Design of Supersonic Transport

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ABSTRACT

DESIGN OF SUPERSONIC TRANSPORT

By Seruvizhi Maharajan

An efficient long range Supersonic Transport (SST) has not been feasible until now because of the difficulties dealing with the sonic boom. However, recent developments in technology make it a real possibility that a technically, environmentally, and economically acceptable SST will be feasible in the near future. This thesis outlines the conceptual and preliminary design of a long range SST, incorporating the latest technologies. The SST is designed to meet low sonic boom and low drag by appropriately designed fuselage shape. The airplane is expected to carry 337 passengers, over 8,700 miles within 4 hours. The stability analysis is included to show that the design is aerodynamically stable. Advanced aerodynamic concepts are incorporated to reduce shock waves and supersonic drag. The proposed SST concept satisfies FAR-25 requirements.

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I would like to thank my parents and teachers for their support and guidance. I was able to achieve my goals and be in this position right now because of their strong belief in me.

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1 INTRODUCTION

A supersonic transport (SST) can travel faster than the speed of sound. Unfortunately, SSTs were banned due to excessive noise (sonic boom) as well as engine exhaust products, which could damage the ozone layer, as SSTs cruise at very high altitude [1]. An additional factor, which made SSTs unattractive to airlines was their high operational costs. SSTs require an improved engine design compared to those used on subsonic transports due to the wide range of operational speeds. In the 1980s, while subsonic engines made great strides in increased efficiency, SST programs failed primarily because they were unable to produce more efficient engine designs for supersonic cruise [2].

The design of an efficient, environmentally acceptable SST remains challenging [3]. A successful design will require a multi-disciplinary approach, seeking improvements in aerodynamics, propulsion, structures and materials, and controls. This paper documents the preliminary design and analysis of an SST. It focuses on longer range and a profitable design for the airliners. An approach to identify any barriers in the development of an SST with advanced technology in the next 25 years is outlined. The approach is based on identifying the customer and design requirements for the aircraft and converting them into requirement for technology [4].

2 LITERATURE REVIEW

In the 1960s, aircraft production was based on maximum time of the aircraft in cruise while the fuel was not considered. This led to the SST concept because an SST could travel three times faster than the average subsonic flight. Thus, manpower and maintenance costs would be reduced by replacing three subsonic aircraft with a signle SST.

However, in the 1970s, the U.S. political parties opposed the SST idea, despite the interest shown by the flying community [6]. Boeing and Lockheed did propose new designs known as Advanced SST [7] but their economics proved to be unprofitable when compared with wide-body transports like the Boeing 747, which carried four times more passengers than an SST. Additionally, the SST engines need to be operated in a wide speed range, which makes it very difficult to be efficient [8]. Unable to overcome these challenges, the idea of the Advanced SST was dropped in the early 1980s. No new projects were proposed until the end of the 20th century. Even the Russian Tu-144, one of the two SSTs ever produced and flown commercially, had very few investors to cover its development costs [10].

2.1 VARIOUS COMMERCIAL SUPERSONIC AIRCRAFT

2.1.1 TUPOLEV TU-144

The Tupolev 144 was the first civilian SST, which flew on December 31, 1962 [11]. The Tupolev was manufactured by Tupolev OKB. Sixteen aircraft were built. These aircraft retired from service in 1984. They carried cargo for a few years and then used to train pilots of the Buran Spacecraft for the Soviet Space program and supersonic research for NASA. Tables 1 and 2 explain the general and performance characteristics of the Tupolev 144. Figure 1 represents the configuration layout of the Tupolev 144.

Table 1 General Characteristics the Tupolev-144 [12]

Crew	3
Capacity	120–140 but normally70~80 passengers
Length	65.50 m (215.54 ft)
Wingspan	28.80 m (94.48 ft)
Height	10.50 m (34.42 ft)
Wing area	438.0 m² (4,715 ft²)
Empty weight	85,000 kg (187,400 lb)
Loaded weight	120,000 kg (264,555 lb)
Max. takeoff weight	180,000 kg (397,000 lb)
Power plant	4 × Kolesov RD-36-51 afterburning turbojet, 200 kN (44,122 lbf)

Table 2 Performance Characteristics of the Tupolev TU-144 [13]

Cruise speed	Mach 2.15 (2,285 km/h (1,420 mph))
Service ceiling	20,000 m (65,600 ft)
Rate of climb	3,000 m/min (9,840 ft/min)
Wing loading	410.96 kg/m² (84.20 lb/ft²)
Thrust/weight	0.44

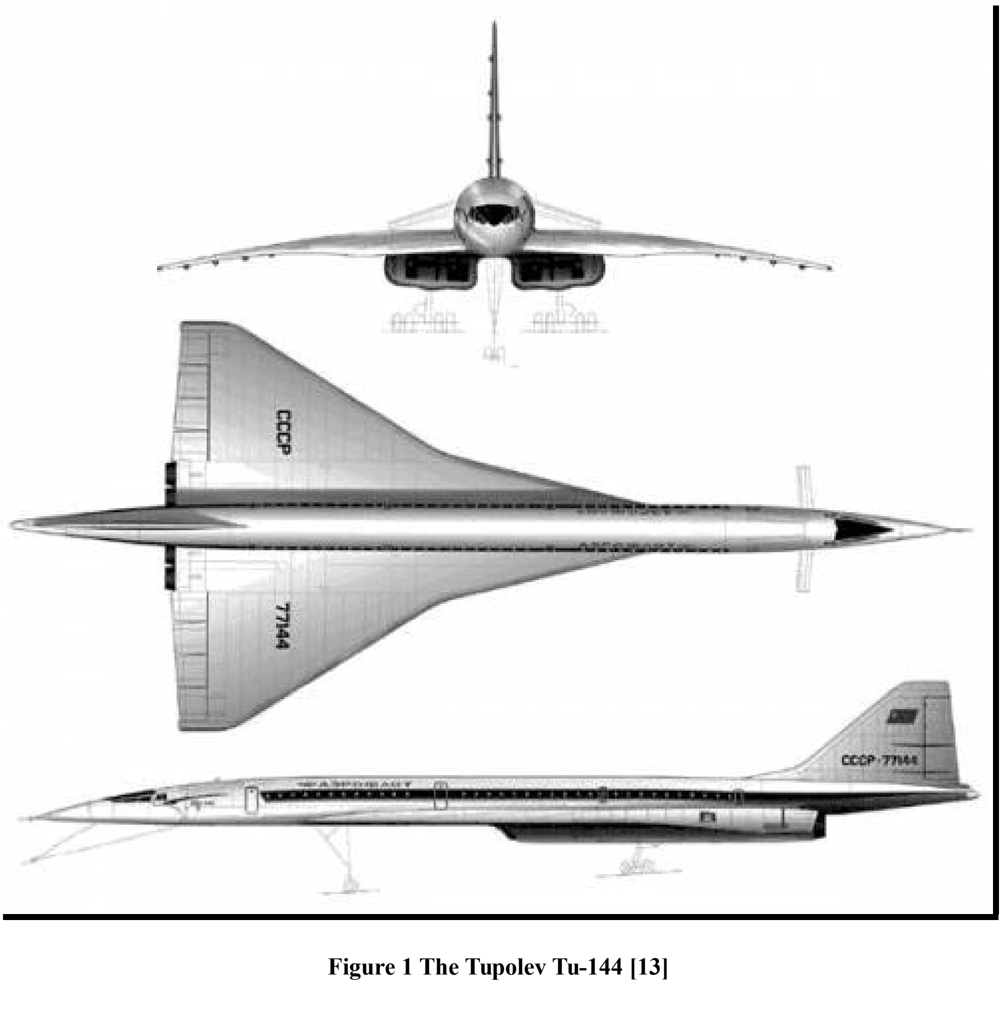

Figure 1 The Tupolev Tu-144 [13]

2.1.2 CONCORDE

Concorde was the second civilian SST to go into service and took its first flight on March 2, 1969 [14]. It remained in service for an amazing 27 years. The Concorde was manufactured by Aerospatiale and the British Aircraft Corporation. The number of aircraft built was 20 with a total cost of $1.3 billion for all the units. Tables 3 and 4 explain the general and performance characteristics of the Concorde. Figure 2 represents the configuration layout of the Concorde.

Table 3 General Characteristics of the Concorde [14]

Crew	3
Capacity	92–120passengers (128 in high-density layout)
Length	202 ft 4 in (61.66 m)
Wingspan	84 ft 0 in (25.6 m)
Height	40 ft 0 in (12.2 m)
Fuselage internal length	129 ft 0 in (39.32 m)
Fuselage width	maximum of 9 ft 5 in (2.87 m) external 8 ft 7 in (2.62 m) internal
Fuselage height	maximum of 10 ft 10 in (3.30 m) external 6 ft 5 in (1.96 m) internal)
Wing area	3,856 ft^2 (358.25 m^2)
Empty weight	173,500 lb (78,700 kg)
Useful load	245,000 lb (111,130 kg)
Power plant	4 × Rolls-Royce/SNECMA Olympus 593 Mk 610 afterburning turbojets
Dry thrust	32,000 lbf (140 kN) each
Maximum fuel load	210,940 lb (95,680 kg)

Table 4 Performance Characteristics of the Concorde [14]

Maximum speed	Mach 2.04 (1,354 mph) at cruise altitude
Cruise speed	Mach 2.02 (1,340 mph) at cruise altitude
Range	3,900 nmi (4,500 mi, 7,250 km)
Service ceiling	60,000 ft (18,300 m)
Rate of climb	5,000 ft/min (25.41 m/s)
lift-to-drag	Mach 0.94–1.47, Mach 2.04 – 7.14
Fuel consumption	46.85 lb/mi (13.2 kg/km) operating for maximum range
Thrust/weight	0.373
Maximum nose tip temperature	260 °F (127 °C)

