ANALYSING AND MINIMIZATION OF SONIC BOOM IN SUPERSONIC COMMERCIAL AIRCRAFT

[*1] Bijay kumar sah, [2]Keerthana Rathinavel, [3]Naveenkumar. M, [4]Ashir Azar , [5]Dr.Venkatesan Saradha Paramashivaiah

[1,2,3,4] UG, Final year student, Department of Aeronautical engineering, Excel engineering college
[5]Associate Professor (HOD), Department of Aeronautical engineering, Excel engineering college
Salem Main Road, Komarapalayam, Namakkal-637303, Tamilnadu, India

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ABSTRACT

The designing and analysing of the process is carried out on a supersonic commercial aircraft with delta wing configuration, which aims to provides depth knowledge of aircraft gross weight calculation, fuel weight, empty weight, wing span, wing area, aspect ratio, lift to drag ratio. As using Carlson method, we get sonic boom overpressures and signature duration results. A simplified method of calculation of sonic boom is been performed both numerically and analytically.

The numerically predicted results are in good agreement with that of experiment data. Sonic-boom overpressures and N-wave signature duration has been predicted for the entire affected ground area for aircraft in level flight path using Carlson theory. In this both theoretical and computational aspect of aerodynamic, structure, propulsion and weight values of an entire aircraft are calculated for the design. Modelling of aircraft is carried out using CATIA V5 where computational method is been performed using ANSYS FLUENT software

Keywords: Supersonic, Aerodynamic aspect, Commercial aircraft, Computational analysis, Carlson theory

1. INTRODUCTION

Over the last few years, air traffic has seen a high increase in capacity, range and efficiency. In addition to these, the need for a faster aircraft. Carry of people and other payloads over a longer distance within a shorter interval of time will be the aim of the most airline industry in the near future. Conceptual design of an aircraft involves many different steps. The Commercial Supersonic aircraft is Structure using CATIA V5 and analysis is been carried out using ANSYS FLUENT software. This program was made up of components that evaluated each performance parameter of the aircraft while solving for the smallest maximum take-off weight. Reducing noise and disturbance, this boom signature is to be limited.

Creation of an environmentally friendly aircraft is an important design parameter. The high Mach number which the Supersonic aircraft will travel at will create the sonic boom overpressure at sea level. Reducing noise and disturbance, this boom signature must be limited. Work was done to analyse and choose the most efficient aerodynamic shape. The sonic boom of the aircraft was approximated using methods described by Carlson. Whereas, Sonic boom approximation by Sea bass provided a rough estimate of boom overpressure using simply aircraft weight, more in-depth analysis was done using Carlson which considers geometric parameters and area distribution of the aircraft. Iteration of design yields an aircraft that satisfies the design mission of 4600km cruise, and an overpressure of 0.547. Control surface area is estimated by using analysis done by Dr. Roskam. The necessary surface area of the delta wing is determined using simple numerical approximations. Key constraints necessary for the aircraft to operate safely under normal flight conditions were analysed for the aircraft configuration. From the thorough analysis of aircraft component create a concept that both satisfied major requirements set in the opportunity description.

Since the retirement of Concorde's airline service in 2003, there is no more civil supersonic transport. The environmental concerns (sonic boom, noise, emissions, etc.) raised by Concorde have been the major barriers for future civil supersonic aircraft. The fundamental problem preventing the return of supersonic flight is the sonic boom at ground level. However, the public's perceived acceptance of the sonic boom intensity is still uncertain. Since the weight and size of a supersonic aircraft have first-order effects on the intensity of the sonic boom signature, it has been deemed nearly impossible to create a low-boom level with a heavier and larger conventional aircraft. Though, for a wide range of customers the low-price airline is attractive, there still exist customers who attribute great value to time. Recognition of the value of time has led to increased interest in the feasibility of supersonic business jets. Double the cruise speed could result in half the time in the air.

Operational flexibility, safety and privacy working environment, and adequate ambience for fostering social contacts add value to supersonic business jets. The unit price and direct operation costs should be viable for both manufacturers and customers. The interest of supersonic civil flight is not only driven by enterprising human spirit or profit seeking but also by technology itself. The basic technical capability for the supersonic cruise has existed for decades and the technology has improved considerably since Concorde. The variable cycle engine concept and acoustic problems caused by the inlet and nozzle require more development to be solve. Sonic boom mitigation concepts still need further ground and flight testing. Therefore, environmentally friendly, economically viable and tech-nonlogically feasible characteristics are required for any future super-sonic airliner. There have been prominent publications on supersonic aircraft design review. The National Research Council analysed the design challenges and critical solutions appropriate to supersonic transport.

1.1 OPPORTUNITY DESCRIPTION

This aircraft sets the certain value of Mach no. 1.6, design range of 4600km with passengers accommodate of 48 and 4 crew members. Also, the aircraft is expected to achieve supersonic cruise efficiency, have a low sonic boom and high lift for take-off or landing making growth to the aviation industry. As, the requirement of the supersonic commercial aircraft is much in need so the design will lead to the massive growth.

1.2 MISSION STATEMENT

It is cost effective, advance and fastest mode of transport. It's expected to hold total number of 52 passenger including crew members. Sonic boom overpressure is estimated to be of 0.547 psf for the designed supersonic commercial aircraft with a signature duration of 0.3 sec.

2. LITERATURE REVIEW

Tupolev Tu -144 was known to be the first supersonic transport aircraft introduced two months before the Concorde with a cruise speed of 2,200 km/hr (1,400 mph, Mach 2). The design of both the airliner were slightly similar with same delta wing design configuration. Its regime ended soon as it experienced an in-flight failure during a pre-delivery test flight, crash-landing on 23 May 1978 with two crew fatalities. As of now the last two remaining aircraft is in Gromov Flight Research Institute in Zhukovsky.

Secondlythe British – French supersonic passenger aircraft with the Mach speed of 2.04 was first introduced on 21st Jan 1976. It took its first flight on 2nd march 1969 and retired in the year 2003. Companies like Airbus, Sud-Aviation (Aerospatiale), British Aircraft Corporation (BAC), Rolls – Royce/ SNECMA etc. were join under the Anglo-French treaty for manufacturing Concorde. Total number of 20 aircraft were built including 6 non-commercial aircraft. Concorde lasted in service for 27 years until 25th July 2000 when Air France flight 4590 was crashed and killed 113 people (109 on aircraft ,4 on ground). The accident was caused by a piece of metal left on the runway after falling from a continental jet. Therefore, after three years of accident the Concorde was retired in the year 2003. The total seating capacity of the aircraft was about 92-128 passengers, 3 crew members. Range of these aircraft was 7,222.8 km, cruising speed of 2,180 km/ hr (1,354 mph, Mach 2.04). Where, the maximum cruising altitude of the Concorde was 60,000ft (18,300m). Concorde was the only aircraft that travel faster than the speed of the sound. Because of this the air pressure and friction could really heat up the aircraft. So, Concorde was made up of high strength and high temperature aluminium alloy named RR58. The temperature at the nose was the highest about 127 degrees Celsius and was 91 degrees Celsius at the end of the fuselage.

Over these ongoing years we can visualize the air traffic has seen a high increase in capacity, range and efficiency. In addition to these, the necessity for a faster aircraft is essential at the present time. Transportation of people over a longer range within a minimum amount of time will be the main goal of the airline industry in the coming future. There are many current ongoing projects like BOOM, SPIKE, Lockheed Martin X-59 Quest expertizing in this field which are expected to be on service till 2024 later or sooner.

Since over the several years there has been a supersonic stagnation for decades, research on supersonic transport has never stopped. Based on the failures of the previous SST program, NASA was given the responsibility to establish the technology base for a viable supersonic cruise aeroplane. As part of the effort, the Supersonic Cruise Research (SCR)program was carried out from 1971 to 1981. The Variable Cycle Engine (VCE) program, a propulsion offshoot of SCR, was conducted from1976 to 1981 to study the promising VCE concepts. Feasibility studies for the next-generation SST were initiated in the late 1980s. The highspeed Research (HSR) program began by NASA in 1989, including in-depth studies from 1995 with Tu-144 test flights.

