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Tesseract
Supersonic Business Transport

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Abstract

This year, the senior level Aerospace Design class at Case Western Reserve University developed a conceptual design of a supersonic business transport. Due to the growing trade between Asia and the United States, a transpacific range has been chosen for the aircraft. A Mach number of 2.2 was chosen too because it provides reasonable block times and allows the use of a large range of materials without a need for active cooling. A payload of 2500 lbs. has been assumed corresponding to a complement of nine (passengers and crew) plus some light cargo.

With these general requirements set, the class was broken down into three groups. The aerodynamics of the aircraft were the responsibility of the first group. The second developed the propulsion system. The efforts of both the aerodynamics and propulsion groups were monitored and reviewed for weight considerations and structural feasibility by the third group. Integration of the design required considerable interaction between the groups in the final stages. The fuselage length of the final conceptual design was 107.0 ft while the diameter of the fuselage was 7.6 ft. The delta wing design consisted of an aspect ratio of 1.9 with a wing span of 47.75 ft and mid-cord length of 61.0 ft. A SNEMCA MCV 99 variable-cycle engine design was chosen for this aircraft.

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Introduction

The Aerospace Design class was given the task of developing a conceptual design of a supersonic business transport. The initial specifications for the design were developed by the class and are listed below.

Table 1: Design Specifications

Range	Transpacific
Mach Number	2.2
Passenger & Crew Capacity	9
Total Payload	2500 lbs.

With these guidelines, the class was broken down into three groups. Each of the three groups was placed in charge of one of the following design areas: 1) Aerodynamics, 2) Propulsion and 3) Structure. The iterative process of aircraft design began with an initial sizing of the aircraft. For the specifications listed above, a takeoff gross weight of 107,000 lbs. was estimated. Also, a fuselage length of 107.0 ft and a diameter of 7.6 ft were determined in the initial study. After the initial sizing was completed each of the three groups began a detailed analysis of their respective design area. During the design process, constant communication between the groups was required to keep the project on line. The final conceptual drawings of Tesseract are presented in Figures 1 thru 4 on the following pages.

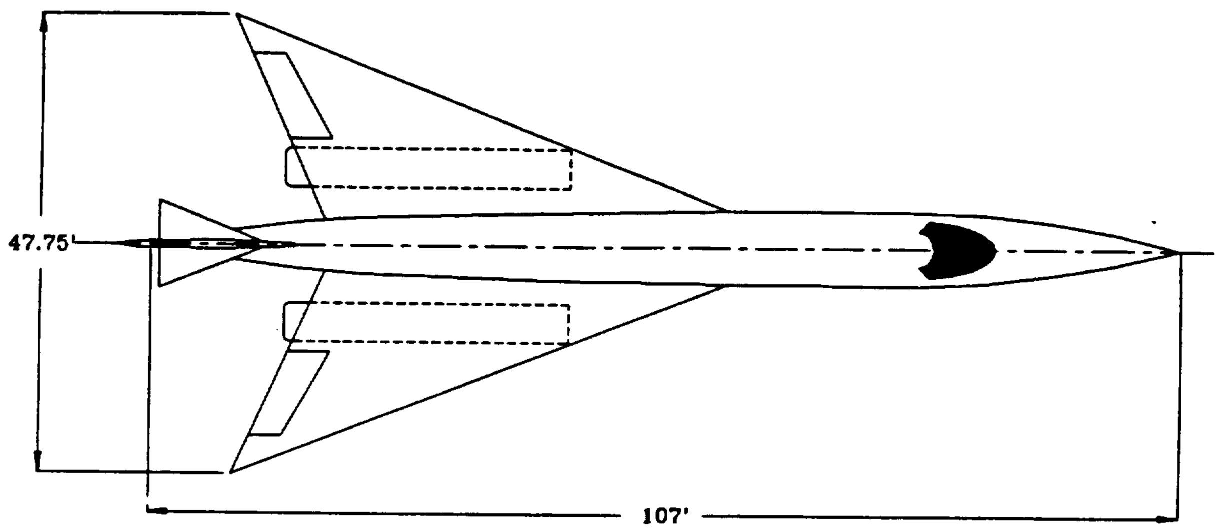

Figure 1: Top View

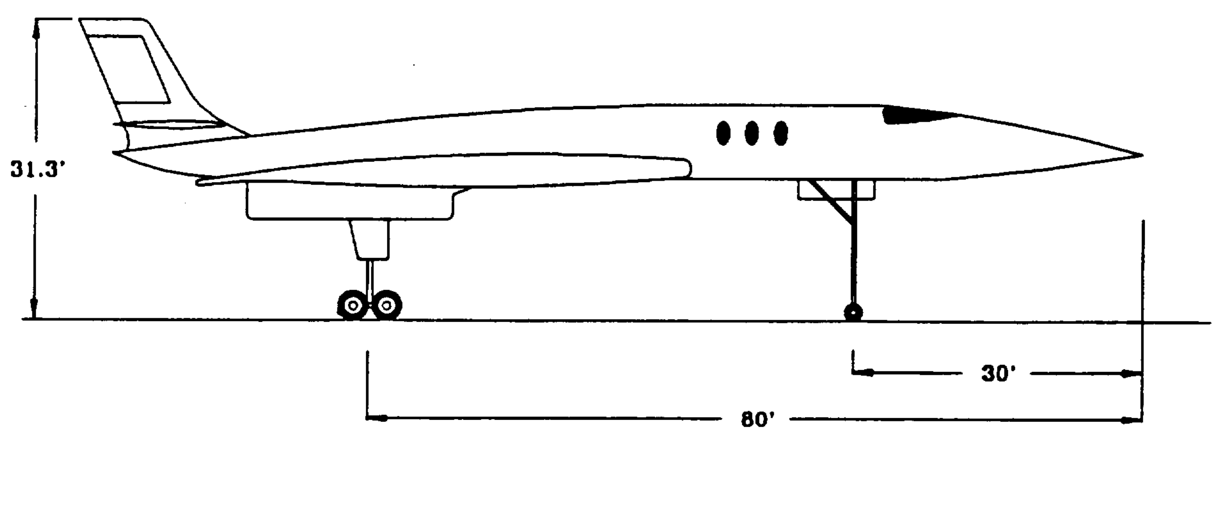

Figure 2: Side View

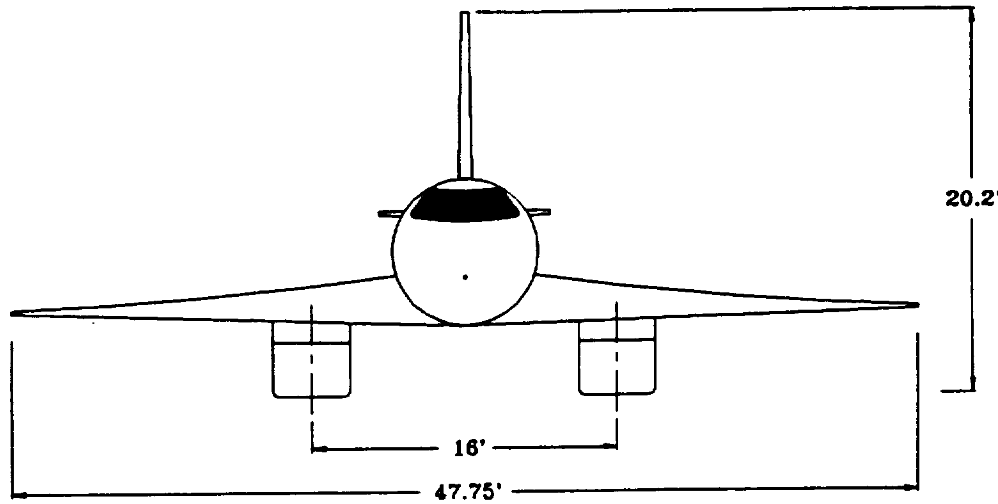

Figure 3: Front View

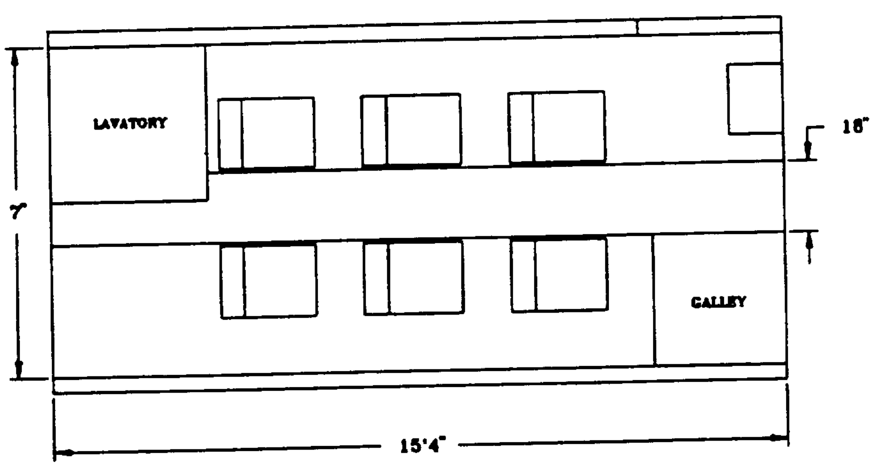

Figure 4: Cabin Layout

Analysis

Section 1: Aerodynamics

During the initial conceptual sizing of the proposed supersonic business jet, similar designs indicated that the jet would have approximately a maximum lift to drag ratio (L/D max) of 8. Historical trends indicated that the most efficient cruise for jet aircraft occurs at velocities higher that those that would generate a maximum lift to drag ratio. This higher velocity is at a L/D of 86.6% of maximum (Raymer[1]). In our case, cruise L/D would be roughly 7.

A design cruise lift coefficient (CL_{cruise}) was now determined from initial mission requirements and basic flight mechanics. For an aircraft with a takeoff gross weight (TOGW) of 107,000 lbs. and a cruising Mach number (M) of 2.2, a reasonable CL_{cruise} needed to be selected. A target range for the cruising CL from 0.12 to 0.13 was selected based on similar designs. After some iteration, a design lift coefficient of 0.128 was determined. This cruising CL was designed for a wing reference area of roughly 1200 square feet and an initial cruising altitude of 55,000 feet using equation 1.

$$CL = \text{Lift} / q\, S_{ref} = \text{Weight} / (0.5 * \rho_{h=55,000} * V^2 * S_{ref}) \quad \text{(Eq. 1)}$$

q = Dynamic Pressure ρ_h = Density of air at 55,000 ft.

V = Velocity S_{ref} = Reference Area

$$CL = 99,800\ \text{lb} / (0.5 * 2.87\text{e-}4\ \text{slug/ft}^3 * (2129.5\ \text{ft/s})^2 * 1200\ \text{ft}^2)$$

Note that a weight of 99,800 lbs. was used in the calculation rather than 107,000 lbs. This smaller value is less the TOGW by a factor of 12% of total fuel weight (60,000 lbs.) which is a conservative estimate of the fuel consumed during takeoff and climb. Any error in this estimation could later be accounted for by slightly adjusting the initial cruising altitude.

As just alluded to, maintaining a constant lift coefficient during the cruising portion of the mission while accounting for a constantly changing weight (fuel consumption) can be accomplished by increasing the altitude of the aircraft periodically as the fuel supply is diminished. Alternately, velocity can be altered (reduced) to accomplish the same effect but obviously this method is not practical. From similar calculations as above, the results in Table 2 were obtained.

Table 2: Altitude as a Function of Fuel and CL_{cruise}

CL Cruise	Fuel Remaining (% weight)	Altitude (ft)
0.128	88	55000
0.128	53	60000
0.128	25	65000

From the above analysis, a change in cruising altitude of roughly 10,000 feet would be required to maintain a constant lift coefficient. Such a flight profile (see Figure 1, Appendix A) might have restrictions due to flight regulations of maintaining constant altitude during all or portions of the mission. Although, at present, such altitudes are not as populated as some lower flight levels, such considerations must be mentioned in the early design stage. Implications of this may result in the aircraft not flying at its design lift coefficient during the entire cruise.

Before the analysis could proceed any further, a wing planform needed to be selected. Several wing planform designs with subsonic or supersonic leading edges were investigated. Forward swept and eccentric wings were considered (primarily for novelty) but were unfortunately discarded due to a lack of literature and data available on these subjects. A delta configuration with subsonic leading edges was chosen for the main reason that theories for wing performance of deltas existed and were readily available.

A subsonic leading edge was desired to minimize the supersonic wave drag. To guarantee this the leading edge sweep back angle must lie within the Mach cone. Based upon a free stream cruise Mach number of 2.2 and a normal to the leading edge Mach number of 0.8, the sweep back angle was calculated to be 68.7°. This lies within the Mach cone which is 63.0°.

Based on the previous discussion, and to minimize induced drag (as shall be discussed hereafter) an aspect ratio of 1.9 was desired. Based on the aforementioned pure delta configuration, an aspect ratio of 1.56 was calculated. This aspect ratio needed to be increased without changing the reference area of the wing. The main motivation for this was to minimize the induced drag which is inversely proportional to the aspect ratio. To accomplish this a triangular section was removed from the trailing edge of the wing (see Figure 2, Appendix A).

There are theories available to predict the performance of delta wings. One such theory developed by Brown[3] for the lift curve slope (a) is as follows:

$$a = 2\,\Pi^2 \tan e \,/\, (\Pi + \text{lambda}) \qquad \text{(Eq. 2)}$$

Lambda is a function of the ratio of one-half the apex angle tangent (tan e) to that of the tangent of the Mach angle (see Figure 3, Appendix A). For the particular configuration shown, lambda is equal to 1.25 which in turn gives a lift curve slope of 1.76 per radian as shown in Figure 4, Appendix A.

Airfoil selection is difficult due to the unavailability of recent airfoil developments. An airfoil must be selected to meet the above mentioned parameters. Based on historical trends for this type of aircraft, a thickness ratio (t/c) between 0.07 and 0.09 is predicted. This range excludes the use of present day supercritical airfoils, because they tend towards higher thickness ratios (roughly 0.15).

The next step in aerodynamic considerations was the calculation of the total drag on the aircraft during cruise. To determine the parasite drag coefficient, the component buildup method as prescribed in Raymer[1] was followed. This method considered each portion of the aircraft separately. The value of the overall coefficient was then found by summing the drag of the individual components. Each component's skin friction drag was determined using flat plate approximations. These values for Mach 2.2 are summarized in Table 3 and are presented in Appendix B.

In lieu of the effects of lights, antennae's, and other manufacturing defects along with other unaccountable factors, an exact coefficient can not be determined. A correction factor of 10 percent can be added to the skin friction drag of the aircraft as prescribed by Raymer[1].

The wave drag of the aircraft was determined using an approximation method described in Raymer[1]. This method is valid only for a cross sectional area distribution of the aircraft similar to a Sears-Haack distribution (see Figure 5, Appendix A). Aimed at minimizing wave drag, the aircraft was designed as close as possible to this ideal distribution. For the aircraft, the wave drag coefficient corrected for Mach number and non-ideal area distribution is 0.0068. The values for the induced drag at supersonic speeds were calculated using a theory developed by Brown[3] similar to that used for determining the lift curve slope. This method produces an induced drag value of 0.0058 for the aircraft (see Figure 4, Appendix A).

Drag calculations for the subsonic and transonic regimes were calculated for varies altitudes using software developed by Kern[4] International entitled Basic Aircraft Performance Analysis . This program calculated parasite drag for Mach numbers ranging from 0.0 to 1.0. These values were predicted by simply smoothing the curve generated from the data above. While this may seem a crude approximation, such a technique will suffice for the preliminary design. All the results of the parasite drag are presented in Figure 6, Appendix A as a function of Mach number.

Table 3: Drag Summary During Cruise

Parasite Drag Coefficient

Skin Friction Drag	.0051
Wave Drag	.0068
Miscellaneous	.0005
TOTAL $C_{D,0}$	.0124

Induced Drag

Induced Drag $C_{D,i}$	.0058

Total Drag Coefficient

Parasite Drag	.0124
Induced Drag	.0058
TOTAL C_D	.0182

Longitudinal static stability of most conventional aircraft require the use of a horizontal stabilizer or simply a horizontal tail. For an aircraft with a delta wing configuration, an actual horizontal tail separate from the wing is not always present. Rather, the horizontal tail surface is part of the delta wing configuration (see Figure 7, Appendix A).

Several difficulties, however, arise from not employing a horizontal tail separate from the wing. To maintain longitudinal static stability, the tail of the aircraft may need to produce a force in the direction of gravity to balance the moments of the aircraft about the center of gravity. This will require a portion of the wing to generate negative lift. This then requires the portion of the wing generating positive lift to balance the negative lift as well as support the weight of the aircraft to maintain level flight.