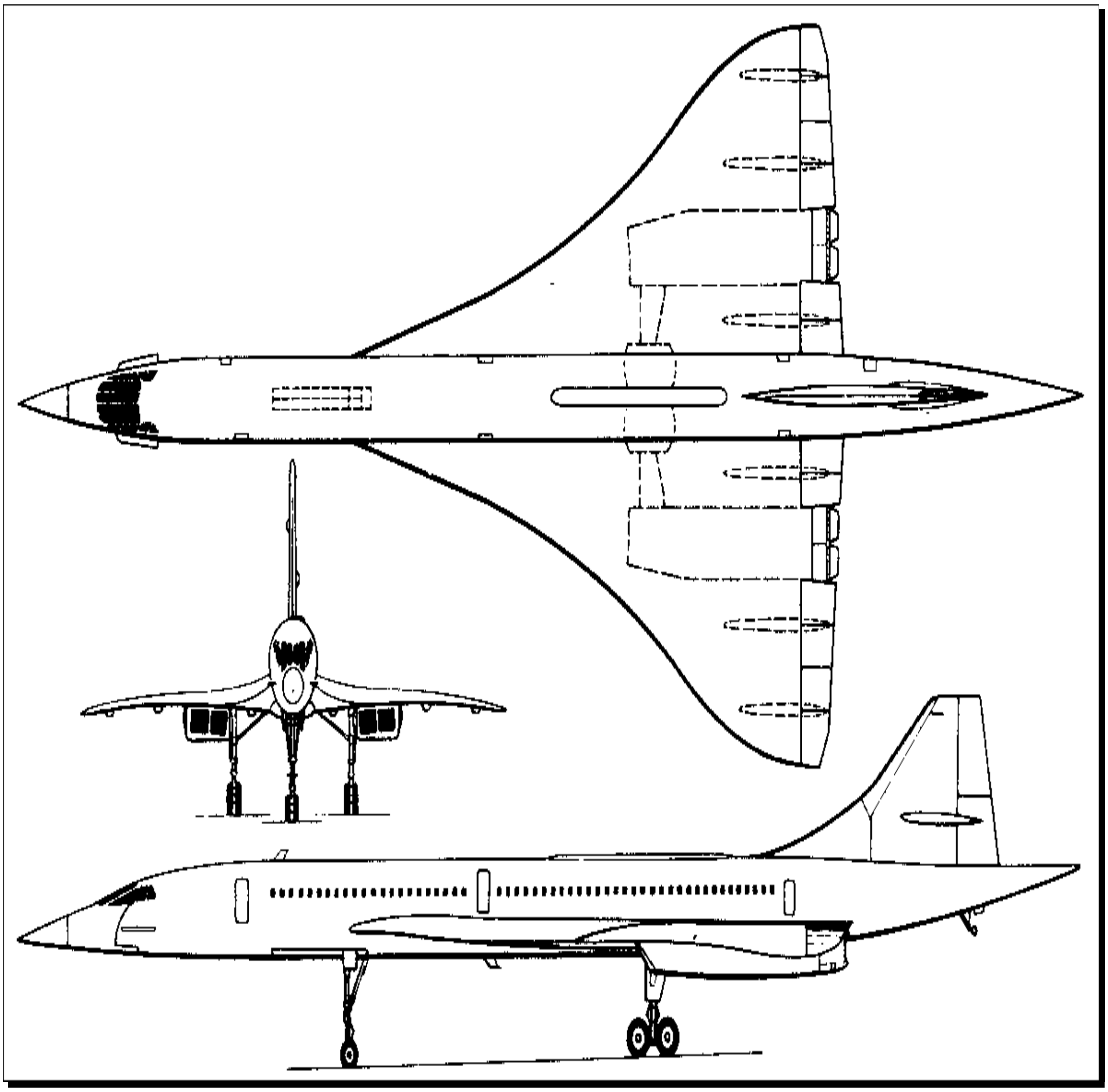

Figure 2 The Concorde [15]

2.1.3 LOCKHEED L-2000

The Lockheed L-2000 featured a compound delta planform and a long fuselage with engines padded under the wing. The Lockheed L-2000 was judged simpler to produce, and its performance was slightly lower and its noise levels slightly higher [16]. By 1966, the design took on its final form as the L- 2000-7A and L-2000-7B. The L-2000-7A with redesigned wing and fuselage lengthened to 273 ft (83 m).

The redesigned fuselage allowed for a mixed-class seating of 230 passengers. The new wing had a large wing twist and curvature. Despite having the same wingspan, the wing area was increased to 9,424 ft² (875 m²), with a slightly reduced 84° sweepback, and an increased 65° main delta wing, with reduced forward sweep along the trailing edge. The leading-edge flap increased the lift at low speeds, which allowed a slight down elevon- (elevator and aileron) deflection. The greater length of fuselage improved its fineness ratio and resulted in reduced drag. The ventral fin present underneath the trailing edge of the fuselage made the aircraft directionally stable. The L-2000-7B was extended to 293 ft (89 m) to reduce the chance of the tail striking the runway. Tables 5 and 6 represent the general characteristics and the performance characteristics of the Lockheed L-2000. Figure 3 represents the configuration layout of the Lockheed L-2000.

Table 5 General Characteristics of the Lockheed L-2000 [16]

Capacity	273 passengers
Length	273 ft 2 in (83.26 m)
Wingspan	116 ft (35.36 m)
Wing area	9,424 ft² (875 m²)
Empty weight	238,000 lb (107,900 kg)
Max. takeoff weight	590,000 lb (276,600 kg)
Power plant	4 × GE4/J5M or Pratt & Whitney JTF17A-21L

Table 6 Performance Characteristics of the Lockheed L-2000 [16]

Cruise speed	Mach 3.0
Range	4,000 nmi (7,400 km)
Service ceiling	76,500 ft (23,317 m)
Wing loading	62.61 lbs/ft^2

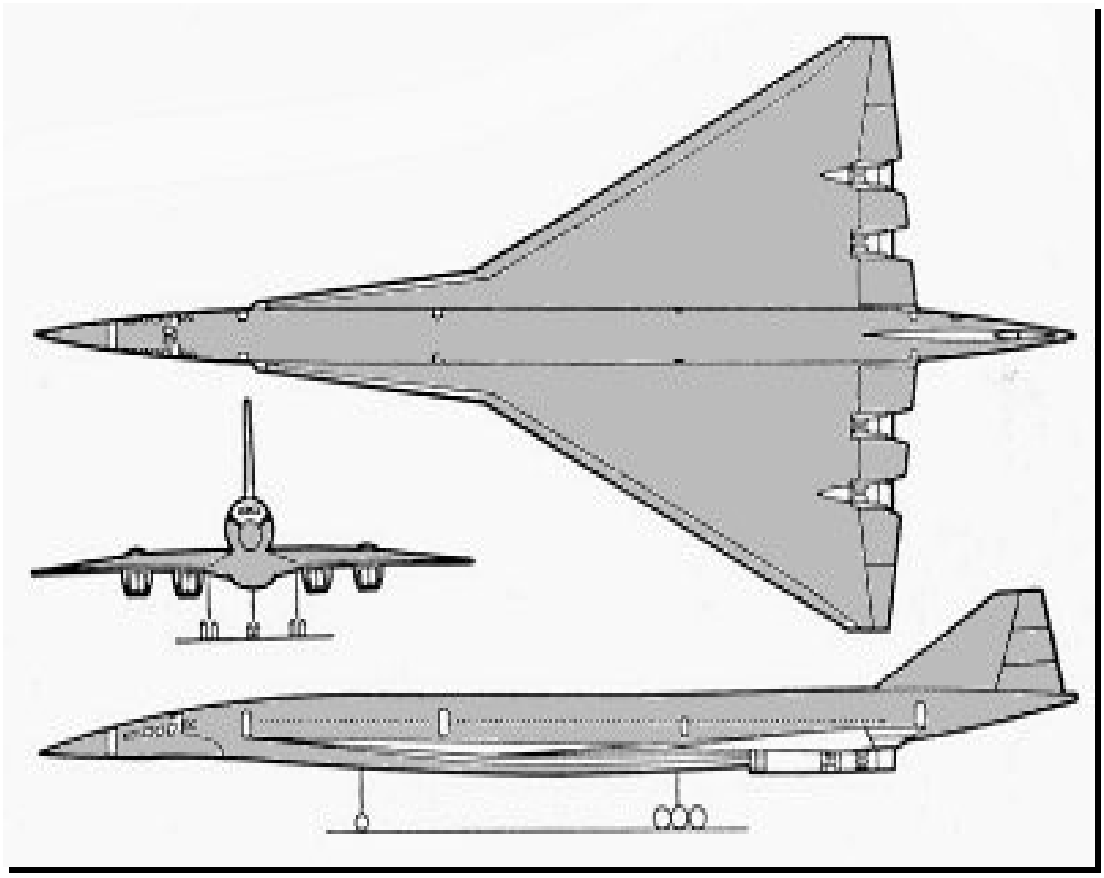

Figure 3 The Lockheed L-2000 [17]

2.1.4 BOEING 2707

The Boeing 2707 was the first supersonic aircraft design in the U.S. [18]. The motivation for the early development of the Boeing 2707 was that supersonic flights would allow the airliners more trips compared to subsonic flights, increasing thus their utility. However, environmental and economical issues combined once more to stop this program from producing a feasible concept. Table 7 represents the general and performance characteristics of the Boeing 2707. Figure 4 represents the configuration layout of the Boeing 2707.