The High-Speed Civil Transport (HSCT) program, the focus of HSR program, aim developing a 300-passenger, Mach 2.4 supersonic airliner. The pro-gram terminated in 1999 on account of environmental challenges and budget problems. The Quiet Supersonic Platform (QSP) was conducted by the US Defence Advanced Research Projects Agency (DARPA) from 2000 to 2006, aimed at developing a low-boom (0.3 PSF) supersonic aircraft both for military and civil applications. In Europe, the next-generation supersonic research program was initiated in 1994 but was stopped in the same period as the HSR program because Europe turned to a large aeroplane. High Speed Aircraft (HISAC), also called environmentally friendly High-Speed Aircraft, was conducted from 2005 to 2009 to research the technology base of a small-size environmentally friendly supersonic transport. Japan Aerospace Exploration Agency (JAXA) initiated a scaled supersonic experimental aeroplane project named NEXST (National Experimental Supersonic Transport) project [16]in 1996 so as to establish advanced design technologies for the next-generation SST. The program ended in 2007. The Silent Supersonic Technology Demonstration (SSTD) program started in 2006 to validate MDO design tools and demonstrate the silent supersonic aircraft concept. The Drop test for Simplified Evaluation of Non-symmetrically distributed sonic boom (D-SEND) project [18,19] started in 2007 to drop models from balloons to validate the sonic boom mitigation technology. Since several decades of Research, it is transparent that a small-size supersonic transport could be the first step into a modern supersonic era. The increasingly stringent noise requirements have created the need for the supersonic jet to the quiet supersonic jet (QSJ) program.

2.1 WING DESIGN:

Sir James Hamilton the designer of the Concorde Ogival delta wing was appointed as the Britain Director – General of the Concorde Project in 1966.This wing design led to cruise the aircraft at twice of speed of the sound, yet provide safe take-off and landings. The feature of delta wing doesn't operate the factors of low speed in take-off, landing quite well. So, Ogival delta was the change for providing greater efficiency at low speed. To maintain efficient lift at low speed (take-off, landing) the angle of attack should be high, which leads to the drawback of high drag at low speed and flow separation at high angle of attack. So, the modification was an update of slender ogival delta wing making the flow separation slow. The idea of slender delta wing was proposed by Kuchemann and Weber, they published referring the strong vortices produced on the upper surface by the delta wing at high angle of attack. The vortex leads to the cause of increase lift, lowering air pressure. Weber found that with the increase in length of the wing, lift from the vortex will be increased. Thus, leading to extend the wing along the fuselage. This design led the drawback of low speed while take off and land due to nose high to generate the required vortex lift. Also, maximum power was required to fly at a low speed or high angle of attack. Flow separation or boundary layer separation is the separation of boundary layer that is formed by the relative motion between the fluid and a solid surface (aerofoil). The flow type of boundary layer can be calculated with the Reynolds no. Whereas, the laminar flow is independent to the Reynolds number, the turbulent flow increases with the increase in Reynolds number. The flow travels for long and stopped at a point and flow reverse, then the flow will be detached and forms of eddies and vortices leading to the increase in induced drag and reduction in lift. Which is generally caused by the pressure differential between front and rear.

Where for Tupolev the wing was replaced by Canard Delta with a double delta wing including spanwise and chordwise camber adding two small retractable surfaces (moustache canard). The advantage of this was to increase lift at low speed. The movement of elevons moving downwards in a delta wing increases the lift, but also pitches its nose downward. Placing the canards at the nose downwards moment thus helps reducing the landing speed of the production of Tu-144. It also had the fuel capacity of 98000kg to 125000kg.

2.2VORTEX FORMATION IN SLENDER DELTA WING:

When the angle of attack is high the built-in ability produces strong vortices over the upper surfaces, lowering air pressure and increasing lift. This effect is knowns as Vortex Lift. At, low speed the air swirling over the wing produced a bouncing motion, at a frequency of about half a second, which can sometimes mistake for light turbulence. This motion soon disappeared once the speed had increased after take-off, but was during final approach. Supersonic aircraft used thinner aerofoil in order to reduce drag and was also designed to fly with separated vortex flow, which is also a method of lift generation. The wing area and the wing span of the aircraft was 358.25m^2(3,856 sq. ft) and 84ft (25.6 m) respectively. The separated vortex flow will form a cone-shaped vortices which may control separation leading to not stall and increase angle of attack up to 40°. The vortices will soon fall off at the higher angles. This method of lift generation took over 5000 hours of wing tunnel tests.

2.3 SHOCK WAVES:

Shock waves are generated when the aircraft flies at the speed of the sound. The large pressure waves (750 miles/hr.) formed from the air flowing through the shock waves is the cause for Sonic Boom and are heard as a loud sound at the ground. Generally, the subsonic aircraft don't generate sonic boom as it flies below speed of sound and the air ahead of it makes a way before the aircraft reaches and avoids from the huge formation of pressure waves. Sonic boom generated by the Concorde was 1.94 psf (Pounds per sq. ft), at speed of Mach 2.

2.4 ENGINE:

The Engine used in Concorde was Olympus 593 produced by the Rolls Royce and SNECMA. There are Total four number of Rolls Royce/SNECMA engine were used in the aircraft. Each engine generates 18.7 tons(180kn) of thrust and the total fuel burned by the four engines was 6,771 gallons (25,629 litres) of fuel per hour.

Concorde had the total number of seventeen fuel tanks that hold 31,569 gallons (119,500 litres) of kerosene fuel. The fuel used was A1 jet fuel. The engine dimension was 1212 mm (47.75 inch) in diameter and length of 4039 mm (159 inch). Afterburning Kuznetsov NK-144 turbofan with a cruise SFC of 1.58 kg/kgp hr was the engine used in Tupolev

2.5MATERIALS USED:

Material used in Tupolev was 15% titanium and 23% non-metallic materials. Structural materials used were aluminum alloys, titanium or stainless steels. Concorde was made up of high strength and high temperature aluminium alloy named RR58. The temperature at the nose was the highest about 127 degrees Celsius and was 91 degrees Celsius at the end of the fuselage. Thus, Concorde used high- reflectivity white paint that having double reflective in comparison to the white paint that applied on the other jets. The head encountered by the Concorde caused the airframe to expand 7 inches (17.8 cm) in flight. Thus, special aluminium alloy (AV_2GN) light weight and more heat resistant than titanium were used. This alloy having relatively low density i.e., 7.75g/cc. The various Components for Concorde were manufactured by several company of the UK and France, and there were two assembly lines, one at Filton and one at Toulouse.

3.AIRCRAFT CONCEPT

3.1 AIRCRAFT DESIGN CONCEPT

A brief overview of key design parameters is given in table 1. The aircraft's maximum take-off weight of 66530.99lbs includes 48 passengers and 4 crew members including luggage.

Table 1. Key design parameters

Design Parameters	Value	Units
Aircraft (MTOW)	30177.95	Kg
Fuel weight fraction	0.299	
Empty weight fraction	0.522	
Wing area	132.201	m^2
Wing length	14.689	M
Wing span	18	M
Root chord	14.689	M
Aspect Ratio	2.4	
Aircraft length	120	Ft
Tip chord	0	
T/W	0.25	
t/c	10%	
MAC distance	3	M
MAC length	9.793	M
Fuselage Diameter	2.87	M

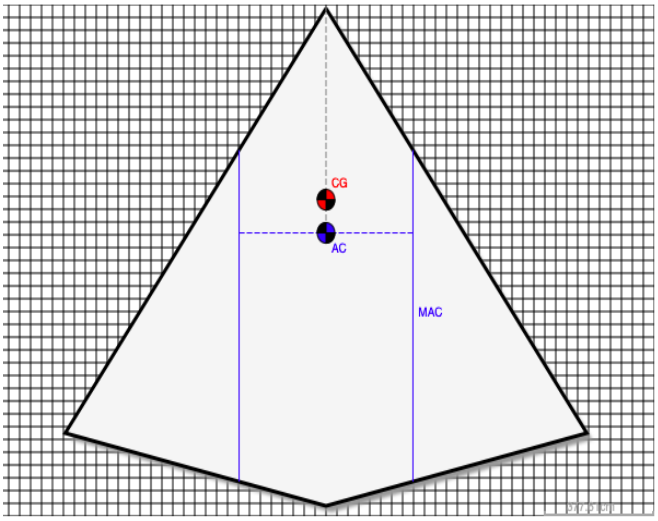


Fig 1: MAC and CG location

3.2 AIRCRAFT DESIGN MISSION

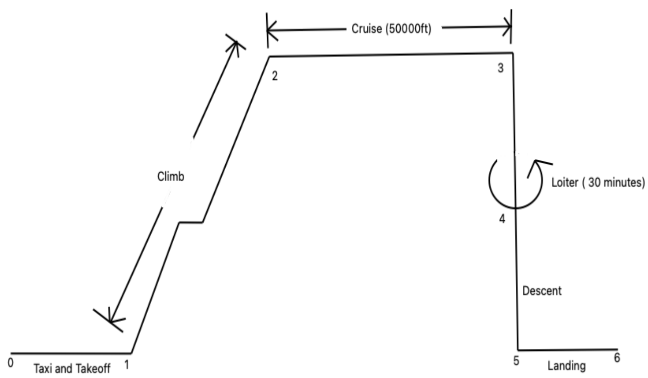


Fig2: Aircraft Design Mission

Table 2. Aircraft Design Mission Values

$\frac{W_1}{W_0} = 0.970$
$\frac{W_2}{W_1} = 0.985$
$\frac{W_3}{W_2} = 0.783$
$\frac{W_4}{W_3} = 0.966$
$\frac{W_5}{W_4} = 0.955$
$\frac{W_6}{W_5} = 0.997$

The above fig-2 represents the typical mission profile for the aircraft. This aircraft will climb to cruise altitude at best rate of climb right after take-off. The mission profile has accounted for air traffic control altitude restrictions in order to clear any traffic in the airspace if necessary. This is important especially when flying out from busy airports that are major worldwide hubs. The aircraft will then continue in a steady level cruise for 1227.63 mph at an altitude of 50000ft. The especially high altitude is essential to the achievement of a low sonic boom overpressure, as will be discussed in the sonic boom section. The mission profile includes a total loiter duration of 30 minutes in order to comply with FAA regulations.

