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THE OHIO STATE UNIVERSITY

STB-White, Final Report

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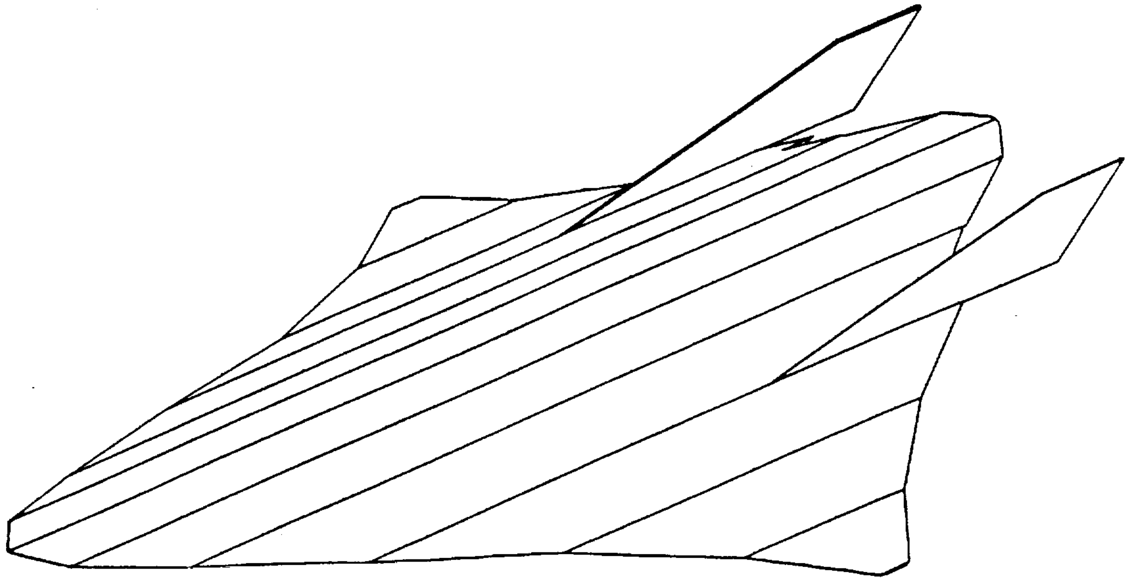

THE OHIO STATE UNIVERSITY

ADVANCED AERONAUTICS DESIGN PROGRAM

STB-White

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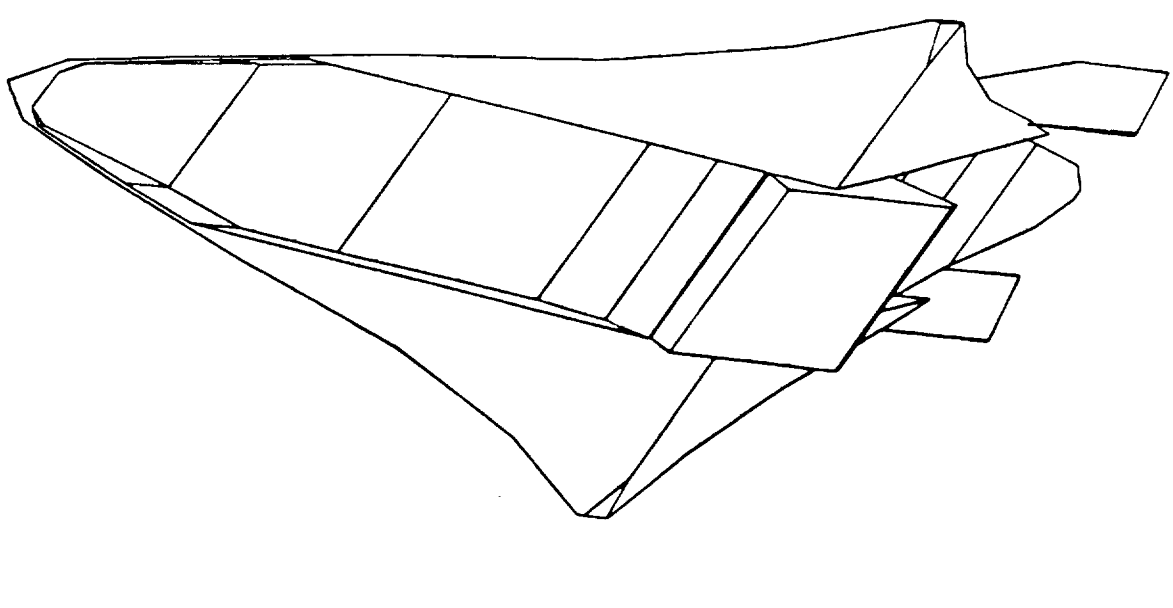

STB - White

Design Team

Final Report

Aero 694, 515H, 416

May 30, 1990

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ABSTRACT

The final design of a hypersonic, SCRAMjet research aircraft, which is to be dropped from a carrier plane, is considered. Topics such as propulsion systems, aerodynamics, component weight analysis, and aircraft design with waverider analyses are stressed with smaller emphasis placed on aircraft systems such as cockpit design and landing gear configurations. Propulsion systems include analysis of the turbofanramjet for acceleration to low hypersonic speed (Mach 6.0) and analysis of the SCRAMjets themselves to carry the aircraft to Mach 10.0. Both analyses include the use of liquid hydrogen as fuel. Inlet design for both propulsion systems is analyzed as well. Aerodynamic properties are found using empirical and theoretical formulas for lift and drag on delta-wing aircraft. The aircraft design involves the integration of all preliminary studies into a modified waverider configuration.

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Compilation of this report was done by John Bryan Roos using WordPerfect 5.1 and Symphony.

ACKNOWLEDGMENTS

For their help with the development of the *STB-White*, the design team would like to thank Dr. G. Gregorek and Robyn Reuss of The Ohio State University. The propulsion group would also like to thank Dr. Hartzell at General Electric for his extensive help in providing much needed data.

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ac - aerodynamic center

AR - aspect ratio

alpha - angle of attack

b - span

c - cord

c.g. - center of gravity

C_{Do} - dimensionless coefficient of profile drag

C_L - dimensionless coefficient of lift

C_{La} - slope of lift vs. angle of attack curve

$C_{L\beta}$ - lateral stability derivative

C_{LP} - damping in roll coefficient

C_{MA} - slope of moment coefficient vs. angle of attack curve

C_{MCG} - dimensionless moment coeff. at the center of gravity

$C_{N\beta}$ - directional stability derivative

ECP - elevon control power

L/D - lift over drag ratio

LH_2 - liquid hydrogen

M - Mach number

psf - pounds per square foot

Q - dynamic pressure

S - reference area

sfc - specific fuel consumption

T - thrust

LIST OF SYMBOLS - CONT.

V - velocity

W - weight

W_l - landing weight

W_{to} - takeoff weight - drop weight

β - side slip angle

1.0 Introduction

For many years the concept of a supersonic combustion ramjet, or SCRAMjet, has been studied. SCRAMjets utilize high speed air flows to compress, burn with fuel, and expand air without the aid of moving parts such as compressors and turbines. The difficulty in designing and running these engines has prevented a SCRAMjet propelled aircraft from being built. Recent research has intensified, and the possibility of a working SCRAMjet is near. Work on the design of an aircraft to carry the SCRAMjet at the hypersonic speeds necessary for these engines to run is also intensifying. The purpose of this report is to present the final design work of a manned, SCRAMjet test aircraft, the *STB-White* (SCRAMjet Test Bed White Group). All aspects that have been analyzed for the last two quarters are included. Each section studied is included with its own discussion and conclusions. The main sections included are aircraft design, propulsion systems, aerodynamics and aircraft systems. The report also includes helpful figures such as 3-view drawings and pie charts at the end of each section to allow for simple study of data.

2.0 Aircraft Design and Development

2.1 Overview of Aircraft Design and Development

The *STB-White* was conceived as a SCRAMjet test bed. Its sole purpose is to carry a prototype SCRAMjet to a high speed, high altitude test run. This was kept in mind in all phases of design. All attempts were made to keep the aircraft small, light, and compact. The design of the aircraft completely centered around carrying the engines and the fuel to power them. Taking a look at Figure 2-1, it can be seen that the engines and the fuel take up 60% of the total weight of the aircraft. The waverider configuration was seen solely as the most efficient way to get to the test run speed and altitude. The testing of a waverider configuration was seen only as a added bonus to the overall design.

2.2 Mission Profile

The mission requirements almost completely define the aircraft's capabilities. The aircraft must be capable of:

- dropped from a French designed carrier plane at Mach 0.8
- climbing and accelerating under its own power to Mach 6.0 and an altitude of approximately 80,000 feet
- carry a prototype SCRAMjet engine that will start a Mach 6.0 that will be able to climb and accelerate to Mach 10.0 and an altitude 100,000 feet, and maintain steady level flight for

two minutes at Mach 10.

- decelerate and land at NASA Kennedy Space Center, Florida.*

* This requirement was added by the *STB-White* design team to ease the problem of supersonic flight over land. The runway at the Kennedy Space Center is near the ocean. This will allow the flight to be completely subsonic over land.

The aircraft meets these requirements.

The current mission profile will be performed in five phases, transport, drop, acceleration, test run, and landing. Phase 1, transport, calls for the French carrier plane to transport the aircraft from Kennedy Space Center, Florida to Maine. In Maine, the aircraft will be fully fueled with LH_2. Phase 2, drop, will have the French carrier plane flying to the drop zone approximately 250 miles east and 100 miles north of Maine. Phase 3, acceleration, will include the acceleration up to Mach 6.0 using the turbofanramjets, and the acceleration to Mach 10.0 using the SCRAMjets. Looking at Figure 2-2, it can be seen that the SCRAMjet ignition is just south of Maine. In the case of the SCRAMjets not igniting the aircraft will be able to make an emergency landing in Maine. Phase 4, test run, will be steady level flight for two minutes at Mach 10.0 and an altitude of 100,000 feet. Phase 5, glide, has the aircraft slowing after the test run. It will approach the coast of Florida at around Mach 1.0, and continue an unpowered glide until it lands at Kennedy Space Center.

2.3 Preliminary Design

The preliminary design, shown in Figure 2-3, was conceived before the preliminary winter quarter presentation in mid-January. It was conceived based purely from concept. Few real calculations had yet been done, no SCRAMjet data was available at the time, little was known about waveriders, and the design was almost completely scrapped in favor of the next design.

2.4 Design Revision 1.0

Design Revision 1.0, shown in Figure 2-4, was a complete rework of the preliminary design presented at the interim winter quarter presentation. This was the first design to show any merit or promise.

The main features of the aircraft is the over-under configuration of the propulsion system, and the waverider configuration. The turbofanramjets had been roughly sized, but little exact SCRAMjet data was available. Engine-out stability was the driving factor for the over-under configuration. Preliminary waverider analysis decided the basic shape of the aircraft. Optimization for Mach 6.0 was decided upon because the high Mach number (8.5 - 10.0) data was not trusted, and the configuration remained relatively constant from Mach 6.0 - 8.0.

The position and type of the landing gear to be used was decided upon. The front gear would be located forward of the pilot and would be a steerable nose wheel. The rear gear, consisting of a pair of sleds, would be located behind the aircraft cg and outside of the SCRAMjets.

A large problem arose from this design - internal space. There was not near the internal space needed to hold the LH_2 fuel.

2.5 Design Revision 2.0

Design Revision 2.0, shown in Figure 2-5, was presented at the final winter quarter presentation. This design solved the liquid hydrogen fuel volume problem.

A complete rework of the waverider analysis had to be done. The length was extend 20 feet from the previous design to bring the total length to 80 feet. The span was also extended 22 feet for a total span of 56 feet. This allowed enough space for the LH_2 needed for the mission. The second analysis of the waverider design produced more realistic data then the first, and it was decided to optimize the aircraft for Mach 10.0. In this set of runs, the waverider configuration remained relatively constant after Mach 8.0.

The optimized waveriders from MAXWARP had somewhat pointed noses. It was felt that a pointed nose would be far to difficult to cool, and it was decided to stay with a blunt nose. Attempts were made to keep all changes in the planform smooth to aid in cooling.

This design also shows the final configuration of the over-under propulsion system. It was decided that two downsized turbofanramjets and five full sized SCRAMjets could provide the necessary propulsion.

2.6 Final Design Configuration

This design was formulated in the first few weeks of Spring Quarter. Over Spring break some flaws were found in Design Revision 2.0. It was almost impossible to fit the turbofanramjets and the SCRAMjets in the over-under configuration that was desired. Another problem was location of the cockpit. Variable geometry was needed for the turbofanramjets, and it would not allow any room for the cockpit. It was also decided to lengthen the aircraft to 85 feet. This length was the maximum allowed by the French carrier plane. The current width was 56 feet, and the maximum allowed by the French carrier plane was 50 feet. The width was narrowed to 48 feet for this reason.

The waverider analysis was redone for the new length. The new geometry allowed enough room for the over-under engine configuration, the fuel, and the cockpit. The final design was set, and final versions of all aspects of aircraft development began. The inlet study was begun, the model was designed and built, and stability and control were done. A 3-D model was created on I-DEAS V4.1 (Figure 2-6), which aided in the internal arrangement of component systems (Figure 2-7).

Table 2-1 shows some of the more important specifications for the aircraft.

Specifications for the *STB-White*

Table 2-1

Length	85 feet
Height	14 feet
Span	48 feet
Planform Area	2100 feet
Wing Sweep	70 degrees
Aspect Ratio	1.1
Wing loading	
At Drop	25.2 lbs/sqft
At Landing	18.6 lbs/sqft
Gross Drop Weight	53,000 lbs
Landing Weight	39,172 lbs
Cruise Altitude	100,000 feet
Cruise Mach Number	10
Range	1400 miles
Landing Velocity	276 ft/sec (188 mph)
Landing Distance	5181 feet
Propulsion	
Turbofanramjet	2 - 20% Downsized General Electric
SCRAMjet	5 - Fullsized General Electric

2.7 WAVERIDER CONFIGURATION

When approaching hypersonic speeds, it is appropriate to have some kind of configuration which will yield as little drag as possible and, hence, more thrust available. For this reason, a partial waverider configuration was chosen for our design. Our analysis was done using the MAXWARP (Maryland AXisymmetric WAveRider Program) program which was setup on the Harris 800 at the Ohio State Aero/Astro Research Laboratory.

In particular, the design was based on flow fields over power-law bodies. The following information was needed in an input file for power-law bodies.

POWER = In which, a power of the power law equation of 0.66 was used to obtain minimum drag characteristics. The power law equation is

$$r = r_{base} * (z/l)^{power}$$

where rbase is the radius at z = 1, z is the axial distance from the nose of the power law body, l is the length of the body, power is the exponent of the power-law equation, and r is the local radius of the power-law body.

YBL = A value of 0.2 was the final number used for the base height/length ratio.

ZPBODY = The length of the power law body was 18.29 meters (60 feet).

ZSL = A value of 0.05 was used as the non-dimensional length of the nosetip to the length of the body ratio.

Through the iteration process, the length and base height were changed several times in order to assure that the volume was large enough to house fuel, engines, and aircraft systems.

To account for different altitudes, the freestream conditions were changed to the appropriate values for various altitudes. The dynamic pressure and temperature were changed for several altitudes between 60,000 and 100,000 feet. Also, the specific heat ratio was changed between 1.36 and 1.4 to account for some disassociation at hypersonic speeds. It was found that changing this value had little affect on the design aspect of the configuration. Another factor in the freestream conditions was the Mach number. Changing this value did have a great affect on the design. For lower Mach numbers, the design was very rough, and the edges were very sharp. As the Mach number increased, the design became smoother, as shown in Figure 2-8. Also, the base height became thicker at lower speeds. This may be due to the increased size in the Mach cone on the body. Our final configuration was taken from the Mach 10.0 case from the waverider, and is shown in Figure 2-9.

Aircraft analysis parameters were specified in the input file consisting of the aircraft length of 60 feet. Analysis was performed on a viscous body with turbulent air flow. The wall temperature was specified at 3000 degrees Rankine.

The aircraft constraints were specified for volume and box size. A minimum volume was set at 5000 cubic feet, and the maximum was set at 7500 cubic feet. Box size is defined as the half span to length ratio. The minimum box size was set at 0.1, and the maximum box size was set at 0.4. These constraints were set to account for the amount of space that was needed for the aircraft fuel and systems. The slenderness ratio was set for a few runs, but the program kicked out these constraints. Therefore, for the final configuration, the slenderness ratio was not used.

After coordinates were obtained from the H800 Super-mini Computer, these files were downloaded to a PC and read into an AutoCad V10.0. Once in AutoCad V10.0, the configuration was rotated and sized to obtain each view.

MAXWARP was used to integrate several variables in order to obtain a design which minimized the drag count. The output conditions calculated from this program were only ideal values and could not theoretically be used as final calculations. This was because the final configuration did not consist solely of the waverider configuration. Powerplants, boat tail, vertical stabilizer, and other details were added to the final configuration. So, the waverider was only used as a basis on which to design the hypersonic vehicle.

2.8 Materials

In order to chose materials, the thermal distribution is essential. The program CDHEAT (Conceptual Design aerodynamic HEATing analysis) finds this distribution. It uses algebraic equations based on empirical data and has a rather simplistic approach. However, for known body shapes and flight conditions it gives accurate results. Heat transfer coefficient ratios for the various body shapes are found and then used in an iterative process to find the heat transfer rates using the following equation:

$$q = h_o \frac{h}{h_o} (T_{aw} - T_w)$$

h_o - *reference coef*

h - *actual coef*

T_{aw} - *adibatic wall temp*

Unfortunately, the code was not completed in time for this report but temperatures at the nose and along the leading edges were approximated. The nose temperature was under 3000 F and the leading edges were roughly 2400 F. This is beneficial since the aircraft probably will not need active cooling. This lack of an active cooling system will also save weight and space. Furthermore, since this vehicle experiences extreme temperatures for only a short time, the life of the surfaces at these temperatures need not be long.

A carbon/carbon composite capable of withstanding temperatures over 3000 F for extended periods will sufficiently protect the nose. Its strength at high temperatures far surpasses any metal, and it can be used along the fuselage. At the leading edge, a refractory alloy of Columbium is effective. It is commercially available with reasonable strength at room temperature. It is also easy to fabricate and weld allowing easy attachment to our desired leading edge shape. Columbium oxidizes rapidly at temperatures over 1200 F. To prevent this, a coating of slurry silicide is fused onto the surface in a vacuum furnace. For the remaining "cool" spots, a titanium alloy can be used.

Use of evacuated insulation for *STB-White's* cryogenic fuel tanks was initially considered, but it was found that this type of tank often develops leaks in the insulation causing severe boil-off. Therefore, a tank with internal insulation is superior. The system consists of a heat shield surrounding a purge gap. The purge gap further surrounds the insulated tank on the inside. The only problem in this design is that LH_2 permeates the insulation raising the thermal conductivity and causing boil-off. To counter this effect, a vapor barrier is installed inside the insulation.

2.9 Subsonic Wind Tunnel Model Construction

At the beginning of Spring Quarter, Dr. Gregorek gave specifications for the subsonic wind tunnel model. A model two feet long made of wood (balsa or pine) that would mount on a 3/8 inch stinger was requested. With these requirements in mind, a model was devised and constructed.

The subsonic wind tunnel model was to be constructed of balsa wood. A 1/4" x 1/4" square spar was constructed. This spar would hold several rib cross-sections. Each rib was cut and placed in a direction perpendicular to the aircraft flight path - with the spar in the flight path direction. Next, 1/8" thick balsa sheets would be conformed to the ribs to make the skin of the aircraft.

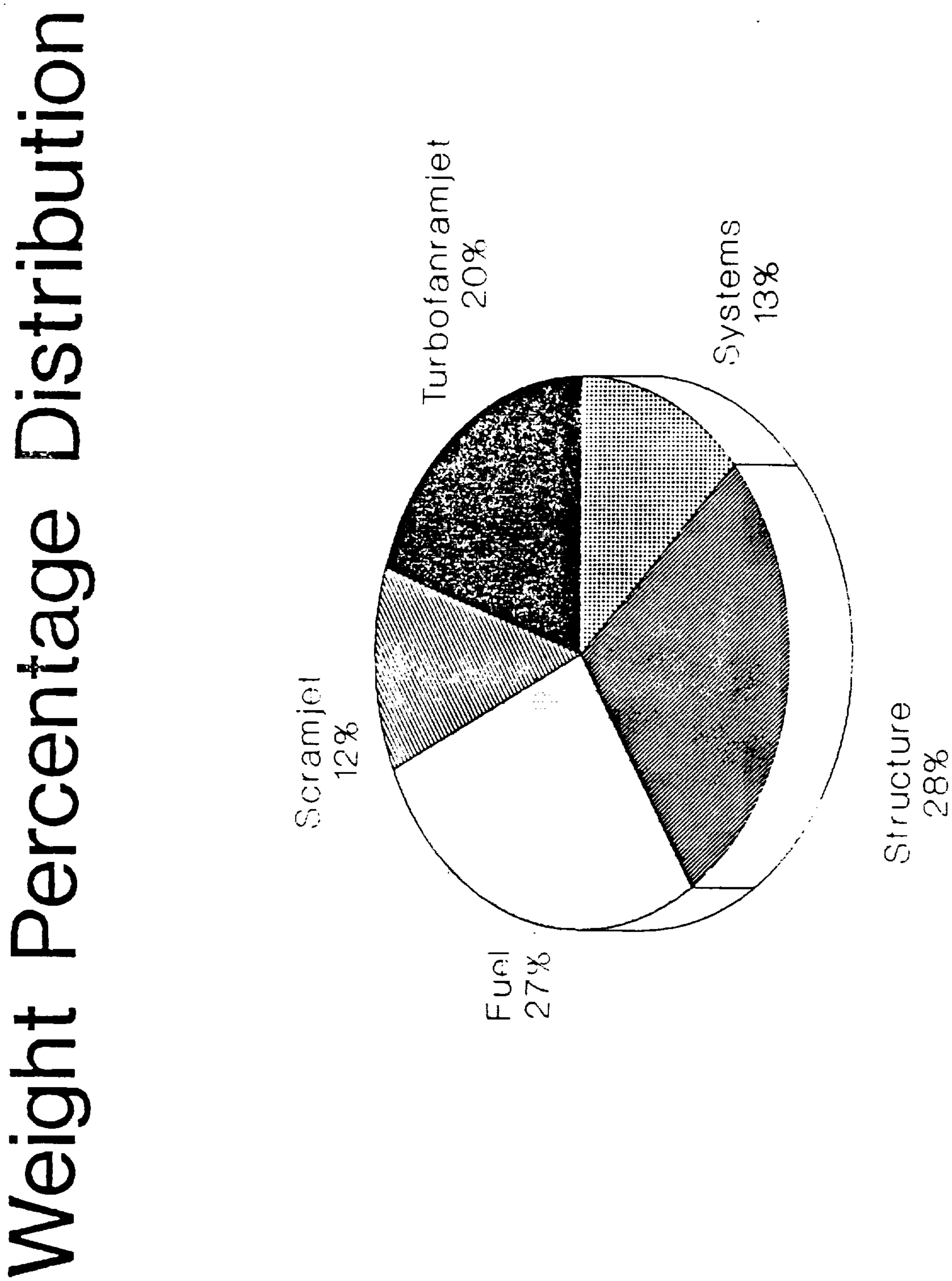
The front half of the aircraft was constructed first (i.e. the waverider portion). The underbody was simply formed by attaching the flat balsa sheets to the rib lower edges. The top of the aircraft was more difficult. The balsa sheet had to be bent to conform to the gently curving ribs that defined the aircraft upper surface. To do this, water was added to the balsa sheets to make them more flexible. This flexible balsa wood was then wrapped over the rib contours and glued to complete the upper surface.

The back half (i.e. the boat-tail) was completed in the same fashion as the front half. The only complication was adding the stinger cavity that must pass through the rear of the model for wind tunnel mounting. This cavity was created by cutting holes in the rib cross-sections and then lining them with sheet metal. Also, it is favorable to affix the stinger at the model center of gravity. This allows for simpler reduction of moment, lift, and drag data obtained from the wind tunnel tests. A hole was drilled through the model upper surface and through the sheet metal-lined cavity to allow a mounting point to the stinger. Next, the balsa sheets were conformed to the rib surfaces as before. Finally, the vertical control surfaces were attached to the model.

Next, an attempt was made to make the model smooth. A micro foam filler was used to fill the balsa grain. The filler was sanded to smoothness with difficulties incurred at wood glue joints. The wood glue sanded more slowly than the balsa wood. The team logo was added to the model at this time by cutting it from a 'sticker'. The letters were glued to the model surface leaving the removable layer attached to the letters. The model was then coated with a clear coat enamel in an attempt to give the paint a less absorbent surface to adhere. Black enamel paint was added in several coats and sanded and repainted to achieve a smooth surface. When painting was complete, the removable paper layer in the logo was removed leaving the logo embedded in the paint.

Weight Percentage Distribution

Figure 2-1



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Flight Path

Figure 2-2

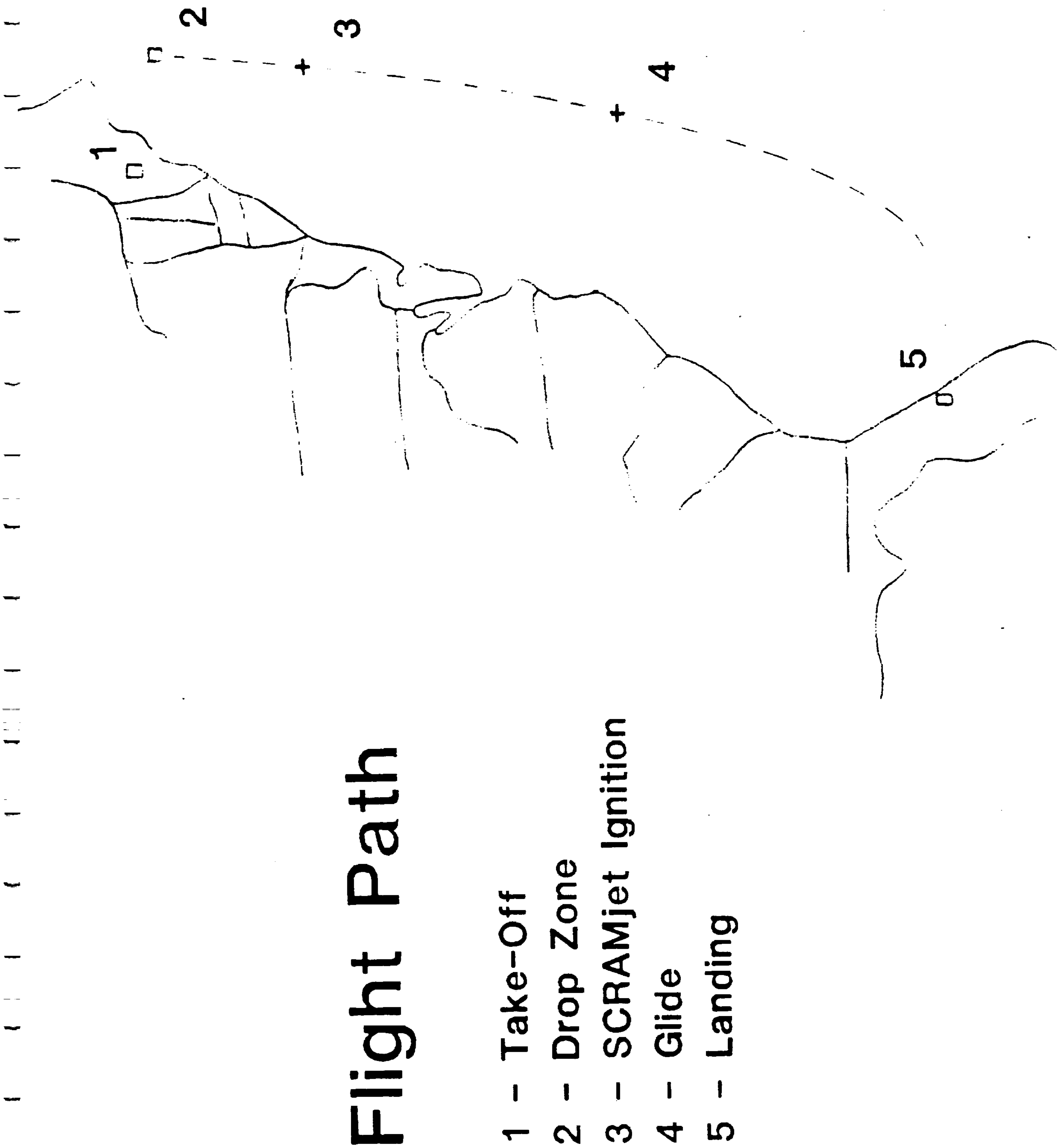

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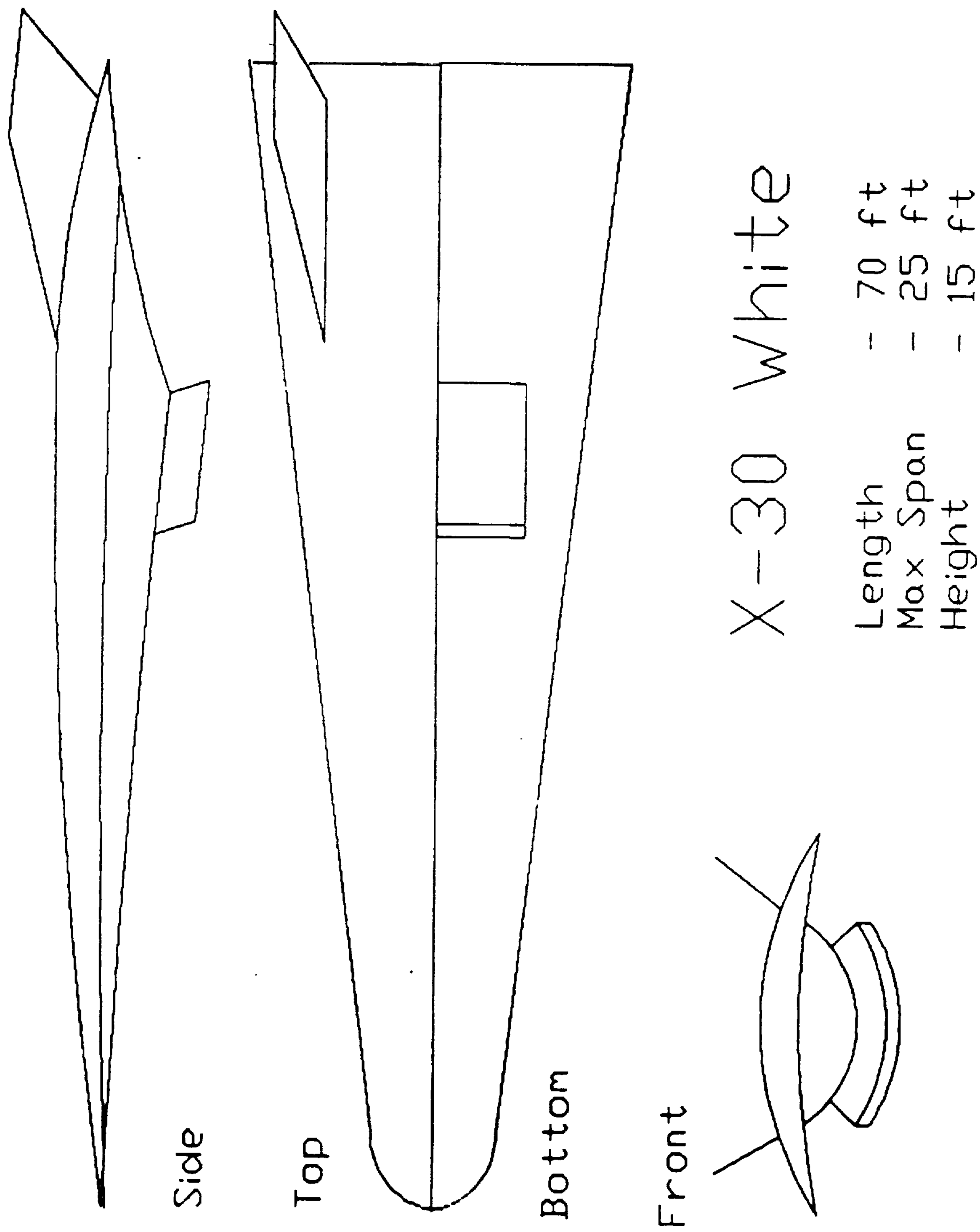
Figure 2-3



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Design Revision 1.0

Figure 2-4

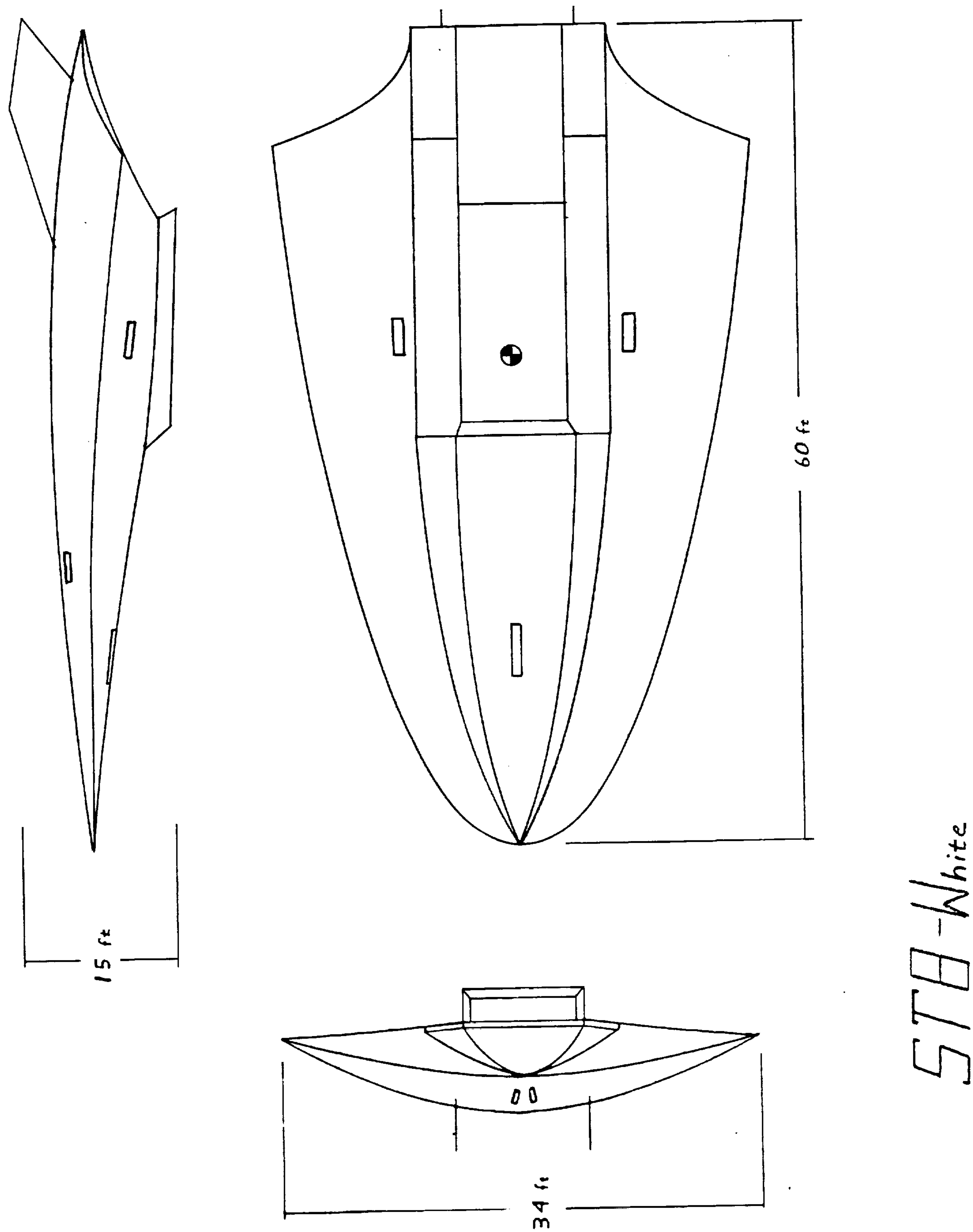
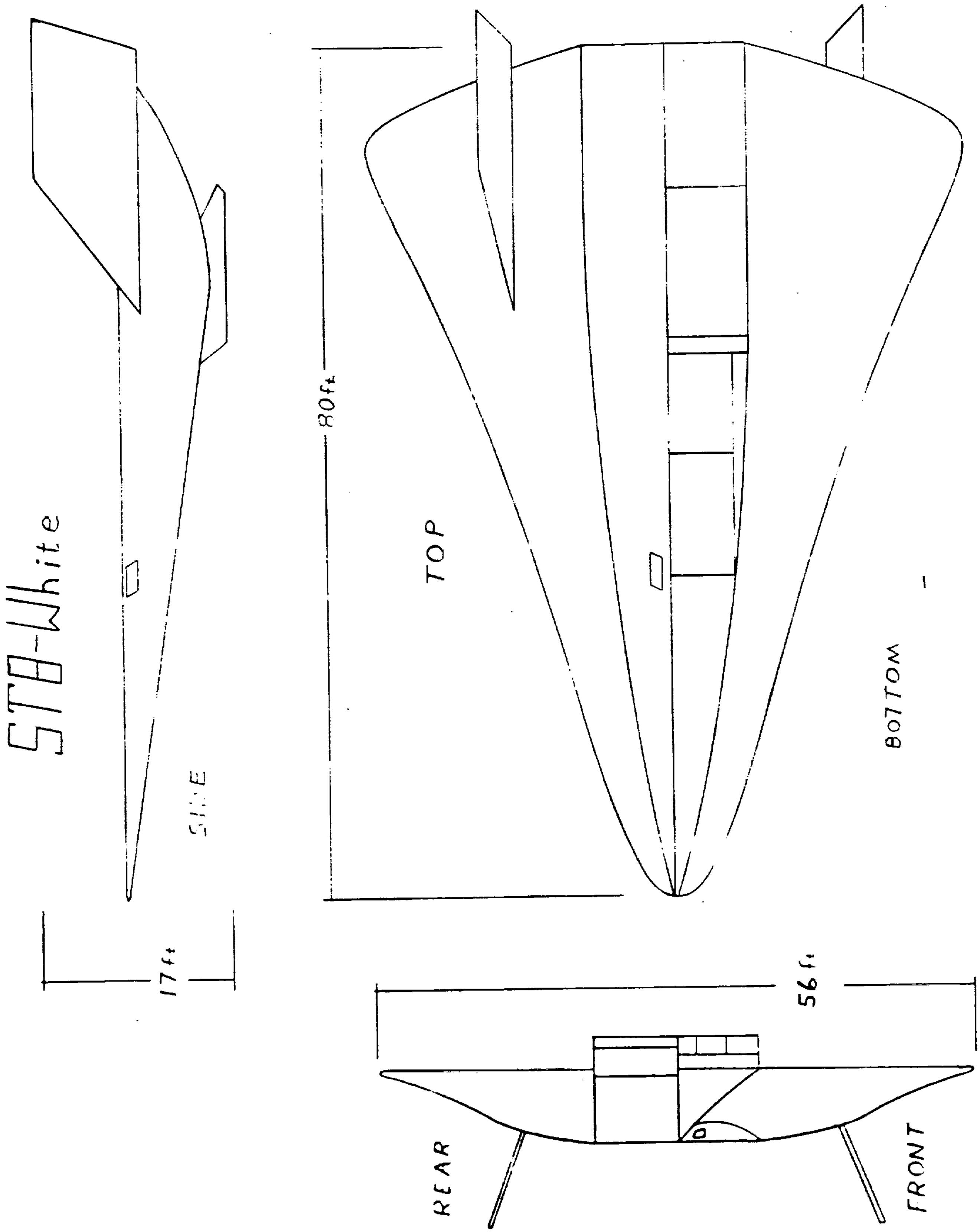


Figure 2-5



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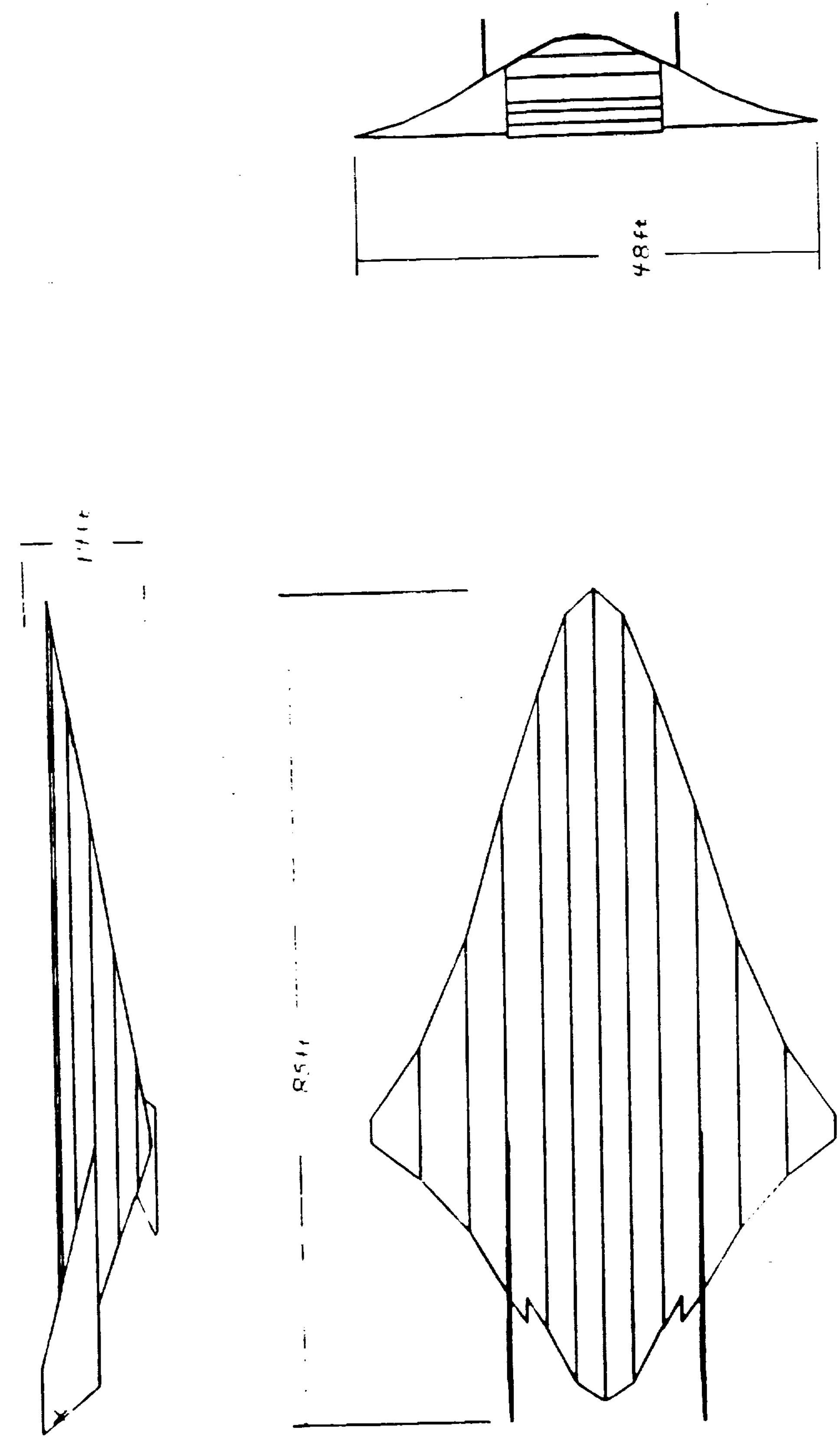
Final Design Configuration

