# SR-SCARLET 1

## Peregrin

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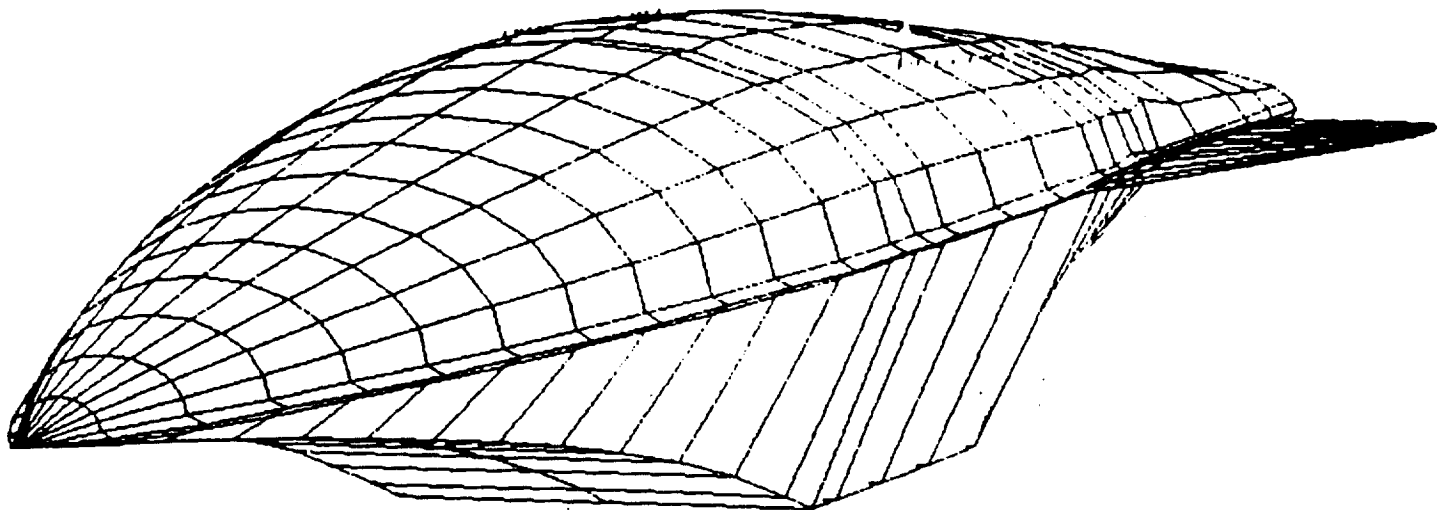P. 112

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# Abstract

This report presents a reconnaissance aircraft with a lifting body configuration. The aircraft is capable of flying a distance of 6000 nmi at Mach 5 with a payload of 7500 lbs. The aircraft does not require a runway for takeoff for it is air launched from a carrier aircraft. Specifically this report addresses the areas of external aerodynamics, cost, thermal protection systems, propulsion, stability and control, and materials. Each area is represented by a separate section; thus; allowing for selective reading.

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# External Aerodynamics

## Summary

The purpose of this section is to address questions affecting configuration selection, the flight regime for cruise, and characteristics of the aircraft in all regimes of flight. Specifically, this section investigates volumetric efficiency and lift and drag characteristics.

## Introduction

Group Scarlet 1 was divided into many sub-groups, one of which was External Aerodynamics. The responsibilities of this group was to decide on the type of configuration that would best suite the cruise environment and to analyze this configuration in all regimes of flight. The results of the research conducted by the External Aerodynamics group is presented in this section. The research addresses the following questions:

1. Which type of aircraft configuration would best suite the Mach 5 flight regime and would also lend itself to be launched by a carrier aircraft?

2. How high a value would the Lift and Drag coefficients of this configuration attain and what Mach Number should this aircraft fly at?

3. What values would the off design Lift and Drag Coefficients of this configuration attain especially through the Drag Rise region and at landing?

Following are the answers to the above questions that were expected at the onset of the research:

1. A lifting body was expected to be the best configuration.

2. Lift to drag ratios of 3.0 and Cl's and Cd's in the orders of .02 and .007 respectively were expected.

3. Off design Cl's and Cd's through the drag rise region were expected to be around .02 and .2 respectively while Lift to drag ratios of around 2.0 were expected at landing.

The results of the research show that a hypersonic reconnaissance aircraft with a lifting body configurations is the most realistic possibility for the requirements given. However, more detailed studies of the aerodynamic characteristics of the configurations need to be conducted before the development of an accurate model can be agreed upon. The remainder of this section contains sections on Research Methods, Research Results, and Conclusions and Recommendations.

## Configuration Selection

### What configuration best suites the requirements for cruise and the requirement for launch from a carrier aircraft?

Primary research into the characteristics of Wave Riders, Lifting Bodies and conventional Wing Body configurations was done using NASA technical memos and articles in *Aerospace America*. These articles were used to establish the base configuration. It was found that a Wave Rider configuration gave very good values of Lift to Drag ratios for the cruise region. These values, if further analysis were performed, would be in the range of 6 to 10. However, Wave Rider configurations, due to there streamline geometry, give poor volumetric efficiencies. A Lifting Body configuration would be expected to give Lift to Drag ratios, for the cruise region, in the range of 2 to 5. Another characteristic of the lifting body is that it is very volumetrically efficient and placement of payload and other required equipment such as avionics and thermal protection

systems would not be a problem. The last configuration considered was a conventional Wing Body configuration. The Lift to Drag ratios expected for this configuration were expected to be in the same range as those for the Lifting Body Configuration. However, Wing Body configurations are not as volumetric efficient as Lifting Bodies. Therefore, due to the requirement that the hypersonic vehicle be dropped from a carried aircraft, the Lifting Body configuration was chosen; the configuration provides reasonable Lift to Drag ratios and high volumetric efficiency. The high volumetric efficiency would mean that the vehicle would be smaller than the equivalent Wave Rider and Wing Body configurations.

## Cruise Characteristics

What are the aerodynamic characteristics of the Lifting Body configuration and what Mach number should the aircraft fly at?

In conducting research on aerodynamic characteristics of a lifting body configuration, various analytical tools had to be used. The configuration itself had to be defined first. The front under surface was of particular importance. For it was from the front under surface that the majority of the aircraft's lift was generated and the required airflow characteristics for the engines were developed. The front under surface of the aircraft was designed using a method of characteristics code. The under surface geometry was altered until the results from the computer code showed no shock waves in the flow. These shocks were observed in the characteristics as cross-over of the characteristic lines (see Figure A1). The under surface geometry was established when the characteristics successfully traversed the length of the front surface without crossover(see Figure A2). The width of the vehicle was governed by the size of the Mach cone at Mach 5. The vehicle was not allowed to pass outside

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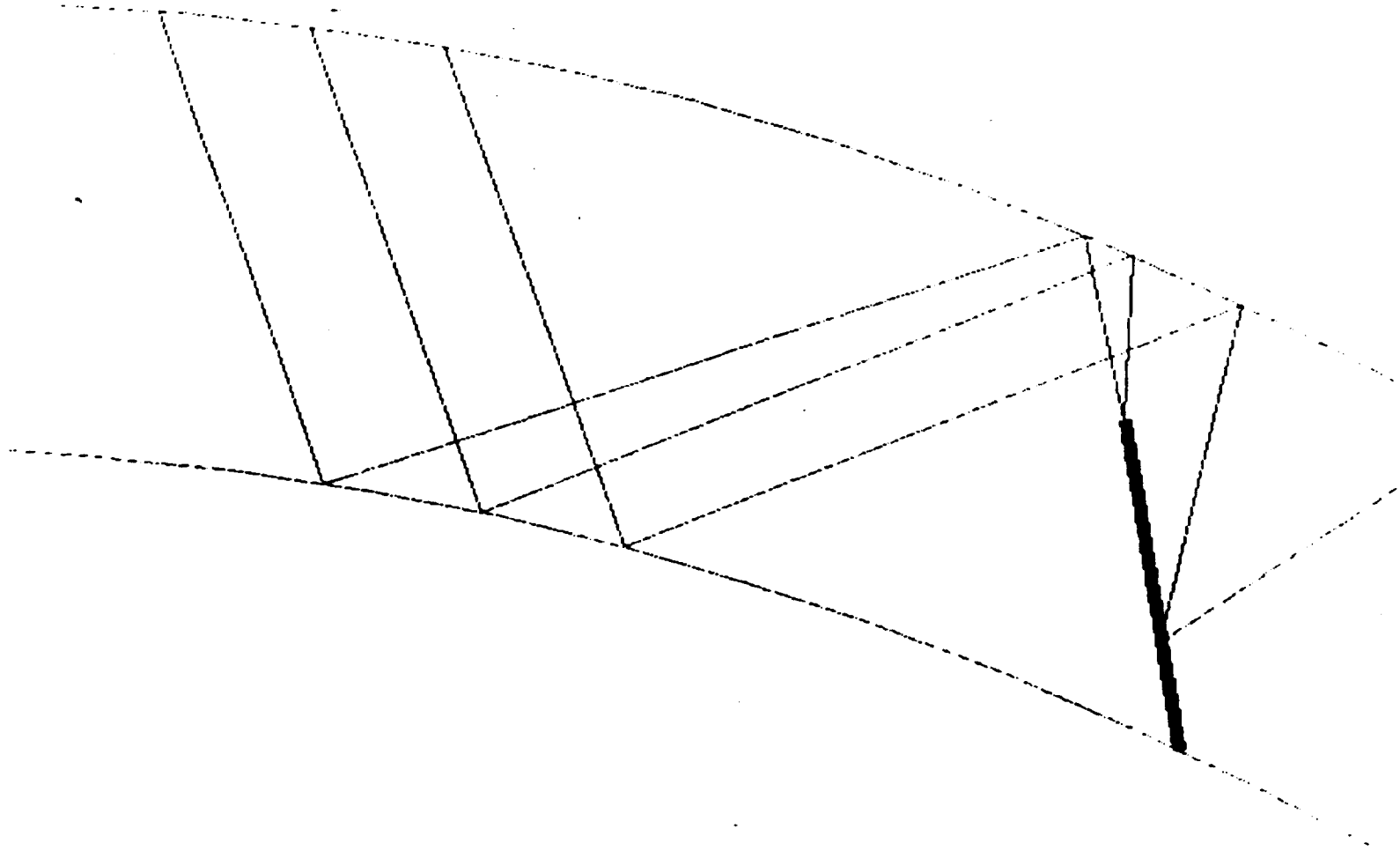
Figure A1 Crossover of Characteristics

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# CHARACTERISTICS ON LOWER SURFACE
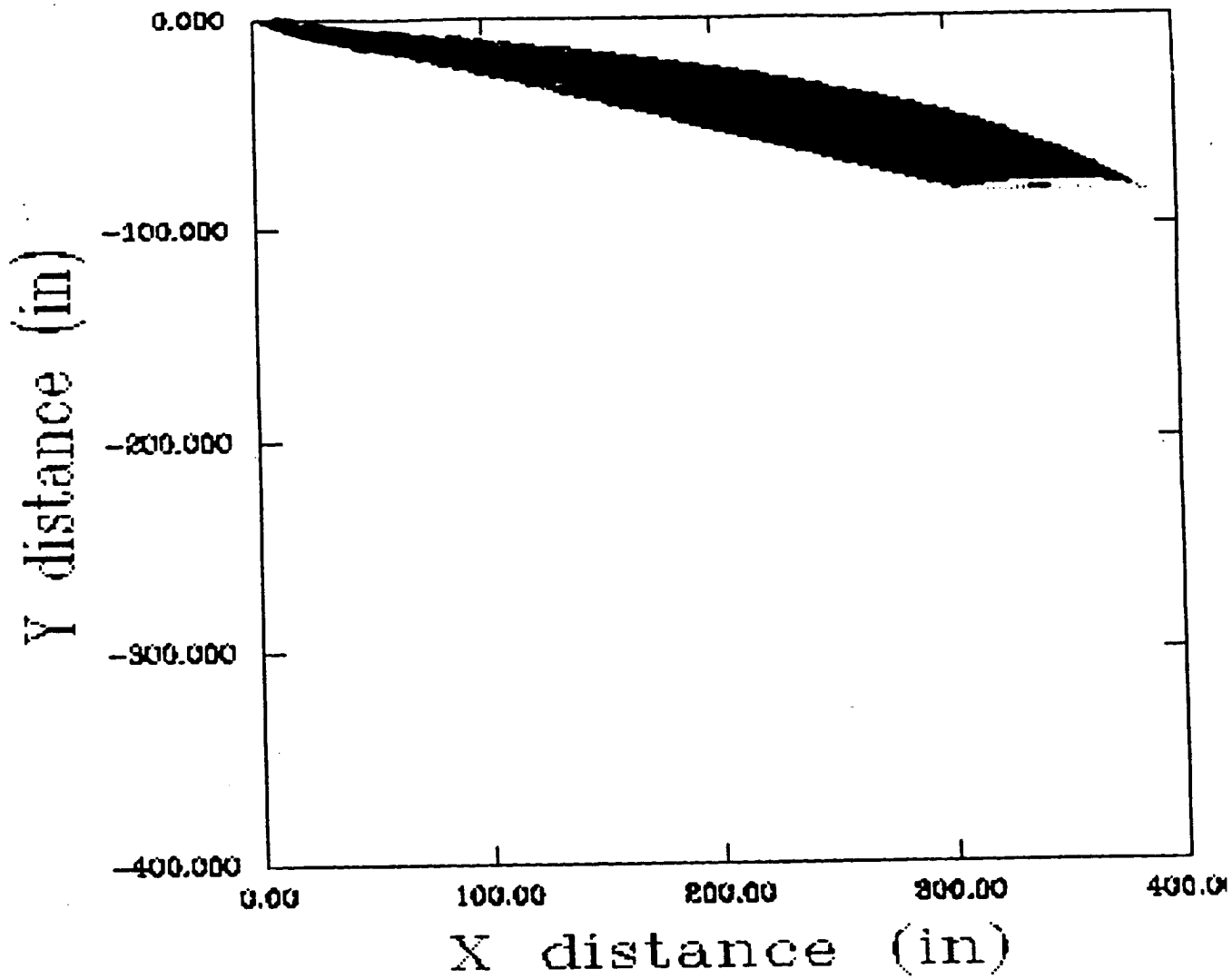


Figure A2 Characteristics off under surface of vehicle

of the Mach cone. The top surface was chosen to be an ellipse because it would increase the volume of the aircraft without adversely effecting the Lift and Drag.

The analysis of the base configuration was done using a computer program called APAS (Aerodynamic Preliminary Analysis System). The part of APAS that was used to analyze the vehicle was HABP (Hypersonic Arbitrary Body Program). The body geometry was input into APAS and divided into six parts: Top front, top mid, top aft, bottom front, bottom mid, and bottom aft. The code then analyzed the various parts using tangent cone, tangent wedge, and Dahlem Buck theories (see Figure A3). The methods chosen to analyze the configuration were governed by the Low Hypersonic Speed Region Pressure Method Selection Rationale stated in the Moore and Williams AIAA paper(Moore and Williams P.4). It should be noted that the bottom aft part of the configuration was not analyzed due to fact that this section is the expansion surface for the engines. Since engine on characteristics of the vehicle were desired for cruise, this surface was omitted from the analysis. From the aerodynamic characteristics obtained from APAS for Mach numbers ranging from 2 to 5, a graph of Cl vs Cd was generated to obtain the optimal Mach number for cruise (see Figure A4). The gain in L/D decreases as the Mach number increases. Therefore the optimal Mach number was decided to be Mach 5 considering the gain in L/D to go to Mach 6 was not substantial enough to justify the requirements imposed on engines and thermal protection systems. For Mach 5, the maximum L/D was calculated to be 3.6 with a Cl and Cd at zero angle of attack to be .02343 and .00732 respectively.

# Analysis Methods

Tangent Cone Emp.

Dahlem-Buck Emp.

Tangent Wedge

Not Analysized

Tangent Cone Emp.
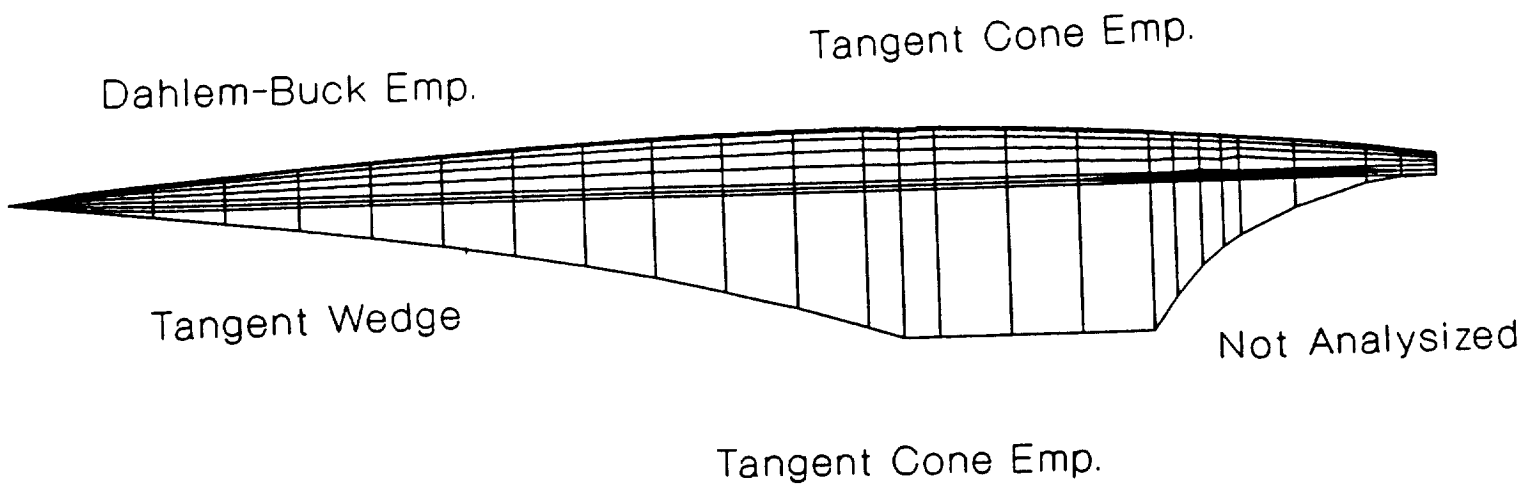
**Figure A3 Methods of Analysis**
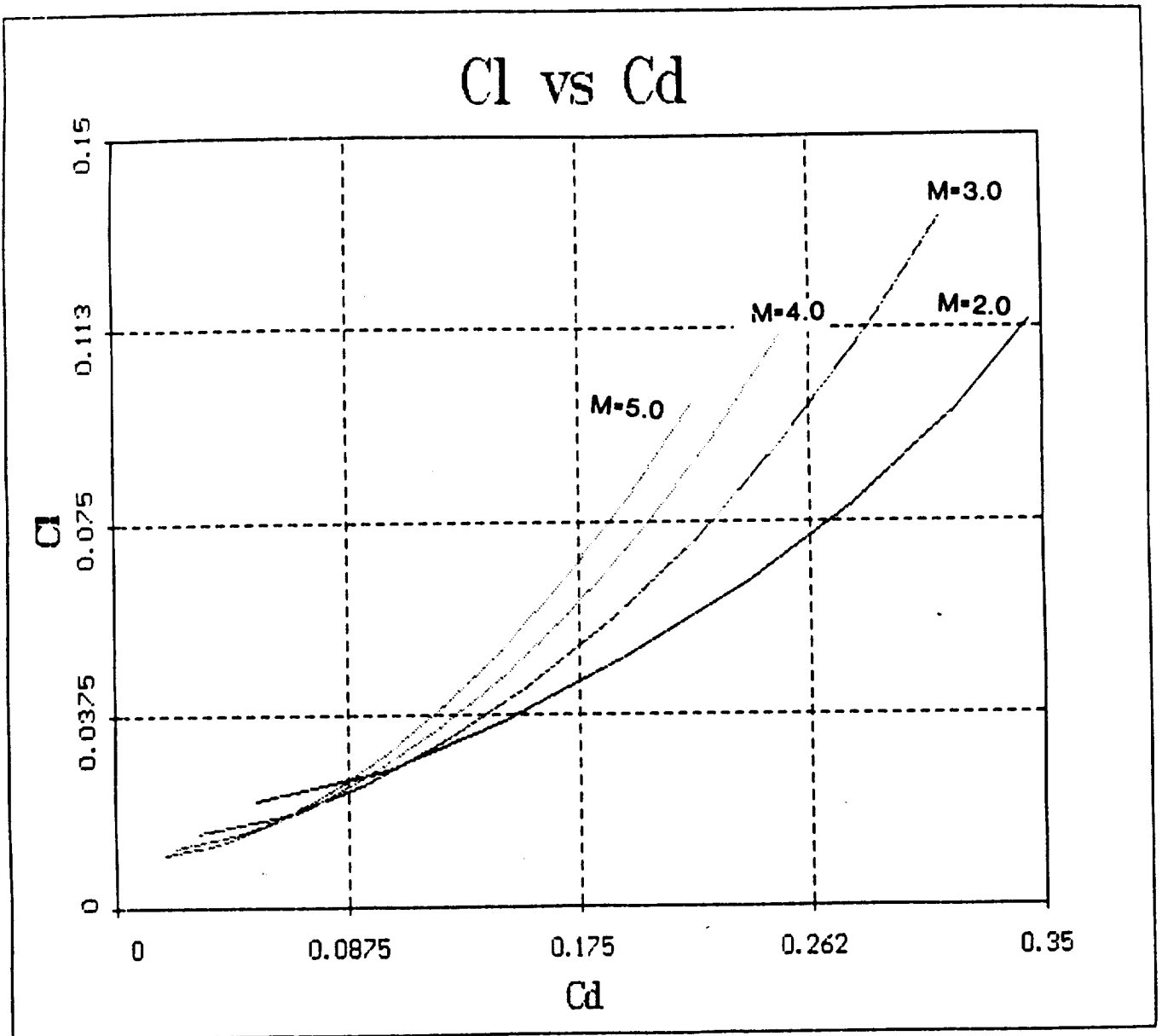
Figure A4 Cl vs Cd for supersonic Mach numbers

## Off Design Characteristics

### What are the off design characteristics of the lifting body configuration especially in the drag rise region and at landing?

The high subsonic and transonic regions up to Mach 2 were analyzed using an equivalent wing body configuration. This configuration consisted of a cone and cylinder for the fuselage and a delta wing matching the planform area and aspect ratio of the lifting body. The fuselage was created to approximately match the cross-sectional areas of the lifting body to get a good approximation of the pressure drag and skin friction drag. The wing was used to get an approximation of the lift on the vehicle at subsonic speeds. Since the lifting body has an effective thickness of 14%, the wing was given a 14% thickness to best approximate this lift. The thickness were approximated so as to give a close approximation of the Cd in the drag rise region. This geometry was then input into an empirical code that returned values for cl and cd for various alpha's and Mach numbers ranging from .8 to 2 (see Figure A5). In the drag rise region, especially at Mach 1, at zero angle of attack, the lift to drag ratio is .08 with a Cl and Cd of .0137 and .1735 respectively. At sixteen degrees angle of attack, however, the lift to drag ratio is 1.56 with a Cl and Cd of .4066 and .2602 respectively.

