aft portion of the booster is a half nozzle, the effects of over and under expansion of the nozzle produces a pitching moment. The effect cannot

be intuitively determined without analysis.

- horizontal thrust force produced by the half nozzle. As with the vertical force, it must be determined by analysis. The contribution to the pitching moment is probably less than that contributed by the vertical nozzle.
- elevon and elevator trim angles. The effect of tail incidence variation.

In order to determine the longitudinal static stability throughout the flight envelope, a program was written to perform the stability calculations. The details of the program are given in the appendix to this section.

The aerodynamic center of the booster/orbiter moves aft approximately 8% of the total booster length at supersonic speeds. This phenomena of high speed flight limits the maximum speed of airplanes due to the need to trim out the pitching moment produced. Shifting of the center of gravity aft by using fuel from the forward fuel tanks first and pumping the fuel in a schedule so as move the CG aft during the supersonic flight portions will minimize the effects of the aft moving aerodynamic center resulting in lower trim drag.

The forward body of the booster vehicle produces positive lift at supersonic speeds due to the expansion above the nose and inlet precompression below. The lower portion of the very front of the nose is sloped upward as with the SR-71 (see REF 8). These provide positive pitching moments. At subsonic speeds, the lift produced by the body can be ignored but it has a substantial pitchup moment at the maximum angle of attack at takeoff.

The leading edge extension (LEX) is sloped upward from the wing toward the forward fuselage a total of 3 degrees. From the data in REF 6, the angle of attack added to the LEX incidence will cause the LEX to have a net positive lift and will operate close to its zero pitching moment angle of attack. The LEX will provide a positive pitching moment to offset the rearward shifting aerodynamic center. The pitching moment of the LEX is more significant than the total lift it produces.

The orbiter produces lift and drag riding on the back of the booster these provide negative and positive moments respectively. The lifting capability of the orbiter is severely limited due the loss in the underside exposure. The lift-slopes at all Mach numbers were considered to be half the value as that in free flight. The orbiter lifting surface is a weak parameter in determining the overall vehicle neutral point. An aerodynamic fairing on the orbiter propulsion system would reduce the overall boattail drag and would also decrease its contribution to the pitching moment.

The inlet thrust, assumed to be in line with the direct engine thrust, combined with this direct engine thrust will produce a substantial positive moment. This turns out to be very useful so the elevons have to provide less pitching moment therefore producing less trim drag. Since this thrusts provides the positive moment, it is possible to remove the other devices that produce positive pitching moments at supersonic speeds such as the LEX. However, the pitching moment will not be provided if the engines fail or reduce their power to idle. Therefore, poweroff moment balance must exist.

The vertical component of the nozzle thrust increases as the nozzle design point is approached. Since the nozzle is probably designed for the higher Mach numbers, nozzle thrust will increase continuously during the assent therefore an increasing pitch down moment will be encountered. This pitching moment will be offset primarily by the increasing effectiveness of the LEX and forward body lift.

In order to compensate for the possible varying trim conditions during power off and power on configurations, the horizontal tails were positioned at the wingtips. The incidence variation makes for a very effective control surface capable of trimming for various irregularities such as premature separation, separation difficulties, and varying power conditions.

LATERAL AND DIRECTIONAL STATIC STABILITY

The dihedral effect of such a large sweepback requires the use of zero or possible a slight negative dihedral to avoid Dutch-roll tendencies. However, the dihedral effect of sweepback decreases with Mach number and at Mach 6 the effect is quite small (see REF 9). With separation stability being very critical, 3 degrees of positive dihedral was designed into the vehicle. Additionally, with the orbiter positioned on top, the booster/orbiter combination is considered to be a low wing. The sweepback effect provides the dihedral needed for the low wing configuration. The aerodynamic center of the vertical tails, being above the CG, provide lateral stability.

Returning to land, the combined effect of sweepback, positive dihedral, low wing loading, and lower CG has the adverse effect of making the landing a tricky situation with an excessive Dutch-roll tendency. The use of active flight controls coupled with yaw dampers should help offset these effects.

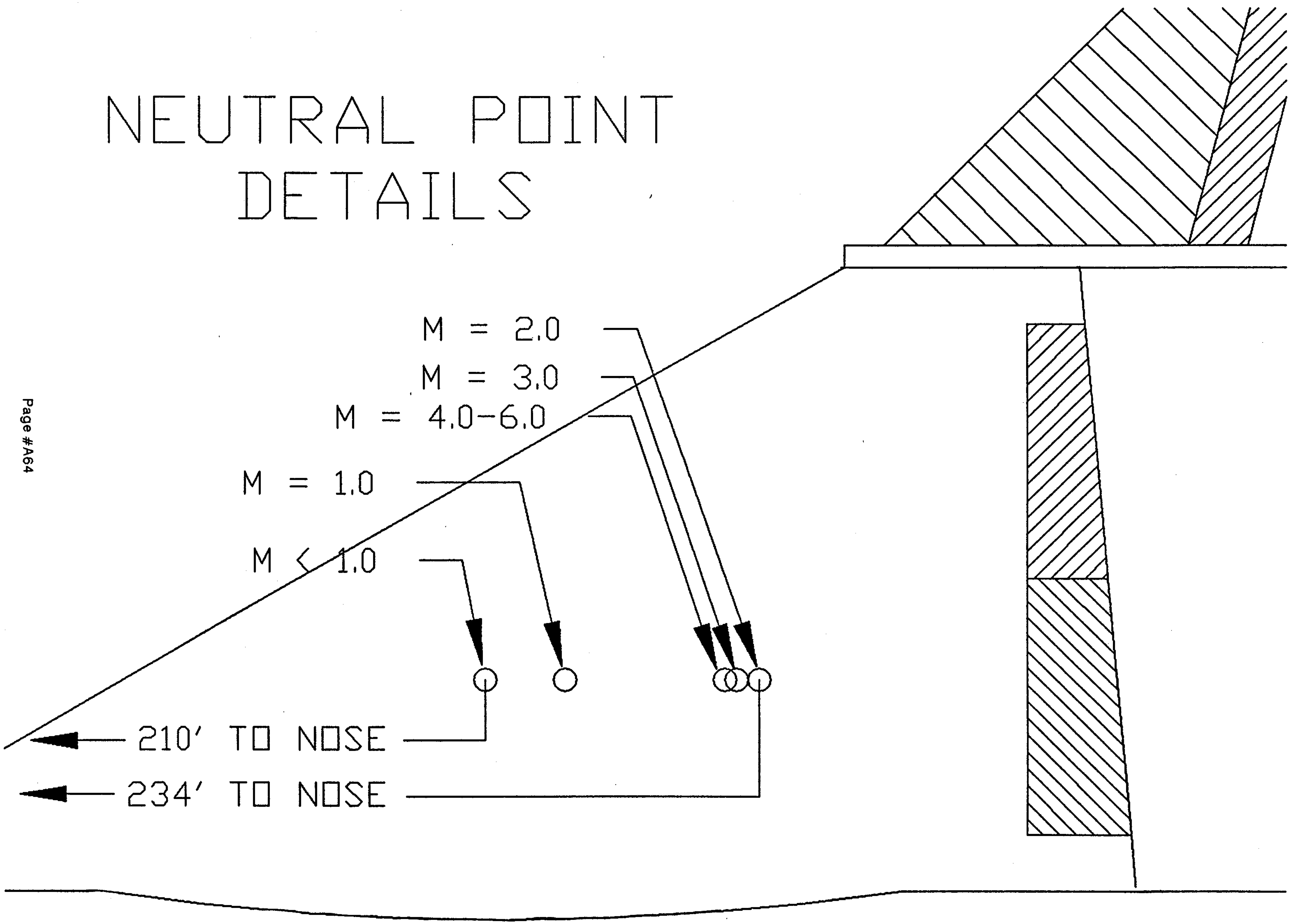
The twin vertical tails were sized using data from current high speed aircraft such as the SR-71 and the XB-70 as well as conceptual design of hypersonic vehicles (see REF 8,10). The large nose of the front of the vehicle produces a substantial yawing moment that must be compensated for by the vertical tails. Additional vertical tail area is added below the wing on both wings. Since the orbiter must have an adequate sized vertical tail for its own stability, directional stability of the booster was analyzed without the orbiter.

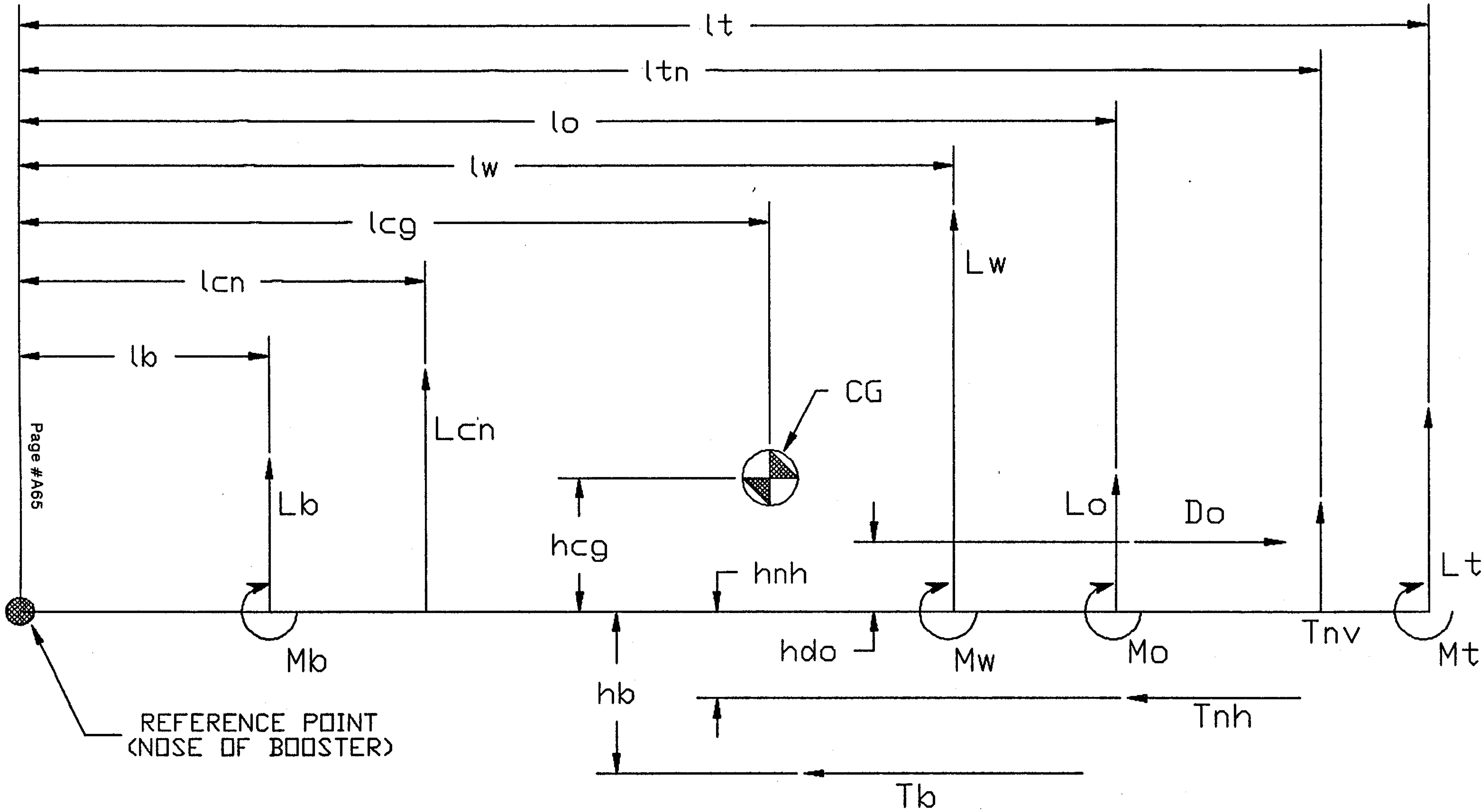
All moving vertical tails for increased control power was rejected due to lack of large single-engine-out moments. The rudders are in two parts. The top rudder is used subsonically to avoid interference with the all-moving horizontal tail. Supersonically, the bottom rudder is used, since the horizontal tails are not used for pitch control. The problem similar to "aileron reversal" for the vertical tails is eliminated with the use of the split rudder (see REF 10).

SEPARATION

The positioning of the orbiter near the CG of the booster, allows for minimum effect on the stability of both vehicles upon separation. The booster would "unload" and enter a zero-g pushover with the engines still producing substantial thrust. The orbiter with its now greater lift-to-weight and greater drag would up and back relative to the booster for a relatively clean separation.

NEUTRAL POINT DETAILS





ASSUMPTIONS: BOOSTER DRAG MOMENT IS SMALL
NO ORBITER THRUST UNTIL AFTER SEPARATION

MOMENT ABOUT C.G.

MOMENT ABOUT CG

$$L_B(l_{cg}-l_b) + L_{CN}(l_{cg}-l_{CN}) - L_W(l_w-l_{cg}) - L_O(l_O-l_{cg}) - L_T(l_t-l_{cg}) + M_W + M_T + M_B + M_O$$
$$+ T_B(h_B+h_{cg}) + T_{NH}(h_{NH}+h_{cg}) - D_O(h_{cg}-h_{OO}) - T_{NV}(l_{TN}-l_{cg}) = \text{MOMENT ABOUT CG}$$

NONDIMENSIONALIZE USING S_{REF}, ρ_∞, V_∞, MEAN AERODYNAMIC CHORD

$$\frac{L_B}{q S_{ref}}\left(\frac{l_{cg}-l_b}{c}\right) + \frac{L_{CN}}{q S_{ref}}\left(\frac{l_{cg}-l_{CN}}{c}\right) - C_{LW}\left(\frac{l_w-l_{cg}}{c}\right) - C_{LO}\left(\frac{S_O}{S_{ref}}\right)\left(\frac{l_O-l_{cg}}{c}\right) - C_{LT}\left(\frac{S_T}{S_{ref}}\right)\left(\frac{l_T-l_{cg}}{c}\right) + C_{MW}$$
$$+ C_{MT}\left(\frac{S_T\bar{C}_T}{S_{ref}C}\right) + \frac{M_B}{q S_{ref} c} + C_{MO}\left(\frac{S_O\bar{C}_O}{S_{ref}C}\right) + \frac{T}{q S_{ref}}\left(\frac{h_B+h_{cg}}{c}\right) + \frac{T_{NH}}{q S_{ref}}\left(\frac{h_{NH}+h_{cg}}{c}\right) - \frac{D_O}{q S_{ref}}\left(\frac{h_{cg}-h_{OO}}{c}\right) \qquad (1)$$
$$- \frac{T_{NV}}{q S_{ref}}\left(\frac{l_{TN}-l_{cg}}{c}\right) = C_{m\,cg}$$

KEEPING THE FIRST 2 TERMS AND LAST TERMS AS THEY ARE

$$\cdots\cdots - a_W\alpha\left(\frac{l_w-l_{cg}}{c}\right) - a_O\alpha\left(\frac{S_O}{S_{ref}}\right)\left(\frac{l_O-l_{cg}}{c}\right) - a_T(\alpha-i)\left(\frac{S_T}{S_{ref}}\right)\left(\frac{l_T-l_{cg}}{c}\right) + \cdots\cdots \qquad (2)$$

PITCHING MOMENT DERIVATIVE

$$\frac{\partial C_{m\,cg}}{\partial\alpha} = -a_W\left(\frac{l_w-l_{cg}}{c}\right) - a_O\left(\frac{S_O}{S_{ref}}\right)\left(\frac{l_O-l_{cg}}{c}\right) - a_T\left(\frac{S_T}{S_{ref}}\right)\left(\frac{l_T-l_{cg}}{c}\right)$$

MOMENT AT ZERO LIFT (C_{m0})

$$L = L_w + L_o + L_T$$

$$C_L = \frac{L}{\frac{1}{2}\rho V_\infty^s S_{ref}} = C_{Lw} + C_{Lo}\left(\frac{S_o}{S_{ref}}\right) + C_{LT}\left(\frac{S_T}{S_{ref}}\right)$$

$$C_L = a_w\alpha + a_o\alpha\left(\frac{S_o}{S_{ref}}\right) + a_T(\alpha - i)\left(\frac{S_T}{S_{ref}}\right)$$

$$0 = \left(a_w + a_o\frac{S_o}{S_{ref}} + a_T\frac{S_T}{S_{ref}}\right)\alpha - a_T\, i\,\frac{S_T}{S_{ref}}$$

$$\alpha = \frac{\left(a_T\, i\,\frac{S_T}{S_{ref}}\right)}{\left(a_w + a_o\frac{S_o}{S_{ref}} + a_T\frac{S_T}{S_{ref}}\right)}$$

ZERO LIFT ANGLE OF ATTACK

SUBSTITUTE THIS α INTO C_{mcg} (2) TO OBTAIN MOMENT COEFFICIENT @ ZERO LIFT

STICK FIXED NEUTRAL POINT $\dfrac{\partial C_m}{\partial \alpha} = 0$

$$0 = -\frac{a_w l_w}{c} + \frac{a_w l_{cg}}{c} - \frac{a_o S_o l_o}{c S_{ref}} + \frac{a_o S_o l_{cg}}{c S_{ref}} - \frac{a_t S_t l_t}{c S_{ref}} + \frac{a_t S_t l_{cg}}{c S_{ref}}$$

$$\text{NEUTRAL POINT} = \left(\frac{a_w l_w + \dfrac{a_o S_o l_o}{S_{ref}} + \dfrac{a_t S_t l_t}{S_{ref}}}{a_w + \dfrac{a_o S_o}{S_{ref}} + \dfrac{a_t S_T}{S_{ref}}} \right)$$

CG POSITION FOR NEUTRAL STATIC STABILITY

STATIC LONGITUDINAL CONTROL

$$C_L = C_{L_\alpha}\alpha + C_{L_\delta}\delta + C_{L_E}E \qquad C_{m_{CG}} = C_{mo} + C_{m_\alpha}\alpha + C_{m_\delta}\delta + C_{m_E}E$$

CASE 1 SET ELEVON ANGLE; OBTAIN ELEVATOR ANGLE

$$\Delta C_L = \Delta C_{LW} + \frac{S_T}{S_{ref}}\Delta C_{LT} = \frac{\partial C_L}{\partial E}E + \frac{S_T}{S_{ref}}\frac{\partial C_{LT}}{\partial \delta}\delta$$

$$\frac{\partial C_L}{\partial \delta} = \frac{S_T}{S_{ref}}\frac{\partial C_{LT}}{\partial \delta}$$

$$\Delta C_M = -\frac{S_T}{S_{ref}}\left(\frac{l_T - l_{cg}}{c}\right)\frac{\partial C_{LT}}{\partial \delta}\delta$$

$$\frac{\partial C_m}{\partial \delta} = -\frac{S_T}{S_{ref}}\left(\frac{l_T - l_{cg}}{c}\right)\frac{\partial C_{LT}}{\partial \delta}$$

$$\delta_{TRIM} = \frac{-C_{mo} - C_{m\alpha}\alpha - C_{mE}E}{C_{m\delta}} \qquad C_{L_{TRIM}} = C_{L_\alpha}\alpha + C_{L_\delta}\delta_{TRIM} + C_{LE}E$$

$$\alpha = \frac{C_{L_{TRIM}} - C_{L_\delta}\delta_{TRIM} - C_{LE}E}{C_{L\alpha}}$$

$$\delta_{TRIM} = \frac{-C_{m0} - C_{m\alpha}\left(\dfrac{C_{LTRIM} - C_{L\delta}\delta_{TRIM} - C_{LE}E}{C_{L\alpha}}\right) - C_{mE}E}{C_{m\delta}}$$

$$C_{m\delta}\delta_{TRIM} = -C_{m0} - \frac{C_{m\alpha}}{C_{L\alpha}}C_{LTRIM} + \frac{C_{m\alpha}}{C_{L\alpha}}C_{L\delta}\delta_{TRIM} + \frac{C_{m\alpha}}{C_{L\alpha}}C_{LE}E - C_{mE}E$$

$$\delta_{TRIM}\left(C_{m\delta} - \frac{C_{m\alpha}}{C_{L\alpha}}C_{L\delta}\right) = -C_{m0} - \frac{C_{m\alpha}}{C_{L\alpha}}\left(C_{LTRIM} - C_{LE}E\right) - C_{mE}E$$

$$\delta_{TRIM} = \frac{-C_{m0}C_{L\alpha} - C_{m\alpha}C_{LTRIM} + C_{m\alpha}C_{LE}E - C_{mE}E\,C_{L\alpha}}{C_{m\delta}C_{L\alpha} - C_{m\alpha}C_{L\delta}}$$

CASE (2) SET ELEVATOR ANGLE; OBTAIN ELEVON ANGLE

$$\Delta C_L = \Delta C_{Lw} = \frac{\partial C_{Lw}}{\partial E}E$$

$$\Delta C_M = \Delta C_{Lw}\left(\frac{l_w - l_{cg}}{c}\right) = \left(\frac{l_w - l_{cg}}{c}\right)\frac{\partial C_L}{\partial E}E + \frac{\partial C_{mw}}{\partial E}E$$

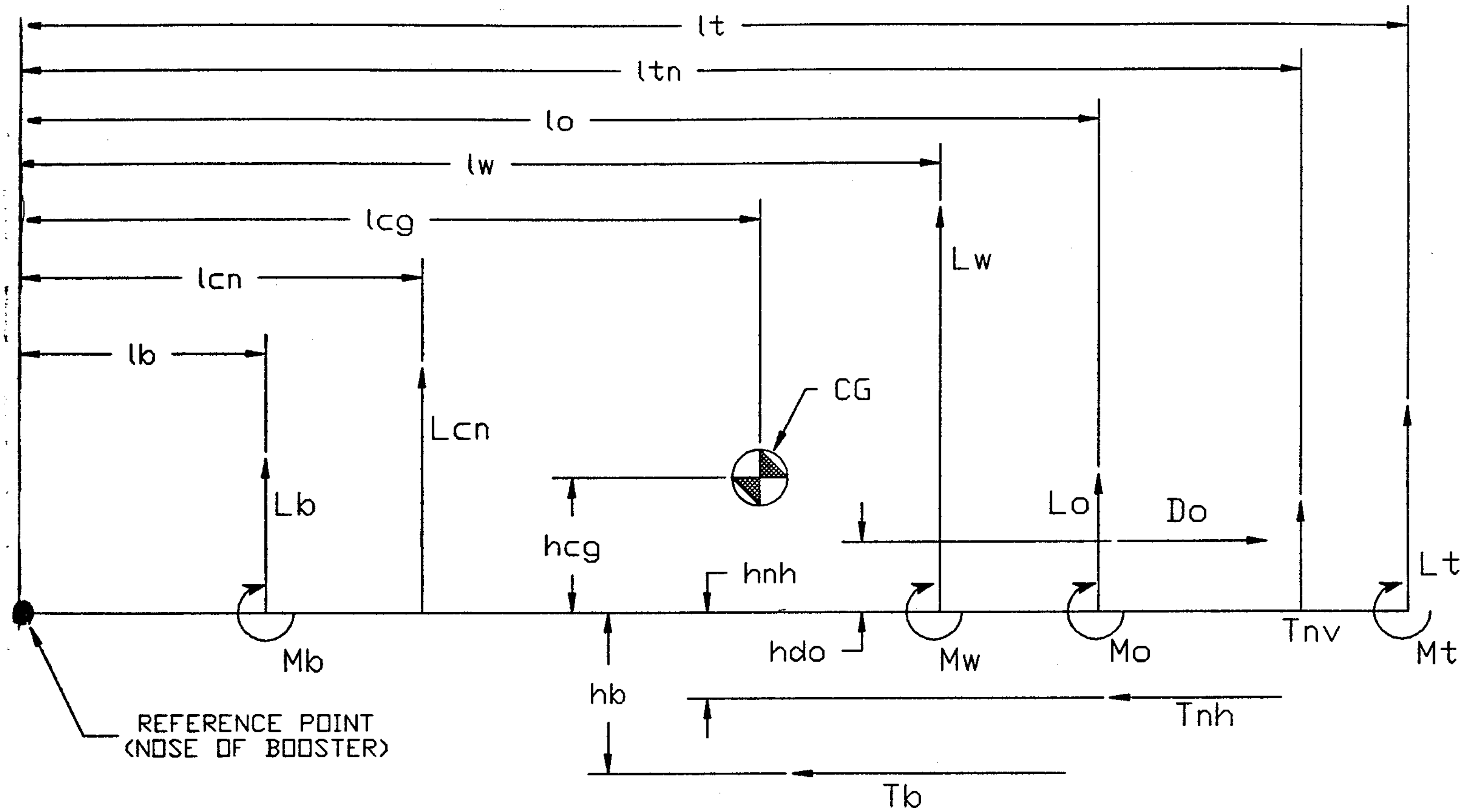
$$\frac{\partial C_M}{\partial E} = \left(\frac{l_w - l_{cg}}{c}\right)\frac{\partial C_L}{\partial E} + \frac{\partial C_{mw}}{\partial E}$$

$$E_{TRIM} = \frac{-C_{m0} - C_{m\alpha}\alpha - C_{m\delta}\delta}{C_{mE}}$$

$$E_{TRIM} = \frac{-C_{m0}C_{L\alpha} - C_{m\alpha}C_{LTRIM} + C_{m\alpha}C_{L\delta}\delta - C_{m\delta}\delta\,C_{L\alpha}}{C_{mE}C_{L\alpha} - C_{m\alpha}C_{LE}}$$

DESCRIPTION OF PROGRAM STABILITY

This program is custom written for the longitudinal static stability analysis of the booster vehicle. It is menu driven allowing the many variables in the stability equations to be changed quickly and in a organized fashion. All the parameters must be determined at chosen Mach numbers and then entered into the program. All of the initial variables must be inputed through a file "in.dat" in the order specified in the appendix. Pertinent stability data as well as the major parameters are outputed to file "out.dat" for further analysis. Critical stability data such as elevator & elevon angle to trim, neutral point, pitching moment derivative, and moment at zero lift are displayed to the screen as well. The program is especially useful for determining the stability of the booster/orbiter combination through the assent where the many parameters vary



ASSUMPTIONS: BOOSTER DRAG MOMENT IS SMALL
NO ORBITER THRUST UNTIL AFTER SEPARATION

MOMENT ABOUT C.G.

INPUT VARIABLE ORDER FOR THE INPUT FILE FOR PROGRAM "STBLTY"

P,V,SREF

LCG,LB,LCN,LW,LO,LT,LTN

HCG,HB,HNH,HDO

MW,MT,MB,MO,DO,T,TNH,TNV

SO,AWB,AO,AT,ST

LFTB,LFTW,CL,DCLDA,DCMDA

DCLDE,DCMDE,DCLDE,DCMDE

VARIABLE DEFINITIONS FOR PROGRAM STABILITY

P:	DENSITY
V:	VELOCITY
SREF:	REFERENCE WING AREA
LCG:	DISTANCE FROM REFERENCE POINT TO CG
LB:	DISTANCE FROM REFERENCE POINT TO BODY LIFT POINT
LCN:	DISTANCE FROM REFERENCE POINT TO CHINE LIFT POINT
LW:	DISTANCE FROM REFERENCE POINT TO WING AC
LO:	DISTANCE FROM REFERENCE POINT TO ORBITER AC
LT:	DISTANCE FROM REFERENCE POINT TO TAIL AC
LTN:	DISTANCE FROM REFERENCE POINT NOZZLE VERTICAL THRUST
HCG:	HEIGHT FROM REFERENCE POINT TO CG
HB:	HEIGHT FROM REFERENCE POINT TO DIRECT THRUST LINE
HNH:	HEIGHT FROM REFERENCE POINT TO NOZZLE HORIZONTAL TRST
HDO:	HEIGHT FROM REFERENCE POINT TO ORBITER DRAG
MW:	MOMENT ABOUT AC OF THE WING
MT:	MOMENT ABOUT AC OF THE TAIL
MB:	MOMENT ABOUT CG OF THE BODY
MO:	MOMENT ABOUT AC OF THE ORBITER
T:	DIRECT THRUST FORCE INCLUDING INLET
TNH:	HORIZONTAL NOZZLE THRUST FORCE
TVN:	VERTICAL NOZZLE THRUST FORCE
SO:	REFERENCE WING AREA OF THE ORBITER
ST:	REFERENCE WING AREA OF THE TAIL
AWB:	LIFT SLOPE OF THE WING
AO:	LIFT SLOPE OF THE ORBITER
AT:	LIFT SLOPE OF THE TAIL
C:	MEAN AERODYNAMIC CHORD
LFTB:	LIFT OF THE BODY
LFTCN:	LIFT OF THE CHINE
CL:	COEFFICIENT OF LIFT
MCG:	MOMENT ABOUT THE CG
CMCG:	COEFFICIENT OF MOMENT ABOUT THE CG
DELTA:	TAIL DEFLECTION ANGLE +DOWN
I:	TAIL INCIDENCE +DOWN
E:	ELEVON DEFLECTION +DOWN
MZL:	MOMENT ABOUT CG @ ZERO LIFT
CMZL:	COEFFICIENT OF MOMENT ABOUT CG @ ZERO LIFT
DCMDA:	$\frac{dCm}{d(alpha)}$
DCMDD:	$\frac{dCm}{d(delta)}$
DCMDI:	$\frac{dCm}{d(incidence)}$
DCLDA:	$\frac{dCl}{d(alpha)}$
DCLDD:	$\frac{dCl}{d(delta)}$
DCLDI:	$\frac{dCl}{d(incidence)}$

```
      PROGRAM STBLTY
      REAL P,V,SREF,LCG,LB,LCN,LW,LO,LT,LTN
     +      HCG,HB,HNH,HDO,C
     +      MW,MT,MB,MO,DO,T,TNH,TNV,SO,AWB,AO,AT,ST
     +      LFTB,LFTCN,LFTW,LFTO,LFTT
     +      NEWVAL,MCG,CMCG,DELTA,I,E,TEMP2,TEMP3
     +      NEUPT,DCMDA,MZL,CMZL,DCLDA,DCMDA
     +      DCMDD,DCMDE,DCLDD,DCLDE
      INTEGER RESP,CON
      OPEN(1,FILE='IN')
      OPEN(2,FILE='OUT')
      WRITE(*,*) 'LONGITUDINAL STATIC STABILITY ANALYSIS'
      READ(1,*)P,V,SREF
      READ(1,*)LCG,LB,LCN,LW,LO,LT,LTN
      READ(1,*)HCG,HB,HNH,HCG,HDO
      READ(1,*)MW,MT,MB,MO,DO,T,TNH,TNV
      READ(1,*)SO,AWB,AO,AT,ST
      READ(1,*)LFTB,LFTCN,CL,DCLDA,DCMDA
      READ(1,*)DCMDD,DCMDE,DCLDD,DCLDE
      WRITE(*,*) 'AIRCRAFT PARAMETERS HAVE BEEN READ FROM INPUT FILE'
10    CALL CLRSCR
      PRINT *, '                    MAIN MENU'
      PRINT *, ' '
      PRINT *, ' '
      PRINT *, 'EDIT VARIABLE PARAMETERS --- 1'
      PRINT *, ' '
      PRINT *, 'CALCULATE STABILITY DATA --- 2'
      PRINT *, ' '
      PRINT *, 'FILE MENU                --- 3'
      PRINT *, ' '
      PRINT *, 'EXIT                     --- 4'
      READ(*,*)RESP
      GOTO (100,200,300,400)RESP
100   CALL CLRSCR
      PRINT *, '            EDIT MODE'
      PRINT *, ' '
      PRINT *, ' '
      PRINT *, '(1) DENSITY'
      PRINT *, '(2) VELOCITY'
      PRINT *, '(3) ZERO LIFT MOMENTS AND ORBITER DRAG'
      PRINT *, '(4) THRUST AND LIFT'
      PRINT *, '(5) CENTER OF GRAVITY AND AERODYNAMIC CENTER DATA'
      PRINT *, '(6) RETURN TO MAIN MENU'
      READ(*,*)RESP
      GOTO (110,120,130,140,141,10)RESP
110   PRINT *, 'NEW DENSITY?'
      READ(*,*)P
      GOTO 100
120   PRINT *, 'NEW VELOCITY?'
      READ(*,*)V
      GOTO 100
130   CALL CLRSCR
      PRINT * ,'        ZERO LIFT MOMENTS AND ORBITER DRAG'
      PRINT * ,' '
      PRINT *, '(1) MOMENT OF WING'
      PRINT *, '(2) MOMENT OF TAIL'
```

```
      PRINT *, '(3) MOMENT OF BODY'
      PRINT *, '(4) MOMENT OF ORBITER'
      PRINT *, '(5) ORBITER DRAG'
      PRINT *, '(6) RETURN TO EDIT MENU'
      READ(*,*)RESP
      IF (RESP .EQ. 6) GOTO 100
      PRINT *, 'ENTER NEW VALUE'
      READ(*,*)NEWVAL
      GOTO(145,150,155,160,165)RESP
145   MW=NEWVAL
      GOTO 130
150   MT=NEWVAL
      GOTO 130
155   MB=NEWVAL
      GOTO 130
160   MO=NEWVAL
      GOTO 130
165   DO=NEWVAL
      GOTO 130
140   CALL CLRSCR
      PRINT *, '              EDIT THRUST AND LIFT'
      PRINT *, ' '
      PRINT *, '(1) DIRECT ENGINE THRUST'
      PRINT *, '(2) NOZZLE VERTICAL THRUST'
      PRINT *, '(3) NOZZLE HORIZONTAL THRUST'
      PRINT *, '(4) LIFT OF THE BODY'
      PRINT *, '(5) LIFT OF THE CHINE'
      PRINT *, '(6) COEFFICIENT OF LIFT TO TRIM'
      PRINT *, '(7) RETURN TO EDIT MENU'
      READ(*,*)RESP
      IF (RESP .EQ. 7) GOTO 100
      PRINT *, 'ENTER NEW VALUE'
      READ(*,*)NEWVAL
      GOTO(170,175,180,185,190,192)RESP
170   T=NEWVAL
      GOTO 140
175   TNH=NEWVAL
      GOTO 140
180   TNV=NEWVAL
      GOTO 140
185   LFTB=NEWVAL
      GOTO 140
190   LFTCN=NEWVAL
      GOTO 140
192   LFTW=NEWVAL
      GOTO 140
141   CALL CLRSCR
      PRINT *, '              EDIT CG AND AC'
      PRINT *, ' '
      PRINT *, ' '
      PRINT *, '(1) CENTER OF GRAVITY LOCATION'
      PRINT *, '(2) WING AERODYNAMIC CENTER'
      PRINT *, '(3) ORBITER AERODYNAMIC CENTER'
      PRINT *, '(4) TAIL AERODYNAMIC CENTER'
      PRINT *, '(5) WING LIFT SLOPE'
      PRINT *, '(6) TAIL LIFT SLOPE'
```

```
      PRINT *, '(7) ORBITER LIFT SLOPE'
      PRINT *, '(8) RETURN TO EDIT MENU'
      READ(*,*)RESP
      IF (RESP .EQ. 8) GOTO 100
      PRINT *, 'ENTER NEW VALUE'
      READ(*,*)NEWVAL
      GOTO(142,143,144,146,500,501,502)RESP
142   LCG=NEWVAL
      GOTO 141
143   LW=NEWVAL
      GOTO 141
144   LO=NEWVAL
      GOTO 141
146   LT=NEWVAL
      GOTO 141
500   AWB=NEWVAL
      GOTO 141
501   AT=NEWVAL
      GOTO 141
502   AO=NEWVAL
      GOTO 141
200   CALL CLRSCR
      PRINT *, '                    CALCULATED STABILITY DATA'
      PRINT *, ' '
      PRINT *, 'SET ELEVON ANGLE'
      READ(*,*)E
      CALL CLRSCR
      TEMP2=MW+MT+MB+MO
      TEMP3=TB*(HB+HCG)+TNH*(HNH+HCG)-DO*(HCG-HDO)-TNV*(LTN-LCG)
      ND=0.5*P*V*V*SREF*C
      DCMDA=-AWB*(LW-LCG)/C-AO*(SO/SREF)*(LO-LCG)/C-(ST/SREF)
     +      *((LT-LCG)/C)*AT
      ALZL=(AT*I*ST/SREF)/(AW+AO*SO/SREF+AT*ST/SREF)
      CMZL=(LFTB*(LCG-LB))/ND+(LFTCN*(LCG-LCN))/ND-AO*ALZL*SO/SREF*
     +      (LO-LCG)/C-AW*ALZL*(LW-LCG)/C-AT*(ST/SREF)*(LT-LCG)/C
     +      +TEMP2/ND+TEMP3/ND
      PRINT *, 'MOMENT COEFFICIENT @ ZERO WING LIFT = ',CMZL
      PRINT *, 'COEFFICIENT OF MOMENT DERIVATIVE = ',DCMDA
      DCLDA=AW+AO*SO/SREF+AT*ST/SREF
      DELTA=-(CMZL*DCLDA+DCMDA*CL-DCMDA*NDCLDE*E-DCMDE*E*DCLDA)/
     +      (DCLCA*DCMDD-DCMDA*DCLDD)
      PRINT *, 'ELEVATOR ANGLE  = ',DELTA
      NEUPT=(AWB*LW+AO*SO*LO/SREF+AT*ST*LT/SREF)/
     +      (AWB+AO*SO/SREF+AT*ST/SREF)
      PRINT *, 'STICK FIXED NEUTRAL POINT = ',NEUPT
      TRIMA=(CL+AT*I*ST/SREF)/(AW+AO*SO/SREF+AT*ST/SREF)
      PRINT *, 'TRIM ANGLE OF ATTACK',TRIMA
      PRINT *, ' '
      PRINT *, ' '
      PRINT *, 'HIT 1 THEN ENTER TO CONTINUE'
      READ(*,*)RESP
290   IF (RESP .EQ. 1) GOTO 10
      GOTO 290
300   CALL CLRSCR
      PRINT *, '        FILE MENU'
      PRINT *, ' '
```

```
      PRINT *, ' '
      PRINT *, '(1) OUTPUT TO FILE OUT'
      PRINT *, '(2) RETURN TO MENU'
      PRINT *, ' '
      PRINT *, 'ENTER CHOICE'
      READ(*,*)RESP
      GOTO (350,10)RESP
350   WRITE (2,*)'          STABILITY  PARAMETERS'
      WRITE (2,*)' '
      WRITE (2,*)'TAIL INCIDENCE ',I
      WRITE (2,*)'TAIL DELTA ',DELTA
      WRITE (2,*)'ELEVON ANGLE ',E
      WRITE (2,*)'DCM/DA ',DCMDA
      WRITE (2,*)'MOMENT COEFF @ ZERO LIFT ',CMZL
      WRITE (2,*)'TRIM ANGLE OF ATTACK ',TRIMA
      WRITE (2,*)'NEUTRAL POINT ',NEUPT
      WRITE (2,*)'DENSITY ',P,' VELOCITY ',V
      WRITE (2,*)'CG POSITION ',LCG
      WRITE (2,*)'WING AC LOCATION ',LW
      WRITE (2,*)'DIRECT ENGINE THRUST ',T
      WRITE (2,*)'HORIZONTAL NOZZLE THRUST ',TNH
      WRITE (2,*)'VERTICAL NOZZLE THRUST ',TVN
      WRITE (2,*)'DCLDA ',DCLDA
      WRITE (2,*)'COEFFICIENT OF LIFT ',CL
      WRITE (2,*)'DCMDD ',DCMDD
      WRITE (2,*)'DCMDE ',DCMDE
      WRITE (2,*)'DCLDE ',DCLDE
      WRITE (2,*)'DCLDD ',DCLDD
      GOTO 10
400   CALL CLRSCR
      PRINT *, 'PROGRAM TERMINATED'
      STOP
      END
*
      SUBROUTINE CLRSCR
      DO 390 N=1,25
         WRITE (*,*)' '
390   CONTINUE
      RETURN
      END
```

SUGGESTIONS FOR FURTHER RESEARCH

Due to the lack of propulsion system data, flight mechanics and some stability calculations could not be done. As soon as this data becomes available, calculations must be done in order to determine such aspects of flight mechanics as takeoff and landing distances and velocities, initial rate of climb, glide angle, and total horizontal distance covered during ascent. A more detailed investigation of the zero-g pushover maneuver must also be done. Also, a more detailed look at the flight profile must be accomplished.

REFERENCES

1. Raymer, Daniel P., AIRCRAFT DESIGN: A Conceptual Approach.

2. Anderson, John D. Jr., INTRODUCTION TO FLIGHT, 3rd ed.

3. Saad, Michel A., Compressible Fluid Flow.

4. NASA DATA ON PROJECT BETA

5. Harloff, G.J. and Petrie, S.L., "Preliminary Aerothermodynamic Design Method for Hypersonic Vehicles", Sverdrup Technology Inc., Lewis Research Center Group.

6. "Low Alpha Supersonic Flow", Journal of Aircraft.

7. Chatput, Armand J., "Preliminary Sizing Methodology for Hyper sonic Vehicles", AIAA-87-2954.

8. Rich, Ben R., "F-12 Series Aircraft Aerodynamic and Thermodynamic Design in Retrospect", Lockheed-California Company.

9. Etkin, Bernard, DYNAMICS OF FLIGHT: Stability and Control.

10. Harloff, Gary J. and Berkowitz, Brian M., "HASA-Hypersonic Aerospace Sizing Analysis for the Preliminary Design of Aerospace Vehicles", NASA Contractor Report 182226.

ACKNOWLEDGMENTS

INSTRUCTOR:	Professor E. Reshotko
GRAD ASSISTANT:	William Humphrey
NASA SPONSORS:	Leo Burkardt
	Art Glassman

PROPULSION GROUP

Hani Alexander
Ralph Jansen
David Smith

PROPULSION

Flight trajectory and hardware analysis needed to be done in order to quantify the propulsion system of the booster for a two stage to orbit behicle. Flight path analysis was the first issue addressed by the propulsion group. The goal of the analysis was to find the flight path that required the minimum fuel consumption to reach the operating target of Mach 6 at 100,000 feet. A method outlined in Daniel P. Raymer's book Aircraft Design: A conceptual Approach was the basis for the analysis. Several computer programs were written to do the necessary numerical computation for each flight path. Different thrust models were used to represent the propulsion system as it evolved.

The flight analysis programs have several features in common. Each program uses a model of the standard atmosphere presented in Introduction to Flight by Anderson. The atmosphere model uses a series of linear fits to model the temperature of the atmosphere. The density and pressure of the atmosphere at each location are correlated to the temperature distribution. An energy height function is used in each program in order to compute the energy height as a function of Mach number and altitude. The plot of energy height becomes slightly distorted in this type of plot because the speed of sound changes along with the properties of the atmosphere. Another function that is incorporated in the program plots the structural limit line which is defined by a maximum dynamic pressure level. Drag computations were based on a model used in the analysis of the Sanger project. All of the flight analysis programs output graphical data with an x-range of 0 to Mach 6 and a y-range of 0 to 120,000 feet. Programs were written to output both graphical and numerical representations of specific power curves and specific fuel consumption curves. Software was written in Pascal with listings included in Appendix B.

A preliminary flight path was chosen after an analysis of specific power and specific fuel consumption was completed. Preliminary plots of specific power curves are included in Appendix C. The optimal flight path began with a climb from takeoff to 26,000 feet and Mach

1.3. The flight then followed the structural limit line to Mach 6.4 and 80,000 feet. The final segment of flight traded off excess speed for altitude as the craft climbs to 100,000 feet at Mach 6. This flight path was determined using a model of engine thrust that was based on the mass flow of air. The thrust was found by calculating mass flow of fuel and using the Isp to fing the engine thrust. This model was valid for the ramjet operation, but it is inaccurate for the turbojet where the mass flow is limited by the turbomachinery.

An alternate model of the turbojet is based on the scaled up performance of an afterburning turbojet presented in Raymer's work. This model of the engine could not be fully incorporated into the work since the software, ONX OFFX, that it is based on was never installed on the IBM mainframe. A crude linear approximation of the engine was used instead, which was only valid up to Mach 2. This lead to difficulties in calculations, since the two engine modeling systems did not precisely correlate at Mach 2.

The engine hardware was broken into three main sections: the inlet, the combustor, and the nozzle. A turbo ramjet burning liquid hydrogen was the type of propulsion system chosen. This choice utilizes current technology and falls in line with the technology levels of other vehicle components. The inlet is a multiple ramp system with mixed compression. Five turbojets are used to provide propulsion up to Mach 3.7. The specifications of the turbojets are included in Appendix A, and are based on the scaling laws presented by Raymer. At Mach 3.7 th turbojets are disengaged and the ramjet combustion begins. The turbojets are allowed to windmill which serves a duel purpose. Low energy boundary layer is diverted away from the ramjet through the windmilling turbojets. The airflow also cools the inner surface of the ramjet. The transition point is chosen where the ramjet specific impulse exceeds the specific impulse of the turbojet. A half nozzle is used to expand the exit flow. The bottom of the fuselage is used as the top half of the nozzle. Prandtl-Meyer relations can be used to find the flow in this type of nozzle. Detailed work on the hardware must be done in order to verify the models that were used.

EXHAUST NOZZLE

For the nozzle of the propulsion system, two different types of nozzles were chosen for the prospective engines. For the turbojet engine an axisymmetric, variable, converging-diverging (C-D) nozzle was chosen. This type of nozzle was desired for multiple reasons. The variable, C-D nozzle is considered to be the nozzle of choice for after-burning engines which must produce very high exhaust velocities such as the booster's. This class of nozzles is structurally efficient pressure vessels and have a very large technology base due to their frequent use. The variable as opposed to fixed option was necessary due to the large variation of area required to expand the gases and maintain a high pressure ratio.

For the ramjet, a non-axisymmetric, "half", C-D nozzle was chosen. This type of nozzle worked well for the need to blend the nozzle with the aft section of the vehicle. The "half" indication is essentially the non-axisymmetric portion. The use of a "half" nozzle was important in reducing the weight of the nozzle by approximately 50% and it also simplified the overall structure while maintaining essentially the same level of performance as a full axisymmetric nozzle. The velocity upon exit is around Mach 2 and the final Mach number ranges from Mach 4 to Mach 7. Prandtl-Meyer expansion occurs at the exit. In addition to both turning and accelerating the flow, the Prandtl-Meyer expansion defines the shape of the nozzle's diverging section. While this type of nozzle simplified the structure, it complicated the analysis. A true analysis of the expansion was not conducted. This was due to both not being able to obtain enough information on Prandtl-Meyer expansion and the procedures involved in analyzing it and a lack of time. This information needs to be obtained and should be researched further.

INLET DESIGN

The primary purpose of the inlet to an airbreathing engine is to slow the airflow entering the engine to usable speeds. For both the turbojet and the ramjet, the speed entering the compressor should be approximately Mach 0.4. To do this, the aircraft utilizes a mixed compression inlet using two external variable geometry compression ramps to initially slow the flow, and then has the flow enter a cowl where it is further slowed by an internal two-dimensional wedge. At Mach numbers greater than one, the external compression ramps are used to turn the flow and cause oblique shocks which reduce the speed of the flow. After the two oblique shocks, the supersonic flow enters the cowl and goes through a series of reflected oblique shock waves until it finally passes through a normal shock and enters the diffuser prior to the engines.

For Mach numbers less than one, the external ramps are contracted so that they are flush with the airframe and no turning of the flow occurs. If the air has a speed of less than or equal to Mach 0.4, then the air enters the engine unchanged. For speeds between Mach 0.4 and Mach 0.7, the internal ramps are used as diffusers to slow the flow down to Mach 0.4. For speeds greater than Mach 0.7, the ramps act to cause the flow to go through a normal shock prior to entering the diffuser.

Upon entering the diffuser section, the flow is directed to either the turbojet or the ramjet, depending upon the velocity of the booster. The configuration of the turbojet over the ramjet allows the air at low booster velocities to enter the turbojet without much turning, and the flow at higher booster velocities, when the ramjet is operating, enters the ramjet after a larger amount of turning. Variable flaps are used to divert the flow to the respective engines.

For design conditions the airflow over the external ramps is Mach 6. The booster's velocity may be greater, but the shock wave from the nose and its resulting boundary layer will allow for the design conditions. The flow goes through two external oblique shocks and four internal oblique shocks prior to the normal shock. The capture area was calculated from the mass flow requirements and the density, pressure, and velocity at 90,000 feet. The exterior compression surfaces were then designed for decent pressure recovery and low boundary layer build-up. A rough approximation was made of the throat area by using isentropic calculations and taking the strengths of the shocks into consideration. From this, the area of the cowl and the angle of the internal wedge was calculated. From the angle of the wedge and the throat area, the number and strngth of the oblique shocks could be estimated. After many iterations of changing the cowl area and the wedge angle for the best internal pressure recovery, the angles of teh external ramps were altered to give the best external pressure recovery compared to the internal pressure recovery. The throat area was then calculated by both the mass flow equations and the isentropic flow calculations using the inverse of the total internal and external pressure recoveries. When these were close, the final cowl area and internal wedge angles were calculated.

In calculating the capture area, both the boundary layer bleed and the secondary airflow were taken into consideration. The mass flow used for boundary layer bleed at design conditions was estimated at 18% of the total airflow. After checking for shock-induced flow separation, no major region of separated flow was found. However, boundary layer build-up on the long external compression ramps and build-up due to the internal reflected shocks would be large. At high speeds when the ramjet is in operation, some of the boundary layer may be diverted through the turbojet. The remaining boundary layer for the high speed flows and the boundary layer for lower speeds when the turbojet is in operation, must be sucked out and expelled behing the engines. Due to the variation of the external ramps and the moving of the internal reflected shocks, bypass doors would not be as effective as open slots for removing the boundary layer. The slots would cover certain areas on the external ramps and would always be open. The slots on the upper surface of the internal wedge (the part behind the cowling of the second external ramp) would only be in use when an oblique shock was incident upon them. In this way, the boundary layer would be removed where it was most crucial.

Off-design conditions were calculated similarly to the design condition, but the length of the external compression ramps and the internal wedges were the same as for the design conditions. For flow entering the ramjet, the final turning angle of the flow should be greater, allowing the flow to enter the diffuser lower because the ramjet is beneath the turbojet. For the reduced velocity conditions, the angles of the external ramps were reduced, and the cowl lip was pivoted upward to decrease the vertical capture area. The internal wedges also had to rotate to give the best pressure recovery ratio between the cowl area and the throat area.

The mixed compression inlet using the two variable geometry compression ramps and the rotating cowl lip were found to be the most effective method of handling both design and off-design conditions. Initially, the two external ramps were designed to be stationary, allowing the cowl to translate horizontally to decrease the capture area and increase the cowl area for off-design conditions. The boundary layer slots would be increased in size for lower speed flows and the free-stream air would also enter the boundary layer ducts. The free-stream flow would then bypass the fixed compression ramps and emerge before the throat. The low momentum boundary layer would still be removed and ejected out of the back. However, for subsonic speeds, the cowl was forced to translate almost 30 feet, which would not have been feasible.

Early calculations of the thrust and the mass flow were too large, forcing a design capture area of 2077 square feet, which would have dropped the cowl almost 50 feet below the nose of the aircraft. To compensate for this, a droop-nose was designed for the booster which would have placed the cowl 27 feet below the airframe. However, the mass flow calculations were corrected, giving a capture area of only 534 square feet, placing the cowl 13 feet below the airframe.

