Design and Analysis of a Supersonic Business Jet

a project presented to
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Master of Science in Aerospace Engineering

by

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LIST OF SYMBOLS

MTOW: Maximum Takeoff Weight
W_E: Empty Weight
W_{TO}: Maximum Takeoff Weight
A and B: Regression coefficients of the Airplane.
M_{ff}: Maximum Fuel Fraction
nm: Nautical Miles
$W^{"}_{\#\$\%\&}$: Maximum Fuel Used
$W^{"}_{)\%\$}$: Maximum Fuel Reserved
$W^{"}$: Amount of Fuel in a journey
W_{*+}: Weight of Payload
$W_{,-../01/2-.}$: Weight of the passengers
$W_{32/4}$: Weight of the crew
$W_{5_{6\%76}}$: Tentative Empty Weight
R: Range
E: Endurance
S: Wing Area
f: Parasite Wing Area
S_{wet}: Wetted Area
$C_{+_{9-:}}$: Maximum Coefficient of Lift
$C_{+_{9-:_{+}}}$: Maximum Coefficient of Lift while Landing
$C_{+_{9-:_{;<}}}$: Maximum Coefficient of Lift while Take-off
T_{TO}: Takeoff Thrust
V_S: Stall speed (knots)
W/S: Wing Loading
S_{TOFL}: Size of Take-off field Length
TOP_{25}: Take-off Parameter for FAR 25 requirements
T/W: Thrust to weight ratio
C_L: Coefficient of Lift
C_D: Coefficient of Drag
$C_{=_{>}}$: Zero lift Coefficient of Drag
$C_{=_{?}}$: Induced Coefficient of Drag
$C^{"}$: Skin Friction Drag
RC: Rate of Climb
$T_{2/AB}$: Thrust Required
c_{VT}: Volume coefficient of the vertical Tail
L_{VT}: Distance between the quarter chord of the vertical with respect to the wing
S_{VT}: Area of the vertical stabilizer
b_w: Wing Span
S_w: Wing Area
c_{HT}: Volume coefficient of the horizontal stabilizer

L_{HT}: Distance between the quarter chord of the horizontal with respect to the wing
S_{HT}: Area of the horizontal stabilizer
C_W: Mean Aerodynamic Chord length of the wing
λ_{HT}: Taper ratio of the horizontal tail wing.
λ_{VT}: Taper ratio of the vertical tail wing.
$\Lambda_{D;}$: Sweep angle of the vertical tail.
$\Lambda_{E;}$: Sweep angle of the horizontal tail.
$F\frac{G}{3}H_{E;}$: Thickness ratio of the horizontal tail.
$F\frac{G}{3}H_{D;}$: Thickness ratio of the vertical tail.
$c2_{D;}$: Vertical stabilizer root chord length.
$cG_{D;}$: Vertical stabilizer tip chord length.
$c2_{E;}$: Horizontal stabilizer root chord length.
$cG_{E;}$: Horizontal stabilizer tip chord length.
$b_{D;}$: Vertical stabilizer span
$b_{E;}$: Horizontal stabilizer span
dB: Decibels

PRELIMINARY (CLASS I) SSBJ DESIGN

CHAPTER 1: Mission Specification and Comparative Study

1.1 INTRODUCTION

Supersonic Business jets are designed to travel at speeds above the speed of sound (Mach 1.0). There is no supersonic business jet available which makes this a conceptual aircraft, though there are many designs proposed by various leading aircraft manufacturer's. SSBJ's are business jets with some modifications that help them to fly at supersonic speeds and higher altitudes.

Only two commercial transports have entered service till date. the British Aerospace Concorde and Tupolev Tu-144. But were eventually shut down due to the high noise produced by them, high manufacturing costs, consumption of a lot of fuel and concerns relating to the environment thus it became difficult for the government to recoup the development costs.

Several Manufacturers believe that these concerns can be tackled if addressed at small scales. Keeping this in mind various companies continue to work on supersonic business jets, their technologies to be installed and overcome the concerns put up during the early development of the supersonic business jet designs.

1.2 MISSION SPECIFICATION

1.2.1 Mission Specification:

Payload Capacity: 24 to 48 passengers
Crew Members: 4
Range: 5500 – 6000 nm (9260 to 11,112 km)
Cruising speed: 1.6
Maximum speed: 3.0 Mach
Cruise altitude: 54000 ft (16460 m)
Take-off field length: 2600m
Landing field length: 1500m
Approach speed: 140 knots
Noise requirements: 40 dB

1.2.2 Mission Profile:

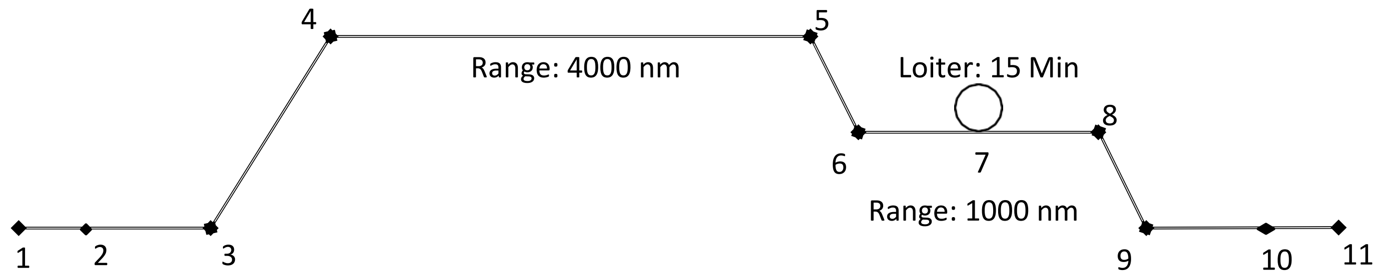


Figure 1: Mission Profile

1: Engine Start
1-2: Taxi
2-3: Take-off
3-4: Climb
4-5: Cruise
5-6: Descent
6-7: Cruise
7: Loiter
7-8: Cruise
8-9: Decent
9-10: Land
10-11 : Taxi, Shutdown

1.2.3 Market Analysis:

The supersonic business jets have a very high demand in the market. Due to their speeds and services provided. The biggest consumers of the supersonic business jets will be the high-value passengers such as the executives, prime ministers, presidents etc. they will find value in the higher speed transportation as described by the flight International in December 2012.

1.2.4 Technical and Economic Feasibility:

The Supersonic business jets can be installed with Engines that provide speeds up to Mach 3. With certain modifications, the engines can be made to run by hydrogen. Which is a highly combustible gas, cheap to obtain, environment friendly. Using this the development of the supersonic business jets will be costly in the beginning but will gradually help provide cheaper transportations at very high speeds making it a win situation for the manufacturer as well as the consumer.

Making the aircraft ecofriendly, economic, provide speedy transportation. It can be installed with the latest technologies for safety purposes and emergency situations. It will provide a huge boost to the aviation industry to provide speedy transportations to the high-value passengers.

The hydrogen powered engine’s is the main problem faced in the mission, various attempts have been made to manufacture such an engine and have been successful up to a certain extent but will take an enormous amount of period to design such an engine.

1.2.5 Critical Mission Requirements

The critical mission requirement for the supersonic business jets are:

- Landing where it would require a longer runway to land the aircraft safely.
- The takeoff distance which is very high for the aircraft to take off from shorter runways which may cause problems for the SSBJ’s to takeoff from airports having shorter runways.
- The other critical mission requirement is the use of hydrogen gas as a fuel which due to its low density, it requires to be stored in pressurized containers which indirectly affect the overall weight of the aircraft.

1.3 COMPARATIVE STUDY OF SIMILAR AIRPLANES

1.3.1 COMPARATIVE STUDY

Some Aircrafts similar to the conventional SSBJ are:
Aerion SBJ

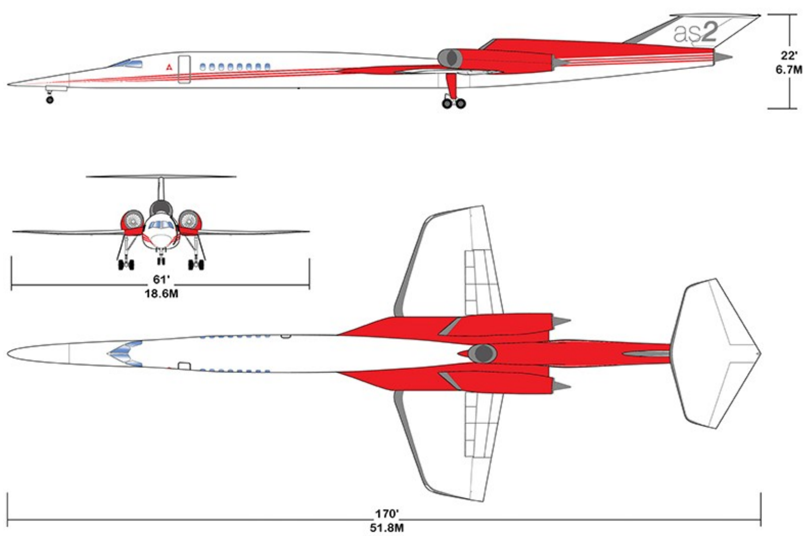


Figure 2: Design of the Aerion SBJ

Aerion AS2

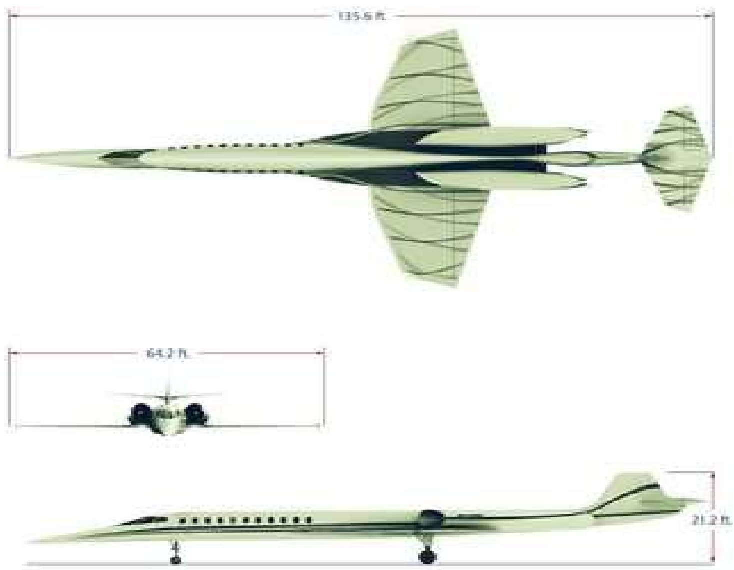


Figure 3: Design of the Aerion AS2

Spike S-512

Figure 4: Design of the Spike S-512

Sukhoi Gulfstream S-21

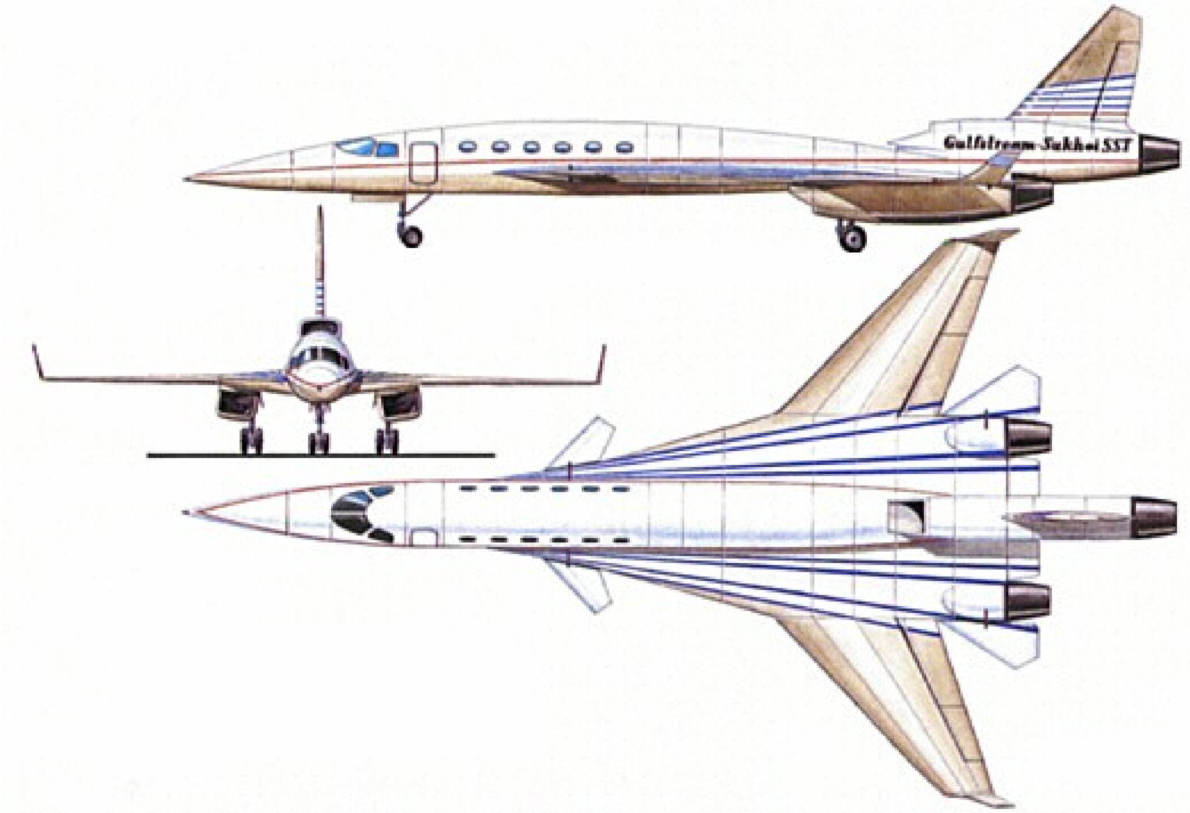


Figure 5: Design of the Sukhoi Gulfstream S-21

Tupolev Tu-444

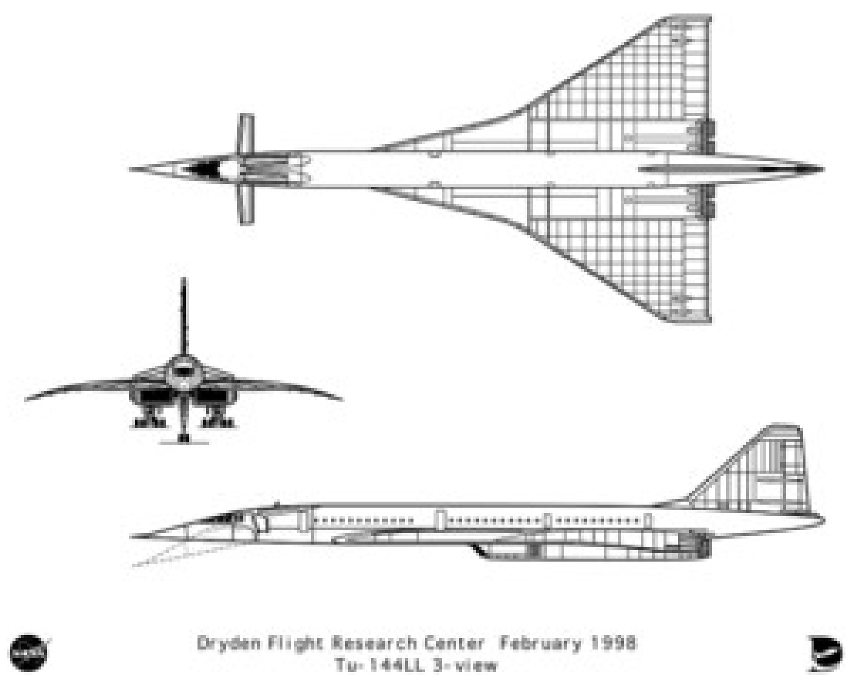


Figure 6: Design of the Tupolev Tu-444

Table 1: Comparison of various SSBJ Designs

SSBJ's : General Characteristics:	Aerion AS2	Aerion SBJ	Spike S-512	Sukhoi Gulfstream S-21	Tupolev Tu-444
Payload Capacity	8-12 passengers	8-12 passengers	18 passengers	6-10 passengers	6-10 passengers
Range	8,797 – 9,816 km	7,800 – 8,500 km	10,334 km	4,369 – 7,403 km	7,500 km
Cruising speed	1.4 Mach	1.7 Mach	1.6 Mach	1.9 Mach	2 Mach
Cruising Altitude	52,000 ft (16,000 m)	51,000 ft (15,500 m)	50,000 ft (15,000 m)	63,900 ft (19,477 m)	-
Height	22 ft	21.2 ft	-	27.1 ft	21.4 ft
MTOW	121,000 lb	90,000 lb	115,000 lb	114,200 lb	90,400 lb
length	170 ft	135 ft	122 ft	124.2 ft	118.1 ft
Wing span	61 ft	64.2 ft	58 ft	65.4 ft	53.1 ft
Maximum speed	1.5 Mach	1.8 Mach	1.8 Mach	1.9 Mach	2 Mach

1.3.2 DISCUSSION

The comparative study in the above table provides various informative parameters to be considered while designing the Supersonic Business Jet. These aircrafts were chosen as majority of the parameters are similar to the SSBJ that is being designed, they have almost a similar maximum take-off weight, it can carry a payload of up to 20 passengers and 4 crew members to a range of 6000 nm with a cruising velocity of 1.6 mach and a maximum cruising velocity of 3.0 mach. Prototypes have been introduced while the only problem being faced is the cost of development of such aircrafts. They can only be developed if the cost of production is reduced up to a certain extent to move forward with the proposed projects.

1.4 **CONCLUSIONS AND RECOMMENDATIONS**

Supersonic business jets are very important in the field of aviation to provide speedy transportation to the people. Thus, the concept of this report is to design an aircraft that is ecofriendly, fuel-efficient as well as economical along with the latest technology available. At present the SSBJ's are only a concept on which a lot of research had been done by various researchers.

CH-2 CONFIGURATION SELECTION

2.1 INTRODUCTION

The purpose of this report is to define the configuration of the supersonic business jet that will be designed later through the course. It defines the configuration of each section of the aircraft according to the needs of the design considering the limitations, the advantages to use the configuration along with its disadvantages.

This report also covers the comparison of the geometry, weights and performance of the aircrafts similar to my supersonic business jet design. It also covers sketches oy my design showing the aspects of my configuration.

2.2 COMPARATIVE STUDY OF AIRPLANES WITH SIMILAR MISSION PERFORMANCE.

2.2.1 Comparison of weights, performance and geometry of similar Airplanes.

Table 2: Performance comparison of similar aircrafts

SSBJ's : General Characteristics:	Aerion AS2	Aerion SBJ	Spike S-512	Sukhoi Gulfstream S-21	Tupolev Tu-444
Payload Capacity	8-12 passengers	8-12 passengers	18 passengers	6-10 passengers	6-10 passengers
Range	8,797 – 9,816 km	7,800 – 8,500 km	10,334 km	4,369 – 7,403 km	7,500 km
Cruising speed	1.4 Mach	1.7 Mach	1.6 Mach	1.9 Mach	2 Mach
Cruising Altitude	52,000 ft (16,000 m)	51,000 ft (15,500 m)	50,000 ft (15,000 m)	63,900 ft (19,477 m)	-
Height	22 ft	21.2 ft	-	27.1 ft	21.4 ft
MTOW	121,000 lb	90,000 lb	115,000 lb	114,200 lb	90,400 lb
length	170 ft	135 ft	122 ft	124.2 ft	118.1 ft
Wing span	61 ft	64.2 ft	58 ft	65.4 ft	53.1 ft
Maximum speed	1.5 Mach	1.8 Mach	1.8 Mach	1.9 Mach	2 Mach

2.2.2 Configuration comparison of similar Airplanes

The airplanes similar to my aircraft are as follows:

i. Aerion AS2
ii. Aerion SBJ
iii. Spike S-512

iv. Sukhoi Gulfstream S-21
v. Tupolev Tu-444

i. Aerion AS2

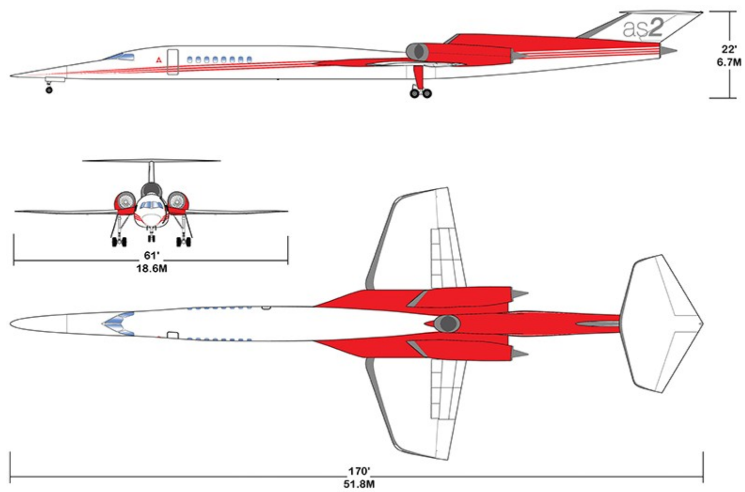


Figure 7: Front view, Side view and Top view of the Aerion AS-2

ii. Aerion SBJ

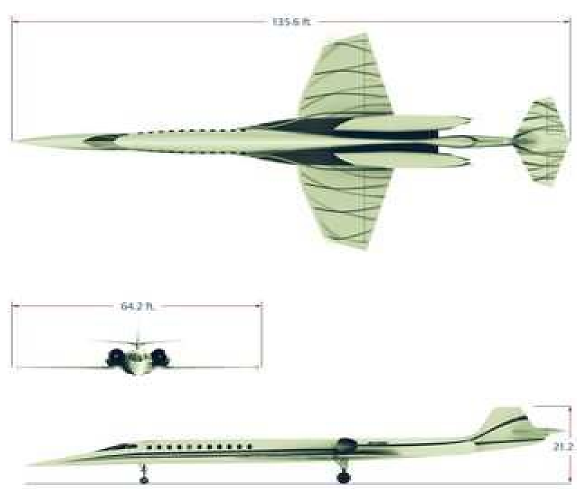


Figure 8: Top view, Front view and Side view of the Aerion SBJ

iii. Spike S-512

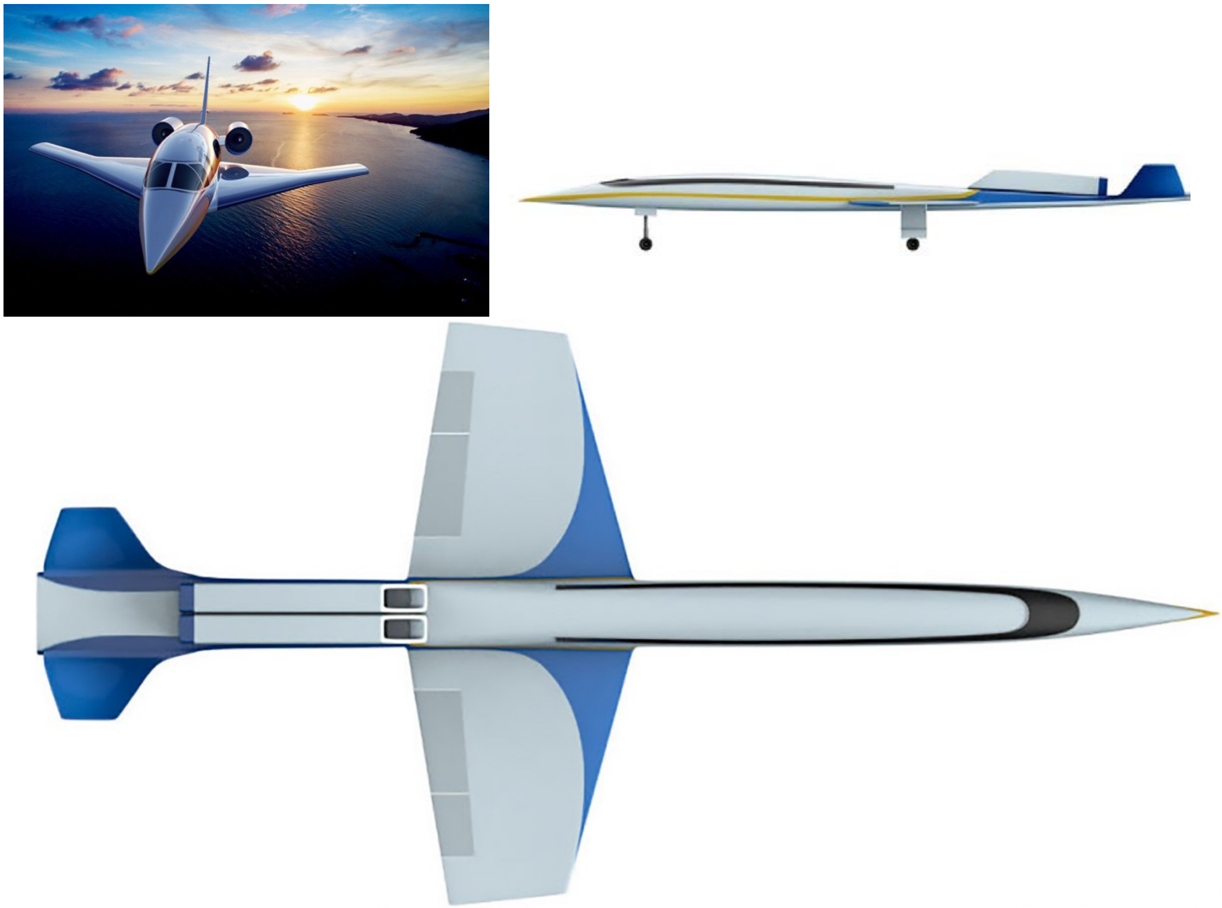


Figure 9: Front view, side view and Top view of the Spike S-512

iv. Sukhoi Gulfstream S-21

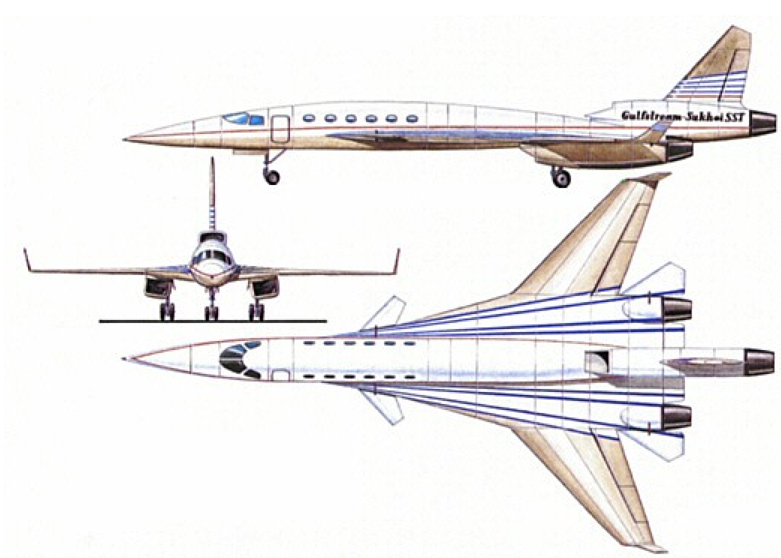


Figure 10: Side view, front view and top view of the sukhoi gulfstream s-21

v. **Tupolev Tu-144**

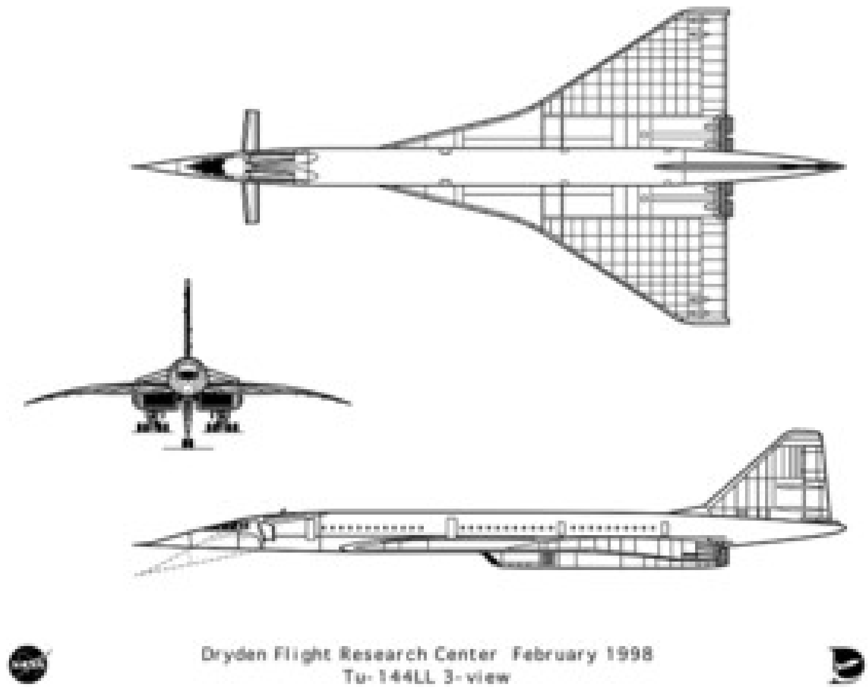


Figure 11: Top view, front view and side view of the Tupolev Tu-144

2.2.3 Discussion

The above section describes the 3 views of the aircrafts similar to mine. Looking at the previous designs, the configurations of the aircrafts are as follows:

All the suggested supersonic business jet designs have a less wing area as well as a low-wing configuration. Low wing configurations were selected as they are a best fit of the business jets. They provide better maneuverability than any mid-wing or high-wing aircraft. They are safer, decreases the take-off distance as well as the landing distance. Not much efforts are to be made from the pilot’s end while landing. It provides easy retraction of the landing gears. Thus, the low-wing configuration is the best configuration of the proposed business jet designs.

The business jets similar to my proposed design have either a vertical tail or a T-tail in the empennage section. They have very thin airfoil shaped fuselage’s. this is because they are supersonic jets thus they need to carry less weight, to provide more speed. Lesser the wing area and the thinner the airfoil, the more speed it will obtain but at a cost of reducing the maneuverability, which is the most important characteristic of an aircraft.

They have sharp nose tips as the lesser the angle formed while cutting the airflow, the more speed it can obtain. The wider the angle while passing through a laminar flow the more drag and less speed will be obtained by any aircraft.

2.3 CONFIGURATION SELECTION

2.3.1 Overall Configuration

Airplanes can be classified as:

1. Land Based
2. Water Based
3. Amphibious

The supersonic business jet is a land based design. Thus, it's overall configuration will be design in such a way that it will satisfy all the requirements for an aircraft that is land based in every manner including the safety while take-off, landing, while in flight, provide a better flight performance to the aircraft etc.

2.3.2 Wing Configuration

From the structural point of view, the wings can be defined as:

1. Cantilever wing
2. Strutted wing

The wings can be classified as:

1. High wing
2. Mid wing
3. Low wing

According to the sweep point of view, wings can be classified as

1. Zero or negligible wing sweep
2. Aft sweep
3. Forward sweep
4. Variable sweep
5. Oblique sweep

the characteristics important to the weight, the performance and the stability control of an airplane are:

1. Aspect ratio
2. Thickness ratio
3. Airfoil(s)
4. Taper ratio
5. Twist

6. Incidence angle
7. Dihedral angle
8. High lift and control surface requirements
9. Winglets

The business jet being designed will have a Cantilever wing with a low wing configuration with a swept-back wing. The characteristics of the wing configuration will be designed in the later section.

2.3.3 Empennage Configuration

The empennage section contains the following parts of the aircraft:

1. Horizontal Tail
2. Vertical Tail
3. Canard's: horizontal or vertical

All the rules that apply to the wing configuration are also applied to the empennage section. The empennage section has the following configurational choices to be made:

1. For the horizontal tail

- Fuselage mounted (usually far aft on the fuselage.
- Boom mounted
- Vertical tail mounted (t-tail/ Cruciform)
- Butterfly or V-Tail

2. For the Vertical Tail

- Fuselage mounted
- Boom mounted
- Single or multiple twin tails
- Butterfly or V-Tail

The supersonic business jet will have a T-tail configuration in the Empennage section. Which is said to be the most suitable design of business jets.

2.3.4 Integration of the Propulsion System

The engines in an aircraft can be arranged as:

1. Tractors
2. Pushers
3. Combination of tractors and pushers

These three basic configurations can be installed in different manners:

1. Pods or Nacelles
2. Buried

Using any configuration or the way of installing them, they can be dispositioned on or in the:
1. Wing
2. Fuselage
3. Empennage

The major consequences created due to the dispositioning of the engines are:
1. Airplane weight
2. Airplane vibration and noise
3. Engine efficiency
4. Handling characteristics
5. Maintenance

The propeller will be located behind the c.g of the aircraft thus it will have a pusher installation configuration. The pusher configuration was selected as they tend to be more stabilizing both in the static longitudinal and static directional stability. This feature can be used to save empennage area.

The engines will be installed in pods/nacelles as they make it very easy and save time during the maintenance of the engines. And help keep them running in the best conditions. They will be installed on the fuselage in the far aft section of the fuselage.

This configuration was selected looking at the major consequences that created due to the dispositioning of the engines. Thus to overcome these consequences, this design is proposed to create the most efficient supersonic business jet.

2.3.5 Landing Gear Disposition

The landing gears on an aircraft can be classified as
1. Fixed or non-retractable
2. Retractable

Depending upon the layouts, landing gears can be classified as:
1. Taildraggers
2. Conventional or tricycle
3. Tandem
4. Outrigger

The landing gears can be mounted in or on the:
1. Wing/Nacelle
2. Fuselage

The proposed supersonic business jet will have a retractable type landing gear as it prevents from producing a lot of drag, gives longer ranges, needs comparatively less power while flying the aircraft.

It will have a conventional or a tricycle type configuration. While is the most used and trusted configuration to balance the aircraft. During take-off/landing or while at rest.
The landing gears will be mounted on the wing beyond the c.g, as the propulsion system is attached behind the c.g. to balance the aircraft, the landing gears will be attached before the c.g.

The number of mail gear struts along with the number of tires to be installed will be decided later while designing the landing gear configuration.

2.3.6 Proposed Configuration

The proposed configuration is a land based design. It has a cantilevered low wing configuration with a swept back angle. It has a T-tail configuration in the empennage section. Having a conventional landing gear composition with the engines attached far aft of the fuselage behind the c.g of the aircraft. The engines will be attached using the pods mounted on the fuselage which makes them easy to maintain or change whenever necessary.

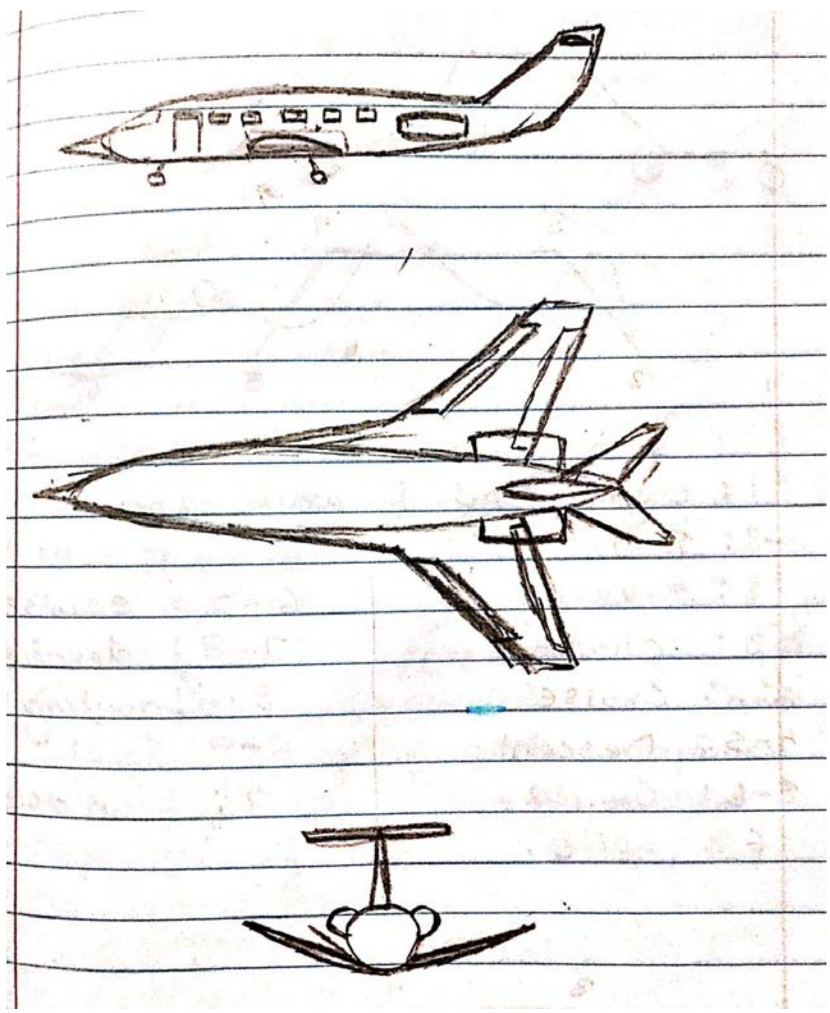


Figure 12: Projected side view, Top view and front view of the supersonic business jet

CHAPTER 3: WEIGHT SIZING AND WEIGHT SENSITIVITIES

3.1 INTRODUCTION

This is the third report in the series of 12. The main purpose of this report is to calculate the preliminary weight estimations using hand written calculations and the AAA program. The preliminary weight estimation process is an important factor while designing an aircraft. It gives a clear idea of what the weight of an aircraft would be after every stage of the flight path for example how much fuel will be required through-out its journey from starting the engine till it lands and shuts off. It gives a rough estimation of how much payload the aircraft will be able to lift through its journey etc.

In this report first the regression points A and B are calculated of different aircrafts similar to the design of my aircraft using the maximum takeoff weights and the empty weights. A graph of this regression points is constructed forming an equation. This equation is then solved using the estimated maximum takeoff weight of my aircraft which will provide a value that will define the allowable/required empty weight of the aircraft. Once this value is obtained, the empty weight of the aircraft will be calculated using the manual calculations using the formulas obtained from the Aircraft Design book by Jan Roskam. The data obtained from these calculations will be compared with the data obtained from the calculations done through the AAA program. The graphs and the regression coefficients obtained through both the methods will be compared.

Once all the data required is obtained, sensitivities of different parameters with respect to the take-off weight will be calculated through both manually and using the AAA program. The sensitivities are useful to obtain various data about the aircraft being designed such as its drawbacks, Advantages etc. parameters such as the empty weight, range, payload, endurance, the L/D ratio, the specific fuel consumption are all compared with the takeoff weight to figure out what changes will be observed if the flight was increased or decreased by a mile throughout it’s journey, how much takeoff weight will be increased if the empty weight was increased, how fast will the airplane climb if the L/D was to be increased or decreased., how much fuel will be used or how much more will the aircraft travel if the payload was increased or decreased from the total allowable weight. All these topics will be discussed in the sections below.

Once the sensitivities are conducted, the trade studies will be done on the existing parameters of the aircraft to see what differences will occur to the design and the mission of the aircraft if certain values are increased or decreased.

3.2 MISSION WEIGHT ESTIMATES

3.2.1 Database for takeoff weights and empty weights of similar Airplanes.

Table 3: Database of takeoff and empty weights of similar airplanes

Airplanes	MTOW (lbs)	Empty Weights (lbs)
Aerion AS2	121000	47250
Aerion SBJ	90000	45100
Spike S-512	115000	47250
Sukhoi Gulfstream S-21	114200	54167
Tupolev Tu-444	90400	42550

3.2.2 Determination of Regression Coefficients A and B.

3.2.2.1 Determination of Regression Coefficients Using Manual Calculations.

The regression coefficients A and B can be found using the natural log of the Maximum Takeoff Weights and the Empty Weights of the aircrafts similar to my aircraft.

Table 4: Database for Natural Log values of the takeoff and empty weights of similar airplanes

Airplanes	Log_{10}(MTOW)	Log_{10}(We)
Aerion AS2	5.08278537	4.67440181
Aerion SBJ	4.954242509	4.65417654
Spike S-512	5.06069784	4.67440181
Sukhoi Gulfstream S-21	5.057666104	4.73373478
Tupolev Tu-444	4.95616843	4.62889956

Comparing these equations and forming a graph with the natural log of the Maximum Takeoff Weight on the X-axis and the Empty Weight on the Y-axis gives the graph obtained below.

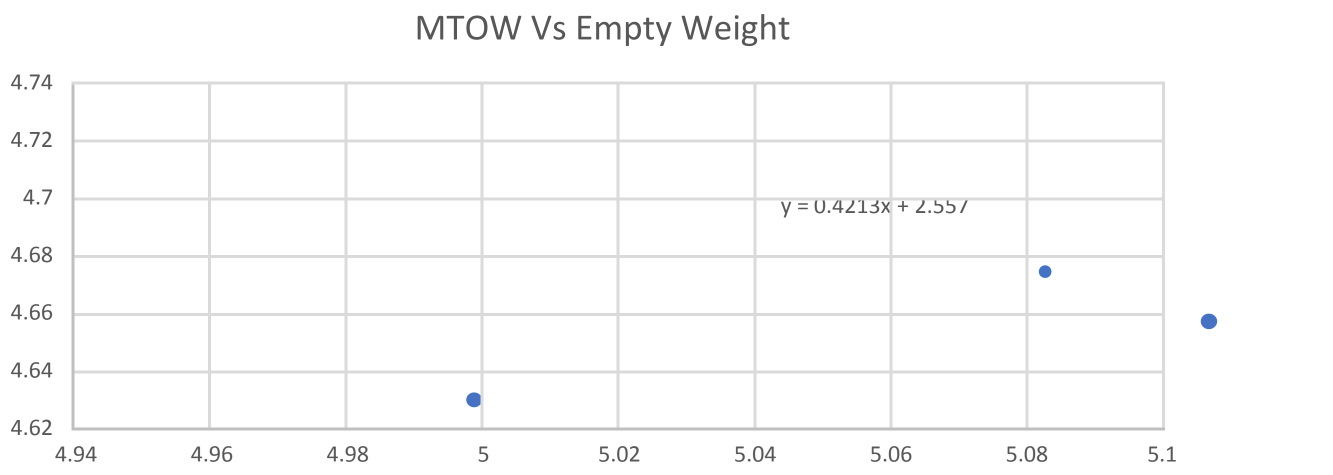


Figure 13: Graph of the MTOW vs Empty weight to calculate the regression coefficients

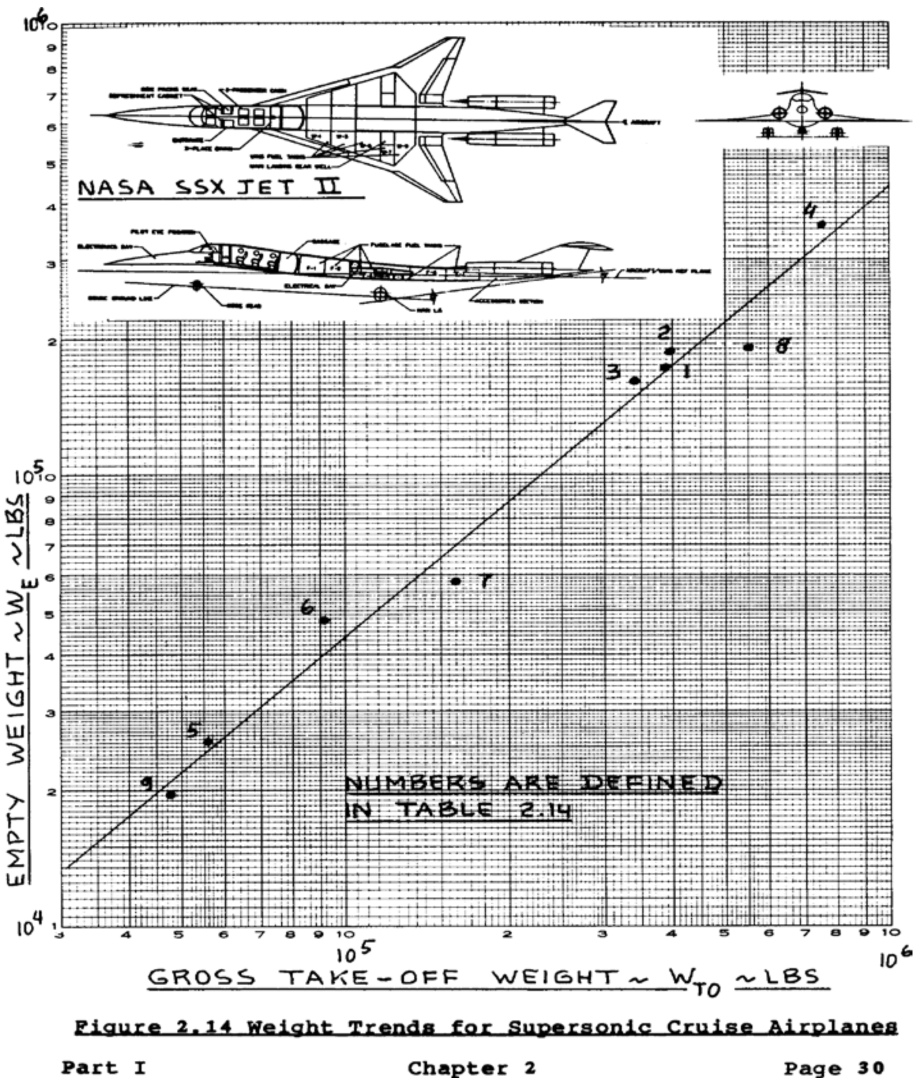


Figure 14: Weight comparison of similar aircrafts obtained from the book

This graph provides an equation in the form of **"y = mx + c"**. The equation obtained from the graph is

$y = 0.4213x + 2.557$ **1**

This equation obtained is then compared with the natural log equation given in the book by Jan Roskam.

$W_e = inv.log_{10}\{(log_{10}W_{TO} - A)/B\}$ **2**

In equation 2 the W_{TO} represents the Maximum Takeoff Weight considered while designing the preliminary design of the airplane.

Comparing the equations 1 and 2 we get the approximate value for the Empty Weight (W_e).

$log_{10}W_e = (1/B) * log_{10}W_{TO} - (A/B)$

From 1
A/B = 2.557
1/B = 0.4213
A = -6.069
B = 2.3736

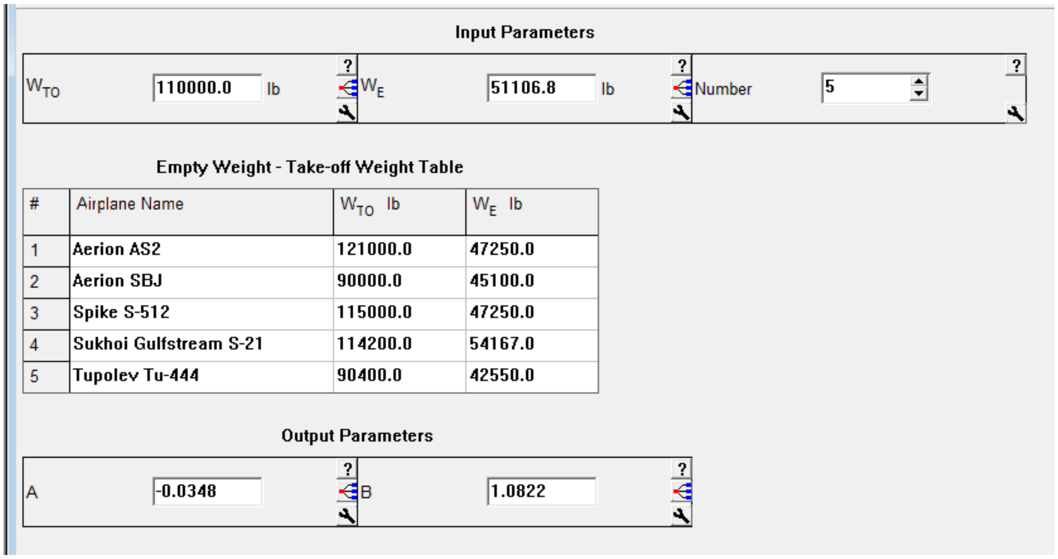
$log_{10}W_e = 0.4213 * log_{10}(1,10,000) + 2.557$

$log_{10}W_e = 0.4213 * 5.041392 + 2.557$

$W_e = 47966.57$ lbs 3

Thus, equation 3 defines the value for the Empty Weight of the airplane. Thus, the manual calculation of the Empty weight estimation using the formulas provided in the Book, should vary about +/- 5% of the Empty Weight obtained through the Regression coefficients.

3.2.2.2 Determination of the Regression Coefficients using the AAA Program.



Input Parameters

W_{TO}	110000.0 lb	W_E	51106.8 lb	Number	5

Empty Weight - Take-off Weight Table

#	Airplane Name	W_{TO} lb	W_E lb
1	Aerion AS2	121000.0	47250.0
2	Aerion SBJ	90000.0	45100.0
3	Spike S-512	115000.0	47250.0
4	Sukhoi Gulfstream S-21	114200.0	54167.0
5	Tupolev Tu-444	90400.0	42550.0

Output Parameters

A	-0.0348	B	1.0822

Figure 15: Table of similar airplanes obtained using the AAA program to calculate the regression coefficients

Entering the take-off and empty weights of the similar airplanes in the AAA program, the regression Coefficients are obtained. The Regression points A and B obtained are different from the AAA program are different from those obtained using the manual calculation technique.

The regression coefficients obtained from the AAA program are:
A= -0.0348
B= 1.0822

Once this data is obtained, graphs are plotted and the design point is defined which plots the SSBJ comparing it with other similar designs. The graphs of the design points are defined below in the graphs obtained from the AAA Program. The design point is the point where the SSBJ stands in comparison with the similar aircrafts.

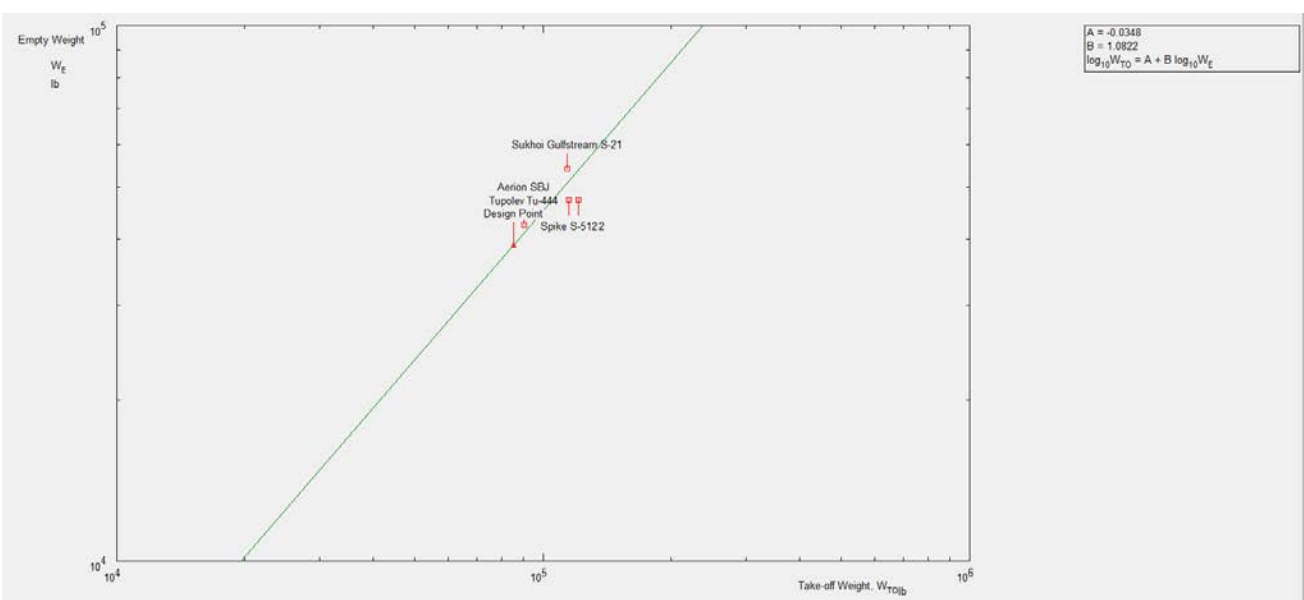


Figure 16: Graph of the regression coefficients obtained using the AAA program

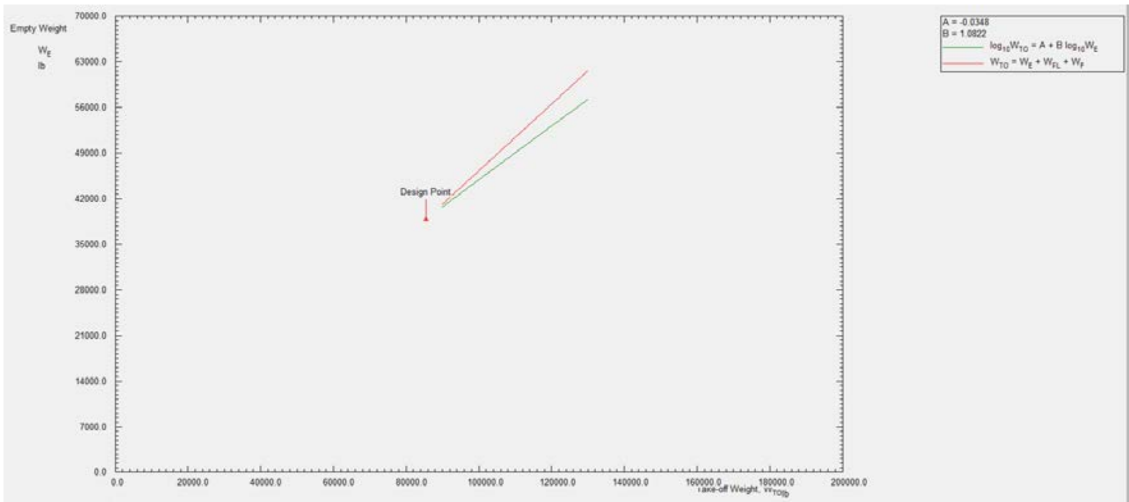


Figure 17: Design point of the SSBJ obtained from the AAA program

3.2.3 Determination of Mission Weights

3.2.3.1 Manual Calculation for the Mission Weights.

The manual calculations of the weight estimation require the maximum takeoff weight of the aircraft, the proposed range of the aircraft, the loiter time of the aircraft, the weight fractions of the aircraft at every stage of the flight path which can be obtained from the book. The cruising speed and more importantly the flight path.

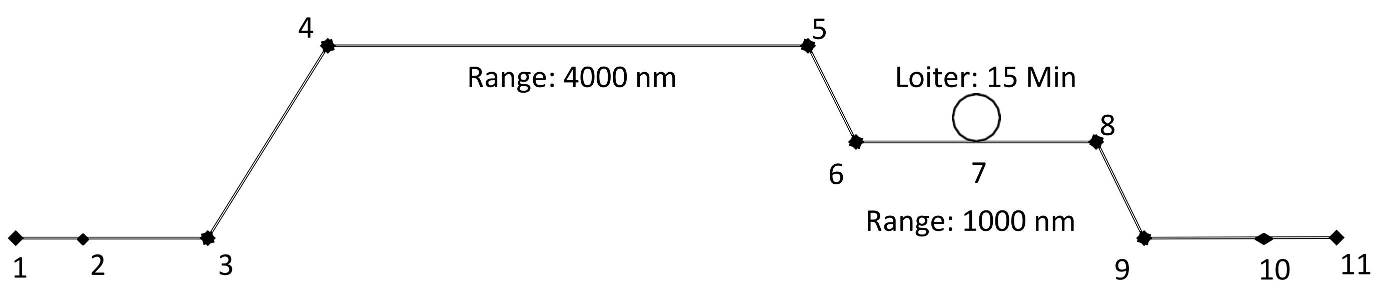


Figure 18: Mission profile of SSBJ Design

1: Engine Start
1-2: Taxi
2-3: Take-off
3-4: Climb
4-5: Cruise
5-6: Descent
6-7: Cruise
7: Loiter
7-8: Cruise
8-9: Decent

9-10: Land
10-11 : Taxi, Shutdown

Thus, the M_{ff} that is the Maximum Fuel Fraction used throughout the path can be obtained by multiplying the fuel fractions at every stage of the flight. The fuel fractions for cruise and loiter are to be calculated using the given formulas:

Cruise: $R_{32} = \mathrm{M}\frac{\mathrm{D}}{\mathrm{N_0}}{}_{32}\mathrm{P}\ \ \mathrm{F}\frac{\pm}{=}\mathrm{H}_{32}\ \ln\left(\frac{\mathrm{T^U}}{\mathrm{T_V}}\right)$ 4

Loiter: $E_{\mathrm{YG2}} = \mathrm{M}\frac{\mathrm{Z}}{\mathrm{N_{0[6)}}}\mathrm{P}\,\mathrm{F}\frac{\pm}{=}\mathrm{H}_{\mathrm{YG2}}\ \ln\left(\frac{\mathrm{T}\backslash}{\mathrm{T]}}\right)$ 5

The Fuel Fractions at different stages of the flight are:

Engine Start, Warmup: $\frac{\mathrm{T}^\wedge}{\mathrm{T}\_}= 0.990$ 6

Taxi: $\frac{\mathrm{T^d}}{\mathrm{T}^\wedge}= 0.995$ 7

Takeoff: $\frac{\mathrm{T^f}}{\mathrm{T_d}}= 0.995$ 8

Climb: $\frac{\mathrm{T^U}}{\mathrm{T_f}}= 0.92$ 9

Cruise for 5000nm (using eqn 4): $\frac{\mathrm{T^V}}{\mathrm{T_U}}=\frac{\mathrm{T^{\wedge h}}}{\mathrm{T_i}}$

- $5753.897 = \frac{\mathrm{Zmmn.op}}{>.\mathrm{q}} * 12 * \ln\left(\frac{\mathrm{T^U}}{\mathrm{T_V}}\right)$
- $\frac{\mathrm{T^U}}{\mathrm{T_V}}= 1.4212$
- $\frac{\mathrm{T^V}}{\mathrm{T_U}}= 0.7036$ 10

Descent: $\frac{\mathrm{T^v}}{\mathrm{T_V}}=\frac{\mathrm{T^{\wedge h}}}{\mathrm{T_i}}= 0.99$ 11

Cruise for 300nm (using eqn 4): $\frac{\mathrm{T}\backslash}{\mathrm{T_v}}$ and $\frac{\mathrm{T^i}}{\mathrm{T]}}$

- $345.234 = \frac{\mathrm{oZp.wZx}}{>.\mathrm{q}} * 12 * \ln\left(\frac{\mathrm{T^V}}{\mathrm{T_v}}\right)$
- $\frac{\mathrm{T^V}}{\mathrm{T_v}}= 1.043$
- $\frac{\mathrm{T^v}}{\mathrm{T_V}}=\frac{\mathrm{T]}}{\mathrm{T}\backslash}= 0.9586$ 12

Loiter (for 1 hour) (using eqn 5): $\frac{T]}{T\backslash}$

- $1 = \frac{Z}{>.w} * 14 * \ln F\frac{T\backslash}{T]}H$
- $\frac{T\backslash}{T]} = 1.0588$
- $\frac{T]}{T\backslash} = 0.9444$ 13

Landing, Taxi, Shutdown: $\frac{T^{\wedge\wedge}}{T^{\wedge}h} = 0.992$ 14

The Mff (Maximum Fuel Fraction) can be found by taking the product of equations 6 through 14. As shown below:

$$M'''' = \frac{T^{\wedge}}{T\_`} * \frac{Td}{T^{\wedge}} * \frac{Tf}{Td} * \frac{TU}{Tf} * \frac{TV}{TU} * \frac{Tv}{TV} * \frac{T\backslash}{Tv} * \frac{T]}{T\backslash} * \frac{Ti}{T]} * \frac{T^{\wedge}h}{Ti} * \frac{T^{\wedge\wedge}}{T^{\wedge}h} \qquad 15$$

$$M'''' = 0.99 * 0.995 * 0.995 * 0.95 * 0.7036 * 0.99 * 0.9586 * 0.9444 * 0.9586 * 0.99 * 0.992$$

$$M'''' = 0.5353 \qquad 16$$

The Maximum Fuel Fraction obtained is 0.5353

The amount of Fuel used ($M''_{\#\$\%\&}$)

$$W''_{\#\$\%\&} = z1 - M''''| * M_{;<} \qquad 17$$

$$W''_{\#\$\%\&} = (1 - 0.5353) * 110000$$

$$W''_{\#\$\%\&} = 51115.55\ lbs \qquad 18$$

$$W''_{)\%\$} = 5\%\ of\ W''_{\#\$\%\&}$$

$$W''_{)\%\$} = 2555.77\ lbs \qquad 19$$

$$W'' = W''_{\#\$\%\&} + W''_{)\%\$} \qquad 20$$

$$W'' = 53671.33\ lbs \qquad 21$$

Eqn 21 provides the weight of fuel carried by the airplane to complete its entire journey.

Calculating the weight of the payload:

$$W_{*+} = W_{,-../01/2.} + W_{32/4} \qquad 22$$

$$W_{,-../01/2.} + W_{32/4} = 20 * 205 + 4 * 205$$

$W_{*+} = 4920\ lbs$ 23

The total weight of the payload consists of the weight of the crew, weight of the passengers considering the weight equal to 175 lbs per person and the baggage weight equal to 30 lbs per person which totals to 205 lbs per person which gives the total payload of the aircraft equal to 4920 lbs as described in the eqn 23.

The total empty weight of an aircraft can be found by removing the fuel weight and the payload from the maximum takeoff weight which leaves us with an empty structure. As shown below:

$W_{5_{6\%76}} = W_{;<} - W_{"} - W_{*+}$ 24

$W_{5_{6\%76}} = 110000 - 53671.33 - 4920$

$W_{5_{6\%76}} = 51408.66\ lbs$ 25

The equation 25 shows the empty weight of an aircraft obtained by the manual calculation of the mission weights.

