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 2,500 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-chord length of 61.0 ft. A SNECMA MCV 99 variable-cycle engine design was chosen for this aircraft.

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 in Table 1.

Table 1 Design Specifications

Range	Transpacific
Mach Number	2.2
Passenger & Crew Capacity	9
Total Payload	2,500 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:

- aerodynamics,
- propulsion, and
- structures.

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 areas. During the design process, constant communication between the groups was required to keep the project on line. Included in this report is an overview of all the work completed by May 14, 1992, by each of the three groups.

Analysis

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.[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.

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.

Table 2 Altitude vs Fuel and CL_{Cruise}

CL_{Cruise}	Fuel Remaining (% weight)	Altitude (ft)
0.128	88	55,000
0.128	53	60,000
0.128	25	65,000

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 (Figure 1) 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.

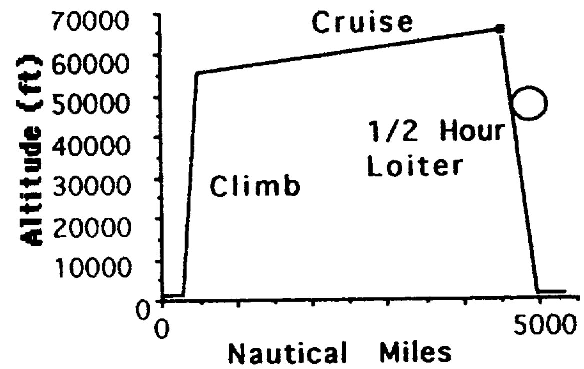

Fig. 1 Tesseract's Mission Profile

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 supporting data available. A delta configuration with subsonic leading edges was chosen primarily because 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 of 63.0°.

Based on the previous discussion, and to minimize induced drag (to be discussed hereafter), an aspect ratio of 1.9 was desired. Based on the 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.

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:

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

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. For the particular configuration shown, lambda is equal to 1.25, which, in turn, gives a lift curve slope of 1.76.

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.

Table 3 Drag Summary during Cruise

Parasite Drag Coefficient	
Skin friction drag	.0051
Wave drag	.0068
Miscellaneous	.0005
Total $C_{D,O}$	.0124
Induced Drag	
Induced drag $C_{D,i}$	.0058
Total Drag Coefficient	
Parasite drag	.0124
Induced drag	.0058
Total C_D	.0182

In lieu of the effects of lights, antennae, and other manufacturing defects, along with other unaccountable factors, an exact coefficient cannot 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 (Figure 2).

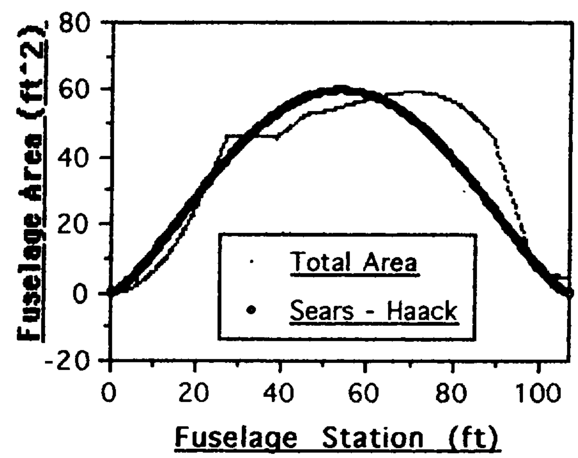

Fig. 2 Area Distribution

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.

Drag calculations for the subsonic and transonic regimes were calculated for various 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.

Longitudinal static stability of most conventional aircraft requires 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.

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 to support the weight of the aircraft to maintain level flight.

Analysis of the static stability for the aircraft showed that a horizontal stabilizer was essentially unnecessary for the cruising speed of Mach 2.2. 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.

For this design, the pertinent stability figures are listed in Table 4.

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	.037
Static Margin at M = 2.2	.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.

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, 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.

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.