APPENDIX A

ENGINE SPECIFICATIONS

Sea level static thrust (lb)	160,800
Sea level TSFC (lb/hr)	1.64
Sea level static airflow (lbm/s)	1,319
Bare engine weight (lb)	16,080
Engine length (in)	315
Maximum diameter (in)	102
Fan-face diameter (in)	93
Overall pressure ratio	22
Fan pressure ratio	4.3
Bypass ratio	0.41

APPENDIX B

PROGRAM LISTINGS

1) Graphical representation of PS curves
2) Graphical representation of FS curves
3) Output of data along flight path

Ralph Jansen
Hani Alexander

Program: PS curves
This program plots specific power curves. The structural limit line and specific energy curves are also plotted.
Thrust calculations are based on a capture area model.

```
program test(input, output);
uses Crt, Graph;
const
  r=287.04;
  g=9.8;
  c=1.94045e8;
  k=1.4;
  deltat=1;
  wgtf=7.11111e6;
  s=1400;
  cpd=0.1;
  pi=3.14159;
  ar=2;
  trans=3.7;
  struck=100000;

  (********** EXPONENT FUNCTION *******)
function pow(x,y:real):real;
  begin
     pow:=exp(x*ln(y));
  end;

(********* CORREPONDING BASE TEMPERATURE & SLOPE ********)
procedure val(var alt,t1,m,h,p1,d1:real);
  var
     a:integer;
  begin
     a:=trunc(alt/10);
     case a of
      0..1100:   begin
            t1:=288.16;h:=0;m:=-6.5e-3;p1:=1.01325e5;d1:=1.225;
            end;
      1101..2500: begin
            t1:=216.66;h:=11000;m:=0;p1:=2.27e4;d1:=3.648e-1;
            end;
      2501..4700: begin
            t1:=216.66;h:=25000;m:=3e-3;p1:=2.5273e3;d1:=4.0639e-2;
            end;
      4701..5300: begin
            t1:=282.66;h:=47000;m:=0;p1:=1.2558e2;d1:=1.5535e-3;
            end;
      5301..7900: begin
```

```
            t1:=282.66;h:=53000;m:=-4.5e-3;p1:=6.1493e1;d1:=7.5791e-4;
            end;
      7901..9000: begin
            t1:=165.66;h:=79000;m:=0;p1:=6.14e1;d1:=7.5791;
            end;
      9001..10500: begin
            t1:=165.66;h:=90000;m:=4e-3;p1:=6.14e1;d1:=7.5791;
            end;
      10501..32000: begin
            t1:=225.66;h:=105000;m:=0;p1:=6.14e1;d1:=7.5791;
            end;
    end;
  end;

(********** TEMPERATURE FUNCTION *********)

function t(alt:real):real;
  var
    tt,ht,mt,pt,dt: real;
  begin
    val(alt,tt,mt,ht,pt,dt);
    t:=tt+mt*(alt-ht)
  end;

(********** PRESSURE FUNCTION *********)

function p(alt:real):real;
var
  ap,tp,mp,hp,pp,dp:real;
begin
  ap:=alt;
  val(ap,tp,mp,hp,pp,dp);
  if (mp=0) then p:=pp*exp(-(g/(r*tp))*(ap-hp))
  else
    p:=pp*pow(-g/(mp*r),t(ap)/tp);
end;

(********* DENSITY FUNCTION *********)

function d(alt:real):real;
var
  ad,td,md,hd,pd,dd:real;

begin
  ad:=alt;
  val(ad,td,md,hd,pd,dd);
  if (md=0) then d:=dd*exp(-(g/(r*td))*(ad-hd))
  else
    d:=dd*pow(-(g/(md*r)+1),t(ad)/td);
end;
```

```
(*********** Isp FUNCTION *************)
function isp(mach:real):real;
  begin
    isp:=4200-15*mach*mach;
  end;

(*********** THRUST **************)
function thrust(alt,mach,yt,yp,yd,yc:real):real;
  var
   ai:real;
  begin
    if mach<3.7 then ai:=40 else ai:=35;
    thrust:=isp(mach)*ai*g*mach*yc*yd/40;
  end;


(********** DRAG CALCULATIONS ***********)
function drag(alt, mach, yt, yp, yd, yc:real):real;

(**** cl = coefficient of lift, ar = aspect ratio, lambda = leading edge sweep ********)
var ar,cd,cdp,cdl,lambda, cl:real;

  begin

     if (mach<0.8) and (mach>0) then cdp:=0.011;
     if (mach>=0.8) and (mach<1.2) then cdp:=-0.0510+0.0762*mach;
     if (mach>=1.2) and (mach<7.0) then cdp:=0.0605 - 0.0177*mach +
0.00163*sqr(mach);

     cl:=2*wgtf/(yd*s*sqr(mach*yc));
     cdl:=(0.1378 + 0.1693*mach - 0.0115*sqr(mach))*sqr(cl);
     cd:=cdl+cdp;

     drag:=0.5*yd*sqr(mach*yc)*s*cd;
  end;


(*********** specific power curve ************)
function ps(alt, mach,yt,yp,yd,yc : real):real;
var thrustvar, dragvar, fsvar : real;
begin
   thrustvar := thrust(alt, mach,yt,yp,yd,yc);
   dragvar := drag(alt, mach,yt,yp,yd,yc);
      fsvar := (mach * yc * (thrustvar - dragvar)) / wgtf;
{   writeln('Thrust = ', thrustvar:8:3, ' Drag := ', dragvar:8:3, ' FS := ', fsvar:8:3);
}   ps := fsvar;
end;

(************ energy height **************)
```

```
function he(alt, mach , yt, yp, yd, yc : real):real;
begin
   he := alt + (sqr(mach * yc) / (2 * g));
end;

(******** MAIN PROGRAM *********)

(************* DECLARATION OF VARIABLES **********)

var
  GraphDriver, GraphMode : integer;
  x, y, value, tolerance, func,func2 : real;
  yalt, xmach, yt,yp,yd,yc : real;
  dx, dy : integer;
  hevar : real;


begin
  GraphDriver := Detect;
  DetectGraph(GraphDriver, GraphMode);
  InitGraph(GraphDriver, GraphMode, '');
  SetColor(3);
  Rectangle(0,0, GetMaxX, GetMaxY);
  line(trunc(trans*100),0,trunc(trans*100),getmaxy);

  (*********** structure limit line ***********)
  y:=0;
  x:=0;
  while (x<640) and (y<480) do
  begin
  x:=sqrt(struck/d(y*82.02))/sqrt((k*r*t(y*82.02)))*100;
  putpixel(trunc(x),trunc(480-y),1);
{  write('m#',x/100:3,' a',y*82.02:5,'
m','air',thrust(y*82.02,x/100)/isp(x/100)*40/g:5,'N','t= ',thrust(y*82.02,x/100):5);
  writeln('dr',drag(y*82.02,x/100));}
  y:=y+3;
  end;

  yalt:=250;
  dy :=480;
    repeat
      yt:=t(yalt*0.3048);
      yp:=p(yalt*0.3048);
      yd:=d(yalt*0.3048);
      yc:=sqrt(k*r*yt);
      xmach:=0.01;
      dx:=1;
      repeat
          tolerance:=0.04+0.18*xmach/6.4;
          hevar := he((yalt * 0.3048), xmach,yt,yp,yd,yc);
```

```
          if (((hevar / 3048.0) - trunc(hevar / 3048.0 )) < tolerance)
         then
           PutPixel(dx, dy, White { ((trunc(hevar / 3048) mod 15) + 1)} );
         xmach:=xmach+0.01;
         dx:=dx+1;
      until (xmach > 6.4);
      yalt:=yalt+250;
      dy:=dy-1;
   until (yalt > 120000);

   yalt:=250;
   dy :=480;
   repeat
      yt:=t(yalt*0.3048);
      yp:=p(yalt*0.3048);
      yd:=d(yalt*0.3048);
      yc:=sqrt(k*r*yt);
      xmach:=0.01;
      dx:=1;
      repeat
        writeln(x:6:2, ' ', y:8:0);
        func:=ps((yalt*0.3048),xmach,yt,yp,yd,yc);
         if (func<2000) and(d(yalt*0.3048)*xmach*xmach*k*r*yt<struck) then
         begin
         func2:=ps((yalt*0.3048),(xmach+0.02),yt,yp,yd,yc);
         tolerance:=(abs((func-func2))/70.0+0.3)/3;
         if (func>0) and (abs(func/200 - trunc(func/200)) < tolerance)
        then
          PutPixel(dx, dy,trunc(func/200) mod 15+1);
         end;
         xmach:=xmach+0.01;
         dx := dx+1;
      until (xmach > 6.4);
      yalt:=yalt+250;
      dy:=dy-1;
   until (yalt >120000);
end.
```

Hani Alexander
Ralph Jansen

Program: FS curves

This program plots the specific fuel consumption curves, and does a numerical intergration of fuel usage along a flight path.
The structural limit line and specific energy curves are also plotted.
Calculations are based on a numerical model of the standard atmosphere.

```
program fscurves(input, output);
uses Crt, Graph;
const
  r=287.04;
  g=9.8;
  c=1.94045e8;
  k=1.4;
  deltat=1;
  wgtf=5.77777e6;
  s=1400;
  cpd=0.1;
  pi=3.14159;
  ar=2;
  trans=3.7;
  struck=100000;
  sf=5.36;
  en=5;

  (********** EXPONENT FUNCTION *******)
function pow(x,y:real):real; {y^x}
  begin
     pow:=exp(x*ln(y));
  end;

(********* CORREPONDING BASE TEMPERATURE & SLOPE ********)
procedure val(var alt,t1,m,h,p1,d1:real);
  var
     a:integer;
  begin
     a:=trunc(alt/10);
     case a of
      0..1100:  begin
              t1:=288.16;h:=0;m:=-6.5e-3;p1:=1.01325e5;d1:=1.225;
             end;
      1101..2500: begin
              t1:=216.66;h:=11000;m:=0;p1:=2.27e4;d1:=3.648e-1;
             end;
      2501..4700: begin
              t1:=216.66;h:=25000;m:=3e-3;p1:=2.5273e3;d1:=4.0639e-2;
             end;
      4701..5300: begin
```

```
            t1:=282.66;h:=47000;m:=0;p1:=1.2558e2;d1:=1.5535e-3;
            end;
      5301..7900: begin
            t1:=282.66;h:=53000;m:=-4.5e-3;p1:=6.1493e1;d1:=7.5791e-4;
            end;
      7901..9000: begin
            t1:=165.66;h:=79000;m:=0;p1:=6.14e1;d1:=7.5791;
            end;
      9001..10500: begin
            t1:=165.66;h:=90000;m:=4e-3;p1:=6.14e1;d1:=7.5791;
            end;
      10501..32000: begin
            t1:=225.66;h:=105000;m:=0;p1:=6.14e1;d1:=7.5791;
            end;
    end;
  end;

(********** TEMPERATURE FUNCTION *********)

function t(alt:real):real;
  var
    tt,ht,mt,pt,dt: real;
  begin
    val(alt,tt,mt,ht,pt,dt);
    t:=tt+mt*(alt-ht)
  end;

(********** PRESSURE FUNCTION *********)

function p(alt:real):real;
var
  ap,tp,mp,hp,pp,dp:real;
begin
  ap:=alt;
  val(ap,tp,mp,hp,pp,dp);
  if (mp=0) then p:=pp*exp(-(g/(r*tp))*(ap-hp))
  else
    p:=pp*pow(-g/(mp*r),t(ap)/tp);
end;

(********* DENSITY FUNCTION *********)

function d(alt:real):real;
var
  ad,td,md,hd,pd,dd:real;

begin
  ad:=alt;
  val(ad,td,md,hd,pd,dd);
  if (md=0) then d:=dd*exp(-(g/(r*td))*(ad-hd))
```

```
  else
    d:=dd*pow(-(g/(md*r)+1),t(ad)/td);
end;

(*********** Isp FUNCTION *************)
function isp(mach:real):real;
  begin
    isp:=4200-15*mach*mach;
  end;

(*********** THRUST **************)
function thrust(alt,mach,yt,yp,yd,yc,wgt:real):real;
  var
   ai:real;
  begin
   if mach<1.3 then thrust:=sf*en*(30000-3846.2*mach)*4.44 else
   begin
     if mach<3.7 then ai:=20 else ai:=35;
     thrust:=isp(mach)*ai*g*mach*yc*yd/40;
   end;
  end;

(********** DRAG CALCULATIONS ***********)
function drag(alt, mach, yt, yp, yd, yc,wgt:real):real;

(**** cl = coefficient of lift, ar = aspect ratio, lambda = leading edge sweep ********)
var ar,cd,cdp,cdl,lambda, cl:real;

  begin
     if (mach<0.8) and (mach>0) then cdp:=0.011;
     if (mach>=0.8) and (mach<1.2) then cdp:=-0.0510+0.0762*mach;
     if (mach>=1.2) and (mach<7.0) then cdp:=0.0605 - 0.0177*mach +
0.00163*sqr(mach);
     cl:=2*wgt/(yd*s*sqr(mach*yc));
     cdl:=(0.1378 + 0.1693*mach - 0.0115*sqr(mach))*sqr(cl);
     cd:=cdl+cdp;
     drag:=0.5*yd*sqr(mach*yc)*s*cd;
  end;

(*********** specific power curve ************)
function ps(alt, mach,yt,yp,yd,yc,wgt : real):real;
var thrustvar, dragvar, fsvar : real;
begin
   thrustvar := thrust(alt, mach,yt,yp,yd,yc,wgt);
   dragvar := drag(alt, mach,yt,yp,yd,yc,wgt);
   fsvar := (mach * yc * (thrustvar - dragvar)) / wgtf;
{   writeln('Thrust = ', thrustvar:8:3, ' Drag := ', dragvar:8:3, ' FS := ', fsvar:8:3);}
   ps := fsvar;
end;
```

```
(*********** specific fuel consumption curve ********)
function fs(alt,mach,yt,yp,yd,yc,wgt : real): real;
begin
   fs:=ps(alt,mach,yt,yp,yd,yc,wgt)*isp(mach)/thrust(alt,mach,yt,yp,yd,yc,wgt);
end;

(************ energy height **************)
function he(alt, mach , yt, yp, yd, yc,wgt : real):real;
begin
   he := alt + (sqr(mach * yc) / (2 * g));
end;
(******** MAIN PROGRAM *********)
(************* DECLARATION OF VARIABLES **********)
var
  GraphDriver, GraphMode : integer;
  x, y, value, tolerance, func,func2, yalt2,dyalt,dhe : real;
  yalt, xmach, yt,yp,yd,yc,fwght,yt2,yp2,yd2,yc2 : real;
  dx, dy : integer;
  hevar,wgt,df : real;
begin
  wgt:=wgtf;
  fwght:=0;
  xmach:=0.3;
  repeat
    begin
      yalt:=20000*xmach;
      yalt2:=(xmach-0.01)*20000;
      yt:=t(yalt*0.3048);
      yp:=p(yalt*0.3048);
      yd:=d(yalt*0.3048);
      yc:=sqrt(k*r*yt);
      yt2:=t(yalt2*0.3048);
      yp2:=p(yalt2*0.3048);
      yd2:=d(yalt2*0.3048);
      yc2:=sqrt(k*r*yt2);
      dhe:=abs(he(yalt*0.3048,xmach,yt,yp,yd,yc,wgt)-he(yalt2*0.3048,xmach-
0.01,yt2,yp2,yd2,yc2,wgt));
      write('mach=',xmach:5,' alt=',yalt:5,'ft Thrust=');
      write(thrust(yalt*0.3048,xmach,yt,yp,yd,yc,wgt)/4.44:5,'lbs Drag=');
      writeln(drag(yalt*0.3048,xmach,yt,yp,yd,yc,wgt)/4.44:5,'lbs');
      writeln('weight=',wgt/4.44:11,'lbs');
      df:=dhe/fs(yalt*0.3048,xmach-0.01,yt,yp,yd,yc,wgt);
      fwght:=fwght+df;
      wgt:=wgt-df;
      xmach:=xmach+0.01;
    end;
  until (xmach>1.28);
  xmach:=1.28;
  repeat
    begin
```

```
      yalt:=12000+(35000*sqrt(xmach-1.2));
      yalt2:=12000+(35000*sqrt(xmach-1.21));
      yt:=t(yalt*0.3048);
      yp:=p(yalt*0.3048);
      yd:=d(yalt*0.3048);
      yc:=sqrt(k*r*yt);
      yt2:=t(yalt2*0.3048);
      yp2:=p(yalt2*0.3048);
      yd2:=d(yalt2*0.3048);
      yc2:=sqrt(k*r*yt2);
      dhe:=abs(he(yalt*0.3048,xmach,yt,yp,yd,yc,wgt)-he(yalt2*0.3048,xmach-
0.01,yt2,yp2,yd2,yc2,wgt));
      write('mach=',xmach:5,' alt=',yalt:5,'ft Thrust=');
      write(thrust(yalt*0.3048,xmach,yt,yp,yd,yc,wgt)/4.44:5,'lbs Drag=');
      writeln(drag(yalt*0.3048,xmach,yt,yp,yd,yc,wgt)/4.44:5,'lbs');
      writeln('weight=',wgt/4.44:11,'lbs');
      df:=dhe/fs(yalt*0.3048,xmach,yt,yp,yd,yc,wgt);
      fwght:=fwght+df;
      wgt:=wgt-df;
      xmach:=xmach+0.01;
    end;
  until (xmach>6.4);
  xmach:=6.4;
  repeat
    begin
      yalt:=xmach*-70000+540000;
      yalt2:=(xmach+0.01)*-70000+540000;
      yt:=t(yalt*0.3048);
      yp:=p(yalt*0.3048);
      yd:=d(yalt*0.3048);
      yc:=sqrt(k*r*yt);
      yt2:=t(yalt2*0.3048);
      yp2:=p(yalt2*0.3048);
      yd2:=d(yalt2*0.3048);
      yc2:=sqrt(k*r*yt2);
      dhe:=abs(he(yalt*0.3048,xmach,yt,yp,yd,yc,wgt)-
he(yalt2*0.3048,xmach+0.01,yt2,yp2,yd2,yc2,wgt));
      write('mach=',xmach:5,' alt=',yalt:5,'ft Thrust=');
      write(thrust(yalt*0.3048,xmach,yt,yp,yd,yc,wgt)/4.44:5,'lbs Drag=');
      writeln(drag(yalt*0.3048,xmach,yt,yp,yd,yc,wgt)/4.44:5,'lbs');
      writeln('weight=',wgt/4.44:11,'lbs');
      df:=dhe/fs(yalt*0.3048,xmach,yt,yp,yd,yc,wgt);
      fwght:=fwght+df;
      wgt:=wgt-df;
      xmach:=xmach-0.01;
    end;
  until (xmach<6);
  writeln('fuel=',fwght/4.44:7);
end.
```

Program: Flight Data

This program outputs thrust,drag and other parameters along the flight path and does a numerical integration of fuel usage along a flight path. The structural limit line and specific energy curves are also plotted. Calculations are based on a numerical model of the standard atmosphere. Thrust is based on a combination of actual performance data and an area change model.

```
program fscurves(input, output);
uses Crt, Graph;
const
  r=287.04;
  g=9.8;
  c=1.94045e8;
  k=1.4;
  deltat=1;
  wgtf=5.77777e6;
  s=1400;
  cpd=0.1;
  pi=3.14159;
  ar=2;
  trans=3.7;
  struck=100000;
  (********** EXPONENT FUNCTION *******)
function pow(x,y:real):real; {y^x}
  begin
     pow:=exp(x*ln(y));
  end;
(********* CORREPONDING BASE TEMPERATURE & SLOPE ********)
procedure val(var alt,t1,m,h,p1,d1:real);
  var
     a:integer;
  begin
     a:=trunc(alt/10);
     case a of
      0..1100:  begin
             t1:=288.16;h:=0;m:=-6.5e-3;p1:=1.01325e5;d1:=1.225;
            end;
      1101..2500: begin
             t1:=216.66;h:=11000;m:=0;p1:=2.27e4;d1:=3.648e-1;
            end;
      2501..4700: begin
             t1:=216.66;h:=25000;m:=3e-3;p1:=2.5273e3;d1:=4.0639e-2;
            end;
      4701..5300: begin
             t1:=282.66;h:=47000;m:=0;p1:=1.2558e2;d1:=1.5535e-3;
            end;
      5301..7900: begin
             t1:=282.66;h:=53000;m:=-4.5e-3;p1:=6.1493e1;d1:=7.5791e-4;
            end;
      7901..9000: begin
```

```
            t1:=165.66;h:=79000;m:=0;p1:=6.14e1;d1:=7.5791;
            end;
      9001..10500: begin
            t1:=165.66;h:=90000;m:=4e-3;p1:=6.14e1;d1:=7.5791;
            end;
      10501..32000: begin
            t1:=225.66;h:=105000;m:=0;p1:=6.14e1;d1:=7.5791;
            end;
    end;
  end;
(********** TEMPERATURE FUNCTION *********)
function t(alt:real):real;
  var
   tt,ht,mt,pt,dt: real;
  begin
   val(alt,tt,mt,ht,pt,dt);
   t:=tt+mt*(alt-ht)
  end;
(********** PRESSURE FUNCTION *********)
function p(alt:real):real;
var
  ap,tp,mp,hp,pp,dp:real;
begin
  ap:=alt;
  val(ap,tp,mp,hp,pp,dp);
  if (mp=0) then p:=pp*exp(-(g/(r*tp))*(ap-hp))
  else
   p:=pp*pow(-g/(mp*r),t(ap)/tp);
end;
(********* DENSITY FUNCTION *********)
function d(alt:real):real;
var
  ad,td,md,hd,pd,dd:real;
begin
  ad:=alt;
  val(ad,td,md,hd,pd,dd);
  if (md=0) then d:=dd*exp(-(g/(r*td))*(ad-hd))
  else
   d:=dd*pow(-(g/(md*r)+1),t(ad)/td);
end;
(*********** Isp FUNCTION *************)
function isp(mach:real):real;
   begin
    isp:=4200-15*mach*mach;
   end;
(*********** THRUST **************)
function thrust(alt,mach,yt,yp,yd,yc:real):real;
   var
    ai:real;
   begin
```

```
    if mach<3.7 then ai:=40 else ai:=35;
    thrust:=isp(mach)*ai*g*mach*yc*yd/40;
  end;
(********** DRAG CALCULATIONS ***********)
function drag(alt, mach, yt, yp, yd, yc:real):real;
(**** cl = coefficient of lift, ar = aspect ratio, lambda = leading edge sweep ********)
var ar,cd,cdp,cdl,lambda, cl:real;
  begin
     if (mach<0.8) and (mach>0) then cdp:=0.011;
     if (mach>=0.8) and (mach<1.2) then cdp:=-0.0510+0.0762*mach;
     if (mach>=1.2) and (mach<7.0) then cdp:=0.0605 - 0.0177*mach +
0.00163*sqr(mach);
     cl:=2*wgtf/(yd*s*sqr(mach*yc));
     cdl:=(0.1378 + 0.1693*mach - 0.0115*sqr(mach))*sqr(cl);
     cd:=cdl+cdp;
     drag:=0.5*yd*sqr(mach*yc)*s*cd;
  end;
(*********** specific power curve ************)
function ps(alt, mach,yt,yp,yd,yc : real):real;
var thrustvar, dragvar, fsvar : real;
begin
   thrustvar := thrust(alt, mach,yt,yp,yd,yc);
   dragvar := drag(alt, mach,yt,yp,yd,yc);
   fsvar := (mach * yc * (thrustvar - dragvar)) / wgtf;
{   writeln('Thrust = ', thrustvar:8:3, ' Drag := ', dragvar:8:3, ' FS := ', fsvar:8:3);}
   ps := fsvar;
end;
(*********** specific fuel consumption curve ********)
function fs(alt,mach,yt,yp,yd,yc : real): real;
begin
   fs:=ps(alt,mach,yt,yp,yd,yc)*isp(mach)/thrust(alt,mach,yt,yp,yd,yc);
end;
(************ energy height **************)
function he(alt, mach , yt, yp, yd, yc : real):real;
begin
   he := alt + (sqr(mach * yc) / (2 * g));
end;
(******** MAIN PROGRAM *********)

(************* DECLARATION OF VARIABLES **********)

var
   GraphDriver, GraphMode : integer;
   x, y, value, tolerance, func,func2 : real;
   yalt, xmach, yt,yp,yd,yc : real;
   dx, dy : integer;
   hevar : real;
begin
   GraphDriver := Detect;
   DetectGraph(GraphDriver, GraphMode);
```

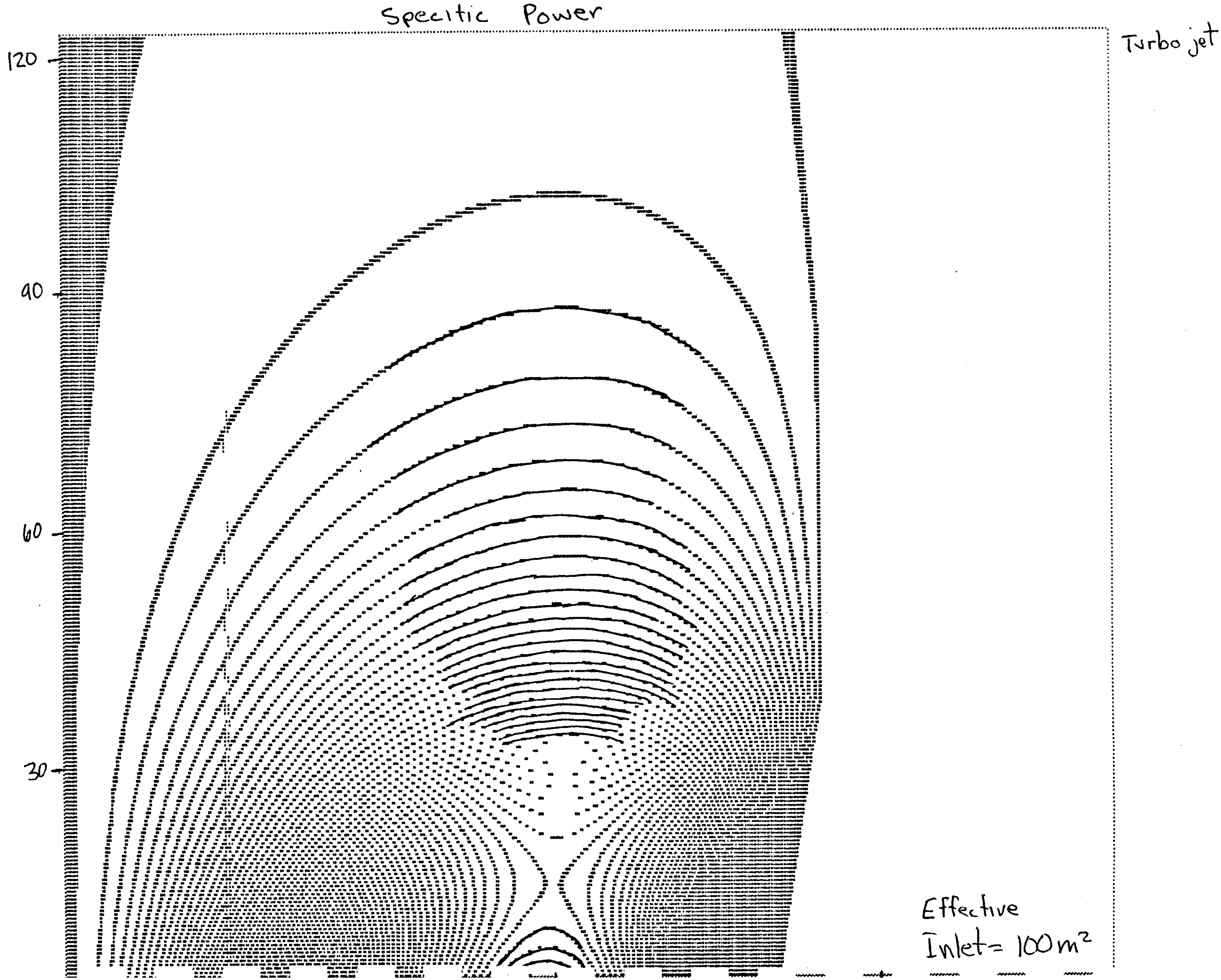
```
InitGraph(GraphDriver, GraphMode, '');
SetColor(3);
Rectangle(0,0, GetMaxX, GetMaxY);
line(trunc(trans*100),0,trunc(trans*100),getmaxy);
(*********** structure limit line ***********)
y:=0;
x:=0;
while (x<640) and (y<480) do
begin
x:=sqrt(struck/d(y*76.2))/sqrt((k*r*t(y*76.2)))*100;
putpixel(trunc(x),trunc(480-y),1);
{  write('m#',x/100:3,' a',y*76.02:5,' m','air',thrust(y*76.2,x/100)/isp(x/100)*40/g:5,'N','t=
',thrust(y*76.2,x/100):5);
writeln('dr',drag(y*76.2,x/100));}
y:=y+3;
end;
xmach:=0.3;
repeat
  begin
    yalt:=xmach*20000;
    putpixel(trunc(xmach*100),480-trunc(yalt/250),red);
    xmach:=xmach+0.01;
  end;
until (xmach>1.5);
xmach:=1.5;
repeat
  begin
    yalt:=12000+(35000*sqrt(xmach-1.2));
    putpixel(trunc(xmach*100),480-trunc(yalt/250),red);
    xmach:=xmach+0.01;
  end;
until (xmach>6.4);
xmach:=6.4;
repeat
  begin
    yalt:=xmach*-70000+540000;
    putpixel(trunc(xmach*100),480-trunc(yalt/250),red);
    xmach:=xmach-0.01;
  end;
until (xmach<6.0);
{  yalt:=250;
dy :=480;
  repeat
    yt:=t(yalt*0.3048);
    yp:=p(yalt*0.3048);
    yd:=d(yalt*0.3048);
    yc:=sqrt(k*r*yt);
    xmach:=0.01;
    dx:=1;
    repeat
```

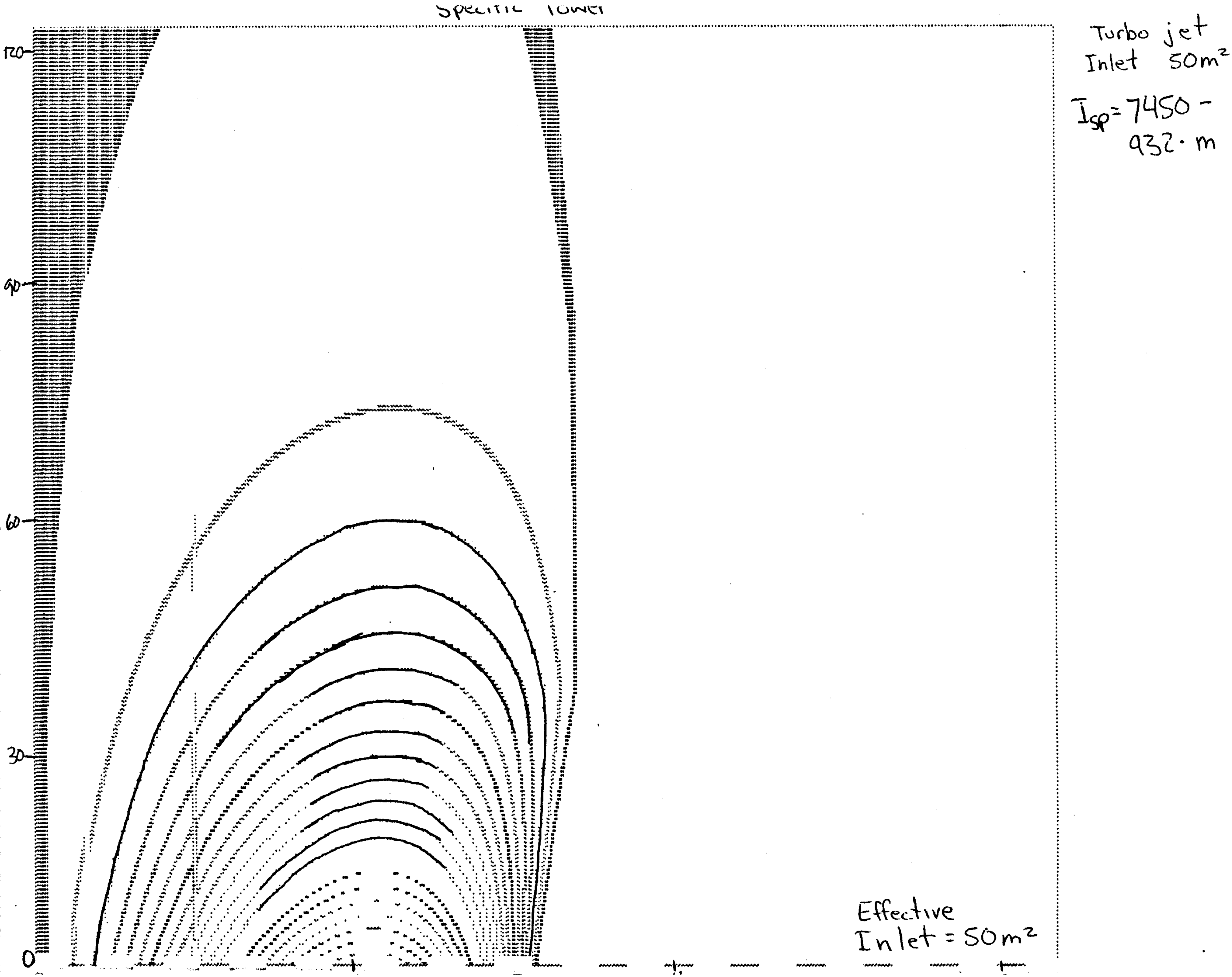
```
          tolerance:=0.04+0.18*xmach/6.4;
          hevar := he((yalt * 0.3048), xmach,yt,yp,yd,yc);
          if (((hevar / 3048.0) - trunc(hevar / 3048.0 )) < tolerance)
         then
           PutPixel(dx, dy, White);
         xmach:=xmach+0.01;
         dx:=dx+1;
      until (xmach > 6.4);
      yalt:=yalt+250;
      dy:=dy-1;
  until (yalt > 120000);
 }
  yalt:=250;
  dy :=480;
  repeat
      yt:=t(yalt*0.3048);
      yp:=p(yalt*0.3048);
      yd:=d(yalt*0.3048);
      yc:=sqrt(k*r*yt);
      xmach:=0.01;
      dx:=1;
      repeat
{         writeln(x:6:2, ' ', y:8:0);
}         func:=fs((yalt*0.3048),xmach,yt,yp,yd,yc);
         if (func<2) then
         begin
         func2:=fs((yalt*0.3048),(xmach+0.02),yt,yp,yd,yc);
         tolerance:=(abs((func-func2))/70.0+0.3)/3;
         if (func>0) and (abs(func/0.1 - trunc(func/0.1)) < tolerance)
        then
           PutPixel(dx, dy,trunc(func/0.1) mod 15+1);
         end;
         xmach:=xmach+0.01;
         dx := dx+1;
      until (xmach > 6.4);
      yalt:=yalt+250;
      dy:=dy-1;
  until (yalt >120000);
end.
```

APPENDIX C

PS CURVES USING PRELIMINARY MODEL

1) **Turbojet 100 square meter inlet**
2) **Turbojet 50 square meter inlet**
3) **Turbojet 25 square meter inlet**
4) **Specific Energy curves**

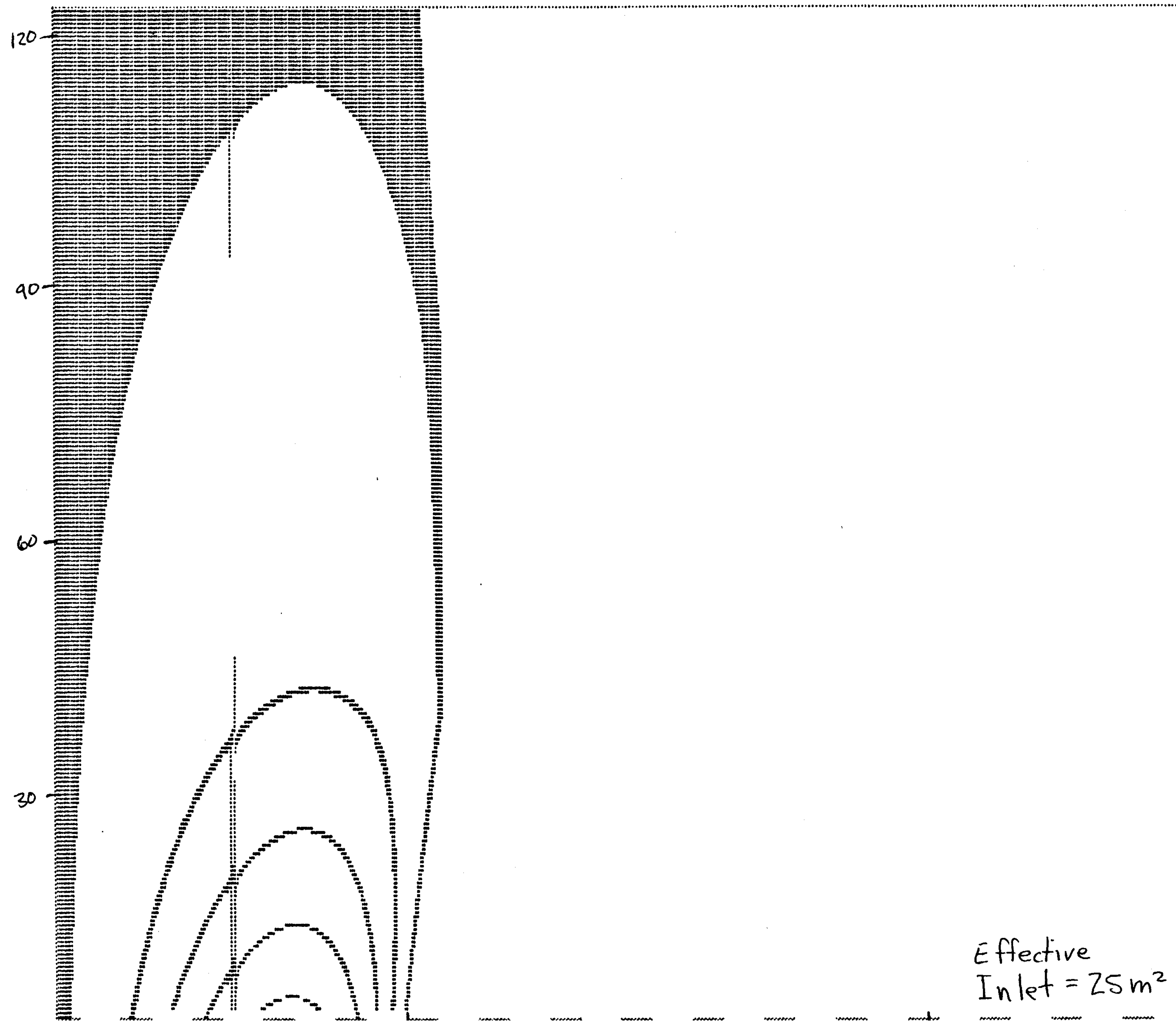

Specific Power
Turbo jet
120
90
60
30
Effective
Inlet = 100 m²

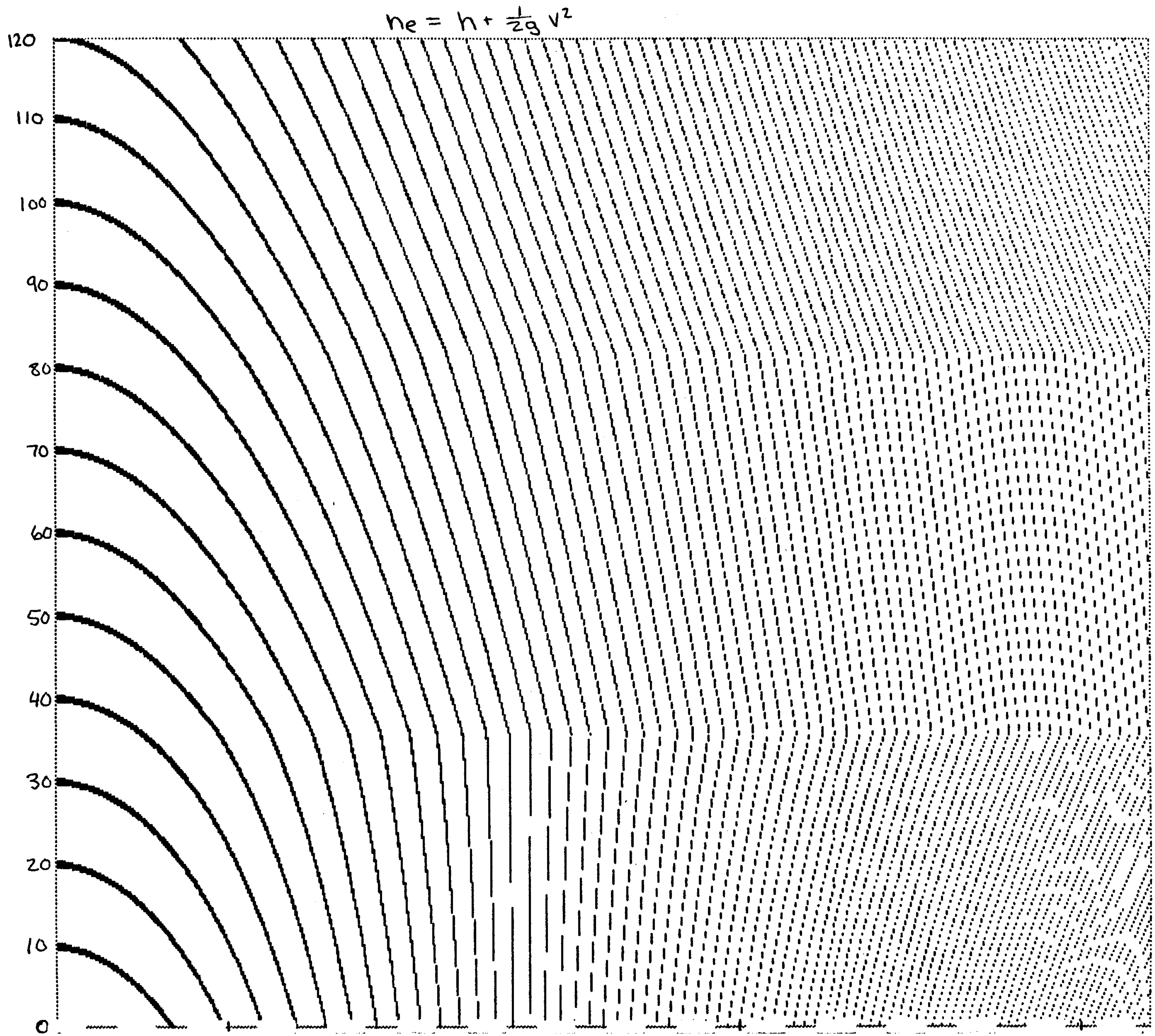


Specific Power
120
90
60
30
0
Effective
Inlet = 50m²
Turbo jet
Inlet 50m²
I_{sp} = 7450 -
932 · m

Specific Power

Turbo Inlet 25m²





he = h + 1/2g V^2
ft x10^3
120
110
100
90
80
70
60
50
40
30
20
10
0

APPENDIX D

INLET SPECIFICATIONS

DESIGN CONDITIONS

MACH 1	=	6						
MACH 2	=	4.93	p/p	=	0.89			
DELTA	=	7.86	r/r	=	2.05			
SIGMA	=	15.60	T/T	=	1.4			
MACH 3	=	4.0	p/p	=	0.89			
DELTA	=	9.53	r/r	=	2.05	p/p	=	0.54
SIGMA	=	19.09	T/T	=	1.4			
MACH 4	=	3.17	p/p	=	0.89			
DELTA	=	11.69	r/r	=	2.05			
SIGMA	=	23.80	T/T	=	1.4			
MACH 5	=	2.55	p/p	=	0.94			
DELTA	=	11.69	r/r	=	1.67			
SIGMA	=	27.83	T/T	=	1.25			
MACH 6	=	2.06	p/p	=	0.96			
DELTA	=	11.69	r/r	=	1.56			
SIGMA	=	32.92	T/T	=	1.21			
MACH 7	=	1.63	p/p	=	0.98			
DELTA	=	11.69	r/r	=	1.51			
SIGMA	=	40.03	T/T	=	1.19			

CAPTURE AREA	=	533.61
WIDTH	=	40.0
DEPTH	=	13.34
COWL AREA	=	124.32
WIDTH	=	40.0
DEPTH	=	3.11
THROAT AREA	=	18.58
WIDTH	=	40.0
DEPTH	=	0.466
COMPRESSOR INTAKE AREA	=	25.66
WIDTH	=	40.0
DEPTH	=	0.64
WEDGE ANGLE	=	11.69

MACH = 4

MACH 1	=	4						
MACH 2	=	3.74	p/p	=	1			
DELTA	=	3.50	r/r	=	1.28			
SIGMA	=	16.89	T/T	=	1.1			
MACH 3	=	3.06	p/p	=	0.93			
DELTA	=	10.5	r/r	=	1.87	p/p	=	0.78
SIGMA	=	23.71	T/T	=	1.32			
MACH 4	=	2.39	p/p	=	0.92			
DELTA	=	13.31	r/r	=	1.92			
SIGMA	=	30.1	T/T	=	1.34			
MACH 5	=	1.85	p/p	=	0.95			
DELTA	=	13.31	r/r	=	1.73			
SIGMA	=	36.55	T/T	=	1.23			
MACH 6	=	1.37	p/p	=	0.97			
DELTA	=	13.31	r/r	=	1.61			
SIGMA	=	46.94	T/T	=	1.22			

CAPTURE AREA	=	447.78
WIDTH	=	40.0
DEPTH	=	11.19
COWL AREA	=	157.15
WIDTH	=	40.0
DEPTH	=	3.94
THROAT AREA	=	39.40
WIDTH	=	40.0
DEPTH	=	0.98

WEDGE ANGLE = 13.31

MACH = 2

MACH 1	=	2						
MACH 2	=	1.91	p/p	=	1			
DELTA	=	2.50	r/r	=	1.10			
SIGMA	=	32.07	T/T	=	1.04			
MACH 3	=	1.79	p/p	=	1			
DELTA	=	3.5	r/r	=	1.14	p/p	=	1
SIGMA	=	34.61	T/T	=	1.06			
MACH 4	=	1.33	p/p	=	1			
DELTA	=	12.75	r/r	=	1.57			
SIGMA	=	48.17	T/T	=	1.21			

CAPTURE AREA	=	442.61
WIDTH	=	40.0
DEPTH	=	11.06
COWL AREA	=	318.29
WIDTH	=	40.0
DEPTH	=	7.97
THROAT AREA	=	203.01
WIDTH	=	40.0
DEPTH	=	5.09