For the landing characteristics, a model was created and tested in the Low Speed Wind Tunnel at The Ohio State University under the supervision of Prof. Haritonidis. The model was suspended from the top of the wind tunnel with very thin wires. The angles that the model deflected for various flow velocities were recorded and were analyzed to produce basic lift and drag characteristics for the model. The back surface was included in the analysis since the engines would be off for landing. From the test data, the drag due to

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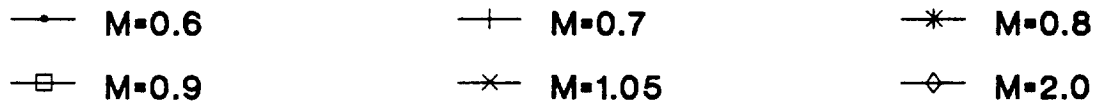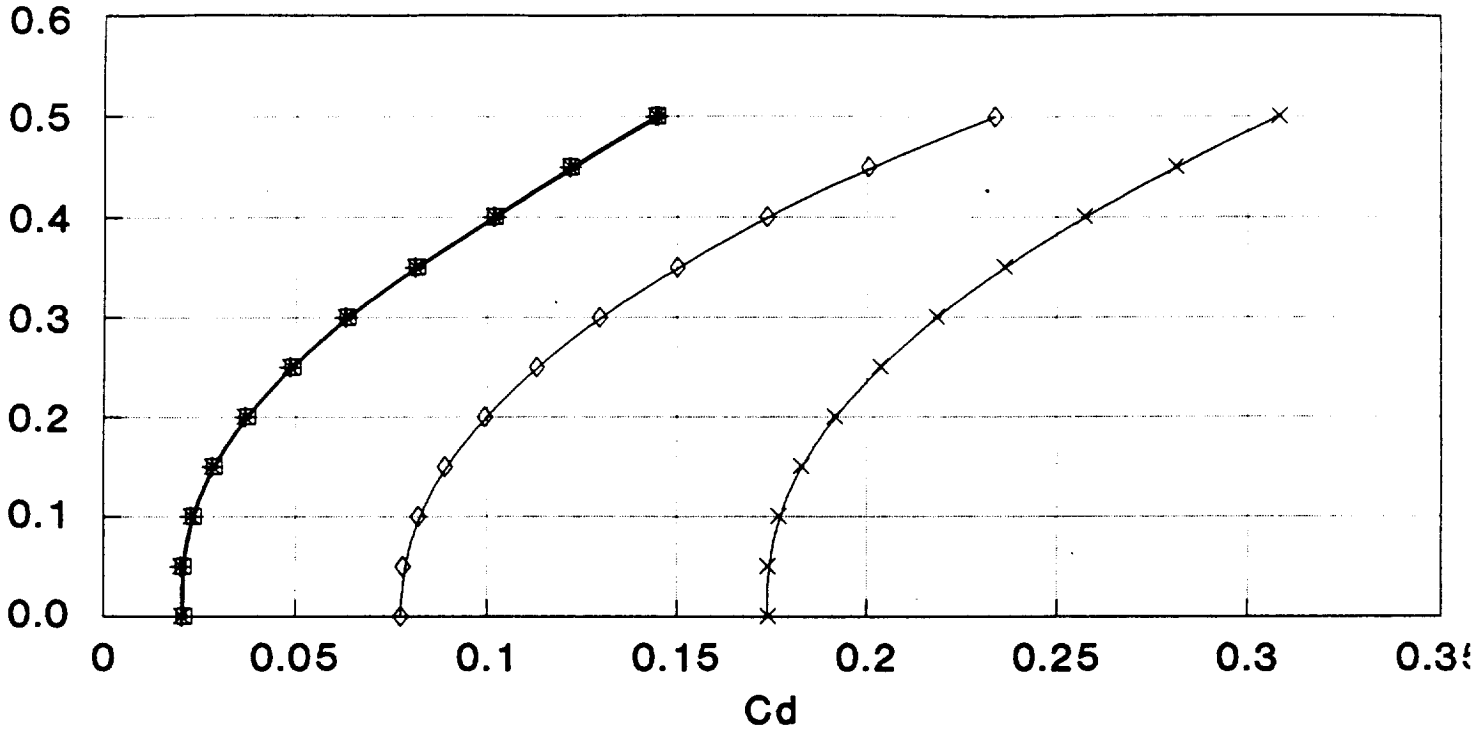
# Cl vs Cd



Figure A5 Cl vs Cd for subsonic and transonic Mach Numbers

the wires were calculated and were subtracted from the total drag. It should be noted that the wind tunnel results were for laminar flow over the model. To get accurate values, the data obtained from the tunnel should be scaled for turbulent boundary layers. However, it was decided not to do this scaling so that a conservative result would be presented. Compressibility corrections were also not performed due to the low landing speeds (under Mach .3). It can be seen that the maximum L/D is approximately 3.0 at a twelve degree angle of attack. Landing at this maximum L/D would put the vehicle at a eighteen degree glide slope. However it can be seen that at zero degree angle of attack, due to the negative lift generated by the negative mean camber line of the vehicle body, and the low drag, that the L/D is approximately -2.6 (see Figure A6).

### Summary and Recommendations

The research results show the lifting body configuration to be an excellent choice for a hypersonic reconnaissance aircraft with the given requirements. The configuration exhibits excellent lift to drag ratios at the cruise conditions and excellent volumetric efficiency. This would allow the vehicle to be small enough to be mounted on various carrier aircraft such as a C-5 or Boeing 747 thereby making its mission profile highly flexible. For further studies, a better approximation of the characteristics of the vehicle through the transonic region is recommended. Also, consideration should be given to landing the vehicle up side down. With a small negative angle of attack, inverted, high L/D's would be generated giving a smaller glide slope and skids instead of landing gear could be used, thus reducing the weight of the vehicle.

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# L/D vs Alpha
## Wind Tunnel Model
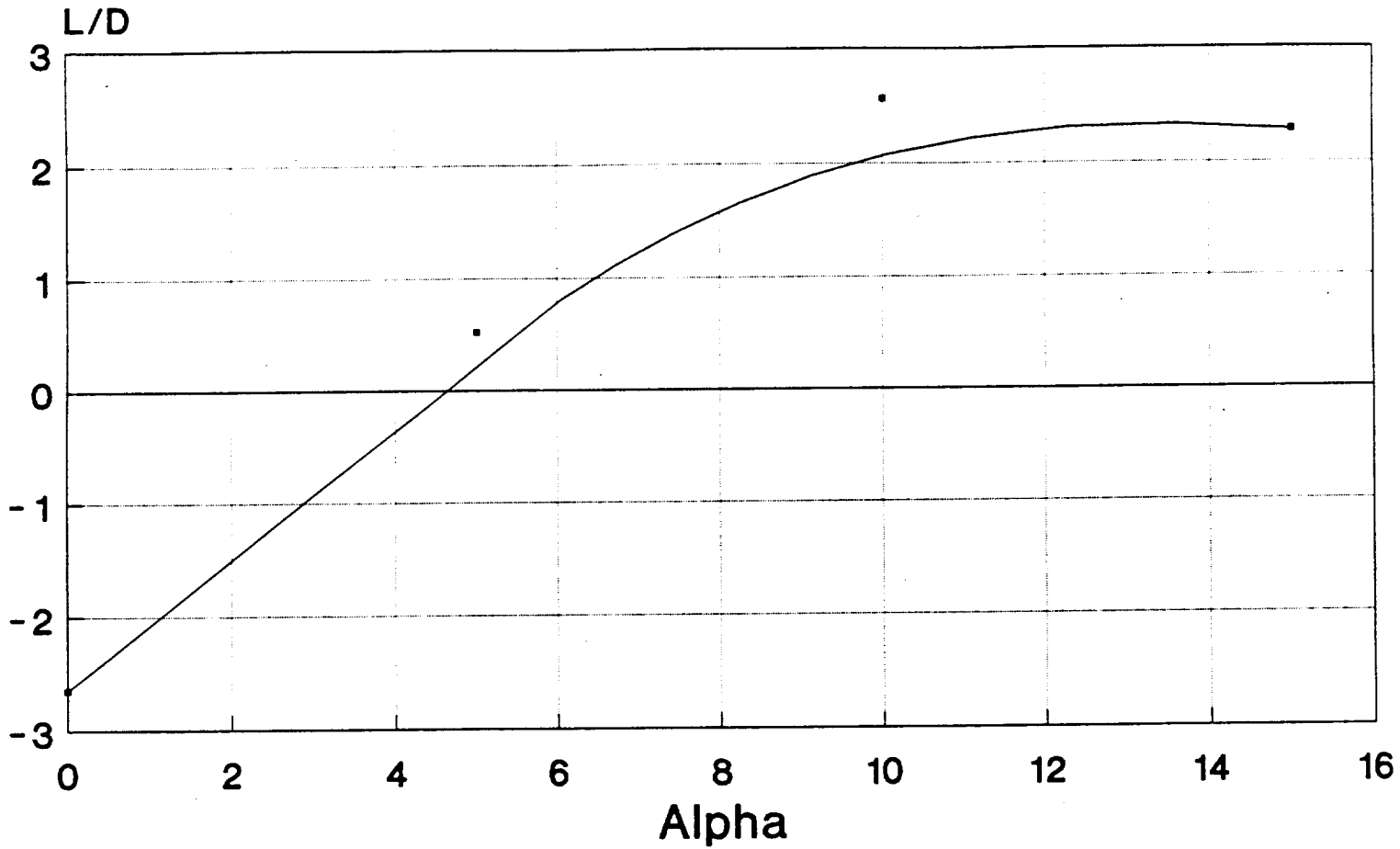


Figure A6 L/D for Wind Tunnel Tests

# References

Moore, W. E. and Williams, J. E. "AIAA Journal," <u>Aerodynamic Prediction Rationale For Analyses of Hypersonic Configurations</u> (1989):4

NASA Langley Research Center. 1991. APAS. Computer Software.

Dr. Michael Foster, Professor Aeronautical/Astronautical Engineering, The Ohio State University, personal consultations, January -June 1992.

Dr. Joseph Haritonidis, Associate Professor Aeronautical/Astronautical Engineering, The Ohio State University, personal consultations, April-June 1992.

Mr. Mondair Yahioui, PhD Candidate,Aeronautical/Astronautical Engineering, The Ohio State University, personal consultations, January -June 1992.

# Propulsion System

# Summary

This research report covers the analysis and process which led to the development of the propulsion system for the reconnaissance aircraft. Reductions in the weight and size of the aircraft considerably reduced ramjet and rocket engine performance requirements. A computer program originating from NASA Lewis Research Center was utilized to get data for this report. The propulsion system which results from the analysis is much smaller in comparison to its predecessors. The system integrates well with the rest of the aircraft and with the design goals of the group. These design goals were as follows:

- Meet the basic mission requirements.
- Use as small amount of fuel as possible.
- Propulsion system should power the aircraft with efficiency.
- The propulsion system should integrate well with the aircraft configurations.

As will be seen in the following report, all of the above design goals were met to the satisfaction of the design team.

## Introduction

A brief history of the engine configuration will now be given. The size of the previous aircraft configuration was much greater than the present configuration. The length of the original aircraft configuration was 105 ft. as compared to 61 ft for the present configuration. This reduction in size decreased the amount of thrust the aircraft needed and therefore decreased the size of the engines. The rockets required to bring the previous aircraft configurations to speed and altitude generated a thrust of 250,000 lbs a piece; with four boosters the total thrust output was 1 million pounds of thrust. In comparison the final configuration requires four rocket boosters generating a combined thrust of just under 50,000 lbs. The research methods used and the results of this research will be outlined in the following sections.

## Research Methods

The two main methods of research were the study of articles and memorandums written on hypersonic engines and inlets, and the use of RAMSCRAM, a program written by Leo Burkardt and Leo Franciscus at the NASA Lewis Research Center. The combination of these two methods and the design goals we were looking for produced the final engine and rocket system.

## Results of the Research

A numerous number of cases were run on RAMSCRAM to establish the characteristics of the ramjet and liquid rocket propulsion system. The propulsion system will be looked at in four basic sections as outlined in the following paragraphs.

## Inlet Design

The basic inlet system that was chosen comprised of a series of 2-D ramps which slow the incoming air flow to subsonic speeds. Two-D inlets were chosen because they require less wetted area and their performance is not adversely affected by changes in angles of attack. The reduced sensitivity to changes in angle of attack was a big factor since the mission profile requires the aircraft to fly at various angles of attack throughout the flight. Once the basic configuration of the inlets was chosen, numbers and concepts needed to be looked at. It was decided that two moveable ramps would be used on the upper and lower surface of the inlets. This was done because the ramjet would operate at different mach numbers at various altitudes and the inlet areas could be changed to give the required airflow into the engine. After careful research and basic number crunching the length of the inlet section was finally determined to be approximately 7.3 ft. The inlet was designed to swallow the normal shock at its "throat" at every operating condition, thereby maintaining a Mach number of 1.3 at that location. After the throat the flow is let into the combustion chamber of the ramjet by another section of ramps. The area of the throat, obtained from RAMSCRAM, was about .277 $ft^2$.

## Engine Design

The idea of using turbofan-ramjets was dropped early on due to weight considerations. Also turbofans are not needed since the aircraft uses rocket assist, at drop-off, to climb to altitude and executes a power off landing. It was determined by cost analysis that a simple ramjet would reduce overall operating costs; also, the use of ramjets would eliminate mechanical complications caused by turbofan machinery. The final engine was found by running RAMSCRAM and since the engines were restricted in length, due to the size of the airplane, an engine configuration in which four engines were mounted side by side was used.

The final specifications of the ramjets are given in table E1 below.

---

Mach = 5 at 80,000 ft. (cruise)

Thrust = 12,900 lbs. per engine

TSFC = 1.777 lbm/hr/lbf

Fuel = Liquid Methane

Mass Flow Rate of Air = 3.10 slugs/sec

Mass Flow Rate of Fuel = .199 slugs/sec

Combustion Chamber  Pressure = 13 atm / Temperature = 4120 F
_____
Table E1 Ramjet Engine Characteristics

This system of engines produces enough thrust to allow the aircraft to cruise at Mach 5 at 80,000 ft. The ramjets come on at 50,000 ft. at Mach 4.2 and accelerate the aircraft to its cruise point. Again, the design goals of less weight and enough thrust for the aircraft were met. The combustion chamber of the ramjet utilizes one fuel injector, injecting fuel at a speed of Mach .7 with airflow coming in at Mach .3. The combustion chamber has an area of

approximately .67 ft$^2$ and a length of 2.5 ft.

Rocket Design

The rockets that were chosen were liquid Hydrogen/Oxygen rockets mounted internally in the aircraft. Internal rockets were chosen since dropping external boosters would pose logistics problems and the external boosters would adversely affect the aerodynamic characteristics of the aircraft. The final design for the liquid rockets was established using RAMSCRAM. No form of air augmentation was used in the rockets since the increases in thrust, caused by air augmentation, was negated by ramdrag. The characteristics of the rocket engines are listed in the table E2 below.

| |
|---|
| Average Thrust = 42,584 lbs./rocket |
| Chamber Pressure = 66.4 atm |
| Burn Time = 94 sec |
| Fuel Consumed = 39000 lbs |
| Specific Impulse = 406 sec |

Table E2 Rocket Engine Characteristics

Fuel Choice

There were two fuel choices for rocket engine. The first choice was a liquid methane liquid oxygen combination. This fuel choice was experimented and was discarded because of high fuel consumption by the rocket. The best SFC's that were achieved ranged from 12.011-14.2. The liquid oxygen liquid hydrogen combination proved to be more practical for the required mission profile. SFC's ranging from 8.1 to 9.5 were achieved. The reduced fuel consumption by the rocket engine allowed for the storage of the propellant on

board the aircraft; thus, eliminating the need for external rocket engines. The mission requirement was that the aircraft be dropped from a carrier plane. This meant that the aircraft had to be kept small in size and the best way to do that was to reduce the amount of space occupied by the engine fuel. Liquid Hydrogen and liquid Oxygen were considered as possible fuels for the ramjet; however, the volumetric inefficiency of liquid hydrogen would required an aircraft 1.5 times the size of an aircraft powered by liquid Methane. Therefore the best compromise for the ramjet fuel was liquid Methane. It was high enough in energy and volumetrically efficient enough to allow for a 6000 nmi mission at Mach 5 with a payload of 7500 lbs.

## Conclusions and Recommendations

The ramjet / rocket engine combination provides an ideal propulsion platform for the reconnaissance vehicle. The ramjet inlet is initially closed off when the plane is dropped from the carrier aircraft. The rocket engine is ignited and propels the aircraft to Mach 4.2 at 50,000 ft. Here the inlets open and the ramjet is started. The rockets burn out and the ramjet gets the aircraft to Mach 5 at 80,000 ft., where it cruises for the duration of the powered flight. When the aircraft lands, the inlet is again closed (see Figure P1).

This overall design that was finally agreed upon best utilized the needs of the aircraft and the overall goals of the design group. Further recommendations would be to use even less space and try to put the rocket engines into the combustion chamber of the ramjets. This would give them a dual purpose that would save weight as well as space.

ROCKET ON/ENGINE OFF AT CLIMB
M=.8, ALT=40,000 ft

.2 FT2

ROCKET OFF/ENGINE ON AT CLIMB
M=4.2, ALT=50,000 ft

4 FT2

POCKET OFF/ENGINE ON AT CRUISE
M=5.0, ALT=80,000 ft

ROCKET OFF/ENGINE OFF LANDING
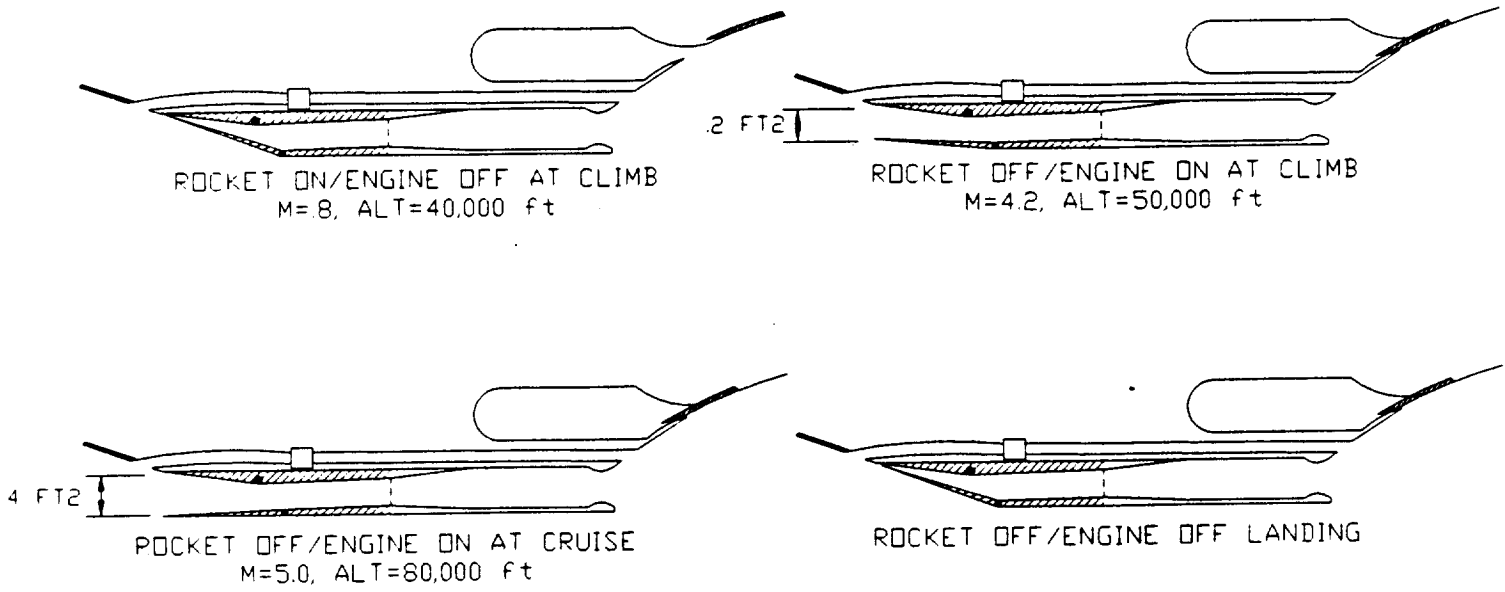
Figure P1 Inlets in various stages of flight

# References

Jensen, Roger. "Weight Estimation of Hypersonic Inlets." Technical Paper No. 655:1968.

Burkhardt, Leo and Franciscus, Leo. "RAMSCRAM - A Flexible Ramjet / Scramjet Engine Simulation Program." Technical Manual, 1990.

Midea, Anthony C., "Mach 6.5 Air Induction System Design for the Beta II Two-Stage-to-Orbit Booster Vehicle.", NASA Contractor Report 187233, 1991.

Weidner, John P.,"Conceptual Study of a Turbojet / Ramjet Inlet.", NASA Technical Memorandum 80141, 1979.

# Weights Analysis

## Summary

The aircraft weight was estimated by adding up the weights of the major aircraft components. These major components were the beams and spars, payload, thermal protection systems, engines, rocket fuel, and fuel required for cruise. The overall aircraft weight found by this technique was 130000 lbs and the empty weight of the aircraft was found to be 28280 lbs.

## Research Methods

There were basically two options in conducting the weights analysis on this aircraft. One was to use either of the two computer codes, HASA and WAATS, to do the analysis. The second way was to take material composition of the individual components and estimate the weight of the component and then add up the weights of the individual components to get the total aircraft weight.. There was one other way which, was not utilized, and that was to compare weights of similar vehicles consistent with the current aircraft configuration.

The reason a comparison method was not used was that the uniqueness in materials and aircraft configuration posed problems for obtaining realistic weight values. The computer codes HASA and WAATS were originally configured for a different computer system other than a VAX. So there were a lot of problems in getting the codes to conform to VAX standards. Since a considerable amount of materials research had been done, the computer code analysis was abandoned in favor of the component wise break down of weights based on material selection.

## Research Results

The components analyzed were the skin, beams and spars, TPS (Thermal Protection System), and Payload. The payload was specified to be 7500 lbs by design requirements. The weight of the skin was found to be 2600 lbs. This weight estimate was obtained from using the material properties for Titanium-6Al-4V/SCS-6, with a skin thickness of 1/16th of an inch. The TPS weight of 6500 lbs was provided for by Thermal Protections Group.

The weight of the aircraft structure was based on the estimate that one percent of the aircraft volume should be used to contribute to the weight of the aircraft structure. One percent may seem small however, the materials used in the structure are 60 % Arotone, a crystalline thermoplastic, and 40% Titanium..

Arotone is almost four times stronger and exhibits 25% weight savings over Titanium. The strength of the material allows for the smaller quantity used in estimating the weight. A look at figure W1 shows that the majority of the aircraft weight is composed of fuel for the rockets and fuel for the aircraft. Figure W2 shows the empty weight of the aircraft while Figure W3 shows the volumetric distribution

Even though the empty weight of the aircraft is 28280 lbs the main landing gear is designed to take a load of 35700 lbs. This is because the glide slope of the aircraft at landing is in the vicinity of eighteen degrees. This steep glide slope will cause the aircraft to make contact with the ground at relatively high speed. Thus, a strong landing gear is needed to absorb the impact with the ground during landing.


Conclusions

The 130000 lbs takeoff weight of the aircraft is reasonable enough to allow for the launching of the aircraft from atop a C-5 or a Boeing 747. The reductions in weight were accomplished by making the proper material and fuels choices (see Figure W6). Even though the lifting body configuration does not exhibit excellent aerodynamic characteristics at landing, the landing gears can feasibly be made strong enough to endure the impact between the ground and the aircraft at landing.