4 AIRCRAFT DETAIL DESIGN

4.1 PRIMARY FUNCTIONS

4.1.1 A/c Geometry

The aim of the aircraft geometry function was to predict a mathematical identification of the aircraft that could be dynamically varies by giving a few key aircraft characteristics. The function used in aircraft, the aspect ratio and wing loading is to determine the second wing sweep and total wing area. The radial distribution of the fuselage is hardcoded. If the radial allotment of the fuselage predicted in extreme results, then the values would be varied, however they were not changed in an iterative manner primarily because of the rise in computation time associated with adding the extra ramification. Thus, the wing and fuselage designed, the performance then took a normal cross sectional area distribution and transported that to other programs. This program then come back to the area distribution and a wireframe depiction of the aircraft to the user. It is key to note that the mathematical presentation of the aircraft is three dimensional, thus wings had a thickness interpreted by the t/c ratio and the fuselage had a cross sectional area distribution predicted by the radial distribution over the aircraft length.

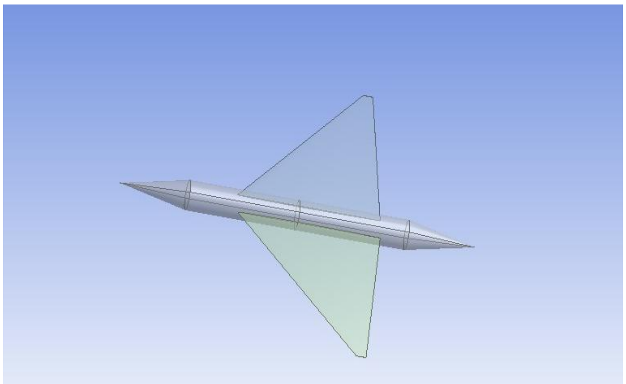


Fig3: Delta wing

To make the function work properly, it is necessary to hold several values constant. The sweep was 59 degrees, mean t/c was 10%, length of a/c was 120ft, and the beginning locations of the wing, engine nacelle, vertical tail and canard were all held constant. The general airplane calculation has been differed in a more complete way to yield a more powerful and point by point model, anyway since time is running short limitations for identication, it is infeasible to carry out such an estimating calculation. The region circulation should be proposed to unchanged in the absence of that capacity. This is undesirable, and therefore this function feed to the program by providing a more precise area distribution of the aircraft which varies with each aircraft configuration iteration.

4.1.2 Mission segment

As per the mission segments identified earlier in the Design Mission portion, the mission function is categorized into 6 parts. These parts include: taxi and take-off, climb, cruise, loiter, descent, and landing. Each and every flight condition has assumptions that go with it; these assumptions will be introduced below. It is important to note that for each flight condition, drag and thrust was calculated, whereby the aircraft is either accelerating or in an equilibrium.

Based on historical values presented in Raymer mission of the segments are taken for the take-off, climb and landing mission segments as illustrated in Table 2. Aircraft Design Mission Values.

CALCULATIONS OF AIRCRAFT MISSION PROFILE

Using Daniel P. Raymer, we get

1.Take-off weight build-up

$W_0 = W_{crew} + W_{Payload} + (W_f/W_o)\, W_o + (W_e/W_0)\, W_0$

$$W_0 = \frac{W_{crew} + W_{payload}}{1-\left(\frac{W_f}{W_o}\right)-\left(\frac{W_e}{W_o}\right)}$$

2.Empty Weight Estimation

$$\frac{W_e}{W_o} = AW_o^c$$

$$= 0.97W_o^{-0.06}$$

3.Fuel – Fraction Estimation

$$\frac{W_f}{W_o} = W_{tf}\left(1 - {W_n}/{W_o}\right)$$

4. Mission Segment Weight Fraction

$$\frac{W_n}{W_O} = \frac{W_6}{W_O} = \frac{W_1}{W_0} = \frac{W_2}{W_1} = \frac{W_3}{W_2} = \frac{W_4}{W_3} = \frac{W_5}{W_4} = \frac{W_6}{W_5}$$

a. Take off:

$$\frac{W_1}{W_0} = 0.970$$

b. climb:

W_0 Guess	W_e/W_o	W_o Calculated
50,000	0.507	29582.47
45,000	0.510	30047.12
43,000	0.511	30205.26
42,000	0.512	30365.08
40,000	0.514	30689.83
	Mean	30177.95

$$\frac{W_2}{W_1} = 0.985$$

c.cruise:

$$\frac{W_3}{W_2} = \exp\left[\frac{-R * TSFC}{V * \frac{L}{D}}\right]$$

$$= \exp\left[\frac{-15091863.5 * 0.00025}{1548.0315 * 9.959}\right]$$

= exp [-0.2447]

= 0.783

d. Descent:

$$\frac{W_4}{W_3} = \exp\left[\frac{-E * TSFC}{L/D}\right]$$

$$= \exp\left[\frac{-1800 * 0.000222}{11.5}\right]$$

= exp [-0.0347]

= 0.966

e. $\frac{W_5}{W_4} = 0.995$

f. Landing:

$$\frac{W_6}{W_5} = 0.997$$

Then,

$$\frac{W_f}{W_O} = k_{tf}\left[1 - \left(\frac{W_6}{W_O}\right)\right.$$

= 1.06 [1-0.717]

= 0.299

Now,

$$W_O = \frac{W_{crew} + W_{payload}}{1 - 0.299 - 0.97W_o^{-0.06}}$$

$$= \frac{5739}{1-0.299-0.97W_o^{-0.06}}$$

Table 3: Gross weight calculation

Taken Gross weight = 66530.991 lbs

Where,

$$\frac{W_f}{W_O} = 0.299$$

$W_f = 0.299 * 30177.95$

$W_f = 9023.21$kg

Therefore, the weight of the fuel is 19892.773 lbs

And,

$$\frac{W_e}{W_o} = 0.97W_o^{-0.06}$$

$W_e = 30177.95 * 0.539$

$W_e = 16252.06$ kg

The empty weight is 35829.659 lbs

4.1.3 Component Weights

The total weight of the aircraft is estimated as 30177.95 kg. The total number of passengers are 48 calculating its weight as 5944kg with its luggage and adding the weight of 4 crew weights 795kg. Book of

Daniel book outlines component weights for both military and air-transport vehicles.

4.1.4 Overpressure

The sizing code used two numerical approximations for determining the sonic boom overpressure created by the aircraft. Two papers written by Carlsonand Seebassoutlined numerical methods for approximating the sonic boom overpressure created by the aircraft. These numerical models were integrated into the sizing code and iterated for each aircraft configuration. The two numerical models were used in parallel because they each predicted a different type of sonic boom signature. The Seebass method predicted the plateau wave sonic boom signature, while the Carlson method predicted the N-wave sonic boom signature.

Each method required different data to predict the sonic boom overpressure. Carlson required a cross sectional area distribution, while Seebass required basic aircraft dimensions including take-off gross weight. From Seebassit was seen that for a supersonic aircraft, it would be more desirable to have a plateau wave rather than an N-wave pressure distribution as it spreads the overpressure over a finite period of time. This would result in a lower average sonic boom overpressure and thus a smaller sonic boom signature. The objective of this function was to give the user an idea on what kind of sonic boom overpressures the aircraft configuration would produce for these two boom signatures. This function completes that objective by using two different numerical models to predict the resulting pressure wave. A more detailed analysis of the methods used and the results from the sizing process is presented in the Sonic Boom section.

4.1.5 Aerofoil and Lift Coefficient

The Aerofoil function takes in the geometry of the new wing and calculates the $C_{L,\ max}$ at take-off and landing. The air foil used in the working model is NACA64A410. Its t/c is 10%.

This is necessary for calculating the landing and take-off field lengths in the Constraints function. A more detailed description on how this function works is outlined in Aerodynamics section.