A SNECMA 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

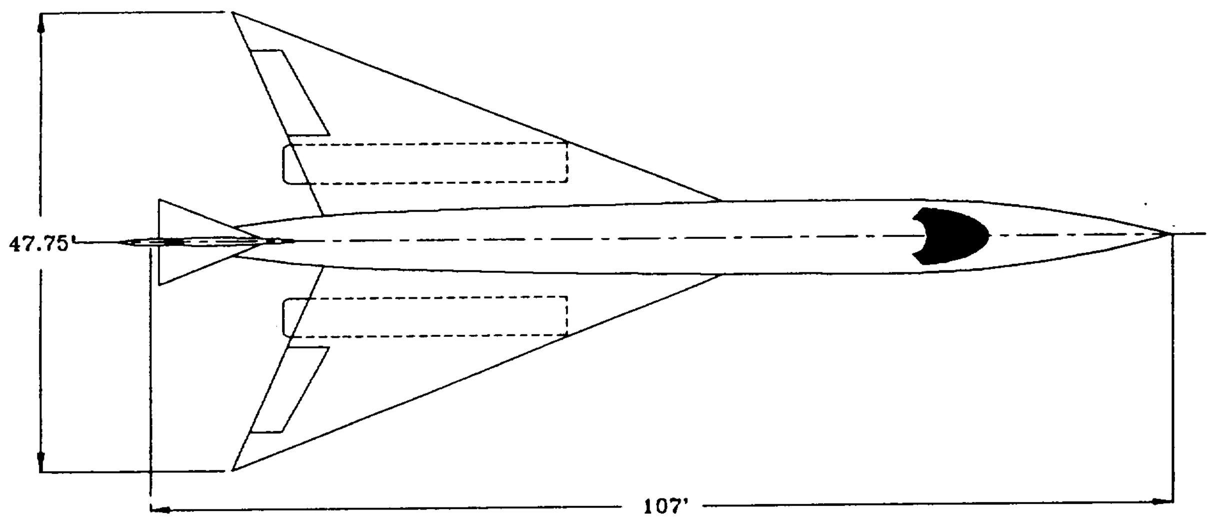

Fig. 3 Top view of Tesseract

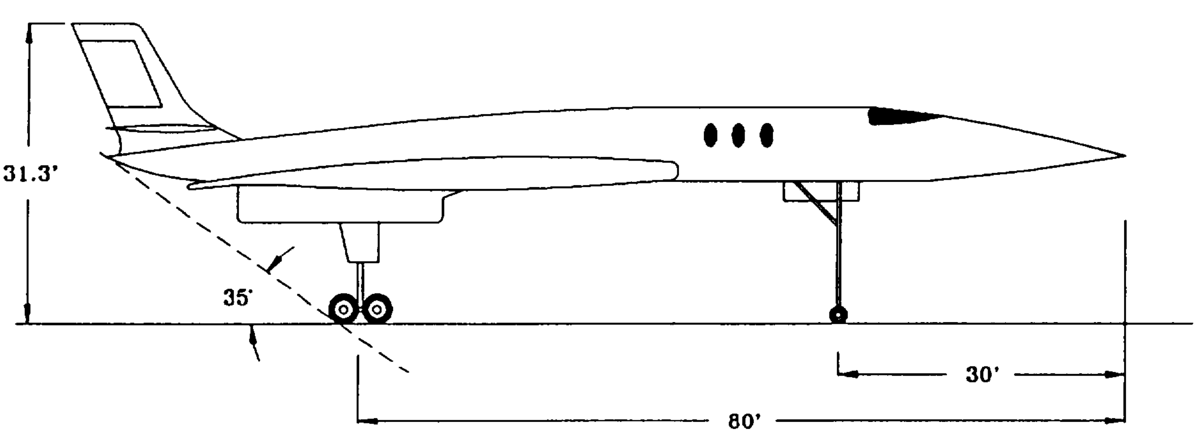

Fig. 4 Side view of Tesseract

premixing before combustion, staged burning, rich burn/quick quench/lean burn combustor, and a variable area geometry reduces pollutant emissions into the atmosphere by 50 percent when compared to other current cycle emissions.[6]

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. 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 angles 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 swallowing 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 30 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 ft 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.