WEDGE ANGLE = 12.75

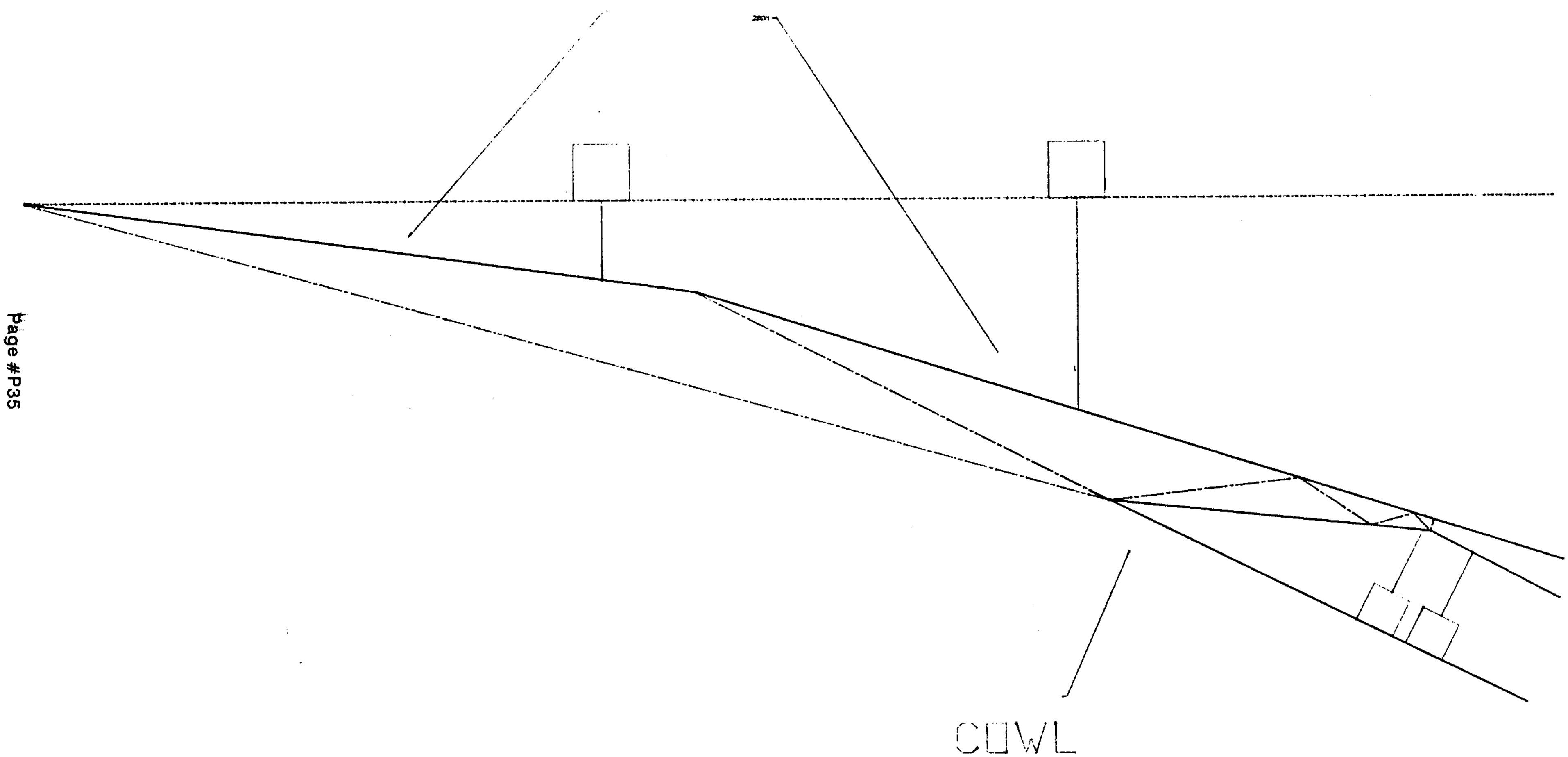
MACH = 0.9

CAPTURE AREA	=	430.56
WIDTH	=	40.0
DEPTH	=	10.76
THROAT AREA	=	426.79
WIDTH	=	40.0
DEPTH	=	9.94
WEDGE ANGLE	=	0.46

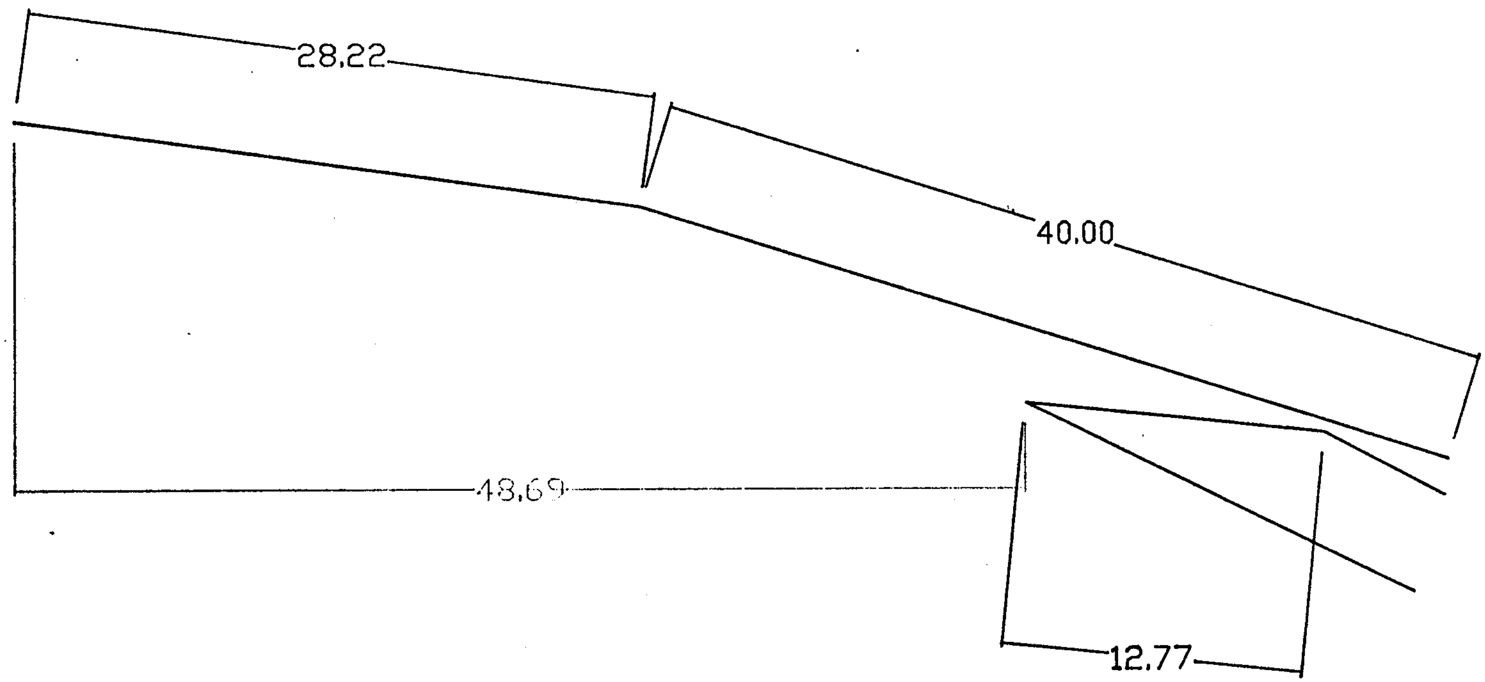
MACH = 0.1

CAPTURE AREA	=	435.62
WIDTH	=	40.0
DEPTH	=	10.89

VARIABLE GEOMETRY COMPRESSION RAMPS

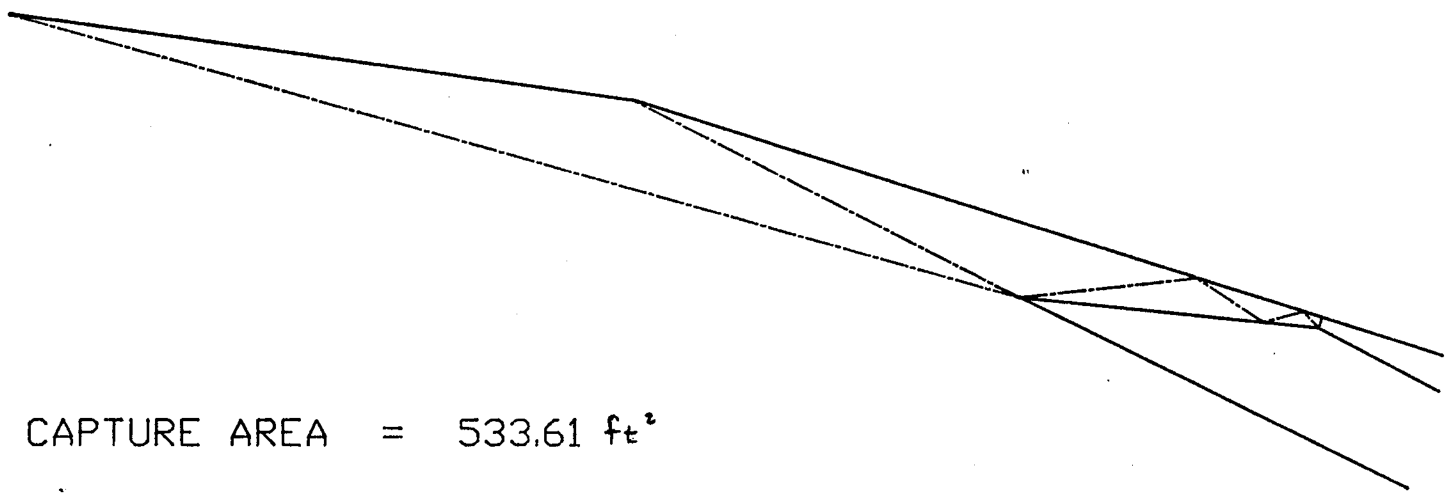


Page #P35


28.22
40.00
48.69
12.77

Page #P37

MACH 6 INLET



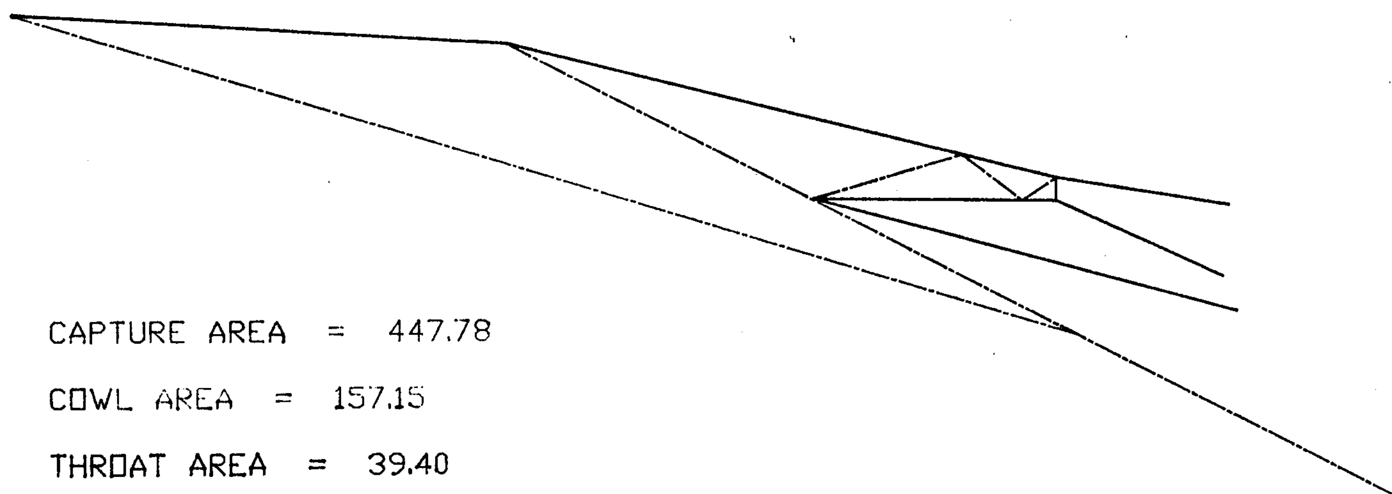
CAPTURE AREA = 533.61 ft^2

COWL AREA = 124.32

THROAT AREA = 18.58

Page #P38

MACH 4 INLET

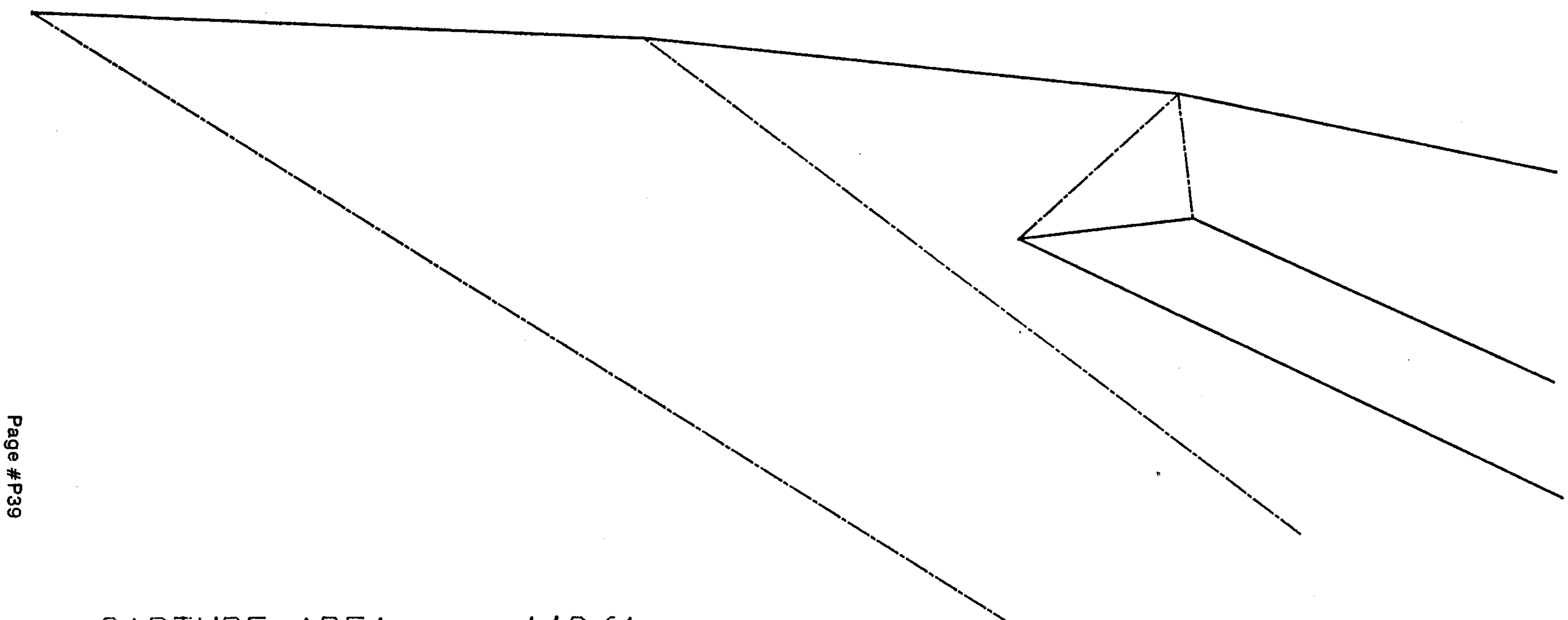


CAPTURE AREA = 447.78

COWL AREA = 157.15

THROAT AREA = 39.40

MACH 2 INLET

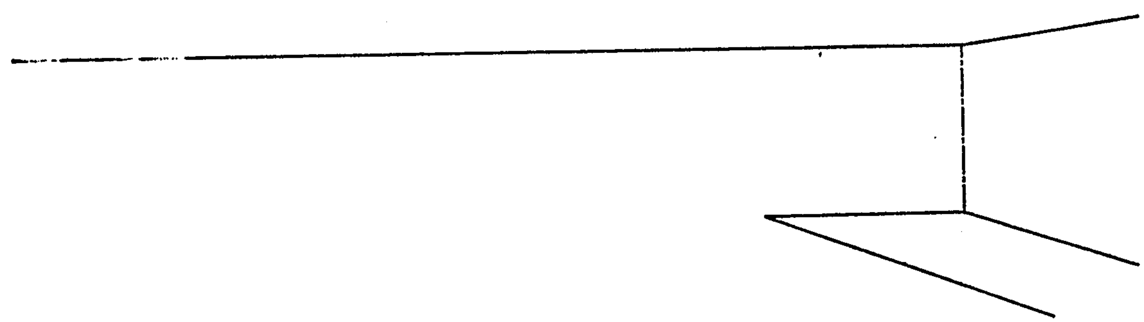


CAPTURE AREA = 442.61

COWL AREA = 318.29

THROAT AREA = 203

MACH .9 INLET

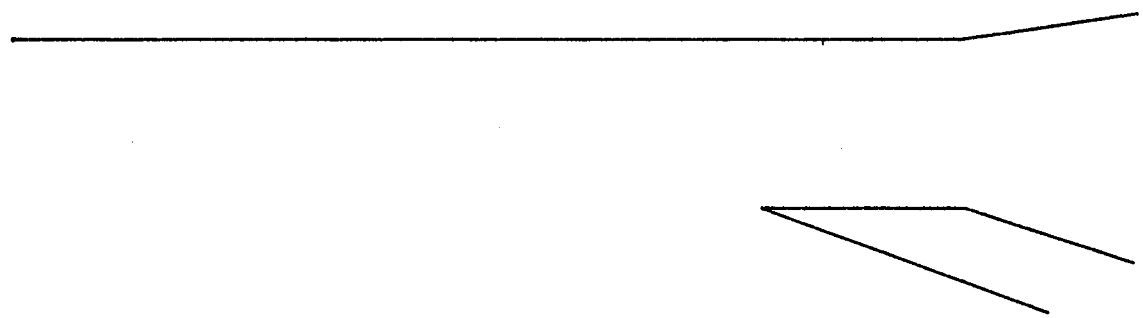


CAPTURE AREA = 430.56

COWL AREA = 430.56

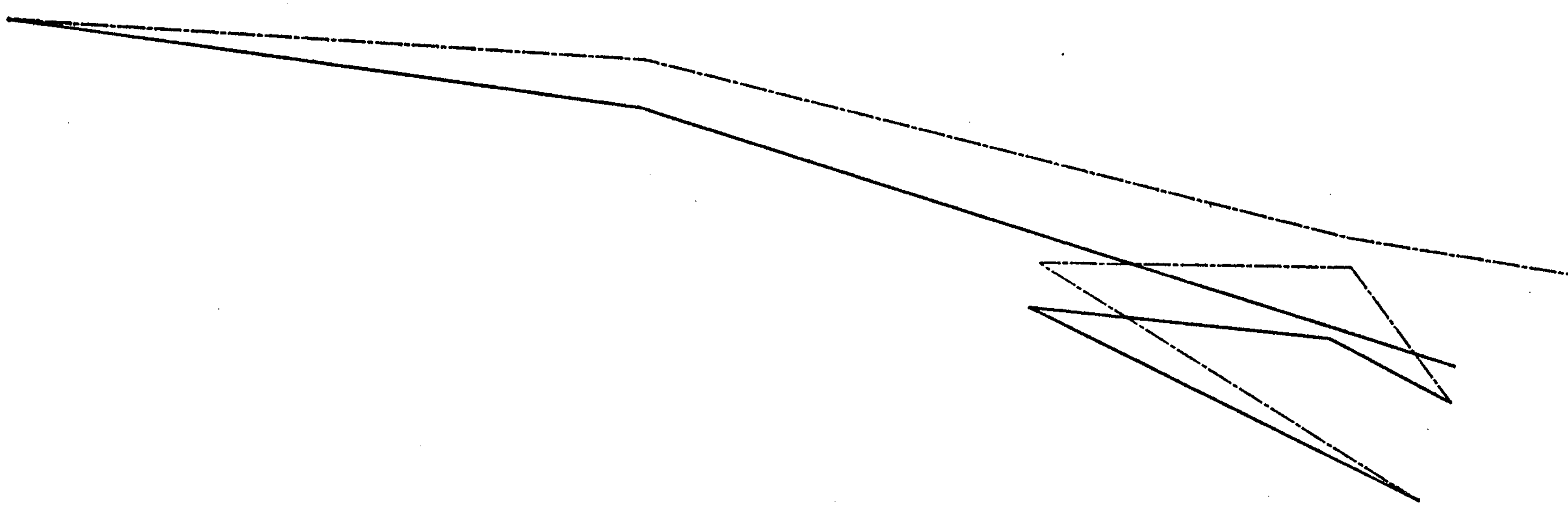
THROAT AREA = 426.79

MACH .1 INLET

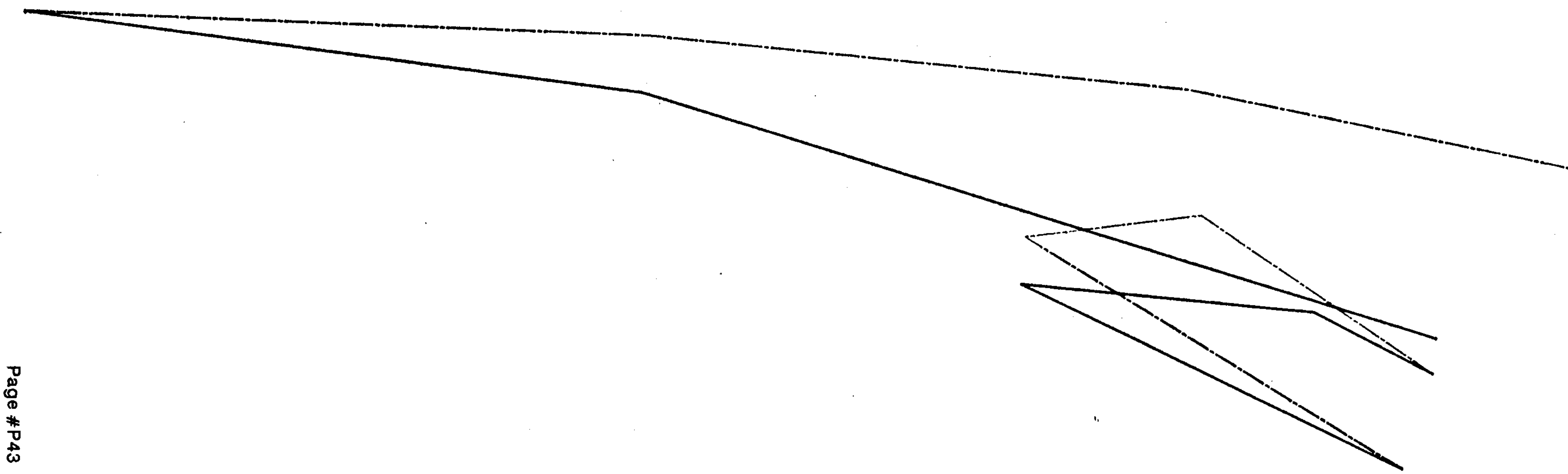


CAPTURE AREA = 426.789

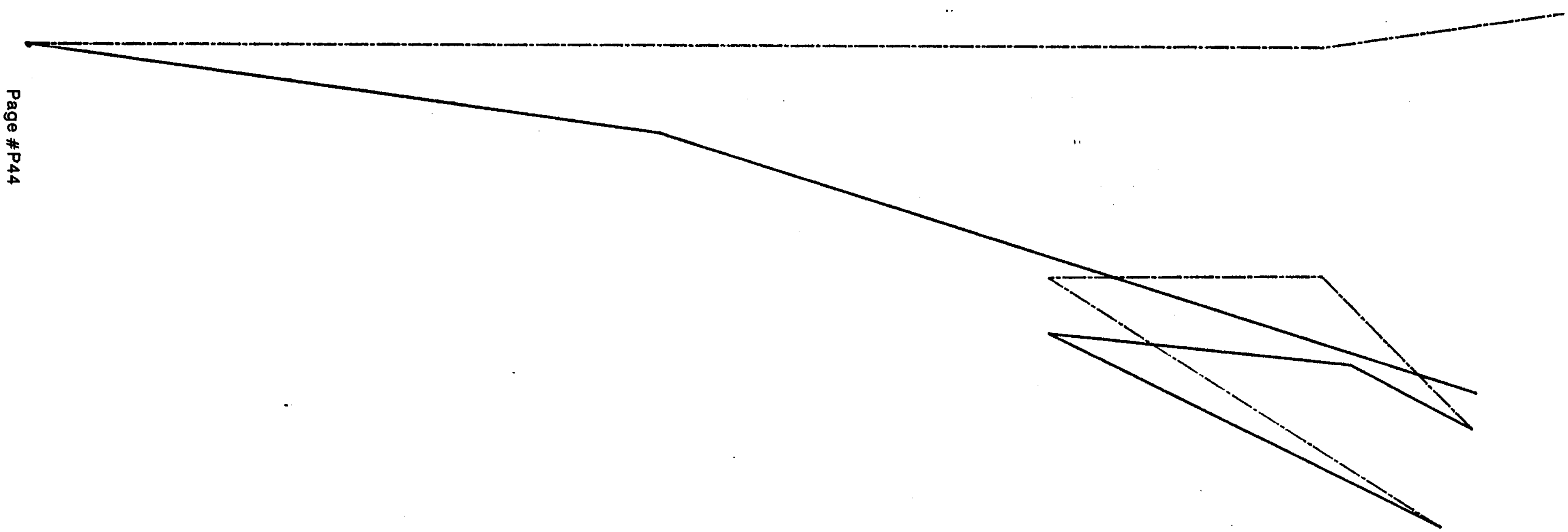
MACH 6 INLET -- MACH 4 INLET



MACH 6 INLET -- MACH 2 INLET

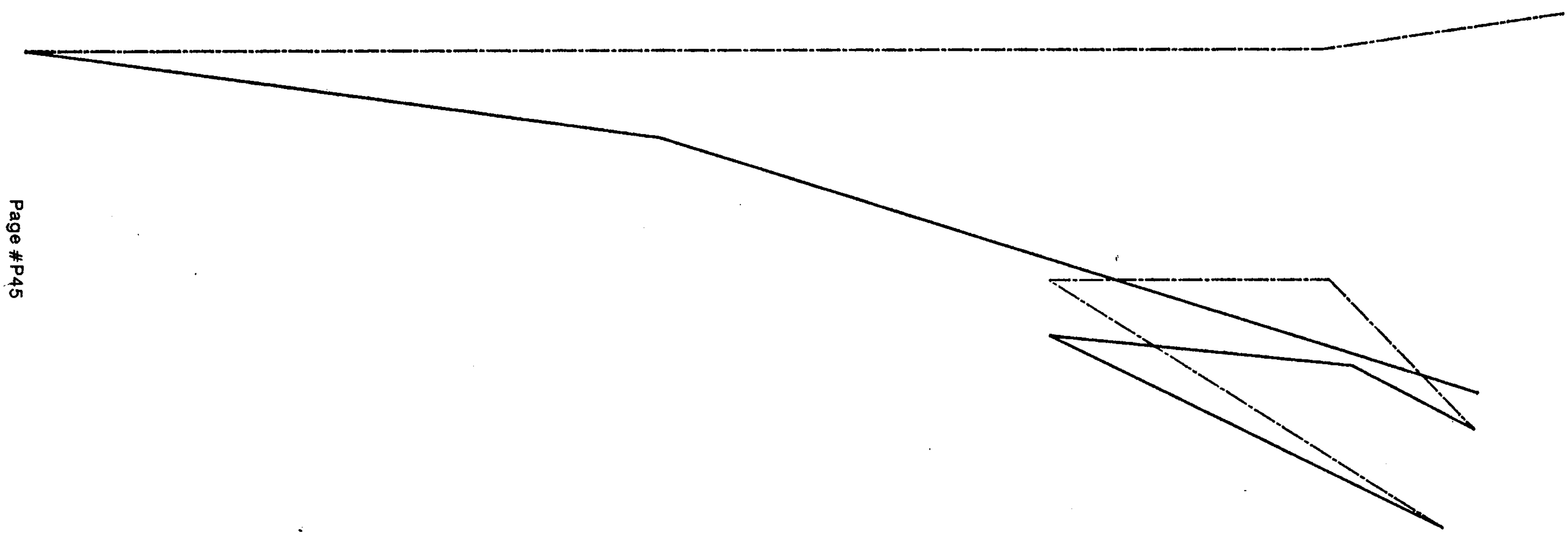


MACH 6 INLET -- MACH .9 INLET



Page #P44

MACH 6 INLET -- MACH .1 INLET



STRUCTURE GROUP

Darren Blue
Jose Rivera

STRUCTURAL ANALYSIS

INTRODUCTION

For the conceptual design of a two-stage land to space vehicle, the structural analysis is very complicated. For the final design of the vehicle, structural considerations play a vital and integral role. Careful study and analysis is necessary to strive for a structurally sound vehicle that must withstand high temperatures and loads while remaining as lightweight as possible. It has been estimated that the booster and orbiter will obtain a speed of Mach 6 at 100,000 feet before separation. This demands consideration of the weight of the vehicle, aerodynamic heating, advanced high-temperature materials, high strength-to-weight materials, active cooling, special coatings, and the strength and design of the understructure of the craft.

Aerodynamic heating was studied according to temperatures that will be encountered and materials were chosen based on those temperatures. After the materials for the craft were chosen, a weight of the booster was estimated. Next, the analysis of the wing understructure was done focusing on spar sizing and control surface strength. In addition, the fuselage hoop and keel structure was analyzed and sized. Finally, the vertical tails and landing gear were taken into account.

AERODYNAMIC HEATING

The booster is designed to achieve a speed of Mach 6 at an altitude of 100,000 feet. This hypersonic speed causes high friction on the skin of the vehicle. In turn, high temperatures on the order of 1000 to 2000 degrees Fahrenheit are encountered. The highest temperatures on the vehicle will occur at stagnation points. Areas of concern will be the leading edges, mainly the nose and the wing leading edges. Because of these high temperatures, materials that can withstand them and still possess good strength and low density characteristics are needed. The materials found to be of primary interest included titanium-aluminide, titanium-based metal-matrix composites, carbon-carbon composites, ceramic-matrix composites, copper-matrix composites, and beryllium alloys. Most of these materials were investigated in NASP studies.

These materials were found to possess the favorable qualities of high strength, low density, endurance to high temperatures, and resistance to creep. The goals in selecting these materials for parts of the booster were to avoid active cooling and resist deterioration from the elements while still being capable of maintaining high strength-to-weight. The main source of cooling will be left up to radiation of heat away from the vehicle. After estimates of the heating and stagnation temperatures were completed, materials were selected.

Titanium-aluminide was chosen for the skin of the booster because of its high temperature characteristics and good strength-to-weight ratio. Titanium-aluminide is capable of withstanding temperatures up to 1800 degrees Fahrenheit and has virtually the same density as titanium. It is also capable of enduring the elements. At the critical area of the leading edge of the wing, it was calculated that a minimum radius of one inch could be used for this alloy without the need for active cooling.

For the nose, a carbon-carbon composite was selected because of its extemely high capacity for temperatures exceeding 2500 degrees Fahrenheit. Carbon-carbon is a brittle material, but the material should not be exposed to any loading or twisting that it is incapable of handling because the stresses in the nose will not be high.

Titanium was chosen for the understructure of the booster mainly because of its high strength-to-weight ratio. Titanium will be used for all spars and remaining structure.

The final consideration of aerodynamic heating was to investigate if and how the booster should be coated. The main reason for coating the vehicle is to increase the emissivity level. Also, the carbon-carbon nose needs protection from oxidation at high temperatures. For protection and increased emissivity, a black coating was chosen to simulate a black body. This type of coating should be able to bring the emissivity level close to a value of one.

WEIGHT ESTIMATION

An important part of any aerospace design is a weight analysis. The weight goal for this vehicle was 1.3 million pounds. This consisted of a one million pound booster and a 300,000 pound orbiter. The weight estimation was calculated using a "Hypersonic Aerospace Analysis for the Preliminary Design of Aerospace Vehicles" paper from NASA Lewis Research Center. This paper proved to be very helpful because it was specifically written for hypersonic vehicles. The results of the weights analysis are shown below.

BODY	=	209,361 lbs
WING	=	230,729
VERTICAL TAIL	=	22,512
LANDING GEAR	=	68,230
TRUST STRUCTURE	=	4,569
TOTAL STRUCTURE	=	535,400
ENGINES	=	80,400
FUEL TANK	=	16,000
SUBSYSTEMS	=	40,506
FUEL	=	160,000
ORBITER	=	301,300
G.L.O.W.	=	1,133,000 lbs

The estimated total take-off weight is well under the target weight of 1.3 million pounds. This figure includes the final orbiter weight of 301,300 pounds which will be presented later in this report.

WING STRUCTURE

In order to analyze the wing, the first decisions made were to determine what loads the wing will see. The three main loadings that will affect the wing structure are inertia, drag, and static loading. The largest by far is the loading due to inertia, therefore, an assumption was made that if the wing was capable of carrying the inertial load, it would be capable of carrying the other loads. The inertial loads were given by the aerodynamics group to be 2 g's. In order to incorporate a factor of safety into our design for the security of the pilots, a determination was made to design for an ultimate inertial loading of 3 g's.

The first step in the wing structure design was to determine the pressure distribution from the root of the wing to the tip. This pressure distribution was determined by using Figure 4.22 in D. Raymer's book, Aircraft Design: A Conceptual Approach. By knowing the taper ratio to be 0.2 and calculating the mean lift, the pressure was calculated at various locations along the wing. The result of these calculations is shown in figure S1. To determine the chordwise pressure distribution, Dr. T. Kicher and the aerodynamics group were consulted and a general shape for this distribution was assumed. This shape is shown in figure S2. From these distributions the loading at any point on the wing could be determined.

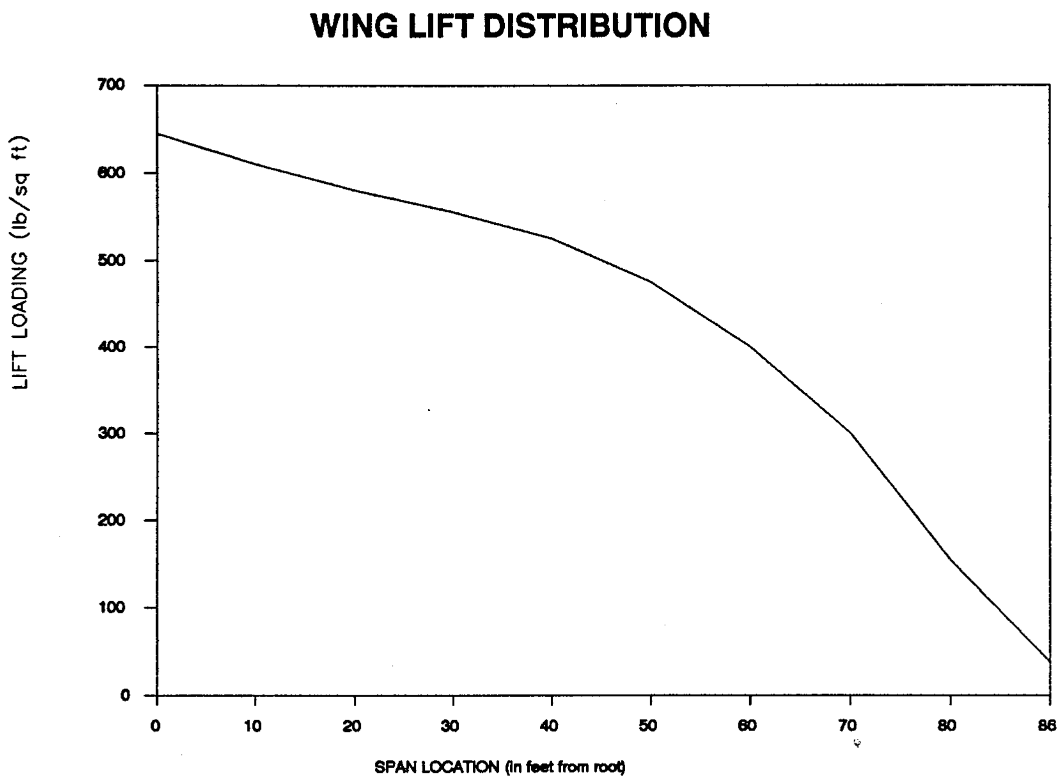


Figure S1

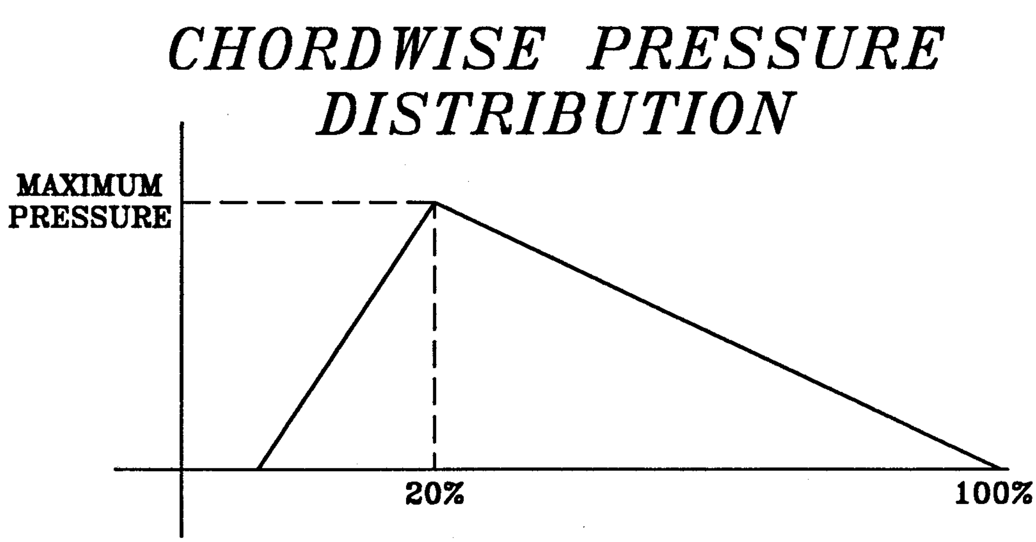


Figure S2

The next step was to determine the number and orientation of spars in the wing. After several conceptual ideas, it was decided that five spars would be used and placed perpendicular to the fuselage. These spars would be boxed in on their ends with another structural member and connected with ribs to transmit the loads from all locations on the wing to the spars. The ribs would also form the shape of the wing to be covered with the skin. The general structural layout of the wing is shown in figure S3.

WING STRUCTURE

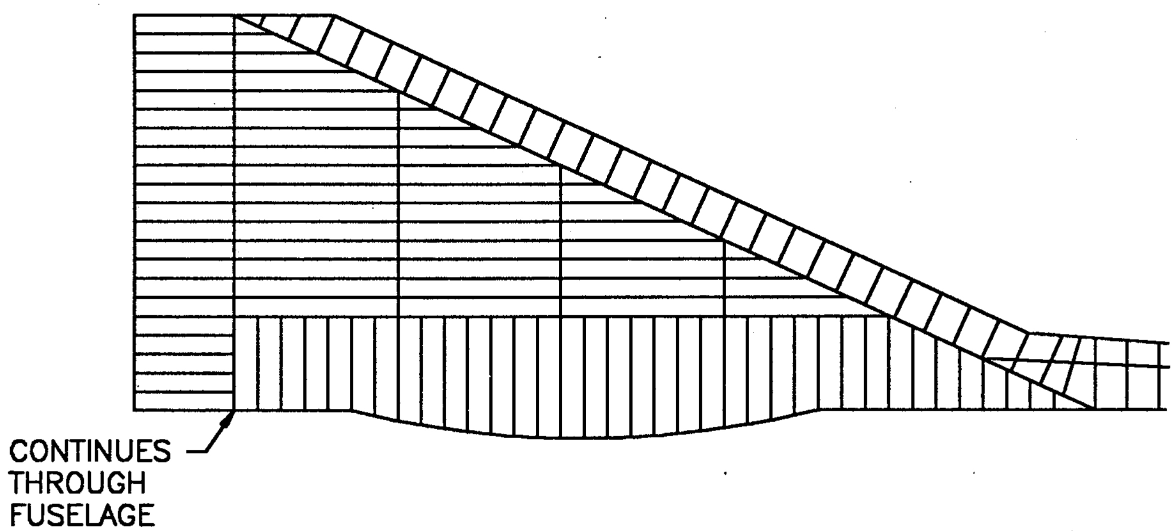
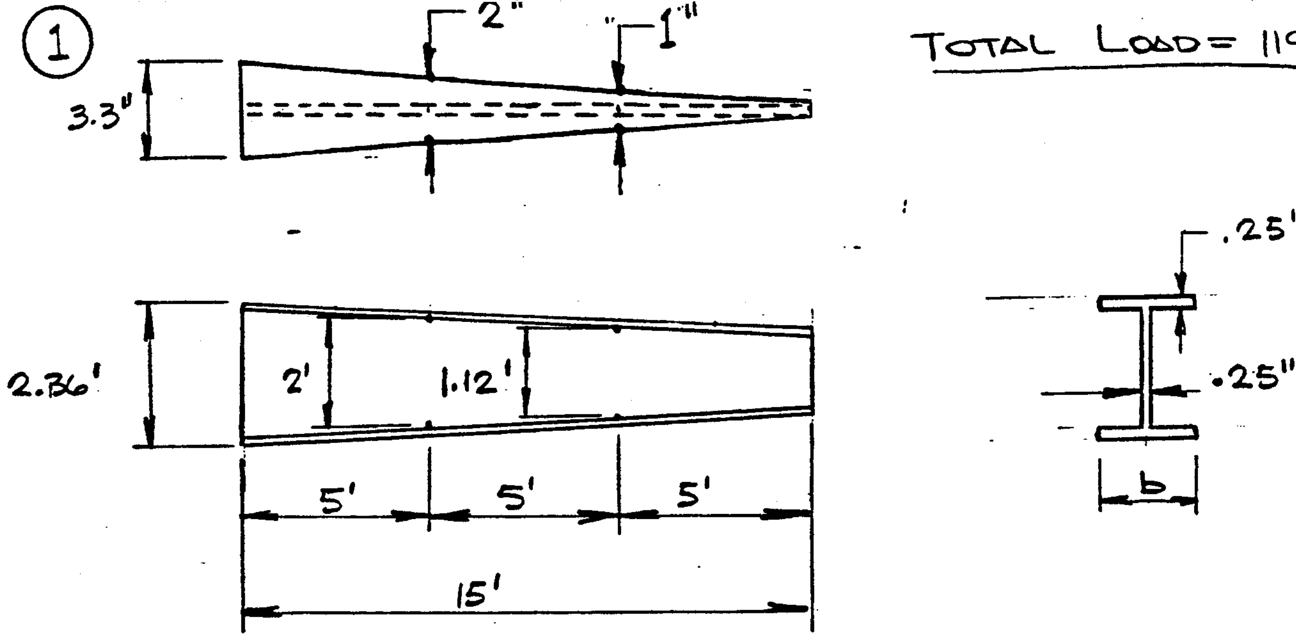


Figure S3

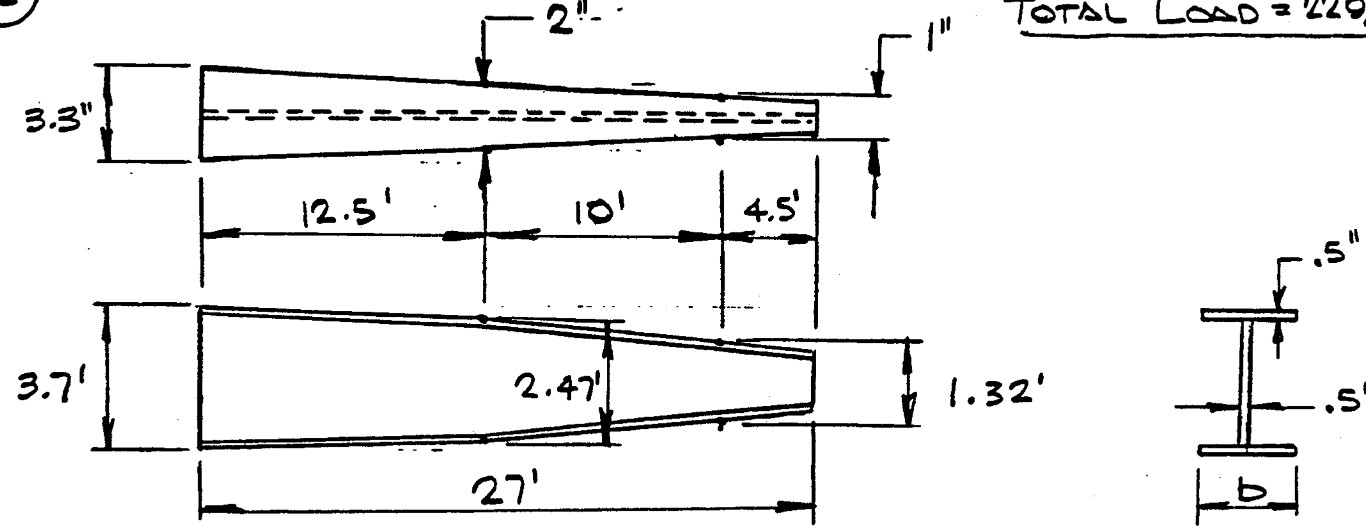
In order to do the actual sizing of the spars, two main assumptions were made. The first assumption was to assume that each wing spar was statically determinate. Actually, the entire wing structure is a statically indeterminate system, but in this case the statically indeterminate system was not readily solvable. The second assumption was to assume the general shape of each spar was similar to an I-beam. These assumptions are reasonable to make for a first cut analysis and enabled the best analysis with the resources available. Using the assumptions and the previously found pressure distributions, the loadings were determined for each individual spar. From the loadings, the moment at several locations on each beam was calculated. Then, using simple beam analysis and the yield strength of titanium, the moments of inertia of each spar at critical points were determined. The calculated moments of inertia were used to define the actual size and taper of each individual spar (see Figure S4).

1



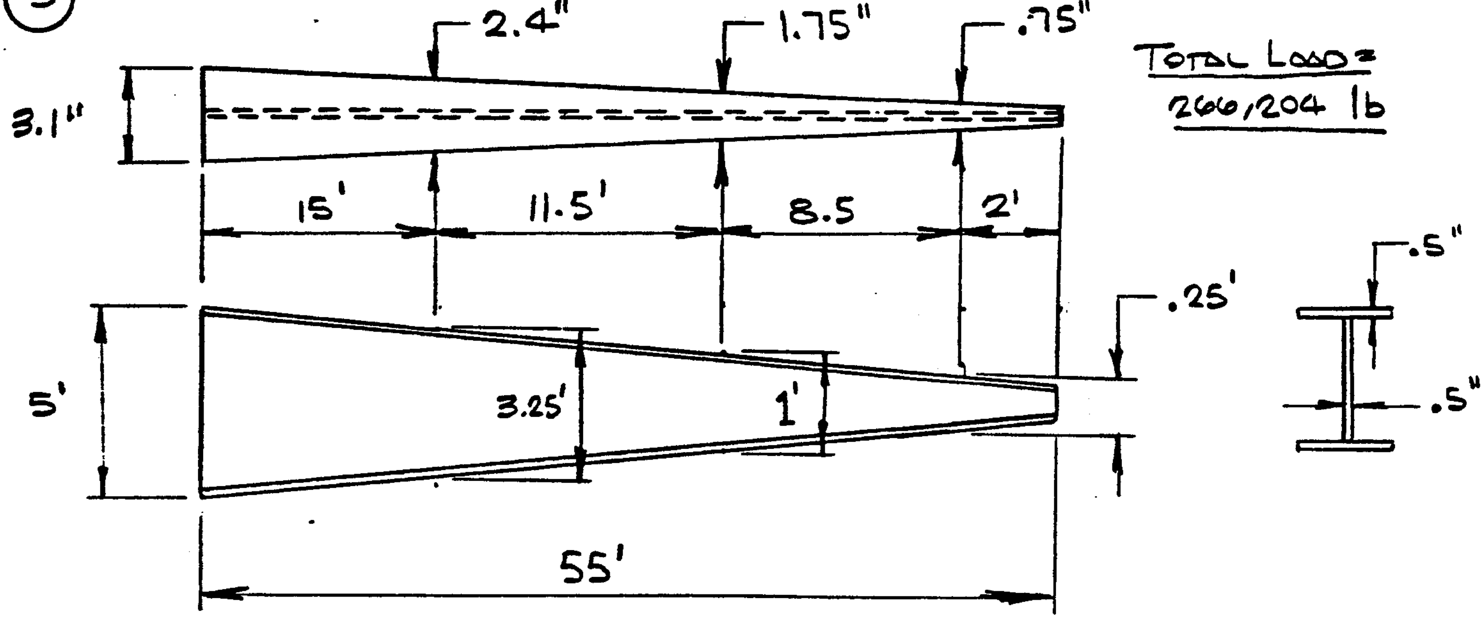
TOTAL LOAD = 119,050 lb

2

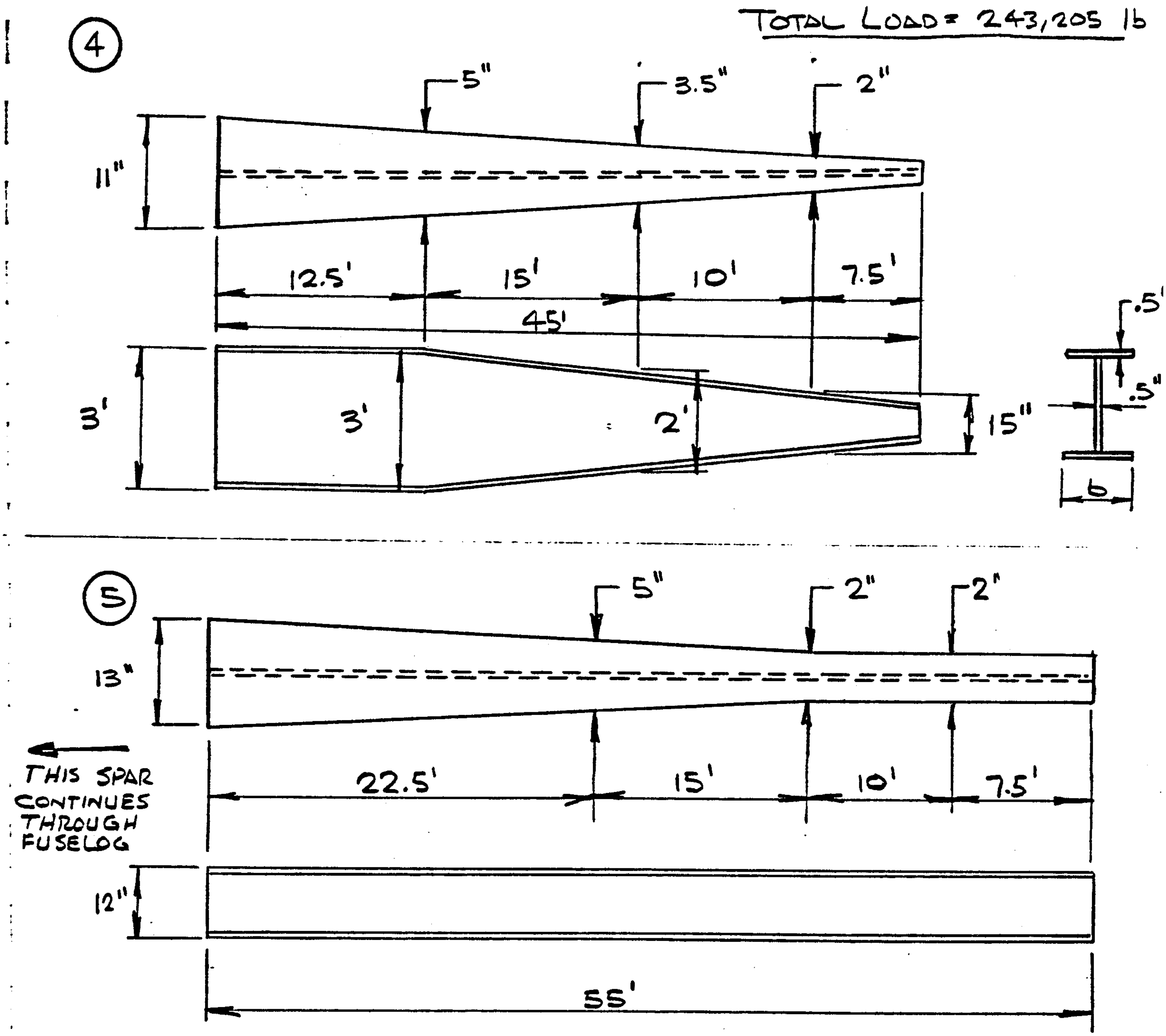


TOTAL LOAD = 229,315

3



TOTAL LOAD = 266,204 lb



TOTAL LOAD = 243,205 lb
4
5"
3.5"
2"
11"
12.5'
15'
10'
7.5'
45'
3'
3'
2'
15"
.5'
.5"
b
5
5"
2"
2"
13"
THIS SPAR CONTINUES THROUGH FUSELOG
22.5'
15'
10'
7.5'
12"
55'

Figure S4

The final step in forming the structure of the wing was to decide how to attach the spars to the fuselage. This turned out to be more than a simple matter. Most designs carry the spars the entire way through the fuselage because it is the easiest and strongest way to attach the spars. This could not be done because the fuel tank is contained in the fuselage and would not permit it. The only spar that could be contiuous through the fuselage was the aft spar. So, with the exception of the aft spar, an alternative method needed to be addressed. Once again Dr. T. Kicher was consulted and a solution was found. In order to carry the load through the fuselage so that the spars on either side would be balanced, the fuel tank would be used as a structural component.

Since the fuselage needs insulation to store the cryogenic liguid hydrogen, a thermos type of design was used. The inner shell is used to contain the liquid hydrogen, then a layer of insulating material is placed between the inside and outside shell. The outside shell will then be used as a structural member to attach the spars and carry the load through the body (see figure S5).