Thus, comparing the empty weights of the aircraft obtained from the regression coefficients and manual calculations of the weight estimation process, the difference obtained is shown in the calculations below,

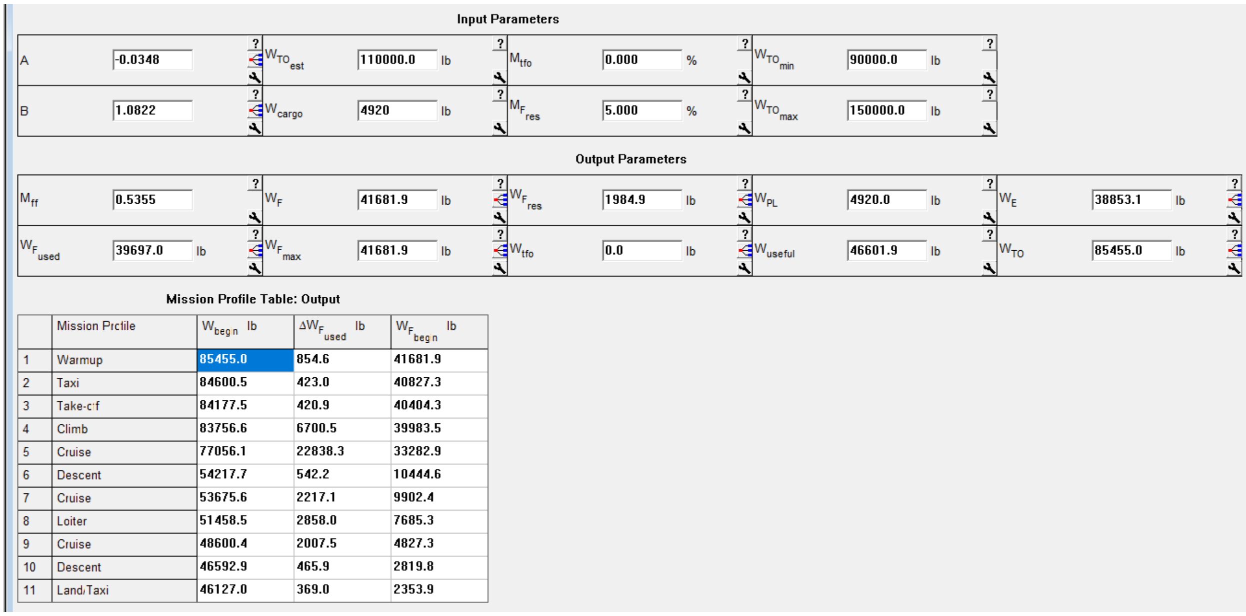
From eqns 3 and 25

$Difference\ in\ empty\ weights = W_{5_{6\%76}} - W_{5} = 51408.66 - 47966.57$

$Difference\ in\ empty\ weights = 3442.09 = \frac{\text{pèèm.>q}}{\text{ènqoo.xn}}$ = 0.071 26

Thus, the difference between the estimated weight and the desired empty weight is 0.071%.

3.2.3.2 Calculations of Mission Weights using the AAA Program.



	Mission Profile	W_{begin} lb	$\Delta W_{F_{used}}$ lb	$W_{F_{begin}}$ lb
1	Warmup	85455.0	854.6	41681.9
2	Taxi	84600.5	423.0	40827.3
3	Take-off	84177.5	420.9	40404.3
4	Climb	83756.6	6700.5	39983.5
5	Cruise	77056.1	22838.3	33282.9
6	Descent	54217.7	542.2	10444.6
7	Cruise	53675.6	2217.1	9902.4
8	Loiter	51458.5	2858.0	7685.3
9	Cruise	48600.4	2007.5	4827.3
10	Descent	46592.9	465.9	2819.8
11	Land/Taxi	46127.0	369.0	2353.9

Figure 19: Mission Weights obtained through the AAA Program

3.3 TAKEOFF WEIGHT ESTIMATIONS

3.3.1 Manual calculations of takeoff weight sensitivities.

To manually calculate the takeoff weight sensitivities, we will need the values for C and D which are constants similar to the A and B regression coefficients. The values of C and D can be found by the following equation:

$$W_5 = W_{;<} ê\ 1 - (1 + M_{2/.})z1 - M_{""}| - M_{G"ëí} - (W_{*+} + W_{32/4})$$

where,

$$C = ê\ 1 - (1 + M_{2/.})z1 - M_{""}| - M_{G"ëí} \quad 27$$

$$D = (W_{*+} + W_{32/4}) \quad 28$$

the values for C and D obtained by substituting the terms in the eqn's 27 and 28 are as follows:

$$C = \{ 1 - (1 + 0.05)(1 - 0.5353)$$

$$C = 0.5122$$

$$D = 4920\ lbs \quad 29$$

Sensitivities of the below mentioned parameters are being studied:

- Payload, W_{PL}
- Empty Weight, W_E

- Range, R
- Endurance, E
- Lift-to-drag ratio, L/D
- Specific fuel consumption, c_j

The reason for conducting the sensitivity study is to find out which parameters drive the design, to determine what changes can be made if ever a new mission capability is to be achieved, they also provide an estimate of the impact caused when changes are made to the design.

➢ **Takeoff weight sensitivities:**

$$log_{10} W_{TO} = A + B log_{10}(C W_{TO} - D) \qquad 30$$

substituting the values of A, B, C and D from the equations 2 and 29, we get the allowable value of the takeoff weight.

$$\log W_{TO} = -6.069 + 2.3736 * \log W_{TO}(0.5122 * W_{TO} - 4920)$$

solving the equation, we get the allowable takeoff weight equal to 99263.5 lbs.

➢ **Sensitivity of takeoff weight to payload weight:**

Taking the derivative of eqn 30 and Considering y $=W_{PL}$, then $\frac{\partial D}{\partial W_{PL}} = 1.0, \frac{\partial C}{\partial W_{PL}} = 0.$

$$\text{Therefore, } \frac{\partial W_{TO}}{\partial W_{PL}} = \frac{B(W_{TO})^2 \frac{\partial C}{\partial W_{PL}} - B W_{TO} \frac{\partial D}{\partial W_{PL}})}{(C(1-B)W_{TO} - D)} \qquad 31$$

The following data can be obtained from the preliminary design,
A= -6.069
B= 2.3736
C= 0.5122
D= 4920 lbs
W_{TO}= 1,10,000 lbs

Substituting the data in equation 31, we get the sensitivity of W_{TO} to W_{PL}.

$$\frac{\partial W_{TO}}{\partial W_{PL}} = 3.60$$

Thus, this means that for every pound of payload added, the airplane take-off gross weight will have to be increased by 3.60 lbs. thus the factor 3.60 is termed as the growth factor due to payload for the SSBJ. According to the results, the mission performance remains the same.

➢ Sensitivity of Takeoff Weight to Empty Weight:

The sensitivity of the empty weight can be calculated by taking the derivative of the equation below:

$$\log_{10} W_{TO} = A + B\log_{10} W_E \qquad 32$$

The derivative of the eqn 32 is as follows:

$$\frac{\partial W_{TO}}{\partial W_E} = B W_{TO}\left\{inv\log_{10}\left[\frac{\log_{10} W_{TO} - A}{B}\right]\right\}^{-1} \qquad 33$$

Substituting the data obtained from the preliminary design and weight sizing of the aircraft, we get,

$$\frac{\partial W_{TO}}{\partial W_E} = 5.44 \qquad 34$$

This shows that to increase each pound in the empty weight, the take-off weight must be increased by 5.44 lbs. the factor 5.44 is the growth factor due to the empty weight.

➢ Sensitivity of Take-off Weight to Range, Endurance and Speed.

For the SSBJ, following data is found:

B= 2.3736
C= 0.5122
D= 4920 lbs
M_{res}= 0.05
M_{ff}= 0.5353
F= 1,96,118.97 lbs
W_{TO}= 1,10,000 lbs

The factor F in the data is found using the equation:

$$F = -B(W_{TO})^2\{C.W_{TO}(1-B) - D\}^{-1}(1+M_{res})M_{ff} \qquad 35$$

Substituting the values of the regression coefficient constants, maximum fuel fraction, reserved fuel and the Maximum takeoff weight in the equation 35, we get the value of F.

$$F = 1,96,118.97\ lbs$$

For Cruise:
c_j = 0.9
L/D = 12
V = 1066.78 knots

For Endurance:

$C_j = 0.8$
L/D = 14

The sensitivities of takeoff weight to the range and endurance can be written as:

$$\frac{\text{òT`}}{\text{ò¨}} = Fc_{ú}\,\mathrm{F}V\,\frac{+}{=}\mathrm{H}^{\dagger Z}$$

$$\frac{\text{òT`}}{\text{ò¨}} = 13.788\ lbs/nm \qquad 36$$

$$\frac{\text{òT`}}{\text{ò5}} = Fc_{ú}\,\mathrm{F}\frac{+}{=}\mathrm{H}^{\dagger Z}$$

$$\frac{\text{òT`}}{\text{ò5}} = 11{,}206.79\ lbs/hr \qquad 37$$

These sensitivities show that if the range in the mission specification is decreased by 1 nm, then the gross weight can be decreased by 13.788 lbs. similarly if the loiter requirements are increased from 1 hour to 1.2 hours, then the take-off gross weight will be increased by (1/5)*11,206.79 = 2,241.35 lbs.

$$\frac{\text{òT`}}{\text{òD}} = -FRc_{ú}\,\mathrm{F}V^{\mathrm{m}}\frac{\pm}{=}\mathrm{H}^{\dagger Z}$$

$$\frac{\text{òT`}}{\text{òD}} = -74.965\ lbs/kt \qquad 38$$

This parameter proves that if the cruise speed can be increased without bringing any change in any other parameter, the gross weight would gradually decrease.

- **Sensitivity of Take-off Weight to the Specific Fuel Consumption and Lift to Drag Ratio:**

With respect to the range requirement:

$$\frac{\text{òT`}}{\text{ò3}_0} = FR\,\mathrm{F}V\frac{+}{=}\mathrm{H}^{\dagger Z} \qquad 39$$

$$\frac{\text{òT`}}{\text{ò3}_0} = 88{,}856.96\ lbs/lbs/lbs/hr$$

If the specific fuel consumption was incorrectly assumed to be 0.8 instead of 0.9, the gross take-off weight will be increased by 0.1*88,856.96 = 8,885.69 lbs.

$$\frac{\text{òT`}}{\text{òF}\frac{\text{ö}}{\text{o}}\mathrm{H}} = -FRc_{ú}\,\mathrm{MV}\,\mathrm{F}\frac{+}{=}\mathrm{H}^{\mathrm{m}\,\dagger Z}\ \mathrm{P} \qquad 40$$

$$\frac{\grave{o}T\ `}{\grave{o}F\ddot{o}H} = -6664.27\ lbs.$$

If the lift to drag ratio of the airplane was 13 instead of 12, the design gross takeoff weight would be decreased by 6664.27 lbs.

With respect to the Loiter Requirements:

$$\frac{\grave{o}T\ `}{\grave{o}3_0} = FE\ F\pm H^{\dagger Z}_{=} \qquad 41$$

$$\frac{\grave{o}T\ `}{\grave{o}3_0} = 14{,}008\ lbs/lbs/lbs/hr$$

If the specific fuel consumption during loiter could be improved to 0.7 from 0.8, the gross empty weight would decrease by 1400.8 lbs.

$$\frac{\grave{o}T\ `}{\grave{o}F\ddot{o}H} = -FEc\ {}_{\acute{u}}\ F\pm H^{\dagger m}_{=} \qquad 42$$

$$\frac{\grave{o}T\ `}{\grave{o}F\ddot{o}H} = -800.48\ lbs$$

If the lift to drag ration during loiter was to be improved from 14 to 15, the gross takeoff weight would be reduced by 800.48 lbs.

3.3.2 Calculation of Take-off Weight Sensitivities using the AAA Program.

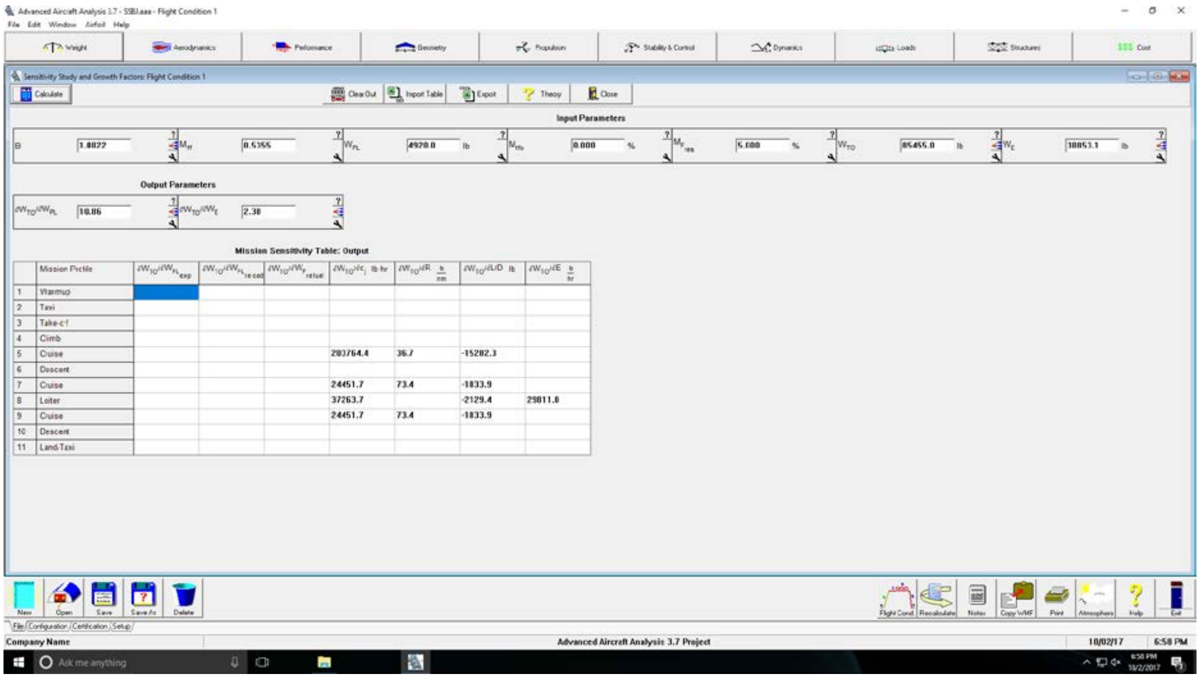


Figure 20: Weight Sensitivities using the AAA program

3.3.3 Trade Studies

Trade studies are done over the parameters considered for the weight estimations. The first trade study is done between the range and the payload. Where the maximum takeoff weight is considered to be constant throughout the process. The main purpose of the trade study is to get the best design point for the aircraft being designed. In this case the aircraft being the Supersonic Business Jet, the best design point is the 5000-nm range carrying a payload weight of 4920 lbs.

As shown in the graph below, it proves that with the increase in the range the payload decreases. Any mission profile can be created using this design from short range-more payload to long range-less payload. This configuration depends on the needs of the consumer.

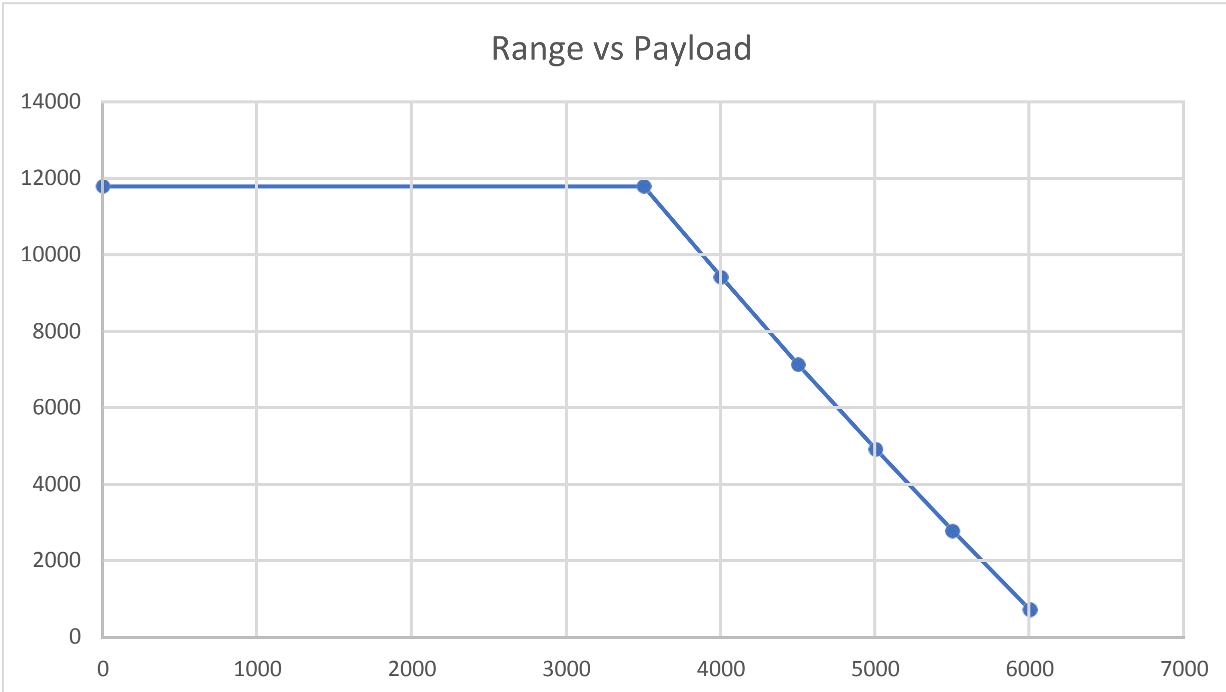


Figure 21: Trade study between the Range vs Payload

The second trade study is done between the maximum takeoff weight and the lift-to-drag ratio. In this case the empty weight of the aircraft is considered to be constant. With the increase in the Lift-to-Drag ratio the weight of the fuel will decrease which will decrease the maximum takeoff weight of the aircraft while decreasing the L/D ratio will increase the takeoff weight of the aircraft. The best value to be considered for the lift-to-drag ratio according to the Aircraft Design book by Jan Roskam is 12.

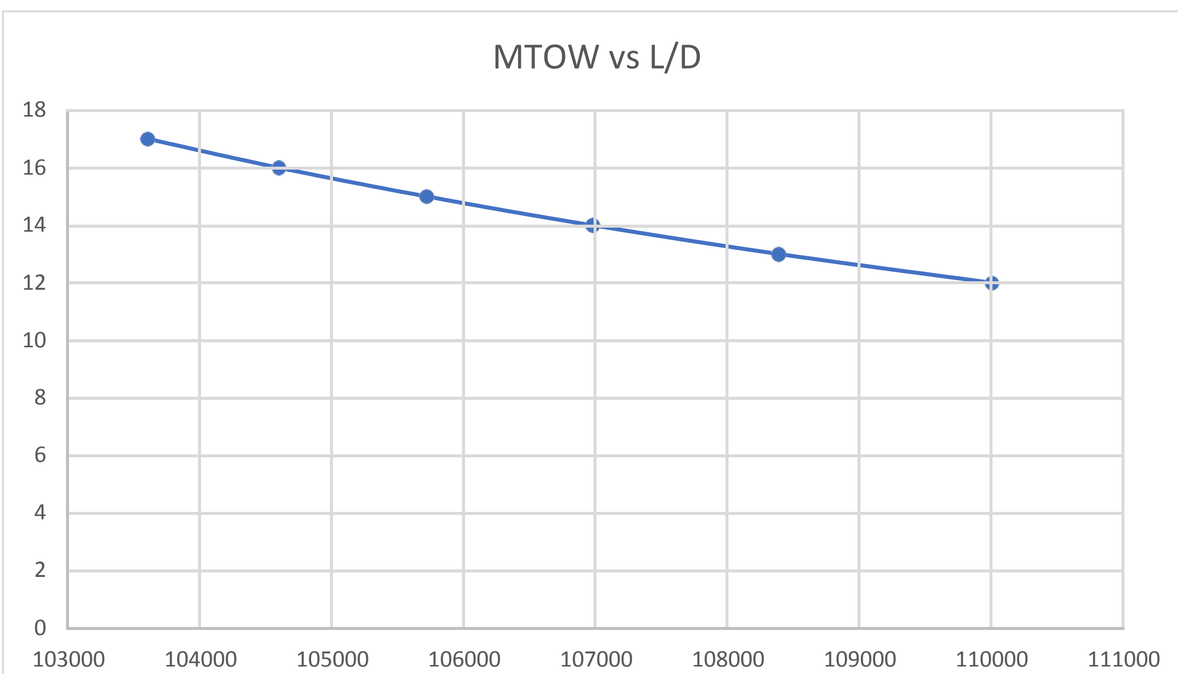


Figure 22: Trade study between the MTOW and L/D

3.4 Discussion

Being the third report of the series, the weight sizing and mission requirements have been discussed in this report. First the regression points were calculated using the data of other similar airplanes. Once the regression coefficients are obtained, the Empty weight and the Takeoff weights were calculated. In the initial stage, certain values were considered as an assumption such as the maximum takeoff weight which was considered to be 1,10,000 lbs. after the manual calculations of the regression coefficients, the AAA program was used for the calculations for the regression coefficients. Both the values obtained through the manual calculations and from the AAA program are comparatively different. The points obtained though the manual calculation are A= -6.069 and B= 2.3736, while the points obtained through the AAA program are A= -0.0348 and B= 1.0822.

After the calculation of the regression coefficients, the mission weights at different stages were calculated using manual calculations and using the AAA program, the maximum fuel fraction remains the same for both the methods. After obtaining the maximum fuel fraction of the aircraft for different stages, the empty weight is calculated by eliminating the fuel weight and the weight of the payload.

Once all the data regarding the weights is available, the sensitivities of the parameters are conducted to see what changes are affected to the design by changing the values of the parameters. These sensitivities are done by two methods, one by the manual calculations and the other using the AAA program. Comparing the sensitivities from both the methods gives minor differences between the values obtained from the manual calculations and the AAA program.

The takeoff weight sensitivities are somewhat similar to the trade studies. The sensitivities show what differences will be observed in the design if the values are changed by a unit difference for example how much weight will be increased or decreased if the range is increased or decreased by 1 nautical mile. While the trade studies show what parameter of the aircraft we are trading for to obtain the other for example the trade study for range vs payload proves that if the range is to be increased, the payload has to be decreased considering the maximum takeoff weight as a constant. While if the payload is to be increased, it has to be traded by the range meaning the range will be decreased with increase in the payload of the aircraft.

3.5 CONCLUSION AND RECOMMENDATIONS

3.5.1 Conclusion

The goal of this report is to calculate the weight sizing of the SSBJ using manual calculation using the formulas obtained from the Airplane design book by Jan Roskam and using the AAA program. Once all the data is obtained, they will be compared between the two methods of calculating the weight estimations to obtain accurate results to design the aircraft. It is also necessary to compare the aircraft being designed to be compared with similar airplanes to check if the calculations or the weight estimations are comparable or not.

The results obtained from all the above calculations can be summarized as follows:

- The maximum takeoff weight and the empty weight are considered to be constant thus the parameters that will change for different kind of missions are the range, payload, total weight of the fuel required.
- The Regression coefficients obtained from the AAA program are very much different from the coefficients obtained by manual calculations. The coefficients obtained from the AAA program are A= -0.0348 and B= 1.0822 while the coefficients obtained from the manual calculations are A= -6.069 and B= 2.3736.
- The sensitivity study shows that minor changes to the parameters affects the airplane design. Improving one parameter can disturb the requirements of the other parameter.

3.5.2 Recommendations

From this report, many lessons were learnt regarding the weight sizing of the aircraft. The results from the trade studies and sensitivities proved that not much freedom is possible to change the parameters freely. The future works consists of designing the other parts of the aircraft in a detailed format.

CHAPTER 4: PERFORMANCE CONSTRAINT ANALYSIS

4.1 INTRODUCTION

This is the fourth report in the series of 12. The main purpose of this report is to make a list of the performance constraints and show the calculations according to which the airplane must me sized. The airplanes are usually designed to meet performance objectives in the following categories:

a) Stall speed
b) Take-off field length
c) Landing field length
d) Cruise speed
e) Climb rate
f) Time to climb up to a certain altitude.
g) Maneuvering

Various calculations will be performed which will allow the rapid estimation of the airplane design parameters that have a major impact in the airplane design. Some of the parameters are:

a. Wing Area, S
b. Take-off thrust, T_{TO}
c. Maximum required take-off lift coefficient, $C_{L_{max}}$ $(CLEAN)$.
d. Maximum Required Lift Coefficient for Take-off, $C_{L_{max_{TO}}}$.
e. Maximum Required Lift Coefficient for Landing, $C_{L_{max_L}}$.

A range of values of wing loading (W/S), thrust loading (T/W) and the maximum lift coefficient $(C_{L_{max}})$ are considered with in which certain performance requirements are met. According to the data obtained, the airplane with the lowest weight and the lowest cost can be obtained by considering the lowest possible thrust loading and the highest possible wing loading which still meets all the performance requirements. The Wing Area (S) and the Takeoff Thrust (T_{TO}) can be obtained from these calculations.

Once the manual calculations are obtained, the AAA program will be used to perform the calculations and obtain the values of the performance Constraints. These values will then be compared with the calculations obtained from the manual calculations. After comparing the final results, they will be summarized.

The propulsion system that is sufficient and that matches the requirements will be selected along with the number of engines to be used according to the design specification to obtain the suitable thrust for the airplane to fly in supersonic speeds. The propulsion system will be specified along with its sizing, the performance and the components of the engine. And how it satisfies the requirements of the airplane.

4.2 MANUAL CALCULATION OF PERFORMANCE CONSTRAINTS

4.2.1 Stall Speed

For airplanes, a stall speed not higher than some minimum value is required. The airplanes certified under the FAR 25 categories have no minimum stall speed requirement. The stall speed can be calculated from the following equation:

$$V_s = \sqrt{\frac{2 * \frac{W}{S}}{\rho * C_{L_{max}}}} \quad 1$$

where,

V_s = Stall speed (knots)

W/S = Wing Loading

$C_{L_{max}}$ = Maximum Coefficient of Lift

substituting the values in the equation, we get the stall speed equal to

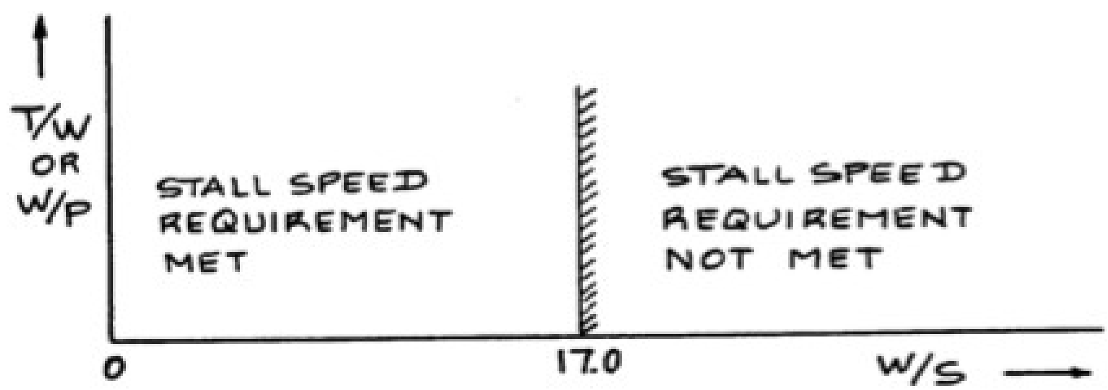


Figure 23: Example of the Stall speed sizing

The values of the $C_{L_{max}}$ can be obtained from the table below.

Airplane Type	$C_{L_{max}}$	$C_{L_{max_{TO}}}$	$C_{L_{max_L}}$
1. Homebuilts	1.2 - 1.8	1.2 - 1.8	1.2 - 2.0*
2. Single Engine Propeller Driven	1.3 - 1.9	1.3 - 1.9	1.6 - 2.3
3. Twin Engine Propeller Driven	1.2 - 1.8	1.4 - 2.0	1.6 - 2.5
4. Agricultural	1.3 - 1.9	1.3 - 1.9	1.3 - 1.9
5. Business Jets	1.4 - 1.8	1.6 - 2.2	1.6 - 2.6
6. Regional TBP	1.5 - 1.9	1.7 - 2.1	1.9 - 3.3
7. Transport Jets	1.2 - 1.8	1.6 - 2.2	1.8 - 2.8
8. Military Trainers	1.2 - 1.8	1.4 - 2.0	1.6 - 2.2
9. Fighters	1.2 - 1.8	1.4 - 2.0	1.6 - 2.6
10. Mil. Patrol, Bomb and Transports	1.2 - 1.8	1.6 - 2.2	1.8 - 3.0
11. Flying Boats, Amphibious and Float Airplanes	1.2 - 1.8	1.6 - 2.2	1.8 - 3.4
12. Supersonic Cruise Airplanes	1.2 - 1.8	1.6 - 2.0	1.8 - 2.2

Figure 24: Typical values for the maximum lift coefficient

The values of the $C_{L_{max}}$ ranges from 1.2 to 2.2 for the supersonic cruise airplane with different stages having the Coefficient of Lifts as:
$C_{L_{max}} = 1.2 - 1.8$
$C_{L_{max_{TO}}} = 1.6 - 2.0$
$C_{L_{max_L}} = 1.8 - 2.2$

The $C_{L_{max}}$ is highly influenced by the factors such as:

a) Wing and Airfoil Design
b) Flap type and Flap Size
c) Center of Gravity Location

Assuming the Stall Speed $(V_s) = 190\ Knots$

$$190 = \sqrt{\frac{2*(\frac{W}{S})}{0.002377*1.8}}$$

$\frac{W}{S} = 77.22\ lb/ft^2$ 2

similarly, the values of the Wing Loading W/S for all the values of $C_{L_{max}}$ in all 3 states, clean, takeoff and landing can be obtained.

Table 5: Calculations of the W/S at different coefficient of Lift

CL	**CL(TO)**	**CL(L)**
1.2	1.7	1.9
1.4	1.8	2
1.6	1.9	2.1
1.8	2	2.2
V(s)	190	knots
rho(SL)	0.002377	slugs/ft3
W/S (clean)	**W/S(TO)**	**W/S(L)**
51.48582	72.938245	81.519215
60.06679	77.22873	85.8097
68.64776	81.519215	90.100185
77.22873	85.8097	94.39067

According to the table above the Values of the wing loading vary at different stages and conditions of the flights path. According to the book Aircraft design by Jan Roskam, the actual value of the W/S considered is the value at maximum coefficient of lift ($C_{L_{max}}$) during the clean stage that is 77.22 lb/ft^2.

The wing loading for similar Aircrafts is shown in the table below:

Table 6: Wing Loading (W/S) for similar Airplanes

Airplane	Wing Loading (lb/ Sqft)
Aerion SBJ	75
Aerion AS2	89.62
Spike S-512	68.37
Sukhoi Gulfstream S-21	53.4
Tupolev Tu-444	61.91

4.2.2 Takeoff Distance

The takeoff distances of airplanes are determined by the following factors:

a) takeoff weight, W_{TO}
b) Takeoff Speed, V_{TO}

c) Thrust-to-weight Ratio at take-off $(T/W)_{TO}$
d) Aerodynamic Drag Coefficient, C_D and ground friction coefficient μ_G.
e) Pilot Technique

The Take-off field lengths differ widely and depend on the type of airplane being considered. For civil airplanes, the requirements for FAR 23 and FAR 25 must be met. The supersonic business jet falls under the FAR 25 category, thus it is proved that the takeoff field length s_{TOFL} is proportional to the wing loading $(W/S)_{TO}$, take-off thrust to weight ratio $(T/W)_{TO}$ and the maximum takeoff life coefficient, $C_{L_{max_{TO}}}$. This can be defined from the equation below:

$$s_{TOFL} = \frac{(W/S)_{TO}}{\sigma \, C_{L_{max_{TO}}} (T/W)_{TO}} = TOP_{25} \quad (3)$$

where, TOP_{25} = take-off parameter for FAR 25 certified airplanes. The unit for the TOP_{25} is lbs/ft^2.
The $s_{TOFL} = 37.5 * TOP_{25}$. 4

The values for $C_{L_{max_{TO}}}$ are given in the figure 1 above.
hence the equation 3 can be written as:

$$s_{TOFL} = 37.5 * \frac{(W/S)_{TO}}{\sigma \, C_{L_{max_{TO}}} (T/W)_{TO}} = 37.5 * TOP_{25} \quad (5)$$

Hence from the above equation, we can say that

$$TOP_{25} = \frac{(W/S)_{TO}}{\sigma \, C_{L_{max_{TO}}} (T/W)_{TO}} \quad (6)$$

The FAR 25 is defined according to the figure shown below.

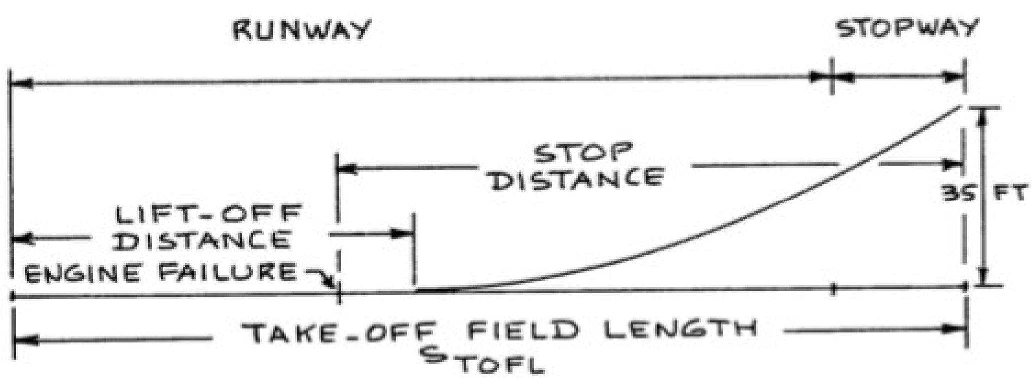


Figure 25: Definition of FAR 25 take-off distance

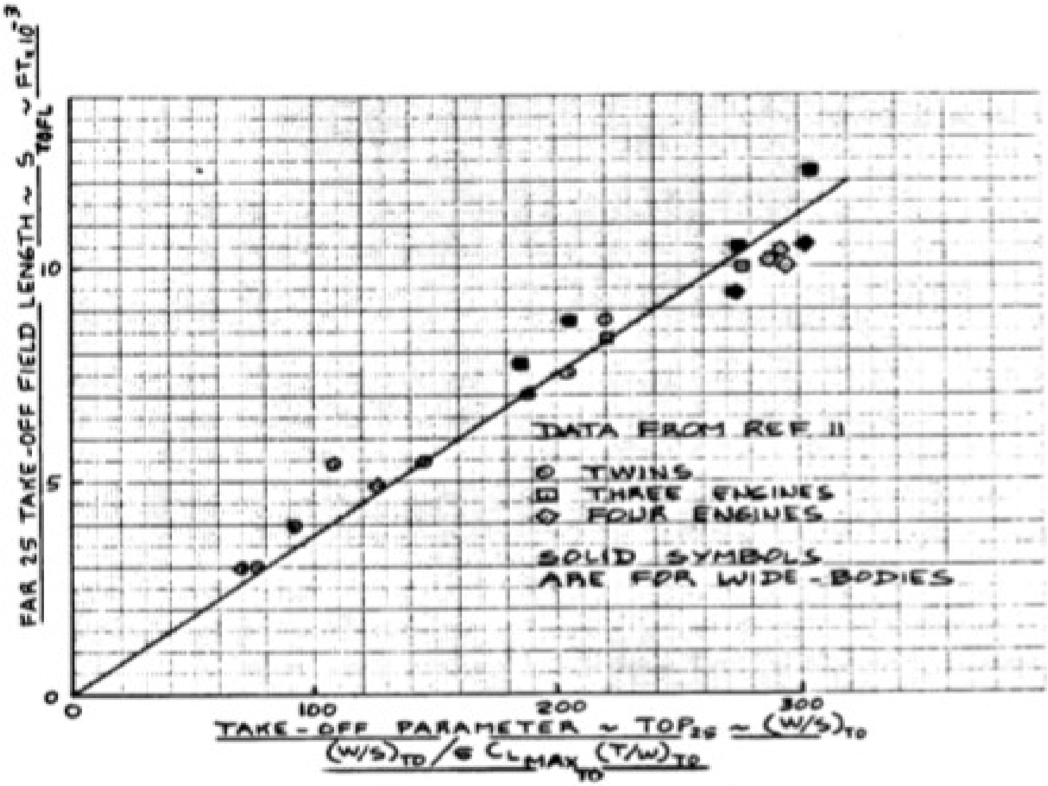


Figure 26: Effect of take-off parameter, TOP 25 on FAR 25 take-off field length

For s_{TOFL} = 5000 ft at sea level,

$$TOP_{mx} = \frac{\text{-`¬ö}}{\text{pn.x}} = \frac{\text{x>>>}}{\text{pn.x}}$$

$$TOP_{mx} = 133.34$$

Substituting the value of TOP_{25} , s_{TOFL}, $C_{+_{\pm®ì;<}}$, W/S_{TO}, and the air density ratio equal to 1 in equation 6, we get the values of the Thrust to Weight Ratio of the airplane at different stages of flight at different coefficient of lifts. It proves that the s_{TOPFL} is proportional to the take-off wing loading.

Calculation of T/W at s_{TOFL} = 5000 ft

Table 7: 〖〖C_L〗_MAX〗_TO Vs W/S to obtain the T/W ratios

C_L_MAX_TO W/S	1.7	1.8	1.9	2
50	0.220577206	0.208322917	0.197358553	0.187490625
55	0.242634927	0.229155209	0.217094408	0.206239688
60	0.264692648	0.249987501	0.236830264	0.224988751

65	0.286750368	0.270819792	0.256566119	0.243737813
70	0.308808089	0.291652084	0.276301974	0.262486876
75	0.33086581	0.312484376	0.29603783	0.281235938
80	0.35292353	0.333316667	0.315773685	0.299985001
85	0.374981251	0.354148959	0.33550954	0.318734063
90	0.397038972	0.374981251	0.355245396	0.337483126

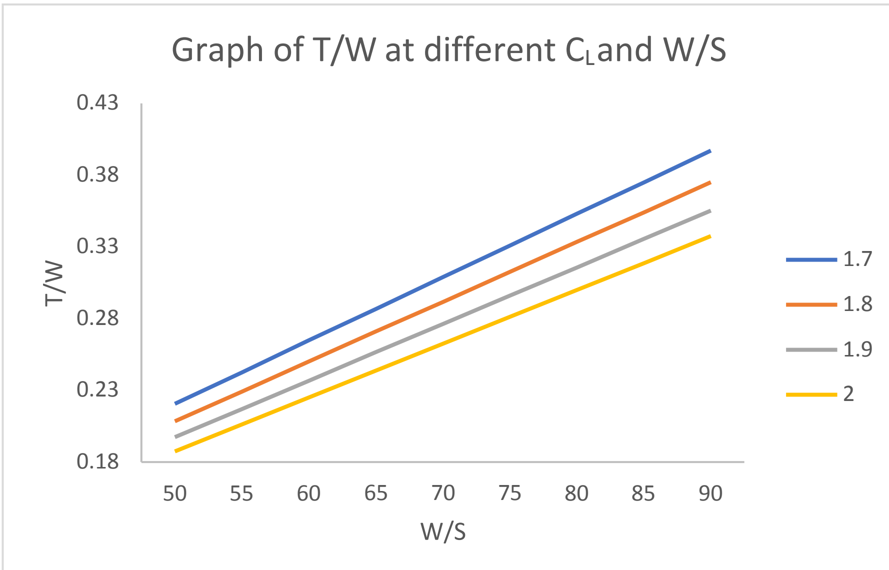


Figure 27: Graph of T/W at different C_L and W/S

The graph of T/W at different wing loading and different coefficient of lifts is obtained according to the calculations shown in table 3 at s_{TOFL} equal to 5000 ft.

4.2.3 Landing Distance

The landing distances of airplanes are dependent on five factors. Those factors are mentioned below:

1. Landing Weight, W_L
2. Approach Speed, V_A
3. Deceleration method used
4. Flying qualities of the airplane
5. Pilot technique

The landing distance of the aircraft is always dependent on the design landing weight of the aircraft. Considering the kinetic energy, the approach speed should have a 'square effect' on the total landing distance. Once the airplane touches the ground, the following methods can be used to decelerate:

a. Brakes

b. Thrust reversers
c. Parachutes
d. Arresting systems
e. Crash barriers

The typical values for landing weights to the take-off weights are shown in the figure below:

Airplane Type	W_L/W_{TO} Minimum	Average	Maximum
1. Homebuilts	0.96	1.0	1.0
2. Single Engine Propeller Driven	0.95	0.997	1.0
3. Twin Engine Propeller Driven	0.88	0.99	1.0
4. Agricultural	0.7	0.94	1.0
5. Business Jets	0.69	0.88	0.96
6. Regional TBP	0.92	0.98	1.0
7. Transport Jets	0.65	0.84	1.0
8. Military Trainers	0.87	0.99	1.1
9. Fighters (jets)	0.78	insufficient data	1.0
(tbp's)	0.57		1.0
10. Mil. Patrol, Bomb and Transports (jets)	0.68	0.76	0.83
(tbp's)	0.77	0.84	1.0
11. Flying Boats, Amphibious and Float Airplanes (land)	0.79	insufficient data	0.95
(water)	0.98		1.0
12. Supersonic Cruise Airplanes	0.63	0.75	0.88

Figure 28: Landing weight V/S take-off weight

The typical values of landing weight to take-off weights for the supersonic business jets are obtained from the figure above:

FTöH T_` 9?0 $= 0.63$

FTöH T_` ®f1 $= 0.75$

FTöH T_` 9-: $= 0.88$

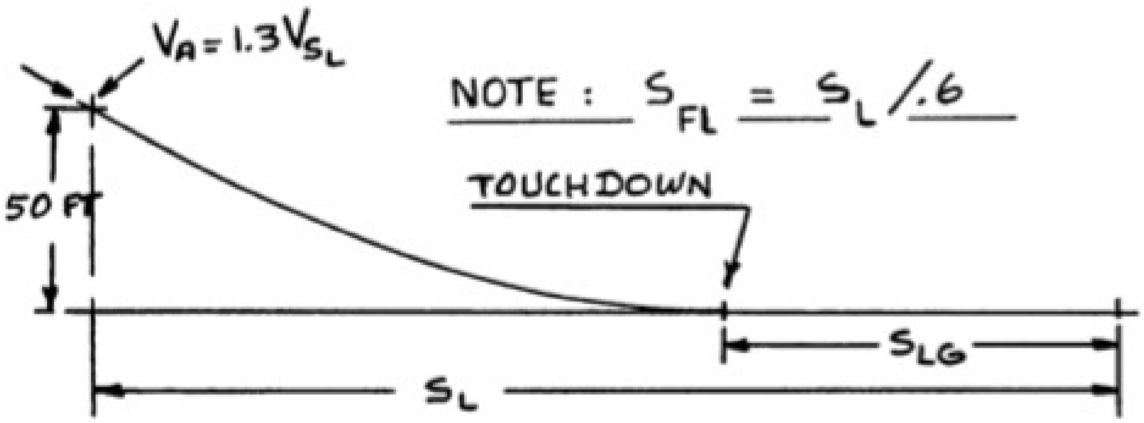


Figure 29: Definition of FAR 25 Landing Distance

The calculations to landing distance sizing is as follows:

V æ+ = μ m*ü » / ≠*Nö≈æÃö ^ ā ″ 7

$V_{-+} = 136.41\ knots$

$V_{®} = 1.3 * V_{-+} = 1.3 * 136.41 = 177.33\ knots$ 8

$s_{ø+} = 0.3\ V_{®}^{m} = 9434.09\ ft$

$s_{+} = s_{ø+} * 0.6 = 5660\ ft$ 9

Hence, the landing distance obtained by the manual calculation method Is 5660 ft, the approach speed obtained is 177.33 knots and the stall speed during landing is 136.41 knots.

Using equation 7, we can formulate the relation between the wing loading and the coefficient of lift.

2 FüH » ö / 36099 >.>>mpnnNö = "Gd / -d

FTH æ + $= 42.90\ C_{+}$ 9-:+ 10

FTH æ ;< = èm.q / >.wx C_{+} 9-:;< $= 50.47\ C_{+}$ 9-:;< 11

From equations 10 and 11 we obtain the relation between the wing loading and the coefficient of life as shown in the table below. For the wing Loading during the landing stage, substituting the coefficients in equation 10, we get:

Table 8: Relation between the lift coefficients and wing loading while landing

Coefficient of Lifts	Wing Loading
1.8	77.22
1.9	81.51
2.0	85.8
2.1	90.09
2.2	94.38

Table 9: Relation between the Lift coefficients and wing loading while take-off

Coefficient of Lift	Wing Loading
1.8	90.846
1.9	95.893
2.0	100.94
2.1	105.987
2.2	111.034

4.2.4 Drag Polar Distance

To size an airplane for climb requirements, it is necessary to have an estimate for the airplane drag polar. The drag polars can be estimated at low speeds by assuming the drag polar to be parabolic and the drag coefficient can be represented by:

$$C_D = C_{D_0} + \frac{C_L^2}{\pi A e} \qquad 10$$

The zero-lift drag coefficient, C_{D_0} can be expressed as:

$$C_{D_0} = \frac{f}{S} \qquad 11$$

where,

f = parasite area

S = wing area

$$\log_{10} f = a + b \log_{10} S_{wet} \qquad 12$$

a and b are themselves a function of the equivalent skin friction coefficient of an airplane, c_f. by estimating the drag, it becomes convenient to predict the value for S_{wet} which correlates well with the W_{TO} for a wide range of airplanes. The wetted area of an aircraft can be determined by the following equation:

$$\log_{Z>} S_{4/G} = c + dlog_{Z>}W_{;<} \quad (13)$$

here in this equation c and d are the regression line coefficients, and can be obtained from the figure 9 as shown below.

Equivalent Skin Friction Coefficient, c_f	a	b
0.0090	-2.0458	1.0000
0.0080	-2.0969	1.0000
0.0070	-2.1549	1.0000
0.0060	-2.2218	1.0000
0.0050	-2.3010	1.0000
0.0040	-2.3979	1.0000
0.0030	-2.5229	1.0000
0.0020	-2.6990	1.0000

Figure 30: Correlation coefficients for parasite area v/s wetted area

Airplane Type	c	d
1. Homebuilts	1.2362	0.4319
2. Single Engine Propeller Driven	1.0892	0.5147
3. Twin Engine Propeller Driven	0.8635	0.5632
4. Agricultural	1.0447	0.5326
5. Business Jets	0.2263	0.6977
6. Regional Turboprops	-0.0866	0.8099
7. Transport Jets	0.0199	0.7531
8. Military Trainers*	0.8565	0.5423
9. Fighters*	-0.1289	0.7506
10. Mil. Patrol. Bomb and Transport	0.1628	0.7316
11. Flying Boats, Amph. and Float	0.6295	0.6708
12. Supersonic Cruise Airplanes	-1.1868	0.9609

Figure 31: Regression line Coefficients for take-off weight v/s the wetted area

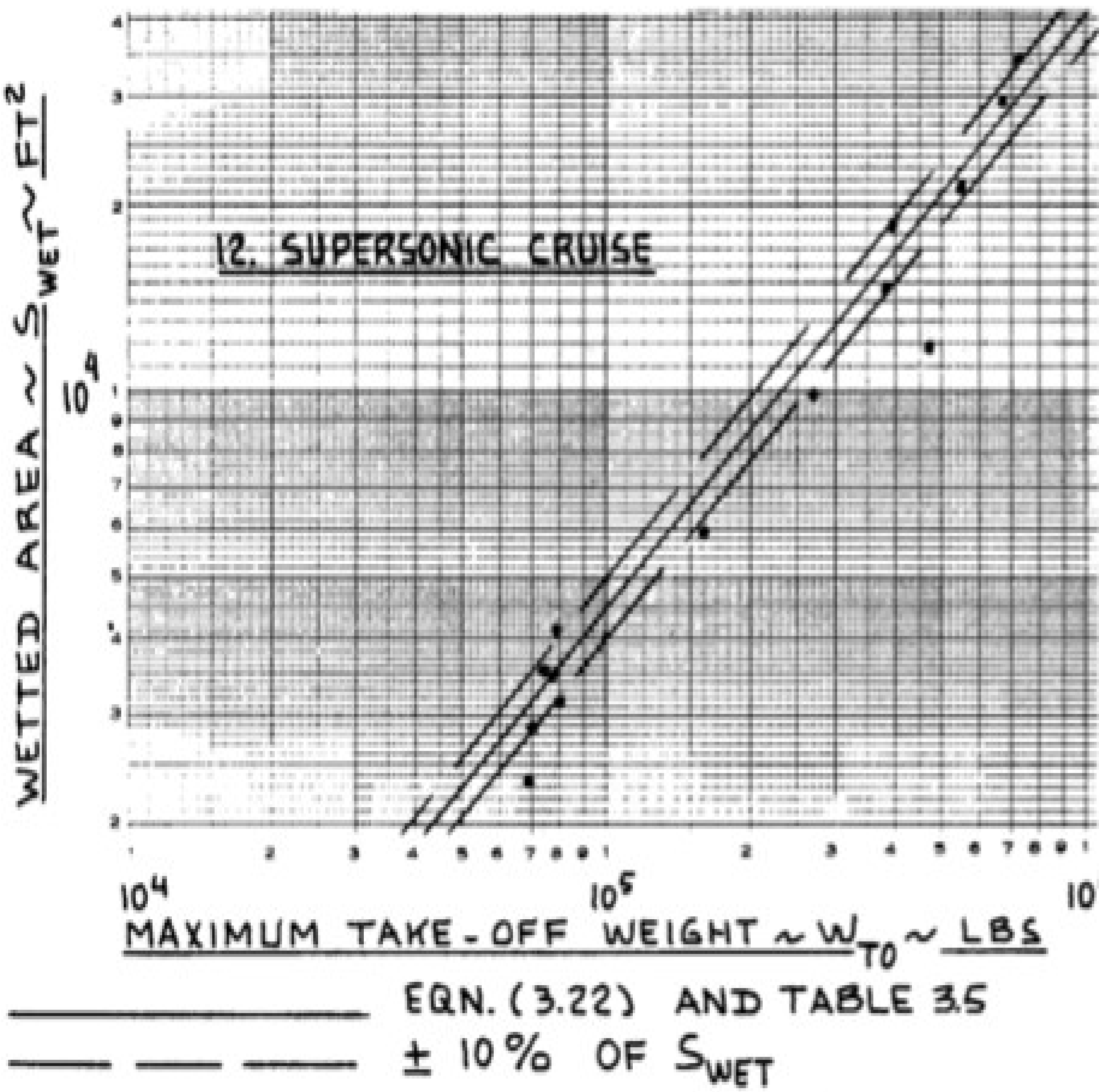


Figure 32: Correlation between the wetted area and take-off weight for a jet with supersonic cruise

Configuration	ΔC_{D_0}	e
Clean	0	0.80 - 0.85
Take-off flaps	0.010 - 0.020	0.75 - 0.80
Landing Flaps	0.055 - 0.075	0.70 - 0.75
Landing Gear	0.015 - 0.025	no effect

Figure 33:estimates for Δ〖C_D〗_0 and e with flaps and gear down.

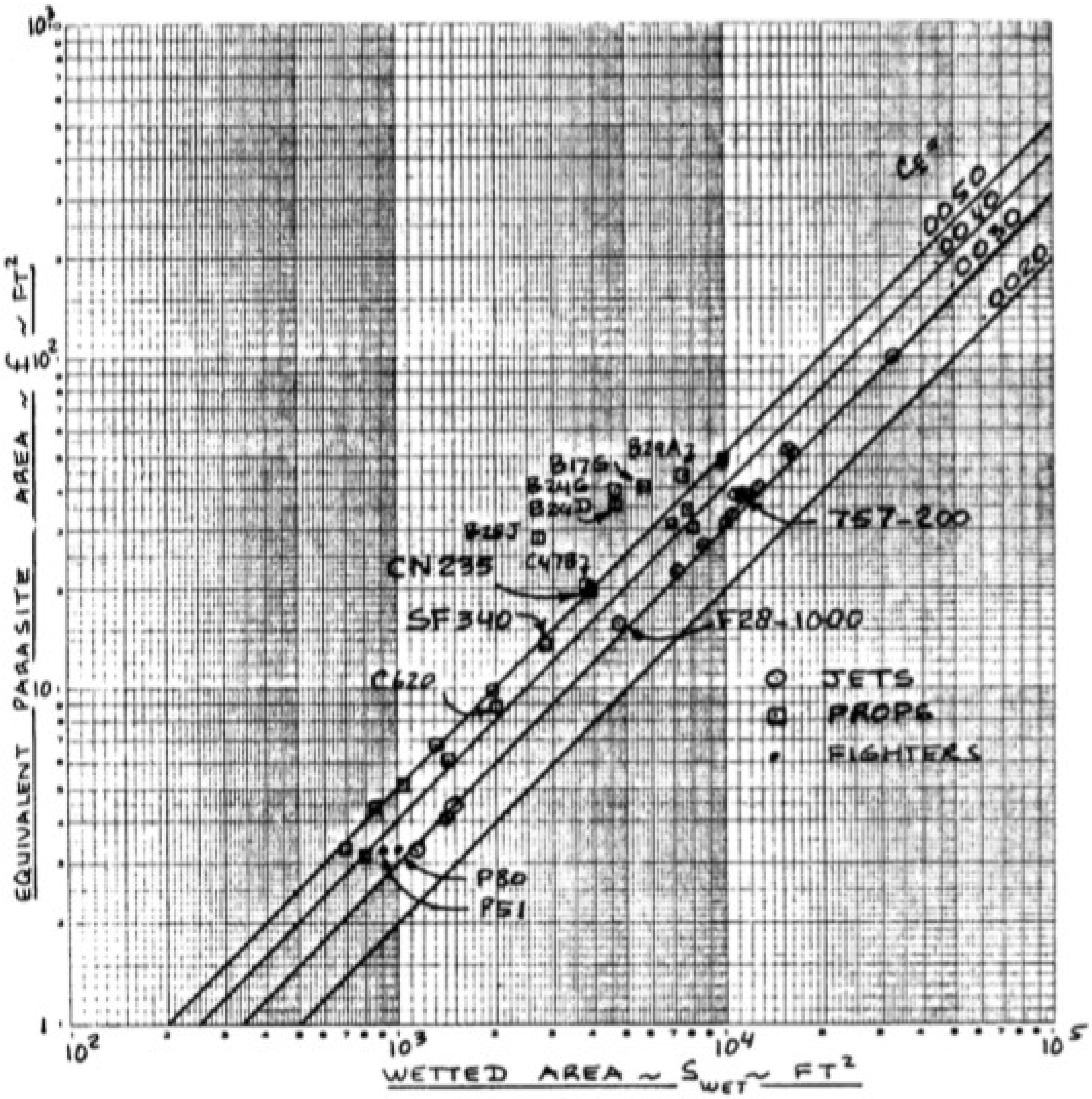


Figure 34: Relation between the wetted area and the equivalent parasite area

The values of the supersonic aircrafts are:

c = -1.1868

d = 0.9609

W_{TO} = 110000 lbs

Substituting the above values in equation 13, we get the value for S_{wet} as:

S_{wet} = 4544.3757 ft^2.

Now, from figure 12 it is evident that the value of the Equivalent skin friction coefficient c_f is equal to 0.0030 hence from figure 8, we can get the value of a and b as:

C_f = 0.0030

a = -2.5229

b = 1
substituting these values in equation 12, we get the value of the equivalent parasite area f:

f = 13.63 ft^2

substituting the values of f and S_{wet} in equation 11, we get the zero lift drag coefficient:
$C_{=_{>}} = 0.00299$

Table 10: Values required for the drag polar calculation

W_{TO}	$(W/S)_{TO}$	S	S_{wet}	F	C_{D_0}
110000	**77.22**	**1424.34**	**4544.3757**	**13.63**	**0.00299**

Configuration	ΔC_{D_o}	e
Clean	0	0.80 - 0.85
Take-off flaps	0.010 - 0.020	0.75 - 0.80
Landing Flaps	0.055 - 0.075	0.70 - 0.75
Landing Gear	0.015 - 0.025	no effect

Figure 35: estimates for Δ 〖C_D〗 _0 and e

Hence, substituting all the values obtained in equation 10 we get the following equations for different conditions of the flight for which the data can be obtained from figure 13 and table 4

For the clean stage: $C_{=} = 0.00299 + 0.0499\ C_{+}^{\mathrm{m}}$ 14

Take-off, Gear up: $C_{=} = 0.01799 + 0.0530\ C_{+}^{\mathrm{m}}$ 15

Take-off, Gear down: $C_{=} = 0.03799 + 0.0530\ C_{+}^{\mathrm{m}}$ 16

Landing, Gear up: $C_{=} = 0.06299 + 0.0566\ C_{+}^{\mathrm{m}}$ 17

Landing, Gear down: $C_{=} = 0.08299 + 0.0566\ C_{+}^{\mathrm{m}}$ 18

4.2.5 Climb Constraints

The climb requirements must be met with the thrust for available removing the installation losses and the accessory operation. The engine thrust must be that for 34 % humidity and standard temperature plus 50 F according to the FAR 25 regulations.

The take-off climb requirements for FAR 25.111 OEI can be summarised as follows:

a. 1.2 percent for two-engine airplanes
b. 1.5 percent for three engine airplanes

c. 1.7 percent for four engine airplanes

The initial climb segment requirements contain the following configurations:

1. Take-off flaps
2. Landing gear retracted
3. Speed
4. Engines at takeoff thrust
5. 35ft to 400ft altitude, ground effect must be accounted
6. Ambient atmospheric conditions
7. Maximum take-off weight

FAR 25.121 (OEI) requirements with the critical engine inoperative is as follows:

a. Positive for two-engine airplanes
b. 0.3 percent for three-3ngine airplanes
c. 0.5 percent for four-engine airplanes

these requirements are for the following conditions:

1. Take-off flaps
2. Landing gear down
3. Remaining engines at take-off thrust
4. Between V_{LOF} and V_2
5. In ground effect
6. Ambient atmospheric conditions
7. At maximum take-off weight

For the second segment climb requirements with one engine inoperative are:

a. 2.4 percent for two-engine airplanes
b. 2.7 percent for three-engine airplanes
c. 3.0 percent for four-engine airplanes

These requirements are for the following configuration:

1. Take-off flaps
2. Landing gear retracted
3. Remaining engines at take-off thrust
4. Speed equal to V_2
5. Out of ground effect
6. Ambient atmospheric conditions
7. At maximum take-off weight

The en-route climb requirement with one engine inoperative, the climb gradient should be no less than the following conditions:

a. 1.2 percent for two-engine airplanes
b. 1.5 percent for three-engine airplanes
c. 1.7 percent for four-engine airplanes

All these apply for the following conditions

1. flaps retracted
2. landing gear retracted
3. remaining engines at maximum continuous thrust
4. speed at $1.25V_S$
5. ambient atmospheric conditions
6. at maximum take-off weight

The landing climb requirements of FAR 25.119 (AEO) is that the climb gradient should be more than 3.2 percent at a thrust corresponding to that obtained eight seconds after moving the throttles from minimum flight. It applies to the following configuration:

1. landing flaps
2. landing gear down
3. speed equal to $1.3\ V_S$
4. ambient atmospheric conditions
5. at maximum design landing weight

FAR 25.121 (OEI) with the critical engine inoperative must be more than the following:

a. 2.1 percent for two-engine airplanes
b. 2.4 percent for three-engine airplanes
c. 2.7 percent for four-engine airplanes

All these must satisfy the following conditions:

1. approach flaps
2. landing gear as defined by normal AEO procedures
3. speed less than $1.5V_{SR}$.
4. V_{SR} must not be more than $1.1V_{S}$
5. Remaining engines at take-off thrust
6. Ambient atmospheric conditions
7. At maximum design landing weight

For jet powered airplanes with one engine inoperative (OEI):

$$\left(\frac{T}{W}\right) = \left(\frac{N}{N-1}\right)\left(\left(\frac{L}{D}\right)^{-1} + CGR\right) \qquad 19$$

For jet powered airplanes with All engines operative (AEO):

$$\left(\frac{T}{W}\right) = \left(\left(\frac{L}{D}\right)^{-1} + CGR\right) \qquad 20$$

where,
CGR: Climb Gradient Required
N: No of engines
L/D: lift to drag ratio of the flight condition
T/W: Thrust to weight ration of the flight condition

To calculate the climb constraints, the following drag polar data will now be assumed:

Table 11: Drag polar data

Configuration	C_{D_0}	A	e	C_{D_i}	$C_{L_{max}}$
Clean	0.00299	7.5	0.85	0.0499 C_L^2	1.8
Take-off flaps	0.01799	7.5	0.8	0.0530 C_L^2	2
Landing flaps	0.06299	7.5	0.75	0.0566 C_L^2	2.2
Gear down	0.08299	7.5	No effect	-	No effect

For FAR 25.111(OEI):

$$\left(\frac{T}{W}\right)_{TO} = 2\left(\frac{1}{L/D} + 0.012\right), \; at \; 1.2\, V_{s_{TO}} \quad 21$$

the value of $C_{L_{max_{TO}}}$ is assumed to be 1.8, the actual lift coefficient in this flight condition is 2/1.44 = 1.389

hence substituting the value of the life coefficient in the equation 15, we get the lift to drag ratio.