Inlet drag is approximated from the inlet drag trends plot for a two-dimensional inlet. This plot was compiled from typical data previously collected.[1] Inlet drag for different modes of flight for this design was 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	1723
0.95	0.18	713
0.10	0.02	3.8

D = Drag q = Dynamic Pressure A = Capture Area

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.[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. The angle geometry of the nozzle was determined from these coefficients. The primary half angle is 10 degrees, and the secondary half angle is 15 degrees.

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. High exhaust velocity allows turbojets to give good supersonic performance, but they are too noisy for civilian use. Turbofans have a lower exhaust velocity and are, therefore, quieter, but their supersonic performance is poor. 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. 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. However, it was decided to use two engines to reduce weight and to eliminate some of the problems involved in mounting an engine to the centerline of the aircraft, such as 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.[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 considerably less than the engine size needed for cruise. The MCV 99 engine has a thrust of 49,455 lb, and was 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

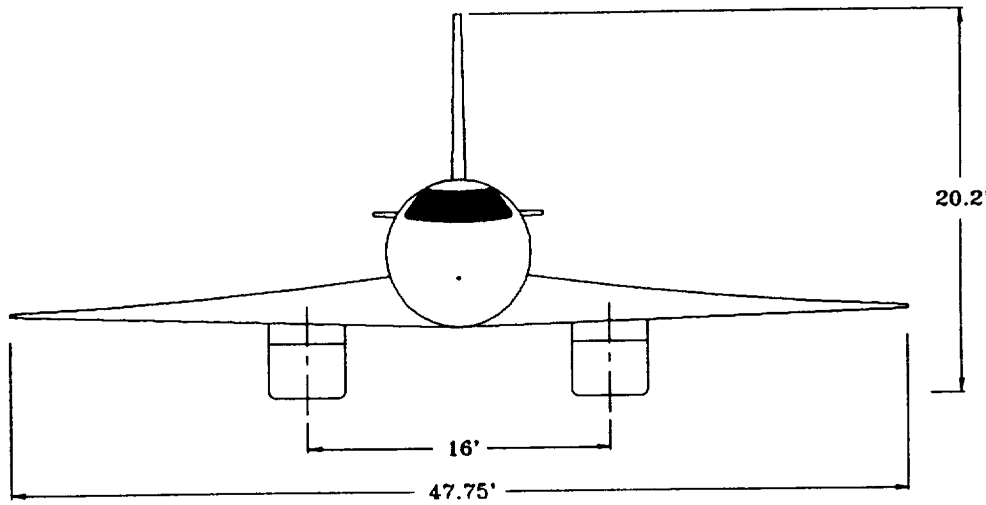

Fig. 5 Front view of Tesseract

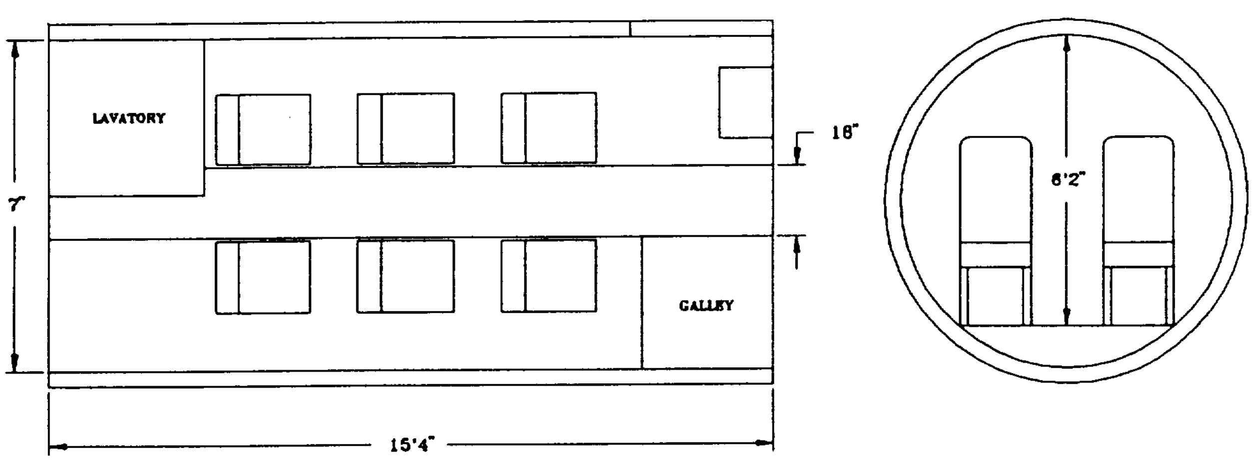

Fig. 6 Cabin layout

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	0.45	0	0.36	0.34

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.[6] The weight of the engine was set at 5,000 lb, based on a historical thrust-to-weight ratio of five for recent supersonic engines.[7,8,9] Blow-in doors are needed to provide correct airflow when the fan is operating, 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. 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[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[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[6,11] 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.

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.

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 and crew	9
Passenger and crew weight	1,800 lbs
Payload weight	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. 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.[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 ft 8 in based on a supersonic fineness ratio of 14.[1] The fineness ratio is the ratio between fuselage length and diameter, which minimizes wave drag. The inner diameter of the fuselage was set to 7 ft after allowing for a 4-in fuselage thickness.

With the initial sizing complete, a cabin layout was generated using the values for economy and high density passenger compartments presented by Raymer.[1] The total passenger cabin length is 15 ft 4 in. A recessed floor was used to allow for a 6-ft-2-in-high aisle 18 in wide. The passenger compartment seats six people; a jump seat is available for the flight attendant. Three seats with a width of 18 inches and a seat pitch of 36 inches were placed on each side of the aisle. The headroom was 5 ft 10 in. The cabin also included a 40 sq in lavatory and a small galley.