In analyzing the static stability for the aircraft, it was discovered that a horizontal stabilizer was essentially unnecessary for the cruising speed of Mach 2.2 (results in Appendix C show that a slightly positive lifting force is required). However, for flight at speeds lower than our cruising speed, the aircraft becomes inherently unstable. This is primarily due to the large shift in the aerodynamic center of the aircraft. The analysis for low speed static stability needs to be evaluated and an appropriate control system needs to be incorporated. For the present, a tail has been added to the design in anticipation of its use in maintaining low speed static stability.

The criteria for longitudinal static stability at cruise were satisfied for our design with the horizontal tail. The necessary criteria are:

1. $C_{M,O}$ must be positive (moment at zero lift)
2. $\partial C_{M,cg}/\partial A_a$

For this design, the pertinent stability figures are listed in the following table at are calculated in Appendix C:

Table 4: Stability Analysis

Location of the center of gravity as a fraction of root chord (Empty Weight)	0.70
Location of the aerodynamic center as a fraction of root chord at M =2.2	0.77
Moment coefficient of the wing body about the aerodynamic center at M=2.2	0.00
Tail Area	50 ft^2
Distance of tail aerodynamic center to the center of gravity	30 ft
Wing Reference Area	1200 ft^2
Mean aerodynamic chord of the delta wing	33.6 ft
Tail Volume Coefficient	0.037
Static Margin at M=2.2	0.09

The aerodynamic center of the delta wing was determined using a graphical method prescribed in Raymer[1]. This method allows us to determine the location of the aerodynamic center of the wing as a fraction of the root chord. The graphical method is shown in Figure 8. For the aircraft, the graph corresponding to a taper ratio of zero was used in the analysis.

To maintain longitudinal static stability the aircraft's center of gravity throughout the flight must remain in front of the neutral point. This distance as a fraction of the chord is known as the static margin and should not go less than 5 percent during any portion of flight. If the static margin falls below 5 percent then the forces required to maintain balance may become too large. However, if the static margin exceeds 15 percent the aircraft becomes 'sluggish'. This essentially means that the restoring forces resulting from changes in angle of attack are small, resulting in slow response time.

Section 2: Propulsion

The propulsion system consists of two variable cycle engines mounted under the wings toward the rear of the aircraft. The system is designed for a flight cruise speed of Mach 2.2. The fuel to air ratio for this system was assumed to be 1/35. The thrust required at Mach 2.2 is 7180 lbf. The mass flow rate of air at cruise is 79.45 lbm/s. The fuel mass flow rate at cruise is 2.27 lbm/s. The propulsion system is designed to handle the one engine out FAA requirement.

The propulsion system was divided into three sections: the inlet, the turbomachinery and the exhaust. Both inlet and exhaust air flows were modeled as adiabatic and compressible. A two-dimensional square inlet controls the velocity and pressure of the air into the engine core. Engine mounting is less complex for a rectangular inlet than for a circular one. The different mass flows associated with the range of flying conditions are accommodated by the use of a variable area ramp. A circular exhaust nozzle controls the velocity leaving the engine. The exhaust nozzle, like the inlet ramp, is variable to allow for the necessary exit velocities required at various flying conditions. Two convergent nozzles are employed when flying at subsonic speeds. Supersonic speeds require the use of a convergent-divergent nozzle. These configurations are shown in the operating mode diagrams (Figure 1, Appendix D)

A SNEMCA MCV 99 variable-cycle engine design was chosen for this aircraft. This variable-cycle engine has four operating modes: take-off, climb, subsonic cruise and supersonic cruise. The climb operating mode is also used for transonic acceleration. This cycle's use of premixing before combustion, staged burning, rich burn, quick quench, lean burn combustor and a variable area geometry reduces pollutant emissions into the atmosphere by fifty percent when compared to present cycles' emissions (Habrard[6]).

Part 1: Inlet Design

The purpose of the inlet is to bring free stream air to the required velocity of Mach 0.5 at the entrance to the compressor with a minimum total pressure loss. Since the aircraft will spend the majority of its flying time at cruise conditions of Mach 2.2 and an altitude of 55,000 feet, the inlet was designed for these conditions. A variable ramp will accommodate the adjustments needed for the other stages of flight (see Figure 2, Appendix D). A square inlet with a width of 3.66 feet and a capture area of 13.4 square feet was designed. Two oblique shocks and a normal shock slow the free stream air flow to Mach 0.5 at the compressor entrance. As suggested by Connors and Meyers[5], the ramp deflection angle are 9.9 and 10 degrees respectively.

To achieve a minimum total pressure loss at supersonic flight conditions, internal contraction was used to swallow the normal shock. The pressure recovery with internal contraction, allowing for some losses, is 0.91. The concept behind using internal contraction as opposed to other types of supersonic inlets is the variation in pressure recovery. By increasing the throat area the normal shock is swallowed further back allowing a higher percentage of pressure recovery. The design method for internal contraction began with the evaluation of pressure, area and temperature ratios of the two oblique shocks and the normal shock. The second step involves the swallowing of the shock by increasing the area of the throat. This is referred to as internal contraction. Area ratios with respect to throat area for isentropic flow were found for the Mach number before the normal shock (M_x) and for the Mach number after the normal shock (M_y). These area ratios were then divided to determine the internal contraction area ratio. The area ratio at Mach 0.5 at the face of the compressor and the isentropic area ratio for M_y were used to calculate the ratio of the compressor face area to the throat area.

The boundary layer is susceptible to separation during supersonic cruise. Separation occurs from the development of a severe pressure gradient. In order to prevent separation, a channel-type boundary layer diverter system on the ramp removes most of the boundary layer before the shocks. In this removal system the boundary-layer air is caught between a splitter plate and the fuselage. This caught air is then removed from the channel by diverting ramps angled at approximately thirty degrees.

There are blow in doors near the fan that only feed into the fan. Therefore, these doors only need to be opened from takeoff to high transonic flight conditions when the fan is in use.

Following the throat, a diffuser with a length of two feet, diffuses the flow from approximately Mach 0.72 after the normal shock to Mach 0.5 at the compressor entrance. A variable inlet ramp adjusts for the varying flight conditions from takeoff through transonic and to cruise at Mach 2.2. For takeoff conditions the ramp is retracted to lead the air directly to the compressor inlet without a contraction. This position allows greater airflow into the engine to achieve the necessary greater thrust level (see Figure 3 and the accompanying analysis in Appendix D).

Inlet drag is approximated from the inlet drag trends plot for a two-dimensional inlet (Figure 4, Appendix D). This plot was compiled from typical data previously collected (Raymer[1]). Inlet drag for different modes of flight for this design were estimated high due to the generality of the sources (Table 5). The maximum drag occurs at approximately Mach 1.3.

Table 5: Inlet Drag Estimates

Mach Number	D/q/A	D (lbs.)
2.20	0.10	911
1.30	0.23	1733
0.95	0.18	713
0.10	0.02	3.8

D = Drag q = Dynamic Pressure A = Capture Area

Part 2: Exhaust

The exhaust nozzle provides back pressure control for the engine and an acceleration device converting gas potential energy into kinetic energy. The throat area is the controlling factor. Since the pressure loss is less for a circular shape, a circular nozzle was chosen instead of a rectangular shape. The circular nozzle assembly also weighs less and is less complex compared to a two dimensional nozzle (Raymer[1]).

A variable-area exhaust nozzle is utilized to accommodate the varying flight conditions. Two convergent nozzles are utilized during subsonic flight, one for the fan and one for the core. A convergent-divergent nozzle is used during supersonic flight. During supersonic cruise at Mach 2.2, the nozzle increases the velocity of the mass flow from approximately Mach 0.5 to Mach 2.8. Since the ratio of specific heat decreases through the engine cycle, an average of exit areas calculated with different specific heat ratios (1.3-1.4) was used. The calculated exit area was 26.9 square feet with a throat area of 6.8 square feet.

Exhaust nozzle analysis involves the use of two dimensionless coefficients, the discharge coefficient and the velocity coefficient. The discharge coefficient represents the difference between ideal mass flow and actual mass flow. The velocity coefficient represents the frictional losses in the boundary layer of the nozzle. From these coefficients the angle geometry of the nozzle was determined. See Figure 5 and the accompanying analysis in Appendix D for the aforementioned information. The primary half angle is ten degrees and the secondary half angle is fifteen degrees.

Part 3: Turbomachinary

Selection of the engine to power the aircraft was constrained by the need to have good fuel efficiency at several flight speeds and altitudes while keeping noise low on takeoff. Single cycle engines (plain turbojets and turbofans) were considered but found lacking in one or more areas. Turbojets give good supersonic performance due to their high exhaust velocity, but are too noisy for civilian use. Turbofans have a lower exhaust velocity and are therefore quieter, but give poor supersonic performance. As a result, a dual-cycle engine that combines the advantages of both turbojet and turbofan was chosen. The design is basically a scaled-down SNECMA MCV99 dual-cycle engine (see Figure 9, Appendix D). At cruise, this gas turbine acts like a normal turbojet, but at lower speeds a fan mounted around the narrow core section is started to give greater efficiency by reducing the exhaust velocity (and therefore noise as well). This concentrically mounted fan is driven by its own combustor and turbine fed by bleed air from the core engine. Cruise efficiency is improved over a turbofan engine because the low-velocity fan, which does not give much thrust at supersonic speeds, is shut down when it is not needed. This engine is also fuel efficient because it does not require an afterburner in any part of its operational envelope.

Originally three engines were to be used for safety in case of engine failure. The decision to use two engines was made for two main reasons. The first was that two engines would give a lower overall weight. The second was the problems involved in mounting an engine to the centerline of the aircraft. These included boundary layer removal, foreign object damage, and accessibility.

Selection of engine thrust was constrained by the cruise condition. At 55,000 ft. in level flight each engine had to deliver 7,180 lb. of thrust. A sea level static thrust of 25,000 lb. was then fixed representing an 8.1% improvement over a sample engine's altitude performance provided by (Raymer[1]). Analysis of Federal Airworthiness Regulations found that the most demanding part of a one engine out takeoff for this aircraft required only approximately 20,000 lb. of thrust per engine at sea level. This is quite less than the engine size needed for cruise. The MCV 99 engine has a thrust of 49,455 lb, and was therefore scaled down for use in this design using a modified "rubber engine" process presented by Raymer[1]. The resulting engine dimensions are in Table 6.

Table 6: Engine Dimensions

Length	12 ft
Compressor Diameter	3.41 ft
Fan Shroud Diameter	4.13 ft
Fan Hub Diameter	2.21 ft

The hub ratio for the fan was found to be 0.535, greater than the 0.5 minimum given by SNECMA as necessary for the compressor (Habrard[6]). The weight of the engine was set at 5,000 lb. based on a historical thrust to weight ratio of 5 for recent supersonic engines (Aviation Week & Space Technology 7,8&9). Blow-in doors are needed when the fan is operating to provide correct airflow, and these have been designed as doors that open out 0.65 ft. on either side of the engine nacelle to give an additional 6.41 square feet of capture area for the fan. These doors close and the fan shuts down as high supersonic speeds are reached.

The figures given for this engine by SNECMA are shown in Table 7.

Table 7: SNECMA Engine Characteristics

	Takeoff (sea level)	M=2.2 Cruise	M=1.3 Climb	M=0.95 Cruise
SFC (lbf/lbm/hr)	0.638	1.138	1.000	0.873
Pressure Ratio	19.2	17	18.8	19
Bypass Ratio	1.0	0	1.04	0.994
Fan Pressure Ratio	2.5	n/a	2.46	2.48
Bleed Ratio for Fan	0.35	0	0.36	0.34

It is assumed that these figures can be held constant even for a lower thrust engine.

At cruise condition, a common air/ fuel ratio of 35/1 was assumed to give an air mass flowrate of 79.45 lbm/s. The fuel flowrate at the same condition is 2.27 lbm/s, and the exit velocity is 4237 ft/s.

Thermodynamic analysis (Cengel, Boles[10]) of the engine gives an ideal Brayton cycle thermal efficiency of 55.49% at cruise. For a maximum constant turbine inlet temperature of 3060 R (Habrard[6]), and assuming an 85% efficient compressor and a 90% efficient turbine, the actual thermal efficiency goes down to 45.9%.

A rich burn/ quick quench/ lean burn combustor has been chosen as a promising solution (Habrard[6], Bahr[11]) to avoid creating large amounts of the pollutant nitrogen oxide. NOx is produced in the largest amounts when combustion is at stoichiometric ratios. To avoid this, the first stage of the combustor is run at over-stoichiometric levels of fuel. This rich mixture is then mixed with air quickly, and the combustion continued at less than stoichiometric levels. The ratios that cause NOx production are then avoided completely during the combustion process.

Section 3: Structure

The structural design team was responsible for the following tasks: 1) estimating an initial takeoff gross weight (TOGW) and the initial sizing of the aircraft; 2) the final weight estimation; 3) landing gear and 4) a finite element analysis of the aircraft.

Part 1: Initial TOGW and Sizing

The initial TOGW of the aircraft was determined by a statistical comparison of current aircraft designs based on the following specifications listed in Table 8.

Table 8: Design Specifications

Range	5,000 nm
Passengers & Crew	9
Passenger & Crew Weight	1,800 lbs.
Payload	700 lbs.

In the initial study, the effects of varying the range, crew and passenger size, cruise altitude, specific fuel consumption, lift to drag ratio and the weight equation constants (either jet transport or jet fighter) were examined with respect to the TOGW. See Tables 1 through 8 in Appendix E for the results of this initial trade study. The estimated TOGW varied from 103,000 lbs. to 117,000 lbs. in this study so a target weight of 107,000 lbs. was set.

Next, the fuselage length was found from the estimated TOGW and a statistical relationship based on current aircraft designs (Raymer[1]). Using this method a fuselage length of 107.0 ft was calculated. With the length set, the diameter of the fuselage was determined to be 7 feet 8 inches based on a supersonic fineness ratio of fourteen (Raymer[1]). The fineness ratio is the ratio between length and diameter of the fuselage which minimizes wave drag. The inner diameter of the fuselage was set to 7 feet after allowing for a 4 inch fuselage thickness. These calculations are presented in Appendix E.

With the initial sizing complete, a cabin layout was generated using the values for economy and high density passenger compartments presented by Raymer[1] (see Figure 4 on page 4). The total passenger cabin length is 15 foot 4 inches. A recessed floor was used to allow for a 6 foot 2 inch high aisle with a width of 18 inches. The passenger compartment seated six people plus a jump seat for the flight attendant. Three seats were placed on each side of the aisle and had a width of 18 inches and a seat pitch of 36 inches. The headroom was 5 feet 10 inches. The cabin also included a 40" X 40" lavatory and a small galley.

Part 2: Final Weight Estimation

After the initial analysis from both the aerodynamics and propulsion groups was completed, a more accurate weight estimate for the aircraft was required. Five different weight approximation methods (Raymer[1], Nicolai[12]) were tested on the Concorde to determine their accuracy for supersonic aircraft. The Concorde was chosen for the comparison because it has a comparable speed of Mach 2.2 but is almost twice the size of our initial TOGW estimate. In each case, a discrepancy of 10% or more was found between the estimated empty weight and the actual Concorde empty weight. To compensate for the large errors in using any of the methods individually, the method used was a combination of the weight estimation equations that best approximated the individual components of the aircraft. The difference between the estimated empty weight and the actual empty weight using the combined method was 3.7%. Applying this technique to our design and using the 3.7% difference as a correction factor, the estimated empty weight of Tesseract was found to be 42,878 lbs. Based on a composite utilization by weight of 55% (Nicolai[12]), the final empty weight of the design was estimated at 37,778 lbs. With the weight of each of the individual components of the aircraft known, the empty weight center of gravity was calculated to be 73.1 feet from the nose. These calculations are presented in Tables 1, 2 and 3 in Appendix F.

Part 3: Landing Gear

Next, the main landing gear is located 80 feet from the nose of the aircraft and is 16 feet off the centerline of the fuselage. It will be positioned on the wing next to the engines. They will fold in towards the fuselage and most likely will need a pod to house part of the gear which does not fit in the wing. The total length of the main landing gear is 20 feet preventing the tail of the airplane from dragging on the ground during takeoff (see Calculations in Appendix G). The main gear was designed using an estimate of the forward center of gravity (CG), aft CG and aerodynamic center. The values used were 76 feet, 80 feet, and 86 feet respectively, from the nose of the airplane. The type of strut used for all the gear is an oleo shock-strut. The main landing gear is comprised of two struts with each strut having two sets of tires for a total of eight tires. The tire size is 37 inches in diameter and 12 inches wide. The maximum static load on each main gear strut was calculated to be 48,600 lbs.