PRACTICAL STEPS FOR WING AIRFOIL SECTION SELECTION:

1. $W_{avg} = ½ (W_i + W_f)$

W_i; aircraft initial weight at the beginning of the cruise

W_f; final aircraft weight at the end of the cruise

2. $C_{lc} = \frac{2W_{avg}}{\rho V_c^2 S}$

C_{lc}; aircraft ideal cruise lift coefficient

V_c; aircraft cruise speed

ρ ; air density at cruising altitude

S; wing planform area

3. $C_{lcw} = \frac{C_{lc}}{0.95}$

C_{lcw}; wing cruise lift coefficient

4. $C_{li} = \frac{C_{lcw}}{0.9}$

C_{li}; wing aerofoil ideal lift coefficient

5. $C_{lmax} = \frac{2W_{TO}}{\rho_o V_s^2 S}$

V_S= aircraft stall speed

W_{TO} = maximum take-off weight

ρ_o = air density at sea level

6. $C_{lmaxw} = \frac{C_{lmax}}{0.95}$

C_{lmaxw} ; wing maximum lift coefficient

7. $C_{lmaxfrom} = \frac{C_{lmaxw}}{0.9}$

$C_{lmaxfrom}$; wing aerofoil gross maximum lift coefficient

8. $C_{lmax} = C_{lmaxfrom} - \Delta C_{LHLD}$

C_{lmax}; wing aerofoil net mass lift coefficient

9. Root chord, $C_r = \frac{2S}{b(1+\lambda)}$

10. Tip chord, $C_t = C_r * \lambda$

LIFT COEFFICIENT:

It relates the angle of attack to the lift force. If the lift force is known at any specific airspeed, then, lift coefficient can be determined from the formula

$$C_L = \frac{2L}{\rho V^2 S}$$

Where,

L; lift force

ρ; fluid density

S; surface area

To find lift coefficient value we refer NACA aerofoil series

C_L, C_D done in wind tunnel effect test "coefficient of lift $(C_L)_{Vs}$ angle of attack "graph.

C_L depends Re, M, angle of attack.

4.2 AUXILIARY FUNCTIONS

4.2.1 Engine modelling

Engines plays a great role when it comes to designing or accurately size of an aircraft. This is primarily due to the need for different fuel consumption data during the many different parts of the design mission. Various steps were identified when modelling of the Supersonic engines.

Fig 4: Engine CF34- 8E

The engine used in this model is General Electric CF34-8E turbofan engine of 14200pounds (62.28 KN) thrust each 2 engines. Maximum take-off by is 5394ft(16444m) and maximum landing is 4072ft (1241m).

To find the time it takes to travel a given distance at a given speed, using the following equation

$$\frac{distance}{speed} = Time$$

$$\frac{4600Km}{1698.624\frac{km}{hr}} = 2.7\ hours$$

Now,

to find the aircraft fuel consumption, we take

$$\frac{Total\ fuel}{Total\ Time} = Fuel\ consumed\ per\ hour$$

$$\frac{9023.21kg}{2.7\ hour} = 3341.93\frac{kg}{hr}$$

= 4643.50 litres

Therefore, the amount of fuel burned in an hour is 4643.50 litres.

4.2.2 Drag modelling

An accurate prediction of drag is essential to the design of the Supersonic aircraft. For steady level flight, a simple force balance reveals that the thrust that the aircraft must have is equal to the drag that acts on it. This in turn drives the engine selection to satisfy the level flight condition during cruise. Further, the amount of fuel required for the mission is dependent on how much thrust is required, and therefore drag calculations have a great influence on the sizing process of the aircraft. Drag prediction is divided into three phases: subsonic, transonic and supersonic flight regimes. These flight regimes correspond to Mach numbers 1.6.

In supersonic to compute parasite drag, same equation as subsonic but form factors and interference factors ignored.

$$e = \frac{4AR\sqrt{(M^2-1)} - 2}{\pi AR^2(M^2-1)Cos\ (\Lambda_{LE})}$$

Where,

e; Oswald efficiency

One of the major components of drag arise due to shock wave is wave drag. While there are NASA codes that predict wave drag, they require a very detached description of the aircraft geometry to work effectively. A simplified area rule methodology developed by Jumper is used in predicting wave drag.

$$D= -\frac{\rho u_\infty^2}{4\pi}\iint_0^1 A''(x_1)A''(x_2)\ln|x_1-x_2|\,dx_1x_2$$

5 AIRCRAFT CONCEPT

5.1AERODYNAMICS

5.1.1 Aerofoil Selection

The Aerofoil selection process for the Commercial Supersonic aircraft requires an in-depth knowledge of aerofoil performance in supersonic, subsonic and transonic flight. The wing is the primary source of lift for the aircraft and wing aerodynamics plays a vital role in deciding the aircraft flying qualities. Very little information was available on aerofoils suited for the required design mission. For this reason, a database of existing supersonic aircraft along with the aerofoils they used was created. Below table provide available informationon aerofoils used in supersonic commercial aircraft. Given the popularity of its application in modern supersonic commercial aircraft, the NACA 6-series and the biconvex aerofoils stand out as the best options for the required design mission.

Type of aerofoil	Root chord	Tip chord
NACA 64A410	14.689 m	0 m

Table 4: Selected aerofoil used in aircraft

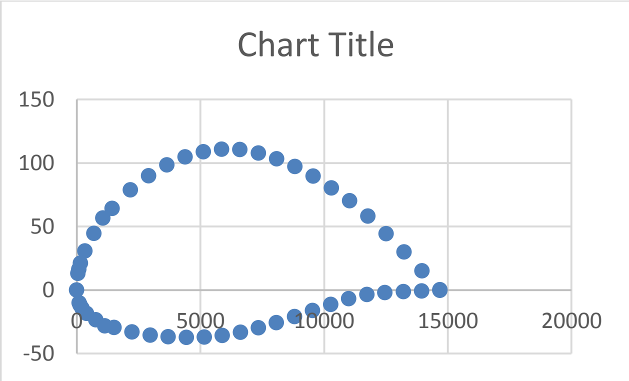


Fig 5.a: NACA 64A410 Root chord of an aerofoil

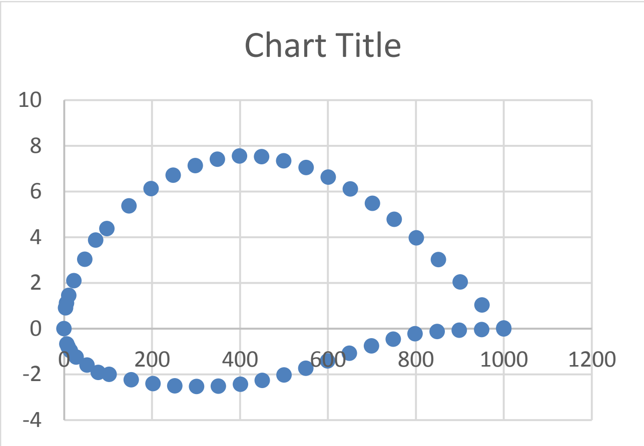


Fig 5.b: NACA 64A410 Tip chord of an aerofoil

To select the best aerofoil for use on the Commercial Supersonic aircraft are:

1. High C_{Lmax} as required during take-off and landing.

2. Delayed stall angle.

5.1.2 Drag build-up

One of the major components of drag arise due to shockwave is wave drag. While there are NASA codes that predict wave drag, they require a very detached description of the aircraft geometry to work effectively.

A simplified area rule methodology developed by Jumper is used in predicting wave drag:

$$D = -\frac{\rho u_\infty^2}{4\pi}\int_0^1\int_0^1 A''(x_1)A''(x_2)\ln|x_1 - x_2|dx_1 x_2$$

5.2 SONIC BOOM

The 0.3 lbs./sq. ft overpressure limit is one of the primary reasons why supersonic aircraft are not in operation in the commercial market today. In order for the Supersonic aircraft to get its passengers to the destination fast, it has to surpass this upper limit on overpressure. A number of different technologies were employed to help reduce the sonic overpressure of this aircraft.

1. **Blunt nose –** The blunt nose design on the Supersonic aircraft will create a bow shock in front of the aircraft which keeps the shock waves from coalescing under the aircraft. This coalescence of shockwaves is the reason for high sonic boom overpressures. They create N-wave shock signatures as in the case of earlier supersonic aircraft. A blunt nose design will help bring the N-wave shock signature to a plateau wave signature with a lower sonic overpressure. Conversely, using a blunt nose increases the wave drag of the aircraft. Optimizing the aerodynamic shaping can reduce the wave drag of the aircraft.