Figure 2-6



Internal Arrangement

Figure 2-7

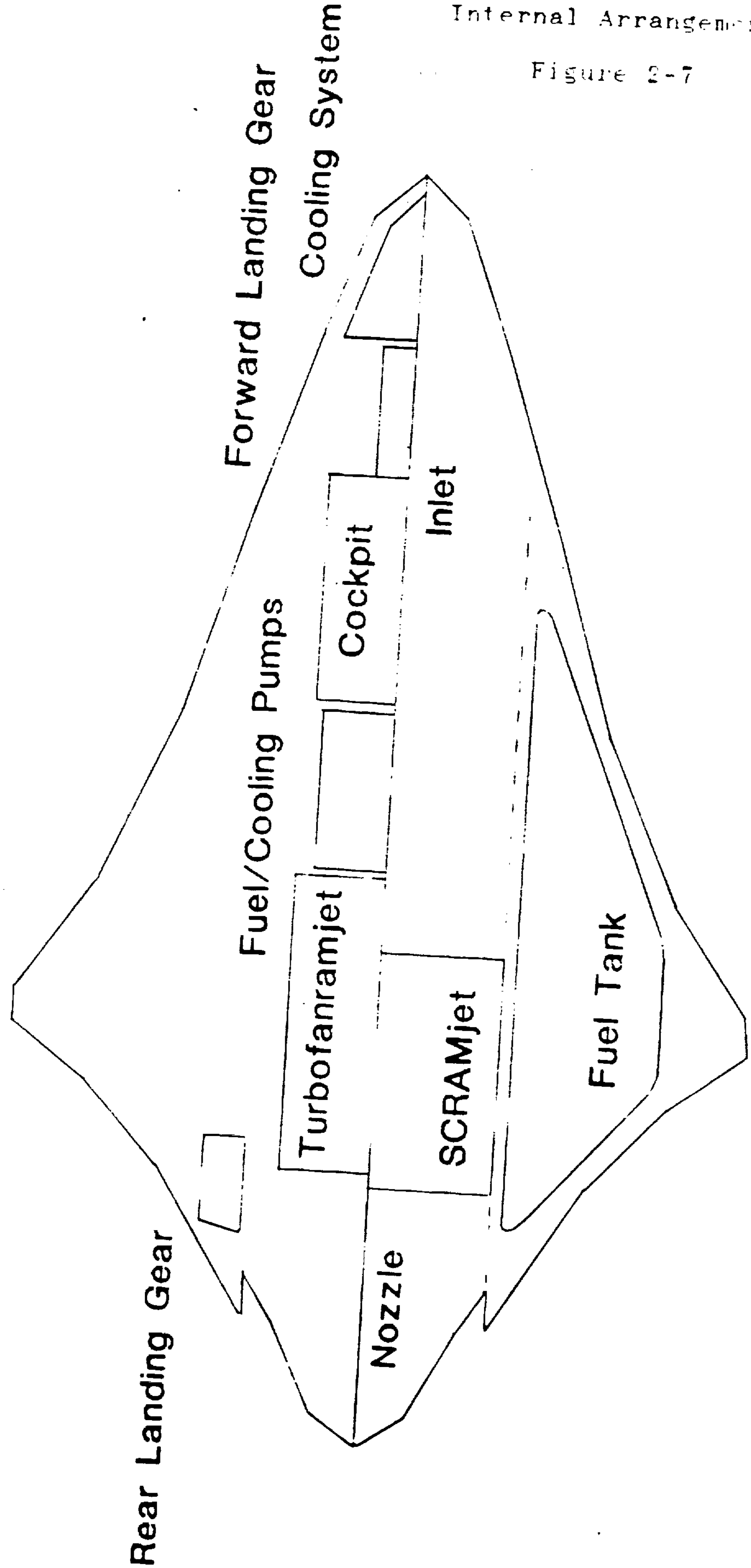
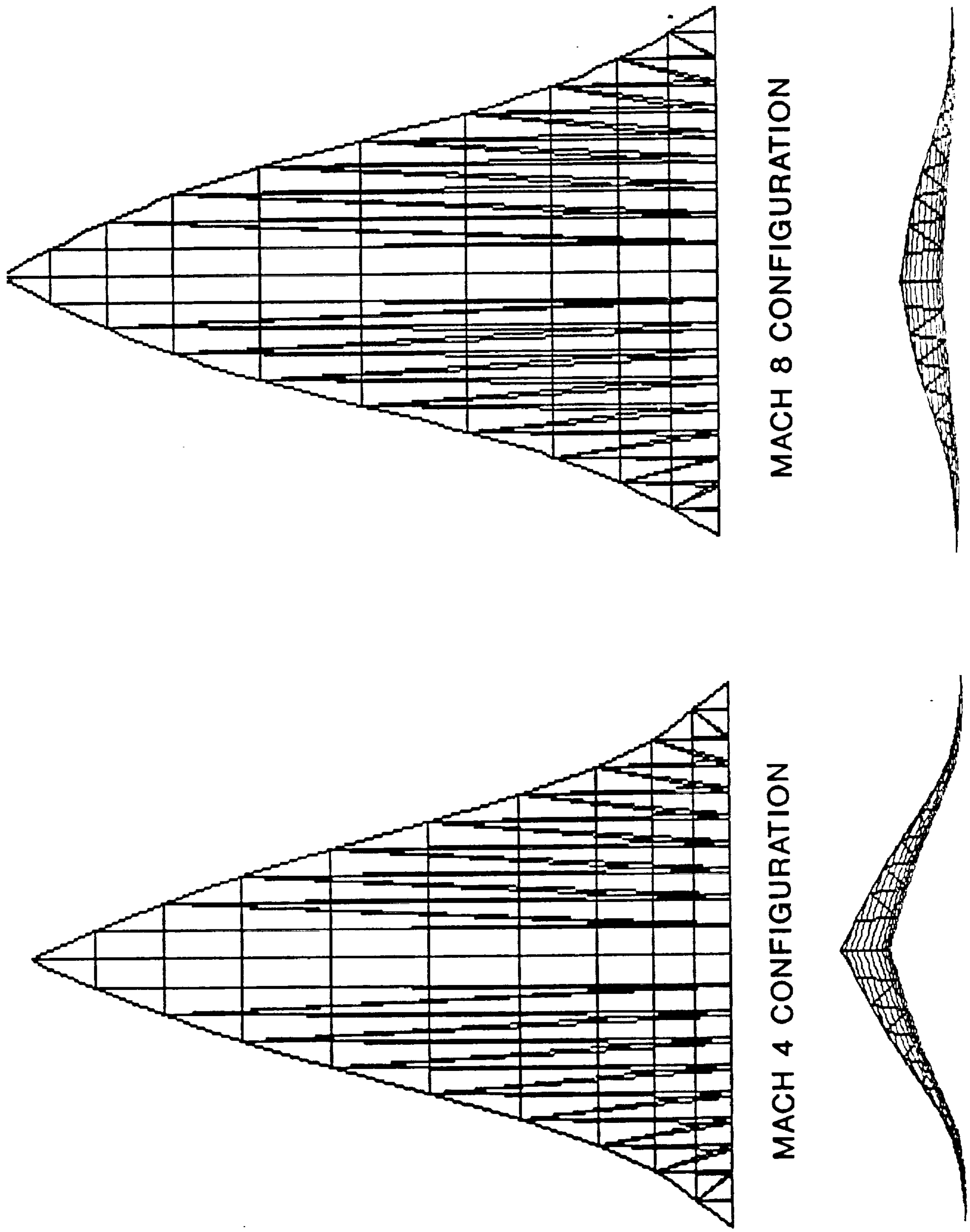
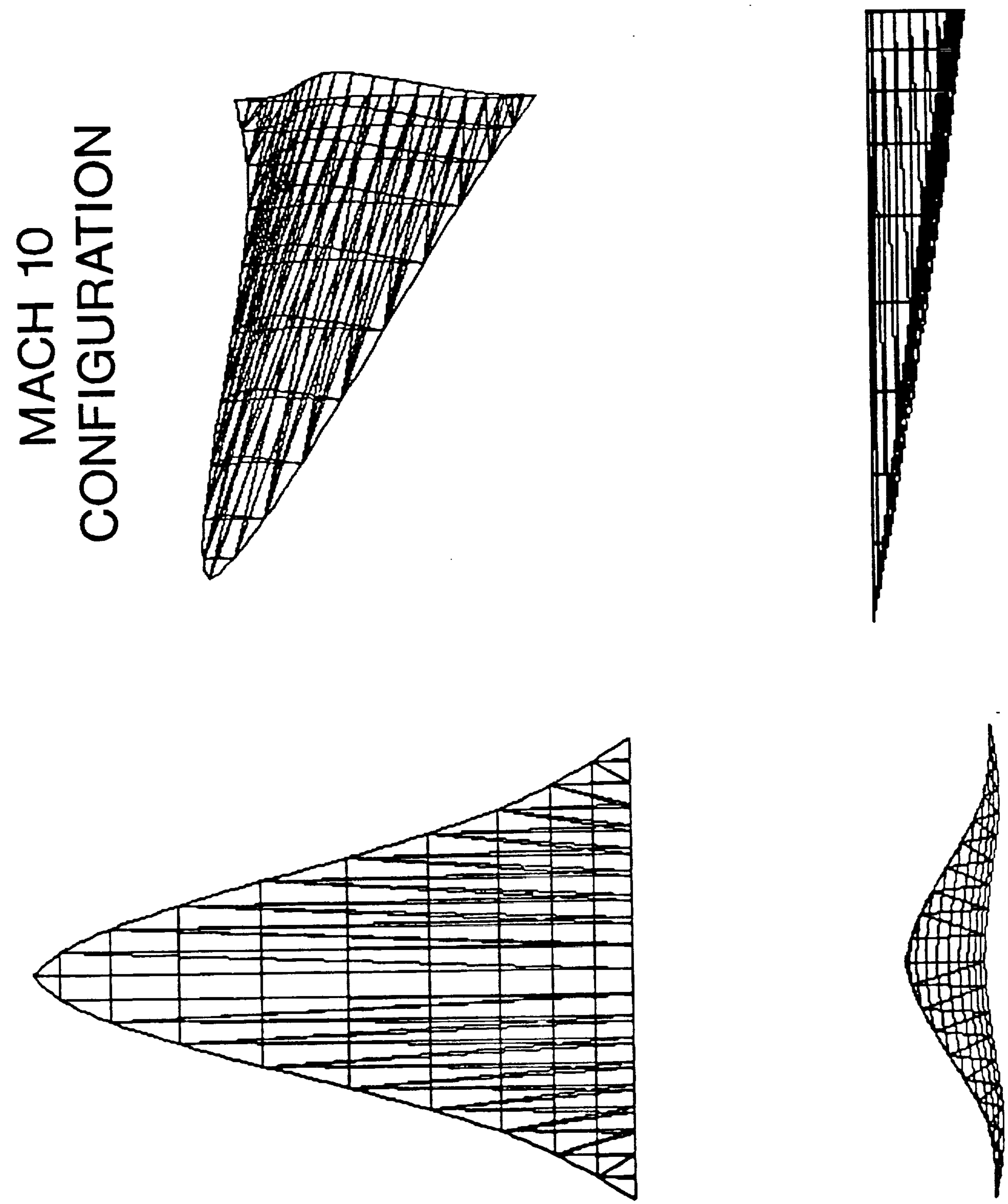


Figure 2-8



Mach 10 Waverider Optimization

Figure 2-9



3.0 Propulsion Systems

3.1 Overview of Propulsion Systems

The propulsion system of the aircraft is broken down into two separate sections - turbofanramjets and SCRAMjets. The turbofanramjets operate from the drop at Mach 0.8 and an altitude of 40,000 feet to Mach 6.0 and an altitude of 80,000 feet. At this speed and altitude the SCRAMjets will ignite and continue accelerating and climbing up to Mach 10 at 100,000 feet. The aircraft will hold this speed and altitude for two minutes and then descend to earth. This flight profile is the most efficient integration of the two propulsion systems.

3.2 Turbofanramjets

3.2.1 Engine Analysis

The engines that will be used to accelerate and climb the aircraft from Mach 0.8 at 40,000 feet to Mach 6.0 at 80,000 feet are Mach 6.0 LH_2-Fueled Augmented Turbofanramjet engines. These turbofanramjets will be produced by General Electric Aircraft Engines in Cincinnati, Ohio. The engines are scheduled to operate initially between the years 2005 and 2010. Dimensions, weights, and engine data for the Mach 6.0 engine were given to each of the four design groups by General Electric. Two different Mach 6.0 engines will be available, an acoustically unconstrained and an acoustically constrained one. A decision was made to use the acoustically unconstrained engine, because it weighed less and was smaller while producing the same thrust. Also, since our flight would take place away from land and people, the excess noise was not a factor in the design process.

Last quarter, the flight profile (see Figure 3-1 and Figure 3-2) was determined and the thrust required and thrust available vs. Mach number plot (see Figure 3-3) was created. A decision was made to use two turbofanramjets for this portion of the mission. However, by using two full size engines, more thrust was available than was needed for this flight profile. Therefore, these engines could be sized down to cut down on weight and also give the aircraft more volume to store the liquid hydrogen fuel. General Electric included engine scaling relationships with the engine data. Upon examining the Thrust Required and Thrust Available vs. Mach number plot for two full size engines, it was determined that these engines could be down-sized by 20 percent with respect to thrust. The engine scaling relationships indicate that the thrust and fuel flow scaled one to one with the airflow. According to these formulas, the full size engine of 6100 lbs. scaled down to 4880 lbs. (20%). Also, additional relationships for diameters and lengths were given and used. The results are as follows for a 20% reduction in engine size with respect to the thrust:

Table 3-1: Turbofanramjet Size Reduction

	Full Size	20% Reduction
Maximum Diameter (in.):	83	74
Length (in.):	265	237
C.G. Location (in.):	119	106
Weight - each (lbs):	6100	4880

The net thrusts given by G.E. can be reduced 20%.

Once the engines were scaled a configuration for the aircraft was made. The drag values that were finalized for our configuration were much higher than the preliminary ones of last quarter. These higher drag numbers gave the group two options. The first, an extra engine could be added to the aircraft to offset the enormous amount of drag in the transonic region. However, by adding an extra engine the entire aircraft would have to be sized up. This option was discarded because the French carrier aircraft design had given each group a size limit. If the aircraft was enlarged, the French limit would not be adhered to. Also, by changing the configuration of the aircraft, the drag analysis would have to be repeated and these drag coefficients would probably increase. Because of the uncertainty of the outcome in adding a third turbofanramjet engine, a second option was carried out.

This second option was to alter the existing flight profile for the turbofanramjet portion of the mission. Since the drag values increased to almost double the available thrust of the two 80%

turbofanramjet engines in the transonic range, the aircraft had to descend until the available thrust was appreciably greater than the drag. Once this was the case, the aircraft could ascend. Since the drag in the transonic region was so large, the aircraft had to descend to an altitude of 15,500 feet to overcome the drag and then ascend to 80,000 feet where the SCRAMjets were ignited. The updated flight profile and the Thrust Available and Thrust Required (drag) vs. Mach number are shown in Figures 3-4 and 3-5 and Figure 3-6, respectively.

3.2.2 <u>Flight Profile</u>

The flight profile was analyzed by a more precise method than used in the past. The thrust available and drag values were known for the aircraft at different altitudes and at different mach numbers. The profile was altered by using only one parameter. This parameter, the angle of ascent or descent, was varied until positive acceleration was possible and when the fuel amount was as low as possible. The entire flight profile analysis was determined by summing the forces along and perpendicular to the flight path and setting these values equal to the mass times the acceleration (F=ma). The analysis was determined using a varying dynamic pressure (Q) until the aircraft ascended above 40,000 feet where the dynamic pressure was set to a constant of 1500 psf. A Q of 1500 psf was determined to be the best for the profile. This Q was where the SCRAMjets performed well. Also this constant Q analysis determined the velocity and thus Mach number at each station of the climb. The fuel estimate for this profile was calculated by knowing the thrust available (used), the specific fuel consumption (sfc) used by the engine, and the duration at which the engine used this fuel flow rate. Even though the previous engine data could be reduced when the engines were sized down, it was unfortunate that the sfc could not also be reduced. If it could have been, the lowered sfc would have translated into a lower amount of fuel needed for the mission. The fuel needed to accelerate from Mach 0.8 at 40,000 feet to Mach 6.0 at 80,000 feet was 6460 lbs of liquid hydrogen. This amount of fuel translates into a storage volume of 1452 cubic feet at a density of 4.43 lbs per cubic foot. Specifications of the flight profile are included in Table 3-2.

Table 3-2: Turbofanramjet Acceleration Data

MAXIMUM ACCELERATION:	1.35	G
TIME (PORTION OF TOTAL MISSION):	6.10	MINUTES
TOTAL FUEL WEIGHT:	6,460	LBS
AIRCRAFT T.O. WEIGHT:	53,000	LBS
DISTANCE TRAVELED (PORTION):	158	MILES

3.3 SCRAMjets

3.3.1 Approach To Analysis

Above Mach 6, the only hope for an air-breathing engine that exists today is the operation of the SCRAMjet engine. At the present time, this idea is still in its preliminary stages. Due to this fact, the data that is available on these engines is sparse and hard to acquire. The mission requirements of this design project call for the acceleration of a hypersonic vehicle to ten times the speed of sound at an altitude of 100,000 feet. Once at this speed and altitude, the vehicle must maintain steady level flight for two minutes. The airplane is to weigh approximately 50,000 lbs. and have an internal volume of 3,000 cubic feet allowed for fuel. With this brief summary of objectives, the possibilities will be analyzed, and the steps taken will be discussed.

General Electric kindly provided the advanced design class with SCRAMjet data that was developed from recent tests. Without this data the following efforts could not have been performed and the whole mission would have been impossible. This data consisted of net thrusts that were developed by a liquid hydrogen burning engine. Since these thrusts were net thrusts, the ram drag was already accounted for and is not considered in this analysis. G.E. provided this information for dynamic pressures ranging from 500 psf to 2,000 psf in 500 psf increments. This covered altitudes from 65,000 feet to 126,000 feet. Combustor exit fuel to air equivalence ratios of 0.8, 1.8, and 2.8 were used in determining the specific impulse for different thrusts. The data covered Mach numbers of 5.0, 7.5, and 10.0. This fit into the regime of flight very well. Combustor exit static pressures and static temperatures were also given. The only regret was that more points in this range were not available for analysis.

This data, along with information from different members of the group, allowed for calculations to be made. The aerodynamicist developed initial values for drag coefficients that were used to calculate the acceleration of the aircraft. Weights were estimated so the required thrusts could be calculated. From the waverider analysis, initial estimates were made on lift to drag ratios at different Mach numbers. The design procedure was a cyclical process that involved many iterations and refinements.

3.3.2 Strategic Requirement Analysis

From the G.E. data, the most thrust was developed for a dynamic pressure of 2,000 psf. This was the data that was used for the first round of calculations. However, it was soon realized that at this dynamic pressure with a velocity of Mach 10.0, the altitude was not as high as what was needed. From simple calculations, it was found that the dynamic pressure that would correspond to Mach

10.0 at 100,000 feet on a standard atmospheric day was 1,615 psf. The closest available data that could be used was at 1,500 psf. Since the turbofanramjet engines are operable to Mach 6.0, this point was decided on to ignite the supersonic combustion ramjet engines. Back solving the dynamic pressure equation showed at Mach 6.0 the aircraft should be flying at approximately 80,000 feet. This meant the vehicle would accelerate from Mach 6.0 to Mach 10.0 while climbing from 80,000 feet to 100,000 feet. The data from G.E. was modified so an estimate could be made for values of Mach 6.0 at 80,000 feet and Mach 9.0 at 97,000 feet. Information was now available at Mach numbers of 6.0, 7.5, 9.0, and 10.0.

3.3.3 Iteration Procedures

Only four points were available in this region, so four divisions were made for calculations. The information from Mach 6.0 would be used from Mach 6.0-6.75. The information from Mach 7.5 would be used from 6.75-8.25. Mach 8.25-9.25 would use data from Mach 9.0, and finally Mach 9.25-10.0 would use the Mach 10.0 data. The initial weight of the airplane, at SCRAMjet ignition, was estimated at 50,000 lbs. All of the data was put into a spreadsheet and examined for the cases of using 1-8 modules (engines). It was quickly narrowed down to examine the realistic choices of 3 to 6 engines. To see this data and the process that was involved refer to Appendix A. Data was analyzed concerning the following items:

Acceleration

(# Eng x Net thrust - C_D x Q x S) / mass

Time of operation

(V_{final} - $V_{initial}$) / Acceleration

Fuel Weight

(# Eng x Net thrust / I_{sp} x 3600 x Time(in hours))

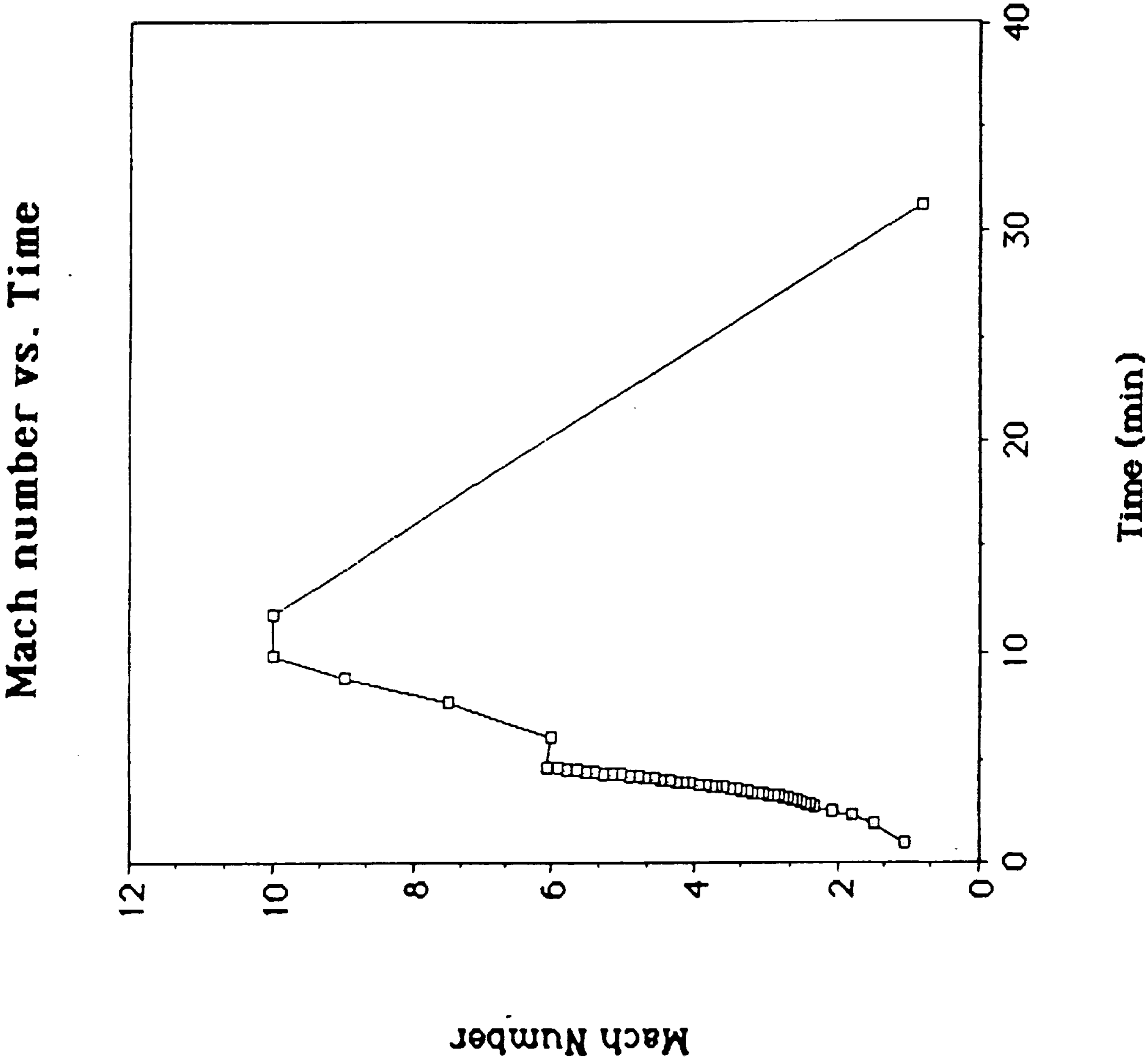
By examining these three major factors, it was seen that the smaller the thrust, the smaller the acceleration. The smaller the acceleration, the longer the operational time. The longer the operational time, the more fuel that was used. There are exceptions depending on equivalence ratio and specific impulse, but this was the general trend. The two minute steady level flight was not considered in these calculations. This was because at cruise, the objective was only to offset the drag. This was possible to do with only two engines. However, by shutting down three of the SCRAMjets, the drag would greatly increase as no flow was directed through the engines. It was quickly realized that running all five engines at a low fuel to air mixture equivalence ratio would solve the problem. No data was supplied from G.E. for equivalence ratios below 0.8. Approximate data was generated for lower values to get a rough estimation that 600 lbs of fuel would be burned to offset the drag if all five engines were operating.

3.3.4 Discussion of Results

It was realized that operating with less than five engines would consume an enormous amount of fuel and operating above five engines would not save enough fuel to offset the 1200 pounds that another SCRAMjet module would weigh. The next problem that was faced was an overabundance of lift. This made it impossible to fly at the requested flight profile. In order to overcome this problem, the angle of attack was changed until an achievable flight profile was reached. This occurred at an angle of attack of -4 degrees. Running through the spreadsheet with the corresponding coefficients of drag for this angle of attack yielded the final results. It was decided to use an extra 500 lbs. of fuel over the optimized value. This was done to increase acceleration and shorten range and flight time. By upping the fuel to air mixture equivalence ratio from 0.8 to 1.8 when the SCRAMjets are accelerating from Mach 6.0 to Mach 8.25, the range was shortened by 175 miles and the total flight time was decreased by 2.13 minutes. To accelerate from Mach 8.25 to Mach 10.0 the optimized value was still used which occurred at a ratio of 1.8. The total range over which the SCRAMjets operated was 584 miles. The vehicle took 5.82 minutes to travel this distance and used 7428 lbs. of LH_2 in the process. All of these numbers take into account the two minute cruise.

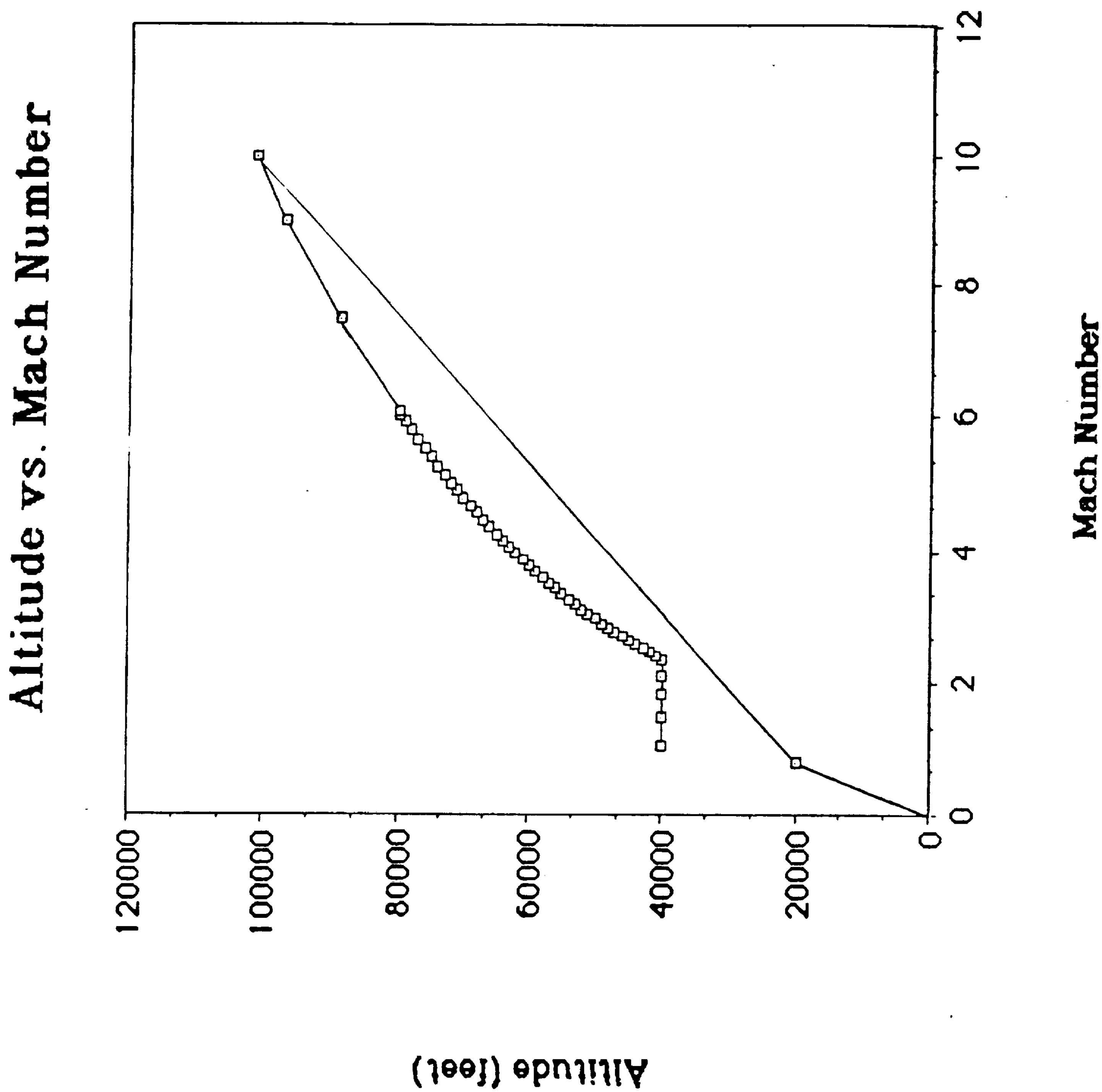
Mach Number vs. Time (Last Quarter)

Figure 3-1



Altitude vs. Mach Number (Last Quarter)

Figure 3-2



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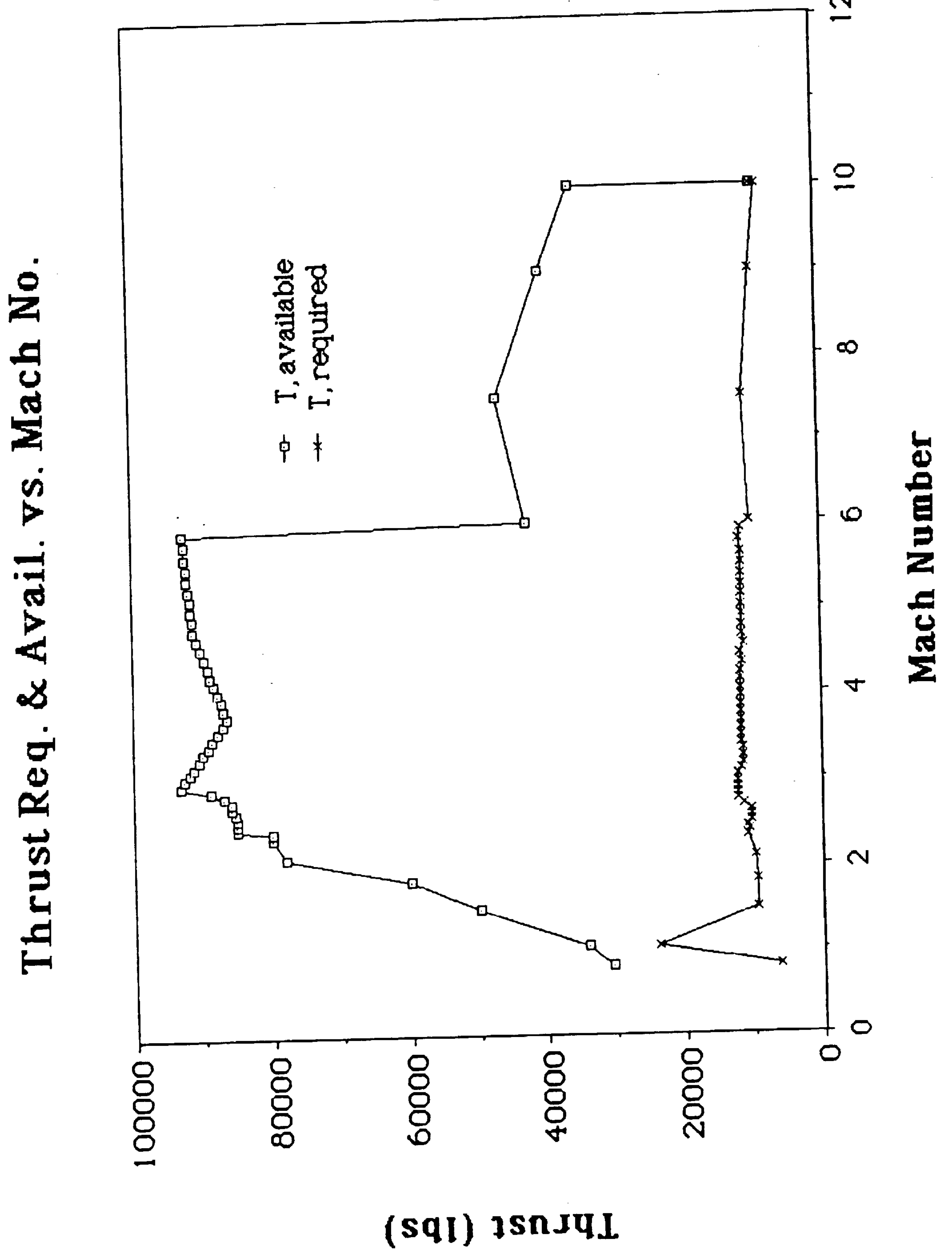
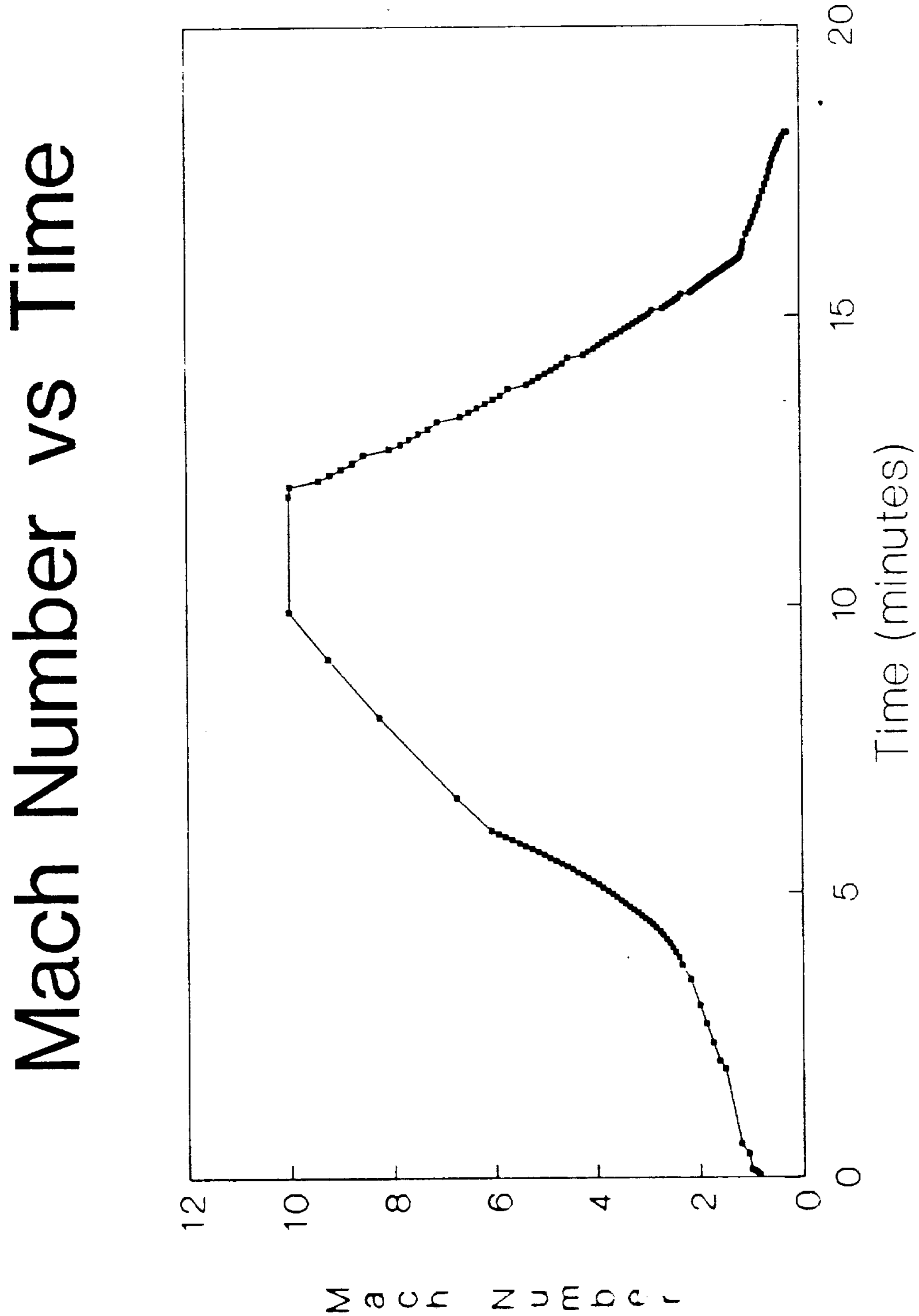
Thrust Required & Available vs. Mach Number (Last Quarter)

Figure 3-3



Mach Number vs. Time

Figure 3-4



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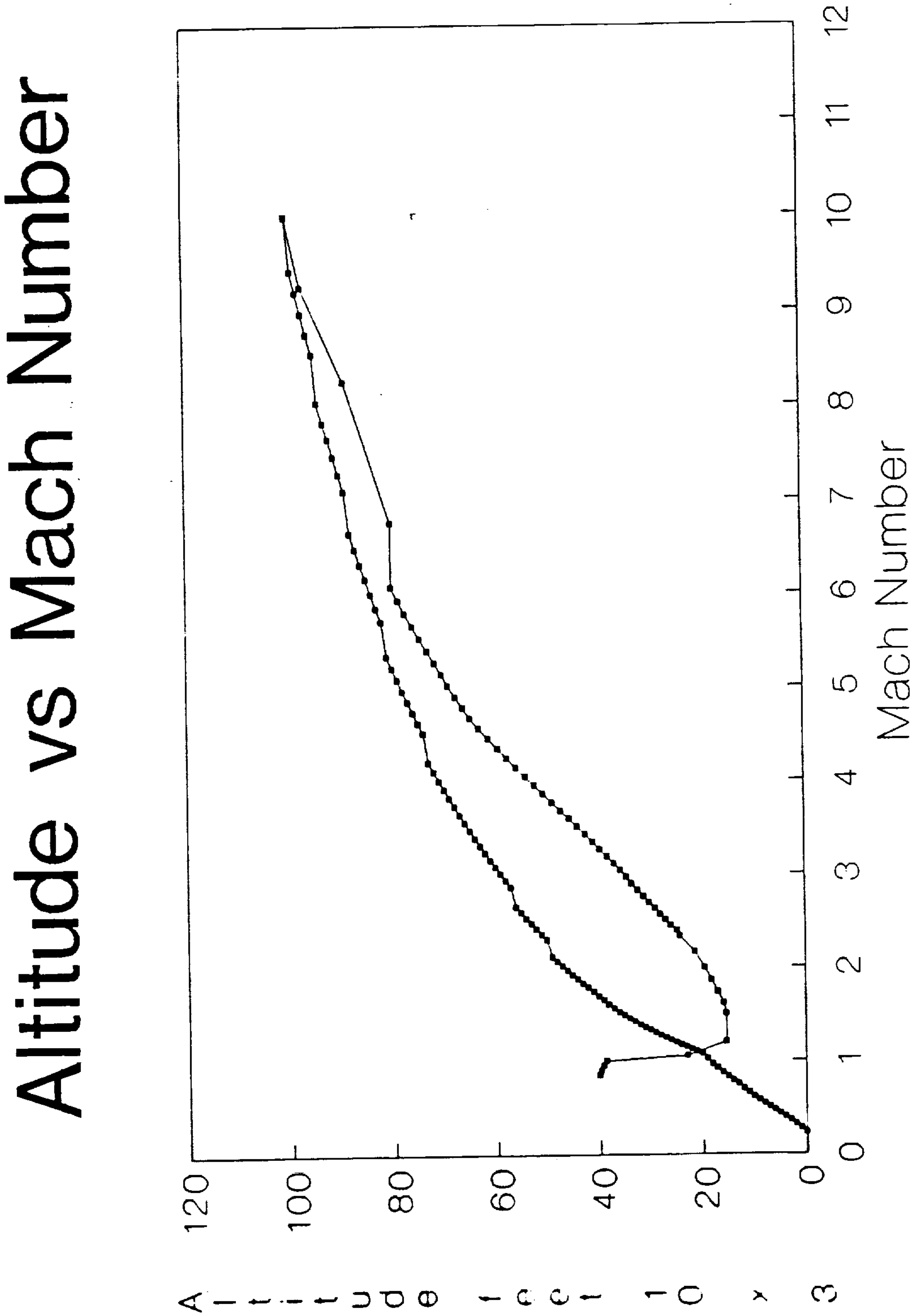
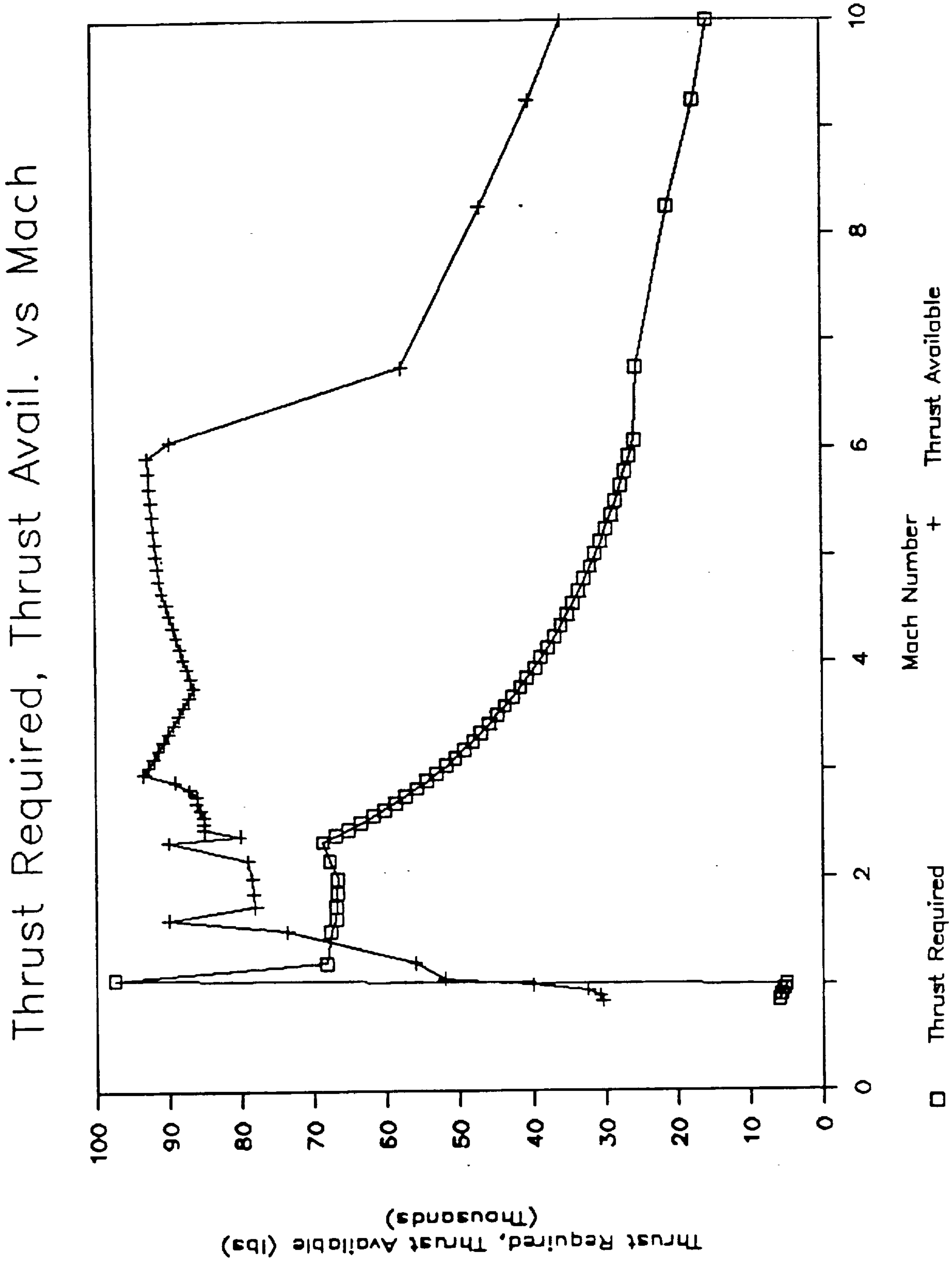
Altitude vs. Mach Number

Figure 3-5



Thrust Required & Available vs. Mach Number

Figure [illegible]



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4.0 Inlet Design

4.1 Overview of Inlet Design

Inlets are a series of ramps which direct the air flow into an engine. Axisymmetric inlets usually consist of flow over a cone in which ramps are formed from the basic cone shape, whereas, two-dimensional inlets consist of flow over wedges. Generally, two-dimensional inlets are considerably easier to analyze. The purpose of an inlet is to deliver the flow to the engine such that the flow is perpendicular to the engine compressor face. Therefore, the ramps are devised to turn the air flow to meet this condition. However, this is not the only function of the ramp system. The shocks that are caused by the ramps are needed to slow the flow down before entering the engine. Unfortunately, mass flow rates cannot be matched at all times during flight without having an extravagant inlet system which consists of many moving parts. The number of moving parts is directly related to the structural weight, so it is impossible to have this "flexible wave" which can be transformed into any shape. On the other hand, if the inlet capture area is not matched up with the needs of the engine, many problems could occur. If not enough flow is delivered to the engine, then the engine will not operate efficiently; if too much flow is delivered to the engine, the air flow will build up and spill out causing spillage drag. There are some ways to correct for the two air flow problems by adding air ducts and vents.