C_D = 0.1202

$$\frac{L}{D} = \frac{C_L}{C_D} = 11.55$$

substituting the value of L/D in equation 16, we get the thrust to weight ratio of the airplane while take-off,

$$\left(\frac{T}{W}\right)_{TO} = 2\left(\frac{1}{L/D} + 0.012\right) = 0.197. \quad 22$$

For FAR 25.121(OEI): (gear down, take-off flaps)

$$\left(\frac{T}{W}\right)_{TO} = 2\left(\frac{1}{L/D} + 0\right) \text{ between } V_{LOF} \text{ and } V_2. \quad 23$$

$V_{LOF} = 1.1\, V_{s_{TO}}$

$$C_{L_{LOF}} = \frac{2}{1.1^2} = 1.653$$

$C_D = 0.183$

$\frac{L}{D} = 9.032$

$\left(\frac{T}{W}\right)_{TO} = 0.221$

For FAR 25.121(OEI): (gear up, takeoff flaps)

$$\left(\frac{T}{W}\right)_{TO} = 2\left(\frac{1}{L/D} + 0.024\right), \; at \; 1.2\, V_{s_{TO}} \quad 24$$

C_D = 0.1202

L/D = 11.55

$\mathrm{F}\frac{;}{\mathrm{T}}\mathrm{H}_{;<} = 0.2211$

FOR FAR 25.121(OEI): (gear up, flaps up)

$\mathrm{F}\frac{;}{\mathrm{T}}\mathrm{H}_{;<} = 2\ \mu\frac{\mathrm{Z}}{\ddot{\mathrm{g}}} + 0.012'', \ at\ 1.25\ V_{æ}$ 25

$C_{+} = 1.8 = \frac{\mathrm{Z\,w}}{\mathrm{Z.mx^{d}}} = 1.152$

$C_{=} = 0.069$

L/D = 16.695

$\mathrm{F}\frac{;}{\mathrm{T}}\mathrm{H}_{;<} = 0.1438$

For FAR 25.119 (AEO): balked landing

$\mathrm{F}\frac{;}{\mathrm{T}}\mathrm{H}_{+} = 2\ \mu\frac{\mathrm{Z}}{\ddot{\mathrm{o}}} + 0.032'', \ at\ 1.3\ V_{-+}$ 26 C_L

= 1.301

C_D = 0.1688

L/D = 7.707

$\mathrm{F}\frac{;}{\mathrm{T}}\mathrm{H}_{+} = 0.3235$

For FAR 25.121 (AEO): balked landing

$\mathrm{F}\frac{;}{\mathrm{T}}\mathrm{H}_{+} = 2\ \mu\frac{\mathrm{Z}}{\ddot{\mathrm{o}}} + 0.021'', \ at\ 1.5\ V_{-®}$ 27

C_L = 0.933

C_D = 0.09975

L/D = 9.353

$\mathrm{F}\frac{;}{\mathrm{T}}\mathrm{H}_{+} = 0.2558$

Airplane Type	h_{abs} (ft)x10^{-3}
Airplanes with piston-propeller combinations:	
normally aspirated	12-18
supercharged	15-25
Airplanes with turbojet or turbofan engines:	
Commercial	40-50
Military	40-55
Fighters	55-75
Military Trainers	35-45
Airplanes with turbopropeller or propfan engines:	
Commercial	30-45
Military	30-50
Supersonic Cruise Airplanes (jets)	55-80

Figure 36: Critical values for the absolute ceiling, h_abs

Finding the rate of climb for jet aircrafts:

$RC = V$ µF ; H − Z″ T ö 28

where, the velocity V = mF ü H » ρzN h π®/ |d 29

substituting the values in equation 29, we get V = 334.07 knots

further substituting the value of V in equation 28, we get,

RC = 8666.56 ft/m^2 30

4.2.6 Maneuvering Constraints

The Maneuvering constraints are specifically for utility, agricultural, aerobatic and for military airplanes.

4.2.7 Speed Constraints

For maximum speed, the following equations are used:

$$T_{2/AB} = C_{=}qS \qquad 31$$

$$W = C_{+}qS \qquad 32$$

Now, if the drag polar is assumed to be parabolic, then the equation is defined as:

$$T_{2/AB} = C_{=_{>}}qS + \frac{\text{N}^{\text{d}}\text{Aæ}\ \text{ö}}{\text{¨®/}} \qquad 33$$

dividing this equation by the weight, we get:

$$\left(\frac{\text{F;H}}{\text{T}}\right)_{2/AB} = \frac{\text{N}^{\circ}\text{h}^{\text{Aæ}}}{\text{T}} + \frac{\text{T}}{\text{Aæ}\pi\text{®/}} \qquad 34$$

arranging equation 34 in a proper format, we get the relation between the thrust-weight ratio and wing loading,

$$\left(\frac{\text{F;H}}{\text{T}}\right)_{2/AB} = \frac{\text{N}\circ_{\text{h}}\text{A}}{\frac{\text{ü}}{\text{»}}} + \frac{\frac{\text{ü}}{\text{»}}}{\text{A}\pi\text{®/}} \qquad 35$$

Substituting the values, we get the equation as:

$$\left(\frac{\text{F;H}}{\text{T}}\right)_{2/AB} = \frac{\text{è.>èè}}{\frac{\text{ü}}{\text{»}}} + \frac{\frac{\text{ü}}{\text{»}}}{\text{mn>nè.pq}} \qquad 36$$

Now, substituting the values of W/S from table 1, we can get different values of the thrust to weight ratio as:

Table 12: Relation between Wing Loading and the required Thrust to weight ratio

W/S	$(T/W)_{reqd}$
51.48	0.08
60.06	0.07
68.64	0.061
77.22	0.055

4.3 CALCULATION OF THE PERFORMANCE CONSTRAINTS WITH THE AAA PROGRAM

4.3.1 Stall Speed

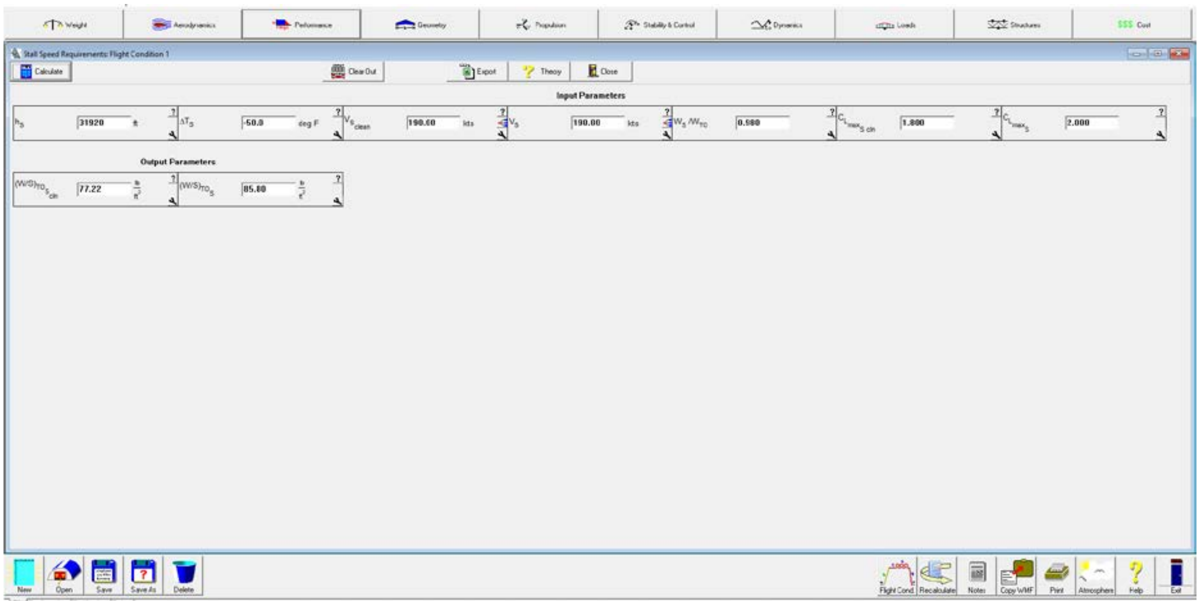


Figure 37: Stall Speed calculation using the AAA program

The stall speed considered is 190 knots. Which gives a wing loading of 77.22 lb/ft^2 and the wing loading during take-off to stall is 85.80 lb/ft^2. The stall is obtained at the height of 31920 ft from the sea level at temperature difference being 50 deg F less than that at sea level.

4.3.2 Takeoff Distance

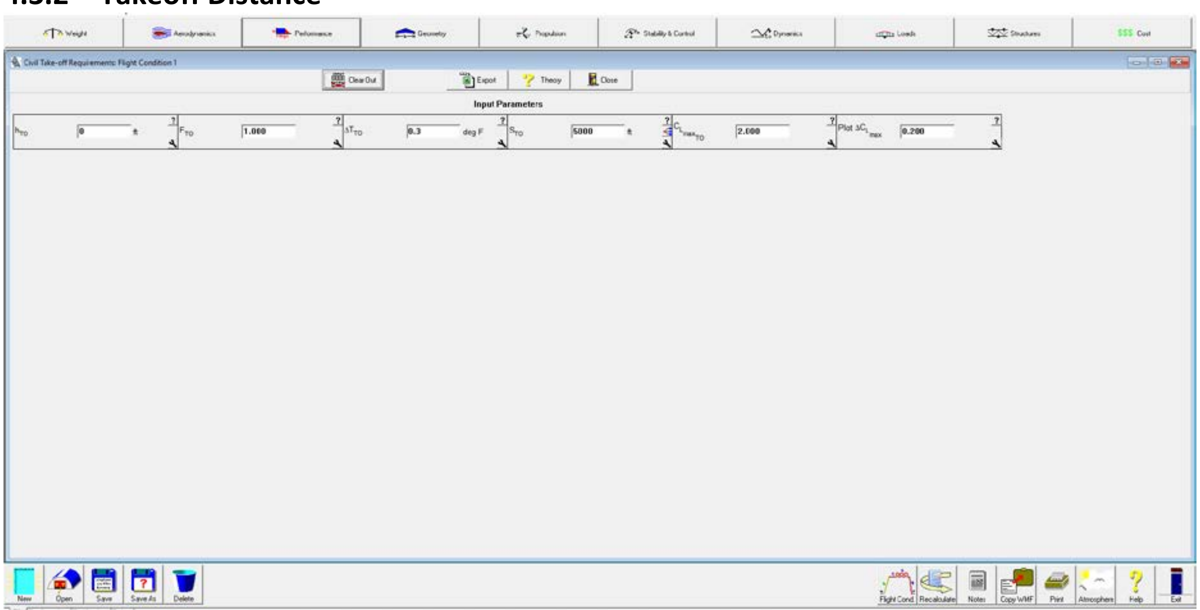


Figure 38: Take-off distance parameters using the AAA program

For Take-off parameters, no output is provided it only requires the input parameter obtained from manual calculations.

S_{TO} = 5000 ft

4.3.3 Landing Distance

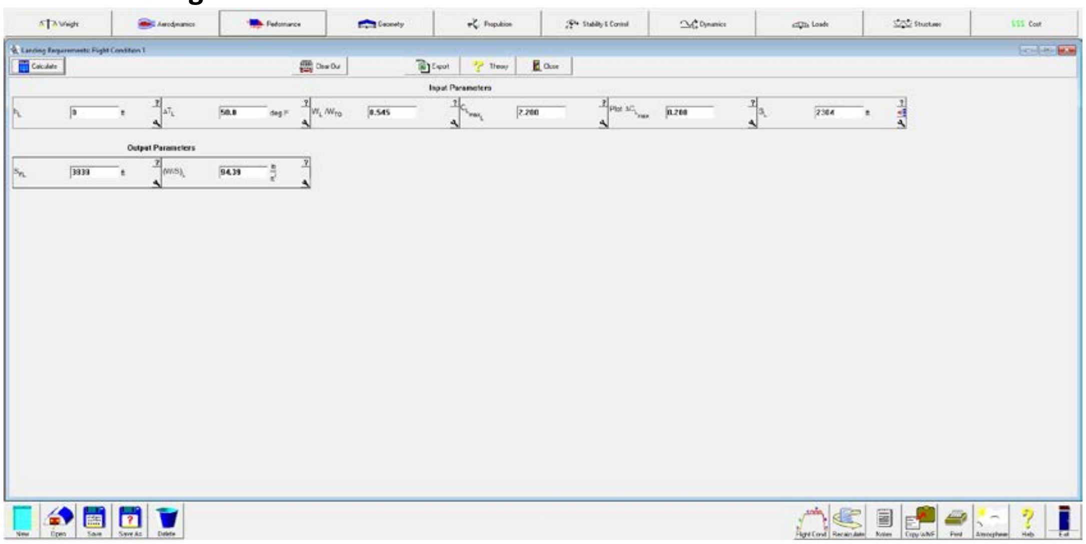


Figure 39: Landing distance calculations using the AAA program

For the landing parameters, the output obtained is the size of the field length and the wing loading required while landing. The outputs are described as follows:

S_{FL} = 3839 ft

$(W/S)_L$ = 94.39 lb/ft^2

4.3.4 Drag Polar Distance

The Drag Polars are included in the Climb constraints in figure 18. The drag polars obtained during the clean, take-off and landing stages are as follows:

$B_{=^*{}_{3Y/-0}} = 0.0499$

$B_{=^*{}_{;<\_Bë40}} = 0.0531$

$B_{=^*{}_{+\_Bë40}} = 0.0566$

4.3.5 Climb Constraints

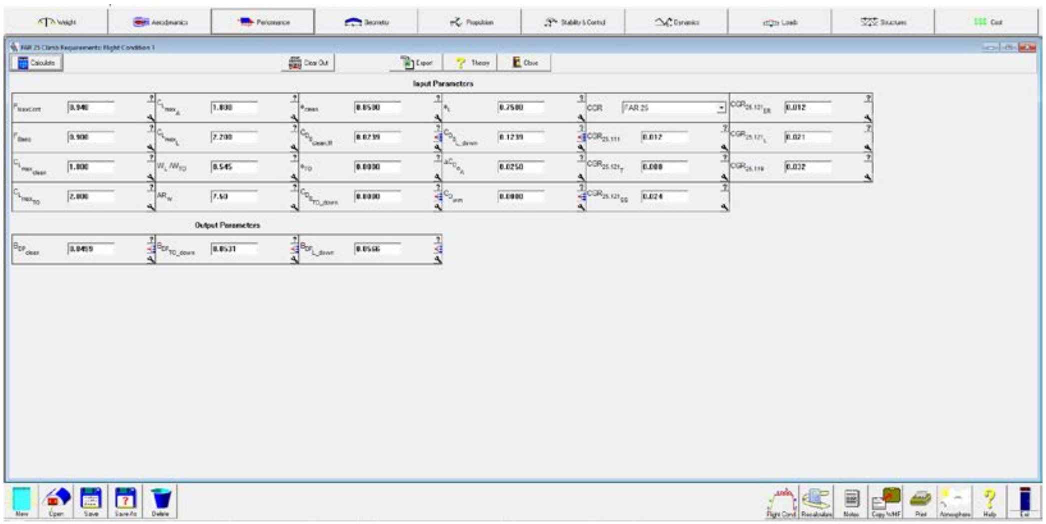


Figure 40: Climb constraint calculations using the AAA program

4.3.6 Maneuvering Constraints

Maneuvering Constraints are not available for a supersonic Jet airplane as the data provided by the book is specifically for the Agricultural, military and training airplanes.

4.3.7 Speed Constraints

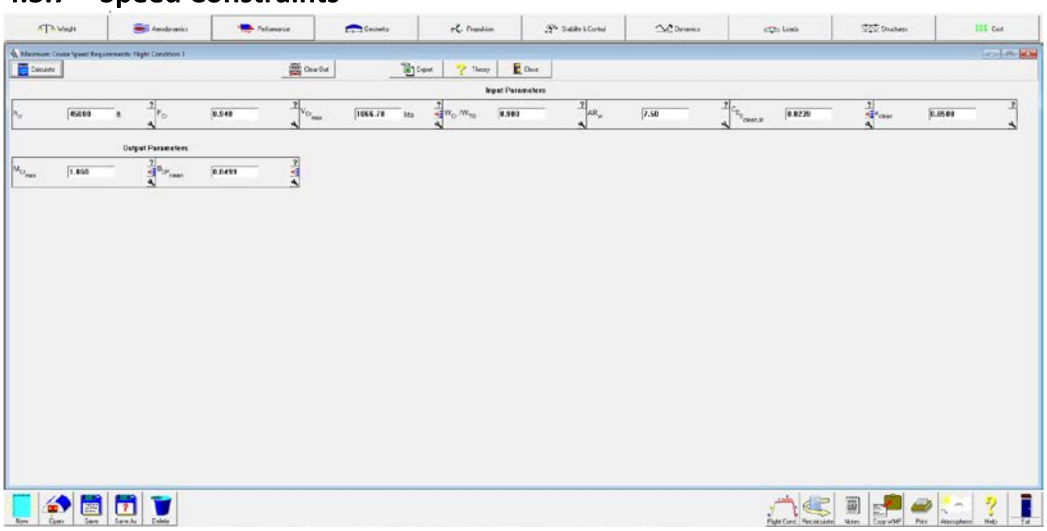


Figure 41: Cruise speed calculations using the AAA program

The cruising speed constraints obtained from the AAA program, provides the Maximum Cruising speed from the given input data such as:

Cruising Altitude (h_{cr}) = 45000 ft

F_{cr} = 0.940

Velocity = 1066.78 knots

W_{cr}/W_{TO} = 0.900

Aspect ratio = 7.5

E_{clean} = 0.8500

Providing this data to the software, give the out of the Maximum Cruising Mach number:
$M_{32_{9-:}} = 1.860\ Mach$

4.3.8 Summary of Performance Constraints

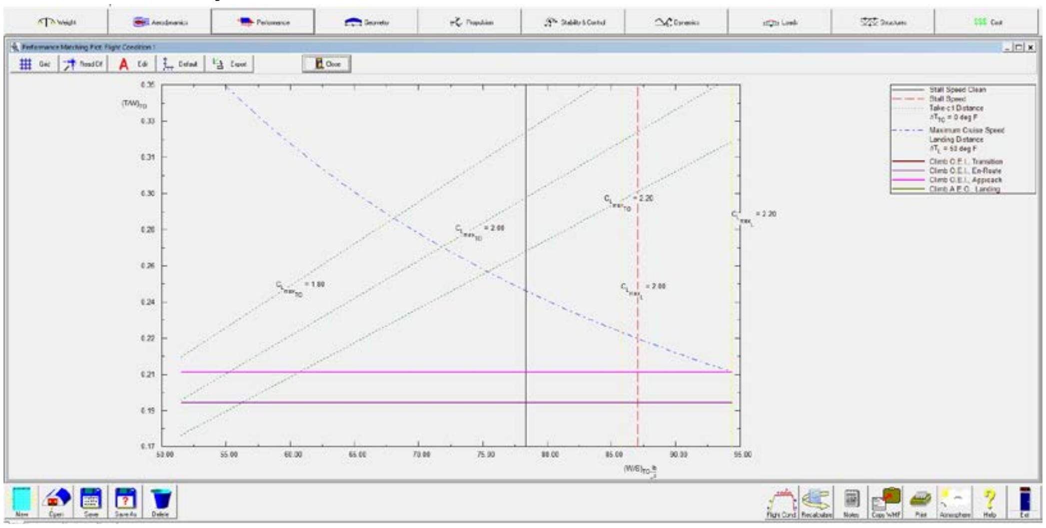


Figure 42: Matching graph from the AAA Program

4.4 SELECTION OF PROPULSION SYSTEM

4.4.1 Selection of Propulsion System Type

Propulsion systems used for similar airplanes are described as follows:

Aerion SBJ: 2 x Pratt & Whitney JT8D-219 turbofans, they produce 19600lbf (87.19 kN) of Thrust which is sufficient for the Aerion SBJ to cruise at supersonic speeds.

Aerion AS2: 3 x GE Aviation low bypass ratio turbofans where each of the turbofan produces 67-76 kN thrust.

HyperMach SonicStar: 2 x hybrid supersonic 4000-X series engines that produce 54,700 lbf of thrust. Though it is still a theoretical model.
Spike S-512: 2 x engines, 20,000 lbf (89 kN) thrust is produced by each of the engines.

Sukhoi Gulfstream S-21: 3 x Aviadvigatel D-21A1 turbofans where each produce thrust equal to 16,535 lbf (73.55 kN).

Tupolev Tu-444: 2 x NPO Saturn AL-32M turbofan where each produces a thrust of 95 kN (21,400 lbf).

For the supersonic business jet being designed, 2 x Pratt and Whitney J58 Engines will be used as each produces 25,000 pound-force (110 kN) of thrust without the after burner and 34,000 (150 kN) of thrust with the after burner. This amount of thrust obtained is sufficient and suitable for a supersonic business jet.

4.4.2 Selection of the Number of Engines

Two Pratt & Whitney J58 Engines will be used as the thrust produced by each of the engine is sufficient for the airplane according to its requirements.

4.4.3 Propeller Sizing

It is an afterburning turbojet with compressor bleed bypass having 9 stage axial flow single spool compressor 8 can, annual combustors, a two-stage axial flow turbine with the sizing specifications as follows:

Length: 17 ft 10 in and an additional 6 in at maximum temperature.
Diameter: 4 ft 9 in
Dry weight: 6000 lb (2,700 kg) approximately

Its performance sizing is as follows:
Maximum thrust: 34,000 pounds-force (150 kN) wet, 25,000 pound force (110 kN) dry
Overall pressure ratio: 7.5 at take-off
Specific Fuel Consumption: 1.9 lb/(lbf.h)
Thrust to weight ratio: approximately 6
Air flow: 300 lb/s at take-off.

4.5 DISCUSSION

This is the fourth report which specifies the performance sizing of the airplane. First the wing loading was calculated using the stall speed requirements. The stall speed of 190 knots was assumed in comparison to similar airplanes. The wing loading (W/S) obtained from the assumed stall speed and the provided coefficient of lifts was then compared to the wing loadings of similar airplanes. The data obtained was then used to calculate the take-off parameters that satisfy the FAR 25 requirements. Once the take-off distance and landing distance are obtained, the relation between the thrust to weight ratio and the wing loading is obtained. The different wing loadings at different coefficients of lift during landing and take-off are then obtained.

Once the final data for the wing loading is obtained, the drag polar distance is then calculated which gives the parasite area, wing area, the wetted area, weight during take-off, weight while landing and the equivalent skin friction coefficient. Using all these terms, along with the aspect ratio assumed to be 7.5, we can calculate the coefficient of drag at different stages of flight with different configurations of the airplane. Using this coefficient of drag, we can calculate the lift to drag ratio required to the airplane at different stages of flight.

The climb constraints are then calculated with the coefficient's of drag equations obtained satisfying the FAR 25 requirements with different configurations like all engines operating, one engine inoperative for this airplane as 2 engines are being used in the supersonic business jet design. These calculations provide the thrust-to-weight ratio at different stages while take-off, landing, using flaps configuration, without flaps configuration, with the landing gear configuration, without the landing gear configuration, and during the clean stage. Using the thrust to weight ratios and the wing loading, we get the rate of climb which is equal to 8666 ft/m^2.

After the climb configurations, comes the manoeuvring constraint which is not applicable to the Supersonic Business Jet. As the data provided only satisfies the agricultural and military aircraft requirements. And then the speed constraints section provides the cruising speed of the airplane at the specified altitude of 45000-55000 ft.

The AAA program was used after the calculations which provides the data similar to the manual calculations. A Matching graph was then obtained in the end which gives the entire data of the airplane performance constraints and shows the area where the airplane will have the best performance according to the data obtained or assumed.

Later on the number of engines to be used was specified and the engine that satisfies all the requirements of the airplane was selected which was the Pratt and Whitney J58 Engines. And its specifications were discussed in the section later.

4.6 CONCLUSIONS AND RECOMMENDATIONS

4.6.1 Conclusions

The goal of this report was to calculate the performance sizing of the SSBJ using manual calculations and the AAA program. Once the data from the hand calculations is obtained, it will be compared with the data obtained from the AAA program to check how valid the data obtained is in both the cases in comparison to similar airplanes. The results obtained in this report can be summarised as follows:

- The wing-loading and the thrust-to-weight ratios are inversely proportional to each other. Thus it is necessary to maintain the ratios or else it can cause complications in the journey of the airplane.
- The requirements for FAR 23 and FAR 25 are totally different hence it is necessary to design the parameters according to the requirements to obtain near accurate data.
- Even minor changes in the aircraft configurations can make a huge difference in the airplane such as from the clean state to the flaps state, the Lift to drag ratio (L/D) can be changed by a huge difference.

- Flying below the Stall Speed can cause the airplane to crash hence it is necessary to fly above the specified stall speed.
- The airplane has different performances according to the altitude hence it is necessary to fly in the satisfactory altitude to obtain the best possible results out of the specified design.
- The wing loading varies at different stage of flight hence the thrust to weight ratios vary according to the wing loading within a certain described range.

6.6.2 Recommendations

From this report, a lot of knowledge was obtained on the performance of the airplane under various conditions and what changes occur with even minor changes to the design. The future work consists of designing the remaining parts of the airplane design.

CHAPTER 5: FUSELAGE DESIGN

5.1 INTRODUCTION

This is the 5th report of the preliminary design of the Supersonic Business Jet that specifies the cockpit and fuselage design. The weight sizing, wing loading, performance parameters were calculated in the previous reports. This report typically specifies the design of the fuselage and cockpit with its dimensions. The design of the fuselage typically depends on the following parameters:

I. The maximum take-off weight of the airplane.
II. The number of passengers.
III. Location of engines
IV. Fuel storage
V. Location of landing gears
VI. Wing placement

While designing the cockpit and the fuselage of the airplane, the following items should be included.

I. Number and weight of cockpit crew members
II. Number and weight of cabin crew members
III. Number and weight of special duty crew members
IV. Number and weight of passengers
V. Weight and volume of 'carry-on' baggage
VI. Weight and volume of 'check-in' baggage
VII. Weight and volume of cargo
VIII. Number, weight and size of cargo containers
IX. Weight and volume of special operational equipment
X. Weight and volume of military payload
XI. Weight and volume of fuel carried in fuselage
XII. Radar equipment
XIII. Auxiliary power unit

In this report the second section describes the preliminary design of the cockpit which includes the cockpit crew requirement, pilot visibility requirements, reachability of the pilot to the essential cockpit controls.

The third section of the report describes the preliminary design of the fuselage section considering the number of passengers, the thickness of the fuselage,

5.2 LAYOUT DESIGN OF THE COCKPIT

The term cockpit is usually associated with small or medium sized airplanes.

The following parameters help to design the layout of the cockpit of an airplane:

I. The pilot and the cockpit crew members should be positioned in such a way that they can reach all the controls comfortably without a lot of effort from the designed position.
II. All the essential instruments to the flight should be visible without undue effort.
III. Communication by voice or touch should be possible without having the pilot put in extra effort.
IV. The visibility from the cockpit must satisfy certain minimum required standards.

The weights and dimensions of the crew members while designing the cockpit is important as It should be ensured that the arm and leg motions needed to carry out control manipulation of throttles, stick or wheel, side arm controller and rudder pedals are indeed feasible.

As Height of the crew members is an important factor while deigning the cockpit of an airplane, it's dimensions can be determined according to the figure described below. The total height of the male crew member is described by A as shown below in figure 2.

For female crew members, the values are to be multiplied by a factor of 0.85 to obtain the weights and dimensions of female crew members.

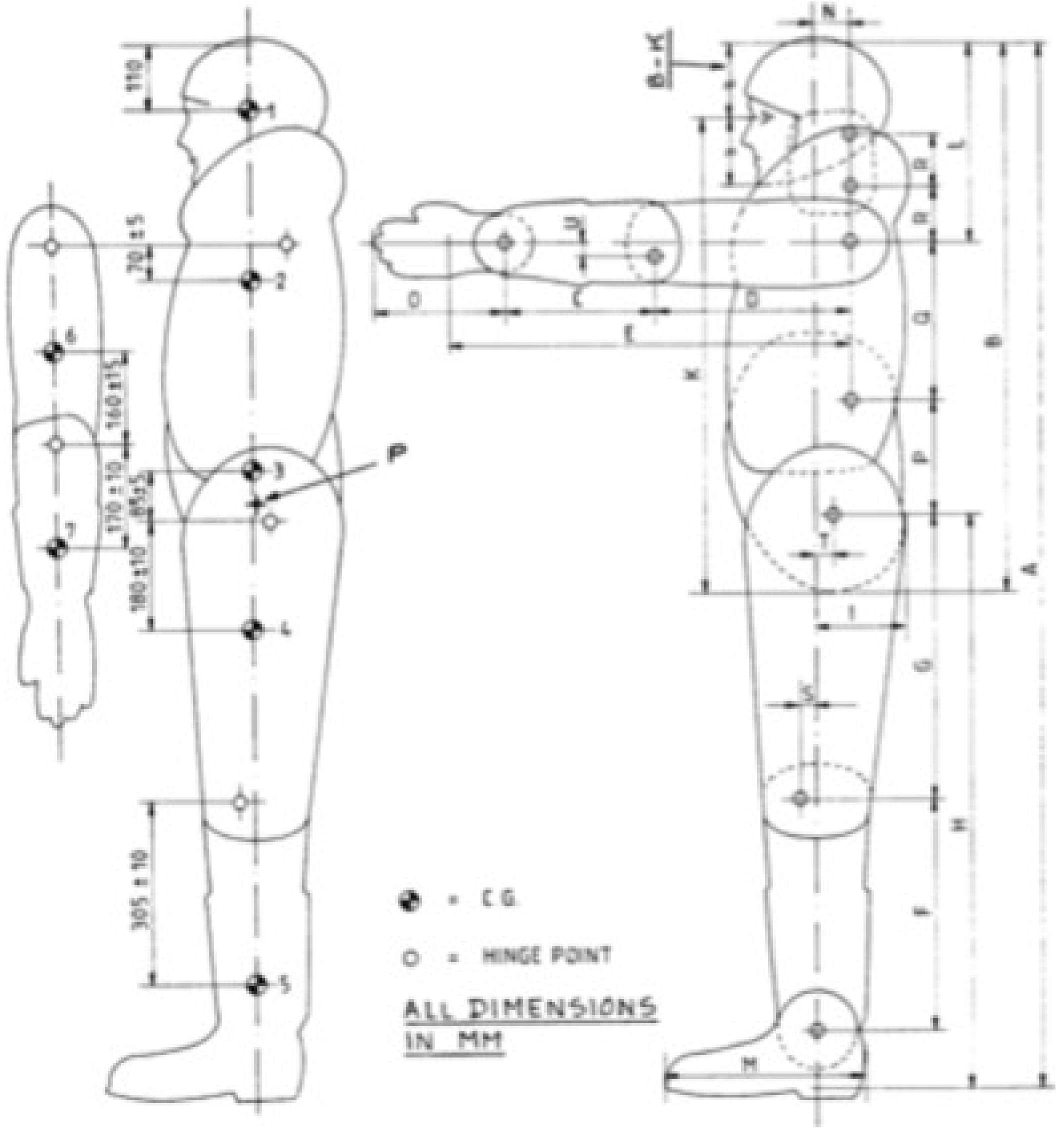


Figure 43: Dimensions of a standing male crew member

The dimensions of male crew members according to the figure 1 can be determined from the figure 2 below.

A	B	C	D	E	F	G	H	I	K	L
1,600	870	230	300	620	350	435	850	140	760	300
1,750	920	255	335	685	390	475	950	150	805	330
1,900	990	280	370	750	430	515	1,050	160	875	360

A	M	N	O	P	Q	R	S	T	U
1,600	300	50	200	190	260	80	25	20	20
1,750	325	60	220	200	270	90	30	30	20
1,900	350	70	240	210	280	100	30	30	20

Figure 44: Dimensions and weight of a male crew member

The cockpit layout must have variation in the dimensional limitations of the human body as the humans come in widely different sizes, due to which the cockpit should allow these variations. This can be obtained by arranging the seat position adjustments and also the rudder paddle adjustments.

The typical arrangement of pilot seat and pilot controls for civil airplanes is shown in the figure below.

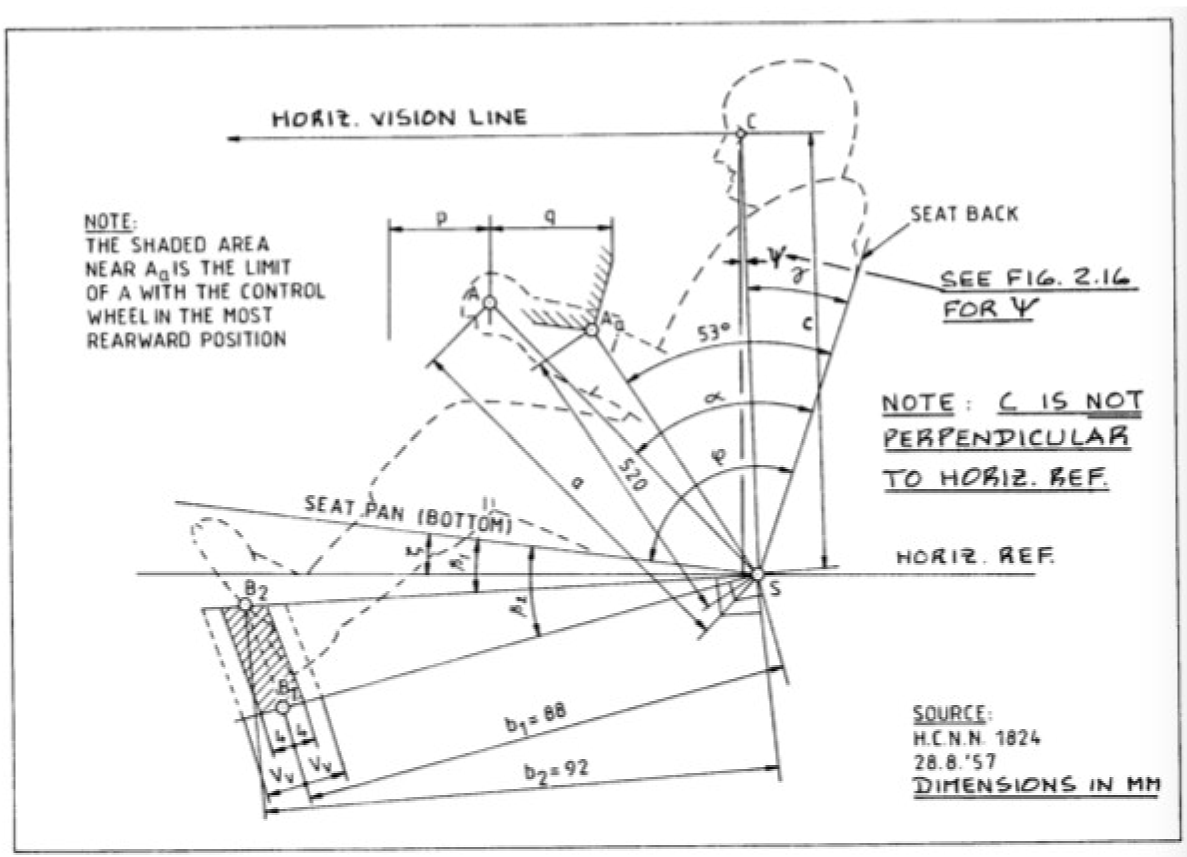


Figure 45: Pilot seat and controls arrangement

The typical weights and dimensions for male crew members for wheel type controllers can be obtained from the figure shown below:

For Wheel Type Controllers:

A	B	C	D deg.	E deg.	F	G	H	I	J	K
37	30.25	5	21	101	29.75	10.00	16.63	19	6	9
39	30.75	5	19	101	30.25	9.75	15.75	19	6	9
41	31.50	5	16	101	31.00	9.75	15.13	19	6	9
43	31.75	5	16	101	31.25	10.00	15.13	19	6	9

A	L	M	N	O	P	Q	R
37	10.00	36.0	5	9.25	15	7	25
39	10.50	35.0	5	9.25	15	7	25
41	10.75	34.5	5	9.25	15	7	25
43	11.00	34.5	5	9.25	15	7	25

Figure 46: Dimensions for wheel type controllers

The areas for good and poor accessibility areas for the pilot seats is described in the figure below according to the divided sections that show how conveniently accessible the section is to the pilot.

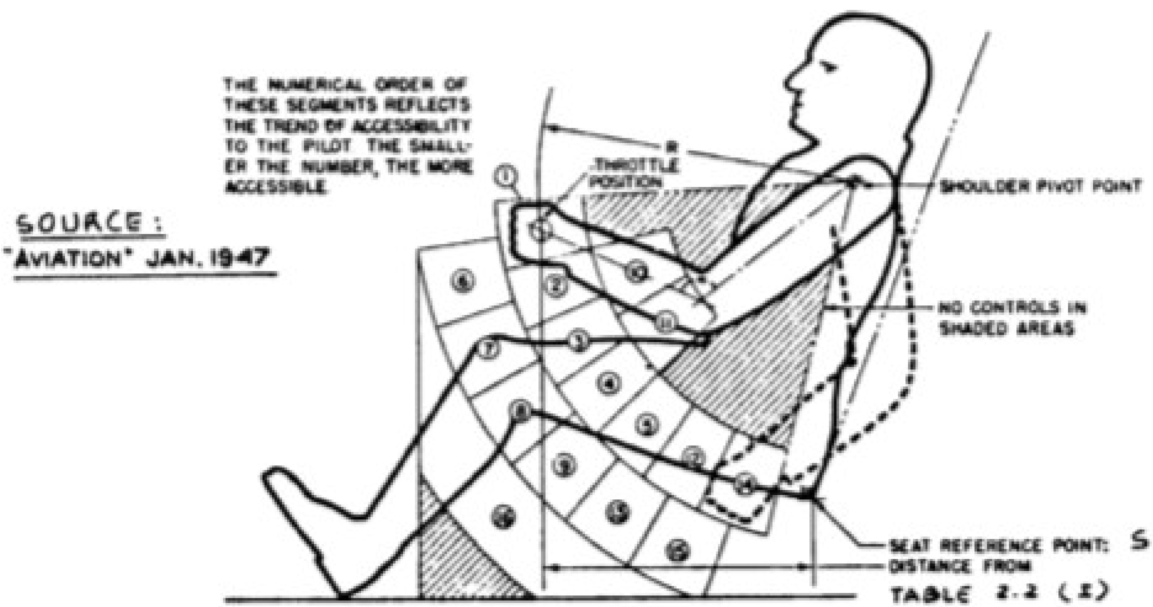


Figure 47: Areas of good and Poor accessibility

the relation between the pilot seat and the pilot controls cannot be directly employed as the human body varies greatly in geometrical dimensions. Some of the variations measured according to figure 1 are:

Variation in arm length (C+D+O): +/- 15 cm
Variation in leg length (H): +/- 20 cm
Variation in seat-eye distance (C): +/- 12 cm

There is no systematic relationship between each of these points hence it implies that a considerable amount of adjustment must be designed into cockpits. The figure above applies to wheel controlled and to center-stick controlled airplanes.

The following should importantly be kept in mind while designing the cockpit of an airplane:

I. Fight essential crew members and their primary cockpit controls should not be located within the 5 degree arcs.
II. According to the 23.771 and FAR 25.771 this are requirement must be met for propeller driven airplanes only.

The dimensions for civil cockpit controls and for the seat adjustments are shown in the figure below:

Symbol	Wheel Control	Stick Control
a	67 (+/- 4)	63 (+/- 4)
ξ	7° (+/- 2°)	7° (+/- 2°)
p = Forward motion of point A:	18 (+/- 2)	16 (+/- 2)
q = Rearward motion of point A:	22 (+/- 2)	20 (+/- 2)
r = Sidewise motion of point A from center•:	-----	15 (+/- 2)
d = Distance between handgrips of wheel•:	38 (+/- 5)	-----
ε = Wheel rotation from center•:	85° (max.)	-----
v = Distance between rudder pedal center lines•:	38 (+/- 12)	45 (+/- 5)
α	64° (+/- 3°)	70° (+/- 3°)
β_1	22°	same
β_2	10°	same
c	77 (+/- 2)	same
γ	21° (+/- 1°)	same
φ	102° (+/- 2°)	same
V_v = Adjustment range of pedals from center position B:	7 (+/- 2)	same
U_v = Forward and aft pedal motion from center position B•:	10 (+/- 2)	same
S_h = Horizontal adjustment range of S from center position•:	< 10	same
S_v = Vertical adjustment range of S from center position•:	8 (+/- 1)	same

Figure 48: Dimensions for civil cockpit controls and for seat arrangements

Determination of Visibility from the cockpit:

A better visibility from the cockpit is essential for a number of reasons:

1. During take-off and Landing operations, a pilot must have a good view of the immediate surroundings.
2. During en-route operations, the pilot must be able to observe conflicting traffic.

3. In fighters, the success in combat may depend on good visibility. Formation flying is impossible without it.

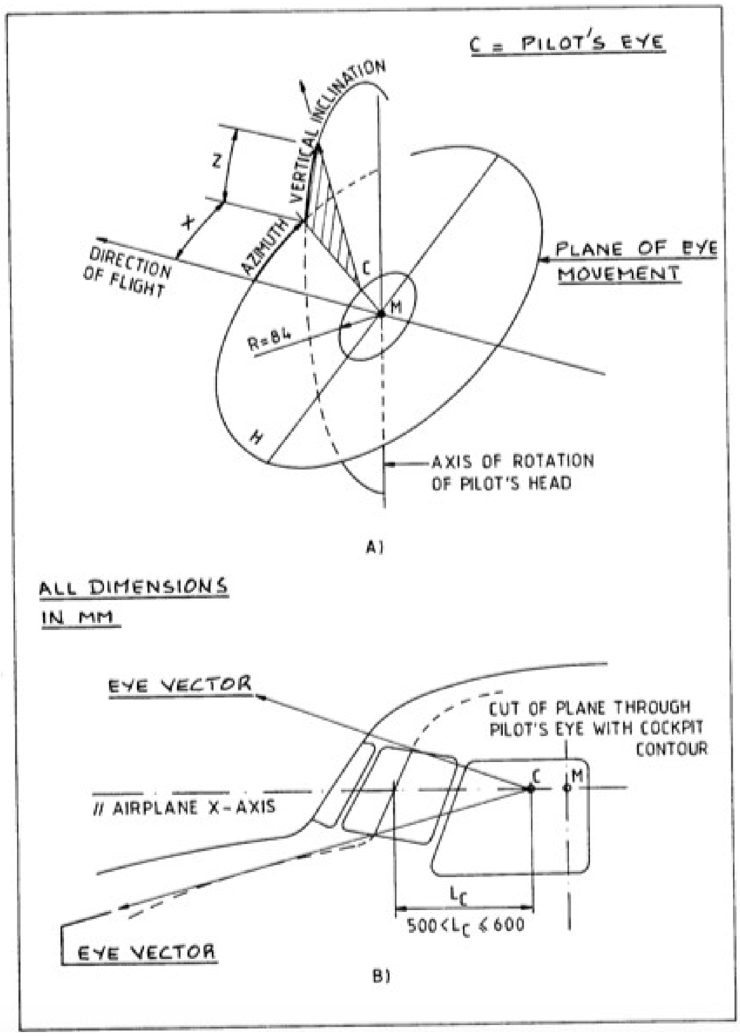


Figure 49: Definition of radial eye vectors

The minimum cockpit visibility rules were brought up for the civil and military airplanes. Different kinds of airplanes have different requirements of visibility which are designed according to the customer requirements. The required cockpit visibility is defined as the angular area that is obtained after intersecting the cockpit with the redial vectors emanating from the eyes of the pilot which are assumed to be centered on the pilot's head. Pilots use both the eyes to see but a point C is used to construct the visibility pattern assuming it as a center of vision. It becomes important to locate the point C as using this point, the seat of the pilot can be located. The seat itself is relative to the floor and to the cockpit controls using the dimensions discussed in figure 2.

The entire process can be described in detail by breaking it down to the following steps:

I. Locate point C on the horizontal vision axis

II. The distance labeled should be within an indicated range.
III. Draw the angle Ψ = 8.75 degrees
IV. Locate point S with the help of the distance 'c' which has a maximum allowable value of 80 cm.
V. Design the pilot seat according to the dimensions
VI. Draw the cockpit controls and seat motions and adjustments within the defined cockpit area.
VII. Check the minimum visibility requirements according to the visibility rules.

Airplanes having side-by-side pilot seating arrangements, no window frames will be located in the area from 30 degrees starboard to 20-degree port.

In the area from 20-degree port to 60-degree port, window frames should not be wider than 2.5 inches.

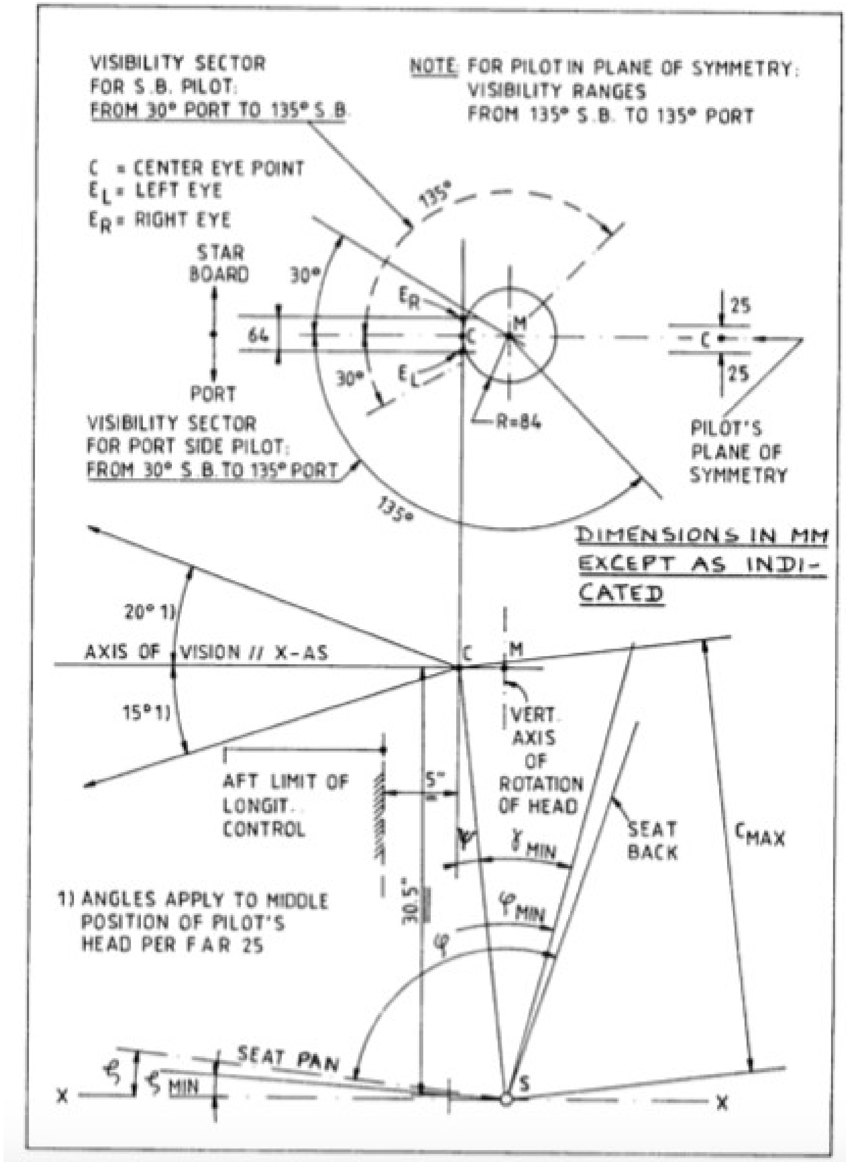


Figure 50: Visibility requirements for the port and starboard side.

Hence, larger windows require very stiff frames. Both the frames and the windows must meet the bird strike requirements which cause in the increase in weight of the airplane. Another problem is the increase in drag which is caused due to flat windows while curved windows can reduce the drag but cause image distortions.

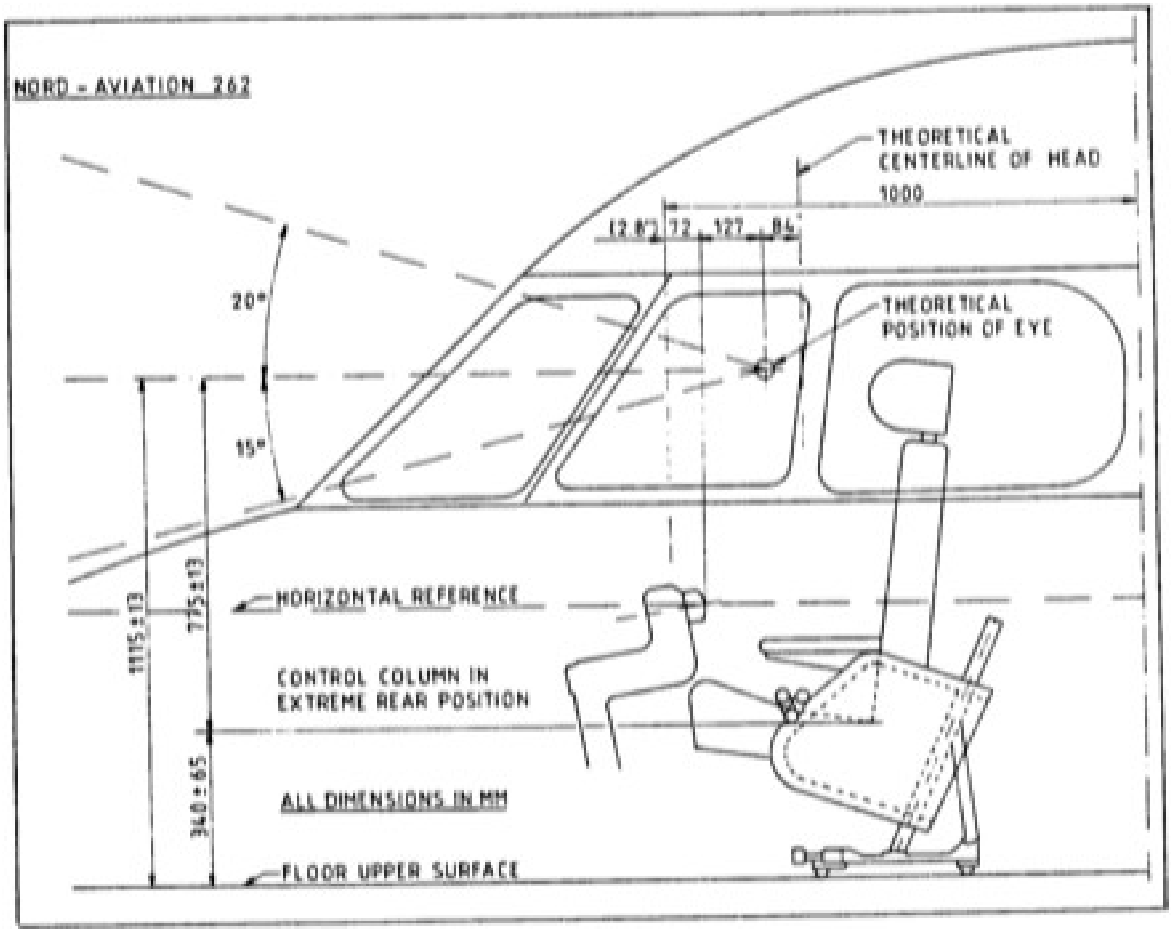


Figure 51: Cockpit layout

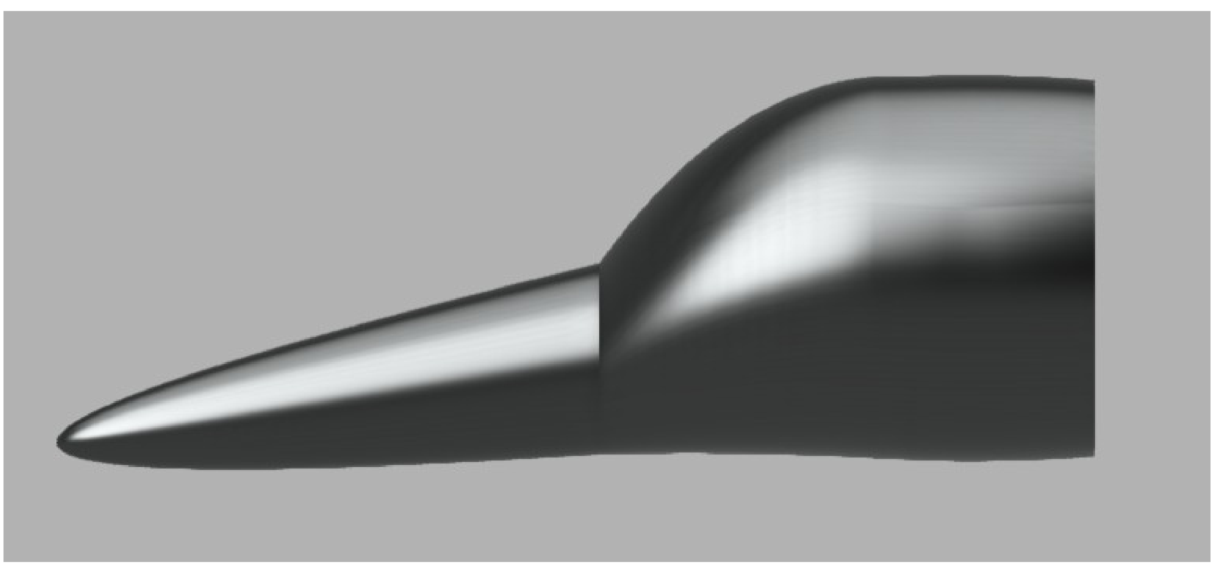


Figure 52: Right view of the cockpit

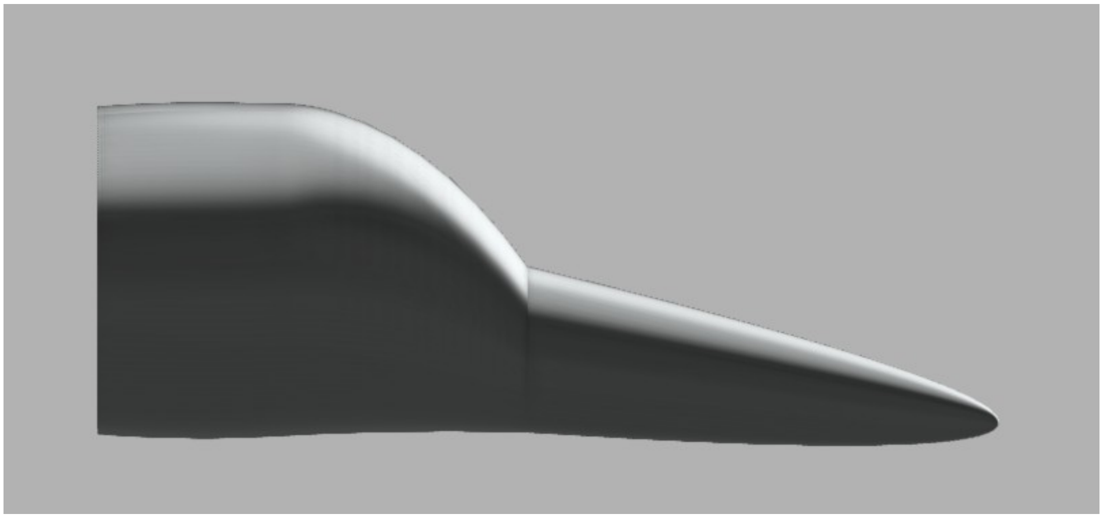


Figure 53: Left View of the Cockpit

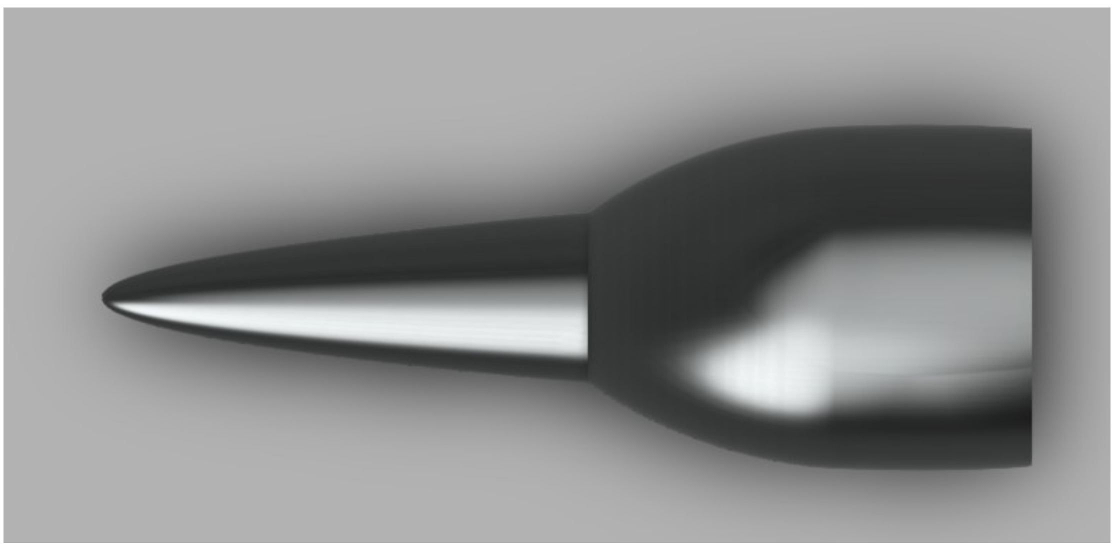


Figure 54: Top View of the cockpit

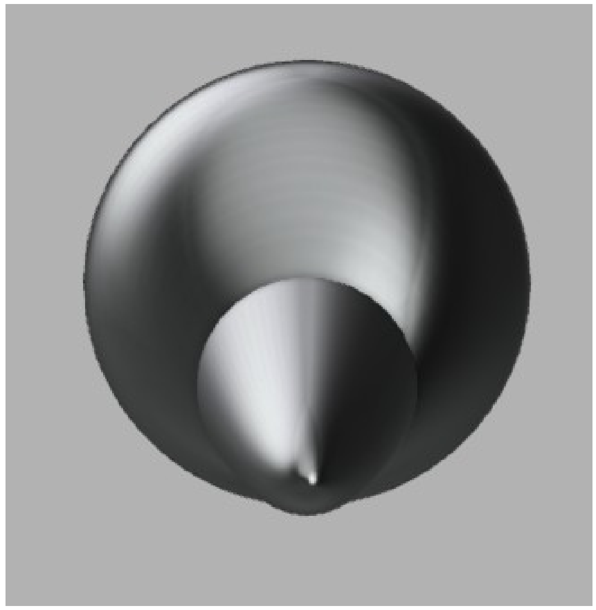


Figure 55: Front view of the cockpit

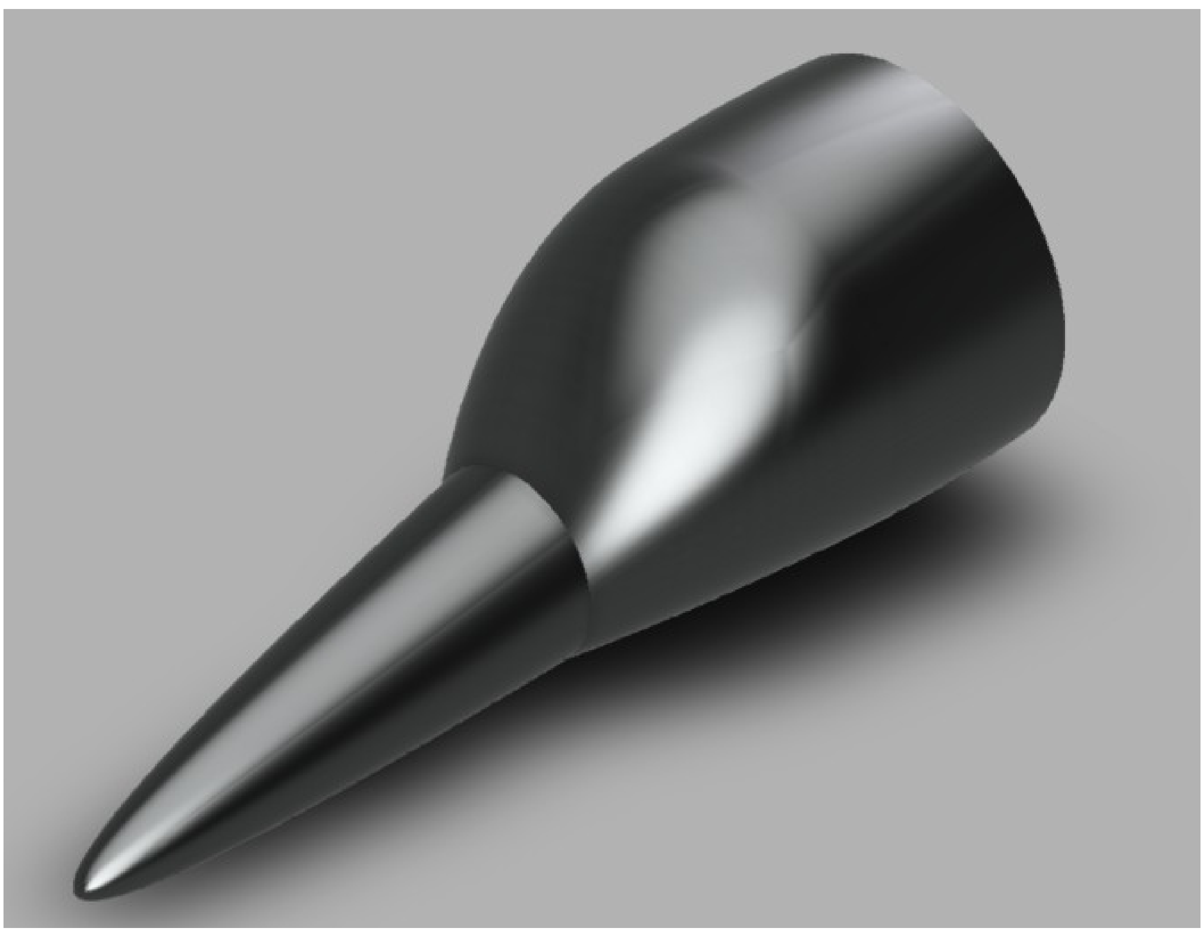


Figure 56: 3D cross-sectional view of the cockpit

5.3 LAYOUT DESIGN OF THE FUSELAGE

While designing the fuselage of the business jet, it is important to carefully consider the following choices:

I. Number of persons abreast
II. Number and size of aisles
III. Type of seating arrangements: first class, business class, economy class.
IV. Cabin provisions required in terms of: closets, toilets, overhead storage compartments, galleys.
V. Seating provisions for the cabin crew.

For small commercial airplanes, such as the business jet, the sufficient structural depth required is 1.5 inches. Aerodynamic fairing of the cockpit exterior to the fuselage exterior causes as little extra drag as possible.

The cone of the fuselage is normally a smooth transition from the maximum fuselage cross section to the end of the fuselage. With low fineness ratio, there will be a very large base drag penalty although the fuselage weight might be reduced while if the fineness ratio of this cone is large, there will be a large penalty in the friction drag of the airplane as well as a large weight penalty. Long fuselage cones tend to increase the tail moment arm there by reducing the tail area and vice versa. Figure 8 describes the typical layout of the exterior of the fuselage.