# Weight Distribution
## Dropoff Weight - 130000 lbs
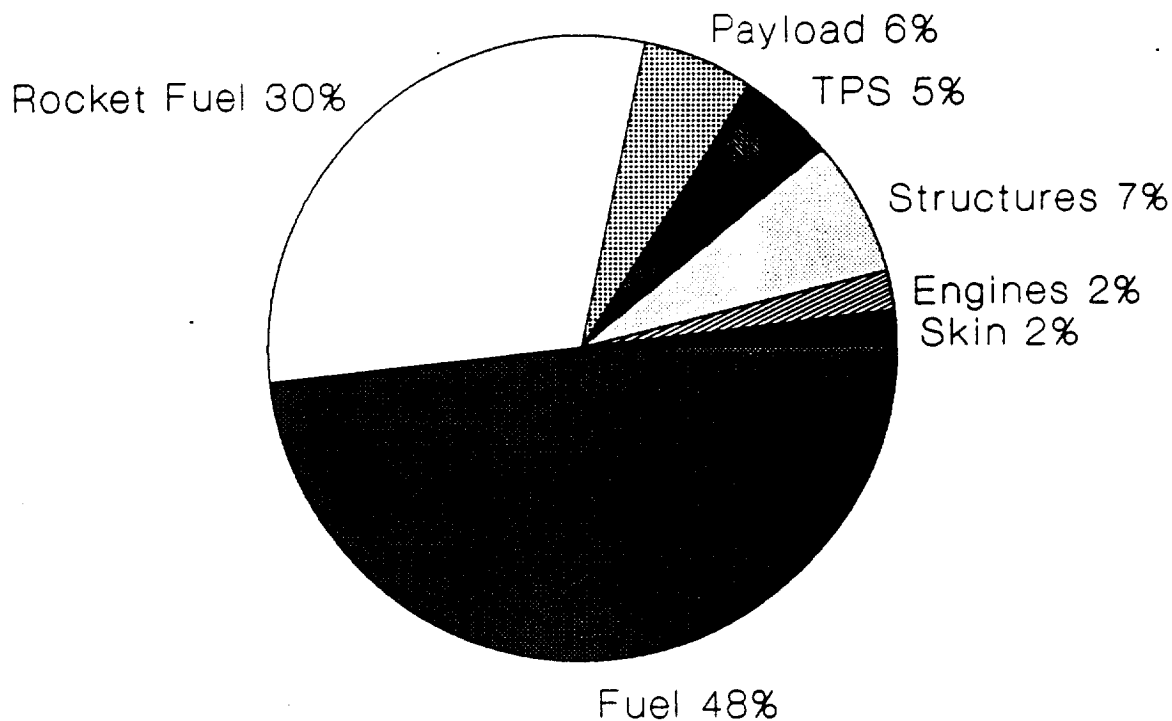


Figure W1 Weight Distribution at Takeoff

# Weight Distribution
## Empty Weight - 28280 lbs.

Structures 32%

Engines 9%

Skin 9%

TPS 23%

Payload 27%
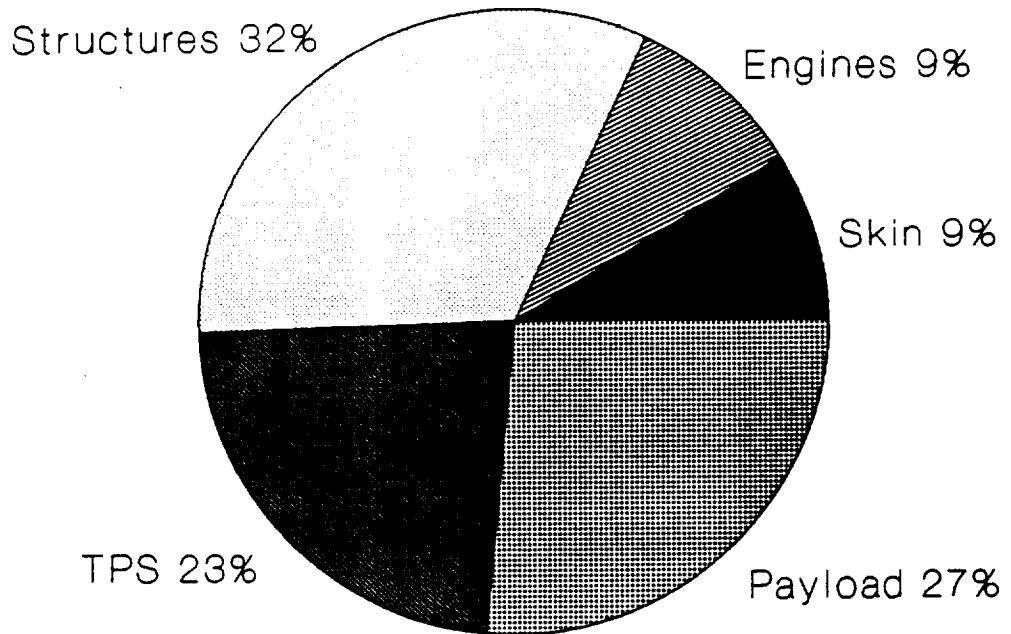
Figure W2 Empty Weight Distribution

# Volume Distribution
## Total Volume 3900 cuft

Methane Fuel 64%

Unusable 10%

TPS 1%

Payload 7%

Avionics 1%

Oxygen Fuel 15%

Hydrogen Fuel 3%
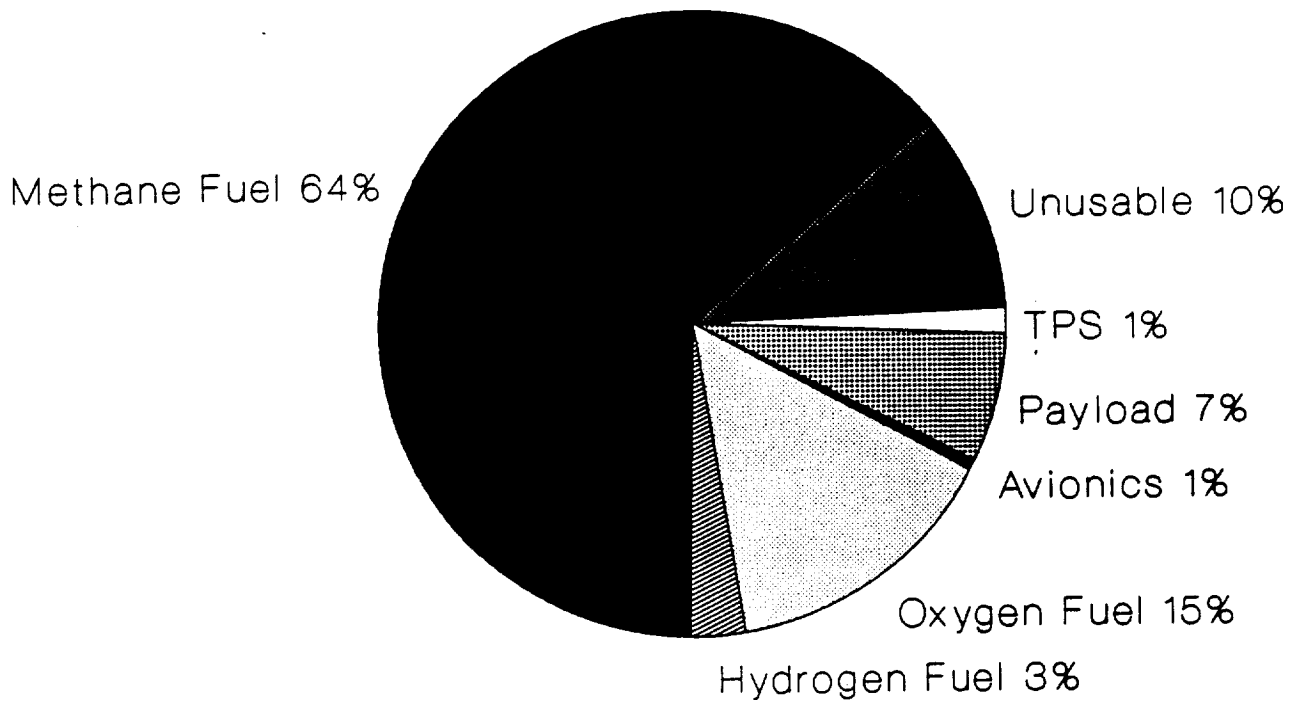
**Figure W3 Volumetric Distribution**

# Main Landing Gear

- Load - 35,700 lbs

- Tire Pressure - 170 psi

- Ply Rating 24

- Tire Size - 46"x16"
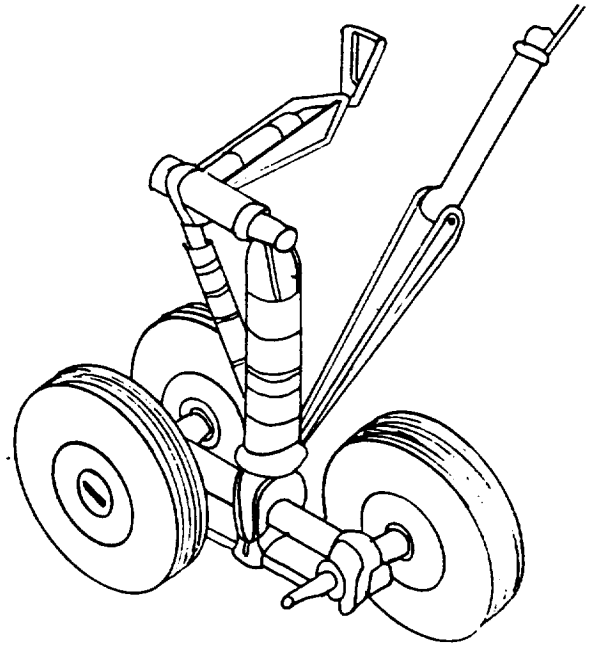
Figure W4 Main Landing Gear

# Nose Landing Gear

- Load - 22,500 lbs

- Tire Pressure - 100 psi

- Ply Rating 14

- Tire Size - 39"x13"
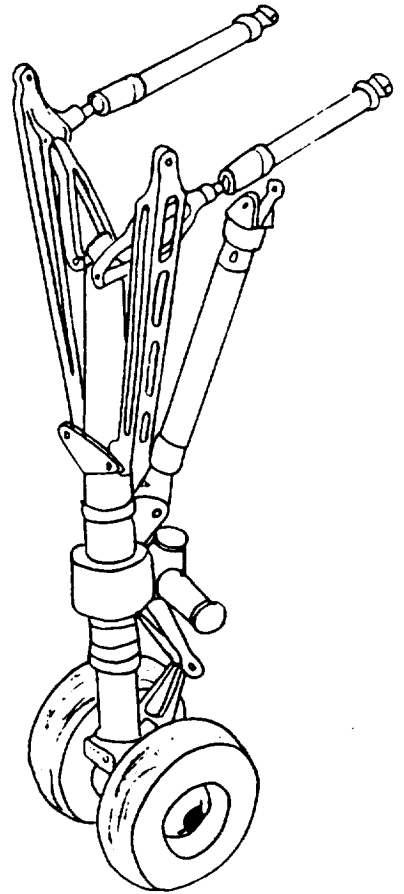
Figure W5 Nose Landing Gear

# Structural Composition

- 60% Structure Arotone
  40% structure Titanium

- Density Arotone 207.36 lbs/ft3
  Density Titanium 276.48 lbs/ft3

- Skin Ti-6Al-4v/scs-6
  Skin Density 228.1 lbs/ft3
  Skin Thickness 1/16"

- Combustion Chamber W-4Re-.35Hf-.35C
  Combustion Chamber Density 1201.77 lbs/ft3
  Combustion Chamber Thickness .072"

Figure W6 structural composition

# References

Jensen, Roger. "Weight Estimation of Hypersonic Inlets." Technical Paper No. 655:1968.

Nicolai, Leland M. "Fundamentals of Aircraft Design". METS, Inc (1975): 24/1-24/35.

# Trajectory Analysis

## Summary

Due to limitation in the ramjet engine, rocket propulsion must be used for most of the climb phase. The transition from rocket to ramjet in the climb trajectory occurs at mach 4.2 at 50,000 feet. The minimum fuel climb trajectory expends 39,000 lbs of rocket fuel, and 3,000 lbs of methane. The steady-level cruise expends 60,000 lbs of methane, and take approximately 2.1 hours to complete.

# Introduction

This section deals with the trajectory analysis of the reconnaissance vehicle from launch from the carrier aircraft, through cruise.

Four question that will be answered by this analysis is as follows:

1) What is the minimum fuel climb trajectory for the ramjet?
2) What is the minimum fuel climb trajectory for the rockets?
3) Determine the transition point from rocket to ramjet?
4) Calculation of the amount of fuel, and thrust required for cruise?

The answers to the above questions determine whether or not the aircraft can carry the amount of fuel required to complete the mission.

The vehicle will be dropped from the carrier aircraft at mach 0.8 at forty thousand feet. The beginning of the climb phase of the trajectory was taken at thirty five thousand feet at mach 0.8, to allow for a safe distance between the carrier plane and the hypersonic vehicle before the ignition of the rockets.

The climb portion of the trajectory utilizes two forms of propulsion, liquid rockets, and ramjets. The rockets accelerate the plane to a velocity, and altitude at which the ramjets can be turned on. It was expected that most of the fuel used in the climb phase would be spent by the rocket, since the rocket must carry their own oxidizer. The ramjet potion of the climb was expected take the most time, since they are less powerful than the liquid rockets.

The aircraft has a steady-level cruise for six thousand nautical miles, at eighty thousand feet, at mach 5.

A sample mission, and a mission altitude vs distance graph are shown in Figures TR1, and TR2, respectively.

Following are the research methods for climb, research methods for cruise, results, conclusions and recommendations of the trajectory analysis.

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## Research methods for climb

The climb portion was analyzed using an energy method which relates the change in specific energy with respect to fuel weight (Nicolai 4-13 to 4-16). A graph of constant fs lines, with units of ft/lbs, were plotted over a range of altitudes and mach numbers in the flight regime. The was done for the portions of the climb utilizing rockets, and the ramjets.

A computer code was written to sweep through the flight regime and calculate the constant fs curves. The flight path perpendicular to the fs curves is the minimum fuel trajectory ( see Figure TR8).

The fs curves for the plane under rocket propulsion were generated by using to the take off weight of 130,000 lbs. An iterative process was used to calculate the angle of attack of the aircraft using the approximation of lift+thrust*sin(alpha)=weight. The decrease in lift vector parallel to the weight vector was neglected. The corresponding drag, sfc, and fs were then calculated.

The fs curves for the ramjet was determined in the same manner as the rockets, except the they were generated by using to a 91,000 lbs

Figure TR1 Sample Mission for Peregrin hypersonic reconnaissance vehicle
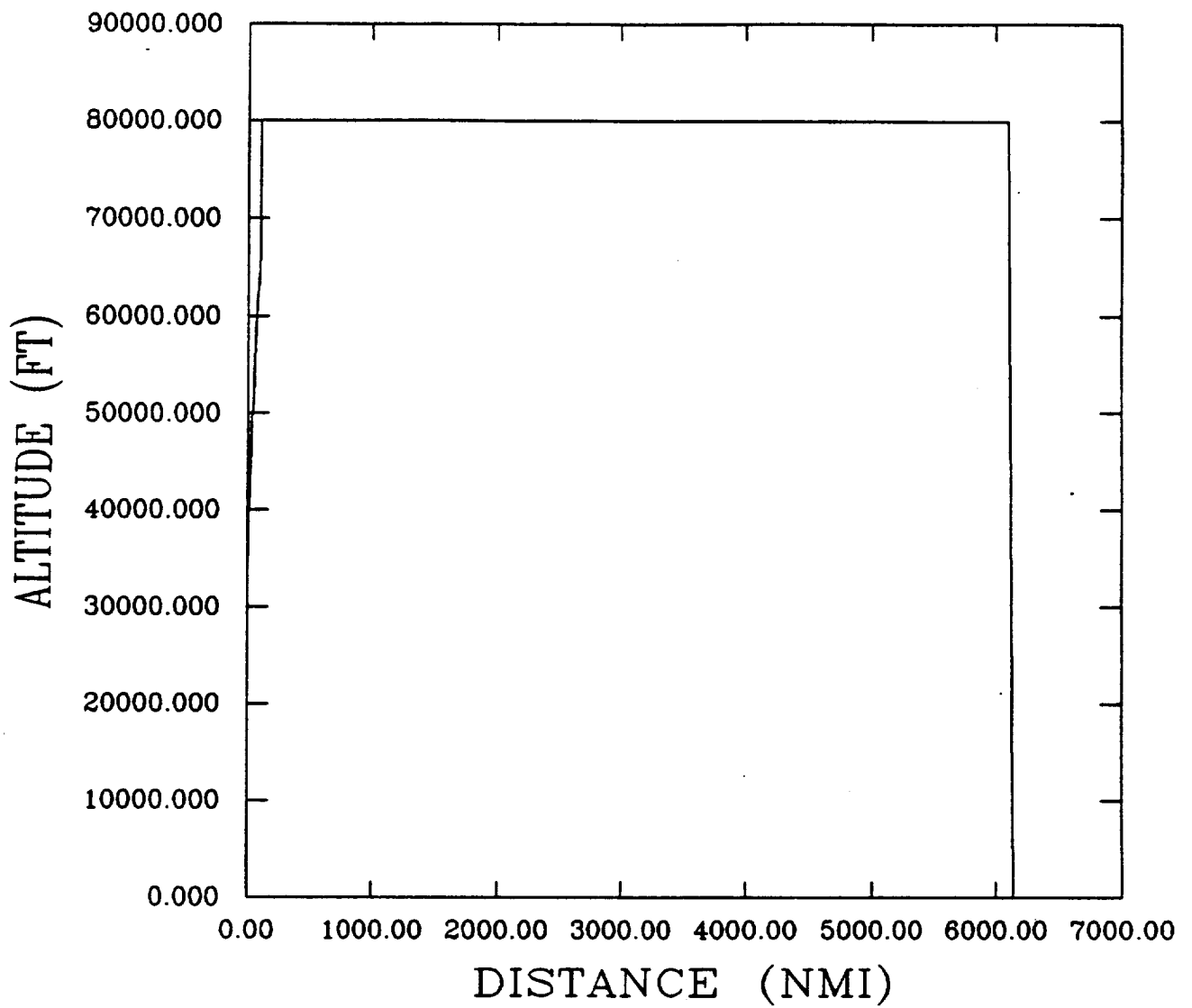
# MISSION DISTANCE VS ALTITUDE



Figure TR2 Mission altitude vs distance

aircraft weight and the sfc and thrust had to be interpolated for different angles of attack, mach numbers, and altitudes.

A maximum angle of attack of 16 degrees was chosen in the flight regime, since the lift and drag characteristics above that angle were not known.

Once the graphs were generated, the flight path was determined. The fuel consumption, weight change, and time to climb were then calculated based on the flight path.

## Research methods for cruise

To determine how to throttle the engine for cruise, the approximation of lift equals weight was used to determine the lift coefficient. The corresponding drag coefficient was then calculated. The amount of thrust required to balance the drag was then determined. A new weight was calculated, and the process repeated until 6000 nmi was reached. When calculating the amount of fuel needed for cruise, a 30 MPH was taken into consideration. This was done to provide for a safety factor in the calculations.

## Results of analysis

The ramjet engine are useful in only a minor portion of the climb trajectory. Most of the climb must use the liquid rocket engines. Because of this, most of the time in the climb phase is taken up by the rockets, and not by the ramjets, as expected. The point at which the ramjets take over for the rockets is at mach 4.2 at 50,000 feet. The cruise portion of the mission takes approximately 2.1 hours. Graphs of lift/drag, and rate of climb for climb for

climb are shown in Figures TR3 and TR4, respectively. The weight time history for the entire mission is shown in figure TR5.

### What is the minimum fuel climb trajectory using ramjets?

There is only a small portion in the flight regime in which the ramjet engines get enough air so that they can ignite. As seen      in Figure TR6, the ramjet has good operating characteristics at

mach 4.1 and above. Just below mach 4.1, the ramjets are inoperable, or do not supply enough thrust to over come drag exerted on the plane. The climb time for the trajectory in Figure TR6 is 74 seconds, using approximately 3000 lbs of liquid methane. Thrust vs mach number for ramjet is depicted in Figure TR7.

### What is the minimum fuel climb trajectory using rockets?

Because of the limited operational range of the ramjets, the rocket must be used during most of the climb. The climb trajectory for      the rocket can be seen in Figure TR8.

The rocket must get the aircraft to mach 4.2 at 50,000 feet.      This portion of the trajectory expends 39,000 lbs of liquid   hydrogen, and oxygen, · and takes approximately 94 seconds.

There is a 3.5 second burn time reserve for the rockets.