Besides all the described problems, boundary layer build up must also be considered when designing an inlet. Generally, the boundary layer is bled off with a splitter plate which is placed a prescribed length parallel to the body. This allows a new growth of boundary layer for a smaller distance on the splitter plate. The losses due to the newly formed boundary layer are not as great as the boundary layer formed from the length of the aircraft. Also, boundary layer suction can be performed using a porous plate in which the change in pressure will "suck" the boundary layer off the surface.

The main objective of the test vehicle was to test a SCRAMjet engine for two minutes at Mach 10.0 at an altitude of 100,000 feet. After being dropped from a French designed carrier aircraft, a turbofanramjet engine will be used to propel the aircraft from 40,000 feet at Mach 0.8 to 80,000 feet at Mach 6.0. Since the two 80 percent full size turbofanramjet engines are placed inside the vehicle, the flow needs to be turned into the engine, whereas, the five full size SCRAMjet engines are located on the lower surface of the *STB-White*. From the design of the plane, all the engines are sharing one inlet surface on the lower side of the plane. Using a series of doors and ramps, the complicated system will have the turbofanramjet engine closed off during the SCRAMjet test. Since the STB will be flying at a negative angle of attack of -4 degrees during the entire time the SCRAMjets will be operating, they will be angled 4 degrees to the top surface of the plane. This will allow for higher pressure recovery for the SCRAMjet inlet. For both the turbofanramjet and the SCRAMjet inlet, a complicated program was written in order to obtain conditions at the entrance of each engine. Static pressure and temperature, combustion Mach number, total temperature and pressure, capture areas, and mass flow rates were calculated from freestream density, pressure, temperature, and Mach number which were input along with the desired ramp angles. Following this analysis of each inlet, an appropriate result was obtained for both engine inlets. (see Appendix B for turbofanramjet inlet Fortran code and Appendix C for SCRAMjet inlet Fortran code).

Because of the basic wedge shape of the waverider design, a two-dimensional inlet using linear compression ramps and variable geometry will be used for both the turbofanramjet and the SCRAMjet inlets. Linear ramps are used for their ease in analysis.

4.2 Turbofanramjet Inlet

Three constraints were placed on the turbofanramjet inlet design based on conditions at the compressor face. First, the airflow needed to be slowed down to subsonic speeds before entering the compressor. Second, the capture area at the compressor face had to be a 3 feet radius for each engine, and third, the flow conditions had to match the engine data provided by General Electric for various speeds and altitudes.

Two design Mach numbers were chosen for the design of the turbofanramjet. First, Mach 2.0 was analyzed using the computer code. Then Mach 4.0 was analyzed in the same manner. The program used oblique shock relations and normal shock relations obtained from Ref. 2 to analyze a series of ramps and angles. First, the freestream Mach number, pressure, density, and temperature were input interactively. Then various ramp angles were chosen until a satisfactory result was obtained. The number of ramps was set at three due to geometry constraints on the lower surface.

With the flow conditions known for the engine, guesses were made for ramp angles for both design Mach numbers. For the Mach 2.0 design condition, by using an initial cone angle of 3.4 degrees (8 degree wedge flying at -4.6 degree angle of attack), an initial ramp angle of 5 degrees, and cowl angles of 7 degrees each, the speed was slowed down to Mach 0.838 with a pressure recovery of 0.981. For the Mach 4 condition, the initial cone angle was set to 5 degrees, the initial ramp angle was 14 degrees, and the cowl angles were 16 and 18 degrees. This slowed the air flow down to Mach 0.766 with a total pressure recovery of 0.972 (see Appendix B). As one can tell, the construction of the lower surface of the STB is going to be quite complicated with variable geometry compression ramps. Following the calculation of the compression angles, the distances from the nose of the aircraft were calculated where the ramps needed to be placed in order to impinge on the cowl lip to form the compression inlet. In order to turn the flow up

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into the turbofanramjet, the flow needed to be subsonic prior to the turn. Therefore, a normal shock formed on the cowl lip (see Figure 4-1). This caused an external compression that was not desired, but was considered to be less of a problem than the expansion that would have otherwise resulted. This expansion would have caused the flow velocity to increase, and the normal shock at the compressor face would have caused entropy gradients that were extremely high. Since the mass flow rate did not match up for the engine, a number of vents were placed on the engine to route the excess air around the entrance of the compressor.

In conclusion, the turbofanramjet inlet did not exactly match the engine needs as far as mass flow rate conditions. It must be noted that the General Electric engine data was unclear as to the definition of the engine mass flow rate. As to which is more costly to the design regarding weight penalties and spillage drag, the answer is beyond the scope of this report. Not enough knowledge is known pertaining to the weight of variable geometry inlets. Therefore, this design entails many losses due to weight penalties.

4.3 SCRAMjet Inlet

Like the turbofanramjet inlet, the SCRAMjet inlet has certain constraints which need to be met. First, the air flow to the engine needs to be slowed down to between Mach 3 and Mach 5.

Second, the capture areas need to be matched to the full-size SCRAMjet, and third, the mass flow rate of the air needs to be equal to that of the required air mass flow rate of the engine.

Again, a linear ramp system with variable geometry is used for the SCRAMjet. With the aid of the computer code, the pressure recovery, mass flow rate of the air flow, and the speed of the air flow to the engine were calculated using freestream conditions and ramp angles. The oblique shock angles were calculated using supersonic oblique shock relations. These were the simplest and best approximations available to calculate the shock angles. The angles were iterated using the theta-beta-Mach number relation (see Ref. 2).

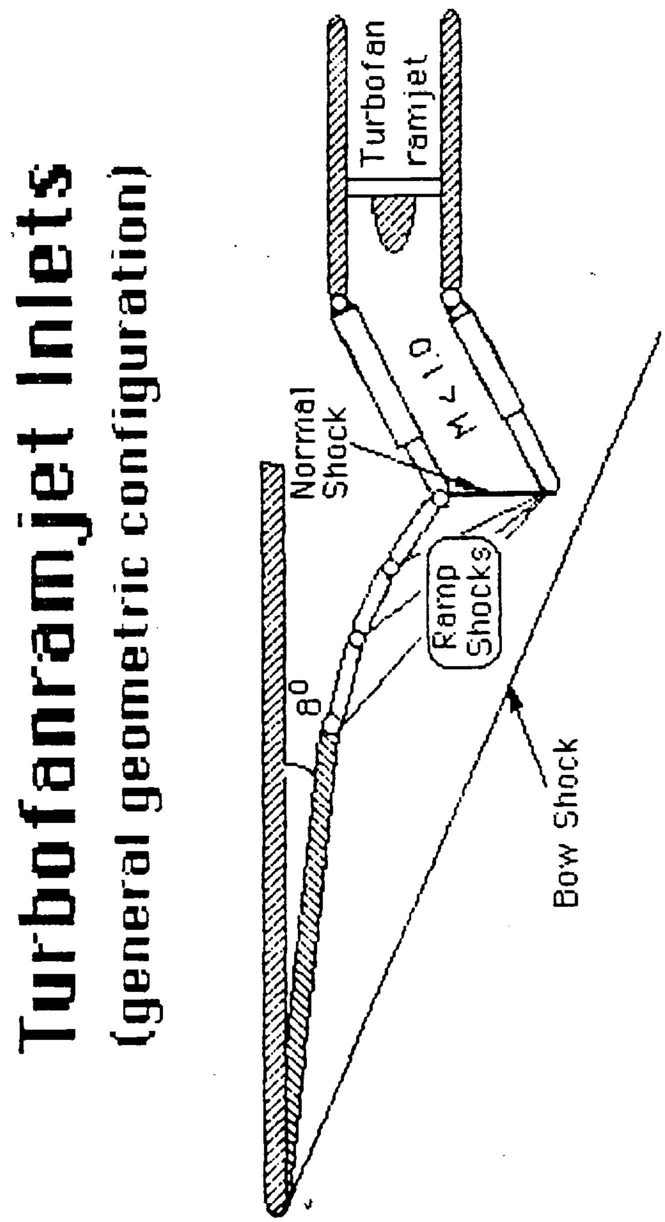
The SCRAMjet inlet was designed to handle many speeds in order to insure fewer losses due to spillage drag. Since the primary function of this aircraft was to test the SCRAMjets, the inlets were designed to eliminate as many air flow problems as possible. The design Mach numbers were chosen as Mach 6.0, Mach 7.0, Mach 8.0, Mach 9.0, and Mach 10.0. A three ramp system was chosen for the SCRAMjet. The turbofanramjet inlet will be completely closed off by connecting the cowl and the inlet surface. This will give a virtually smooth surface to work with for the SCRAMjet. The shock which occurs off the nose of the *STB-White* will completely miss the bottom edge of the SCRAMjet at all times. Therefore, the ramp system begins with one ramp on the lower surface of the aircraft. The other two ramps are located on the cowl of the SCRAMjet. Three ramps were chosen on account of the variable geometry system limitations.

For the Mach 6.0 design case, the cone angle was set at 4 degrees (8 degree wedge flying at -4 degree angle of attack), and the first ramp angle was 6.5 degrees. The two cowl angles were 7 and 3.5 degrees. This gave a total pressure recovery of 0.885 and a speed of Mach 3.8 entering the SCRAMjet. The pressure inside the combustion chamber was 703.0 psf, and the mass flow rate was matched at 142.0 pounds per second. Therefore, the intake capture area was matched to the SCRAMjet inlet area at 8 inches high and 30 inches wide. For the Mach 7.0 case, the cone angle was 4 degrees with the first ramp angle at 5 degrees. The cowl angles were 5 and 4 degrees respectively. This resulted in a total pressure recovery of 0.90 and a speed of Mach 4.5. The pressure inside the combustion chamber was 560 psf, and the mass flow rate was matched at 135 lb/s. For the Mach 8.0 case, the first cone angle was 4 degrees, and the first ramp angle was 5 degrees. The cowl angles were calculated to be 6 and 3 degrees respectively. For the Mach 9.0 case, the first cone angle was 4 degrees with a first ramp angle of 4 degrees. The cowl angles were both 4 degrees. For the Mach 10.0 case, the first cone angle was 4 degrees with a ramp angle of 3.5 degrees. The cowl angles were 3.5 and 2 degrees, respectively. See Appendix D for the computer code output that is referred to above.

In conclusion, the turbofanramjet engine took much larger losses in spillage drag than the SCRAMjet. The SCRAMjet flow was matched for every design Mach number. This was done in order to concentrate on the abilities of the SCRAMjet without complications in air flow.

Turbofanramjet Inlet Configuration

Figure 4-1



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5.0 Aerodynamics

5.1 Subsonic/Supersonic/Hypersonic Aerodynamics

Working independently from the engine groups and hoping to cover any future changes in flight profile the approach to the aerodynamic problem was formulated for a range of possible Mach numbers and angles of attack at varying altitudes. Dimensionless aerodynamic parameters were found through shock expansion theory, DATCOM methods, and Nicolai's approximations. Comparison of the different methods showed that the DATCOM 2-D wedge approximations resulted in the highest $C_{L\alpha}$'s. The comparison also showed that Nicolai's and DATCOM's delta-wing approximations corresponded to one another almost exactly. The delta-wing $C_{L\alpha}$'s were significantly lower than those from the 2-D wedge, however, it was believed that they represented the actual flow field the best. Therefore, the delta-wing approximation was used to calculate the supersonic and hypersonic aerodynamic characteristics of the *STB-White*. Subsonic characteristics were approximated assuming a linear lift curve slope.

Our goal was accomplished by using a spreadsheet, which calculated the C_L's, C_D's, and L/D's for every Mach number at each altitude and angle of attack. Subsonic and Supersonic C_L's were calculated from equations B.1 and B.2 respectively. Subsonic $C_{L\alpha}$'s were calculated using the aspect ratio, Mach number, and sweep angle. The supersonic and hypersonic $C_{L\alpha}$'s came from a delta-wing lift curve graph provided by Nicolai and DATCOM. The $C_{L\alpha}$'s were then incorporated with angles of attack ranging from -4 to 6 degrees, to calculate a variety of C_L's at each Mach number and altitude. From the density and Mach number at each altitude, a velocity and Reynolds number were calculated. Using equations B.3 and B.4 subsonic, and super/hypersonic C_{D0}'s were calculated respectively. Subsonic C_{D0}'s were based exclusively on friction drag, having Reynolds number as their only parameter. Supersonic and hypersonic C_{D0}'s were composed of friction drag and wave drag, having Mach number, thickness to chord ratio, Reynolds number, and the B factor as their parameters. With C_{D0}'s and C_L's for each Mach number, angle of attack, and altitude, the total C_D was calculated using equation B.5. The total C_D's were then multiplied by an efficiency factor of 1.1 to account for cowl drag along with any other miscellaneous drag items. The aerodynamic calculations done here considered the entire upper surface of the *STB-White* to be a lifting surface. The expansion lift from the exhaust ramp was neglected. Inquiries were made into the effects of centripetal lift, however, they were small enough to be considered negligible. Aerodynamic graphs can be seen on the accompanying pages. To see data, refer to Appendix F.

Subsonic C_L

$$C_l = \frac{2AR(\alpha - \alpha_{ol})}{2\sqrt{4 + AR^2(1 - M^2)(1 + \frac{\tan^2(\frac{\delta t}{c})}{(1 - M^2)}}} + (\alpha - \alpha_{ol})^2 \tag{B.1}$$

Supersonic C_L

$$C_l = C_{l\alpha}(\alpha - \alpha_{ol}) \tag{B.2}$$

Subsonic C_{D0}

$$C_{do} = \frac{0.074}{Re^{.2}}[1.082]\frac{S_{wet}}{S_{ref}} + \frac{280}{S_{ref}}\delta C_{do} \tag{B.3}$$

Supersonic and Hypersonic C_{D0}

$$C_{do} = \frac{0.074}{Re^{.2}}\left[\frac{1}{(1 + 0.144M^2)^{0.65}}\right] + B\left[\frac{1}{\sqrt{M^2 - 1}}\right]\left(\frac{t}{c}\right)^2 \tag{B.4}$$

Total C_D

$$C_d = K\left(C_{do} + \frac{C_l^2}{\pi eAR}\right) \tag{B.5}$$

5.2 Glide Analysis

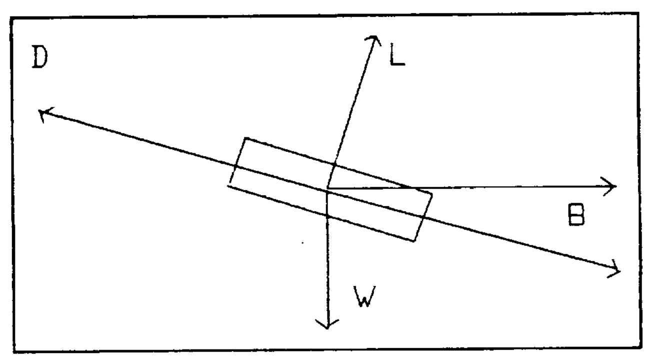
An unpowered glide and land was designated after the two minute SCRAMjet test to cut down on fuel volume. The glide trajectory was calculated with a spreadsheet using aerodynamic data.

The spread sheet contained the altitudes from 100,000 feet to 0 feet in 1000 feet increments and their corresponding densities. The analysis began with a dynamic pressure, Q, of 1500 psf at 100,000 feet and was gradually lowered to 60 psf at 0 ft. From the dynamic pressures and the density, a velocity could be found. A free body diagram gave the equations to solve for lift and acceleration. The angle of descent was arbitrarily chosen.

$$L = W \cos \Theta$$

$$ma = W \sin \Theta - D$$

$$Cl = W \cos \Theta / q S$$



Next, the dimensionless lift and drag were obtained from our aerodynamicist. The CL and CD values gave the angle of attack using the equation: $CL = CLo\ (\alpha - \alpha_{L0})$. Then, acceleration, or in our case, deceleration, was found using the dynamic equation $F = ma$. Now, using the equations of motion:

$$Vf^2 - Vi^2 = 2as \qquad s = Vi\ t + .5\ a\ t^2$$

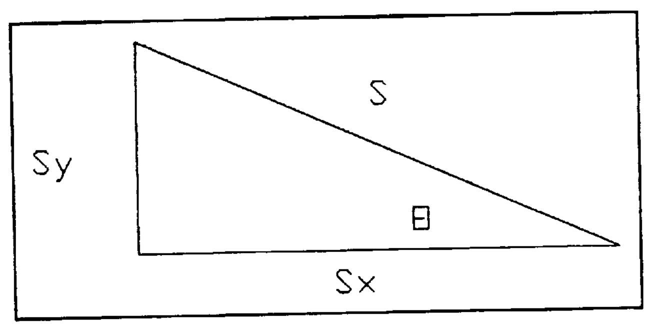
where s = distance travelled

Vf = final velocity

Vi = initial velocity

The distance travelled and the expended time were calculated. This distance was along the path travelled, so the x-y components could be found using the angle of descent.

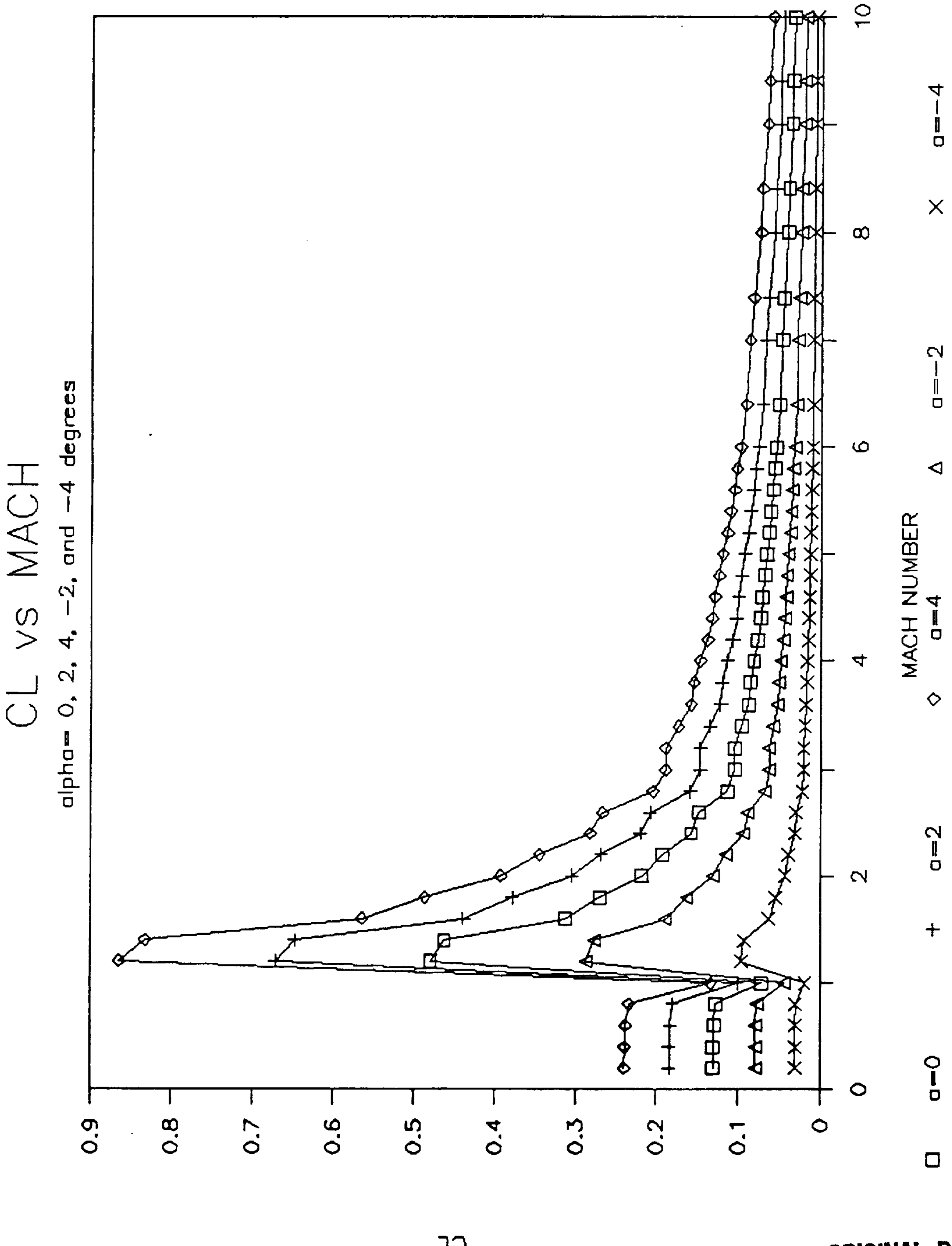
$Sx = s \cos \theta$



$Sy = s \sin \theta$

Now, this distance in the y-direction should equal 1000 feet. When it did not equal 1000 feet, the angle of descent was varied until it did. These calculations were made for all cases.

Figure 5-1



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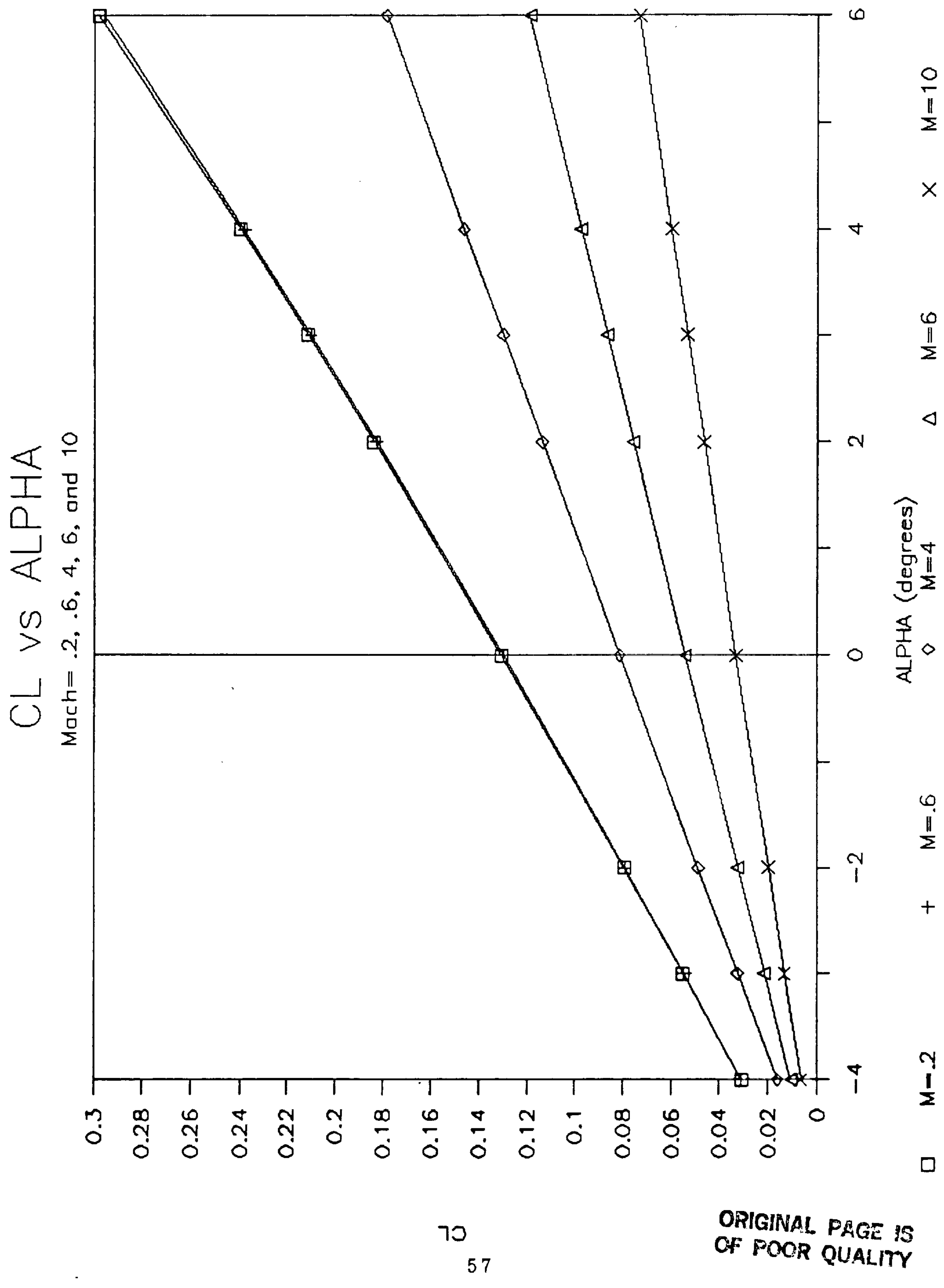
CL vs. Alpha

Figure 5-2



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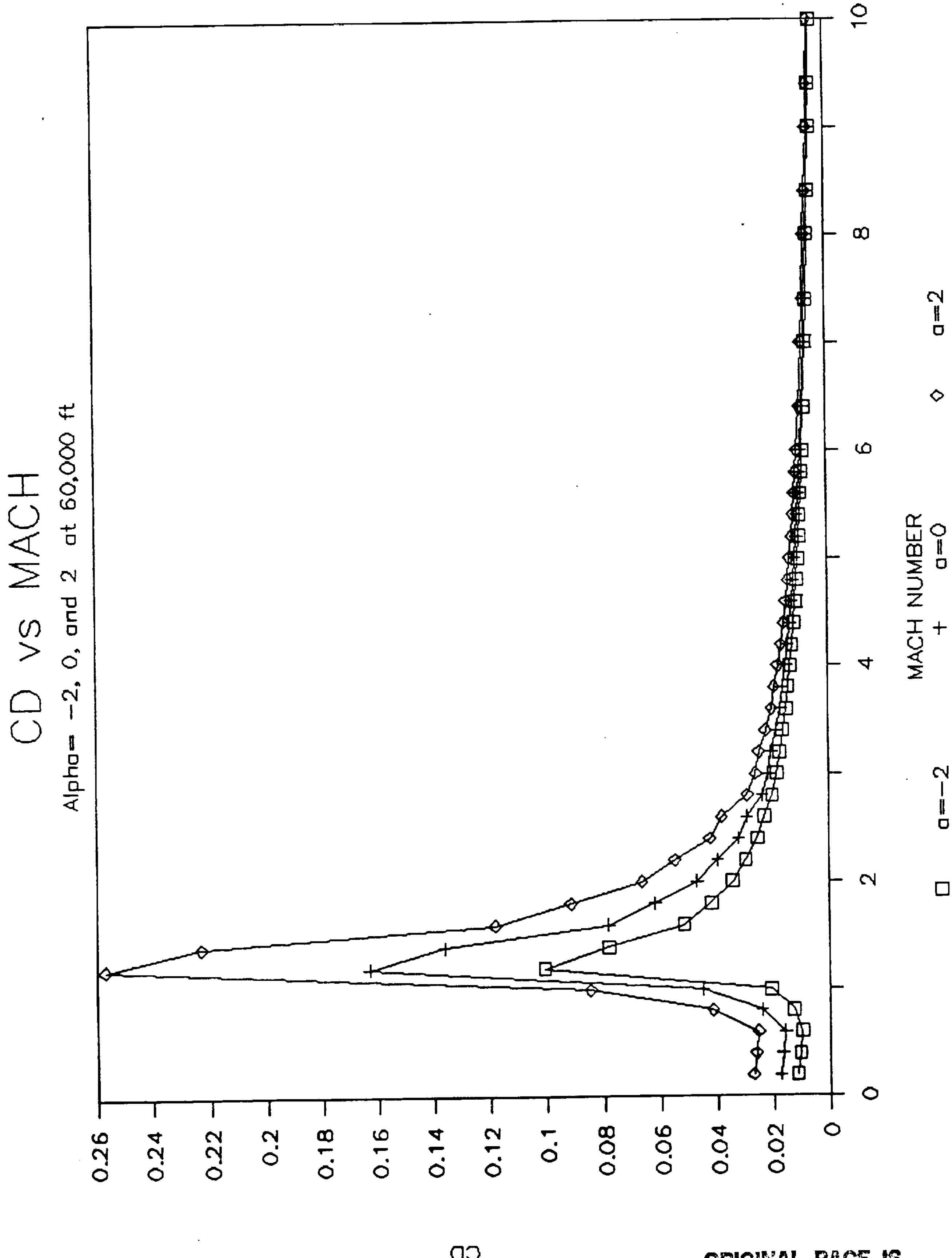
CD vs. Mach - Varying Angle of Attack

Figure 5-3



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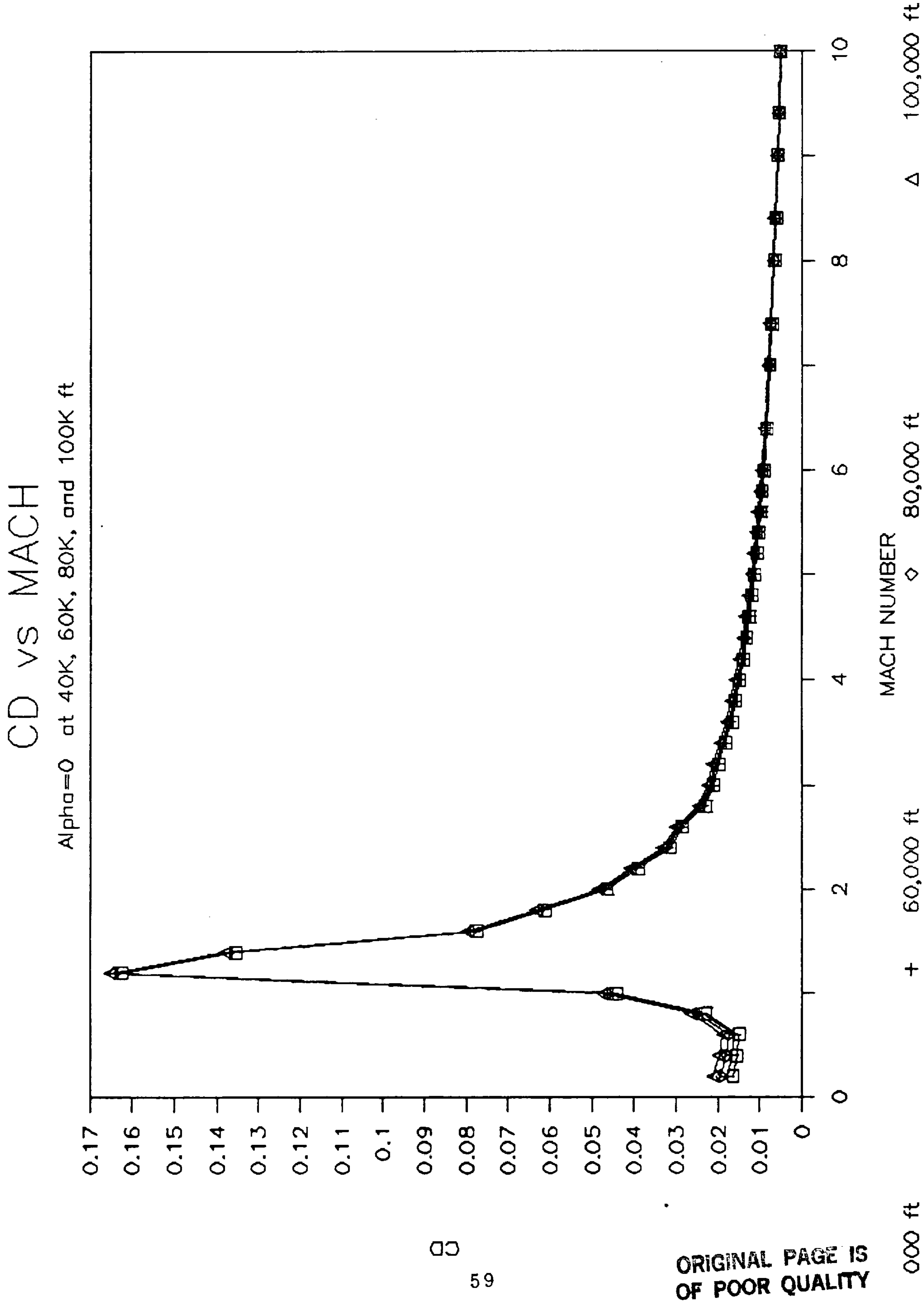
CD vs. Mach - Varying Altitude

Figure 5-4



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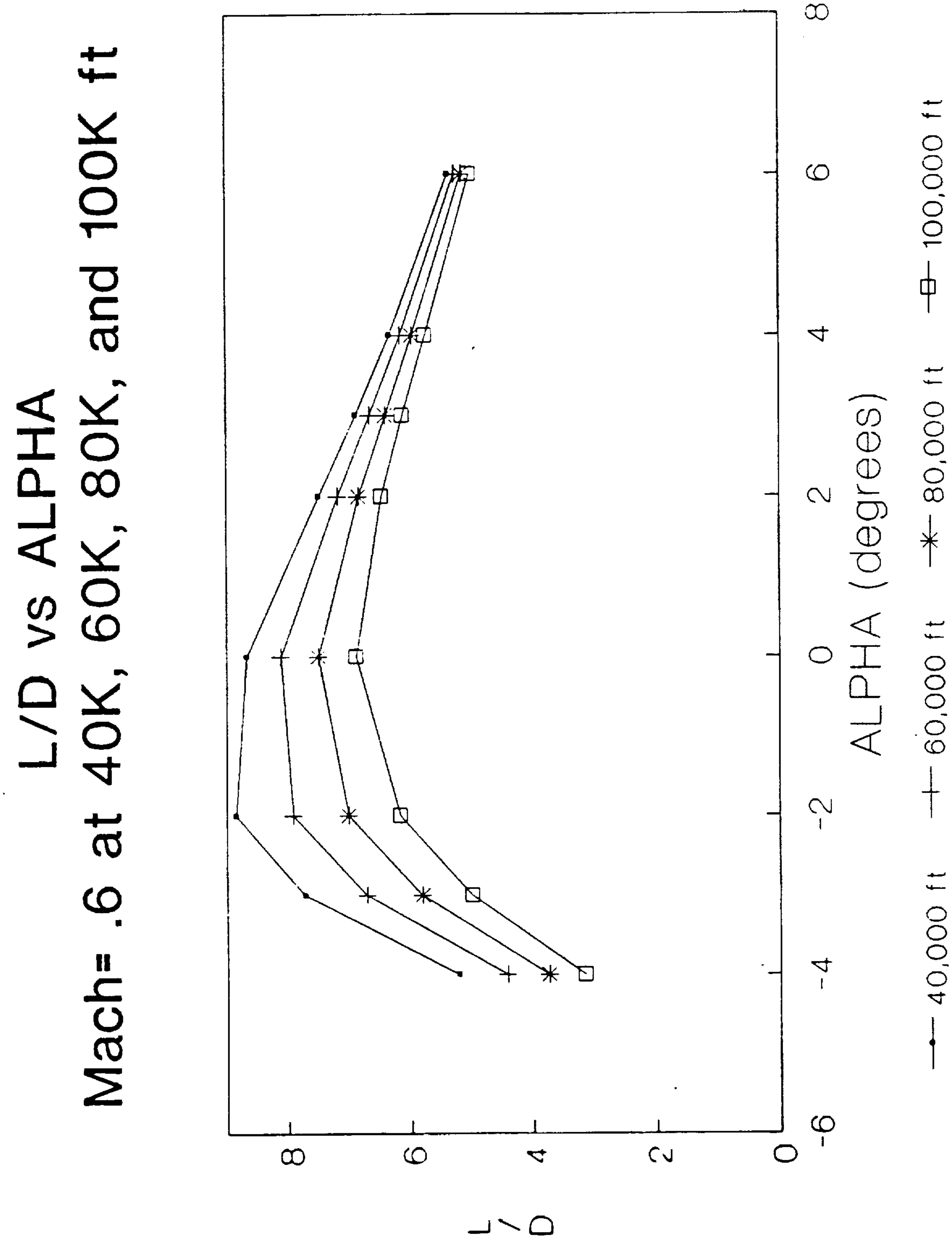
L/D vs. Alpha - Mach 0.6

Figure 5-5



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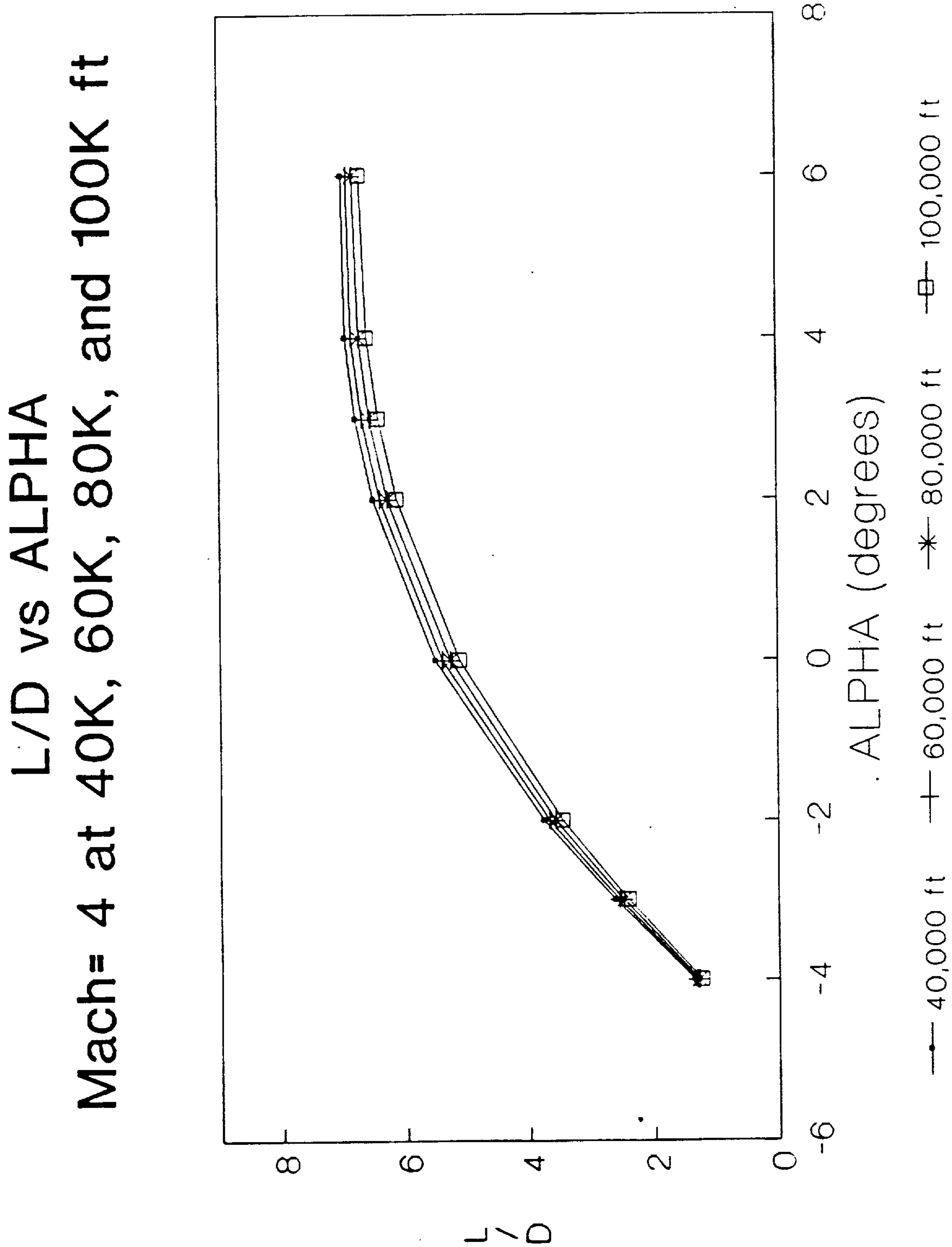
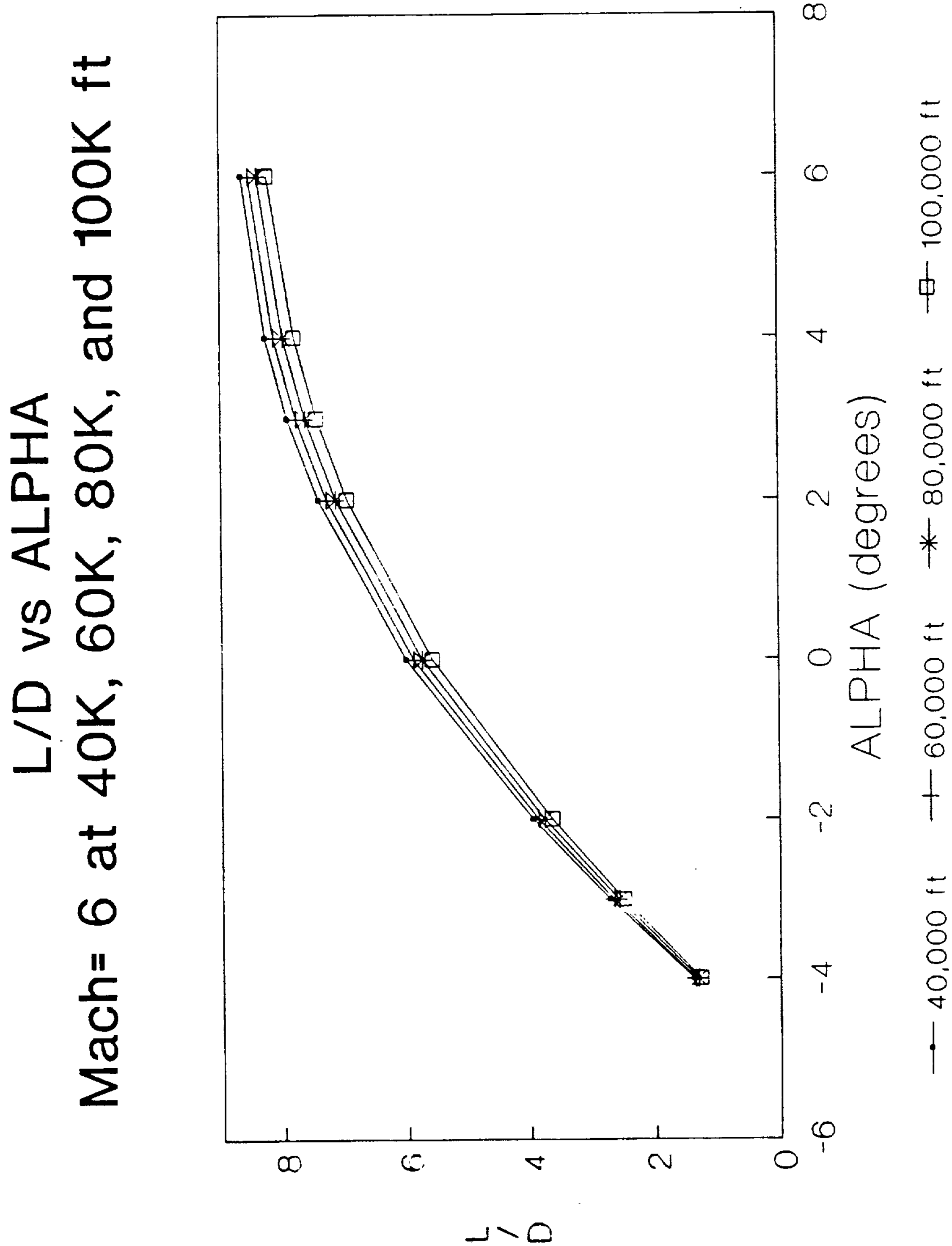
L/D vs. Alpha - Mach 4.0