The nose gear is located 30 feet from the nose of the airplane. It is located on the fuselage and will fold forward into the fuselage to allow the gear to free-fall down in case of a failure in the extension system. The nose gear will be slightly longer than the main gear. It will also have an oleo shock-strut and have two tires. The size of the tires are 22 inches in diameter and 8 inches wide. The maximum static load calculated for the nose gear was 17,500 lbs. which is 18% of the maximum static load for the main gear. This percentage is higher then the suggested 14% or less. The minimum static nose gear load is 9,700 lbs. and the maximum braking load is 12,000 lbs. All the landing gear calculations are based on information presented by Raymer[1] and Currey[14] and can be found in Appendix G.

Part 4: Finite Element Analysis

A finite element analysis was completed on the fuselage and wing using the software "GIFTS[13]". Three sections of the aircraft were modeled during the analysis. These sections included the cabin section, the fuselage wing root section and the internal wing structure. Aluminum Alloy 2014-T6 was used for all of the structural members used in the analysis. Due to the fact that this analysis coincided with the aerodynamic and propulsion studies, the initial numbers used in the finite element analysis do not reflect the most recent changes in the design.

The cabin section was idealized with 96 nodes and 160 elements. The bulkhead and stringers were idealized as hollow square cross sections which were evenly spaced in a circular configuration. The maximum bending moment the airplane would experience and the shear load were calculated using a maximum load factor of 2.5. The internal cabin pressure was assumed to be small compared to the force of the bending moment and therefore was ignored.

The calculated stress due to pressure exerted on the cabin was found to be 8,500 psi. A value of 31,000 psi. was obtained for the total stress of the airplane at lift-off based on the maximum moment and shear stresses. Therefore, the total stress that the plane would have to withstand would be 40,000 psi.

The second test section, where the wing attaches to the fuselage, was modeled in a more simplistic manner. It had eight booms in a hexagonal shape with "I" beams as internal support. Furthermore, "I" beams were used to represent the wing. The maximum bending moment and shear forces were also applied to this section.

The finite element analysis of the wing was completed by modeling the spars as "I" beams. The "I" beams varied in size from the largest at the root (2'-0") to the smallest at the outermost rib (0'-6") (see Figure 1, Appendix H). The ribs were idealized as 3/16" flat plates which also ranged in height through the structure (see Figure 2, Appendix H). Over 150 elements were used for the interior of the wing to improve the accuracy of the results.

The following design specifications shown in Table 9 were used in the analysis.

Table 9: Finite Element Design Specifications

Wing Loading	100 psf
Aspect Ratio	1.7
Wing Span	47.31 ft
Center Line Chord	57.73 ft
Maximum Load Factor	2.5

Using GIFTS, the maximum deflection for the interior of the wing was six inches. Figures 4 and 5 in Appendix H shows the wing tip deflection and the wing outer chord warping. At the root, the maximum normal stress for the spars ranged from 1.24xE6 psf to 1.61xE6 psf. The wing also showed warping at the outer trailing edge with the distributed 100 lb/ft^2 load.

The skin of the wing was also examined for our wing configuration but was not included in the report because the software used would not allow the marriage of the internal structure and the skin to be joined in one complete structure. This inability of the software resulted in the skin analysis to be inconclusive in the overall design of the wing.

Conclusion

The initial iteration of the Tesseract Supersonic Business Transport was a success. However, to complete the conceptual design of this aircraft a final iteration of the data is required to mesh the simultaneous work of the three design groups. For Example, the initial takeoff gross weight estimates may have been to high. Initially, the weight of the aircraft structure was estimated between 40% to 50% of the takeoff gross weight. During the final weight estimation, based on a composite utilization by weight of 55%, the aircraft structural weight was estimated at 35% of the takeoff gross weight. Furthermore, the specific fuel consumption for the SNEMCA MCV 99 variable-cycle engine was lower than the 1.3/hr expected resulting in further reduction of the required takeoff gross weight for the aircraft. Also, the aerodynamic analysis for low speed static stability needs to be evaluated and an appropriate control system needs to be employed. Even though the conceptual design of this aircraft was not completed to incorporate the latest changes of each of the design groups, this project has developed the bases for a future supersonic business transport design.

Acknowledgements

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Mr. Art Glassman of NASA Lewis

NASA/USRA

Summary report prepared by James Fellenstein and Peter Struk

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Appendix A (Aerodynamics)
Figures

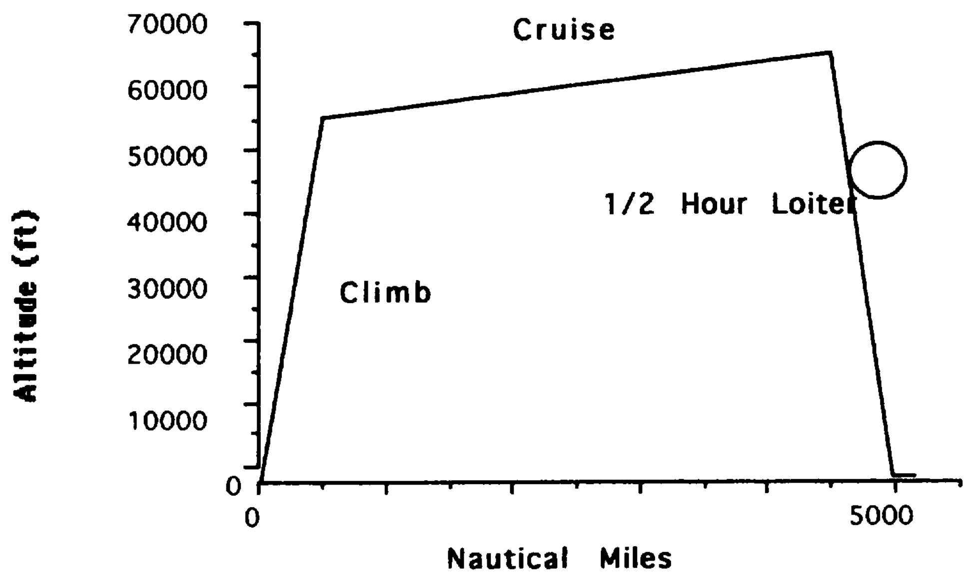


Figure 1: Tesseract's Mission Profile

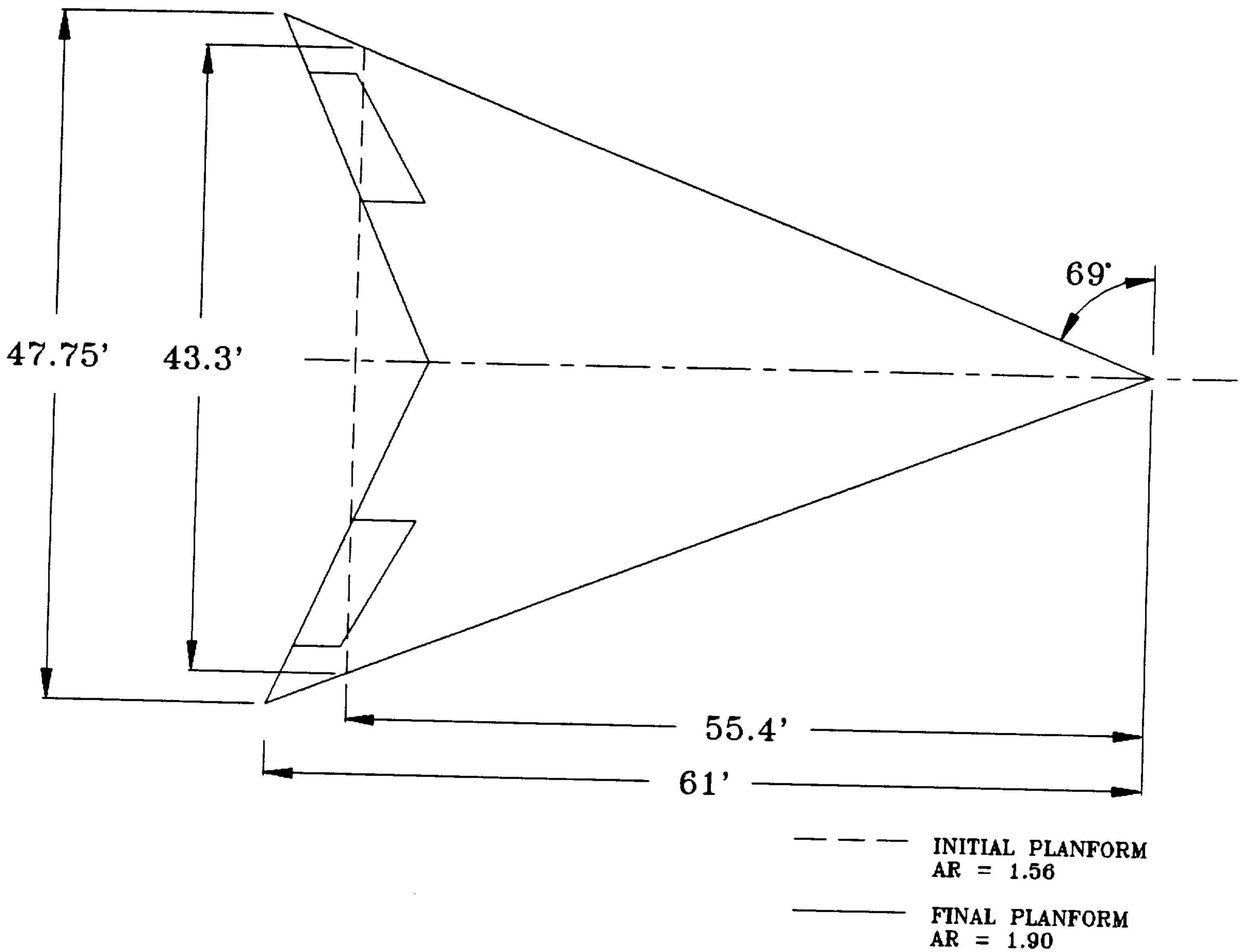
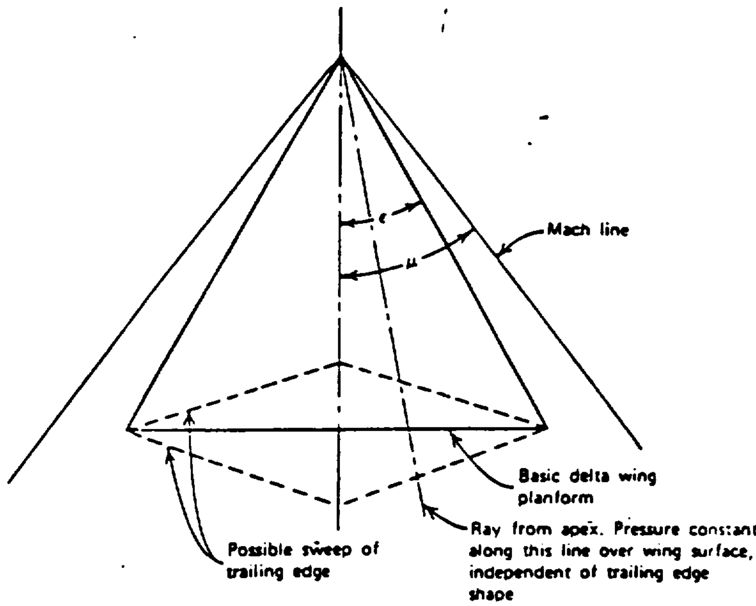
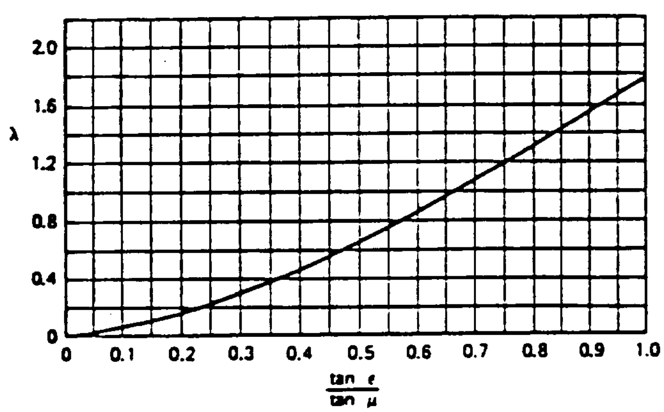


Figure 2: Wing Planform



Supersonic delta wing.



Variation of λ with tan ϵ/tan μ.

Figure 3: Lambda Function (Brown, McCormick[2])

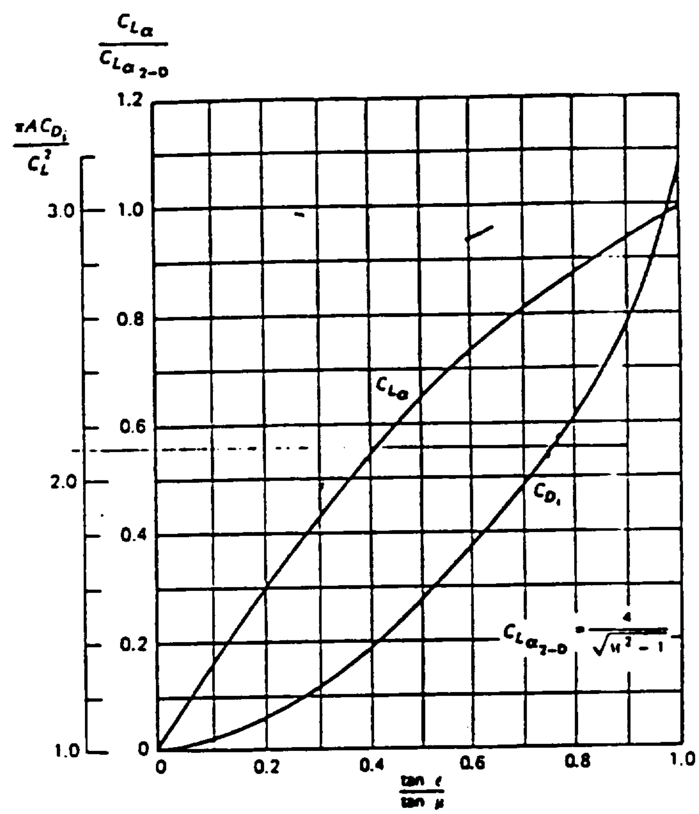


Figure 4: Lift Curve Slope and Induced Drag Coefficient for Delta Wings (Brown, McCormick[2])