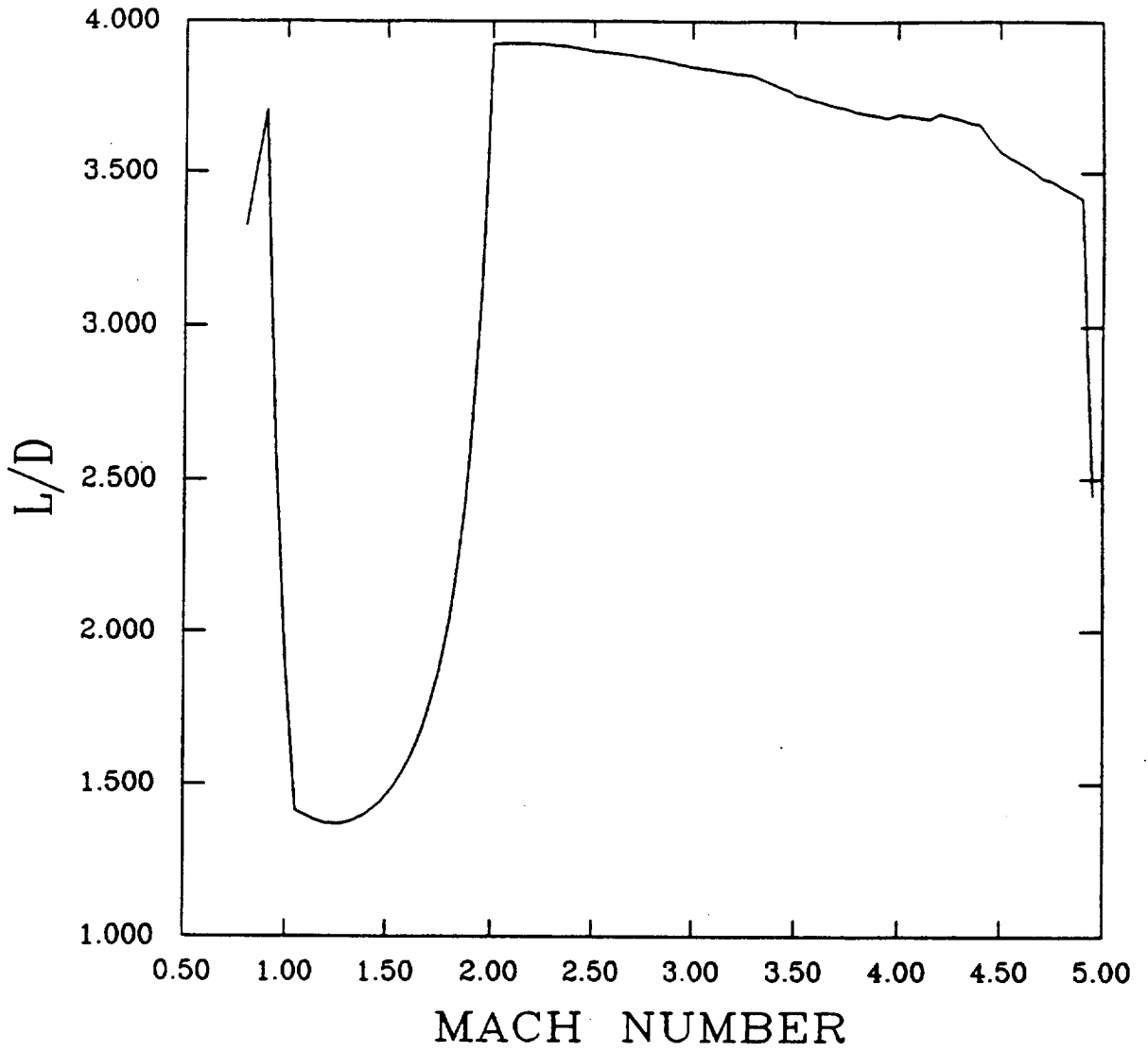# L/D VS MACH NUMBER FOR CLIMB



Figure TR3 L/D vs mach number for climb trajectory

# RATE OF CLIMB VS MACH FOR CLIMB TO CRUISE



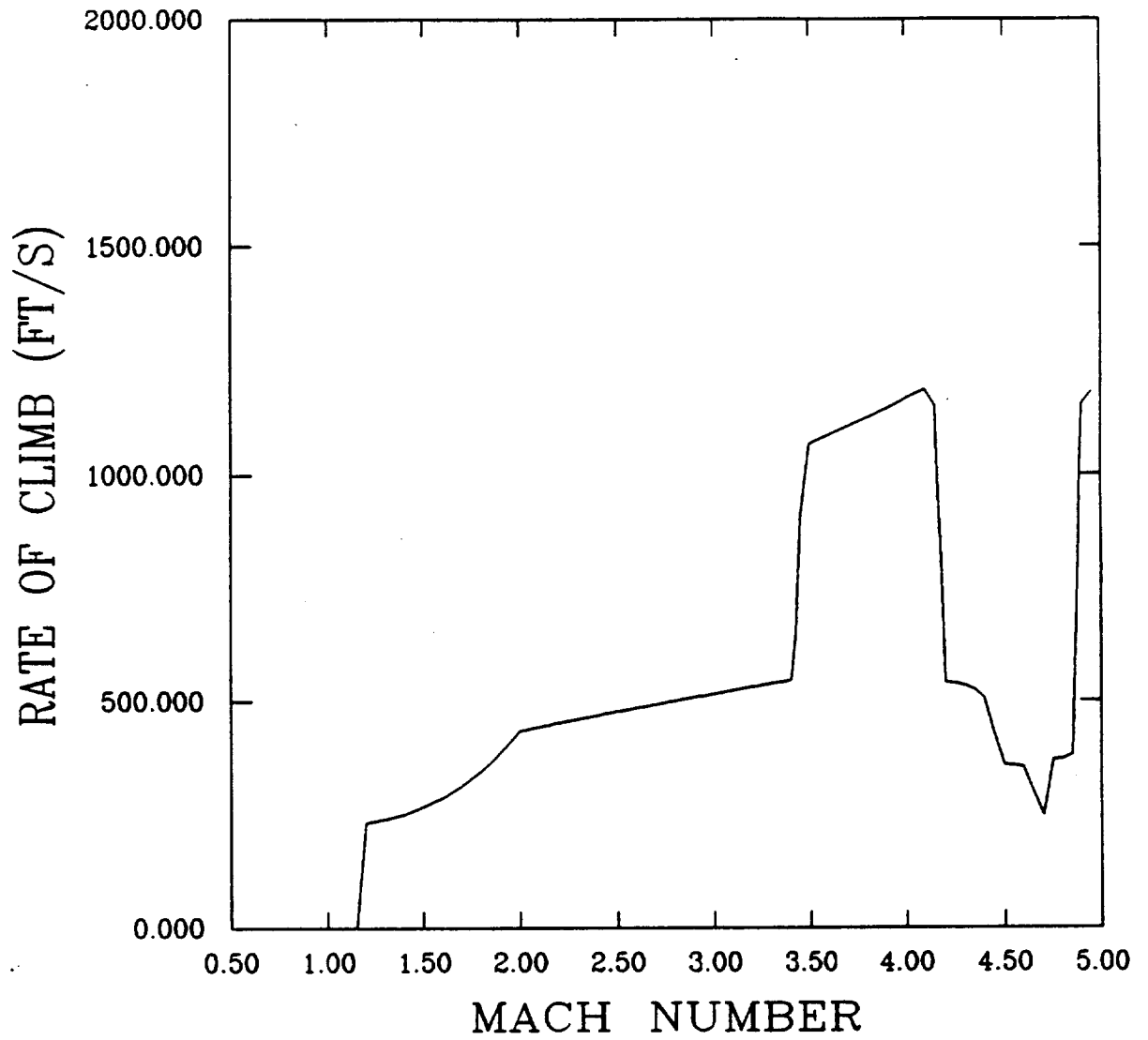Figure TR4 Rate of climb vs mach number for climb
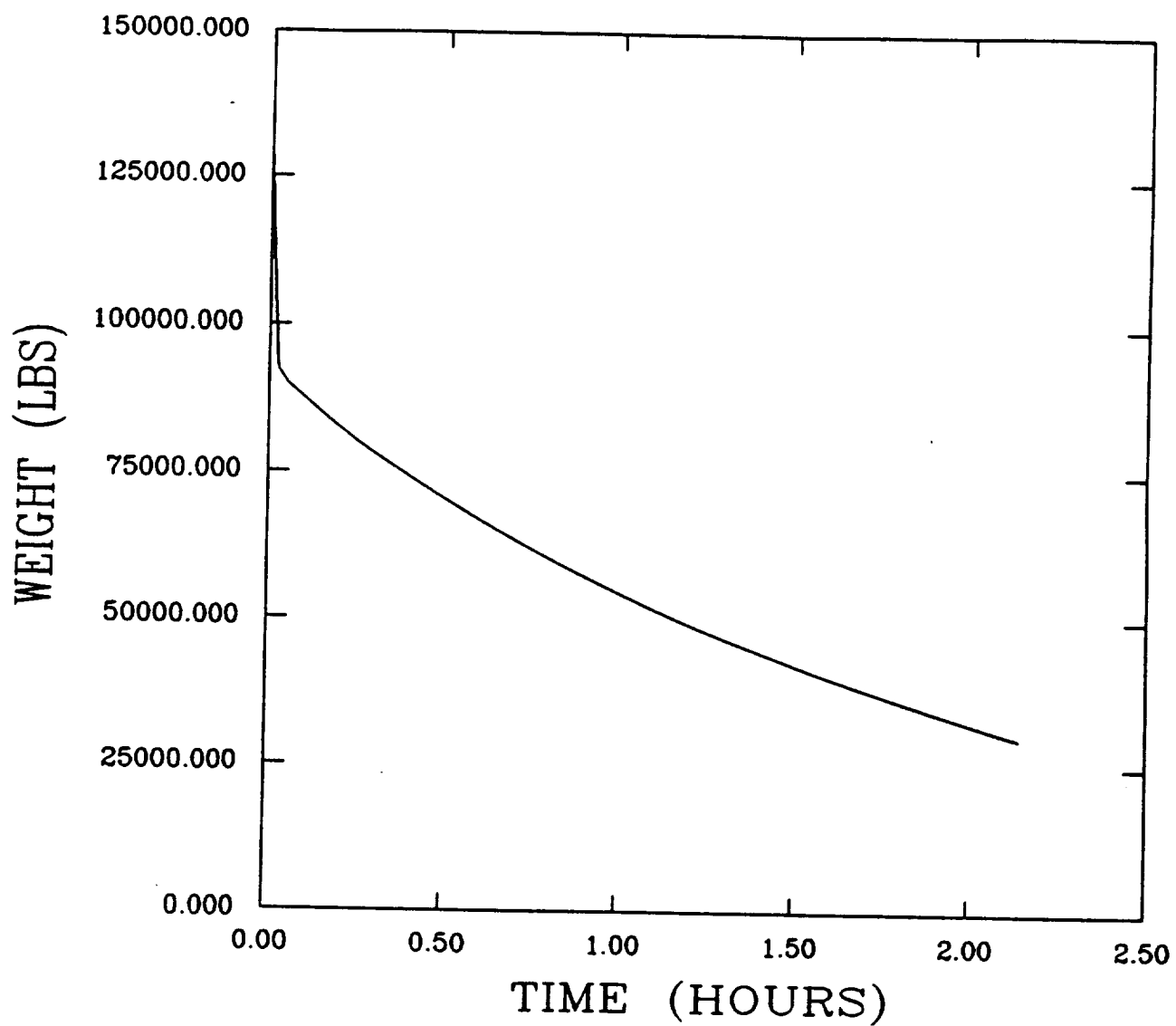
# MISSION WEIGHT VS TIME



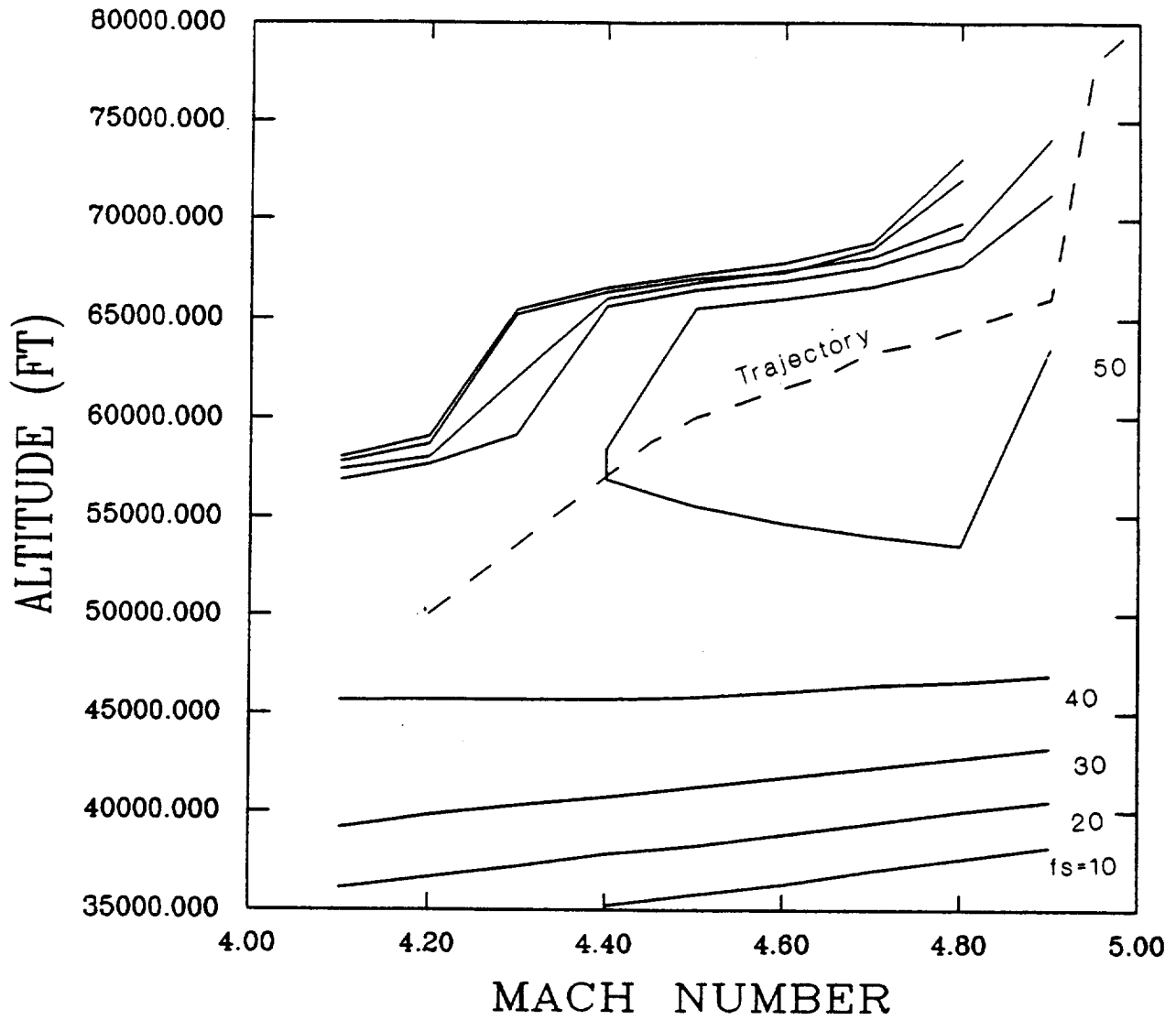Figure TR5 Mission weight time history

# RAMJET CLIMB TRAJECTORY



Figure TR6 Minimum fuel trajectory for ramjet portion of climb

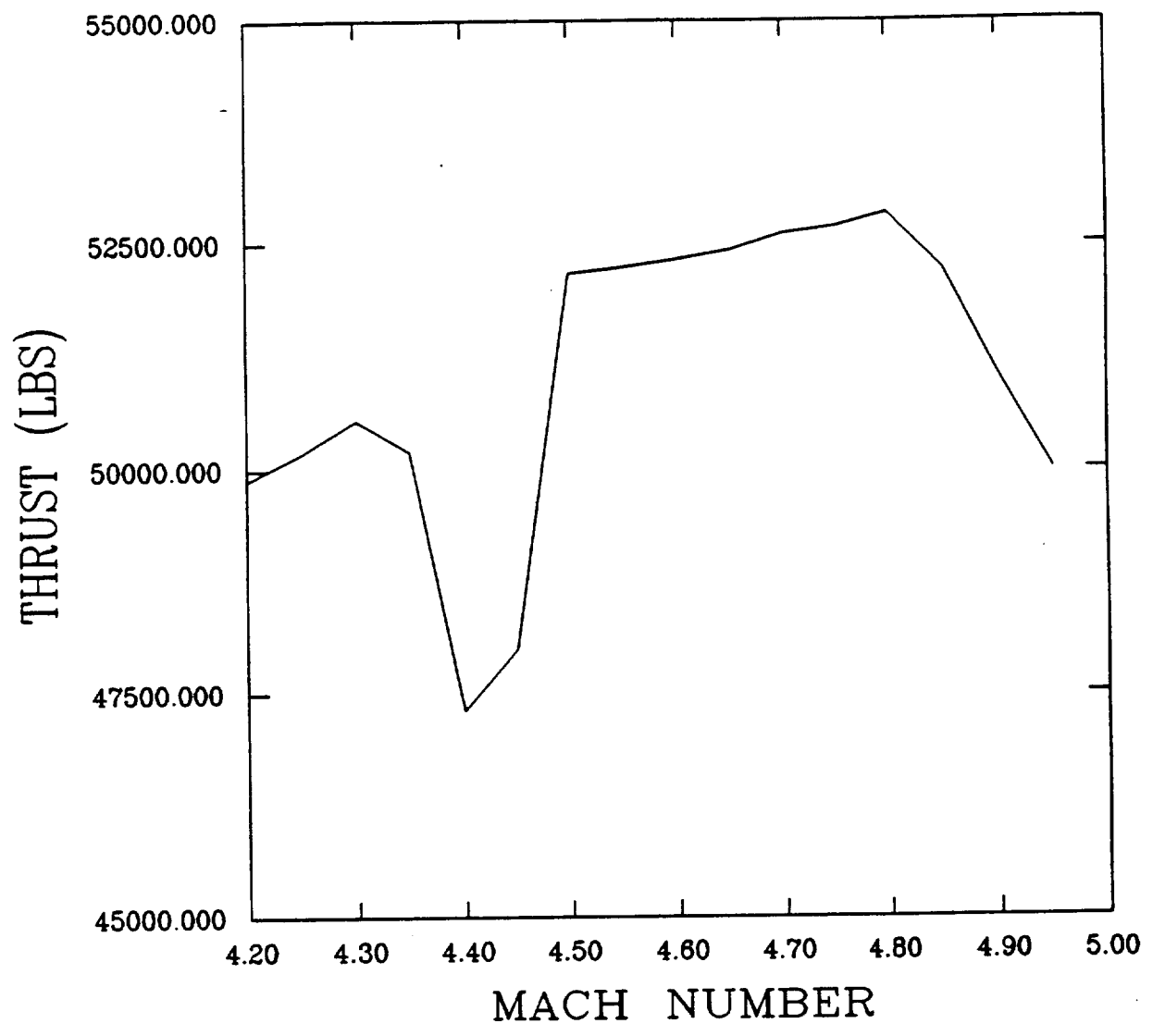# THRUST VS MACH NUMBER FOR CLIMB



Figure TR7 Thrust vs mach number for ramjet portion of climb

# ROCKET CLIMB TRAJECTORY
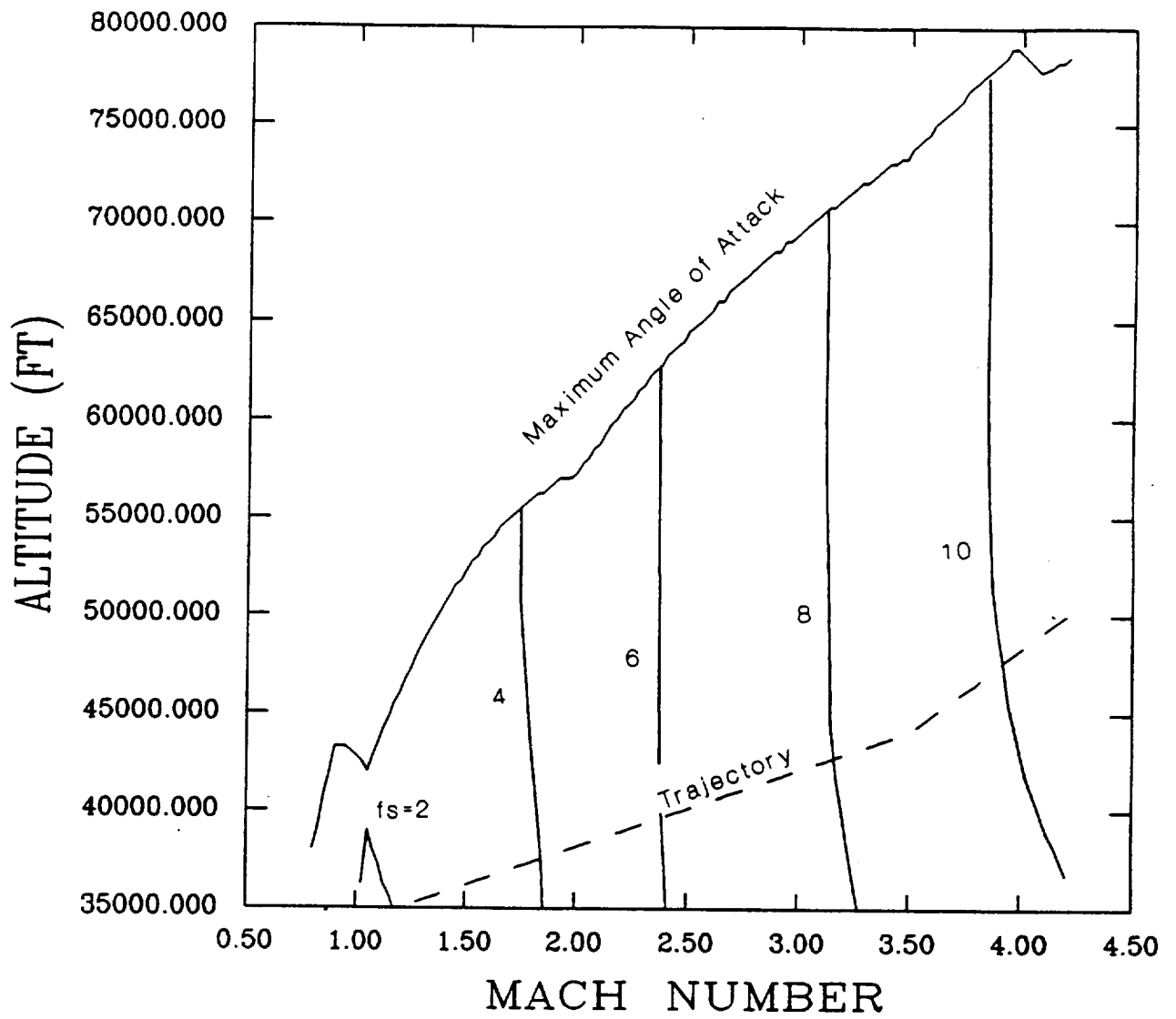


Figure TR8 minimum fuel trajectory for rocket portion of climb

## Where is the transition point?

The transition point occurs at mach 4.2 at 50,000 feet. Although the ramjets can be used at mach 4.1, allowances, had to be made for nonstandard days, and to allow enough time for the ramjets to ignite.

## What is the cruise characteristics?

The amount of fuel needed for cruise is approximately 60,000 lbs. The maximum amount of thrust required is approximately 27,500 lbs. The thrust time history for cruise is shown in Figure TR9.

The maximum angle of attack for cruise is seven degrees, and the minimum is about zero. Besides taking into account a 30 MPH headwind, there is 1500 lbs of methane in reserve.

## Conclusions and recommendations

A rocket assist should be employed to climb to mach 4.2 at 50,000 feet, at which point the ramjet engine will take over. The trajectory, as shown in figures TR6, and TR8 will minimize the fuel usage in the climb stage.

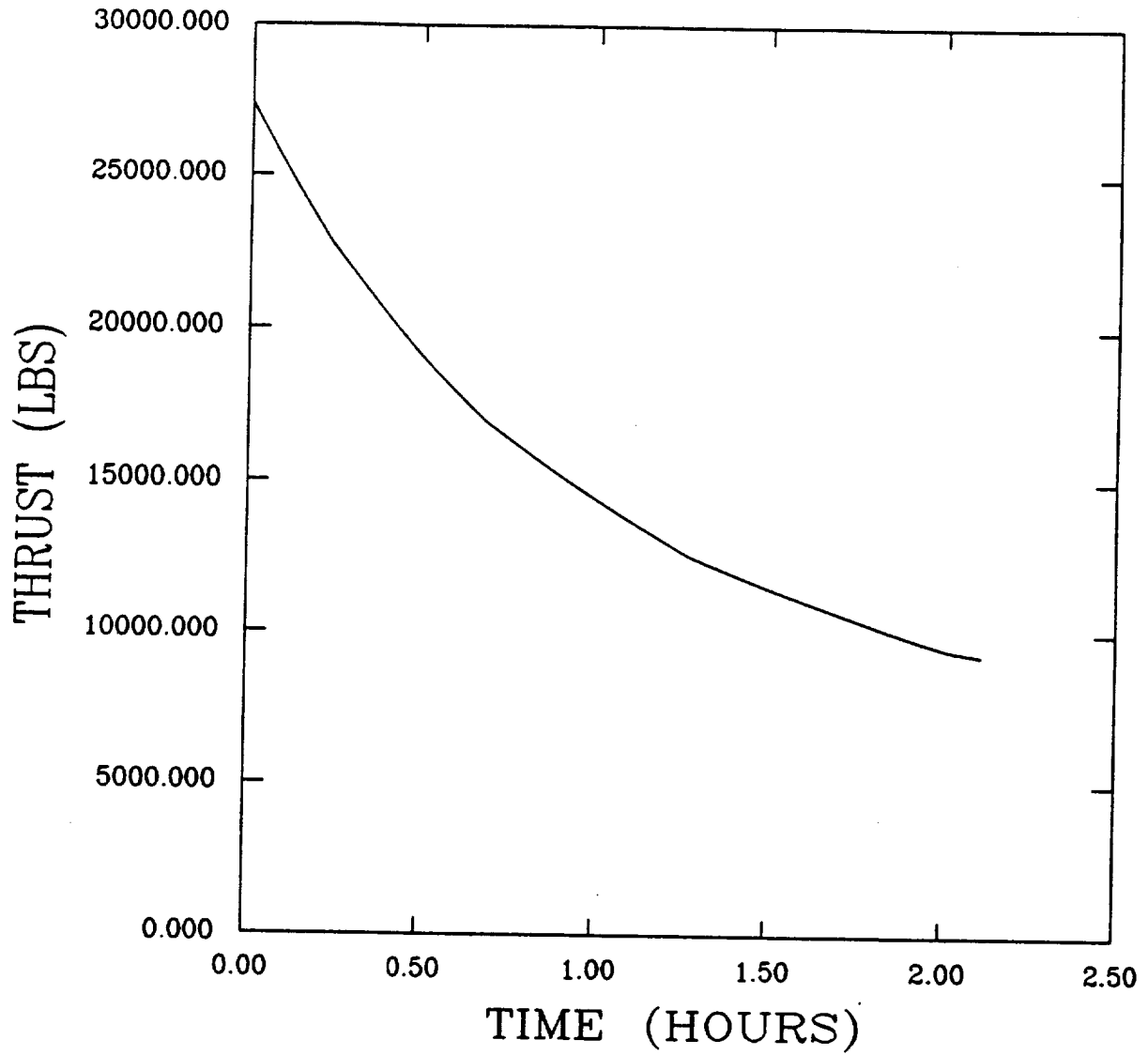# THRUST VS TIME FOR CRUISE



Figure TR9 Thrust time history for cruise

# References

Nicolai, Leland M. "Fundamentals of Aircraft Design". METS, Inc (1975): 4/13-4/20.

Johnson, James. 1992. Sram. Computer Software.

Johnson, James. 1992. Rocket. Computer Software.

Johnson, James. 1992. Throttle. Computer Software.

# THERMAL PROTECTION SYSTEMS

The primary goal of the thermal protection system is to cool the skin of the aircraft to an acceptable level to make hypersonic flight possible. Based on information from materials analysis and aerodynamics the maximum skin temperature was chosen to be 1500 degrees R. This temperature lies well with the safe temperature region of the aircraft's structural materials, even under the stress of extreme wing loadings.

A secondary goal of the thermal protection system is to achieve the primary goal at a minimum weight cost. This parameter is extremely vital to the overall mission. Since this aircraft is designed to be dropped from a carrier aircraft the overall weight must be the bare minimum, or no suitable carrier will be available.

The choice of which type of thermal protection system this aircraft is to employ was arrived at after investigating several types of systems. These systems were broken down into two categories: active and passive. Passive systems are those systems which require no movement of a cooling fluid, while active systems employ a cooling fluid to carry heat away from the aircraft's skin.