Figure 5-6



L/D [illegible] - Mach 6.0

Figure 5-7



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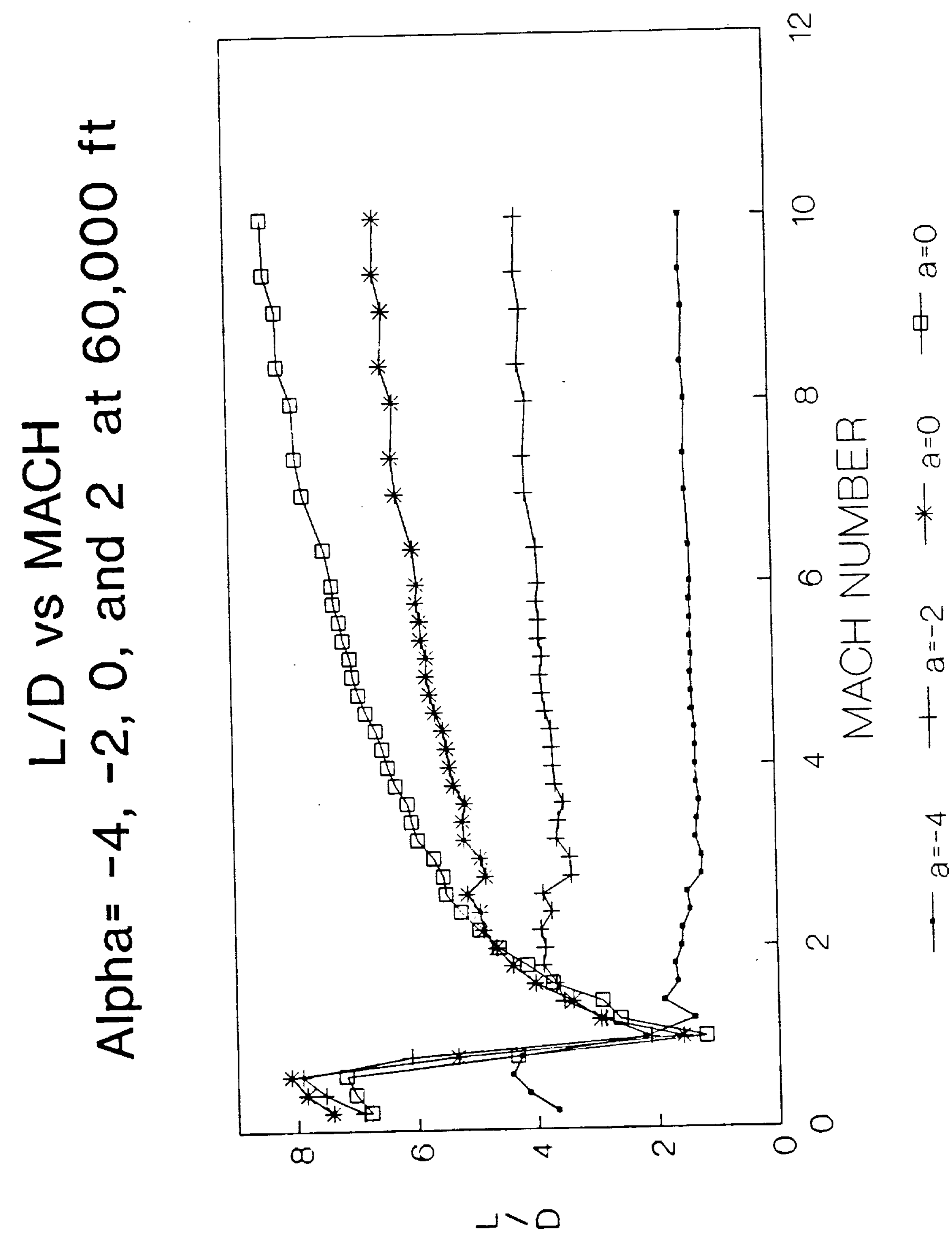
L/D vs. Mach

Figure 5-8



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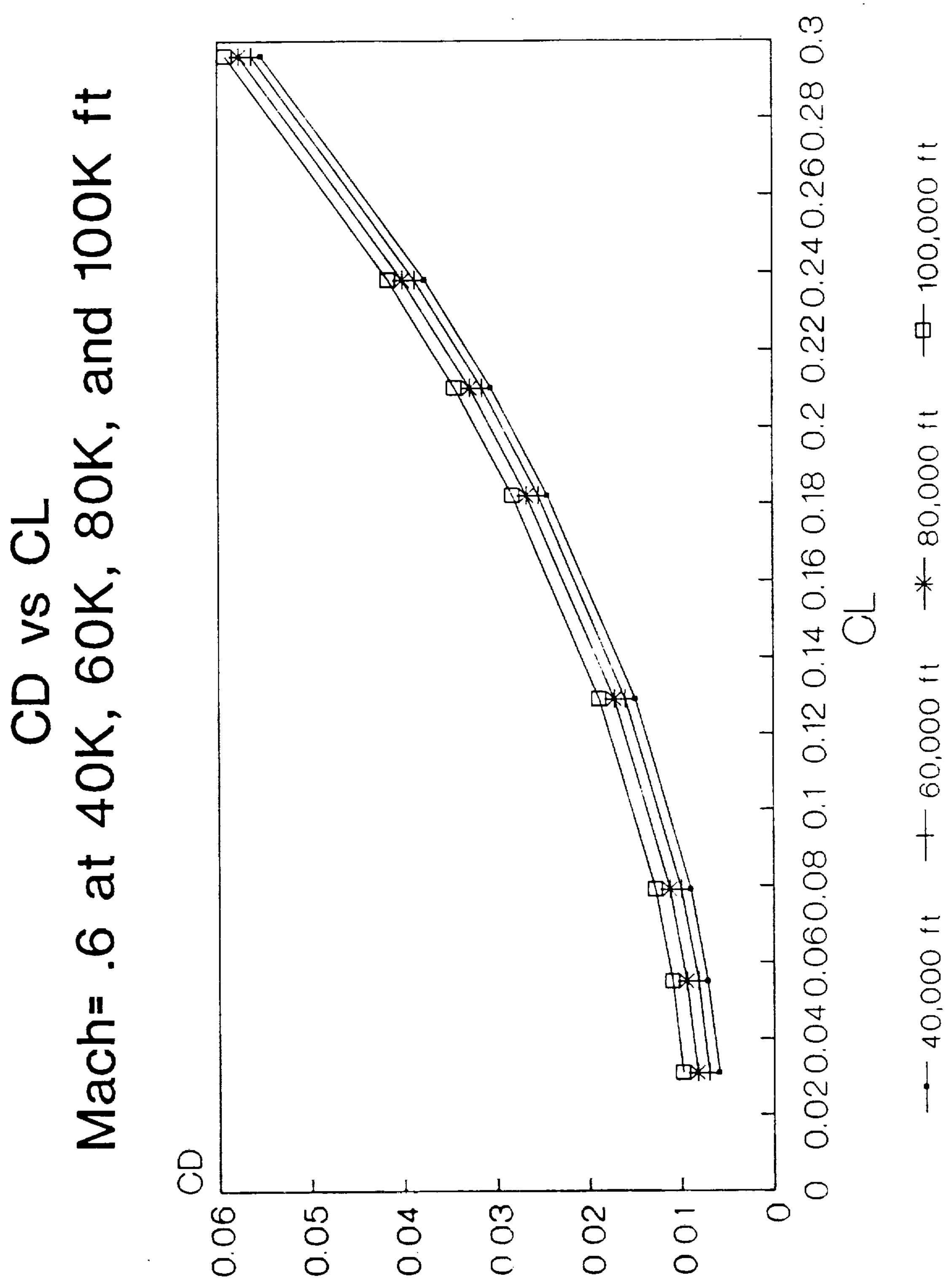
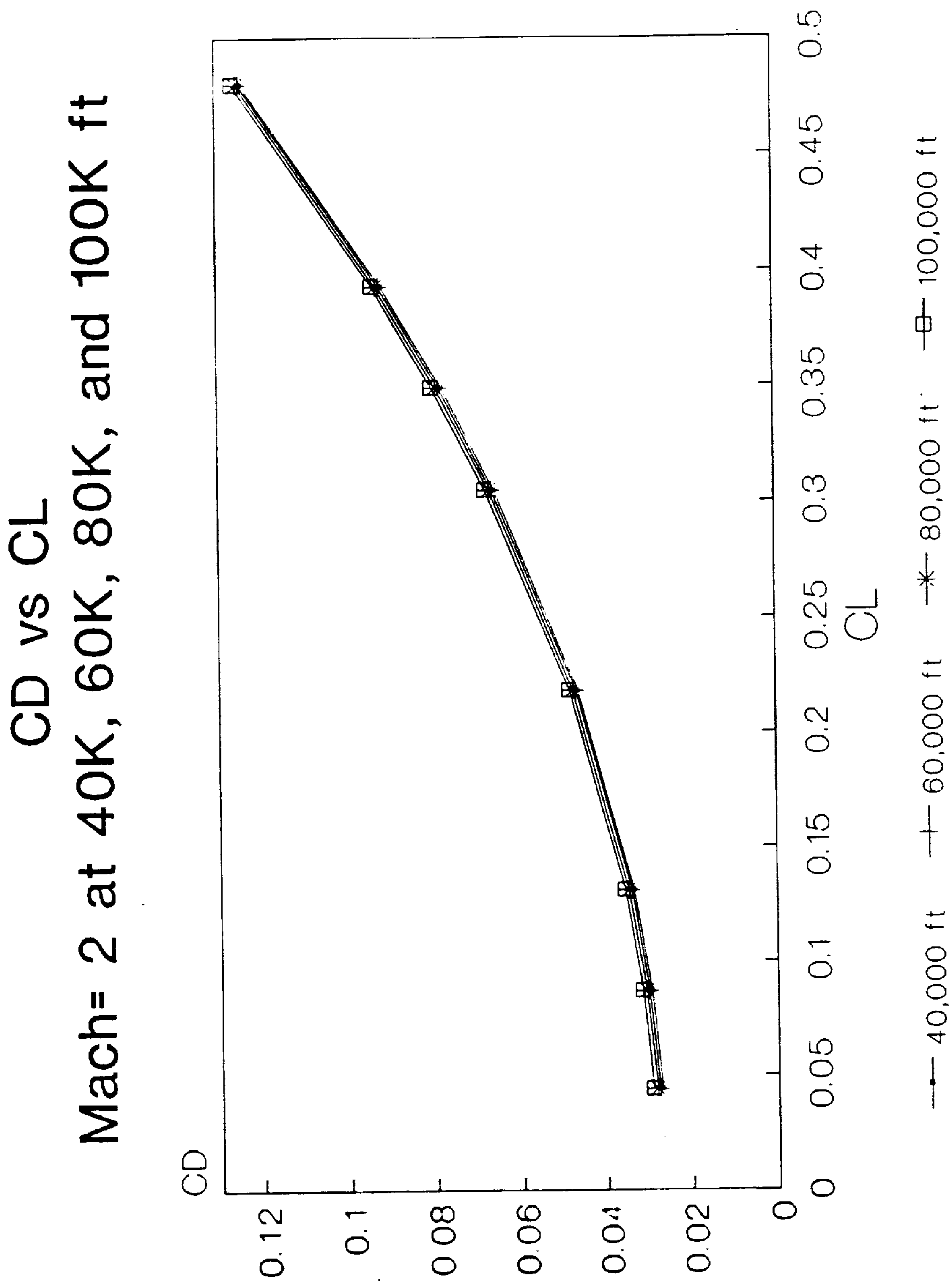
CD vs. CL - Mach 0.6

Figure 5-9



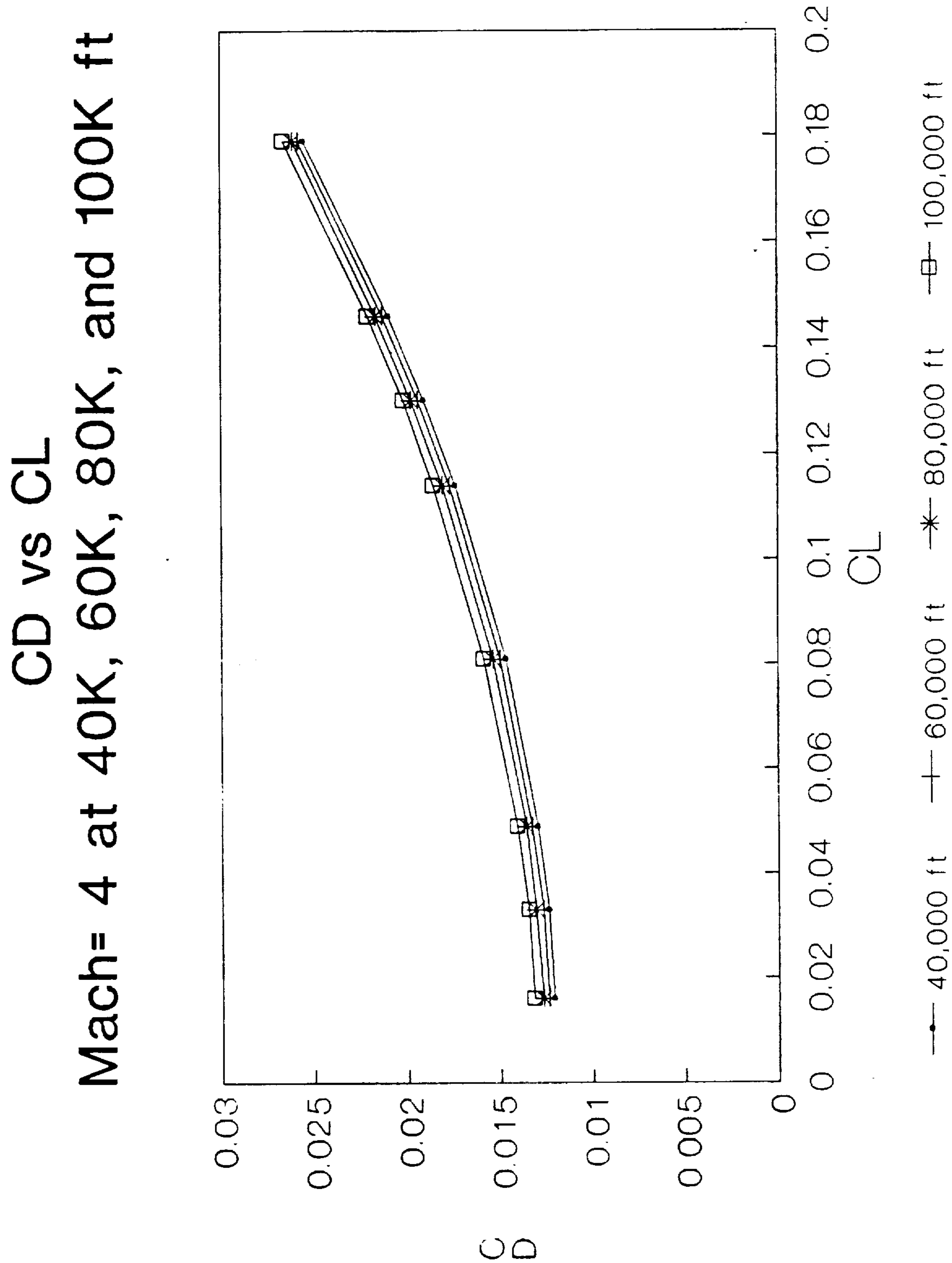
CD vs. CL - Mach 2.0

Figure 5-10



CD vs. CL - Mach 4.0

Figure 5-11



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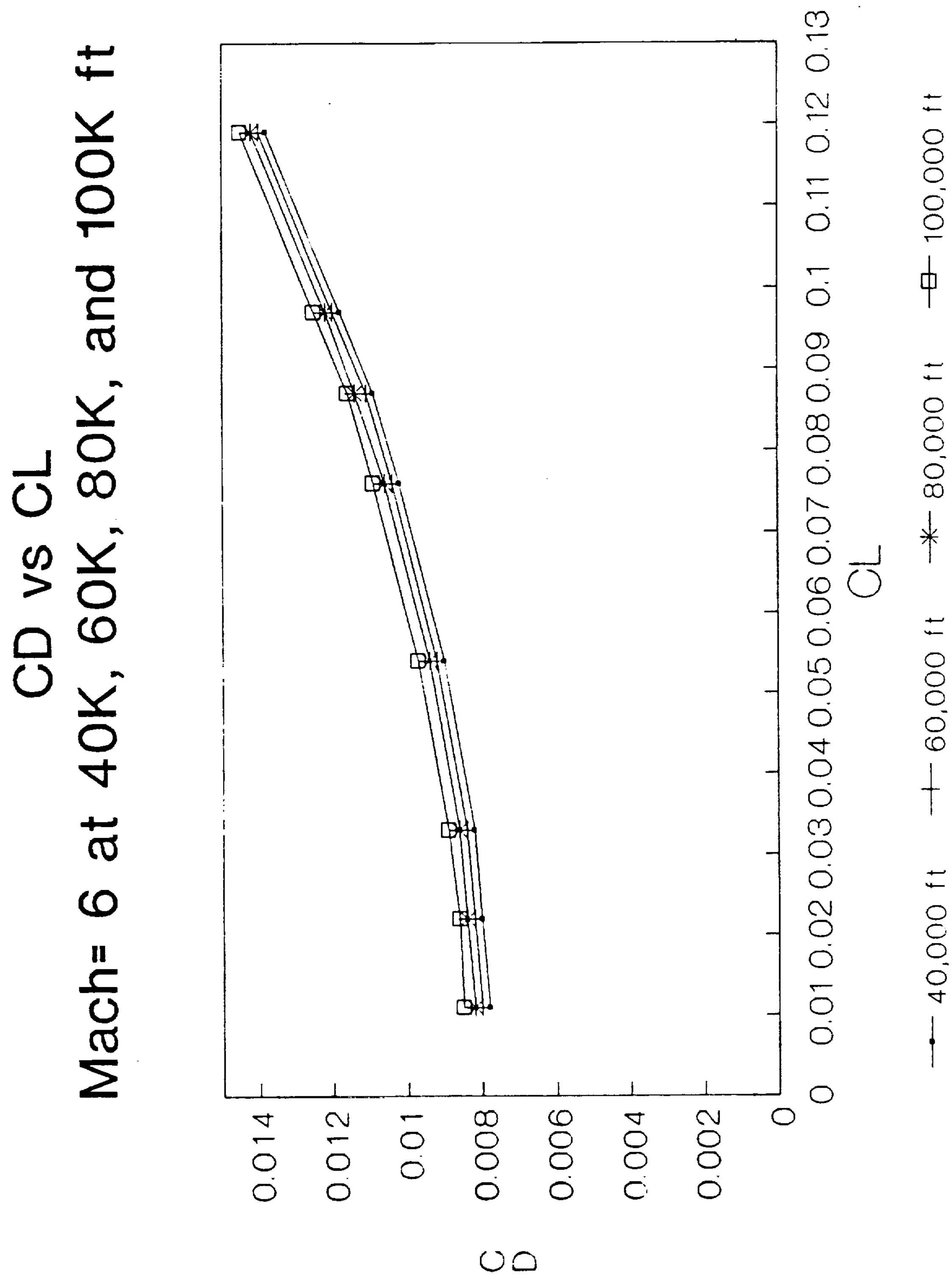
CD vs. CL - Mach 6.0

Figure 5-12



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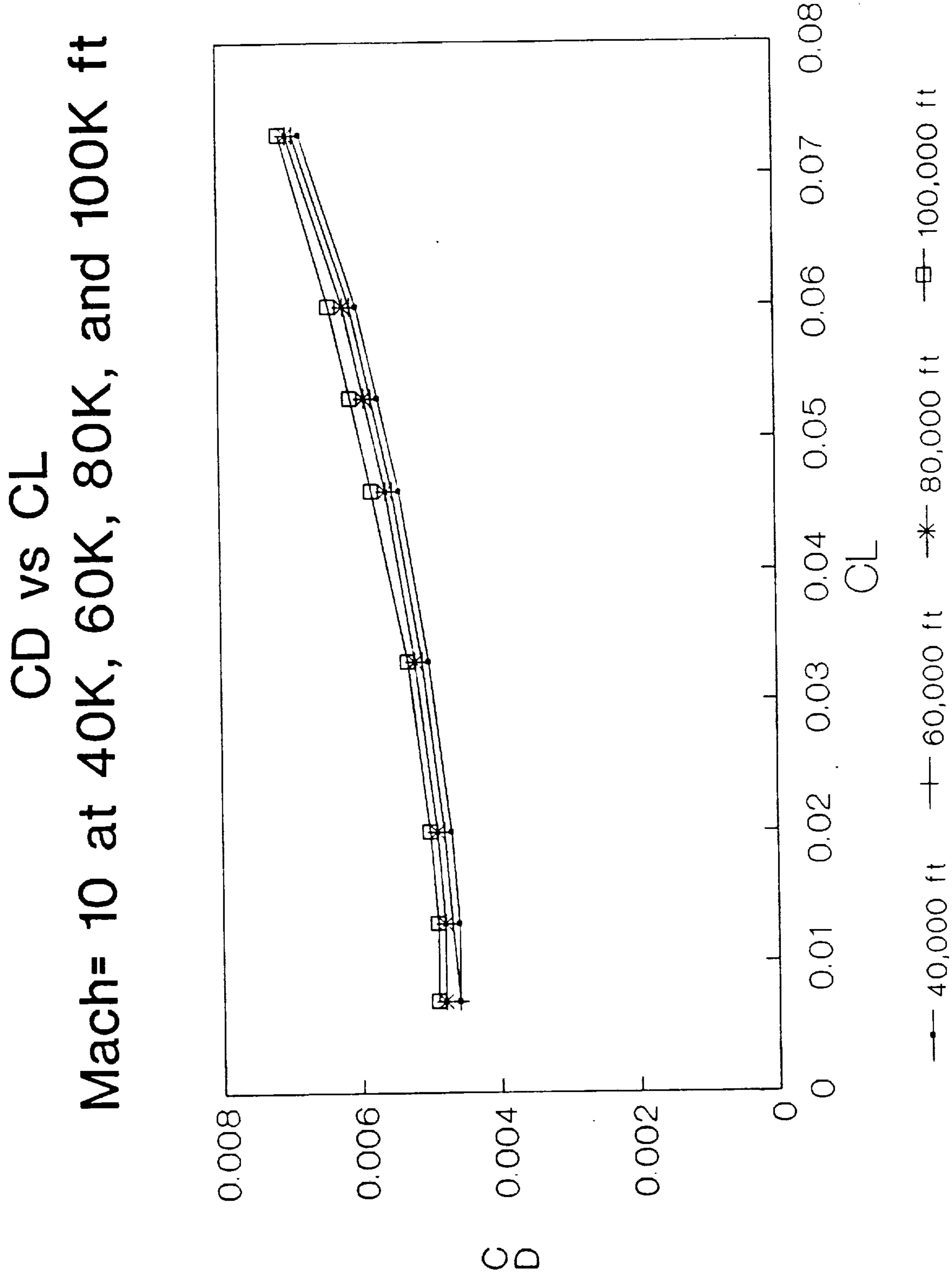
CD vs. CL - Mach 10.0

Figure 5-13



6.0 Stability and Control Analysis

6.1 Weight Analysis

A preliminary estimate of the take off weight of the aircraft was made using historical trends for past experimental and high speed aircraft. This method was outlined in Reference 9. This preliminary value for the take-off weight was then used in subsequent configuration calculations. After a configuration was determined, it was possible to determine a more accurate take-off weight for the aircraft using a component breakdown. A new configuration was then calculated using the value for the take off weight determined by the component breakdown method. This process was repeated several times to produce an accurate result. The final value of the take off weight was 53,000 lbs.

The basic design of the aircraft was determined by the waverider program plots. Given the basic outline of the aircraft, it was then necessary to configure the aircraft for optimum stability and fuel volume. A preliminary design proved to be inadequate for meeting the fuel requirement, so the entire design was enlarged. The final configuration consisted of a front mounted nose wheel, followed by the cockpit, with the twin turbofanramjets placed side by side in the rear. Conformal fuel tanks were placed in the wing area of the aircraft. In addition to meeting our fuel volume requirements, the final configuration also included several mission considerations. The turbofanramjets were mounted side by side to increase engine out stability and to allow one inlet to be used for both turbofanramjet engines. An inlet system length of eleven feet was accounted for in the configuration. The cockpit dimensions included adequate space for the data gathering computers and the cockpit cooling systems. In addition, a coverable window was included for use during landing. The nose wheel was mounted with a forward hinge to permit the wheel to be in proximity with the cockpit cooling system. The placement of the fuel tanks allowed for a small center of gravity (c.g.) travel. This was because the fuel tanks had a c.g. which was very close to the c.g. of the structure.

Once the final configuration was determined, it was possible to perform a component breakdown to determine the weight of the aircraft. The basic areas of weight on the aircraft were; structure, engines, fuel, and systems. The fuel weight was determined by the amount of fuel required for the mission (13960 lbs), and the basic weights for the engines were determined to be 10000 lbs for the ramjets, and 6000 lbs for the SCRAMjets. Estimates for the weight of the remainder of the components were made using the weight estimation methods outlined Reference 9. These estimates were based on parameters involving the geometry, aerodynamics and estimated take-off weight of the aircraft. The weight values for the inlet, vertical tails, computer systems and cooling systems were estimates made from historical trends, combined with information determined in our studies of these components. Pie graphs of the weight breakdowns for total weight, structural weight and systems weight are shown in Figures 6-1 through 6-3.

Once the component weights were determined, it was possible to determine the c.g. of the empty plane by performing a moment balance on the plane. This involved adding the product of the mass and c.g. distance of each component, and then dividing this sum by the total mass of the aircraft. The empty c.g. of the aircraft was determined to be 53.2 feet from the nose of the aircraft. The c.g. of the full plane was determined by also adding in the fuel mass-c.g. product into the c.g. determination. The c.g. of the full aircraft was found to be 51.4 feet from the nose of the aircraft. Thus, the total distance traveled by the c.g. was only 1.8 feet.

The component breakdown of the weight of the aircraft, and the c.g. determination of the aircraft are presented in Appendix H.

6.2 Stability and Control

6.2.1 Overview of Stability and Control

In determining the stability of the aircraft, the main concern was for static stability at low speeds, particularly landing speeds. The main reason for this concern was due to the fact that the aerodynamic center was at it's most forward position at low speeds, which produced the smallest static margin, and therefore the least stability. Also, the stability of the aircraft would be critical during landing to ensure a safe return for the aircraft and the pilot. Dynamic stability was not checked, but stability augmentation systems will be used to correct dynamic instabilities. The static stability of the aircraft was determined using the methods presented in Reference 9. The various stability derivatives were determined using References 9, 10, and the USAF DATCOM volumes on aircraft stability. Since the aircraft had no horizontal tail, elevons were chosen to provide both longitudinal and lateral control. For directional control a pair of vertical stabilizers were mounted in the rear of the aircraft. For this configuration, it was necessary to use the stability equations for a tailless aircraft. The aircraft was checked for stability in all three planes of motion: longitudinal, lateral and directional. The stability of each of these planes will be discussed separately. The aircraft was found to be stable in all three planes, and some performance criteria were determined. A listing of the important static stability parameters is shown in Table 6-1, and a complete listing of the stability and control calculations is provided in Appendix I.

6.2.2 Longitudinal Stability

The primary factor in determining longitudinal static stability was the distance between the aerodynamic center and the center of gravity, or the static margin. For a tailless aircraft, the aerodynamic center and the center of gravity are the same point, and for longitudinal stability it is essential that the center of gravity, or c.g., be in front of the aerodynamic center. However, if the static margin were to become too large, the aircraft would be too stable, and thus hard to control. To prevent this it was necessary to ensure that the static margin remain relatively constant throughout the flight. This will be accomplished by fuel staging to match the flight profile of the aircraft.

The positive longitudinal stability criterion is that the slope of C_M vs. angle of attack, or C_{MA}, be negative. This insures that the aircraft will tend to correct its pitch attitude when disturbed from the equilibrium position. The value of C_{MA} determined for the aircraft was -0.105, thus meeting this condition. Also, the value of C_{MCG} at zero angle of attack should be positive to ensure longitudinal stability. The value of C_{MCG} at zero angle of attack was determined to be 0.03, and this condition was met also. A plot of C_{MCG} vs. angle of attack is shown in Figure 6-4.

6.2.3 Lateral Stability

The primary requirement for lateral stability was that the lateral stability derivative, or $C_{L\beta}$, be negative. This requirement not only insures positive lateral static stability, but also keeps the spiral mode from becoming divergent. The value of $C_{L\beta}$ determined for the aircraft was -0.726, thus meeting this condition. For calculation of the steady state roll the primary criterion was the damping in roll coefficient, or C_{LP}. A value of -0.105 was determined for C_{LP}, thus producing a roll rate of 87 deg/sec upon landing. This high roll rate was due to the low aspect ratio of the aircraft, and also because the elevons were sized for longitudinal stability, which provided a great deal more lateral control that would otherwise be present. A plot of roll rate vs velocity is shown in Figure 6-5.

6.2.4 Directional Stability

The primary requirement for directional (weathercock) stability was that the directional stability derivative, or $C_{N\beta}$, be positive. This requirement insures that the moments generated will rotate the aircraft to reduce the sideslip angle, or β. $C_{N\beta}$ was determined to be 0.102, thus this condition was met, also.

For sizing the aircraft rudders, it was necessary to check for adequate control under two conditions: maintain ß of zero for one engine out, and maintain ß of zero for landing in a crosswind. From this, a maximum value of τ was determined, and the rudder area could be calculated. The total rudder area was about 125 square feet, or 0.28 of the total vertical stabilizer area.

6.2.5 Trim Requirements

The trim requirements were determined from the elevon control power, or ECP. The required elevator deflections for flight at various angles of attack could be determined by use of the ECP. A plot of the required elevator deflection vs angle of attack is shown in Figure 6-6.

The turn radius required for various maneuvers was also calculated using force balance equations in combination with the control parameters. The maneuvers were the turn, pull-up and pull-down. The maneuvers were calculated for an artificial gravity of 3 g's. Plots of the radii vs velocity are shown in Figure 6-7.

Table 6-1

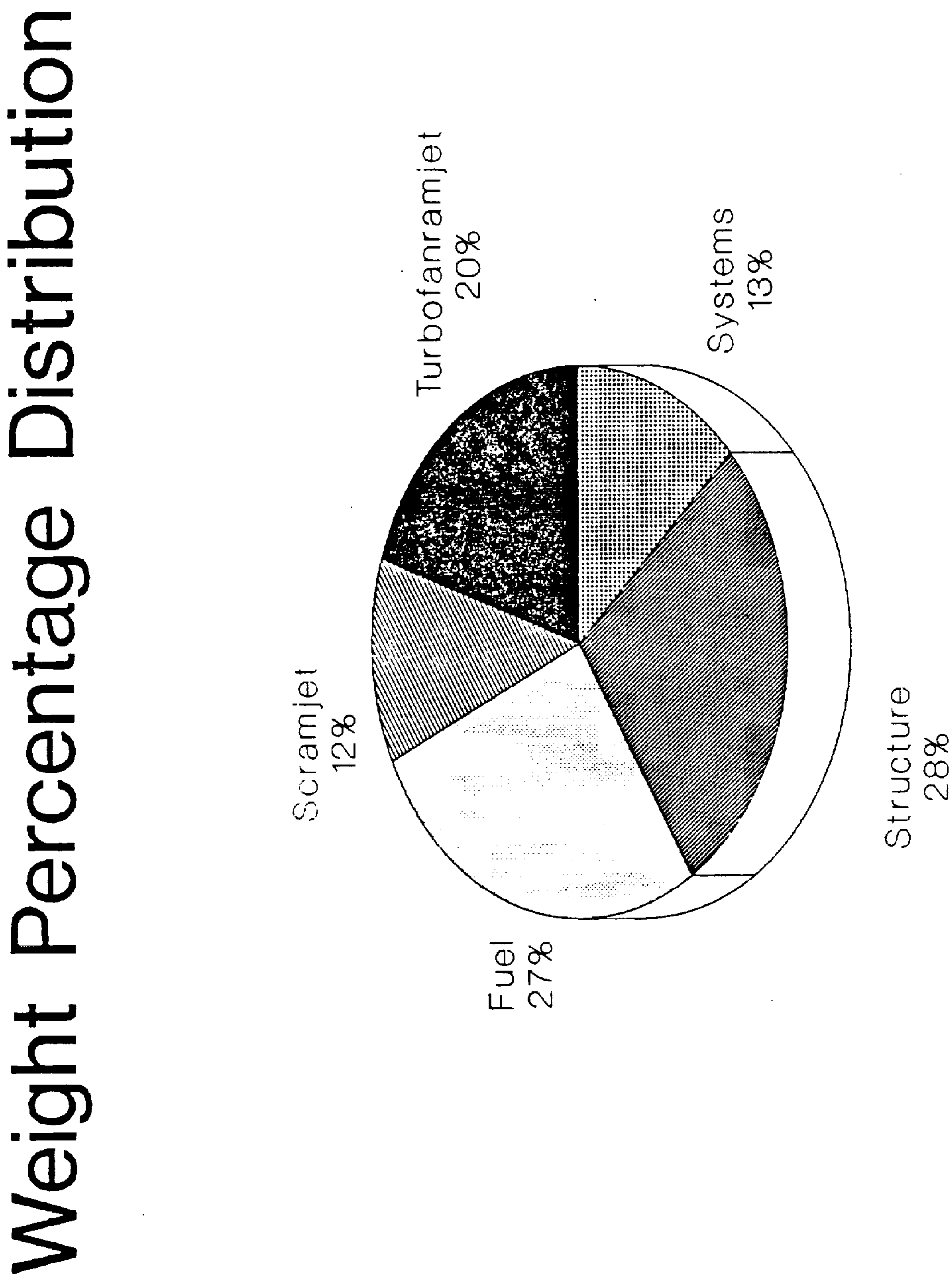
STATIC STABILITY PARAMETERS		NEEDS TO BE +/- FOR STABILITY
LONGITUDINAL STABILITY		
Static Margin (Fully Fueled):	0.139	+
Static Margin (Empty):	0.094	+
CMA (Slope of CM vs Alpha):	- 0.105	-
CM at Zero Angle of Attack:	0.030	+
LATERAL STABILITY		
Rolling Moment Coefficient:	0.069	+
Lateral Stability Derivative:	- 0.726	-
DIRECTIONAL STABILITY		
Directional Stability Derivative:	0.102	+

6.3 Subsonic Wind Tunnel Testing

Subsonic wind tunnel testing was to be completed using the subsonic wind tunnel model. When mounting was attempted, certain problems were found. First, the model was too long and the stinger was not long enough to reach the model center of gravity. Next, the model has a large flat plate lifting surface. This would be a problem when the model was placed at high angle attack possibly destroying the delicate strain gages under high loads. With these problems, the model needed to be torn apart and a different mounting technique used. Since there was little time left in the project, these modifications could not be made and the subsonic wind tunnel testing could not be completed.

Weight Percentage Distribution

Figure 6-1



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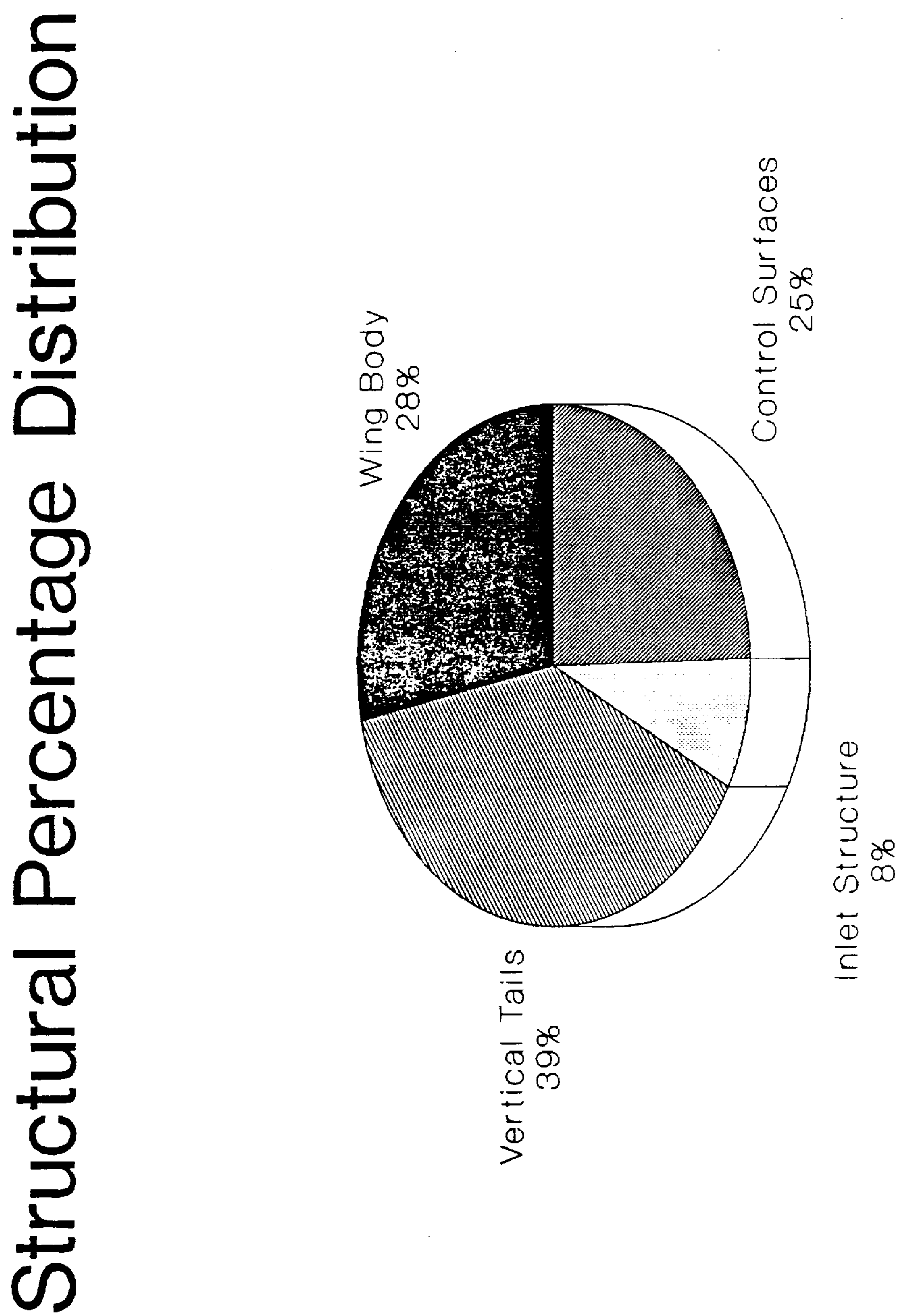
Structural Percentage Distribution

Figure 6-2



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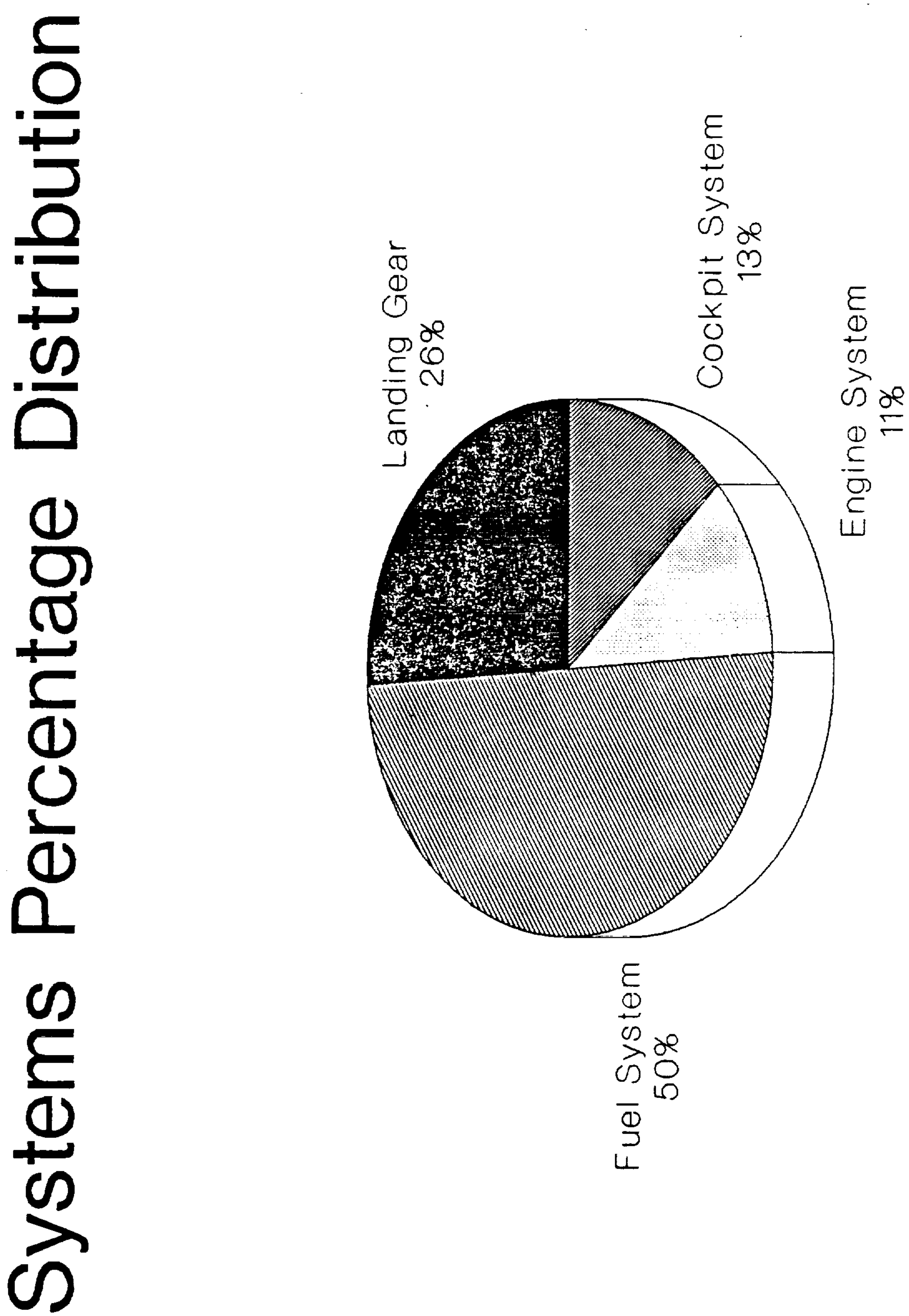
Systems Percentage Distribution