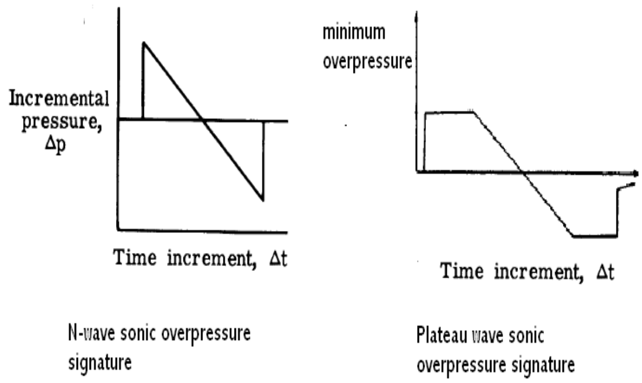


Fig 6: shows the different sonic overpressure signatures discussed.

2.**Dihedral angle –** A dihedral angle on the lifting surface of the aircraft can reduce the sonic overpressure by making the area distribution smoother, which has a high effect on the sonic overpressure signature. Also, a dihedral angle has the effect of increasing the effective length of the aircraft.

3.**Low AR, high sweep** – A high aspect ratio, low sweep wing has the effect of increasing lift rapidly over the wing. This is another major reason for the creation of N-wave shaped sonic signature. A low aspect ratio, highly swept wing brings the aircraft sonic boom overpressure signature to that of a plateau wave.

4.**Smooth area distribution** – A smooth area distribution is vital to reduce creating multiple shocks at multiple locations on the aircraft. Smooth area distribution coupled with the blunt nose design will help bring down the chances of multiple shocks originating all over the surface of the aircraft, which could coalesce together to give a high sonic overpressure.

Two different techniques were employed to calculate the sonic overpressure on the aircraft. One technique was based on the "Simplified sonic boom prediction" paper by Harry W. Carlson. The other method used was developed by R. Seebass and A.R. George in the paper titled "Sonic-boom minimization". While the Carlson method makes use of the area distribution of the aircraft to determine the shape factor of the aircraft and use the shape factor to calculate the N-wave overpressure signature, the Seebass method makes use of basic aircraft parameters (Weight, length, Mach number, etc...) to determine the plateau overpressure signature of the aircraft.

Method	Overpressure (lb/sq. ft)
Carlson	0.28
Seebass	0.71

Signature Duration, Δt 0.03s

Table 5: Seebass and Carlson Overpressures and duration

Since the Carlson method was used to calculate the aircraft sonic overpressure based on the aircraft geometry, it is chosen as the more reliable of the two methods and is explained in more detail. The Carlson method was obtained from NASA Technical Paper 1122 titled "Simplified Sonic-Boom Prediction". Although this method gives a rough estimate of the sonic overpressure and signature duration, much more research and analysis in supersonic sonic boom mitigation is required to develop the final aircraft design. The Carlson method involves three major steps to calculate the sonic boom overpressure and its time signature.

1. Determine Shape factor – In order to calculate the shape factor of the aircraft,

a.Generate axis normal cross-sectional area distribution – The cross-sectional area distribution along the length of the aircraft was generated by the A/C Geometry function in the sizing code. Details about this process were discussed earlier in the aircraft geometry section. The wireframe area

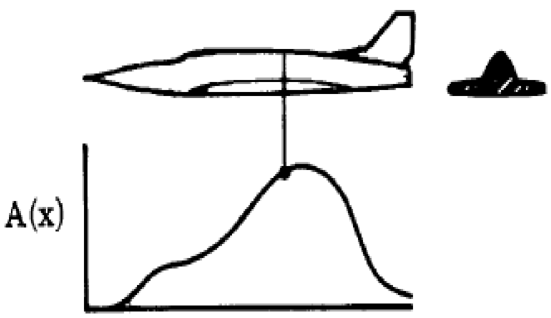


Fig 7: Aircraft Wireframe area distribution

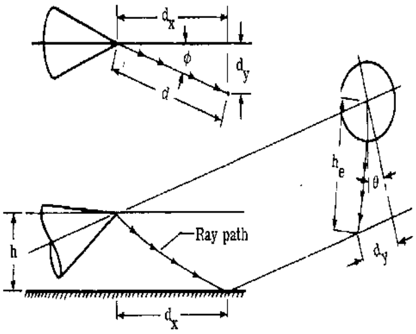


Fig 8: Propagation geometric parameter

b.Equivalent area due to lift – A reasonably accurate approximation of the equivalent area due to lift is calculated from the span distribution along the length of the aircraft. This has been described in Figure 24, where b(x) is the span wise distribution along the length of the aircraft. This is used to calculate the equivalent area distribution, B(x). B(x) is calculated using the equation

$$B(x) = \frac{\sqrt{M^2-1}}{1.4\rho_V M^2}\frac{W\ \mathrm{COS}\ \gamma\ \mathrm{COS}\theta}{S}\int_0^x b(x)dx$$

where M is the supersonic cruise Mach number of the aircraft, ρ_V is the atmospheric pressure at vehicle altitude, W is the weight of the aircraft, S is the planform area, γ is the flight path angle (0° for steady- level flight) and θ is the initial ray path angle (0° if directly under flight path)

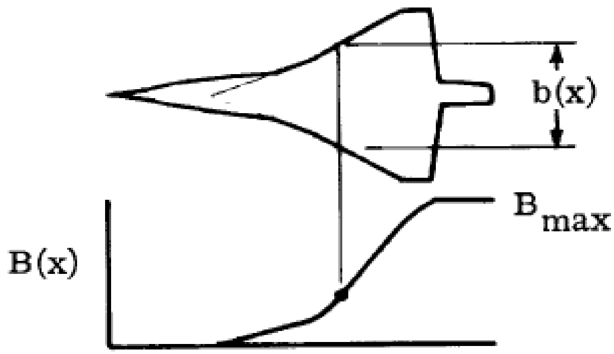


Fig 9: Span distribution and effective area

c. Combined effective area – The geometric area combined with the equivalent area due to lift gives the effective area. The combined effective area for the Supersonic aircraft is given in Figure. This distribution curve was smoothened to yield a better plot with lesser kinks.

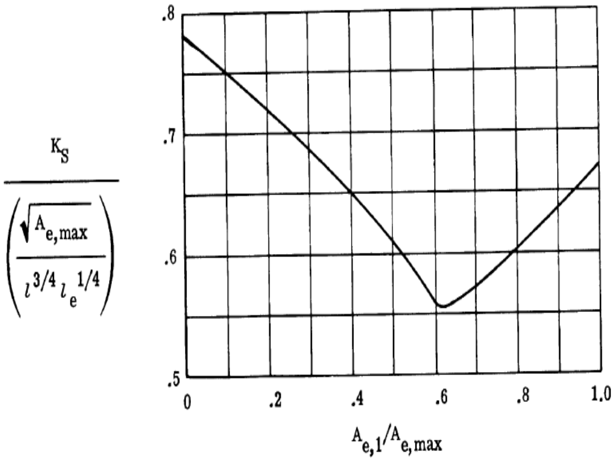


Fig 10: Shape factor as a function of effective area parameters

2. Calculate effect of atmosphere on propagation – The effect of atmosphere on boom propagation needed to be calculated. This was done by determining the effective Mach number and effective altitude of supersonic cruise. The effective Mach number is given by the formula (Carlson).

$$M_e = \sqrt{1+\frac{[A(1-B\tan\gamma)]^2}{[A(\tan\gamma+B)]^2+(CD)^2}}$$

Where,

$$A = \frac{1}{\cos\gamma\sqrt{M^3-1}}$$

$$B = \frac{1}{\cos\theta\sqrt{M^2-1}}$$

$$C = \frac{\tan\theta}{\sqrt{M^2-1}}$$

$$D = \tan^2\gamma + 1$$

Or,

$$d = \frac{K_d(h_v - h_g)}{\sqrt{M_e^2-1}}$$

$$d_x = d\cos\emptyset$$

$$d_y = d\sin\emptyset$$

$$h_e = \sqrt{d_y^2 + [(h_v - h_g)\cos\gamma + d_x\sin\gamma]^2}$$

Or,

$$K_d = K_{d,c} + (K_{d,\infty} - K_{d,c})\left(\left(\frac{M_e-M_c}{M_e-1}\right)^{n_d}\right.$$

$$K_p = K_{p,\infty}\left(\frac{M_e-1}{M_e-M_c}\right)^{n_p}$$

$$K_t = K_{t,\infty}\left(\frac{M}{M-1}\right)^{n_t}$$

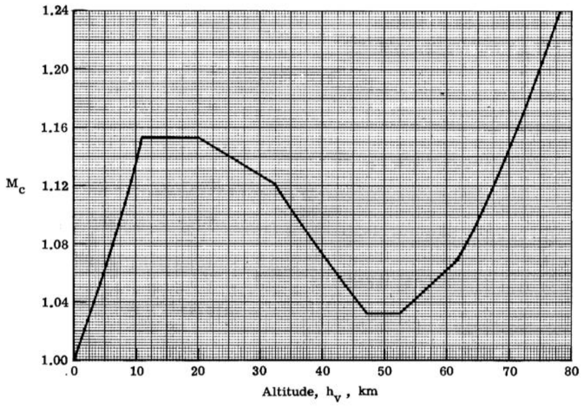


Fig 11: cut off Mach number, M_c

Thus, we find the formulae to calculate the Peak shock overpressure and time signature are,

$$\Delta p_{max} = K_P K_R \sqrt{\rho_v \rho_g}(M^2-1)^{1/8} h_e^{-3/2} l^{3/4} K_S$$

$$\Delta t = K_t \frac{3.42}{a_v} \frac{M}{(M^2-1)^{3/8}} h_e^{1/4} l^{3/4} K_S$$