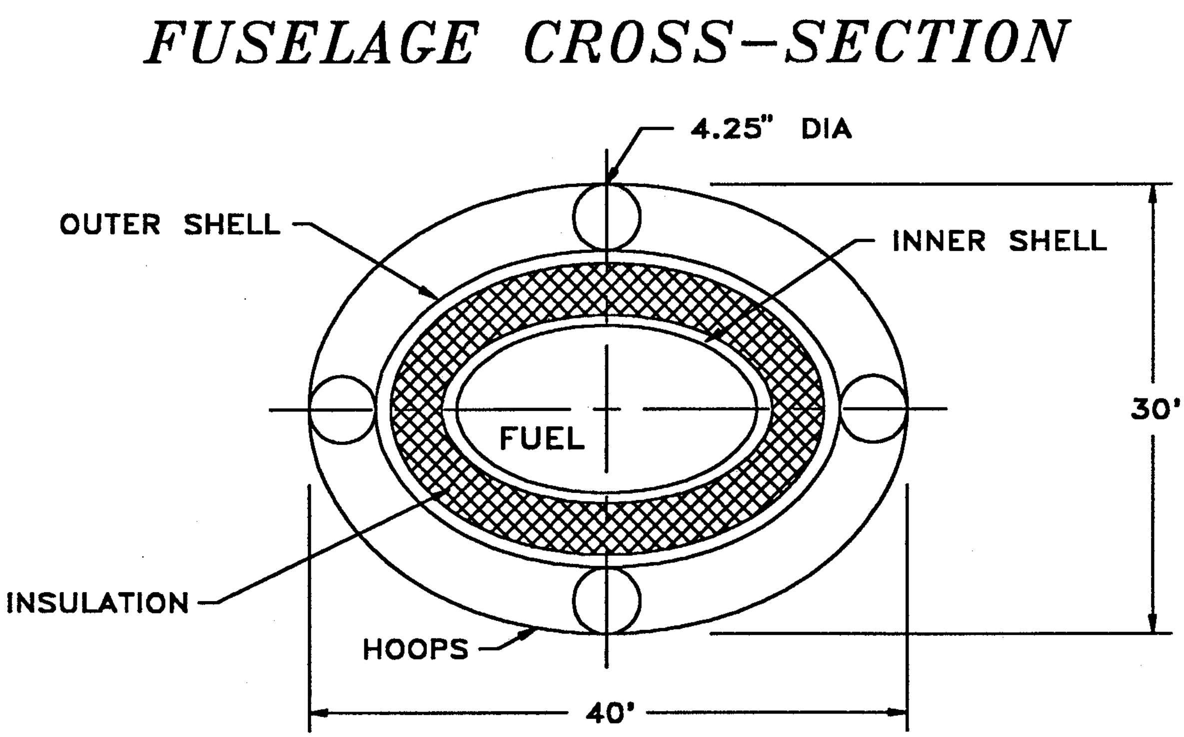


Figure S5

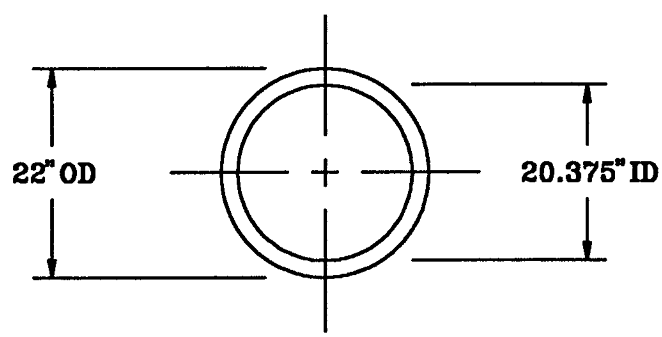
CONTROL SURFACE STRUCTURE

The first structural problem when dealing with the control surfaces came about in attaching the surfaces to the wing. This was a problem because the control surface needed to rotate approximately 30 degrees in either direction. This required that the pin withstand moment loadings from any direction. In order to accomplish this, a hollow, cylindrical pin was chosen. The pin was dimensioned as follows:

outside diameter	=	22 in
inside diameter	=	20.375 in

The pin is made of titanium; the same material used for the spars and the rest of the understructure (see figure S6).

CONTROL SURFACE HINGE PIN



Page S6

The understructure of the control surface also had to be able to take loading from a number of different directions. This led to the decision of using a tapered, hollow cylinder for the main structural member. The actual member resembles a hollow cone. The cone has the following dimensions:

outside diameter = 36" at root - 1" at tip

Material is 1/4" thick titanium

The shape of the control surface is formed by a box and rib structure similar to the wings. The hollow cone and the general layout of the control surface structure is shown in figure S7.

CONTROL SURFACE STRUCTURE

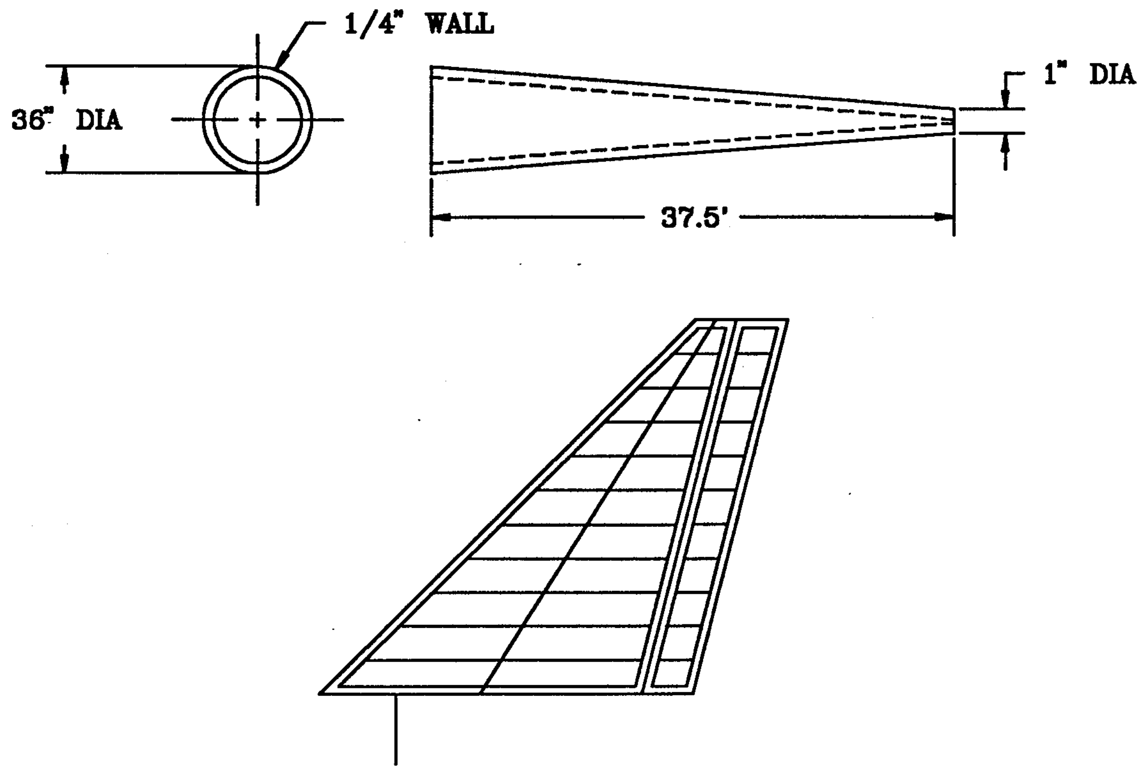


Figure S7

FUSELAGE STRUCTURE

Analysis of the fuselage proved to be too complicated to do a thorough design. An attempt to use simplifying assumptions was done so as to gain some idea of what the loads on the fuselage would be. Despite these assumptions, the only part that could be designed was the keel near the center of gravity of the entire aircraft. The main assumption was to treat the fuselage as a simple beam (see figure S8). The loads from all portions of the aircraft were simply applied to the beam at the locations at which they act. The plane was then assumed to have no pitching moment around the center of gravity. This allowed the sum of the moments

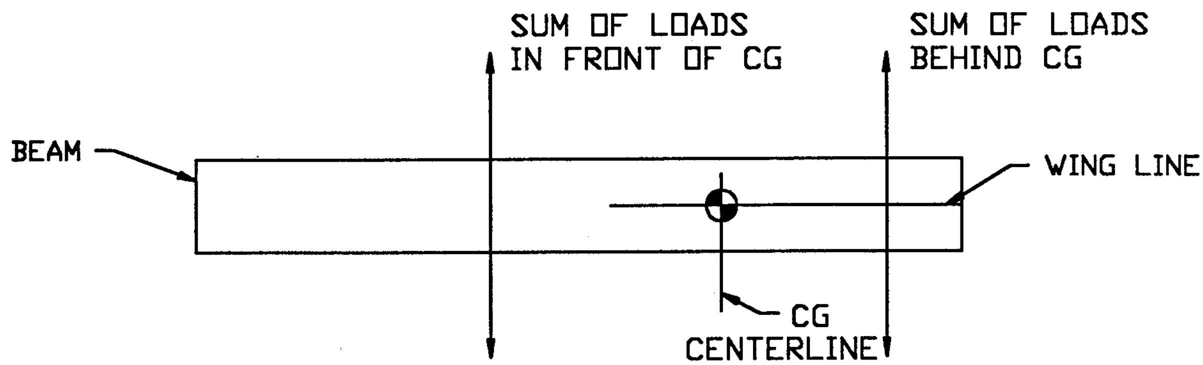


Figure S8

around this point to be analyzed so as to obtain a moment of inertia that would be used to calculate the amount of material needed at this location.

The moments that were found were about the vertical center of gravity and did not take into account the moments about the horizontal center. Initially, vertical plates were going to be used at the top and bottom of the fuselage to handle the load. The size of the material for the keel was then designed to take the entire moment at these points. It was then decided to use round stock at the top, bottom and sides of the structure to take into account the moments about the horizontal axis.

It was found that all members would be 4.25 inch diameter round stock. It is believed that the side members may not have to withstand the same loads as the top and bottom members but it was decided to use the same size to account for unknown moments.

The hoops that give shape to the body of the craft could not be readily analyzed at this time because of time constraints and the complexity of the problem. It is known, however, that the fuel tank will be used as part of the fuselage structure to give higher strength.

VERTICAL TAIL AND LANDING GEAR

For the vertical tails and landing gear no calculations were done, but a preliminary layout was conceived. The tails use the same box and rib structure as the wings and control surfaces. Figure S9 shows the tail structural layout. The main landing gear will have to be placed in the

TAIL STRUCTURE

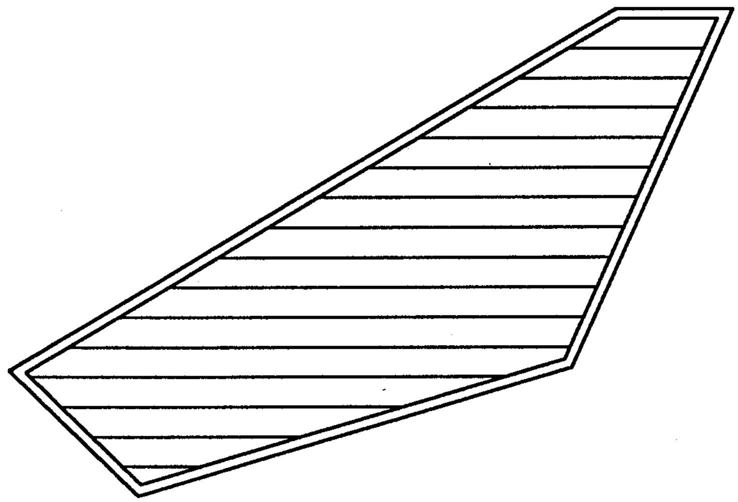


Figure S9

wings and be approximately 33 feet long to reach the ground.

CONCLUSIONS

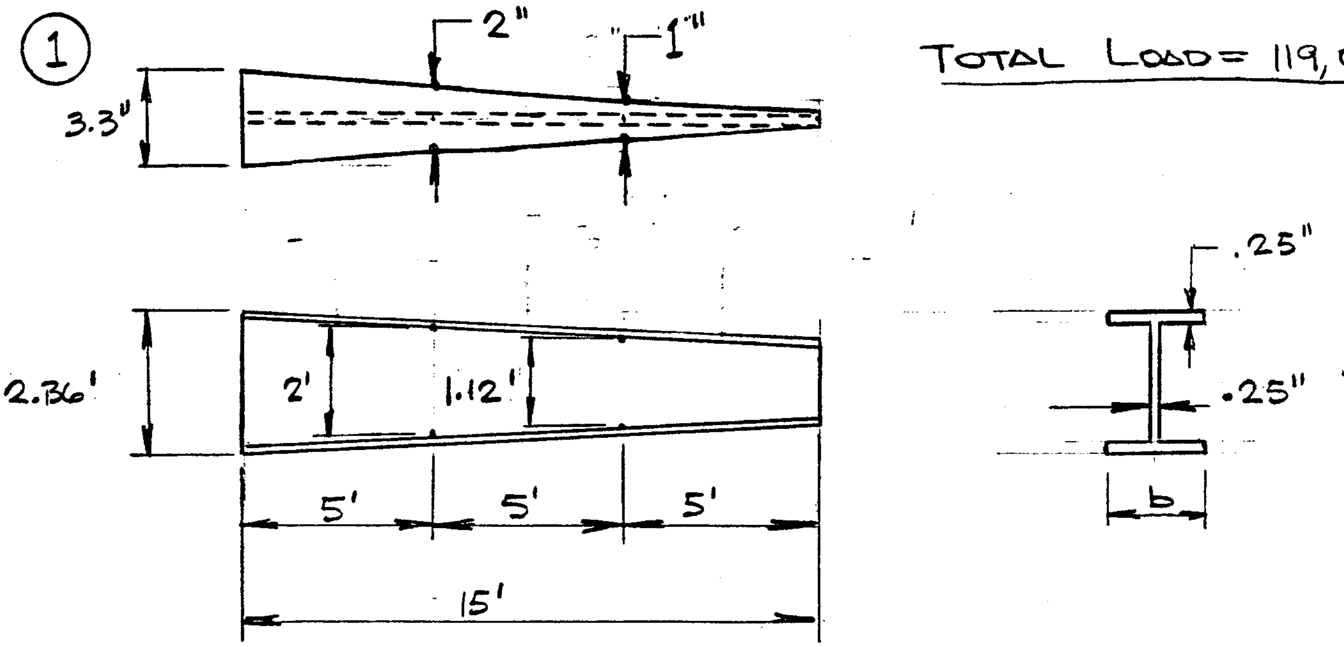
All the calculations that were done for the structural analysis for the two-stage to orbit vehicle are only first cut approximations. A first approximation is all that could be completed in the allotted time period. Due to the nature of structural analysis, it must be completed last, after the other groups complete their respective sections. This is necessary because specifications of the vehicle are needed to complete the structural analysis.

If more time were available, some of the simplifying assumptions could be dealt with in a more complete manner by using finite element analysis to estimate stresses over the vehicle. For a first cut estimation, however, the solutions found are reasonable.

STRUCTURAL APPENDIX

WING SPAR ANALYSIS

LOADS WERE FOUND AT 5 FOOT INCREMENTS BEGINNING AT 2.5 FEET FROM THE ROOT TO THE TIP. (FIGURES ARE NOT TO SCALE)



TOTAL LOAD = 119,050 lb

AT THE ROOT HEIGHT $h = 2' - 4.34'' = 28.34'$

$$\sigma = \frac{Mc}{I}$$

$$M = 53040(2.5) + 45419(7.5) + 20568(12.5) = 730{,}343 \ \text{lb-ft}$$

TITANIUM YEILD STRENGTH $\sigma = 2.3 \times 10^7$ PSF

$$\sigma = 2.3 \times 10^7 = \frac{730{,}343\,(1.181\ \text{ft})}{I} \Rightarrow I = 0.0375\ \text{ft}^4 = 777.6\ \text{in}^4$$

$I = \frac{bh^3}{12}$ b = FLANGE WIDTH h = HEIGHT

$$I = 777.6 = \frac{b(28.34'')^3}{12} - \left[\frac{(b - .25'')(28.34'' - .5'')^3}{12}\right]$$

$\underline{b = 3.3''}$ FLANGE

5' FROM ROOT $h = 2' = 24''$

$M = 267{,}807.5 \text{ lb-ft}$, $\sigma = 2.3\times10^7 = \dfrac{267{,}807.5(1')}{I}$

$I = 0.01164 \text{ FT}^4 = 241.4 \text{ IN}^4$

$I = 241.4 = \dfrac{b(24)^3}{12} - \dfrac{(b-.25)(23.5)^3}{2}$

$b = -0.4$ USE 2'' FLANGE

10' FROM ROOT $h = 1.12' = 13.44''$

$M = 51420 \text{ lb-ft}$

$\sigma = 2.3\times10^7 = \dfrac{51420(.56')}{I} \Rightarrow I = .0013 \text{ FT}^4 = 26 \text{ IN}^4$

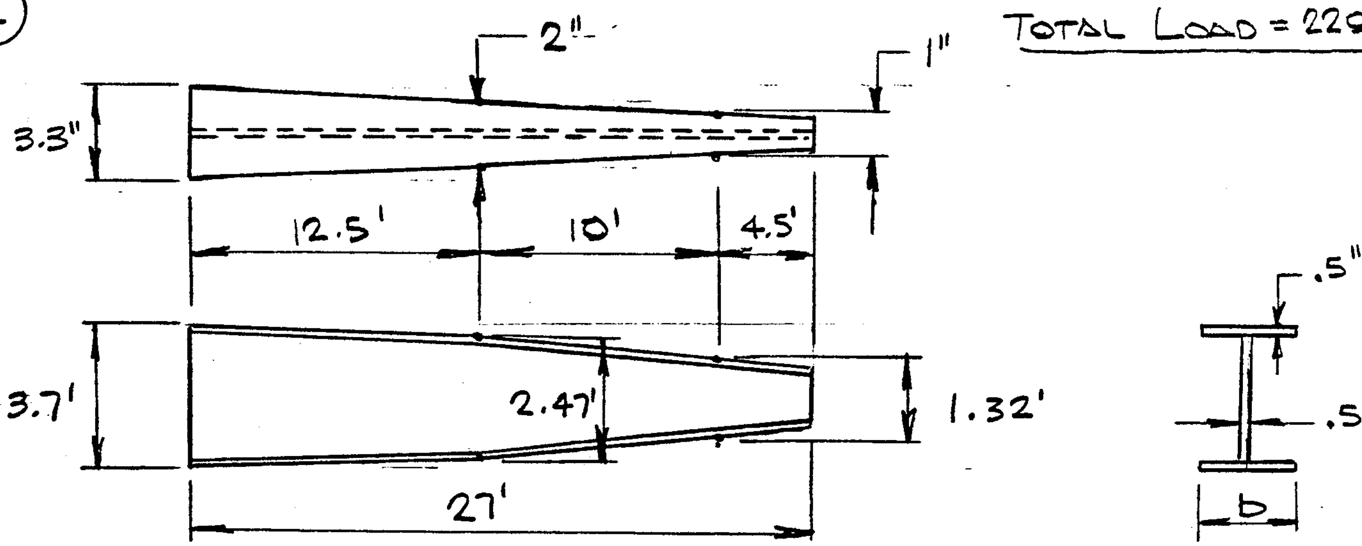
$I = 26 \text{ IN}^4 = \dfrac{b(13.4)^3}{12} - \dfrac{(b-.25)(12.94)^3}{12}$

$b = -\#$ USE 1'' FLANGE

---

(2)

TOTAL LOAD = 229,315



AT THE ROOT $h = 3.7'$

$M = 3\times10^6 \text{ lb-ft.}$, $\sigma = 2.3\times10^7 = \dfrac{3\times10^6(1.85')}{I}$

$I = .241 \text{ FT}^4 = 5000 \text{ IN}^4$

$5000 = \dfrac{b(44.4)^3}{12} - \dfrac{(b-.5)(43.1)^3}{12}$ $b = 3.3''$ FLANGE

AT 12.5' FROM ROOT   h = 2.7'

$M = 697662$ lb-ft, $\sigma = 2.3\times10^7 = \dfrac{697662(1.24')}{I}$

$I = .0375\ FT^4 = 777\ IN^4$

$777 = \dfrac{b(29.64'')^3}{12} - \dfrac{(b-.5)(28.64)^3}{12}$

b = – #   USE 2" FLANGE

AT 22.5' FROM ROOT

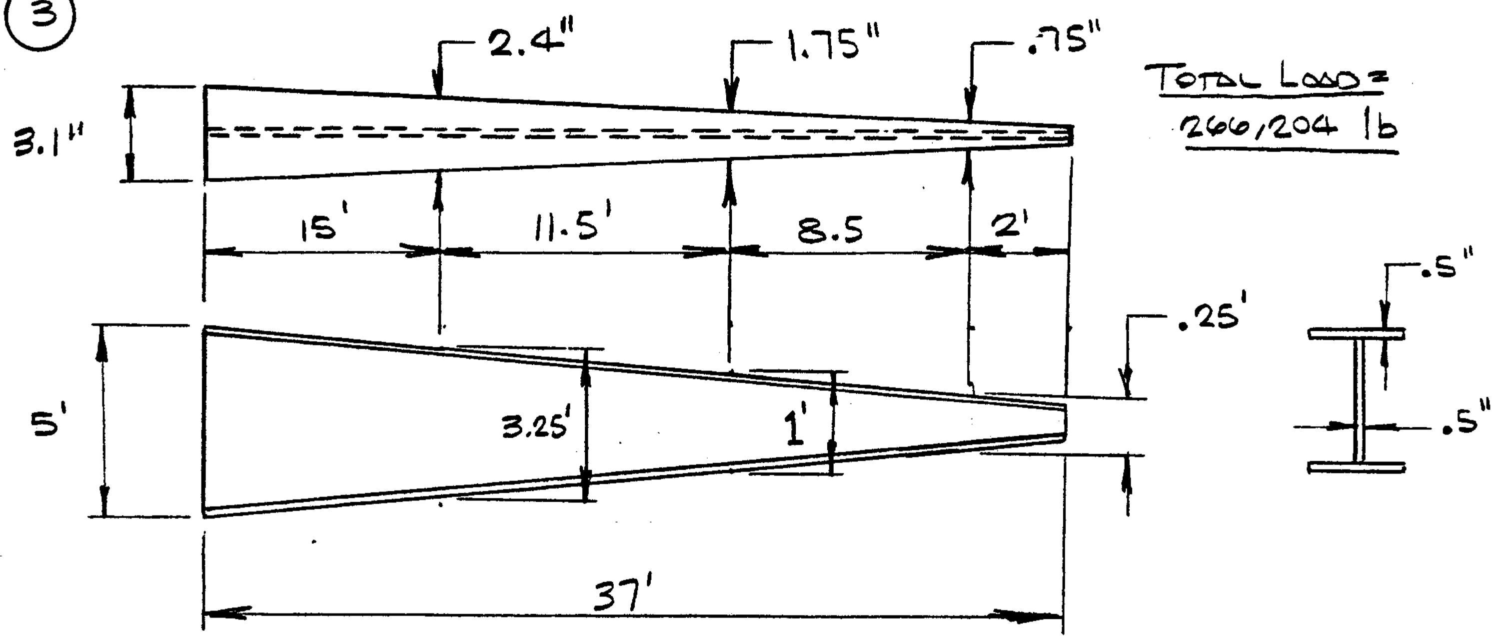
$M = 35693$ lb-ft, $\sigma = 2.3\times10^7 = \dfrac{35693(.658)}{I}$

$I = 21.2\ IN^4$

b = – #   USE 1" FLANGE

---

(3)



TOTAL LOAD = 266,204 lb

AT THE ROOT   h = 5'

$M = 5\times10^6$, $\sigma = 2.3\times10^7 = \dfrac{5\times10^6(2.5')}{I}$

$I = .544\ FT^4 = 11270\ IN^4$

$11270 = \dfrac{b(60)^3}{12} - \dfrac{(b-.5)(59)^3}{12} \Rightarrow$ b = 3.1" FLANGE

AT 15' FROM ROOT $h = 3.25'$

$$M = 2\times10^6 \text{ lb-ft} \; ; \; \sigma = 2.3\times10^7 = \frac{2\times10^6(1.625)}{I}$$

$$I = .141 \text{ ft}^4 = 2930 \text{ in}^4$$

$$2930 = \frac{b(39)^3}{12} - \frac{(b-.5)(38)^3}{12}$$

$b = 2.4''$ FLANGE

AT 26.5' FROM ROOT $h = 1'$

$$M = 450{,}000 \text{ lb-ft} \; , \; \sigma = 2.3\times10^7 = \frac{450{,}000(.5)}{I}$$

$$I = .00978 \text{ FT}^4 = 203 \text{ IN}^4$$

$$203 = \frac{b(12)^3}{12} - \frac{(b-.5)(11)^3}{12}$$

$b = 1.75''$ FLANGE

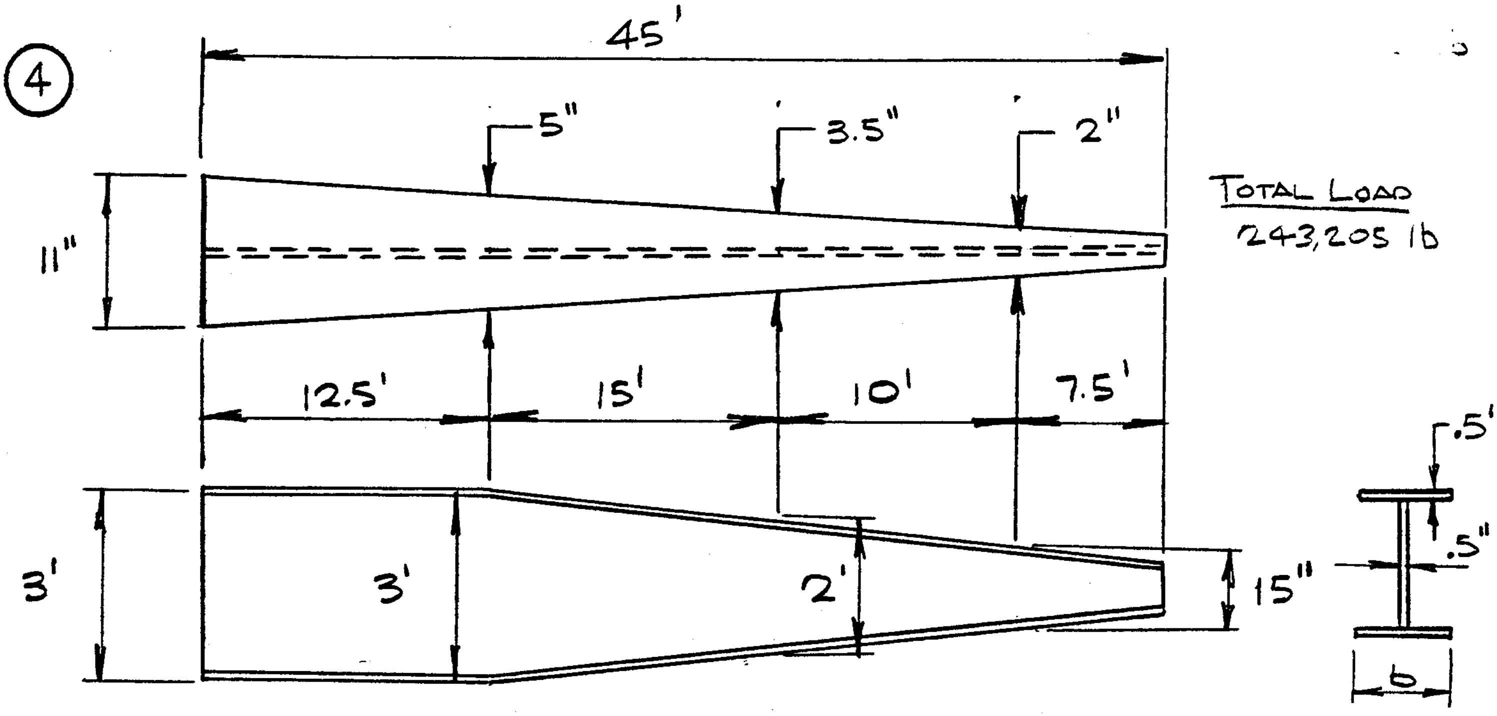
AT 35' FROM ROOT $h = .25'$

$$M = 13263 \; , \; \sigma = 2.3\times10^7 = \frac{13263(.125)}{I}$$

$$I = 1.5 \text{ IN}^4 = \frac{b(3)^3}{12} - \frac{(b-.5)(2)^3}{12}$$

$b = .75''$ FLANGE

(4)



AT THE ROOT $h = 3'$

$M = 6.22 \times 10^6$ lb-ft, $\sigma = 2.3 \times 10^7 = \frac{6.22 \times 10^6 (1.5)}{I}$

$I = .4057$ FT^4 $= 8411.6$ IN^4

$8411.6 = \frac{b(36)^3}{12} - \frac{(b-.5)(35)^3}{12}$

b = 11" FLANGE

AT 12.5' FROM ROOT $h = 3'$

$M = 3.47 \times 10^6$, $\sigma = 2.3 \times 10^7 = \frac{3.47 \times 10^6 (1.5)}{I}$

$I = .2263$ FT^4 $= 4693$ IN^4

$4693 = \frac{b(36)^3}{12} - \frac{(b-.5)(35)^3}{12}$

b = 5" FLANGE

AT 27.5' FROM ROOT $h = 2'$

$$M = 1.1\times10^6 \text{ lb-ft}, \quad \sigma = 2.3\times10^7 = \frac{1.1\times10^6(1)}{I}$$

$$I = .0476 \text{ ft}^4 = 975 \text{ in}^4$$
$$975 = \frac{b(24)^3}{12} - \frac{(b-.5)(23)^3}{12}$$

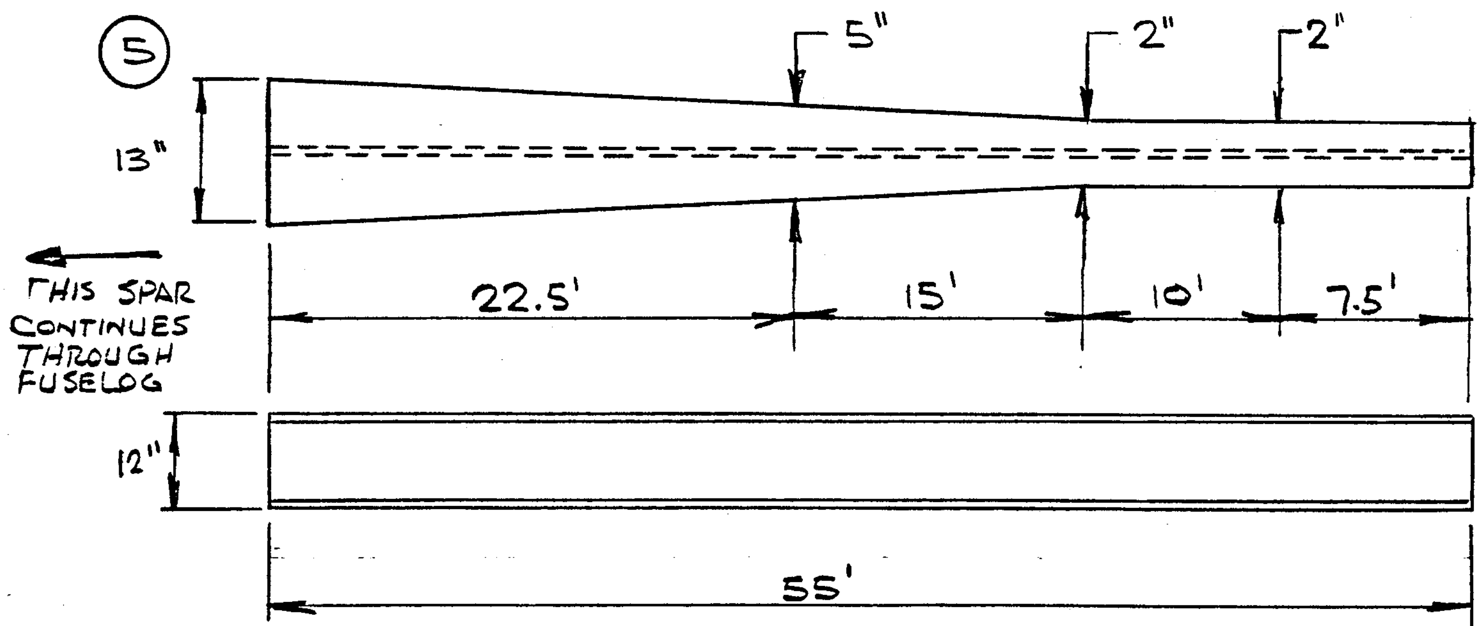
b = 3.5' FLANGE

AT 37.5' FROM ROOT $h = 15''$

$$M = 151{,}325 \text{ lb-ft}, \quad \sigma = 2.3\times10^7 = \frac{151{,}325(.625)}{I}$$

$$I = .0041 \text{ ft}^4 = 85.26 \text{ in}^4$$
$$85.26 = \frac{b(15)^3}{12} - \frac{(b-.5)(14)^3}{12}$$

b = - # USE 2" FLANGE



AT THE ROOT $h = 12''$ TOTAL LOAD = 95,885 lb

$$M = 3.04\times10^6 \text{ lb-ft}, \quad \sigma = 2.3\times10 = \frac{3.04\times10^6(.5)}{I}$$

$$I = .0658 \text{ ft}^4 = 1364 \text{ in}^4$$
$$1364 = \frac{b(12)^3}{12} - \frac{(b-1)(8)^3}{12}$$

b = 13" FLANGE

AT 22.5' FROM ROOT $h = 12''$

$M = 1.235 \times 10^6$ lb-ft , $\sigma = 2.3 \times 10^7 = \frac{1.235 \times 10^6 (.5)}{I}$

$I = .0268 \text{ ft}^4 = 556.7 \text{ IN}^4$

$556.7 = \frac{b(12)^3}{12} - \frac{(b-1)(8)^3}{12}$

$b = 5''$ FLANGE

AT 37.5' FROM ROOT $h = 12''$

$M = 4.00 \times 10^5$, $\sigma = 2.3 \times 10^7 = \frac{4 \times 10^5 (.5)}{I}$

$I = .0087 \text{ ft}^4 = 180.4 \text{ IN}^4$

$180.4 \text{ IN}^4 = \frac{b(12)^3}{12} - \frac{(b-1)(8)^3}{12}$

$b = 1.35''$ USE 2" FLANGE

AT 47.5' FROM ROOT $h = 12''$

$M = 67{,}375$ lb-ft , $\sigma = 2.3 \times 10^7 = \frac{67{,}375 (.5)}{I}$

$I = .0015 \text{ ft}^4 = 30.37 \text{ IN}^4$

$30.37 = \frac{b(12)^3}{12} - \frac{(b-1)(8)}{12}$

$b = -\#$ USE 2" FLANGE

THE FOLLOWING LOADS WERE FOUND.

SPAR # 1

@ 2.5' ⇒ $\frac{1}{2}(2)(18.25)(581.26)(5) = 53040$ lb
@ 7.5' ⇒ $\frac{1}{2}(16.5+18.125)(524.7)(5) = 45,419$ lb
@ 12.5' ⇒ $\frac{1}{2}(9+18.125)(303.3)(5) = 20,567.5$ lb

TOTAL = 119,050 lb

SPAR #2

@ 2.5 ⇒ $\frac{1}{2}(2)(18.125)(507)(5) = 45947$ lb
7.5 ⇒ $\frac{1}{2}(2)(18.125)(500)(5) = 45,312$ lb
12.5 ⇒ $\frac{1}{2}(2)(18.125)(521.7)(5) = 47,279$ lb
17.5 ⇒ $\frac{1}{2}(18.125+16.5)(586.5)(5) = 50,770$ lb
22.5 ⇒ $\frac{1}{2}(14.25+18.125)(363.8)(5) = 29,445$ lb
27.5 ⇒ $\frac{1}{2}(.75+18.125)(33.1)(5) = 1562$ lb

TOTAL = 220,315 lb

SPAR #3

@ 2.5 ⇒ $\frac{1}{2}(2)(18.125)(365)(5) = 33,078$ lb
7.5 ⇒ $\frac{1}{2}(2)(18.125)(362.7)(5) = 32,870$ lb
12.5 ⇒ $\frac{1}{2}(2)(18.125)(383.25)(5) = 34,732$ lb
17.5 ⇒ $\frac{1}{2}(2)(18.125)(419.75)(5) = 38040$ lb
22.5 ⇒ $\frac{1}{2}(2)(18.125)(430.7)(5) = 39,032$ lb
27.5 ⇒ $\frac{1}{2}(2)(18.125)(503.7)(5) = 45,648$ lb
32.5 ⇒ $\frac{1}{2}(10.5+18.125)(524)(5) = 37499$ lb
37.5 ⇒ $\frac{1}{2}(1.875+18.125)(106.1)(5) = 5305$ lb

TOTAL = 266,204 lb

SPAR # 4

@ 2.5 ⇒ $\frac{1}{2}(2)(18.125)(218)(5) = 19,756$ lb
7.5 ⇒ $\frac{1}{2}(2)(18.125)(218)(5) = 19,756$ lb
12.5 ⇒ $\frac{1}{2}(2)(18.125)(231.8)(5) = 21010$ lb
17.5 ⇒ $\frac{1}{2}(2)(18.125)(253)(5) = 22930$ lb
22.5 ⇒ $\frac{1}{2}(2)(18.125)(259)(5) = 23,525$ lb
27.5 ⇒ $\frac{1}{2}(2)(18.125)(303.6)(5) = 27,515$ lb
32.5 ⇒ $\frac{1}{2}(2)(18.125)(343)(5) = 31,080$ lb
37.5 ⇒ $\frac{1}{2}(2)(18.125)(521)(5) = 47,250$ lb
42.5 ⇒ $\frac{1}{2}(7.5+18.125)(420)(5) = 30265$ lb

TOTAL = 243,205

SPARE #5

@ 2.5 ⇒ $\frac{1}{2}(18.125+9.375)(74.6)(5) = 5{,}130$ lb

7.5 ⇒ $\frac{1}{2}(18.125+9.375)(74.6)(5) = 5{,}130$ lb

12.5 ⇒ $\frac{1}{2}(18.125+9.375)(79)(5) = 5{,}431$ lb

17.5 ⇒ $\frac{1}{2}(18.125+9.375(86.2)(5) = 5{,}930$ lb

22.5 ⇒ $\frac{1}{2}(18.125+9.375)(88.5)(5) = 6{,}085$ lb

27.5 ⇒ $\frac{1}{2}(18.125+9.375)(103)(5) = 7{,}115$ lb

32.5 ⇒ $\frac{1}{2}(18.125+9.375)(116.9)(5) = 8{,}040$ lb

37.5 ⇒ $\frac{1}{2}(18.125+9.375)(132.6)(5) = 9115$ lb

42.5 ⇒ $\frac{1}{2}(18.125+9.375)(160.3)(5) = 11{,}020$ lb

47.5 ⇒ $\frac{1}{2}(18.125+9.375)(204)(5) = 14{,}295$ lb

52.5 ⇒ $\frac{1}{2}(120)(269.5)(5) = 13{,}475$ lb

TOTAL = 95,885 lb

CONTROL SURFACE

ANALYSIS

A CYLINDRICAL SPAR WAS CHOSEN TO WITHSTAND LOADS IN ALL DIRECTIONS.

$$\sigma = \frac{Mc}{I} \quad , \quad I = \frac{\pi}{4}(r_o^4 - r_i^4) \qquad \sigma = 160 \times 10^3 \text{ PSI}$$

$$M = 200{,}000 \times 15' = 3 \times 10^6 \text{ lb-ft} \qquad \text{O.D.} = 36''$$

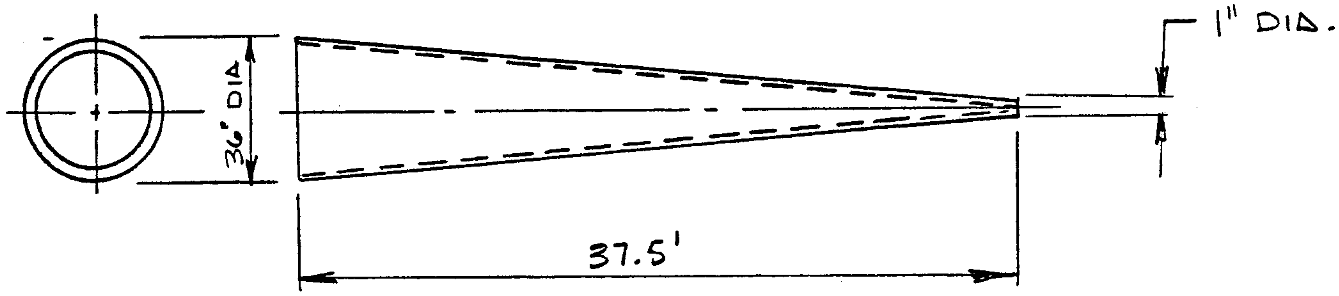
$$= 36 \times 10^6 \text{ lb-IN}$$

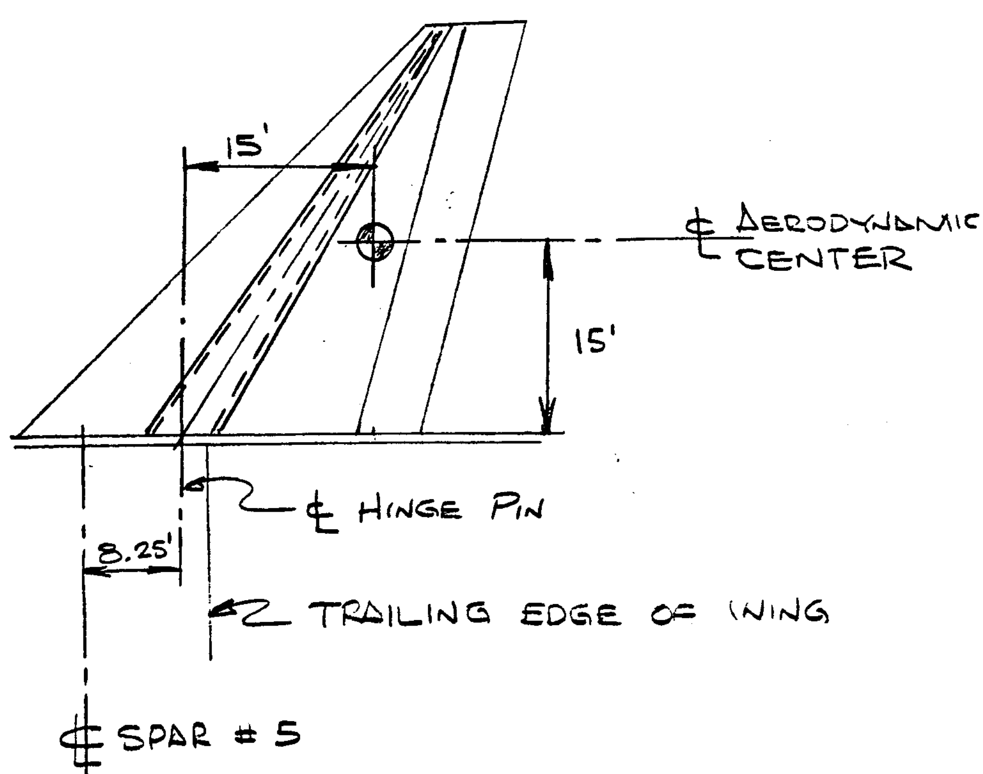
$$160 \times 10^3 = \frac{36 \times 10^6 (18)}{I}$$

$$I = 4050 \text{ IN}^4$$

$$4050 = \frac{\pi}{4}(18^4 - r_i^4) \Rightarrow r_i = 17.8'', \ \text{I.D.} = 35.6$$

USE 35.5" I.D. $\Rightarrow$ $\frac{1}{4}$" THICK WALL





CONTROL SURFACE
HINGE PIN ANALYSIS

$\sigma = \dfrac{Mc}{I}$

$M = 250{,}000 \times 15 \times 12$

$= 4.5\times10^7$ lb·in

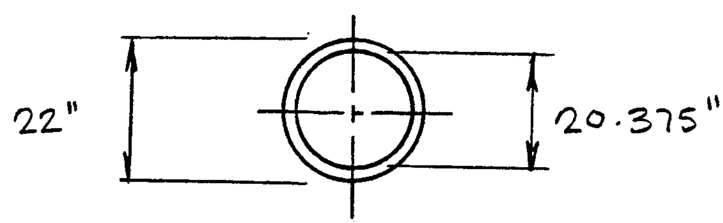
$I = \dfrac{\pi}{4}\left(r_o^4 - r_i^4\right)$

O.D. = 22"

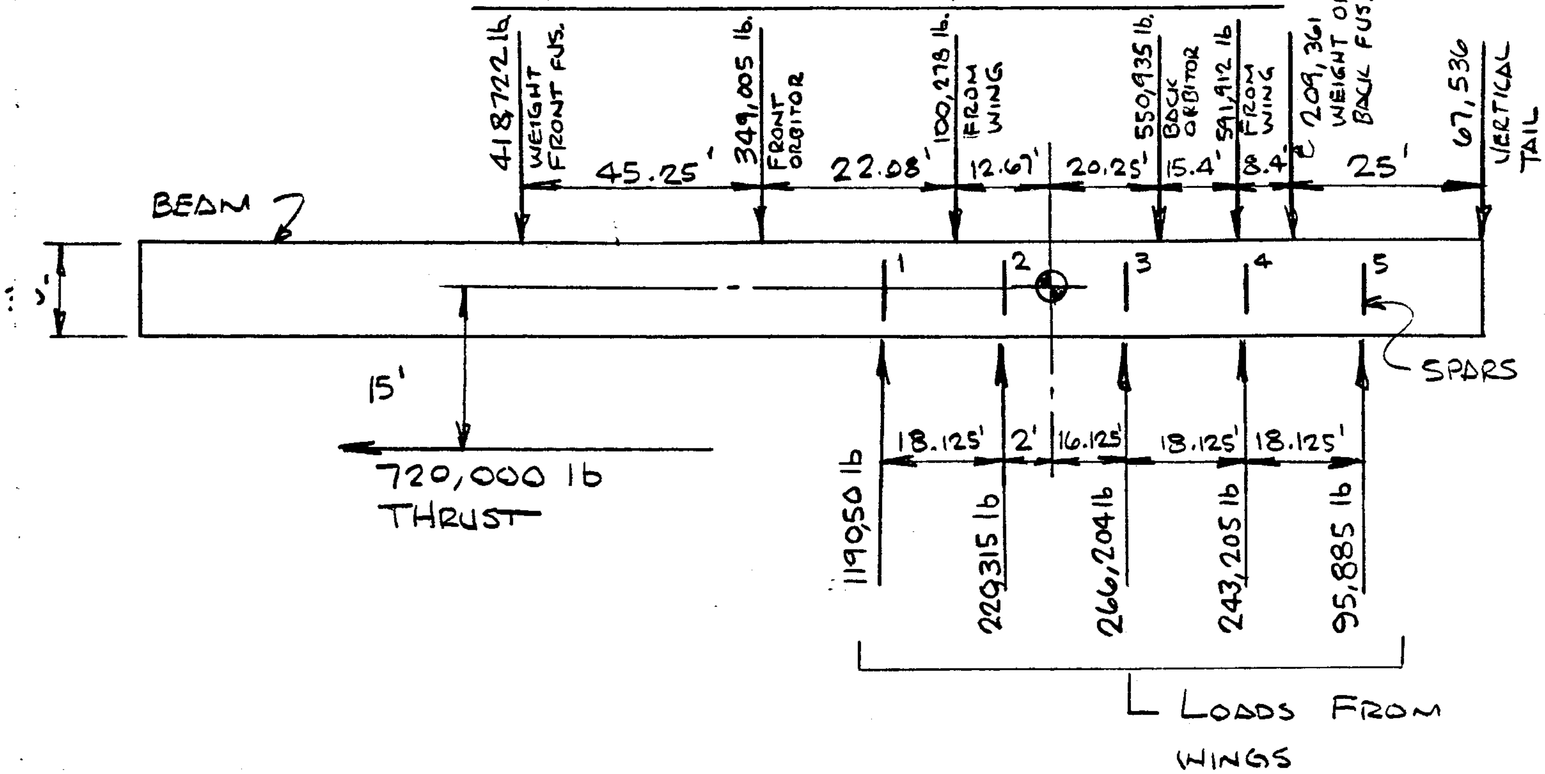
$= 160\times10^3$ psi $= \dfrac{4.5\times10^7\,(11)}{I} \Rightarrow I = 3094$ in^4

$3094 = \dfrac{\pi\left(11^4 - r_i^4\right)}{4}$

$r_i = 10.1875" \Rightarrow$ I.D. = 20.375"



FUSELAGE ANALYSIS



THE MOMENT THAT THE FUSELAGE CENTER OF GRAVITY HAS TO WITHSTAND EQUALS:

$$M = 95{,}885(52) + 243{,}205(34) + 266{,}204(16) + 349{,}005(34.75) + 100{,}278(12.67) + 418{,}722(80)$$

$$= \underline{6.1275 \times 10^7 \text{ lb-ft}}$$

$$\sigma = 2.3 \times 10^7 \text{ PSF} = \frac{6.1275 \times 10^7 (15)}{I} \Rightarrow I = 39.96 \text{ FT}^4 = 828611 \text{ IN}^4$$

$$I = \Sigma(I_x + Ad^2)$$

2.125 RADIUS IS TRIED

$$828611 < 2\left(\tfrac{1}{4}\pi(2.125)^4 + \pi(2.125)^2(177.9)^2\right)$$

$$828611 < 897{,}868 \text{ IN}^4$$

2" RADIUS WAS FOUND TO BE TOO SMALL.

2.125 RAD OR <u>4.25" DIA ROUND IS USED</u>

CONCEPTUAL DESIGN OF

TWO-STAGE-TO-ORBIT

HYBRID LAUNCH VEHICLE

ORBITER DESIGN

July 1, 1991

Gary P. Garbinski
Project Manager

AERODYNAMIC GROUP

Cynthia Sparks
Timothy Mooney
Gary Garbinski

AERODYNAMICS

INTRODUCTION

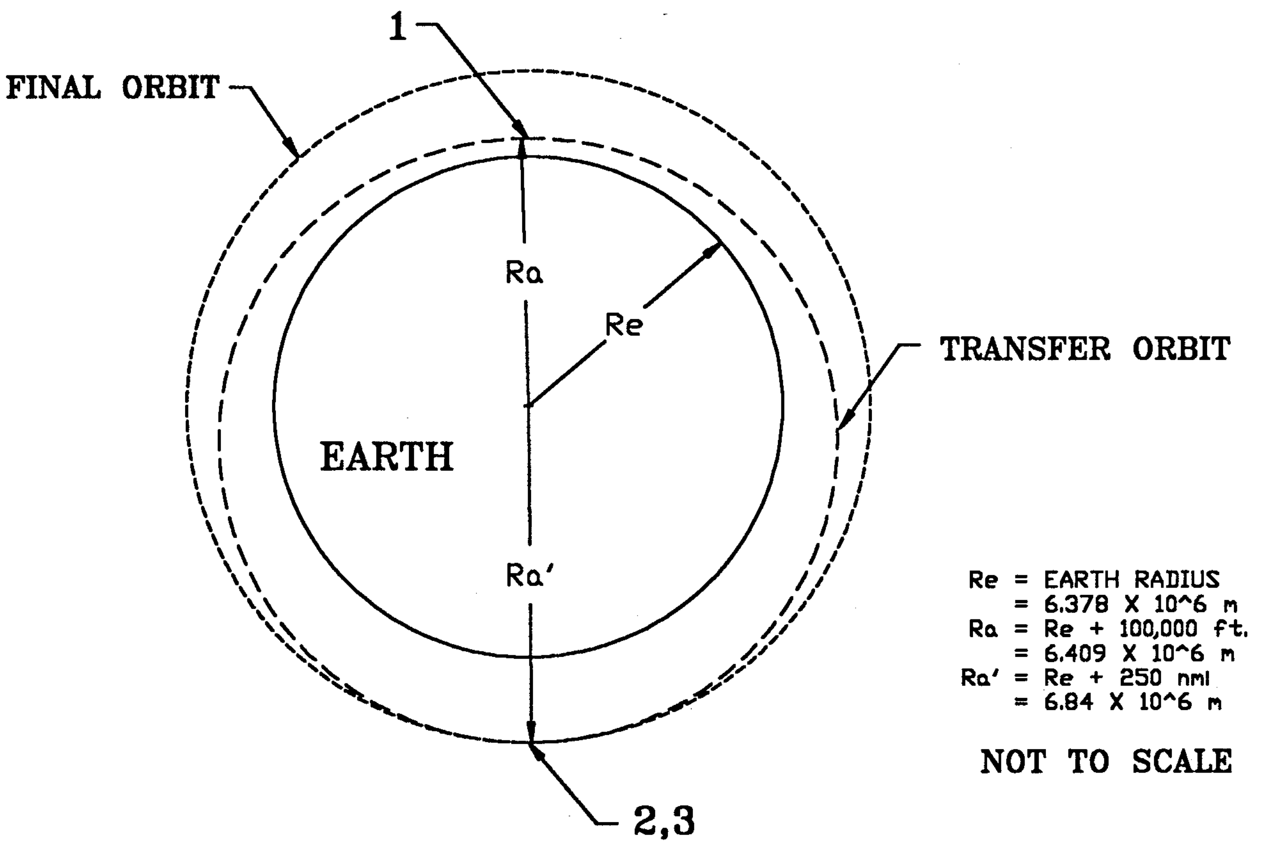
The Orbiter Aerodynamics Group was responsible for a number of areas. These responsibilities included the following: determination of the orbiter trajectory for ascent, which was required in order to know the total necessary velocity impulse; initial weight estimates for the major areas of the orbiter, which included the fuel, structure, and payload; determination of the re-entry trajectory, which was divided into two parts, above 300,000 feet (aerodynamic forces are small relative to gravitational forces) and below 300,000 feet (aerodynamic forces); aerodynamic heating for the nose and the wing leading edge; sizing of the wing, tail, and control surfaces; and a stability analysis.