The two passive systems that were investigated were ablation and radiation. Ablation is the use of heat shields to insulate the aircraft's skin from the viscous heating effects encountered during hypersonic flight. These type of systems are currently used on reentry vehicles such as the space shuttle. This type of system was quickly discarded for several reasons. First, the heat shielding tiles necessary for sustained hypersonic flight would be extremely thick, hence, heavy. This violates the second goal of the system's design. Also, the shields would change the aerodynamic properties of the aircraft. They would increase drag, possibly decrease lift, and even increase the viscous heating effects by increasing leading edge thicknesses. Finally, the shields

would decay in flight. This would necessitate their replacement between flights, and it would decrease aircraft availability, as well as increasing operatinal costs.

Radiation met with much more favorable reviews. Radiation relies on the emissivity of the aircraft's surface to radiate heat energy from the hot skin to the cooler atmosphere. In its simplest form, radiation cooling is reduced to painting the skin black to increase emissivity. This system adds no weight and doesn't change the aerodynamics of the vehicle. Radiation was chosen to cool this particular aircraft, however, it was determined that this system alone could not dissipate enough heat to be feasible. Therefore some active cooling schemes were investigated.

All the active cooling systems analyzed were basically a form of convection. Several were discarded almost immediately. Both transpiration and spraying were discarded because of their weight inefficiency. Transpiration involves pumping a cool fluid through the surface of the aircraft to directly cool the boundary layer. Spraying involves the constant misting of the inner skin surface with a cool fluid to utilize evaporation to cool the aircraft. Both systems simply boil away the cooling fluid which wastes weight during the critical early stages of the mission.

Ram air convection was briefly looked into. This system was again quickly discarded due to its complexity and the excessive drag it would generate.

This led to the investigation of direct and indirect liquid cooling. Both systems utilize the pumping of a super cooled liquid beneath the aircraft's skin to convect away heat. Direct cooling relies on the cryogenic fuel of the aircraft itself to convect heat away from the skin. The "heated" fuel is then simply combusted in the engines. Indirect cooling relies on a cooling fluid which

circulates under the skin. This fluid carries the heat to a heat exchanger, where the heat is again transferred to the cryogenic fuel before it is burned. Both systems offer the added bonus of increased engine output because the "hot" fuel burns more efficiently. The choice between these two systems again reduced to weight and simplicity. The indirect system requires considerably more weight to operate. It doubles the pumping systems of the aircraft, introduces one or more heat exchangers, and the cooling fluid must be counted in the vehicles empty weight. The direct system was clearly the choice for simplicity and weight savings. However, it has disadvantages as well. It relies on the heat carrying capacity of the fuel which may be a small fraction of that of the indirect cooling medium. It also entails the pumping of a highly combustible liquid into contact with superheated structures. This can be potentially dangerous. Still, the direct cooling system showed the greatest potential and was utilized for this initial design.

Two methods are in use to analyze and design a thermal protection system for this vehicle. The first is the solution of the 2-D boundary layer equations. The second method is to use empirical modeling to approximate the surface heating. The boundary layer solution utilizes a fortran program written by Dr. Richard Bodonyi. The only input it requires is the edge velocity profile along the surface of the vehicle. This is obtained from the aerodynamicist in the form of a pressure coefficient. From this the edge velocity can be calculated. The program output includes the Stanton number at the surface which can be used to calculate the heat flux. This method does have some limitations. The program only solves for a laminar boundary layer. Using a transition point approximation found in Anderson's Hypersonic Flight, it was determined that only the leading 5% of the aircraft has a laminar boundary layer. Therefore, even though the program performs an exact solution, it is not

exact for this aircraft. By cooling the surface below the design point boundary layer transition can be delayed, but the effect is not pronounced enough to make the program's analysis accurate over the entire surface. Another characteristic of this program is that it fails at the separation point. This is both useful and troubling. It is a valuable tool to know where separation occurs on the aircraft. The program yielded a separation point 10% behind the leading edge.This phenomena was correctable by lowering the skin temperature.

The second method for analysis of the aircraft's surface is empirical modeling. This involves modeling the aircraft as a series of spheres, cylinders, and delta wings which have experimental curves for their heating characteristics. These curves are fit with an empirical formula. These formulas are good for these shapes within a specified Mach range. For this particular aircraft the nose was modeled as a one inch radius sphere traveling at 80,000 feet. The Mach number was varied from one to seven and the associated adiabatic wall temperatures and heat fluxes calculated. These results are found in figures TP1 and TP2.

The leading edge was also modeled empirically. It was treated as a one inch radius cylinder yawed at an angle of 76 degrees (the sweep angle of the · leading edge) traveling at 80,000 feet. Again the Mach number was varied as well as the angle of attack and the adiabatic wall temperature and heat flux calculated. These results are shown in figures TP3 and TP4.

The future work mainly includes improving the available analysis techniques. Liason is being made with Dr. Bodonyi to discuss the possibility of modifying his program for a turbulent boundary layer. Also the rest of the aircraft must be modeled empirically, possibly as a delta wing. The possibility still exists of using indirect liquid cooling if the cooling efficiency warrants the

extra weight. Finally, the coatings available to increase emissivity must be evaluated to maximize radiation cooling.
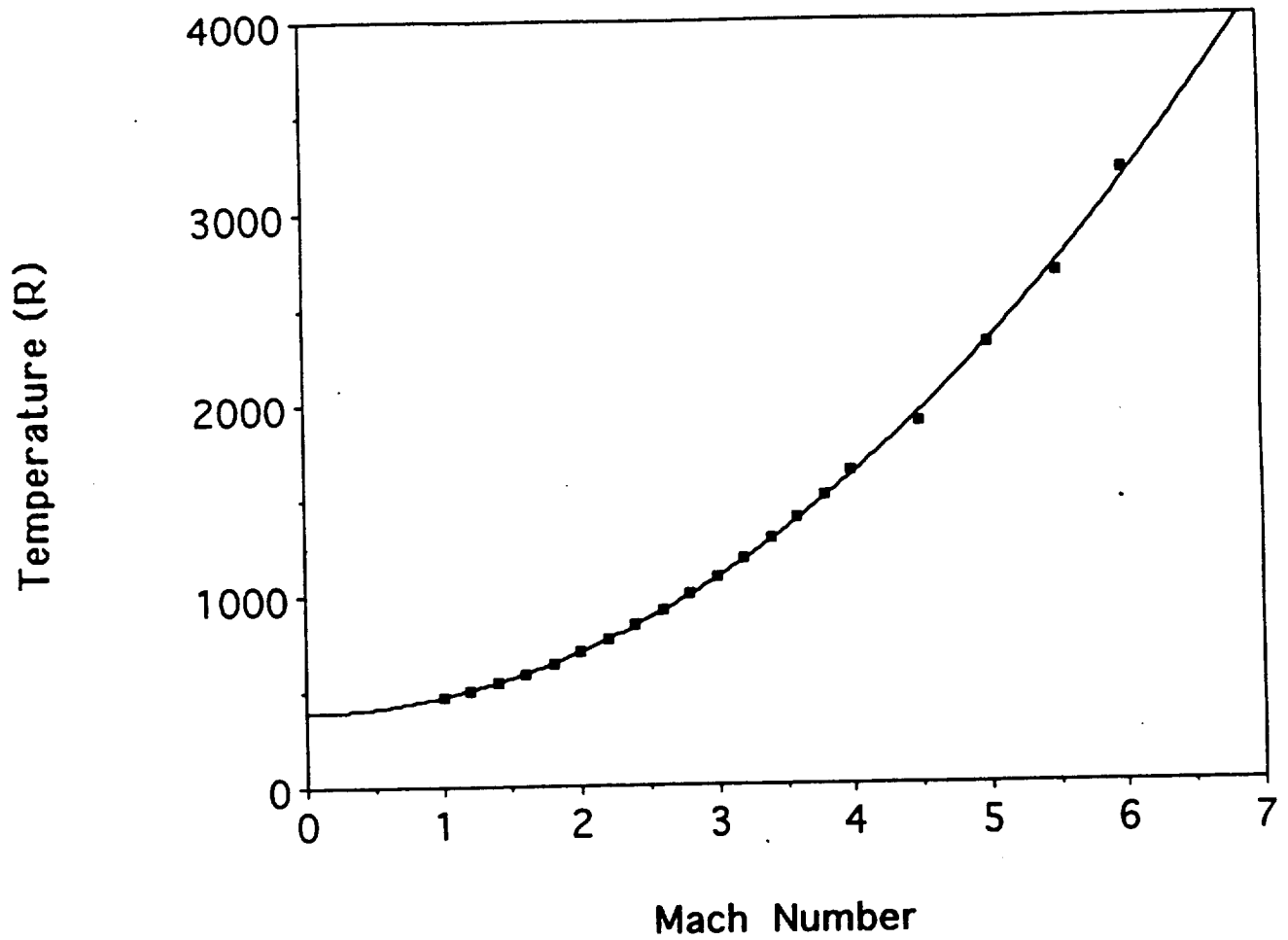
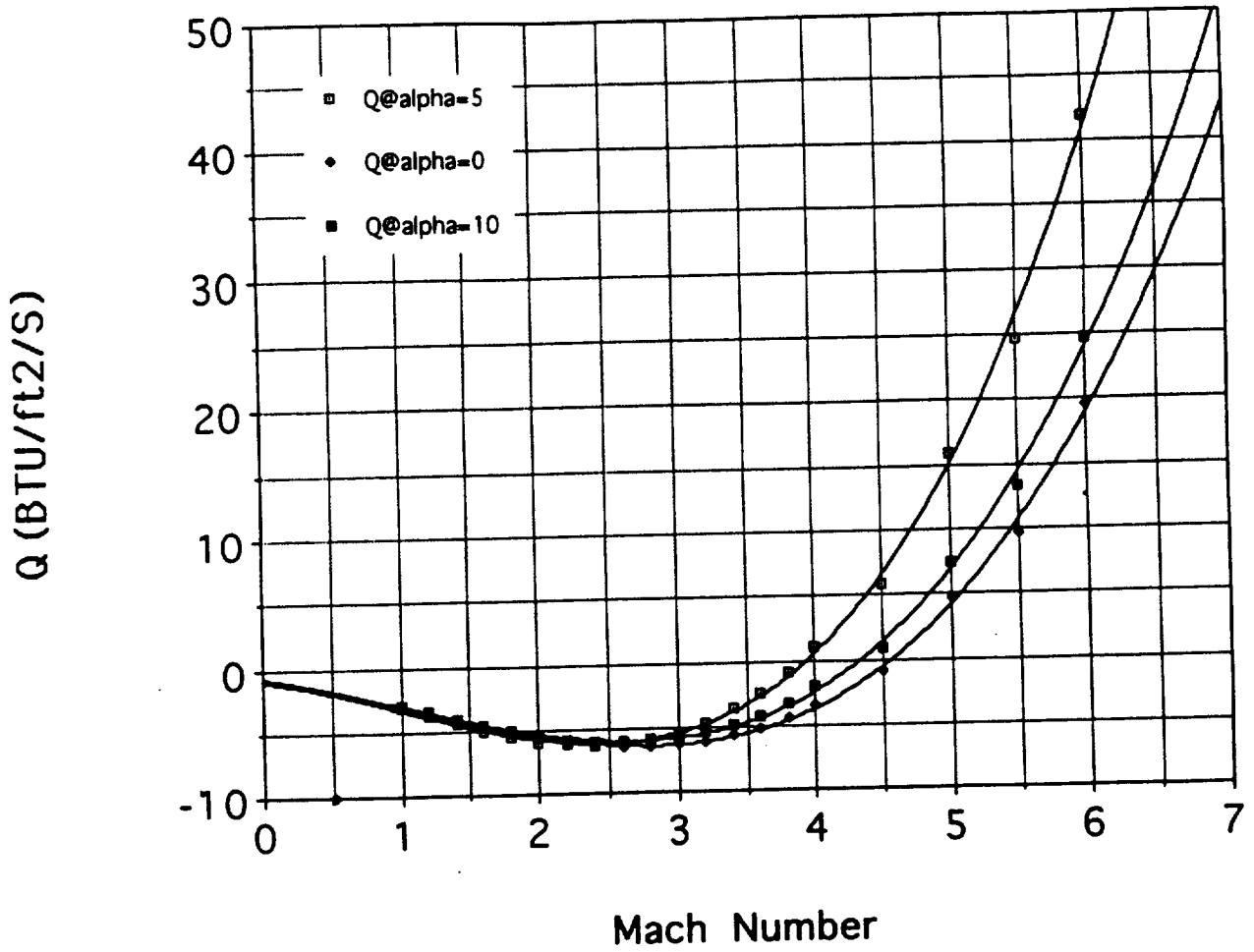Figure TP1  Adiabatic Wall Temp

58
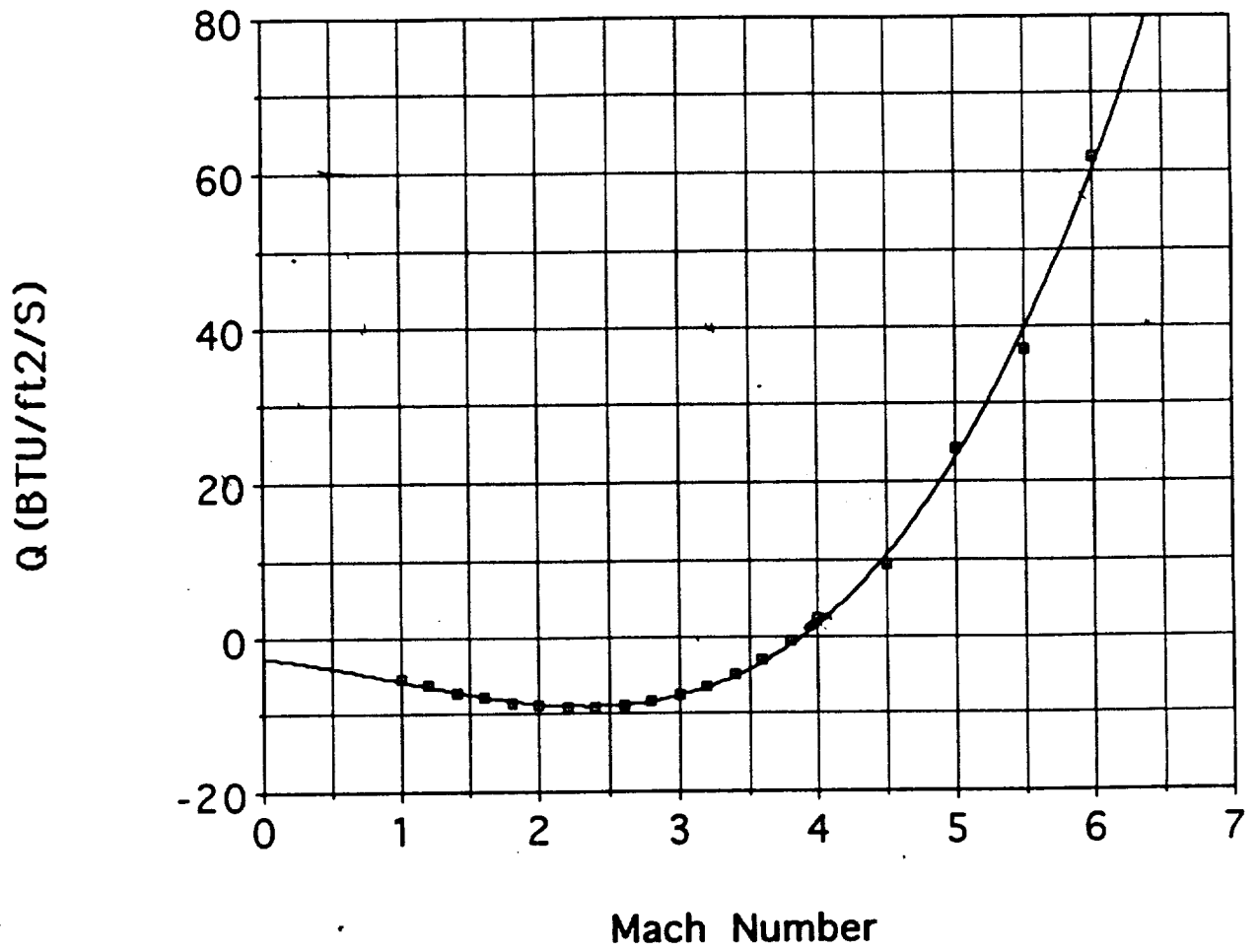
Figure TP2  Q vs Mach

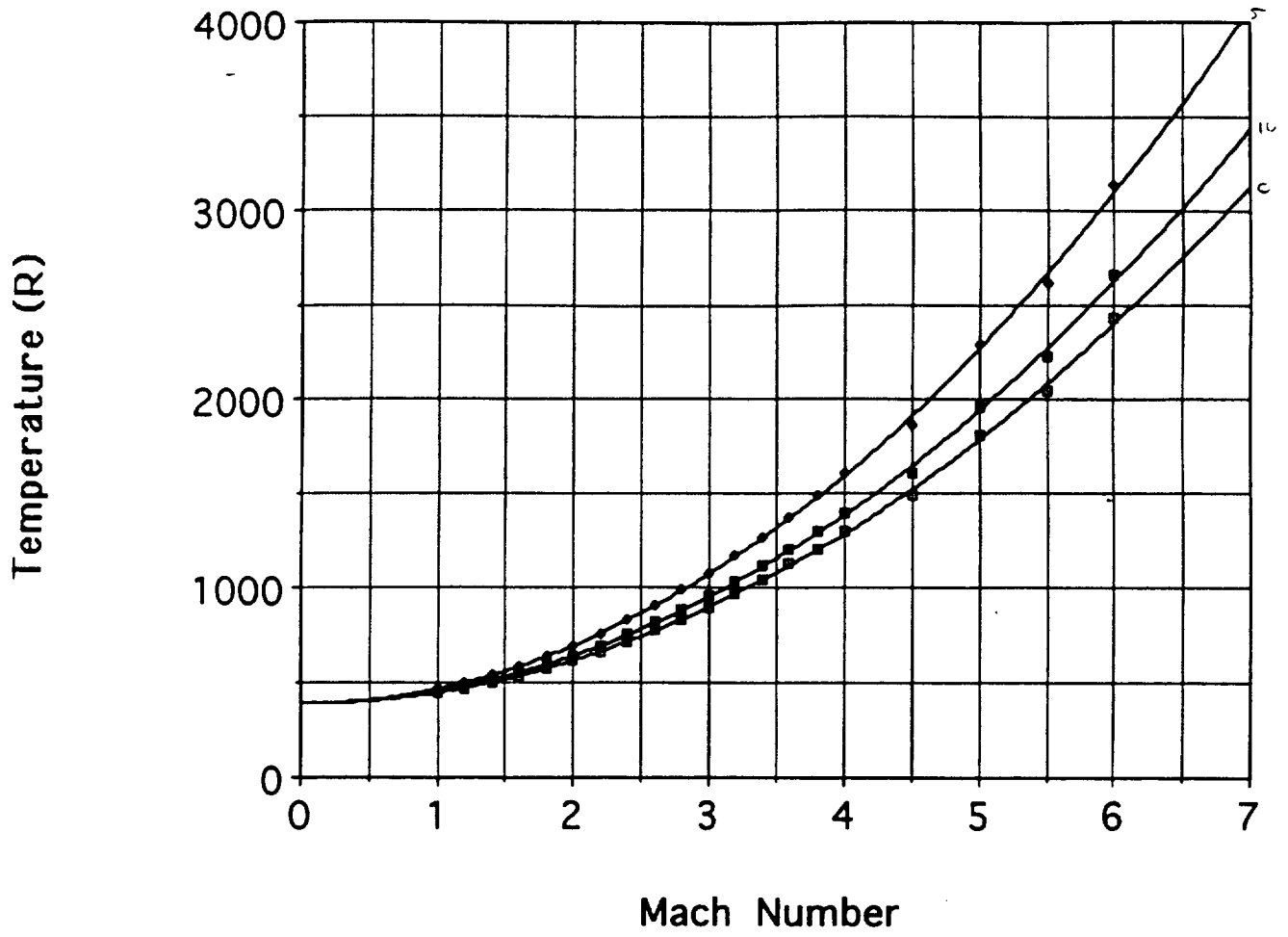Figure TP3  Q vs Mach at Leading Edge

Figure TP4  Temp at leading Edge vs Mach

# References

NASA Langley Research Center. 1991. APAS. Computer Software.

Dr. Richard Bodonyi, Professor Aeronautical/Astronautical Engineering, The Ohio State University, personal consultations, January -April 1992.

Nicolai, Leland M. "Fundamentals of Aircraft Design". METS, Inc (1975): 4/6-4/8.

# Stability & Control

# Summary

This research answers questions regarding the stability and control of a lifting body configuration for a reconnaissance aircraft. The purpose of this research is to obtain the cg as well as neutral point shift throughout the flight regime, determining the stability of the aircraft under cruise conditions, as well as examining some possible vertical control surface options. As a result of this research the cg location was found to move forward during flight while the neutral point was found to move aft. This in turn leads to an unstable aircraft during separation and climb conditions, and a stable aircraft during cruise. Until now research has been conducted on the use of clamshells or thrust vectoring instead of a vertical tail to attain stability. While thrust vectoring was ruled out due to weight criteria the effective use of clamshells will have to weight until further research is conducted on the aerodynamic characteristics at speeds below Mach .8.

## Introduction

Of critical importance in the succesful design of a hypersonic vehicle is the consideration of the stability and control criterion. Without the addressing of these criterion, the aircraft would not be able to maneuver, trim at cruise, or even land. Due to the lack of information on lifting bodies this became an increasingly interesting problem. It became apparent that traditional preliminary control surface sizing techniques would not be applicable since there was no aircraft this configuration was closely related to. Because of this it became necessary to conduct subsonic wind tunnel tests and obtain a good hypersonic analysis code to obtain valid stability and performance criteria. The following stability and control research answers the following questions: What is the cg. shift throughout the flight?, How does the neutral point shift throughout the flight?, What is the static longitudinal stability of the aircraft at cruise conditions?, What control devices have been considered for use?

Using the results of this research other people will be able to further study the best solution on the stability and control of a lifting body. This section of the report will explain the research methods, the results of the research, and the conclusions and recommendations.

## Research Methods

In conducting this research several sources were used. Basic information on stability and control criteria was obtained from Robert C. Nelson's *Flight Stability and Automatic Control* as well as from Donald McLean's *Automatic Flight Control Systems*. Moment as well as lift coefficients for various angles of attack ranging from $0^\circ$ to $20^\circ$ were obtained from an empirical code for Mach numbers from .8 to 2.0 and from a program called APAS for Mach numbers from 2.0 to 5.0. For further information on these computer codes please refer to the aerodynamics part of this report.

## Results of the Research

The research shows that there is a marked shift in the neutral point location with increasing Mach number. The research also shows that as a result of fuel usage, the cg. location will shift throughout the flight. It then became possible to calculate the static longitudinal stability of the aircraft at cruise conditions. Before a final analysis on stability and control could be achieved it was necessary to consider various control surfaces that might be used to trim the aircraft.

### How does the neutral point location shift with increasing Mach number?

Since empirical codes were available to calculate the moment coefficients at various angles of attack and Mach numbers for a given cg location, these were also utilized to obtain the neutral point of the aircraft. By plotting the moment coefficients vs. angle of attack for mach numbers between .8 and 5.0 and seeing at what cg locations the slopes of these curves became zero the neutral point was determined. A sample of one of these curves is shown in figure S1. As can be seen from the results shown in figure S2 the neutral point moves aft for increasing Mach numbers. This is an extremely desirable quality since it means that the aircraft will become more stable towards the cruise portion of the flight.