Table 7 Characteristics of the Boeing 2707 [18]

Power plant	Four General Electric GE4/J5P turbojets, each of 63,200 lb. st (28677 kgf) each, with augmentation.
Empty Operating Weight	287,500 lb (130308 kg)
Max. Ramp Weight	675,000 lb (306175 kg)
Max. Landing Weight	430,000 lb (195045 kg)
Max. Payload	75,000 lb (34020 kg)
Normal Cruising Speed	Mach 2.7 1,800 mph (2900 km/h) at 64,000 ft / 21000m
Range	4,250 miles (6840 km) with 277 passengers
Takeoff Length	5,700 ft (1870 m)
Landing Length	6,500 ft (2133 m)
Span	180 ft 4 in (54.97 m) spread, 105 ft 9 in (32.23 m) swept.
Length	306 ft 0 in (93.27 m)
Height	46 ft 3 in (14.1 m)

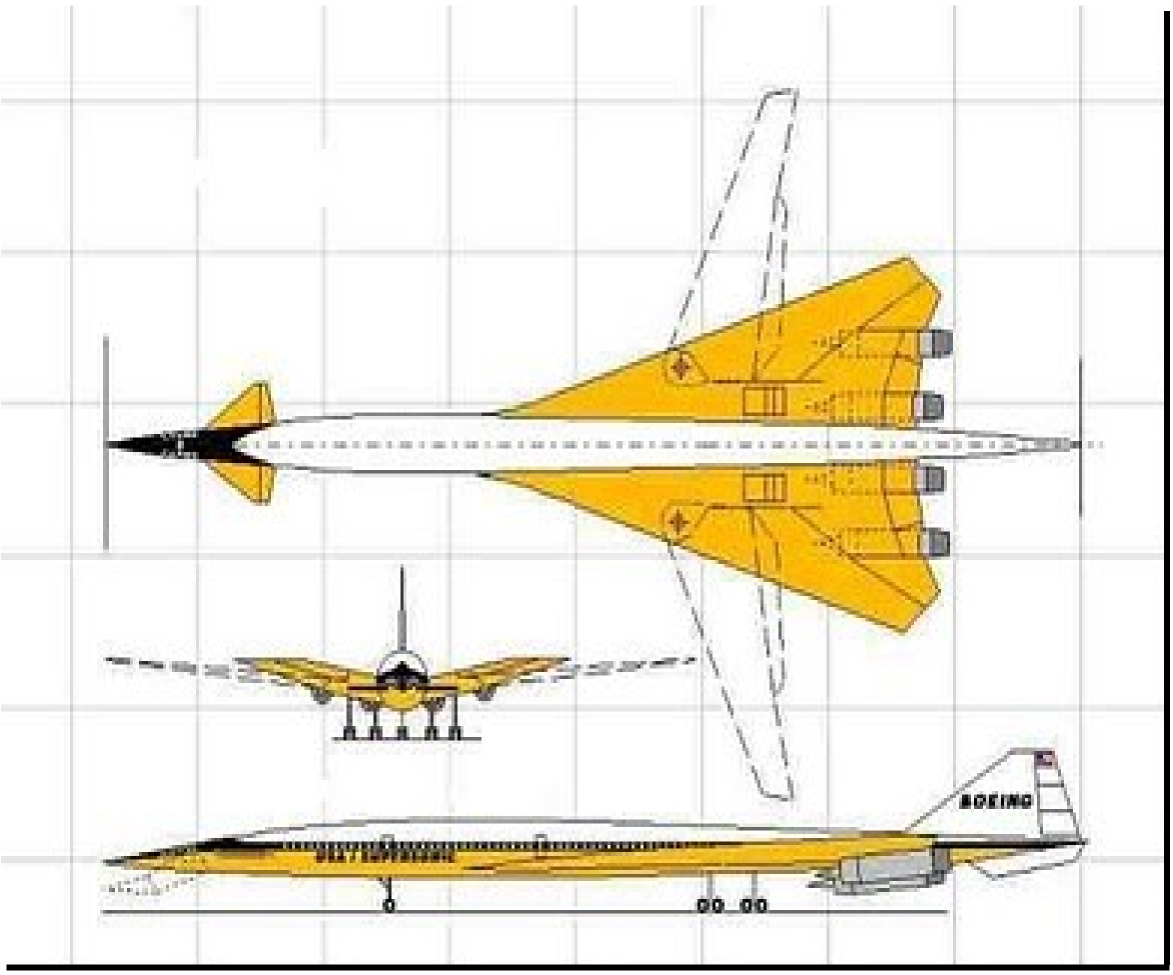

Figure 4 The Boeing 2707 [18]

3 MISSION REQUIREMENTS AND PROFILE

3.1 PAYLOAD CAPACITY

The proposed supersonic design aircraft is expected to carry 337 passengers weighing 175 lbs each and with a baggage weight of 50 lbs per passenger.

3.2 CREW MEMBERS

The proposed SST will have a cockpit crew of 3 and 10 cabin attendants weighing 175 lbs each and with a luggage weight of 50 lbs per crew member.

3.3 CRUISE SPEED

Mach 3.7 cruise speed is chosen for the proposed concept, as it appears to be the most efficient speed for an SST from the comparison of similar aircraft.

3.4 CRUISE ALTITUDE

According to FAR requirements, the minimum cruise altitude for supersonic flight is 42,000 ft. Similar Aircraft were compared, and the cruise altitude is set to 70,000 ft.

3.5 RANGE & ENDURANCE

The flight range is 8,700 miles at cruise speed. The duration is calculated to be 3 hours and 45 minutes.

3.6 MISSION PROFILE

The mission profile for the supersonic design aircraft is shown in Figure 7.

PHASE 1: Taxi out, Take-off, and Climb to 2000 ft.

PHASE 2: Subsonic cruise

PHASE 3: Acceleration and Climb to 35,000 ft

PHASE 4: Transonic cruise

PHASE 5: Acceleration and Climb to 70,000 ft

PHASE 6: Supersonic cruise

PHASE 7: Descent to 35,000 ft

PHASE 8: Subsonic cruise

PHASE 9: Approach, Landing, Taxiing

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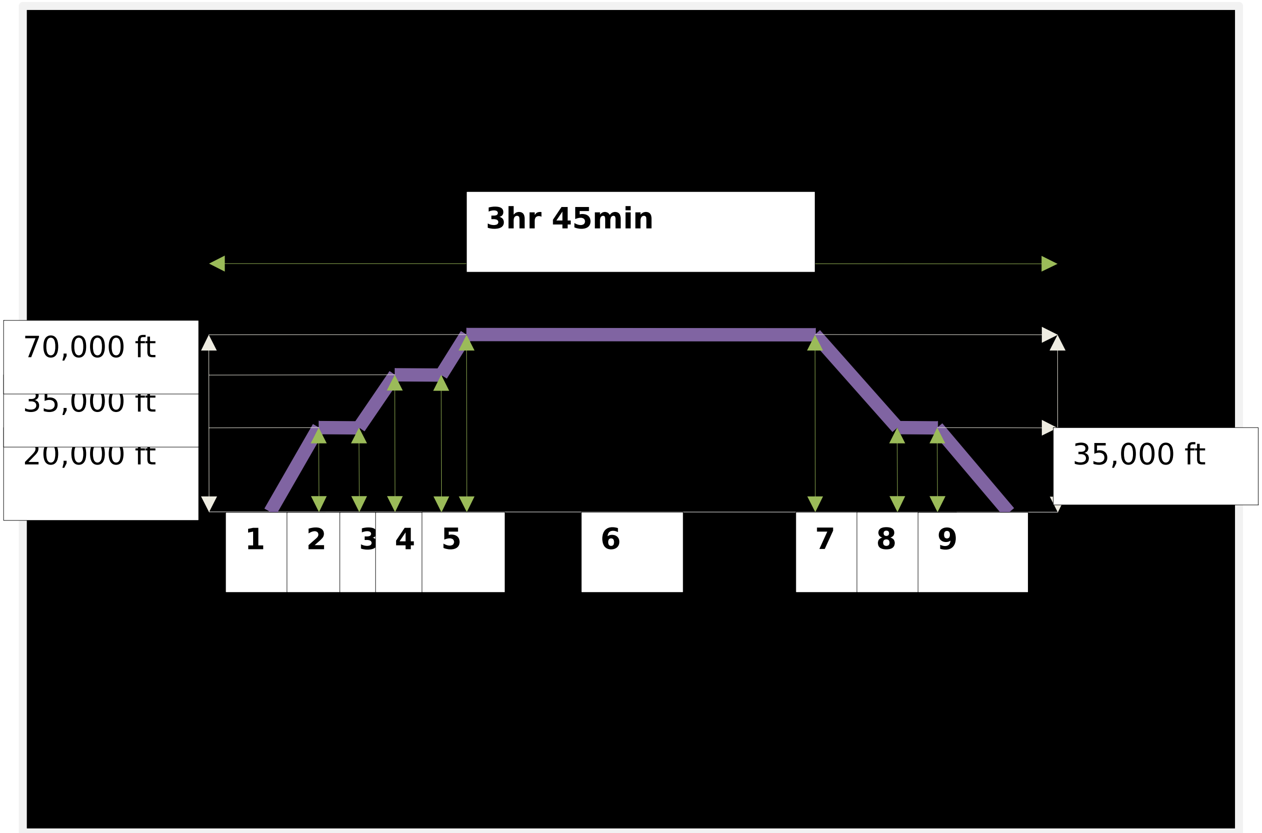

Figure 5 Mission Profile

4 MARKET ANALYSIS

NASA has identified specific environmental and technological performance objectives that have to be met in order to build have an economically viable and environmentally acceptable SST [19]. Aerion and Gulfstream performed an analysis for the need of supersonic jets and found that around 350 aircraft will be needed in 10 years [20].

Teal group conducted a case study and found that 400 jets will be needed in 20 years [21]. These case studies clearly demonstrate the demand for supersonic jets.

5 CONSTRAINTS

5.1 SOCIAL & ECONOMIC

Although the proponents of the supersonic transport assure that 50,000 jobs would be created, the rate of return would be less compared to the investment in other fields the nation would have attained without it [22]. Economic factors are not nearly as limiting for business jets as they are for commercial transport. Prospective manufacturers believe the market will support paying about twice as much for a supersonic aircraft that can cruise at twice the speed of current subsonic business jets.

5.2 ENVIRONMENTAL

The shock wave created by a supersonic aircraft, as it flies through the air propagates to the ground and causes what is known as the sonic boom. The sonic boom must eliminated or reduced to acceptable levels for an SST to be environmentally acceptable. Noise produced near the airports during takeoff, climb out, approach, and landing is also a concern as is high altitude emissions.

5.3 POLITICAL

Generally, the government does not invest money in the SST projects due to fewer profits compared to other projects in different fields. This political factor also depends on economic, social, environmental, and technical factors.

5.4 TECHNICAL

The development of the best SST configuration is based on the structures, the advanced airframe materials, high lift-to-drag ratio (L/D), the propulsion airframe integration, and acceptable takeoff and landing characteristics. Gulfstream and Lockheed Martin Skunk Works are well matched to tackle the challenge of defining a supersonic business jet. Lockheed Martin had 1997 sales surpassing $28 billion. In 1997, Gulfstream reported revenue of $1.9 billion [23]. Although the initial stages require more finance, the technology has many economic advantages to the country and the people.