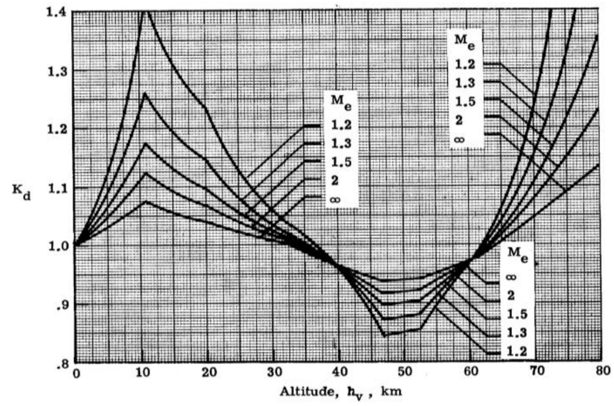


Fig 12: Ray- Path distance factor K_d

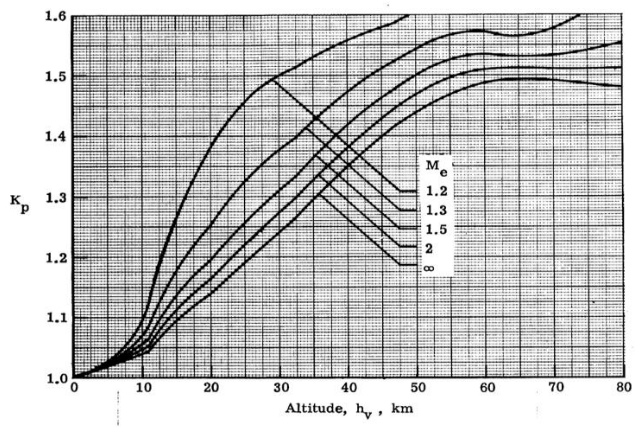


Fig 13: Pressure amplification factor, k_p

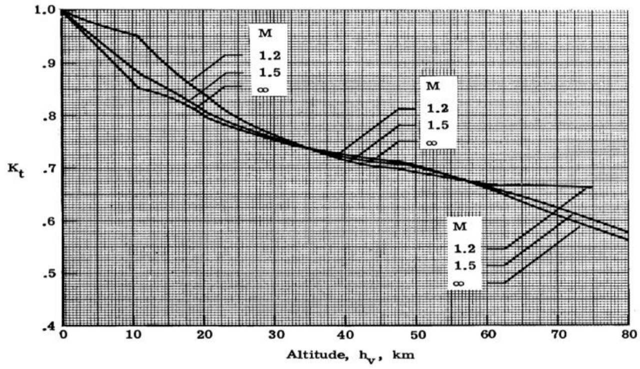


Fig 14: Signature duration factor k_t

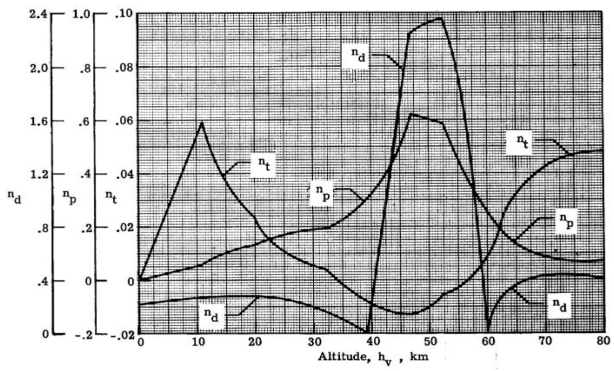


Fig 15: Atmospheric factor curve fit exponents

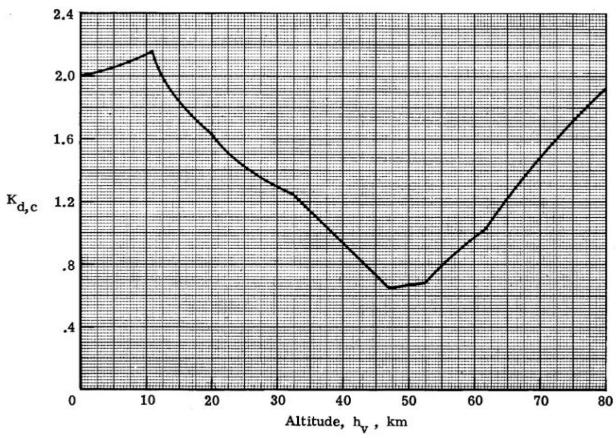


Fig 16: Cut-off ray path distance factor $k_{d,c}$

Using the above formulae, we calculate the overpressure and time signature of the aircraft we get,

a. $$K_L = \frac{\sqrt{M^2-1}}{1.4\,\rho_V}\frac{W\cos\gamma\,\cos\theta}{M^2 l^2}$$

$$= \frac{\sqrt{1.6^2-1}}{1.4*(11.1*10^3)}\frac{30177.95\ \mathrm{Cos}\ 15^o \mathrm{Cos}\ 52^o}{(1.6^2)*(36.576^2)}$$

$$= \frac{22429.006}{53221044.94}$$

$K_L = 0.000421$

b. $$K_S = 0.74\sqrt{K_L + 0.027}$$
$= 0.123$

c. $$M_e = \sqrt{1 + \frac{[A(1-B\tan\gamma)]^2}{[A(\tan\gamma+B)]^2+[CD]^2}}$$
$= 1.049$

d. $$d = K_d\left(\frac{h}{\sqrt{M_e^2-1}}\right)$$

$= 12.529$km

e. $$\emptyset = \tan^{-1}(\tan\theta\cos\theta\sqrt{M^2-1}$$

$= 44.56$ degree

f. $dy = d\ \mathrm{Sin}\theta$

$= 12.529\ \mathrm{Sin}\ 44.56 = 8.791$ km

g. $$h_e = \sqrt{dy^2 + h^2}$$

$= 15.892$ km

h. $K_d = 0.3$

i. $K_p = 1.5$

j. $K_t = 0.9$

Now, substituting all the values we get,

$$\Delta p_{max} = K_P K_R \sqrt{\rho_v \rho_g}(M^2-1)^{1/8} h_e^{-3/2} l^{3/4} K_S$$