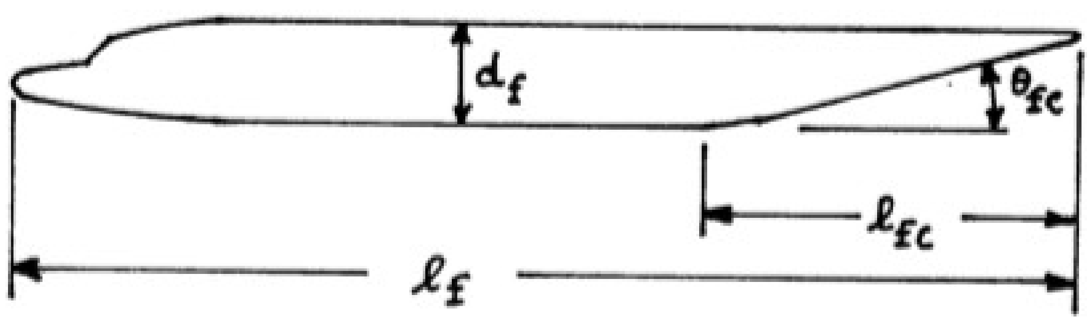


Figure 57: Dimensions of an aircraft

To design the fuselage, the following parameters can be used as mentioned in the figure below:

Airplane Type	l_f/d_f	l_{fc}/d_f	θ_{fc} (deg)
Homebuilts	4 - 8	3*	2 - 9
Single Engine	5 - 8	3 - 4	3 - 9
Twins	3.6** - 8	2.6 - 4	6 - 13
Agricultural	5 - 8	3 - 4	1 - 7
Business Jets	7 - 9.5	2.5 - 5	6 - 11
Regionals	5.6 - 10	2 - 4	15 - 19***
Jet Transports	6.8 - 11.5	2.6 - 4	11 - 16
Mil. Trainers	5.4 - 7.5	3*	up to 14
Fighters	7 - 11	3 - 5*	0 - 8
Mil. Transports, Bombers and Patrol Airplanes	6 - 13	2.5 - 6	7 - 25****
Flying Boats	6 - 11	3 - 6	8 - 14
Supersonics	12 - 25	6 - 8	2 - 9

Figure 58: Fuselage Parameters

The length of the fuselage can be obtained from the following equation and table:

Length $= aW_0^C$	a	C
Sailplane—unpowered	0.86	0.48
Sailplane—powered	0.71	0.48
Homebuilt—metal/wood	3.68	0.23
Homebuilt—composite	3.50	0.23
General aviation—single engine	4.37	0.23
General aviation—twin engine	0.86	0.42
Agricultural aircraft	4.04	0.23
Twin turboprop	0.37	0.51
Flying boat	1.05	0.40
Jet trainer	0.79	0.41
Jet fighter	0.93	0.39
Military cargo/bomber	0.23	0.50
Jet transport	0.67	0.43

Figure 59: Fuselage Length with respect to the maximum take-off weight

For the supersonic business jet carrying a maximum take-off weight of 1,10,000 lbs can be obtained as follows,

$$Length = a * W_{>}^{N} = 0.67 * (1{,}10{,}000)^{>.\text{èp}} = \ 98.6\ ft \qquad 1$$

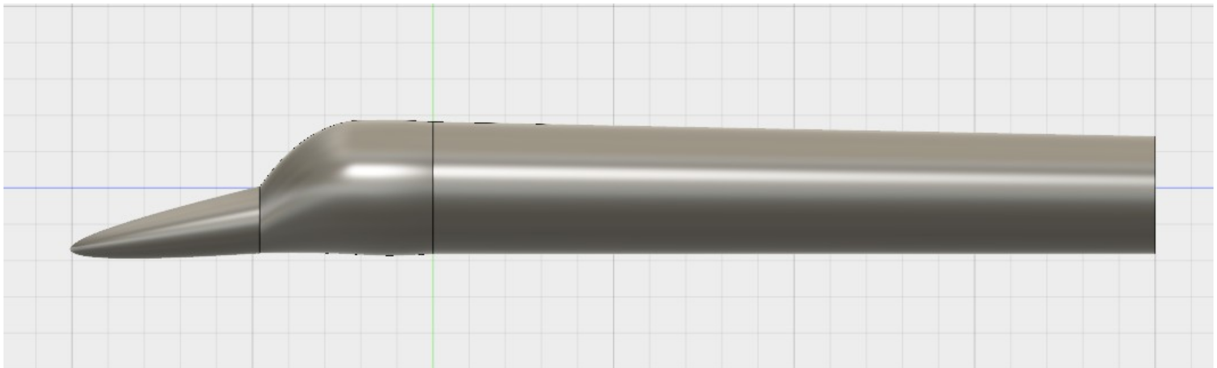


Figure 60: Right view of the fuselage and cockpit

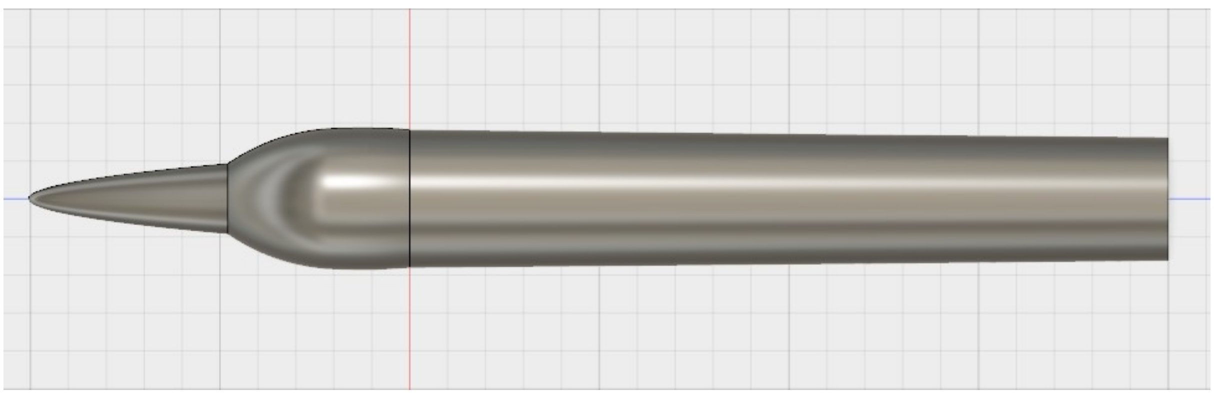


Figure 61: Top view of the fuselage and cockpit

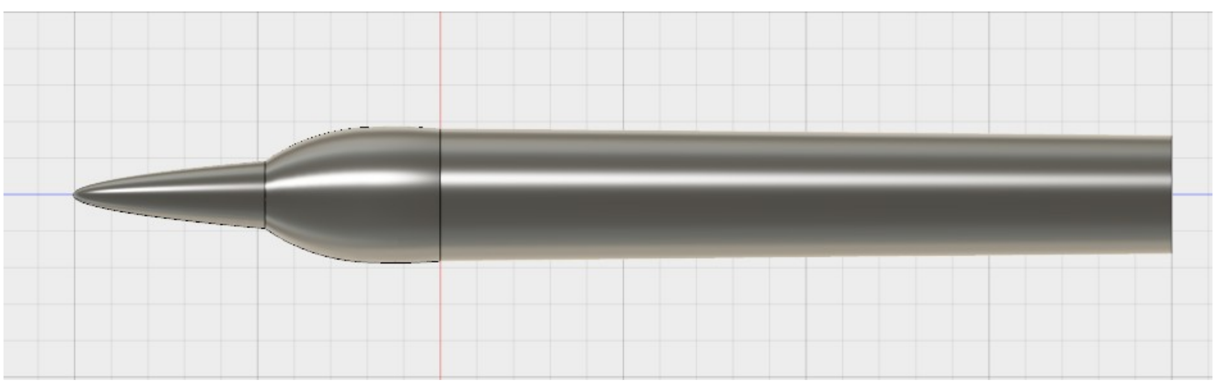


Figure 62: Bottom view of the fuselage and cockpit

Figure 63: Front view of the aircraft

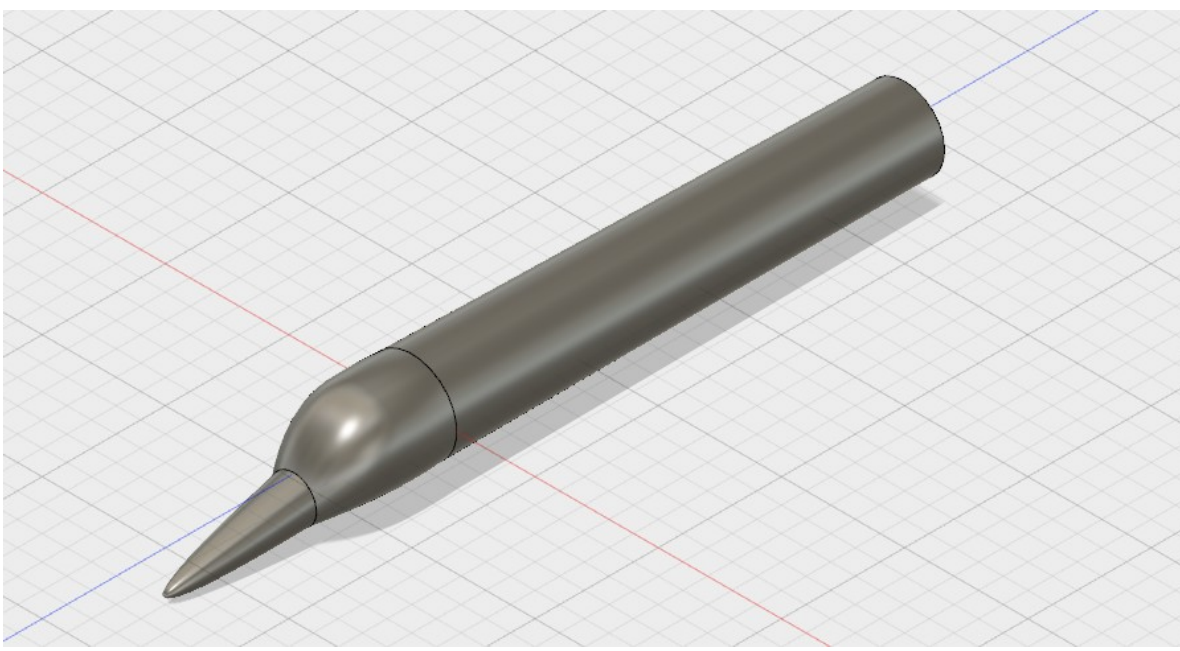


Figure 64: 3D view of the fuselage and cockpit of the SSBJ

AERODYNAMIC DRAG CONSIDERATIONS

The fuselage of an airplane is responsible for a large percentage of the overall drag. Majority airplanes produce a drag of 25-50 percent. Since it is required for an airplane to produce drag as little as possible, the fuselage should be shaped and sized accordingly. The following types of drag are produced by the fuselage of an airplane.

i. Friction drag
ii. Profile drag
iii. Base drag
iv. Compressibility drag
v. Induced drag

The friction drag is directly proportional to the wetted area as the wetted area is directly related to the fuselage length and to the perimeters of fuselage cross sections. This can be reduced by shaping the fuselage in such a manner that laminar flow is obtained or by reducing the length and perimeter of the fuselage as much as possible.

The fuselage fineness ratio plays an important role in determining the fuselage friction drag. The fineness ratio of an airplane gradually increases as the cruise speed increases.

The profile and base drag depend on the front and aft body shape. Blunt fore-bodies and blunt aft bodies promote flow separations which lead to high profile and base drag. Fore-body bluntness can be caused by poor cockpit window and requirement for front end loading.

The ideal 'streamline' nose shape can be obtained only If the windshields are integrated smoothly into the fuselage.

With large fuselage fineness ratio, the drag increases due to the unsweep in the aft-body of the fuselage. This unsweep can lead to vortex induced separations. These vortices not only increase the drag but also tend to increase the problems caused by lateral oscillations. These problems can be stabilized using the use of sharp corners. The sharp corners solve the problems caused by lateral oscillations and also reduce the drag issues. The unsweep is applied to the airplane

i. To facilitate take-off rotations
ii. To facilitate rear cargo loading.

A bulge is sometimes necessary to be added to the upper rear section of the fuselage if large unsweep angles are detected by rear loading considerations. This bulge is needed to obtain sufficient structural depth in the fuselage to resist tail loads.

Compressibility drags are obtained at very high subsonic Mach numbers. It arises from the existence of shocks on the fuselage. These shocks are strongly due to the sweep and thickness of the wing in the area of wing/fuselage juncture. The area rule concept must be used to minimize compressibility drag.

The fuselage contributes the most to induced drag due to the adverse effect on the wings span load distribution.

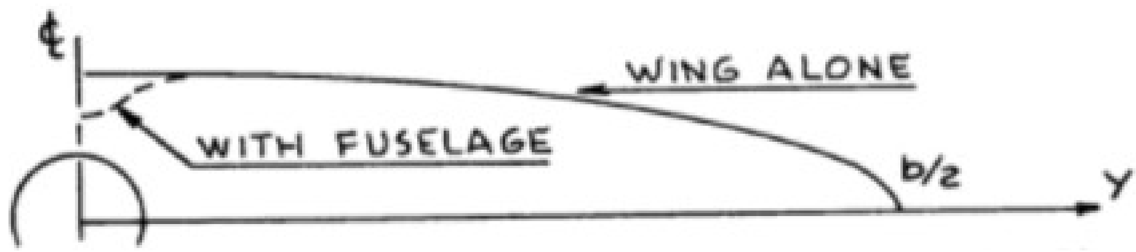


Figure 65: Effect of fuselage on wing span loading

INTERIOR LAYOUT DESIGN OF THE FUSELAGE

The fuselage carries the crew, the passengers, the payload and many of the systems needed for the operation of an airplane. The interior of the airplane design reflects a compromise between level of creature comforts and the weights and sizes required to create the creature comforts.

The ability to load and unload cargo plays an important role. The problems associated with servicing and maintenance dictate where access must be designed into the fuselage. The design for good access, maintenance and inspectability usually conflicts directly with design for low structural weight, low complexity and low drag.

The interior of the fuselage layout design contains the following:

i. Layout of the cross section
ii. Seating layouts, seats and restraint systems
iii. Layout of doors and emergency exits
iv. Galley, lavatory and wardrobe layouts
v. Layout of cargo, baggage holds, including data on cargo containers
vi. Maintenance and servicing considerations

The fuselage cross sections are the result of compromises between weight, drag, systems and creature comfort considerations.

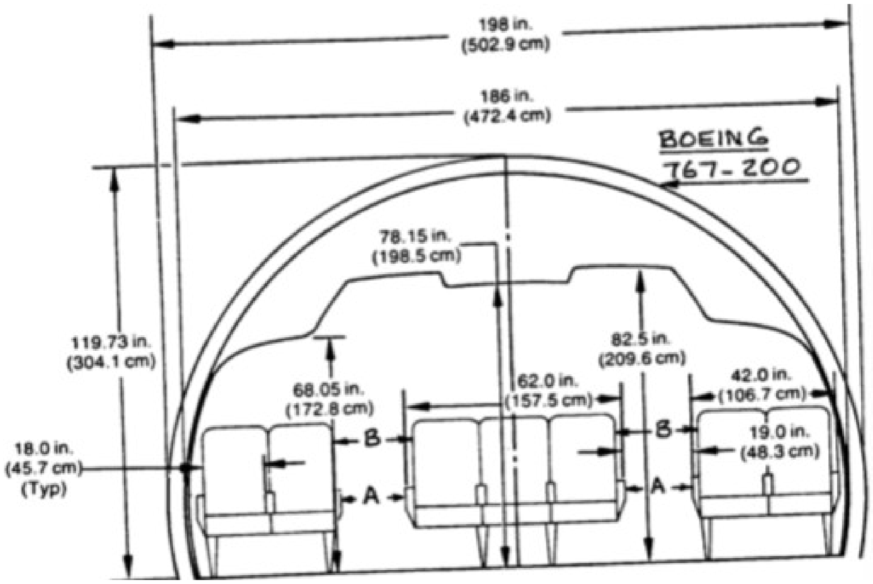


Figure 66: Maximum aisle width requirement

The seating arrangement in an airplane has a direct impact on the cabin length and cabin width, fuselage weight and drag, future potential growth of an airplane, passenger appeal and therefore market acceptance.

Symbol	Unit	De Luxe	Normal	Economy
a	in.	20(18.5-21)	17(16.5-17.5)	16.5(16-17)
b	in.	47(46-48.5)	40(39-41)	39(38-40)
		for two seats per block		
b	in.	---	60(59-63)	57
		for three seats per block		
l	in.	2.75	2.25	2.0
h	in.	42(41-44)	42(41-44)	39(36-41)
k	in.	17	17.75	17.75
m	in.	7.75	8.5	8.5
n	in.	32(24-34)	32(24-34)	32(24-34)
p/p_{max}	in./in.	28/40	27/37.5	26/35.5
α/α_{max}	deg/deg	15/45	15/38	15/38

Figure 67: Seat Classification and seat dimensions

Frame Depths:

For small commercial airplanes: 1.5 inches.
For fighters and trainers: 2.0 inches.
For large transports: $0.02d_f + 1.0$ inches.

Frame Spacings:

For small commercial airplanes: 24 - 30 inches.
For fighters and trainers: 15 - 20 inches.
For large transports: 18 - 22 inches.

Longeron Spacings:

For small commercial airplanes: 10 - 15 inches.
For fighters and trainers: 8 - 12 inches.
For large transports: 6 - 12 inches.

Figure 68: Frame depths, frame spacing's and longeron spacing's

Table 13: Dimensions of the fuselage components

Component	Dimensions
Seat height	45 in
Seat pitch	40 in
Aisle height	73.29 in
Aisle width	20 in
Seat width	25 in
Galley	25 in x 40 in
lavatory	34 in x 40 in

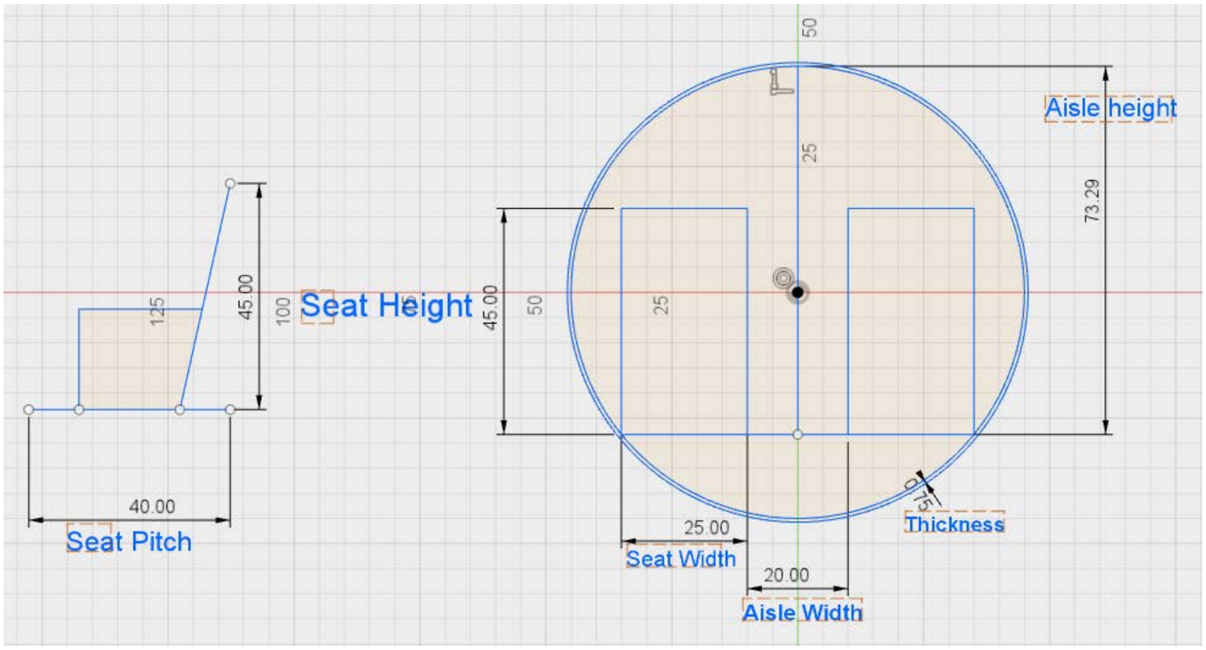


Figure 69: Cross-sectional view of the Fuselage (all units in inches)

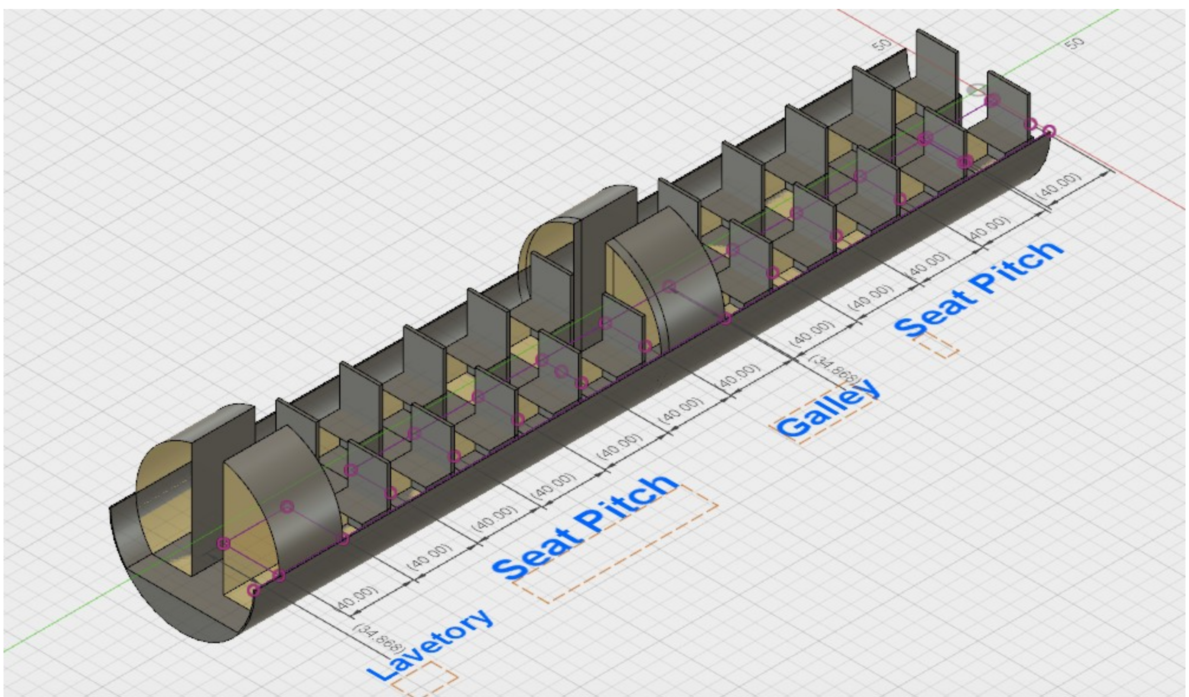


Figure 70: 3D view of the interior layput of the fuselage (all units in inches)

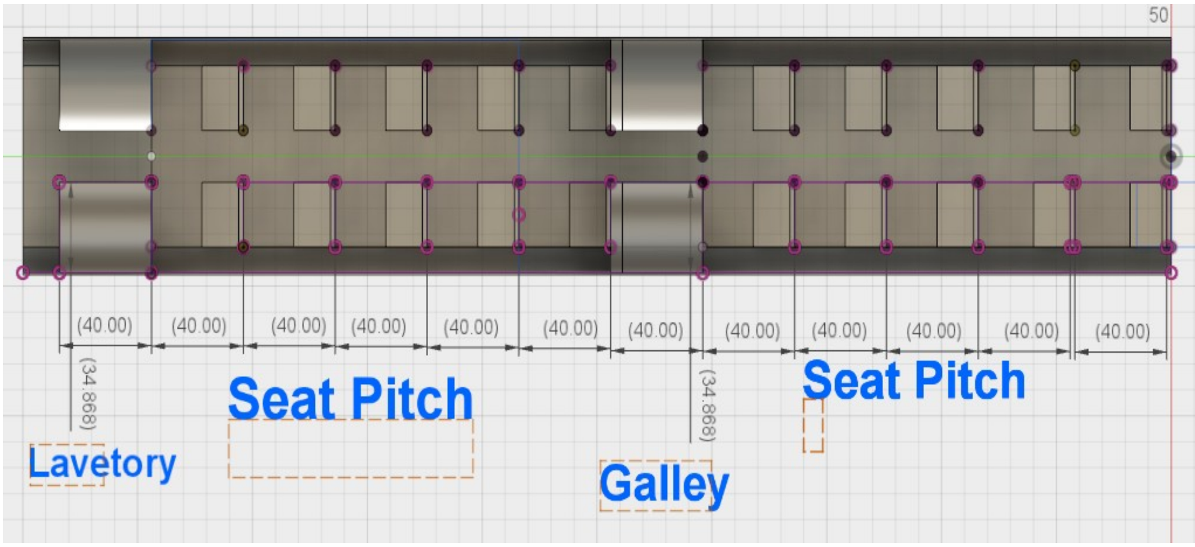


Figure 71: Top view of the interior layout of the fuselage (all units in inches)

5.4 DISCUSSION

This is the 5th report of the supersonic business jet that explains the cockpit and fuselage design. The formula to calculate the length of the fuselage was obtained from the book airplane design: A Conceptual design by Daniel P Raymer. The first section is the introduction part that explains the parameters required to obtain the design of the cockpit and the fuselage.

The second section is the cockpit design section that explains the design and requirement of the cockpit. It explains the dimensions of a male and female crew members seat arrangement requirements, the visibility requirements according to FAR 25 requirements. Wheel/stick control dimensions and requirements and the design of the cockpit. The 3D geometry of the cockpit was designed on Fusion 360. Different views of the cockpit were attached in the above section, the font view, top view, left view, right view and a 3D view of the cockpit.

The last section explains the design of the fuselage. The fuselage was designed according to the requirements within the obtained dimensions of the length of the airplane as obtained in the section above. Different views of the fuselage were attached to the fuselage section, the cross-sectional view explaining the dimensions of the inner part of the fuselage, the right view, the left view, the top view, the bottom view, the back view and the front view. The overall design till date was attached in the fuselage section that contains all the progress in the design section till now.

CHAPTER 6: WING, HIGH-LIFT SYSTEM AND LATERAL CONTROL DESIGN

6.1 INTRODUCTION

This is the sixth report of the series of 12 for the preliminary design of the SuperSonic Business Jet that specifies the wing design of the airplane. The weight sizing, performance parameters, fuselage and cockpit were calculated and designed in the previous reports. This report describes the dimensions and design of the airfoil to be used and wing of the airplane along with the high lift system devices and the lateral control design.

The characteristic required to design the wing of an airplane are:

I. Size (area)
II. Aspect ratio
III. Sweep angle
IV. Thickness ratio
V. Airfoils
VI. Taper ratio
VII. Incidence angle and twist angle
VIII. Dihedral angle
IX. Lateral control surface size and layout.

Certain parameters were discussed in the sections earlier such as the wing area and the aspect ratio of the wing. The design of the wing for the supersonic business jet contains a low-wing configuration.

First the taper ratio, dihedral angle, sweep angle, the thickness ratio will be calculated considering the minimum requirements according to the design of the airplane. All the results will be plotted in a graph and will be justified. In the third section the airfoil will be selected and discussed there after the $C_{+_{9-:}}$ of the wing will be verified using the AAA program

In the fifth section of the report, the high lift devices will be determined and designed according to the obtained dimensions. The sixth section of the report specifies the lateral control surfaces and their sizes keeping in mind that they are compatible with the high-lift devices designed.

The last section of the report explains the calculations and the designs of the parameters important in the design of a wing such as the following:

I. Span, b
II. Root chord, c_r
III. Tip chord, c_t
IV. Mac (Mean Aerodynamic Chord)

V. Mgc (Mean Geometric Chord)
VI. Leading-edge sweep angle
VII. Trailing-edge sweep angle
VIII. Coordinates of the aerodynamic center (x_{ac}, y_{ac})

All these parameters will then be designed along with all the other parameters that apply on the design of the wing

6.2 WING PLANFORM DESIGN

The wing planform of the supersonic business jet is a cranked arrow wing.

The wing size directly affects the following characteristics of the airplane:

- Take-off/ Landing field length
- Cruise performance (L/D)
- Ride through turbulence
- Weight

While designing the planform of the wing, the important parameters to be considered are the following:

- Gross area S
- Aspect ratio A
- Taper ratio
- Dihedral angle

Amongst these parameters, the Gross area and the aspect ratio were already determined in the report 4.

Gross Area: The gross area is the total area of the wing of the airplane being designed which was obtained from the wing loading in the earlier report no 4. The gross area obtained for the Supersonic Business Jet is 1424.34 ft^2

$$S = 1424.34\ ft^2 \quad (1)$$

Aspect ratio (A): The aspect ratio of the wing is the ratio of the width of the wing to the thickness of the wing. The aspect ratio of the wing for the supersonic business jet is obtained to be 2.1 from the previous report

$$A = 2.1 \quad (2)$$

Taper Ratio (λ): The taper ratio is the ratio between the tip chord to the root chord of the airplane. The effect of taper ratio as shown in the figure 3 below describes that higher taper ratios have a high wing weight provide a good wing tip stall and a provide a good wing fuel volume. Whereas low taper ratios help in reducing the weight of the wing and provide poor tip stalls and poor wing fuel volumes. As the SSBJ is supersonic, it requires less weight hence a lower taper ratio will be selected.

According to the figures 1 and 2, the taper ratio is selected according to the airplanes having similar wing area and similar flight requirements hence the taper ratio is considered to be equal to 0.15

$\lambda = 0.15$ 3

$\lambda = \frac{3^6}{3)}$

c_t= length of the tip chord

c_r= length of the root chord

Type	Dihedral Angle, Γ_w, deg.	Incidence Angle, i_w, root/tip deg.	Aspect Ratio, A	Sweep Angle, $\Lambda_{c/4}$, deg.	Taper Ratio, λ_w	Max. Speed, V_{max}, kts	Wing Type
DASSAULT/BREGUET							
Falcon 10	1.5	NA	7.1	27	0.36	492(25K)	ctl/low
Falcon 20F	2	1.5	6.4	30	0.31	465(25K)	ctl/low
Falcon 50	0	NA	7.6	24	0.32	475	ctl/low
CESSNA							
Citation I 500	4	2.5/-0.5	7.8	0	0.39	277(28K)	ctl/low
Citation II	4.7	NA	8.3	2	0.32	277(28K)	ctl/low
Citation III	2.8	NA	8.9	25	0.35	472(33K)	ctl/low
GATES LEARJET							
24	2.5	1	5.0	13	0.50	473(31K)	ctl/low
35A	2.5	1	5.7	13	0.50	464	ctl/low
55	2.9	NA	7.3	13	0.42	470(30K)	ctl/low
IAI							
1124 Westw. I	2	1/-1	6.5	5	0.33	471	ctl/mid
1125 Astra	2.6 (out)	NA	8.8	34/25 at LE	0.30	472(35K)	ctl/low
Canadair CL601	2.3	3	8.5	25	0.26	450	ctl/low
BAe 125-700	2	2.1/-0.3	6.3	20	0.28	436(28K)	ctl/low
GA Gulfst. III	3	3.5/-0.5	6.5	28	0.31	487	ctl/low
Mu Diamond I	2.7	3/-3.5	7.5	20	0.35	431(30K)	ctl/low
L. Jetstar II	2	1/-1	5.3	30	0.37	475(30K)	ctl/low

ctl = cantilever (30K) = 30,000 ft altitude

Figure 72: Wing Geometric Data for Business Jets

Type	Dihedral Angle, Γ_w, deg.	Incidence Angle, i_w, root/tip deg.	Aspect Ratio, A	Sweep Angle, $\Lambda_{c/4}$, deg.	Taper Ratio, λ_w	Max. Speed, V_{max}, kts	Wing Type
NORTH AMERICAN AVIATION (ROCKWELL)							
XB-70A	-3	NA	1.8	65.6(LE)	0.02	M = 2+	ctl/low
RA-5C	0	NA	4.0	37.5	0.19	1,204(40K)	ctl/high
B-1B	0	NA	??	??	0.32	M = 2+	ctl/low
BOEING							
SST	NA	NA	3.4*	30-72	0.21	1,565(75K)	ctl/low
AST-100	get data from NASA reports						
NASA							
SSXJet I	0	NA	1.84	72(LE)	0.08	M =	ctl/
SSXJet II	0	NA	1.84	72(LE)	0.08	M =	ctl/
SSXJet III	0	NA	1.84	72(LE)	0.08	M =	ctl/
TUPOLEV							
Tu-144	8.3 (out)	NA	1.9	76/57	0.13	1,350(50K)	ctl/low
Tu-22M	0	NA	8.0*	20-65	0.28	1,446	ctl/mid
Dassault MIVA	-1.5	NA	1.8	60(LE)	0.11	1,261(36K)	ctl/low
GD F-111A	0	NA	7.5*	16-72	0.33	1,432+	ctl/high
GD B-58	0	NA	2.2	59(LE)	0	M = 2+	ctl/low
Aerospatiale/British Aerospace							
Concorde	0	NA	1.7	ogive	0.12	1,259(55K)	ctl/low

ctl = cantilever (30K) = 30,000 ft altitude
* taken at lowest sweep angle

Figure 73: Wing Geometric Data for Supersonic Airplanes

Item	Effect of Taper Ratio	
	High	Low
Wing weight	High	Low
Tipstall	Good	Poor
Wing fuel volume	Good	Poor

Figure 74: Effect of Taper Ratio

Dihedral Angle (Γ): Dihedral angle is the angle of the tip chord with respect to the root chord. Dihedral angles may be positive or negative depending upon the position of the wing.

Swept wing airplanes tend to have too much dihedral effect due to the sweep. As the wing for the supersonic business jet has a lot of sweep, a dihedral, won't be added to the SSBJ design. The dihedral will be equal to 0

$\Gamma = 0$ 4

	Wing position		
	Low	Mid	High
Unswept (civil)	5 to 7	2 to 4	0 to 2
Subsonic swept wing	3 to 7	−2 to 2	−5 to −2
Supersonic swept wing	0 to 5	−5 to 0	−5 to 0

Figure 75: Dihedral guidelines

1.1 SWEEP ANGLE- thickness ratio combination

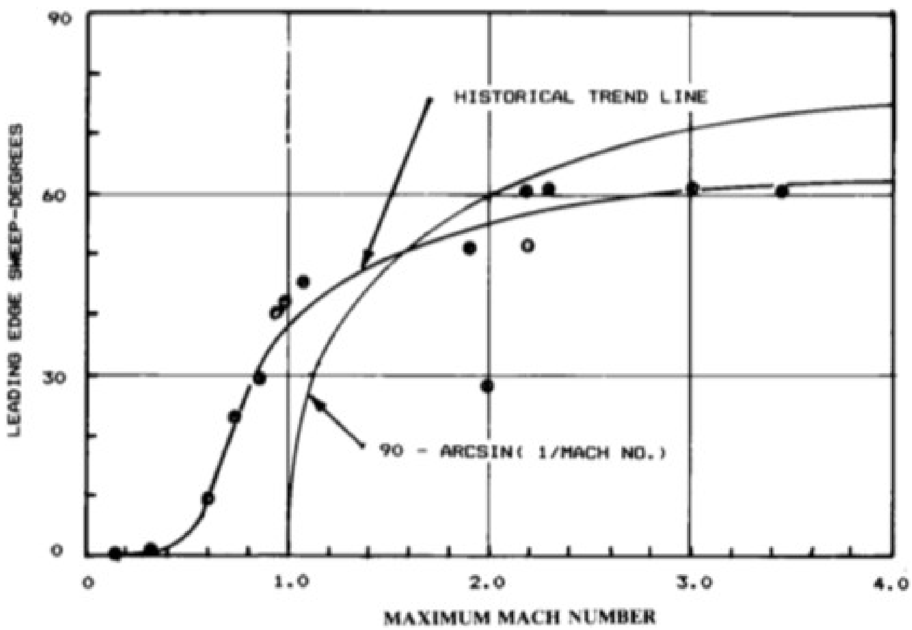


Figure 76: Mach number v/s leading edge sweep

The sweep angle of a supersonic airplane can be obtained by the formula as shown above in the figure obtain from the book Raymer

$$\Lambda = 90 - \arcsin\left(\frac{1}{M}\right)$$

For Mach 1.6,

$$\Lambda = 90 - \arcsin\left(\frac{1}{1.6}\right)$$

$$\Lambda = 51.32^{o} \quad (5)$$

The sweep angle of the supersonic business jet obtained is equal to 51.32 ^{o}.

To obtain the quarter chord sweep angle, $\left(\Lambda_{c/4}\right)$

$$tan\ \Lambda_{›fi} = \tan \Lambda_{\underset{U}{\le}} + £\frac{Z†\lambda}{®(Z‡\lambda)}©$$

$\Lambda_{\underset{U}{\le}} = 41.59\,^{o}$ 6

The thickness ratio can be obtained according to the design Mach number as described in the figure below,

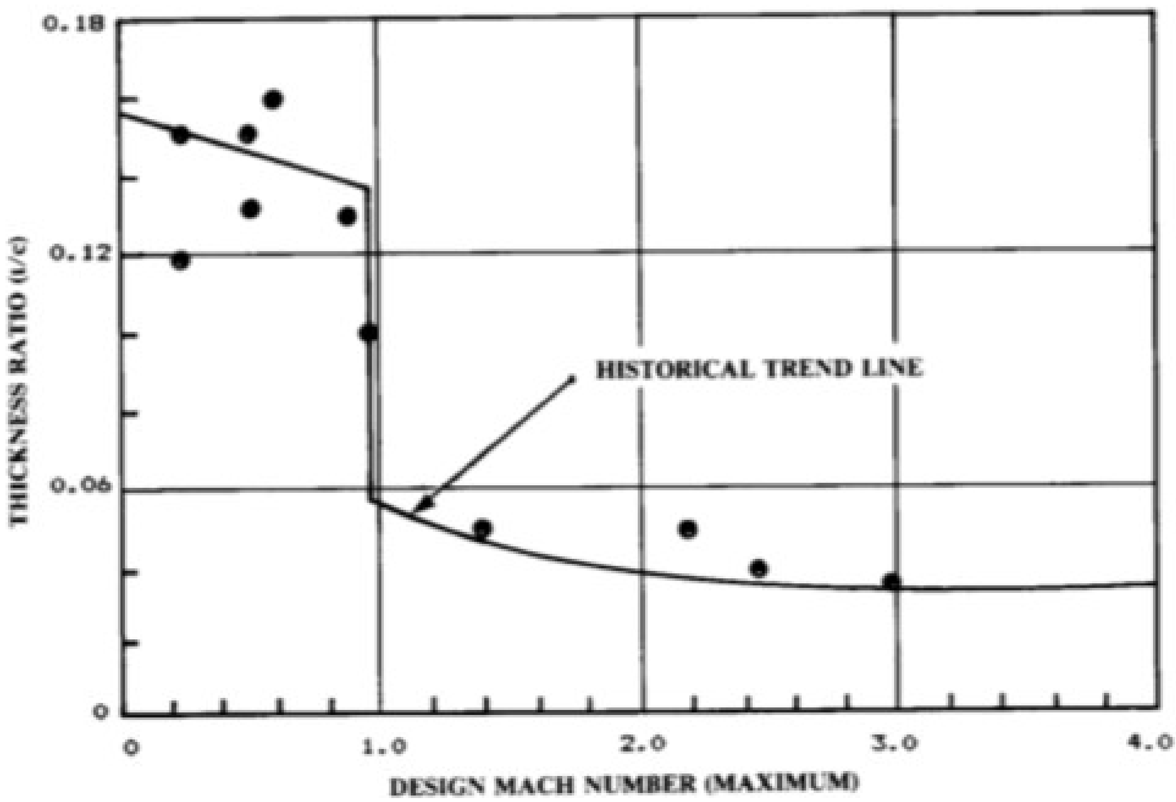


Figure 77: Design Mach number verses the thickness ratio

The thickness ratio $(t/c)_r = 0.04$ $(t/c)_t = 0.04$ is acceptable for this design as observed in figure 5.

Hence, an airfoil with 4% thickness at the root and 3% thickness at the tips is to be selected for the airplane.

The spars on the wing can be defined as front spars and rear spares. The front spar can be located at 0.2c and the trailing edge spars are located at 0.695c , where c is the chord length of the airfoil. The leading edge and trailing edge spars of both the tip and root airfoils can be obtained as described below,

For the leading-edge spars:
Along the root chord: 0.2c = 0.2 * 45.29 = 9.0 ft 7
Along the tip chord: 0.2c = 0.2 * 6.79 = 1.35 ft 8

For the trailing-edge spars:
Along the root chord: 0.695c = 0.695 * 45.29 = 31.47 ft (from the leading edge) 9
Along the tip chord: 695c = 0.695 * 6.79 = 4.72 ft (from the leading edge) 10

6.3 AIRFOIL SELECTION

This section explains the type of airfoil to be selected, the incidence angle (i) and the twist angle.

Type of airfoil(s):

The selection of the airfoil can be done using the following formulas:

The Reynolds number at the root and tip of the airfoils can be determined by,

$$R0_2 = \frac{\rho D3_)}{\mu} = \frac{\text{è.om∗Z>}^{-\text{U}}\text{∗Z>oo.nw∗èx.mq}}{\text{m.qoq∗Z>}^{-\backslash}} = 75.18 * 10^{o} \quad 11$$

$$R0_G = \frac{\rho D3_6}{\mu} = \frac{\text{è.om∗Z>}^{-\text{U}}\text{∗Z>oo.nw∗o.nq}}{\text{m.qoq∗Z>}^{-\backslash}} = 11.0 * 10^{o} \quad 12$$

$$C_{+9-:,} = 0.95\,\frac{\text{FNö}_{„‰Â)} \ddagger \text{Nö}_{,,‰Â6}{}^{\text{H}}}{\text{m}} = 0.95\,\frac{\text{m.xm‡m.pm}}{\text{m}} = 2.3 \quad 13$$

$$C_{+9-:4} = 2.3\sin(51.32) = 1.8 \quad 14$$

As the thickness ratio obtained is 0.04 for the roots and the tips, an airfoil having 4% thickness respectively will be used to design the wing of the supersonic business jet.

For the roots and tips, NASA SC(2)-0404 airfoil is selected that has a 4% thickness at 37% chord.

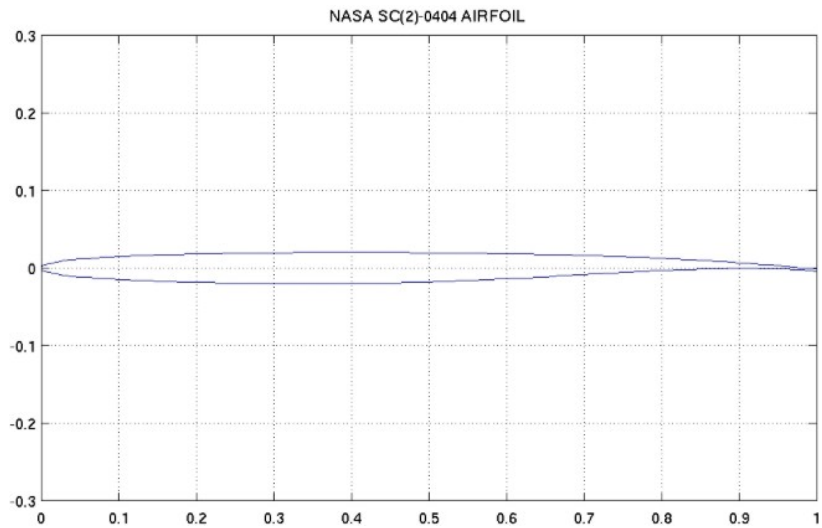


Figure 78: NASA SC(2)-0404

Incidence angle (i)

Item	Large i_w	Small i_w
Cruise drag	High	Low
Cockpit visibility	Good	Watch out
Landing attitude in terms of nose gear hitting runway first	Watch out	No problem

Figure 79: Effect of incidence angle on the aircraft

As observed in the figure above, the cruise drag increases with the increase in the incidence angle which is a huge drawback to the supersonic business jet hence no incidence will be applied to the wing of the supersonic business jet. Hence

$$i_W = 0 \qquad (15)$$

Twist (aerodynamic and geometric) (ε_t):

Twist on the wing is useful to prevent tip stall. There are two types of twist:

i) Aerodynamic twist

The angle between the zero-life angle of both the tip airfoil and the root airfoil is termed as the aerodynamic twist.

ii) Geometric twist

The change in the angle of incidence of the airfoil is termed as the geometric twist. It is measured with respect to the root chord of the wing.

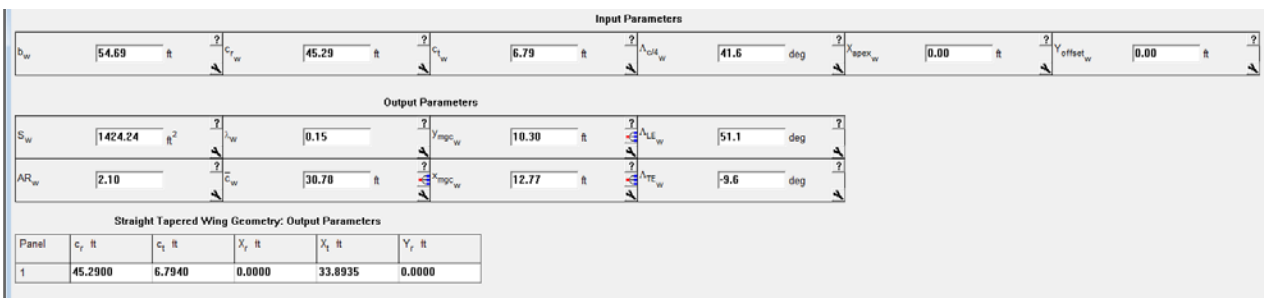
The more the twist is applied, the less it will perform at other lift coefficients. This is the reason large amount of twists should be avoided.

As the supersonic business jet has a compound wing structure, no twist either aerodynamic or geometric is applied to the wing.

$\varepsilon_t = 0$ 16

6.4 WING DESIGN EVALUATION

For the wing design evaluation, the AAA program will be used to verify the CLmax on the wing of the airplane calculated.


Input Parameters

b_w	54.69	ft	c_{r_w}	45.29	ft	c_{t_w}	6.79	ft	$\Lambda_{c/4_w}$	41.6	deg	X_{apex_w}	0.00	ft	Y_{offset_w}	0.00	ft

Output Parameters

S_w	1424.24	ft^2	λ_w	0.15		Y_{mgc_w}	10.30	ft	Λ_{LE_w}	51.1	deg
AR_w	2.10		$\bar{c}_w$	30.78	ft	X_{mgc_w}	12.77	ft	Λ_{TE_w}	-9.6	deg

Straight Tapered Wing Geometry: Output Parameters

Panel	c_r ft	c_t ft	X_r ft	X_t ft	Y_r ft
1	45.2900	6.7940	0.0000	33.8935	0.0000

Figure 80: Wing Parameters AAA

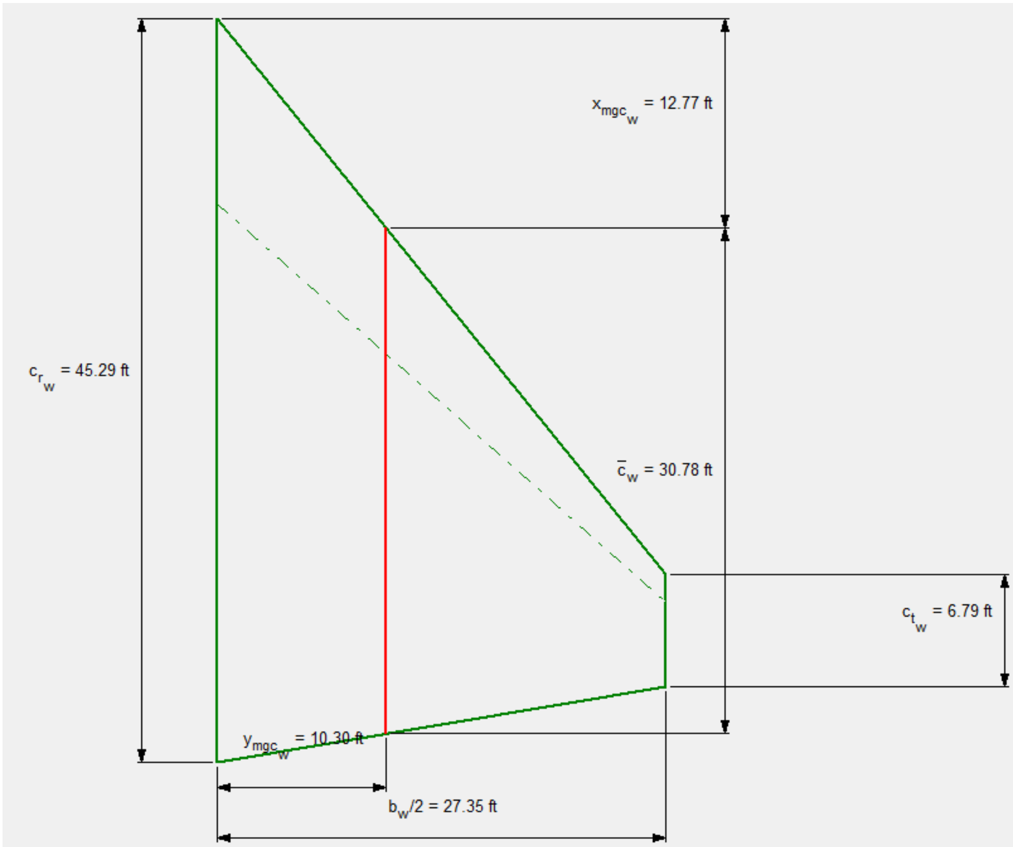


Figure 81: Wing Design Obtained from AAA

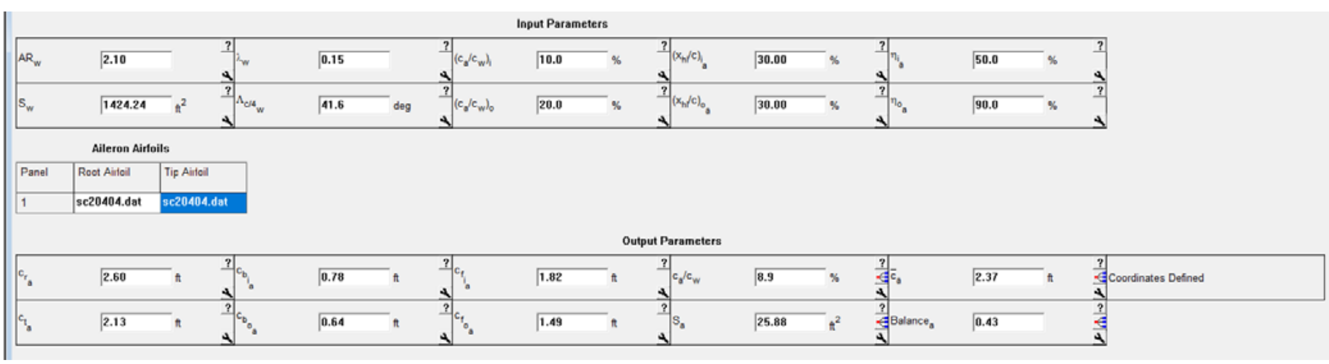


Figure 82: Aileron Geometry Sizing

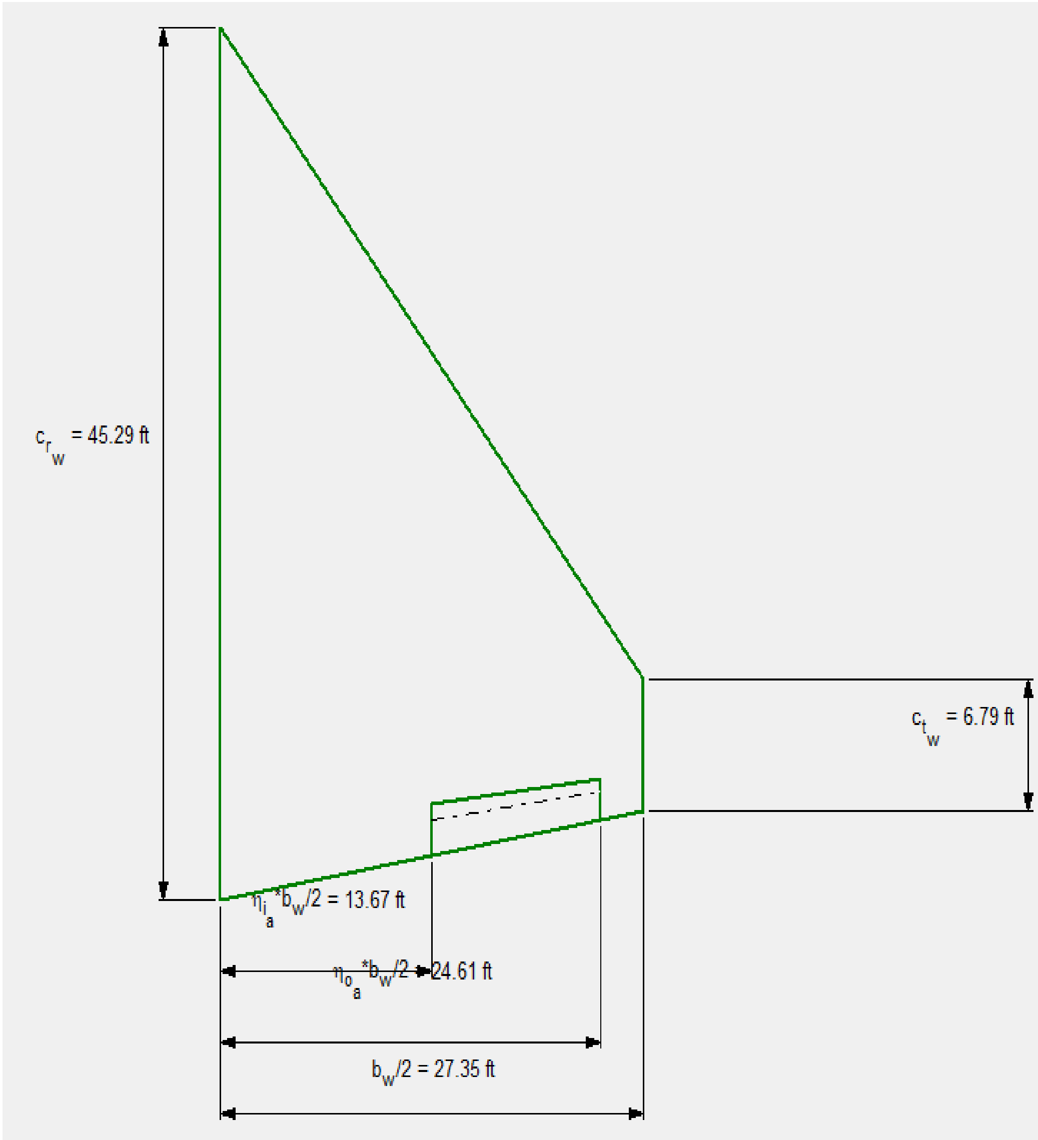
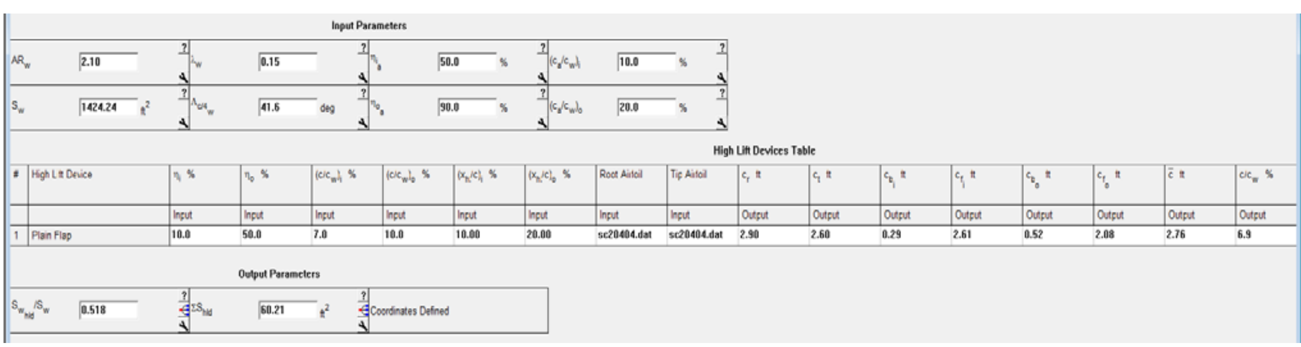


Figure 83: Aileron sizing obtained from AAA



Input Parameters

AR_w	2.10	λ_w	0.15	η_{i_a}	50.0 %	$(c_a/c_w)_i$	10.0 %
S_w	1424.24 ft^2	$\Lambda_{c/4_w}$	41.6 deg	η_{o_a}	90.0 %	$(c_a/c_w)_o$	20.0 %

High Lift Devices Table

#	High Lift Device	η_i %	η_o %	$(c/c_w)_i$ %	$(c/c_w)_o$ %	$(x_h/c)_i$ %	$(x_h/c)_o$ %	Root Airfoil	Tip Airfoil	c_r ft	c_t ft	c_{b_i} ft	c_{r_i} ft	c_{b_o} ft	c_{r_o} ft	$\bar{c}$ ft	c/c_w %
		Input	Input	Input	Input	Input	Input	Input	Input	Output	Output	Output	Output	Output	Output	Output	Output
1	Plain Flap	10.0	50.0	7.0	10.0	10.00	20.00	sc20404.dat	sc20404.dat	2.90	2.60	0.29	2.61	0.52	2.08	2.76	6.9

Output Parameters

$S_{w_{hld}}/S_w$	0.518	ΣS_{hld}	60.21 ft^2	Coordinates Defined

Figure 84: Flap Geometry Sizing

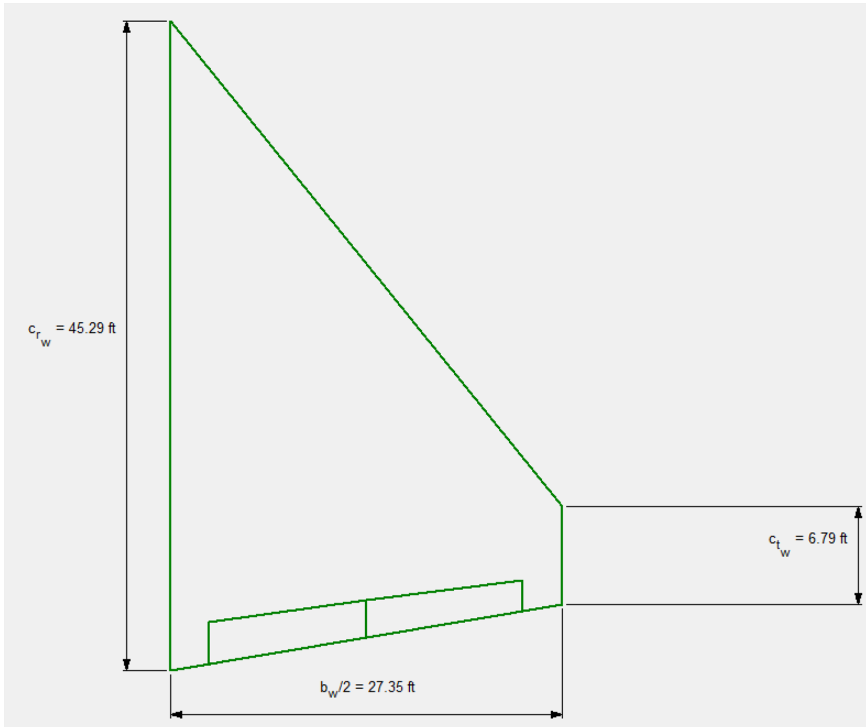


Figure 85: Aileron and Flap Locations Obtained from AAA

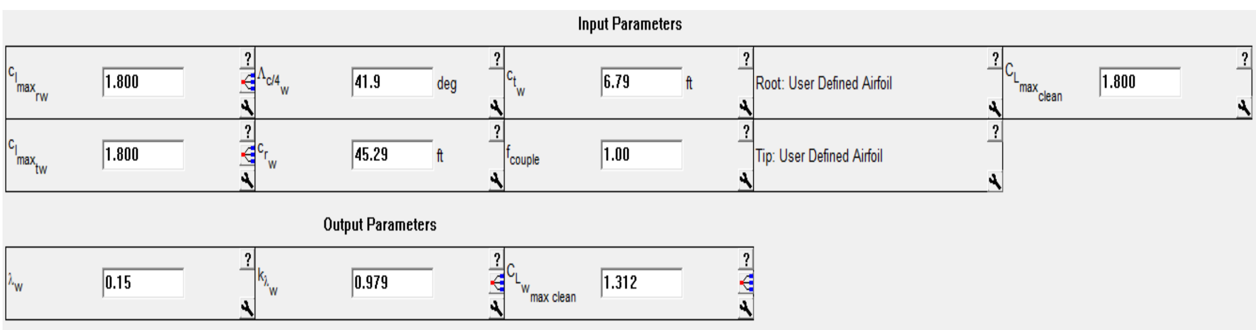


Figure 86: Cl_max requirements

6.5 DESIGN OF THE HIGH-LIFT DEVICES

The high lift devices consist of the flaps and the ailerons on the aft section of the wing. The location of the high lift devices on the wing are described in the figure below:

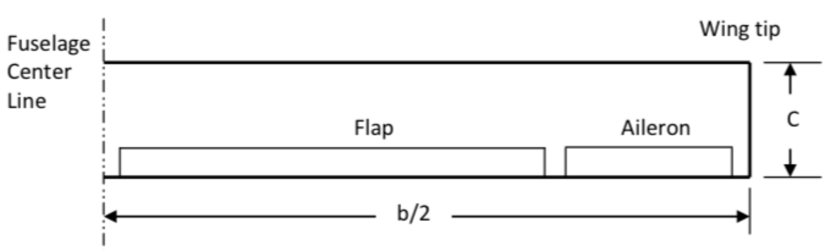


Figure 87: location of high-lift devices on the wing.