ORBITER TRAJECTORY:

The first necessary task for the orbiter aerodynamics group was to determine a suitable trajectory, after separation from the booster, which would place the orbiter into its circular rendezvous orbit. Since the fuel weight is a large percentage of the total weight of the orbiter, a fuel efficient vehicle is desired. In order to achieve a fuel efficient trajectory, elliptical transfer orbits were utilized.

During the entire trajectory, three major velocity impulses will be necessary. Other small impulses will be required from the thrusters for smaller manuevers. This would include a rendezvous with the Space Station. A drawing of the total trajectory can be found on page A2. The first major velocity impulse will be fired soon after separation from the booster. At the release point, the orbiter will be at 100,000 feet traveling at Mach 6. At the maximum altitude

ORBITER TRAJECTORIES



of the booster's trajectory, the orbiter will be released, and the booster will quickly dive down out of the orbiter's way so that the orbiter can fire its engines as soon as possible. See page A4 for a conceptual drawing of this manuever. The sooner the orbiter can fire its engines, the less velocity it will lose. In order to achieve the most efficient transfer, the velocity impulse at the separation point will place the orbiter into an elliptical orbit, which will have the 100,000 foot altitude as its perigee. Due to the thrust that the orbiter will have at this point, aerodynamic forces will be ignored.

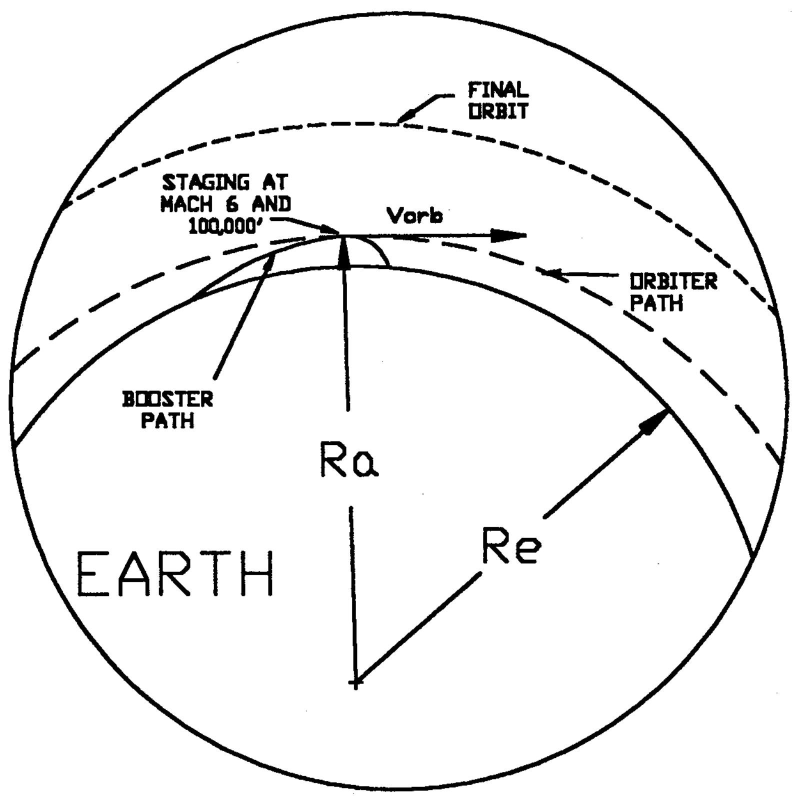
Once again, for the most efficient orbit transfer, the second major velocity impulse will occur at the apogee of the elliptical transfer orbit. The apogee will be located at 250 nautical miles (1,500,000 feet). The velocity impulse will place the orbiter into a circular orbit at this altitude. The velocity in the orbit will be 7.64 km/sec (approximately Mach 23.4). The timing of the mission must be such that the orbiter and Space Station rendezvous at approximately this transfer point. In order to allow for a more flexible launch time, the orbiter may be placed in a circular holding orbit. Although doing so will increase the necessary amount of fuel.

A third velocity impulse will be required to place the orbiter back into the same elliptical transfer orbit so that it may deorbit. At about 300,000 feet, the orbiter will be far enough into the atmosphere so that aerodynamic forces will become dominant.

Impulse Number	Velocity Impulse (km/sec)	Velocity Impulse (kft/sec)
1	6.1703	20.245
2	1.2571	4.1245
3	1.2560	4.1209
TOTALS:	6.4216	28.4904

The orbiter aerodynamics group performed these initial calculations using a specific impulse of 460 seconds (hydrogen), which resulted in an effective exhaust velocity of 4.5126 km/sec (14.806 kft/sec). A detailed copy of the calculations is included in Appendix A1.

BOOSTER-ORBITER SEPARATION DETAIL



SCALE IS EXAGERATED TO SHOW DETAIL

ORBITER WEIGHT ESTIMATES:

The first estimate made for the orbiter weights was based on an assumption that the structure weight would be three times the payload weight. This assumption was made based on the fact that the Shuttle's structure weight is about 2.3 times its payload. Since the Shuttle's weight is greater than that of this orbiter, the estimate that was made was higher than 2.3 since certain portions of the structure weight cannot be scaled down directly. Upper to lower limits of 20,000 pounds and 12,000 pounds for the payload were decided upon by the entire aerospace design group.

m_{pay}		m_{struc}		m_{prop}		m_{tot}	
lb	kg	lb	kg	lb	kg	lb	kg
20000	9072	60000	27216	252320	114430	332330	150720
12000	5443	36000	16330	151290	68600	199300	90390

These first weight estimates were not used based on the fact that the assumption that the structure weight was three times the payload weight was not well enough supported.

The next estimate was made based on information presented in the Airbreathing Transatmospheric Vehicle Concept Studies on the Project Beta. Project Beta has an orbiter structure weight that is 1.74 times its payload (50,000 pounds). Since the maximum for this orbiter's payload weight is to be 20,000 pounds, the ratio of structure to payload weight must be greater than that of Project Beta, as explained earlier. Through consultation with Professor Reshotko, an estimate for the structure weight of 2.25 times the payload was made.

m_{pay}	m_{struc}	m_{prop}	$m_{fuel\ res}$	m_{tot}
lb/kg	lb/kg	lb/kg	lb/kg	lb/kg
20000/ 9072	45000/ 20410	204730/ 55715	20000/ 9072	289000/ 131070
12000/ 5443	27000/ 12245	122850/ 55715	20000/ 9072	181850/ 82470

After this assumption was decided upon, it was necessary to recognize a number of structure factors that could possibly vary. The total weight of the orbiter is composed of the following sub-weights:

$$W_o = W_{FUEL} + W_{LAND\ GEAR} + W_{STRUC} + W_{SYS} + W_{PAY} + W_{PROP}$$

Based on the required total velocity impulse, a fuel weight to total weight ratio of 0.7 is required. See Appendix A2 for any weight assumptions and calculations. Three structural factors were varied to find an acceptable combination. The three factors were structure fraction (ratio of weight of actual structure to total weight), propulsion weight (includes propulsion system and fuel reserve), and payload weight.

Structure Ratio

Assume, W_{PROP} = 30,000 pounds

W_{PAY} = 20,000 pounds

W_s/W_o	0.05	0.10 (Project Beta)	0.20	0.30	0.40 (Shuttle)
W_o (klb)	280	355	767	< 0	< 0

Propulsion

Assume, $W_s/W_0 = 0.105$ (Project Beta)

$W_{PAY} = 20{,}000$ pounds

W_{PROP}(klb)	8	15	22	29	36
W_o (klb)	39	278	12	350	389

Payload

Assume, $W_s/W_0 = 0.105$ (Project Beta)

$W_{PROP} = 30{,}000$ pounds

W_{pay}(klb)	2	14	16	18	20
W_o (klb)	314	325	335	346	356

It was decided to go with the future technology that was used by Project Beta for structure ratio ($W_s/W_0 = 0.105$) . Reductions will be made to both the payload weight and the fuel reserve weight in order to achieve the total goal weight of 300,000 pounds. This requires a combined reduction of 12,000 pounds from the maximum payload of 20,000 pounds and maximum fuel reserve of 20,000 pounds.

RE-ENTRY TRAJECTORY:

The re-entry trajectory from 250 nautical miles (1.5 million feet) is calculated in two parts. The first section of the re-entry is from orbital altitude to 300,000 feet. At these high altitudes, the atmosphere is so thin that there are negligible aerodynamic effects. In the second section, below 300,000 feet, aerodynamic forces are no longer negligible. Above 300,000 feet, the re-entry trajectory is calculated using equations of motion for a two-body problem. The mass of the earth is much greater than that of the orbiter, therefore, the mass of the orbiter is neglected. A vehicle's motion about the earth is determined by the vehicle's initial conditions and the gravitational attraction of the earth. The equations of motion are the following:

$$r'' = -u/r^2 + h^2/r^3$$

$$\text{theta}'' = h/r^2$$

where $-u/r^2$ is a potential force, h^2/r^3 is a kinetic force, and h/r^2 describes the angular momentum.

These equations are integrated to determine r, r'(v), and theta. The r" equation is integrated twice. The integration is done by using the predictor/corrector method. The predictor values are calculated by marching through time and substituting in values calculated in the previous step. These values are then used to calculate corrector values and an average is taken of the predictor and corrector values. The initial conditions are determined from an initial altitude and velocity. The computer program that calculates these values can be seen in Appendix A3.

For the 1.5 million foot to 300,000 foot deorbit, the orbiter will be put into an elliptical trajectory. The following plots of this deorbit maneuver can be seen in Figures A1, A2, and A3.

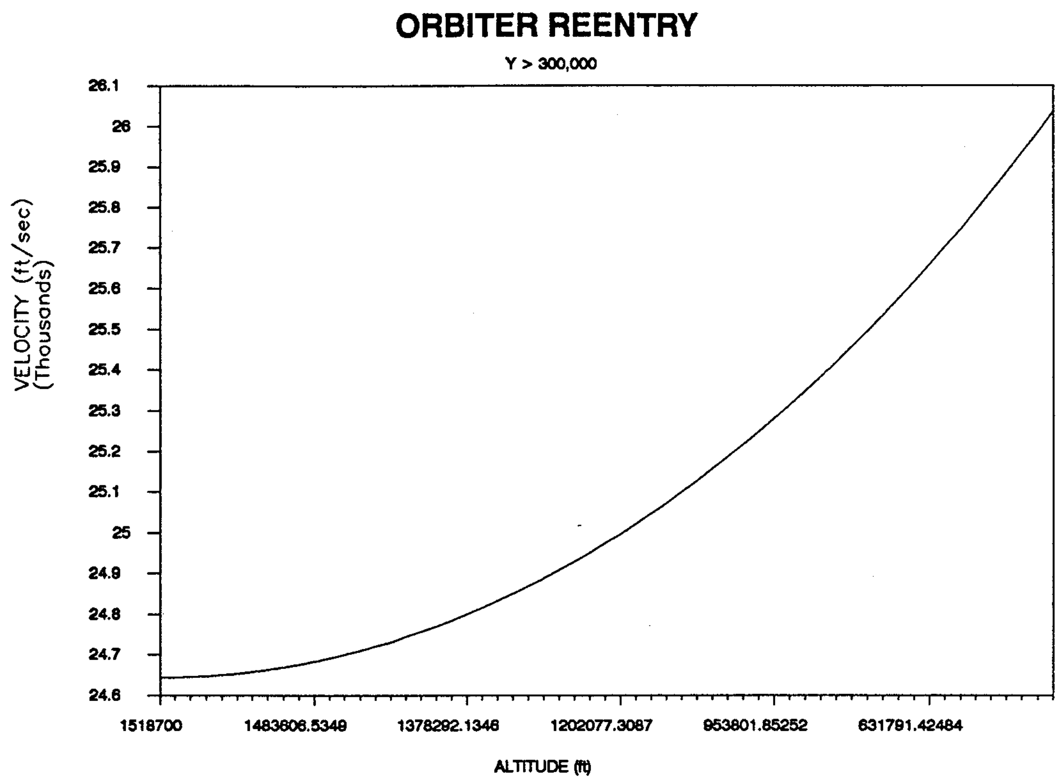


Figure A1

As mentioned previously, below 300,000 feet the re-entry is determined by aerodynamic forces. The equations of motion are the following:

$$m*v' = -D \quad W*\sin(\theta)$$

$$= -1/2*C_D*rho*v^2*A + W*\sin(\theta)$$

$$m*v*\theta' = -L + W*\cos(\theta)$$

$$= -1/2*C_L*rho*v^2*A + W*\cos(\theta)$$

these equations are integrated, again using the predictor/corrector method, to determine velocity (v) and heading angle (theta). The computer program that was written to perform these calculations can be found in Appendix A4. Velocity is again integrated and multiplied by the

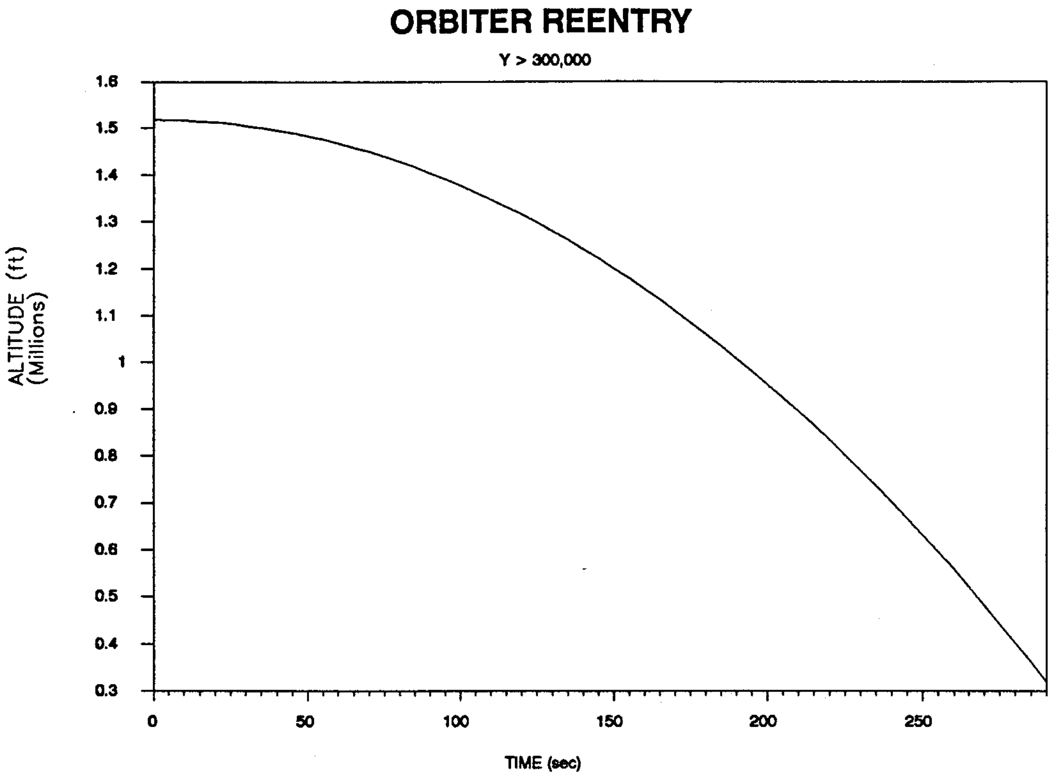


Figure A2

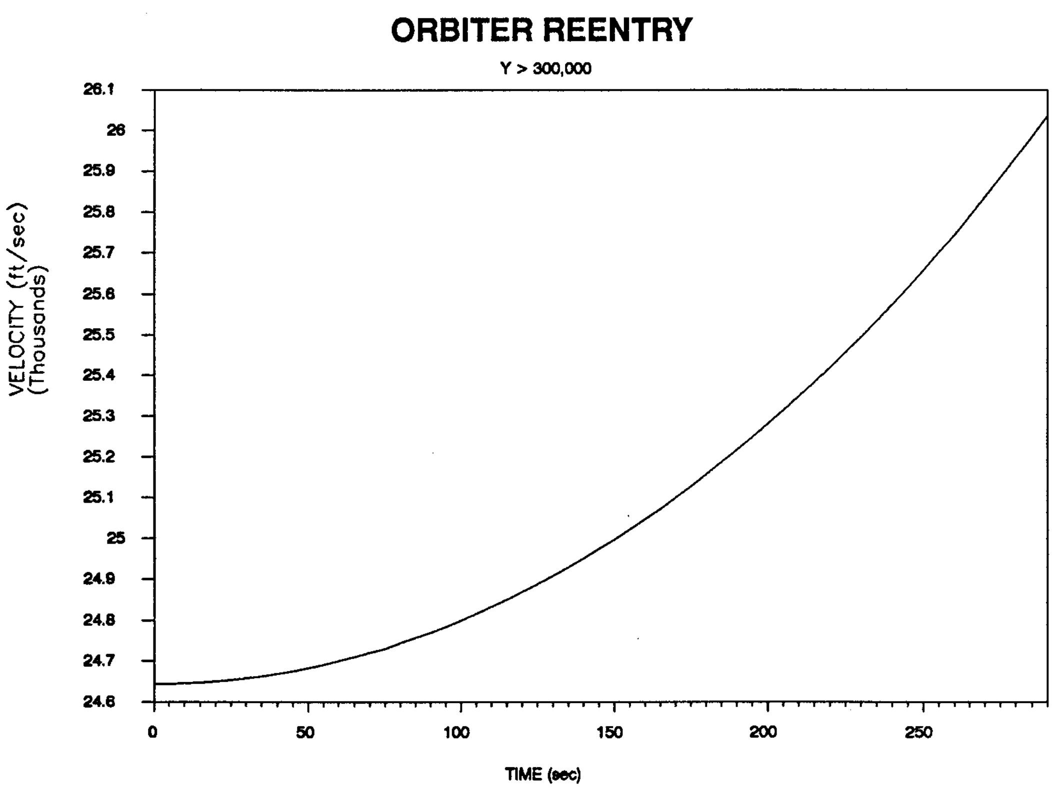


Figure A3

cosine and sine of theta to determine the positions x and y, respectively. The initial x and y positions in this part of the re-entry are defined to be (0, 300000). A check is included in the program which will not allow the deceleration to go above 3g's so that the astronauts will not lose consciousness.

This integration procedure is called from the main program, and a value for the angle of attack (a) is sent to the procedure. The method for determining the angle of attack will be explained later. After calculating the velocity and altitude at a time step, these values along with the angle of attack are used to determine the appropriate lift coefficient. The function for achieving this uses the velocity (converted to Mach number) and altitude to determine which of three methods should be used to determine the lift coefficient.

For Mach numbers less than 1.2, the subsonic lift-curve slope equation, found in Raymer's aircraft design book (see References), was used to determine the lift coefficient for the specified angle of attack:

$$C_{L\alpha} = \frac{(2*pi*AR*(S_{exp}/S_{ref})*F)}{(2+(4+(AR^2*B^2/n*(1+(\tan^2(lambda_{maxt})/B^2))))^{1/2})}$$

where

$$B^2 = 1 - M^2$$

$$n = C_{l\alpha}/(2*pi/B)$$

$$F = 1.07(1 + d/b)^2$$

Although this equation is only accurate up to Mach 1, it was used to estimate the lift coefficients up to Mach 1.2. Therefore, these lift coefficients, which affect drag coefficients as well, between Mach 1 and Mach 1.2 are only estimates.

Above Mach 1.2, but below 100,000 feet, the subsonic lift-curve slope equation, also found in Raymer's book, was used to determine the required lift coefficient from the given angle of attack:

$$C_{L\alpha} = 4/B$$

$$B = (M^2 - 1)^{1/2}$$

for M > 1/cos($lambda_{LE}$). Based on the previous constraint, this equation would be accurate down to Mach 1.4. This equation was also used to estimate the lift coefficients from Mach 1.4 to Mach 1.2.

Above 100,000 feet, Newtonian flow was used to calculate the lift coefficient (as well as the drag coefficient as will be explained later) as a function of angle of attack. The pressure coefficient on the upper side of the wing (shadow region) is zero, and on the lower side it is given by

$$C_{pl} = 2*\sin^2(a)$$

The normal force coefficient on the wing is given by

$$C_N = (C_{pl} - C_{pu})dx$$

for a wing that is composed of the same airfoil, such as that of the orbiter, C_N reduces to

$$C_N = 2*\sin^2(a)$$

Since lift is the vertical component of the normal force, the lift (and drag) components can be determined from C_N and a:

$$C_L = C_N \cos(a) = 2*\sin^2(a)\cos(a)$$

Once the lift coefficient is determined, it can be used, along with velocity, altitude, and angle of attack to determine the drag coefficient. The drag coefficient function again uses Mach number and altitude to determine which of four methods will be used to calculate the coefficient.

The first three methods calculate, for various Mach number ranges, the parasite drag coefficient and drag coefficient due to lift, which can then be summed. These equations were found in a paper on sizing hypersonic vehicles (see References).

Parasite Drag:

$0.8 > M$	$C_{Dmin} = 0.011$
$0.8 <= M <= 1.2$	$C_{Dmin} = -0.0510 + 0.0762M$
$1.2 < M <= 6.0$	$C_{Dmin} = 0.0605 - 0.0177M + 0.00163M^2$

Drag Due to Lift: (for all Mach numbers)

$$C_{DL} = (0.1378 + 0.1693M - 0.01155M^2)C^2_L$$

$$C_D = C_{Dmin} + C_{DL}$$

The fourth method would be used for altitudes higher than 100,000 feet. The method is derived from Newtonian flow theory, as explained earlier:

$$C_D = C_N \sin(a) = 2*\sin^3(a)$$

Plots of C_L and C_D vs angle of attack can be seen in Figures A4 and A5.

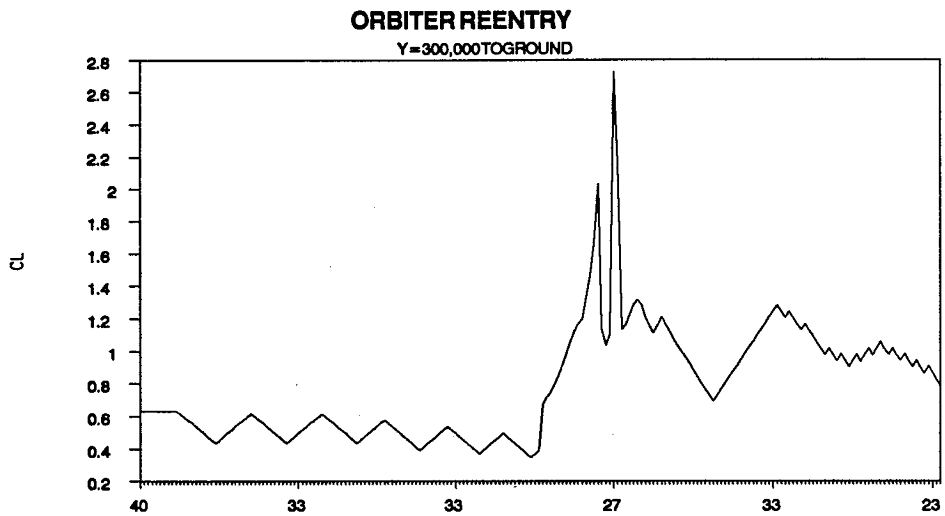


Figure A4

These newly calculated lift and drag coefficients, along with the newly calculated values of velocity, altitude, and heading angle, are used to determine v' and theta' for the next time step.

The next task that is performed by the integrating procedure is to calculate heating information for both the nose and the leading edges. A detailed explanation of the heating calculations is given in the next section. Basically, for each altitude and corresponding velocity, the heating rate and wall temperature for both the nose and leading edge are calculated. Within the main program a running sum is kept of the heat loads:

$$qload(i) = qload(tot) + q'*dt$$

The orbiter re-entry trajectory attempts to keep the temperature and load below certain values, which will be given in the next section along with the results of the calculations.

The main program attempts to maintain a particular re-entry path by specifying certain conditions. This re-entry path, and therefore the conditions, are based on the Shuttle re-entry. The Shuttle was used as a guide so that it would be known if the path was reasonable. Although

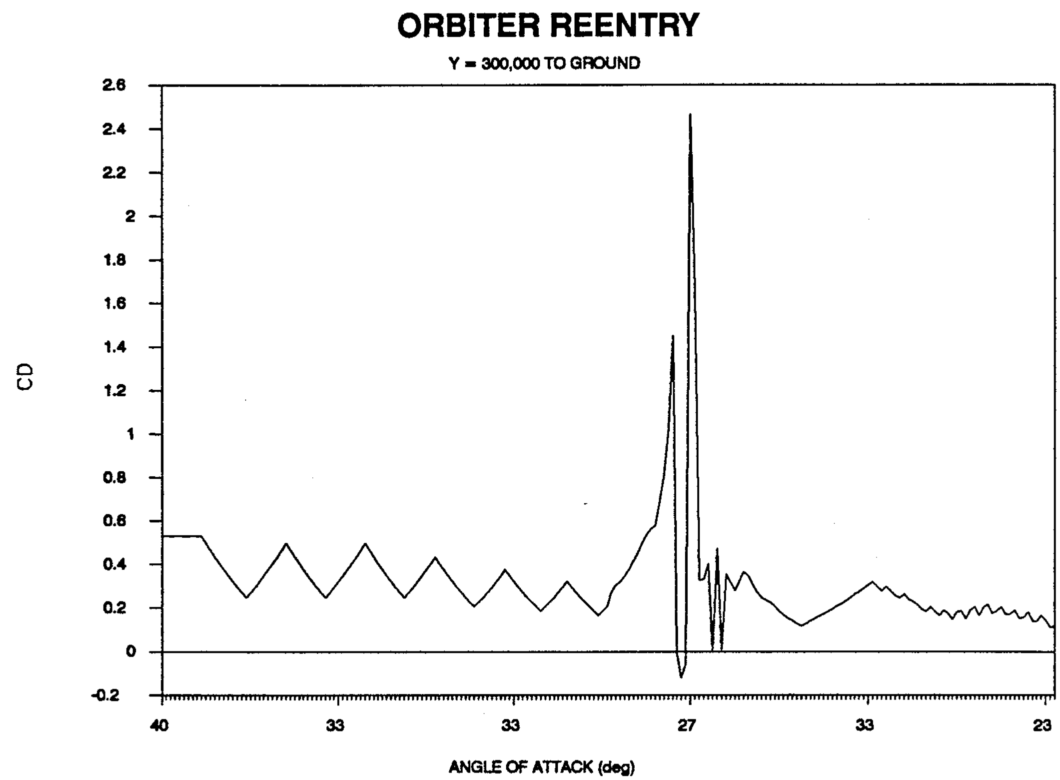


Figure A5

the orbiter tries to maintain this reasonable path, it exceeds what the Shuttle does as far as maximum temperatures encountered. This would require future developments in the area of materials.

The conditions that the orbiter must meet in its re-entry path determine the angle of attack. The initial angle of attack is assigned to be 40 degrees (based on Shuttle data). The conditions are the following:

altitude > 30,000 ft	altitude <= 30,000 ft
$v_y <= 200$ ft/sec	$v_y <= 300$ ft/sec
a > 0 deg: a = a-1	a > 0 deg: a = a-1
a = 0 deg: a = a	a = 0 deg: a = a

$v_y > 200$ ft/sec

$a < 40$ deg: $a = a+1$

$a = 40$ deg: $a = a$

$v_y > 300$ ft/sec

$a < 40$ deg: $a = a+1$

$a = 40$ deg: $a = a$

Upon meeting the proper condition, the appropriate angle of attack is determined and the integration procedure is called to determine the flight conditions at that time step. This process is reiterated until the altitude is no longer greater than zero. The re-entry path can be seen in the following figures: A6, A7, A8. Overall the orbiter's entry path is similar to that of the Shuttle's. As can be seen from the data, there is a great deal of oscillation occurring. These oscillations can be damped out with a controller that is more complex than what there was time to develop presently. This is a project that can be worked on further during the coming semester .

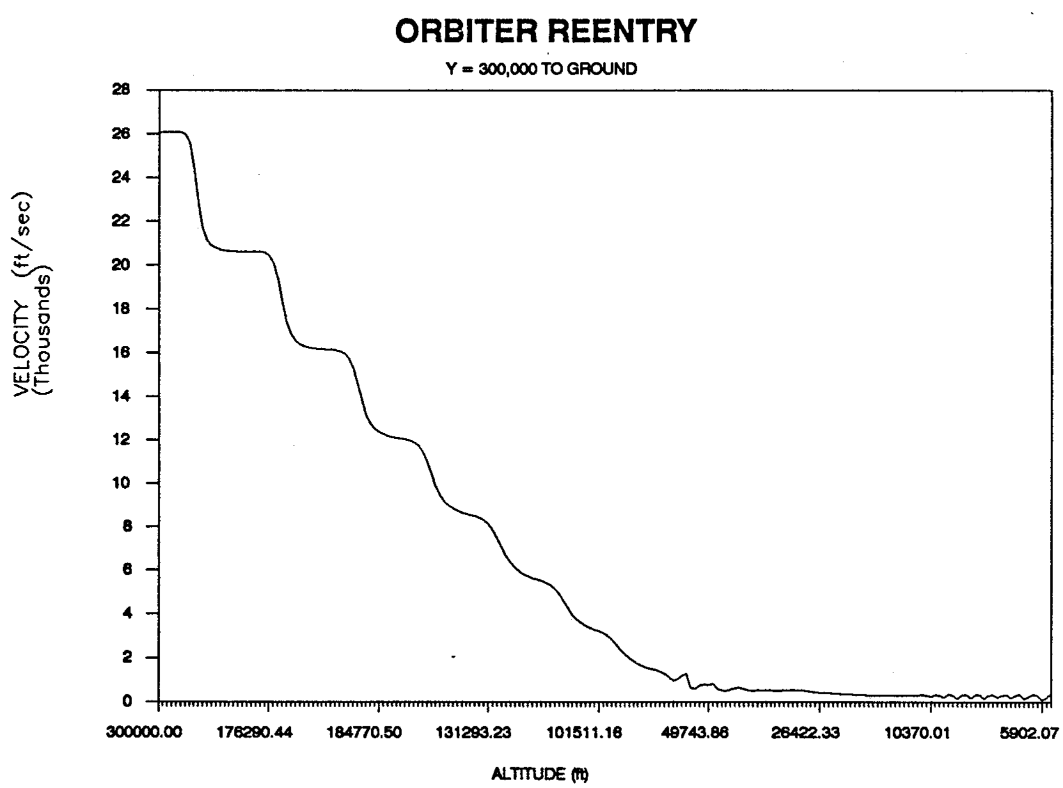


Figure A6

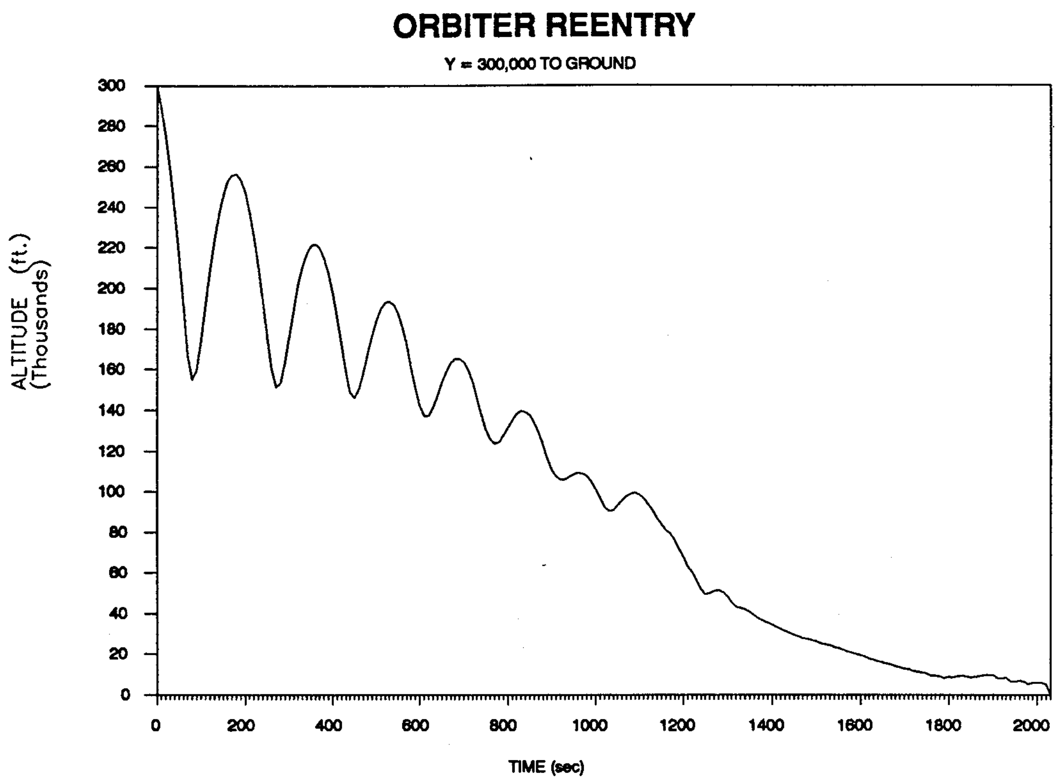


Figure A7

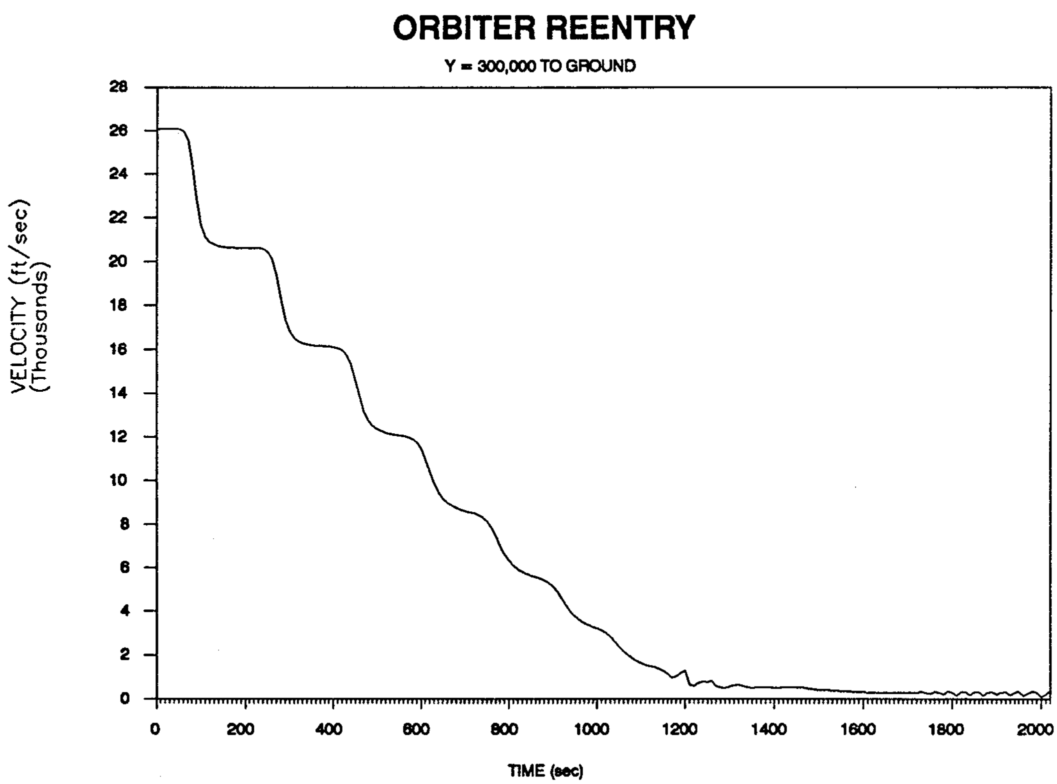


Figure A8

AERODYNAMIC HEATING:

The aerodynamic heating incurred upon atmospheric re-entry affects many aspects of the orbiter design, most importantly, structure weight. The temperatures that would occur during a typical re-entry require that the windward surfaces of the orbiter be covered with a temperature resistant material. The available materials can result in a large weight penalty, therefore the aero-thermodynamic environment of the orbiter was carefully considered.

Both temperature and heat transfer rate, along with heat load, were calculated at the orbiter nose and wing leading edge, two areas of most severe heating effects. These quantities are functions of velocity, altitude, material emmissivity, and airframe geometry. The computer program, previously discussed, which numerically determined the re-entry trajectory, provided the basis for the heating calculations.

The heating rate was approximated by

$$q' = c*rho^{N}*v^{M}*\cos(theta)$$

with

$$c = 3.73E\text{-}09*r_n^{-1/2}*(1-g_w)$$

and the wall temperature by

$$T_w = (q'/e*sigma)^{1/4}$$

Since T_w in part determines c, it was necessary to iterate between the equations for q and T_w.

The results of the heating calculations can be seen in Figures A9, A10, and A11. The maximum heat load that could be withstood in order to keep the inside skin temperature below melting is 7.048E+04 Btu/ft² (80 kJ/cm²). The heat loads for the orbiter's nose and for its leading

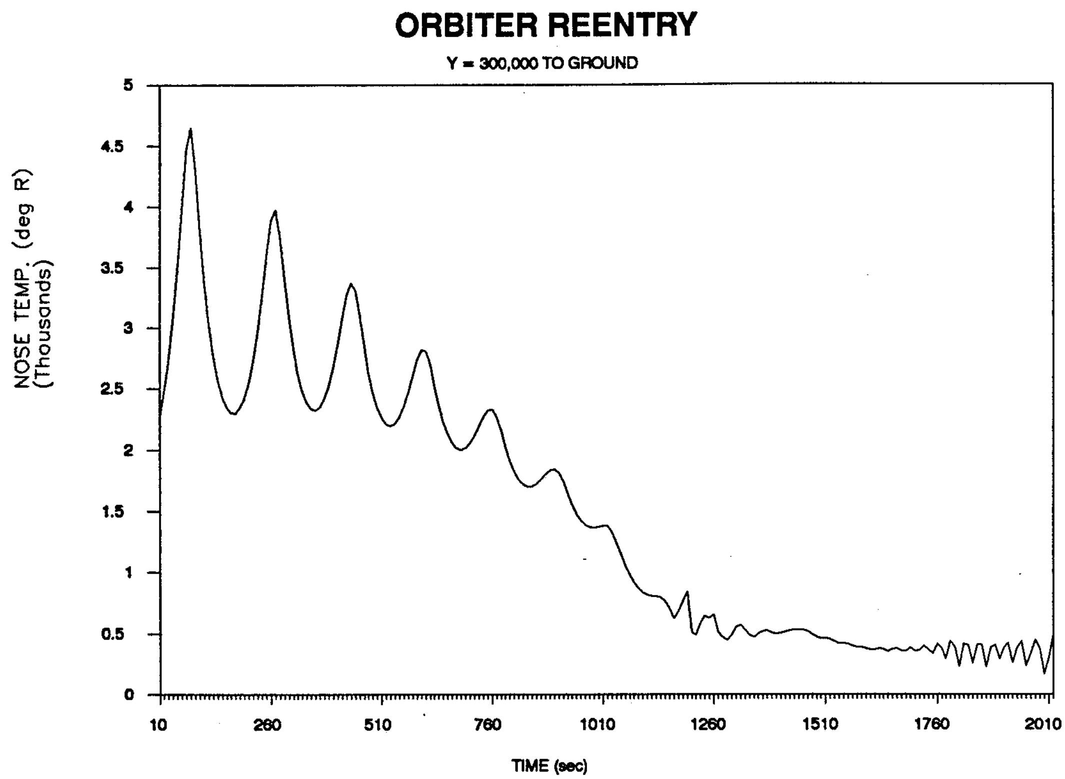


Figure A9

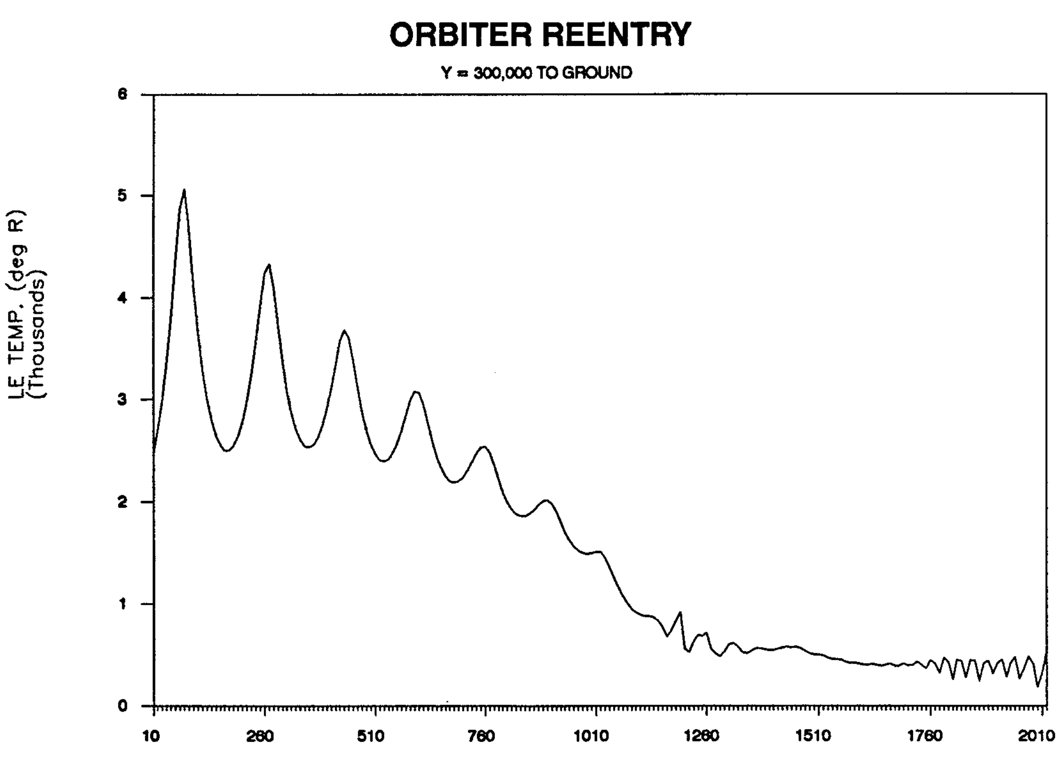


Figure A10

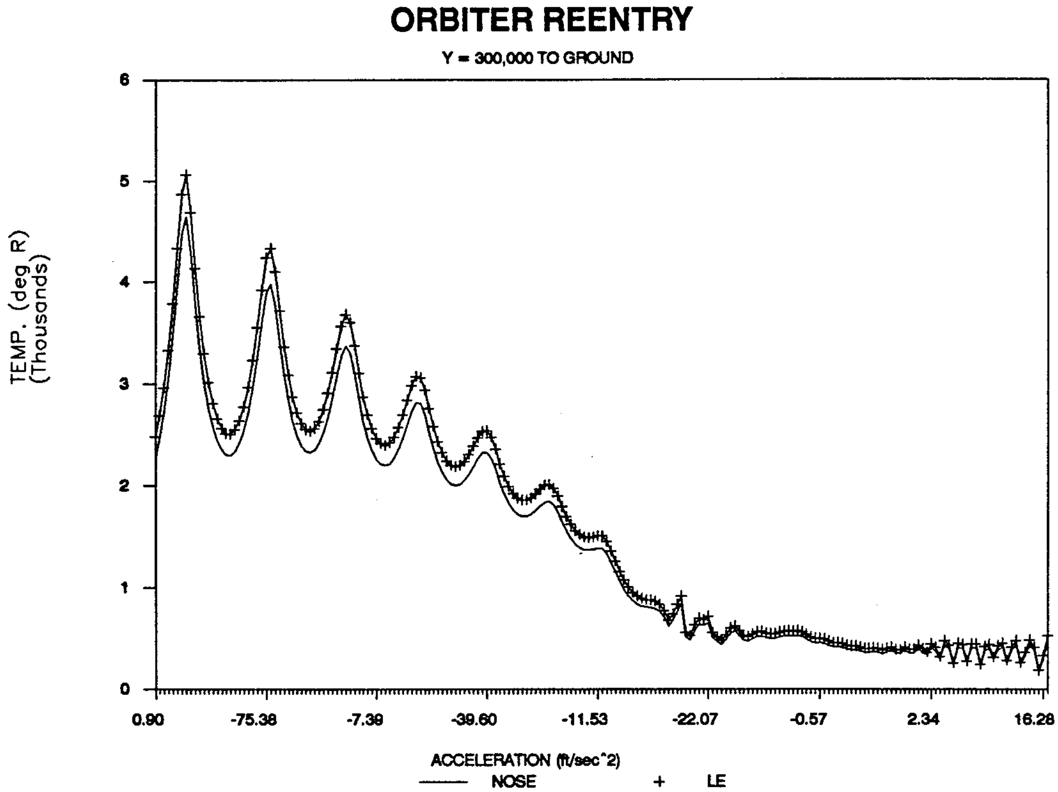


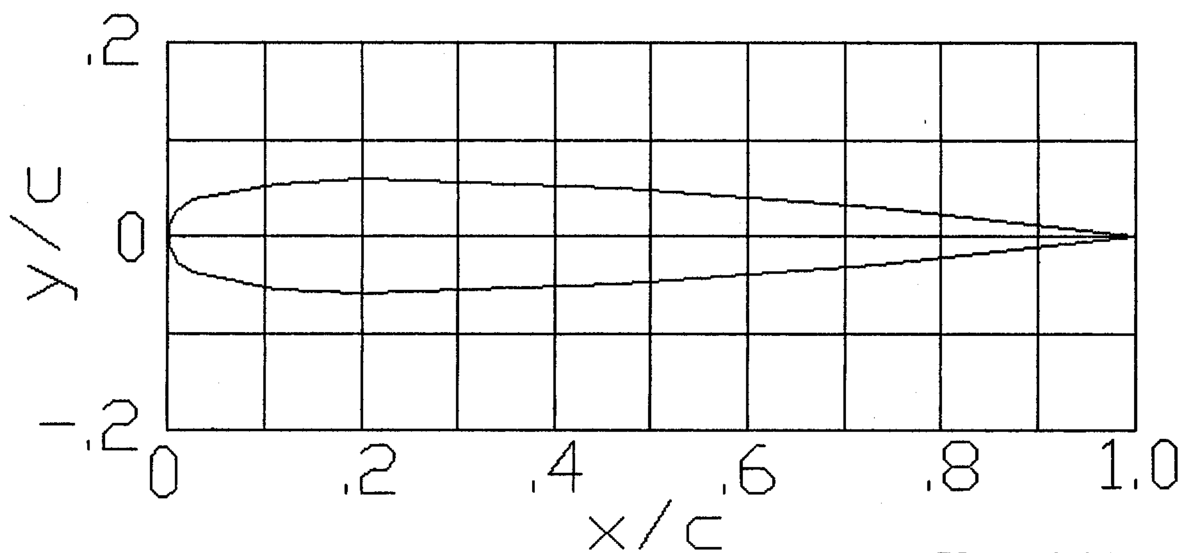
Figure A11

edge are both well below this value. While the maximum values for T_w at the nose (radius = 2 feet) and wing leading edge (radius = 6 inches) appear excessively high, this is due in large part to the oscillatory nature of the re-entry trajectory. If the re-entry could be managed more effectively and the oscillations in velocity and altitude reduced, the maximum values for T_w would be more reasonable. Wall temperatures around 3000 degrees F would be possible. This temperature is within the range that materials available in the near future will withstand without a severe weight penalty.

Based on the heating of a number of airfoils that were investigated with the previously mentioned computer program (NACA 0006, NACA 0009, NACA 64-006, and an unspecified airfoil with a 6" leading edge radius), the smallest leading edge that seemed to be able to maintain acceptable heating values was six inches. The orbiter group then designed an airfoil with this leading edge radius. The airfoil also has a maximum thickness to chord ratio of 12% (6% above and below the centerline). This maximum height is located at 20% of the chord. This airfoil can be seen on page A22

As well as a better controlled re-entry, the heating at higher altitudes should also be investigated. All the calculations performed and data presented represent altitudes below 300,000 feet. This was believed to be the upper limit of significant aerodynamic heating effects. However, the fact that the maximum wall temperatures occur at 230,000 feet indicates that still higher temperatures may be encountered at greater altitudes. Another area that needs to be investigated is heating at locations on the orbiter other than the nose and wing leading edges.

AIRFOIL CROSS-SECTION



$R_{le} = 0.0444C$

$t_{max}(.2C) = 0.06C$

INITIAL WING AND TAIL SIZING:

The initial size estimate for the orbiter wing was based on the conditions at landing. For a lift coefficient of 1.0, a density at zero altitude of 2.3675E-03 slugs/cubic feet, and a landing speed of 300 feet/second,

$$
\begin{aligned}
W/S(ref) &= (1/2)C_L(rho)V^2 \\
&= (1/2)(1)(2.3675E\text{-}03)(300)^2 \\
&= 106.5 \text{ pounds/square foot}
\end{aligned}
$$

For W = 88,200 pounds (with payload)

S(ref) = 880 square feet.

If this reference area was implemented, however, the majority of the wing area would be contained within the fuselage, leaving very little exposed wing area for control (~200 square feet). This area was insufficient for control. Therefore, the wing area was increased by nearly fifty percent to 1200 square feet.