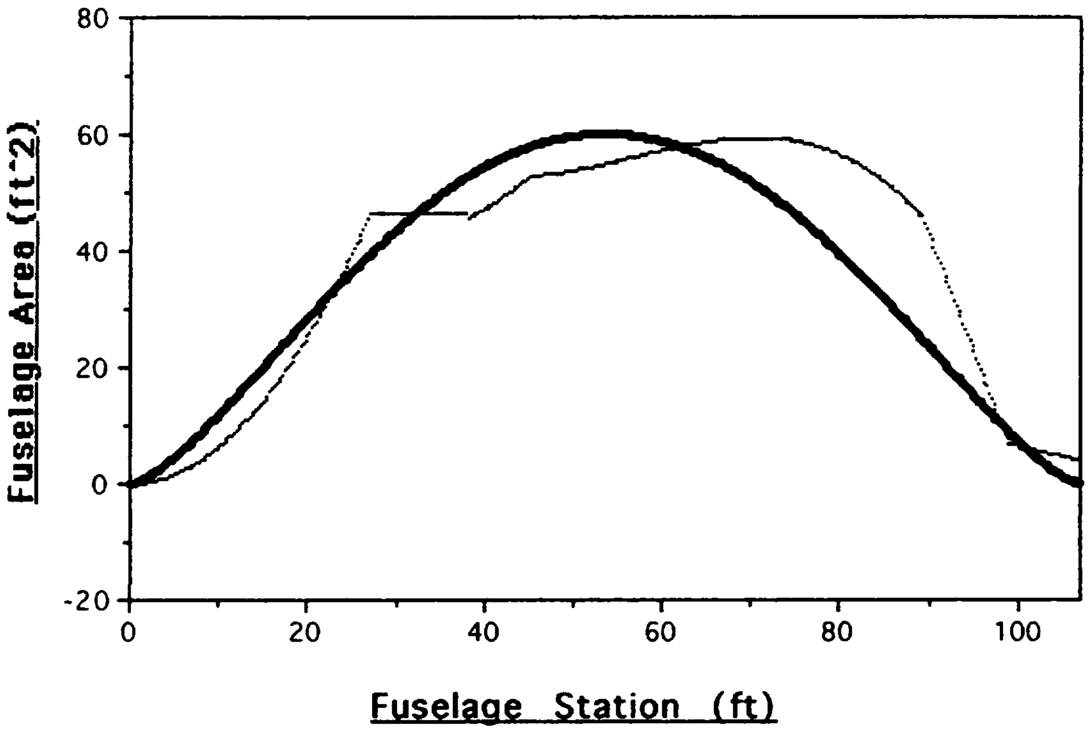


Figure 5: Area Distribution

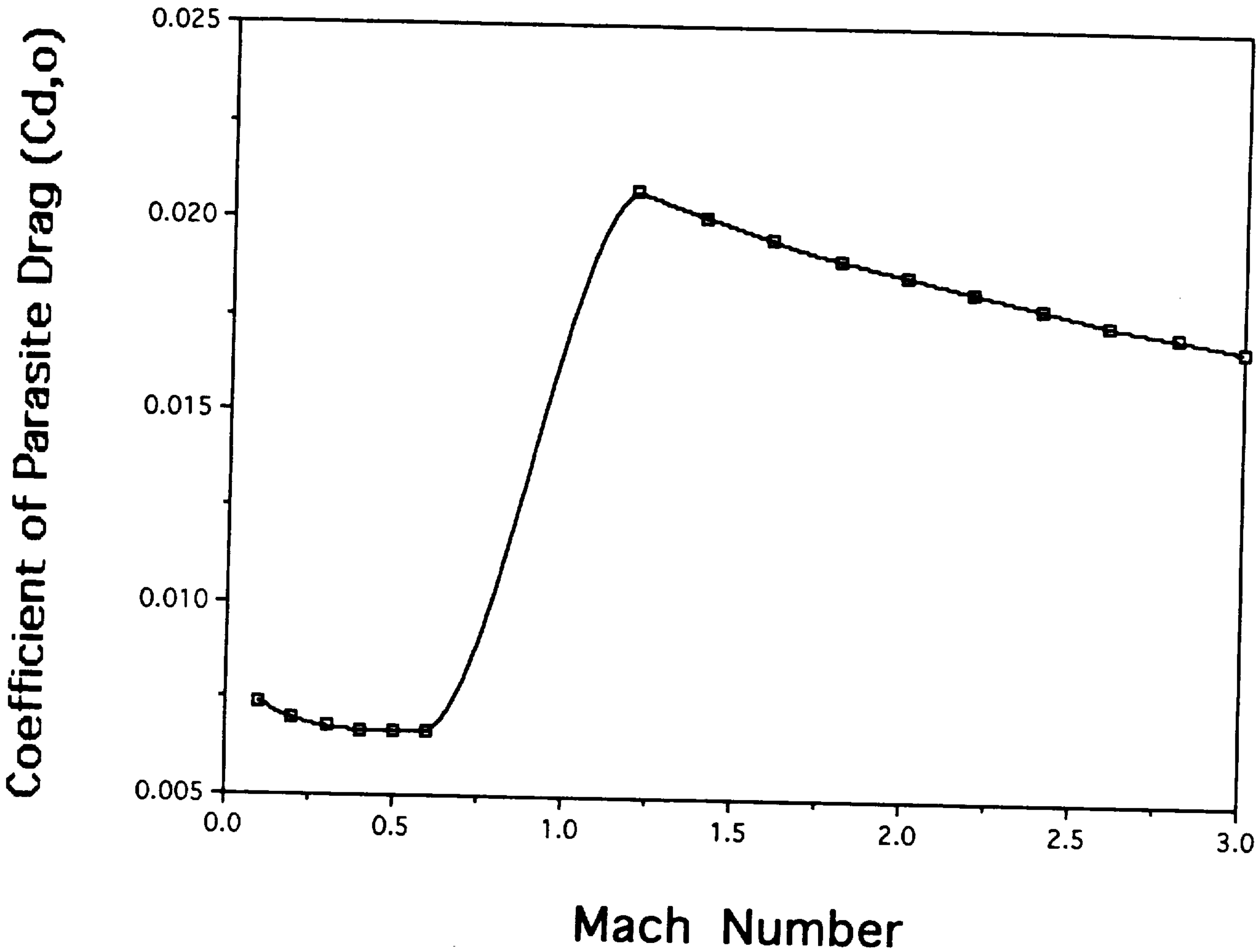


Figure 6: Drag Estimation for Various Flight Regimes

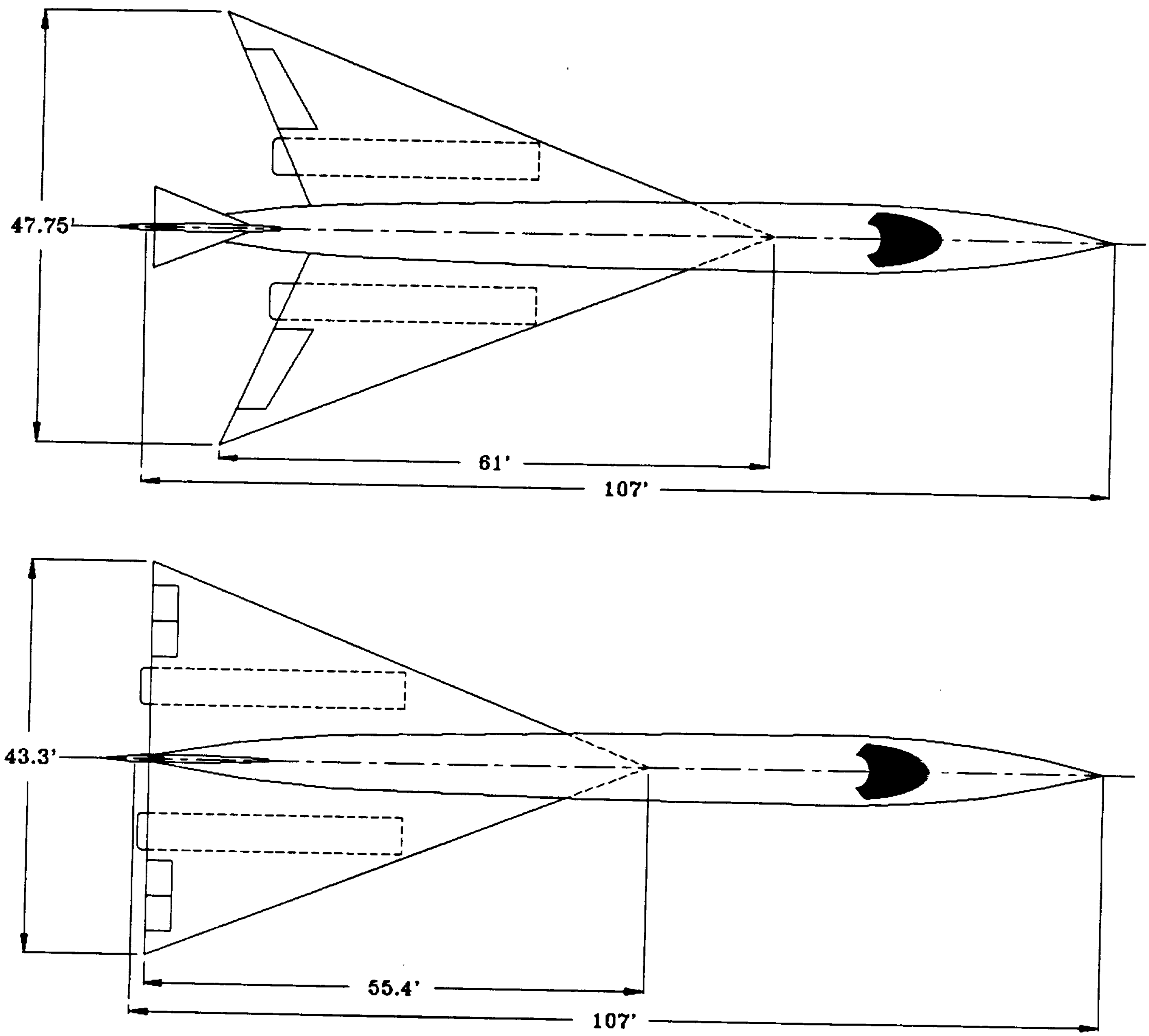
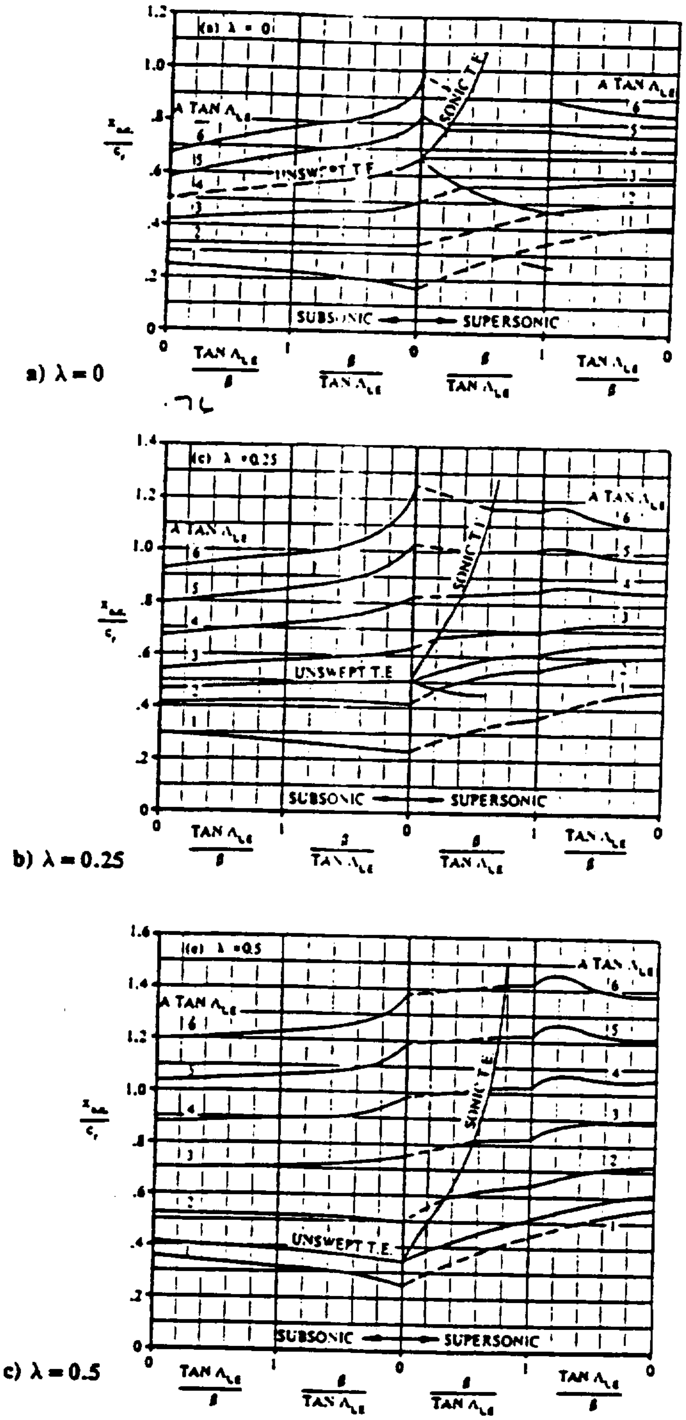


Figure 7: Tail versus Tailless



<u>**Figure 8: Aerodynamic Center (Raymer[1])**</u>

Appendix B (Aerodynamics)
Drag Analysis

Drag Calculations

Parasite Drag
Component Buildup Method (Raymer[1])

Aerodynamic Data

Wing Reference Area	1200 ft^2
Standard Density (55000 ft)	2.9 e-4 slug / ft^3
Standard Temperature (55000 ft)	389.9 °R
Speed of Sound (55000 ft)	967.9 ft /s
Velocity (Mach 2.2)	2129.5 ft/s
Dynamic Viscosity	3.0 e-7 slug/ft/s

Skin Friction

Component	Char. Length (ft)	Reynolds Number	Coefficient of Friction	Wetted Area (sq. ft)	# of Compo-nents	Coefficient of Drag
Wing	33.6	6.9e+7	0.00159	1748.9	1	0.0023
Fuselage	107.0	2.2e+8	0.00135	1570.0	1	0.0018
Nacelle	24.0	4.9e+7	0.00167	240.0	2	0.0007
Vert. Tail	10.0	2.1e+7	0.00190	100.0	1	0.0002
Horz. Tail	10.0	2.1e+7	0.00190	100.0	1	0.0002
TOTALS						0.0051

Wave Drag

Maximum Cross Sectional Area	59.3 ft^2
Characteristic Length	89.0 ft
SEARS-HAACK WAVE DRAG	0.0052
Correction Factor (Ewd)	1.4
Leading Edge Sweep	68.7°
CORRECTED WAVE DRAG	0.0068

Induced Drag
(See Figures 4 & 5, Appendix A)

Induced Drag	0.0058

Totals Drag for Cruise

Coefficient of Drag (Cruise)	0.0182

Subsonic & Transonic Drag Data
(from Kern[4])

Mach	Altitude (ft)	Cd,0
0.1	0	0.0074
0.2	0	0.0070
0.3	0	0.0068
0.4	0	0.0067
0.5	0	0.0067
0.6	0	0.0067
0.7	35000	0.0081
0.8	35000	0.0082
0.85	35000	0.0082
0.9	35000	0.0083
0.95	35000	0.0084
0.95	55000	0.0096
1.0	55000	0.0097
1.1	55000	0.0102
1.2	55000	0.0109
1.3	55000	0.0117

Appendix C (Aerodynamics)
Stability Analysis

Nomenclature

$C_{M,CG}$	:	moment coefficient about center of gravity
C_{LW}	:	lift coefficient of wing
h_R	:	location of center of gravity as a fraction of the root chord
h_{ACR}	:	location of aerodynamic center as a fraction of the root chord
$C_{M,ACWB}$	:	moment coefficient about aerodynamic center
$C_{MW\partial f}$	:	moment coefficient per radian of flap deflection
∂f	:	flap deflection in radians
T	:	thrust
Z_T	:	horizontal distance from engines to the center of gravity
q	:	dynamic pressure
S_W	:	wing reference area
C_R	:	root chord length
C	:	mean aerodynamic chord
n_H	:	ratio of dynamic pressure at tail to that at wing
V_H	:	tail volume coefficient
C_{LH}	:	lift coefficient of tail
S_H	:	tail area
L_T	:	lift of tail
l_T	:	length of tail
i_W	:	wing incidence angle
i_H	:	tail incidence angle
$\partial e / \partial a$	:	rate of change of downwash angle with respect to angle of attack
a_{OLH}	:	zero lift angle of attack

Tail Determination Required to Maintain Static Stability at Cruise

Equation 1

$C_{M,CG} = C_{LW} (h_R - h_{ACR}) + C_{M,ACWB} + C_{MW\partial f} \partial f + T Z_T / q S_W C_R - n_H V_H C_{LH}$

$C_{M,CG} = C_{LW} (h_R - h_{ACR}) + C_{M,ACWB} + C_{MW\partial f} \partial f + T Z_T / q S_W C_R - l_H S_H C_{LH} / S_W C_R$

$0 = C_{LW} (.70 - .77) + 0 + 0 + 0.001 - 0.54 S_H / S_W C_{LH}$

Equation 2

$C_{Lw} + S_H / S_W C_{LH} = 0.128$

Iteration of Equations 1 & 2

S_W	S_H	C_{LW}	C_{LH}	C_{LTOTAL}
1200	100	0.1276	0.004	0.128
1200	50	0.1276	0.009	0.128

Thus a tail is not essentially needed during cruise!