5.5 SUSTAINABILITY

After the retirement of the Concorde, the sustainability of supersonic jets is low [24]. Factors that had an impact on sustainability are limited seats, expensive tickets, and fuel costs, which made the SST unattractive. Sustainability will greatly improve if a quieter, cheaper, and more fuel efficient SST becomes available.

6 COMPARATIVE STUDY OF SIMILAR AIRPLANES

Table 8 represents a comparison of all the aircraft discussed in the previous sections.

Table 8 Comparative Study of Similar Airplanes

	Tupolev Tu-144	Concorde	L-2000 A	BOEING 2707
Crew	3	3	3	3
Capacity	70~80 passengers	92–120passengers	273 passengers	277 passengers
Length	215.54 ft	202 ft 4 in	273 ft 2 in)	306 ft 0 in
Wingspan	94.48 ft	84 ft 0 in	116 ft (35.36 m)	
Height	34.42 ft	40 ft 0 in	-	46 ft 3 in (14.1 m)
Wing area	4,715 ft²	3,856 ft^2	9,424 ft²	-
Empty weight:	85,000 kg (187,400 lb)	173,500 lb (78,700 kg)	238,000 lb (107,900 kg)	287,500 lb (130308 kg)
Useful load	120,000 kg (264,555 lb)	245,000 lb (111,130 kg)		675,000 lb (306175 kg)
Power plant	4 × Kolesov RD-36-51 afterburning turbojet	4 × Rolls-Royce/SNECMA Olympus 593 Mk 610 afterburning turbojets	4 × GE4/J5M or Pratt & Whitney JTF17A-21L	Four General Electric GE4/J5P turbojets
Cruise speed:	Mach 2.15 (2,285 km/h (1,420 mph))	Mach 2.04 (≈1,354 mph, 2,179 km/h) at cruise altitude	Mach 3.0	Mach 2.7
Range:		3,900 nmi (4,500 mi, 7,250 km)	4,000 nmi (7,400 km)	4,250 mls (6840 km)
Service ceiling	(65,600 ft)	60,000 ft	76,500 ft	-

7 CONFIGURATION DESIGN

7.1 FUSELAGE

The Sears-Haack body shape is finalized as the fuselage shape because, theoretically, it produces less wave drag and it satisfies area rule [25]. A double-bubble cross-sectional layout is considered because it holds separate areas for cabin and cargo.

7.2 WING

Considering the compressibility effects, the wing is swept to delay transonic drag rise. The aft sweep is considered to be the conventional type because of its wide use in subsonic, transonic, and supersonic range. It helps in the reduction of maximum cross-sectional area of the wing [25].

Although swept wing has many advantages, the wing weight increases with increasing wingspan. It causes a nose up pitching moment when the wing stops lifting behind the center of gravity, as in the case of tip stall. Placing the engine location on pylons below the wing corrects this drawback. This arrangement allows the engine weight to counteract the wing lift, reducing the wing root bending moment, resulting in a lighter wing. The engine location is designed in such a way that there is essentially no adverse aerodynamic interference. Forward sweep is not used because for a supersonic transport, volumetric wave drag is high due to the Mach cut, which results in a lower effective fineness ratio.

7.3 SELECTION AND INTEGRATION OF THE PROPULSION SYSTEM

The selection of a propulsion system is finalized by comparing it with the speed-altitude envelope [26]. Supersonic cruise requires high thrust specific fuel consumption, maintenance, and low bypass ratio turbofan considering maximum thrust and minimum fuel consumption. Four RB-199s are used to generate the thrust for the maneuver.

7.4 CONTROL CONFIGURATION

Considering high speed performance and longitudinal control stability, all flying is chosen as the best empennage configuration [25]. Directional stability and control is achieved using a conventional vertical tail and rudder. Lateral control stability is achieved by ailerons and spoilers.

7.5 LANDING GEAR TYPE AND DISPOSITION

The landing gear is a conventional twin wheel nose and main gear units [25]. The nose gear is configured as fuselage mounted, which folds forward ahead of the flight deck. The main gear is configured as wing mounted tandem units, which retract sideways into the wing/fuselage unit. The disposition of the landing gear enables the aircraft to rotate at take-off speed. It also prevents the tail from scraping during rotation. This arrangement results in low trimmed drag. The visibility of the pilot is clear because the nose of the aircraft is in level condition.

8 MISSION WEIGHT ESTIMATES

8.1 MISSION PAYLOAD WEIGHT ESTIMATION

The payload weight is based on baggage, cargo, and the number of passengers. The passenger-(175 lbs. per person) and baggage weight-(50 lbs. per person) is assumed to be 225 lbs. The crew members are based on FAR-25, satisfying minimum requirements [25]. For long distance flights, the aircraft is expected to have 3 cabin members and 10 attendants weighing 175 lbs. each with luggage weight of 50 lbs. per person.

$W_{PL} = W_P + W_B + W_C$

W_{PL} - payload weight

W_P - passenger weight

W_B - baggage weight

W_C - cargo weight

8.2 REFERENCE TAKE-OFF WEIGHT ESTIMATION

The guess take-off weight is found by comparing the design aircraft mission specifications with similar mission specifications aircraft.

8.3 MISSION FUEL WEIGHT ESTIMATION

The mission fuel weight is the sum of the fuel reserve and fraction of fuel needed for the mission to be accomplished [25]. Sometimes, the aircraft may need extra fuel to loiter or land in a different location, and therefore, some amount of fuel is reserved for that purpose.

$W_{FUEL} = W_{RES\_FUEL} + W_{MISSION\_FUEL}$

W_{FUEL} - total amount of fuel weight used for the mission

W_{RES_FUEL} - the weight of fuel reserved for emergency during the mission

$W_{MISSION\_FUEL}$ - the weight of fuel used for the mission

Fuel fraction is the method used to calculate the amount of fuel used during the mission. It is the ratio of the end weight the beginning weight. The fuel fraction is calculated for various phases of the mission profile.

The fuel fraction for phase 1 is $W_1/W_{TO} = 0.992$

The fuel fraction for phase 2 is $W_2/W_1 = 0.909$

The fuel fraction for phase 3 is $W_3/W_2 = 0.985$

The fuel fraction for phase 4 is $W_4/W_3 = 0.8335$

The fuel fraction for phase 5 is $W_5/W_4 = 0.920$

The fuel fraction for phase 6 is $W_6/W_5 = 0.4927$

The fuel fraction for phase 7 is $W_7/W_6 = 0.920$

The fuel fraction for phase 8 is $W_8/W_7 = 0.837$

The fuel fraction for phase 9 is $W_9/W_8 = 0.911$

The Mission fuel fraction is calculated using the formula,

$$M_{ff} = \left(\frac{W_1}{W_{¿}}\right) \prod_{i=1}^{i=7} \left(W_{i+1}/W_i\right) \quad = 0.2139$$

The Mission fuel weight is calculated using the formula,

$$W_{MISSION_{FUEL}} = \left(1 - M_{ff}\right) W_{¿} \quad = 0.7861 \quad W_{¿}$$

8.4 EMPTY WEIGHT ESTIMATION

The empty weight is given by the formula [25],

$$W_{¿} - A$$
$$\log_{10} ¿/B\}$$
$$¿$$
$$\{¿$$
$$W_E = inv.\log_{10} ¿$$

8.5 TAKE-OFF WEIGHT ESTIMATION

The take-off weight is given by the iteration process that when the tentative empty weight and empty weight has less difference, the take-off weight is finalized. An analytical approach of using the formula can be done to calculate the estimated take-off weight [25]. Table 9 represents the summary of the mission weights.

$$\log_{10} W_{¿} = A + B \log_{10}\left(C W_{¿} - D\right)$$

A = 0.4221; B = 0.9876

$$C = \left\{1 - \left(1 + M_{res}\right)\left(1 - M_{ff}\right) - M_{tfo}\right\} = 0.7811$$

$$D = W_{PL} + W_{CREW} = 61{,}250\, lbs$$

Table 9 Summary of Mission Weights

Fuel Weight	656,868 lbs
Crew Weight	2275 lbs
Empty Weight	370642.315 lbs
Payload Weight	58,975 lbs

9 TAKEOFF WEIGHT SENSITIVITIES

9.1 SENSITIVITY OF TAKE-OFF WEIGHT TO PAYLOAD WEIGHT W_{PL}

The substitution of A, B, C, and D in the analytical equation gives a take-off weight of 1,097,250 lbs. The growth factor due to payload is calculated using the formula from Roskam part-I and found to be 8.94 [25]. Thus, for each pound of payload added, the aircraft take-off gross weight should be increased by 8.94 lbs. In this case, when the mission performance is the same, the aircraft growth factor is said to be 8.94.

9.2 SENSITIVITY OF TAKE-OFF WEIGHT TO EMPTY WEIGHT W_E

The substitution of A, B, and W_{TO} in equation 2.29 of Roskam part-I results in the value 2.237 [25]. For each pound increase in the empty weight, the take-off weight should be increased by 2.237 lbs to keep the mission performance the same. The factor 2.237 is called the growth factor due to empty weight for this supersonic transport.

9.3 SENSITIVITY OF TAKE-OFF WEIGHT TO RANGE, ENDURANCE AND SPEED

Range (R), Endurance (E), and Speed (V) are specified in the mission specifications for the jet transport. For the supersonic transport, the following data are found. The sensitivity of W_{TO} to Range given is 51.1 lbs/nm [25]. The sensitivity of W_{TO} Endurance is 125,170 lbs/hr. The sensitivity of W_{TO} to speed is -89.8 lbs/knot.

10 PRELIMINARY SIZING OF ALL REQUIREMENTS

10.1 SIZING TO STALL SPEED REQUIREMENTS

For supersonic aircraft, there are no requirements for minimum stall speed in the case of FAR-25 certified airplanes [25]. By assuming a maximum allowable stall speed for a given value of maximum lift co-efficient, the wing loading can be found using the relation below. It is found that the maximum lift co-efficient depends on the wing and airfoil selection, flap type and size, and the center of gravity location.

$$V_S = \sqrt{2\left(\frac{W}{S}\right)/\rho C_{L_{max}}}$$

During the preliminary sizing, the maximum lift co-efficient is assumed to be consistent with mission requirements and the types of flaps and slats deployed. Table 10 gives calculation for stall speed sizing.