$= 0.547$ psf or 26.23 Pa

Also,

$$\Delta t = K_t \frac{3.42}{a_v}\frac{M}{(M^2-1)^{3/8}} h_e^{1/4} l^{3/4} K_S$$

$= 0.3$ sec

Hence, using Carlson we found certain values

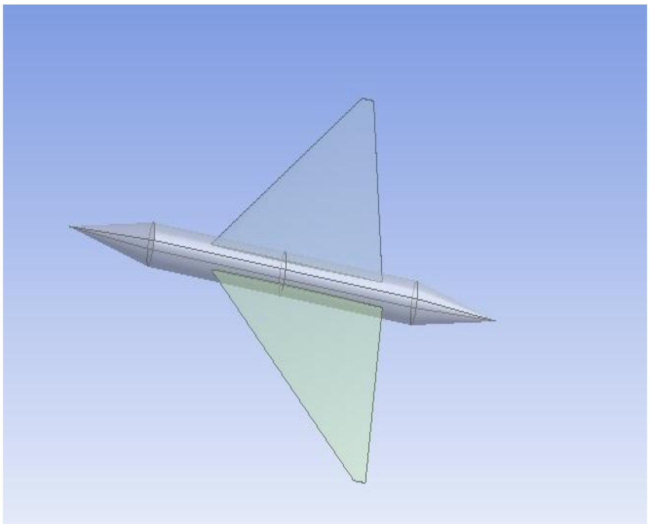


Fig 17: Model of aircraft in Ansys

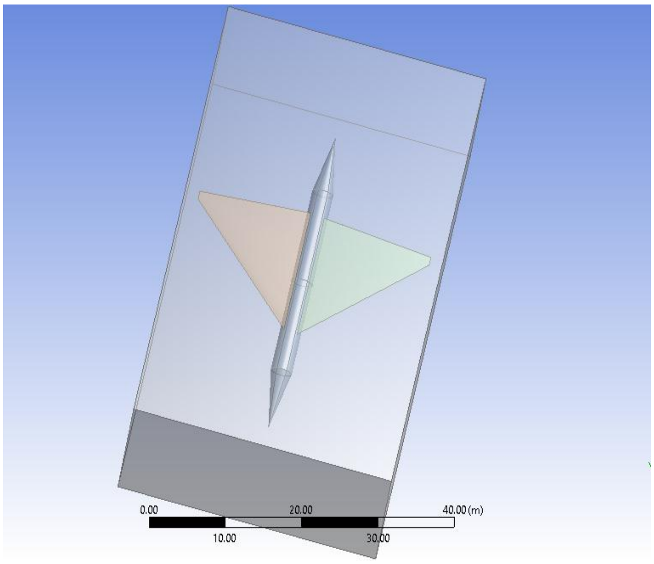


Fig18: Aircraft enclosure

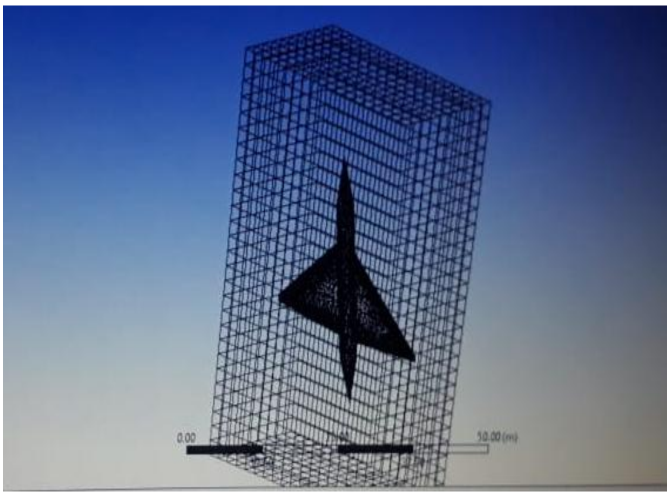


Fig 19: Meshing of an aircraft

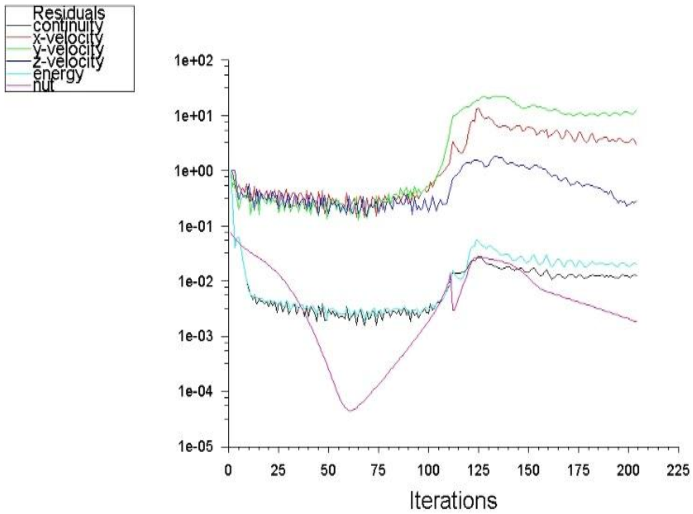


Fig 20: Graph of iterations

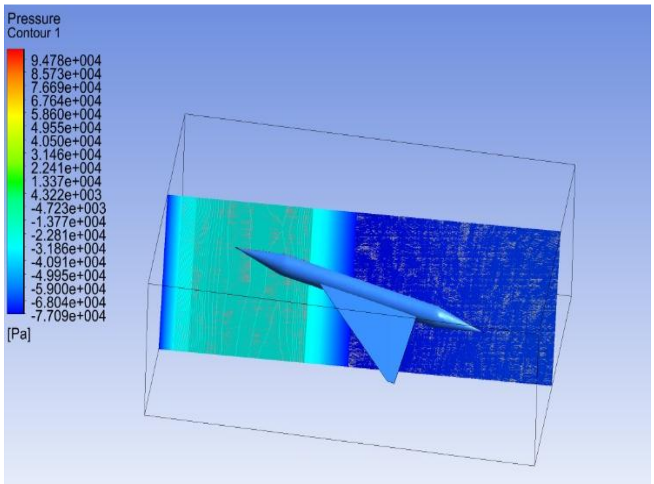


Fig 21: Pressure difference over aircraft

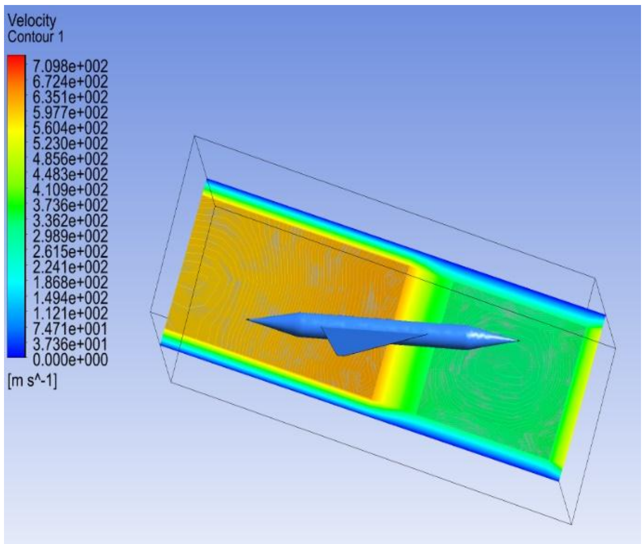


Fig 22: Velocity over aircraft

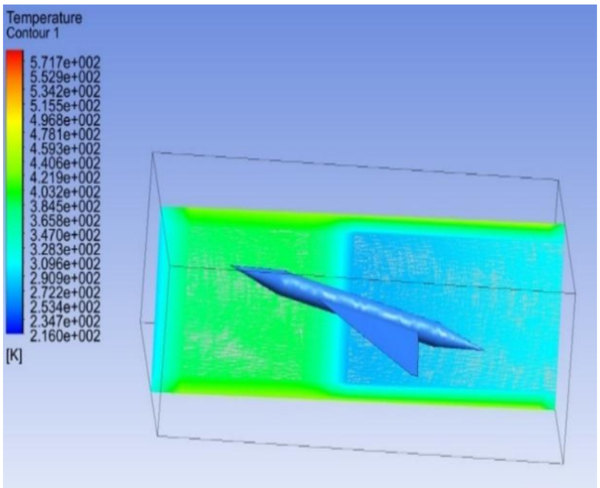
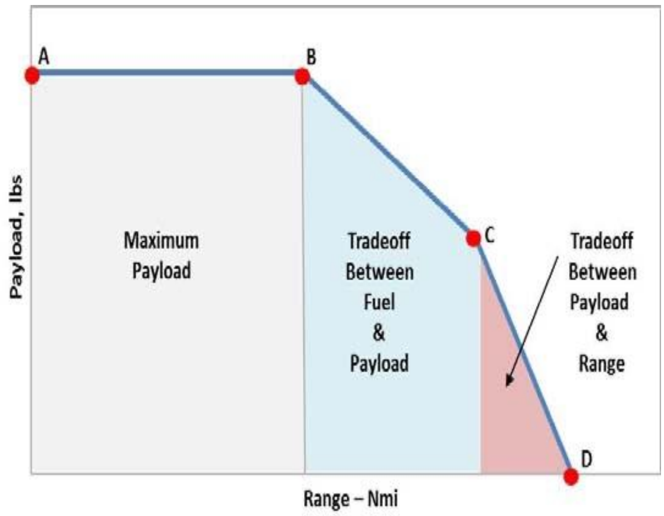


Fig 23: Temperature distribution over the aircraft

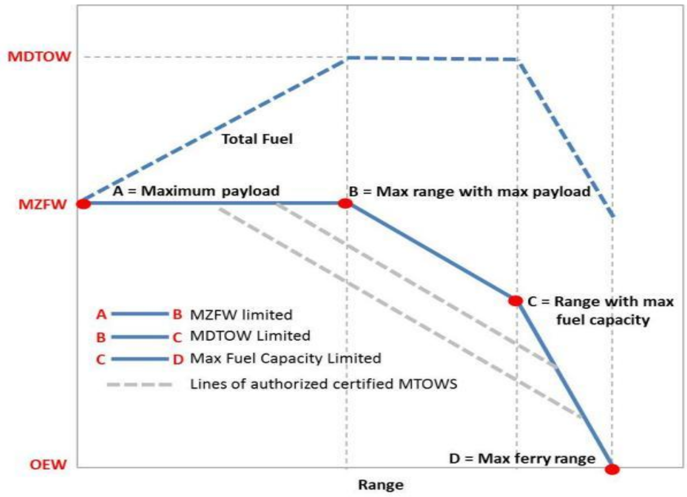
5.3 PERFORMANCE

5.3.1 Range Diagram

Creation of the range diagram provides an outline for cargo loading capability of the aircraft with corresponding range. The numerical values were derived from the Breguet's range equation from the start of cruise to the end of cruise. The initial aircraft weight with fuel weight has to correspond to the mission statement at the beginning of cruise end of climb. The maximum zero range for this aircraft is the maximum cargo load. There is no horizontal steady range for the commercial Supersonic aircraft because of the aircraft's fuel requirements. The Supersonic aircraft cannot reach its operational altitude with a cargo weight equal to the amount of fuel. The aircraft range at maximum take-off weight is 30177.95 kg, with the maximum range of 4600km. All ranges presented are cruise ranges only.