Neutral Point Analysis at Cruise

$$C_{M,CG} = C_{LW} (h_R - h_{ACR}) + C_{M,ACWB} + C_{MW\partial f} \, \partial f + T Z_T / q S_W C_R - V_H C_{LH}$$

$$C_{LH} = C_{LHa} [(a + i_W)(1 - \partial e / \partial a) + i_H - i_W - a_{OHL}] V_H$$

Rewritten in terms of angle of attack (a):

$$C_{M,CG} = C_{LWa} (a + i_W) (h_R - h_{ACR}) + C_{M,ACWB} + C_{MW\partial f} \, \partial f + T Z_T / q S_W C_R - V_H C_{LH}$$

Differentiate with respect to angle of attack

$$\partial C_{M,CG} / \partial a = C_{LWa} (h_R - h_{ACR}) - C_{LHa} (1 - \partial e / \partial a) V_H$$

Find Neutral Point

$$0 = h_R - h_{ACR} - C_{LHa} / C_{LWa} (1 - \partial e / \partial a) V_H$$

$$h_R = h_{ACR} + C_{LHa} / C_{LWa} (1 - \partial e / \partial a) V_H$$

$$h_R = 0.77 + (\approx 1) (.037) (1 - .478)$$

Tail Volume Coefficient

$$V_H = l_T S_H / C S_W = .037$$

Downwash at Supersonic Speeds

(Raymer page 420)

$$\partial e / \partial a = 0.478$$

Static Stability Analysis at Cruise

$$C_{M,CG} = C_{LWa} (a + i_W) (h_R - h_{ACR}) + C_{M,ACWB} + C_{MW\partial f}\, \partial f + T\, Z_T / q\, S_W\, C_R - V_H\, C_{LH}$$

at zero lift

$$C_{M,CG} = T\, Z_T / q\, S_W\, C_R - V_H\, C_{LHa} [i_H - i_W - a_{OHL}]$$

The tail incidence must be designed so that the above bracketed term is positive.

$$\partial C_{M,CG} / \partial a = C_{LWa} (h_R - h_{ACR}) - C_{LHa} / C_{LWa} (1 - \partial e / \partial a)\, V_H$$

$$\partial C_{M,CG} / \partial a = 1.76\, (.70 - .77) - (\approx 1)\, (1 - .478)\, (.037)$$

$$\partial C_{M,CG} / \partial a = -0.104$$

Appendix D (Propulsion)
Figures

SNECMA MCV 99
VARIABLE-CYCLE ENGINE

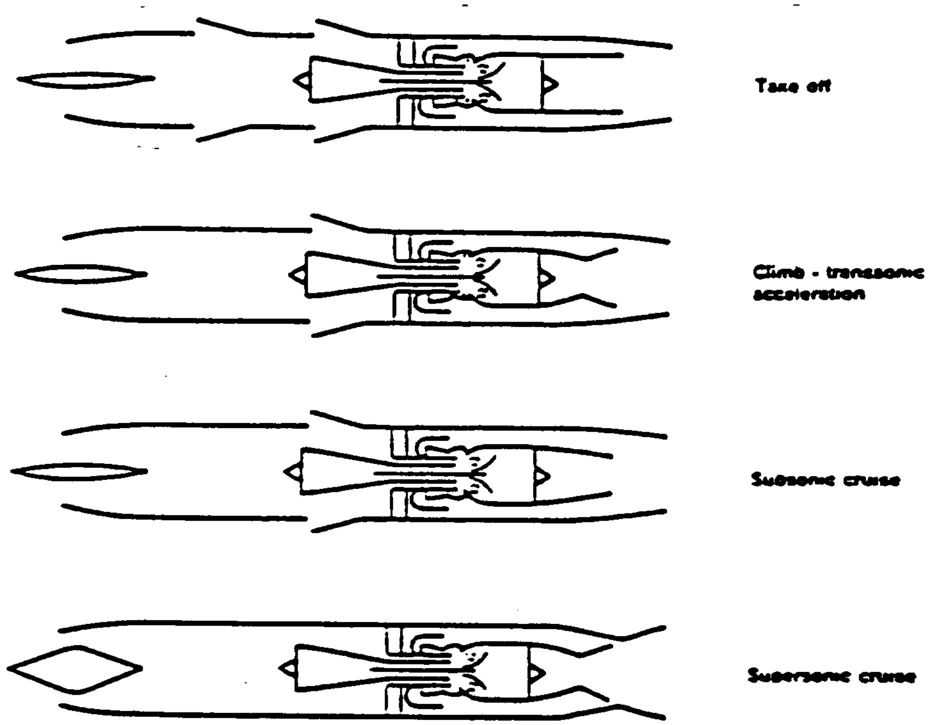


Figure 1: MCV 99 Operating Mode

Reference: Habard[6]

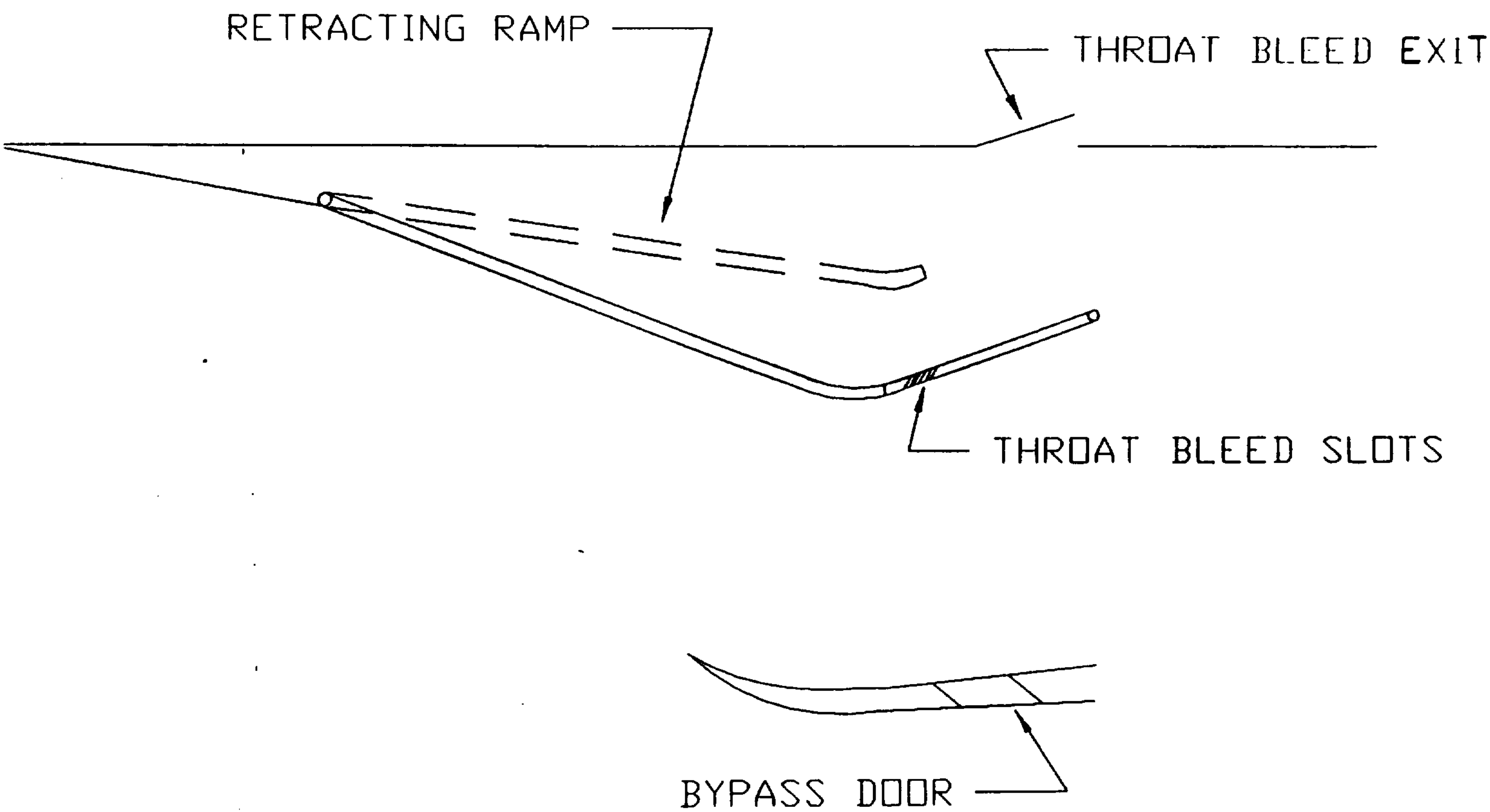
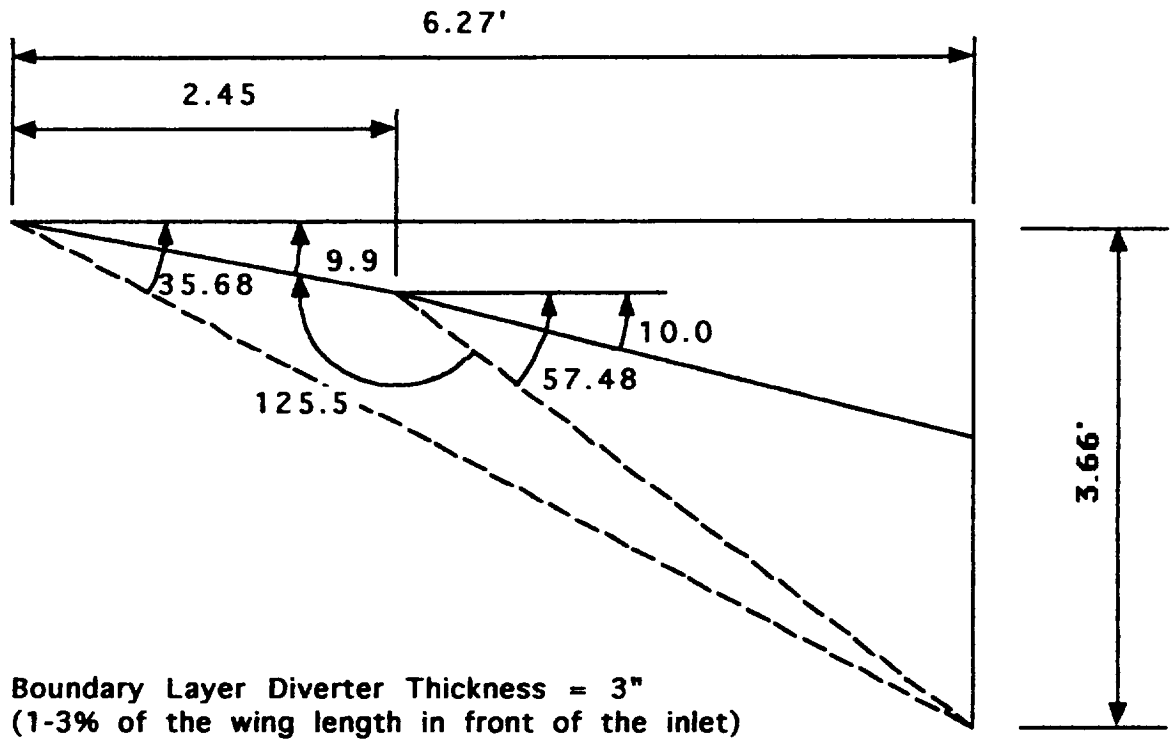


Figure 2: Flow Control Devices



<u>Figure 3: Inlet Geometry</u>

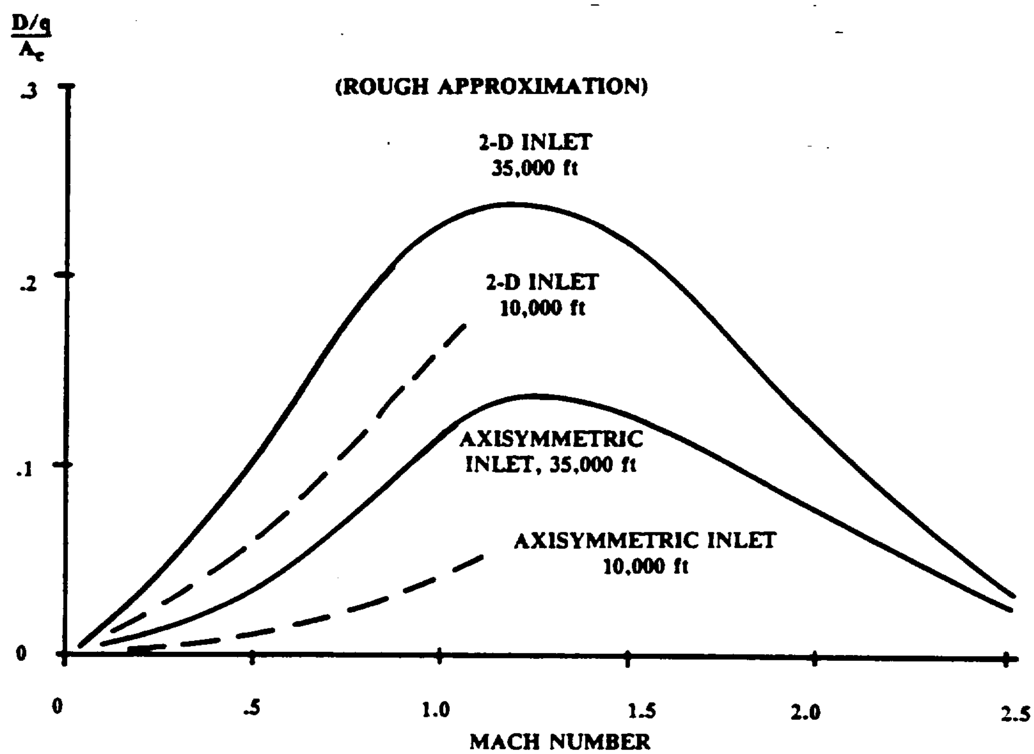


Figure 4: Inlet Drag Trends

Reference: Raymer[1]

Inlet (Method: Internal Contraction)

First Oblique Shock
M1 = 2.2 Delta 1 = 9.9°
M2 = 1.8266
Po2/Po1 = 0.98216
Sigma = 35.685°

Second Oblique Shock
M2 = 1.8266 Delta 2 = 11.0°
M3 = 1.4379
Po3/Po2 = 0.98236
Sigma = 44.576°

Normal Shock
M3 = 1.4379
M4 = 0.72428
Po4/Po3 = 0.94823

Isentropic Flow
M4i = 0.72428
A/A* = 1.0778
P/Po = 0.70526

Isentropic Flow
M3i = 1.4379
A/A* = 1.1366

Internal Contraction
IC = 1.1366/1.0778 = 1.0546
M = 1.2711
My = 0.8105
Poy/Pox = 0.9845

Overall
Po4/Po1 = 0.9499

Overall With Losses
Po4/Po1 = 0.9119

Angles Suggested by Connors and Meyers[5]

Inlet (Method: External)

First Oblique Shock
M1 = 2.2 Delta 1 = 11.8°
M2 = 1.7532
Po2/Po1 = 0.971
Sigma = 37.655°

Second Oblique Shock
M2 = 1.7532 Delta 2 = 13.2°
M3 = 1.274
Po2/Po1 = 0.97092
Sigma = 50.067°

Normal Shock
M3 = 1.274
M4 = 0.7995
Po4/Po3 = 0.98361

Po4/Po1 = 0.927

First Oblique Shock
M1 = 2.2 Delta 1 = 10.0°
M2 = 1.8228
Po2/Po1 = 0.98165
Sigma = 35.786°

Second Oblique Shock
M2 = 1.8228 Delta 2 = 10.0°
M3 = 1,4719
Po2/Po1 = 0.98663
Sigma = 43.431°

Normal Shock
M3 = 1.4719
M4 = 0.71128
Po4/Po3 = 0.93845

Po4/Po1 = 0.9089

Inlet Area

Compressor Area = 9.18 ft^2

$M = 0.5$ $\quad$ $A/A^* = 1.3398$
$P/Po = 0.84302$
$e/eo = 0.88517$
$T/To = 0.95238$

$$\frac{\text{Compressor Area}}{\text{Throat Area}} = \frac{1.3398}{1.0778} = 1.24309$$

$$\text{Throat Area} = 7.385\ ft^2$$

$$\frac{\text{Area of Tip}}{\text{Area of Throat}} = 1.0546$$

$$\text{Area of Tip} = 7.788\ ft^2$$

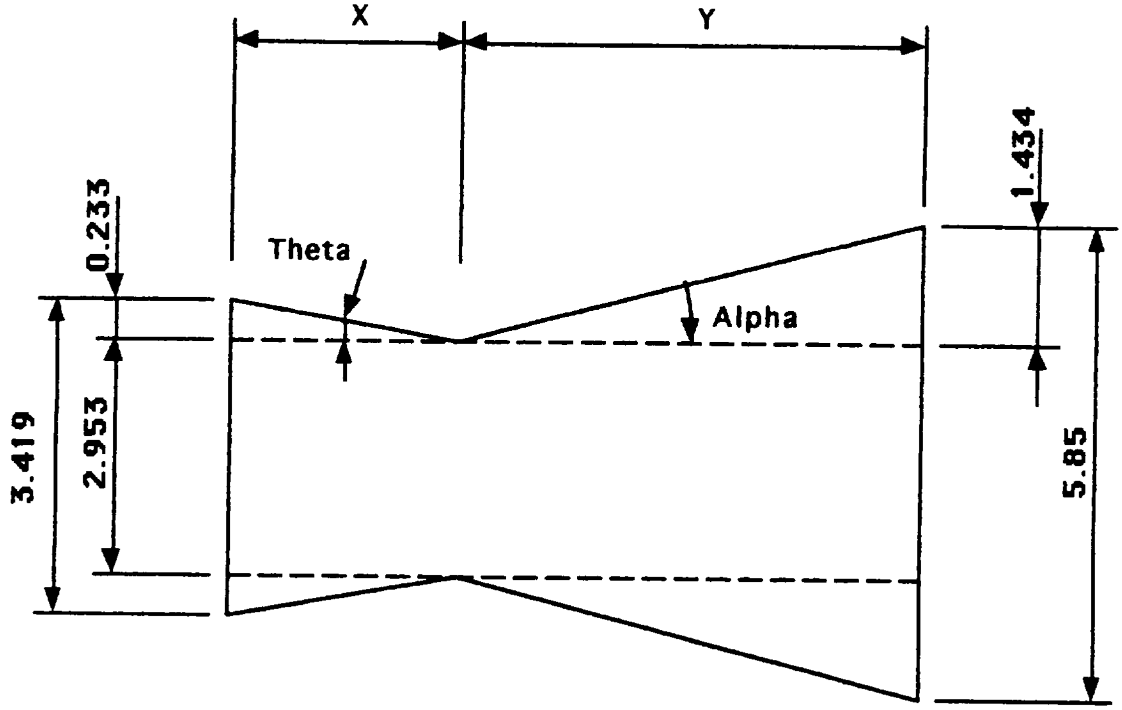


Figure 5: Nozzle Geometry

Nozzle Geometry

At Turbine

Mt = 0.5

At/A* = 1.3398

At Exit

Me = 2.8953

Ae/A* = 4.208

A(turbine exit) = At= 9.18 ft^2

D(turbine exit) = Dt = 3.419 ft

$A^* = A(\text{throat}) = \frac{At}{1.3398} = 6.85\ ft^2$ (D* = 2.953 ft)