Figure 6-3



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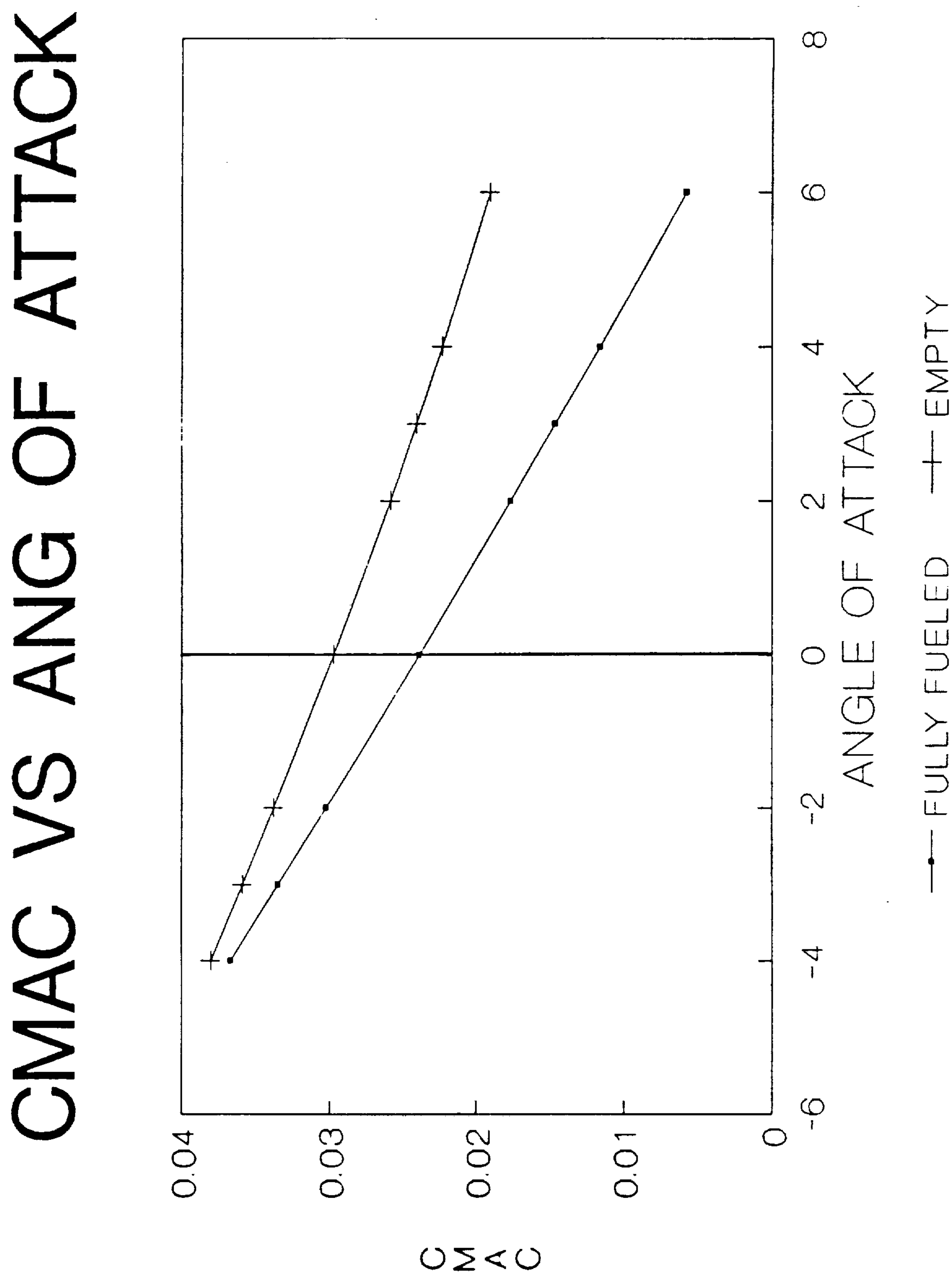
CMAC vs. Angle of Attack

Figure 6-4



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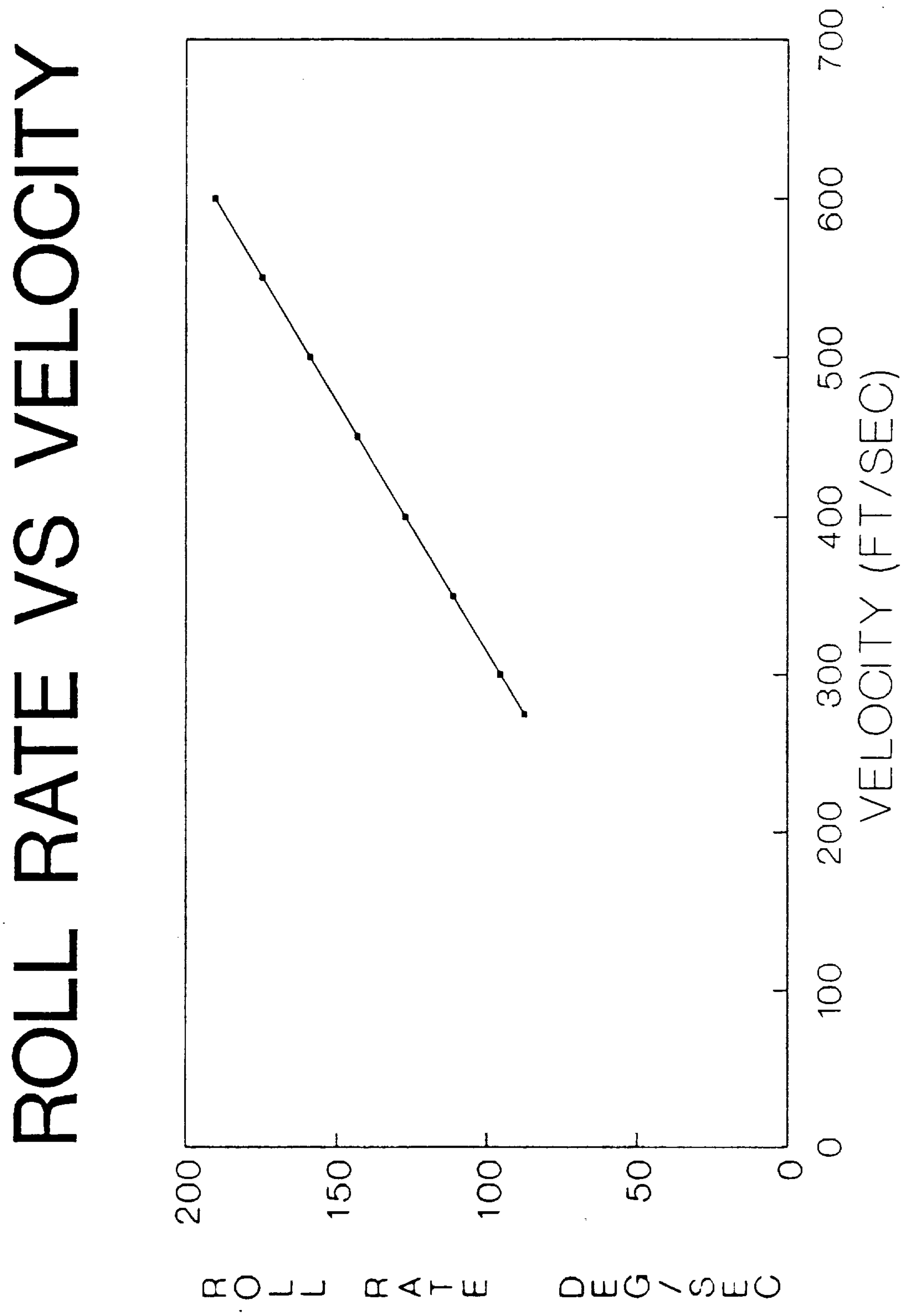
Roll Rate vs. Velocity

Figure 6-5



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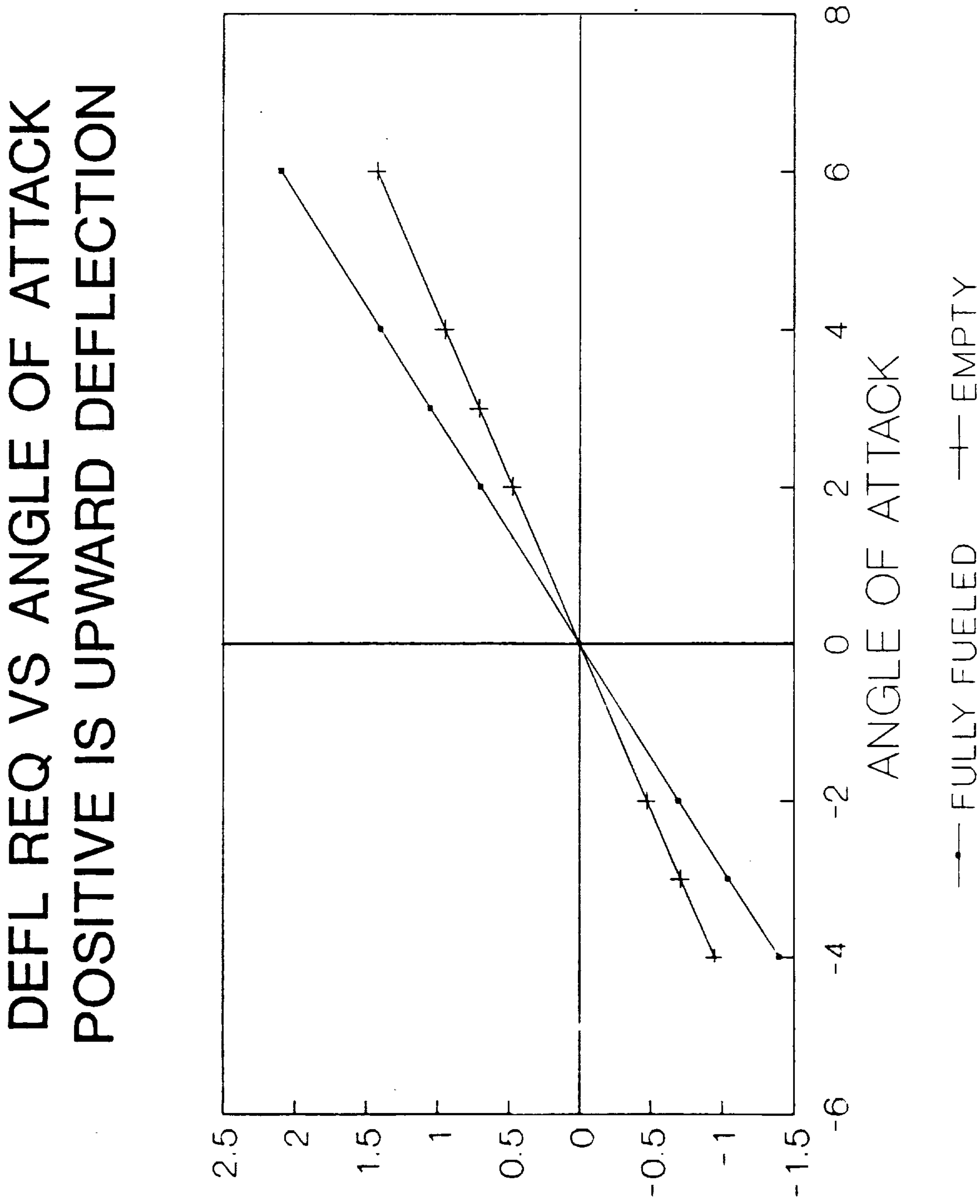
Deflection Required vs. Angle of Attack

Figure 6-6



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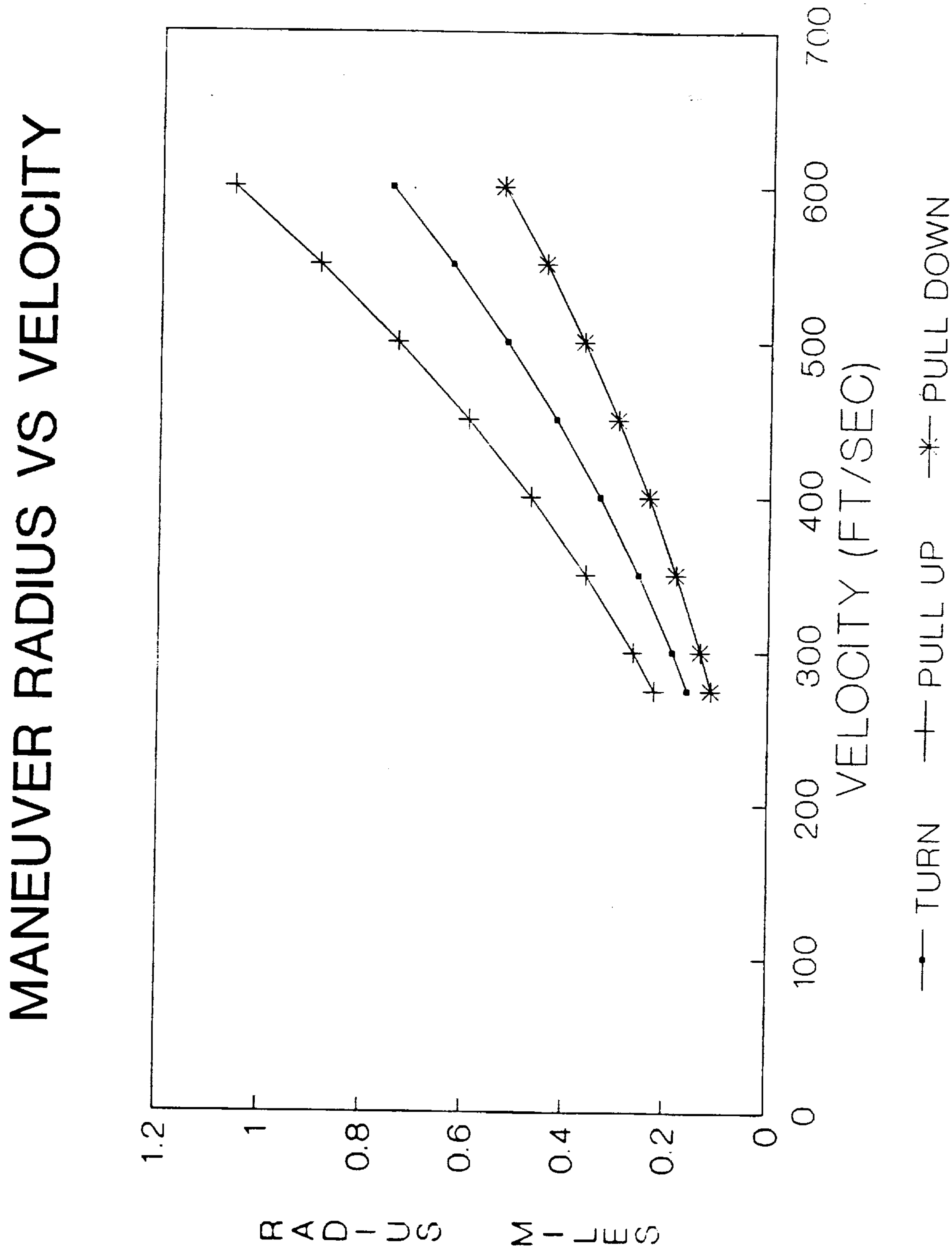
Maneuver Radius vs. Velocity

Figure 6-7



7.0 Aircraft Systems

7.1 Cockpit

A pilot was needed for the aircraft for a number of reasons. Some of these reasons were; performance of landing maneuvers, and handling any problems that might occur during the high speed ionization communications blackout. The cockpit was designed to accommodate both the pilot and the various computers needed to record the experimental flight data. The cockpit was sized to an average length, width and height of 11', 5', and 5' respectively. Included in this space were provisions for cockpit instrumentation, an oxygen system, an air conditioning system, 200 lbs. of on-board flight computers, emergency equipment, and an ejection seat. A glass window will be included for use at velocities up to Mach three. For higher speeds a metal covering plate will protect the observation window. The pilot will be wearing a pressure suit, so ejection may be accomplished for speeds up to Mach two (at high altitude). Weights of the various cockpit furnishings are shown in Appendix H.

7.2 Landing Gear

Due to an inadequate allotment of time in the previous quarter, one of the areas that needed to be addressed this quarter was landing gear. Reference 10 was used for this analysis. This analysis encompassed many aircraft variables such as landing speed, center of gravity location, and length of aircraft. Possible locations for the landing gear were taken from the drawing of the aircraft which indicated where there was adequate room to store the gear. The location for the nose gear was set at 17 feet from the nose of the aircraft. The location of the main gear (skids) was set at 65 feet from the nose. A decision to use two tires for the nose gear was made to aid in the controllability of the aircraft upon landing. Skids were used for the rear because there was not adequate space in the aircraft to house conventional wheeled gear. Drawings of the nose gear and the main gear during operation are shown in Figures 7-1 & 7-2 respectively. Reference 10 used an energy method to determine the sizing of the shock absorber. The total amount of energy that needed to be absorbed at touch-down was determined. The energy absorbed by each set of gear was set to the total value that needed to be absorbed. The reason for this was because one set will touch down before the other one does.

There were many variables in determining the size of our shock absorbers. Table 7-1 shows some of the values that were used in the analysis.

Table 7-1: Landing Gear Design Factors

Vertical landing speed	= 10 ft/sec	
Landing load factor	= 8	
Absorbing medium efficiency	= 0.85	(liquid springs)

After these values in Table 7-1 were decided upon, the analysis could be completed. Values of the sizes of the shock absorbers are shown in Table 7-2. The length of the struts shown in the drawings is arbitrary as long as the shock absorber stroke is met, and there is no danger of the aircraft contacting the landing surface upon full compression of the shock absorber.

Table 7-2: Landing Gear Parameters

Nose gear (2 tires)	
Required shock absorber length (stroke)	= .992 feet
Required shock absorber diameter	= .311 feet
Energy absorbed	= 66770 BTU
Main gear (skids)	
Required shock absorber length (stroke)	= .737 feet
Required shock absorber diameter	= .354 feet
Energy absorbed	= 66770 BTU

Landing Gear - Front Nosewheel

Figure 7-1

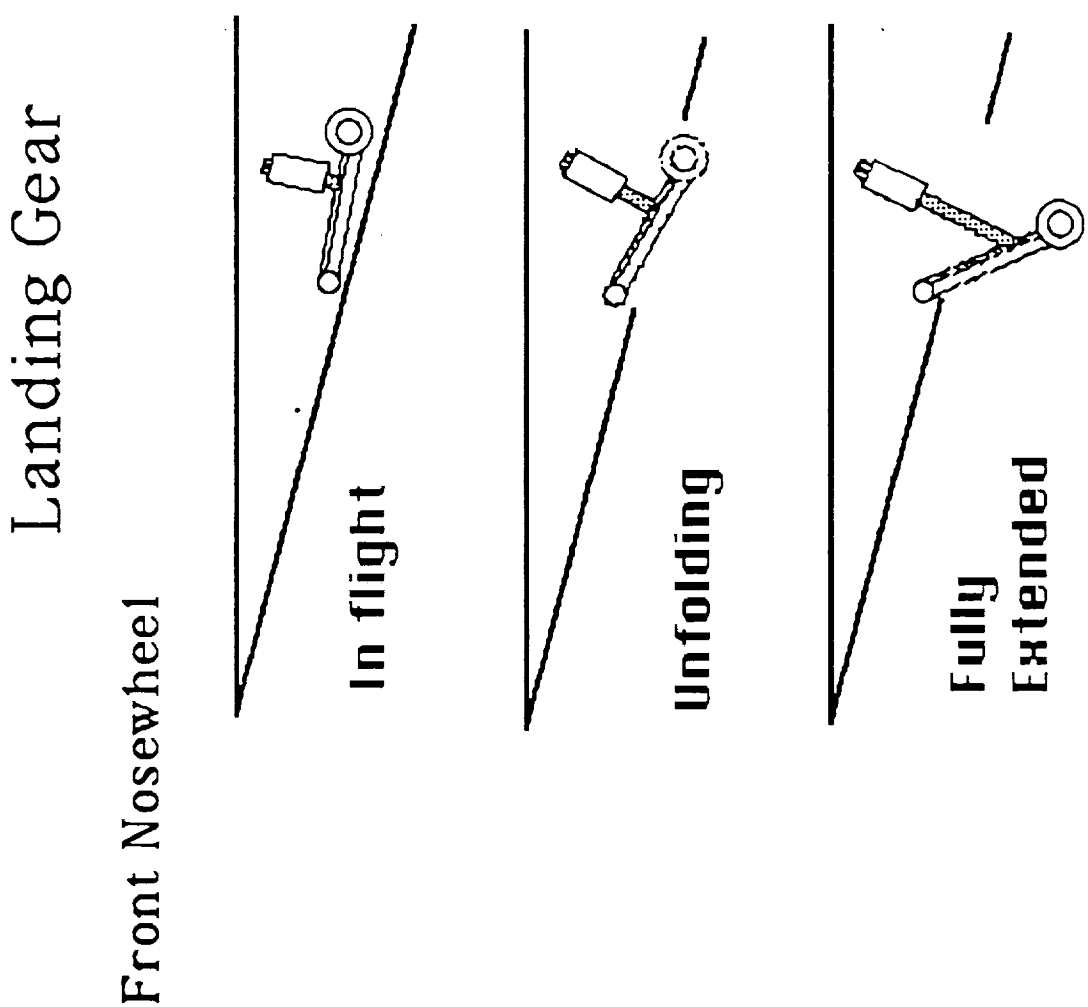
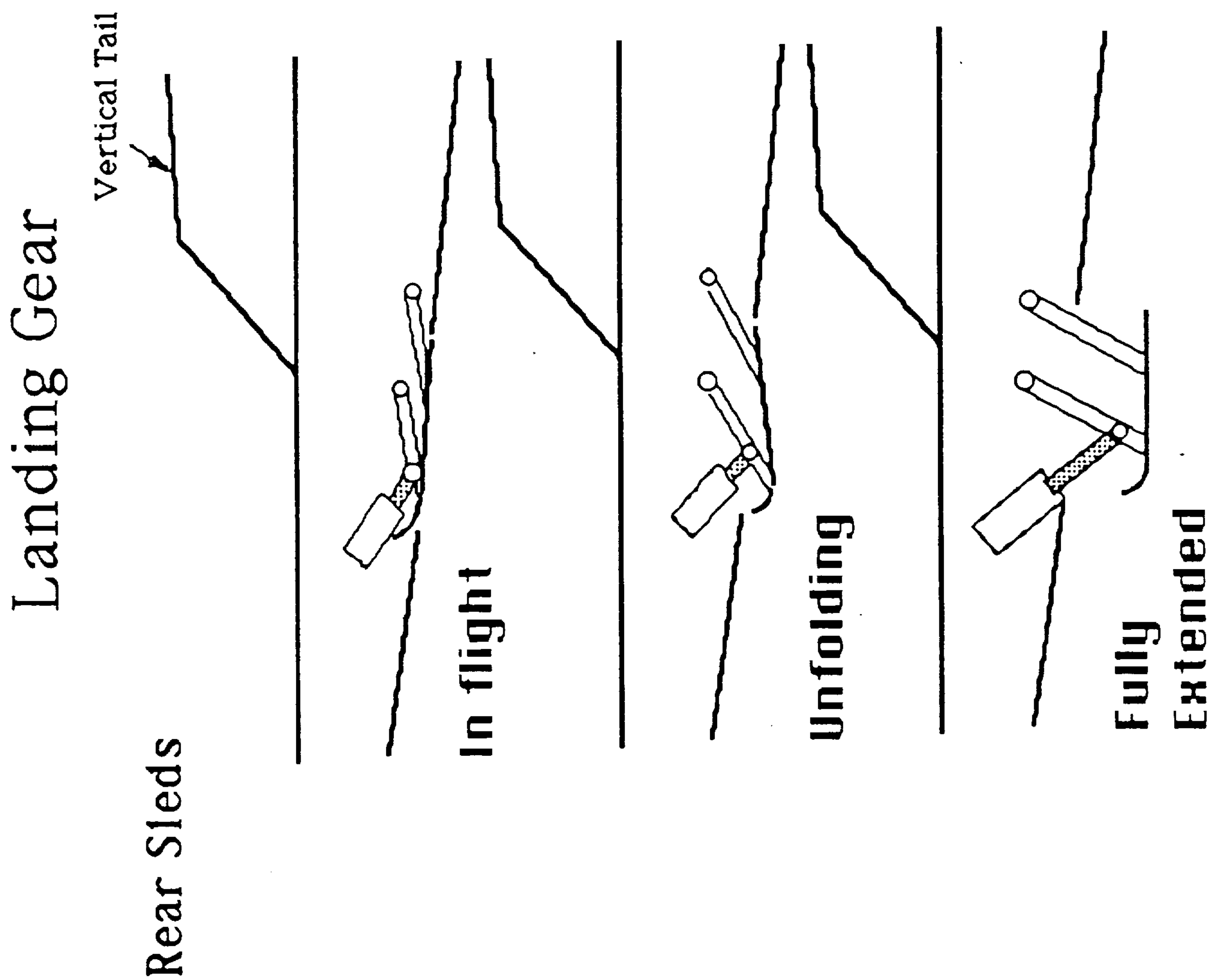


Figure 7-2


Landing Gear
Rear Sleds
Vertical Tail
In flight
Unfolding
Fully
Extended

8.0 Environmental Aspects

The major reason that supersonic or hypersonic aircraft cannot fly over land is because of a sonic boom which creates overpressures. This sonic boom is due to the sudden rise and fall of pressure waves. The maximum increase in pressure is called the overpressure, and is measured in terms of psf. An overpressure of less then 1.0 psf is considered relatively safe.

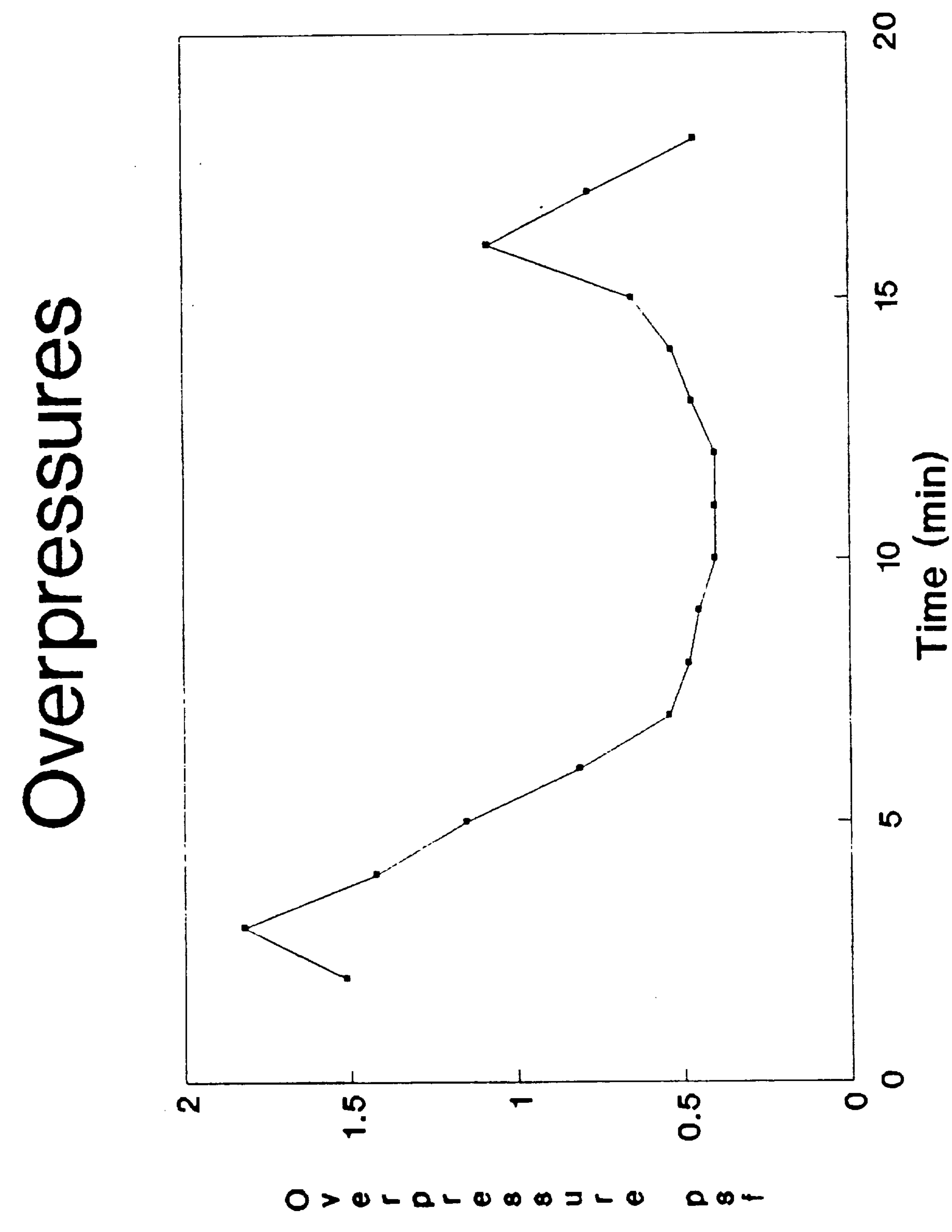
There are many ways that the overpressure of an aircraft can be calculated. A method by Morris was chosen. The rise in pressure is due to either volume or lifting effects, whichever is greater. It was determined that the rise in pressure due to lift would be greater, and the following equation was used:

$$dP_L = 0.363\,K_R\sqrt{\frac{P_a}{P_g}}\,K_L\frac{(M^2-1)^{0.375}}{M}\sqrt{W}\,l_W^{-0.25}\,P_a^{-0.5}\,h^{-0.75}\,Pg$$

A plot of the overpressure related to the time in flight can be seen in Figure 8-1.

Because a sonic boom can break glass, frighten animals, and harm the human eardrum, it was decided that the flight path of *STB-White* must take place over water. Even though the overpressures created are almost always less then 1.0 psf, it was decided to keep the flight far from land for safety purposes.

Figure 8-1


Overpressures
Time (min)
0
5
10
15
20
Overpressure psi
0
0.5
1
1.5
2

9.0 Economic Analysis

An economic analysis was performed using the methods outlined in Reference 9. The calculations were done using 1974 dollars, and then converted using a consumer price index value determined from Fundamentals of Macroeconomics by L. Fleisher. Three basic parameters were used for estimating the aircraft cost: the AMPR weight, the maximum speed in knots and the quantity of aircraft produced. The aircraft had an AMPR weight of 20100 lbs, and a maximum speed of 5935 knots. Calculations were performed to determine the cost of producing either one or two aircraft, and these are compared in Table 9-1. The cost of producing an additional proto-type is about 1.15 billion dollars in addition to the base cost of 5.45 billion dollars. From this information, it would appear that a second aircraft would indeed be produced to aid in the SCRAMjet testing.

A pie chart of the aircraft cost breakdown is shown in Figure 9-1. From this figure it can be seen that the majority of the cost for the aircraft is in tooling costs. A pie chart of the entire development cost breakdown is shown in Figure 9-2. In this figure it is apparent that the main expenditure is for the engineering required to develop the aircraft. Once the engineering and tooling has been completed for one aircraft, it is relatively easy to produce other aircraft at a much reduced cost. This is the reason for only an approximate increase of 20% for an additional aircraft.

This cost estimation may be somewhat optimistic considering that many new technologies will need to be developed to complete the aircraft. A listing of economic analysis is shown in Appendix J.

Table 9-1: Aircraft Development Costs

TOTAL DEVELOPMENT COST TO PRODUCE ONE AIRCRAFT		
AIRFRAME ENGINEERING		$2,631,030,667
DEVELOPMENT SUPPORT		$1,958,839,157
AIRCRAFT		$805,571,066
ENGINES & AVIONICS	$17,202,500	
MANUFACTURING LABOR	$126,512,434	
MATERIAL & EQUIPMENT	$16,298,046	
TOOLING	$629,111,470	
QUALITY CONTROL	$16,446,616	
FLIGHT TEST OPERATIONS		$55,340,978
TEST FACILITIES		$0
	TOTAL COST =	$5,450,781,868

TOTAL DEVELOPMENT COST TO PRODUCE TWO AIRCRAFT		
AIRFRAME ENGINEERING		$2,986,856,785
DEVELOPMENT SUPPORT		$2,489,746,643
AIRCRAFT		$996,027,024
ENGINES & AVIONICS	$17,202,500	
MANUFACTURING LABOR	$181,916,851	
MATERIAL $ EQUIPMENT	$28,219,628	
TOOLING	$745,038,854	
QUALITY CONTROL	$23,649,191	
FLIGHT TEST OPERATIONS		$134,482,648
TEST FACILITIES		$0
	TOTAL COST =	$6,607,113,101

Figure 9-1

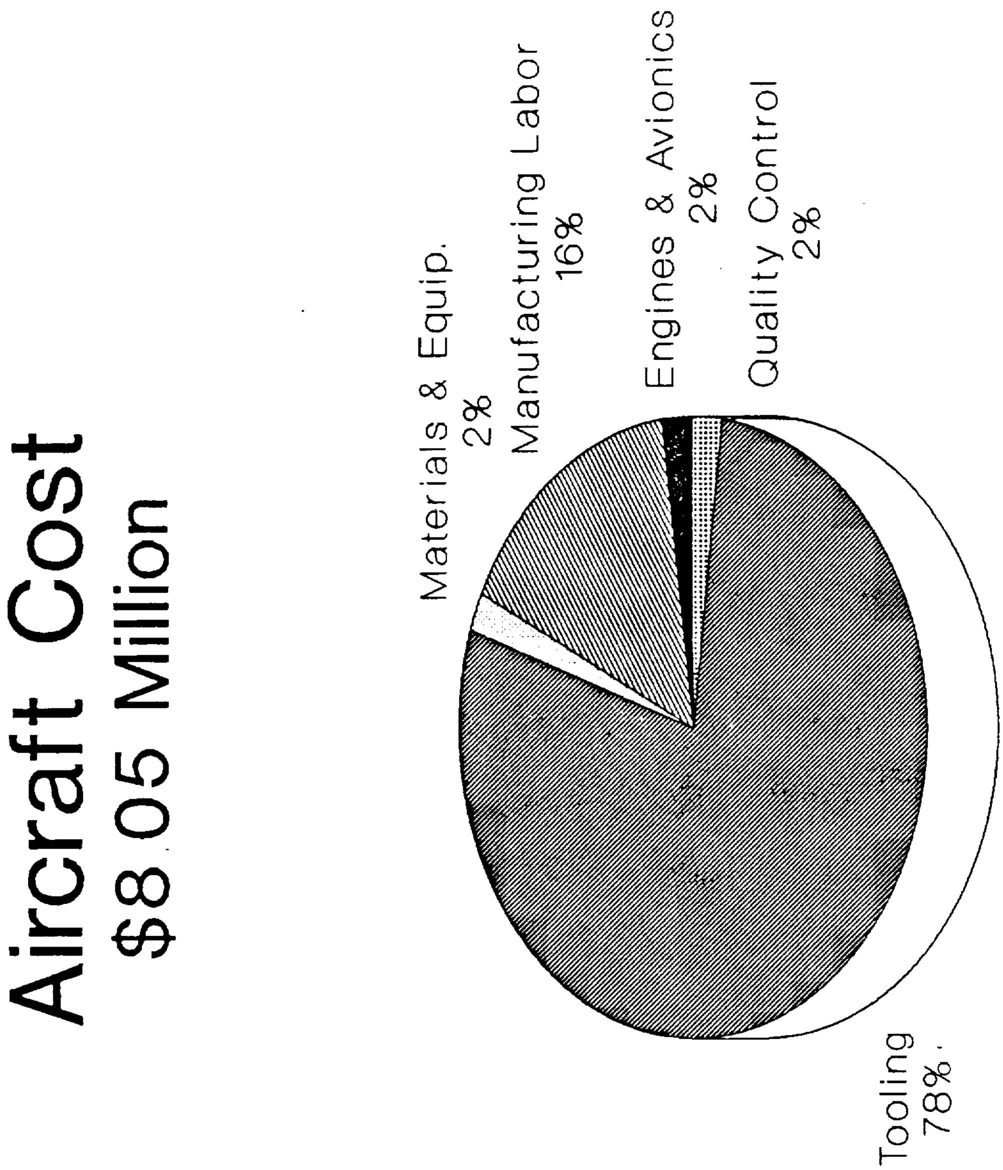
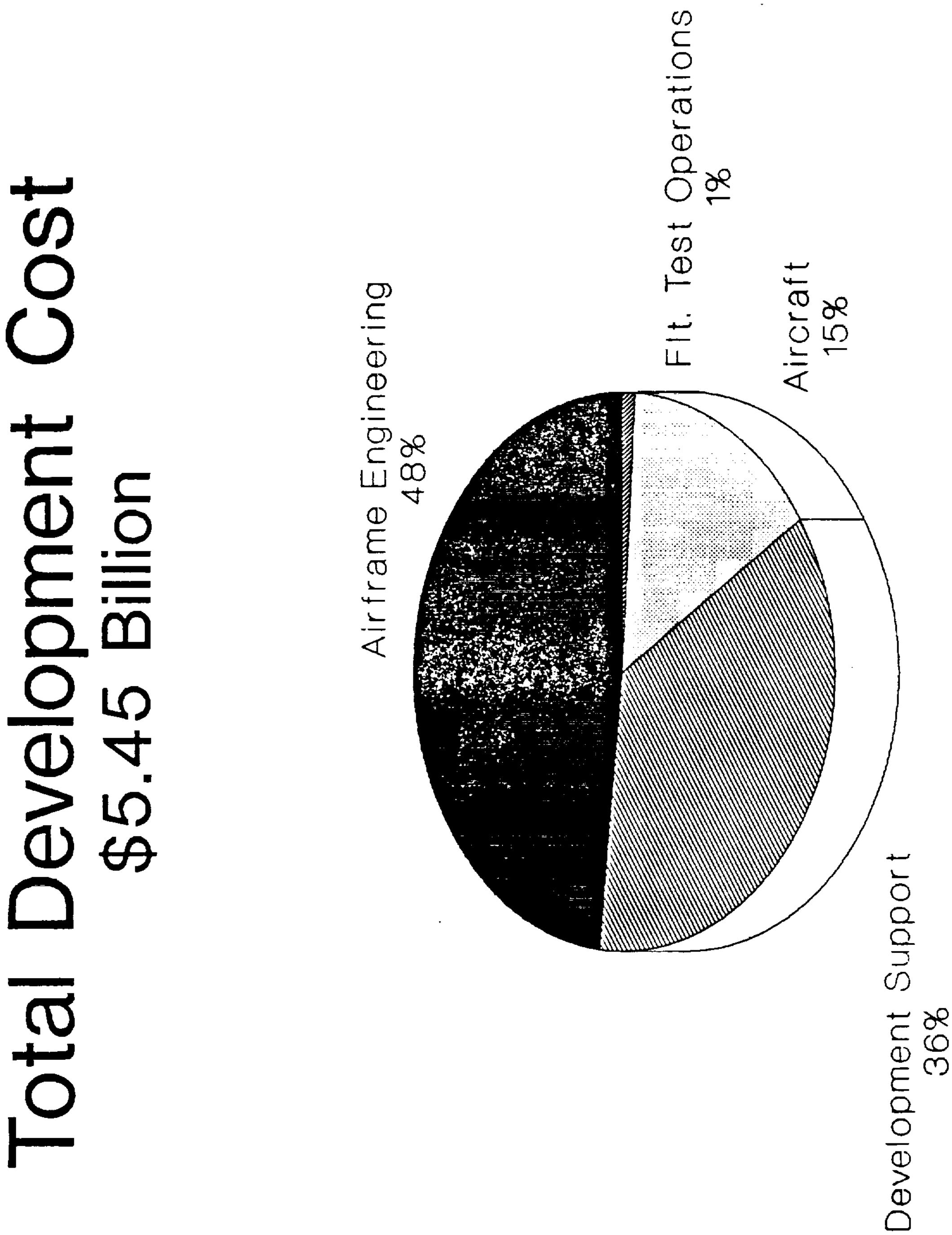

Aircraft Cost
$8.05 Million
Materials & Equip.
2%
Manufacturing Labor
16%
Engines & Avionics
2%
Quality Control
2%
Tooling
78%

Figure 9-2


Total Development Cost
$5.45 Billion
Airframe Engineering
48%
Flt. Test Operations
1%
Aircraft
15%
Development Support
36%

CONCLUSIONS

With design work complete, the configuration of the *STB-White* has been finalized. Implementing propulsion, waverider studies with aerodynamics, cockpit, and landing systems design, the aircraft configuration is complete. The aircraft utilizes two General Electric turbofanramjet engines and five full-size SCRAMjet combustion cells, also from General Electric, occupying a Mach 10.0 modified waverider configuration. The turbofanramjets use 6,460 pounds of liquid hydrogen to carry the aircraft from Mach 0.8 at an altitude of 40,000 feet to Mach 6.0 at 80,000 feet. The five SCRAMjets are then activated accelerating the aircraft from Mach 6.0 to Mach 10.0 at 100,000 feet. The SCRAMjets hold the aircraft at this altitude and Mach number for two minutes during which important data is collected and after which all engines are shut down for a glide descent and landing. The total fuel expenditure for the SCRAMjet phase is 7428 pounds of liquid hydrogen. Since the modified Mach 10.0 waverider has roughly 6,500 cubic feet internal volume, the 13,828 total pounds of fuel at 3,121 cubic feet of fuel volume can be contained in an airframe/fuel tank integrated structure. Problems involving the pressurization of the fuel and building the airframe from materials resistant to liquid hydrogen corrosion are solved. The aircraft is configured with a diamond-shape planform with a length of 85 feet, maximum width of 48 feet, and height (including vertical tails) of 14 feet. A planform area of 2,100 square feet gives the aircraft an aspect ratio of roughly 1.1. The cockpit will include avionics, pilot ejection systems, a pilot pressure suit, and a cockpit cooling system. The landing system will include a front nose wheel for directional control upon landing and rear sleds. Furthermore, the landing gear configurations will be as simple and compact as possible to conserve space and reduce weight. With these design aspects complete, the *STB-White* design is complete.