### How does the cg shift throughout the flight?

Determining the location of the center of gravity was very important since it dictated the static stability of the aircraft. The first step was to locate the empty cg location by summing the moments of all the integral components of the aircraft. From this an empty cg of 32.27 feet from the nose of the aircraft was obtained. Knowing this, the fuel tanks were arranged so that the aircraft would be unstable only for separation and climb conditions under which high maneuverability of the aircraft would
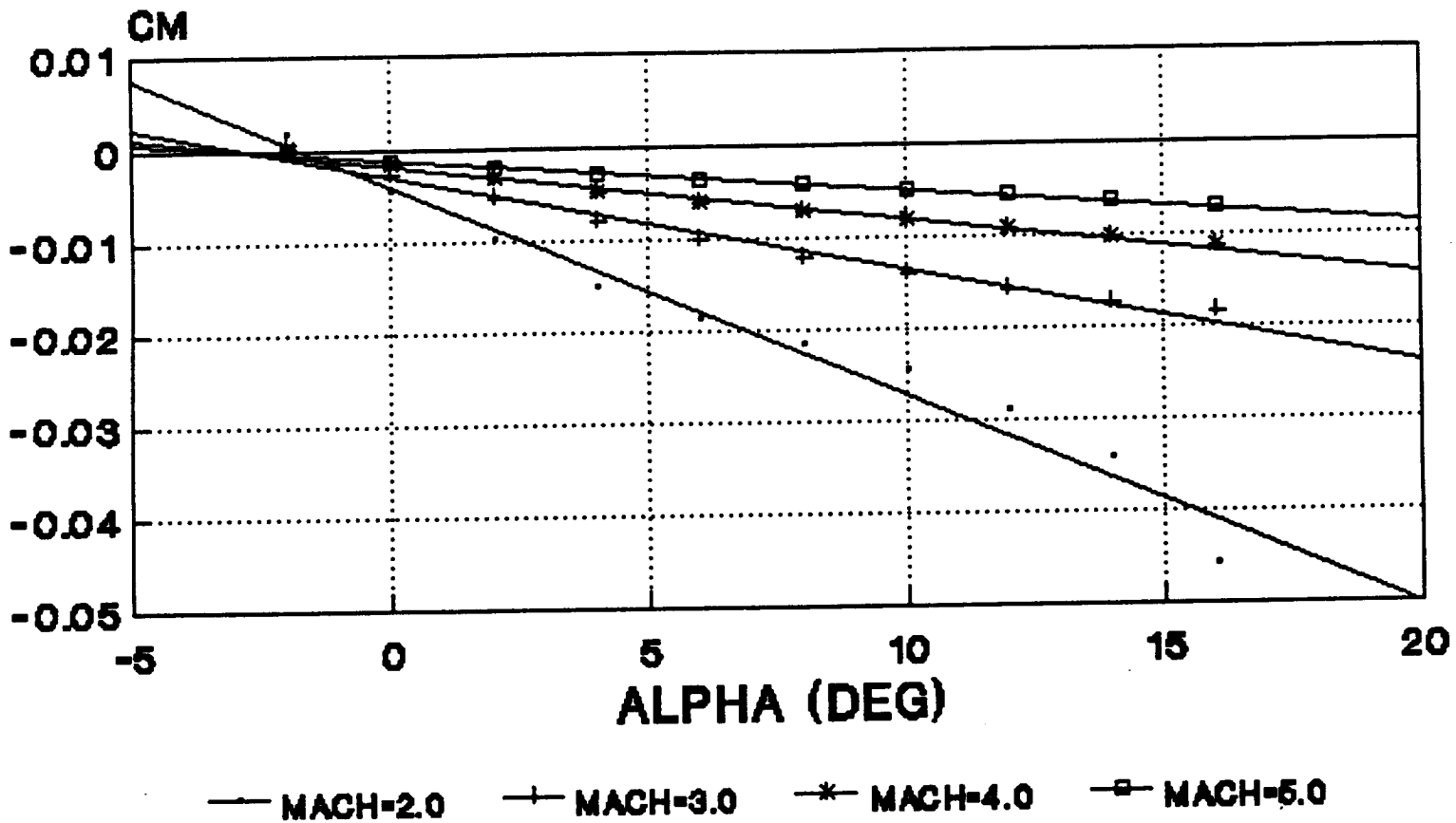
# CM VS ALPHA
## XCG=32.3,ZCG=-1.72



Figure S1 Cm vs. Alpha
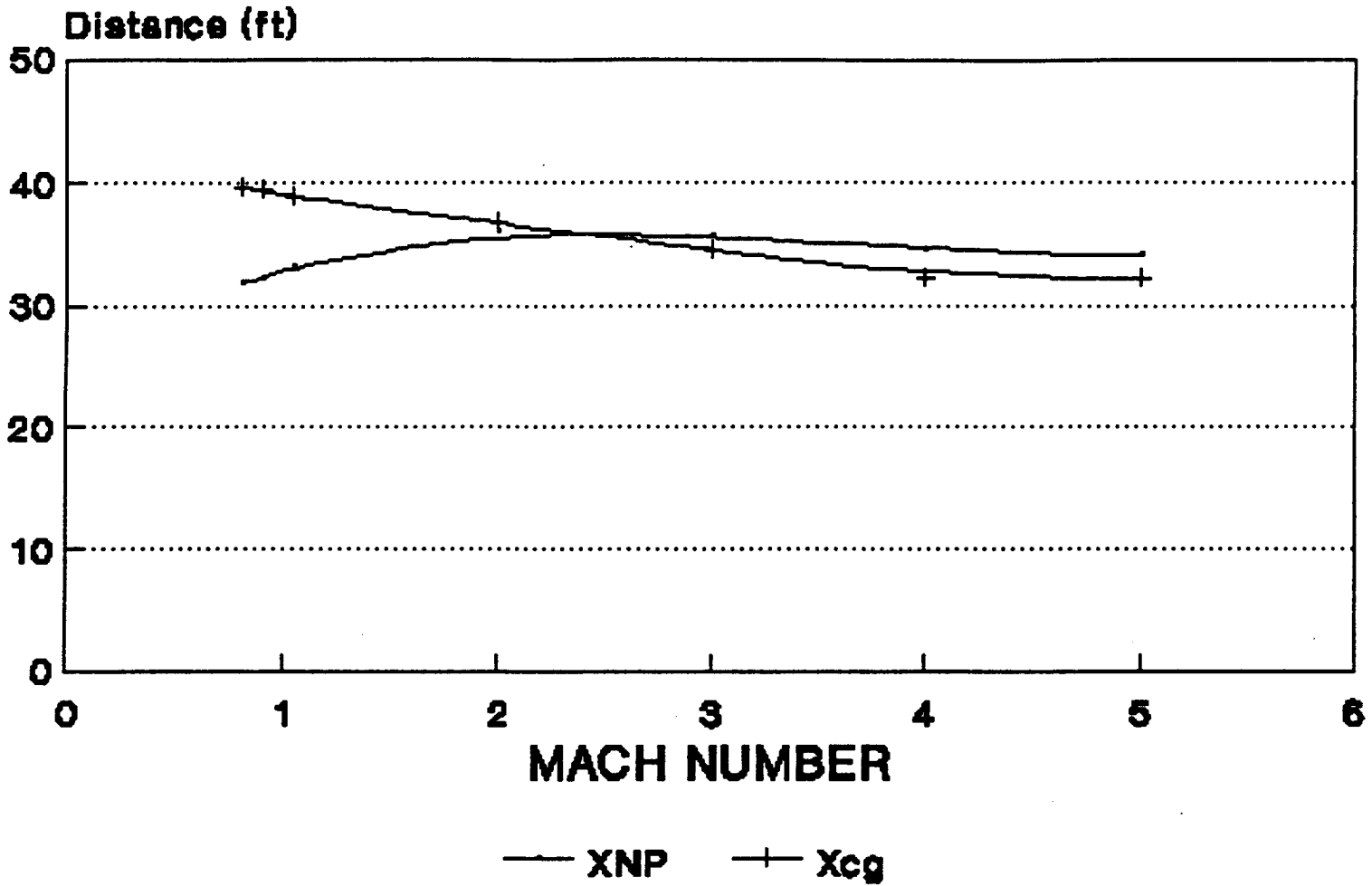
# NP & CG vs. Mach Number



Figure S2 NP & CG vs. Mach Number

be required(see figure S3). This in turn was done by placing all of the rocket fuel as far back as possible giving a cg location that would be aft of the neutral point. The outcome of the cg shift is shown on figure S2 and is described in this section. As a result of having the rocket fuel as far back as possible the aircraft is only unstable for the first 94 seconds during which separation and climb occur. During cruise, by using fuel only from the appropriate fuel tanks, the cg location was moved up to and kept at a statically stable position. The position chosen was the empty cg location of 32.27 feet. This was done so that minimum adjustments were required to be performed by the control systems.

<u>What is the static longitudinal stability of the aircraft at cruise conditions?</u>

As mentioned above the aircraft is statically stable at cruise. This can further be seen from the Cm vs. Alpha curve shown in figure S4. Here it can be seen that the slope of the line is negative thus ensuring the static stability of the aircraft. Also apparent from this graph is a positive $Cm_0$ which means that a positive elevator deflection is required to trim the aircraft. Also of importance are figures S5 and S6 which show CL vs. Alpha as well as Cm/CL vs. Mach number for the particular flight path taken.

<u>What control surfaces were considered for stabilizing the aircraft?</u>

In order to stabilize the aircraft there was never a question about having horizontal control surfaces since separation from the carrier aircraft requires a moderate pitching moment. Vertical control surfaces were eliminated from the design for a variety of reasons. First, since the mean camber line of the aircraft is reversed there is the question of whether this aircraft should land upside down in order to improve the
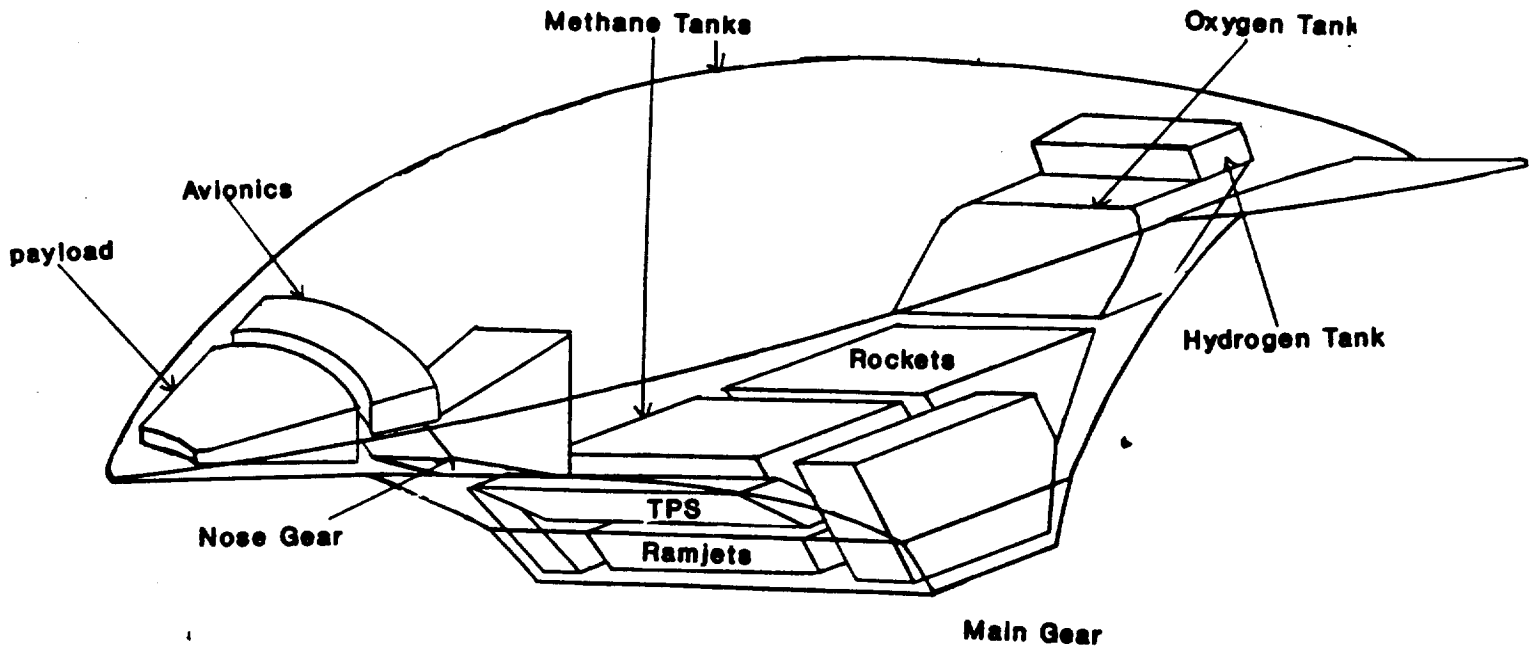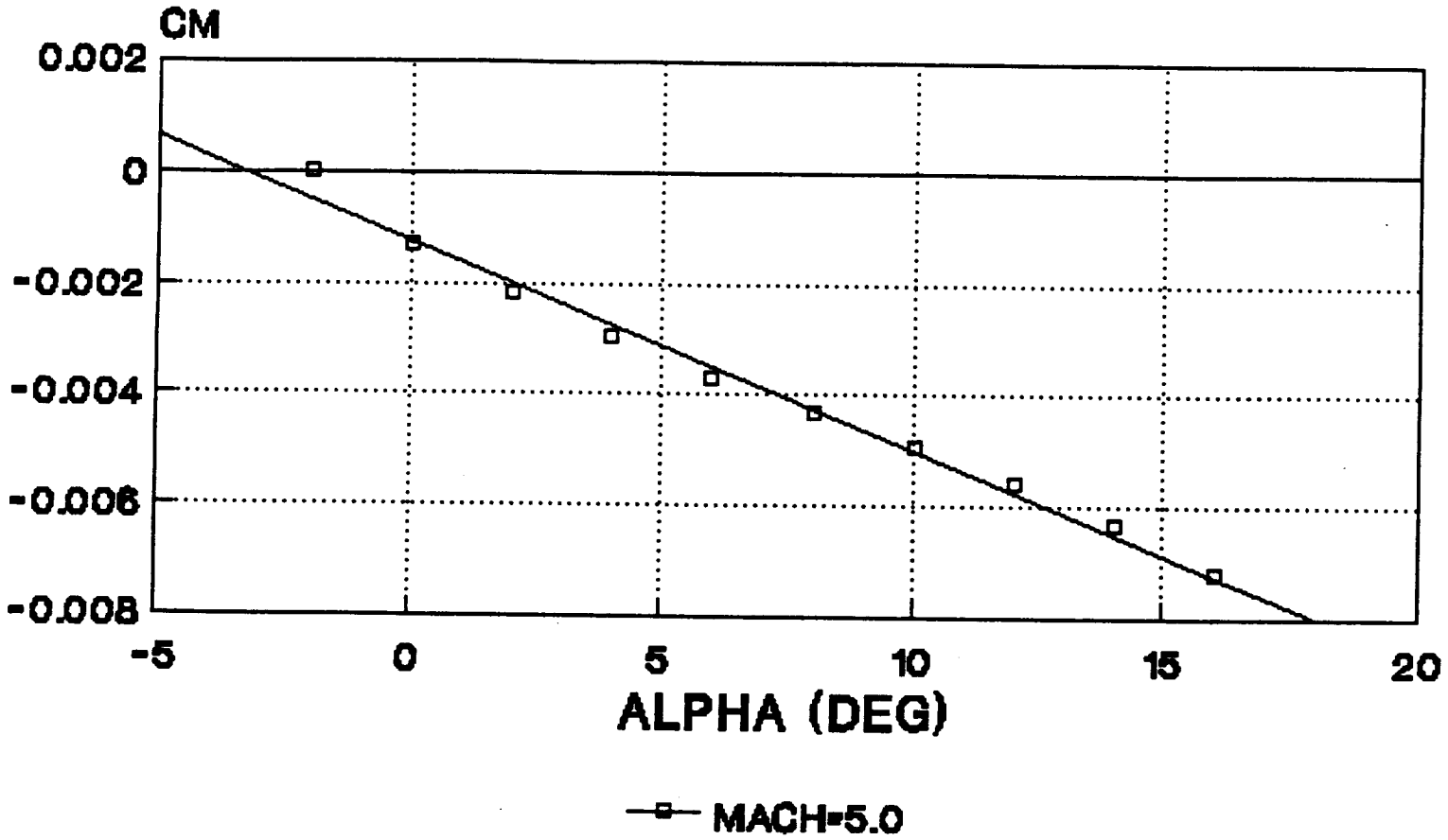
# Inboard Layout



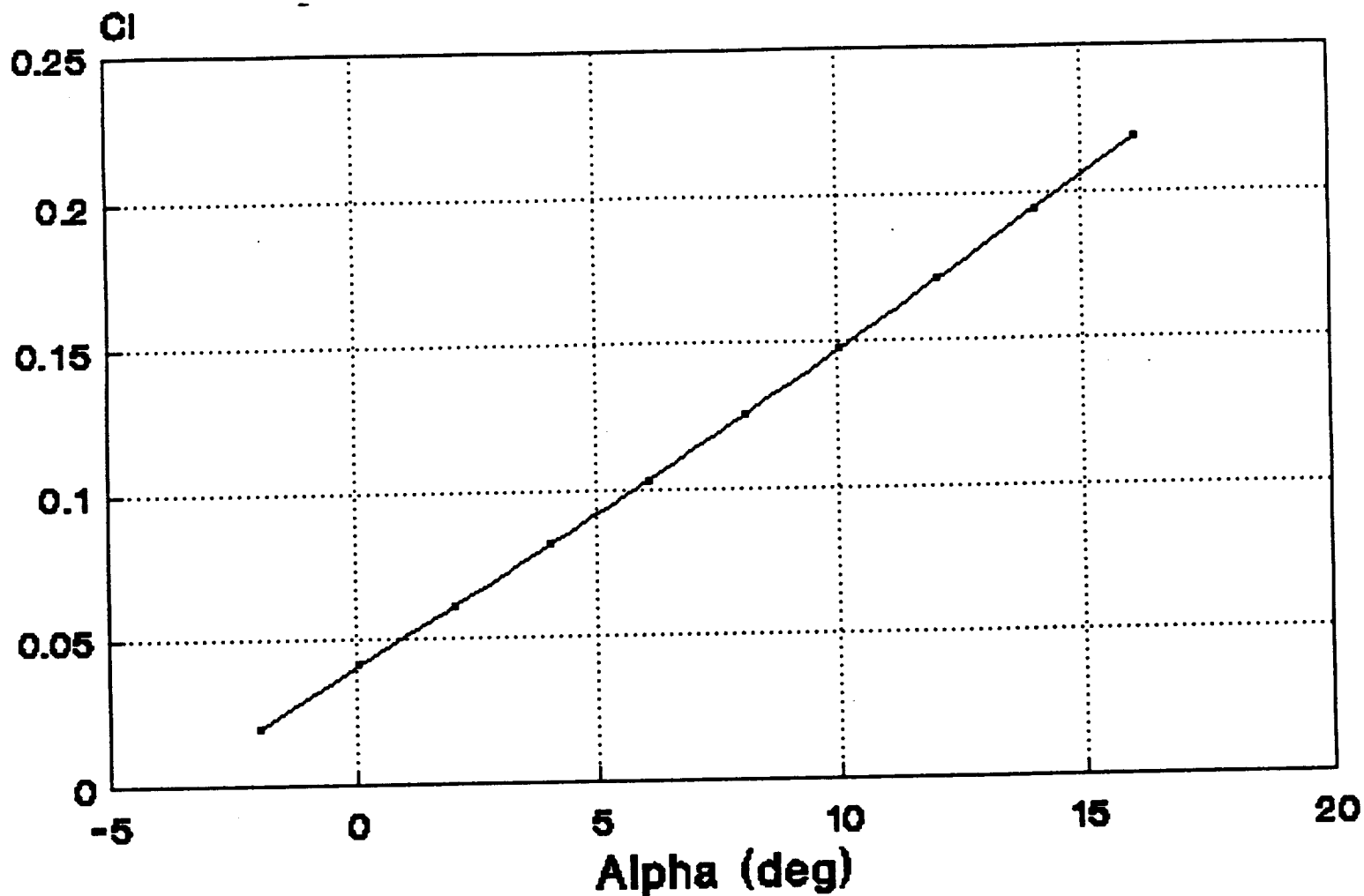Figure S3 Inboard Layout

Figure S4 CM vs Alpha

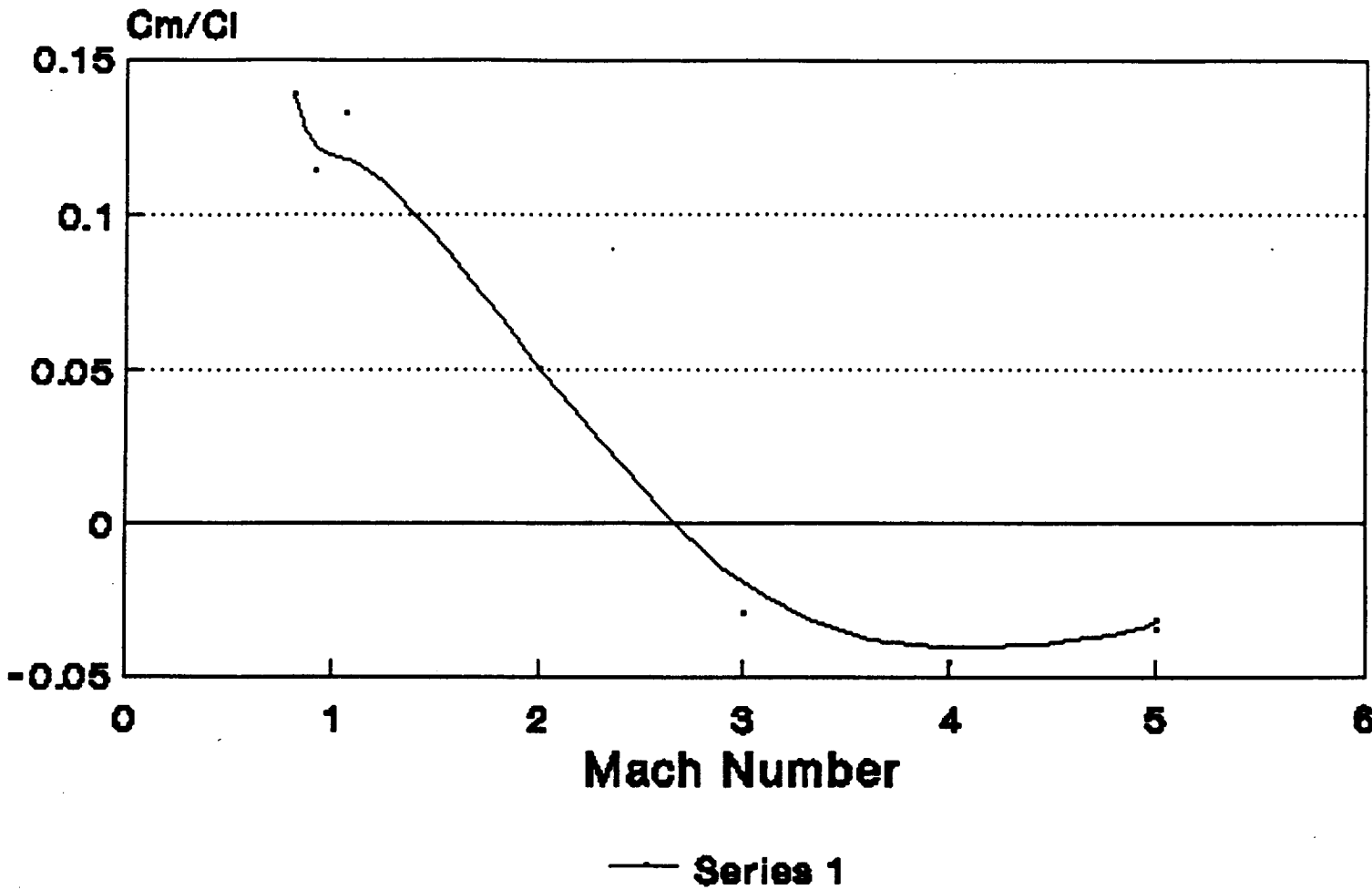# Cl vs Alpha



Figure S5 Cl vs Alpha

# Cm/Cl vs. Mach Number



Figure S6 Cm/Cl vs. Mach Number

aerodynamic characteristics. Also, due to the availability of present day control systems the question of a vertical tail being required came up. This is very important since the lack of a vertical tail will keep the drag to a minimum. Some systems that were looked at so that no vertical tail would be needed were thrust vectoring and clamshells on the horizontal control surfaces. Thrust vectoring proved to be inefficient since it required the addition of a heavy and complex system of ramps aft of the engines. This system though useful in aiding the maneuverability of high performance fighters sacrificed too much of the weight restriction imposed on the aircraft. Clamshells seem to be a better alternative but further research will have to be conducted in the future to determine their feasability. The problem with clamshells is in the configuration of the aircraft. Since the "wings" are extremely small it is in doubt that the clamshells could produce enough of a moment about the cg to be effective.