Table 10 Sizing to Stall Speed Requirements

Vs	Cl_max	W/S
130	2.2	125.91
130	2.4	137.35
130	2.6	148.8

10.2 SIZING TO TAKE-OFF DISTANCE REQUIREMENTS

The take-off distance of the supersonic aircraft depends on the take-off weight, take-off speed, thrust-to-weight ratio, aerodynamic drag co-efficient, ground friction, and pilot technique [25]. The take-off of the supersonic aircraft is assumed to take place on a hardened surface, such as concrete or asphalt.

The take-off requirements are based on FAR-25, which is generally known as ground run requirements in combination with minimum climb capability. The sizing for the take-off distance is calculated using Roskam part-I [25]. The passenger aircraft is required to have a take-off field length less than 5,000 ft at sea-level conditions. Table 11 illustrates a range of $(W/S)_{TO}$, $(T/W)_{TO}$ and $C_{L_{max}}$ for which the field length requirement is satisfied.

Table 11 Sizing to Take-off Distance Requirements

(W/S) Cl_max	**(W/S) Cl_max**	**(W/S) Cl_max**	**(W/S) Cl_max**	**(W/S) Cl_max**	**(W/S) Cl_max**		**(T/W)**
1	**1.2**	**1.6**	**2**	**2.2**	**2.4**		
0.30	0.25	0.19	0.15	0.14	0.13		40
0.45	0.38	0.28	0.22	0.20	0.19		60
0.60	0.50	0.38	0.30	0.27	0.25		80
0.75	0.63	0.47	0.38	0.34	0.31		100
0.90	0.75	0.56	0.45	0.41	0.38		120
1.05	0.88	0.66	0.53	0.48	0.44		140

10.3 SIZING TO LANDING DISTANCE REQUIREMENTS

The landing distance requirements are always based on the landing weight of the aircraft. Landing distance is calculated using the relation between the landing weight and take-off weight from Table 3.3 of the Roskam part-I [25]. The landing distance of an aircraft depends on landing weight, approach speed, deceleration method, flying qualities of the aircraft, and pilot's technique.

The FAR-25 landing field length is the ratio of the total landing distance and 0.6. This 0.6 factor of safety is included for variations in pilot technique and other critical conditions. The landing distance (S_L) is assumed to be 3,889.62 ft. The landing field length (S_{FL}) is calculated to be 6483 ft. The approach speed is 1.3 times the stall speed (V_{SL}). Using S_{FL}, the approach speed is calculated. The V_{SL} is used to calculate (W/S) for various values of the $C_{Lmax\_landing}$. The below equation gives the relation between the landing lift co-efficient and (W/S). Table 12 explains the relationship between the landing lift co-efficient and the wing loading.

$$\left(\frac{W}{S}\right)=33.4\,C_{L_{max_{landing}}}$$

Table 12 Sizing to Landing Distance Requirements

$C_{L_{max_{landing}}}$	$\left(\frac{W}{S}\right)$
1.9	108.74
2.1	120.18
2.3	131.63

10.4 SIZING TO RATE-OF-CLIMB REQUIREMENTS

To size an aircraft for climb requirements, it is necessary to have a drag polar for the aircraft [25]. The FAR-25 requirements are met for the supersonic design aircraft. The zero lift drag co-efficient equation is given below.

$$C_{Do}=f/S$$

$$f \rightarrow equivalent\ parasite\ area$$

$$S \rightarrow Wing\ area$$

The drag polar for a clean aircraft can be determined using the take-off weight [25]. The effects of flaps and landing gear are taken into account for the calculation of the drag polar. The zero lift drag due to flaps and landing gear is also added to the total drag.

Using the drag polar equations:

At 1.2 V_{STO}, C_D = 0.1639 (FAR 25.111 (OEI))

At $V_{LOF = 1.1}$ V_{STO}, C_D = 0.232 (FAR 25.121 (OEI) (gear down, take-off flaps up))

At V_2, C_D = 0.125 (FAR 25.121 (OEI) (gear down, take-off flaps up))

At 1.2 $V_{STO,}$ C_D = 0.109 (FAR 25.121 (OEI) (flaps up, gear up))

At 1.25 V_{SA}, C_D = 0.058 (FAR 25.121 (OEI) (flaps up, gear up))

At 1.3 V_{SL}, C_D = 0.1937 (FAR 25.119 (AEO) (Balked Landing))

At 1.25 V_{SA}, C_D = 0.1194 (FAR 25.121 (OEI) (Balked Landing))

10.5 SIZING TO TIME-TO-CLIMB REQUIREMENTS

There is a linear relationship between the rate-of-climb and altitude. The rate-of-climb depends on the engine of the aircraft and the speed, at which the climb occurs [25]. The rate-of-climb at a given altitude is given below as

$$h_{|¿|}$$
$$1-\frac{h}{¿}$$
$$RC = RC_o ¿$$

For a supersonic aircraft, $\overset{|¿|}{h_{¿}}$ ranges from 55-80 ft*10^{-3}. The rate-of-climb can be related to (T/W) and (W/S) using the below equation [25]. Table 13 gives sizing to climb.

$$RC = \left[\frac{2\left(\frac{W}{S}\right)}{\left\{\frac{\rho\left(C_{Do}\pi Ae\right)^{\frac{1}{2}}}{2}\right\}}\right]^{\frac{1}{2}}\left\{\left(\frac{T}{W}\right)-\frac{1}{\frac{L}{D}}\right\}$$

Table 13 Sizing to Climb Requirements

W/S	T/W	T/W	T/W
	1200	2500	2000
40	0.39	0.76	0.62
60	0.28	0.52	0.43
80	0.22	0.40	0.33
100	0.18	0.33	0.28

120	0.16	0.29	0.24
140	0.15	0.25	0.21

10.6 SIZING TO CRUISE SPEED REQUIREMENTS

A cruise speed of Mach 3.7 at sea level is desired for take-off . Since the cruise speed is very high, the effects of increased drag are taken into account [25]. The wetted area is found from the take-off weight and assuming a low skin friction, the parasitic area is found. The wing area is calculated by taking an arbitrary wing loading value. From the area of the wing the C_{Do} value is calculated. By assuming an Aspect ratio and Ostwald efficiency factor, the (T/W) and the (W/S) relations for which the cruise speed requirement is met is calculated. Table 14 shows the relationship between (T/W) and (W/S) for which the cruise speed requirement is met.

Table 14 Sizing to Cruise Speed Requirements

(W/S)	(T/W)
60	0.6986754
70	0.5989705
80	0.524206
90	0.4660686
100	0.41957
110	0.3815361
120	0.3498507

10.7 SIZING TO CEILING REQUIREMENTS

The absolute ceiling is 0 fpm for the minimum climb rate. The service ceiling is 500 fpm for the minimum climb rate [25]. The combat ceiling for supersonic aircraft occurs at M > 1 with minimum climb rate of 1,000 fpm, and the cruise ceiling for supersonic aircraft occurs at M > 1 with a minimum climb rate of 1,000 fpm. The below equations are used to get (T/W) and (W/S) relations for which the ceilings requirement is met.

$$h_{|¿|}$$
$$1-\frac{h}{¿}$$
$$RC=RC_o¿$$

$$RC=\left[\frac{2\left(\frac{W}{S}\right)}{\left\{\frac{\rho\left(C_{Do}\pi Ae\right)^{1/2}}{2}\right\}^{1}}\right]^{1/2}\left\{\left(\frac{T}{W}\right)-\frac{1}{\frac{L}{D}}\right\}$$

11 MATCHING OF ALL SIZING REQUIREMENTS

The sizing of all the requirements are overlaid on each other and the best combination is selected for the lowest possible thrust-to-weight ratio and highest possible wing loading. This process of obtaining the best design point is also known as the matching process. Figure 6 provides the matching graph for all sizing requirements. Table 15 provides the summary of the results obtained from the design point.

Table 15 Summary of Performance Sizing graph

Take-off weight	1,097,250 lbs
Area	9500 ft^2
Thrust Required	482,790 lbs
Take-off Cl	2.1
Landing Cl	2.2
R/C	2000 ft/min
Stall Speed	130 knots