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Appendix A

SCRAMjet Information

Appendix A:
SCRAMjet Information

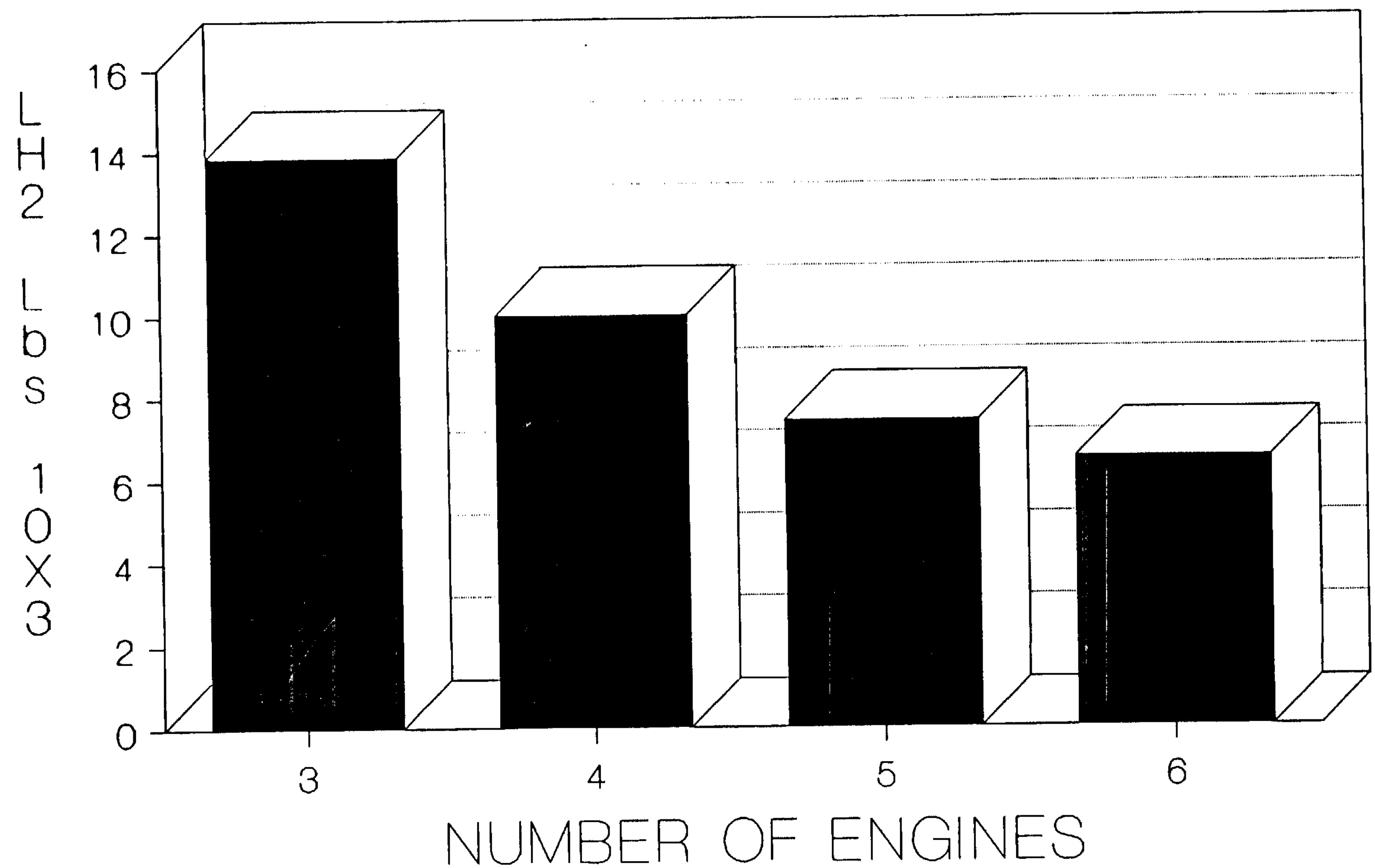
The following material does not represent the present conditions of the aircraft. It has been included to show how the process of selection was reached and should not be taken as an accurate depiction of fuel weights, accelerations, drags or any thing else. It was not revised because the employment of five engines had been finalized and only the corresponding data was revised throughout the quarter. A graph of total fuel versus number of SCRAMjets can be seen in Figure A-1. This figure should put the importance of the iteration procedure in perspective. The following page contains a section of the spreadsheet which is discussed in the preceding sections. This page contains all of the information for the cases of 3 to 6 engines. It is broken up in to the four divisions previously mentioned. Under the column titled "MACH #", the first line in each division represents the range over which the data is used. Halfway down the column in each division is the Mach number that was used to calculate data in that division. The accuracy of this spread sheet could be greatly increased if data was available allowing 40 or 50 divisions as opposed to the 4 divisions the author has developed. The acceleration is dependent upon the mass of the airplane. The amount of fuel used in the previous section by five engines is subtracted from the total mass of the airplane. This allows for a constantly changing mass. The mass of the fuel from five engines was subtracted because that appeared to be the number of engines that would be used.

3/6/90 PJG SCRAMJET INFORMATION

# ENG	NET F	ISP	TIME	WEIGHTf	ALT(ft)	MACH #	ACC
5	8500	3000	0.023904	1219.112	80000	6-6.75	8.436466
6	8500	3000	0.014498	887.2916	80000		13.90974
3	8500	3000	-0.08034	-2458.47	80000		-2.51008
4	8500	3000	0.068057	2776.737	80000		2.963189
5	11500	1800	0.011144	1281.648	80000		18.09519
6	11500	1800	0.007908	1091.363	80000		25.50021
3	11500	1800	0.061387	4235.731	80000	6	3.285146
4	11500	1800	0.018864	1735.550	80000		10.69016
5	12500	1250	0.009461	1703.044	80000		21.31476
6	12500	1250	0.006867	1483.464	80000		29.36370
3	12500	1250	0.038656	4174.899	80000		5.216891
4	12500	1250	0.015201	2189.082	80000		13.26582
5	6600	2497	0.065029	3093.899	88875	6.75-8.25	6.300590
6	6600	2497	0.038641	2206.160	88875		10.60306
3	6600	2497	-0.17780	-5075.60	88875		-2.30436
4	6600	2497	0.205054	7804.718	88875		1.998113
5	9337	1570	0.026916	2881.417	88875		15.22171
6	9337	1570	0.019228	2470.016	88875	7.5	21.30841
3	9337	1570	0.134409	8633.005	88875		3.048308
4	9337	1570	0.044851	3841.056	88875		9.135010
5	10236	1106	0.022571	3760.223	88875		18.15195
6	10236	1106	0.016504	3299.394	88875		24.82471
3	10236	1106	0.085244	8520.465	88875		4.806457
4	10236	1106	0.035692	4756.803	88875		11.47920
5	5300	2150	0.055838	2477.693	97000	8.25-9.25	4.974620
6	5300	2150	0.032531	1732.166	97000		8.538843
3	5300	2150	-0.12896	-3433.58	97000		-2.15382
4	5300	2150	0.196950	6991.270	97000		1.410396
5	8000	1350	0.019766	2108.374	97000		14.05330
6	8000	1350	0.014293	1829.623	97000		19.43326
3	8000	1350	0.084344	5398.029	97000	9	3.293382
4	8000	1350	0.032026	2732.937	97000		8.673342
5	8500	980	0.017654	2756.187	97000		15.73453
6	8500	980	0.012949	2426.060	97000		21.45074
3	8500	980	0.064567	6048.268	97000		4.302125
4	8500	980	0.027726	3463.039	97000		10.01833
5	4673	1850	0.050613	2301.251	101340	9.25-10	4.116151
6	4673	1850	0.028438	1551.642	101340		7.325629
3	4673	1850	-0.09046	-2468.02	101340		-2.30280
4	4673	1850	0.229777	8357.850	101340		0.906673
5	7086	1247	0.016797	1718.125	101340		12.40255
6	7086	1247	0.012063	1480.717	101340	10	17.26931
3	7086	1247	0.078055	4790.301	101340		2.669035
4	7086	1247	0.027645	2262.178	101340		7.535793
5	8025	908	0.013331	2120.854	101340		15.62714
6	8025	908	0.009855	1881.442	101340		21.13881
3	8025	908	0.045252	4319.429	101340		4.603788
4	8025	908	0.020595	2621.166	101340		10.11546

WEIGHT FOR 2 MINUTE CRUISE
595.6027

NUMBER OF ENGINES VS. FUEL WEIGHT FOR SCRAMJET OPERATION



Appendix B:

Turbofanramjet Inlet Code

```
c*******************************************************************
      program TURBOFAN
c*******************************************************************
c     AERONAUTICAL & ASTRONAUTICAL ENGINEERING                     *
c     416  HYPERSONIC DESIGN                                       *
c     SPRING QUARTER 1990                                          *
c                                                                  *
c     AUTHORS: GOUHIN, P. & WHITE, H.                              *
c                                                                  *
c     PURPOSE: to AID IN THE DESIGN OF AN INLET FOR A TURBOFANRAMJET *
C              ENGINE ON A HYPERSONIC TEST BED VEHICLE.  THIS      *
c              CODE ALLOWS FOR THE ADDITION OF SEVERAL RAMPS, ON   *
C              THE UNDERSIDE OF THE VEHICLE, to INDUCE SHOCKS      *
c              to INCREASE PRESSURE RATIOS AND REDUCE FLOW         *
c              VELOCITY.                                           *
C                                                                  *
C     PRESENTLY: THE PROGRAM IS DESIGNED TO CONSIDER THREE RAMPS ON *
C              THE UNDER SIDE OF THE VEHICLE.  FROM THIS POINT THE *
C              FLOW CURVES INTO THE COMBUSTION CHAMBER AT SUBSONIC *
C              SPEEDS.                                             *
c                                                                  *
c     INPUT:   AS DEFINED IN THE program(set up for five ramps)    *
C              INTERACTIVE VARIABLES FOR FREESTREAM MACH NUMBER    *
C              AND RAMP ANGLES ARE INPUT FROM KEYBOARD.            *
c                                                                  *
C     OUTPUT:  DATA IS WRITTEN to TF.DAT                           *
c                                                                  *
c*******************************************************************
      implicit real *8 (a-h,l-z)
      integer i,j,k
      open(1,file='TF.DAT',STATUS='NEW')
30    format(5X,'VALUES FROM PROGRAM TURBOFAN FOR')
40    format(3X,'FREESTREAM MACH # = ',F6.3)
50    format(3x,'beta1=',f6.3,3x,'M2=',f5.3,3x,'Pt2/Pt1=',f6.4
     F,3X,'P2/P1=',F6.4,3X,'RO2/RO1=',F7.4)
60    format(3x,'beta2=',f6.3,3x,'M3=',f5.3,3x,'Pt3/Pt2=',f6.4
     F,3X,'P3/P2=',F6.4,3X,'RO3/RO2=',F7.4)
70    format(3x,'beta3=',f6.3,3x,'M4=',f5.3,3x,'Pt4/Pt3=',f6.4
     F,3X,'P4/P3=',F6.4,3X,'RO4/RO3=',F7.4)
80    format(3x,'beta4=',f6.3,3x,'M5=',f5.3,3x,'Pt5/Pt4=',f6.4
     F,3X,'P5/P4=',F6.4,3X,'RO5/RO4=',F7.4)
90    format(3x,'M6=',f5.3,3x,'Pt6/Pt5=',f6.4
     F,3X,'P6/P5=',F8.4,3X,'RO6/RO5=',F7.4)
100   format(3x,'beta6=',f6.3,3x,'M7=',f5.3,3x,'Pt7/Pt6=',f6.4
     F,3X,'P7/P6=',F6.4,3X,'RO7/RO6=',F7.4)
C105   format(17X,'M8=',F5.3,17X,'P8/P7='F6.4,'RO8/RO7=',F7.4)
110   format(5X,'THE TOTAL PRESSURE RECOVERY IS ',F6.4)
120   format(5X,'THE STATIC PRESSURE RECOVERY IS ',F8.4)
130   format(5X,'THE DENSITY RATIO IS ',F8.4)
140   format(5X,'THE ITERATED BETA2 IS ',F6.3)
150   format(5x,'THE MASS FLOW RATE IS ',F10.4)
160   format(5x,'THE PRESSURE IN THE COMBUSTION CHAMBER IS ',F10.4)
```

```
C       OTHERWISE KNOWN AS PI.
        pi = 3.14159265358979323846264338327950288
C      THE FREESTREAM FLOW IS DEFINED
        WRITE(*,*)" ENTER FREESTREAM MACH NUMBER!!"
        READ(5,*)M1
        WRITE(*,*)" ENTER AMBIENT TEMPERATURE!!"
        READ(5,*)T1
        WRITE(*,*)" ENTER FREESTREAM DENSITY!!!!"
        READ(5,*)RO1
        RO1=RO1*32.2
        WRITE(*,*)" ENTER AMBIENT PRESSURE!!!!!!!!!"
        READ(5,*)P1
        AREA=3059./144.0
        write(1,30)
        write(1,40)M1
C     THETA1 IS THE HALF-WEDGE NOSE ANGLE OF THE VEHICLE
        WRITE(*,*)" ENTER WEDGE ANGLE OF VEHICLE!!!!"
        READ(5,*)THETA1
        theta1 = THETA1*pi/180.
C     BETA IS NEVER EXPECTED to EXCEED 70 DEG. AND THUS IT IS
C      ITERATED DOWN FROM THIS INITIAL guess
        beta1 = 70.*pi/180.
C     LEFT AND RIGHT REFER to THE L.H.S & R.H.S. OF THE THETA-
C     BETA-MACH NUMBER EQUATION
        left1 = dtan(theta1)
        rt1 = 1.0
    5   beta1=beta1*dabs(left1/rt1)**0.1
        rta=((m1**2*dsin(beta1)**2)-1)/(m1**2*(1.4+dcos(2*beta1))+2)
        rtb=2/dtan(beta1)
        rt1=rta*rtb
C     if THE TWO SIDES ARE NOT EQUAL WITHIN THIS RANGE then THE
C      ITERATION PROCESS MUST continue AND A NEW BETA IS CHOSEN
        if (dabs(left1-rt1).gt.1d-6) goto 5
C     THE NORMAL FREESTREAM MACH NUMBER IS CALCULATED
        mn1=m1*dsin(beta1)
C     THE NORMAL MACH NUMBER AFTER THE SHOCK IS CALCULATED
        mn2=dsqrt((1+.2*mn1**2.)/(1.4*mn1**2.-.2))
C     THE PRESSURE RATIO ACCROSS THE SHOCK IS CALCULATED
        pt2pt1=(mn1/mn2)*((2+(0.4*mn2**2))/(2+(.4*mn1**2)))**3.00
        p2p1=1.0+2.8/2.4*(mn1**2.0-1.0)
c       FOR A CALORICALLY PERFECT GAS.
        ro2ro1=2.4*mn1**2./(0.4*mn1**2.+2.)
        T2T1=P2P1/RO2RO1
        TTA=((1+1.4*mn1**2.)/(1+1.4*mn2**2.))**2*(mn2/mn1)**2
        TTB=(1.+.2*mn2**2)/(1.+.2*mn1**2)
        TT2TT1=TTA*TTB
C     THE MACH NUMBER BEHIND THE SHOCK IS CALCULATED
        m2 = mn2/dsin(beta1-theta1)
        beta1 = beta1*180./pi
        write(1,50) beta1,m2,pt2pt1,P2P1,RO2RO1
c*****************************************************************************
C    VALUES ARE DEFINED AND THE FIRST RAMP IS ANALYZED
```

```
      WRITE(*,*)" ENTER FIRST RAMP ANGLE!!!!"
      READ(5,*)THETA2
      theta2 = THETA2*pi/180.
      beta2 = 70*pi/180.
      left2 = dtan(theta2)
      rt2 = 1.0
 6    beta2=beta2*dabs(left2/rt2)**0.1
      rta=((m2**2*dsin(beta2)**2)-1)/(m2**2*(1.4+dcos(2*beta2))+2)
      rtb=2/dtan(beta2)
      rt2=rta*rtb
      if (dabs(left2-rt2).gt.1d-5) goto 6
      mn2=m2*dsin(beta2)
      mn3=dsqrt((1+.2*mn2**2.)/(1.4*mn2**2.-.2))
      pt3pt2=(mn2/mn3)*((2+(0.4*mn3**2))/(2+(.4*mn2**2)))**3.00
      p3p2=1.0+2.8/2.4*(mn2**2.-1.0)
      ro3ro2=2.4*mn2**2./(0.4*mn2**2.+2.)
      T3T2=P3P2/RO3RO2
      TTA=((1+1.4*mn2**2.)/(1+1.4*mn3**2.))**2*(mn3/mn2)**2
      TTB=(1.+.2*mn3**2)/(1.+.2*mn2**2)
      TT3TT2=TTA*TTB
      m3 = mn3/dsin(beta2-theta2)
      beta2 = beta2*180./pi
      write(1,60) beta2,m3,pt3pt2,P3P2,RO3RO2
c*******************************************************************
C   VALUES ARE DEFINED AND THE SECOND RAMP IS ANALYZED
      WRITE(*,*)" ENTER SECOND RAMP ANGLE!!!!"
      READ(5,*)THETA3
      theta3 = THETA3*pi/180.
      beta3 = 70.*pi/180.
      left3 = dtan(theta3)
      rt3 = 1.0
 7    beta3=beta3*dabs(left3/rt3)**0.1
      rta=((m3**2*dsin(beta3)**2)-1)/(m3**2*(1.4+dcos(2*beta3))+2)
      rtb=2/dtan(beta3)
      rt3=rta*rtb
      if (dabs(left3-rt3).gt.1d-4) goto 7
      mn3=m3*dsin(beta3)
      mn4=dsqrt((1+.2*mn3**2.)/(1.4*mn3**2.-.2))
      pt4pt3=(mn3/mn4)*((2+(0.4*mn4**2))/(2+(.4*mn3**2)))**3.00
      p4p3=1.0+2.8/2.4*(mn3**2.0-1.0)
      ro4ro3=2.4*mn3**2./(0.4*mn3**2.+2.)
      T4T3=P4P3/RO4RO3
      TTA=((1+1.4*MN3**2.)/(1+1.4*MN4**2.))**2*(MN4/MN3)**2
      TTB=(1.+.2*MN4**2)/(1.+.2*MN3**2)
      TT4TT3=TTA*TTB
      m4 = mn4/dsin(beta3-theta3)
      beta3 = beta3*180./pi
      write(1,70) beta3,m4,pt4pt3,P4P3,RO4RO3
C*******************************************************************
C   VALUES ARE DEFINED AND THE THIRD RAMP IS ANALYZED
      WRITE(*,*)" ENTER THIRD RAMP ANGLE!!!!"
      READ(5,*)THETA4
```

```
      theta4 = THETA4*pi/180.
C     theta4 = 3.0*pi/180.
      beta4 = 70.*pi/180.
      left4 = dtan(theta4)
      rt4 = 1.0
 8    beta4=beta4*dabs(left4/rt4)**0.1
      rta=((m4**2*dsin(beta4)**2)-1)/(m4**2*(1.4+dcos(2*beta4))+2)
      rtb=2/dtan(beta4)
      rt4=rta*rtb
          WRITE(*,*)LEFT4,RT4
      if (dabs(left4-rt4).gt.1d-4) goto 8
      mn4=m4*dsin(beta4)
      mn5=dsqrt((1+.2*mn4**2.)/(1.4*mn4**2.-.2))
      pt5pt4=(mn4/mn5)*((2+(0.4*mn5**2))/(2+(.4*mn4**2)))**3.00
      p5p4=1.0+2.8/2.4*(mn4**2.0-1.0)
      ro5ro4=2.4*mn4**2./(0.4*mn4**2.+2.)
      T5T4=P5P4/RO5RO4
      TTA=((1+1.4*MN4**2.)/(1+1.4*MN5**2.))**2*(MN5/MN4)**2
      TTB=(1.+.2*MN5**2)/(1.+.2*MN4**2)
      TT5TT4=TTA*TTB
      m5 = mn5/dsin(beta4-theta4)
      beta4 = beta4*180./pi
      write(1,80) beta4,m5,pt5pt4,P5P4,RO5RO4
C*********************************************************************
C   VALUES ARE DEFINED AND THE FOURTH RAMP IS ANALYZED
C     theta5 = 3.0*pi/180.
C     beta5 = 70.*pi/180.
C     left5 = dtan(theta5)
C     rt5 = 1.0
C 9   beta5=beta5*dabs(left5/rt5)**0.2
C     rta=((m5**2*dsin(beta5)**2)-1)/(m5**2*(1.4+dcos(2*beta5))+2)
C     rtb=2/dtan(beta5)
C     rt5=rta*rtb
C     if (dabs(left5-rt5).gt.1d-4) goto 9
C     mn5=m5*dsin(beta5)
C     mn6=dsqrt((1+.2*mn5**2.)/(1.4*mn5**2.-.2))
C     pt6pt5=(mn5/mn6)*((2+(0.4*mn6**2))/(2+(.4*mn5**2)))**3.00
C     p6p5=1.0+2.8/2.4*(mn5**2.0-1.0)
C     ro6ro5=2.4*mn5**2./(0.4*mn5**2.+2.)
C     T6T5=P6P5/RO6RO5
C     TTA=((1+1.4*MN5**2.)/(1+1.4*MN6**2.))**2*(MN6/MN5)**2
C     TTB=(1.+.2*MN6**2)/(1.+.2*MN5**2)
C     TT6TT5=TTA*TTB
C     m6 = mn6/dsin(beta5-theta5)
C     beta5 = beta5*180./pi
C     write(1,90) beta5,m6,pt6pt5,P6P5,RO6RO5
C*********************************************************************
C   VALUES ARE DEFINED AND THE FIFTH RAMP IS ANALYZED
C     theta6 = 3.0*pi/180.
C     beta6 = 70.*pi/180.
C     left6 = dtan(theta6)
C     rt6 = 1.0
```

```
C  10  beta6=beta6*dabs(left6/rt6)**0.2
C      rta=((m6**2*dsin(beta6)**2)-1)/(m6**2*(1.4+dcos(2*beta6))+2)
C      rtb=2/dtan(beta6)
C      rt6=rta*rtb
C      if (dabs(left6-rt6).gt.1d-4) goto 10
C      mn6=m6*dsin(beta6)
C      mn7=dsqrt((1+.2*mn6**2.)/(1.4*mn6**2.-.2))
C      pt7pt6=(mn6/mn7)*((2+(0.4*mn7**2))/(2+(.4*mn6**2)))**3.00
C      p7p6=1.0+2.8/2.4*(mn6**2.0-1.0)
C      ro7ro6=2.4*mn6**2./(0.4*mn6**2.+2.)
C      T7T6=P7P6/RO7RO6
C      TTA=((1+1.4*MN6**2.)/(1+1.4*MN7**2.))**2*(MN7/MN6)**2
C      TTB=(1.+.2*MN7**2)/(1.+.2*MN6**2)
C      TT7TT6=TTA*TTB
C      m7 = mn7/dsin(beta6-theta6)
C      beta6 = beta6*180./pi
C      write(1,100) beta6,m7,pt7pt6,P7P6,RO7RO6
c*****************************************************************
C      NORMAL SHOCK OCCURS AND VALUES ARE CALCULATED FOR COMBUSTION
       m6=dsqrt((1+.2*m5**2.)/(1.4*m5**2.-.2))
       pt6pt5=(m5/m6)*((2+(0.4*m6**2))/(2+(.4*m5**2)))**3.00
       p6p5=1.0+2.8/2.4*(m5**2.0-1.0)
       ro6ro5=2.4*m5**2./(0.4*m5**2.+2.)
       T6T5=P6P5/RO6RO5
       TTA=((1+1.4*M5**2.)/(1+1.4*M6**2.))**2*(M6/M5)**2
       TTB=(1.+.2*M6**2)/(1.+.2*M5**2)
       TT6TT5=TTA*TTB
       write(1,90) m6,pt6pt5,P6P5,RO6RO5
C     THE FOLLOWING PART OF THE PROGRAM WAS WRITTEN FOR THE
C     EXPANSION WAVE THAT WOULD BE GENERATED AS THE FLOW IS
C     TURNED BACK INTO THE SCRAMJET COMBUSTION CHAMBER.
C     HOWEVER, A COWL WAS ADDED AT THE BOTTOM OF THE SCRAMJET
C     MODULE TO GET AWAY FROM THIS EXPANSION. THEREFORE, AT THIS
C     POINT IN TIME THE EXPANSION WAVE IS NOT VALID.
c*****************************************************************
C       THE CONDITIONS FOR THE EXPANSION WAVE ARE CALCULATED
C       USING THE PRANDTL-MEYER function.
C       PART=DSQRT(0.4/2.4*(M7**2.-1.0))
C       PART1=DATAN(PART)
C       NU1=SQRT(6.0)
C       NU2=DATAN(DSQRT(M7**2.-1.))
C       NU7=PART1*NU1-NU2
C       THETA7=21.*PI/180.
C       NU8=THETA7+NU7
C               NU8=76.92
C       GNU8=NU8-5.0
C       M8=3.0
C12     M8=M8*(NU8/GNU8)**.3
C       NU3=SQRT(6.0)*DATAN(DSQRT(0.4/2.4*(M8**2.-1.0)))
C       NU4=DATAN(DSQRT(M8**2.-1.))
C       GNU8=(NU3-NU4)*180./PI
C               write(*,*)GNU8,NU8,M8
```

```
C        if (DABS(GNU8-NU8).gt.1D-5) goto 12
C        NOTE THAT THE TOTAL PRESSURES AND TEMPERATURES do NOT
C        CHANGE THROUGH A PRANDTL-MEYER EXPANSION WAVE.
C        T8T7=1/((1+.4/2.0*M8**2.)/(1+.4/2.*M7**2.))
C        P8P7=1/(((1+.4/2.*M8**2.)/(1+.4/2.*M7**2.))**3.5)
C        RO8RO7=P8P7/T8T7
C        write(1,105) M8,P8P7,RO8RO7
C        THE FOLLOWING VALUES ARE SET TO ONE SO THOSE RAMPS COULD
C        BE "C"ED OUT AND THE TOTAL RATIOS ARE STILL CALCULATED
C        AT THE END OF THE PROGRAM.  FOR EACH ADDITIONAL RAMP
C        THAT IS ANALYZED, THE CORRESPONDING NUMBERS MUST BE DELETED
C        SO THAT THE VALUES ARE NOT RESET TO ONE.
         P8P7=1.0
         RO8RO7=1.0
         T8T7=1.0
         PT7PT6 =1.0
         P7P6=1.0
         TT7TT6=1.0
         RO7RO6=1.0
         T7T6 =1.0
C        PT6PT5=1.0
C        P6P5=1.0
C        TT6TT5=1.0
C        RO6RO5=1.0
C        T6T5=1.0
C        PT5PT4=1.0
C        P5P4=1.0
C        TT5TT4=1.0
C        RO5RO4=1.0
C        T5T4=1.0
        PTRATIO=PT2PT1*PT3PT2*PT4PT3*PT5PT4*PT6PT5*PT7PT6
        PRATIO=1./(P2P1*P3P2*P4P3*P5P4*P6P5*P7P6*P8P7)
        TTRATIO=TT2TT1*TT3TT2*TT4TT3*TT5TT4*TT6TT5*TT7TT6
        RORATIO=RO2RO1*RO3RO2*RO4RO3*RO5RO4*RO6RO5*RO7RO6*RO8RO7
        TRATIO=T2T1*T3T2*T4T3*T5T4*T6T5*T7T6*T8T7
        RO5=RO1*RORATIO
        T5=T1*TRATIO
C       THE SPEED OF SOUND IS CALCULATED
        A=DSQRT(1.4*1716.*T5)
C       THE VELOCITY RIGHT BEFORE THE COMBUSTION CHAMBER IS CALCULATED
        V=M5*A
C       MASS FLOW RATE ENTERING THE COMBUSTION CHAMBER.
        MDOT=RO5*V*AREA
        PRESSURE=P1/PRATIO
        write(1,110)PTRATIO
        write(1,120)PRATIO
        write(1,130)RORATIO
        write(1,150)MDOT
        WRITE(1,160)PRESSURE
C*********************************************************************
C        POSITIONING THE SHOCK WAVE SO THAT IT IMPINGES ON THE LIP OF
C        THE SCRAMJET.  THIS IS AN ITERATIVE PROCESS.
```

```
C
C       THIS IS THE CALCULATION OF THE DOWNWARD ANGLE OF THE
C       HORIZONTAL to EACH SHOCK WAVE.
        TH1=THETA1
C       TH2=TH1+THETA2
C       TH3=TH2+THETA3
C       TH4=TH3+THETA4
C       TH5=TH4+THETA5
C       TH6=TH5+THETA6
C       BT1=BETA1
        BT2=TH1+BETA2
        BT2=BT2*PI/180.
C       BT3=TH2+BETA3
C       BT4=TH3+BETA4
C       BT5=TH4+BETA5
C       BT6=TH5+BETA6
C       USING SIMULAR TRIANGLES, THE RAMPS CAN BE PLACE SO THAT
C       THE OBLIQUE SHOCKS HIT AT THE LIP OF THE ENGINE.
C       THESE ARE THE LENGTHS OF THE ORIGINAL TRIANGLE THAT
C       DEFINES THE AREA BEFORE THE FIRST RAMP BEGINS.
        L1=47.0
        H1=8.287
C       DEFINITION OF THE DIMENSIONS OF THE FIRST RAMP.
        L2=50.2-L1
        H2=10.6-H1
        H2L2=1.0
        AH2L2=1.0
   11   BT2=BT2*DABS(H2L2/AH2L2)**0.3
        H2L2 = H2/L2
        AH2L2=DTAN(BT2)
        if (DABS(H2L2-AH2L2).gt.1D-6) goto 11
        EBETA2=BT2-TH1
        EBETA2=EBETA2*180./PI
       WRITE(1,140)EBETA2
       stop
       end
```

Appendix C:

SCRAMjet Inlet Program

```
c***********************************************************************
        program INLET
c***********************************************************************
c      AERONAUTICAL & ASTRONAUTICAL ENGINEERING                         *
c      416  HYPERSONIC DESIGN                                           *
c      SPRING QUARTER 1990                                              *
c                                                                       *
c      AUTHORS: GOUHIN, P. & WHITE, H.                                  *
c                                                                       *
c      PURPOSE: to AID IN THE DESIGN OF AN INLET FOR A SCRAMJET         *
C               ENGINE ON A HYPERSONIC TEST BED VEHICLE.  THIS          *
c               CODE ALLOWS FOR THE ADDITION OF SEVERAL RAMPS, ON       *
C               THE UNDERSIDE OF THE VEHICLE, to INDUCE SHOCKS          *
c               to INCREASE PRESSURE RATIOS AND REDUCE FLOW             *
c               VELOCITY.                                               *
C
C      PRESENTLY: THE PROGRAM IS DESIGNED TO CONSIDER ONE RAMP ON THE *
C               UNDER SIDE OF THE VEHICLE.  THE SECOND RAMP TAKES IN- *
C               TO ACCOUNT THE COWL ANGLE.  THE THIRD RAMP TAKES INTO *
C               ACCOUNT THE SECOND ANGLE ON THE COWL.  FROM THIS POINT*
C               THE FLOW GOES DIRECTLY INTO THE COMBUSTION CHAMBER    *
C               FOR A CLEARER EXPLANATION WITH FIGURES SEE THE FINAL  *
C               REPORT FOR THE WHITE GROUP (AAE 416 SPRING 1990)      *
c                                                                     *
c      INPUT:   AS DEFINED IN THE program(set up for five ramps)      *
C               INTERACTIVE VARIABLES FOR FREESTREAM MACH NUMBER      *
C               AND RAMP ANGLES ARE INPUT FROM KEYBOARD.              *
c                                                                     *
C      OUTPUT:  DATA IS WRITTEN to INLET.DAT                          *
c                                                                     *
c***********************************************************************
        implicit real *8 (a-h,l-z)
        integer i,j,k
        open(1,file='INLET.DAT',STATUS='NEW')
30      format(5X,'VALUES FROM PROGRAM INLET FOR')
40      format(3X,'FREESTREAM MACH # = ',F6.3)
50      format(3x,'beta1=',f6.3,3x,'M2=',f5.3,3x,'Pt2/Pt1=',f6.4
     F,3X,'P2/P1=',F6.4,3X,'RO2/RO1=',F7.4)
60      format(3x,'beta2=',f6.3,3x,'M3=',f5.3,3x,'Pt3/Pt2=',f6.4
     F,3X,'P3/P2=',F6.4,3X,'RO3/RO2=',F7.4)
70      format(3x,'beta3=',f6.3,3x,'M4=',f5.3,3x,'Pt4/Pt3=',f6.4
     F,3X,'P4/P3=',F6.4,3X,'RO4/RO3=',F7.4)
80      format(3x,'beta4=',f6.3,3x,'M5=',f5.3,3x,'Pt5/Pt4=',f6.4
     F,3X,'P5/P4=',F6.4,3X,'RO5/RO4=',F7.4)
90      format(3x,'beta5=',f6.3,3x,'M6=',f5.3,3x,'Pt6/Pt5=',f6.4
     F,3X,'P6/P5=',F6.4,3X,'RO6/RO5=',F7.4)
100     format(3x,'beta6=',f6.3,3x,'M7=',f5.3,3x,'Pt7/Pt6=',f6.4
     F,3X,'P7/P6=',F6.4,3X,'RO7/RO6=',F7.4)
C105     format(17X,'M8=',F5.3,17X,'P8/P7='F6.4,'RO8/RO7=',F7.4)
110     format(5X,'THE TOTAL PRESSURE RECOVERY IS ',F6.4)
120     format(5X,'THE STATIC PRESSURE RECOVERY IS ',F8.4)
130     format(5X,'THE DENSITY RATIO IS ',F8.4)
```

```
140    format(5X,'THE ITERATED BETA2 IS ',F6.3)
150    format(5x,'THE MASS FLOW RATE IS ',F8.4)
160    format(5x,'THE PRESSURE IN THE COMBUSTION CHAMBER IS ',F10.4)
C      OTHERWISE KNOWN AS PI.
       pi = 3.1415926535897932384626433832795028 8
C     THE FREESTREAM FLOW IS DEFINED
       WRITE(*,*)" ENTER FREESTREAM MACH NUMBER!!"
       READ(5,*)M1
       WRITE(*,*)" ENTER AMBIENT TEMPERATURE!!"
       READ(5,*)T1
       WRITE(*,*)" ENTER FREESTREAM DENSITY!!!!"
       READ(5,*)RO1
       RO1=RO1*32.2
       WRITE(*,*)" ENTER AMBIENT PRESSURE!!!!!!!!!"
       READ(5,*)P1
       AREA=8.0*30.0/144.0
       write(1,30)
       write(1,40)M1
C    THETA1 IS THE HALF-WEDGE NOSE ANGLE OF THE VEHICLE
       WRITE(*,*)" ENTER WEDGE ANGLE OF VEHICLE!!!!"
       READ(5,*)THETA1
       theta1 = THETA1*pi/180.
C    BETA IS NEVER EXPECTED to EXCEED 70 DEG. AND THUS IT IS
C     ITERATED DOWN FROM THIS INITIAL guess
       beta1 = 70.*pi/180.
C    LEFT AND RIGHT REFER to THE L.H.S & R.H.S. OF THE THETA-
C    BETA-MACH NUMBER EQUATION
       left1 = dtan(theta1)
       rt1 = 1.0
    5  beta1=beta1*dabs(left1/rt1)**0.3
       rta=((m1**2*dsin(beta1)**2)-1)/(m1**2*(1.4+dcos(2*beta1))+2)
       rtb=2/dtan(beta1)
       rt1=rta*rtb
C    if THE TWO SIDES ARE NOT EQUAL WITHIN THIS RANGE then THE
C     ITERATION PROCESS MUST continue AND A NEW BETA IS CHOSEN
       if (dabs(left1-rt1).gt.1d-6) goto 5
C    THE NORMAL FREESTREAM MACH NUMBER IS CALCULATED
       mn1=m1*dsin(beta1)
C    THE NORMAL MACH NUMBER AFTER THE SHOCK IS CALCULATED
       mn2=dsqrt((1+.2*mn1**2.)/(1.4*mn1**2.-.2))
C    THE PRESSURE RATIO ACCROSS THE SHOCK IS CALCULATED
       pt2pt1=(mn1/mn2)*((2+(0.4*mn2**2))/(2+(.4*mn1**2)))**3.00
       p2p1=1.0+2.8/2.4*(mn1**2.0-1.0)
c      FOR A CALORICALLY PERFECT GAS.
       ro2ro1=2.4*mn1**2./(0.4*mn1**2.+2.)
       T2T1=P2P1/RO2RO1
       TTA=((1+1.4*mn1**2.)/(1+1.4*mn2**2.))**2*(mn2/mn1)**2
       TTB=(1.+.2*mn2**2)/(1.+.2*mn1**2)
       TT2TT1=TTA*TTB
C    THE MACH NUMBER BEHIND THE SHOCK IS CALCULATED
       m2 = mn2/dsin(beta1-theta1)
       beta1 = beta1*180./pi
```

```
      write(1,50) beta1,m2,pt2pt1,P2P1,RO2RO1
c*********************************************************************
C   VALUES ARE DEFINED AND THE FIRST RAMP IS ANALYZED
      WRITE(*,*)" ENTER FIRST RAMP ANGLE!!!!"
      READ(5,*)THETA2
      theta2 = THETA2*pi/180.
      beta2 = 70*pi/180.
      left2 = dtan(theta2)
      rt2 = 1.0
    6 beta2=beta2*dabs(left2/rt2)**0.3
      rta=((m2**2*dsin(beta2)**2)-1)/(m2**2*(1.4+dcos(2*beta2))+2)
      rtb=2/dtan(beta2)
      rt2=rta*rtb
      if (dabs(left2-rt2).gt.1d-5) goto 6
      mn2=m2*dsin(beta2)
      mn3=dsqrt((1+.2*mn2**2.)/(1.4*mn2**2.-.2))
      pt3pt2=(mn2/mn3)*((2+(0.4*mn3**2))/(2+(.4*mn2**2)))**3.00
      p3p2=1.0+2.8/2.4*(mn2**2.-1.0)
      ro3ro2=2.4*mn2**2./(0.4*mn2**2.+2.)
      T3T2=P3P2/RO3RO2
      TTA=((1+1.4*mn2**2.)/(1+1.4*mn3**2.))**2*(mn3/mn2)**2
      TTB=(1.+.2*mn3**2)/(1.+.2*mn2**2)
      TT3TT2=TTA*TTB
      m3 = mn3/dsin(beta2-theta2)
      beta2 = beta2*180./pi
      write(1,60) beta2,m3,pt3pt2,P3P2,RO3RO2
c*********************************************************************
C   VALUES ARE DEFINED AND THE SECOND RAMP IS ANALYZED
      WRITE(*,*)" ENTER COWL ANGLE!!!!"
      READ(5,*)THETA3
      theta3 = THETA3*pi/180.
      beta3 = 70.*pi/180.
      left3 = dtan(theta3)
      rt3 = 1.0
    7 beta3=beta3*dabs(left3/rt3)**0.3
      rta=((m3**2*dsin(beta3)**2)-1)/(m3**2*(1.4+dcos(2*beta3))+2)
      rtb=2/dtan(beta3)
      rt3=rta*rtb
      if (dabs(left3-rt3).gt.1d-4) goto 7
      mn3=m3*dsin(beta3)
      mn4=dsqrt((1+.2*mn3**2.)/(1.4*mn3**2.-.2))
      pt4pt3=(mn3/mn4)*((2+(0.4*mn4**2))/(2+(.4*mn3**2)))**3.00
      p4p3=1.0+2.8/2.4*(mn3**2.0-1.0)
      ro4ro3=2.4*mn3**2./(0.4*mn3**2.+2.)
      T4T3=P4P3/RO4RO3
      TTA=((1+1.4*MN3**2.)/(1+1.4*MN4**2.))**2*(MN4/MN3)**2
      TTB=(1.+.2*MN4**2)/(1.+.2*MN3**2)
      TT4TT3=TTA*TTB
      m4 = mn4/dsin(beta3-theta3)
      beta3 = beta3*180./pi
      write(1,70) beta3,m4,pt4pt3,P4P3,RO4RO3
C*********************************************************************
```

```
C   VALUES ARE DEFINED AND THE THIRD RAMP IS ANALYZED
        WRITE(*,*)" ENTER SECOND COWL ANGLE!!!!"
        READ(5,*)THETA4
        theta4 = THETA4*pi/180.
C       theta4 = 3.0*pi/180.
        beta4 = 70.*pi/180.
        left4 = dtan(theta4)
        rt4 = 1.0
    8   beta4=beta4*dabs(left4/rt4)**0.1
        rta=((m4**2*dsin(beta4)**2)-1)/(m4**2*(1.4+dcos(2*beta4))+2)
        rtb=2/dtan(beta4)
        rt4=rta*rtb
            WRITE(*,*)LEFT4,RT4
        if (dabs(left4-rt4).gt.1d-4) goto 8
        mn4=m4*dsin(beta4)
        mn5=dsqrt((1+.2*mn4**2.)/(1.4*mn4**2.-.2))
        pt5pt4=(mn4/mn5)*((2+(0.4*mn5**2))/(2+(.4*mn4**2)))**3.00
        p5p4=1.0+2.8/2.4*(mn4**2.0-1.0)
        ro5ro4=2.4*mn4**2./(0.4*mn4**2.+2.)
        T5T4=P5P4/RO5RO4
        TTA=((1+1.4*MN4**2.)/(1+1.4*MN5**2.))**2*(MN5/MN4)**2
        TTB=(1.+.2*MN5**2)/(1.+.2*MN4**2)
        TT5TT4=TTA*TTB
        m5 = mn5/dsin(beta4-theta4)
        beta4 = beta4*180./pi
        write(1,80) beta4,m5,pt5pt4,P5P4,RO5RO4
C*********************************************************************
C   VALUES ARE DEFINED AND THE FOURTH RAMP IS ANALYZED
C       theta5 = 3.0*pi/180.
C       beta5 = 70.*pi/180.
C       left5 = dtan(theta5)
C       rt5 = 1.0
C   9   beta5=beta5*dabs(left5/rt5)**0.2
C       rta=((m5**2*dsin(beta5)**2)-1)/(m5**2*(1.4+dcos(2*beta5))+2)
C       rtb=2/dtan(beta5)
C       rt5=rta*rtb
C       if (dabs(left5-rt5).gt.1d-4) goto 9
C       mn5=m5*dsin(beta5)
C       mn6=dsqrt((1+.2*mn5**2.)/(1.4*mn5**2.-.2))
C       pt6pt5=(mn5/mn6)*((2+(0.4*mn6**2))/(2+(.4*mn5**2)))**3.00
C       p6p5=1.0+2.8/2.4*(mn5**2.0-1.0)
C       ro6ro5=2.4*mn5**2./(0.4*mn5**2.+2.)
C       T6T5=P6P5/RO6RO5
C       TTA=((1+1.4*MN5**2.)/(1+1.4*MN6**2.))**2*(MN6/MN5)**2
C       TTB=(1.+.2*MN6**2)/(1.+.2*MN5**2)
C       TT6TT5=TTA*TTB
C       m6 = mn6/dsin(beta5-theta5)
C       beta5 = beta5*180./pi
C       write(1,90) beta5,m6,pt6pt5,P6P5,RO6RO5
C*********************************************************************
C   VALUES ARE DEFINED AND THE FIFTH RAMP IS ANALYZED
C       theta6 = 3.0*pi/180.
```

```
C       beta6 = 70.*pi/180.
C       left6 = dtan(theta6)
C       rt6 = 1.0
C  10   beta6=beta6*dabs(left6/rt6)**0.2
C       rta=((m6**2*dsin(beta6)**2)-1)/(m6**2*(1.4+dcos(2*beta6))+2)
C       rtb=2/dtan(beta6)
C       rt6=rta*rtb
C       if (dabs(left6-rt6).gt.1d-4) goto 10
C       mn6=m6*dsin(beta6)
C       mn7=dsqrt((1+.2*mn6**2.)/(1.4*mn6**2.-.2))
C       pt7pt6=(mn6/mn7)*((2+(0.4*mn7**2))/(2+(.4*mn6**2)))**3.00
C       p7p6=1.0+2.8/2.4*(mn6**2.0-1.0)
C       ro7ro6=2.4*mn6**2./(0.4*mn6**2.+2.)
C       T7T6=P7P6/RO7RO6
C       TTA=((1+1.4*MN6**2.)/(1+1.4*MN7**2.))**2*(MN7/MN6)**2
C       TTB=(1.+.2*MN7**2)/(1.+.2*MN6**2)
C       TT7TT6=TTA*TTB
C       m7 = mn7/dsin(beta6-theta6)
C       beta6 = beta6*180./pi
C       write(1,100) beta6,m7,pt7pt6,P7P6,RO7RO6
c*****************************************************************
C      THE FOLLOWING PART OF THE PROGRAM WAS WRITTEN FOR THE
C      EXPANSION WAVE THAT WOULD BE GENERATED AS THE FLOW IS
C      TURNED BACK INTO THE SCRAMJET COMBUSTION CHAMBER.
C      HOWEVER, A COWL WAS ADDED AT THE BOTTOM OF THE SCRAMJET
C      MODULE TO GET AWAY FROM THIS EXPANSION. THEREFORE, AT THIS
C      POINT IN TIME THE EXPANSION WAVE IS NOT VALID.
c*****************************************************************
C        THE CONDITIONS FOR THE EXPANSION WAVE ARE CALCULATED
C        USING THE PRANDTL-MEYER function.
C        PART=DSQRT(0.4/2.4*(M7**2.-1.0))
C        PART1=DATAN(PART)
C        NU1=SQRT(6.0)
C        NU2=DATAN(DSQRT(M7**2.-1.))
C        NU7=PART1*NU1-NU2
C        THETA7=21.*PI/180.
C        NU8=THETA7+NU7
C                NU8=76.92
C        GNU8=NU8-5.0
C        M8=3.0
C12      M8=M8*(NU8/GNU8)**.3
C        NU3=SQRT(6.0)*DATAN(DSQRT(0.4/2.4*(M8**2.-1.0)))
C        NU4=DATAN(DSQRT(M8**2.-1.))
C        GNU8=(NU3-NU4)*180./PI
C                write(*,*)GNU8,NU8,M8
C        if (DABS(GNU8-NU8).gt.1D-5) goto 12
C        NOTE THAT THE TOTAL PRESSURES AND TEMPERATURES do NOT
C        CHANGE THROUGH A PRANDTL-MEYER EXPANSION WAVE.
C        T8T7=1/((1+.4/2.0*M8**2.)/(1+.4/2.*M7**2.))
C        P8P7=1/((((1+.4/2.*M8**2.)/(1+.4/2.*M7**2.))**3.5)
C        RO8RO7=P8P7/T8T7
C        write(1,105) M8,P8P7,RO8RO7
```

```
C       THE FOLLOWING VALUES ARE SET TO ONE SO THOSE RAMPS COULD
C       BE "C"ED OUT AND THE TOTAL RATIOS ARE STILL CALCULATED
C       AT THE END OF THE PROGRAM.  FOR EACH ADDITIONAL RAMP
C       THAT IS ANALYZED, THE CORRESPONDING NUMBERS MUST BE DELETED
C       SO THAT THE VALUES ARE NOT RESET TO ONE.
        P8P7=1.0
        RO8RO7=1.0
        T8T7=1.0
        PT7PT6 =1.0
        P7P6=1.0
        TT7TT6=1.0
        RO7RO6=1.0
        T7T6 =1.0
        PT6PT5=1.0
        P6P5=1.0
        TT6TT5=1.0
        RO6RO5=1.0
        T6T5=1.0
C       PT5PT4=1.0
C       P5P4=1.0
C       TT5TT4=1.0
C       RO5RO4=1.0
C       T5T4=1.0
       PTRATIO=PT2PT1*PT3PT2*PT4PT3*PT5PT4*PT6PT5*PT7PT6
       PRATIO=1./(P2P1*P3P2*P4P3*P5P4*P6P5*P7P6*P8P7)
       TTRATIO=TT2TT1*TT3TT2*TT4TT3*TT5TT4*TT6TT5*TT7TT6
       RORATIO=RO2RO1*RO3RO2*RO4RO3*RO5RO4*RO6RO5*RO7RO6*RO8RO7
       TRATIO=T2T1*T3T2*T4T3*T5T4*T6T5*T7T6*T8T7
       RO5=RO1*RORATIO
       T5=T1*TRATIO
C      THE SPEED OF SOUND IS CALCULATED
       A=DSQRT(1.4*1716.*T5)
C      THE VELOCITY RIGHT BEFORE THE COMBUSTION CHAMBER IS CALCULATED
       V=M5*A
C      MASS FLOW RATE ENTERING THE COMBUSTION CHAMBER.
       MDOT=RO5*V*AREA
       PRESSURE=P1/PRATIO
       write(1,110)PTRATIO
       write(1,120)PRATIO
       write(1,130)RORATIO
       write(1,150)MDOT
       WRITE(1,160)PRESSURE
C*******************************************************************
C       POSITIONING THE SHOCK WAVE SO THAT IT IMPINGES ON THE LIP OF
C       THE SCRAMJET.  THIS IS AN ITERATIVE PROCESS.
C
C       THIS IS THE CALCULATION OF THE DOWNWARD ANGLE OF THE
C       HORIZONTAL to EACH SHOCK WAVE.
        TH1=THETA1
C       TH2=TH1+THETA2
C       TH3=TH2+THETA3
C       TH4=TH3+THETA4
```

```
C       TH5=TH4+THETA5
C       TH6=TH5+THETA6
C       BT1=BETA1
        BT2=TH1+BETA2
        BT2=BT2*PI/180.
C       BT3=TH2+BETA3
C       BT4=TH3+BETA4
C       BT5=TH4+BETA5
C       BT6=TH5+BETA6
C       USING SIMULAR TRIANGLES, THE RAMPS CAN BE PLACE SO THAT
C       THE OBLIQUE SHOCKS HIT AT THE LIP OF THE ENGINE.
C       THESE ARE THE LENGTHS OF THE ORIGINAL TRIANGLE THAT
C       DEFINES THE AREA BEFORE THE FIRST RAMP BEGINS.
        L1=47.0
        H1=8.287
C       DEFINITION OF THE DIMENSIONS OF THE FIRST RAMP.
        L2=50.2-L1
        H2=10.6-H1
        H2L2=1.0
        AH2L2=1.0
   11   BT2=BT2*DABS(H2L2/AH2L2)**0.3
        H2L2 = H2/L2
        AH2L2=DTAN(BT2)
        if (DABS(H2L2-AH2L2).gt.1D-6) goto 11
        EBETA2=BT2-TH1
        EBETA2=EBETA2*180./PI
       WRITE(1,140)EBETA2
       stop
       end
```