The wing was designed to have an initial leading edge sweep angle of 60 degrees which tapered off to 45 degrees. Also, the exposed area aft of the 60 degree portion was designed to equal the exposed area aft of the 45 degree portion. Finally, the aspect ratio was set at 1.75. With these conditions, a wing was designed with a relative area of 1047 square feet and an actual area of 1200 square feet. The difference in these areas stemmed from the fact that the 60 degree portion of the wing was small as compared to the total wing area. As a result, this portion was ignored in all calculations in an attempt at simplification. Other specifications of this wing included:

Taper ratio	=	0.33
Y	=	9.564 feet
M.A.C.	=	24.71 feet
1/4 chord	=	6.18 feet aft of leading edge
Tip chord	=	11.27 feet
Root chord	=	29.25 feet

In order to find the aerodynamic center of this wing, an interpolation was required using the data located in Figure 16.5 of Aircraft Design: A Conceptual Approach. With an aspect ratio of approximately two and a sweep angle of 45 degrees, it was found that the aerodynamic center was located 11.24 feet aft of the leading edge along the mean aerodynamic chord for subsonic speeds and 15.76 feet aft for supersonic speeds.

The next task was to achieve static pitch stability. This was achieved by insuring that the total pitching moment was zero. This was done by placing the aerodynamic center aft of the center of gravity of the aircraft. The minimum distance separating the aerodynamic center and the center of gravity, expressed as a percentage of the mean aerodynamic chord, is called the static margin. Ten percent is a normal value for the static margin, and since the mean aerodynamic chord was approximately 25 feet in length, the aerodynamic center was placed 2.5 feet aft of the center of gravity. This 2.5 feet was measured from the furthest aft center of gravity (landing without a payload) to the furthest forward aerodynamic center (the subsonic case).

Trim was another factor that had to be considered. For stability, the total moment about the center of gravity must equal zero. This was accomplished through the use of horizontal and/or vertical tails. In order to size these tails, two coefficients had to be selected. These were the coefficient of volume for the vertical tail (C_{vt}) and the coefficient of volume for the horizontal tail (C_{ht}). These coefficients related the tail sizes to the size and orientation of the wing. To select these coefficients, the orbiter was modeled after a jet trainer, and from Table 6.4 in Aircraft Design: A Conceptual Approach, values of 0.7 and 0.06 for C_{ht} and C_{vt}, respectively, were found.

The equations relating wing size and orientation to tail size were as follows:

$$C_{vt} = (L_{vt}S_{vt})/(b_wS_w)$$

$$C_{ht} = (L_{ht}S_{ht})/(C_wS_w)$$

where
- L_{vt} = Length from wing quarter chord point to vertical tail quarter chord point.
- L_{ht} = Length from wing quarter chord point to horizontal tail quarter chord point.
- S_{vt} = Vertical tail area.
- S_{ht} = Horizontal tail area.
- b_w = Wing span.
- C_w = Wing mean chord length.
- S_w = Wing area.

The only unknowns in the above equations were the tail areas and the lengths between the quarter chord points. The distance was approximated, a tail area was found, the actual lengths between quarter chord points was evaluated, and the process was repeated until the distances were equivalent. A distance of 41 feet was chosen and areas equaling 70.6 square feet and 442.6 square feet for the vertical and horizontal tails was calculated.

In an effort to minimize weight, the vertical and horizontal tails were replaced by a V-tail. The V-tail was intended to reduce the wetted area of the tails. With a V-tail, the horizontal and vertical tail forces are the result of horizontal and vertical projections of the force exerted upon the "V" surfaces. In addition, V-tails offer reduced interference drag. However, extensive research has determined that in order to obtain satisfactory stability and control, the V-tail must be oversized to about the same equivalent area as the separate horizontal and vertical tails.

A trapezoid was selected as the basic shape of the V-tail in order to avoid complexity. An eleven foot span from the outer edge of the fuselage was arbitrarily selected and used in the initial calculations. This resulted in a V-tail which spanned twenty feet outward from the center of the fuselage.

Since the required horizontal tail area was larger than the required vertical tail area, the V-tail had to project a mimimum of 442.6 square feet in the horizontal direction. As a result, the projected vertical tail area was larger than required. The actual area for both projected vertical tails was 219 square feet. This gave a volume coefficient of 0.184; more than three times the starting value. The horizontal projected area was 525 square feet, which produced a volume coefficient of 0.82. The final specifications of each V-tail included:

Total area	=	371.6 square feet
Exposed area	=	154.88 square feet
Taper ratio	=	0.3
Y	=	8.21 feet
M.A.C.	=	20.37 feet
1/4 chord	=	5.09 feet aft of leading edge

After the wing and tail were sized, they required control surfaces. The wings needed ailerons to control roll, and the V-tail needed "ruddervators" to control both yaw and pitch. Ailerons are typically 50-90% of the span. Therefore, the aileron was initially designed with a length equal to 70% of the span from the fuselage wall to the tip (14 feet). The width of ailerons

LANDING PERFORMANCE:

The final task of the aerodynamics team was to calculate the landing distances required by the orbiter for two conditions: landing with a payload and landing without a payload. For the first of these two conditions a distance of 8600 feet was found, and for the second a landing distance of 6800 feet was found.

AERODYNAMIC

APPENDICES

APPENDIX A1
RE-ENTRY TRAJECTORY VELOCITY IMPULSE CALCULATIONS

IMPULSE NUMBER 1:

Before:

altitude	=	100,000 feet = 30,480 m
r_A	=	r_E + 100,000 ft = 6.378E+06 m + 30,480 m
	=	6.4085E+06 m
$(v_1)_{before}$	=	Mach 6 = 1848 m/sec

After:

$(v_1)_{after}$	=	$(2*(-u/(r_A+r_{A'}) + u/r_A))^{1/2}$
u	=	3.99E+14
r_A	=	6.4085E+06
$r_{A'}$	=	r_E+250 nmi = 6.378E+06 m+4.63E+05 m = 6.841E+06 m
$(v_1)_{after}$	=	8.0183E+03 m/sec

IMPULSE NUMBER 2:

Before:

$(v_2*r_{A'})_{Before}$	=	$(v_1*r_A)_{After}$
$(v_2)_{Before}$	=	7.5114E+03 m/sec

After:

$(v_2)_{After}$	=	$(u/r_{A'})^{1/2}$ = 7.6371E+03 m/sec = Mach 23.4

IMPULSE NUMBER 3:

Before:

$(v_3)_{Before}$	=	$(v_2)_{After}$ = 7.6371E+03 m/sec

After:

$(v_3)_{After}$	=	$(2(-u/(r_{A'}+r_A) + u/r_{A'}))^{1/2}$
	=	7.5114E+03 m/sec

Delta v_1 = 6.1703E+03 m/sec

Delta v_2 = 1.2571E+02 m/sec

Delta v_3 = 1.256E+02 m/sec

Delta v_{tot} = 6.4216E+03 m/sec

APPENDIX A2
ORBITER WEIGHT CALCULATIONS

$$\frac{m_{pay} + m_{struc} + m_{fuel}}{m_{pay} + m_{struc}} = e^{deltav/ve} = 4.15$$

FIRST ATTEMPT:

Assume $m_{struc} = 3m_{pay}$

$$\frac{4m_{pay} + m_{fuel}}{4m_{pay}} = 4.15$$

$m_{fuel} = 4m_{pay}*4.15 - 4m_{pay}$

for m_{pay} = 20,000 lbs

m_{fuel} = 252,320 lbs
m_{struc} = 60,000 lbs

for m_{pay} = 12,000 lbs

m_{fuel} = 151,290 lbs
m_{struc} = 36,000 lbs

SECOND ATTEMPT:

Assume $m_{struc} = 2.25m_{pay}$

$$\frac{3.25m_{pay} + m_{fuel}}{3.25m_{pay}} = 4.15$$

$m_{fuel} = 3.25m_{pay}(4.15) - 3.25m_{pay}$

for m_{pay} = 20,000 lbs

m_{fuel} = 204,730 lbs
m_{struc} = 45,000 lbs

for mpay = 12,000 lbs

m_{fuel} = 122,850 lbs
m_{struc} = 27,000 lbs

WEIGHT LIMITATIONS:

$W_O = W_F + W_{LG} + W_S + W_{SYS} + W_{PAY} + W_{PROP}$

F	=	FUEL
LG	=	LANDING GEAR
S	=	STRUCTURES
SYS	=	SYSTEMS
PAY	=	PAYLOAD
PROP	=	PROPULSION SYSTEM

RESTRICTIONS:

W_F/W_O	$= 0.70$	(from required velocity impulses)
W_{LG}	$= 0.05W_E$	$W_E = W_S + W_{SYS} + W_{PROP}$
W_{SYS}	= 16,000 LBS	(includes crew, life support, avionics, control, etc.)

$$1=W_F/W_O+.05(W_S+W_{SYS}+W_{PROP})/W_O+W_S/W_O+W_{SYS}/W_O+W_{PAY}/W_O+W_{PROP}/W_O$$

$$1 = W_F/W_O + 1.05W_S/W_O + 1.05W_{SYS}/W_O + 1.05W_{PROP}/W_O + W_{PAY}/W_O$$

$$1 = 0.7 + 1.05W_S/W_O + 17/W_O + 1.05W_{PROP}/W_O + W_{PAY}/W_O$$

$$0.3 = 1.05W_S/W_O + 17/W_O + 1.05W_{PROP}/W_O + W_{PAY}/W_O$$

Various combinations of W_S/W_O, W_{PROP}/W_O, AND W_{PAY}/W_O tried and results shown in report.

APPENDIX A3

PROGRAM HIGHORB

```
program highorb (input, orbit);

(* This program performs numerical integrations in order to chart the *)
(* re-entry trajectory of the orbiter from orbit height to 180,000 ft. *)

var
   orbit: text;
   z, r, theta, y, zdot, thdot, dt, zp, rp, thp, zcdot, zc, rc: real;
   thc, time, v, h, thcdot: real;
   i: integer;

BEGIN
(* Initialize *)
     zdot := -0.27843;
     z := 0.0;
     r := 6841000.0;
     thdot := 0.001098;
     theta := 0.0;
     y := 1500000.0;
     dt := 10.0;
     i := 1;
     zp := 0.0;
     rp := 0.0;
     thp := 0.0;
     zcdot := 0.0;
     thcdot := 0.0;
     zc := 0.0;
     rc := 0.0;
     thc := 0.0;
     time := 0.0;
     v := 2.4645E+04;
     h := 5.1385E+10;
     assign(orbit, 'c:\turbo\highorb.p');
     rewrite(orbit);
(* Write initial values to screen and save in data file. *)
     writeln(i, time, y, v);
     writeln(orbit, i, time, y, v);
     while (r > 6.4696E+06) do begin
           zp := z + zdot * dt;
           rp := r + zp * dt;
           thp := theta + thdot * dt;
           zcdot := (-3.99E+14/(rp*rp))+(2.6404E+21/(rp*rp*rp));
           thcdot := (5.1385E+10/(rp*rp));
           zc := z + zcdot*dt;
           rc :=  r + zc*dt;
           thc := theta + thcdot * dt;
           i := i + 1;
           z := 0.5 * (zp + zc);
           r := 0.5 * (rp + rc);
           theta := 0.5 * (thp + thc);
           zdot := (-3.99E+14/(r*r))+(2.6404E+21/(r*r*r));
           thdot := (5.1385E+10/(r*r));
           y := (r-6.37812E+06)*3.281;
           v := (h/r)*3.281;
           time := time + dt;
(* Data writen to screen and saved in a data file. *)
           writeln(i, time, y, v);
           writeln(orbit, i, time, y, v);
        end;
END.
```

APPENDIX A4

PROGRAM SLOWSP

```
program slowsp (input, slowsp);

(* This program performs numerical integrations in order to chart the *)
(* re-entry trajectory of the orbiter from 230,000 ft to landing.     *)

var
   lowsp, heat: text;
   v, r, theta, y, vdot, thdot, dt, vp, rp, thp, vcdot, vc, rc: real;
   thc, time, cd, a, m, g, gw, thcdot, cl, b, rho, rhoo, yp, yc: real;
   c, Rn, q, Tw1, Tw2, e, sigma, T, hw, hi, qload, x, xc, xp : real;
   base, expo, Tw3, Tw4, Rle, qle, qleload, delta, alpha : real;
   i: integer;
   stop: char;



function enthalpy(T : real) : real;
var
    h : array [1..130] of real;
    m : real;
    n , m2 : integer;

begin
        h[1] := 47.67;
        h[2] := 52.46;
        h[3] := 57.25;
        h[4] := 62.03;
        h[5] := 66.82;
        h[6] := 71.61;
        h[7] := 76.40;
        h[8] := 81.18;
        h[9] := 85.97;
        h[10] := 90.75;
        h[11] := 95.53;
        h[12] := 100.32;
        h[13] := 105.11;
        h[14] := 109.90;
        h[15] := 114.69;
        h[16] := 119.48;
        h[17] := 124.27;
        h[18] := 129.06;
        h[19] := 133.86;
        h[20] := 138.66;
        h[21] := 143.47;
        h[22] := 148.28;
        h[23] := 153.09;
        h[24] := 157.92;
        h[25] := 162.73;
        h[26] := 167.56;
        h[27] := 172.39;
        h[28] := 177.23;
        h[29] := 182.08;
        h[30] := 186.94;
        h[31] := 191.81;
        h[32] := 196.69;
        h[33] := 201.56;
        h[34] := 206.46;
        h[35] := 211.35;
        h[36] := 216.26;
```

```
h[37] := 221.18;
h[38] := 226.11;
h[39] := 231.06;
h[40] := 236.02;
h[41] := 240.98;
h[42] := 245.97;
h[43] := 250.95;
h[44] := 255.96;
h[45] := 260.97;
h[46] := 265.99;
h[47] := 271.03;
h[48] := 276.08;
h[49] := 281.14;
h[50] := 286.21;
h[51] := 291.30;
h[52] := 296.41;
h[53] := 301.52;
h[54] := 306.65;
h[55] := 311.79;
h[56] := 316.94;
h[57] := 322.11;
h[58] := 327.29;
h[59] := 332.48;
h[60] := 337.68;
h[61] := 342.90;
h[62] := 348.14;
h[63] := 353.37;
h[64] := 358.63;
h[65] := 363.89;
h[66] := 369.17;
h[67] := 374.47;
h[68] := 379.77;
h[69] := 385.08;
h[70] := 390.40;
h[71] := 395.74;
h[72] := 401.09;
h[73] := 406.45;
h[74] := 411.82;
h[75] := 417.20;
h[76] := 422.59;
h[77] := 428.00;
h[78] := 433.41;
h[79] := 438.83;
h[80] := 444.26;
h[81] := 449.71;
h[82] := 455.17;
h[83] := 460.63;
h[84] := 466.12;
h[85] := 471.60;
h[86] := 477.09;
h[87] := 482.60;
h[88] := 488.12;
h[89] := 493.64;
h[90] := 499.17;
h[91] := 504.71;
h[92] := 510.26;
h[93] := 515.82;
h[94] := 521.39;
h[95] := 526.97;
h[96] := 532.55;
```

```
      h[97] := 538.15;
      h[98] := 543.74;
      h[99] := 549.35;
      h[100] := 554.97;
      h[101] := 560.59;
      h[102] := 566.23;
      h[103] := 571.86;
      h[104] := 577.51;
      h[105] := 583.16;
      h[106] := 588.82;
      h[107] := 594.49;
      h[108] := 600.16;
      h[109] := 605.84;
      h[110] := 611.53;
      h[111] := 617.22;
      h[112] := 622.92;
      h[113] := 628.62;
      h[114] := 634.34;
      h[115] := 640.05;
      h[116] := 645.78;
      h[117] := 660.12;
      h[118] := 674.49;
      h[119] := 688.90;
      h[120] := 703.35;
      h[121] := 717.83;
      h[122] := 732.33;
      h[123] := 746.88;
      h[124] := 761.45;
      h[125] := 776.05;
      h[126] := 790.68;
      h[127] := 805.34;
      h[128] := 820.03;
      h[129] := 834.75;
      h[130] := 849.48;
      if T <= 2500.0 then m := (T/20)-9
        else m := (T/50)+66;

      m2 := trunc(m);
      if m2 > 129 then m2 := 129;
      enthalpy := ((h[m2+1]-h[m2])*(m-m2))+h[m2];
end;


    function temperature(y : real): real;
    var
      T : array [1..51] of real;
      y2 : real;
      n, y3 : integer;

    begin
      T[1] := 519.0;
      T[2] := 483.0;
      T[3] := 447.0;
      T[4] := 412.0;
      T[5] := 390.0;
      T[6] := 390.0;
      T[7] := 390.0;
      T[8] := 392.0;
      T[9] := 398.0;
      T[10] := 403.0;
```

```
   T[11] := 409.0;
   T[12] := 418.0;
   T[13] := 434.0;
   T[14] := 449.0;
   T[15] := 464.0;
   T[16] := 479.0;
   T[17] := 487.0;
   T[18] := 487.0;
   T[19] := 479.0;
   T[20] := 468.0;
   T[21] := 457.0;
   T[22] := 438.0;
   T[23] := 416.0;
   T[24] := 395.0;
   T[25] := 373.0;
   T[26] := 352.0;
   T[27] := 330.0;
   T[28] := 325.0;
   T[29] := 325.0;
   T[30] := 325.0;
   T[31] := 333.0;
   T[32] := 349.0;
   T[33] := 365.0;
   T[34] := 383.0;
   T[35] := 409.0;
   T[36] := 435.0;
   T[37] := 460.0;
   T[38] := 510.0;
   T[39] := 560.0;
   T[40] := 611.0;
   T[41] := 694.0;
   T[42] := 795.0;
   T[43] := 896.0;
   T[44] := 996.0;
   T[45] := 1094.0;
   T[46] := 1194.0;
   T[47] := 1292.0;
   T[48] := 1391.0;
   T[49] := 1489.0;
   T[50] := 1586.0;
   T[51] := 1663.0;

   y2 := (y/10000)+1;
   y3 := trunc(y2);
   temperature := ((T[y3+1]-T[y3])*(y2-y3))+T[y3];
end;

function lift (v,y, alpha:real) :real;
var
   mach, temp, a, AR, F, betasq, eta, lambda, sexp, sref : real;
   lift1, lift2, lift3, Cn, beta :real;

begin
     temp := temperature ( y );
     a := 49.0 * sqrt ( temp );
     mach := v / a;
     if mach < 1.2 then
     begin
       AR := 2.0;
       F := 2.07;
```

```
          betasq := 1 - sqr(mach);
          eta := 0.95;
          lambda := 3.1416/180*51.4;
          sexp := 511.6;
          sref := 1047.0;
          lift1 := 2*pi*AR*(sexp/sref)*F;
          lift2 := 2 + sqrt(4+(AR*AR*betasq/(eta*eta)*(1+(sqr(sin(lambda)/
                   cos(lambda))/betasq))));
          lift3 := lift1/lift2;
          lift := lift3*(alpha*pi/180);
          lift1 := lift3*(alpha*pi/180);
        end;
        if ((mach >1.2) and (y <=  100000)) then
        begin
           beta := sqrt(sqr(mach) -1);
           lift := (4/beta)*(alpha*pi/180);
        end;
        if y > 100000.0 then
        begin
           Cn := 2*sqr(sin(alpha*pi/180));
           lift := Cn*cos(alpha*pi/180);
        end;
    end;


function drag (v,y, alpha, cl:real) :real;
    var
       mach, temp, a, cdmin, cdl, Cn : real;

    begin
         temp := temperature ( y );
         a := 49.0 * sqrt ( temp );
         mach := v / a;
         if mach < 0.8 then
         begin
           cdmin := 0.011;
           cdl := sqr(cl)*(0.1378+(0.1693*mach)-(0.0115*sqr(mach)));
           drag := cdmin + cdl;
         end;
         if ((mach >= 0.8) and (mach <= 1.2)) then
         begin
           cdmin := -0.510+(0.0767*mach);
           cdl := sqr(cl)*(0.1378+(0.1693*mach)-(0.0115*sqr(mach)));
           drag := cdmin + cdl;
         end;
         if ((mach > 1.2) and (y <= 100000.0)) then
         begin
           cdmin := 0.0605 - (0.0177*mach) + (0.00163*sqr(mach));
           cdl := sqr(cl)*(0.1378+(0.1693*mach)-(0.0115*sqr(mach)));
           drag := cdmin + cdl;
         end;
         if y > 100000 then
         begin
           Cn := 2*sqr(sin(alpha*pi/180));
           drag := Cn*sin(alpha*pi/180);
         end;
    end;
```

```
function power (base, expo: real) :real;
(* performs exponentiation *)
     begin
          power := exp(ln(base)*expo);
     end;

procedure integrate(var alpha, cl, cd, vdot, thdot, Tw2, q, qload, Tw4, qle,
                    qleload, v, theta, y, x, rho: real);

     var
        rhoo, b, a, g, m, Rn, dt, vp, thp, vcdot, vc, xp, yp, xc, yc, thc: real;
        Tw1, e, sigma, Tw3, Rle, delta, T, hi, hw, gw, c: real;

     begin
          (* initialize *)
           rhoo := 3.4E-03;      (* slugs/ft^3 *)
           b := 0.0000455;       (* ft^-1 *)
           a := 1047.42;         (* ft^2 *)
           g := 32.2;            (* ft/sec^2 *)
           m := 2739.1;          (* slugs *)
           Rn := 2.0;            (* feet *)
           dt := 10.0;           (* sec *)
           vp := 0.0;
           thp := 0.0;
           vcdot := 0.0;
           vc := 0.0;
           xp := 0.0;
           yp := 0.0;
           xc := 0.0;
           yc := 0.0;
           thc := 0.0;
           Tw1 := 0.0;
           e := 0.3;             (* emissivity *)
           sigma := 0.476E-12;   (* Stephan-Boltzman Const.  Btu/sec*ft^2*R^4*)
           Tw3 := 0.0;
           Rle := 0.5;            (* feet *)
           delta := 0.785;        (* radians *)


           vp := v + vdot * dt;
           thp := theta + thdot * dt;
           xp := x + vp * dt * cos(thp);
           yp := y - vp * dt * sin(thp);
(*  Ensure that deceleration is not greater than 6g (-193.2 ft/sec^2).  *)
           vcdot := ((-cd*rho*vp*vp*a)/(2*m))+(g*sin(thp));
           if vcdot < -193.2 then vcdot := -193.2;
           thcdot := (-(cl*rho*vp*a)/(2*m))+((g/vp)*cos(thp));
           vc := v + vcdot*dt;
           thc := theta + thcdot * dt;
           xc := x + vc * dt * cos(thc);
           yc := y - vc * dt * sin(thc);
           v := 0.5 * (vp + vc);
           theta := 0.5 * (thp + thc);
           x := 0.5 * (xc + xp);
           y := (0.5 * (yc + yp));
           rho := rhoo * (exp(-(b*y)));
(*  Go to function to calculate lift coefficient *)
           cl := lift ( v , y, alpha );
(*  Go to function to calculate drag coefficient *)
           cd := drag ( v , y, alpha, c );
```

```
(*  Ensure that deceleration is less then 6g (-193.2 ft/sec^2).   *)
           vdot := ((-cd*rho*v*v*a)/(2*m))+(g*sin(theta));
           if vdot < -193.2 then vdot := -193.2;
           thdot := (-(cl*rho*v*a)/(2*m))+((g/v)*cos(theta));
(* Calculate heating information for nose. *)
  (* Call function to calculate temperature at altitude y *)
           T := temperature(y);
  (*  Call function to calculate enthalpy at freestream temperature T *)
           hi := enthalpy(T);
           while (abs((Tw2-Tw1)/Tw2) > 0.01) do begin
               Tw1 := Tw2;
  (* Call function to calculate enthalpy at wall temperature T *)
               hw := enthalpy(Tw2);
               gw := (hw/(hi+((v*v)/2)));
               c := (3.7263E-09/(sqrt(Rn)))*(1-gw);
               q := c*(sqrt(rho))*v*v*v;
               Tw2 := sqrt(sqrt(q/(e*sigma)));
           end;
           Tw1 := 0.0;
           while (abs((Tw4-Tw3)/Tw4) > 0.01) do begin
               Tw3 := Tw4;
               hw := enthalpy(Tw4);
               gw := (hw/(hi+((v*v)/2)));
               c := (3.7263E-09/(sqrt(Rle)))*(1-gw);
               qle := c*(sqrt(rho))*v*v*v*cos(delta);
               Tw4 := sqrt(sqrt(qle/(e*sigma)));
           end;
           Tw3 := 0.0;
      end;

BEGIN
(* Initialize *)

     i := 1;
     time := 0.0;
     alpha := 40.0;
     vdot := 0.015;
     v := 26054;
     theta := 0.0301;
     y := 300000.0;
     x := 0.0;
     cd := 0.531;
     cl := 0.633;
     Tw2 := 395.0;
     qload := 0.0;
     q := 0;
     Tw4 := 3100.0;
     qle := 0;
     qleload := 0.0;
     thdot := 1.22E-03;
     dt := 10.0;
     rho := 5.15E-09;      (* slugs/ft^3 *)


     assign(lowsp, 'lowsp.p');
     assign(heat, 'heat.p');
     rewrite(lowsp);
     rewrite(heat);
(* Write initial values to screen and save in data file. *)
     writeln(i, time, y, v, x, cl, cd, Tw2, q, qload,vdot, alpha, Tw4);
```

```
    writeln(lowsp, i, time, y, v, x, theta, cl, cd, alpha, vdot);
    writeln(heat, time, Tw2, q, qload, Tw4, qle, qleload);
    while (y > 0.0) do begin
     if (y >= 30000.0) then
     begin
       if (v*sin(theta) <= 200.0) then
       begin
          if (alpha > 0.0) then
          begin
            alpha := alpha - 1.0;
            integrate(alpha, cl, cd, vdot, thdot, Tw2, q, qload, Tw4, qle,
                      qleload, v, theta, y, x, rho);
          end
          else
             integrate(alpha, cl, cd, vdot, thdot, Tw2, q, qload, Tw4, qle,
                       qleload, v, theta, y, x, rho);
       end
       else begin
          if (alpha < 40.0) then
          begin
             alpha := alpha + 1;
             integrate(alpha, cl, cd, vdot, thdot, Tw2, q, qload, Tw4, qle,
                       qleload, v, theta, y, x, rho);
          end
          else
             integrate(alpha, cl, cd, vdot, thdot, Tw2, q, qload, Tw4, qle,
                       qleload, v, theta, y, x, rho);
       end;
     end
     else begin
       if (v <= 300.0) then
       begin
          if (alpha > 0.0) then
          begin
            alpha := alpha - 1.0;
            integrate(alpha, cl, cd, vdot, thdot, Tw2, q, qload, Tw4, qle,
                      qleload, v, theta, y, x, rho);
          end
          else
             integrate(alpha, cl, cd, vdot, thdot, Tw2, q, qload, Tw4, qle,
                       qleload, v, theta, y, x, rho);
       end
       else begin
          if (alpha < 40.0) then
          begin
             alpha := alpha + 1;
             integrate(alpha, cl, cd, vdot, thdot, Tw2, q, qload, Tw4, qle,
                       qleload, v, theta, y, x, rho);
          end
          else
             integrate(alpha, cl, cd, vdot, thdot, Tw2, q, qload, Tw4, qle,
                       qleload, v, theta, y, x, rho);
       end;
     end;
      qload := qload + q*dt;
      qleload := qleload + qle*dt;
      time := time + dt;
      i := i + 1;
(* Data writen to screen and saved in a data file. *)
      writeln(i, time, y, v, x, cl, cd, alpha, Tw2, q, qload,vdot);
```

```
      writeln(lowsp, i, time, y, v, x, theta, cl, cd, alpha, vdot);
      writeln(heat, time, Tw2, q, qload, Tw4, qle, qleload);
    end;
    readln(stop);
end.
```

APPENDIX A5
VARIABLE DEFINITIONS

M_{PAY} = PAYLOAD WEIGHT
M_{STURC} = M_s = STRUCTURE WEIGHT
M_{FUEL} = $M_{PROPELLANT}$ = FUEL WEIGHT
M_{TOT} = M = TOTAL WEIGHT
r = RADIAL POSITION OF VEHICLE W.R.T. THE CENTER OF THE EARTH
u = G*M_E G = GRAVITATIONAL CONSTANT M_E = MASS OF EARTH
h = ANGULAR MOMENTUM (v*r)
THETA = VEHICLE HEADING ANGLE
D = DRAG FORCE
L = LIFT FORCE
v = VEHICLE VELOCITY
a = ANGLE OF ATTACK
C_{La} = WING LIFT SLOPE
AR = WING ASPECT RATIO
S_{EXP} = EXPOSED WING AREA
S_{REF} = A = WING REFERENCE AREA
C_{la} = SECTION LIFT SLOPE
$LAMBDA_{max\,t}$ = WING SWEEP ANGLE AT MAXIMUM THICKNESS
b = WING SPAN
$LAMBDA_{LE}$ = WING LEADING EDGE SWEEP ANGLE
C_{pl} = PRESSURE COEFFICIENT ON LOWER SIDE OF WING
C_{pu} = PRESSURE COEFFICINET OF UPPER SIDE OF WING
CD_{min} = PARASIT DRAG COEFFICIENT
C_{DL} = LIFT INDUCED DRAG COEFFICIENT
qload = HEAT LOAD
q' = HEATING RATE
v_y = Y COMPONENT OF VELOCITY
r_n = NOSE RADIUS
M = 3 (FOR HEATING CALCULATIONS)
N = 0.5 (FOR HEATING CALCULATIONS)
g_w = RATIO OF WALL ENTHALPY TO TOTAL ENTHALPY
e = EMMISSIVITY
sigma = STEPHAN-BOLTZMAN CONSTANT

APPENDIX A6

WING SIZING CALCULATIONS

Design of wing

Refer to the autocad drawing, which follows these calculations, for all variables concerning the wing design.

$$
\begin{aligned}
Y2 &= 14 - Y1 \\
X2 &= X1 - Y1/TAN30 \\
X3 &= X2 - Y2 \\
&= X1 - Y1/TAN30 - 14 + Y1
\end{aligned}
$$

$$(1/2)\{X1 + X2\}Y1 = (1/2)\{X2 + X3\}Y2$$

$$(X1 + X1 - Y1/TAN30)Y1 = (X1 - Y1/TAN30 + X1 - Y1/TAN30 - 14 + Y1)\{14 - Y1\}$$

This simplifies to:

$$Y1^2(1 - 3/TAN30) + Y1(28/TAN30 - 28 + 4X1) - 28X1 + 196 = 0$$

However,

$$1/2(2X1 + 15.59)9 + (2X1 - Y1/TAN30)Y1 = 600$$

Which yields,

$$X1 = (529.845 + Y1^2/TAN30)/(9 + 2Y1)$$

Substituting this equation for X1 into the previous equation and simplifying yields:

$$-1.4644Y1^3 - 45.2688Y1^2 + 2695.853Y1 - 13071.66 = 0$$

Solving yields values of:

$$
\begin{aligned}
Y1 &= 5.431 \text{ feet} \\
Y2 &= 8.569 \text{ feet} \\
\\
X1 &= 29.25 \text{ feet} \\
X2 &= 19.84 \text{ feet} \\
X3 &= 29.25 \text{ feet}
\end{aligned}
$$

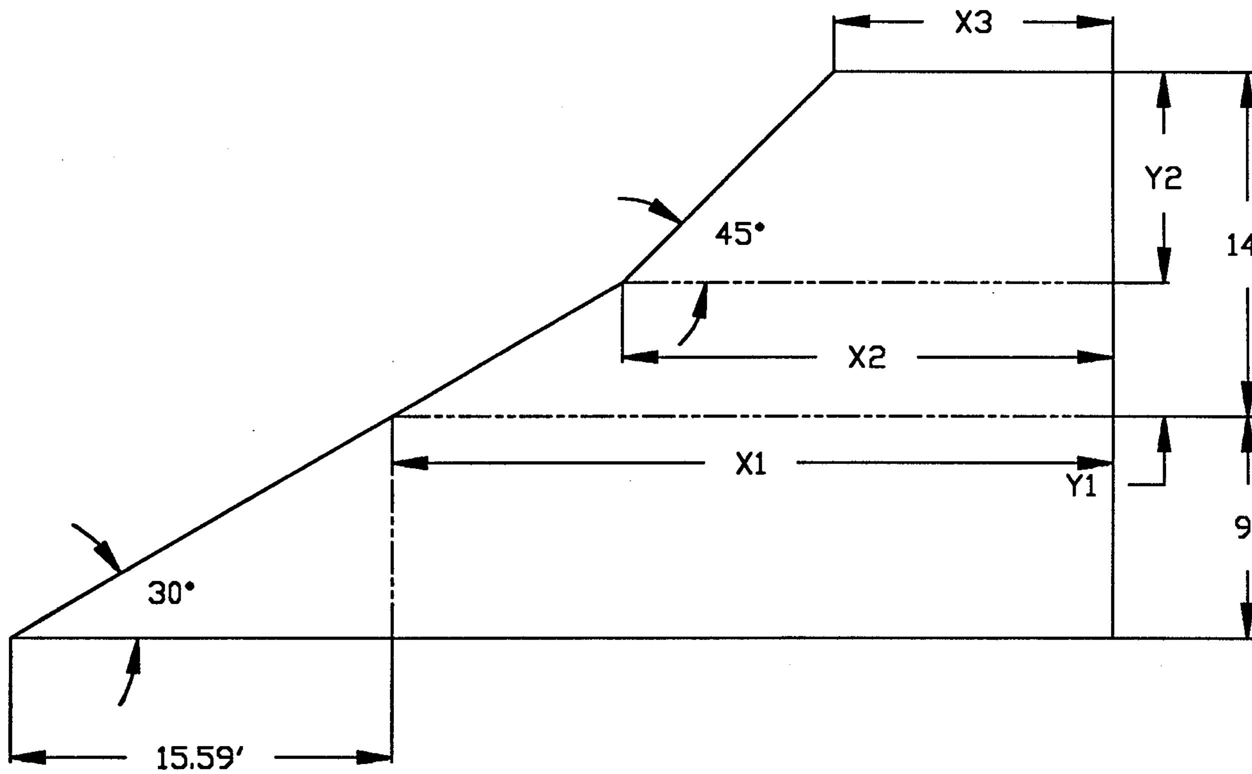
Aerodynamic parameters of this wing are calculated as follows:

$$
\begin{aligned}
\text{Taper ratio (T)} &= \text{tip chord/root chord} \\
&= 11.27/34.27 \\
&= 0.329
\end{aligned}
$$

$$
\begin{aligned}
\text{Vertical distance to M.A.C.(Y)} &= (b/6)\{(1+2T)/(1+T)\} \quad \text{(where b=46 feet)} \\
&= 9.564 \text{ feet}
\end{aligned}
$$

$$
\begin{aligned}
\text{M.A.C.(C)} &= (2/3)(34.27)\{(1+T+T^2)/(1+T)\} \\
&= 24.71 \text{ feet}
\end{aligned}
$$

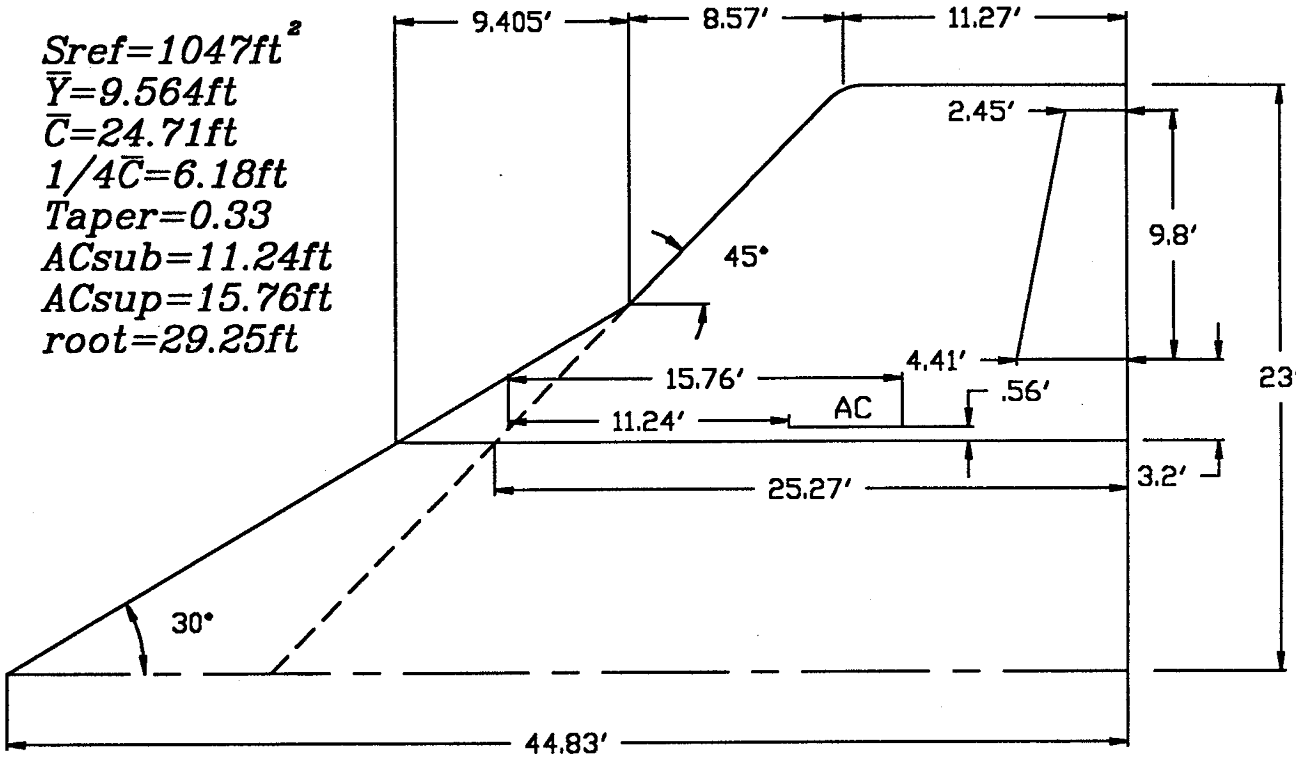
WING USED FOR INITIAL SIZING CALCULATIONS



APPENDIX A7

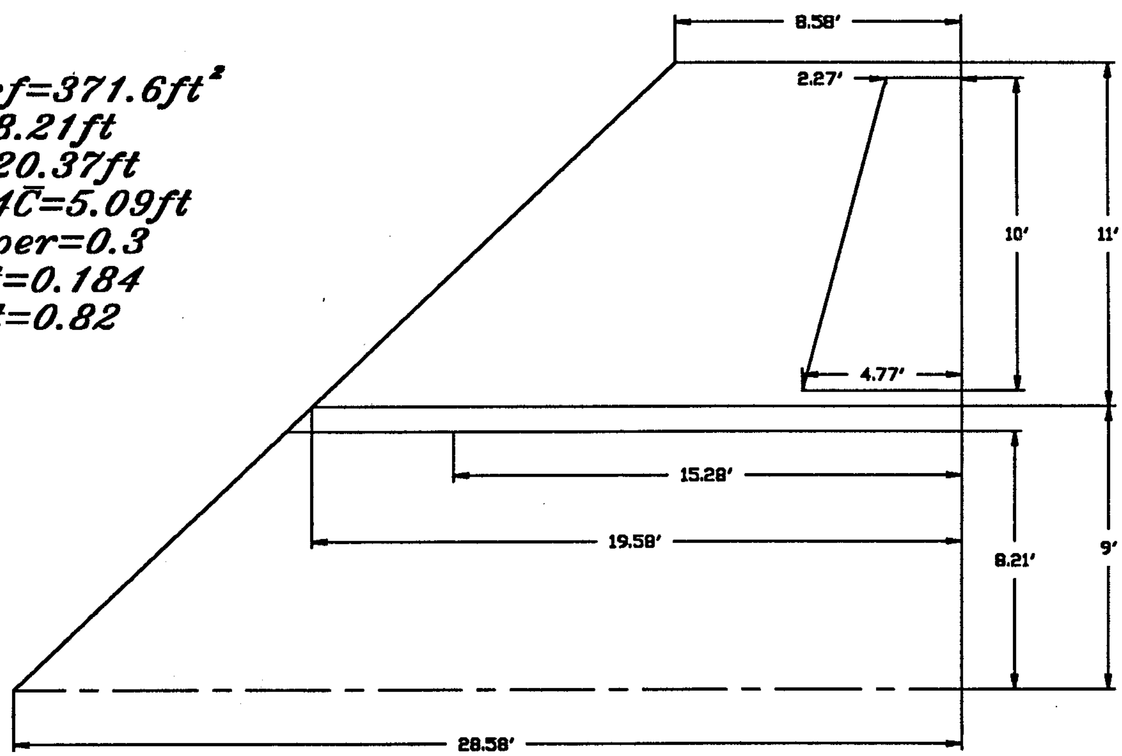
COMPUTER GENERATED GRAPHICS

WING DIMENSIONS

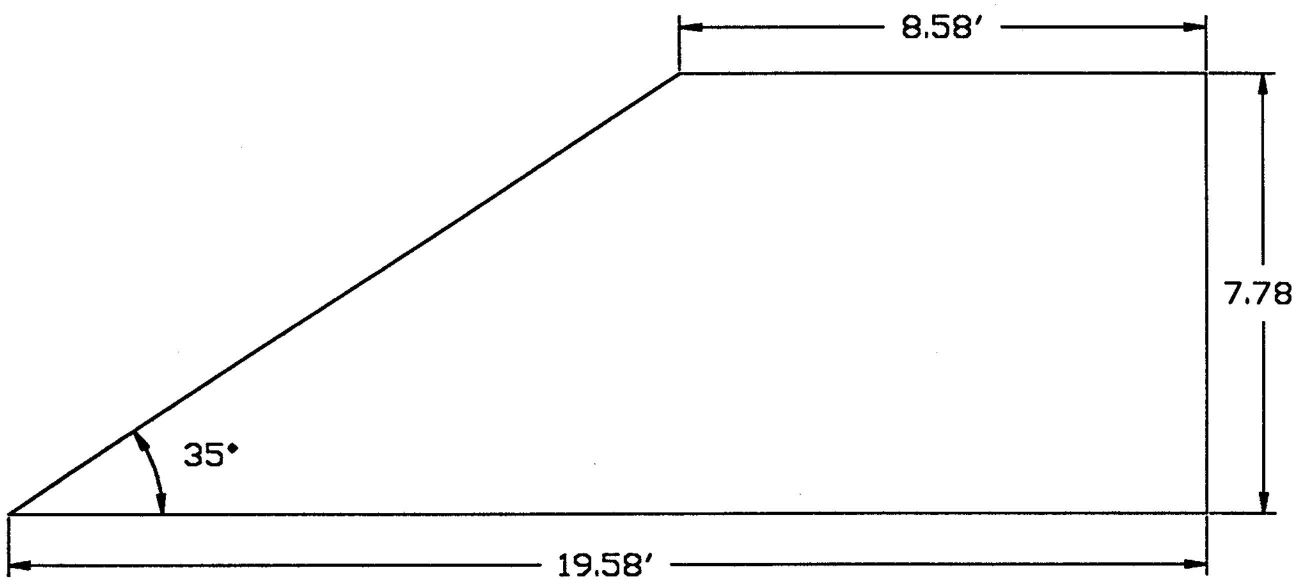


V-TAIL DIMENSIONS

$S_{ref}=371.6ft^2$
$\overline{Y}=8.21ft$
$\overline{C}=20.37ft$
$1/4\overline{C}=5.09ft$
$Taper=0.3$
$C_{vt}=0.184$
$C_{ht}=0.82$

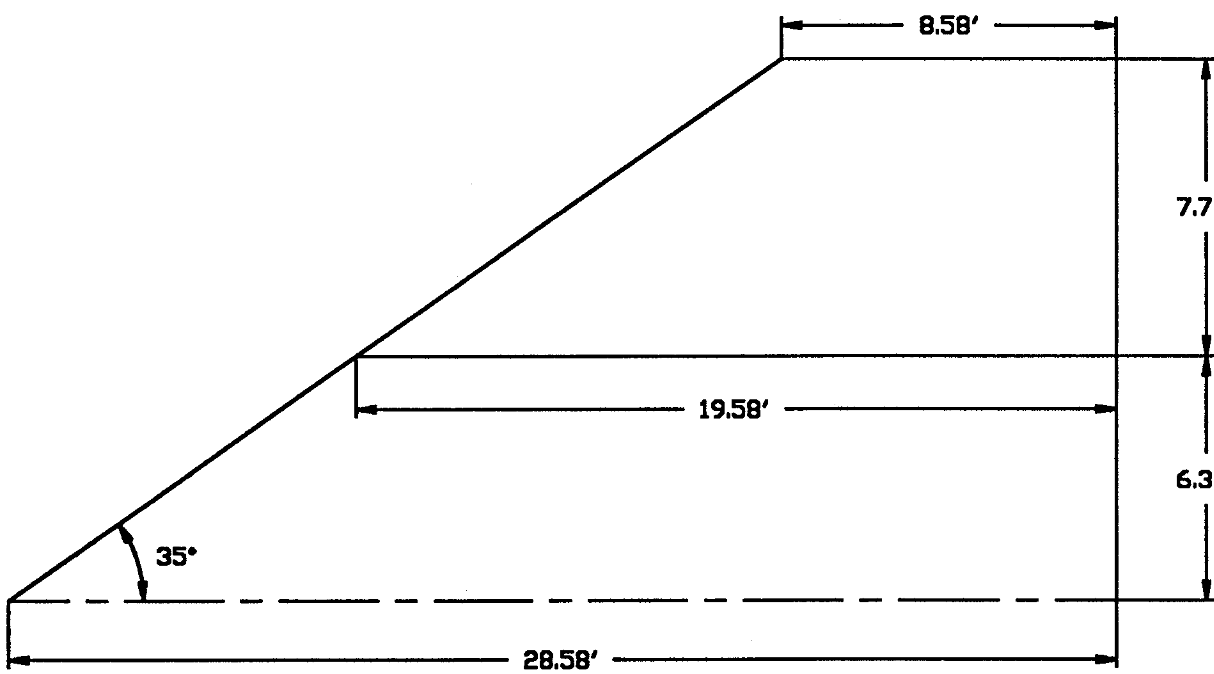


PROJECTED AREA
VERTICAL TAIL



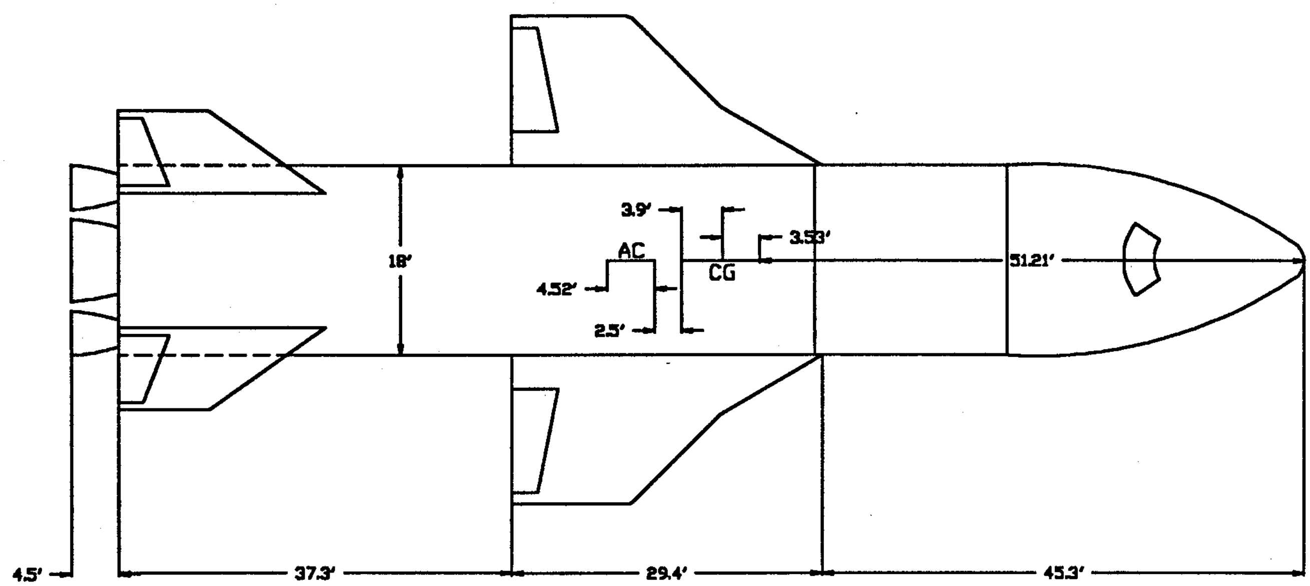
AREA = 109.5ft²

PROJECTED AREA
HORIZONTAL TAIL

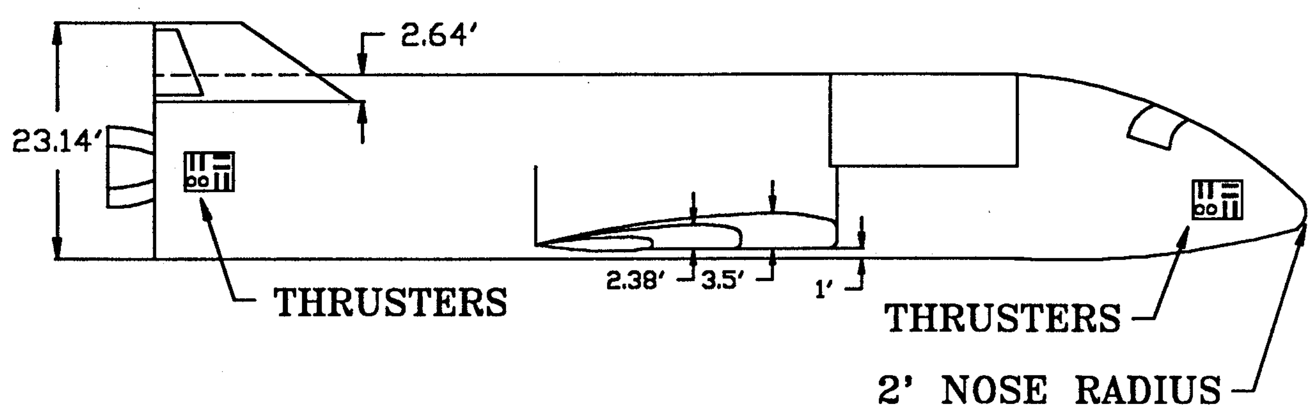


AREA = 262.7 ft^2

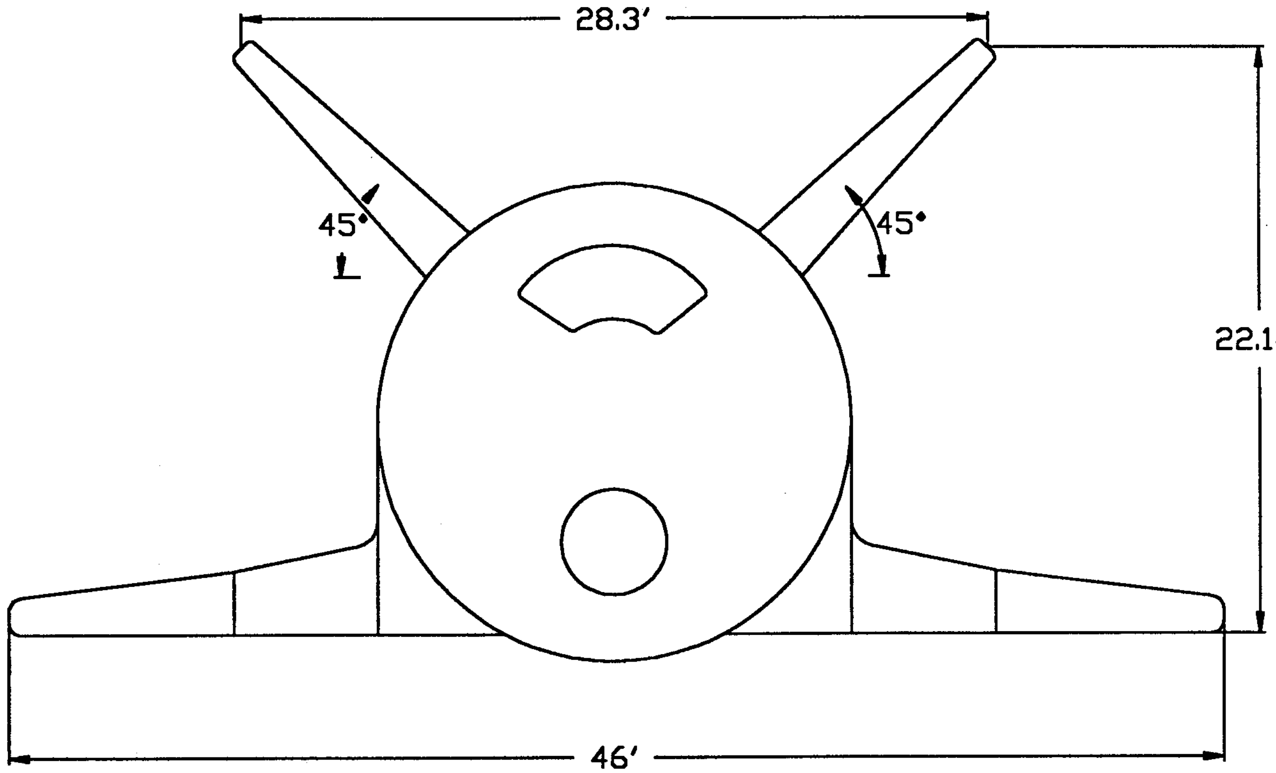
ORBITER TOP-VIEW

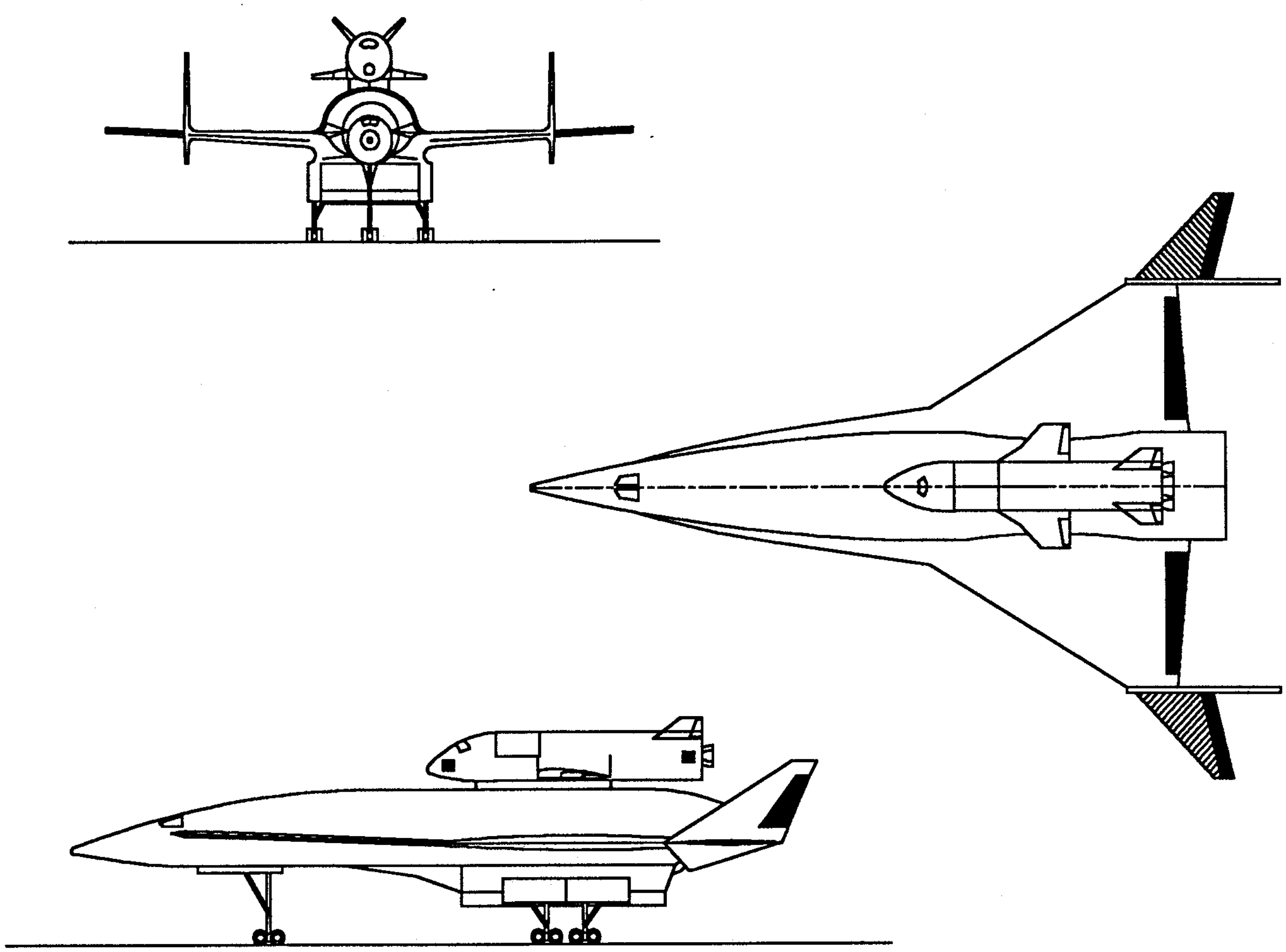


ORBITER SIDE-VIEW



FRONT VIEW





PROPULSION GROUP

Steven Eldridge

PROPULSION SYSTEM

In designing the propulsion system for the orbital craft, several performance requirements had to be met. It was of primary importance that the system be sufficient to provide the necessary thrust to obtain orbital velocity and to facilitate orbital transfers. A second requirement was that the vehicle be equipped with control thrusters to allow for translation and rotation in all axes. Encompassing all other requirements was the stipulation that the overall system weight, including propellants, be as low as possible.