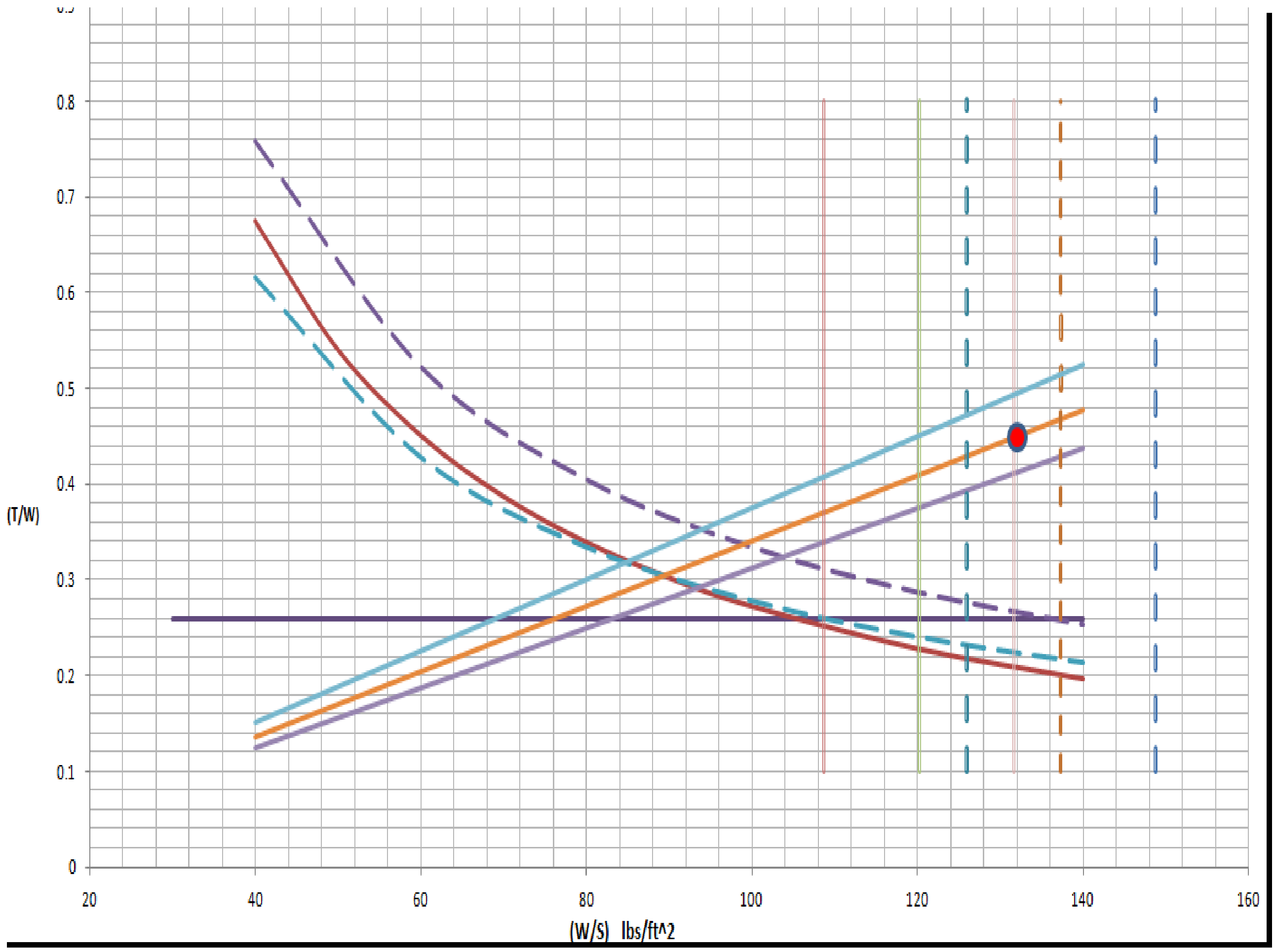

Figure 6 Performance Sizing graph

12 PRELIMINARY DESIGN

12.1 FUSELAGE LAYOUT

The shape and dimensions of the aircraft have very great impact on the wave drag generated. The fuselage design of a supersonic aircraft depends on the wave drag which increases rapidly as the fuselage volume increases. The fuselage design of the Boeing SST and Concorde were used to get an idea for reduction in drag [15] [27]. The fuselage layout was designed using chapter 2 in preliminary design sequence I and chapter 4 in preliminary design sequence II [28]. The cabin consists of 3 classes of passengers. The first class is seated as 2-2-2, the business class is seated as 2-3-2, and the economy class is seated as 3-3-3. First class consists of 6 rows, the business class consists of 15 rows and economy consists of 22 rows. Table 16 gives the summary of the fuselage dimensions.

Table 16 Summary of Fuselage Dimensions

Fuselage Diameter	22 ft
Fuselage Length	297ft
Fuselage Area	3250 ft^2
Cabin width	21.26 ft
Cabin height	7.7 ft
Cabin length	136 ft
Seat width	16.5 inch
Aft body + fore body	53ft

12.2 SIZING FOR HIGH LIFT DEVICES

The fowler flap is chosen as the best type based on weight and maximum co-efficient of lift during take-off and landing. The calculations are performed based on preliminary design process [29]. The range of the fowler flap varies from 1 to 1.3 which is enough to produce an incremental lift co-efficient. The take-off deflection angle is 10 degrees, and the landing deflection angle is 40 degrees. The flap size parameter is 0.9, and the flap chord ratio is 0.31.

12.3 WING DESIGN

A mid swept back wing is chosen due to high speed and compressibility effects. The step- by-step process given in chapter 6 of preliminary design sequence is used for determining the following plan form design characteristics of the wing [29]. The taper ratio, dihedral angle, incidence angle, and sweep angle were obtained from similar aircraft configuration [28]. Table 17 gives the summary of the wing parameters.

Table 17 Summary of Wing Parameters

Wing Area	9500 sq. ft
Aspect Ratio	2
Wing Span	135 ft
Sweep Angle	65 degree
Taper Ratio	0.12

Dihedral angle	5 degree
Chord root	125 ft
Chord tip	15ft
Mean aerodynamic chord	70 ft

12.4 EMPENNAGE DESIGN

The calculations are based on chapter 8 of class I design process [29]. The distances of the horizontal and vertical stabilizers from the center of gravity are 95 ft and 98 ft. Table 18 gives the summary of the horizontal stabilizer parameters. Table 19 gives the summary of the vertical stabilizer parameters.

Table 18 Summary of Horizontal stabilizer Calculation

AR	2
Taper ratio	0.35
Sweep angle	50 degree
Area	1194 sq. ft
Span	48ft
Chord	24 ft

Table 19 Summary of Vertical stabilizer Calculation

Area	991 sq. ft
AR	1.5
Taper ratio	0.40
Sweep angle	45 degree
Span	39 ft
Chord	25 ft

12.5 AIRFOIL SELECTION

The airfoil selection is based mainly on the ideal lift co-efficient and maximum lift co-efficient required during take-off and landing. It is also based on attached flow over the wing. Since the camber and thickness of the airfoil leads to more drag, the airfoil chosen is thin symmetrical airfoil. The best supersonic airfoil is based on maximum lift and considerable amount of reduced drag that could make the mission achievable. Many airfoil combinations were studied to produce high lift and low drag.

After a detailed analysis of different types of supersonic airfoils, the diamond shaped airfoil and bi-convex airfoil were further analyzed in detail. The bi-convex airfoil was found to reduce the bow wave by keeping the flow attached to the leading edge of the airfoil. At 2 degrees angle of attack, the lift co-efficient required is obtained using class I design process [29].

The symmetrical airfoil with 3 % thickness, with leading edge slats of 0.17 % of the chord, and trailing edge flaps of 0.26 % of the chord provides the necessary lift. Figure 7 represents the airfoil shape used in the wing design process.

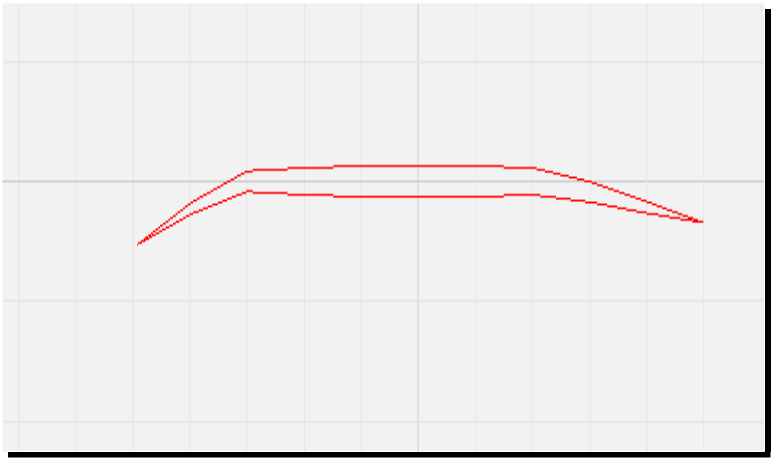

Figure 7 Airfoil design using slats and flap

12.6 LANDING GEAR DESIGN

The landing gear characteristics such as number, type, size of tires, length and diameter of struts, preliminary disposition, and retraction feasibility are found using the Class I design [28]. Table 20 provides summary for static load per unit strut calculation. Table 21 gives summary for landing gear parameters.

Table 20 Summary of Static load per unit strut Calculation

Pn/Wto	0.06
Pm/Wto	0.97

Table 21 Landing gear Calculation

Nose gear length	20ft
Main gear length	25 ft
Main gear distance from the nose tip	165 ft
Nose gear distance from the nose tip	65 ft
Main gear tire area	49 * 17 inch
Nose gear tire area	46* 16 inch
Number of nose gear tires	4
Number of main gear tires	6

13 WEIGHT AND BALANCE ANALYSIS

The weight of each component and the distance of each component from the aircraft nose are tabulated. The weight and balance method used to calculate the center of gravity for various scenarios are based on class I design [28]. The moment of each component was calculated from the nose of the cockpit to the center of gravity of each component. Table 22 shows the weight of each component and the distance from the nose to each component. Figure 8 shows the center of gravity weight excursion diagram.

Table 22 Weight and Balance Analysis summary

Component	Weight (lbs)	x(inches)	Wx(lbs. inches)
Wing	80000	1120	89600000
Empennage	11000	2280	25080000
Fuselage	55000	180	9900000
Nacelles	17000	1100	18700000
Landing gear	27000	240	6480000
power plant	60000	1120	67200000
Empty Weight	371000	1130	419230000
Fuel weight	657000	1130	742410000
Crew weight	2350	420	987000
Luggage+Payload	68000	720	48960000

The center of gravity for various scenarios such as take-off weight, take-off weight with fuel weight, empty weight, and with 50% passengers were calculated and analyzed. The final center of gravity excursion diagram is shown in the figure10.