The following equations are already known to us from assignment number 3:

$C_{L_{max_{TO}}} = 2.0$ (Take-off) 17

$C_{L_{max_L}} = 2.2$ (landing) 18

$C_{L_{max}} = 1.8$ (clean stage/ cruise stage) 19

The type and size of high-lift devices needed to meet the $C_{L_{max_{TO}}}$ and $C_{L_{max_L}}$ requirements can be calculated using the following equations:

This equation denotes the wing maximum lift coefficient. Now, to see if the wing can produce its own lift or not, the following equation is used,

$\Delta C_{L_{max_{TO}}} = 1.05 \left(C_{L_{max_{TO}}} - C_{L_{max}}\right) = 1.05\ (2.0 - 1.8) = 0.21$ 20

$\Delta C_{L_{max_L}} = 1.05 \left(C_{L_{max_L}} - C_{L_{max}}\right) = 1.05\ (2.2 - 1.8) = 0.42$ 21

c_f/c is the fraction of the flap chord length and the chord of the airfoil. Cf can be obtained from the trailing edge spars as defined in equation 9 and 10. Hence we get,

$c_f/c = 0.3$ 22

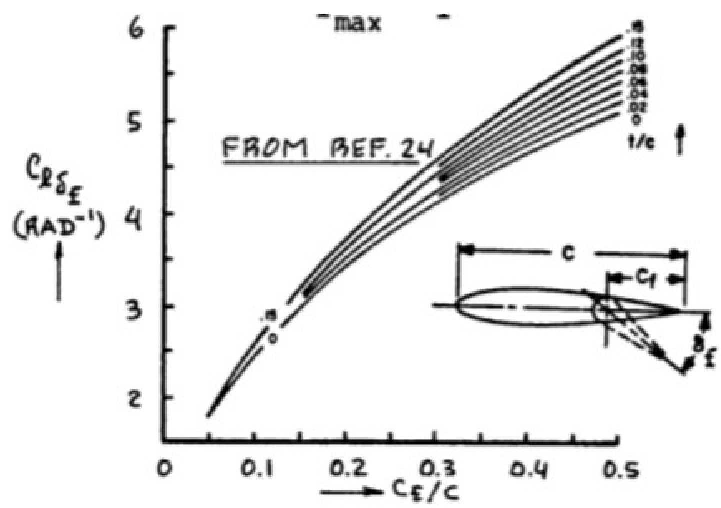


Figure 88: Effect of thickness ratio and chord ratio on $C_{l_{\delta_f}}$.

$C_{l_{\delta_f}} = 4.5\ (RAD^{-1})$ 23

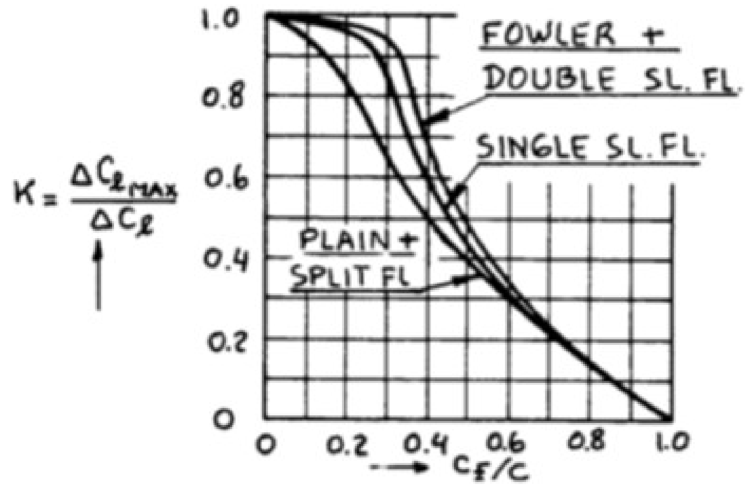


Figure 89: Effect of Flap chord ratio and Flap type on K.

where K can be found from the figure above,
K = 0.85 for cf/c = 0.3 using the single slotted flaps 24

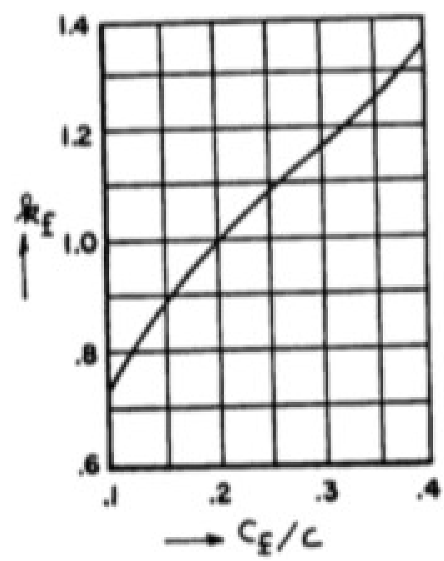


Figure 90: Relation between kf and cf/c

K_f can be obtained from the figure above comparing it with cf/c where,
$K_f = 1.16$ 25

for plain flaps:
K' can be obtained from the figure

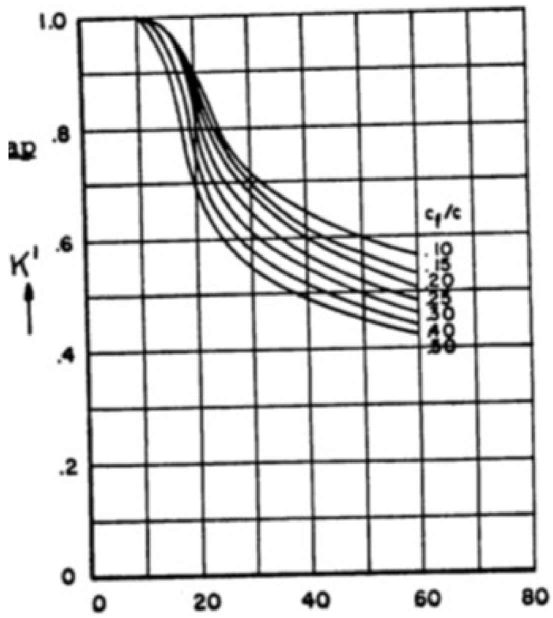


Figure 91: Relation between the flap angle and K'

While landing: $\Delta C_Y = C_{Y_{\delta'}} * \delta' * K' = 4.5 * rad(40) * 0.55 = 1.728$ 26

While Take-off: $\Delta C_Y = C_{Y_{\delta'}} * \delta' * K' = 4.5 * rad(20) * 0.75 = 1.178$ 27

$K_{\Lambda} = (1 - 0.08 \cos^{2} \Lambda) \cos^{\frac{3}{4}} \Lambda = 0.955 * 0.801 = 0.765$ 28

$\Delta C_{Y_{max}} = \Delta C_{l_{max}} \left(\frac{S_w}{S}\right) K_{\Lambda} = 0.765 * 2.0 * 0.77 = 1.178$ 29

hence the value of the maximum coefficient of lift obtained in equation 27 is equal to the coefficient of lift obtained in equation 29, hence the ratio Sw/S = 0.77 is said to be sufficient for the wing to obtain the required lift during the take-off stage.

$\Delta C_Y = \left(\frac{1}{k}\right) \Delta C_{Y_{max}}' = \left(\frac{1}{0.85}\right) * 1.8 = 2.11$ 30

hence, equation 28 gives a value of 2.11 which is within the 5% range of $C_{l_{max}} = 2.2$ hence this proves that the wing can fly on its own.

Table 8.12b) Supersonic Cruise Airplanes: Vertical Tail Volume, Rudder, Aileron and Spoiler Data

Type	Wing Area S ft^2	Wing Span b ft	Vert. Tail Area S_v ft^2	S_r/S_v	x_v ft	$\bar{V}_v$	Rudder Chord root/tip $fr.c_v$	S_a/S	Ail. Span Loc. in/out fr.b/2	Ail. Chord in/out $fr.c_w$
NORTH AMERICAN AVIATION (Now Rockwell)										
XB-70A	6,297	105	468	0.75	48.5	0.034	****	0.067	.33/.72	.13/.31*
RA-5C	700	53.0	102	1.0**	21.8	0.060	1.0**	no ailerons		
BOEING***										
SST	9,000	174	866	0.26	88.5	0.049	.23/.46	0.014	.78/.96	.32/.43
AST-100	11,630	138	890	1.0**	121	0.067	1.0**	0.017	.72/1.0	.15/.29
NASA***										
SSXjet I	965	42.1	75.0	1.0**	38.3	0.071	1.0**	0.018	.76/1.0	.21/.26
SSXjet II	965	42.1	75.0	1.0**	35.5	0.066	1.0**	0.018	.76/1.0	.21/.26
SSXjt III	1,128	45.6	97.0	1.0**	32.1	0.061	1.0**	0.017	.74/1.0	.19/.26
TUPOLEV										
Tu-144	4,715	94.5	648	0.19	55.6	0.081	.20/.35	0.100	.31/.97	.11/.51*
Tu-22M	1,585	113	437	0.17	35.6	0.087	.39/.36	NA	.80/.95	.24/.28
Tu-22	2,062	90.9	376	0.14	29.6	0.059	.25/.33	0.051	.66/.95	.29/.31
Dassault										
Mirage IVA	840	38.9	129	0.12	14.1	0.056	.14/.24	0.120	.30/.96	.17/.63*
GD F-111A	530	63.0	115	0.25	18.6	0.064	.27/.29	no ailerons		
Concorde	3,856	84.0	477	0.24	54.1	0.080	.18/.47	0.089	.51/1.0	.15/.27*
Rockw.B1B	1,950	137	230	0.30	45.8	0.039	.29/.38	no ailerons		
Conv. B58	1,481	57.0	153	0.24	31.8	0.057	.32/.31	0.120	.18/.69	.16/.28*

* Elevon equipped ** Slab vertical tail ***Study projects only
**** Rudder hingeline skewed

Figure 92: Supersonic cruise Airplanes Wing/ Tail data

According to the data compared with different airplanes as shown in the figure above, the best place to locate the ailerons is 0.80 - 0.95 (b/2).

b/2 = 27.345 ft
therefore, the location of the ailerons can be determined by,
0.8 * b/2 = 21.87 (from the root chord) 31
0.95 * b/2 = 25.97 (from the root chord) 32
the length of the ailerons is 4.1 ft along the spars

6.6 DESIGN OF THE LATERAL CONTROL SURFACES

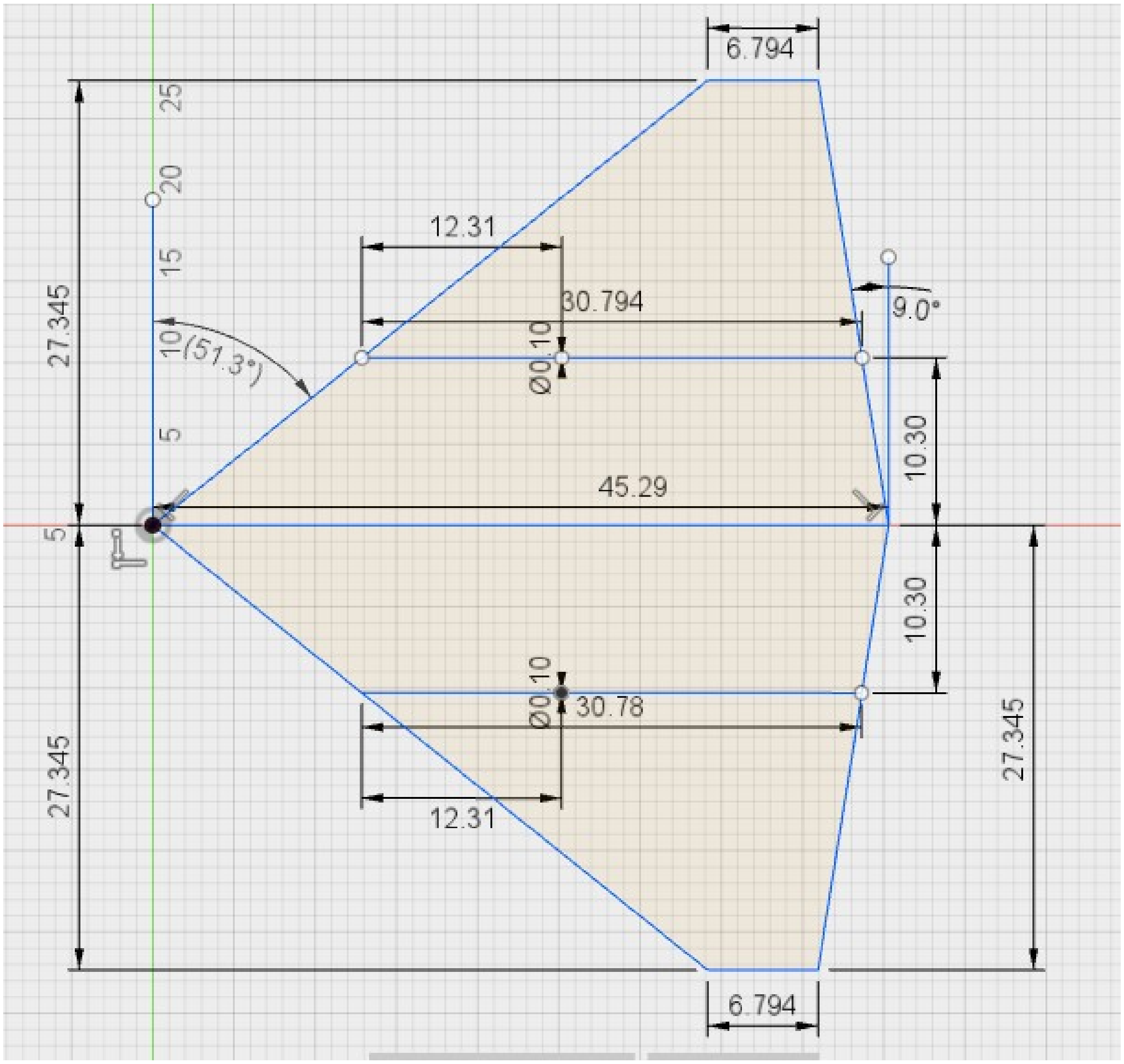


Figure 93: Wing Layout of the SSBJ

6.7 DRAWINGS

To draw the wing of the airplane, the following parameters must first be calculated:

i) Span, b

The span of the airplane can be calculated using the equation:

$$b = (AS)^{\frac{1}{2}} = (2.1 * 1424.34)^{\frac{1}{2}} = 54.69\ ft \quad (33)$$

ii) Root chord, c_r

The root chord of the airplane wing can be calculated using the equation:

$c_2 = \frac{\text{m∗æ}}{\text{Ì(Z‡Ó)}} = \frac{\text{m∗Zèmè.pè}}{\text{xè.oq∗Z.Zx}} = 45.29\ ft$ 34

iii) Tip chord, c_t

The tip chord of the airplane wing can be calculated using the equation:
$c_G = \lambda * c_2 = 0.15 * 45.29 = 6.794\ ft$ 35

iv) Mac (mean aerodynamic chord)

$Mac(c) = \frac{\text{m}}{\text{p}} c_2 \frac{\text{Z‡}\lambda\text{‡}\lambda^{\text{d}}}{\text{Z‡}\lambda} = 30.78\ ft$ 36

where, Mac is the length of the mean aerodynamic chord.

$Y = \frac{\text{Ì}}{\text{o}}[(1+2\lambda)(1+\lambda)] = 13.62\ ft$ 37

Y is the distance of the mean aerodynamic center from the center line of the airplane.

v) Mgc (mean geometric chord)

$Mgc = \frac{\text{æ}}{\text{Ì}} = \frac{\text{Zèmè.pè}}{\text{xè.oq}} = 26.04\ ft$ 38

vi) Leading-edge sweep angle

$\Lambda_{+5} = 90 - \arcsin \text{F}\frac{\text{Z}}{\pm}\text{H} = 51.32°$ 39

vii) Trailing edge sweep angle

$\Lambda_{;5} = -9°$ 40

The trailing edge sweep angle was obtained from the figure of the wing above.

viii) Co-ordinates of the aerodynamic center (xac,yac)

The x-coordinate of the aerodynamic center is 0.4(Mac) for supersonic aircrafts. Therefore
Xac = 0.4 * 30.78 = 12.312 ft (from the leading edge of the mean aerodynamic chord)
Yac = 13.62 ft (as obtained from the equation 37 above)

Hence the co-ordinates of the aerodynamic center are: (12.312 ft, 13.62ft)

6.8 DISCUSSION

This report specifies the design of the wing. In the first section, the parameters required to design the wing were obtained from the 4th report such as the aspect ratio and the wing area. Then certain parameters such as the incidence angle, taper ratio, thickness ratio, dihedral angle, etc were calculated using the provided formulas and certain parameters were assumed form similar aircrafts data.

Once the data for the wing is obtained, an airfoil was selected that satisfies all the required conditions such as the coefficient of lift. The airfoil selected was the NASA SC(2)-0404. After selecting the airfoil, the AAA program was used to evaluate the wing parameters and compare them with the obtained results to check the accuracy of the results. After comparing the results, the high-lift devices were designed to match the requirements to obtain the required lift. The high lift devices contain the ailerons and flaps. The high lift devices obtained are shown in the AAA section figures along with the parameters and the drawings.

As all the data is obtained, the wing was designed as shown above using the root chord, tip chord, wing area, span, sweep angle, quarter chord angle, taper ratio and the aspect ratio. The aerodynamic center was located in the design of the wing along with the center of gravity of the wing.

6.9 CONCLUSIONS

For the supersonic business jet, a cranked arrow wing is a required configuration rather than a delta wing. Hence for future design purpose, a cranked arrow wing will be designed due to which the weight of the wing can be reduced by a certain extent and adding the control surfaces on the wing surface which neglects he use of horizontal on the empennage section hence reducing weight, cost, maintainability etc.

CHAPTER 7: DESIGN OF THE EMPENNAGE AND THE LONGITUDINAL AND DIRECTIONAL CONTROLS

7.1 INTRODUCTION

This is the seventh report of the series of 12 for the preliminary design of the Supersonic Business Jet. This report describes the design of the empennage and the longitudinal and directional stability controls. The weight sizing, performance parameters, fuselage and cockpit, and the wing design were calculated and designed in the previous reports. This report focuses on the empennage section and the horizontal and vertical stabilizers for the directional and longitudinal controls.

The parameters required to design the horizontal and vertical stabilizers are the following:

I. Wing Area
II. Aspect ratio
III. Wing span
IV. Taper ratios
V. Thickness ratios
VI. Dihedral angles
VII. Airfoils
VIII. Incidence angles
IX. Sweep angles
X. Control surface sizes and their layouts

The parameters such as the wing area, wing aspect ratio, wing mean aerodynamic chord length, thickness ratios of the wing airfoil, wing sweep angles, etc were discussed in the previous report number 6 that specifies the wing design.

The empennage section of the supersonic airplane contains a T-tail configuration as it is considered to be the suggested configuration for business jets to obtain better directional and longitudinal stability.

The empennage section of the supersonic business jet will be worked on as discussed in report number 2 then the calculations of the vertical and horizontal stabilizers will be calculated based on certain assumptions made from similar aircrafts and the obtained data from the previous reports and then CAD drawings will be designed based on the calculations obtained.

Once the horizontal and vertical stabilizers are designed, the directional and longitudinal control surfaces will be worked on to determine the size and the layout according to the requirements of the supersonic business jet.

The similar process will be worked on the AAA program. These calculations and data obtained will then be compared with the data obtained from the manual calculations and then final plan forms will be created according to the data obtained.

7.2 OVERALL EMPENNAGE DESIGN

As discussed in the second report, the empennage section will have a T-tail configuration as the T-tail is the best preferred configuration for business jets.

The reason for selecting the T-tail configuration is that it gets a direct clean flow over the horizontal stabilizer rather than the turbulent flow created after passing the wing section of the aircraft. Due to the direct clean flow obtained, a huge amount of induced drag created on the surface of the horizontal stabilizer due to the turbulent flow from the wing is decreased in the case of a T-tail configuration.

Determine the location of the empennage (L_{HT}, L_{VT}, L_c).

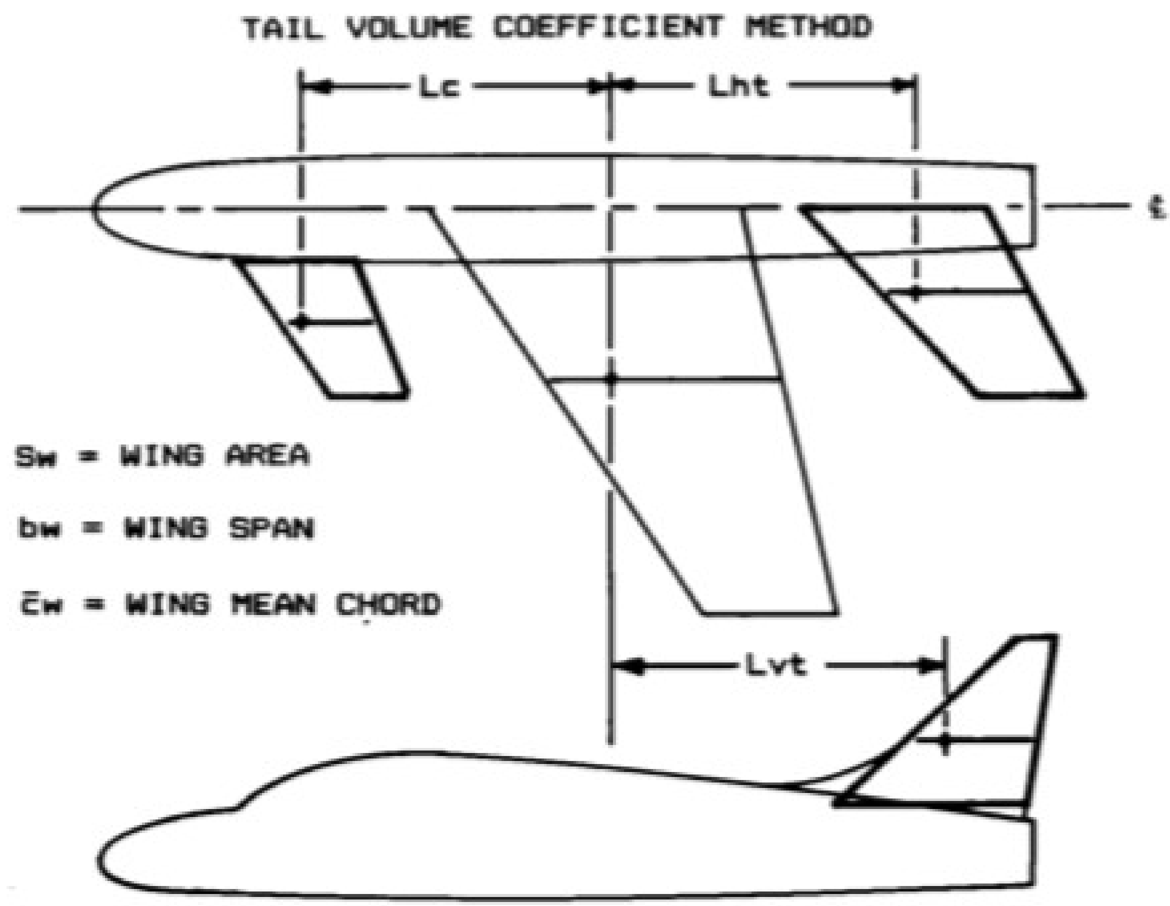


Figure 94: Tail volume coefficient method

L_h: location of the horizontal stabilizer with respect to the quarter chord of the wing. The location of the horizontal stabilizer can be determined as shown in the figure above.

L_v: location of the vertical stabilizer with respect to the quarter chord of the wing. The location of the vertical stabilizer can be determined as shown in the figure above.

L_c: location of the canard on the aircraft with respect to the quarter chord of the wing of the airplane as described in the figure.

The supersonic business design does not have a canard configuration so the canard design will not be discussed in this report.

For aircrafts having aft mounted engines, the vertical moment arm can be considered to be between 45% - 50% the fuselage length. Hence the location of the quarter chord of the vertical stabilizer will be 50% of the fuselage length from the quarter chord of the airplane wing. Hence the vertical stabilizer is considered to be at a distance of:

L_{VT} = 50% of fuselage length = 50% of 50ft (from the quarter chord of the wing)
L_{VT} = 25ft 1

The location of the horizontal stabilizer depends on the location of the tip chord of the vertical stabilizer due to the configuration being a T-tail configuration as the horizontal stabilizer is mounted on the tip of the vertical stabilizer. Since no much data is provided for the time being a wild guess will be considered comparing with other airplanes as described in the figures below obtained from the aircraft design book by Jan Roaskam.

L_{HT} = 40.78ft 2

Determine the size of the empennage (S_s, S_v, S_c):

The size of the empennage contains the area of the vertical stabilizer, horizontal stabilizer and the area of the canard section. As described earlier the supersonic business jet has no canard configuration so the canard area will not be discussed.

The tail volume coefficients can be obtained from the following figure:

Type	Wing Area S ft^2	Wing Span b ft	Vert. Tail Area S_v ft^2	S_r/S_v	x_v ft	$\bar{V}_v$	Rudder Chord root/tip fr.c_v	S_a/S	Ail. Span Loc. in/out fr.b/2	Ail. Chord in/out fr.c_w
NORTH AMERICAN AVIATION (Now Rockwell)										
XB-70A	6,297	105	468	0.75	48.5	0.034	****	0.067	.33/.72	.13/.31*
RA-5C	700	53.0	102	1.0**	21.8	0.060	1.0**	no ailerons		
BOEING***										
SST	9,000	174	866	0.26	88.5	0.049	.23/.46	0.014	.78/.96	.32/.43
AST-100	11,630	138	890	1.0**	121	0.067	1.0**	0.017	.72/1.0	.15/.29
NASA***										
SSXjet I	965	42.1	75.0	1.0**	38.3	0.071	1.0**	0.018	.76/1.0	.21/.26
SSXjet II	965	42.1	75.0	1.0**	35.5	0.066	1.0**	0.018	.76/1.0	.21/.26
SSXjt III	1,128	45.6	97.0	1.0**	32.1	0.061	1.0**	0.017	.74/1.0	.19/.26
TUPOLEV										
Tu-144	4,715	94.5	648	0.19	55.6	0.081	.20/.35	0.100	.31/.97	.11/.51*
Tu-22M	1,585	113	437	0.17	35.6	0.087	.39/.36	NA	.80/.95	.24/.28
Tu-22	2,062	90.9	376	0.14	29.6	0.059	.25/.33	0.051	.66/.95	.29/.31
Dassault										
Mirage IVA	840	38.9	129	0.12	14.1	0.056	.14/.24	0.120	.30/.96	.17/.63*
GD F-111A	530	63.0	115	0.25	18.6	0.064	.27/.29	no ailerons		
Concorde	3,856	84.0	477	0.24	54.1	0.080	.18/.47	0.089	.51/1.0	.15/.27*
Rockw.B1B	1,950	137	230	0.30	45.8	0.039	.29/.38	no ailerons		
Conv. B58	1,481	57.0	153	0.24	31.8	0.057	.32/.31	0.120	.18/.69	.16/.28*

* Elevon equipped ** Slab vertical tail ***Study projects only
**** Rudder hingeline skewed

Figure 95: Tail Volume Coefficients for vertical Stabilizer

S_{VT} = Area of the vertical stabilizer

Hence the area of the vertical stabilizer can be determined by the following equation:

$$c_{VT} = \frac{L_{VT} S_{VT}}{b_w S_w} \qquad 3$$

where,
c_{VT} = Volume coefficient of the vertical Stabilizer = 0.09
L_{VT} = Distance between the quarter chord of the vertical with respect to the wing = 25ft
S_{VT} = Area of the vertical stabilizer
b_w = Wing Span = 54.69ft
S_w = Wing Area = 1424.34 ft^2

Substituting these values obtained in equation 3 will give the area for the vertical stabilizer. The volume coefficient of the vertical stabilizer can be obtained from the figure above:

For the T-tail configuration, the tail volume coefficient can be reduced by 5% for the vertical stabilizer due to the end plate effect hence 5% of 0.09 = 0.0855.

$$S_{VT} = \frac{c_{VT} b_w S_w}{L_{VT}} = \frac{0.0855 \times 54.69 \times 1424.34}{25} = 266.4 ft^2 \qquad 4$$

the span of the vertical stabilizer can be determined from the equation:

$$b_{VT} = \sqrt{A * S_{VT}} = 17.88\ ft \quad (5)$$

the root chord of the vertical stabilizer can be obtained from the equation below:

$$c_{r_{VT}} = \frac{2 * S}{b(1+\lambda)} = 15.75\ ft \quad (6)$$

the tip chord of the vertical stabilizer is:

$$c_{t_{VT}} = \lambda * c_{r_{VT}} = 0.9 * 15.75 = 14.175\ ft \quad (7)$$

the mean aerodynamic chord of the vertical stabilizer can be obtained from the equation,

$$c = \frac{2}{3} * c_{r_{VT}} \left(\frac{1+\lambda+\lambda^2}{1+\lambda}\right) = 14.979\ ft \quad (8)$$

The distance of the mean aerodynamic chord from the root chord can be found from the equation below. In the case of vertical stabilizers, the equation becomes 2 time the original equation as explained below:

$$Y_{VT} = 2 * \left(\frac{b}{6}\right)\left(\frac{1+2\lambda}{1+\lambda}\right) = 8.783\ ft \quad (9)$$

Type	Wing Area S ft^2	Wing mgc $\bar{c}$ ft	Wing Airfoil root/tip	Hor. Tail Area S_h ft^2	S_e/S_h	x_h ft	$\bar{V}_h$	Elevator Chord root/tip fr.c_h
NORTH AMERICAN AVIATION (Now Rockwell)								
XB-70A	6,297	78.5	NA	delta with elevons and small canard				
RA-5C	700	15.7	NA	356	1.0	17.1	0.56	stabilator
BOEING								
SST*	9,000	29.0**	NA	592	0.16	161	0.36	.24/.74
AST-100*	11,630	96.2	NA	547	1.0	107	0.052	stabilator
NASA*								
SSXjet I	965	30.6	.002/.003	65.0	1.0	47.2	0.10	stabilator
SSXjet II	965	30.6	.002/.003	80.0	1.0	41.2	0.09	stabilator
SSXjet III	1,128	33.1	.002/.003	80.0	1.0	41.9	0.09	stabilator
TUPOLEV								
Tu-144	4,715	58.3		delta with elevons and folding canard				
Tu-22M	1,585	15.4**	NA	727	1.0	37.2	1.11	stabilator
Tu-22	2,062	23.7***	NA	620	0.12	34.7	0.44	.29/.30
Dassault								
Mirage IVA	840	24.7	NA	delta with elevons				
GD F-111A	530	9.12**	NA	352	1.0	17.6	1.28	stabilator
Concorde	3,856	61.7	NA	ogive with elevons				
Rockwell B1B	1,950	15.8**	NA	494	1.0	49.9	0.80	stabilator
Convair B58	1,481	34.6	NA	delta with elevons				

* Study projects only ** Measured at forward sweep *** Fixed sweep airplane
See Refs. xx - yy

Figure 96: Tail Volume Coefficients for horizontal stabilizers

S_{HT} = Area of the horizontal Stabilizer

The area of the horizontal stabilizer can be obtained from the following equation:

$$c_{HT} = \frac{L_{HT} * S_{HT}}{C_w * S_w} \qquad 10$$

where,
c_{HT} = Volume coefficient of the horizontal stabilizer = 0.377
L_{HT} = Distance between the quarter chord of the horizontal with respect to the wing = 40.78ft
S_{HT} = Area of the horizontal stabilizer
S_w = Wing Area = 1424.34 ft^2
C_W = Mean Aerodynamic Chord length of the wing = 30.78 ft

Substituting these values in equation 5 provides the area of the horizontal stabilizer.

For T-tail configurations, the tail volume coefficient can be decreased by 5% due to clean air obtained on the surface of the horizontal stabilizer. Hence 5% of 0.377 = 0.359

$$S_{HT} = \frac{c_{HT} * C_w * S_w}{L_{HT}} = \frac{0.359*30.78*1424.34}{30.78} = 385.94 ft \qquad 11$$

the span of the horizontal stabilizer can be calculated using the equation below:

$$b_{HT} = \sqrt{A * S_{HT}} = 34.027\ ft \qquad 12$$

$$\frac{b_{HT}}{2} = 17.0\ ft \qquad 13$$

Being a T-tail configuration, the root chord of the horizontal stabilizer cannot be greater than 1.1 times the tip chord of the vertical stabilizer. Hence the root chord of the horizontal stabilizer can be obtained using the equation below:

$$c_{r_{HT}} = \frac{2*S}{b(1+\lambda)} = 14.177\ ft \qquad 14$$

$$c_{t_{HT}} = c_{r_{HT}} * \lambda = 8.5\ ft \qquad 15$$

the mean aerodynamic chord of the horizontal stabilizer can be obtained as shown below:

$$\bar{c}_{HT} = \frac{2}{3} * c_{r_{HT}} * \left(\frac{1+\lambda+\lambda^2}{1+\lambda}\right) = 11.5778\ ft \qquad 16$$

The distance of the mean aerodynamic center from the root chord can be found from the equation below,

$$\bar{Y}_{HT} = \left(\frac{b}{6}\right)\left(\frac{1+2\lambda}{1+\lambda}\right) = 7.7978\ ft \qquad 17$$

7.3 DESIGN OF THE HORIZONTAL STABILIZER

Type	Dihedral Angle, Γ_h deg.	Incidence Angle, i_h deg.	Aspect Ratio, A_h	Sweep Angle, $\Lambda_{c/4_h}$ deg.	Taper Ratio, λ_h
Homebuilts	+5 - -10	0 fixed to variable	1.8 - 4.5	0 - 20	0.29 - 1.0
Single Engine Prop. Driven	0	-5 - 0 or variable	4.0 - 6.3	0 - 10	0.45 - 1.0
Twin Engine Prop Driven	0 - +12	0 fixed to variable	3.7 - 7.7	0 - 17	0.48 - 1.0
Agricultural	0 - +3	0	2.7 - 5.4	0 - 10	0.59 - 1.0
Business Jets	-4 - +9	-3.5 fixed	3.2 - 6.3	0 - 35	0.32 - 0.57
Regional Turbo-Props.	0 - +12	0 - 3 fixed to variable	3.4 - 7.7	0 - 35	0.39 - 1.0
Jet Transports	0 - +11	variable	3.4 - 6.1	18 - 37	0.27 - 0.62
Military Trainers	-11 - +6	0 fixed to	3.0 - 5.1	0 - 30	0.36 - 1.0
Fighters	-23 - +5	0 fixed to variable	2.3 - 5.8	0 - 55	0.16 - 1.0
Mil. Patrol. Bomb and Transports	-5 - +11	0 fixed to variable	1.3 - 6.9	5 - 35	0.31 - 0.8
Flying Boats, Amph. and Float Airplanes	0 - +25	0 fixed	2.2 - 5.1	0 - 17	0.33 - 1.0
Supersonic Cruise Airplanes	-15 - 0	0 fixed to variable	1.8 - 2.6	32 - 60	0.14 - 0.39

Figure 97: Planform design parameters for horizontal tails.

The design of the horizontal stabilizer can be obtained by considering the parameters described below:

I. Aspect ratio: The aspect ratio is the ratio of the width of the airfoil to the length of the airfoil. For the horizontal stabilizer, the aspect ratio considered is,
   $AR = 3$ 18

II. Taper ratio: The ratio of the tip chord to the root chord of the wing is termed as the taper ratio. For T-tail configurations the taper ratio should be high due to the direct clean flow observed by the horizontal tail. Hence the horizontal stabilizer taper ratio considered for the supersonic business jet is,
   $\lambda_{HT} = 0.6$ 19

III. Sweep angle: The sweep provided to the wing tips with respect to the line perpendicular to the centerline of the airplane is termed as the sweep angle. For T-tail configurations, the sweep angle should be 5 degrees more than the sweep angle obtained for the wing of the airplane.

Λ_{LE} = 56.32° 20

IV. Thickness ratio: The thickness ratio is the thickness of the airfoil used for the design of the wing. For horizontal stabilizers, the thickness ratio should be 2% less than the thickness ratio of the wing used.

$$\left(\frac{t}{c}\right)_{H} = 2\% \tag{21}$$

V. Airfoils: The airfoils used for the horizontal stabilizer depends on the thickness ratio obtained. Hence an airfoil having 2% thickness will be used on the tips and roots of the horizontal stabilizer.

VI. Incidence angle: The incidence angle is the angle the wing is attached to the body of the aircraft with respect to the free-stream direction. 0 incidence is applied to the horizontal stabilizer. This was obtained from comparing with similar aircraft, supersonic cruise airplanes do not have any incidence on the horizontal stabilizer.

VII. Dihedral angle: The dihedral angle is the angle obtained when the tips of the wing are banked at a particular angle with respect to the root of the wing. Hence for the T-tail configuration, the dihedral considered to be is 0.

7.4 DESIGN OF THE VERTICAL STABILIZER

Type	Dihedral Angle, Γ_v deg.	Incidence Angle, i_v deg.	Aspect Ratio, A_v	Sweep Angle, $\Lambda_{c/4_v}$ deg.	Taper Ratio, λ_v
Homebuilts	90	0	0.4 - 1.4	0 - 47	0.26 - 0.71
Single Engine Prop. Driven	90	0	0.9 - 2.2	12 - 42	0.32 - 0.58
Twin Engine Prop Driven	90	0	0.7 - 1.8	18 - 45	0.33 - 0.74
Agricultural	90	0	0.6 - 1.4	0 - 32	0.43 - 0.74
Business Jets	90	0	0.8 - 1.6	28 - 55	0.30 - 0.74
Regional Turbo-Props.	90	0	0.8 - 1.7	0 - 45	0.32 - 1.0
Jet Transports	90	0	0.7 - 2.0	33 - 53	0.26 - 0.73
Military Trainers	90	0	1.0 - 2.9	0 - 45	0.32 - 0.74
Fighters	75 - 90	0	0.4 - 2.0	9 - 60	0.19 - 0.57
Mil. Patrol, Bomb and Transports	90	0	0.9 - 1.9	0 - 37	0.28 - 1.0
Flying Boats, Amph. and Float Airplanes	90	0	1.2 - 2.4	0 - 32	0.37 - 1.0
Supersonic Cruise Airplanes	75 - 90	0	0.5 - 1.8	37 - 65	0.20 - 0.43

Figure 98: Planform design parameters for vertical tails.

the design of the vertical stabilizer includes the following parameters to be

I. Aspect ratio: For the vertical stabilizer in a T-tail configuration, the aspect ratio is considered to be between 0.7-1.2. Hence to provide more stability to the horizontal stabilizer attached at the tips of the vertical stabilizer, the highest aspect ratio term is considered for the design purpose.
   AR = 1.2 (22)

II. Taper ratio: The taper ratio of the vertical stabilizer is considered to be very high to accommodate the horizontal stabilizer on top of it.
   $\lambda_{VT} = 0.9$ (23)

III. Sweep angle: The sweep angle of the vertical stabilizer cannot be considered too high in the case of the T-tail configuration.
   The sweep angle at the quarter-chord of the vertical stabilizer is to be considered between a range of 37° – 65° as shown in the figure above for supersonic cruise airplanes.

The configuration being a T-tail, a lower sweep value will be considered to provider better strength to the structure of the T-tail.

$$\Lambda_{\frac{c}{4}_{VT}} = 40^{\circ} \quad (24)$$

hence, the sweep angle can be obtained from the equation,

$$tan\Lambda_{VT} = tan\Lambda_{\frac{c}{4}_{VT}} + \left(\frac{1-\lambda_{VT}}{AR(1+\lambda_{VT})}\right) = 0.8829 \quad (25)$$

$$\Lambda_{VT} = 41.44^{\circ} \quad (26)$$

IV. Thickness ratio: The thickness ratio of the vertical stabilizer is the same as the thickness ratio of the horizontal stabilizer hence the vertical stabilizer will have a thickness of 2%.

$$\left(\frac{t}{c}\right)_{VT} = 2\% \quad (27)$$

V. Airfoils: The airfoils used on the vertical stabilizer will be such that the shape can provide better strength to the horizontal stabilizer on top of it, hence symmetrical airfoils will be used having a thickness ratio of 2%.

VI. Incidence angle: The vertical stabilizer has no incidence angle so the incidence angle is 0 in this case.

VII. Dihedral angle: Being a T-tail configuration, the dihedral angle is 90° in the case of a vertical stabilizer.

7.5 EMPENNAGE DESIGN EVALUATION

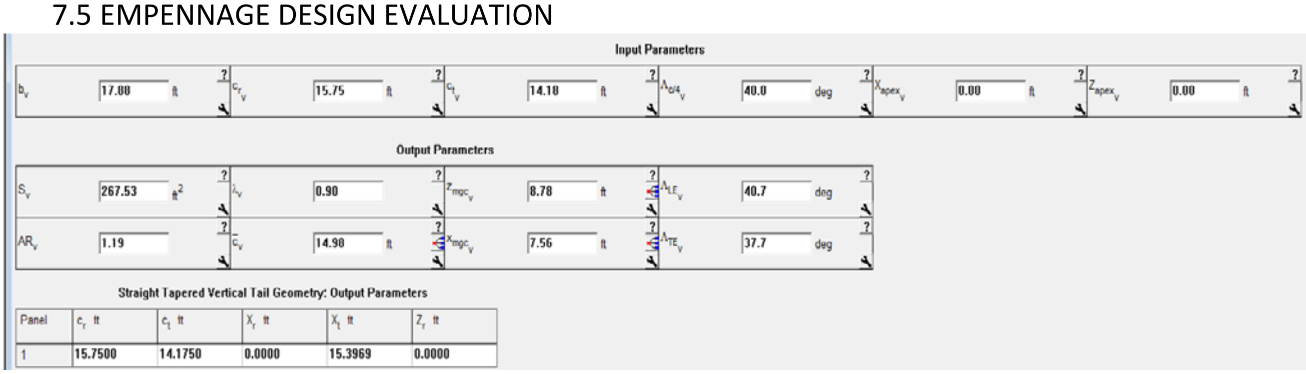


Figure 99: Parameters to Design a Vertical Stabilizer

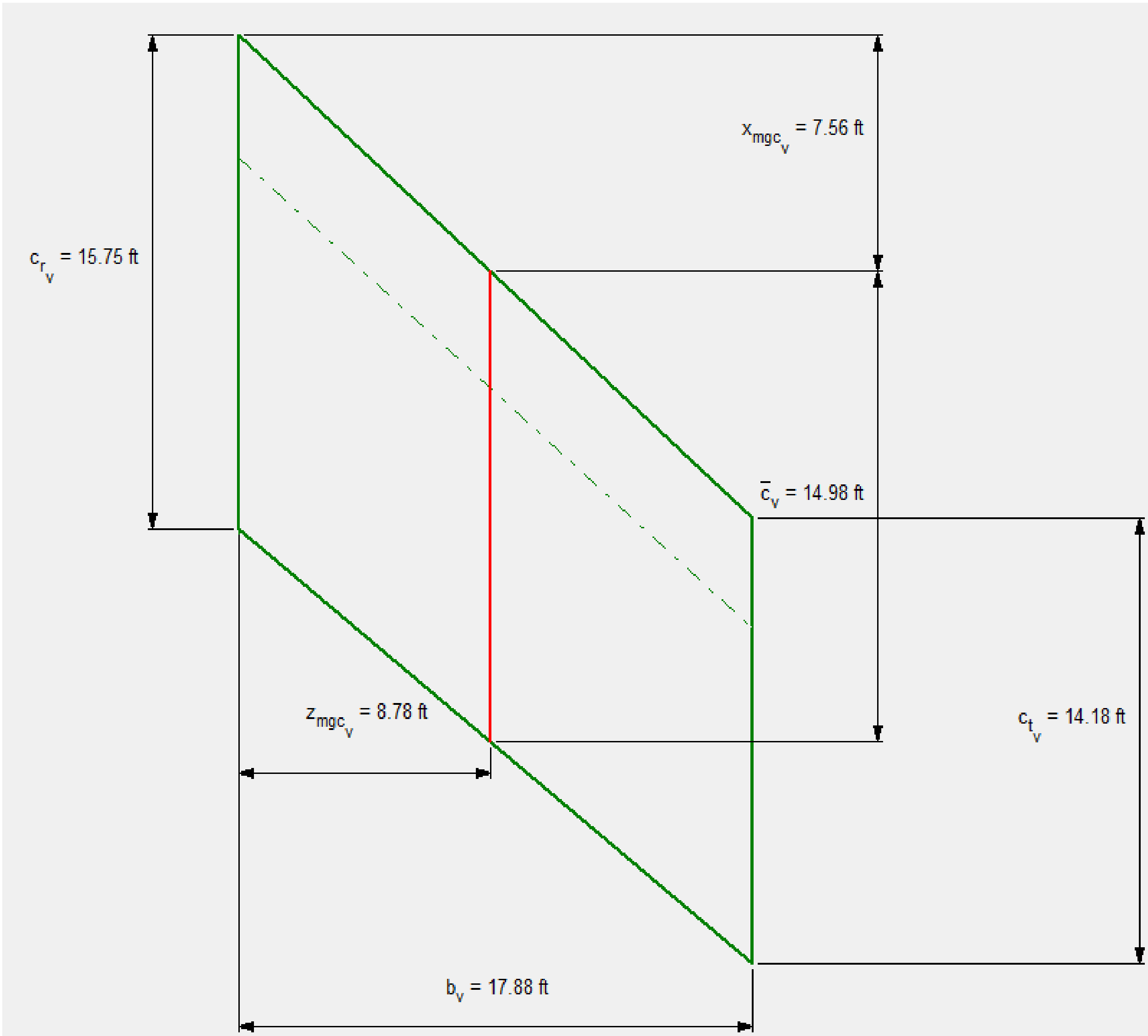


Figure 100: Vertical Tail Obtained From AAA

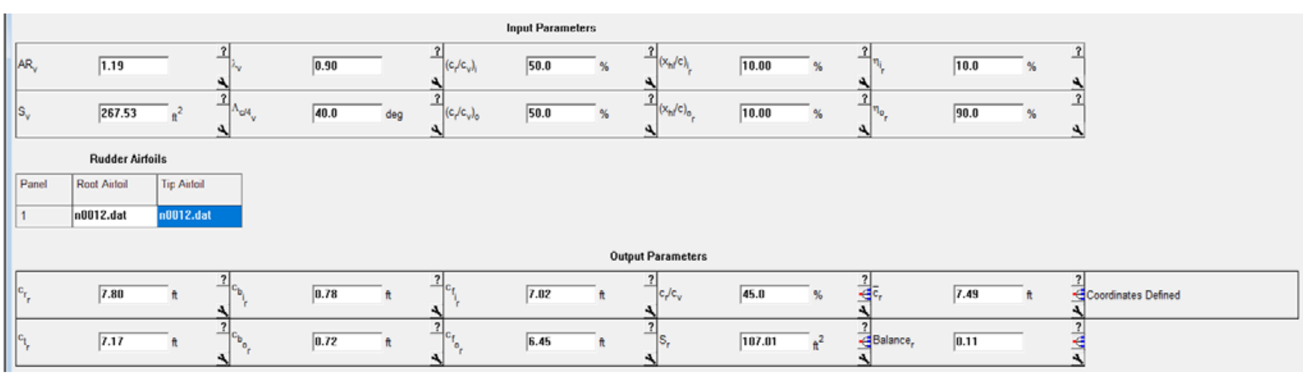


Figure 101: Rudder Sizing Parameters

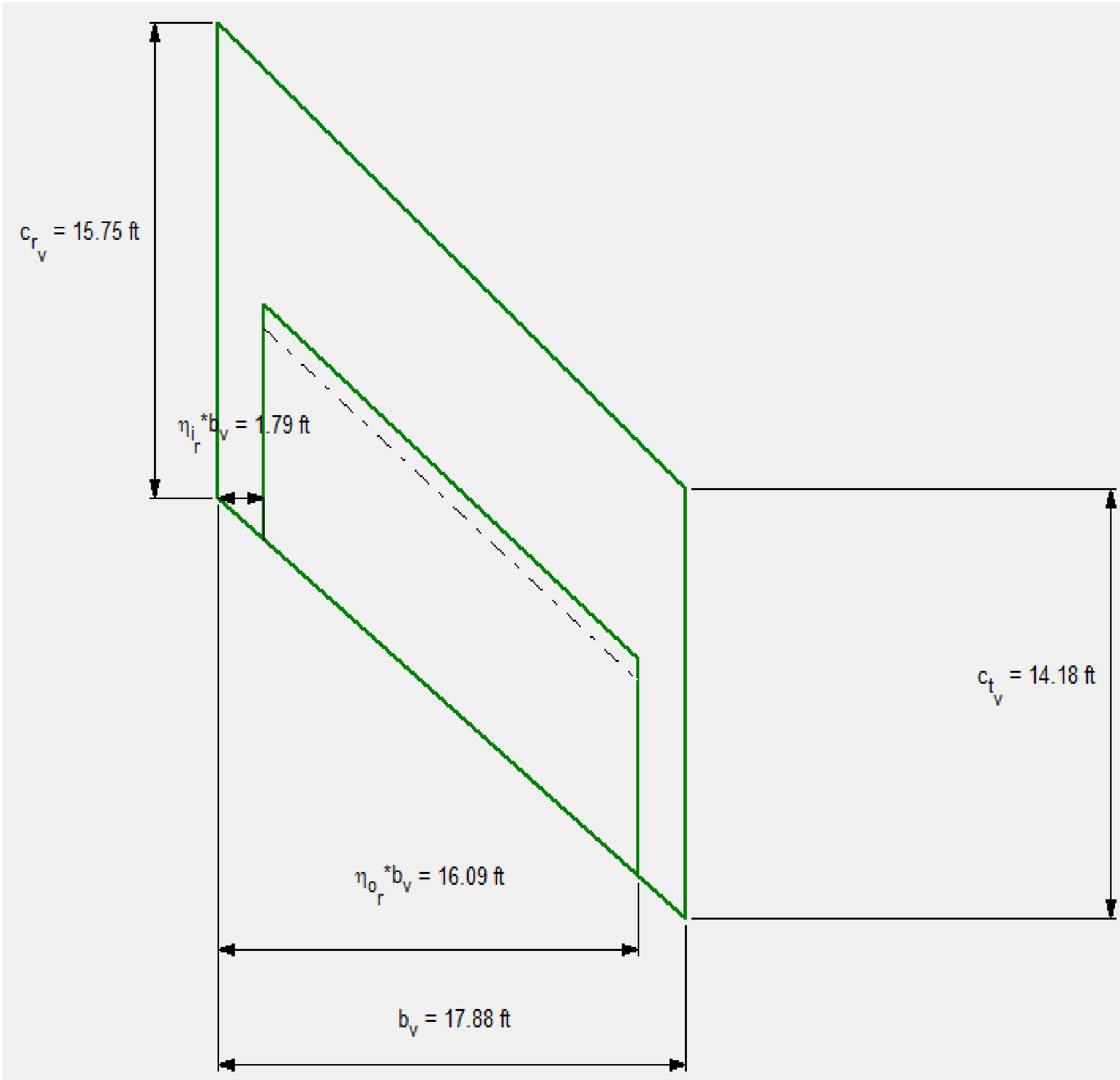
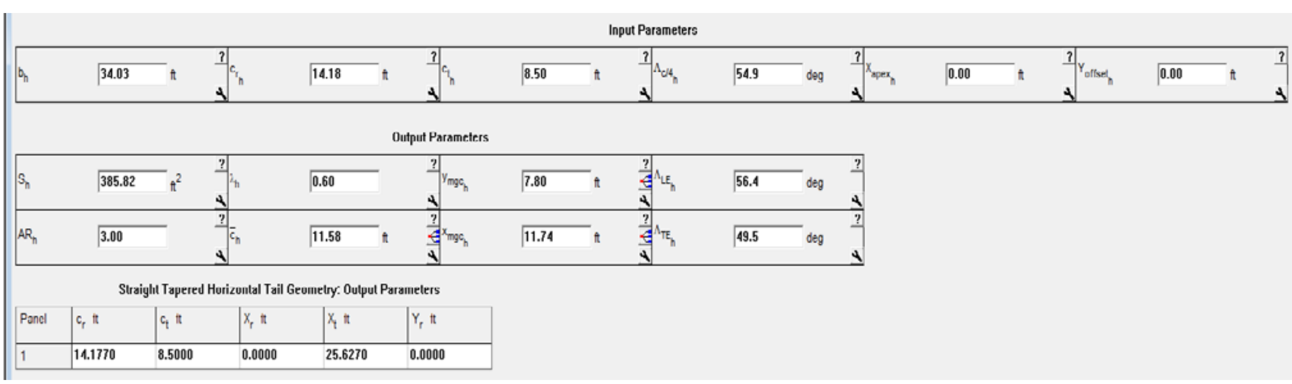


Figure 102: Rudder Location



Input Parameters

b_h	34.03	ft	c_{r_h}	14.18	ft	c_{t_h}	8.50	ft	$\Lambda_{c/4_h}$	54.9	deg	X_{apex_h}	0.00	ft	Y_{offset_h}	0.00	ft

Output Parameters

S_h	385.82	ft^2	λ_h	0.60		y_{mgc_h}	7.80	ft	Λ_{LE_h}	56.4	deg
AR_h	3.00		$\bar{c}_h$	11.58	ft	x_{mgc_h}	11.74	ft	Λ_{TE_h}	49.5	deg

Straight Tapered Horizontal Tail Geometry: Output Parameters

Panel	c_r ft	c_t ft	X_r ft	X_t ft	Y_r ft
1	14.1770	8.5000	0.0000	25.6270	0.0000

Figure 103: Horizontal Tail sizing Parameters

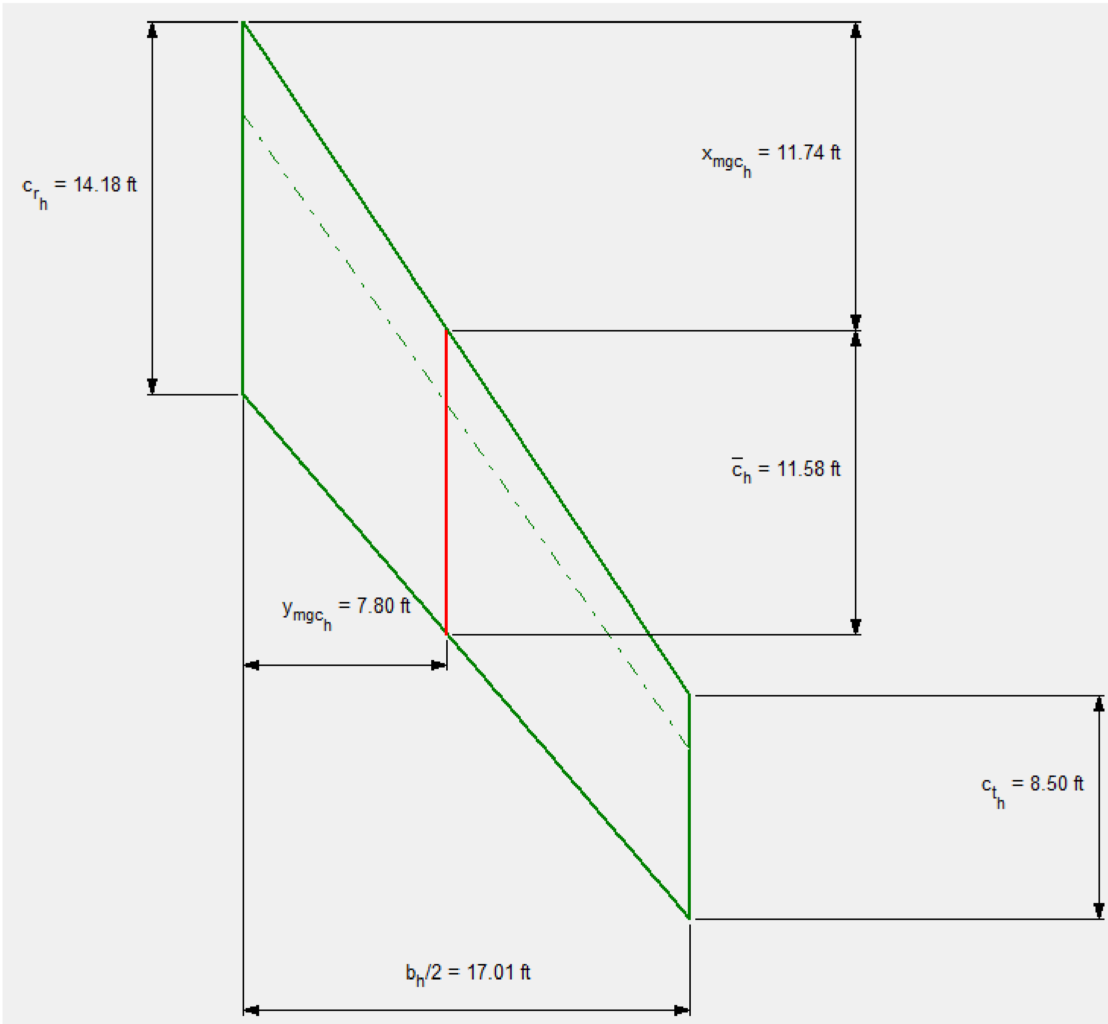


Figure 104: Horizontal Tail Obtained From AAA

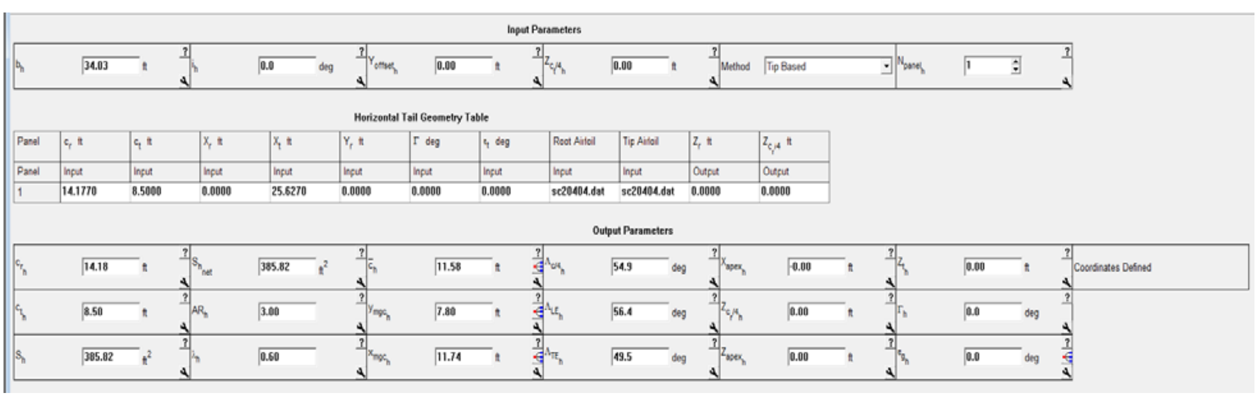


Figure 105: Sizing Parameters for Elevators

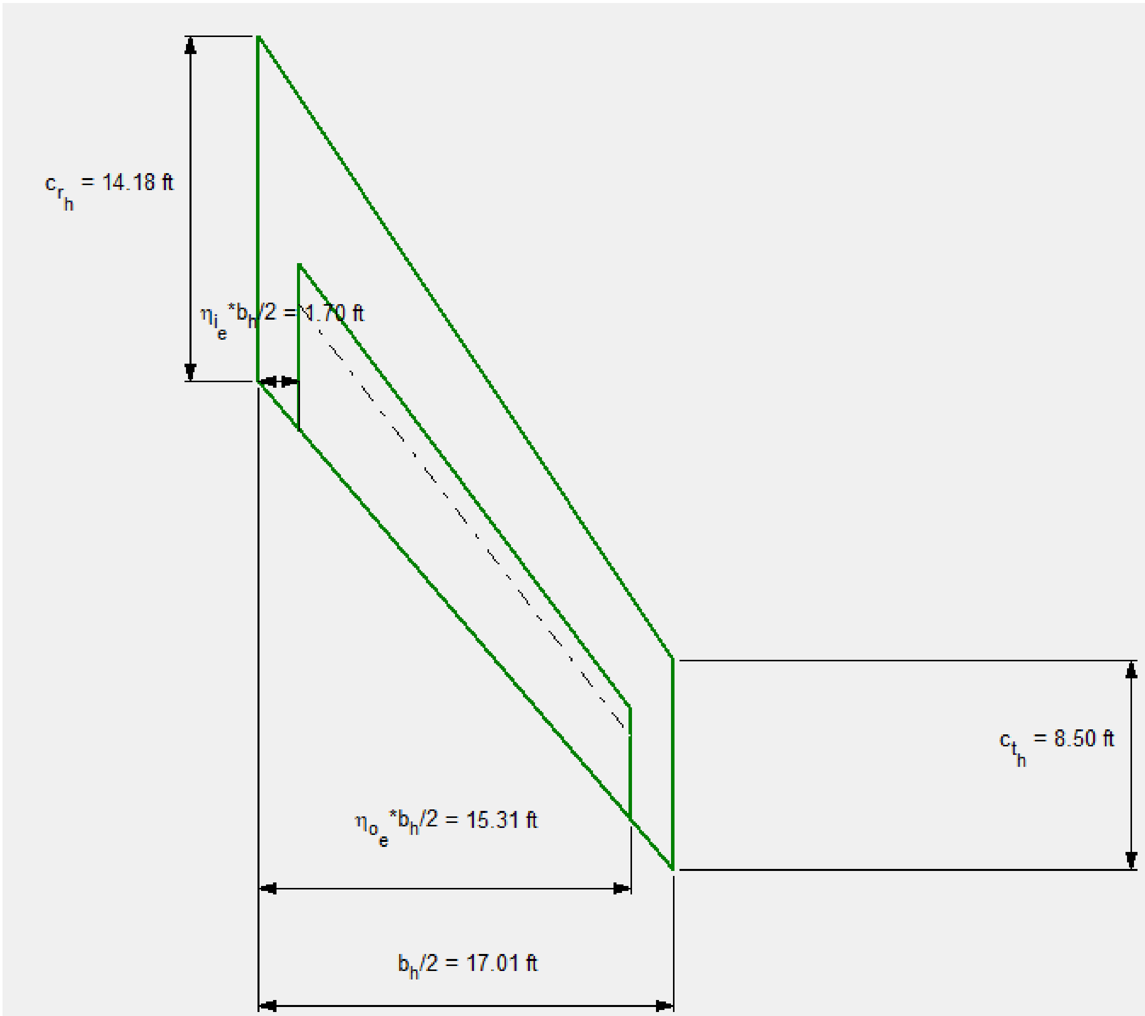


Figure 106: Elevator location and sizing obtained from AAA

7.6 DESIGN OF THE LONGITUDINAL AND DIRECTIONAL CONTROLS

The longitudinal and directional control surfaces are the elevators and the rudders respectively. The elevators and the rudders generally begin at the side of the fuselage and extend up to 90% of the tail wing span. And are typically 25-50% of the wing span. High speed aircrafts use large chord rudders that extend up to 50% of the chord.

For the supersonic business jet design, the elevators and the rudder require large area to provide better performance at supersonic speeds. So, the rudders and the elevator extend to 50% of the chord. As the design has a T-tail configuration, more strength is required at the roots of both the horizontal and vertical stabilizers hence the span of the elevators and the rudders will begin at 10% from the fuselage and extend up to 90% of the span of the horizontal and vertical stabilizer respectively.