The following is a list of the specific initial requirements relating to the propulsion system:

1) OVERALL CRAFT WEIGHT W/PAYLOAD = 300,000 LB.
2) RELEASE VELOCITY = 6000 FT/SEC.
3) RELEASE ALTITUDE = 100,000 FT.
4) VELOCITY CHANGE (DELTA Vo) FOR ORBIT = 20,000 FT/SEC.
5) VELOCITY CHANGE (DELTA Vt) FOR ORBITAL TRANSFERS = 820.2 FT/SEC.
6) MAXIMUM TOTAL FUEL WEIGHT = 225,000 LB.
   (Fuel weight = .75 * Total weight)

The first step taken in the design of the system was to research the available propellants and to choose that primary fuel/oxidizer combination that would best meet the stated requirements. After examining several possible combinations, it was decided to use liquid hydrogen as the fuel and liquid oxygen as the oxidizer. This choice was made on the basis of several factors, including the high obtainable specific impulse and proven performance of this combination in existing vehicles. Other advantages are the wide availability and relatively low cost of production of both Hydrogen and Oxygen. A third factor in the propellant selection was the abundance of performance data available on Hydrogen-Oxygen engines for use in subsequent design calculations.

Having selected the propellants it was then possible to proceed with the design of the system components. It was decided that the areas to be examined could be placed into the following categories:

1) Storage chambers for fuel.
2) Propellant feed mechanisms.
3) Thrust Chambers, Injectors, Nozzles.
4) Piping to transfer liquids.
5) Structure to transmit thrust forces.
6) Power source(s) to run feed mechanisms.
7) Control devices to regulate propellant flows.

An analysis of each of the above components was subsequently performed and the results utilized in determining the overall engine configuration for the vehicle. Certain components, such as flow controllers, power sources, and piping, were not examined in great detail due to the wide availability of such components.

MAIN ENGINE:

The function of the main engine in this vehicle is to provide the necessary thrust to reach orbital velocity from an initial release velocity of approximately 6000 ft/sec. It was decided that the main engine would not be used for orbital transfers and thus could be designed specifically for high-thrust, long duration firing, eliminating a major concern for transient operation and wide throttling requirements. This functional requirement can be met by the Space Shuttle Main Engine(SSME) and this was the first engine configuration examined.

Current performance measures for the SSME include:

1) Isp = 460 sec. (at altitude).

2) Maximum Thrust = 470,000 lb. (at altitude).

3) Maximum Chamber Pressure = 3240 psi.

These values were used to obtain initial estimates for the fuel required to reach orbital velocities, and it was found that the values obtained were close to those desired. A further investigation of recent publications for a similar engine which could be used for comparison proved unsuccessful, as most recent data on Hydrogen-Oxygen engines has been compiled for lower thrust applications. Thus it was decided that the "next-generation SSME" would be specified as the main engine, and that estimates would be made on the engines' performance based upon recent data and expected advances in materials technology in future years. The final engine specifications are presented on the following page.

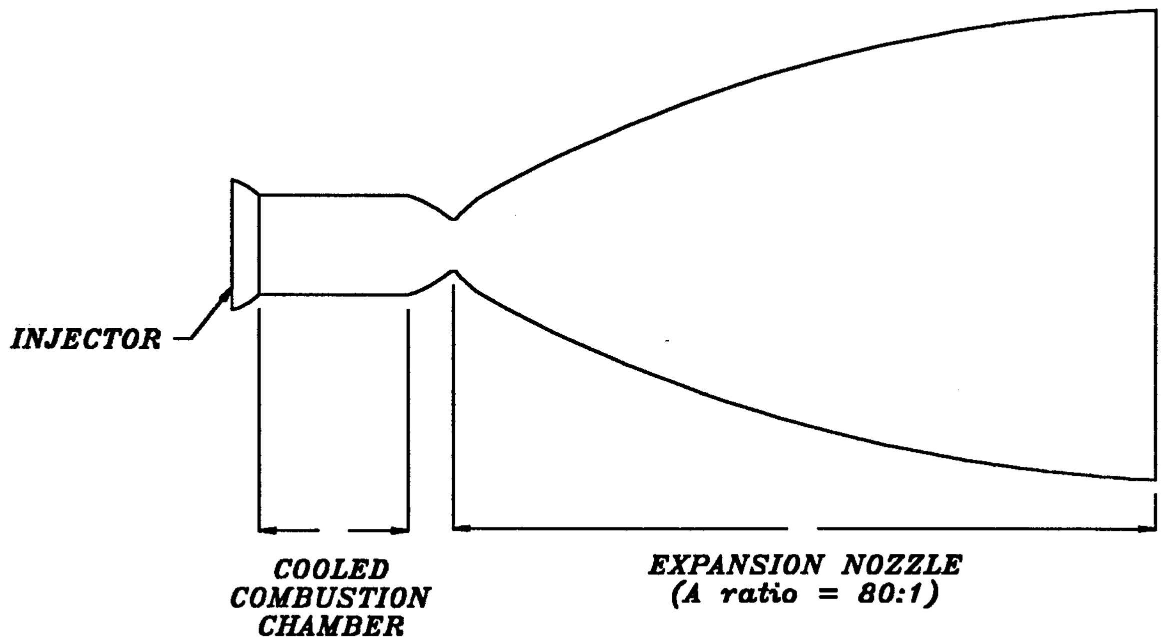
MAIN ENGINE SPECIFICATIONS:

FEED SYSTEM :	FOUR (4) TURBOPUMPS (2 booster and 2 main) TWO (2) PRE-COMBUSTION CHAMBERS (used to drive high pressure turbopumps)

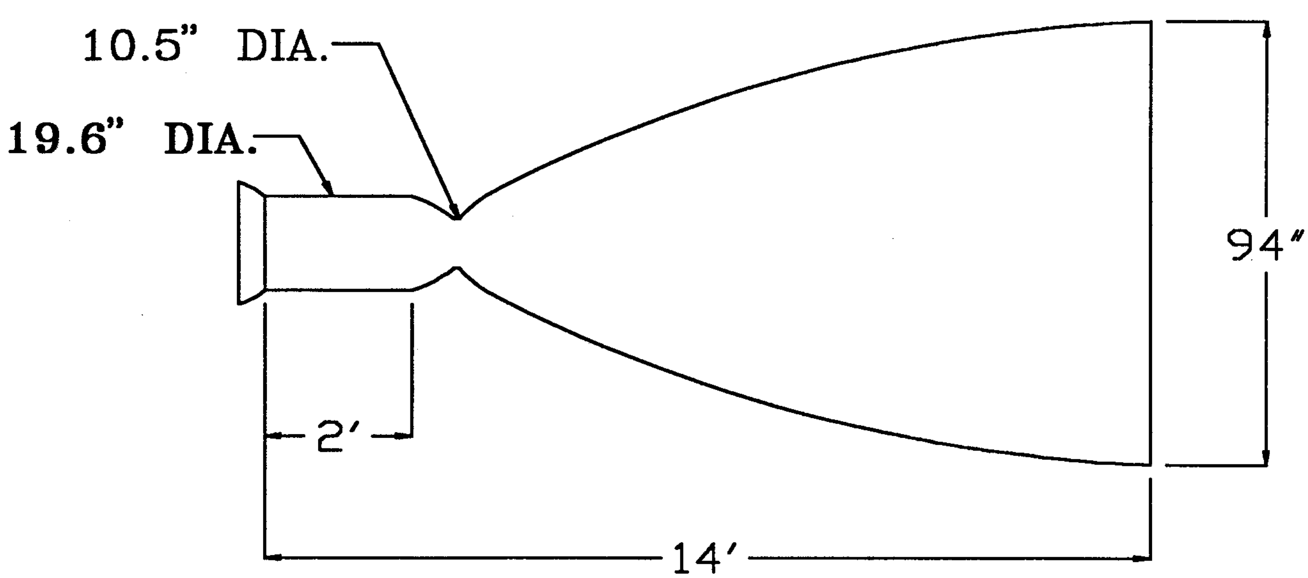
NUMBER OF ENGINES:	1
FUEL:	LIQUID HYDROGEN
OXIDIZER:	LIQUID OXYGEN
THRUST (MAX):	493,000 lb.
Isp:	493 sec.
FLOW RATE (MAX):	1000 lb./sec.
CHAMBER PRESSURE:	3000 PSI (MAX)
WEIGHT:	5362 lb.
OVERALL LENGTH:	168 in.
AREA RATIO:	80:1
THROAT DIAMETER:	10.5 in.
EXIT DIAMETER:	94 in.
COMBUSTOR DIAMETER:	19.6 IN.
COMBUSTOR LENGTH:	24 IN.

COOLING:	REGENERATIVELY COOLED, USING LIQUID HYDROGEN AS COOLANT, IN TWO SEPARATE SECTIONS (One inlet at Aratio = 10 cools upper nozzle section and combustion chamber, and three additional inlets cool remainder of expansion nozzle).
INJECTOR:	IMPINGING STREAM TYPE
IGNITION:	SPARK PLUG
MOUNTING:	RIGIDLY TO BULKHEAD (i.e., no gimbal)

MAIN ENGINE
(SHOWN IN HALF SECTION)



MAIN ENGINE



ORBITAL MANEUVERING ENGINES:

The Orbital Maneuvering Engines(OME's) serve to provide the required thrust for orbital transfers and major maneuvering. Thus the engines must be designed to be restartable and to have controllable thrust vectors for precise maneuvering. Based on the preliminary configuration of the orbiter fuselage, it was decided that a pair of engines, mounted on each side of the main engine would be the most effective configuration. An advantage of this type of mounting is that the maneuvering engines need only be gimballed to obtain pitching moments, since yaw moments can be obtained by simply decreasing the thrust from one of the OME's.

Having decided upon an effective mounting scheme, it was necessary to examine the applicable propellants and proceed with the actual engine design. Initially, it appeared that a combination of Monomethyl Hydrazine(MMH), as the fuel, with Nitrogen Tetroxide(N_2O_4), as the oxidizer, would be ideal, due to the storability, hypergolic reaction, and relatively high specific impulse of these propellants. Upon further investigation it was learned that most vehicles using this combination, including the Space Shuttle, had no other fuel, such as hydrogen, aboard. Thus it was decided that
since the Hydrogen and Oxygen tanks were a permanent part of our vehicle, it would be advantageous to design the maneuvering engines to operate with Hydrogen and Oxygen as well, and hence take advantage of the higher obtainable specific impulse and the existing fuel storage tanks.

Having determined the propellants to be utilized, it was then possible to explore the applicable engine configurations. Calculations were performed, using various thrust values in conjunction with an applicable Isp of 490 seconds, to determine an engine size that could provide the necessary velocity change for orbital transfer in a reasonable time allotment. A decision was made to use two engines, having 7500 lbs. of thrust, each, as this thrust level produced an acceptable velocity change in under 3 minutes. It was further decided that it would be advantageous to fire both OME's during initial ascent, in conjunction with the main engine.

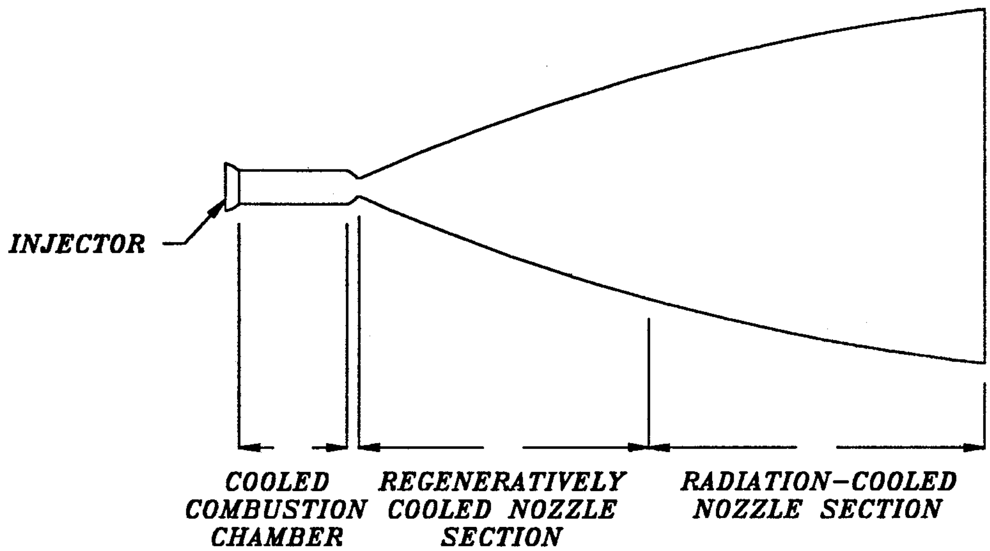
The actual engine design was based primarily on information presented in a paper entitled, Advanced LO2/LH2 Space Engine Characteristics, published in July, 1989, by Chris Erickson and Ron Pauckert, of Rocketdyne. This paper presented extensive data relating to engines having thrust levels below 50,000 lb.. It was first decided that both engines be gimballed, for thrust vector control, and that the exit plane of each be the same as that of the main engine, to avoid external nozzle damage. Since the OME's function essentially as one engine, a single propellant feed system was chosen, with SEPARATE flow control valves for each engine, to facilitate thrust modulation. The final engine specification is on the following page.

ORBITAL MANEUVERING ENGINE SPECIFICATIONS:

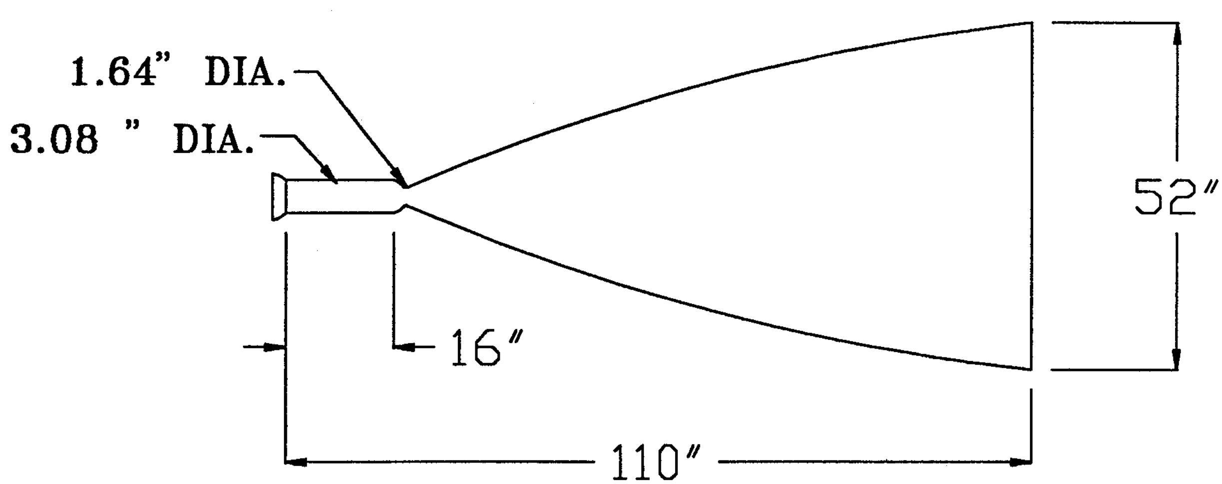
FEED SYSTEM:	FOUR (4) TURBOPUMPS (2 BOOSTER AND 2 MAIN) ONE (1) GAS GENERATOR (used to drive main turbopumps) FLOW VALVES REGULATE PROPELLANT FEED TO EACH ENGINE. NOTE: THIS SYSTEM SUPPLIES BOTH ENGINES.
NUMBER OF ENGINES:	2
FUEL:	LIQUID HYDROGEN
OXIDIZER:	LIQUID OXYGEN
THRUST (MAX):	7500 LB.
Isp:	490 sec.
FLOW RATE (MAX):	15.2 lb./sec.
CHAMBER PRESSURE:	1900 PSI (MAX)
WEIGHT:	266 lb.
OVERALL LENGTH:	110 in.
AREA RATIO:	1000:1
THROAT DIAMETER:	1.64 in.
EXIT DIAMETER:	52 in.
COMBUSTOR DIAMETER:	3.08 in.
COMBUSTOR LENGTH:	16 in.
COOLING:	REGENERATIVELY COOLED, USING LIQUID HYDROGEN AS COOLANT, FROM Aratio=500 TO INJECTOR. RADIATION COOLED FROM Aratio=500 TO EXPANSION NOZZLE EXIT.
INJECTOR:	IMPINGING STREAM TYPE
IGNITION:	SPARK PLUG
MOUNTING:	GIMBALLED MOUNT ON EACH ENGINE.

MANEUVERING ENGINE

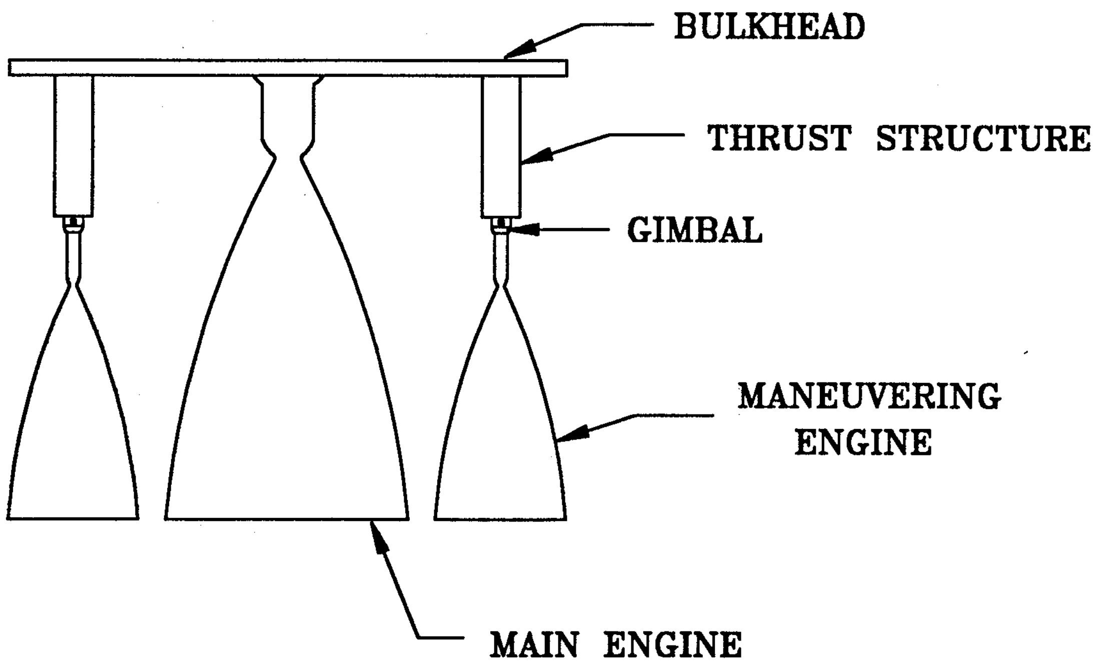
(SHOWN IN HALF SECTION)



MANEUVERING ENGINE



ORBITAL ENGINE MOUNTING



ATTITUDE CONTROL ROCKETS:

The attitude control rockets have the primary function of controlling the orientation of the vehicle while in space. Thus, a main requirement of these rockets is that enough thrust is produced such that maneuvers may be performed within a reasonable time allotment. A second requirement is that each rocket can be restarted thousands of times, and that each burn time be controllable to within a few hundredths of a second, providing precise maneuvering capability.

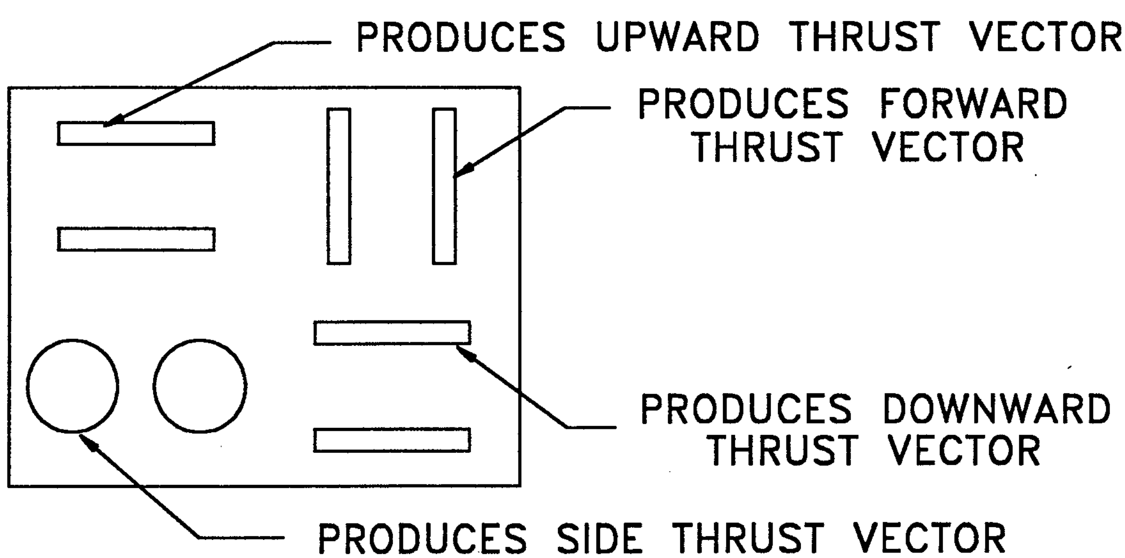
An analysis of the vehicle was first performed in order to determine the minimum number of such thrusters that would facilitate translation and rotation of the craft in all three coordinate axes. It was determined that a minimum of 16 thrusters would be needed, and that a redundant thruster for each required should be included, insuring that no one thruster failure could disable the attitude control system.

The next matter to be examined was the choice of propellant and size of each engine. Since the firing time of each engine can be a fraction of a second, it was decided to use a Monomethyl Hydrazine(MMH) / Nitrogen Tetroxide(N_2O_4) combination, with a gas pressure feed system. It was then determined that a thrust level of 900 lb. per engine would be sufficient for maneuvering, providing 180 degrees of rotation about any axis in less that 10 seconds. The final specifications are now presented.

ATTITUDE CONTROL ROCKET SPECIFICATIONS:

FEED SYSTEM:	GAS PRESSURE FEED SYSTEM USING HELIUM AS PRESSURANT.
NUMBER OF ENGINES:	32
FUEL:	MONOMETHYL HYDRAZINE (MMH)
OXIDIZER:	NITROGEN TETROXIDE (N_2O_4)
THRUST (MAX):	900 lb.
Isp:	300 sec.
FLOW RATE (MAX):	3 lb./sec.
CHAMBER PRESSURE:	1000 PSI (MAX)
WEIGHT:	30 lb. (approximate)
Aratio:	40
MOUNTING:	PLACED IN 4 PODS, HAVING 8 ENGINES EACH, ON EACH SIDE OF THE NOSE AND TRAILING END OF THE FUSELAGE. EACH POD PRODUCES FOUR DIFFERENT THRUST VECTORS (AS SHOWN IN ATTACHED FIGURE).

ATTITUDE CONTROL ROCKET FACE-PLATE CONFIGURATION



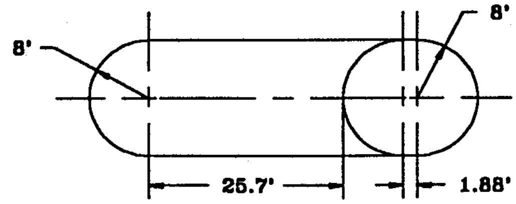
FUEL TANK DESIGN:

The design of the fuel tanks began with a calculation of the total weight and volume of each propellant required for the entire mission. These calculations are included in the appendix and yield the following results:

PROPELLANT	TOTAL WEIGHT	TOTAL VOLUME
HYDROGEN	29,930 lb.	6760 cu. ft.
OXYGEN	179,600 lb.	2524 cu. ft.
MMH	490 lb.	8.94 cu. ft.
N_2O_4	1,010 lb.	11.18 cu. ft.

The tank design was then selected, and the dimensions, operating limits, and weight of each tank was calculated. The tank diameter was limited to 16 ft. since the fuselage O.D. is 18 ft. and an annular space is required for the vehicle structure and tank insulation. It was decided to use a tandem tank configuration for the Hydrogen and Oxygen tanks in which the two share a common bulkhead, shown in structural views of the craft, for the most efficient use of the available space. Since the MMH and N_2O_4 volumes were sufficiently small, a spherical tank was designed for each. An external helium tank, chosen to hold the pressurant for the MMH/N_2O_4 gas pressure delivery system, is also utilized to pressurize the Hydrogen and Oxygen tanks. The specifications for each tank are included on the following page.

FUEL TANK CONFIGURATION



TANK SPECIFICATIONS:

MATERIAL:	ALUMINUM ALLOY 2014 T6
	DENSITY = 174.53 lb./cubic ft.
	YIELD STRENGTH = 60 ksi

<u>HYDROGEN AND OXYGEN TANKS</u>

H_2 TANK WEIGHT:	3811 lb.
O_2 TANK WEIGHT:	<u>1634 lb.</u>
TOTAL WEIGHT:	**5445 lb.**
OVERALL LENGTH:	52 ft.
DIAMETER:	16 ft.
WALL THICKNESS:	.125 in.
PRESSURE (MAX):	39 PSI (factor of safety = 2)
CONFIGURATION:	CYLINDRICAL TANKS WITH SPHERICAL ENDS, IN TANDEM ARRANGEMENT.

MMH AND N_2O_4 TANK

MMH TANK WEIGHT:	49.5 lb.
N_2O_4 TANK WEIGHT:	42.6 lb.
TOTAL WEIGHT:	**92.1 lb.**

MMH TANK DIAMETER:	2.8 ft.
N_2O_4 TANK DIAMETER:	2.6 ft.
WALL THICKNESS:	.278 in.

PRESSURE (MAX): 1000 PSI (factor of safety = 2)

CONFIGURATION: SPHERICAL

PROPULSION

APPENDIX

PROPULSION SYSTEM CALCULATIONS

variables: Av - Vehicle Accelertion
Fm - Main Engine Fuel Flow (lb/sec.)
Fo - Maneuvering Engine Fuel Flow (lb/sec.)
Fa - Attitude Control Rocket Fuel Flow (lb/sec.)
g - Gravitational Acceleration
Isp - Specific Impulse
Mv - Vehicle Mass
t - time (sec.)
Tm - Main Engine Thrust (lb)
To - Manuvering Engine Thrust
Ta - Attitude Control Rocket Thrust
Vo - Velocity change to reach orbit
Vt - Velocity change for orbital transfer
Wv - Vehicle Weight
Wa - Attitude Control Rocket Weight
Wm - Main Engine Weight
Wo - Maneuvering Engine Weight

FUEL CONSUMPTION (INITIAL ASCENT)

Basic Equations: Sum of forces = Mv * Av
Fuel consumed = (Fm + Fo) * t
Vo = Av * t

Given Values: Tm = 493,000
To = 2 * 7500 = 15,000
Fm = 1000 1b/sec
Fo = 2 * 15.2 lb/sec = 30.4 lb/sec
Wv = 300,000
Vo = 20,000 ft/sec

Calculation: Note: The main engine and both OME's are fired during initial ascent.

sum of forces = T - W = Mv * Av

T = Tm + To = 508,000 lb

W = 300,000 - Fm*t - Fo*t
W = 300,000 - 1030.4 * t

Mv = W/g = (300,000 - 1030.4t)/ 32.2

Av = (T- W) / m
Av = (208,000 + 1030.4*t)* 32.2 / (300,000 - 1030.4 *t)

Vo = Av * t = 20,245

(208,000t+1030.4*t**2)*32.2 = 20,245*(300,000-1030.4*t)
6.698e6 * t + 33179 * t**2 = 6.0735e9 - 2.086 e7 * t
33179 * t**2 + 2.776e7 * t - 6.0735e9 = 0

t = 180 sec.

Fuel consumed =	1030.4 * t =	185,560 lb.
	5% reserve	9,280 lb
	5% for drag	9,280 lb
	TOTAL	**204,120 lb.**

FUEL CONSUMPTION (ORBITAL TRANSFERS)

Basic Equations: Sum of forces = Mv * Av
Fuel Consumed = Fo * t
Vt = Av * t

Given values: Fo = 15.2 lb/sec.
Vt = 820.2 ft/sec
Wv = 100,000 lb
To = 7500 * 2 = 15,000 lb

Calculation: Note: The main engine is not fired during orbital transfer.

sum of forces = To = 15000

Mv = (100,000 - Fo * t) / g = (100,000 - 30.4*t) / 32.2

Av = To / Mv
Av = 15,000*32.2 / (100,000 - 30.4*t)
Av = 483,000 / (100,000 - 30.4*t)

Vt = Av * t = 820.2 ft/sec

483,000*t = 820.2*(100,000 - 30.4*t)
507,934*t = 8.202e7

t = 161.5 sec.

Fuel consumed =	Fo * t = 30.4 * t =	4910 lb
	10% reserve	491 lb
	Fuel Consumed	**5401 lb**

FUEL CONSUMPTION (ATTITUDE CONTROL)

flow rate calculation:

fuel: MMH
oxidizer: N_2O_4

mix ratio: 2.05:1

flow rate (max) = 3 lb / sec.
MMH flow rate = .98 lb /sec.
N_2O_4 flow rate = 2.02 lb /sec.

fuel consumption:

A total burn time of 500 seconds is selected for the operation of the attitude control system. This is based upon calculated times to perform various maneuvers, presented later.

$t = 500$ s

Fuel Consumed = 500 * 3 = **1500 lb.**

FUEL VOLUME CALCULATION

variables: DH - HYDROGEN DENSITY (lb. / cu. ft.)
DO - OXYGEN DENSITY (lb. / cu. ft.)
DM - MONOMETYL HYDRAZINE DENSITY (lb. / cu. ft.)
DN - NITROGEN TETROXIDE DENSITY (lb / cu. ft.)
VH - Hydrogen volume (cu. ft.)
VO - Oxygen volume (cu. ft.)
VM - MMH volume (cu. ft.)
VN - N2O4 volume (cu. ft.)

Hydrogen and Oxygen

Mix Ratio = 6:1 (main and maneuvering engines)
Total Fuel Consumed = 204,120 + 5401 = **209,521 lb**
Oxygen Consumed = 179,600 lb.
Hydrogen Consumed = 29,930 lb.

density values:
DH = 4.43
DO = 71.14

Volumes:
VH = 29,930 / 4.43 = **6756 cu. ft.**
VO = 179,600 / 71.14 = **2524 cu. ft.**

MMH AND N2O4 VOLUMES

Mix Ratio = 2.05:1
Total Fuel Consumed = **1500 lb.**
N2O4 Consumed = 1010 lb.
MMH Consumed = 490 lb

density values:
DM = 54.84
DN = 90.30

Volumes:
VM = 490 / 54.84 = **8.94 cu. ft.**
VN = 1010 / 90.3 = **11.2 cu. ft.**

ENGINE SIZING CALCULATIONS (main and maneuvering)

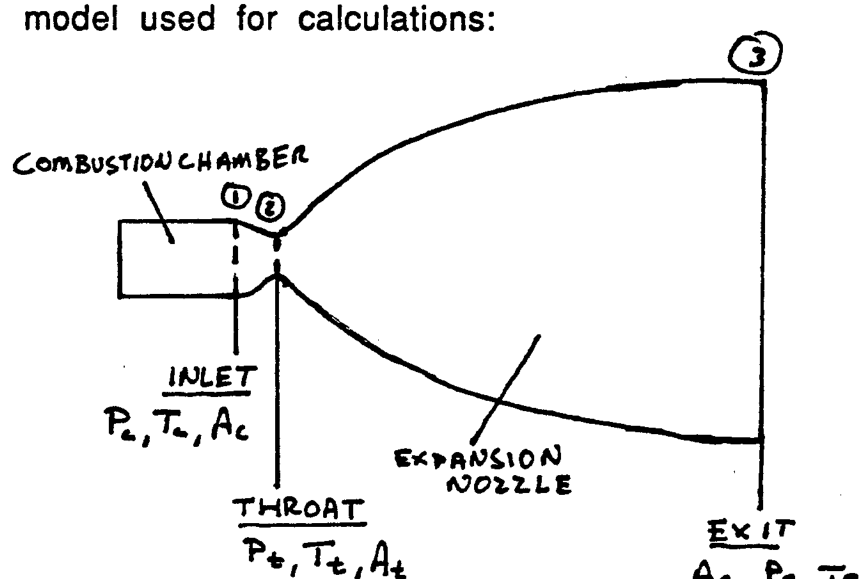
variables: Ac = cobustin chamber c.s. area
At = throat area
Ae = exit area
k = ratio of specific heats
mc = inlet mach number (chamber)
M = molecular weight
Pc = combustion chamber pressure
Pt = throat pressure
Po = stagnation pressure
Ru = universal gas constant
R = gas constant
rhoc = inlet density (combustion chamber)
rhot = throat density
rhoo = stagnation density
Tc = inlet temperature (combustion chamber)
Tt = throat temperature
To = stagnation temperature

known values:

design mix ratio = 6

M (H2 + O2) = 17.43 kg / kg•mole
k = 1.26
Ru = 8314.3 J / kg•mole•degree K
R = Ru/M = 477 J / kg•degree K

model used for calculations:



MAIN ENGINE NOZZLE SIZING

design values: Overall length = 168 in.
Ac/At = 3.5 (for minimal Isp loss)
Ae/At = 80
Exit diameter = 94 in. = 238.8 cm
Pc = 3000 psi = 20.68 MPa
Tc = 4500 F = 2755 K

basic equations (isentropic):

1) A1/At =1/m1*[(2/(k+1)* (1 + (k-1)*m1**2/2)]**(k+1)/2*(k-1)

2) To/T = [1 + (k - 1)m**2 /2]**

3) Po/p = [1 + (k - 1)m**2 /2]**(k/(k-1))

4) rhoo/rho = 1 + (k - 1)m**2 /2]**(1/(k-1))

5) P = rho*R*T (ideal gas)

6) mass flow = rho*area*velocity

7) m = V / (k*R*T)**.5

calulations:

using equation 1 with Ac/At = 3.5, k=1.26, gives
mc = .17

using equation 2 with Tc = 2755 k, k=1.26, gives
To = 2765 K

using equation 3 with Pc = 20.68 MPa, k=1.26, gives
Po = 21.06 Mpa

using equation 5 with Po, R, and To known, gives
rhoo = 15.97 kg / m**3

•utilizing the basic equations for the inlet and throat of the nozzle, with the stagnation properties known, gives the following values:

<u>Combustion Chamber (Inlet)</u>

Pc = 20.68 MPa = 3000 psi
Tc = 2755 K = 4500 F
rhoc = 15.74 kg / m**3
dc = 49.9 cm = 19.6 in.
Ac = .1955 m**s

<u>Nozzle Throat</u>

Pt = 11.22 Mpa
Tt = 2447 K
rhot = 9.98 kg/m**3
dt = 26.7cm = 10.5 in.
At = .056 m**s

<u>Nozzle Exit</u>

Pe ≈ 0 (approximating as a vacuum)
de = 238.8 cm. = 94 in.
Ae = 4.469 m**2

engine weight:

equation: Weight = .00766*T + .00033*T*Aratio**.5 +130
(found in <u>Hypersonic Aerospace Sizing Analysis for the Preliminary Design of Aerospace Vehicles</u>)

Thrust(T) = 493,000 lb
Aratio = 80

Weight = 5362 lb.

MANEUVERING ENGINE NOZZLE SIZING

design values:
Pc = 1900 psi = 13.1 MPa
Tc = 4500 F = 2755 K
Ae/At = 1000
Ac/At = 3.5 (for minimal Isp loss)
overall length = 110 in.
de = 52 in.

calculation:

•using the design values in conjunction with the basic equation presented, the following results can be obtained from an isentropic analysis.

Combustion Chamber (Inlet)
Pc = 1900 psi = 13.1 Mpa
Tc = 4500F = 2755 K
rhoc = 9.965 kg/m**3
dc = 7.82 cm. = 3.08 in.
Ac = 7.45 in**2

Nozzle Throat
Pt = 7.38 MPa = 1070 psi
Tt = 2447 K = 3945 F
rhot = 6.32 kg/m**3
dt = 4.17 cm = 1.64 in.
At = 2.11 in**2

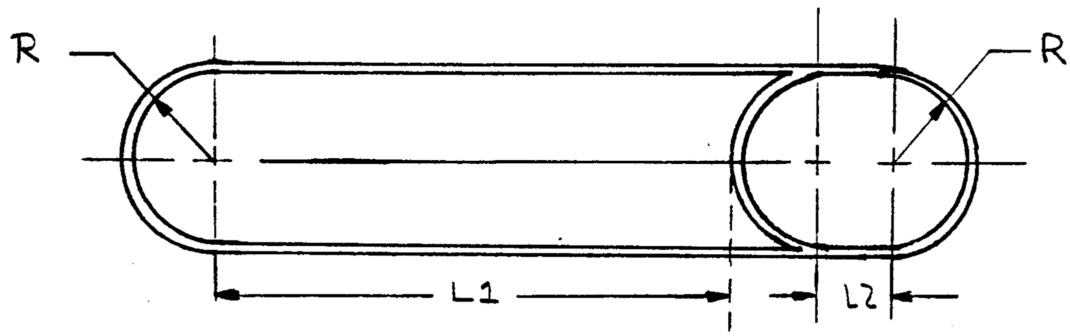
Nozzle Exit
Pe ≈ 0 (approximating as a vacuum)
de = 52 in.
Ae = 2120 in**2

weight calculation:
Thrust (T) = 7500 lb.
Aratio = 1000

Weight = 266 lb. (using same eqn. as for main engine sizing)

HYDROGEN AND OXYGEN TANK CALCULATIONS

Design: Tandem cylindrical tanks with spherical ends.



length calculation:

H2 tank volume (inside) = π * R**3 + π * R**2 * L1
H2 fuel volume = 6756 cu. ft. (calculated)
L1 = 25.7 ft.

O2 tank volume (inside) = 1.333*π*R**3 + π*R**2 *L2
O2 fuel volume = 2524 cu. ft. (calculated)
L2 = 1.88 ft.

material properties:

Aluminum Alloy 2014-T6
Yield Strength = 60 ksi
Density = 174.53 lb. / cu. ft.

material volume:

wall thickness (t) = .125 in.

Mat'l. volume (H2 tank) = 4*π*R**2*t + 2*π*R*L1*t
(for t<<R) = 21.83 cu. ft.

Mat'l volume (O2 tank) = 4*π*R**2*t + 2*π*R*L2*t
= 9.36 cu. ft.

tank weights:

W(H2 tank) = 21.83 * 174.53 = 3811 lb.
W(O2 tank) = 9.36 * 174.53 = <u>1634 lb.</u>

Total tank weight = 5445 lb.

<u>Calculation of allowable pressures</u>

basic pressure vessel equations:

hoop stresses: stress = P * R / t
longitudinal stresses: stress = P * R / 2 * t

P = internal pressure
R = radius
t = wall thickness

Note: since H2 & O2 tanks have hoop and longitudinal stresses present, only hoop stresses need be considered in calculating allowable pressure.

calculation: P = stress * t / R
Pmax = max. stress * t / (R * n)

n = factor of safety
n = 2 (chosen design value)

max. stress = 60 ksi
t = .125 in.
R = 8 ft.

Pmax = 60000 * .125/12 / (8 * 2)
Pmax = 39 psi

MMH & N204 TANK CALCULATIONS

design: spherical tanks with external pressurant

radius calculation:

tank volume (inside) = 1.333 * π * R**3

MMH volume = 8.94 cu. ft. (calculated)
R1 = R (MMH tank) = 1.29 ft.

N2O4 volume = 11.18 cu. ft. (calculated)
R2 = R (N2O4 tank) = 1.39 ft.

material properties:

Aluminum Alloy 2014-T6
yield strength = 60 ksi
density = 174.53 lb / cu. ft.

wall thickness calculation:

basic equation: stress = P * R / (2 * t)

design pressure (Pmax) = 500 psi (max)

t = Pmax * R * n / (2 * max. stress)

n = factor of safety
n = 2 (design value)

N2O4 tank t = 500 * 1.39 * 2 / (2 * 60000)
t = .139 in.

MMH tank t = 500 * 1.29 * 2 / (2 * 60000)
t = .129

Specifying a thickness of .139 in. for each tank insures that the yield stress of the material will not be exceeded.

material volume:

Mat'l. volume (MMH tank) = $4 * \pi * R1^{**}2 * t$
$= 4 * \pi * 1.29^{**}2 * .139/12$
$= .245$ cu. ft.

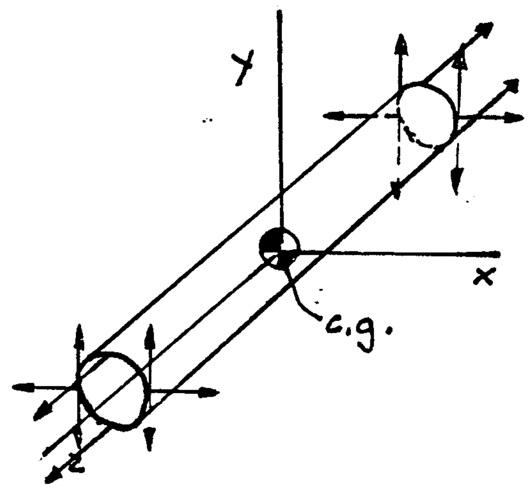
Mat'l. volume (N2O4 tank) = $4 * \pi * R2^{**}2 * t$
$= 4 * \pi * 1.39^{**}2 * .139/12$
$= .284$ cu. ft.

tank weights:

W (MMH tank) =	.245 * 174.53 =	**42.6 lb.**
W (N2O4 tank) =	.284 * 174.53 =	**49.5 lb.**
Total Weight =		**92.1 lb.**

CALCULATION OF ATTITUDE CONTROL MANEUVERING TIMES

model used for calculations:



given values:

c.g. location (payload) = 51.4 ft back from nose
c.g. location (no payload) = 58.5 ft back from nose
overall length = 112 ft.
mass of craft (payload) = 3105.6 lb*sec**2/ft
mass of craft (no payload) = 2484.5 lb*sec**2/ft
Thrust = 900 lb. (per engine)

basic equations:

sum of moments about c.g. = I*alpha
I = moment of inertia about c.g.
alpha = angular acceleration
theta = .5 * alpha * t**2
t = elapsed time
theta = angle of turning

calculations:

Ix = Iy = 3.337e6 lb*ft*sec**2

Iz = 1.258e5 lb*ft*sec**2

rotation about x-axis:

thrusters firing = 2 on each end

alpha(max) = .061 rad/sec**2

time for 180 degree rotation = 14 sec.

rotation about y-axis:

thrusters firing = 1 on each end

alpha (max) = .032 rad/sec**2

time for 180 degree rotation = 20 sec.

rotation about z-axis:

thrusters firing = 2 on each side

alpha (max) = .256 rad/sec**2

time for 180 degree rotation = 7 sec.

NOTE: All of the above calculations are based upon the thrust produced by the required thrusters(16), and hence represent a "worst-case scenario" which would occur in the event that one of the redundant thrusters should fail. Thus, the actual maneuvering times would be less than those calculated, since the thrust level would be twice that assumed.

STRUCTURE GROUP

Cary Brown
Donald Currey

STRUCTURES

INTRODUCTION

-Design Goals-

The goal of this group was to place 20,000 pounds of payload into a low earth orbit from a separation point at 100,000 feet with a velocity of Mach 6. The structures group delt with weight estimates, component configuration, component design, and thermal protection systems.

The most important role of the structures group was to design a structural configuration that satisfies the weight constraints as well as achieving the needs of the aerodynamics and propulsion groups. The structures section had the difficult task of making the craft perform up to the specifications of the other groups. This multitude of data from the other groups required our structural concepts to remain flexible. The design, as expected, was altered many times throughout the course of the semester.

Throughout the design process many assumptions and estimations were made. Initial component weights were largely based on calculations made with the weights estimating program, HASA - Hypersonic Aerospace Sizing Analysis. Weights which could not be determined from this program were taken proportionally from existing crafts or concepts. These included the Space Shuttle, General Dynamics Orbiter, Shuttle II, Hermes, and Sanger.

DISCUSSION

Initial Design Considerations:

The design process began with the configuration of the fuel tanks. The fuel tanks were the largest part of the orbiter, and would influence the design configuration greatly. It was believed that a spherical design for the LOX tank would be the most efficient shape to minimize the structural weight and maximize the strength. The two major tanks where incorporated by using a common bulkhead system. See the Orbiter drawings in the appendix. After enclosing the specified volume of LOX in a spherical tank, a fuselage diameter of 18 feet was chosen.