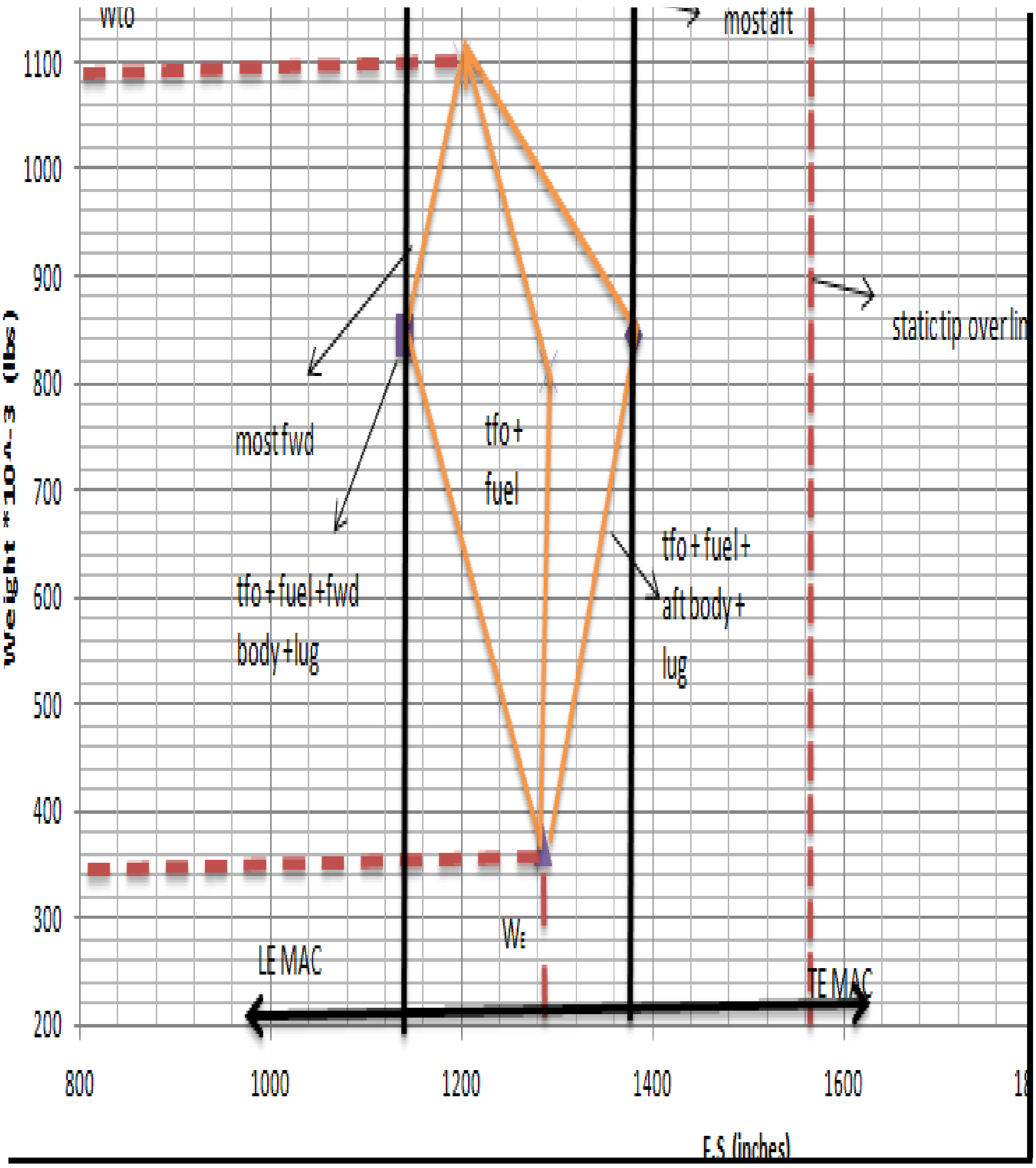

Figure 8 CG Excursion diagram

14 STABILITY AND CONTROL ANALYSIS

14.1 STATIC LONGITUDINAL STABILITY

The static longitudinal plot is used to find the horizontal stabilizer area with respect to a certain amount of static margin. The method used to obtain the X-plot is based on class I preliminary design [30]. The relationship between the horizontal stabilizer area and weight is given in the below equation.

$$W_h = 0.0034\left\{ (W_{¿})^{0.813} (S_h)^{0.584} \left(\frac{b_h}{t_{rh}}\right)^{0.033} \left(\frac{c}{l_h}\right)^{0.28} \right\}^{0.915}$$

The equation below is used to obtain the values for the horizontal stabilizer area and aerodynamic center relationship.

$$X_{ac} = \frac{X_{ac_{wf}} + \dfrac{C_{L_{\alpha_h}}\left(1 - \dfrac{\partial \varepsilon_h}{\partial \alpha}\right)\left(\dfrac{S_h}{S}\right) X_{ac_h}}{C_{L_{\alpha_h}}}}{1 + \dfrac{C_{L_{\alpha_h}}\left(1 - \dfrac{\partial \varepsilon_h}{\partial \alpha}\right)\left(\dfrac{S_h}{S}\right)}{C_{L_{\alpha_{wf}}}}}$$

The below three equations are used to support the above equations under static longitudinal stability. The horizontal stabilizer area from the stability plot is 2000 ft^2. The initial design horizontal stabilizer area is 1194 ft^2. Since these two values are close the horizontal stabilizer area will be incremented to 800 ft^2 for better stability control. Figure 9 shows the static longitudinal X-plot.

$$C_{L_{\alpha_{wf}}} = \left(1 + 0.025\left(\frac{d_f}{b}\right) - 0.25\left(\frac{d_f}{b}\right)^2\right)\left(\frac{2\pi A}{2 + \left[\left(\dfrac{A^2 B^2}{K^2}\right) + 4\right]^2}\right)$$

$$K = C_{L_\alpha} M \sqrt{(1 - M^2)} / 2\pi$$

$$\frac{2l_h}{b}$$

$$\left(1-\frac{h_h}{b}\right)/\sqrt[3]{¿}$$

¿

$$\left(\frac{1}{A}-\frac{1}{1+A^{1.7}}\right)\left[\frac{10-3\lambda}{7}\right][\sqrt{\cos\theta_{0.25}¿}]$$

¿

¿1.19

¿

4.44¿

$$\frac{\partial\varepsilon_h}{\partial\alpha}=¿$$

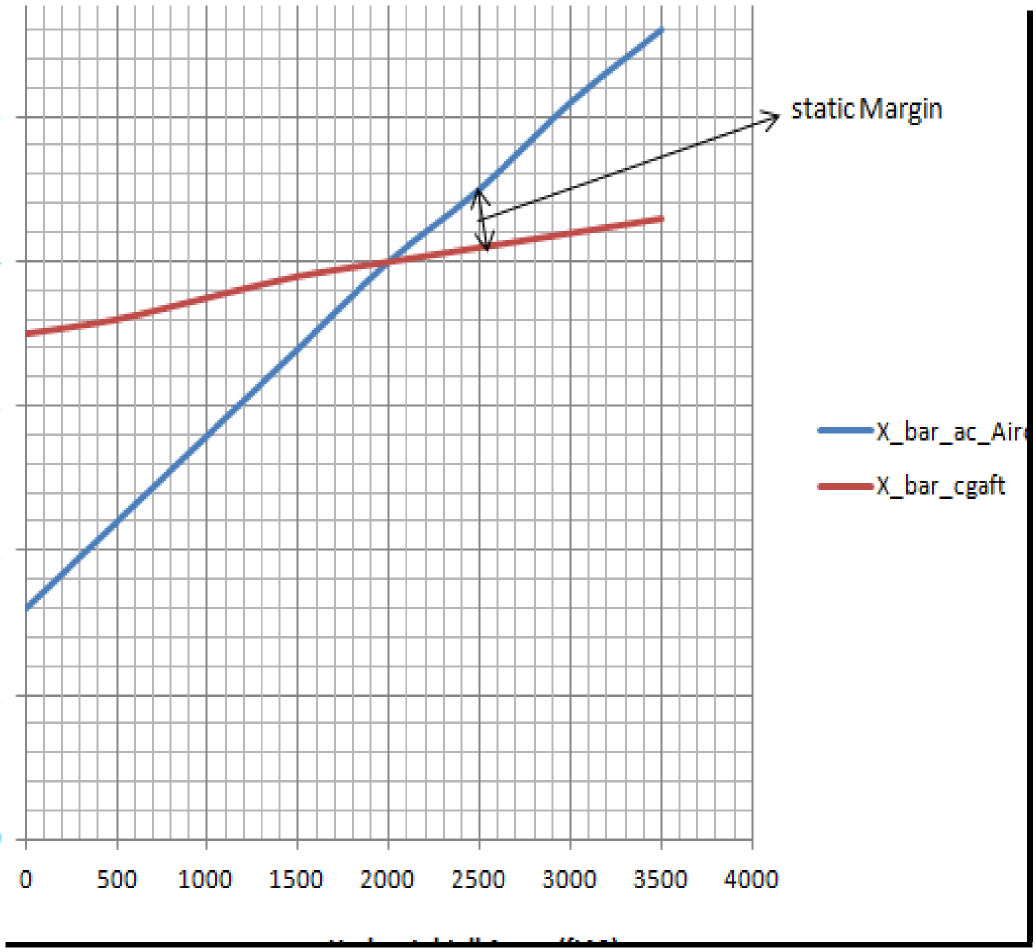

Figure 9 Static Longitudinal X-plot

14.2 STATIC DIRECTIONAL STABILITY

The relationship between the yaw side-slip moment co-efficient and vertical stabilizer area is given in the below equation. The method used to obtain the vertical stabilizer area is based on class I design [30].

$$C_{n_\beta} = C_{n_{\beta_{wf}}} + C_{L_{\alpha_v}} \left(\frac{S_V}{S}\right)\left(\frac{X_V}{b}\right)$$

The yaw side slip angle is assumed to be zero at high angle of attack. Therefore the yaw side-slip moment co-efficient of the fuselage reduces to the equation below.

$$C_{n_{\beta_f}} = -57.3 K_N K_{R_l} \left(\frac{S_{f_s} L_f}{Sb}\right)$$

The value of K_N is determined from the graph which shows the relationship between wing-fuselage interference with respect to directional stability. The value of K_{R_l} is determined from the graph which shows the relationship between effects of fuselage Reynolds number with respect to the wing-fuselage directional stability. Figure 10 shows the directional stability X-plot obtained using the above two equations. The preliminary vertical stabilizer area calculated is 991 square ft. The vertical stabilizer area obtained from the X-plot is 1100 square ft. The lesser area is chosen in order to reduce the weight of the vertical stabilizer.