For the vertical stabilizer,

The span of the vertical stabilizer is 17.88 ft.
The rudder will start at a distance 1.788 ft. from the root chord of the vertical stabilizer and extend up to 16.092 ft. from the foot chord. Whereas, they will start from 50% of the chord and extend till the end. The reason to start at 50% of the chord is to avoid the rudders effectiveness problems that are caused at supersonic speeds. As described in the 2D design in the CAD Drawings section

For the horizontal stabilizer,

The span of the horizontal stabilizer is 34.027 ft. the span of one side of the horizontal stabilizer is 17.0135 ft as the elevator starts at 10% distance from the root chord of the horizontal stabilizer hence they start at a distance 1.7 from the root chord and extend till 90% of the span that is 15.31215 ft from the root chord. This defines the span of the elevators while they start at 50% distance of the chord and extend till the end as described in the CAD Drawings section.

7.7 CAD DRAWINGS

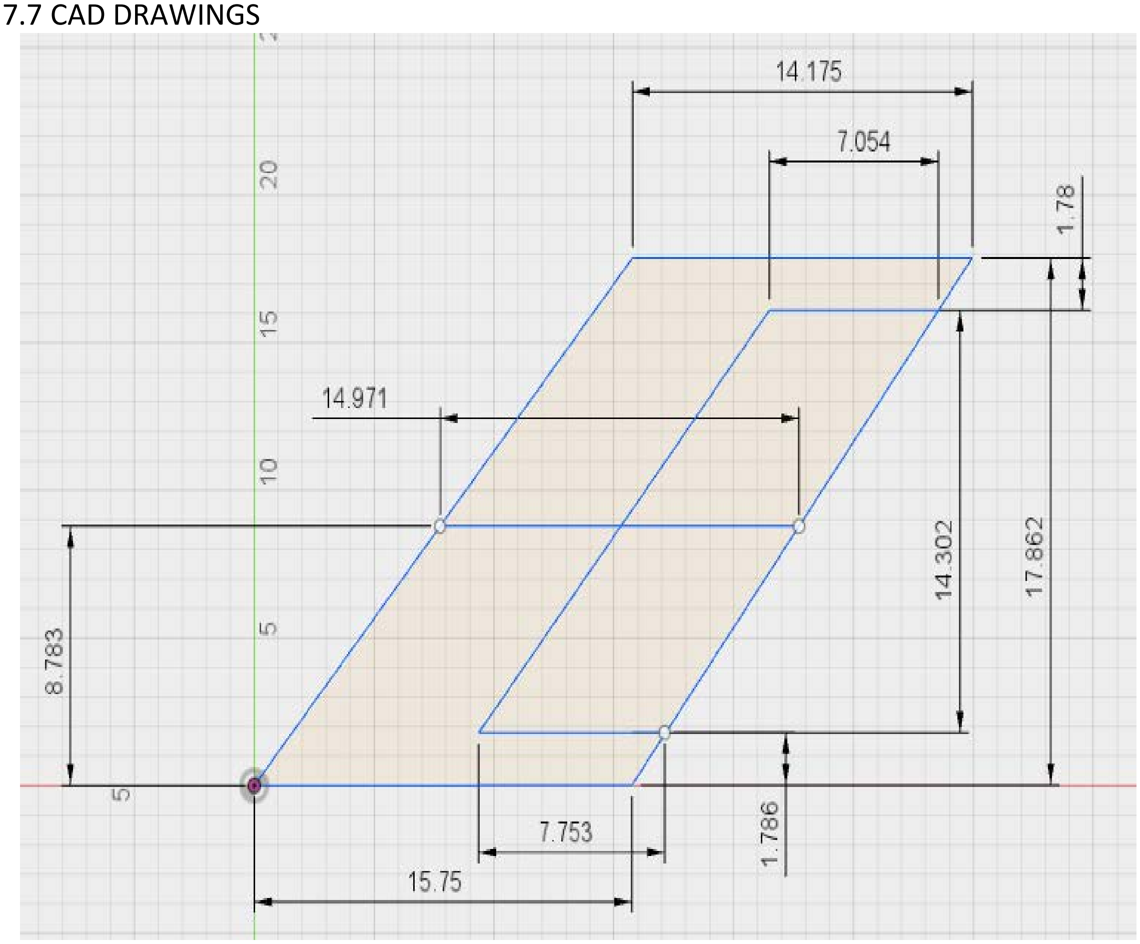


Figure 107: Rudder and vertical stabilizer dimensions (all dimensions in fts)

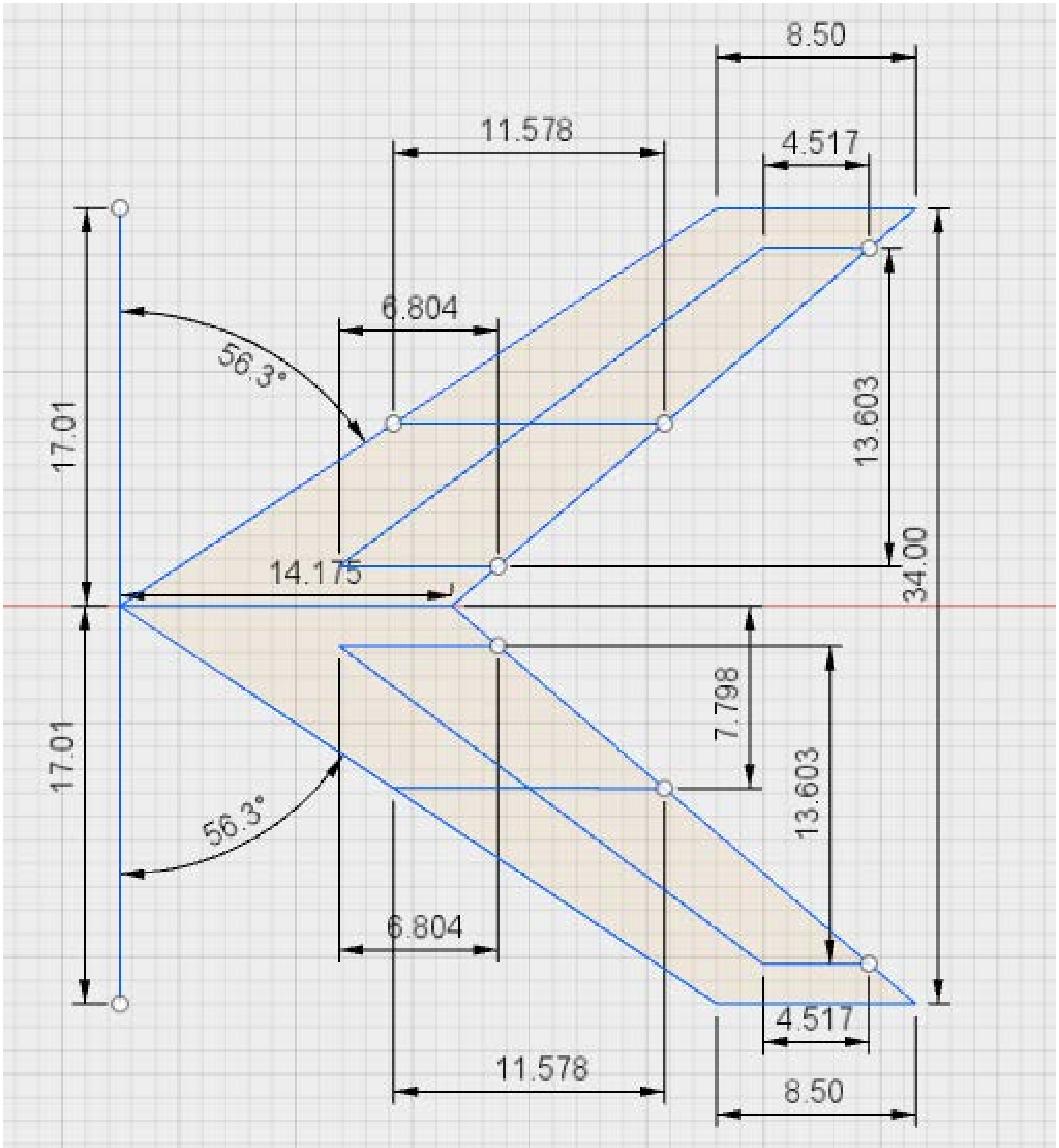


Figure 108: Elevators and horizontal stabilizer dimensions (all dimensions in fts)

7.8 DISCUSSION

This report specifies the Empennage design. The tail configuration used is a T-tail configuration. The report contains all the required data for the vertical as well as the horizontal stabilizers and the directional and longitudinal control surfaces that are the rudders and elevators.

Parameters were calculated using the equations obtained both from Roskam and Raymer. The parameters obtained are the wing areas of both the horizontal and vertical stabilizer, sweep angles, taper ratios, thickness ratios, span lengths, quarter chord sweep angles, mean aerodynamic chords, mean aerodynamic center etc.

Once the parameters are obtained, we proceed with the designing of the empennage section and the directional and longitudinal stabilizers. The data obtained using the manual calculations is then compared with the data obtained from the AAA program. The differences in the data obtained through both the process is negligible hence no changes are made in the design parameters.

After deciding the final values used, the CAD drawing of the horizontal tail, vertical tail, the rudders and the elevators is designed as observed in the section above.

7.9 CONCLUSIONS

From the results obtained from the above sections, we can conclude that no changes are required in the design of the empennage section according to class-1 Design. The data satisfies the requirements according to class-1 design. Further calculations with minor changes will be carried on after the stability and control design report.

CHAPTER 8: LANDING GEAR DESIGN, WEIGHT AND BALANCE ANALYSIS

8.1 INTRODUCTION

This is the 8^{th} report of the series of 12 for the design of the supersonic business jet. It specifies the landing gear design of the SSBJ. To design the landing gears, first a suitable configuration is to be selected. The following parameters are an important factor while designing the landing gears:

I. Number, type and the size of tires.
II. Length and the diameter of struts.
III. The landing gears preliminary disposition
IV. Retraction feasibility

Once this Is obtained. Select the landing gear system to be installed. For the supersonic business jet, as specified in the 1^{st} report, a retractable tricycle landing gear configuration will be used. Retractable landing gears will be used because at speeds higher than 150 knots, a very high drag penalty is observed in fixed landing gears, hence to avoid that drag penalty a retractable configuration is used.

While designing the landing gears, two main criterions should be satisfied. Those are
I. Tip-over Criteria.
II. Ground Clearance Criteria.

The landing gear design requires an iteration process to obtain the actual CG location of the airplane to place the landing gears to maintain the center of gravity of the airplane. The entire process is explained in the sections below.

8.2 ESTIMATION OF THE CENTER OF GRAVITY LOCATION OF THE AIRPLANE

The center of gravity depends on the weights of the components hence it is necessary to breakdown the maximum takeoff weight into parts to obtain weights of all the components. The ratio of every component was obtained from similar airplanes which is explained in the table 1 below. All the weights of the components along with the center of gravity distance is posted in the table. This was the initial step for designing the landing gear. Once the new CG is obtained, the landing gear will be re designed according to the new CG.

The Center of gravity of the major components can be obtained as explained in the figure below and the locations of CG of the supersonic business jet are plotted in the table below:

Table 14: List of CG of Major Components

Components	Equation	CG Location (from the nose)
Wing (supersonic)	0.4 * c_bar	54.869 ft
Horizontal Tail	0.3 * c_bar_h	112.166 ft
Vertical Tail	0.3 * c_bar_v	93.502 ft
Fuselage	0.5* L	49.3 ft
Nacelles	0.4 * Length of Nacelles	82.128 ft

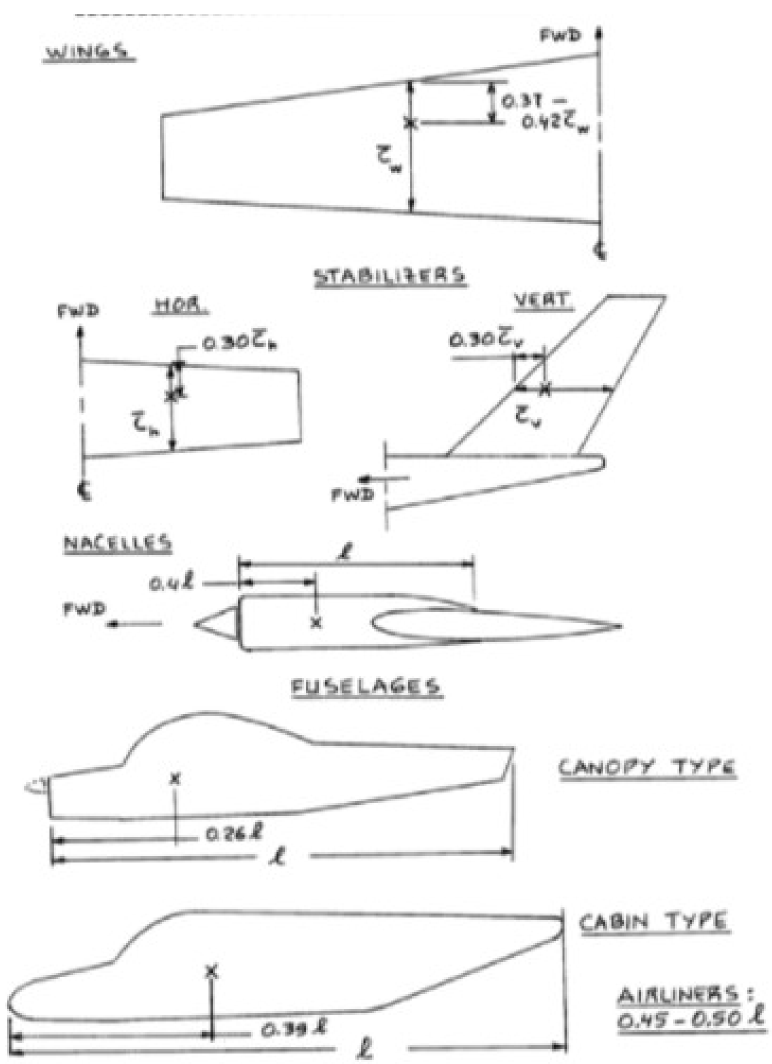


Figure 109: Locations of CG's of major components

The moment arms can be obtained by multiplying the CG distance of the component from the nose by the weight of the component. This process provides the main CG of the airplane.

Table 15: Moment arms of Different Airplane Components

	distance	WEIGHT	WiXi	Yi	WiYi
Fuselage group	50	8030	401500	0	0
wing group	54.869	13200	724270.8	0	0
empennage group	84.206	1650	138939.9	0	0
engine group	82.129	9130	749837.77	0	0
landing gear group	52.769	4180	220574.42	0	0
misc	50	9130	456500	0	0
empty weight		47960.57		0	0
trapped fuel and oil	55	1073.4266	59038.463	0	0
crew	4	700	2800	0	0
operating empty weight		47966.57		0	0
fuel	54.869	53671.33	2944892.206	0	0
passengers	50	3500	175000	0	0
baggage	50	720	36000	0	0
		110000	5909353.559	0	0

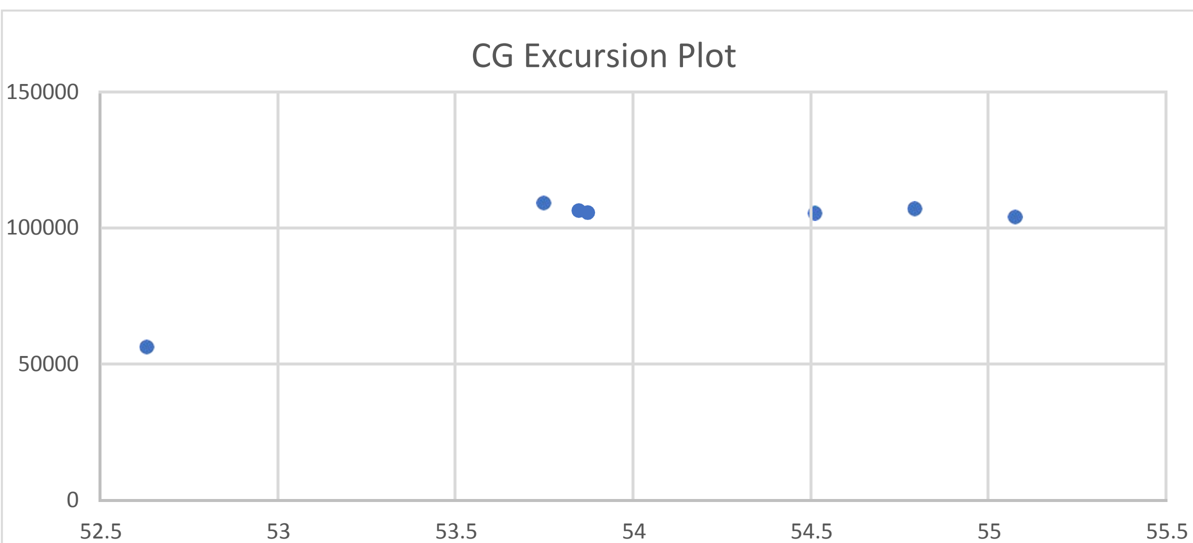


Figure 110: Initial CG Excursion Plot

According to the CG excursion plot obtained, the locations for most forward and the most aft CG's are as below:

Most Forward CG: 52.6279 ft from the nose tip of the airplane.
Most Aft CG: 55.07165 ft from the nose tip.

The CG range of the Supersonic Business Jet is 2.45 ft (29.4 in) form the allowable range of 20 to 100 ft.

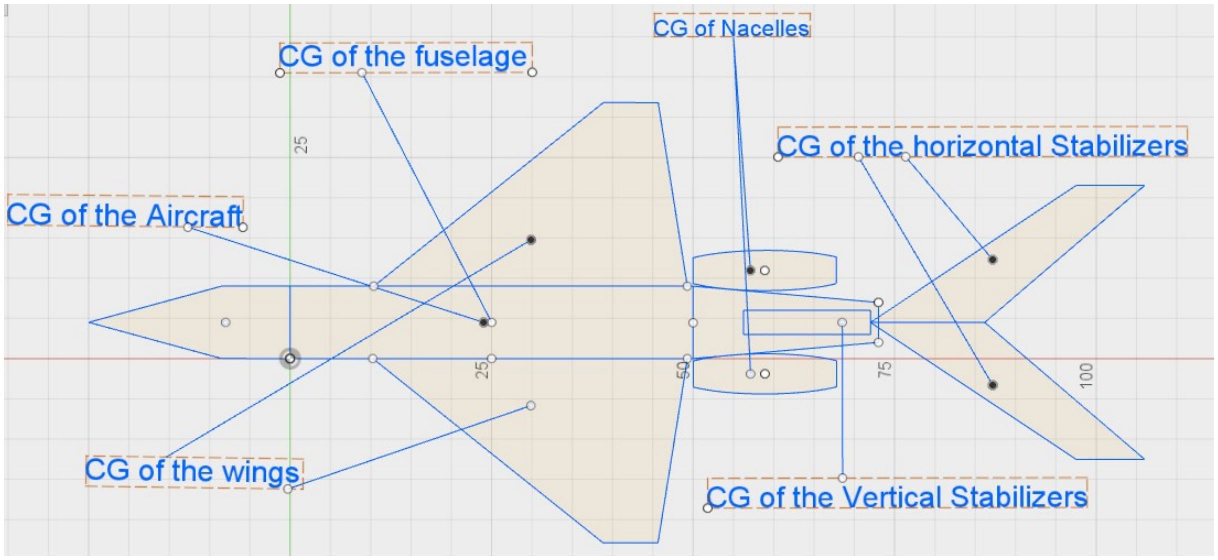


Figure 111: CG Locations of Major Components of the SSBJ

8.3 LANDING GEAR DESIGN

The landing gears for the Supersonic Business Jet has a retractable landing gear with a tricycle configuration. This is the most commonly used configuration in general aviation. This configuration provides a better inclination to the airplane due to a low wing configuration. The load of the airplane is distributed between the nose and the main landing gears. The maximum load the nose landing gear can resist is 20% of the total load where as 10% of the total load is considered to be ideal for an airplane while the remaining 90% of the load is over the main landing gears. Nose landing gear is designed over the CG os the cockpit section where as the main landing gears are designed behind the main CG of the aircraft obtained by the iteration process that will be carried in the report below.

The tip-over criteria is satisfied by placing the landing gears 7-15 degrees behind the main CG of the airplane.

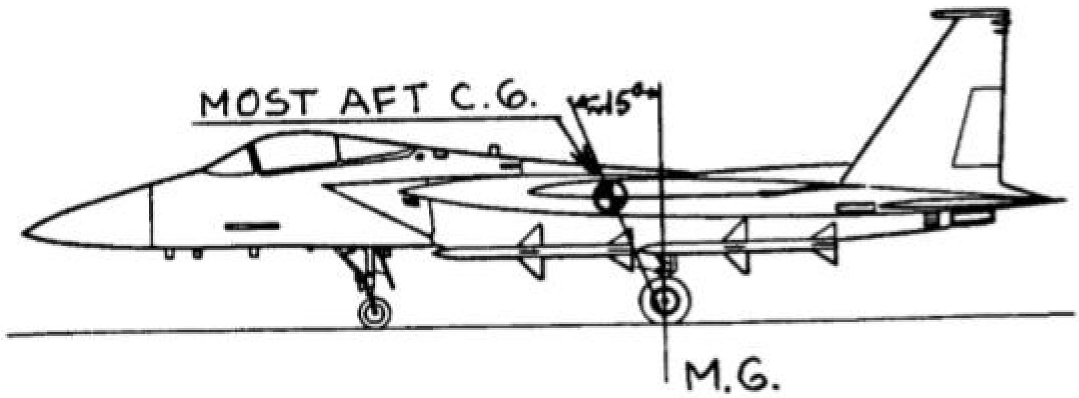


Figure 112: Longitudinal Tip over Criteria

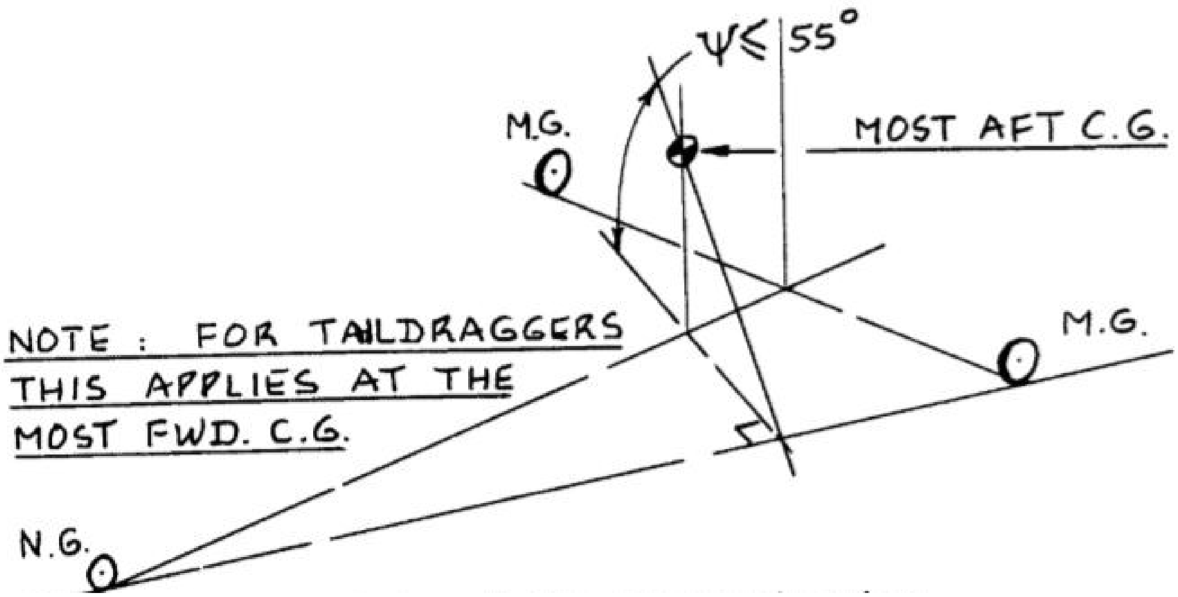


Figure 113: Lateral Tip-Over Criteria

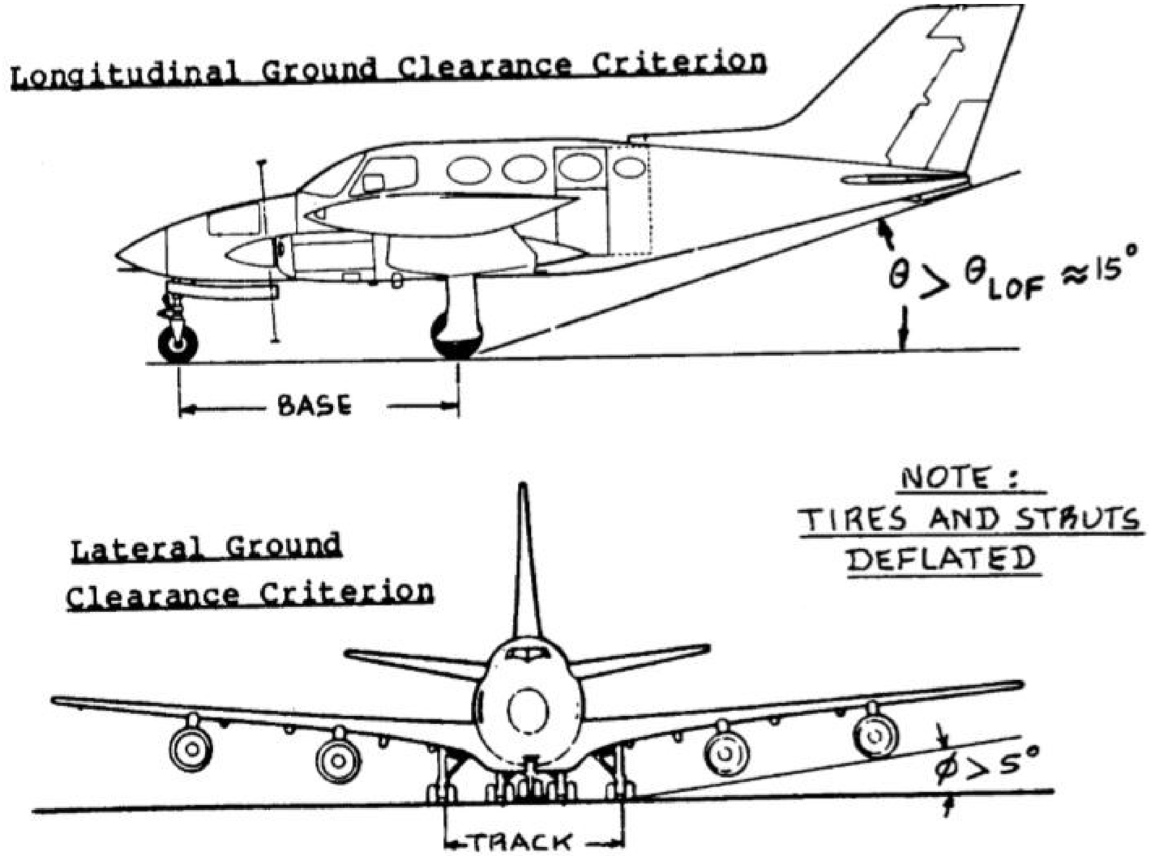


Figure 114: Longitudinal Ground Clearance Criterion

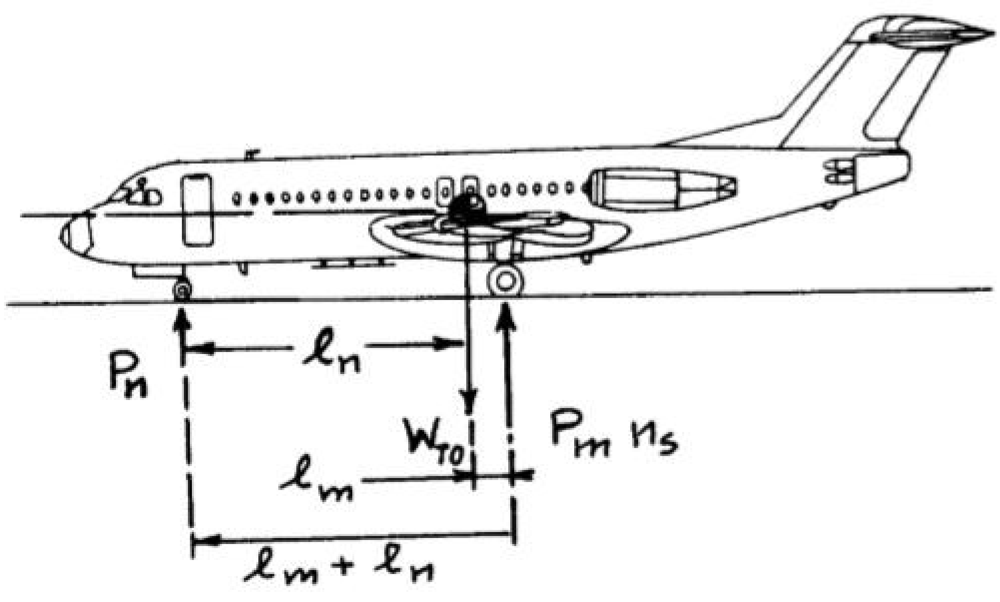


Figure 115: Geometric Wheel location Definition

The size of the wheels can be obtained from the following table:

Table 11.1 Statistical tire sizing

Main wheels diameter or width (in.) = $A\ W_W^B$

	Diameter		Width	
	A	B	A	B
General aviation	1.51	0.349	0.7150	0.312
Business twin	2.69	0.251	1.170	0.216
Transport/bomber	1.63	0.315	0.1043	0.480
Jet fighter/trainer	1.59	0.302	0.0980	0.467

W_W = Weight on Wheel

Figure 116: Statistical Tire sizing

Table 16: Main Wheel Dimensions

Wheel Parameters	Dimensions
No of wheels	4
diameter	40 inches
Width	14 inches
Rolling radius	16.5 inches
pressure	129.92 psi
Weight on Wheel	24750 lbs
Load Distributed	0.9 (90% of total weight)

Table 17: Nose Wheel Dimensions

Wheel Parameters	Dimensions
No of wheels	2
diameter	24 in
Width	5.5 in
Rolling radius	10.6 in
pressure	148.67 psi
Weight on Wheel	5500 lbs
Load Distributed	0.1 (10% of total weight)

Nose wheel strut: $P_0 = \frac{\text{T–\`{}*Ÿ,,}}{\text{Ÿ,,‡Ÿ7}} = 101065.33 \text{ lbs}$

Main gear strut: $P_9 = \frac{\text{T–\`{}*Ÿ7}}{\text{O\~{}9Ì/2 ë" .G2\~{}G*(Ÿ,,‡Ÿ7)}} = 48914 \text{ lbs}$

The tire contact area can be obtained from the following equations:

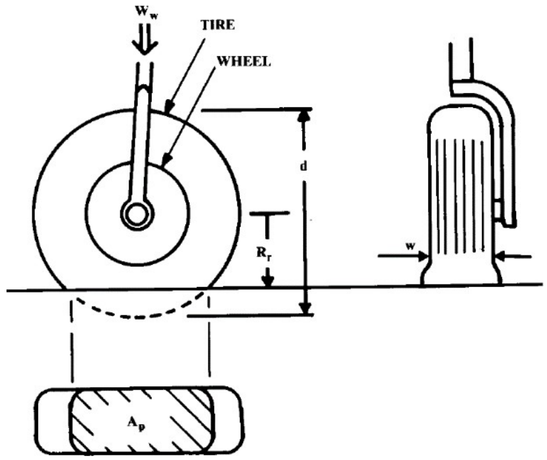


Figure 117: Tire Contact Area Definition

Main Wheel:

$W_4 = PA_* = 24750$ lbs

$A_* = 2.3\sqrt{wd}\, \mathrm{F}\frac{\mathrm{B}}{\mathrm{m}} - R_2\mathrm{H} = 190.5\ in^{\mathrm{m}}$

Nose Wheel:

$W_4 = PA_* = 5500$ lbs

$A_* = 2.3\sqrt{wd}\, \mathrm{F}\frac{\mathrm{B}}{\mathrm{m}} - R_2\mathrm{H} = 36.994\ in^{\mathrm{m}}$

The placements of the landing gears are shown in the figure 10 below. As seen the configuration satisfies the tip over and the ground clearance criterion.

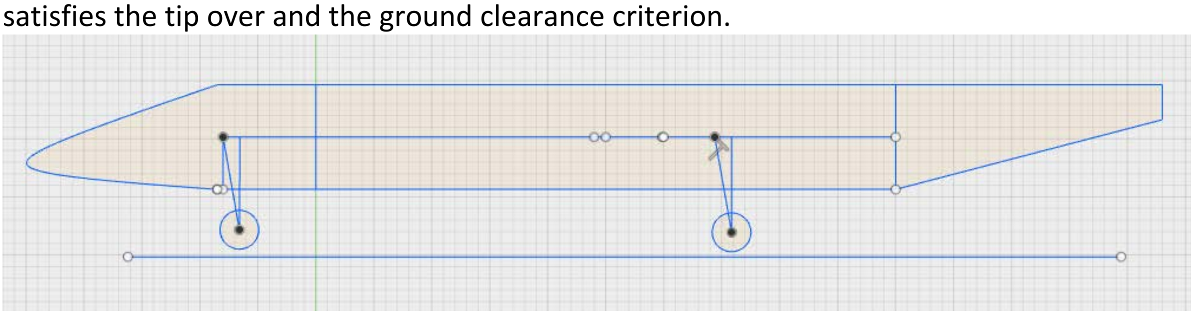


Figure 118: Side view of the SSBJ showing the landing gear Placements

8.4 WEIGHT AND BALANCE

The parameters used to calculate the weight balance and decide the center of gravity of the airplane are:

Table 18: Parameters Deciding the CG of the SSBJ

half passengers	105521.3266	5761478.637
half luggage	107265.3266	5886578.637
0 passegers+full luggage	104125.3266	5743978.637
empty	105780	5707978.637
TFO + passengers + luggage	56328.67	2974086.431
passengers only	109280	5882978.637
cargo only	106500	5743978.637

Table 19: Final Loading Scenario according to new CG

	distance	WEIGHT	WiXi	Yi	WiYi
Fuselage group	50	8030	401500	0	0
wing group	54.869	13200	724270.8	0	0
empennage group	84.206	1650	138939.9	0	0
engine group	82.129	9130	749837.77	0	0
landing gear group	55.0716502	4180	230199.4978	0	0
misc	50	9130	456500	0	0
empty weight		47960.57		0	0
trapped fuel and oil	55	1073.4266	59038.463	0	0
crew	4	700	2800	0	0
operating empty weight		47966.57		0	0
fuel	54.869	53671.33	2944892.206	0	0
passengers	50	3500	175000	0	0
baggage	50	720	36000	0	0
		110000	5918978.637	0	0

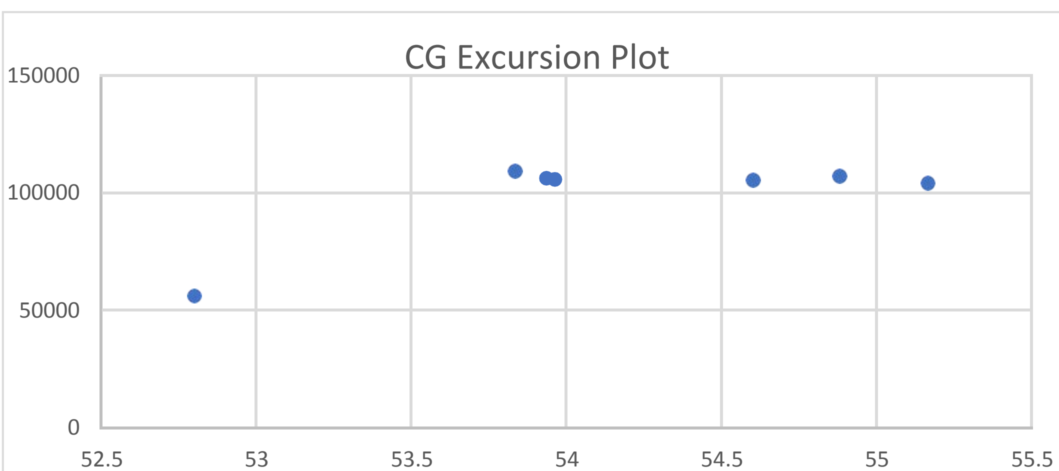


Figure 119: Final CG Excursion Plot

After the iterative process, the CG's of the airplane obtained are as shown in the excursion plot above:
Most Forward CG: 52.79 ft from the nose tip
Most Aft CG: 55.16 ft from the nose tip

Which provides a range of 2.37 ft(28.44 inches which is an acceptable value from the provided range of 20-100 inches for supersonic airplanes.

8.5 DISCUSSION

The report explains the locations of the Center of gravity of various components, of the airplane, the weights of various sections of the airplane and the location of the landing gears. The landing gear configuration considered for the supersonic business jet is the retractable tri-cycle landing gear configuration. To locate the CG of an entire airplane, an iterative process is to be worked upon considering the weight and distance from the nose tip to obtain a range of CG's from which a final CG of the airplane will be decided upon. Once the CG is obtained, the landing gears will be designed behind the most aft CG obtained to maintain the balance of the airplane and satisfy the tip over and the ground clearance criterion. The loads acting on each tire is also calculated from the equations as mentioned in the main landing gear and the nose landing gear wheel dimensions' table. The landing gear data is then calculated using the equations and tables obtained from Raymer and Roskam. Using those dimensions obtained, the CAD is designed as observed in the sections above.

8.6 CONCLUSION

The CG range obtained is 28.7 inches which is within the allowable range of 20 – 100 inches for supersonic airplanes. Hence it can be said that the Airplane is much stable with such a low variation in the range of the CG of the airplane for future work more number of iterations will be worked on to obtain the most exact CG and place the landing gears accordingly to obtain accurate results while designing the Supersonic Business Jet.

CHAPTER 9: STABILITY AND CONTROL ANALYSIS

9.1 INTRODUCTION

This is the 9th report of the preliminary airplane design. The report specifies the stability and control analysis of the Supersonic Business Jet. The design process will be carried following the steps provided by Roskam in the book Airplane design part II satisfying the class 1 design requirements. The longitudinal and directional stabilities will be calculated in the report. There are two types of stability:

i. Static Stability
ii. Dynamic stability

The static stability deals with the initial tendency of the object to return to the equilibrium position after being disturbed whereas the dynamic stability deals with the time history of the vehicles motion after it initially responds to its own static stability.

A dynamically stable airplane should always be statically stable hence static stability is not sufficient to ensure dynamic stability.

Control means the study of deflections of the high lift devices and the directional and longitudinal devices necessary to make the aircraft under a controlled situation and obtain the desired output as required by the pilot.

This chapter also explains the x-plots to determine and changes in the horizontal area with respect to the center of gravity and the aerodynamic centers and also changes in the vertical tail area with respect to the side-slip. Iteration process are necessary in this process to determine the exact values to be used while designing the landing gears and the empennage section of the airplane.

9.2 STATIC LONGITUDINAL STABILITY

This section defines the static longitudinal stability. To obtain the directional x-plot, the required parameters are the aerodynamic center of the airplane and the center of gravity of the airplane which are directly proportional to the change in the horizontal tail area. The aerodynamic center is obtained from the following equation,

$x_{-3æ} =$ ‾‰‹¸ ‡ ûö M^ - ù∈ - PM » - PÂ‰‹ / ‰ - ù » - / ûö‰,Í / ø

Where,

$$F = 1 + \frac{\text{Nö}_{‰}\text{-}\ \overset{\text{F}}{\text{Z†}}\ \frac{\text{ù∈-}}{\text{ù}_{\text{,}}}\text{HF}\ \frac{\text{»-}}{\text{»}}\text{H}}{\text{Nö}_{‰,\text{í}}}$$

whereas the CG of the aircraft is obtained by varying the area of the horizontal tail with respect to the change in its weight. On comparing the aerodynamic center as a function of the horizontal tail area and the center of gravity with respect to the weight of the horizontal tail area, the graph obtained is termed as the longitudinal X-plot. The area of the horizontal tail can be obtained at the point where there is a variation of 10% in the graph after the interaction of the points.

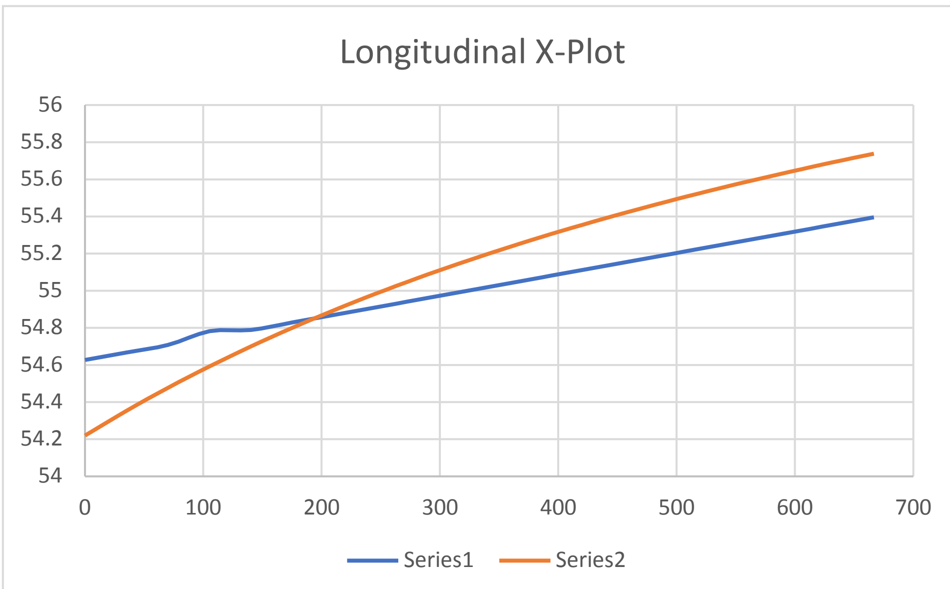


Figure 120: Longitudinal X-Plot

The area of the horizontal stabilizer for the Supersonic Business Jet is 385.94 ft^2. As observed from the X-plot the 10% variation is obtained around 380 ft^2. Hence there is not much change in the area of the horizontal tail which concludes that the aircraft is longitudinally stable according to class-1 stability requirements and the horizontal tail area will be maintained at 385.94 ft^2.

9.3 STATIC DIRECTIONAL STABILITY

This section defines the Static directional stability of the aircraft. For the static directional stability, the vertical tail area is directly proportional to the side-slip feedback system. For the preliminary design purpose, the side-slip of the wing is considered to be equal to zero whereas the side-slip of the fuselage and the vertical tail can be obtained from the equations described below.

$C_{0_{\text{‹}}} = C_{0_{\text{‹¸}}} + C_{0_{\text{‹î}}} + C_{0_{\text{‹˘}}}$

Where,

$C_{0_{\text{‹¸}}} = 0 \; (\,For\ priliminary\ Sizing)$

$C_{0_{\text{‹î}}} = -57.3 K_{\text{õ}} K_{\text{··}_{\text{Y}}} \text{F}\frac{\text{æî}_{\$}{}^{\text{Y}}\text{î}}{\text{æì}}\text{H}$

$C_{0_{\text{‹˘}}} = -\text{F}C_{\text{''}_{\text{‹˘}}}\text{H}\,\text{F}\frac{\text{Y˘\_\#¶\$(\%)±\&˘\_\$'(}(\%)}{\text{ì}}\text{H}$

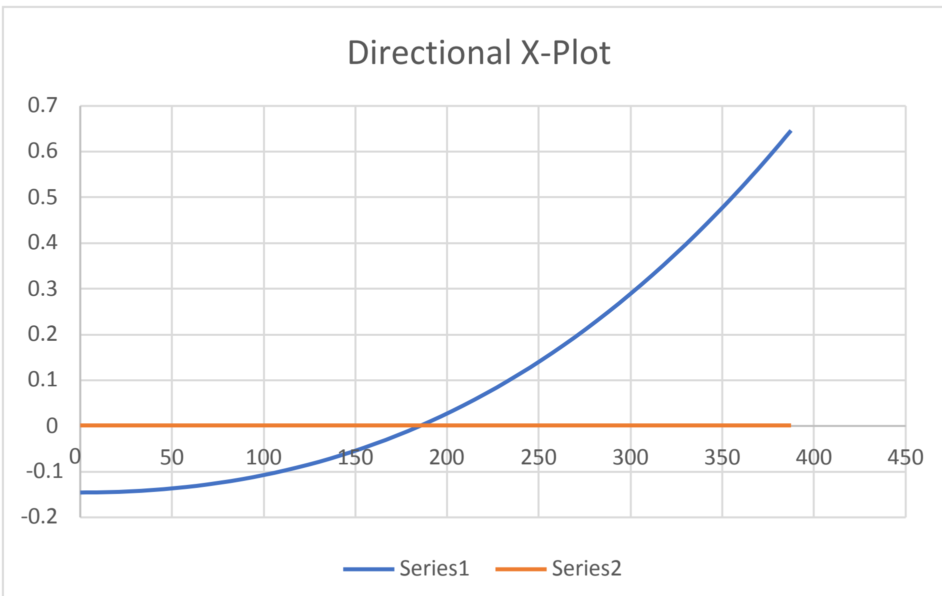


Figure 121: Directional X-Plot

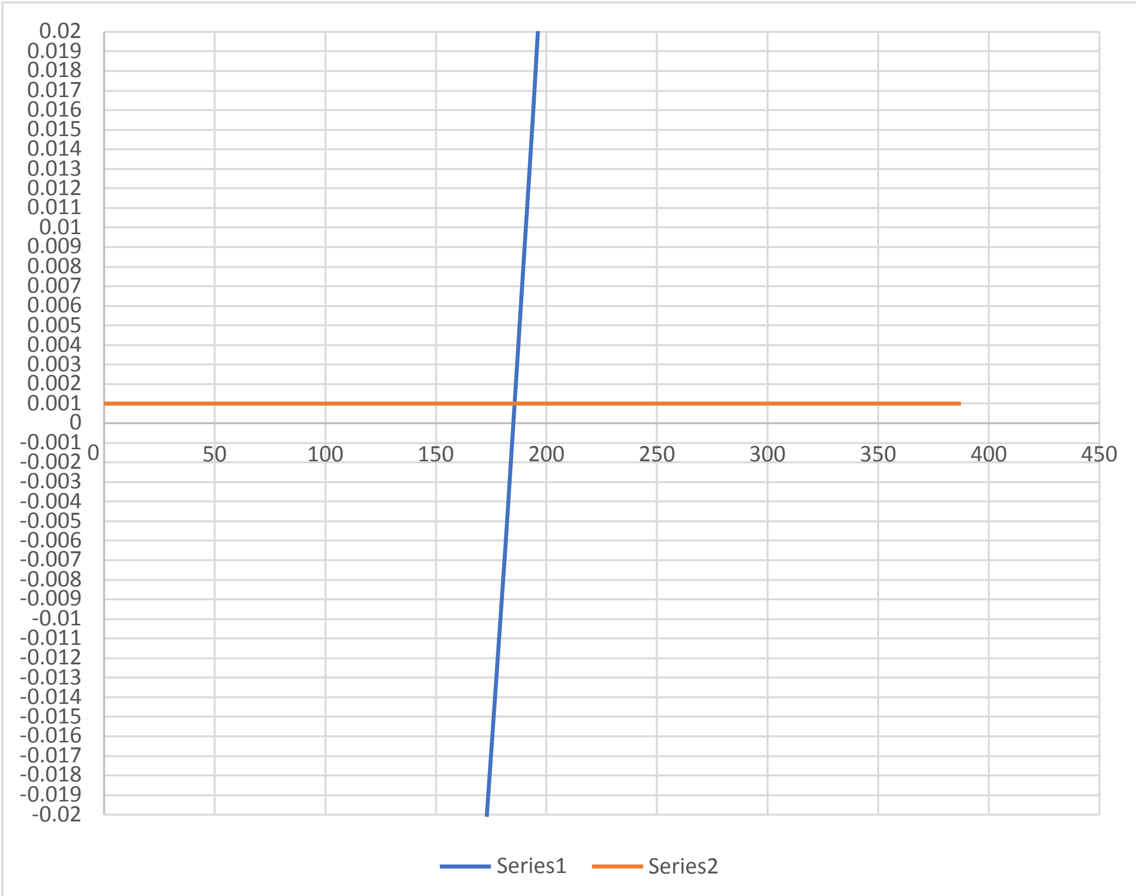


Figure 122: Zoomed view of the directional X-plot

At side-slip $C_{0_)} = 0.001$ the area of the vertical stabilizer obtained is 190 ft^2 which is slightly less than the area used to design the vertical stabilizer hence that wouldn't make much of a difference on the design of the supersonic business jet. Which concludes that this much difference is slightly negligible in the preliminary design process hence according to class 1 design process, the current design is directionally stable. The deflection obtained on the vertical is

$$k = -\frac{\text{Z.oèpoZqo}}{\text{†Z.oèmoZqo}} = 1\ deg/deg$$

$k_) = 1.$

From the design of the SSBJ, the y_t that's the distance of the thrust line from the centerline of the aircraft is 6.351 ft. the maximum thrust was defined in report number 4 which is 39600 lbs per engine.

The critical engine out yawing-moment is therefore 39600 * 6.351 = 251499.6 ft.lbs.
The total yawing moment of the SSBJ to be held at N_D is therefore 1.25 * 251499.6 = 62874.9 ft.lbs

The landing stall speed which is the lowest speed for the SSBJ is 190 knots hence foe one engine out requirements to satisfy, the minimum required speed Is 1.2* 190 = 228 knots.

From the vertical tail and the rudder geometry, the value for rudder control power derivative is computed to:

$C_{0_{\hat{\text{E}}\,2}} = -1.6436196 \text{ deg}^{-1}$

this yields a rudder deflection of 1 degree at the required V_{mc} as explained above. Which is acceptable hence the vertical is not critical from a viewpoint of engine-out control.

9.4 EMPENNAGE DESIGN-WEIGHT & BALANCE – LANDING GEAR DESIGN – LONGITUDINAL STABILITY AND CONTROL CHECK

As the longitudinal and directional stability conditions are satisfied as proved in the above sections, no change in the horizontal or the vertical stabilizer is to be made. Hence no iterative process is required for the class 1 design process.

9.5 DISCUSSION

In this report, the static longitudinal and the static directional stability is discussed as well as one engine stall speed requirements to satisfy the stability and control of the SSBJ. To obtain the static longitudinal stability, the X-plot was generated using the center of gravity of the airplane with respect to different horizontal tail weights and respective areas and comparing it with the plot of the aerodynamic center of the airplane obtained from the different weights and horizontal tail area which proves that the SSBJ is statically longitudinally stable according to class 1 preliminary design as is satisfies all the requirements with in the provides range.

Next the static directional stability was calculated using the sideslip with respect to the vertical tail areas with their respective weights. This condition is satisfied obtaining the required area of the vertical tail which makes it longitudinally stable according to the preliminary design requirements providing a deflection angle of 1 degree from the maximum allowable 5 degrees. Once the stability is satisfied, the one engine out requirements were calculated to obtained the minimum requirements to satisfy the controls requirements.

9.6 CONCLUSION

After calculating the static longitudinal and static directional X-plots, it can be concluded that the Supersonic Business Jet is both longitudinally and directionally stable according to class 1 requirements. For future work, the areas of the horizontal and vertical tails will be sized accurately to overcome the minor changes in the areas.

CHAPTER 10: DRAG POLAR ESTIMATION

10.1 INTRODUCTION

This is the 10th report of the Preliminary Aircraft Design. The drag polars of the supersonic business jet were already calculated in the performance sizing report that is report number 4. Those results will be used to compare the final results after all the changes made in the airplane after all the iteration process considering the CG locations and the required parameters. This report specifies in the drag polar estimation of the airplane. Drag is produced by every component of the airplane. But this report specifies the drag produced due to the use of high lift devices and landing gears while take-off and landing.

Then the AAA program will be used to compare the results with the manual calculations and obtain the final data required.

10.2 AIRPLANE ZERO LIFT DRAG

The total wetted area of the airplane can be calculated by adding up the wetted area of different parts of the airplane. The wetted area of different parts can be calculated using the equations provided in the book by roskam. The components that contribute to the wetted area of the airplane are:

i) Fuselage
ii) Wing(s)
iii) Empennage
iv) Nacelles

The wetted area for these components can be calculated using the equations as explained below:

Calculating the Wetted Area of the Wing:

$$S_{wet,plf} = 2 * S_{exp,plf} * 1 + \frac{0.25 (t/c)_r (1+\tau\lambda)}{1+\lambda} +$$

where, $\tau = \frac{(t/c)_t}{(t/c)_r}$ $\quad and\ \lambda = \frac{c_t}{c_r}$

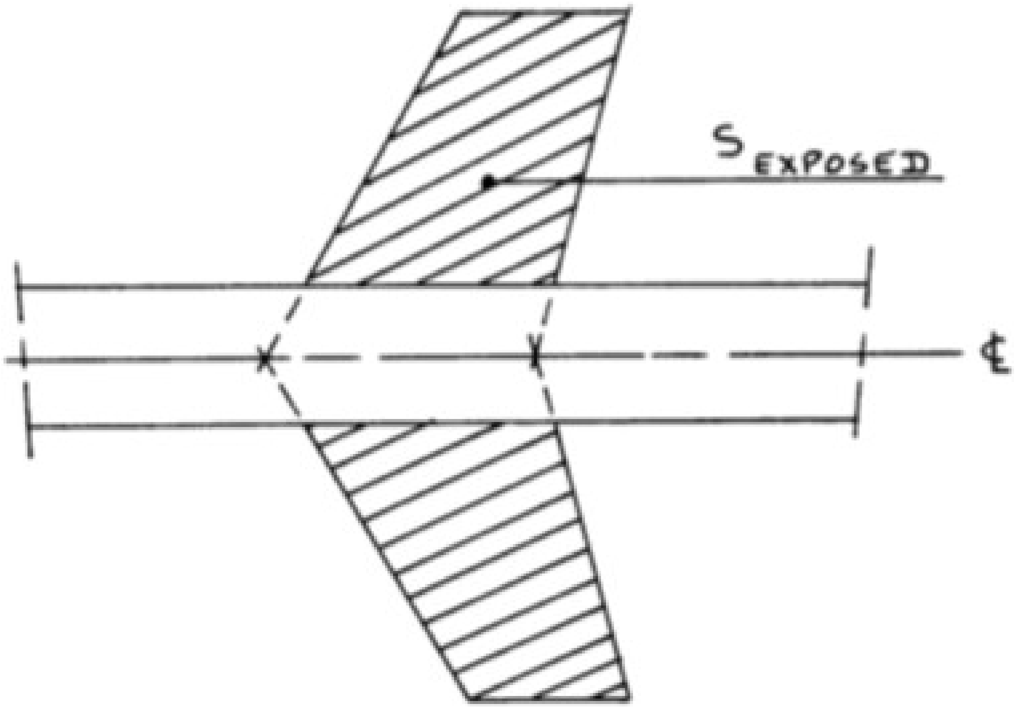


Figure 123: Definition of the Exposed Planform

Table 20: Wing Wetted Area

WING Parameters	**Dimensions**
Total Area	1424.34
Exposed Area	1041.996
Span	54.69
root chord	45.29
tip chord	6.974
root thickness ratio	0.04
tip thickness ratio	0.04
Taper ratio	0.153985427
tau	1
Wetted Area	**2104.83192**
Airfoil Area incident on the fuselage	57.155

The wetted area of the wing can be calculated using the parameters mentioned in the table 1 above.

Similarly the wetted area for the horizontal and Vertical tails can be calculated using the same equations but with parameters as described in the tables below,

Table 21: Vertical Tail Wetted Area

VERTICAL TAIL	
Total Area	266.4
Exposed Area	197.54
span	17.88
root chord	15.75
tip chord	14.175
root thickness ratio	0.12
tip thickness ratio	0.12
Taper ratio	0.9
tau	1
Wetted Area	**406.9324**
Airfoil (Fuselage)	20.02
Airfoil (Horizonatal)	16.216

Table 22: Horizontal Tail Wetted Area

HORIZONTAL TAIL	
Total Area	385.94
Exposed Area	385.94
span	34.027
root chord	14.177
tip chord	8.5
root thickness ratio	0.06
tip thickness ratio	0.06
Taper ratio	0.599562672
tau	1
Wetted Area	**783.4582**
Actual Wetted Area	767.2422

To obtain the wetted area of the Fuselage, the equation and parameters used are as follows:

For Fuselage's with cylindrical Mid-sections,

$$S_{wet_{fus}} = \pi * D_f * l_f \left(1 - \frac{2}{\lambda_f}\right)^{\frac{2}{3}} \left(1 + \frac{1}{\lambda_f^2}\right)$$

where, $\lambda_f = \frac{l_f}{D_f}$

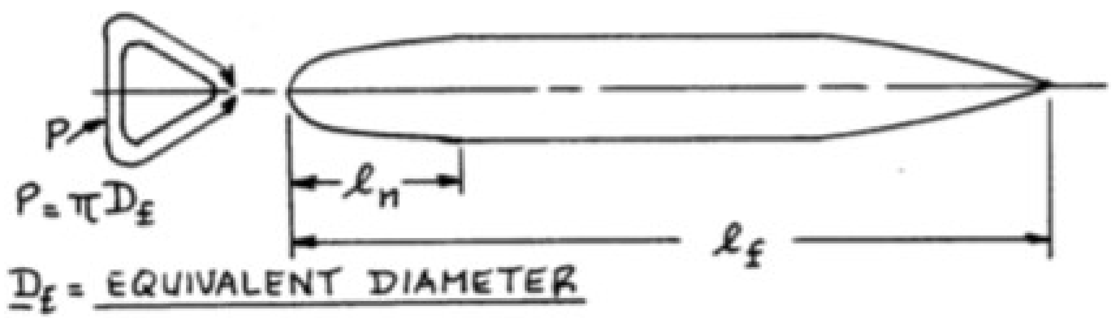


Figure 124: Fuselage Wetted Area Parameters

Table 23: Fuselage Wetted area Parameters

FUSELAGE (CYLINDRICAL MID-SECTIONS)	
pi	3.14
fuselage diameter	9.25
fuselage length	98
fineness ratio	10.59459459
Wetted Area	**2497.910319**
Actual Wetted Area	2204.678319

The Wetted Area of the Externally Mounted Nacelles Can be obtained From the Equation below:

$$S_{wet_{fan\,cowl}} = l_n D_n \left(2 + 0.35 \left(\frac{l_1}{l_n}\right) + 0.8 \left(\frac{l_1 D_{hl}}{l_n D_n}\right) + 1.15 \left(1 - \frac{l_1}{l_n}\right) \left(\frac{D_{ef}}{D_n}\right)\right)$$

$$S_{wet_{gas\,gen}} = \pi * l_g * D_g \left\{1 - \left(\frac{1}{3}\right)\left(1 - \frac{D_{eg}}{D_g}\right)\left[1 - 0.18\left(\frac{D_g}{l_g}\right)^{\frac{5}{3}}\right]\right\}$$

$$S_{wet_{plug}} = 0.7 * \pi * l_p * D_p$$

Table 24: Nacelles Wetted Area Parameters

EXTERNALLY MOUNTED NACELLES	
engine length	17.83

engine diameter	4.75
ln	7.132
lg	3.566
Lp	2.1396
dp	1.1875
deg	1.9
dg	3.0875
dhl	4.95
dh	6.75
def	5.75
l1	7.132
S_wet_fan_cowling	269.634175
S_wet_gas_gen	30.76672539
S_wet_plug	5.58462345
nacelle attached to the Fuselage	158.902

The total wetted Area obtained by adding the wetted areas of all the above components and subtracting the regions where they intersect, we get the actual wetted area of the entire aircraft as:

$S_{T/G} = 5789.67\ ft^{m}$

using the wetted area of the airplane obtained and the figure 3, the equivalent parasite area (f) is obtained as

f = 13.6 ft^2

hence the clean zero lift Drag Coefficient is obtained by dividing the equivalent parasitic drag to the Area of the wing which gives us:

$C_{=_{>}} = \frac{"}{æ} = \frac{Zp.o}{Zèmè.pè} = 0.00954$

$C_{=_{>}} = 0.00954$

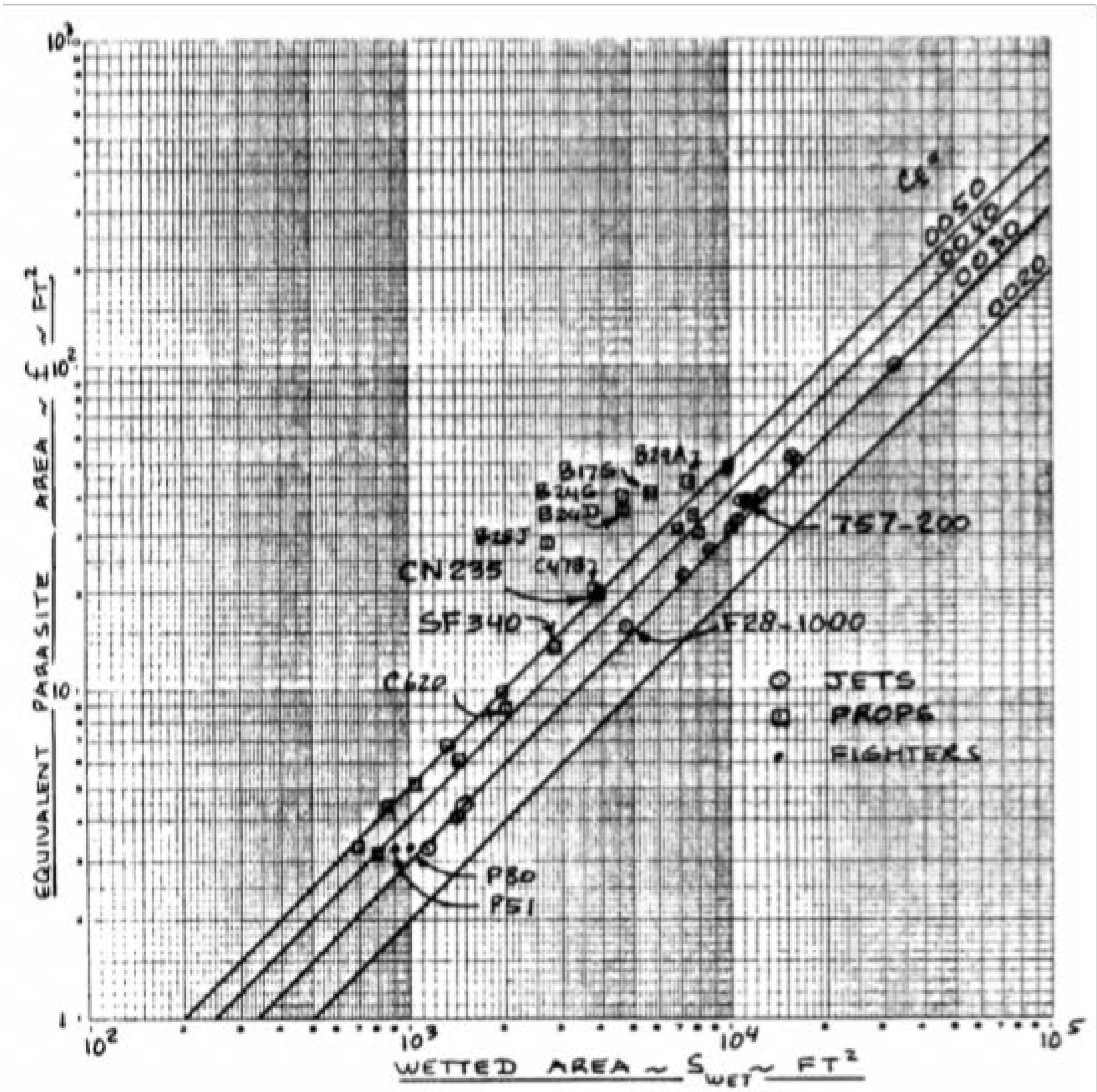


Figure 125: Equivalent Parasite Area v/s Wetted Area

Configuration	ΔC_{D_o}	e
Clean	0	0.80 - 0.85
Take-off flaps	0.010 - 0.020	0.75 - 0.80
Landing Flaps	0.055 - 0.075	0.70 - 0.75
Landing Gear	0.015 - 0.025	no effect

Figure 126: Incremental zero-lift drag values at different stages and the Oswald’s coefficient

10.3 LOW SPEED DRAG INCREMENTS

10.3.1 High Lift Device Drag Increment for Take-off and Landing

The Drag increment due to high lift devices is explained in the table below as:

Table 25: Drag Increment Due to High-Lift Devices

Component	$\Delta C_{=_{>}}$	e	Aspect ratio	Drag Polar
Take-off Flaps	0.020	0.8	2.1	$C_{=} = 0.0419 + 0.1895\ C_{+}^{\mathrm{m}}$
Landing-Flaps	0.075	0.75	2.1	$C_{=} = 0.0969 + 0.2021\ C_{+}^{\mathrm{m}}$

10.3.2 Landing Gear Drag

Table 26: Drag Increment Due to the Landing- Gear

Component	$\Delta C_{=_{>}}$	e	Aspect ratio	Drag Polar
Landing Gears	0.025	No Effect	2.1	$C_{=} = 0.03454$

10.4 COMPRESSIBILITY DRAG

The Compressibility Drag section is defined for airplanes flying at Subsonic Speeds. The SSBJ cruises at Supersonic Speeds hence the Compressibility drag section won't apply to this design.