Ae = 4.208 (A*) = 28.82 ft^2

Theta = Primary Half Angle

$\text{Tan (Theta)} = \frac{0.233}{X}$

See Figures 6, 7 & 8

Theta = 5° X = 2.66 ft

Theta = 10° X = 1.32 ft

Discharge Coefficient (Cd = A8e/A8) = 0.98

(A8 = A(throat)) New A(throat) = 6.99 ft^2

A(exit)/A(throat) = 4.12 a = 15° Cv = 0.998 Y = 5.35 ft

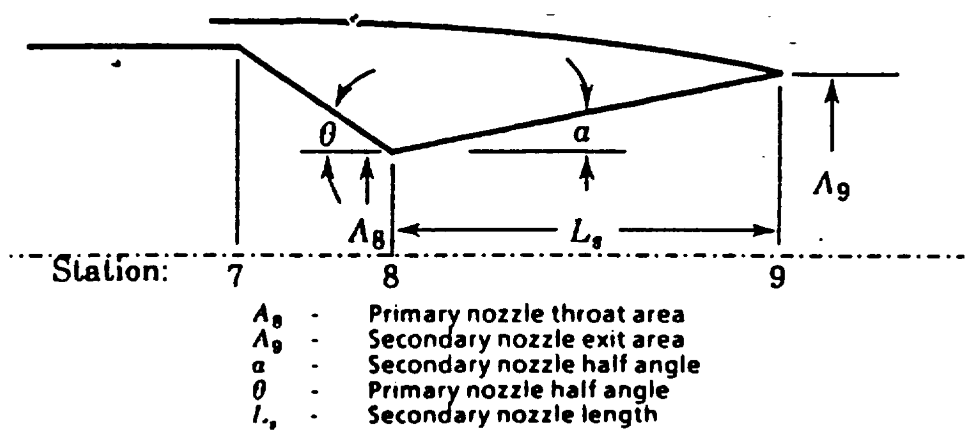


Figure 6: Nozzle Geometric Parameters

Reference: Mattingly

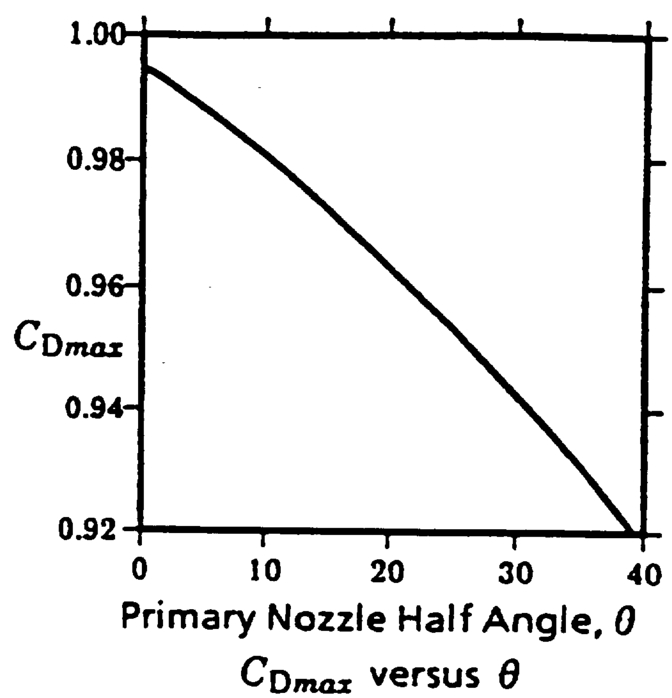


Figure 7: Nozzle Discharge Coefficient

Reference: Mattingly

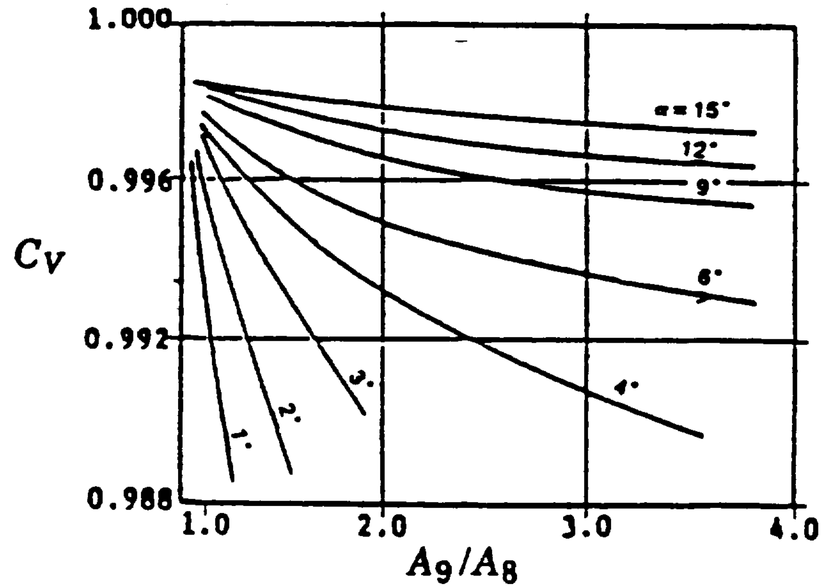


Figure 8: Convergent - Divergent Nozzle Velocity Coefficient

Reference: Mattingly

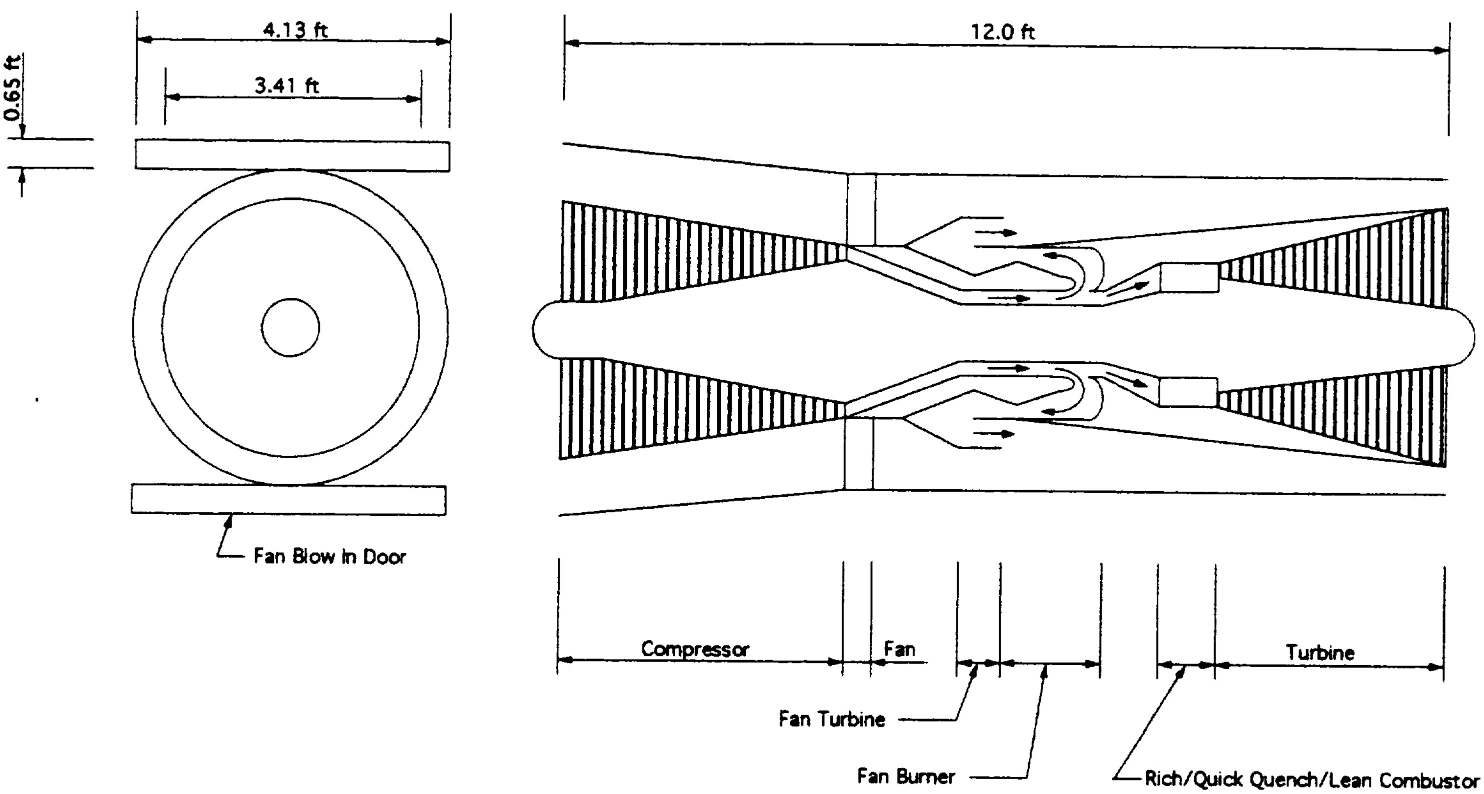


Figure 9: Powerplant Configuration

Appendix E (Structure)
Initial Sizing

Table 1: Conceptual Sizing for SST (Standard Conditions)

Design Criteria

Crew Members= 9	Wc=	1800	lbs.	
Payload	Wp=	700	lbs.	
Range	R=	5000	nm	
	R=	30380000	ft	
Speed of Sound	a=	994.8		Alt 30000 ft
Cruise Velocity	M=	2.2	M	
	V=	2188.56	ft/s	
Specific Fuel Consumption	C=	1.3	1/hr	Turbojet
	C=	0.00036111	1/s	
Endurance	E=	0.3333	hr	
	E=	1199.88	s	
Lift/Drag	L/D=	8		

Mission Segment Weight Fractions

Total Weight	W0=	103548	lbs.	
1) Warmup and Takeoff	W1/W0=	0.97		Table 3.2
2) Climb	W2/W1=	0.985		Table 3.2
3) Cruise	W3/W2=	0.48503241		0.866 L/D
4) Loiter	W4/W3=	0.94727935		L/D Max.
5) Land	W5/W4=	0.995		Table 3.2
	W5/W0=	0.43679723		
	Wf/W0=	0.59699494		6% Fuel allowance
	We/W0=	0.37886176		Table 3.1 (Jet Fighter/Trainer)
Verification of Total Weight	W0=	103548	lbs.	
Fuel Weight	Wf=	61818	lbs.	
Empty Weight	We=	39231	lbs.	

Table 2: Conceptual Sizing for SST (Range = 4,500 nm)

Design Criteria

Crew Members=	9	Wc=	1800	lbs.	
Payload		Wp=	700	lbs.	
Range		R=	**4500**	**nm**	
		R=	27342000	ft	
Speed of Sound		a=	994.8		Alt 30000 ft
Cruise Velocity		M=	2.2	M	
		V=	2188.56	ft/s	
Specific Fuel Consumption		C=	1.3	1/hr	Turbojet
		C=	0.00036111	1/s	
Endurance		E=	0.3333	hr	
		E=	1199.88	s	
Lift/Drag		L/D=	8		

Mission Segment Weight Fractions

Total Weight	W0=	67367	lbs.	
1) Warmup and Takeoff	W1/W0=	0.97		Table 3.2
2) Climb	W2/W1=	0.985		Table 3.2
3) Cruise	W3/W2=	0.5214272		0.866 L/D
4) Loiter	W4/W3=	0.94727935		L/D Max.
5) Land	W5/W4=	0.995		Table 3.2
	W5/W0=	0.46957265		
	Wf/W0=	0.56225299		6% Fuel allowance
	We/W0=	0.40063675		Table 3.1 (Jet Fighter/Trainer) A=1.7 C=-0.13
Verification of Total Weight	W0=	67367	lbs.	
Fuel Weight	Wf=	37877	lbs.	
Empty Weight	We=	26990	lbs.	

Table 3: Conceptual Sizing for SST (Range = 4,000 nm)

Design Criteria

Crew Members=	9	Wc=	1800	lbs.	
Payload		Wp=	700	lbs.	
Range		R=	**4000**	**nm**	
		R=	24304000	ft	
Speed of Sound		a=	994.8		Alt 30000 ft
Cruise Velocity		M=	2.2	M	
		V=	2188.56	ft/s	
Specific Fuel Consumption		C=	1.3	1/hr	Turbojet
		C=	0.00036111	1/s	
Endurance		E=	0.3333	hr	
		E=	1199.88	s	
Lift/Drag		L/D=	8		

Mission Segment Weight Fractions

Total Weight	W0=	46110	lbs.	
1) Warmup and Takeoff	W1/W0=	0.97		Table 3.2
2) Climb	W2/W1=	0.985		Table 3.2
3) Cruise	W3/W2=	0.5605529		0.866 L/D
4) Loiter	W4/W3=	0.94727935		L/D Max.
5) Land	W5/W4=	0.995		Table 3.2
	W5/W0=	0.50480741		
	Wf/W0=	0.52490415		6% Fuel allowance
	We/W0=	0.42087735		Table 3.1 (Jet Fighter/Trainer) A=1.7 C=-0.13
Verification of Total Weight	W0=	46110	lbs.	
Fuel Weight	Wf=	24203	lbs.	
Empty Weight	We=	19407	lbs.	

Table 4: Conceptual Sizing for SST (Crew & Passengers = 11)

Design Criteria

Crew Members= 11	Wc=	2200 lbs.	
Payload	Wp=	700 lbs.	
Range	R=	5000 nm	
	R=	30380000 ft	
Speed of Sound	a=	994.8	Alt 30000 ft
Cruise Velocity	M=	2.2 M	
	V=	2188.56 ft/s	
Specific Fuel Consumption	C=	1.3 1/hr	Turbojet
	C=	0.00036111 1/s	
Endurance	E=	0.3333 hr	
	E=	1199.88 s	
Lift/Drag	L/D=	8	

Mission Segment Weight Fractions

Total Weight	W0=	108918 lbs.	
1) Warmup and Takeoff	W1/W0=	0.97	Table 3.2
2) Climb	W2/W1=	0.985	Table 3.2
3) Cruise	W3/W2=	0.48503241	0.866 L/D
4) Loiter	W4/W3=	0.94727935	L/D Max.
5) Land	W5/W4=	0.995	Table 3.2
	W5/W0=	0.43679723	
	Wf/W0=	0.59699494	6% Fuel allowance
	We/W0=	0.3763795	Table 3.1 (Jet Fighter/Trainer) A=1.7 C=-0.13
Verification of Total Weight	W0=	108918 lbs.	
Fuel Weight	Wf=	65023 lbs.	
Empty Weight	We=	40994 lbs.	

Table 5: Conceptual Sizing for SST (Cruise Altitude = 60,000 ft)

Design Criteria

Crew Members=	9	Wc=	1800	lbs.	
Payload		Wp=	700	lbs.	
Range		R=	5000	nm	
		R=	30380000	ft	
Speed of Sound		a=	**968.1**	**ft/s**	**Alt 36000 to 80000 ft**
Cruise Velocity		M=	2.2	M	
		V=	2129.82	ft/s	
Specific Fuel Consumption		C=	1.3	1/hr	Turbojet
		C=	0.00036111	1/s	
Endurance		E=	0.3333	hr	
		E=	1199.88	s	
Lift/Drag		L/D=	8		

Mission Segment Weight Fractions

Total Weight	W0=	117679 lbs.	
1) Warmup and Takeoff	W1/W0=	0.97	Table 3.2
2) Climb	W2/W1=	0.985	Table 3.2
3) Cruise	W3/W2=	0.47544948	0.866 L/D
4) Loiter	W4/W3=	0.94727935	L/D Max.
5) Land	W5/W4=	0.995	Table 3.2
	W5/W0=	0.4281673	
	Wf/W0=	0.60614266	6% Fuel allowance
	We/W0=	0.37261305	Table 3.1 (Jet Fighter/Trainer) A=1.7 C=-0.13
Verification of Total Weight	W0=	117679 lbs.	
Fuel Weight	Wf=	71330 lbs.	
Empty Weight	We=	43849 lbs.	