Conclusions and Recommendations

As a result of this research it is apparent that the aircraft performs effectively in the Mach .8 to 5.0 regime. The aircraft is unstable during separation and climb when high maneuverability of the aircraft is required. The aircraft then becomes stable during the cruise portion so that minimum control is required. From the low maneuverability that is required by this aircraft's mission profile I believe that clamshells on the horizontal surfaces might be used instead of a vertical control surface though further research at speeds below Mach .8 is clearly warranted.

# References

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


McLean, Donald. "Automatic Flight Control Systems." New York:Prentice Hall. (1990)

# Materials

The need for hypersonic vehicles, spurred research in lighter, stronger, and more heat resistant materials. These aircraft will need materials with high strength to density ratios to minimize the weight of their interior structures. Materials for the skin of hypersonic vehicles will have to endure high temperatures and be strong and light. The engines that power these aircraft will produce excessive temperatures in the combustion chambers where pressure is greatest. The material used in the combustion chamber must be able to withstand not only the temperatures produced, but also the pressures encountered.

This problem was attacked by separating materials into three categories. "Cold," "hot," and "engine" materials needed to be considered separately. The criteria for the cold material to be used for interior structures was that it needed to be strong and light. Metals and metal alloys could not provide the same strength to density ratios as metal matrix composites. An aluminum lattice structure with silicon carbide fibers (6061 Al/SCS-2) seemed to be the most promising material until thermoplastics were considered. Arotone, a crystalline thermoplastic composite, was found to have a strength to density ratio of two and a half times that of the aluminum/silicon carbide matrix while possessing the same operating temperature of 600 F. Although most of the information about Arotone is classified by its manufacturer, DuPont, it is known to have a tensile strength of 650,000 psi and a density of .12 pounds per cubic inch. This criteria along with a maximum operating temperature of 600 F make Arotone the obvious choice for interior structures. The hot material needed similar characteristics with the addition of an operating temperature between 1500 and 1800 F. Thermoplastics were disregarded because of their relatively low operating temperatures.

Figure M1 was used as a starting point. In the Mach four to five range the best material is shown to be titanium metal matrix composites. But ordered intermetallics, such as the SR-71 employed, were absent from the figure. Titanium aluminides provided the best results of the ordered intermetallics. The gamma type (TiAl) has an acceptable maximum usage temperature of 1650 F and a density of .137 pounds per cubic inch, but a strength of only 108,000 psi. The alpha-2 type, (Ti$_3$Al), has a better strength of 154,000 psi, but also a greater density of .159 pounds per cubic inch and an unacceptable maximum operating temperature of 1200 F. Titanium metal matrix composites were then studied so that they could be compared to the titanium aluminides. This is approached by first studying fibers used in composites. The maximum temperatures in oxidizing temperature (figure M2) were compared. SiC, ZrO$_2$, and Al$_2$O$_3$ were found to have the highest maximum usage temperatures. Silicon carbide, a widely used ceramic fiber, was found to have an expansion coefficient that closely matched that of titanium, so it was chosen as the fiber. After investigating several titanium alloy-silcon fiber matrix composites a six percent aluminum/four percent vanadium/ninety percent titanium alloy lattice structure was chosen. The silicon fiber content of the matrix composite is found to be optimum at 37 percent using discontinuous fibers. This material, Ti-6Al-4V/SCS-6, can withstand up to 210,000 psi, has a density of .132 lbs per cubic inch and a maximum usage temperature of 1800 F. The performance of this titanium metal matrix composite exceeded the titanium aluminides in every category so it was chosen as our hot material.

The engine material was by far the most demanding of the three. Temperatures of 3,000 F and pressures of 15 atm were targeted as design specifications. Basic engine materials were compared (figure M3) as a starting point. Since carbon is not a reasonable material for an engine, tungsten

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appeared to be our best choice, even when its high density was considered. The strength in the tungsten was alloyed with hafnium carbon. At 3500 F, less than one percent HfC raises the strength of tungsten from 9,000 psi to 50,000 to 60,00 psi (see figure M4 for detailed graph.) Unfortunately, this HfC addition raises the ductile-brittle transition temperature of tungsten from 200 to 400 F. The further addition of rhenium was found to improve low temperature ductility while further improving the high temperature strength. The optimum tungsten alloy was found to contain 4 percent rhenium, .35 percent hafnium, and .35 percent carbon. Figure M5 shows that W- 4Re-35Hf-.35C has a tensile strength of 70,000 psi at 3500 F, a 10,000 psi improvement over W-.35Hf-.35C and a 61,000 psi improvement over unalloyed tungsten while figure M6 shows that ductile-brittle transition temperature has been brought back down to 200 F by the addition of rhenium. NASA was quoted as saying that W-4Re-.35Hf-.35C "is the strongest known metallic material in the 3000 to 4000 F range." This high strength at operating temperatures will enable the engine combustion chamber walls to be thinner and therefore lighter than expected.

The design criteria for all three types of materials were matched as closely as possible. The cold material, Arotone, used for interior structures was found to have excellent strength to density characteristics and can operate at 600 F. The hot material chosen, Ti- 6Al-4V/SCS-6, was found to be lightweight and maintain high strength at up to 1800 F. The engine material for the combustion chambers of the ramjet engines was found to have excellent strength at temperatures exceeding the 3000 F design target. Overall, the material design goals were met successfully.
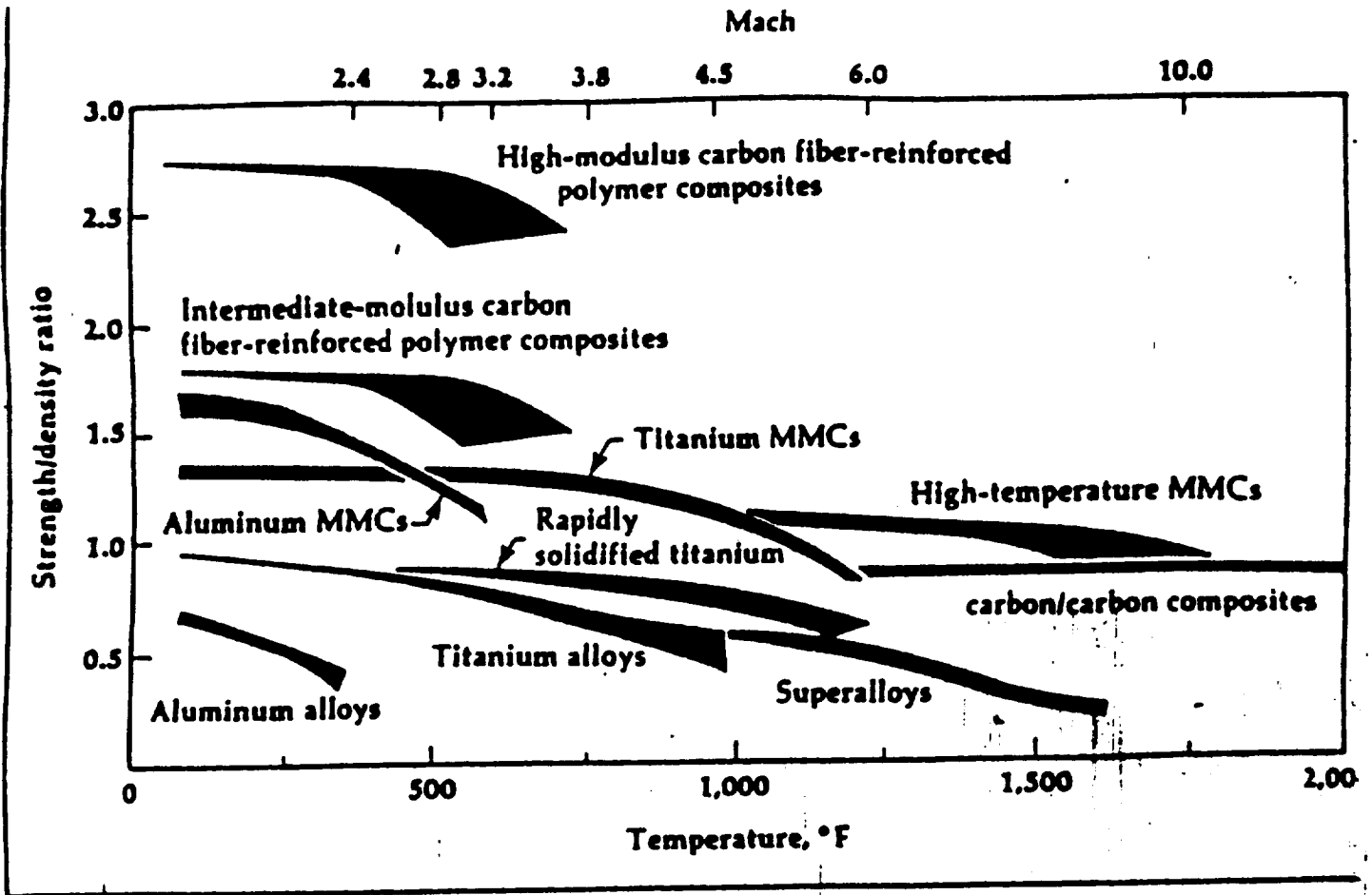
Figure M1  Material Properties

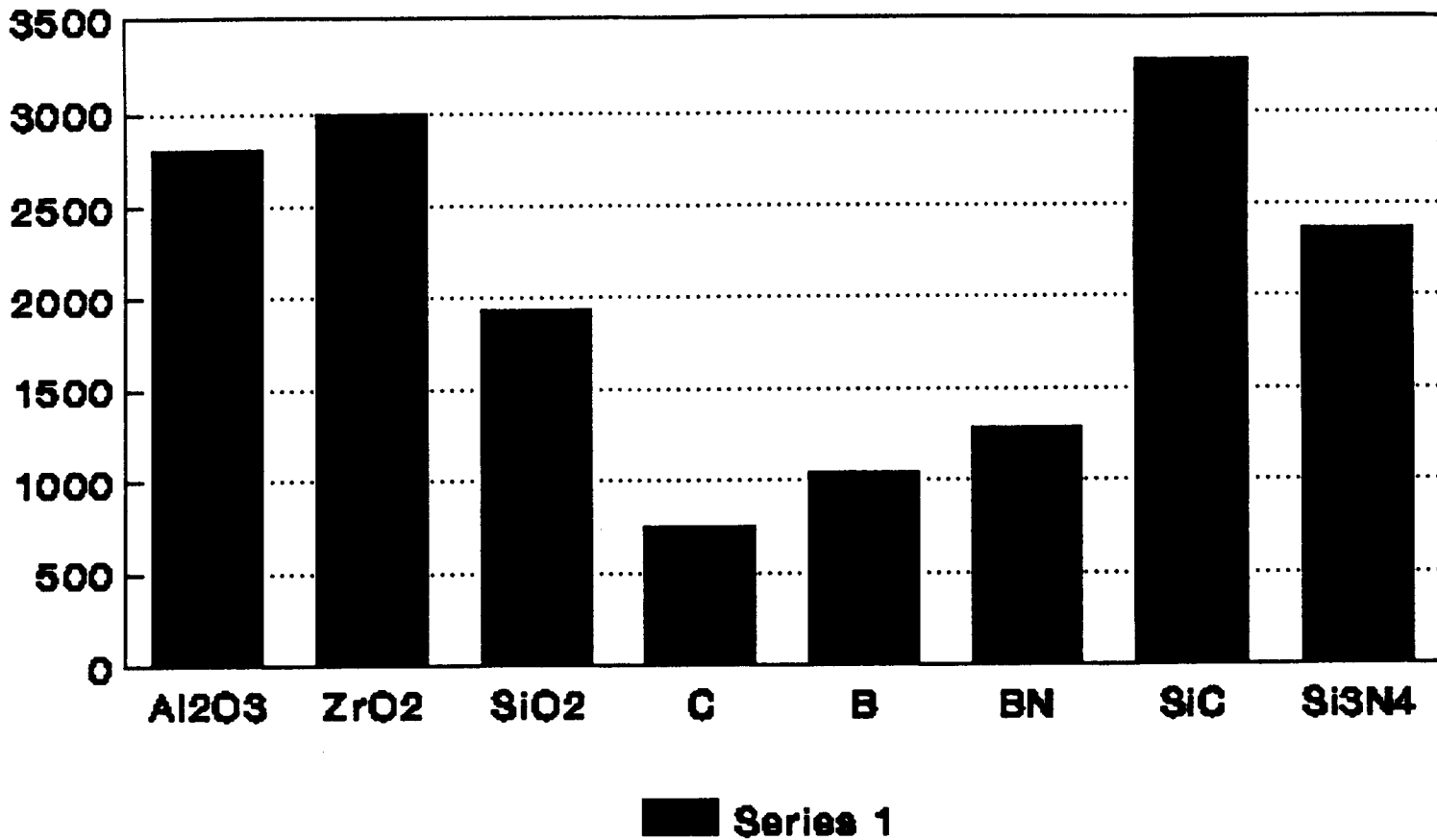# Fibers
## Max Temp – Oxidizing Atmosphere



Figure M2  Oxidizing Effects
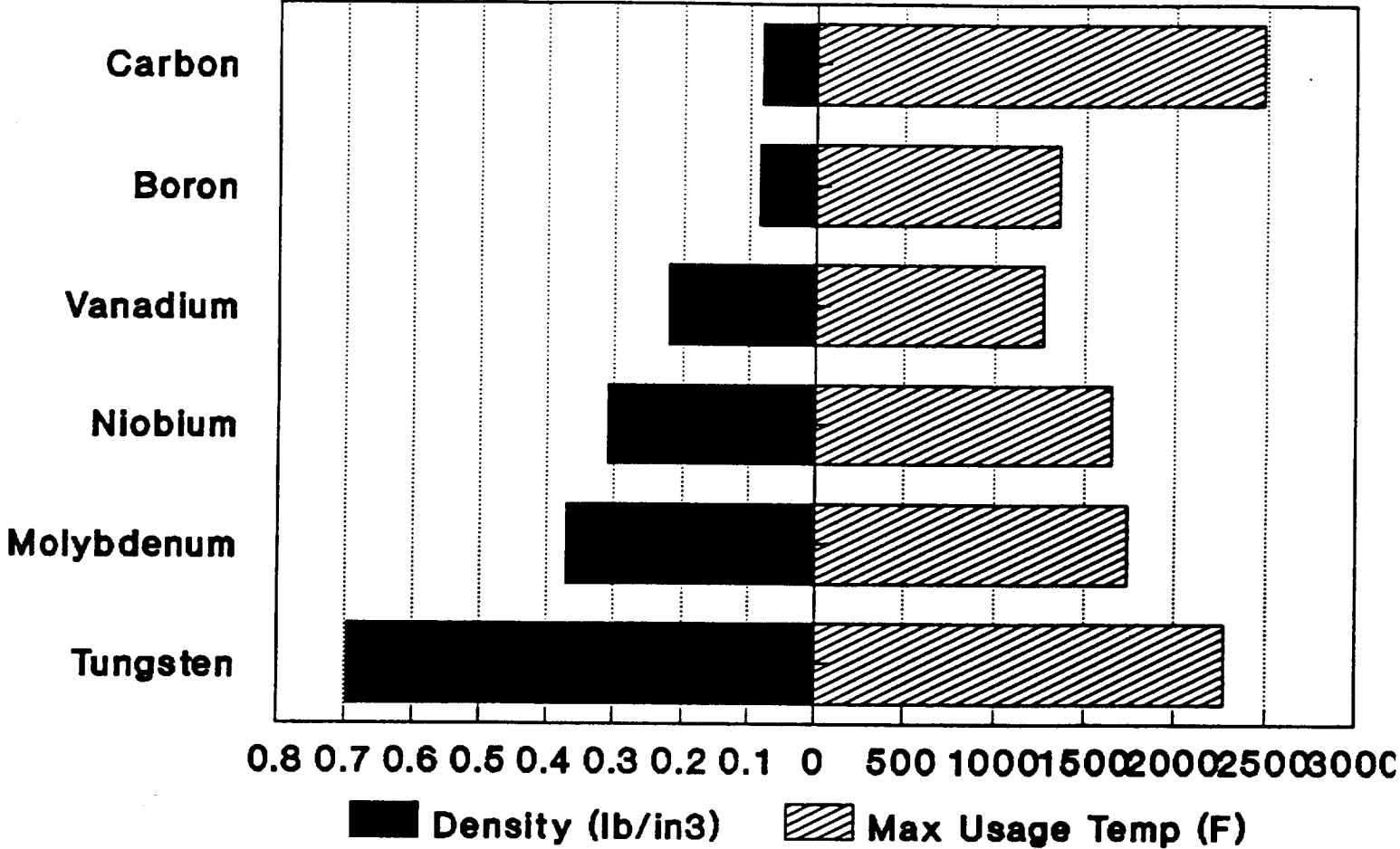
# Basic Engine Materials



| | 0.8 0.7 0.6 0.5 0.4 0.3 0.2 0.1 0 | 500 1000 1500 2000 2500 3000 |
|---|---|---|

■ Density (lb/in3)   ▨ Max Usage Temp (F)
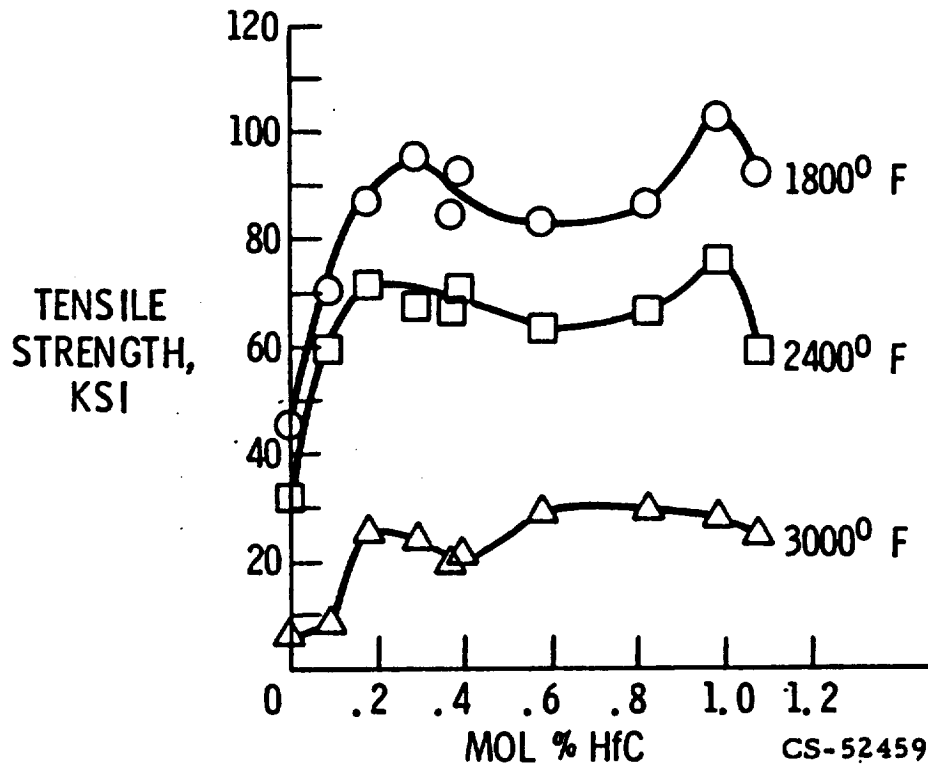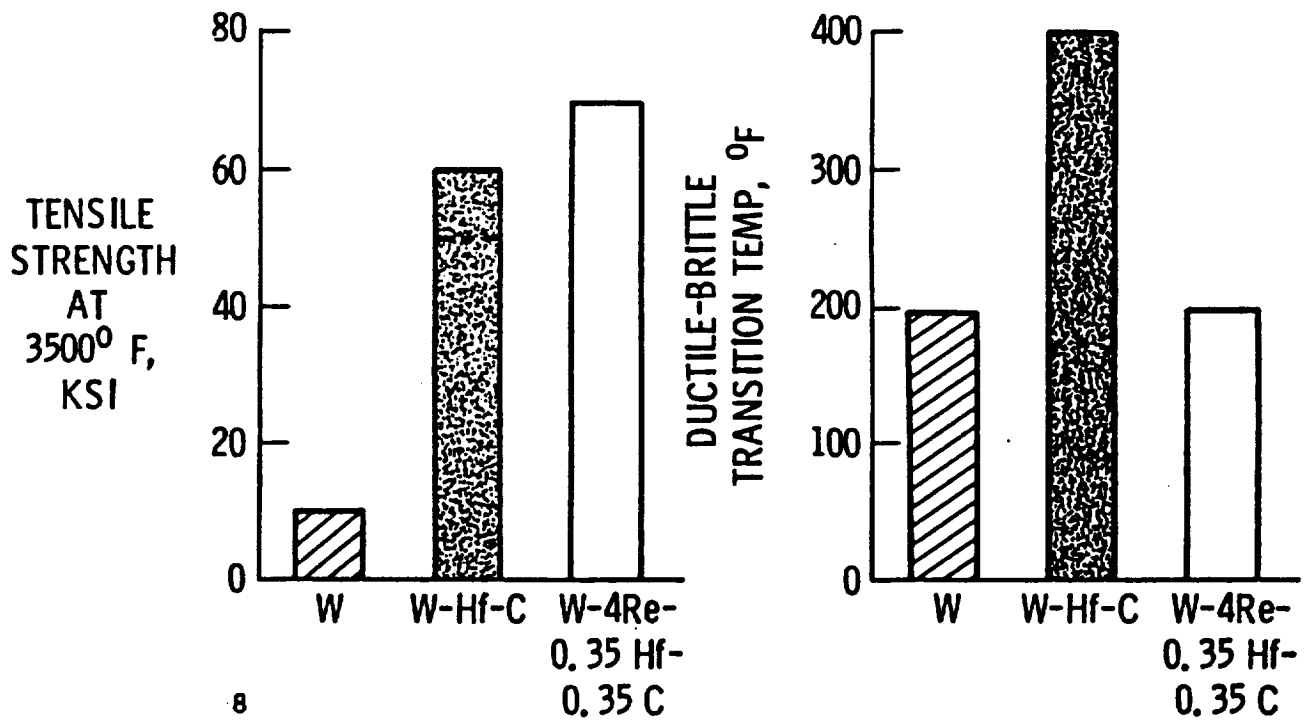
Carbon
Boron
Vanadium
Niobium
Molybdenum
Tungsten

Figure M3  Engine Materials

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Effect of hafnium carbide content on high-temperature tensile strength of Mo-Hf-C alloys.

Figure M4  Tensile Strength

Comparison of mechanical properties of W-RHC alloy with unalloyed tungsten and W-Hf-C alloy in swaged condition.

Figure M5 and M6  Strength and Brittleness Comparisions

# Model Making

An accurate wind tunnel model was required to establish the subsonic aerodynamic characteristics of the aircraft. Several steps were involved in the construction of the model. Beginning with the definition of the aircraft's geometry in CATIA. CATIA is a CAD/CAM developed by DeSault Systems in France, which generates computer instructions for the Okada NC/Mill machine. The computer instructions allow the Okada to construct any geometry defined in CATIA.

The aircraft's geometry was manually defined in CATIA by inputting points obtained from the configuration analyzed in APAS. The process was long and tedious and required the expertise of Tom Merrick for completion. Once the geometry was defined in CATIA, milling surfaces were defined over the geometry. These surfaces would serve as the paths that the cutting bit would take to construct the model.