10.5 AREA RULING

10.6 AIRPLANE DRAG POLARS

The Aircraft Drag Polars obtained are as follows:

Table 27: Parameters Required to calculate the Drag Polars

W_{TO} (lbs)	$(W/S)_{TO}$	S (ft^2)	S_{Wet} (ft^2)	f (ft^2)	$C_{=_{>}}$
110000	77.22	1424.34	5789.67	13.6	0.00954

The drag polar equations and graphs obtained are as follows:

The Drag Polar obtained for different stages of the flight are:

Table 28: Drag Polars For different Configurations

Configuration	e	Aspect Ratio	Increment	Drag Polar
Clean Stage	0.85	2.1	0	$C_{=} = 0.1219 + 0.1783\, C^{m}_{+}$
TO Flaps-Gear Down	0.8	2.1	0.045	$C_{=} = 0.0669 + 0.1895\, C^{m}_{+}$
TO Flaps-Gear Up	0.8	2.1	0.020	$C_{=} = 0.0419 + 0.1783\, C^{m}_{+}$
Landing Flaps Gear-Up	0.75	2.1	0.075	$C_{=} = 0.0969 + 0.2021\, C^{m}_{+}$
Landing Flaps Gear- Down	0.75	2.1	0.1	$C_{=} = 0.1219 + 0.2021\, C^{m}_{+}$

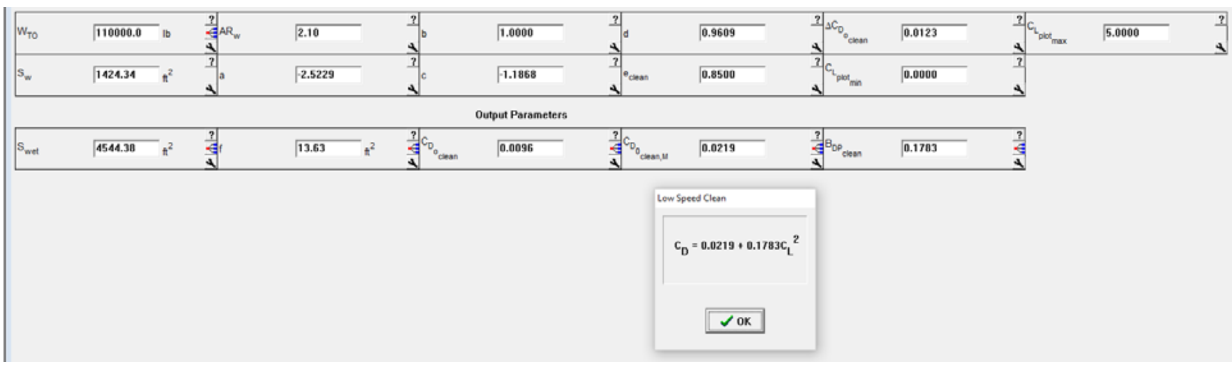


Figure 127: Drag Polar Calculation for Clean Stage

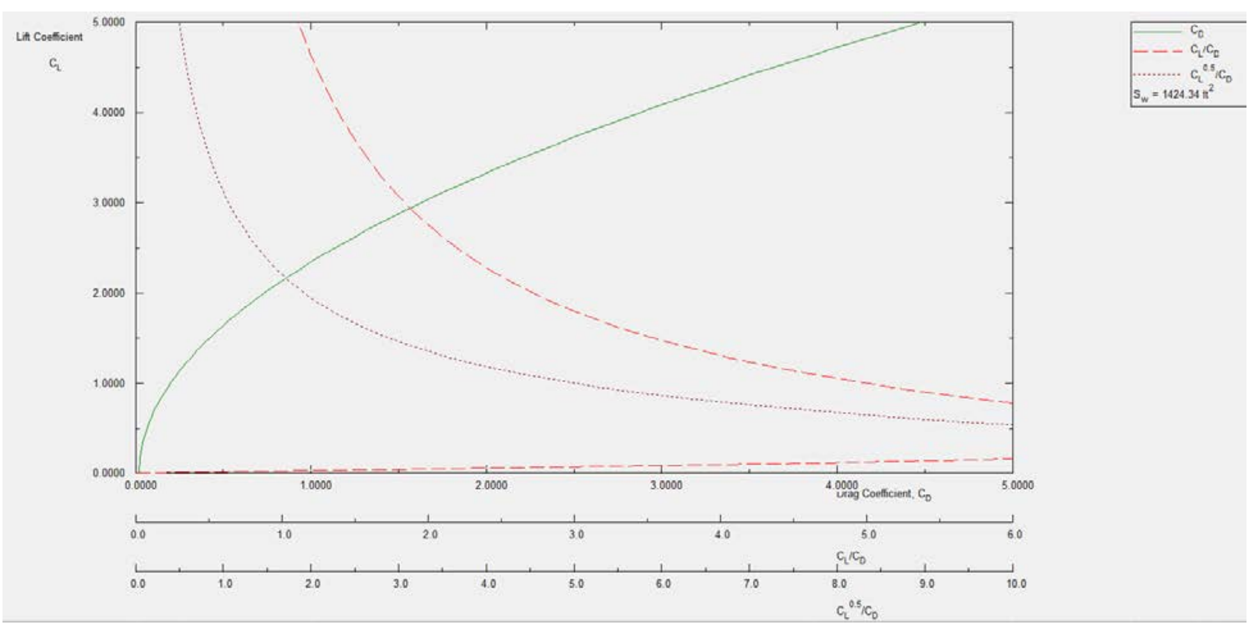


Figure 128: Plot of Clean stage CL v/s CD

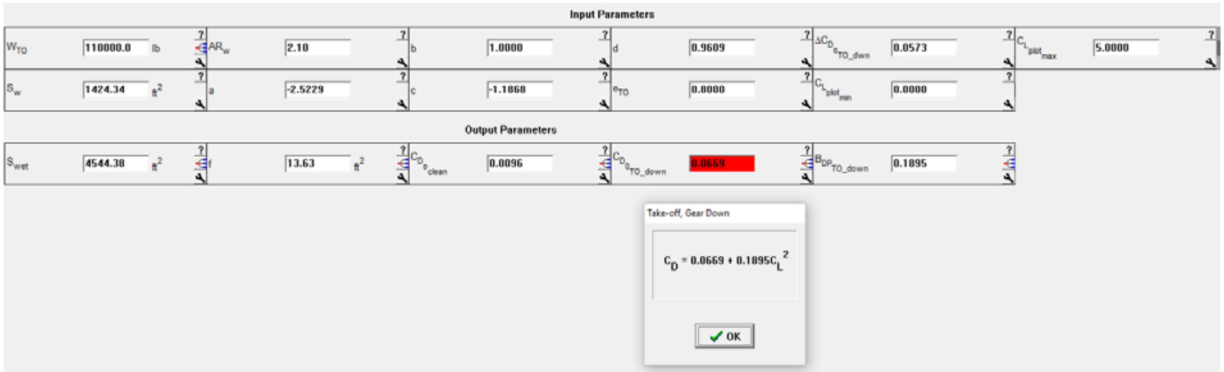


Figure 129: Drag Polar Calculations for Take-off-flaps- Gear down configuration

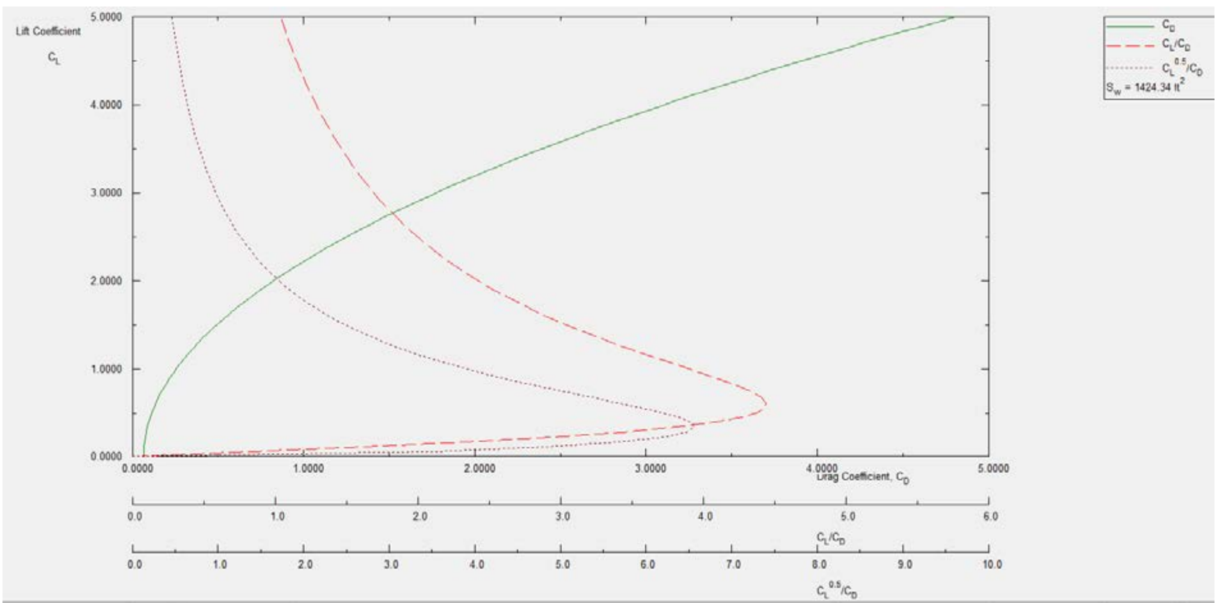


Figure 130: CL vs CD plots for take-off flaps-gear down configuration

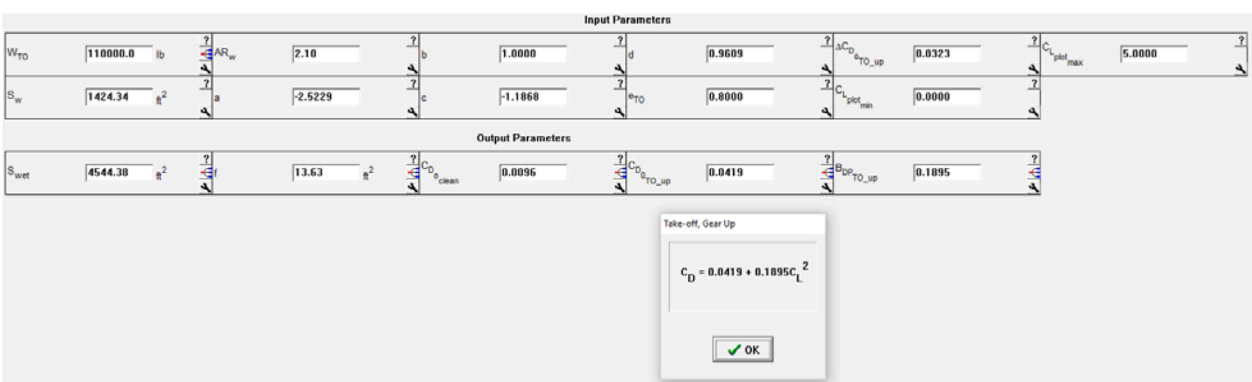


Figure 131: Drag Polar Calculations for Take-off flaps - Gear-Up Configuration

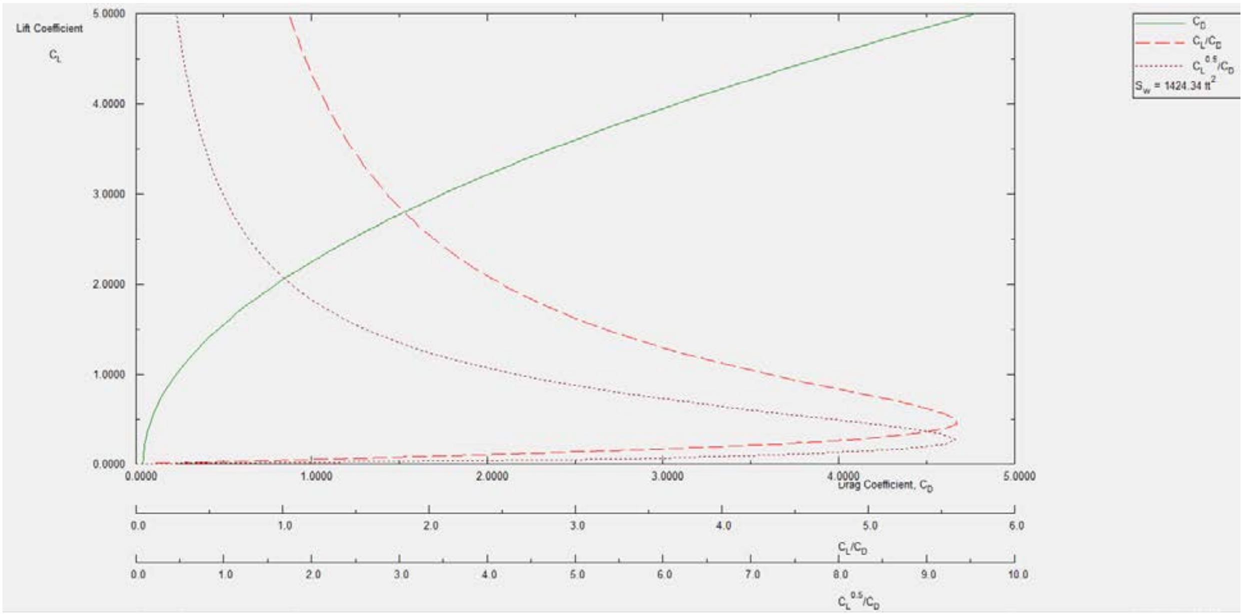


Figure 132: CL vs CD Plots for Take-off Flaps- Gear-up Configuration

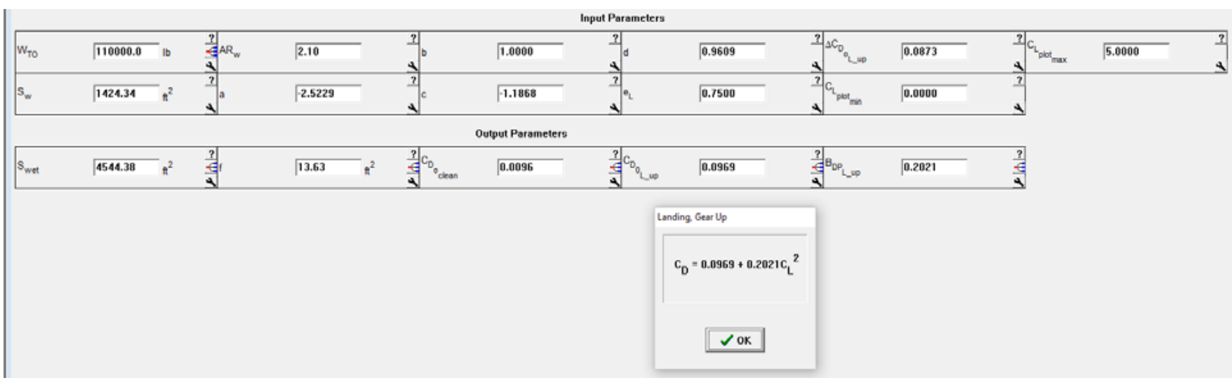


Figure 133: Drag Polar Calculation for Landing-Flaps gear-up configuration

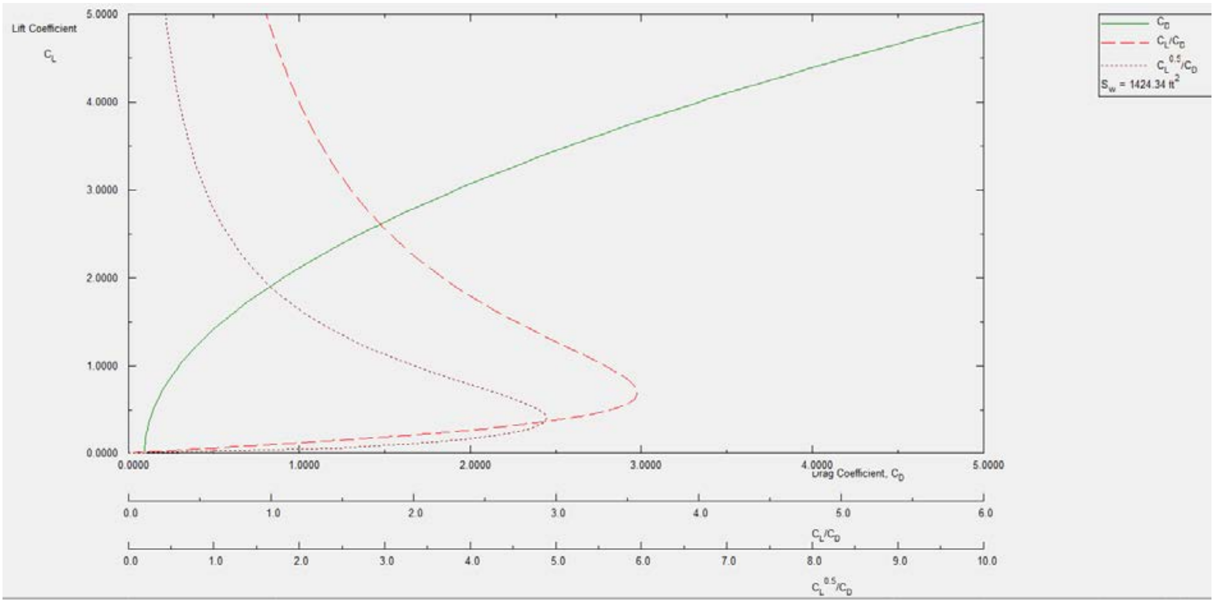


Figure 134: CL v/s CD plots for Landing-Flaps gear up configuration

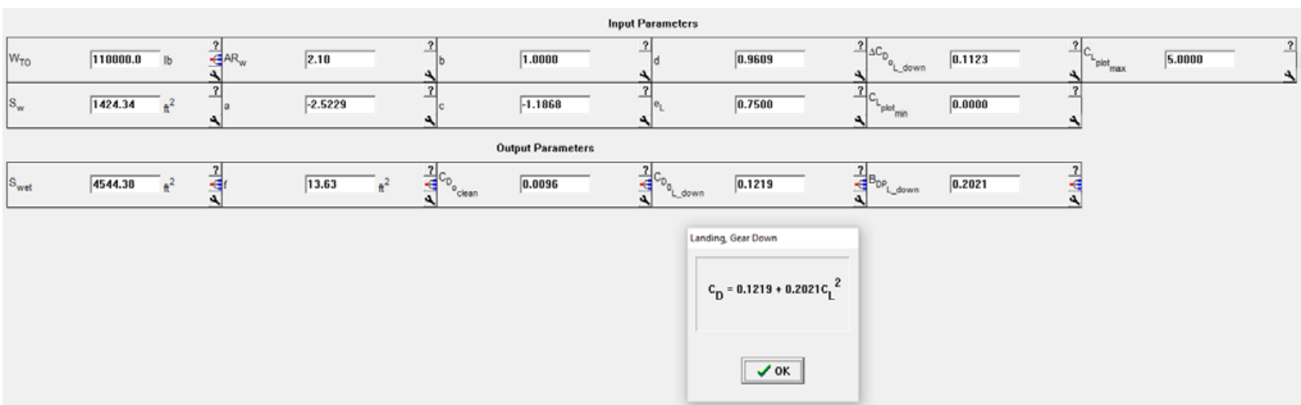


Figure 135: Drag Polar Calculation For Landing-Flaps gear-down configuration

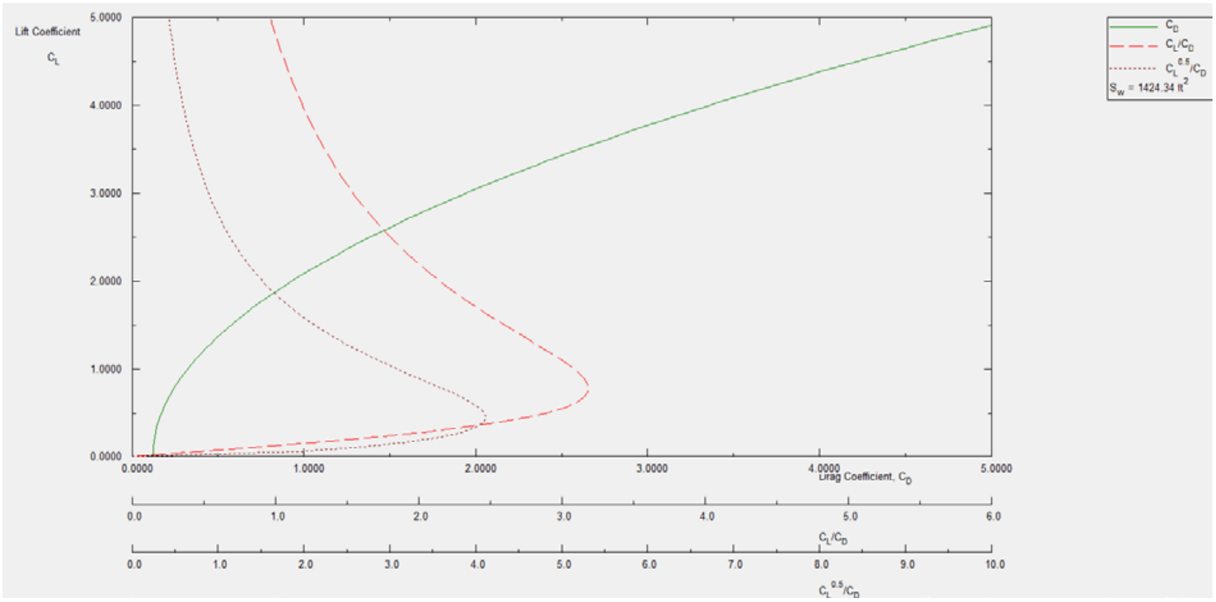


Figure 136: CL v/s CD plots for Landing-Flaps gear down configuration

10.7 DISCUSSION

In this report, the drag polar's of the high lift devices and the landing gears were calculated and obtained using manual calculations as well as using the AAA program. The results obtained from both the process provide the same results and even on comparing them with the results obtained earlier, all the results are the same with minor changes in the maximum lift to drag ratios. Hence no further change is required in the design of the supersonic business jet.

10.8 CONCLUSION

The required performances already satisfy the previous results as obtained in the earlier performance constraints reports. So, no change is further needed in the design of the airplane.

CHAPTER 11: FINAL DESIGN REPORT – ENVIRONMENTAL/ ECONOMIC TRADEOFFS; SAFETY/ ECONOMIC TRADEOFFS

11.1 DRAWINGS & SUMMARY OF MOST IMPORTANT DESIGN PARAMETERS

Table 29: Aircraft Component Parameters

	Wing	Horizontal Tail	Vertical Tail
Area	**1424.34 ft^2**	**385.95 ft^2**	**266.4 ft^2**
Span	**54.69 ft**	**34.027 ft**	**17.88 ft**
Aspect Ratio	**2.1**	**3**	**1.2**
Taper Ratio	**0.15**	**0.6**	**0.9**
Thickness Ratio	**0.04**	**0.06**	**0.12**
Airfoil	**NASA SC(2)-0404**	**NASA SC(2)-0406**	**NACA-0012**
Dihedral Angle	**0**	**0**	**90**
Incidence Angle	**0**	**0**	**0**
Root Chord	**45.29 ft**	**14.177 ft**	**15.75 ft**
Tip Chord	**6.8 ft**	**8.5 ft**	**14.175 ft**
	Fuselage		
Total length	**98 ft**		
Diameter	**9 ft**		
Width	**9 ft**		
Fineness Ratio	**10.89**		
Tail Cone Length	**23 ft**		
Cabin Length	**25 ft**		
Nose Length	**5 ft**		

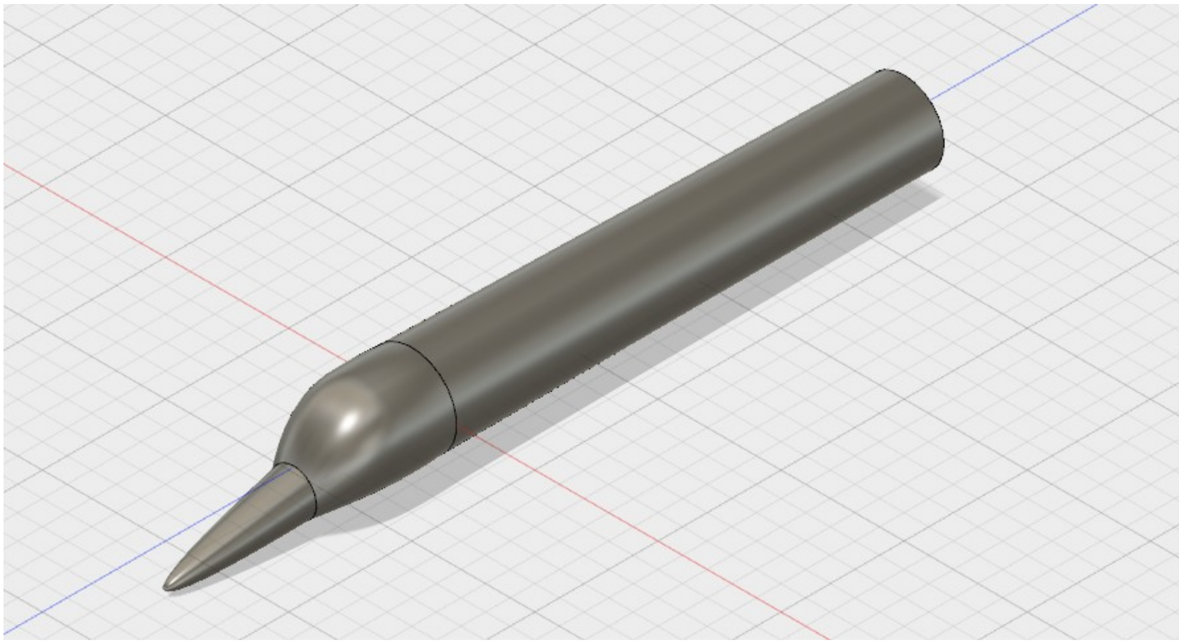


Figure 137: 3D view of the designed cockpit and fuselage

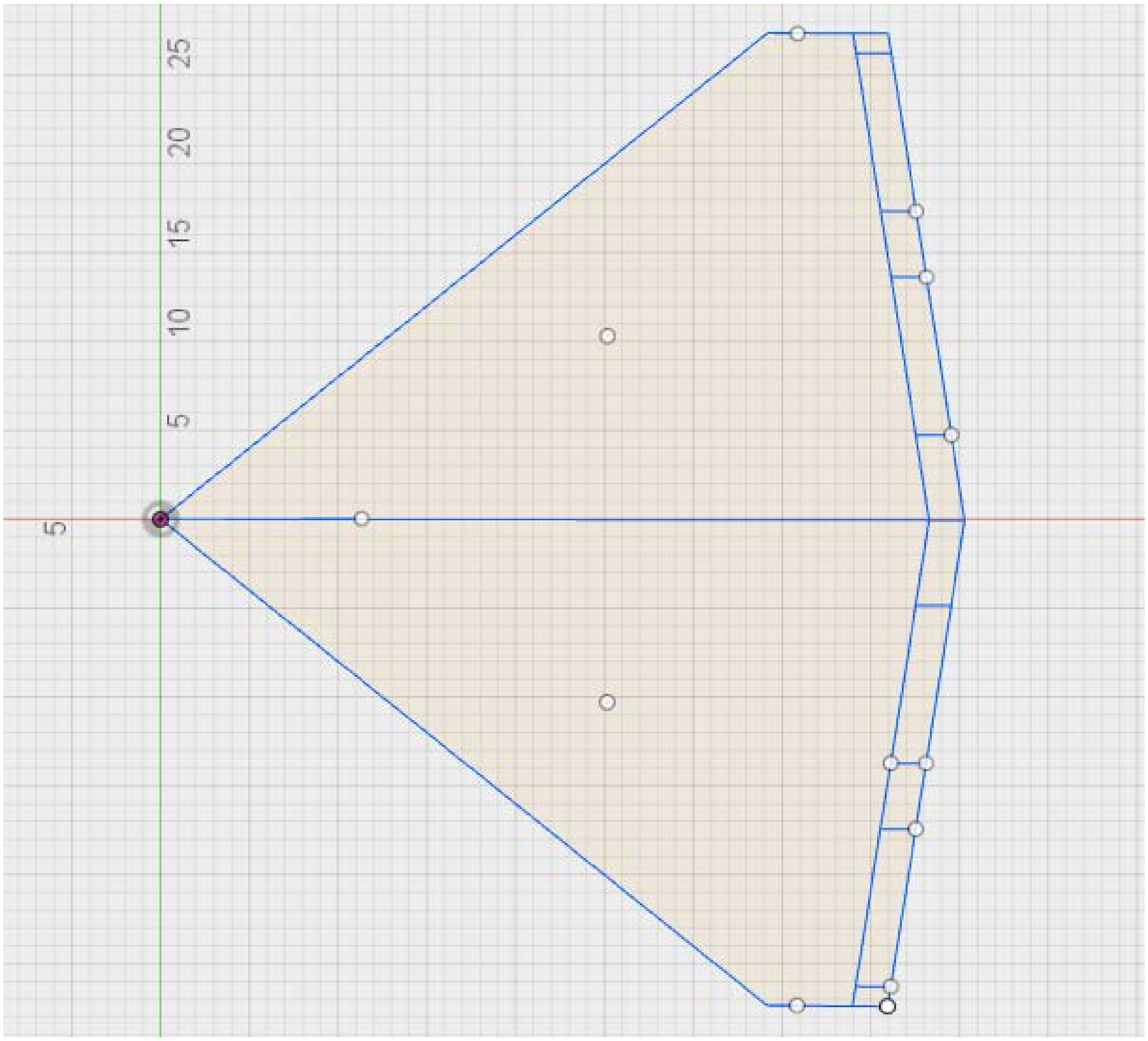


Figure 138: 2D view of the Designed Wing

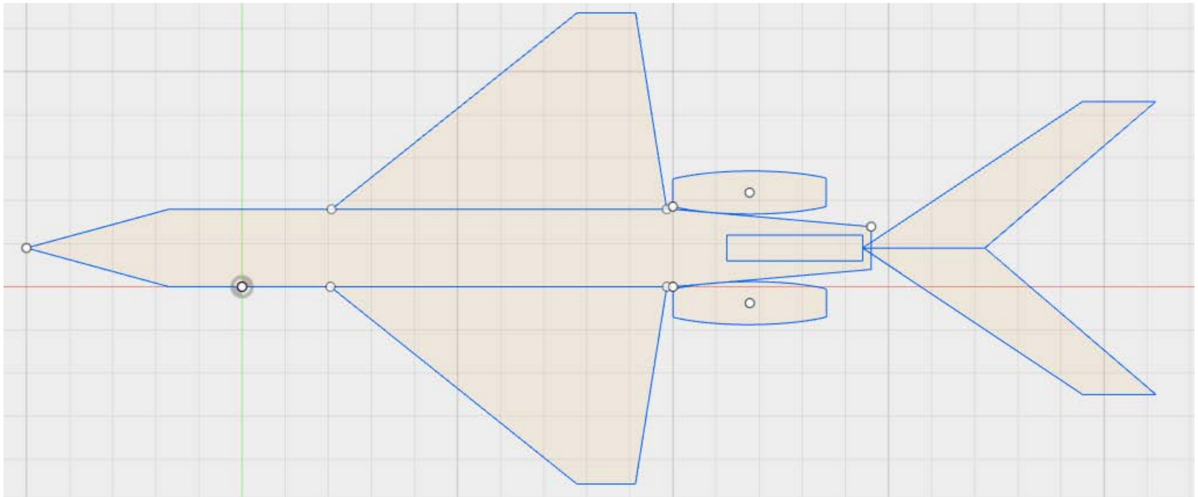


Figure 139: Top view of the designed SSBJ

Figure 140: Side view of the Designed SSBJ

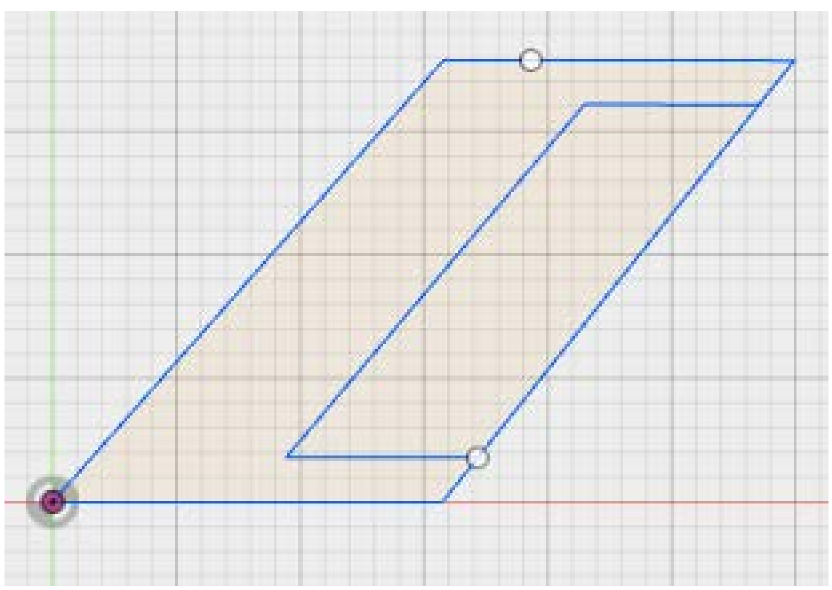


Figure 141: Side view of the vertical stabilizer

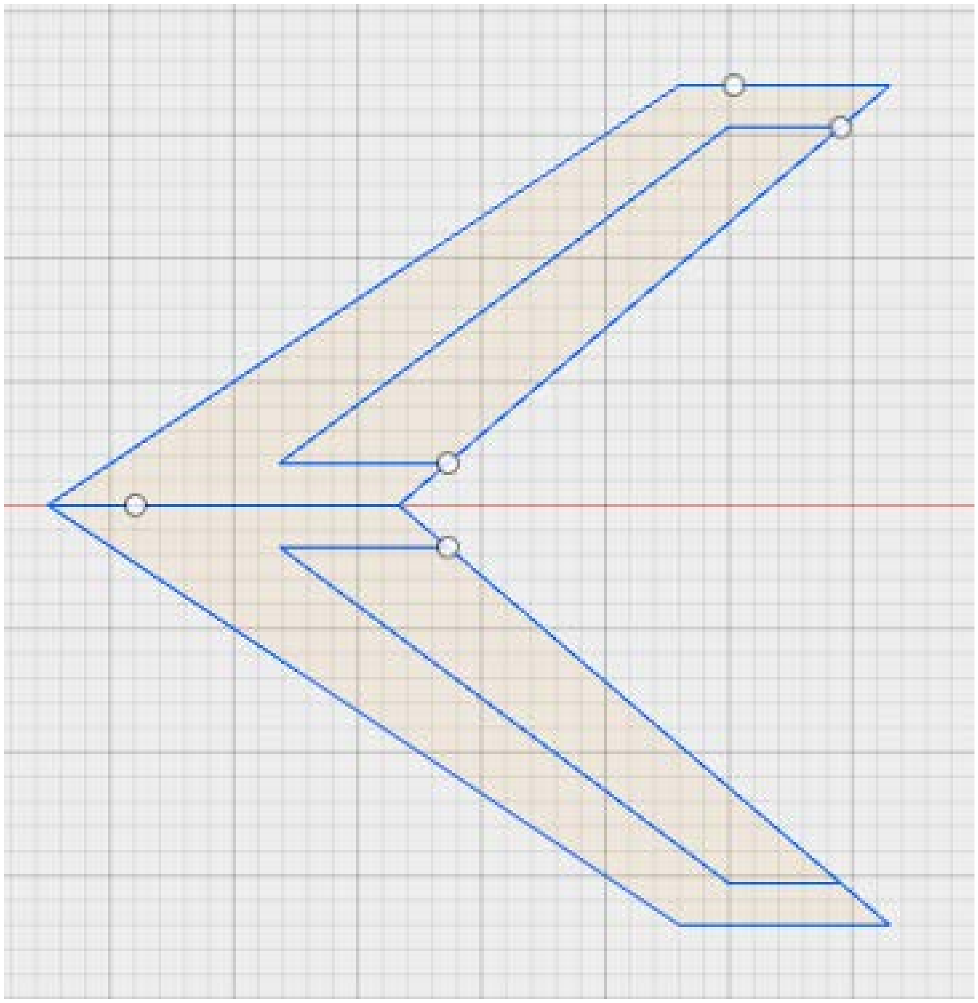


Figure 142: Top view of the horizontal Staibilizer

11.2 RECOMMENDATIONS

After designing the Supersonic Business Jet, as observed in all the previous reports, it can be concluded that it satisfies all the critically required parameters of the Preliminary design process. Hence the design can be further worked upon with a detailed view of all the parameters and characteristics of the Business Jet. For further design, it can be Recommended that a lot of changes are supposed to be done on the design of the supersonic business jet. Looking at all the reports in a list as they were worked upon, they can be recommended as:

1. For the initial sizing, it is highly recommended to reduce the number of passengers on board from 20 to 12. Reducing won't make much of a difference in the design of the airplane rather just provide more comfort to the passengers onboard and get the best of what they are paying for. It is recommended to consider 12 passengers as most of the supersonic business jets being designed carry passengers ranging from 12-16. Hence the lowest value is selected to provide better performance to the airplane adjusting the

weight in other components which is always required in an airplane. It is also recommended to increase the range.

2. For the configuration design, it is recommended to use a cranked arrow wing in the design of the airplane which will help to reduce the weight of the airplane by a great extent by installing the elevators on the cranked part of the wing which helps to remove the horizontal stabilizer and reduce the weight of the vertical stabilizer.

3. For the weight sizing of the airplane, it is recommended to use composites on major components of the airplane which helps reduce the weight of the total aircraft and also provide better performances and higher speeds, low maintenance and higher life cycles, more strength etc.

4. For the performance of the airplane, the stall speed should be reduced to provide better low speed performance of the airplane. Which also helps to reduce the wing loading on the airplane, increases the lift coefficient, reduces the drag produced. Provides better take-off and landing performances to the airplane.

5. For the cockpit/fuselage design of the airplane, it recommended to further reduce the maximum height to provide better aerodynamic performances.

6. For the wing design, it is recommended to use the cranked arrow wing rather than a delta wing as recommended earlier for the configuration design.

7. For the empennage section, it is recommended to reduce the weights after a lot of iterations in the weight and balance section considering every minute change in the area of the horizontal and vertical design according to the CG obtained that satisfying the airplanes stability requirements.

8. For the landing gear design, it is recommended to use the retractable landing gears having a tricycle configuration. Which is already used in the present design as at higher speeds, fixed landing gears produce a lot of drag which is not suggested for supersonic airplanes. It is also suggested to consider a lot of iterations to obtain the exact CG location of the airplane and consider every small factor that affects the change in weight of the airplane and place the landing gears behind the CG to maintain the stability of the airplane and satisfy the nose over and ground clearance criterion as the wing configuration is a mid-wing configuration so the it should provide better ground clearance to obtain better ground effects while take-off of the airplane.

9. for the stability and control part, it is recommended to iterate the longitudinal and directional X-plots up to a number of times till the best configuration that’s feasible to the design of the airplane is obtained hence it should satisfy every critical error in the design that could affect the design of the airplane.

10. For the drag polar estimation, it is suggested to consider the highest possible zero lift drag incremental factors to obtain better performances to the design.

11.3 ENVIRONMENTAL/ECONOMIC TRADEOFFS

- The most important environmental issue related to the airplane is the fuel consumption. As the Business Jet cruises at Supersonic speeds, it consumes a lot of fuel producing a lot of noise inside the engine section which contributes to the noise and air pollution. These issues are faced by all the airplanes hence it is suggested to do some research on the engine sections and reduce the usage of fuel and using hydrogen powered engines which reduce the air pollution up to a very great extent. And for the noise pollution, it is suggested to use spikes in the outlet sections of the engines which helps reduce noise and the vibrations produced.

- These issues where addressed by doing a lot of research work in this section. To reduce the air pollution battery powered airplanes were introduced while to reduce the noise effects spikes were introduced and a lot of research work has been done on the noise reducers.

- According to me, the best proposed solution to the air pollution is the use of hydrogen powered airplanes for which the fuel costs reduces by a very great extent. As hydrogen is known as the most combustible gas, it also provides better performances and also reduces the air pollution.

- The trade-off for the environmental issue is the cost of research to be done and the cost to manufacture the proposed design as any minor change in the design can cost millions to redesign and manufacture.

11.4 SAFETY/ECONOMIC TRADEOFFS

- The most important safety issue related to the airplane is the control of the airplane at supersonic speeds. It is very important to keep in the airplane under the control of the pilot.

- For to satisfy the safety requirements, high speed controllers are to be designed which can maintain the aircraft steady even without the help of pilots. It should perform on its own such that it can predict every minor change required to keep the aircraft steady in any condition and it should be controllable.

- To design such a system, it costs millions to produce such a controller.

- As there aren’t any supersonic business jets currently available, only research work has been carried on them. All the proposed trade-offs were an assumption made from the problems faced by airplanes flying at similar speeds.

CLASS II SSBJ DESIGN

CHAPTER 12: LANDING GEAR DESIGN

12.1 Introduction

Landing gears in an aircraft are used for the following reasons:

- To absorb landing shocks and taxing shocks.
- To provide better ground maneuvering that is: taxi, take-off roll, landing roll and steering.
- To provide better braking capability.
- To allow for airplane towing.
- To protect the ground surface.

The landing gears should be designed in such a manner that the zero touch down rate can absorb landing and taxi loads and also transmit a part of these loads to the airframe. The magnitude of the loads depends on the type of airplane and it's mission profile.

The types of loads to be considered while designing the landing gears are the:

1. Vertical loads: caused by non-zero touch down rates and taxing over rough surfaces. (For transport aircraft's the touchdown speed should be around 10 fps.)
2. Longitudinal loads: caused by the spin-up loads, braking loads and rolling friction loads.
3. Lateral loads: caused by crabbed landing, cross wind taxing and ground turning.

12.2 Vertical Landing Gear Loads:

The magnitude of the loads acting on the vertical landing gears depends mainly on the touchdown rate. For the business Jet, the touch down rate should be,

w_t = 17 fps. (1)

Landing Gear types:

To design the type of the landing gear, the main considerations to keep in mind are:

a. Fixed of retractable landing gear.
b. Deciding the configuration of the landing gear (tricycle, bicycle, tailwheel or unconventional gear.

As discussed earlier the supersonic business jet has a retractable landing gear with a tricycle landing gear configuration.

The tricycle landing gear was selected due to the following reasons:

a. Good visibility over the nose during ground operations.
b. Stability against ground loops.
c. Good steering characteristics.

d. Level floor while on the ground.
e. Easy take-off procedure.

Retractable landing gears were selected for the design as they provide minimal aerodynamic drag.

12.3 Compatibility of landing gears and runway surface:

The load on each landing gear as well as the load on each tire must not exceed the values which:
- Cause structural damage to the gear or to the airplane.
- Cause tire damage.
- Cause runway damage or excessive surface deformations.

Description of Surface	Maximum Allowable Tire Pressure	
	kg/cm^2	psi
Soft, loose desert sand	1.8 - 2.5	25 - 35
Wet, boggy grass	2.1 - 3.2	30 - 45
Hard desert sand	2.8 - 4.2	40 - 60
Hard grass depending on the type of subsoil	3.2 - 4.2	45 - 60
Small tarmac runway with poor foundation	3.5 - 5.0	50 - 70
Small tarmac runway with good foundation	5.0 - 6.3	70 - 90
Large, well maintained concrete runways	8.5 - 14	120 - 200

Figure 143: Recommended tire pressures for various surfaces

12.4 Nose gear steering loads:

To obtain adequate nose wheel steering, a minimum normal force must act on the nose gear to generate appropriate levels of friction forces needed for steering.
The normal force on the nose gear should not be less than $0.08W_{TO}$ for adequate steering.

Hence, for the SSBJ, the normal force on the nose gear should not be less than:
$0.08 * W_{;<} = 0.08 * 110{,}000 = 8800\ lbs\ (3991.6\ kg)$ (2)

12.5 Gear loads from a surface viewpoint:

Three types of runways will be considered:

- Type 1 surface: Runways with unprepared or simply prepared surfaces such as grassy surfaces or gravel surfaces. Surface failure occurs due to severe local indentation caused by excessive tire loads.
- Type 2 surface: Runways with flexible pavement (asphalt or tarmacadam). These are normally very thick surfaces. The surface failure occurs due to local indentation caused by excessive tire loads. This may result in severe surface waviness.
- Type 3 surface: th1ese are the surfaces with rigid pavement (concrete). The thickness of these surfaces if half of the thickness of the flexible pavement.

12.6 Allowable gear loads according to the type of surfaces:

- Type 1: the load per strut cannot increase more than 10000 lbs.
- Type 2 & 3: depend on the LCN (load classification number). Every runway has a load classification number. The LCN number must not exceed the lowest LCN number it is intended to run on.

As the SSBJ has a Twin-Tricycle configuration, the Equivalent single wheel load wil be equal to,

$$ESWL = \frac{*7}{m} \; or \; \frac{*,,}{m} \quad (3)$$

where P_n and P_m are the loads acting on the nose wheel and the main landing gear wheels.

$$P_0 + n_{.}P_9 = W \quad (4)$$

For the supersonic business jet, the wheel configuration is the Twin Tricycle layout (twin delta tandem).

The number of struts and the number of wheels per strut can be determined according to the range of the airplane.

P_n = the pressure on the nose wheel.
P_m = the pressure on the main wheel.
L_n = the distance between the nose wheel and the CG of the airplane.
L_m = the distance between the main wheel and the CG of the airplane.
n_s = no. of main gear struts.

The tires manufactured are rated in terms of:

- Ply rating
- Maximum allowable static loading
- Recommended inflation pressure
- Maximum allowable runway speed

Certain geometric parameters of the tires are:
D_o or D_t: outside diameter

W or b_t: maximum width
D: the tire rim diameter

The structure of tires is affected due to the severe static and dynamic load during taxing, during take-off roll and during landing roll. The tires also help to absorb shock during touchdown. How much shock the tires absorbs, depends on the design of the shock absorbers.

Each tire is designed to operate at a so-called maximum allowable static load. This load must not exceed the most critical weight/c.g. combination. A 25% growth in the tire load is allowed while selecting a tire for a new airplane.

Nose-wheel tires are designed for maximum allowable dynamic loads.
Dynamic load = $f_{B10}(static\ load)$
For tires for new designs: $f_{B10} = 1.50$ (5)
The allowable tire deflection may be computed from: $s_t = D_o - 2$ (loaded radius)

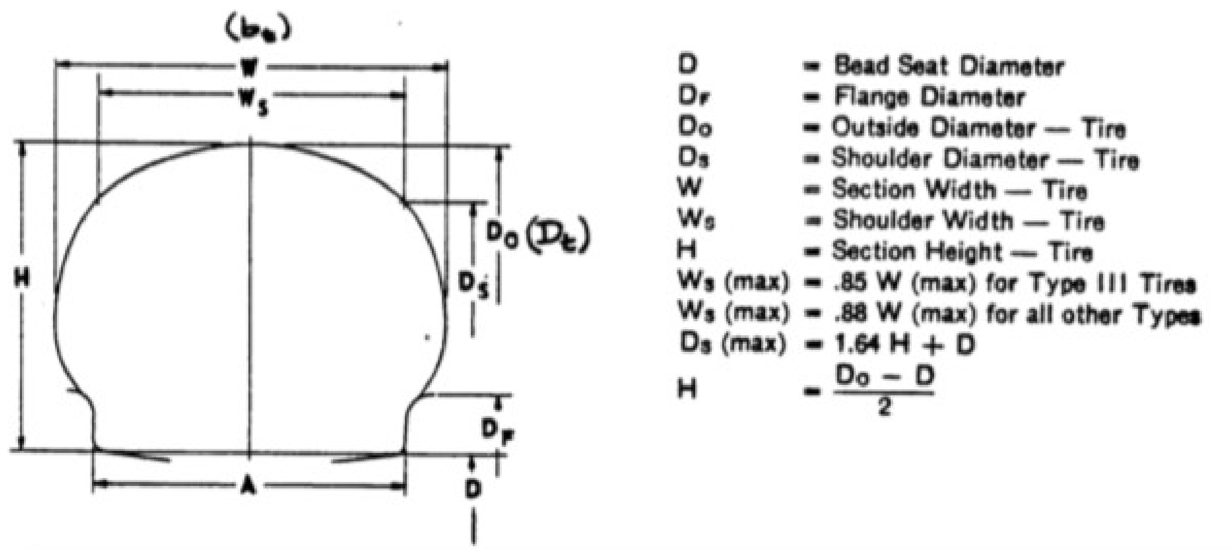


Figure 144: Tire Geometry parameters

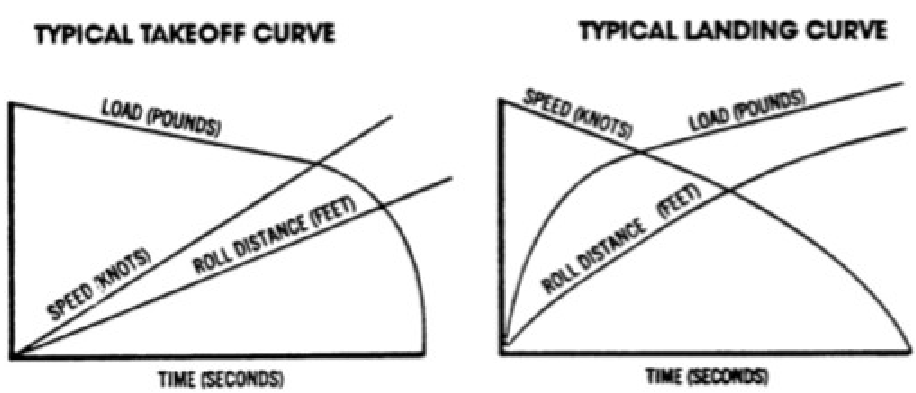


Figure 145: Tire Performance requirements while take-off and landing

12.7 Tire Clearance Requirements

1. Wheel well clearance (after retraction)
2. Tire-to-fork and tire to strut clearance
3. Tire-to-tire clearance in multiple wheel arrangements

The tire grows in size during its service life. 4 percent in width and 10 percent in diameter. It grows under the influence of centrifugal forces. This forces depend on the maximum tire operating speed on the ground.

For preliminary design proposes, it is acceptable to account for the following tire clearances:
Width: 0.04W + lateral clearance due to centrifugal forces + 1 inch
Radius: $0.1D_o$ + radial clearance due to centrifugal forces + 1 inch

12.7.1 For Main gear tires:
For FAR 25 certified airplanes, the loads are to be multiplied by 1.07.
To provide allowance for the growth in the airplane weight, the design loads are to be multiplied by 1.25.

The maximum static load for the main gear of the SSBJ is 101065.33 lbs as discussed in the class I design of the airplane.
Multiplying the maximum static load by 1.07 = $101065.33 * 1.07 = 108139.9\ lbs$
To allow for growth in the weight, multiplying the result by 1.25 which gives,
$108139.9 * 1.25 = 135174.9\ lbs$
The load on each tire of the main gear = $\frac{\text{ZpxZnè ww}}{\text{è}} = 33793.7\ lbs$ (6)

12.7.2 For Nose gear tires:
For FAR 25 certified airplanes, multiply the loads by 1.07.
To provide allowance for the growth in the airplane weight, the design loads are to be multiplied by 1.25.

The maximum static load for the nose gear of the SSBJ is $12175.1 * 1.07 = 13027.35\ lbs$
To allow growth in the weight, multiplying the result by 1.25 which gives,
$13027.35 * 1.25 = 16284.2\ lbs$
the load on each tire of the nose gear is $= \frac{\text{Zomwè m}}{\text{m}} = 8142\ lbs$ (7)

The maximum static load per nose gear tire can be determined from:

$$P_{0_{B10_G}} = \frac{\text{T\_ˋMY,,‡}\frac{\text{‰Â}}{\text{-zˉ<-l}}\text{P}}{\text{0}_6\text{(Y,,‡Y7)}}$$

a_x can be obtained from the following:

$$\frac{\text{-Â}}{1} = 0.35\ (for\ dry\ concrete\ with\ simple\ brakes)$$

$$\frac{\text{-Â}}{1} = 0.45\ (for\ dry\ concrete\ with\ anti-skid\ brakes)$$

The SSBJ design has anti-skid brakes installed due to which $a_x/g = 0.45$

Hence, the maximum dynamic load per nose gear tire is:

$$P_{0_{B10_G}} = 110000 * \frac{\mathrm{Fm.noZ‡}\frac{\mathrm{h.UV}}{\mathrm{\hat{}h.fd}}\mathrm{H}}{\mathrm{m(m.noZ‡pZ.mm)}} = 4539.39\ lbs \quad (8)$$

the design maximum static load may be obtained from the maximum dynamic load by diving the following factor:

for Type I to III tires: 1.45

for type II tires: 1.25

for type VI, VII, VIII: 1.50

For new design tires: 1.50

The maximum tire operating speed is the highest take-off and landing speed of the aircraft:

For landing: $V_{G?2/_{„‰Â}} = 1.2\ V_{-_+} = 1.2 * 190 = 228\ knots$ (9)

For take-off: $V_{G?2/_{9-:}} = 1.1\ V_{-_{;<}} = 1.1 * 190 = 209\ knots$ (10)

12.8 Devices used for Shock Absorption

- Tires
- Air springs
- Oleo-pneumatic struts
- Shock chords and rubbers
- Cantilever springs
- Liquid springs

Shock absorption capability of tires and shock absorbers:

The maximum kinetic energy that needs to be absorbed when the airplane touches down is:

$$E_G = 0.5 * W_+ * w_G^{\frac{d}{-}} \quad (11)$$

where, W_L = landing weight

w_t = design vertical touchdown rate

12.8.1 For the main Landing gear

$$E_G = n_- P_9 N_1 (\eta_G s_G + \eta_- s_-) \quad (12)$$

where, $W_+ = n_- P_9$ (13)

n_s = no. of main gear struts (assumed to be equal to the number of shock absorbers)

P_m = maximum static load per main gear strut.

N_g = landing gear load factor
η_t = tire energy absorption efficiency
η_s = shock absorber energy absorption efficiency
s_t = maximum allowable tire deflection
s_s = stroke of the shock absorber.

$$s_{-} = \frac{34\frac{h.VM\frac{ü}{-}öP(_{\text{\textbackslash}}6)^{d}}{7\$\hat{o}_{,,}5_{-}}6\dagger\ \eta_{-}\,_{6}\,_{\bar{6}}7}{\eta_{\$}} \tag{14}$$

$$s_{-\,B/-?10} = s_{-} + \frac{Z}{Zm} \tag{15}$$

The diameter of the shock absorber (strut) may be estimated from:

$$d_{-} = 0.041 + 0.0025(P_{9})^{\hat{\bar{d}}} \tag{16}$$

Element:	Energy Absorption Efficiency:
Tires:	η_t = 0.47
Shock absorbers:	
air springs	η_s = 0.60 t0 0.65
metal springs with oil damping	= 0.70
liquid springs	= 0.75 to 0.85
oleo-pneumatic	= 0.80
cantilever spring	= 0.50

Figure 146: Energy absorption efficiency of tires and shock absorbers

Certification Base:	Landing Gear Load Factor, N_g:
FAR 23	N_g = 3.0
FAR 25	N_g = 1.5 to 2.0
Fighters and Trainers	N_g = 3.0 - 8.0: See Fig.2.25 for more details
Military transports	N_g = 1.5 - 2.0

Figure 147: Landing gear load factors

equation x assumes that the main gear reaction load is transferred directly into the shock absorber. This condition is not satisfied for the gears where the reaction line is not in line with the shock absorbers.

hence, from equation 13, we get the weight of the airplane while landing as

$$W_{+} = 2 * 67587.43 = 135174.879\ lbs \qquad (17)$$

using equation 11, we get the maximum kinetic energy to be absorbed as,

$$E_G = 0.5 * W_{+} * (w_G)^{\frac{d}{-}} = 0.5 * 135174.879 * (17)^{\frac{d}{i.]\wedge}} = 120426.494\ lbs.ft \qquad (18)$$

for the main landing gears, the maximum kinetic energy is explained in equation 12 and 14. For the SSBJ, the stroke of the absorber is,

$$s_{-} = \frac{89\dfrac{0.5\ M\frac{W_{+}}{g}P\ (w_G)^{m}}{n_{-}P_9N_1} - \eta\, s_G;}{\eta_{-}} = \frac{3\dfrac{0.5 * \frac{135174.879}{9.81} * (17)^{m}}{2 * 67587.43 * 2} - 0.47 * 387}{0.8} = -13.12$$

$$s_{-} = -13.12 \qquad (19)$$

$$E_G = n_{-}P_9N_1(\eta_G s_G + \eta_{-}s_{-}) = 2 * 67587.4394 * 2 * (0.47 * 38 + 0.8 * 13.12)$$

$$E_G = 1991108.05\ lbs \qquad (20)$$

The diameter of the shock absorber (strut) can be obtained from the following equation,

$$d_{-} = 0.041 + 0.0025(P_9)^{\frac{\wedge}{d}} = 0.69\ ft \qquad (21)$$

12.8.2 For the nose gear

Replace W_L with P_n, the load P_m must be replaced by the maximum dynamic load of the nose gear.
For the nose gear n_s=1

$$s_{-} = \frac{3>\frac{h.VF\frac{\hat{o}7}{-}H(\_6)^{d}}{7\$\hat{o}7\&?7_{6}5-}@\dagger\eta_{6}-6^{7}}{\eta_{\$}} = \frac{-A\frac{Fh.V*F\frac{\hat{}vdlU.\hat{}i}{I.J}H*(\hat{}\backslash)dH}{\hat{}*ih\backslash].\backslash i*d}B\dagger>.\grave{e}n*mm"}{>.w}$$

$$s_{æ} = 3.58 \tag{22}$$

$$d_{-} = 0.041 + 0.0025\,F2 * P_{0}\,{}_{B10_{G}}H^{\hat{\bar{d}}} = 0.28\,ft \tag{23}$$

12.9 Brakes and Braking Capability
The purpose of brakes is to:

- Help stop an airplane.
- Help steer an airplane by differential braking action.
- Hold the airplane when parked.
- Hold the airplane while running up the engine.
- Control speed while taxing.

Brakes turn kinetic energy into heat energy through friction. This friction generated heat is dissipated to the immediate environment brake: wheel, tire and surrounding air. The capacity of wheel and tire to absorb heat is limited. This must be accounted for in the design of the wheel and tire.