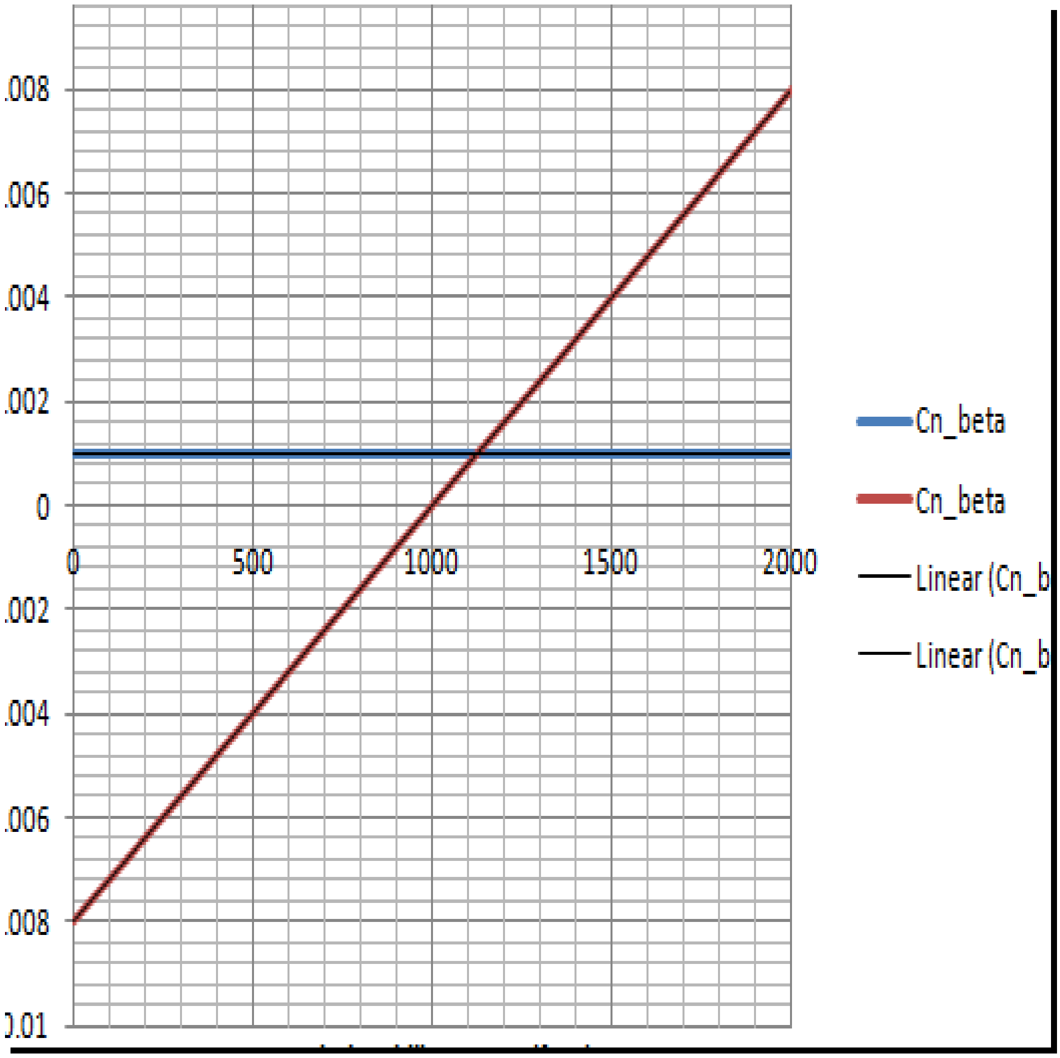

Figure 10 Static Directional X-plot

15 DRAG POLAR

The overall drag co-efficient is calculated using the zero lift drag co-efficient. The total wetted area is calculated to get the zero lift drag co-efficient. The parasitic area is also calculated in order to find the value of the zero lift drag co-efficient [29]. A 10% drag caused due to interference is also added to the total drag.

The estimated total wetted area is 16,886 ft^2, and the parasitic area estimated is 45 ft^2. The total drag is estimated for various configurations such as ideal, take-off, and landing. The below equations give the relationship between the drag polar and the lift co-efficient used to obtain the values for various configuration [28]. Table 23 shows the zero-lift drag co-efficient estimated for various configurations. Table 24 shows the relationship between lift and drag for different configurations.

$$C_{D_o} = f / S_{wetted surface}$$

$$C_D = C_{D_o} + \frac{C_L^2}{\pi e AR}$$

Table 23 Zero Lift Drag co-efficient for various configurations

Configuration	C_{DO}
Clean	0.03
Take-off	0.035
Landing	0.043

Table 24 Drag Polar Analysis for Different Aircraft Configuration

Configuration	C_D	C_L	L/D
Clean	0.13	1.8	13.8
Take-off	0.15	2.0	12.5
Landing	0.19	2.1	11.0

16 PRELIMINARY DESIGN LAYOUT

Table 25 provides the summary of the supersonic aircraft layout parameters. Figure 11 shows the front view of the supersonic aircraft. Figure 12 shows the side view and the top view layout of the supersonic transport.

Table 25 Preliminary Design Results

	Wing		Horizontal tail		Vertical tail
Area	9500 ft^2		1194 ft^2		991 ft^2
Span	137 ft		48 ft		39 ft
MGC L.E.	69 feet 3 inch		24 ft 9 inch		25ft 4 inch
Aspect Ratio	2.0		1.9		1.5

Sweep Angle	65 degree		65 degree		45 degree
Taper Ratio	0.12		0.35		0.4
Thickness Ratio	0.03		0.03		0.12
Airfoil	bi-convex		bi-convex		bi-convex
Dihedral angle	5 degree		0 degree		not appl.
Spoiler hinge ratio	1				
Spoiler chord ratio	0.9				
Spoiler Span ratio	0.45				
Flap Chord Ratio	0.2				
Flap Span ratio	0.3				
	Fuselage		**Cabin Interior**		
Maximum Length	297 ft		213 ft		
Maximum Height	19 ft		8 ft		
Maximum width	20 ft		16 ft		

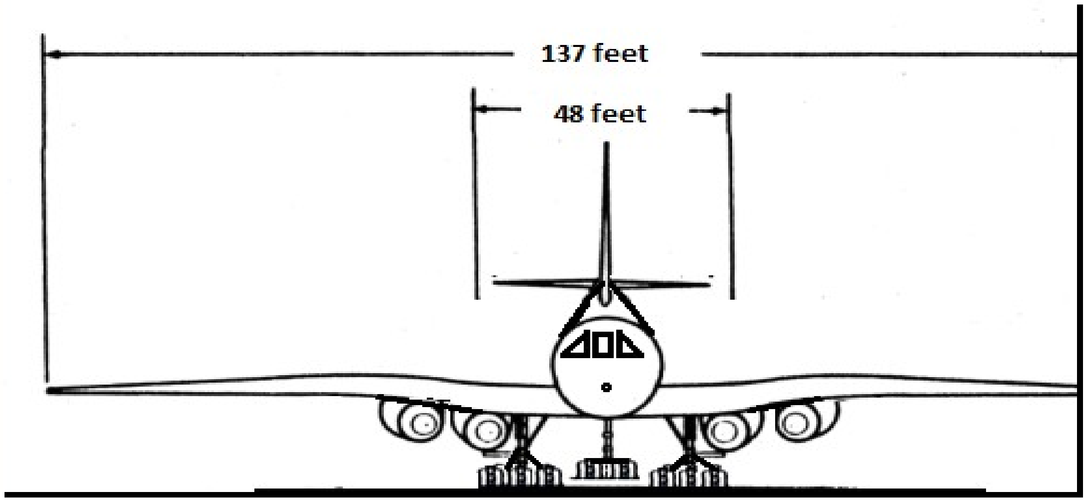

Figure 11 Front view of the SST

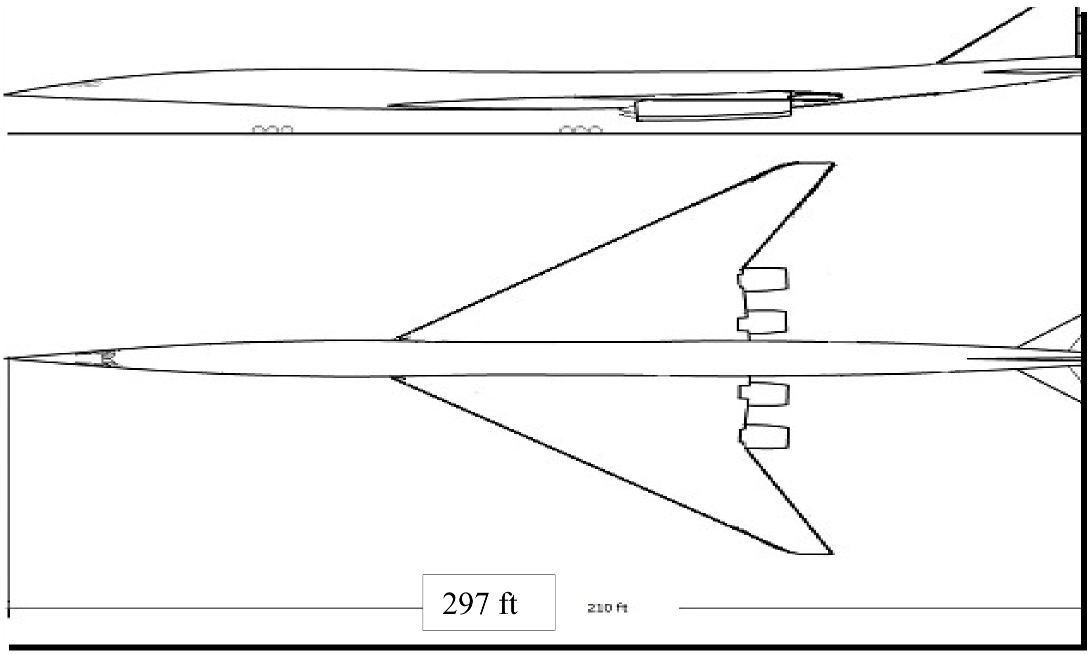

Figure 12 Side view and Top view of the SST

17 CONCLUSION/RECOMMENDATIONS

This thesis has given a brief summary of a Class I preliminary design for an SST. The nominal design point chosen seems to be reasonable although the thrust to weight ratio seems to be high. The shape of the aircraft nose is designed based on the nose cone modification method. Therefore, the SST design holds good for acceptable noise level. The center of gravity estimation holds good for subsonic and supersonic speeds. Though the preliminary design of the SST is safe, the design has to be refined more before it reaches the market. The SST drag is reduced based on the conventional design process but the material incorporated will not insulate the structure at high speeds. The design process should be taken to next level by considering advancements in structure and prevention of aerodynamic heating inside the structure. A detailed design process will incorporate all possible limitations and challenges to produce an efficient SST design.

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