(a)



(b)

Fig 24: Payload and Range graph

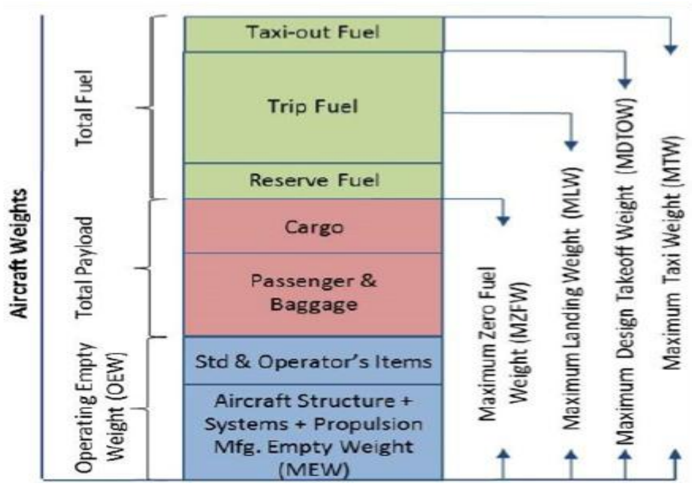


Fig 25: Aircraft weight build-up

Types of range are:

a. Harmonic Range

Range with maximum possible payload

b. Ferry Range

Range with zero payload, and including reserve fuel

c. Gross Still Air Range

Range assuming all the mission fuel is utilized for cruise flight alone.

6.PROPULSION

6.1 Engine description

The engine used in this model is General Electric CF34-8E turbofan engine of 14200pounds (62.28 KN) thrust each 2 engines. Maximum take-off by is 5394ft(16444m) and maximum landing is 4072ft (1241m).

To find the time it takes to travel a given distance at a given speed, using the following equation

$$\frac{distance}{speed} = Time$$

$$\frac{4600Km}{1698.624\frac{km}{hr}} = 2.7\ hour$$

Now,

to find the aircraft fuel consumption, we take

$$\frac{Total\ fuel}{Total\ Time} = Fuel\ consumed\ per\ hour$$

$\frac{9023.21kg}{2.7\ hour} = 3341.93\frac{kg}{hr}$ = 4643.50 litres

Concorde was powered by four Rolls-Royce/SNECMA Olympus 593 engines. This engine is the direct descendant of the Bristol Siddeley Olympus, the world's first two-spool axial-flow turbojet engine, designed and built in Patchway. The Olympus 593 was flight tested from Filton Airfield fitted to the underside of a Vulcan bomber.

Therefore, the amount of fuel burned in an hour is 4643.50 litres. F34-8E is an advanced 14,500 pound thrust class turbofan propulsion system and a member of GE's popular CF34engine family. It is the system that powers Embraer's 70-90 passenger airliners, the EMBRAER 170/175. The -8E takes full advantage of its CF34 design and operational experience lineage as well as its relationship with other advanced CF34 models. It incorporates all of the service-proven reliability, environmental and operational characteristics that have earned the CF34 engine family an excellent global reputation with airline and corporate operators for exceptional performance. The -8E propulsion system incorporates a nacelle design specifically tailored to the EMBRAER 170/175 underwing installation. The new design maximizes LRU accessibility, resulting in enhanced maintainability

Table 6: Performance Specifications

Components	Value
Bypass ratio	14500lb
Maximum overall pressure ratio Thrust/weight ratio	5:1
Fan diameter	28:5:1
Length	5:6:1
Weight	46.2
Noise	53
Emissions	121

7.RESULT

The engineering analysis although not satisfy all of the requirements of the aircraft but it's close to those goals. Therefore, the concept of the supersonic commercial aircraft transport is definitely worth designing.

As, we calculated both numerically and computationally we see aerodynamic flow over the body is currently approximated and no shock interactions have been considered. Fuselage structures must be further researched to include specialized structural load path at location of high stress concentration. CATIA model of fuel tanks, cargo space, centre of gravity travel i.e., aircraft load configuration must be allotted for more precise calculation. Material selection should be inputted into CATIA providing a finalized moments about the x, y and z axis. The results from finalized CATIA model can be used to validate the dynamic stability and control of the aircraft. The results can provide a more accurate aircraft diagram as well mission performance. These results will provide insight on further feasibility of the Supersonic commercial aircraft.

8.CONCLUSION

A simplified Carlson method for calculation of the overpressure and time signature characteristics in supersonic commercial aircraft configuration is calculated and observed in the CFD flow. The procedure done step by step relies greatly to the great extent on use of the charts to provide the necessary sonic-boom generation and propagation factors for use in relatively simple expressions for signature characteristics. With a bit inaccuracy in numerical complete calculations can often be obtained in less time than is required for the preparation of computer input data for the more accurate calculation methods.

LIST OF SYMBOLS

b = Wing span

AR =Aspect ratio

T/W = Thrust / Weight

R/C = Rate of Climb

TSFC = Thrust Specific Fuel Consumption

C_L = Coefficient of lift

C_D = Coefficient of Drag

M = Mach number

L/D = Lift/ Drag

W_0 = Total Weight

W_f = Weight of the fuel

W_e = Empty Weight

R = Range

E = Endurance

K_L = Lift parameter

W = Weight of an aircraft

γ = flight path angle

θ = ray path azimuth angle

ρ_v = atmospheric pressure at aircraft altitude

l = length of an aircraft

K_S = aircraft shape factor

M_e = a/c effective Mach no. governing sonic boom atmosphere propagation characteristics

d = distance between a/c ground trade position at time of sonic boom generation and location of ground impactpoint.

K_d = ray- path distance factor

h = h_v- h_g (altitude of aircraft above ground)

Φ = angle between a/c ground track and ground projection of ray path.

dy = Component of d in a direction perpendicular to a/c ground track

h_e = effective altitude

K_d = ray path distance factor

K_p = pressure amplification factor

K_t = signature duration factor

ΔP_{max} = bow shock overpressure

K_R =reflection factor

ρ_g = atmospheric pressure at ground level

Δt = signature duration

MAC = Mean Aerodynamic Chord

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BIOGRAPHIES

Mr. Bijay Kumar Sah was born on 27th may 2001 and currently pursuing B.E Final year of Aeronautical Engineering. Had presented 5 papers at National and International Technical Symposium. My research interest includes Aero modelling (RC and UAV), Aerodynamics, Aircraft & Rocket propulsion, Aircraft Structure, Design and fabrication of Satellite, Fluid mechanics, Thermodynamics, Computational Fluid dynamics, Composite Material, Rocket and Missile

Contact: sahbijay2001@gmail.com

Ms. Keerthana Rathinavel was born on 11th June 2001 and currently pursuing B.E Final year of Aeronautical Engineering. My research Area includes Aerodynamics, Aircraft structure, Propulsion, Thermodynamics, Fluid Mechanics, Aero Modelling (RC & UAV), and CFD.

Contact: keerthanarvel@gmail.com

Mr. Naveen kumar. M was born on 05-09-2000 and currently pursuing B.E Final year of Aeronautical Engineering. My research area includes in Drone Fabrication, Aerodynamics.

Mr. Ashir Azar was born on 16/03/2000 and currently pursuing B.E Aeronautical engineering. My research interest includes Aerodynamics, Propulsion, Drone fabrication.

Dr.S.P.Venkatesan was born on 4th November in 1976 has completed his Ph.D(Thermal Design)in Anna University,ChennaiTamil Nadu,India in the year July 2018 and M.E (Engineering Design)in the year 2000 from Kongu Engineering College,Perundurai,Tamil Nadu,India under Bharathiyar University,Coimbatore,Tamil Nadu,India and B. E (Mechanical Engineering) in the year 1999 Madras University,Chennai,TamilNadu,India.

Has got nearly14years of Teaching experience with Research experience of 6 years and Industrial experience of 5 years

Published more than 35 paper in International Journal in various topics in the field of Mechanical engineering

Contact:spvensphd@gmail.com