Appendix D:

Inlet Code Output

```
 VALUES FROM PROGRAM TURBOFAN FOR
FREESTREAM MACH # =  2.000
beta1=32.856    M2=1.878    Pt2/Pt1=0.9993
beta2=36.686    M3=1.704    Pt3/Pt2=0.9981    P2/P1=1.2069    RO2/RO1= 1.1436
beta3=43.197    M4=1.463    Pt4/Pt3=0.9956    P3/P2=1.3022    RO3/RO2= 1.2069
beta4=52.804    M5=1.206    Pt5/Pt4=0.9957    P4/P3=1.4205    RO4/RO3= 1.28[illegible]
M6=0.838    Pt6/Pt5=0.9922    P6/P5=  1.5301    P5/P4=1.4174    RO5/RO4= 1.28[illegible]4
 THE TOTAL PRESSURE RECOVERY IS 0.9811    RO6/RO5= 1.3520
 THE STATIC PRESSURE RECOVERY IS    0.2065
 THE DENSITY RATIO IS    3.0685
 THE MASS FLOW RATE IS   4337.9894
 THE PRESSURE IN THE COMBUSTION CHAMBER IS   4913.0079
```

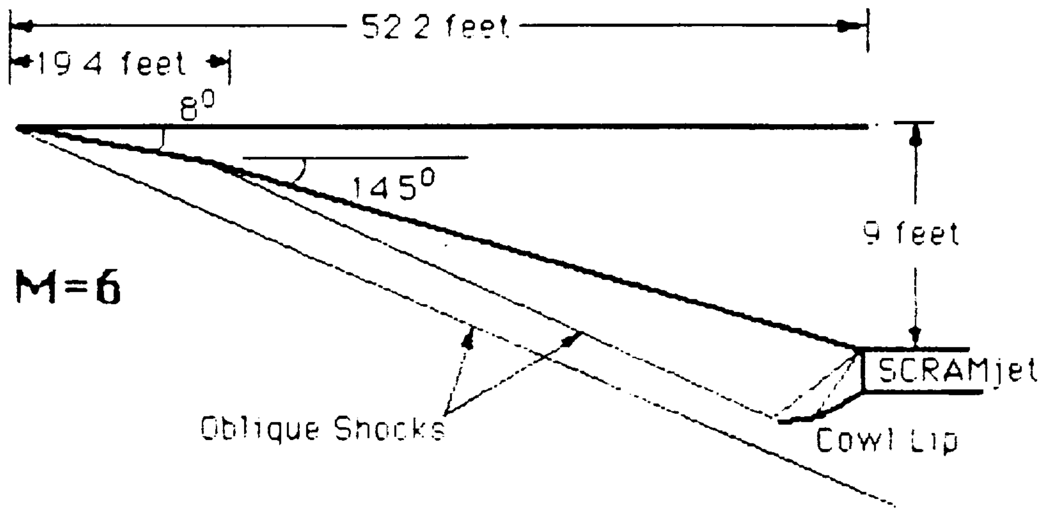
```
 VALUES FROM PROGRAM TURBOFAN FOR
FREESTREAM MACH # =  4.000
beta1=18.021    M2=3.638    Pt2/Pt1=0.9887
beta2=27.522    M3=2.761    Pt3/Pt2=0.8635    P2/P1=1.6199    RO2/RO1= 1.4068
beta3=35.286    M4=2.029    Pt4/Pt3=0.8972    P3/P2=3.1307    RO3/RO2= 2.1668
beta4=48.996    M5=1.341    Pt5/Pt4=0.9197    P4/P3=2.8003    RO4/RO3= 2.0229
M6=0.766    Pt6/Pt5=0.9717    P6/P5=  1.9303    P5/P4=2.5678    RO5/RO4= 1.9149
 THE TOTAL PRESSURE RECOVERY IS 0.6844    RO6/RO5= 1.5865
 THE STATIC PRESSURE RECOVERY IS    0.0142
 THE DENSITY RATIO IS  18.7330
 THE MASS FLOW RATE IS  5726.8474
```

```
  VALUES FROM PROGRAM INLET FOR
FREESTREAM MACH # =   6.000
beta1=12.372    M2=5.449    Pt2/Pt1=0.9818    P2/P1=1.7614    RO2/RO1= 1.4905
beta2=15.349    M3=4.696    Pt3/Pt2=0.9470    P3/P2=2.2608    RO3/RO2= 1.7631
beta3=17.452    M4=4.058    Pt4/Pt3=0.9561    P4/P3=2.1478    RO4/RO3= 1.7043
beta4=16.676    M5=3.795    Pt5/Pt4=0.9957    P5/P4=1.4155    RO5/RO4= 1.2801
  THE TOTAL PRESSURE RECOVERY IS 0.8851
  THE STATIC PRESSURE RECOVERY IS   0.0826
  THE DENSITY RATIO IS   5.7335
  THE MASS FLOW RATE IS 142.6178
  THE PRESSURE IN THE COMBUSTION CHAMBER IS   703.6571


  VALUES FROM PROGRAM INLET FOR
FREESTREAM MACH # =  7.000
beta1=11.018    M2=6.281    Pt2/Pt1=0.9723    P2/P1=1.9215    RO2/RO1= 1.5816
beta2=12.739    M3=5.541    Pt3/Pt2=0.9619    P3/P2=2.0712    RO3/RO2= 1.6636
beta3=13.947    M4=4.942    Pt4/Pt3=0.9727    P4/P3=1.9141    RO4/RO3= 1.5775
beta4=14.443    M5=4.538    Pt5/Pt4=0.9893    P5/P4=1.6057    RO5/RO4= 1.3982
  THE TOTAL PRESSURE RECOVERY IS 0.8999
  THE STATIC PRESSURE RECOVERY IS   0.0818
  THE DENSITY RATIO IS   5.8035
  THE MASS FLOW RATE IS 135.2251
  THE PRESSURE IN THE COMBUSTION CHAMBER IS   560.5341


  VALUES FROM PROGRAM INLET FOR
FREESTREAM MACH # =  8.000
beta1=10.019    M2=7.087    Pt2/Pt1=0.9602    P2/P1=2.0935    RO2/RO1= 1.6755
beta2=11.728    M3=6.173    Pt3/Pt2=0.9475    P3/P2=2.2543    RO3/RO2= 1.7598
beta3=13.723    M4=5.313    Pt4/Pt3=0.9406    P4/P3=2.3355    RO4/RO3= 1.8011
beta4=12.875    M5=4.970    Pt5/Pt4=0.9942    P5/P4=1.4687    RO5/RO4= 1.3137
  THE TOTAL PRESSURE RECOVERY IS 0.8509
  THE STATIC PRESSURE RECOVERY IS   0.0618
  THE DENSITY RATIO IS   6.9769
  THE MASS FLOW RATE IS 127.6593
  THE PRESSURE IN THE COMBUSTION CHAMBER IS   511.9135


  VALUES FROM PROGRAM INLET FOR
FREESTREAM MACH # =  9.000
beta1= 9.255    M2=7.868    Pt2/Pt1=0.9456    P2/P1=2.2776    RO2/RO1= 1.7717
beta2=10.136    M3=6.981    Pt3/Pt2=0.9619    P3/P2=2.0701    RO3/RO2= 1.6630
beta3=11.036    M4=6.271    Pt4/Pt3=0.9726    P4/P3=1.9168    RO4/RO3= 1.5790
beta4=11.963    M5=5.674    Pt5/Pt4=0.9794    P5/P4=1.8047    RO5/RO4= 1.5155
  THE TOTAL PRESSURE RECOVERY IS 0.8664
  THE STATIC PRESSURE RECOVERY IS   0.0613
  THE DENSITY RATIO IS   7.0507
  THE MASS FLOW RATE IS 120.9392
  THE PRESSURE IN THE COMBUSTION CHAMBER IS   430.4089


  VALUES FROM PROGRAM INLET FOR
FREESTREAM MACH # = 10.000
beta1= 8.653    M2=8.623    Pt2/Pt1=0.9283    P2/P1=2.4742    RO2/RO1= 1.8698
beta2= 9.119    M3=7.704    Pt3/Pt2=0.9661    P3/P2=2.0119    RO3/RO2= 1.6315
beta3=11.536    M4=6.535    Pt4/Pt3=0.9164    P4/P3=2.6023    RO4/RO3= 1.9313
beta4=10.118    M5=6.206    Pt5/Pt4=0.9968    P5/P4=1.3711    RO5/RO4= 1.2517
  THE TOTAL PRESSURE RECOVERY IS 0.8193
  THE STATIC PRESSURE RECOVERY IS   0.0563
  THE DENSITY RATIO IS   7.3747
  THE MASS FLOW RATE IS 114.5754
  THE PRESSURE IN THE COMBUSTION CHAMBER IS   383.7050
```

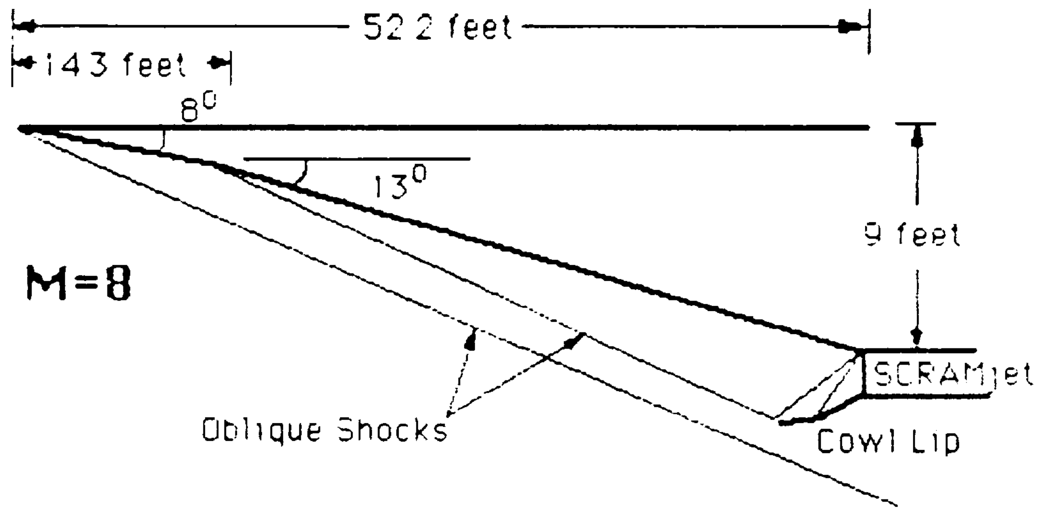
Appendix E:

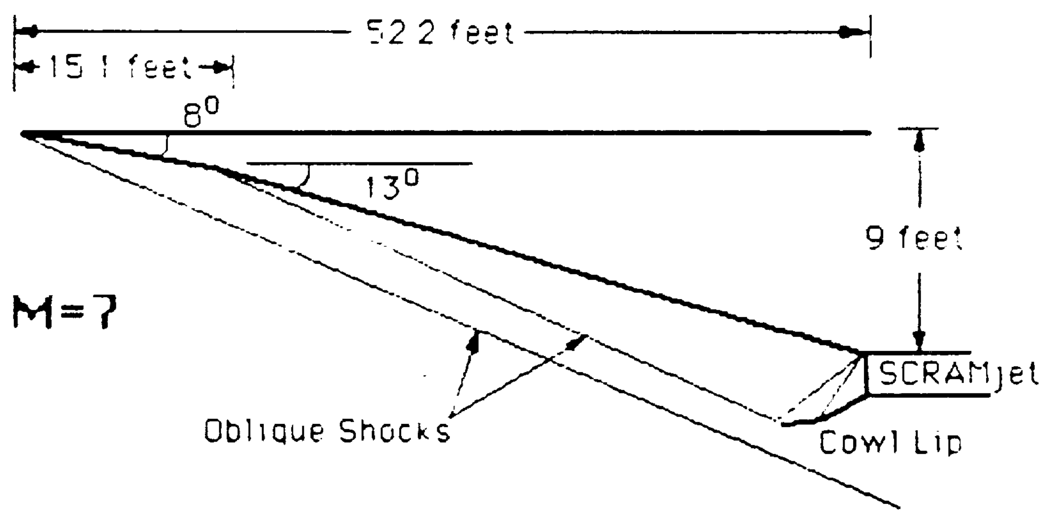
SCRAMjet Inlet Representative Configurations

SCRAMjet Inlets
(not drawn to scale)

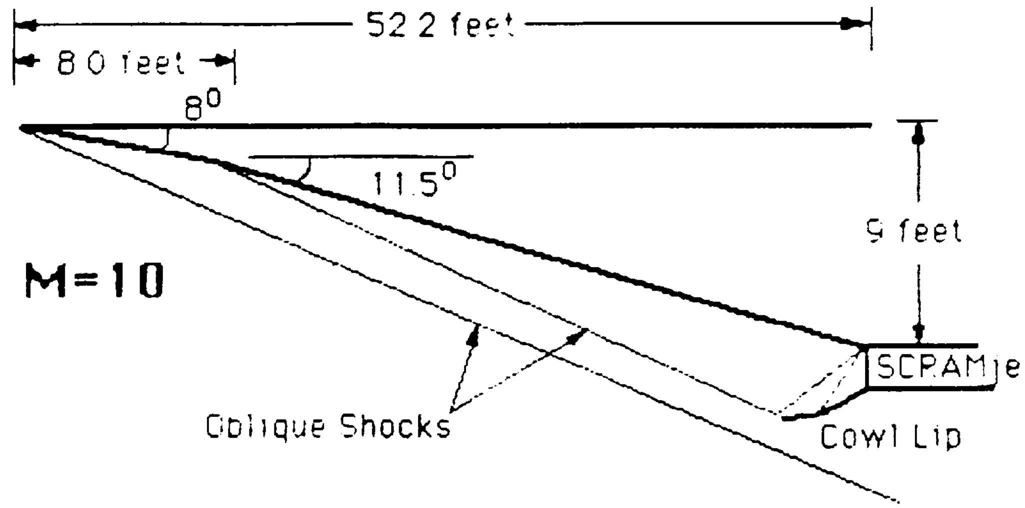


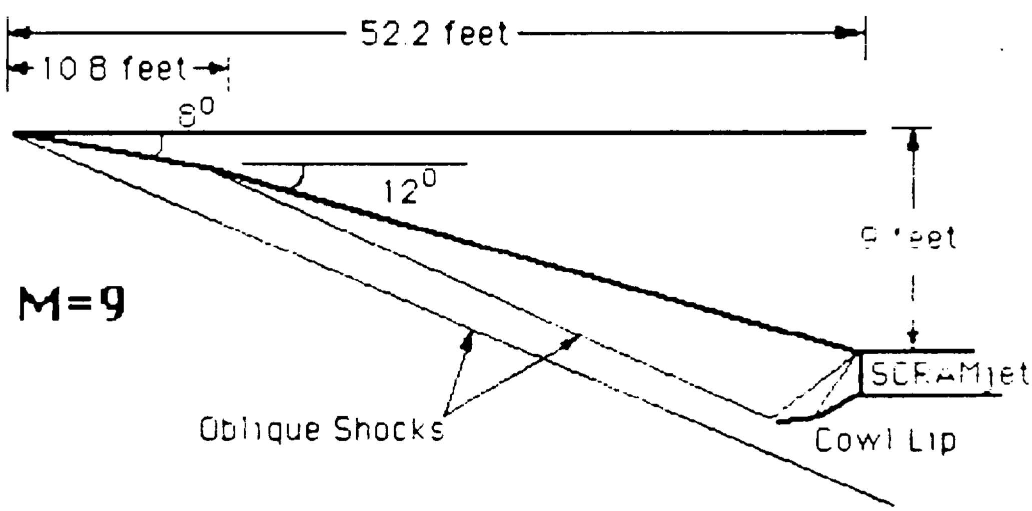
SCRAMjet Inlets
(not drawn to scale)





SCRAMjet Inlets
(not drawn to scale)





Appendix F:

Aerodynamic Data

AR=1.1 alpha=4

MACH	40K CD	50K CD	60K CD	70K CD	80K CD	90K CD	100K CD
0.2	0.039	0.040	0.041	0.041	0.042	0.043	0.044
0.4	0.038	0.039	0.040	0.040	0.041	0.042	0.043
0.6	0.038	0.038	0.039	0.039	0.040	0.041	0.042
0.8	0.065	0.066	0.066	0.067	0.067	0.068	0.069
1.0	0.126	0.127	0.127	0.128	0.128	0.129	0.130
1.2	0.381	0.381	0.382	0.382	0.383	0.383	0.384
1.4	0.338	0.339	0.339	0.340	0.340	0.341	0.341
1.6	0.171	0.171	0.172	0.172	0.173	0.173	0.174
1.8	0.131	0.131	0.131	0.131	0.132	0.132	0.133
2.0	0.092	0.092	0.092	0.092	0.093	0.093	0.094
2.2	0.074	0.074	0.074	0.075	0.075	0.075	0.076
2.4	0.055	0.055	0.055	0.056	0.056	0.056	0.057
2.6	0.049	0.050	0.050	0.050	0.050	0.051	0.051
2.8	0.035	0.036	0.036	0.036	0.036	0.037	0.037
3.0	0.031	0.032	0.032	0.032	0.032	0.033	0.033
3.2	0.030	0.030	0.031	0.031	0.031	0.031	0.032
3.4	0.027	0.027	0.027	0.027	0.028	0.028	0.028
3.6	0.024	0.024	0.024	0.024	0.025	0.025	0.025
3.8	0.023	0.023	0.023	0.023	0.023	0.024	0.024
4.0	0.021	0.021	0.021	0.021	0.022	0.022	0.022
4.2	0.019	0.020	0.020	0.020	0.020	0.020	0.021
4.4	0.018	0.018	0.018	0.019	0.019	0.019	0.019
4.6	0.017	0.017	0.018	0.018	0.018	0.018	0.018
4.8	0.016	0.016	0.017	0.017	0.017	0.017	0.017
5.0	0.015	0.016	0.016	0.016	0.016	0.016	0.016
5.2	0.014	0.015	0.015	0.015	0.015	0.015	0.015
5.4	0.014	0.014	0.014	0.014	0.014	0.014	0.015
5.6	0.013	0.013	0.013	0.013	0.014	0.014	0.014
5.8	0.012	0.013	0.013	0.013	0.013	0.013	0.013
6.0	0.012	0.012	0.012	0.012	0.012	0.012	0.012
6.4	0.011	0.011	0.011	0.011	0.011	0.011	0.011
7.0	0.010	0.010	0.010	0.010	0.010	0.010	0.010
7.4	0.009	0.009	0.009	0.009	0.009	0.009	0.010
8.0	0.008	0.008	0.008	0.008	0.008	0.008	0.009
8.4	0.008	0.008	0.008	0.008	0.008	0.008	0.008
9.0	0.007	0.007	0.007	0.007	0.007	0.007	0.007
9.4	0.007	0.007	0.007	0.007	0.007	0.007	0.007
10.0	0.006	0.006	0.006	0.006	0.006	0.006	0.006

AR=1.1 alpha=4

MACH	40K L/D	50K L/D	60K L/D	70K L/D	80K L/D	90K L/D	100K L/D
0.2	6.096	6.000	5.899	5.791	5.677	5.555	5.429
0.4	6.230	6.143	6.050	5.951	5.846	5.733	5.616
0.6	6.322	6.239	6.150	6.055	5.954	5.845	5.733
0.8	3.584	3.558	3.530	3.500	3.468	3.432	3.394
1.0	1.056	1.052	1.048	1.044	1.039	1.033	1.028
1.2	2.267	2.265	2.262	2.260	2.257	2.254	2.251
1.4	2.460	2.457	2.454	2.452	2.448	2.445	2.441
1.6	3.304	3.298	3.292	3.284	3.277	3.268	3.258
1.8	3.730	3.722	3.713	3.703	3.692	3.680	3.668
2.0	4.291	4.278	4.264	4.249	4.233	4.214	4.195
2.2	4.677	4.661	4.644	4.625	4.604	4.581	4.557
2.4	5.152	5.130	5.106	5.079	5.051	5.019	4.986
2.6	5.408	5.384	5.358	5.329	5.298	5.263	5.227
2.8	5.785	5.751	5.714	5.674	5.631	5.583	5.533
3.0	6.006	5.968	5.928	5.884	5.837	5.785	5.730
3.2	6.227	6.189	6.149	6.104	6.056	6.003	5.948
3.4	6.425	6.384	6.339	6.291	6.238	6.181	6.120
3.6	6.587	6.543	6.494	6.442	6.385	6.322	6.257
3.8	6.789	6.743	6.694	6.640	6.582	6.518	6.451
4.0	6.957	6.909	6.857	6.801	6.740	6.673	6.603
4.2	7.109	7.059	7.005	6.946	6.882	6.813	6.740
4.4	7.260	7.208	7.152	7.091	7.025	6.953	6.877
4.6	7.433	7.380	7.323	7.261	7.194	7.120	7.043
4.8	7.582	7.527	7.469	7.405	7.336	7.261	7.182
5.0	7.720	7.665	7.604	7.539	7.469	7.391	7.310
5.2	7.822	7.764	7.702	7.634	7.562	7.482	7.398
5.4	7.963	7.905	7.841	7.773	7.699	7.617	7.532
5.6	8.065	8.005	7.940	7.870	7.794	7.711	7.624
5.8	8.189	8.128	8.062	7.991	7.914	7.829	7.741
6.0	8.263	8.200	8.133	8.060	7.982	7.896	7.805
6.4	8.458	8.393	8.324	8.249	8.168	8.080	7.987
7.0	8.850	8.784	8.713	8.636	8.553	8.462	8.366
7.4	9.025	8.958	8.885	8.807	8.722	8.629	8.532
8.0	9.180	9.111	9.036	8.956	8.868	8.773	8.673
8.4	9.436	9.366	9.291	9.209	9.121	9.024	8.923
9.0	9.550	9.479	9.402	9.319	9.229	9.130	9.027
9.4	9.765	9.693	9.616	9.532	9.441	9.342	9.238
10.0	9.877	9.805	9.726	9.642	9.550	9.450	9.344

AR=1.1 alpha=2

MACH	40K CD	50K CD	60K CD	70K CD	80K CD	90K CD	100K CD
0.2	0.026	0.026	0.027	0.028	0.029	0.030	0.031
0.4	0.025	0.026	0.026	0.027	0.027	0.028	0.029
0.6	0.024	0.025	0.025	0.026	0.027	0.027	0.028
0.8	0.040	0.041	0.041	0.042	0.043	0.043	0.044
1.0	0.084	0.084	0.085	0.085	0.086	0.087	0.087
1.2	0.256	0.257	0.257	0.257	0.258	0.258	0.259
1.4	0.222	0.223	0.223	0.224	0.224	0.225	0.225
1.6	0.118	0.118	0.118	0.119	0.119	0.120	0.120
1.8	0.091	0.091	0.091	0.092	0.092	0.093	0.093
2.0	0.066	0.066	0.066	0.067	0.067	0.067	0.068
2.2	0.054	0.054	0.054	0.055	0.055	0.055	0.056
2.4	0.041	0.042	0.042	0.042	0.043	0.043	0.043
2.6	0.037	0.038	0.038	0.038	0.038	0.039	0.039
2.8	0.028	0.029	0.029	0.029	0.029	0.030	0.030
3.0	0.025	0.026	0.026	0.026	0.026	0.027	0.027
3.2	0.024	0.025	0.025	0.025	0.025	0.025	0.026
3.4	0.022	0.022	0.022	0.022	0.023	0.023	0.023
3.6	0.020	0.020	0.020	0.020	0.020	0.021	0.021
3.8	0.019	0.019	0.019	0.019	0.019	0.020	0.020
4.0	0.017	0.018	0.018	0.018	0.018	0.018	0.019
4.2	0.016	0.016	0.017	0.017	0.017	0.017	0.017
4.4	0.015	0.015	0.016	0.016	0.016	0.016	0.016
4.6	0.015	0.015	0.015	0.015	0.015	0.015	0.016
4.8	0.014	0.014	0.014	0.014	0.014	0.015	0.015
5.0	0.013	0.013	0.013	0.013	0.014	0.014	0.014
5.2	0.012	0.012	0.013	0.013	0.013	0.013	0.013
5.4	0.012	0.012	0.012	0.012	0.012	0.012	0.013
5.6	0.011	0.011	0.011	0.012	0.012	0.012	0.012
5.8	0.011	0.011	0.011	0.011	0.011	0.011	0.011
6.0	0.010	0.010	0.010	0.010	0.011	0.011	0.011
6.4	0.009	0.009	0.010	0.010	0.010	0.010	0.010
7.0	0.009	0.009	0.009	0.009	0.009	0.009	0.009
7.4	0.008	0.008	0.008	0.008	0.008	0.008	0.008
8.0	0.007	0.007	0.007	0.007	0.007	0.008	0.008
8.4	0.007	0.007	0.007	0.007	0.007	0.007	0.007
9.0	0.006	0.006	0.006	0.006	0.006	0.006	0.007
9.4	0.006	0.006	0.006	0.006	0.006	0.006	0.006
10.0	0.005	0.005	0.006	0.006	0.006	0.006	0.006

AR=1.1 alpha=2

MACH	40K L/D	50K L/D	60K L/D	70K L/D	80K L/D	90K L/D	100K L/D
0.2	7.111	6.944	6.768	6.585	6.395	6.194	5.993
0.4	7.347	7.190	7.026	6.853	6.673	6.482	6.289
0.6	7.494	7.343	7.184	7.016	6.841	6.655	6.466
0.8	4.432	4.381	4.326	4.267	4.204	4.135	4.065
1.0	1.208	1.201	1.194	1.187	1.178	1.169	1.160
1.2	2.623	2.619	2.615	2.611	2.606	2.601	2.595
1.4	2.910	2.906	2.901	2.896	2.890	2.884	2.877
1.6	3.739	3.729	3.719	3.707	3.694	3.680	3.664
1.8	4.169	4.156	4.141	4.125	4.108	4.089	4.069
2.0	4.648	4.629	4.608	4.585	4.561	4.533	4.505
2.2	4.987	4.964	4.938	4.911	4.881	4.848	4.813
2.4	5.299	5.269	5.236	5.201	5.162	5.120	5.075
2.6	5.547	5.514	5.479	5.440	5.399	5.353	5.304
2.8	5.608	5.567	5.523	5.475	5.423	5.366	5.307
3.0	5.763	5.719	5.671	5.620	5.565	5.504	5.440
3.2	6.027	5.982	5.933	5.880	5.823	5.760	5.695
3.4	6.137	6.089	6.037	5.981	5.920	5.854	5.784
3.6	6.197	6.146	6.091	6.032	5.968	5.898	5.824
3.8	6.400	6.348	6.291	6.230	6.165	6.092	6.017
4.0	6.520	6.466	6.407	6.344	6.276	6.202	6.124
4.2	6.618	6.562	6.502	6.437	6.367	6.290	6.210
4.4	6.725	6.667	6.606	6.539	6.467	6.389	6.307
4.6	6.885	6.826	6.763	6.696	6.622	6.542	6.459
4.8	6.999	6.940	6.875	6.806	6.732	6.650	6.565
5.0	7.100	7.039	6.974	6.904	6.828	6.745	6.658
5.2	7.141	7.079	7.013	6.941	6.864	6.779	6.691
5.4	7.257	7.195	7.127	7.055	6.977	6.891	6.801
5.6	7.311	7.248	7.180	7.106	7.027	6.940	6.850
5.8	7.405	7.341	7.272	7.198	7.118	7.030	6.938
6.0	7.427	7.362	7.293	7.217	7.137	7.048	6.956
6.4	7.552	7.486	7.416	7.339	7.257	7.167	7.073
7.0	7.893	7.826	7.754	7.675	7.591	7.499	7.403
7.4	8.007	7.939	7.866	7.787	7.702	7.609	7.512
8.0	8.054	7.985	7.912	7.832	7.747	7.653	7.555
8.4	8.284	8.215	8.141	8.060	7.974	7.879	7.780
9.0	8.303	8.234	8.160	8.079	7.993	7.898	7.799
9.4	8.488	8.419	8.344	8.263	8.175	8.080	7.980
10.0	8.523	8.454	8.379	8.298	8.211	8.116	8.016

AR=1.1 t/c=1/12 alpha=0

MACH	40K CD	50K CD	60K CD	70K CD	80K CD	90K CD	100K CD
0.2	0.016	0.017	0.018	0.018	0.019	0.020	0.021
0.4	0.015	0.016	0.017	0.017	0.018	0.019	0.020
0.6	0.015	0.015	0.016	0.017	0.017	0.018	0.019
0.8	0.023	0.023	0.024	0.024	0.025	0.026	0.027
1.0	0.044	0.045	0.045	0.046	0.046	0.047	0.048
1.2	0.162	0.163	0.163	0.164	0.164	0.165	0.165
1.4	0.135	0.136	0.136	0.137	0.137	0.138	0.138
1.6	0.077	0.078	0.078	0.079	0.079	0.079	0.080
1.8	0.061	0.061	0.062	0.062	0.062	0.063	0.063
2.0	0.046	0.047	0.047	0.047	0.048	0.048	0.048
2.2	0.039	0.039	0.039	0.040	0.040	0.040	0.041
2.4	0.031	0.032	0.032	0.032	0.033	0.033	0.033
2.6	0.028	0.029	0.029	0.029	0.030	0.030	0.030
2.8	0.023	0.023	0.024	0.024	0.024	0.024	0.025
3.0	0.021	0.021	0.021	0.022	0.022	0.022	0.022
3.2	0.020	0.020	0.020	0.020	0.021	0.021	0.021
3.4	0.018	0.018	0.019	0.019	0.019	0.019	0.019
3.6	0.017	0.017	0.017	0.017	0.017	0.018	0.018
3.8	0.016	0.016	0.016	0.016	0.016	0.017	0.017
4.0	0.015	0.015	0.015	0.015	0.015	0.016	0.016
4.2	0.014	0.014	0.014	0.014	0.014	0.015	0.015
4.4	0.013	0.013	0.013	0.014	0.014	0.014	0.014
4.6	0.012	0.013	0.013	0.013	0.013	0.013	0.013
4.8	0.012	0.012	0.012	0.012	0.012	0.013	0.013
5.0	0.011	0.011	0.012	0.012	0.012	0.012	0.012
5.2	0.011	0.011	0.011	0.011	0.011	0.011	0.012
5.4	0.010	0.010	0.010	0.011	0.011	0.011	0.011
5.6	0.010	0.010	0.010	0.010	0.010	0.010	0.011
5.8	0.009	0.010	0.010	0.010	0.010	0.010	0.010
6.0	0.009	0.009	0.009	0.009	0.009	0.010	0.010
6.4	0.008	0.008	0.009	0.009	0.009	0.009	0.009
7.0	0.008	0.008	0.008	0.008	0.008	0.008	0.008
7.4	0.007	0.007	0.007	0.007	0.007	0.008	0.008
8.0	0.006	0.007	0.007	0.007	0.007	0.007	0.007
8.4	0.006	0.006	0.006	0.006	0.006	0.006	0.007
9.0	0.006	0.006	0.006	0.006	0.006	0.006	0.006
9.4	0.005	0.005	0.005	0.006	0.006	0.006	0.006
10.0	0.005	0.005	0.005	0.005	0.005	0.005	0.005

AR=1.1 alpha=0

MACH	40K L/D	50K L/D	60K L/D	70K L/D	80K L/D	90K L/D	100K L/D
0.2	8.006	7.710	7.410	7.105	6.798	6.483	6.177
0.4	8.426	8.139	7.846	7.547	7.244	6.931	6.626
0.6	8.675	8.393	8.104	7.808	7.506	7.194	6.889
0.8	5.557	5.444	5.326	5.201	5.071	4.932	4.792
1.0	1.617	1.600	1.583	1.564	1.543	1.521	1.498
1.2	2.954	2.947	2.940	2.932	2.924	2.915	2.904
1.4	3.414	3.406	3.397	3.387	3.376	3.363	3.351
1.6	4.055	4.038	4.021	4.001	3.981	3.957	3.933
1.8	4.429	4.408	4.385	4.361	4.334	4.304	4.272
2.0	4.706	4.679	4.649	4.617	4.582	4.544	4.503
2.2	4.935	4.903	4.868	4.831	4.790	4.746	4.699
2.4	4.992	4.955	4.915	4.871	4.824	4.772	4.718
2.6	5.207	5.167	5.123	5.076	5.026	4.970	4.912
2.8	4.914	4.870	4.823	4.772	4.717	4.657	4.594
3.0	4.991	4.945	4.896	4.842	4.785	4.722	4.656
3.2	5.272	5.224	5.171	5.115	5.055	4.989	4.920
3.4	5.289	5.239	5.185	5.127	5.065	4.997	4.926
3.6	5.251	5.200	5.145	5.086	5.023	4.953	4.881
3.8	5.436	5.383	5.326	5.265	5.200	5.128	5.053
4.0	5.503	5.449	5.391	5.329	5.262	5.189	5.113
4.2	5.546	5.491	5.432	5.369	5.301	5.226	5.149
4.4	5.606	5.551	5.491	5.426	5.358	5.282	5.204
4.6	5.739	5.682	5.621	5.556	5.486	5.409	5.329
4.8	5.814	5.757	5.695	5.629	5.558	5.480	5.399
5.0	5.875	5.817	5.755	5.688	5.616	5.537	5.456
5.2	5.865	5.807	5.744	5.677	5.605	5.526	5.444
5.4	5.950	5.891	5.828	5.760	5.687	5.608	5.525
5.6	5.963	5.904	5.841	5.773	5.700	5.620	5.537
5.8	6.025	5.965	5.902	5.833	5.760	5.679	5.596
6.0	6.006	5.947	5.883	5.815	5.742	5.662	5.578
6.4	6.068	6.009	5.945	5.877	5.803	5.722	5.639
7.0	6.335	6.275	6.210	6.139	6.064	5.982	5.896
7.4	6.394	6.334	6.269	6.199	6.123	6.041	5.955
8.0	6.363	6.304	6.239	6.170	6.096	6.015	5.930
8.4	6.550	6.490	6.425	6.355	6.279	6.197	6.111
9.0	6.506	6.447	6.383	6.314	6.240	6.159	6.075
9.4	6.650	6.590	6.526	6.456	6.382	6.300	6.215
10.0	6.633	6.574	6.511	6.442	6.369	6.289	6.205

AR=1.1 alpha=-2

MACH	40K CD	50K CD	60K CD	70K CD	80K CD	90K CD	100K CD
0.2	0.010	0.011	0.012	0.012	0.013	0.014	0.015
0.4	0.009	0.010	0.011	0.011	0.012	0.013	0.014
0.6	0.009	0.009	0.010	0.011	0.011	0.012	0.013
0.8	0.012	0.012	0.013	0.013	0.014	0.015	0.015
1.0	0.020	0.020	0.021	0.021	0.022	0.023	0.023
1.2	0.100	0.100	0.101	0.101	0.102	0.102	0.103
1.4	0.077	0.078	0.078	0.079	0.079	0.080	0.080
1.6	0.051	0.051	0.051	0.052	0.052	0.053	0.053
1.8	0.041	0.042	0.042	0.042	0.043	0.043	0.043
2.0	0.033	0.034	0.034	0.034	0.035	0.035	0.036
2.2	0.029	0.029	0.029	0.030	0.030	0.030	0.031
2.4	0.025	0.025	0.025	0.026	0.026	0.026	0.027
2.6	0.023	0.023	0.023	0.023	0.024	0.024	0.024
2.8	0.020	0.020	0.020	0.020	0.021	0.021	0.021
3.0	0.018	0.018	0.018	0.019	0.019	0.019	0.020
3.2	0.017	0.017	0.017	0.017	0.018	0.018	0.018
3.4	0.016	0.016	0.016	0.016	0.016	0.017	0.017
3.6	0.015	0.015	0.015	0.015	0.015	0.016	0.016
3.8	0.014	0.014	0.014	0.014	0.014	0.015	0.015
4.0	0.013	0.013	0.013	0.013	0.014	0.014	0.014
4.2	0.012	0.012	0.013	0.013	0.013	0.013	0.013
4.4	0.012	0.012	0.012	0.012	0.012	0.012	0.013
4.6	0.011	0.011	0.011	0.011	0.012	0.012	0.012
4.8	0.011	0.011	0.011	0.011	0.011	0.011	0.011
5.0	0.010	0.010	0.010	0.010	0.011	0.011	0.011
5.2	0.010	0.010	0.010	0.010	0.010	0.010	0.010
5.4	0.009	0.009	0.009	0.010	0.010	0.010	0.010
5.6	0.009	0.009	0.009	0.009	0.009	0.009	0.010
5.8	0.009	0.009	0.009	0.009	0.009	0.009	0.009
6.0	0.008	0.008	0.008	0.009	0.009	0.009	0.009
6.4	0.008	0.008	0.008	0.008	0.008	0.008	0.008
7.0	0.007	0.007	0.007	0.007	0.007	0.007	0.008
7.4	0.007	0.007	0.007	0.007	0.007	0.007	0.007
8.0	0.006	0.006	0.006	0.006	0.006	0.006	0.006
8.4	0.006	0.006	0.006	0.006	0.006	0.006	0.006
9.0	0.005	0.005	0.005	0.005	0.006	0.006	0.006
9.4	0.005	0.005	0.005	0.005	0.005	0.005	0.005
10.0	0.005	0.005	0.005	0.005	0.005	0.005	0.005

AR=1.1 t/c=1/12 alpha=-2

MACH	40K L/D	50K L/D	60K L/D	70K L/D	80K L/D	90K L/D	100K L/D
0.2	7.793	7.343	6.905	6.480	6.069	5.666	5.291
0.4	8.457	7.994	7.540	7.096	6.666	6.240	5.842
0.6	8.846	8.376	7.913	7.460	7.018	6.580	6.169
0.8	6.623	6.366	6.105	5.842	5.578	5.308	5.048
1.0	2.221	2.171	2.119	2.065	2.008	1.948	1.888
1.2	2.879	2.869	2.858	2.845	2.832	2.817	2.802
1.4	3.582	3.567	3.550	3.532	3.512	3.490	3.467
1.6	3.716	3.694	3.669	3.642	3.614	3.582	3.548
1.8	3.936	3.909	3.879	3.847	3.812	3.774	3.734
2.0	3.913	3.882	3.848	3.811	3.772	3.728	3.683
2.2	3.985	3.950	3.913	3.872	3.829	3.782	3.732
2.4	3.804	3.768	3.729	3.688	3.643	3.594	3.543
2.6	3.952	3.914	3.872	3.827	3.779	3.727	3.672
2.8	3.474	3.437	3.398	3.356	3.311	3.261	3.210
3.0	3.489	3.452	3.412	3.368	3.322	3.272	3.219
3.2	3.720	3.680	3.637	3.591	3.541	3.487	3.432
3.4	3.680	3.640	3.596	3.550	3.500	3.446	3.390
3.6	3.598	3.558	3.515	3.469	3.420	3.366	3.311
3.8	3.732	3.691	3.646	3.598	3.548	3.492	3.435
4.0	3.757	3.715	3.670	3.622	3.571	3.514	3.457
4.2	3.762	3.720	3.675	3.627	3.575	3.519	3.461
4.4	3.785	3.743	3.698	3.650	3.598	3.541	3.483
4.6	3.875	3.832	3.786	3.736	3.683	3.626	3.566
4.8	3.914	3.871	3.824	3.775	3.721	3.663	3.603
5.0	3.942	3.898	3.851	3.801	3.748	3.690	3.629
5.2	3.910	3.867	3.820	3.771	3.718	3.660	3.601
5.4	3.960	3.917	3.871	3.821	3.767	3.709	3.649
5.6	3.951	3.908	3.862	3.812	3.759	3.702	3.642
5.8	3.984	3.941	3.894	3.845	3.792	3.734	3.674
6.0	3.952	3.909	3.863	3.814	3.762	3.705	3.645
6.4	3.972	3.929	3.884	3.835	3.783	3.726	3.667
7.0	4.143	4.100	4.053	4.004	3.950	3.892	3.832
7.4	4.164	4.122	4.076	4.026	3.973	3.916	3.856
8.0	4.109	4.067	4.023	3.975	3.924	3.868	3.809
8.4	4.232	4.190	4.145	4.096	4.044	3.987	3.928
9.0	4.173	4.133	4.089	4.042	3.992	3.936	3.879
9.4	4.265	4.224	4.180	4.133	4.082	4.026	3.969
10.0	4.232	4.192	4.149	4.103	4.054	3.999	3.943