The supporting frames of the tank separate it from the craft's outer skin and provide room for tank insulation. This integral tank design creates a strong and lightweight structure. This configuration encompasses nearly half of the main body. The initial diameter for the tank was 16'8" which was later reduced to 16 feet to accommodate a reduced fuel requirement. The final tank has a diameter of 16 feet, which leaves 1 foot of clearance between the tank and the inside of the fuselage. The space will be used for insulation for the cryogenic liquids in the tank system, as well as any necessary plumbing or wiring.

The payload bay of the Space Shuttle carries 60,000 lbs. Since the orbiter's payload requirement is 20,000 lbs, its payload volume was set at roughly one third of the Shuttle's. Consequently, the payload bay was 16' wide by 18' long. This will enable the orbiter to carry payloads of the same diameter as the Shuttle's.

The complete fuselage of the orbiter is separated by four major compartments; nose, payload bay, fuel tanks, and engine. Each of these sections is separated from one another by a full bulkhead.

Weights Estimates/Configurations:

The original weight breakdown was from the Hypersonic Aerospace Sizing Analysis program. From this the weights for the fuel tanks, thermal protection system, landing gear, subsystems, wings, vertical tail, and engines were estimated. The weights for the landing gear or subsystems have not been altered since that initial estimation. However the weights for the other components have been calculated explicitly. The propulsion group determined more accurate weights for the fuel tanks by analyzing strength requirements with pressurized fuel. They also established the weight for the overall propulsion system by determining weights of each of its components. The structure group calculated the weights of the other components. The final weight breakdown of the orbiter is as follows:

STRUCTURE:	Wings	4984.6 lbs.
	Wings (TPS)	2740.0
	V-Tail	1500.0
	V-Tail (TPS)	529.0
	Fuselage	10,063.0
	Fuselage (TPS)	9203.0
	Nose Component	3000.0
	Nose (TPS)	2510.0
	Nose Gear	1000.0
	Main Gear	4000.0
	LOX Tank	1634.0
	LH_2 Tank	3811.0
	N_2O_4 Tank	49.5
	MMH Tank	42.6
	TOTAL STRUCTURE	45,066.7
PROPULSION:		
	Main engine	5362.0
	Orbital Engines(2)	532.0
	TOTAL PROPULSION	5894.0
SUBSYSTEMS:		
	Subsystems (Control)	10,000.0
	Subsystems (Hydraulic)	5000.0
	Crew Compartment	4280.0
	TOTAL SUBSYSTEMS	19,280.0
TOTAL EMPTY WEIGHT		70,241.0
PAYLOAD:		20,000.0

FUEL:

	LOX	179,600.0
	LH_2	29,930.0
	N_2O_4	1010.0
	MMH	490.0
	TOTAL FUEL	211,030.0

TOTAL GROSS WEIGHT 301,271 lbs

The CG was calculated using slightly different values for the weights. The CG calculation was done before many of the calculated weights had been established. The position of the CG was determined by summing the moments about the nose. This was done for the separation stage with the craft fully loaded with fuel. The position of the CG for the landing configurations both with and without payload were also determined.

Wing and V-Tail Placement:

Once the wing was designed and its aerodynamic characteristics were determined, it was placed on the Orbiter. The Aerodynamics group concluded that the subsonic aerodynamic center of the wing was to be located 2.5 feet behind the most aft center of gravity of the orbiter. This occurs at landing with no payload or fuel. The AC range extends from 45% to 63.8% of the Mean Aerodynamic Chord. This range is equivalent to 15.77 to 20.29 feet from the leading edge of the wing. The center of gravity of the wing was 19.0 feet from the root leading edge of the wing. The wing was then located on the orbiter so that its AC was 2.5 feet behind the orbiter CG. This corresponded to locating the wing CG 5.73 feet behind the orbiter CG. However this caused a shift of the Orbiter CG, and consequently, the CG to AC offset, due to the additional weight of the wing. The position of the wing was adjusted until the wing AC and Orbiter CG were separated by 2.5 feet.

The original HASA estimation for the surface area of the V-tails was underestimated. To find the proper value the distance from the quarter chord of the wing to the quarter chord of the V-tail was calculated. From this information the aerodynamics group was able to establish the required increase in area, thus a better weight estimate was obtained. This increased the weight from 1000 pounds to an estimated 1500 pounds.

Wing Design:

Total area and shape of the wing was determined by the aerodynamics group. From this it was possible to make estimates of the internal structural layout and the skin thicknesses.

A sparwise lift distribution was found to range from 2080.7 lbs. to 988.7 lbs. at the tip of the wing (see figures in the appendix). These lift calculations were based on area increments from the root to the tip of the wing, using a wing loading of 84 lbs/ft^2. The wing structure weight was estimated to be 4.5 lbs/ft^2 and the total wing weight was determined. Then the net lift was calculated by subtracting the incremental wing weight from the lift. From this, the shear force and bending moment distributions were found.

An ultimate load factor (maximum acceleration plus a factor of safety) of 3.6 was selected. This was chosen due to the maximum deceleration calculated by the aerodynamics group, which was roughly 3 g's. Adding a Factor of Safety of 1.2 gave an ultimate load factor of 3.6.

The aerodynamics group designed the orbiter's airfoil geometry. From this, the maximum allowable spar thickness was determined. An internal structure consisting of three spars was used. See *Wing Structure* drawing in the appendix. Two of these are load bearing and are located at 0.2 and 0.8 chord lengths from the leading edge. The third spar is smaller and non load carrying. It is used for the lateral support of the ribs. The two load bearing spars are located on the lower edge of the wing, while the third is on the top edge. These spars were modeled as I-beams.

By multiplying the bending moments by the ultimate load factor, the ultimate moment was found. The spars were modeled as cantilever beams. By using the equation stress=Mc/I, the moment of inertia for selected spars(I-beams) was found by using the yield strength of Titanium at the maximum moment.

The airfoil distribution allows for a specific maximum spar thickness. This maximum range is 3.03 to 1.35 feet. The space needed for the thermal protection system also needed to be accounted for. A 12 inch high I-beam was used for the leading edge spar which encounters no clearance problems. However, the trailing edge spar had to be tapered from a height of 12 inches to 3 inches at the tip. For this reason, a tip cap was employed which acts as an end rib on the wing that connects the three spars. The leading edge spar has a flange width of 6 inches and a thickness of 0.4375 inch. Its web thickness is 0.25 inch. The trailing edge spar has a flange dimension of 6 inches by 0.3125 inch at the root chord and tapers to 6 inches by 0.3125 inch at the tip. Similarly, the web is 0.25 inch thick and tapers from 12 inches to 3 inches. The center spar has a flange that is 3 inches by 0.25 inch and a web of 6 inches 0.25 inch.

The total spar weights are:

Leading edge	213 lbs (x2)
Trailing edge	121 lbs (x2)
Center	79 lbs (x2)
L. E. Carrythrough	284 lbs
T. E. Carrythrough	232 lbs

The two load bearing spars of the wings are connected by 12 inch I-beam carrythroughs. They have a combined weight of 516 pounds and are fabricated of titanium. The carrythroughs are fastened to the fuselage frame members as well as the wing fillets.

To model the ribs, the cross sectional areas of the wing airfoil at the tip and the root were found. The average of these was used as the average rib size. There are 10 ribs per wing. Each of these is fabricated from a sheet of 1/8 inch thick titanium with 70% of the area cutout for weight reduction. Each rib has a 2 inch wide flange around it to mount the skin to. The weight of the average size rib was found to be 43 pounds, which gives a total of 851 pounds for all 20 ribs. An additional 100 pounds was included for fasteners and other components.

The strucuture of the V-Tails was similar to that of the wings. See*V-Tail Structure* in the appendix.

Fuselage:

Originally it was uncertain of how to design the internal structure of the fuselage. After a complete wing structural component analysis, it was realized that it was possible to make some reasonable assumptions. The framework system that was designed includes circular frame elements which are connected by a network of twelve stringers placed equally around the circumferance of the fuselage. Refer to *Structural Cross-Section* and *Structural Cross-Section (Payload Bay)* in the appendix.

Modelling the fuselage would require finding the moment of the cylinder which is composed of 12 rigidly fixed stringers. The moment of inertia of the individual stringers was neglected, and they were modeled as tension and compression members located in a circle around the center of bending. Then, the sum of the moments from the center of the fuselage (I = Ad^2) was used to find the moment of inertia of the fuselage which is 140,112 in^4. The total bending moment about the center of the fuselage is 10.7x10^6 in-lbs. This gave a tensile and compressive stress of 8.24 ksi for the upper and lower stringers under ultimate load conditions. This means that further weight reduction can be achieved by reducing the size of these stringers.

The frame element was designed to fit the size requirements of the tanks, payload area and outer shell. The frame element consists of a fabricated circular I-beam with a 12 inch high web and 2 inch wide flanges on both I.D. and O.D. sides. Both the web and flanges were constructed of 1/8 inch thick Titanium. To minimize the weight, 24 - 6 inch diameter cutouts, which are located in pairs between the spars, were included. These frame members weigh 192 lbs. each. A 4.5 to 4.75 foot spacing between frame members, depending on bulkhead spacing, was decided upon for the 88 foot fuselage section. This required 16 frame members weighing a total of 3072 lbs.

The frame members are connected by a series of 12 stringers. These I-beam shaped members have a 12 inch high web and 2 inch wide flanges, each of which are 1/8 inch thick. Each 88 foot long stringer section weighs 342 lbs. This gives a total stringer weight of 4104 lbs.

The fuselage contains three bulkheads. Two are fore and aft of the payload bay and the third serves as the thrust structure to which the engines are mounted. The two forward bulkheads consist of titanium I-beams in a criss-cross pattern attatched at the stringer intersection points. See *Structural Cross Section (Fore and Afr Payload Bay Bulkheads)* drawings in the appendix. A 1/16 inch plate of titanium covers the entire bulkhead. Each of these weighs 675 lbs. The thrust structure bulkhead is bulkier in order to transmit the thrust of the engines to the fuselage. It was modeled as two I-beams in a cross pattern which transmit the thrust loads of the main engine (493,000 lbs) and of the orbital engines (7500 lbs each). Refer to *Structural Cross Section (Thrust Structure Bulkheads)* drawing in appendix. From this, the necessary moment of inertia of the I-beams was calculated based on maximum moments generated by the thrust and the yield strength of titanium. A 12" high I-beam with a web thickness of 0.375" and a 12" wide flange of thickness 0.625" satisfied the forementioned condition. This structure weighs 1537 lbs including the two load bearing I-beams, fuselage support ring, and supporting members similar to those on the forward bulkheads. This gives a total fuselage structural weight of 10,063 lbs. This is higher than the initial estimation of 8000 lbs. This, however, was a very good initial estimation.

The thermal protection system for the orbiter is the area which required the most estimating. The group began by choosing an advanced thermal protection system from the paper "Heatshield Design for Transatmospheric Vehicles" by Pitts and Murbech. The TPS selection was based on weight, maximum temperature limit, system simplicity, and available information. The Fiber-Fiber Rigid Composite Insulation (FRCI) system was selected. FRCI is a ceramic composite silica and aluminaborosilicate fiber. This serves as the insulation layer.

A coating of RCG is applied to the FRCI insulation to protect it from aerodynamic stresses. The structural skin of the orbiter was considered as part of the thermal protection system since it is a major factor in the choice and weight of the entire TPS. Polymide Graphite was chosen as the skin of the orbiter. This is currently an experimental material for use in re-entry vehicles. It has an operational temperature limit of over 500 degrees F and is very lightweight. The insulation tiles are glued to the polymide graphite skin with RTV 560, a high temperature adhesive.

The weight of the thermal protection system is the greatest area of uncertainty as far as component weights of the orbiter. The weight of the TPS was estimated by dividing the orbiter into six sections: fuselage top and bottom, wing top and bottom, and V-tail top and bottom. The weight of the TPS covering each of these areas was then found by using the density of each material, thickness of each of the four layers, and surface area. The RCG coating was estimated to be 0.020 inch thick, the RTV 560 glue was .010 inch thick and the polymide graphite skin was 1/16 inch thick. The paper mentioned above gave insulation thicknesses required to maintain maximum operational skin temperatures at various heat loads. Several assumptions about the heating loads on the various surfaces of the craft had to be made because actual data was not available. These assumptions are included in the TPS weight estimating spreadsheet. The thermal protection weight estimates are:

Wing Top	1086 lbs.
Wing Bottom	1654
Fuselage Top	4123
Fuselage Bottom	7591
V-Tail Top (Inside)	200
V-Tail Bottom (Outside)	329
Total TPS Weight	14,983 lb

CONCLUSION

Some of the actual weight estimates differed from the original estimates given by percentages from the General Dynamics Orbiter and other means. These were the weights that were used in the spreadsheet to calculate the Center of Gravity of the Orbiter. This data is still acceptable due to the various ways found to reduce some of the respective component weights; such as reducing the length of the orbiter by as much as 15 feet due to the reduction in the size of the fuel tanks and over estimated crew and payload bay compartments. When the stress on the fuselage structure due to bending moment was finally calculated to be as low as 8.24 ksi, this meant that the stringer size could be reduced and thus the total fuselage structural weight would be reduced.

STRUCTURE

APPENDICES

AT SEPARATION

COMPONENT	WEIGHT (lbs)	X-AXIS (ft)	MOMENT (ft-lbs)
WING	4984.6	64.5	321506.7
WING(THERMAL PROTECTION)	2561	64.5	165184.5
V-TAIL STRUCTURE	1500	105	157500
V-TAIL (THERMAL)	750	105	78750
FUSELAGE	7000	68	476000
FUSELAGE(THERMAL PROTEC)	8000	68	544000
NOSE STRUCTURE	3000	16	48000
NOSE (THERMAL)	2000	8	16000
NOSE GEAR	1000	16	16000
MAIN GEAR	4000	71.5	286000
LOX TANK	1634	52	84968
LH2 TANK	3811	79	301069
N2O4 TANK	49.5	96	4752
MMH TANK	42.6	96	4089.6
MAIN ENGINE	5362	108.5	581777
ORBITAL ENGINES	532	108	57456
PAYLOAD	20000	27	540000
CREW COMPARTMENT	4280	14	59920
LOX FUEL	179600	52	9339200
LH2 FUEL	29930	79	2364470
N2O4	1010	96	96960
MMH	490	96	47040
SUBSYSTEMS (CONTROL)	10000	17	170000
SUBSYSTEMS (HYDRAULIC)	5000	92	460000
TOTAL	296536.7		16220642.8
		C.G. =	54.70

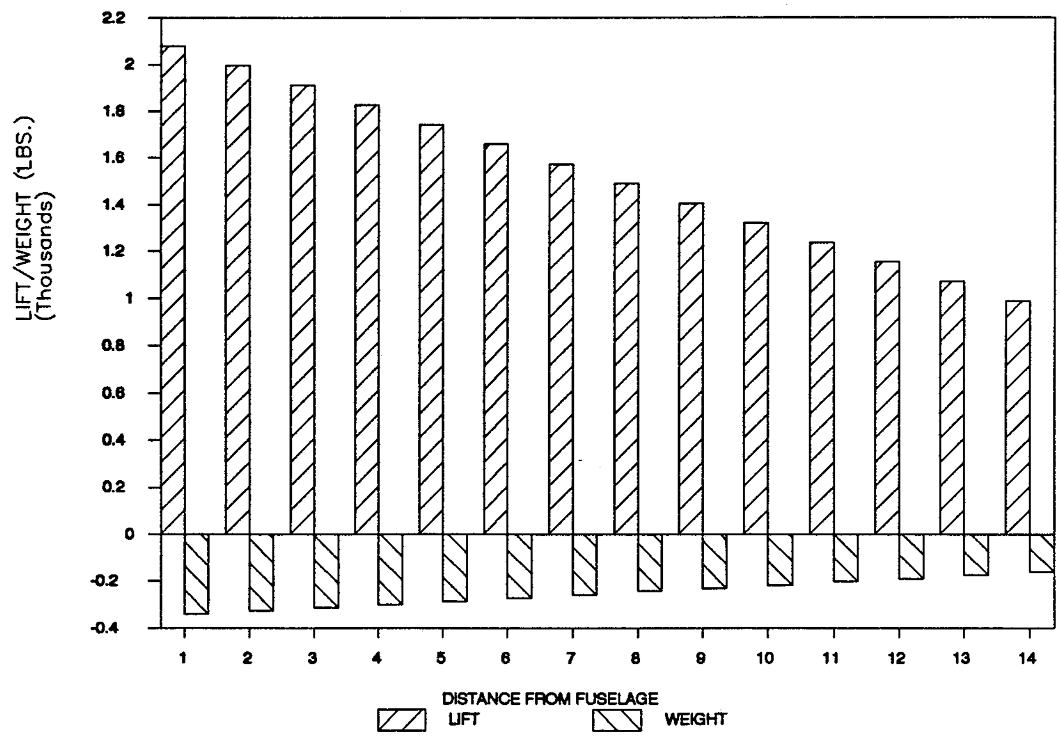
LANDING WITHOUT PAYLOAD

COMPONENT	WEIGHT (lbs)	X-AXIS (ft)	MOMENT (ft-lbs)
WINGS	4984.6	64.5	321506.7
WING(THERMAL PROTECTION)	2561	64.5	165184.5
V-TAIL STRUCTURE	1500	105	157500
V-TAIL(THERMAL PROTECTION)	750	105	78750
FUSELAGE	7000	68	476000
FUSELAGE(THERMAL PROTEC)	8000	68	544000
NOSE STRUCTURE	3000	16	48000
NOSE(THERMAL PROTEC)	2000	8	16000
NOSE GEAR	1000	16	16000
MAIN GEAR	4000	71.5	286000
LOX TANK	1634	52	84968
LH2 TANK	3811	79	301069
N2O4 TANK	49.5	96	4752
MMH TANK	42.6	96	4089.6
MAIN ENGINE	5362	108.5	581777
ORBITAL ENGINES	532	108	57456
PAYLOAD	0	27	0
CREW COMPARTMENT	4280	14	59920
LOX FUEL	0	52	0
LH2 FUEL	0	79	0
N2O4	0	96	0
MMH	0	96	0
SUBSYSTEMS (CONTROL)	10000	17	170000
SUBSYSTEMS (HYDRAULIC)	5000	92	460000
TOTAL	65506.7		3832972.8
		C.G. =	58.51

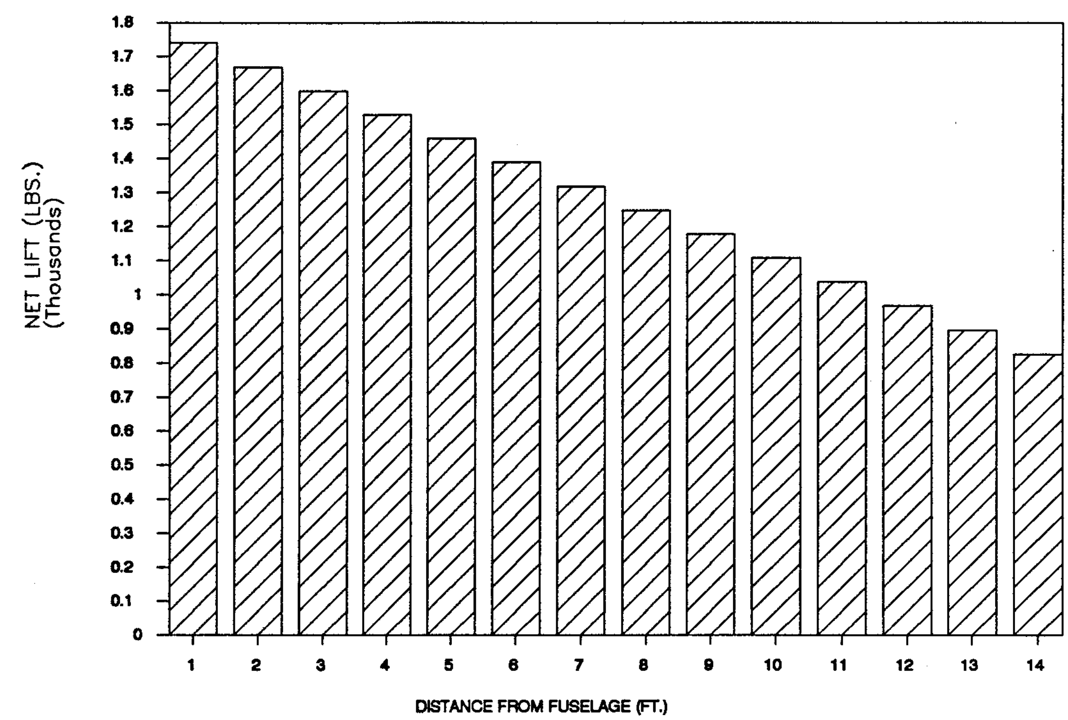
LANDING WITH PAYLOAD

COMPONENT	WEIGHT (lbs)	X-AXIS (ft)	MOMENT (ft-lbs)
WING	4984.6	64.5	321506.7
WING(THERMAL PROTECTION)	2561	64.5	165184.5
V-TAIL STRUCTURE	1500	105	157500
V-TAIL (THERMAL)	750	105	78750
FUSELAGE	7000	68	476000
FUSELAGE(THERMAL PROTEC)	8000	68	544000
NOSE STRUCTURE	3000	16	48000
NOSE (THERMAL)	2000	8	16000
NOSE GEAR	1000	12	12000
MAIN GEAR	4000	71.5	286000
LOX TANK	1634	52	84968
LH2 TANK	3811	79	301069
N2O4 TANK	49.5	96	4752
MMH TANK	42.6	96	4089.6
MAIN ENGINE	5362	108.5	581777
ORBITAL ENGINES	532	108	57456
PAYLOAD	20000	27	540000
CREW COMPARTMENT	4280	14	59920
LOX FUEL	0	52	0
LH2 FUEL	0	79	0
N2O4	0	96	0
MMH	0	96	0
SUBSYSTEMS (CONTROL)	10000	17	170000
SUBSYSTEMS (HYDRAULIC)	5000	92	460000
	85506.7		4368972.8
TOTAL		C.G. =	51.10

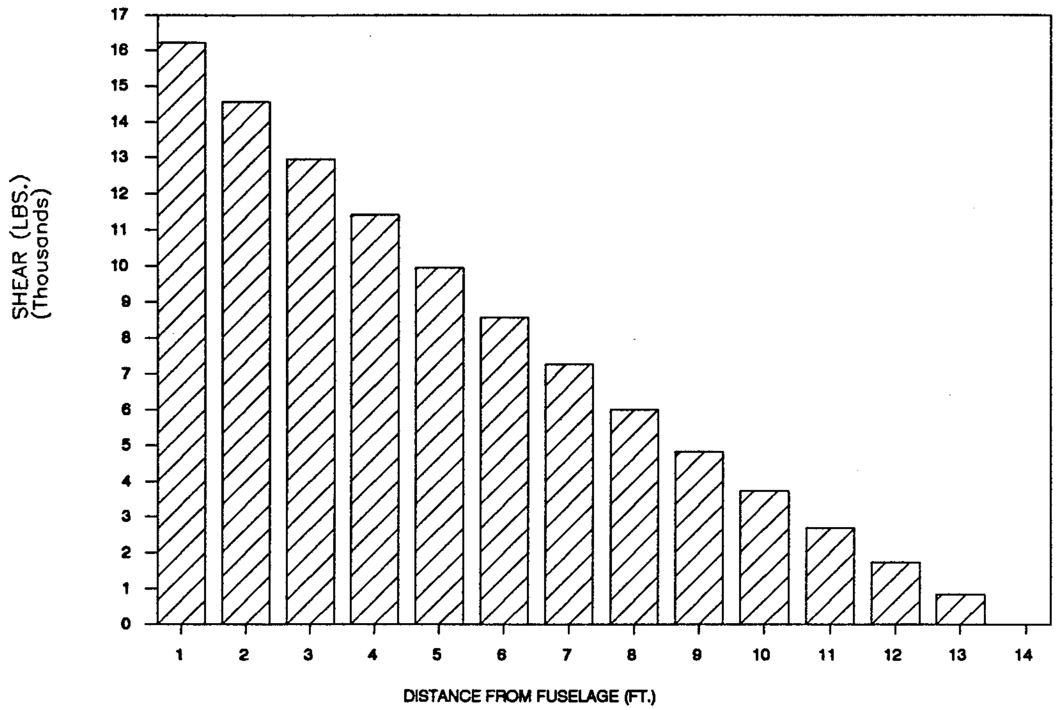
LIFT AND WEIGHT ACROSS WING



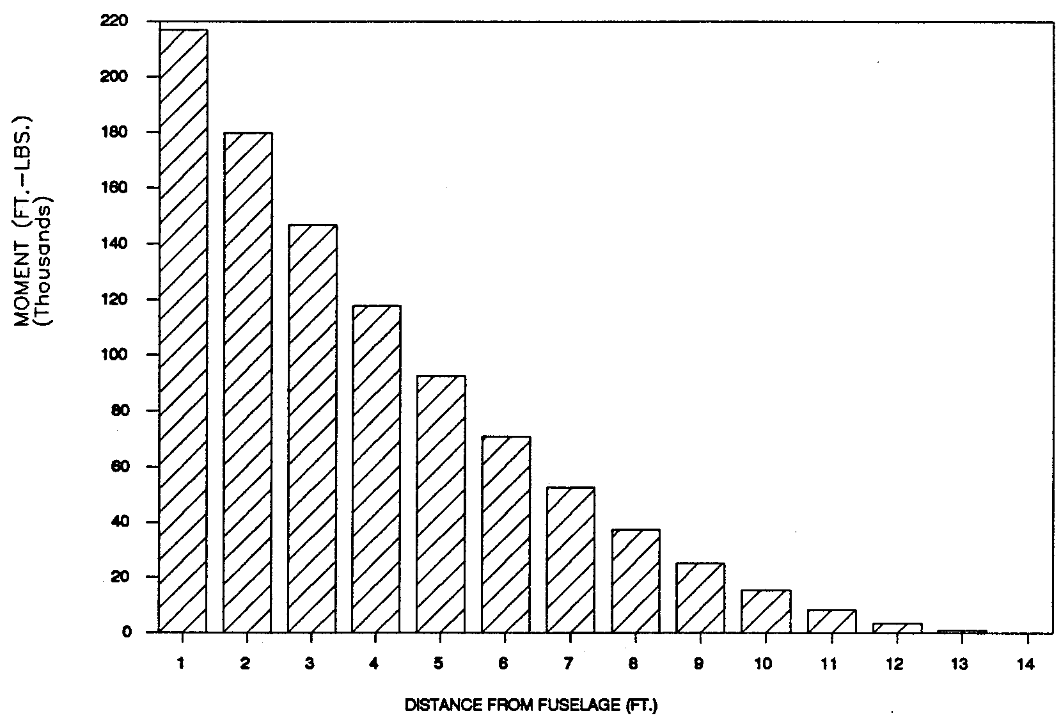
NET LIFT ACROSS WING



SHEAR LIFT ACROSS WING



BENDING MOMENT ACROSS WING



SPAR CLEARANCE

Distance From Fuselage (ft) (FEET)	CHORD	SPAR: LEADING (FEET)	CENTER (FEET)	TRAILING (FEET)
0	25.27	3.03	2.44	1.19
1	24.27	2.91	2.34	1.15
2	23.27	2.79	2.24	1.10
3	22.27	2.67	2.15	1.05
4	21.27	2.55	2.05	1.00
5	20.27	2.43	1.95	0.96
6	19.27	2.31	1.86	0.91
7	18.27	2.19	1.76	0.86
8	17.27	2.07	1.66	0.82
9	16.27	1.95	1.57	0.77
10	15.27	1.83	1.47	0.72
11	14.27	1.71	1.38	0.67
12	13.27	1.59	1.28	0.63
13	12.27	1.47	1.18	0.58

WING WEIGHT CALCULATION

Distance From Fuselage (ft)	Wing Area delta A (ft^2)
0	
1	24.8
2	23.8
3	22.8
4	21.8
5	20.8
6	19.8
7	18.8
8	17.8
9	16.8
10	15.8
11	14.8
12	13.8
13	12.8
14	11.8

Area total 1 side
255.8 sq ft

Area total 2 sides
511.6 sq ft

Wing Spar+Rib =3.729 (lb/ft^2)
Spar&Rib density =0.162 (lb/in^3)
(TITANIUM)

Weight (lb) OF INTERNAL WING	Weight w/skin
92.4	
88.6	
84.9	953.8
81.2	(Lbs)
77.5	
73.7	For two
70.0	1907.6
66.3	(Lbs)
62.5	
58.8	
55.1	
51.3	
47.6	
43.9	

Wing Thermal Protect	
top	1017
bottom	1544
Total	2561

Total Wing (lbs)
3514.8

Total for both wings (lbs)
7029.6

W/CARRYTHROUGH=
7545.6

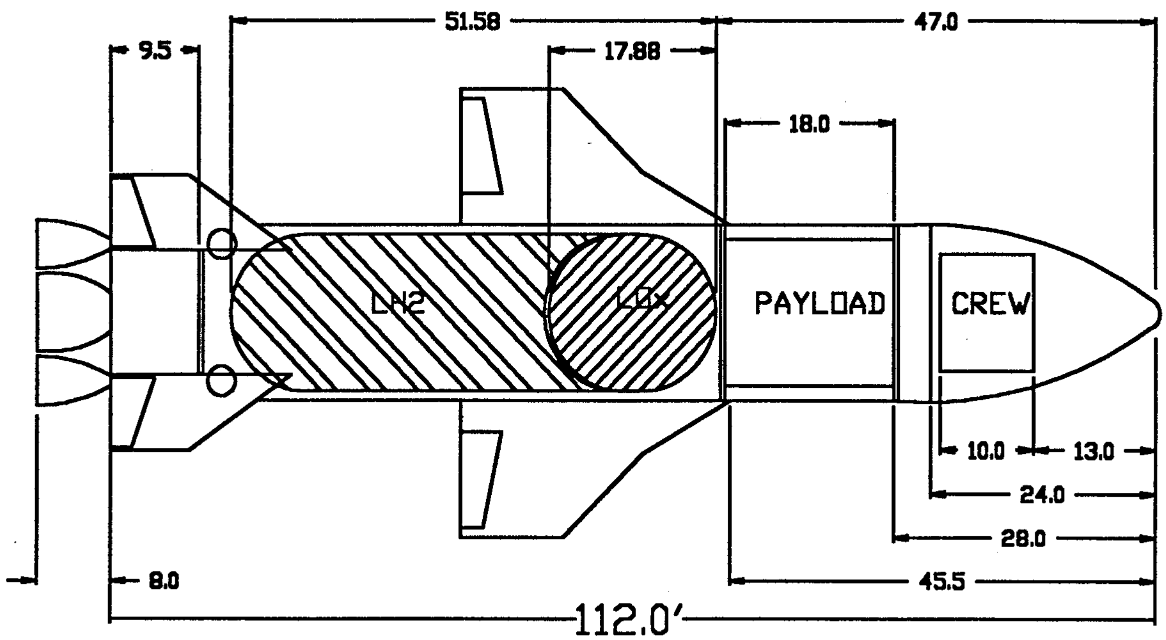
TOTAL WING DENSITY (LBS/FT^2) 13.74

WING LOADING ANALYSIS Wing Loading:

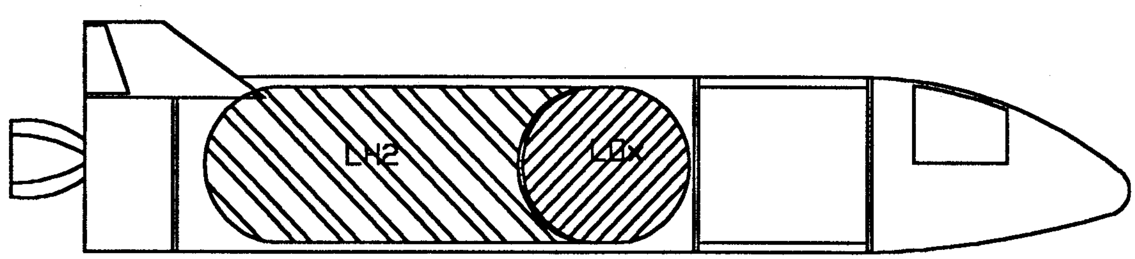
Distance From Fuselage (ft)	Lift (Lbs)	TotalLift (Lbs)	Wing Weight (Lbs)	Net Lift (Lbs)	Tot Net (Lbs)
0					
1	2080.68	21485.5	340.4	1740.3	17970.7
2	1996.68	Per Wing	326.6	1670.0	Per Wing
3	1912.68		312.9	1599.8	
4	1828.68		299.2	1529.5	
5	1744.68		285.4	1459.3	
6	1660.68		271.7	1389.0	35941.4
7	1576.68		257.9	1318.8	Total
8	1492.68		244.2	1248.5	
9	1408.68		230.4	1178.2	
10	1324.68		216.7	1108.0	
11	1240.68		203.0	1037.7	
12	1156.68		189.2	967.5	
13	1072.68		175.5	897.2	
14	988.68		161.7	826.9	

Shear	Moment	ULT. LOAD	ULTIMATE MOMENT	OF
(Lbs)	(lb-ft)	FACTOR =3.6	MOMENT	INTERTIA
			(FOR Ti)	(FOR Al)
16230.4	216932.2	780956.0	351.4	865.1
14560.4	179783.0	647219.0	291.2	716.9
12960.6	146817.1	528541.4	237.8	585.5
11431.1	117823.5	424164.6	190.9	469.8
9971.8	92591.6	333329.6	150.0	369.2
8582.8	70910.5	255277.8	114.9	282.8
7264.0	52569.5	189250.2	85.2	209.6
6015.5	37357.8	134488.2	60.5	149.0
4837.3	25064.7	90233.0	40.6	100.0
3729.3	15479.4	55725.7	25.1	61.7
2691.6	8391.0	30207.5	13.6	33.5
1724.1	3588.8	12919.7	5.8	14.3
826.9	862.1	3103.5	1.4	3.4
0.0	0.0	0.0	0.0	0.0

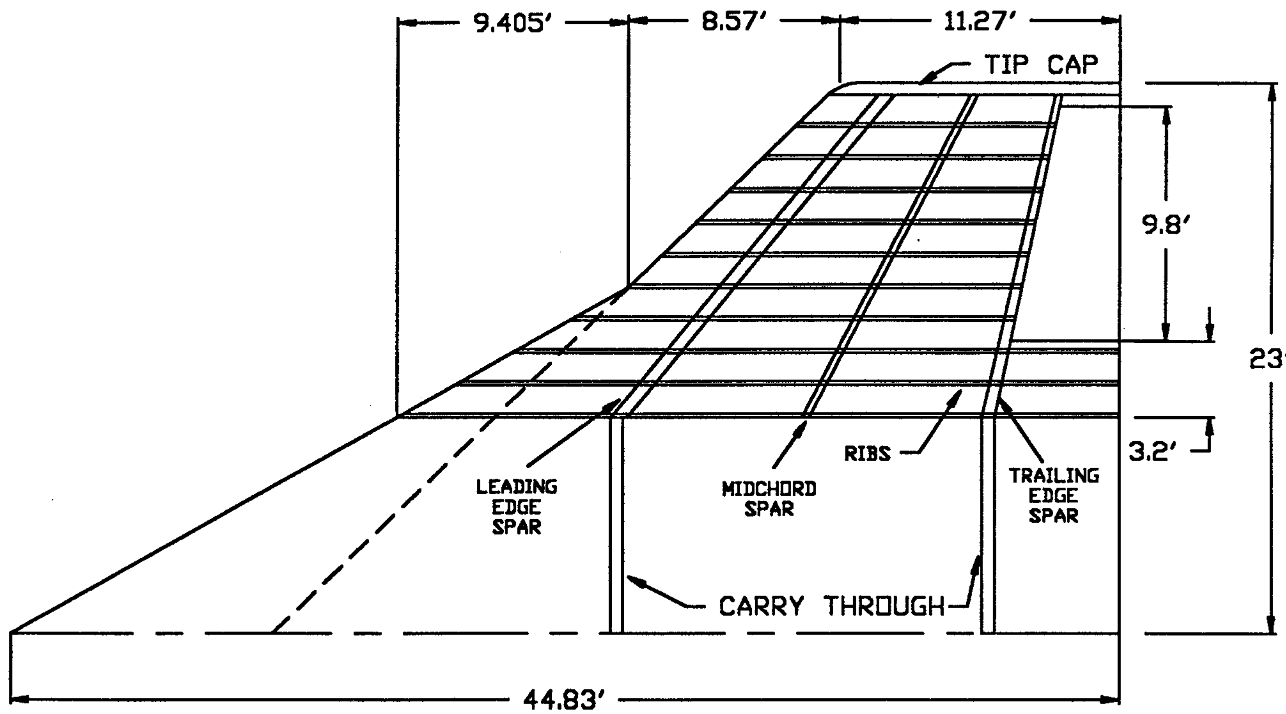
ORBITER TOP-VIEW



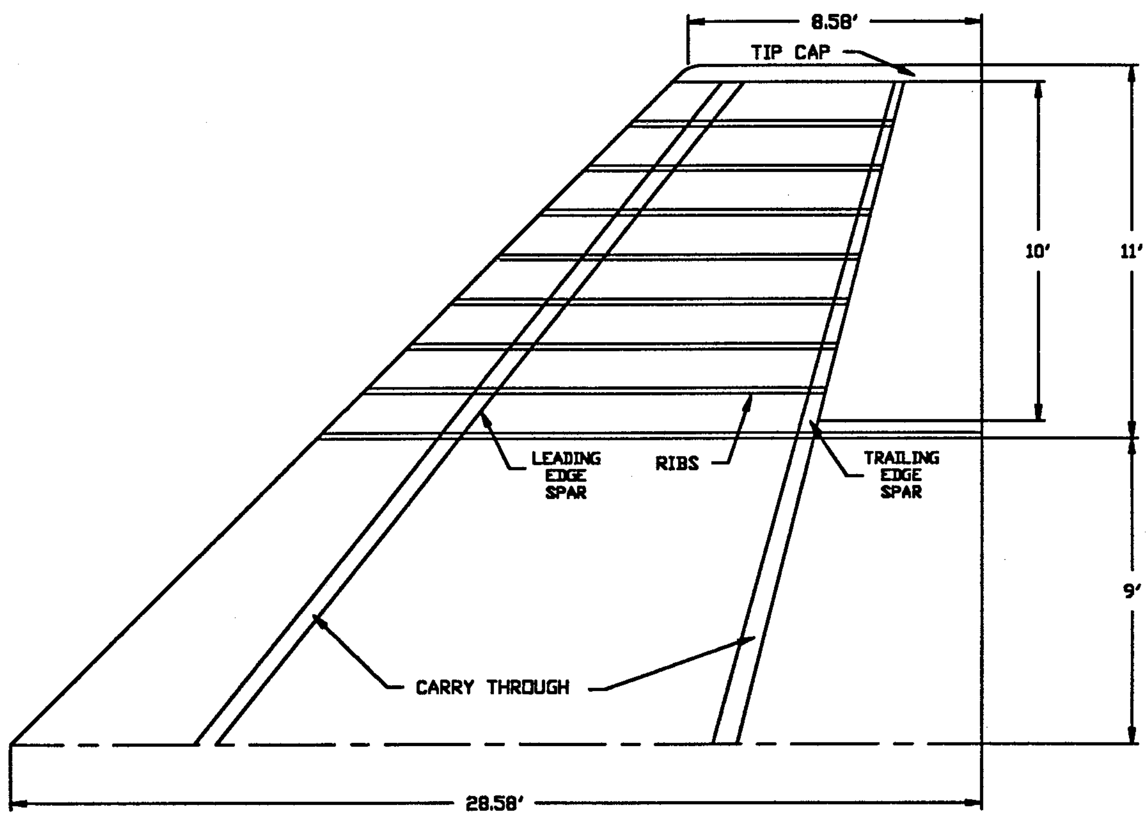
ORBITER SIDE-VIEW



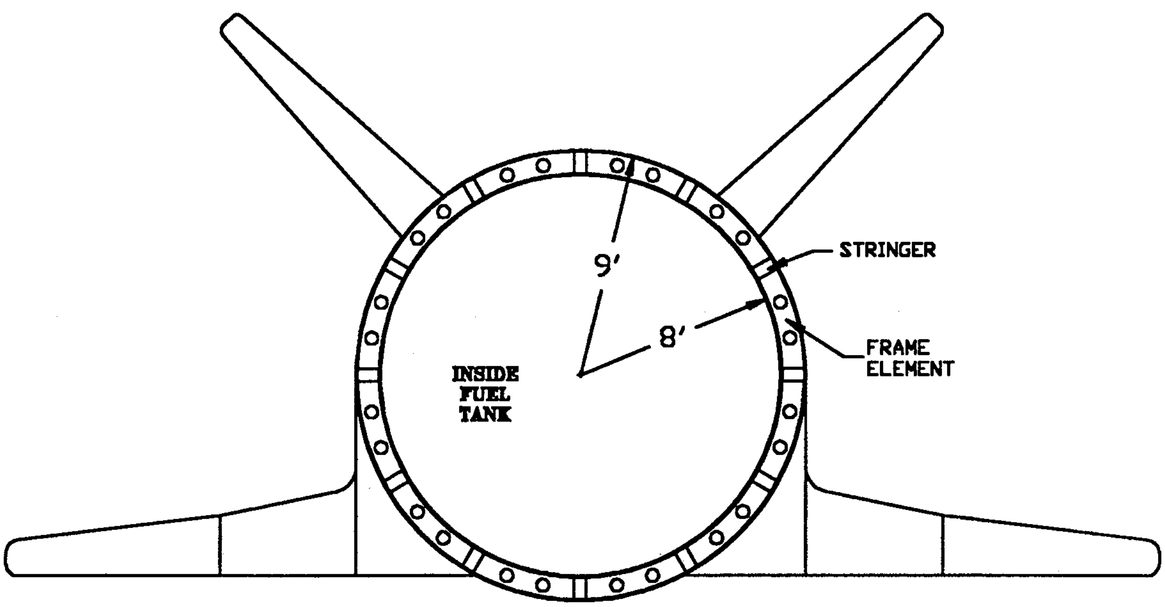
WING STRUCTURE



V-TAIL STRUCTURE

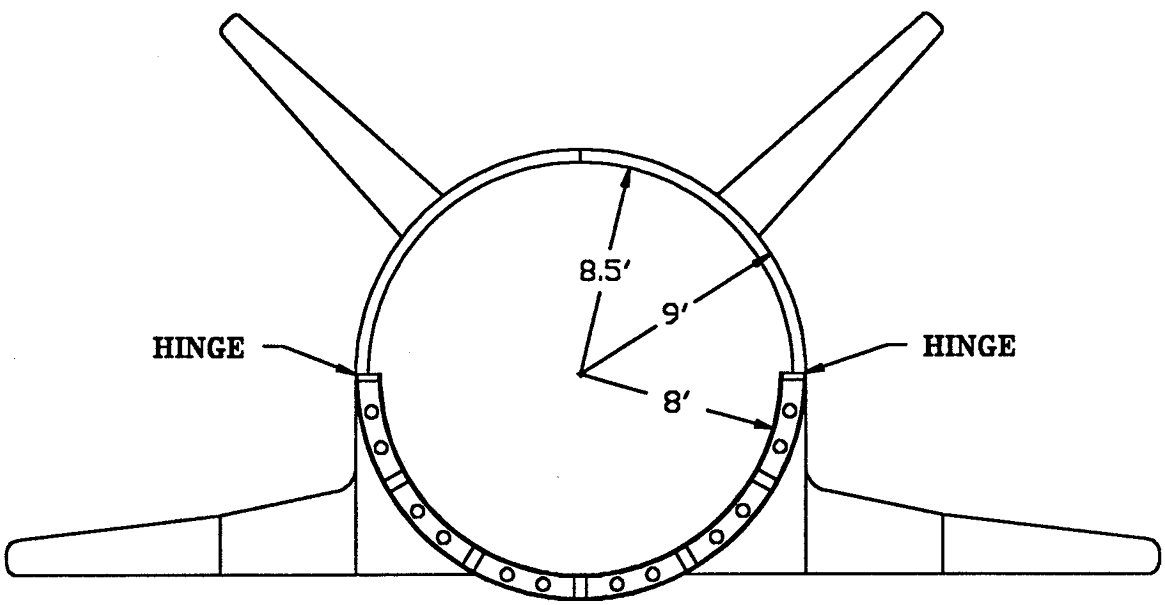


STRUCTURAL CROSS-SECTION



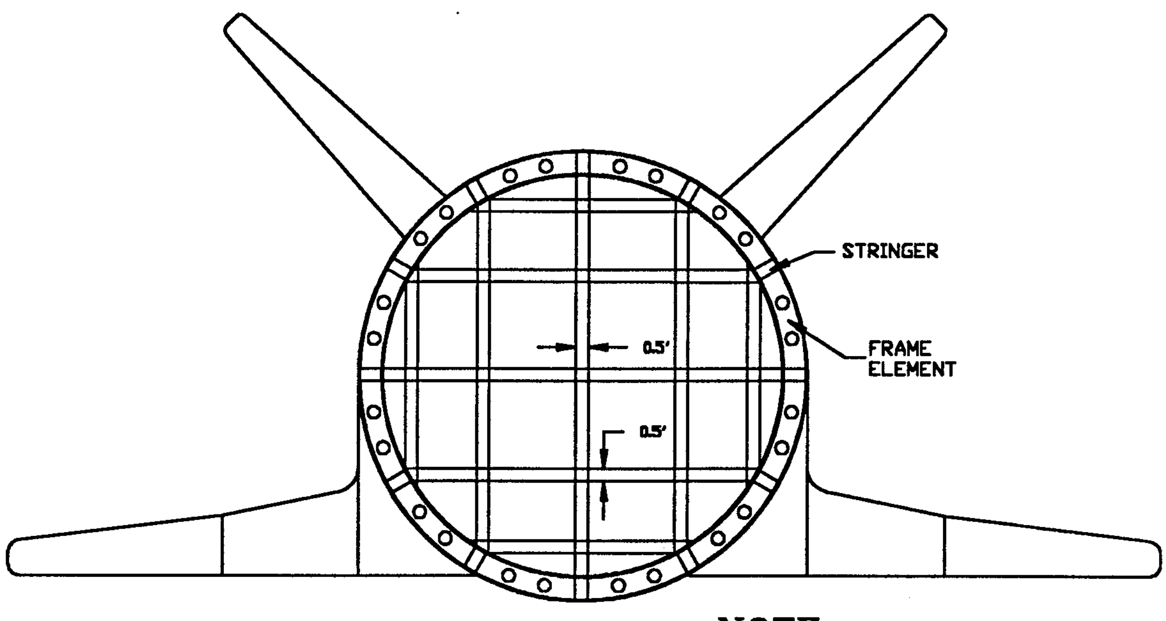
NOTE:
FRAME ELEMENTS BETWEEN X=47' AND X=98.6'

STRUCTURAL CROSS-SECTION



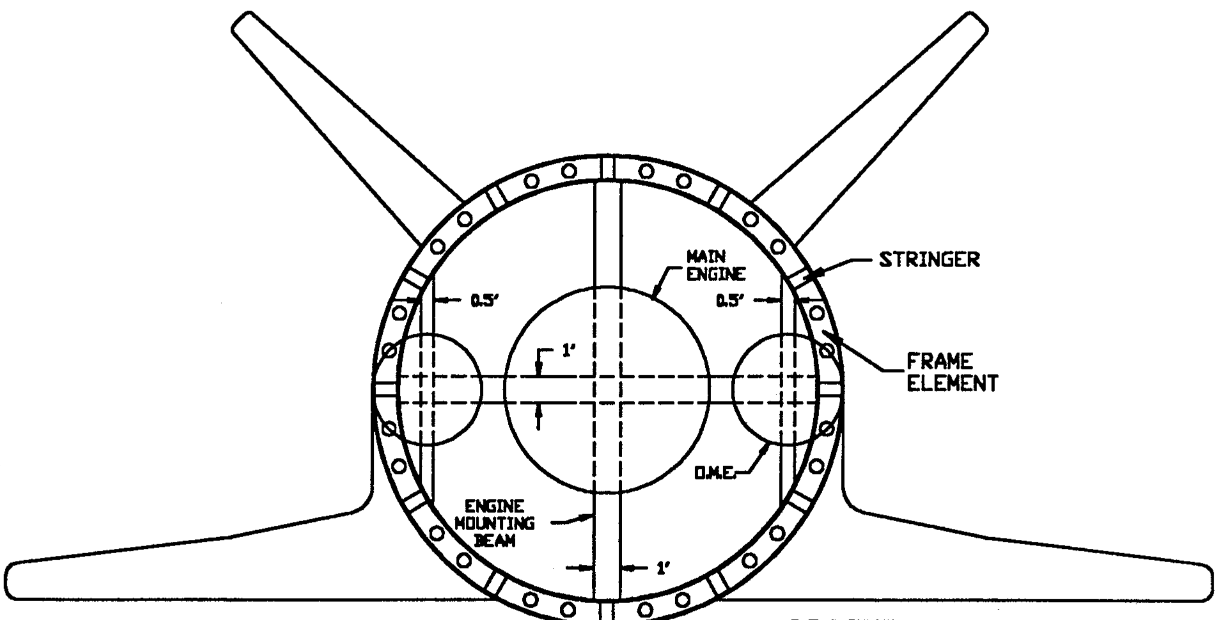
NOTE:
PAYLOAD BAY DOORS
X=28' TO X=46'

STRUCTURAL CROSS-SECTION



NOTE:
FORE AND AFT PAYLOAD BAY BULKHEADS

STRUCTURAL CROSS-SECTION



NOTE:
THRUST STRUCTURE BULKHEAD

ADVANTAGES

This conceptual design has the following three advantages over the Space Shuttle. First, the use of high specific impulse air-breathing engines during its initial ascent reduces the mission's fuel requirements and thus the cost of placing the payload into orbit. Second, unlike the Space Shuttle, this vehicle was designed to be completely reusable; thereby further reducing its cost by eliminating the need for substantial refurbishment after each mission. Third, due to the reduction in turn-around time, NASA's profitability would increase both in absolute and in per unit terms by expanding the capacity to launch more missions per year and reducing the cost of each as well.

Unlike proposed single-stage-to-orbit vehicles, the orbiter of this design does not have the added weight of air-breathing engines to carry into space. This reduces the fuel requirement thereby further reducing the cost of each mission. This conceptual design of a two-stage-to-orbit vehicle appears to be a viable option for the next generation space shuttle.

REFERENCES

Martin, John J., Atmospheric Reentry, Prentice-Hall, Inc., Englewood Cliffs, New Jersey, 1966.

Nelson, Robert C., Flight Stability and Automatic Control, McGraw-Hill, 1989

"Preliminary Sizing Methodology for Hypersonic Vehicles", Armand J. Chapat, General Dynamics, Fort Worth, Texas, AIAA - 87 - 2954.

Raymer, Daniel P., Aircraft Design: A Conceptual Approach, AIAA Education Series, 1989.

Wylen, Gordan J. and Sonntag, Richard E., Fundamentals of Classical Thermodynamics, John Wiley and Sons, New York, NY, 1986.