Table 6: Conceptual Sizing for SST
(Specific Fuel Consumption = 1.2 / hr)

Design Criteria

Crew Members=	9	Wc=	1800 lbs.	
Payload		Wp=	700 lbs.	
Range		R=	5000 nm	
		R=	30380000 ft	
Speed of Sound		a=	994.8	Alt 30000 ft
Cruise Velocity		M=	2.2 M	
		V=	2188.56 ft/s	
Specific Fuel Consumption		C=	**1.2 1/hr**	Turbojet
		C=	0.00033333 1/s	
Endurance		E=	0.3333 hr	
		E=	1199.88 s	
Lift/Drag		L/D=	8	

Mission Segment Weight Fractions

Total Weight	W0=	72302 lbs.	
1) Warmup and Takeoff	W1/W0=	0.97	Table 3.2
2) Climb	W2/W1=	0.985	Table 3.2
3) Cruise	W3/W2=	0.51279318	0.866 L/D
4) Loiter	W4/W3=	0.95123418	L/D Max.
5) Land	W5/W4=	0.995	Table 3.2
	W5/W0=	0.46372524	
	Wf/W0=	0.56845125	6% Fuel allowance
	We/W0=	0.39697155	Table 3.1 (Jet Fighter/Trainer) A=1.7 C=-0.13
Verification of Total Weight	W0=	72302 lbs.	
Fuel Weight	Wf=	41100 lbs.	
Empty Weight	We=	28702 lbs.	

Table 7: Conceptual Sizing for SST (Lift/Drag = 9)

Design Criteria

Crew Members=	9	Wc=	1800 lbs.	
Payload		Wp=	700 lbs.	
Range		R=	5000 nm	
		R=	30380000 ft	
Speed of Sound		a=	994.8	Alt 30000 ft
Cruise Velocity		M=	2.2 M	
		V=	2188.56 ft/s	
Specific Fuel Consumption		C=	1.3 1/hr	Turbojet
		C=	0.00036111 1/s	
Endurance		E=	0.3333 hr	
		E=	1199.88 s	
Lift/Drag		L/D=	9	

Mission Segment Weight Fractions

Total Weight	W0=	62344 lbs.	
1) Warmup and Takeoff	W1/W0=	0.97	Table 3.2
2) Climb	W2/W1=	0.985	Table 3.2
3) Cruise	W3/W2=	0.52563602	0.866 L/D
4) Loiter	W4/W3=	0.95299718	L/D Max.
5) Land	W5/W4=	0.995	Table 3.2
	W5/W0=	0.47622016	
	Wf/W0=	0.55520663	6% Fuel allowance
	We/W0=	0.40469335	Table 3.1 (Jet Fighter/Trainer) A=1.7 C=-0.13
Verification of Total Weight	W0=	62344 lbs.	
Fuel Weight	Wf=	34614 lbs.	
Empty Weight	We=	25230 lbs.	

Table 8: Conceptual Sizing for SST (Equation Constants)

Design Criteria

Crew Members= 9	Wc=	1800	lbs.	
Payload	Wp=	700	lbs.	
Range	R=	5000	nm	
	R=	30380000	ft	
Speed of Sound	a=	994.8		Alt 30000 ft
Cruise Velocity	M=	2.2	M	
	V=	2188.56	ft/s	
Specific Fuel Consumption	C=	1.3	1/hr	Turbojet
	C=	0.00036111	1/s	
Endurance	E=	0.3333	hr	
	E=	1199.88	s	
Lift/Drag	L/D=	8		

Mission Segment Weight Fractions

Total Weight	W0=	76984	lbs.	
1) Warmup and Takeoff	W1/W0=	0.97		Table 3.2
2) Climb	W2/W1=	0.985		Table 3.2
3) Cruise	W3/W2=	0.48503241		0.866 L/D
4) Loiter	W4/W3=	0.94727935		L/D Max.
5) Land	W5/W4=	0.995		Table 3.2
	W5/W0=	0.43679723		
	Wf/W0=	0.59699494		6% Fuel allowance
	We/W0=	0.37053064		Table 3.1 (Jet Transport) **A=1.02 C=-0.09**
Verification of Total Weight	W0=	76984	lbs.	
Fuel Weight	Wf=	45959	lbs.	
Empty Weight	We=	28525	lbs.	

Calculation of Fuselage Length and Diameter

Length = A W_o^c

W_o = Takeoff Gross Weight = 107,000 lbs.
A = 0.83 (Raymer[1])
C = 0.43

Length = 107.0 ft

$$\text{Fineness Ratio} = \frac{\text{Fuselage Length}}{\text{Diameter}} = 14$$

[Fineness Ratio for Supersonic Aircraft = 14 (Raymer[1])]

Diameter = 7.6 ft

Appendix F (Structure)
Final Sizing

Table 1: Concorde Weight Approximation

Weight Equations Terminology:

A=	1.70		Aspect Ratio
Ah=	0.00		Horizontal Tail Aspect Ratio
Av=	0.89		Vertical Tail Aspect Ratio
Bh=	0.00	ft	Horizontal Tail Span
Bw=	83.83	ft	Wing Span
D=	0.33	ft	Fuselage Structural Depth
Fw=	7.70	ft	Fuselage Width at Horizontal Tail Intersection
Ht/Hv=	0.00		Not a "T" Tail
Iy=	41620083.4	lb ft^2	Yawing Moment of Inertia (Eq. 16.51)
Kdoor=	1.20		Three Side Doors
Kdw=	0.768		Delta Wing
Kdwf=	0.774		Delta Wing
Klg=	1.00		Location of Landing Gear
Kmp=	1.00		
Kng=	1.00		
Knp=	1.00		
Kr=	1.00		
Ktp=	1.00		
Kuht=	1.00		
Kws=	0.45		
Ky=	21.66		Pithching Radius of Gyration
Kz=	72.20	ft	Yawing Radius of Gyration
L/D=	7.00		
L=	203.75	ft	Fuselage Structural Length
La=	190.00	ft	Electronic Routing Distance
lambda=	0.15		Wing Taper Ratio (Tip Chord/ Centerline root Chord)
Lambda=	1.28		Wing Sweep at 25% MAC
Lambda ht=	0.00		Horizontal Tail Wing Sweep at 25% MAC
Lambda vt=	0.79		Vertical Tail Wing Sweep at 25% MAC
Lec=	98.00	ft	Total Length from engine front to Cockpit
Lm=	123.60	in	Length of Main landing Gear
Ln=	154.80	in	Nose Gear Length
Lt=	72.20	ft	Tail Length
M=	2.20		Mach Number
Nc=	6.00		Number of Crew
Nen=	4.00		Number of Engines
Nf=	5.00		Number of Functions Performed by Controls (4-7)
Ngen=	4.00		Number of Generators (=Nen)
Nl=	1.00		Ultimate Landing Load Factor
Nlt=	41.30	ft	Nacelle Length

Nm=	1.00		Number of Mechanical Functions (0-2)
Nmss=	2.00		Number of Main Gear Shock Struts
Nmw=	8.00		Number of Main Wheels
Nnw=	2.00		Number of Nose Wheels
Np=	147.00		Number of Crew and Passengers
Nt=	17.00		Number of Fuel Tanks
Nw=	3.90	ft	Nacelle Width
Nz=	3.24		Ultimate Load Factor (Table 12.1 Megson)
Q=	511.68	lb/ft^2	Dynamic Pressure
Rkva=	160.00	kV A	System Electrical Rating
Scs=	821.44	ft^2	Total Area of Control Surfaces
Scsw=	344.44	ft^2	Control Surface Area (Wing Mounted)
Se=	0.00	ft	Elevator Area
Sfwet=	5601.23	ft^2	Fuselage Wetted Area
Sht=	0.00	ft^2	Horizontal Tail Area
Sn=	426.00	ft^2	Nacelle Wetted Area
Sr=	112.00	ft^2	Rudder Area
Svt=	365.00	ft^2	Vertical Tail Area
Sw=	3856.00	ft^2	Wing Area
t/c=	0.09		Wing Thickness
Vi=	31647.50	gal	Integral Tanks Volume
Vpr=	12025.73	ft^3	Volume of Pressurized Section
Vstall=	150.00	ft/s	
Vt=	31647.40	gal	Total Fuel Volume (6.5 lbs/gal)
W=	10.30	ft	Fuselage Structural Width
Wapu (uninstalled)=	1000.00		Auxillary Power Unit
Wc=	28000.00	lbs	Maximum Cargo Weight
Wdg=	408000.00	lbs	Design Gross Weight
Wec=	12042.57	lbs	Weight of Engine and Contents (per nacelle)
Wen=	11000.00	lbs	Engine Weight (Each)
Wfw=	51427.03	lbs	Weight of Fuel in Wings
Wl=	245000.00	lbs	Landing Design Gross Weight
Wuav=	1200.00	lbs	Uninstalled Avionics Weight (800-1400)

Combined Weight Method:

W wing=	24850 lbs.	1	6.44 lbs/ft^2
W horizontal tail=	0 lbs.		
W vertical tail=	2761 lbs.	1	7.56 lbs/ft^2
W fuselage=	43548 lbs.	1	
W main Landing gear=	10108 lbs.	1	
W nose landing gear=	1648 lbs.	1	
W installed engine=	54828 lbs.	2	13.4 Engine % of TDGW
W fuel system=	3360 lbs.	3	
W surface control & hyd=	5032 lbs.	3	(1.1 Correction for Variable Nose)
W apu installed=	2200 lbs.	1	
W instruments=	806 lbs.	1	
W electrical=	2721 lbs.	3	
W avionics=	1840 lbs.	1	
W furnishings=	8071 lbs.	3	
W air con & anti-ice=	4103 lbs.	1	
W handling gear=	1224 lbs.	1	

Total Weight: **167101 lbs.**

Actual Operational Weight Empty: **173500 lbs**

Percent Difference: **-3.69%**

Table 2: Tesseract Weight Estimation

Weight Equations Terminology:

A=	1.90	Aspect Ratio
Ah=	0.00	Horizontal Tail Aspect Ratio
Av=	0.89	Vertical Tail Aspect Ratio
Bh=	0.00 ft	Horizontal Tail Span
Bw=	47.75 ft	Wing Span
D=	0.33 ft	Fuselage Structural Depth
Fw=	3.83 ft	Fuselage Width at Horizontal Tail Intersection
Ht/Hv=	0.00	Not a "T" Tail
Iy=	3160541.2 lb ft^2	Yawing Moment of Inertia (Eq. 16.51)
Kdoor=	1.00	One Side Door
Kdw=	0.768	Delta Wing
Kdwf=	0.774	Delta Wing
Klg=	1.00	Location of Landing Gear
Kmp=	1.00	
Kng=	1.00	
Knp=	1.00	
Kr=	1.00	
Ktp=	1.00	
Kuht=	1.00	
Kws=	0.37	
Ky=	12.00	Pithching Radius of Gyration
Kz=	40.00 ft	Yawing Radius of Gyration
L/D=	7.00	
L=	107.00 ft	Fuselage Structural Length
La=	85.00 ft	Electronic Routing Distance
lambda=	0.00	Wing Taper Ratio (Tip Chord/ Centerline root Chord)
Lambda=	1.12	Wing Sweep at 25% MAC
Lambda ht=	0.00	Horizontal Tail Wing Sweep at 25% MAC
Lambda vt=	0.79	Vertical Tail Wing Sweep at 25% MAC
Lec=	60.00 ft	Total Length from engine front to Cockpit
Lm=	123.60 in	Length of Main landing Gear
Ln=	154.80 in	Nose Gear Length
Lt=	40.00 ft	Tail Length
M=	2.20	Mach Number
Nc=	3.00	Number of Crew
Nen=	2.00	Number of Engines
Nf=	5.00	Number of Functions Performed by Controls (4-7)
Ngen=	2.00	Number of Generators (=Nen)
Nl=	1.00	Ultimate Landing Load Factor
Nlt=	24.00 ft	Nacelle Length
Nm=	1.00	Number of Mechanical Functions (0-2)

Nmss=	2.00		Number of Main Gear Shock Struts
Nmw=	4.00		Number of Main Wheels
Nnw=	2.00		Number of Nose Wheels
Np=	9.00		Number of Crew and Passengers
Nt=	8.00		Number of Fuel Tanks
Nw=	4.00	ft	Nacelle Width
Nz=	3.46		Ultimate Load Factor (Table 12.1 Megson)
Q=	649.60	lb/ft^2	Dynamic Pressure
Rkva=	160.00	kV A	System Electrical Rating
Scs=	400.00	ft^2	Total Area of Control Surfaces
Scsw=	170.00	ft^2	Control Surface Area (Wing Mounted)
Se=	0.00	ft	Elevator Area
Sfwet=	1570.23	ft^2	Fuselage Wetted Area
Sht=	0.00	ft^2	Horizontal Tail Area
Sn=	240.00	ft^2	Nacelle Wetted Area
Sr=	35.00	ft^2	Rudder Area
Svt=	100.00	ft^2	Vertical Tail Area
Sw=	1200.00	ft^2	Wing Area
t/c=	0.08		Wing Thickness
Vi=	9230.00	gal	Integral Tanks Volume (6.5 lbs/gal)
Vpr=	1151.51	ft^3	Volume of Pressurized Section
Vstall=	237.18	ft/s	
Vt=	9230.00	gal	Total Fuel Volume
W=	7.66	ft	Fuselage Structural Width
Wapu (uninstalled)=	1000.00		Auxillary Power Unit
Wc=	500.00	lbs	Maximum Cargo Weight
Wdg=	107000.00	lbs	Design Gross Weight
Wec=	5918.29	lbs	Weight of Engine and Contents (per nacelle)
Wen=	5000.00	lbs	Engine Weight (Each)
Wfw=	14998.75	lbs	Weight of Fuel in Wings
Wl=	50000.00	lbs	Landing Design Gross Weight
Wuav=	1000.00	lbs	Uninstalled Avionics Weight (800-1400)

Combined Weight Method:

			Composite Corrections:	
W wing=	3488 lbs.	2.91 lbs/ft^2	0.75	2616 lbs.
W horizontal tail=	0 lbs.		0.75	0 lbs.
W vertical tail=	605 lbs.	6.05 lbs/ft^2	0.75	454 lbs.
W fuselage=	10422 lbs.		0.75	7816 lbs.
W main Landing gear=	2066 lbs.		0.88	1818 lbs.
W nose landing gear=	590 lbs.		0.88	519 lbs.
W installed engine=	13251 lbs.	12.4 % of TDGW	1.00	13251 lbs.
W fuel system=	1359 lbs.		1.00	1359 lbs.
W surface control & hyd=	2428 lbs.		0.60	1457 lbs.
W apu installed=	2200 lbs.		1.00	2200 lbs.
W instruments=	203 lbs.		1.00	203 lbs.
W electrical=	1923 lbs.		1.00	1923 lbs.
W avionics=	1538 lbs.		1.00	1538 lbs.
W furnishings=	482 lbs.		1.00	482 lbs.
W air con & anti-ice=	476 lbs.		1.00	476 lbs.
W handling gear=	321 lbs.		1.00	321 lbs.
Total Weight:	41352 lbs.			36434
Correction Factor	1.037			1.037

Corrected Total Weight:

42878 Lbs — 37778 Lbs

Total Design Gross Weight:

107000 lbs — 107000 lbs

Percent of TDGW:

40.07% — 35.31%

Table 3: Empty Weight Center of Gravity

Component	Weight (lbs)	Dis. From Nose (ft)	Moment (ft/lb)
Wing	2616	83.3	218008.7
Vertical Tail	454	100.0	45400.0
Fuselage	7816	49.6	387461.8
Main Landing Gear	1818	87.5	159075.0
Nose Landing Gear	519	10.0	5190.0
Installed Engine	13251	88.5	1172713.5
Fuel System	1359	80.0	108720.0
Surface Control & Hyd.	1457	90.0	131130.0
APU Installed	2200	90.0	198000.0
Instruments	203	17.0	3451.0
Electrical	1923	45.0	86535.0
Avionics	1538	50.0	76900.0
Furnishings	482	34.5	16629.0
Air Con & Anti-Ice	476	72.0	34272.0
Handling Gear	321	60.0	19260.0
Total:	36433		2662746.0

Empty Weight Center of Gravity:

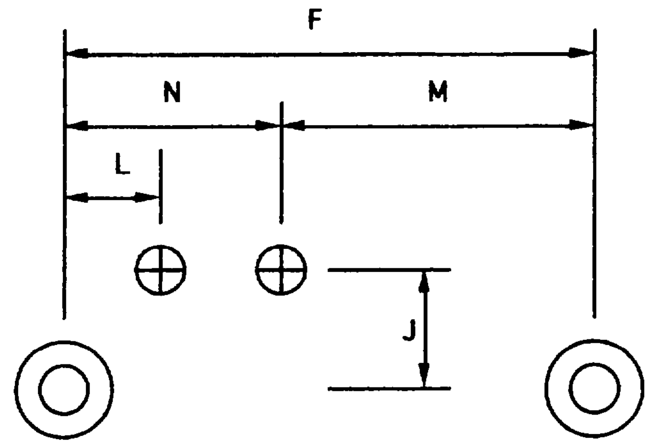
Xcg= 73.1 ft

Appendix G (Structure)
Landing Gear

Landing Gear

Distance From Nose:

Forward CG = 70 ft
Aft CG = 75 ft
AC = 86 ft
Nose Gear = 30 ft
Main Gear = 80 ft



F = 55 ft L = 46 ft
M = 5 ft N = 50 ft
J = 20 ft

Maximum Static Main Gear Load = W(F-M)/2F = 48,600 lbs/ Strut

Maximum Static Nose Gear Load = W(F-L)/F = 17,500 lbs
(18% Maximum Main Load)

Minimum Static Nose Gear Load = W(F-N)/F = 9,700 lbs
(10% Maximum Main Load)

Braking Load Nose Gear = 10 J W /(32.2 F) = 12,000 lbs

Tire Sizing

Main Gear:

Diameter = A WwB

Ww = Weight on Wheel
A = 1.60
B = 0.31

Diameter = 36.5 in

Ww - Main gear takes 90% of the total load. Therefore, each of the four tires will carry 24,075 lbs.