AR=1.1 alpha=-4

MACH	40K CD	50K CD	60K CD	70K CD	80K CD	90K CD	100K CD
0.2	0.007	0.008	0.008	0.009	0.010	0.011	0.012
0.4	0.006	0.007	0.007	0.008	0.009	0.010	0.011
0.6	0.006	0.006	0.007	0.008	0.008	0.009	0.010
0.8	0.006	0.007	0.007	0.008	0.008	0.009	0.010
1.0	0.008	0.008	0.009	0.009	0.010	0.010	0.011
1.2	0.069	0.069	0.070	0.070	0.070	0.071	0.072
1.4	0.048	0.049	0.049	0.050	0.050	0.051	0.051
1.6	0.037	0.038	0.038	0.038	0.039	0.039	0.040
1.8	0.031	0.032	0.032	0.032	0.033	0.033	0.034
2.0	0.027	0.027	0.028	0.028	0.028	0.029	0.029
2.2	0.024	0.024	0.024	0.025	0.025	0.025	0.026
2.4	0.021	0.022	0.022	0.022	0.023	0.023	0.023
2.6	0.020	0.020	0.020	0.020	0.021	0.021	0.021
2.8	0.018	0.018	0.018	0.019	0.019	0.019	0.019
3.0	0.017	0.017	0.017	0.017	0.017	0.018	0.018
3.2	0.015	0.016	0.016	0.016	0.016	0.017	0.017
3.4	0.014	0.015	0.015	0.015	0.015	0.015	0.016
3.6	0.014	0.014	0.014	0.014	0.014	0.015	0.015
3.8	0.013	0.013	0.013	0.013	0.013	0.014	0.014
4.0	0.012	0.012	0.012	0.013	0.013	0.013	0.013
4.2	0.011	0.012	0.012	0.012	0.012	0.012	0.013
4.4	0.011	0.011	0.011	0.011	0.011	0.012	0.012
4.6	0.010	0.011	0.011	0.011	0.011	0.011	0.011
4.8	0.010	0.010	0.010	0.010	0.010	0.011	0.011
5.0	0.009	0.010	0.010	0.010	0.010	0.010	0.010
5.2	0.009	0.009	0.009	0.009	0.010	0.010	0.010
5.4	0.009	0.009	0.009	0.009	0.009	0.009	0.010
5.6	0.008	0.009	0.009	0.009	0.009	0.009	0.009
5.8	0.008	0.008	0.008	0.008	0.009	0.009	0.009
6.0	0.008	0.008	0.008	0.008	0.008	0.008	0.009
6.4	0.007	0.007	0.007	0.008	0.008	0.008	0.008
7.0	0.007	0.007	0.007	0.007	0.007	0.007	0.007
7.4	0.006	0.006	0.006	0.006	0.007	0.007	0.007
8.0	0.006	0.006	0.006	0.006	0.006	0.006	0.006
8.4	0.005	0.006	0.006	0.006	0.006	0.006	0.006
9.0	0.005	0.005	0.005	0.005	0.005	0.005	0.005
9.4	0.005	0.005	0.005	0.005	0.005	0.005	0.005
10.0	0.005	0.005	0.005	0.005	0.005	0.005	0.005

AR=1.1 alpha=-4

MACH	40K L/D	50K L/D	60K L/D	70K L/D	80K L/D	90K L/D	100K L/D
0.2	4.332	3.984	3.660	3.360	3.083	2.821	2.586
0.4	4.876	4.491	4.132	3.799	3.489	3.196	2.933
0.6	5.205	4.799	4.419	4.065	3.737	3.425	3.146
0.8	4.966	4.609	4.271	3.952	3.653	3.367	3.106
1.0	2.483	2.346	2.211	2.080	1.952	1.827	1.709
1.2	1.395	1.388	1.380	1.372	1.363	1.352	1.342
1.4	1.909	1.896	1.881	1.866	1.850	1.832	1.813
1.6	1.683	1.669	1.654	1.638	1.620	1.601	1.581
1.8	1.728	1.712	1.695	1.676	1.657	1.635	1.613
2.0	1.616	1.600	1.583	1.564	1.544	1.523	1.500
2.2	1.606	1.589	1.571	1.551	1.531	1.508	1.484
2.4	1.466	1.450	1.433	1.414	1.395	1.373	1.351
2.6	1.519	1.502	1.483	1.464	1.443	1.420	1.396
2.8	1.271	1.256	1.241	1.224	1.206	1.186	1.166
3.0	1.268	1.253	1.237	1.220	1.202	1.182	1.162
3.2	1.360	1.344	1.326	1.308	1.288	1.267	1.245
3.4	1.333	1.317	1.300	1.282	1.263	1.242	1.220
3.6	1.291	1.275	1.259	1.241	1.222	1.202	1.181
3.8	1.341	1.325	1.307	1.289	1.270	1.248	1.226
4.0	1.345	1.329	1.312	1.293	1.274	1.252	1.230
4.2	1.342	1.326	1.308	1.290	1.270	1.249	1.227
4.4	1.346	1.330	1.313	1.295	1.275	1.254	1.232
4.6	1.378	1.362	1.344	1.326	1.306	1.284	1.261
4.8	1.389	1.373	1.356	1.337	1.317	1.295	1.272
5.0	1.396	1.380	1.362	1.344	1.324	1.302	1.279
5.2	1.380	1.364	1.347	1.328	1.308	1.287	1.265
5.4	1.396	1.380	1.363	1.344	1.325	1.303	1.281
5.6	1.390	1.374	1.357	1.338	1.319	1.297	1.275
5.8	1.399	1.383	1.366	1.348	1.328	1.307	1.285
6.0	1.384	1.368	1.352	1.334	1.314	1.293	1.272
6.4	1.387	1.371	1.355	1.337	1.318	1.297	1.276
7.0	1.446	1.430	1.413	1.395	1.376	1.355	1.333
7.4	1.450	1.434	1.418	1.400	1.381	1.360	1.338
8.0	1.424	1.409	1.393	1.376	1.357	1.337	1.316
8.4	1.467	1.452	1.436	1.418	1.399	1.379	1.358
9.0	1.441	1.426	1.411	1.394	1.376	1.356	1.336
9.4	1.472	1.458	1.442	1.425	1.407	1.387	1.367
10.0	1.457	1.443	1.427	1.411	1.394	1.374	1.354

Appendix G:

Flight Profile & Glide Analysis

AAE 416 - FLIGHT PROFILE FOR GLIDE PORTION

SURFACE A

EMPTY WEIGHT = 43000 LBS
EMPTY MASS = 1335.40 SLUGS

RANGE	Q	DENSITY	RANGE VELOCITY	mach	ANGLE OF DESCENT DEGREES	CL CALCED
100000	1500	3.2114E-05	9665.257	9.985	0.0000	0.013650
99000	1400	3.3704E-05	9114.614	9.416	0.1300	0.014625
98000	1400	3.5379E-05	8896.234	9.191	0.3300	0.014625
97000	1400	3.7145E-05	8682.180	8.970	0.3500	0.014625
96000	1400	3.9007E-05	8472.425	8.753	0.3800	0.014625
95000	1400	4.0970E-05	8266.963	8.541	0.4300	0.014625
94000	1300	4.3041E-05	7772.227	8.030	0.1800	0.015750
93000	1300	4.5525E-05	7557.213	7.808	0.4100	0.015750
92000	1300	4.7530E-05	7396.099	7.641	0.6500	0.015749
91000	1300	4.9963E-05	7213.772	7.453	0.7200	0.015749
90000	1300	5.2531E-05	7035.238	7.268	0.7000	0.015749
89000	1300	5.5243E-05	6860.378	7.088	0.6000	0.015750
88000	1200	5.8106E-05	6426.805	6.640	0.3000	0.017063
87000	1200	6.1132E-05	6265.724	6.473	0.7500	0.017062
86000	1200	6.4328E-05	6108.092	6.310	0.8000	0.017061
85000	1200	6.7706E-05	5953.769	6.151	1.0000	0.017060
84000	1200	7.1277E-05	5802.710	5.995	1.0000	0.017060
83000	1200	7.5053E-05	5654.856	5.842	1.0000	0.017060
82000	1200	7.8931E-05	5514.191	5.697	1.3000	0.017059
81000	1100	8.2787E-05	5155.018	5.326	0.5000	0.018614
80000	1100	8.6831E-05	5033.544	5.200	1.6000	0.018607
79000	1100	9.1073E-05	4914.920	5.078	1.6000	0.018607
78000	1100	9.5523E-05	4799.073	4.958	1.8000	0.018605
77000	1100	1.0019E-04	4685.966	4.841	1.8000	0.018605
76000	1100	1.0509E-04	4575.416	4.727	2.0000	0.018603
75000	1100	1.1022E-04	4467.670	4.616	2.0000	0.018603
74000	1100	1.1561E-04	4362.281	4.507	2.4000	0.018598
73000	1000	1.2126E-04	4061.217	4.196	0.9000	0.020473
72000	1000	1.2719E-04	3965.413	4.097	2.5000	0.020456
71000	1000	1.3341E-04	3871.870	4.000	3.0000	0.020448
70000	1000	1.3993E-04	3780.589	3.906	3.0000	0.020448
69000	1000	1.4678E-04	3691.318	3.814	3.0000	0.020448
68000	1000	1.5395E-04	3604.335	3.724	3.6000	0.020435
67000	1000	1.6148E-04	3519.294	3.636	3.6000	0.020435
66000	1000	[illegible]	[illegible]	[illegible]	3.8000	0.020431
65000	1000	[illegible]	[illegible]	[illegible]	4.0000	[illegible]
64000	1000	[illegible]	[illegible]	[illegible]	[illegible]	[illegible]
[illegible]	1000	[illegible]	[illegible]	[illegible]	[illegible]	[illegible]
[illegible]	1000	[illegible]	[illegible]	[illegible]	[illegible]	[illegible]
[illegible]	1000	[illegible]	[illegible]	[illegible]	[illegible]	[illegible]
[illegible]	[illegible]	[illegible]	[illegible]	[illegible]	[illegible]	[illegible]
[illegible]	[illegible]	[illegible]	[illegible]	[illegible]	[illegible]	[illegible]
[illegible]	[illegible]	[illegible]	[illegible]	[illegible]	[illegible]	[illegible]
[illegible]	[illegible]	[illegible]	[illegible]	[illegible]	[illegible]	[illegible]
[illegible]	900	[illegible]	[illegible]	[illegible]	[illegible]	[illegible]
[illegible]	900	[illegible]	[illegible]	[illegible]	[illegible]	[illegible]
54000	900	[illegible]	[illegible]	[illegible]	[illegible]	[illegible]
53000	900	3.1527E-04	[illegible]	2.469	10.0000	0.022405
52000	900	3.3072E-04	[illegible]	2.410	10.0000	0.022405

ORIGINAL PAGE IS OF POOR QUALITY

51000	900	3.4692E-04	2277.831	2.353	10.0000	0.022405
50000	900	3.6391E-04	2224.022	2.298	10.0000	0.022405
49000	800	3.8175E-04	2047.248	2.115	5.0000	0.025497
48000	800	4.0045E-04	1998.875	2.065	14.0000	0.024834
47000	800	4.2080E-04	1949.943	2.015	15.0000	0.024723
46000	800	4.4067E-04	1905.474	1.969	15.0000	0.024723
45000	800	4.6227E-04	1860.424	1.922	20.0000	0.024051
44000	800	4.8493E-04	1816.437	1.877	20.0000	0.024051
43000	800	5.0871E-04	1773.474	1.832	20.0000	0.024051
42000	800	5.3365E-04	1731.536	1.789	20.0000	0.024051
41000	795	5.5982E-04	1685.288	1.741	20.0000	0.024202
40000	790	5.8727E-04	1640.248	1.695	20.0000	0.024356
39000	785	6.1610E-04	1596.335	1.649	20.0000	0.024511
38000	780	6.4630E-04	1553.621	1.605	20.0000	0.024668
37000	775	6.7800E-04	1511.997	1.562	20.0000	0.024827
36000	770	7.1030E-04	1472.446	1.521	20.0000	0.024988
35000	765	7.3820E-04	1439.655	1.487	20.0000	0.025152
34000	760	7.6700E-04	1407.745	1.454	20.0000	0.025317
33000	755	7.9660E-04	1376.791	1.422	20.0000	0.025485
32000	750	8.2700E-04	1346.768	1.391	20.0000	0.025655
31000	745	8.5840E-04	1317.492	1.361	20.0000	0.025827
30000	740	8.9070E-04	1289.036	1.332	20.0000	0.026001
29000	735	9.2390E-04	1261.380	1.303	20.0000	0.026178
28000	730	9.5800E-04	1234.507	1.275	20.0000	0.026357
27000	725	9.9310E-04	1208.335	1.248	20.0000	0.026539
26000	720	1.0290E-03	1182.969	1.222	15.0000	0.027470
25000	715	1.0660E-03	1158.215	1.197	14.0000	0.027787
24000	710	1.1040E-03	1134.121	1.172	13.0000	0.028100
23000	705	1.1440E-03	1110.188	1.147	12.0000	0.028409
22000	700	1.1840E-03	1087.397	1.123	11.0000	0.028714
21000	695	1.2250E-03	1065.220	1.100	10.0000	0.029014
20000	690	1.2670E-03	1043.641	1.078	9.0000	0.029310
19000	640	1.3110E-03	988.1062	1.021	8.0000	0.031682
18000	590	1.3550E-03	933.1927	0.964	8.0000	0.034367
17000	555	1.4010E-03	890.1074	0.920	3.0000	0.036843
16000	520	1.4480E-03	847.4857	0.876	3.0000	0.039323
15000	485	1.4960E-03	805.2302	0.832	3.0000	0.042161
14000	450	1.5460E-03	762.9858	0.788	3.0000	0.045440
13000	415	1.5690E-03	727.3234	0.751	3.0000	0.049272
12000	380	1.6480E-03	679.0913	0.702	3.0000	0.053810
11000	345	1.7010E-03	636.9016	0.658	3.0000	0.059269
10000	310	1.7560E-03	594.7012	0.614	3.0000	0.065961
9000	285	1.8110E-03	561.0198	0.580	3.0000	[illegible]
8000	260	[illegible]	[illegible]	[illegible]	[illegible]	[illegible]
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[illegible]	[illegible]	[illegible]	[illegible]	[illegible]	[illegible]	[illegible]

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REA = 2100 FT^2

Cla-super x-subs	ANGLE OF ATTACK (DEGREES)	Cd MATCHED	DRAG	DECEL	DELTA DIST(FT.)	DIST(FT)	DELTA XDIST
0.380	-2.926	0.0049	15417	-11.545	0.000		0.00
0.380	-2.779	0.0052	15303	-11.386	4.54E+05		454103
0.380	-2.780	0.0053	15682	-11.558	1.70E+05	624254	170148
0.380	-2.780	0.0055	16073	-11.839	1.59E+05	783163	158906
0.380	-2.780	0.0056	16475	-12.123	1.48E+05	931566	148400
0.380	-2.780	0.0057	16888	-12.405	138627	1070193	138623
0.410	-2.784	0.0061	16730	-12.427	3.19E+05	1389460	319265
0.410	-2.784	0.0063	17211	-12.658	1.30E+05	1519661	130198
0.410	-2.784	0.0064	17589	-12.806	9.41E+04	1613727	94061
0.410	-2.784	0.0066	18037	-13.102	1.02E+05	1715383	101647
0.410	-2.784	0.0068	18498	-13.459	9.45E+04	1809891	94501
0.410	-2.784	0.0069	18973	-13.871	8.76E+04	1897479	87583
0.420	-2.657	0.0074	18754	-13.875	2.08E+05	2105085	207603
0.420	-2.657	0.0076	19239	-13.985	7.31E+04	2178180	73089
0.420	-2.657	0.0078	19738	-14.331	6.81E+04	2246232	68044
0.420	-2.657	0.0080	20253	-14.604	6.37E+04	2309960	63719
0.420	-2.657	0.0082	20783	-15.001	5.92E+04	2369152	59183
0.420	-2.657	0.0085	21330	-15.411	5.50E+04	2424115	54955
0.460	-2.860	0.0087	21877	-15.652	5.02E+04	2474304	50176
0.460	-2.666	0.0093	21520	-15.834	1.21E+05	2595312	121003
0.460	-2.667	0.0095	22042	-15.607	3.97E+04	2634963	39635
0.460	-2.667	0.0098	22577	-16.007	3.69E+04	2671825	36848
0.460	-2.667	0.0100	23125	-16.305	3.45E+04	2706334	34492
0.460	-2.667	0.0103	23686	-16.725	3.21E+04	2738406	32056
0.470	-2.717	0.0105	24261	-17.044	3.00E+04	2768441	30017
0.470	-2.717	0.0108	24850	-17.485	2.79E+04	2796304	27846
0.470	-2.717	0.0110	25454	-17.713	2.63E+04	2822573	26246
0.470	-2.489	0.0119	24944	-18.173	6.98E+04	2892346	69764
0.470	-2.491	0.0122	25552	-17.730	2.17E+04	2914032	21665
0.470	-2.492	0.0125	26176	-17.916	2.05E+04	2934492	20432
0.520	-2.732	0.0128	26814	-18.[illegible]	1.90E+04	2953479	18961
0.520	-2.732	0.0131	27471	-18.886	1.77E+04	2971138	17635
0.520	-2.733	0.0134	28143	-19.053	1.67E+04	2987792	16621
0.520	-2.733	0.0137	28834	-19.570	1.55E+04	3003269	15447
[illegible]	-2.734	0.0141	29541	-19.[illegible]	1.44E+04	3017719	14418
[illegible]	-2.734	0.0144	[illegible]	[illegible]	1.35E+04	[illegible]	[illegible]
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[illegible]	[illegible]	[illegible]	[illegible]	[illegible]	[illegible]	[illegible]	[illegible]
[illegible]	[illegible]	[illegible]	[illegible]	[illegible]	[illegible]	[illegible]	[illegible]
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0.620	-2.914	0.0212	[illegible]	-24.4[illegible]	5.46E+03	[illegible]	5378

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0.620	-2.914	0.0218	41158	-25.229	5.04E+03	3163257	4960
0.620	-2.914	0.0224	42278	-26.068	4.65E+03	3167903	4576
0.620	-2.628	0.0247	41454	-28.236	1.34E+04	3181274	13320
0.620	-2.690	0.0254	42644	-24.144	4.05E+03	3185327	3933
0.620	-2.700	0.0262	43934	-24.566	3932.825	3189260	3799
0.650	-2.805	0.0269	45190	-25.506	3360.915	3192621	3246
0.650	-2.865	0.0277	46554	-23.848	3556.991	3196178	3342
0.650	-2.865	0.0286	47985	-24.920	3245.106	3199423	3049
0.650	-2.865	0.0295	49491	-26.048	2960.626	3202383	2782
0.650	-2.865	0.0304	51078	-27.236	2698.462	3205082	2536
0.650	-2.851	0.0315	52656	-28.418	2780.292	3207862	2613
0.670	-2.902	0.0328	54339	-29.678	2523.457	3210386	2371
0.670	-2.889	0.0341	56143	-31.029	2290.245	3212676	2152
0.670	-2.875	0.0355	58079	-32.479	2071.328	3214747	1946
0.670	-2.862	0.0370	60172	-34.046	1873.999	3216621	1761
0.670	-2.848	0.0386	62377	-35.697	1653.320	3218274	1554
0.670	-2.834	0.0401	64342	-37.169	1284.537	3219559	1207
0.7	-2.912	0.0416	66453	-38.749	1172.437	3220731	1102
0.7	-2.899	0.0433	68722	-40.448	1065.448	3221797	1001
0.7	-2.885	0.0452	71172	-42.284	966.940	3222764	909
0.7	-2.871	0.0472	73852	-44.290	880.522	3223644	827
0.7	-2.856	0.0449	69811	-41.264	898.769	3224543	845
0.7	-2.842	0.0471	72763	-43.475	811.202	3225354	762
0.72	-2.887	0.0496	76053	-45.938	730.036	3226084	686
0.72	-2.873	0.0524	79766	-48.719	656.149	3226741	617
0.72	-2.799	0.0555	83985	-54.557	555.907	3227296	537
0.72	-2.773	0.0592	88887	-58.772	493.039	3227789	478
0.72	-2.749	0.0635	94669	-63.648	433.882	3228223	423
0.72	-2.724	0.0237	35105	-19.593	1370.755	3229594	1341
0.76	-2.820	0.0192	28215	-14.984	1671.245	3231265	1641
0.76	-2.797	0.0161	23426	-11.951	1997.338	3233263	1967
0.83	-2.961	0.0061	8806	-1.557	14611	3247873	14431
0.83	-2.798	0.0064	8617	-1.971	28617	3276490	28338
0.83	-2.612	0.0068	8450	-1.846	28569	3305059	28291
0.83	-2.441	0.0050	5834	-2.684	14636	3319695	14616
0.83	-2.270	0.0051	5597	-2.506	14774	3334469	14754
0.9	-2.301	0.0053	5370	-2.336	14950	3349419	14929
0.9	-2.092	0.0055	5152	-2.173	15244	3364663	15223
0.9	-1.848	0.0057	4954	-2.025	13125	3377789	13107
0.9	-1.559	0.0060	4761	-1.880	18039	3395827	18014
0.9	-1.211	0.0063	4596	-1.757	15804	3411631	15782
0.9	-0.785	[illegible]	4460	-1.655	15884	3427515	15867
0.9	[illegible]	[illegible]	[illegible]	-1.598	11990	3439505	11974
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1.1	[illegible]	[illegible]	[illegible]	[illegible]	[illegible]	[illegible]	[illegible]
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1.1	[illegible]	[illegible]	[illegible]	[illegible]	[illegible]	[illegible]	[illegible]
1.1	[illegible]	[illegible]	[illegible]	[illegible]	[illegible]	[illegible]	[illegible]
1.1	[illegible]	[illegible]	[illegible]	[illegible]	[illegible]	[illegible]	[illegible]
1.1	[illegible]	[illegible]	[illegible]	[illegible]	[illegible]	[illegible]	[illegible]
1.1	[illegible]	[illegible]	[illegible]	[illegible]	[illegible]	[illegible]	[illegible]
1.1	[illegible]	[illegible]	[illegible]	[illegible]	[illegible]	[illegible]	[illegible]

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XDIST	YDIST	DELTA TIME	TIME	
0.00				86986.43
86.00	1030.3	9.834704	9.834704	
118.23	980.0	6.147228	15.98193	
148.33	970.7	6.013320	21.99525	
176.43	984.2	5.882483	27.87773	
202.69	1040.4	5.755518	33.63325	
263.15	1003.0	8.923089	42.55634	
287.81	931.7	5.828732	48.38507	
305.63	1067.1	5.016247	53.40132	
324.88	1277.4	5.275605	58.67693	
342.77	1154.6	5.150777	63.82770	
359.36	917.2	5.021254	68.84896	
398.68	1087.0	7.905578	76.75454	
412.52	956.8	4.799560	81.55410	
425.41	950.1	4.690247	86.24434	
437.48	1112.2	4.597155	90.84150	
448.69	1033.0	4.487683	95.32918	
459.10	959.2	4.380471	99.70966	
468.60	1138.7	4.239611	103.9492	
491.52	1056.0	6.735510	110.6847	
499.02	1107.1	3.945481	114.6302	
506.00	1029.3	3.849848	118.4801	
512.53	1084.0	3.769615	122.2497	
518.61	1007.4	3.677664	125.9273	
524.29	1048.2	3.601703	129.5290	
529.56	972.4	3.510632	133.0397	
534.54	1100.0	3.449621	136.4893	
547.75	1095.9	5.756073	142.2454	
551.85	945.9	3.287403	145.5328	
555.72	1070.8	3.231479	148.7643	
559.31	993.7	3.150359	151.9146	
562.65	924.2	3.074675	154.9893	
565.80	1045.7	3.021715	158.0110	
568.73	971.8	2.948013	160.9590	
571.46	957.6	2.882629	163.8417	
574.00	940.9	2.818588	166.6602	
576.41	1044.4	[illegible]	[illegible]	
[illegible]	969.1	[illegible]	[illegible]	
[illegible]	899.1	[illegible]	[illegible]	
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[illegible]	[illegible]	[illegible]	[illegible]	
[illegible]	[illegible]	[illegible]	[illegible]	
[illegible]	1112.5	[illegible]	[illegible]	
596.90	1027.0	2.211603	197.9896	
597.92	948.3	2.150731	200.1403	

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598.85	874.7	2.090388	202.2307
599.72	806.8	2.031844	204.2626
602.24	1165.3	3.538553	207.8011
602.99	980.6	2.001751	209.8029
603.71	1017.9	1.995945	211.7988
604.32	869.9	1.867338	213.6662
604.96	1216.6	1.943734	215.6099
605.53	1109.9	1.878909	217.4888
606.06	1012.6	1.816267	219.3051
606.54	922.9	1.754864	221.0599
607.04	950.9	1.804113	222.8641
607.48	863.1	1.742195	224.6063
607.89	783.3	1.682393	226.2886
608.26	708.4	1.621817	227.9105
608.59	640.9	1.563707	229.4742
608.89	565.5	1.488595	230.9628
609.12	439.3	1.328311	232.2911
609.33	401.0	1.283365	233.5744
609.52	364.4	1.237142	234.8116
609.69	330.7	1.191684	236.0033
609.84	301.2	1.149772	237.1530
610.00	307.4	1.174417	238.3275
610.15	277.4	1.127947	239.4554
610.28	249.7	1.081657	240.5371
610.40	224.4	1.036533	241.5736
610.50	143.9	0.964302	242.5379
610.59	119.3	0.917809	243.4557
610.67	97.6	0.870115	244.3258
610.92	285.0	1.563034	245.8889
611.23	318.9	1.744123	247.6330
611.60	346.8	1.926514	249.5595
614.34	2285.6	5.264283	254.8238
619.71	3982.7	7.505969	262.3298
625.06	3976.1	7.712261	270.0420
627.83	766.0	5.666421	275.7084
630.63	773.2	5.831865	281.5403
633.45	782.4	6.015197	287.5555
636.34	797.8	6.235618	293.7911
638.82	686.9	5.935303	299.7264
642.23	944.1	7.162704	306.8891
645.22	827.1	6.930794	313.8199
648.22	[illegible]	[illegible]	321.0039
650.49	627.5	[illegible]	[illegible]
652.70	611.8	[illegible]	[illegible]
654.79	578.6	[illegible]	340.5816
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[illegible]	[illegible]	[illegible]	[illegible]

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Appendix H:

Weight and C.G. Analysis

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REFINED WEIGHT ESTIMATE - COMPONENT BREAKDOWN

   WTO=     53180.0 LB
    WL=     39000.0 LB
  FUEL=     13960.0 LB
 PILOT=       220.0 LB

          --- WINGS ---
NPIV=            1.0           SWEEP=          70.0 DEGREES
AR=              1.1           TR=             0.08
N=               4.5           TC=             0.08
S             2100.0
WING WT=                       3321.9 LB

          --- VERTICAL TAILS ---
VERT TAILS WT=                 1300.0 LB

          --- FUSELAGE ---
L=              85.0           KINL=            1.25
H=              10.0           Q=            1500.0
FUSELAGE WT=                   6138.7 LB

          --- LANDING GEAR ---
GEAR WT=                       1751.8 LB

          --- ENGINES ---
RAMJETS BASIC WT=             10000.0 LB
SCRAMJETS BASIC WT=            6000.0 LB

AIR CONTROL SYS:
INLET WT=                      2000.0 LB
NOZZLE WT=                      700.0 LB

FUEL SYSTEM:
FUEL CELL WT=                  2482.0 LB
BACKING AND SUPPORT WT=         536.9 LB
DUMP AND DRAIN SYS WT=           70.9 LB
   CONTROL SYS WT=              251.8 LB

ENGINE CONTROLS:
                85
                 7                                (ASSUMING SCRAMJET = RAMJET)
              [illegible]
       [illegible] WT=          [illegible] LB

              [illegible]
                               [illegible]              [illegible]
                 [illegible]   [illegible]

          [illegible]           [illegible]

               [illegible]
              [illegible]
   [illegible] WT               [illegible] LB
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--- INSTRUMENTS ---

FLIGHT INST WT= 16.7 LB
ENGINE INST WT= 35.8 LB
MISCELLANEOUS INST WT= 8.0 LB
ON BOARD COMPUTERS WT= 200.0 LB

TOTAL INSTRUMENT WT= 260.5 LB

--- FURNISHINGS ---

EJECTION SEAT WT= 171.2 LB
EMERGENCY EQUIPMENT WT= 73.7 LB
OXYGEN SYSTEM WT= 16.9 LB
AIR CONDITIONING WT= 114.1 LB
CABIN WINDOW AND COVER WT= 242.9 LB (FOR LOW SPEED USE ONLY)

TOTAL FURNISHINGS WT= 618.7 LB

--- COOLING SYSTEMS ---

NOSE COOLING SYS WT= 900 LB
WING COOLING SYS WT= 700 LB
VERTICAL STAB SYS WT= 150 LB
INLET SYS COOLING= 200 LB

TOTAL COOLING SYS WT= 1950 LB

TOTAL WEIGHT (NO FUEL)= 39066.4 LB
TOTAL WEIGHT (FULL)= 53026.4 LB

AN ADDITIONAL 153.6 LB IS AVAILABLE FOR STRUCTURAL USE

WEIGHT DETERMINED WITHOUT ASSUMING COMPOSITE MATERIALS

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CENTER OF GRAVITY DETERMINATION

COMPONENT	WEIGHT	DISTANCE (FROM NOSE)	MOMENT
WING	3321.9 LB	50.1 FT	166424.9 FT-LB
TAILS	1300.0 LB	73.5 FT	95550.0 FT-LB
FUSELAGE	6138.7 LB	41.2 FT	252915.8 FT-LB
FRONT GEAR	525.5 LB	17.0 FT	8934.3 FT-LB
REAR GEAR	1226.3 LB	65.3 FT	80075.6 FT-LB
RAMJETS	10000.0 LB	55.0 FT	550000.0 FT-LB
SCRAMJETS	6000.0 LB	57.0 FT	342000.0 FT-LB
INLET SYS	2000.0 LB	37.5 FT	225000.0 FT-LB
NOZZLE SYS	700.0 LB	71.0 FT	49700.0 FT-LB
FUEL SYS	3341.6 LB	46.2 FT	154380.2 FT-LB
ENGINE CONTROLS	108.1 LB	48.9 FT	5285.1 FT-LB
STARTING SYS	607.9 LB	47.1 FT	28630.5 FT-LB
CONTROLS	747.3 LB	68.3 FT	51038.0 FT-LB
INSTRUMENTS	260.5 LB	22.1 FT	5757.3 FT-LB
NOSE COOLING SYS	900.0 LB	3.1 FT	2790.0 FT-LB
WING COOLING SYS	700.0 LB	31.5 FT	22050.0 FT-LB
STAB COOLING SYS	150.0 LB	51.1 FT	7665.0 FT-LB
INLET COOLING SYS	200.0 LB	47.1 FT	9420.0 FT-LB
FURNISHINGS	618.7 LB	26.0 FT	16087.1 FT-LB
PILOT	220.0 LB	26.0 FT	5720.0 FT-LB
	39066.4 LB		2079424 FT-LB
	XCG (NO FUEL)=	53.2 FT	
FORWARD FUEL	13960.0 LB	46.2 FT	644952.0 FT-LB
	53026.4 LB		2724376 FT-LB
	XCG (FULL)=	51.4 FT	
	AVERAGE XCG=	52.3 FT	

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Appendix I:

Stability and Control Calculations

STABILITY AND CONTROL OF A/C:

--- AIRCRAFT PARAMETERS --

S=	2100 FT^2	WING PLANFORM AREA
AR=	1.1	ASPECT RATIO
B=	48 FT	WING SPAN
SWEEP=	70 DEGREES	SWEEP ANGLE
TR=	0.083	TAPER RATIO
CF=	8 FT	MEAN FLAP CHORD
CR=	60 FT	ROOT CHORD
SVS=	444 FT^2	TOTAL VERTICAL STAB AREA
ZV=	5 FT	VERT DISTANCE BETWEEN CG AND VERT STAB AC
LVS=	20.8 FT	HORZ DISTANCE BETWEEN CG AND VERT STAB AC
Z=	5 FT	DISTANCE BETWEEN THRUST LINE AND CG
HCG=	53 FT	DISTANCE TO CG
HAC=	57 FT	DISTANCE TO AC (BELOW M = .4)
VOL=	40 FT^3	VOLUME OF FUSELAGE
H=	7 FT	MEAN FUSELAGE HEIGHT
W=	12 FT	MEAN FUSELAGE WIDTH
CLA=	1.06	LIFT CURVE SLOPE (BELOW M = .4)
CL=	0.13	LIFT COEFFICIENT (BELOW M = .4)
CD=	0.02	DRAG COEFFICIENT (BELOW M = .4)
CM=	0.04	MOMENT COEFFICIENT (BELOW M = .4)
Q=	1500 LB/FT^2	DYNAMIC PRESSURE
QVS=	1000 LB/FT^2	STABILIZER DYNAMIC PRESSURE
M=	0.1	MACH NUMBER

-- LONGITUDINAL STABILITY --

MAC=	40.25 FT	MEAN AERODYNAMIC CHORD = 2/3*CR * ((1+TR+TR^2) / (1+TR))
SM=	0.099	STATIC MARGIN (AVERAGE) = (HAC-HCG) / MAC
SMF=	0.139	STATIC MARGIN (FULLY FUELED)
SME=	0.094	STATIC MARGIN (EMPTY)
CMA=	-0.105	ALPHA MOMENT COEFFICIENT (SHOULD BE <0) = -SM * CLA
CMCG=	0.030	CG MOMENT COEFFICIENT (SHOULD BE >0) =CM - SM*CL + Z/MAC*CD

[illegible]

[illegible]	[illegible]	[illegible]	[illegible]	[illegible]
[illegible]	[illegible]	[illegible]	[illegible]	[illegible]
[illegible]	[illegible]	[illegible]	[illegible]	[illegible]
		[illegible]	[illegible]	[illegible]
[illegible]	[illegible]	[illegible]	[illegible]	[illegible]
[illegible]	[illegible]	[illegible]	[illegible]	[illegible]
[illegible]	[illegible]	[illegible]	[illegible]	[illegible]
4	[illegible]	0.039	[illegible]	[illegible]
[illegible]	[illegible]	0.057	0.006	0.019

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-- LATERAL STABILITY --

THE LATERAL STABILITY DERIVATIVE:

     CLR=      0.069          ROLLING MOMENT COEFFICIENT
                                = CL/4 - 2 * LVS/B * ZV/B*CYB

       L= 10374000            ROLLING MOMENT
                                = CLR * (Q*S*B)

    CLBW=     -0.750          WING LATERAL STABILITY DERIVATIVE
                                FIG. 21-8 (NICOLAI)

    CLBF=      0.046          FUSELAGE LATERAL STABILITY DERIVATIVE
                                LOW WING EQN 21-13 (NICOLAI)

      SP=      1.401          STABILIZER PARAMETER

    CLBS=     -0.022          STABILIZER LATERAL STABILITY DERIVATIVE
                                = - CLA * SP * QVS/Q * SVS/S * ZV/B

     CLB=     -0.726          TOTAL LATERAL STAB DERIVATIVE (SHOULD BE <0)
                                = CLBW + CLBF + CLBS

THE STEADY STATE ROLL RATE:

     CLP=     -0.105          DAMPING IN ROLL COEFFICIENT
                                FIG 21-9 (NICOLAI)

    CLDA=      0.040          AILERON CONTROL POWER DERIVATIVE
                                FIG 4.28 (ROSKAM)

      DA=      20.00          MAXIMUM AILERON DEFLECTION (DEGREES)

       P=      31.75          ROLL RATE (DEGREES/SEC)   V=     100
                                = -2*V/B * CLDA/CLP * DA

       P=      63.49          ROLL RATE (DEGREES/SEC)   V=     200
                                = -2*V/B * CLDA/CLP * DA

       P=     126.98          ROLL RATE (DEGREES/SEC)   V=     400
                                = -2*V/B * CLDA/CLP * DA
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         -- DIRECTIONAL (WEATHERCOCK) STABILITY --

     VVS=      0.092           VERTICAL TAIL VOLUME COEFFICIENT
                                 = (LVS*SVS) / (B*S)

THE DIRECTIONAL STABILITY DERIVATIVE:

    CNBF=     -0.004           FUSELAGE DIR. STABILITY DERIVATIVE
                                 = -1/3 * (VOL/S) * (H/W)

    CNBW=      0.015           WING DIR. STABILITY DERIVATIVE
                                 EQN 21-22 (NICOLAI)

     CNB=      0.102           DIRECTIONAL STAB DERIVATIVE (SHOULD BE >0)
                                 = CNBF + CNBW + (VVS*CLA*SP*QVS/Q)

RUDDER REQUIREMENTS:

      DR=      20.00           MAXIMUM RUDDER DEFLECTION (DEGREES)

  CNDREO=      0.001           RUDDER CONTROL POWER (ENGINE OUT CONDITION)
                                 = (THRUST+ENG DRAG) / (Q*S*B*DR)

  CNDRCW=      0.001           RUDDER CONTROL POWER (CROSSWIND CONDITION)
                                 = -CNB * BETA / DR    (BETA=11.5 DEGREES)

       T=      0.498           TAU (DETERMINED USING MAXIMUM CNDR VALUE)
                                 = CNDR / (.9*CLAVS*VVS)

   RATIO=       0.28           TAIL AREA RATIO (SR/SVS)
                                 FIG 21.12 (NICOLAI)

      SR=     124.32           TOTAL RUDDER AREA
                                 = SVS * RATIO
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-- TRIM REQUIREMENTS --

CLD PARAMETER:

CLDT= 3.6 THEORETICAL CLD VALUE
FIG 6.1.1.1-7A (DATCOM)

CLDR= 0.7 CORRECTION RATIO OF CLD/CLDT
FIG 6.1.1.1-7B (DATCOM)

B= 0.949 BETA
= ((1 - M^2))^.5

CLD= 2.656 CLD
= 1/B * CLDR * CLDT

ELEVON CONTROL POWER:

CFC= 0.199 FLAP CHORD RATIO
= FLAP CHORD / AIRFOIL CHORD

CMD= -0.159 CMD
= -2 * ((1-CFC) * (CFC)^3)^.5

ECP= -0.423 ELEVON CONTROL POWER
= -SM*CLD +CMD

DEREO= 0.000 REQUIRED ELEVATOR DEFLECTION (ALPHA = 0)
= (SM*CLA*ALPHA) / ECP

DCO= 0.000 TRIM DRAG (ALPHA = 0)
FIG 9.25 (NICOLAI)

DERE5= -1.246 REQUIRED ELEVATOR DEFLECTION (ALPHA = 5)
= (SM*CLA*ALPHA) / ECP

DC5= 0.002 TRIM DRAG (ALPHA = 5)
FIG 9.25 (NICOLAI)

[illegible] VS CL FOR SM (MACH = .4):

ALPHA	CL	CD	SMF DERED	SME DERED	-SMF DERED	-SME DERED
[illegible]	[illegible]	[illegible]	[illegible]	[illegible]	-1.798	-0.947
[illegible]	0.05[illegible]	[illegible]	[illegible]	[illegible]	[illegible]	[illegible]
[illegible]	[illegible]	[illegible]	[illegible]	[illegible]	[illegible]	[illegible]
[illegible]	[illegible]	[illegible]	[illegible]	[illegible]	[illegible]	[illegible]
[illegible]	[illegible]	[illegible]	[illegible]	[illegible]	[illegible]	[illegible]
[illegible]	[illegible]	[illegible]	[illegible]	[illegible]	[illegible]	[illegible]
[illegible]	[illegible]	[illegible]	[illegible]	[illegible]	[illegible]	[illegible]
[illegible]	[illegible]	[illegible]	[illegible]	[illegible]	[illegible]	[illegible]

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MANEUVERS:

CLMAX = 0.2978 VSTALL = 229.5706 FT/SEC = 156.5254 MI/HR
WF = 39172 VLAND = 275.4847 FT/SEC = 187.8304 MI/HR
S = 2100
RHO = 0.002377

-- MAXIMUM ROLL RATE --

V (FT/SEC)	P (DEG/SEC)
275	87.30
300	95.24
350	111.11
400	126.98
450	142.86
500	158.73
550	174.60
600	190.48

-- MINIMUM TURNING RADIUS (3 G) --

V (FT/SEC)	R (MILES)	V (FT/SEC)	R (MILES)
275	0.16	1000	2.08
300	0.19	2000	8.31
350	0.25	3000	18.70
400	0.33	4000	33.25
450	0.42	5000	51.96
500	0.52	6000	74.82
550	0.63	7000	101.83
600	0.75	8000	133.01
		9000	168.34
		10000	207.82

MINIMUM PULL UP RADIUS (3 G) --

V (FT/SEC)	R (MILES)	V (FT/SEC)	R (MILES)
275	0.22	1000	2.94
300	0.26	2000	11.76
350	0.36	3000	26.45
400	0.47	4000	47.03
450	0.60	5000	73.45
[illegible]	0.73	6000	105.81
[illegible]	0.89	7000	144.01
[illegible]	1.06	8000	188.10

[illegible] DOWN RADIUS [illegible]

V (FT/SEC)	R (MILES)	V (FT/SEC)	R (MILES)
[illegible]	[illegible]	[illegible]	[illegible]
[illegible]	[illegible]	[illegible]	[illegible]
[illegible]	[illegible]	[illegible]	[illegible]
[illegible]	[illegible]	[illegible]	[illegible]
[illegible]	[illegible]	[illegible]	[illegible]
[illegible]	[illegible]	[illegible]	51.9[illegible]
[illegible]	0.44	[illegible]	72.01
[illegible]	0.53	8000	94.05

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Appendix J:

Cost Analysis

COST ANALYSIS (1990 DOLLARS)

ENGINEERING HOURS:

S=	5935.00	MAXIMUM SPEED (KNOTS)
A=	20100.00	AMPR WEIGHT
Q=	1.00	QUANTITY PRODUCED
E=	57512638	# OF ENGINEERING HOURS
RATE=	$45.75	HOURLY ENGINEERING RATE
COST=	$2,631,030,667	ENGINEERING LABOR COST

DEVELOPMENT SUPPORT:

D=	$1,958,839,157	DEVELOPMENT SUPP COST

FLIGHT TEST OPERATIONS:

F=	$55,340,978	FLIGHT TEST OP COSTS

TOOLING:

T=	19203647	# OF TOOLING HOURS
RATE=	$32.76	HOURLY TOOLING RATE
COST=	$629,111,470	TOOLING COSTS

MANUFACTURING LABOR:

L=	4960105	# OF MANUF HOURS
RATE=	$25.51	HOURLY MANUF RATE
COST=	$126,512,434	MAUF LABOR COSTS

QUALITY CONTROL:

Q/C=	644814	# OF Q/C HOURS
RATE=	$25.51	HOURLY Q/C RATE
COST=	$16,447,615	QUALITY CONTROL COSTS

MATERIAL AND EQUIPMENT:

M=	$16,258,046	MATERIAL & EQUIP COSTS

ENGINE AND AVIONICS:

[illegible]	[illegible]	[illegible]
[illegible]	[illegible]	[illegible]
COST=	$10,202,500	[illegible] COSTS
AVIONICS=	$1,700,000	AVIONICS COSTS

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