Final Weight Estimation

After the initial analysis from both the aerodynamics and propulsion groups was completed, it was decided that a more accurate weight estimate for the aircraft was required. Five different weight approximation methods[1,12] 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, a combination of the weight estimation equations that best approximated the individual components of the aircraft was calculated. 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, we estimated the empty weight of Tesseract to be 42,878 lbs. Based on a composite utilization by weight of 55%[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 ft from the nose.

Landing Gear

The main landing gear is located 80 ft from the nose of the aircraft and is 16 ft off the centerline of the fuselage. It will be positioned on the wing next to the engines. It will fold in towards the fuselage and most likely will need a pod to house part of the gear that does not fit in the wing. The total length of the main landing gear is 20 ft, preventing the tail of the airplane from dragging on the ground during takeoff. 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 ft, 80 ft, and 86 ft, respectively, from the nose of the airplane. An oleo shock-strut is used for all the gear. The main landing gear is comprised of two struts with each strut having two sets of tires for a total of eight tires. Diameter of the tires is 37 in; width, 12 in. 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 two tires, with a diameter of 22 in and width of 8 in. 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 than 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]

Finite Element Analysis

A finite element analysis was completed on the fuselage and wing using the software "GIFTS."[13] The cabin section, the fuselage wing root section, and the internal wing structure were modeled during the analysis. 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 that 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 was therefore ignored.

Stress due to pressure exerted on the cabin was calculated 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, 40,000 psi should be the total stress that the plane would have to withstand.

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"). The ribs were idealized as 3/16" flat plates that also ranged in height through the structure. Over 150 elements were used for the interior of the wing to improve the accuracy of the results.

The 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

GIFTS showed the maximum deflection for the interior of the wing to be six inches. 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 too 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 SNECMA MCV 99 variable-cycle engine was lower than the 1.3 lbf/16m/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 basis for a future supersonic business transport design.

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