Width = A WwB

Ww = Weight on Wheel
A = 0.10
B = 0.47

Width = 11.5 in

Nose Gear:

Nose wheels are 60% of the main wheel tire size.
Diameter = 21.9 in
Width = 6.9 in

Appendix H (Structure)
Finite Element Analysis

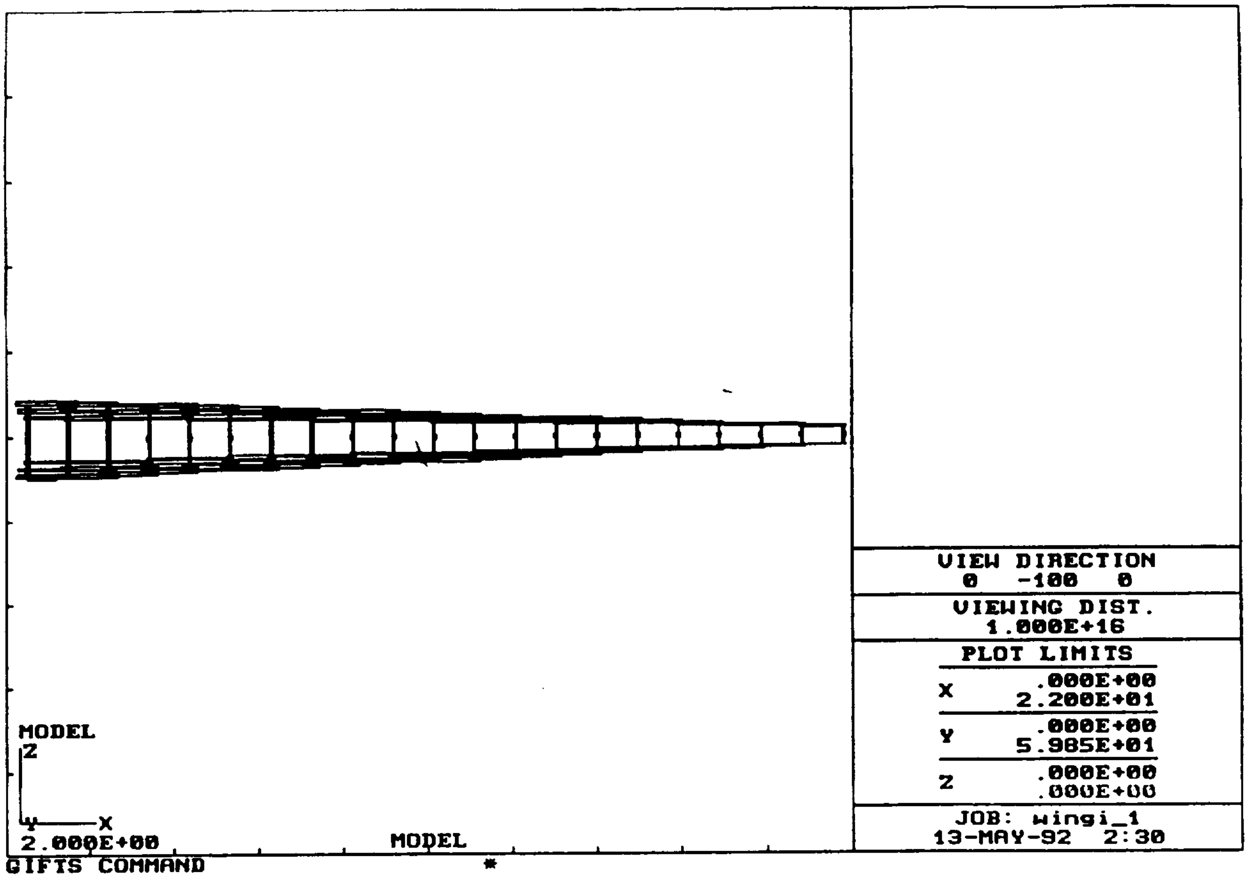


Figure 1: Internal Wing Model

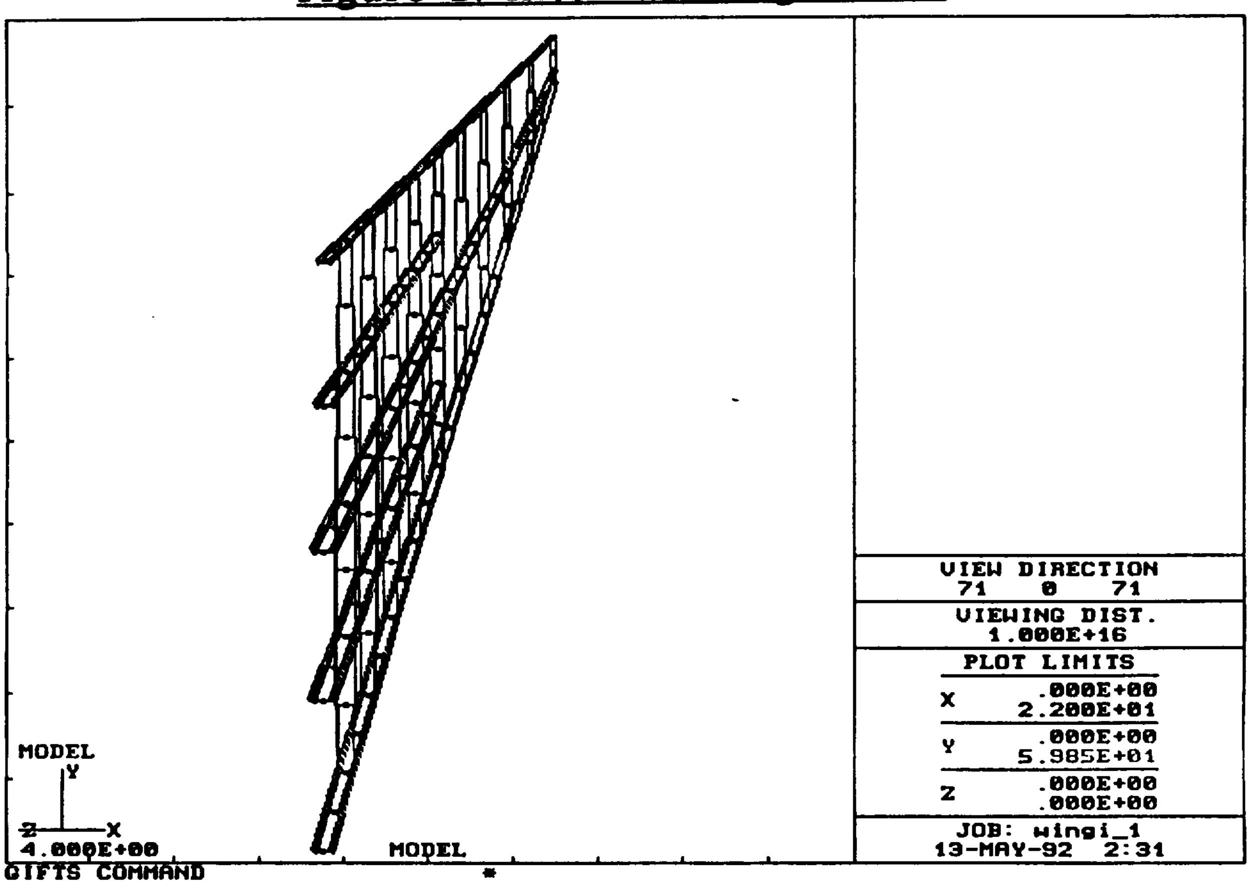


Figure 2: Internal Wing Model (Spars & Ribs)

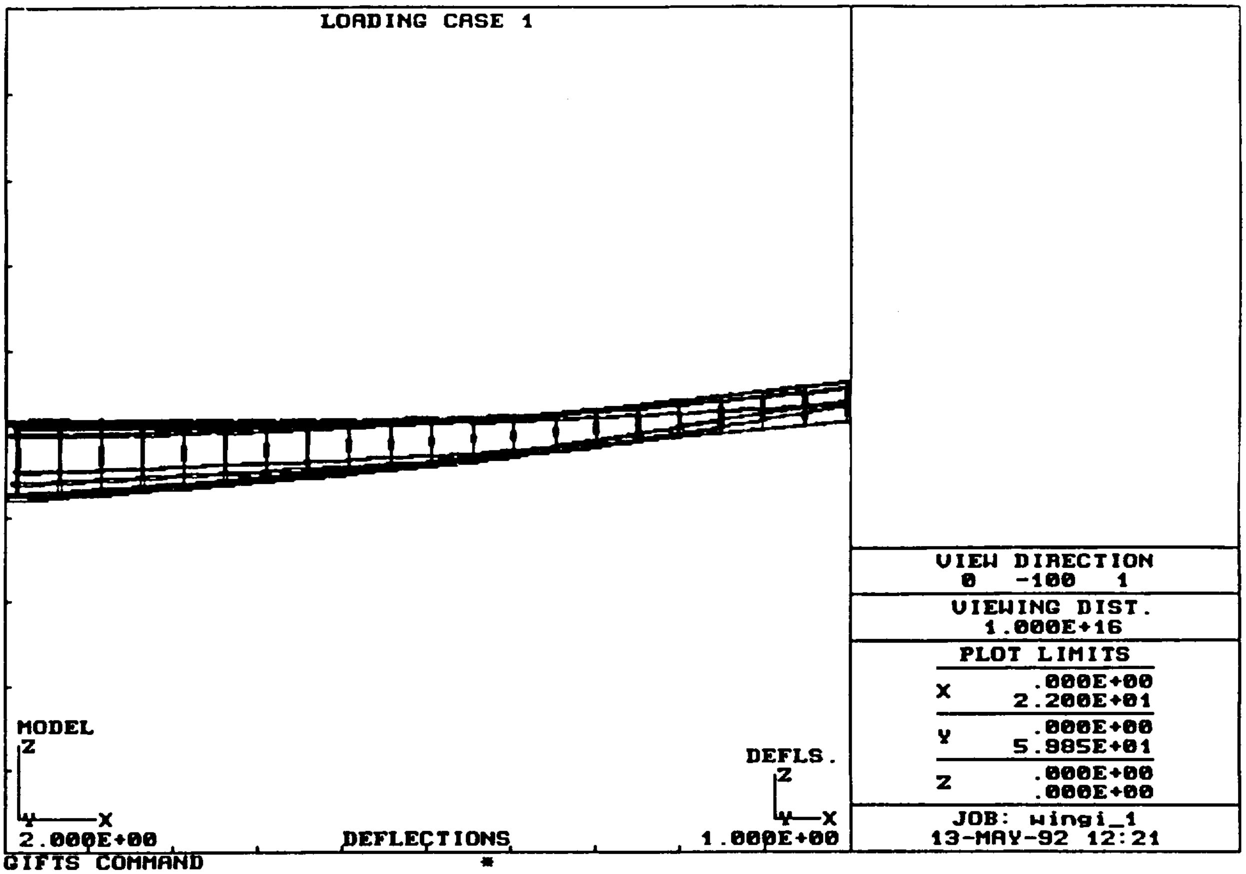


Figure 3: Wing Deflection

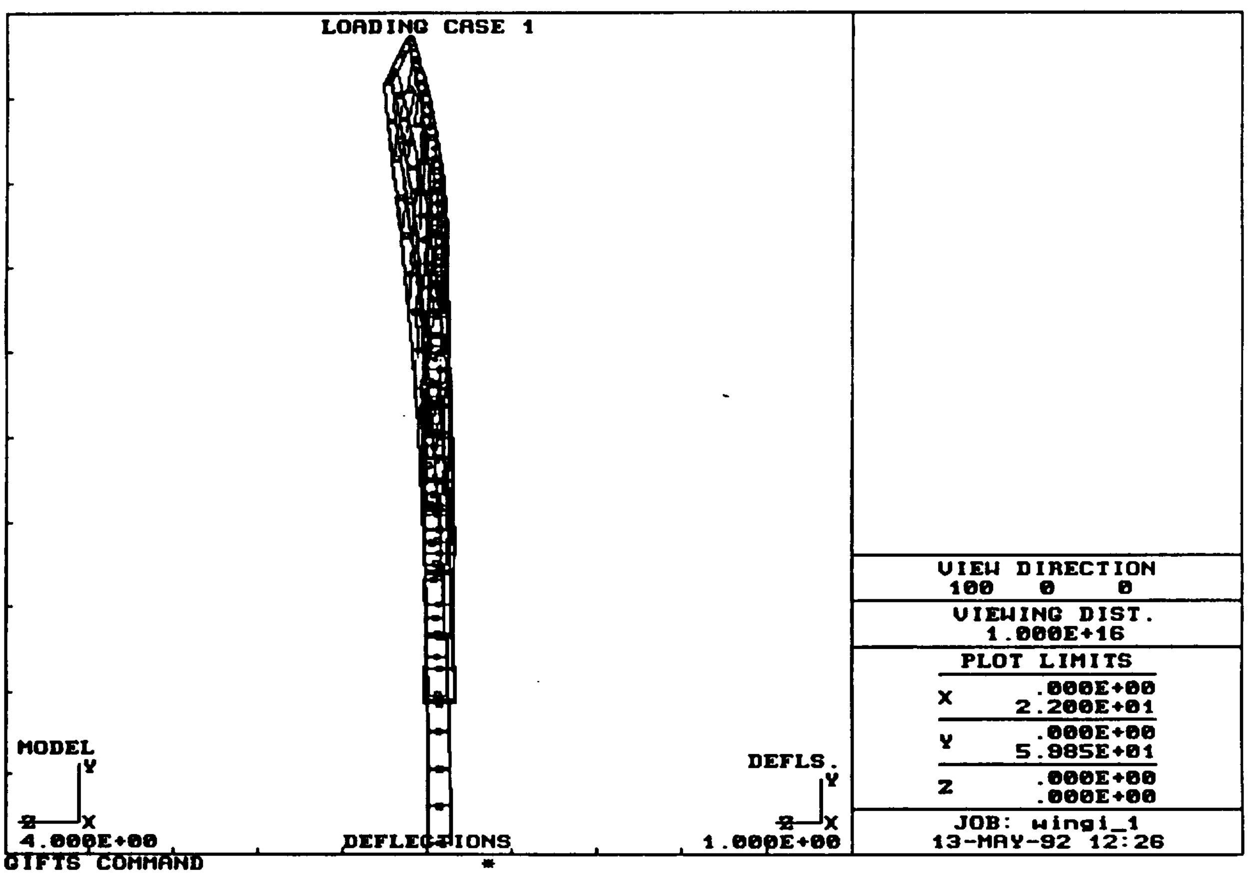


Figure 4: Wing Warping