Upon establishment of the cutting paths, CATIA was instructed to generate the NC milling code needed by the Okada to mill the surfaces. This code was then transferred to the Okada milling machine. At this point Shelby Davis mounted a pine block measuring 16"x4"x4" on a wooden base in the milling machine. The rest of the process was computerized in that the milling machine followed the instructions generated by CATIA. The first set of cuts were bulk cuts designed to remove most of the excess material and define the general shape of the model. The final cut produced a finished model accurate to 1/1000". The process is more involved than what is depicted on this page. A better understanding of it can be established by viewing figures M1-M3. The process would have cost several thousand dollars had it been initiated by industry, however, due to its academic nature allowed for a considerable reduction in this cost.
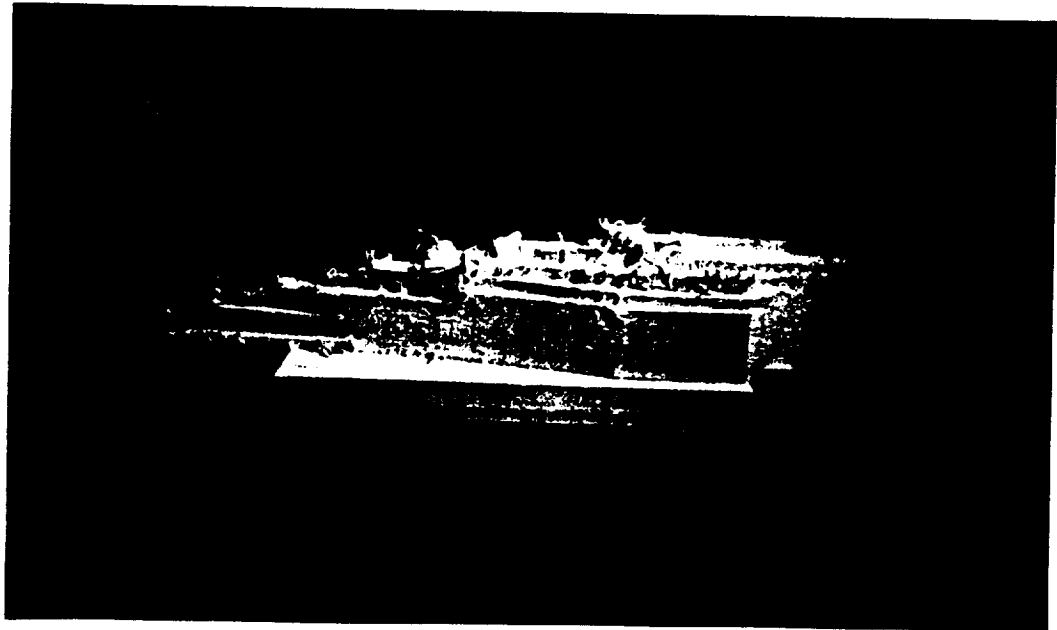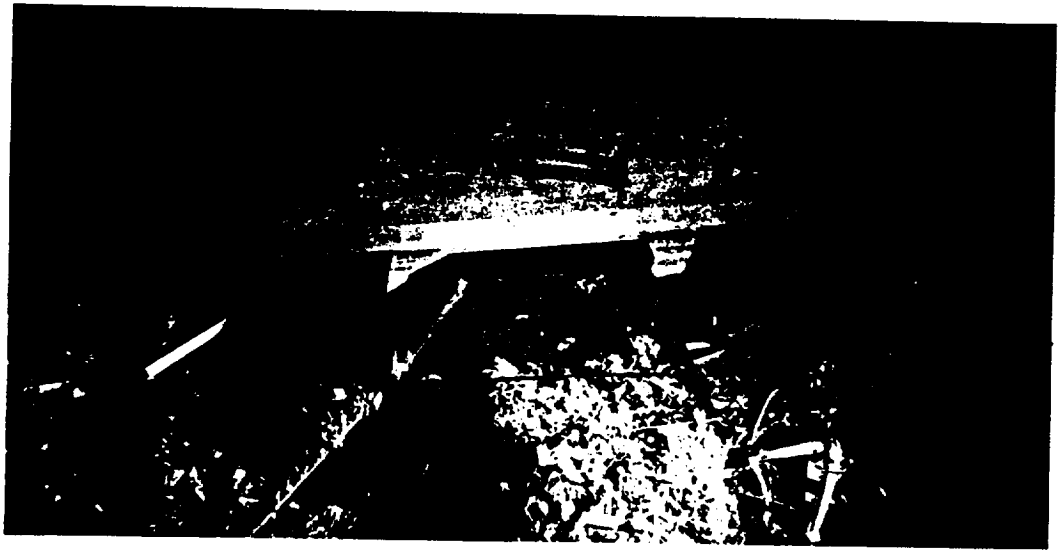
Figure M1

87

Figure M2

Figure M3

89    C-2
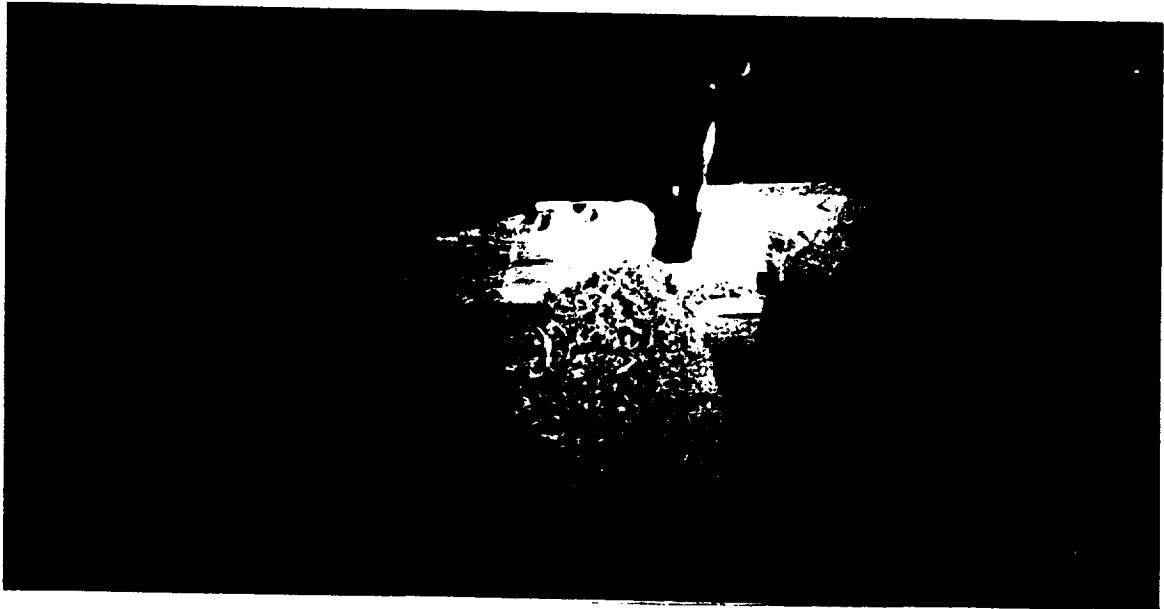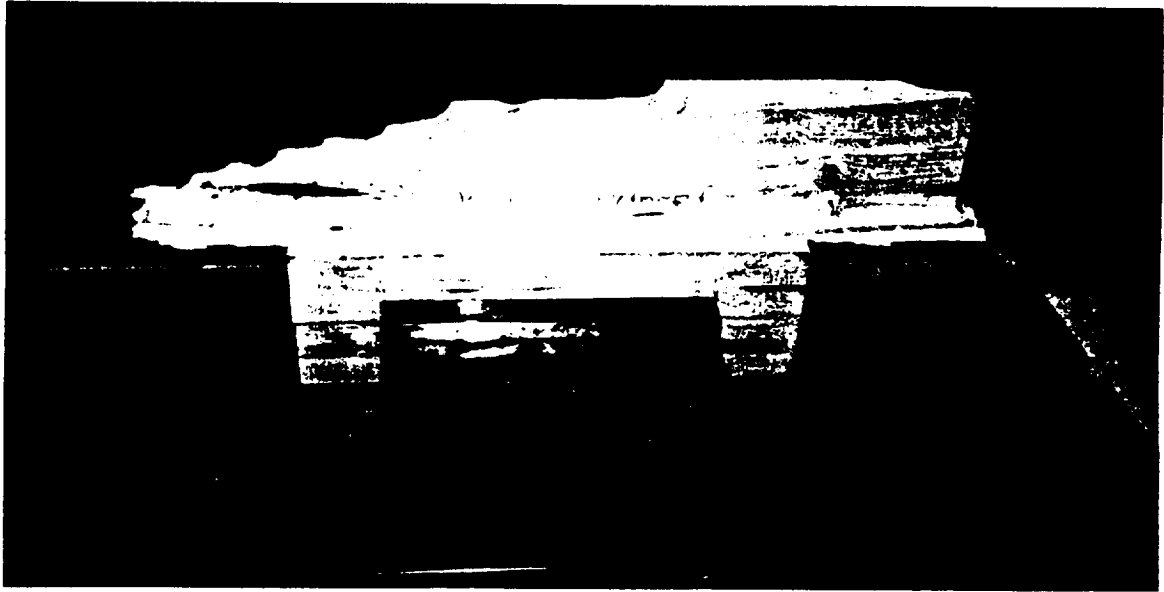
# Cost Analysis

## Summary

The purpose of this section is to explain the methods by which the developmental, production, and operational costs of the Peregrin were derived. The estimates of the costs are broken down and tabulated in the following sections of this document.

## Introduction

The aim of Scarlet 1 is to keep the cost of the Peregrin project low while maintaining the highest quality standards. The first decision was to limit the number of prototypes to one. It was decided that more than one developmental aircraft would not be cost effective. The idea of eliminating the prototype by making all aircraft production line aircraft was eliminated because it was felt that this could injure the quality of the Peregrin and increase costs in the future.

### Research Methods

The cost of outfitting the United States with twelve Peregrins and sustaining these aircraft is estimated by using modified versions of existing cost analysis formulas (Nicolai 24-1 to 24-35). The total cost of producing these aircraft includes the research and development costs of producing a single prototype and eleven additional aircraft are assembled. In addition to the purchasing cost, the operational cost, which must stay below $300 million per year, are included in the total cost estimate.

The initial and sustaining engineering hours were calculated from:

$$E = .25\left[.0396A^{.791}S^{1.526}Q^{.183}\right]$$

where   A = AMPR weight in lbs = 11644 lbs

S = maximum speed in knots at flight altitude = 9408knots

Q = cumulative quantity produced

$$= Qd \text{ for developmental phase}$$

$$= Qd + Qp \text{ for production phase}$$

* AMPR (Aeronautical Manufacturers Planning Report)
weight is the empty weight of the aircraft less the
engines, TPS, starter, instruments, fuel cells, and
payload.

These engineering hours include the time spent on development of the airframe, avionics, engines, and TPS. The sustaining engineering hours, Ep, include the additional engineering time needed during the production phase of the aircraft. The factor of .25 is included because the technology is near term. Aircraft engineering costs are found by multiplying this number by the engineering dollar rate of $48.00.

The developmental support costs, which include all labor and materials used in the engineering work above, was found from:

$$D = .25 \left[ .008325 A^{.873} S^{1.890} Q^{.346} C \right]$$

where     C = $3.40, which is the conversion factor from 1970 to 1992 dollars from the Consumer Price Index.

The factor of .25 is included here also because engineering technology is near term.

The costs of producing a single engine was:

$$P = KT^{.8356} C$$

where     K = 80 for the ramjet engine

     T = thrust at 80,000 ft

The cost estimate considered six engines for the prototype and four engines each for each of the production aircraft.

The manufacturing labor hours include the time required to fabricate, machine, process, and assemble the major parts of the aircraft. These hours

were found from:

$$L = 28.984 R^{.748} S^{.543} Q^{.524}$$

The manufacturing labor cost is found by multiplying L by the hourly rate of $30.00. Quality control costs were estimated as 13% of the manufacturing labor costs.

The cost of the materials and equipment needed for the production of the aircraft was found from:

$$M = 25.672 R^{.689} S^{.624} Q^{.792} C$$

The amount of tooling hours needed for aircraft production was found from:

$$T = 4.0127 R^{.764} S^{.899} Q^{.178} R^{.066}$$

where      R = Production rate in deliveries per month

The production of the prototype was estimated at 18 months and the production of the next eleven aircraft was estimated at 18 additional month total. The testing costs are found by multiplying the tooling hours by the hourly rate of $38.50.

The cost of the prototype flight test operations was found using:

$$F = .001244 R^{1.160} S^{1.371} Q_D^{1.281} C$$

Ten percent of the total was added to the price as profits for the manufacturing company.

The operational costs of the aircraft were found by estimating that maintenance and parts for each Peregrin to be 10% of the labor, materials and equipment, tooling and quality control costs of a production aircraft. A flight operations cost of one million dollars per flight was estimated for 12 flights per year per aircraft.

## Results of the Research

The tabulated results for the costs of the development, test, and evaluation and production are shown in Table 1. The total cost for production

of the first twelve Peregrines is just over $4.3 billion translating to a cost of $362 million for each aircraft.

The yearly operational costs are shown in Table 2. The total operational costs for twelve Peregrines each performing twelve flights per year is $242 million. This is well under the operational cost limit of $300 million specified as a design goal.

Total Development Test and Evaluation

| | | | |
|---|---|---|---|
| Aircraft Engineering | | $ | 905 Million |
| Development Support | | | 812 Million |
| 1 Prototype | | | 711 Million |
|     Engines | $ 4 Million | | |
|     Manual Labor | 128 Million | | |
|     Materials & Equipment | 16 Million | | |
|     Tooling | 546 Million | | |
|     Quality Control | 17 Million | | |
| Flight Test Operations | | | 62 Million |
| | Subtotal | $ 2.49 Billion | |

Total Production and Unit Cost for 11 Additional Aircraft

| | | |
|---|---|---|
| Engines | $ | 34 Million |
| Manufacturing Labor | | 342 Million |
| Materials & Equipment | | 103 Million |
| Sustaining Engineering | | 521 Million |
| Tooling | | 410 Million |
| Quality Control | | 44 Million |
| Subtotal | $ 1.454 Billion | |

| | | |
|---|---|---|
| Total Production for 12 Aircraft | $ 3.944 Billion | |
| Profit (10% of Total) | $ | 394 Million |
| Total Production Cost | $ 4.338 Billion | |
| Average Cost | $ | 362 Million |

Table C1 Production Cost analysis

Yearly Operational Cost

| | | |
|---|---|---:|
| Manufacturing Labor | | 3,105,000 |
| Materials & Equipment | | 933,000 |
| Tooling | | 3,730,000 |
| Quality Control | | 403,000 |
| | Subtotal | $ 8,171,000 |
| Total Cost for 12 Aircraft | | $ 98 Million |

Flight Operations:

| | |
|---|---:|
| Cost Per Flight | 1 Million |
| Flights per Aircraft | 12 |
| Number of Aircraft | 12 |
| Total Yearly Flight Cost | $ 144 Million |
| Total Operational Costs | $ 242 Million |

Table C2 Yearly Maintenance Analysis

# References

Nicolai, Leland M. "Fundamentals of Aircraft Design". METS, Inc (1975): 24/1-24/35.

# Appendix A
# Aircraft configurations

# Configuration - Three View

configuration 2
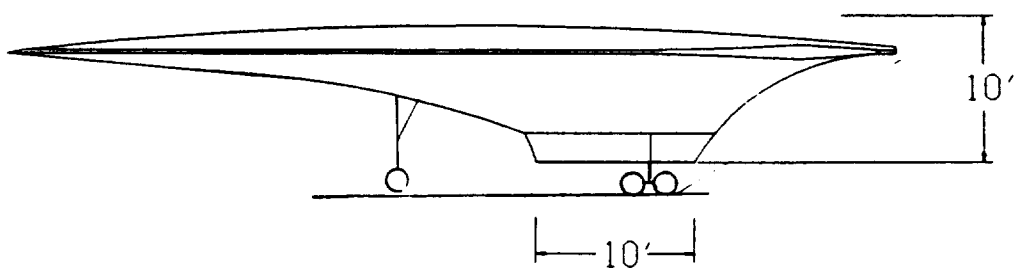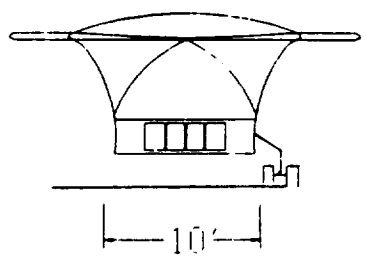
48'

105'

30'
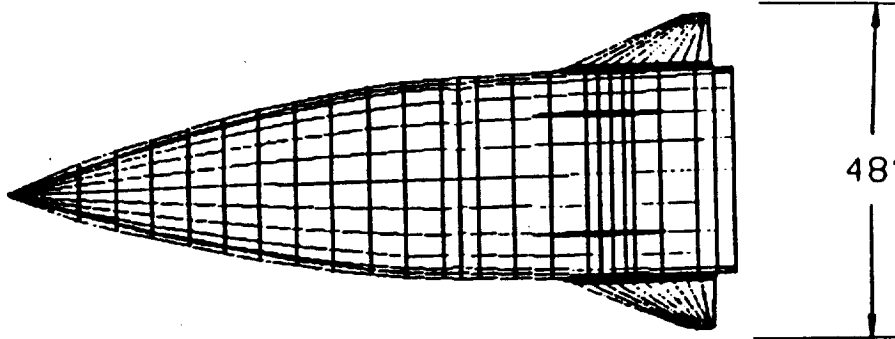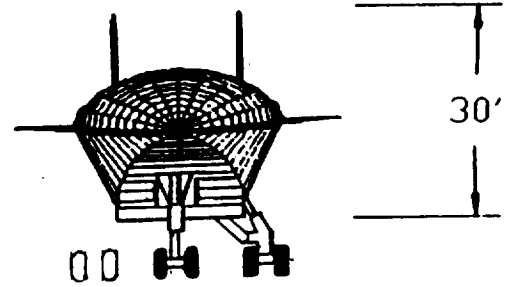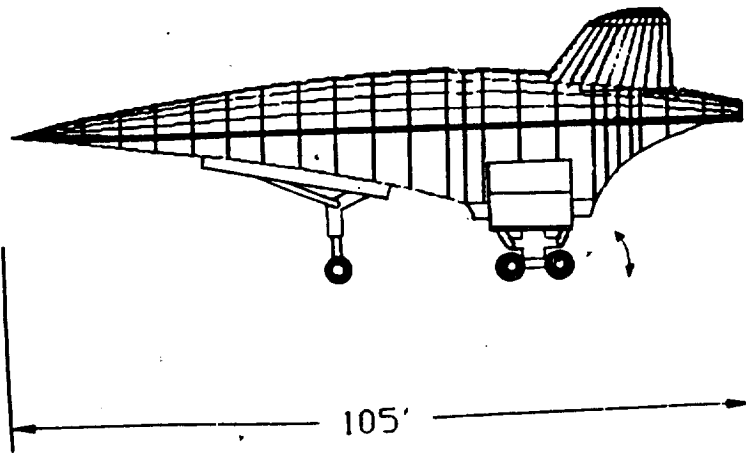
# GROUP SCARLET I
# HYPERSONIC RECONNAISSANCE
# AIRCRAFT

28'

61'

**configuration 4**

10'

10'

10'
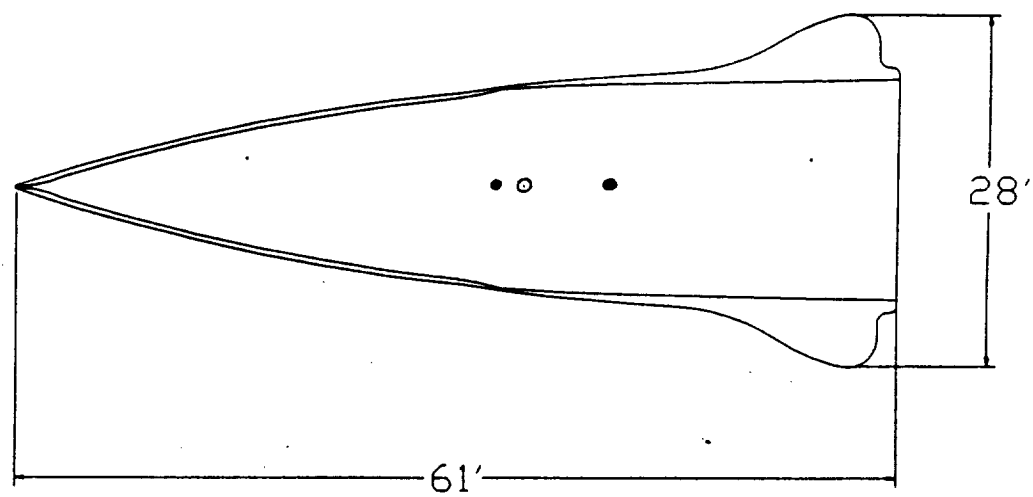
# Conformal Temperature Map
## Temperature (R)



1,600

1,400　　1,300　　1,200　　1,150

1,700　1,500　　　　　　　　　　　　　1,100

1,500　　1,300
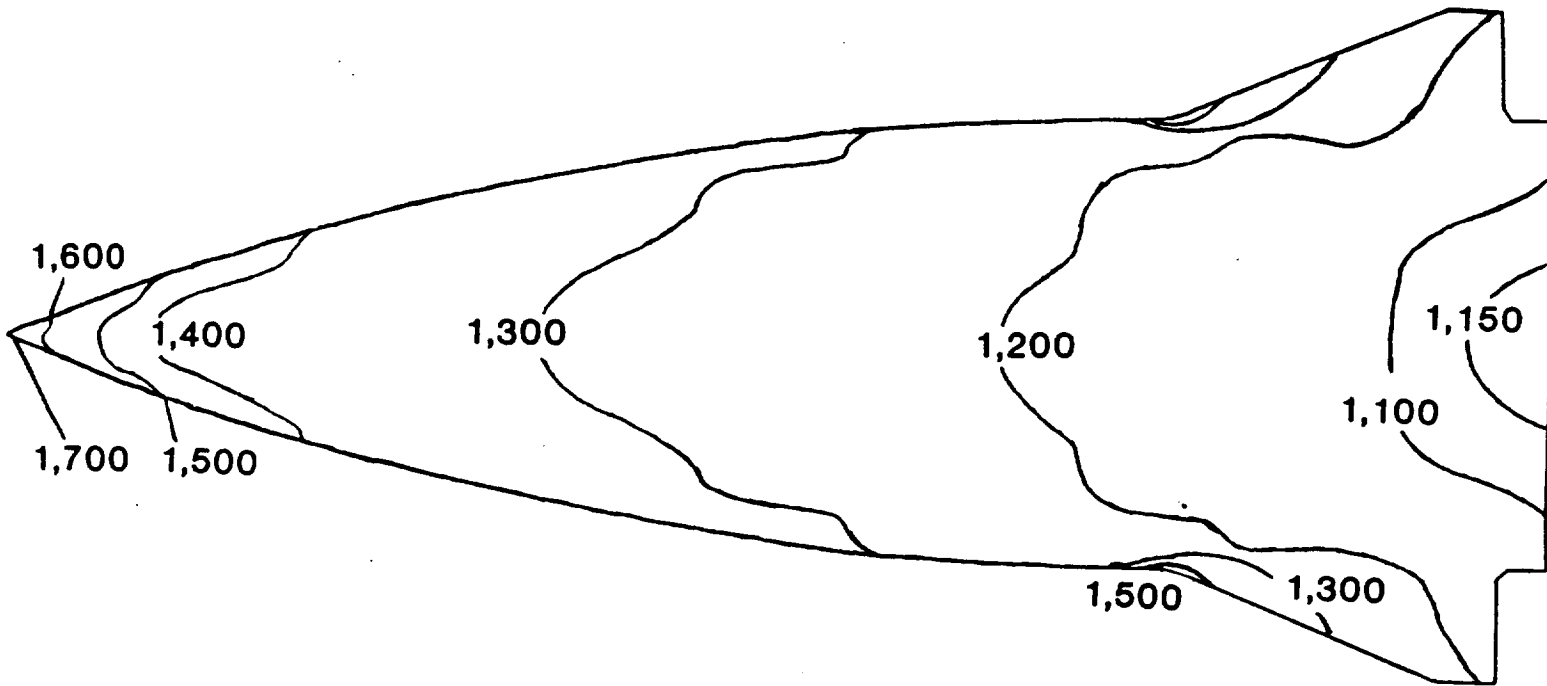
# Appendix B
# Development Histories

# Development History of Engines

Thrust (lbs Thousands)

# Rocket History

Thrust (lbs/rocket Thousands)

# Weight History



Weight (lbs Thousands)