The rolling friction generated between the rolling tires and the runway is what causes the airplane to slow down. The rolling friction coefficient is related to the slip ratio. Where the slip ratio of the wheel is defined as:

$$Slip\ Ratio = <1 - \frac{TC//Y\ddot{}*\pm\grave{I}2\text{-}D/.\,\ddot{e}0}{TC//Y\ddot{}*\pm\grave{I}2\text{-}D/.\,\ddot{e}""} =$$

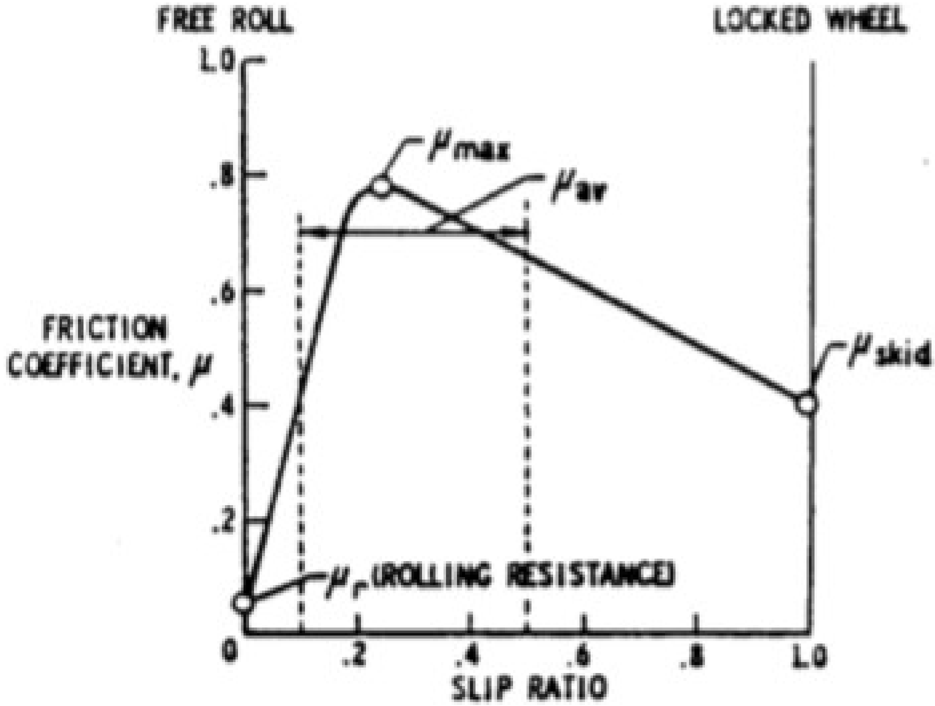


Figure 148: Effect of slip ratio on ground friction coefficient

The zero-slip ratio the coefficient is 0.02 to 0.05 depending on the surface characteristics. At a slip ratio of 1.0, the brakes have 'locked' the wheel and the friction coefficient is about 0.4, corresponding to a skidding condition (this will wear out a tire and cause tire blow-out in then 100 ft). if an anti-skid system is used to control wheel RPM during braking, the average value of friction coefficient which can be attained is about 0.70.

Accounting for the fact that tires will be a bit worn and that brakes do not operate at their best efficiency; the following deceleration values can be obtained during roll-out.

Conventional brakes: 0.35g on a dry surface.
Carbon brakes: 0.40g on a dry surface.
Anti-skid brakes: 0.45g on a dry surface.
Anti-skid carbon brakes: 0.50g on a dry surface.

Note: Carbon brakes offer a significant improvement. Carbon brakes are also 40% lighter than conventional brakes. Their cost is about twice the cost of conventional brakes.

Brake Actuation:
Brakes are actuated with the help of a hydraulic system.

CHAPTER 13: CLASS II AIRPLANE WEIGHT COMPONENTS

13.1 INTRODUCTION

To calculate the Class II weight of the SSBJ, the following data is required:

i. Airplane take-off gross weight
ii. Wing and empennage design parameters such as:
   - Area
   - Sweep angle
   - Taper ratio, λ
   - Thickness ratio, t/c
iii. Load factor, n_{lim} or n_{ult}
iv. Design cruise and dive speed, V_C or V_D
v. Fuselage configuration and interior requirements.
vi. Power plant installation
vii. Landing gear design and disposition
viii. System requirements
ix. Preliminary structural arrangement

The following data can be obtained from the class I design of the SSBJ described in the previous reports.

Some basic weight definitions useful as obtained in clas I design of the airplane are as follows:

$$W_{TO} = W_E + W_F + W_{PL} + W_{tfo} + W_{crew} \quad (1)$$

where,
W_E = Empty weight of the aircraft.

W_F = Mission fuel weight

W_{PL} = Payload weight

W_{tfo} = Trapped fuel weight

W_{crew} = Crew weight

Where,

$$W_E = W_{struct} + W_{pwr} + W_{feq} \quad (2)$$

W_{struct} = Structure weight

W_{pwr} = Powerplant weight

W_{feq} = Fixed Equipment weight

Airplane Component Weights

The following weight items are already known:

Payload Weight: $W_{*+} = 4220\ lbs$

Crew Weight: $W_{32/4} = 700\ lbs$

Fuel weight: $W_{"\sim/Y} = 53671.33\ lbs$

Trapped fuel and oil: $W_{G"ë} = 1073.4\ lbs$

Weights are needed to be estimated for the following items:

Structural weight (W_{struct}):

i. Wing
ii. Adjustments for fowler flaps
iii. Empennage
iv. Fuselage
v. Nacelles
vi. Landing gear

Powerplant weight (W_{pwr}):

i. Engines
ii. Fuel system
iii. Propulsion system
iv. Accessory drives, starting and ignition systems
v. Thrust reversers

Fixed Equipment Weight, (W_{fwq}):

i. Flight controls
ii. Electrical systems
iii. Instrumentation, avionics and electronics
iv. Air-conditioning, pressurization and de-icing
v. Oxygen
vi. APU
vii. Furnishings
viii. Baggage and cargo handling
ix. Operational items

x. Paint

13.2 V-n Diagram

$V_{\text{-Z}}$= +1-g stall speed or the minimum steady flight speed which can be obtained.

V_{N}= Design cruising speed

$V_{=}$= Design diving speed

$V_{®}$= Design maneuvering speed

$V_{\text{ú}}$= Design speed for maximum gust intensity

13.1.1 Calculating the Stall Speed (V_S):

$$V_{\text{æF-\$}} = \text{G}\frac{\text{m}*\text{F}\frac{\text{Hü}}{\text{»}}\text{H}}{\neq*\text{N5}_{\text{„‰Â}}}\text{I}^{\hat{\text{d}}} \quad (3)$$

GW = Design Gross Weight
S = Wing Area, ft^2
ρ = air density in slugs, ft^3
$C_{\text{õ}\,9\text{-:}}$ = maximum normal force coefficient

The maximum normal force coefficient can be obtained from the equation,

$$C_{\text{Õ}\,9\text{-:}} = \text{Gz}C_{+\,9\text{-:}}\,\text{l}^{\text{m}} + \text{M}C_{=\text{-G}\,\text{N}_{\text{ö„‰Â}}}\,\text{P}\,\text{I}^{\text{m}\,\frac{\hat{}}{\text{d}}} \quad (4)$$

$$C_{\text{õ}\,9\text{-:}} = 2.86 \quad (5)$$

therefore, we can get the stall speed with the help of the maximum normal force coefficient

$$V_{\text{æF-\$}} = \text{G}\frac{\text{m}*\text{F}\frac{\text{^^hhhh}}{\text{^UdU.fU}}\text{H}}{\text{>.>>mpnoq*Z.w>nx}}\text{I}^{\frac{\hat{}}{\text{d}}} = 420.45\ knots \quad (6)$$

the negative stall speed line can be determined from the following equation:

$$V_{\text{æ}0/1} = \text{G}\frac{\text{mF}\frac{\text{Hü}}{\text{»}}\text{H}}{\neq\text{N5}_{\text{„‰Â7\%}}}\text{I}^{\hat{\text{d}}} \quad (7)$$

$$C\tilde{o}_{9-:0/1} = \mathrm{AF}C_{+9-:0/1}\mathrm{H}^{\mathrm{m}} + {}^{*}C_{=\text{-GN}_{+9-:0/1}}{}^{\mathrm{m}\,\hat{\bar{\mathrm{d}}}} + \mathrm{B} \quad (8)$$

in equation 8, $C_{+9-:0/1} = -1.0$ will be assumed

substituting that in Equation 8, we get

$$C\tilde{o}_{9-:0/1} = 1.00279735 \quad (9)$$

hence, $V_{æ_{0/1}}$ can be obtained as,

$$V_{æ_{0/1}} = \mathrm{A}\frac{\mathrm{Mm}{*}\mathrm{F}\frac{\hat{}\hat{}\mathrm{hhhh}}{\hat{}\mathrm{UdU.fU}}\mathrm{HP}}{\mathrm{>.>>mpnoq{*}Z.>>mnqnpx}}\mathrm{B}^{\hat{\bar{\mathrm{d}}}}$$

$$V_{æ_{0/1}} = 704.9\ knots \quad (10)$$

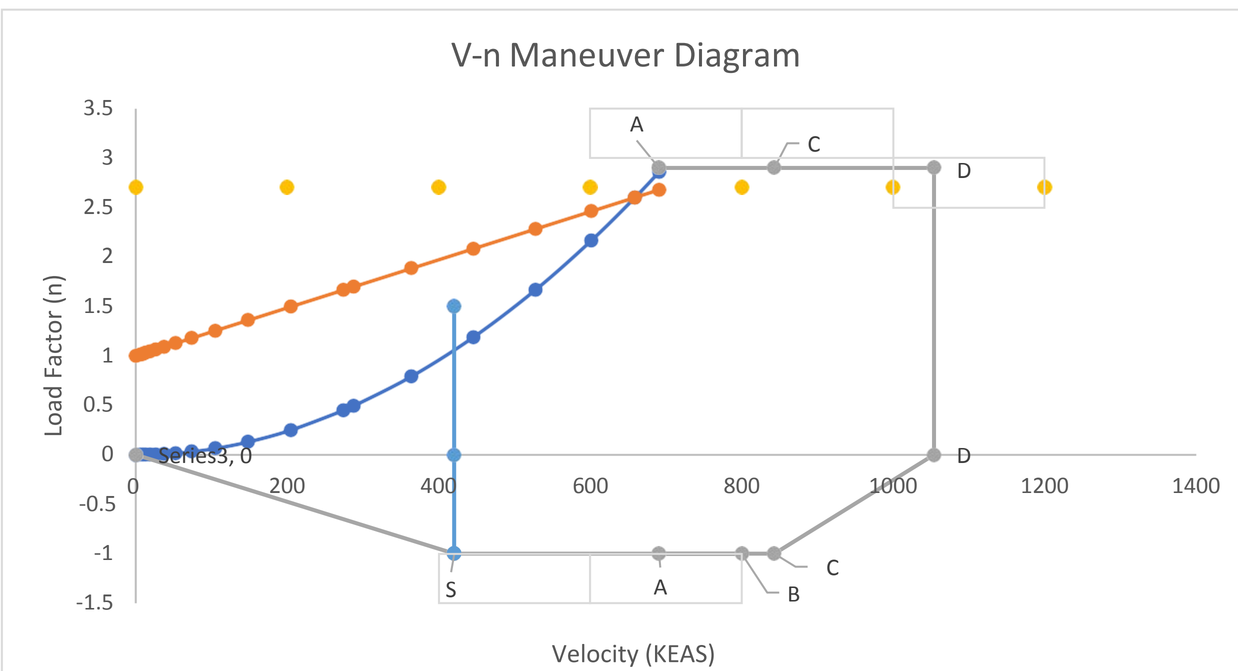


Figure 149: V-n Maneuver Diagram

13.2.1 Calculating the design speed for maximum gust intensity, V_B

V_B should not be greater than V_C.
V_B should be greater than the speed determined from the intersection of the $C\tilde{o}_{9-:}$ line and the gust line marked V_B.

The V$_B$ marked gust line intersects the Cõ$_{9-:}$ line at around 210 knots

$$V_{ú} = 800\ knots \tag{11}$$

13.2.2 Calculating the design cruising speed, V$_C$

V$_c$ must be sufficiently greater than V$_B$ to provide for inadvertent speed increases likely to occur as a result of severe atmospheric turbulence.

$$V_{N} \geq V_{ú} + 43\ kts \tag{12}$$

$$V_{N} \geq 800 + 43 = 843\ knots \tag{13}$$

13.2.3 Calculating the design diving speed, V$_D$

$$V_{=} \geq 1.25\ V_{N} \tag{14}$$

$$V_{=} \geq 1.25 * 843 = 1053.75\ knots$$

13.2.4 Calculating the design maneuvering speed, V$_A$

$$V_{®}(+ve) \geq V_{æ}\,(n_{\text{•'E}})^{\hat{\text{FH}}}_{\text{d}} \tag{15}$$

where, n$_{lim}$ is the limit maneuvering load factor at V$_C$.

$$V_{®}(+ve) \geq 420.45 * 2.3^{\hat{\bar{\text{d}}}}$$

$$V_{®}(+ve) \geq 690.87\ knots \tag{16}$$

$$V_{®}(-ve) \geq 704.90 * -1^{\hat{\bar{\text{d}}}}$$

$$V_{®}(-ve) \geq 704.90\ knots \tag{17}$$

13.2.5 Calculating the design limit load factor, n$_{lim}$.

$$n_{\text{•'E,ë.}} \geq 2.5 + £\frac{\text{mè,>>>}}{\text{T‡Z>,>>>}}© \tag{18}$$

$$n_{\text{Y?9,ë.}} \geq 2.5 + £\frac{\text{mè,>>>}}{\text{ZZ>,>>>‡Z>,>>>}}©$$

$n_{Y?9_{,\ddot{e}.}} \geq 2.7$ (19)

the negative, design limit load factor is determined from:

$n_{Y?9_{0/1}} \geq \ -1.0\ up\ to\ V_{N}$ (20)

$n_{Y?9_{0/1}}\ varies\ linearly\ from\ the\ value\ at\ V_{N}\ to\ zero\ at\ V_{=}$

13.2.6 Construction of gust load factor lines

For the gust line marked V_B:

$U_{B/} = 84.67 - 0.000933h = 84.67 - 0.000933(50{,}000)\ =\ 34.288$ (21)

For the gust line marked V_C:

$U_{B/} = 66.67 - 0.000833h = 66.67 - 0.000833(54{,}000)\ =\ 21.688$ (22)

For the gust line marked V_D:

$U_{B/} = 33.34 - 0.000417h = 33.34 - 0.000417(54{,}000)\ =\ 10.822$ (23)

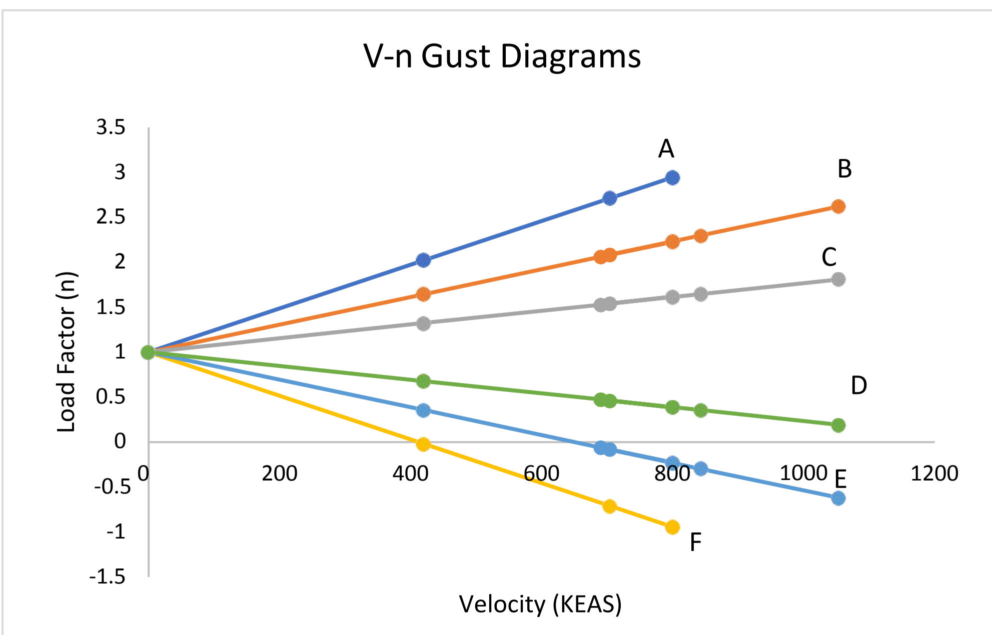


Figure 150: V-n Gust Diagram (in Knots)

13.3 METHODS FOR ESTIMATING THE STRUCTURE WEIGHTS

The airplane structure weight, $W_{.G2\sim 3G}$ consists of the following component weights:

i. Wing, W_{T}
ii. Empennage, $W_{/9,}$
iii. Fuselage, $W_{''}$
iv. Nacelles, W_{0}
v. Landing gears, W_{1}

$$W_{.G2\sim 3G} = W_{T} + W_{/9,} + W_{''} + W_{0} + W_{1} \qquad (19)$$

i. Weight of the Wing:

$$W_{T} = 0.0017\, W_{\pm\&ø} \left(\frac{Ì}{3ëN\Lambda_{d}}\right)^{>.nx} \left[\left(1 + 4\frac{o.p\ \#¶\$*\Lambda_{d}+}{Ì}6\right)^{d} (n_{YG})^{>.xx} \left(\frac{Ìæ}{G)T\approx V\neg 3ë\Lambda_{d}}\right)^{>.p>}\right] \qquad (20)$$

where, W_{MZF} = maximum zero fuel weight = $W_{;<} - W_{ø}$

$$W_{\pm\&ø} = 110000 - 53671.33 = 56328.67\ lbs \qquad (21)$$

substituting the values of the terms obtained into equation 20, we get the weight of the wing to be equal to:

$$\boldsymbol{W_{W} = 6387.4\ lbs} \qquad (22)$$

ii. Weight of the Empennage:

$$W_{/9,} = W_{C} + W_{f} \qquad (23)$$

where, the weight of the horizontal stabilizer can be obtained using Torenbeeks method as follows,

$$W_{C} = K_{C} S_{C} \left[\frac{p.wZz(æ)^{h.d} D}{Z>>>*3ë\ddot{E}_{d}+} - 0.287\right] \qquad (24)$$

K_h = 1.1 variable incidence stabilizers

Substituting the values in the equation, we obtain the weight of the horizontal equal to:

$$\boldsymbol{W_{h} = 7125.13\ lbs} \qquad (25)$$

The weight of the vertical stabilizer can be obtained using Torenbeeks method as follows,

$$W_{D} = K_{D} S_{f D}\left[3.81 \frac{(æú)^{h.d} D}{\left(Z>>>*3ë.Ë_{\hat{d}ú} + \hat{d}\right)} - 0.287\right] \quad (26)$$

$K_D = 1$ $(for\ fuselage\ mounted\ horizontal\ tails)$

$$\boldsymbol{W_V = 3605.2\ lbs} \quad (27)$$

the weight of the empennage can be obtained from the equations 25 and 27.

$$\boldsymbol{W_{emp} = W_V + W_h = 10730.3\ lbs} \quad (28)$$

iii. Weight of the Fuselage

The weight of the fuselage can be obtained from the following equation.

$$W_{"} = 0.021 K_{"} \left| M \frac{D^{\circ Y}}{4 í ‡ C í} P^{ȡ} z S_{"1.} \right|^{Z.m} \quad (29)$$

where, $S_{"1.} = fuselage\ gross\ shell\ area\ in\ ft^{m}$

the fuselage gross shell area is obtained from the previous sections as

$S_{"1.} = 2204.67832\ \frac{Yì}{"_{G}^{d}}$

hence, substituting the values in the equation, we obtain the weight of the Fuselage as

$$\boldsymbol{W_f = 11392.3542\ lbs} \quad (30)$$

iv. Weight of Nacelles

The supersonic business jet has 2 low-by pass turbofan engines due to which the weight of the nacelles can be obtained from the equation

$$W_0 = 0.055\ T_{;<} \quad (31)$$

where T_{TO} is, the total required take-off thrust, hence the equation accounts for all the nacelles.

$T_{;<} = 56933.02 \frac{Yì}{"Gd}$

there the total weight of the nacelles is equal to:

$W_0 = 0.055 * 56933.02$

$$\boldsymbol{W_n = 3131.3161\ lbs} \tag{32}$$

v. Weight of the Landing Gears

The weight of the landing gears for a business jet with main landing gears attached to the wing and the nose gear mounted to the fuselage, the following equation is used:

$$W_1 = K_{1_2}\,\mathrm{F}A_1 + B_1(W_{;<})^{\overset{\mathbf{f}}{\mathrm{FH}}}_{\mathrm{U}} + C_1W_{;<} + D_1(W_{;<})^{\mathrm{d}}\overset{\mathbf{f}}{\mathrm{H}} \tag{33}$$

where,

$K_{1_2} = 1\ (for\ low\ wing\ aircrafts)$

the constants $A_1\ through\ D_1$ are obtained from the table as described below:

Table 5.1 Constants in Landing Gear Weight Eqn.(5.42)

Airplane Type	Gear Type	Gear Comp.	A_g	B_g	C_g	D_g
Jet Trainers and Business Jets	Retr.	Main	33.0	0.04	0.021	0.0
		Nose	12.0	0.06	0.0	0.0
Other civil airplanes	Fixed	Main	20.0	0.10	0.019	0.0
		Nose	25.0	0.0	0.0024	0.0
		Tail	9	0.0	0.0024	0.0
	Retr.	Main	40.0	0.16	0.019	1.5×10^{-5}
		Nose	20.0	0.10	0.0	2.0×10^{-6}
		Tail	5.0	0.0	0.0031	0.0

Figure 151: Constants in Landing Gear Weight

from the table, the values for A_g through D_g for the business jets with retractable landing gears are obtained as follows:

$Main$: $A_1 = 33.0$
$B_1 = 0.04$
$C_1 = 0.021$
$D_1 = 0.0$

$$W_{g} = K_{g_r}\left[A_g + B_g (W_{TO})^{3/4} + C_g W_{TO} + D_g (W_{TO})^{3/2}\right] \quad (34)$$

$$\boldsymbol{W_{g_m} = 2584.6\ lbs} \quad (35)$$

$Nose$: $A_1 = 12.0$
$B_1 = 0.06$
$C_1 = 0.0$
$D_1 = 0.0$

$$\boldsymbol{W_{g_n} = 374.4\ lbs} \quad (36)$$

The total weight of the landing gears can be obtained by adding equation (35) and (36).

$$\boldsymbol{W_g = W_{g_m} + W_{g_n} = 2959\ lbs} \quad (37)$$

The Structural weight of the airplane can be obtained as explained in equation (19) as:

$$W_{struct} = W_w + W_{emp} + W_f + W_n + W_g$$

hence adding the values obtained in equation (22), (28), (30), (32) and (37), we get

$$\boldsymbol{W_{struct} = 34600.3\ lbs} \quad (38)$$

13.4 METHOD FOR ESTIMATING THE POWERPLANT WEIGHT

The airplane power plant weight, W_{pwr} consists of the following components:

- i. Engines, W_e
- ii. Fuel system, W_{fs}
- iii. Propulsion system, W_p
- iv. Accessory drives, Starting and Ignition system, W_{acc}
- v. Thrust reversers, W_{tr}

Hence,

$$W_{pwr} = W_e + W_{fs} + W_{p} + W_{acc} + W_{tr} \quad (39)$$

To estimate the weight of the power plant, it is recommended to obtain the weight data from the engine manufacturers.

i Engine Weight Estimation

The SSBJ uses two Pratt and Whitney J58 Engines that weigh around 6000 lbs each. The exact values can be obtained directly from the manufacturers but for an estimate, these values can be considered.

$$W_{/} = W_{/01} + N_{/01} \tag{40}$$

where,
$W_{/01}$ *is the weight of each engine* $\sim 6000\ lbs$
$N_{/01}$ *is the no of engines* $= 2$

$$W_{/} = 12000\ lbs \tag{41}$$

ii Fuel System

The SSBJ has internally integrated fuel tanks due to which to calculate the weight of the fuel system, the following equation will be used:

$$W_{".1.} = 80(N_{/} + N_{G} - 1) + 15(N_{G})^{>.x} \mathrm{M}\frac{\mathrm{T}}{\mathrm{Ï}_{\mathrm{Í\$F}}}\mathrm{P}^{>.ppp} \tag{42}$$

where,
$N_{/01} = No.\ of\ engines$
$N_{G} = No.\ of\ tanks$
$W_{ø} = Weight\ of\ fuel$
$K_{..} = 6.55\frac{YÌ}{1\text{-}Y}\ (constant\ for\ JP-4\ Fuel)$

Substituting the values in the equation 42, we get the weight of the fuel system equal to:

$$\boldsymbol{W_{fsys} = 1298.5\ lbs} \tag{43}$$

iii Propulsion System

The propulsion system contains the engine controls, the engine starting system, propeller controls, the oil system and oil cooler.

The weight of the propulsion system is the summation of the above systems.

$$W_{,2ë,..1.} = W_{/3} + W_{/..} + W_{,3} + W_{ë.3} \quad (44)$$

where,
The weight for the engine controls for an aircraft using fuselage mounted jet engines can be obtained from the following equation:

$$W_{/3} = K_{/3}\ \mathrm{z}l_{"} N_{/}|^{>.nqm} \quad (45)$$

$K_{/3} = 1.080\ (for\ afterburning\ engines)$
$l_{"} = length\ of\ the\ fuselage$
$N_{/} = No.\ of\ engines$

$W_{/3} = 1.080 * (2 * 98.6)^{>.nqm}$
$W_{/3} = 70.95\ lbs$ (46)

The weight for the engine starting system can be determined using the equation:

$$W_{/..} = 9.33\,\mathrm{F}\frac{\mathrm{T}^{\%}}{\mathrm{Z}>>>}\mathrm{H}^{\mathrm{Z}.>nw} \quad (for\ engines\ using\ Pneumatic\ Starting\ Systems) \quad (47)$$

$$W_{/..} = 38.93\,\mathrm{F}\frac{\mathrm{T}^{\%}}{\mathrm{Z}>>>}\mathrm{H}^{>.q\mathrm{Z}w} \quad (for\ engines\ using\ electric\ starting\ system) \quad (48)$$

As, it is not decided upon the system being used for the Supersonic Business Jet, calculations using both pneumatic and Electric systems will be worked upon and a final decision will be made depending on the usable weight.

The values of the weights obtained for both the systems are as follows:

$W_{/..} = 135.9\ \ lbs\ (for\ engines\ having\ pneumatic\ starting\ systems)$ (49)

$W_{/..} = 381\ \ lbs\ (for\ engines\ having\ electric\ starting\ systems)$ (50)

Weight of the oil system and oil cooler can be obtained using:

$$W_{ë.3} = K_{ë.3} W_{/} \quad (51)$$

$K_{ë.3} = 0.000\ (for\ Jet\ Engines)$

hence that gives us,

$$W_{ë.3} = 0.00\ lbs \quad (52)$$

the total weight of the propulsion system can be obtained as defined in equation (44) by adding the equations (46), (48), and (52)

We get as follows:

$$W_{,2ë,..1.} = 452\ lbs \tag{53}$$

iv Thrust Reversers

An estimate of the weight of the thrust reversers can be made using the equation:

$$W_{GC2} = 0.18\ W_{/} \tag{54}$$

$$W_{GC2} = 15910\ lbs \tag{55}$$

hence, the total weight of the power plant system can be obtained as explained in equation (39) by adding the terms above:

$$W_{,42} = 15910\ lbs \tag{56}$$

13.5 METHOD FOR ESTIMATING FIXED EQUIPMENT WEIGHT

The SSBJ fixed equipment's contain the following components:

- i. Flight control systems, $W_{"3}$
- ii. Electrical systems, $W_{/Y.}$
- iii. Instrumentation, avionics and electronics, $W_{?-/}$
- iv. Air-conditioning + deicing, $W_{-,?}$
- v. Oxygen system, $W_{ë:}$
- vi. Furnishings, $W_{"\sim 2}$
- vii. Paint, $W_{,G}$
- viii. APU, $W_{-,\sim}$
- ix. Baggage and cargo handling, $W_{Ì3}$

The weight of the fixed equipment's of the SSBJ can be obtained by the following equation,

$$W_{"/A} = W_{"3} + W_{/Y.} + W_{?-/} + W_{-,?} + W_{ë:} + W_{"\sim 2} + W_{,G} + W_{-,\sim} + W_{Ì3} \tag{57}$$

i. Flight Control System

The weight of the flight control system for an SSBJ can be estimated from the following equation,

$$W_{"3} = 56.01\,\mathrm{MF}\frac{(\mathrm{T}\_\grave{}\,)(\mathrm{A}\circ)}{\mathrm{Z}>>>>>}\mathrm{H}^{>.\mathrm{xno}}\,\mathrm{P} \quad (58)$$

where, $q_{=}$ $is\ the\ design\ dive\ dynamic\ pressure\ in\ psf$ which can be obtained from the design speed obtained from the V-n diagram and the density obtained at the altitude of the flight.

$$q_{=} = \frac{\mathrm{z}}{\mathrm{m}}\rho_{\sim}\,V_{\sim}^{\mathrm{m}} \quad (59)$$

$$q_{=} = 169.6\ psf$$

$$W_{"3} = 1138.34\ lbs \quad (60)$$

ii. Electrical System Weight Estimation

The weight of electrical systems in a jet transport can be derived from the

$$W_{/\mathrm{Y}\text{-}} = 10.8\mathrm{z}V_{,-:}\,|^{>.\mathrm{n}}\,\mathrm{F}1 - 0.018\mathrm{z}V_{,-:}\,|^{>.\mathrm{px}}\mathrm{H} \quad (61)$$

here, $V_{,-:}$ $is\ the\ passenger\ cabin\ volume\ in\ ft^{\mathrm{p}}$

$$W_{/\mathrm{Y}\text{-}} = 1902\ lbs \quad (62)$$

iii. Instrumentation, Avionics and Electronics

The weight for the instrumentation, avionics and Electronics can be calculated using Torenbeek's method for jet transport.

$$W_{?\text{-}/} = 0.575(W_5)^{>.\mathrm{xxo}}(R)^{>.\mathrm{mx}} \quad (63)$$

where, $W_5 = empty\ weight\ in\ lbs\ and\ R = maximum\ range\ in\ nautical\ miles$

$$W_{?\text{-}/} = 1974.6\ lbs \quad (64)$$

iv. Weight estimation for Air-Conditioning, and De-icing

For pressurized airplanes flying at subsonic speeds

$$W_{-,?} = 469\,\mathrm{F}\frac{\mathrm{D_{F\text{‰}\hat{A}}z\tilde{O}_{<)}\ddagger\tilde{O}_{F\text{‰}\hat{A}}}|}{\mathrm{Z}>>>>}\mathrm{H}^{>.\grave{e}\mathrm{Zq}} \quad (65)$$

substituting the values in the equation gives the total weight of the air-conditioning and De-icing systems being equal to:

$$W_{api} = 972.8\ lbs \tag{66}$$

For Pressurized airplanes flying at Supersonic Speeds, the weight for the air-conditioning and De-icing systems can be obtained using the equation:

$$W_{api} = 202\left(\frac{W_{pax} + 200N_{cr}}{1000}\right)^{0.735} \tag{67}$$

hence, substituting the values, we get the estimated weight for this system being equal to:

$$W_{api} = 427.6\ lbs \tag{68}$$

v. Oxygen System

For business jets flying above 25,000 ft, the following equation will be used to estimate the weight of the oxygen system.

$$W_{ox} = 40 + 2.4N_{pax} \tag{69}$$

substituting the values, we obtain the total weight of the oxygen system

$$W_{ox} = 83.2\ lbs \tag{70}$$

vi. Furnishings

$$W_{fur} = 55N_{fdc} + 32N_{pax} + 15N_{cc} + K_{lav}(N_{pax})^{1.33} + K_{buf}(N_{pax})^{1.12} + 109\left\{\frac{N_{pax}(1 + P_c)}{100}\right\}^{0.505} + 0.771\left(\frac{W_{TO}}{1000}\right) \tag{71}$$

where,

$N_{fdc} = No.\ of\ flight\ deck\ crew$

$N_{pax} = No.\ of\ passengers$

$N_{cc} = No.\ of\ Cabin\ Crew$

$K_{lav} = 3.90\ (for\ business\ Airplanes)$

$K_{buf} = 5.68\ (for\ long\ range\ airplanes)$

$P_c = Design\ Ultimate\ cabin\ pressure$

$W_{TO} = Max.\ Take-off\ weight$

substituting the values of the following terms in equation 71 we get the weight of furnishings equal to:

$W_{\text{'~2}} = 1281.5\ lbs$ (72)

vii. Paint

The weight of paint is generally considered to be between 0.003 to 0.006 times the maximum take-off weight

$W_{\text{,G}} = (0.003\ to\ 0.006)W_{\text{;<}}$ (73)

hence to utilize the minimum weight for every component, the weight of the paint will initially be considered to be equal to 0.003 times the maximum take-off weight.

$W_{\text{,G}} = 0.003 * W_{\text{;<}} = 330\ lbs$ (74)

viii. APU

The weight of the APU (auxiliary power unit) ranges typically between 0.004 to 0.013 times the maximum take-off weight. To obtain the list possible weight estimate for a component in the initial stage of this design, the minimum limit will be considered.

$W_{\text{®*Ä}} = 0.004 * W_{\text{;<}}$ (75)

$W_{\text{®*Ä}} = 440\ lbs$ (76)

ix. Baggage and cargo handling

For the SSBJ, the cargo will be carried along with the passengers in to the passenger cabin hence cargo handling will have no extra weights other than the over-head baggage compartments.

$W_{\text{Ì3}} = 0\ lbs$ (77)

hence according to equation (57), the total weight of the furnishings can be obtained by adding the above individual terms which provide the total weight equal to

$W_{\text{"/A}} = 7577.6\ lbs$ (78)

CHAPTER 14: FIXED EQUIPMENT LAYOUTS

14.1 INTRODUCTION

The purpose of this chapter is to define the fixed equipment's attached to the SSBJ and conduct a study on the configurations and their locations on the design of the airplane. Layouts of the systems as described below will be prepared for the SSBJ design satisfying the weight requirements as obtained in chapter 13.

The Equipment's attached to the SSBJ are as follows:

1. Flight Control System
2. Fuel System
3. Hydraulic System
4. Electrical System
5. Environmental Control System
6. Cockpit Instrumentation, Flight Management and Avionics System
7. De-Icing, Anti-Icing, Rain Removal and Defog System
8. Escape System
9. Water and Waste System
10. Safety and Survivability

14.2 FLIGHT CONTROL SYSTEM

The flight control system can be divided into two sections:

1. Primary Flight Control system
2. Secondary Flight Control System

The primary flight controls are as follows:

i. Ailerons (Lateral)
ii. Spoilers (Lateral)
iii. Differential Stabilizers (Lateral)
iv. Elevator (Longitudinal)
v. Stabilizer (Longitudinal)
vi. Canard (Longitudinal)
vii. Rudder (Directional)

The secondary flight controls are as follows:

i. Primary flight controls
ii. Trailing and leading edge flaps (High Lift Devices)
iii. Engine fuel controls (Thrust)
iv. Manifold gates (Thrust)

v. Propeller blade incidence (Thrust)

The flight control system can be defined as:

i. Reversible flight control systems
ii. Irreversible flight control systems

14.2.1 REVERSIBLE FLIGHT CONTROL SYSTEM

In a reversible flight control system, any movement in the cockpit controls also changes the positions of the aerodynamic controls and vice-versa. The reversible flight control system is typically mechanized with cables, push-rods or a combination.

The major problems associated with this kind of design are:

a. Friction
b. Cable stretch
c. Weight
d. Handling qualities
e. Flutter

Major advantages with this type of the flight control system are:

a. Simplicity (Reliability)
b. Low cost
c. Relatively maintenance free

14.2.2 IRREVERSIBLE FLIGHT CONTROL SYSTEM

In an irreversible flight control system, the aerodynamic control surfaces move only with the change in the cockpit controls and not vice-versa. The irreversible flight control system is hydraulic and/or electrical hence it is an irreversible process. The actuator moves the aerodynamic control surfaces in this system.

The major design problems associated with the irreversible flight control system are:

a. Complexity
b. Reliability
c. Redundancy
d. Cost
e. Accessibility for repair
f. Susceptibility to lighting strikes (for electrically signaled systems)

The major advantages with this type of flight control are:

a. Flexibility in combining pilot control commands with automatic control commands
b. Ability to tailor handling qualities
c. Potential of lower weight (using electrical or optical signaling)

14.2.3 DESIGN FLIGHT CONTROL SYSTEM

Considering the both the flight control systems, reversible and irreversible and the design and specification of the airplane, the irreversible flight control system will be used to design the flight control system for the SSBJ.

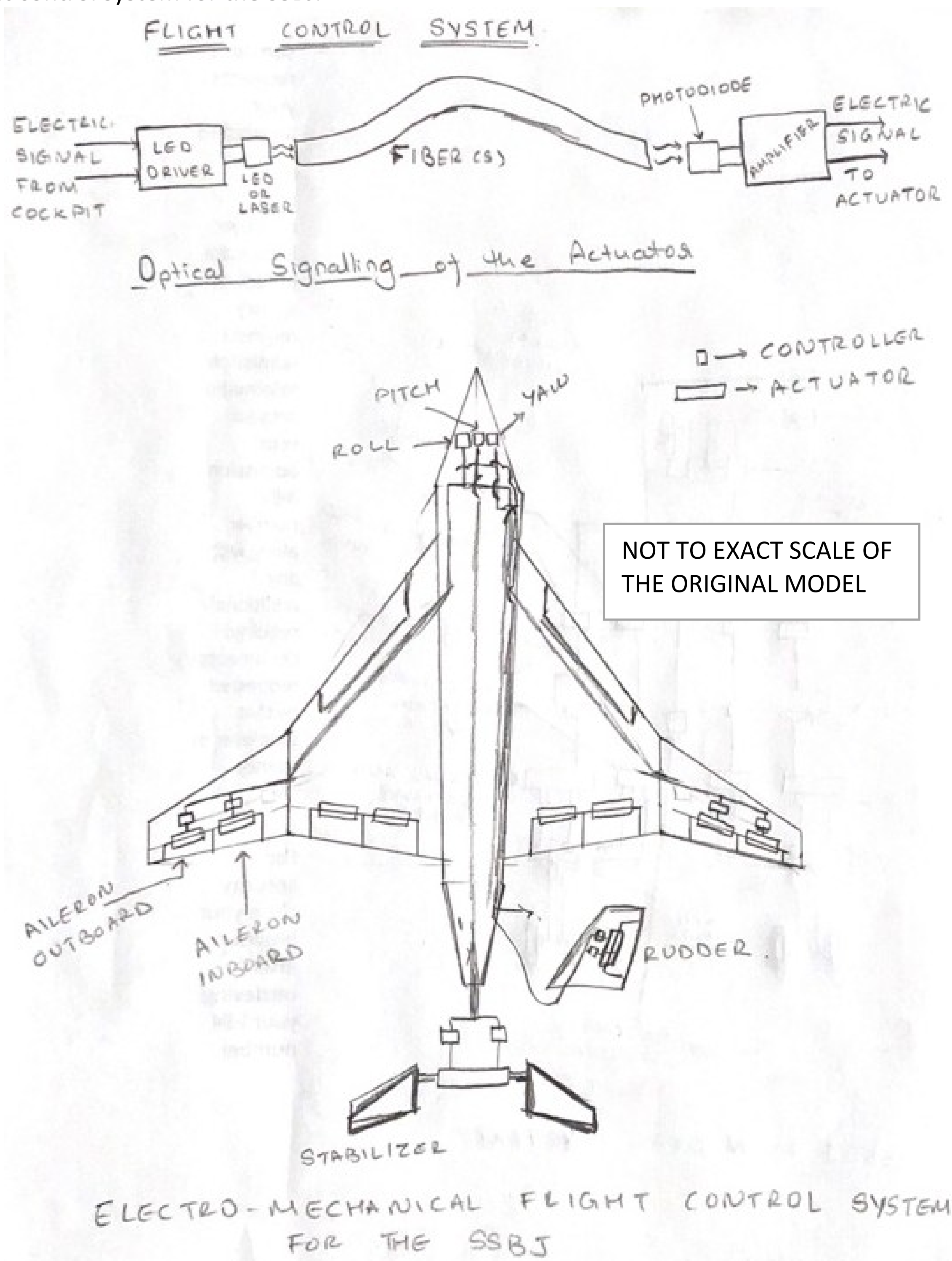


Figure 152 : Electro-Mechanical Flight Control System of the SSBJ

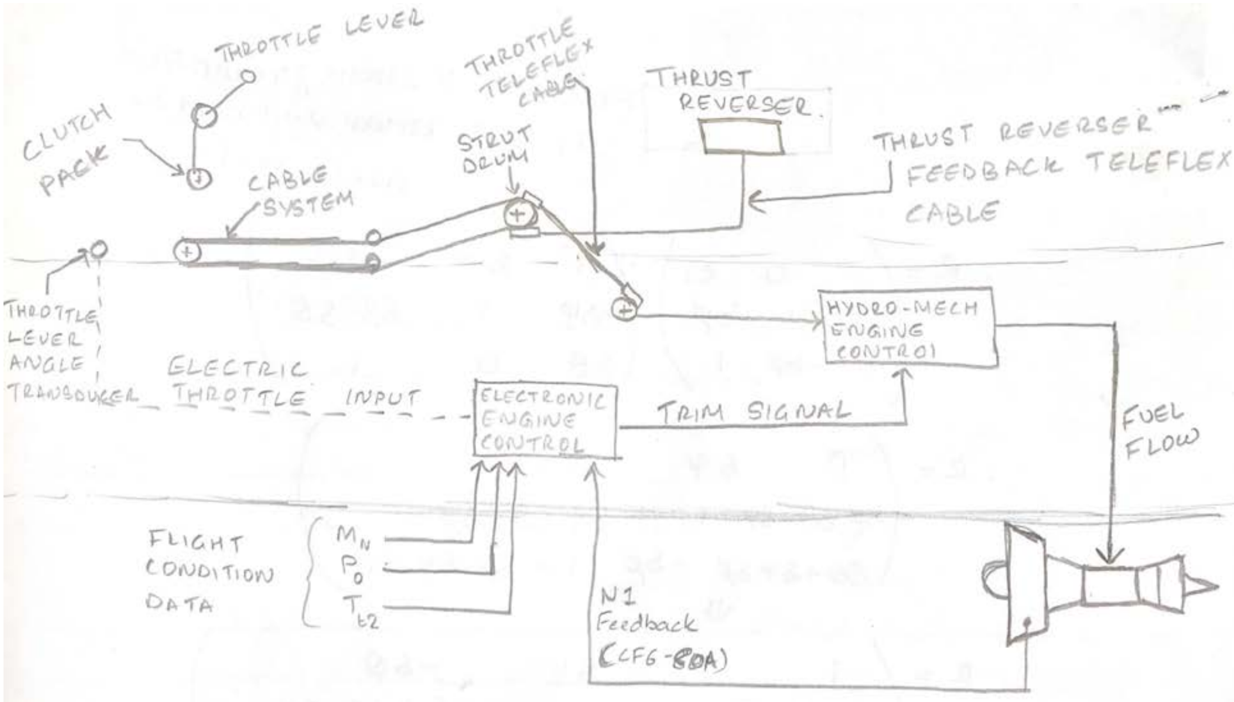


Figure 153: Thrust Control System for the SSBJ

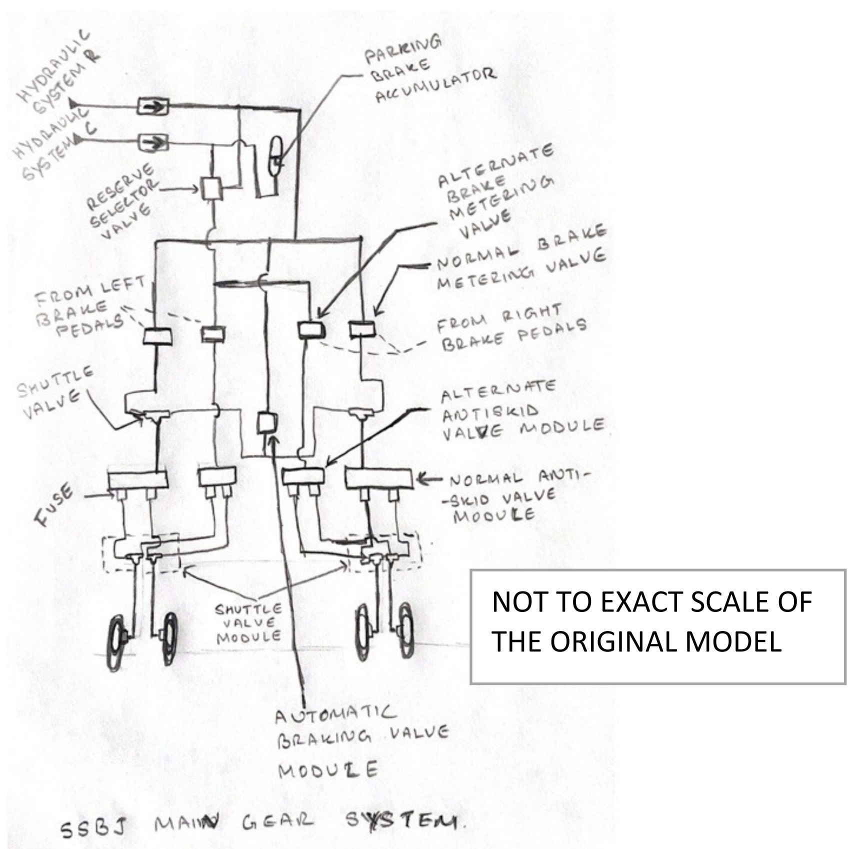


Figure 154: SSBJ Main landing gear control system.

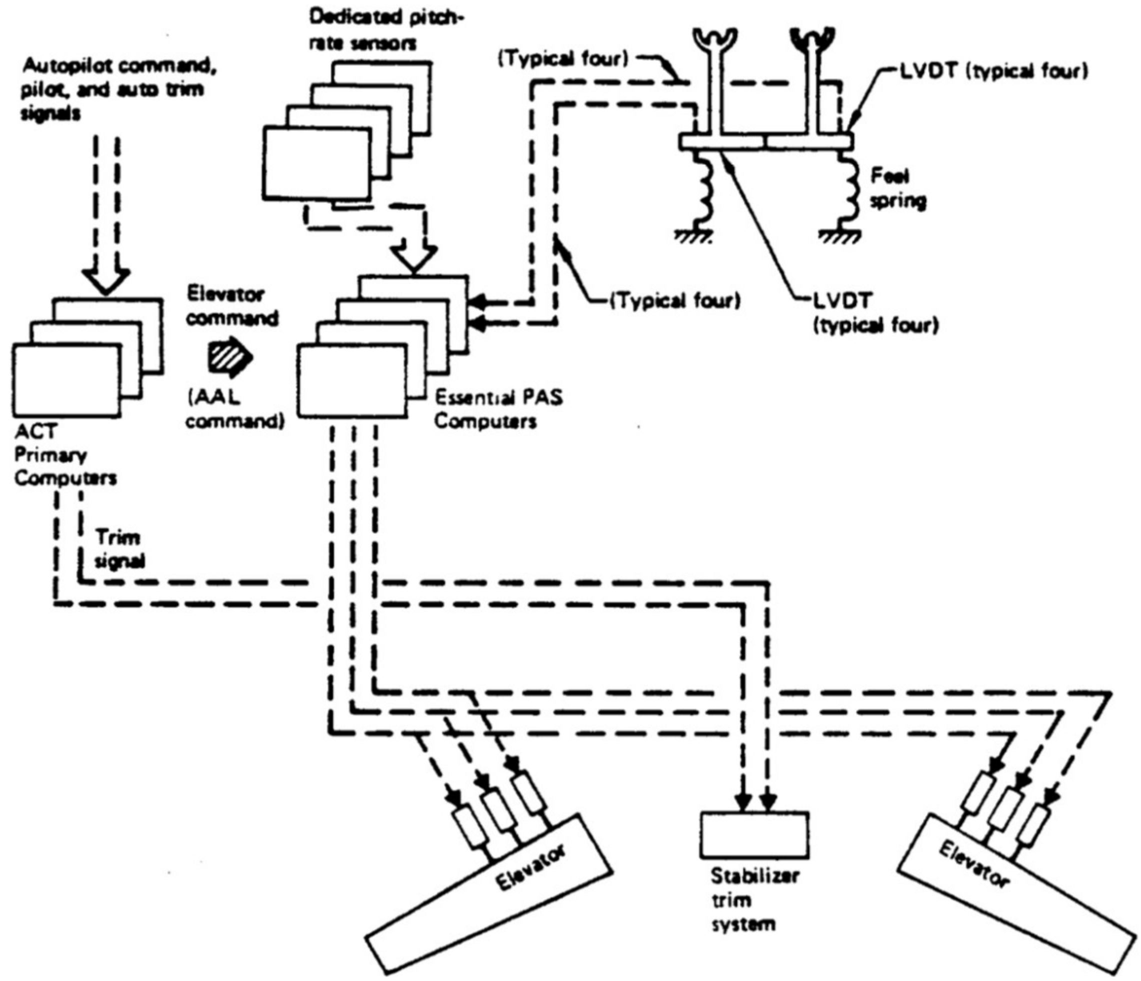


Figure 155: Fly-by-Wire Hydraulic Elevator Control system of the SSBJ

14.3 FUEL SYSTEM

This section specifies the fundamental principles for the fuel system layout design. Since airplane fuels are very combustible liquids, the design, location, operation, accessibility and maintenance aspects are of great importance for the aircraft safety as well as the airplane economy.

To operate properly, most fuel systems need the following components:

I. Fuel tanks that can carry a sufficient amount of fuel to cover the design ranges.
II. Fuel pumps and fuel lines to carry the fuel from the tank to the propulsion system. (dimensioned 1.5 times the maximum required fuel flow)
III. Fuel venting system to prevent excessive pressures from building up in the tanks. Should also provide positive pressure inside the tank during flight.
IV. Fuel quantity indicating system as well as fuel flow indicators.

V. Fuel management system, to allow the crew to regulate the flow from various tanks to different engines (including a shut-off system).
VI. An easy method for refueling must be provided.
VII. If the airplane ramp weight exceeds the maximum design landing weight by more than 5%, a fuel dumping system must be provided.

14.3.1 SIZING OF THE FUEL SYSTEM

The sizing of the fuel system depends on the following design decisions:

i. Total fuel volume required.
ii. Size, location and number of fuel tanks needed.
iii. No. of fuel pumps, location of fuel pumps and required capacity of fuel pumps and fuel lines.

The maximum fuel flow can be obtained by multiplying the maximum required thrust (T) by the associated fuel consumption (c_j).

$$Max.\,Fuel\,Flow = T_{;<}(c_{\acute{u}})$$

the number of fuel tanks and weight of the fuel tanks should be considered to be a minimum from the weight and cost view point.

14.3.2 GUIDELINES FOR FUEL SYSTEM LAYOUT DESIGN

Fuel systems should be equipped with fuel vents and fuel sump systems where the fuel vent system prevents excessive pressure from building up in the fuel tanks. It also serves to maintain the ram-air pressure in the tanks while in flight whereas the surge tanks are used to collect and condense any excess fuel vapor before it exits through the overboard fuel vents.

The following should be kept in mind for the preliminary design of fuel tanks in the aircraft.

1. Fuel tanks cannot easily rupture in otherwise survivable crashes.
2. Fuel lines should be away from easily damaged structure in the case of a crash.
3. Fuel lines should be placed away from any component creating sparks.
4. Should not be near the landing gear wells.
5. The tanks should be away from the engines.
6. Fuel vent lines and fuel dump lines should be located such that the fuel and the fuel vapor can be easily separated.
7. Ram-air inlets should be provided to avoid large asymmetric pressures.
8. Fuel quantity sensors should be placed inside the fuel tank to indicate the correct level of fuel at all stages of flight.
9. Fuel pumps should be added to the fuel systems so the fuel flow can be obtained at all times in any condition of the flight.

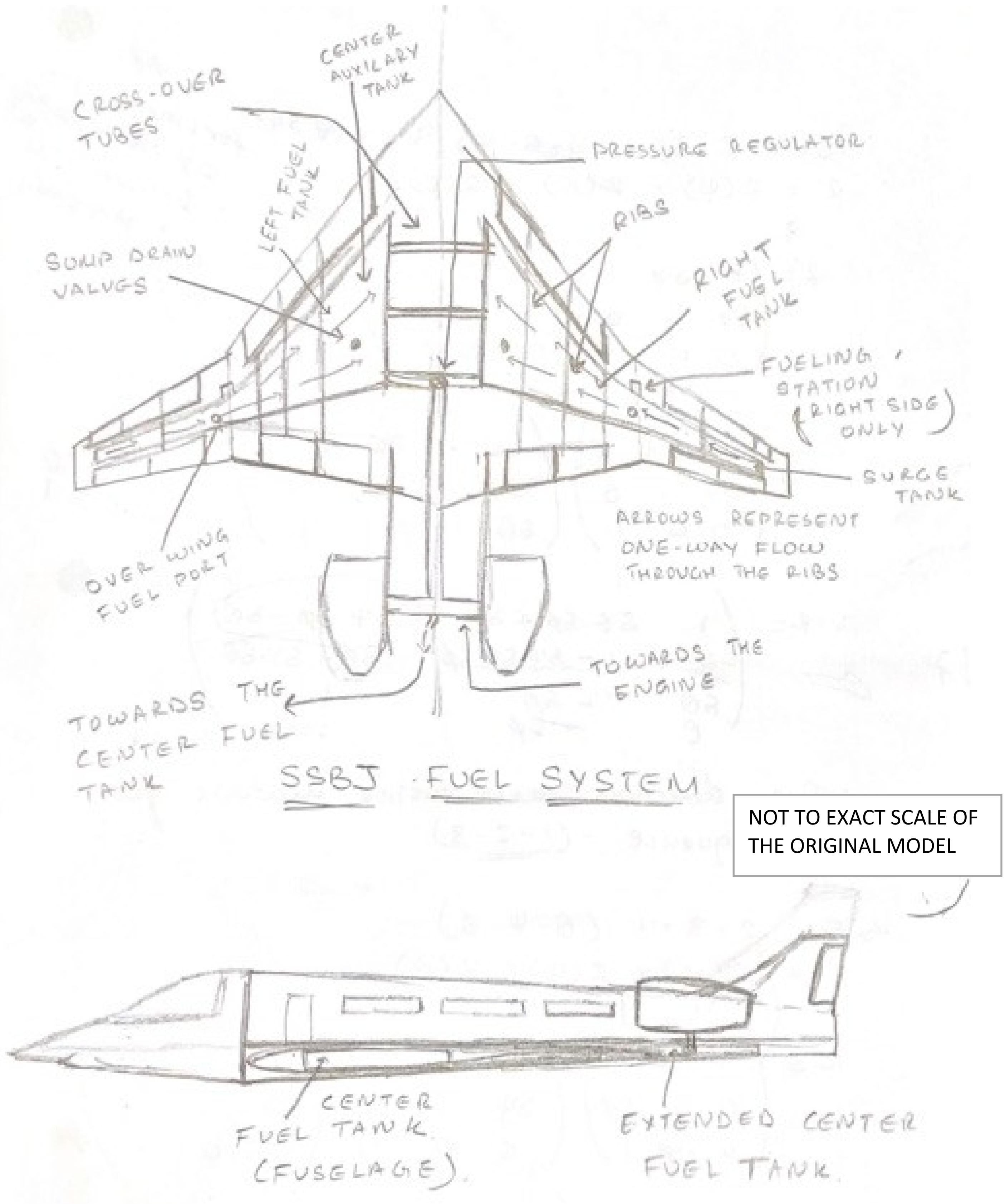


Figure 156: SSBJ Fuel System Layout.

14.4 HYDRAULIC SYSTEM

14.4.1 FUNCTIONS OF HYDRAULIC SYSTEMS

The functions of a hydraulic system vary from aircraft to aircraft. The typical function of a hydraulic system is to provide hydraulic power to the actuators to function the following:

- Moving primary flight controls: ailerons, elevators, stabilizers, rudder and spoilers.
- Moving secondary flight controls: flaps, trim controls and speed brakes
- Extending and retracting the landing gears.
- Controlling the wheel brakes.
- Landing gear steering.
- Operating thrust reversers.

The hydraulic system usually consists of the following components:

- Hydraulic fuel reservoir.
- Hydraulic pumps.
- Accumulators
- Lines and valves for fluid distribution to all operating points.

The number of hydraulic pumps to be used depends on the criticality of hydraulic system to safe flight operations.

14.4.2 SIZING OF HYDRAULIC SYSTEMS

The maximum amount of hydraulic fluid flow required for the operation of an airplane normally occurs during the landing phase to operate both the primary and secondary flight controls, landing gears, speed brakes and may have to be operated simultaneously.

A convenient way to analyze the total fluid flow requirements is to create a list of actuator rate and force requirements from which the gallons per minute flow requirements is obtained.

The power requirements for the hydraulic system for the SSBJ is usually up to the range of 700 hp.

14.4.3 GUIDELINES OF HYDRAULIC SYSTEM DESIGN

- It is essential to make a list of functions to be served by the hydraulic system under normal and under emergency operating conditions.
- Hydraulic system components require service and maintenance due to which they should be easily accessible.
- Hydraulic supply lines should not be close to each other.
- Each system should be independent of each other.

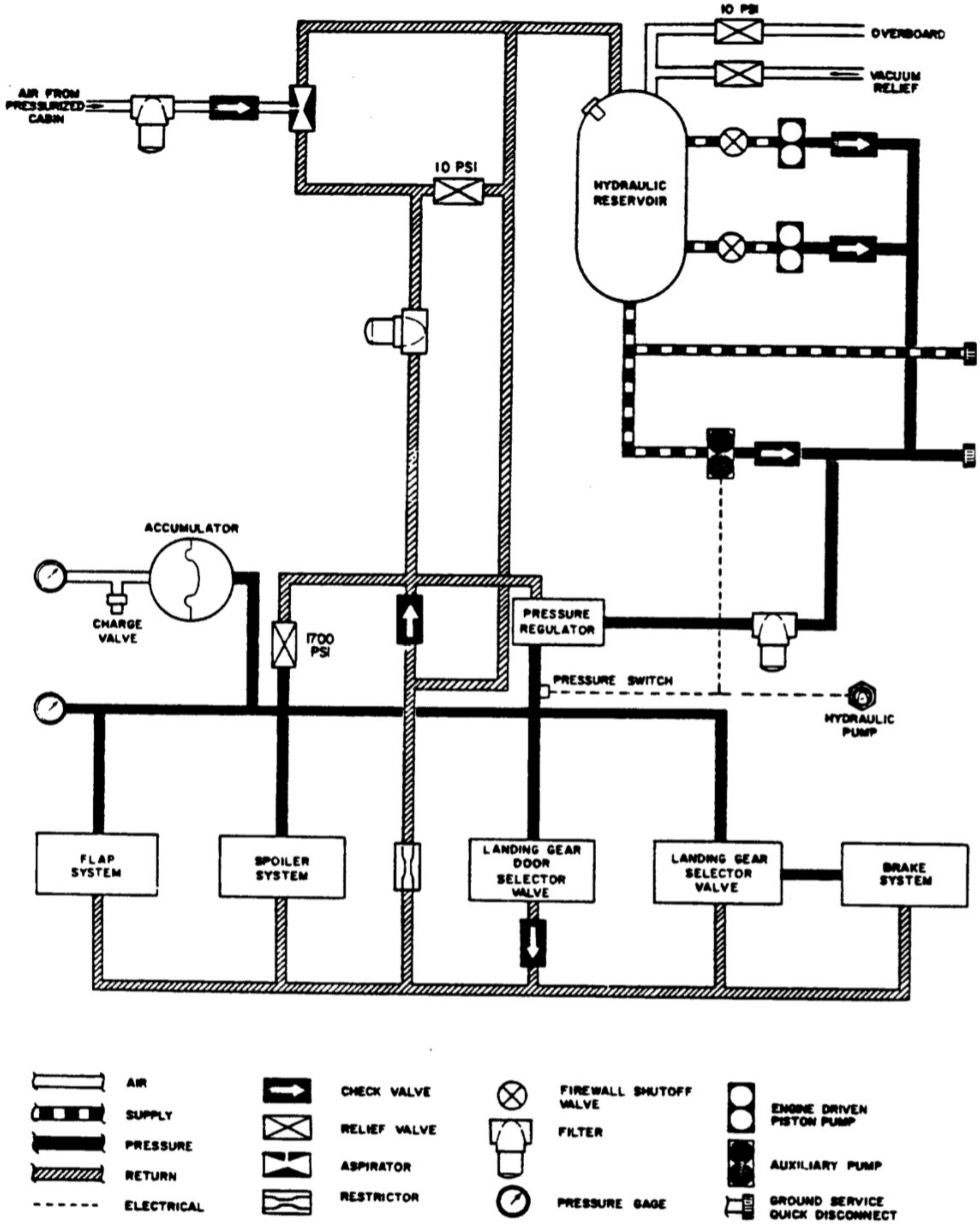


Figure 157: Hydraulic Reservoir of the SSBJ

14.5 ELECTRICAL SYSTEM

All airplanes require electrical power for the operation of a large number of systems. For the Supersonic Business Jet, the Electrical system will be used to provide power to the following systems:

- Internal and External lighting.
- Flight instruments and avionics systems.
- Food and beverages heating system.
- Engine starting system.
- Flight control system (primary and secondary)

The electrical power is mainly generated by two systems:

1. Primary power generating system: engine driven generators
2. Secondary power generating system: Battery, APU, RAT (Ram-air Turbine)

14.5.1 MAJOR COMPONENTS OF THE ELECTRICAL SYSTEMS

Under normal operating conditions, the electrical power is generated by the engine driven generators or the alternators. These devices may be designed to generate DC/AC power.

In the DC generators, their primary power is fed to the DC buses of the airplane and to inverters to derive AC power.

In the AC generators, their primary power, is fed to the AC buses of the airplane and to the transformer/rectifier systems to derive DC power.

14.5.2 SIZING OF ELECTRICAL SYSTEMS

To determine the electrical power requirements of the aircraft, it is necessary to construct an electric power load profile. Electric power requirements should be obtained for each phase of the mission profile of the airplane.

The electrical systems are designed for two types of load requirements:

1. Essential load requirements: determined by the sum of all electric loads which are essential for the safe operation of the flight.
2. Normal operating load requirements: determined by the maximum load requirement for each phase of the flight.

The following must be kept in mind while designing the electrical system of the airplane:

1. Electrical systems must be shielded from the effects of lightning strike.
2. Should be designed in such a way that they are shielded from each other.
3. Electrical systems must be designed so that airplane dispatch is possible with certain system components failed.

4. Servicing and accessibility of electrical system components must be easy and safe.
5. Flight crucial buses and/or wiring bundles should be widely separated to avoid catastrophic results under the following scenarios:
    - Uncontained failure of engine components
    - Terrorist action
    - Failure of adjacent structure
    - Localized in-flight fires
6. Batteries should be provided for various stand-by functions. They should be physically shielded from the primary structure as any kind of leak could cause corrosive effect.

14.5.3 SSBJ ELECTRICAL SYSTEM LAYOUT

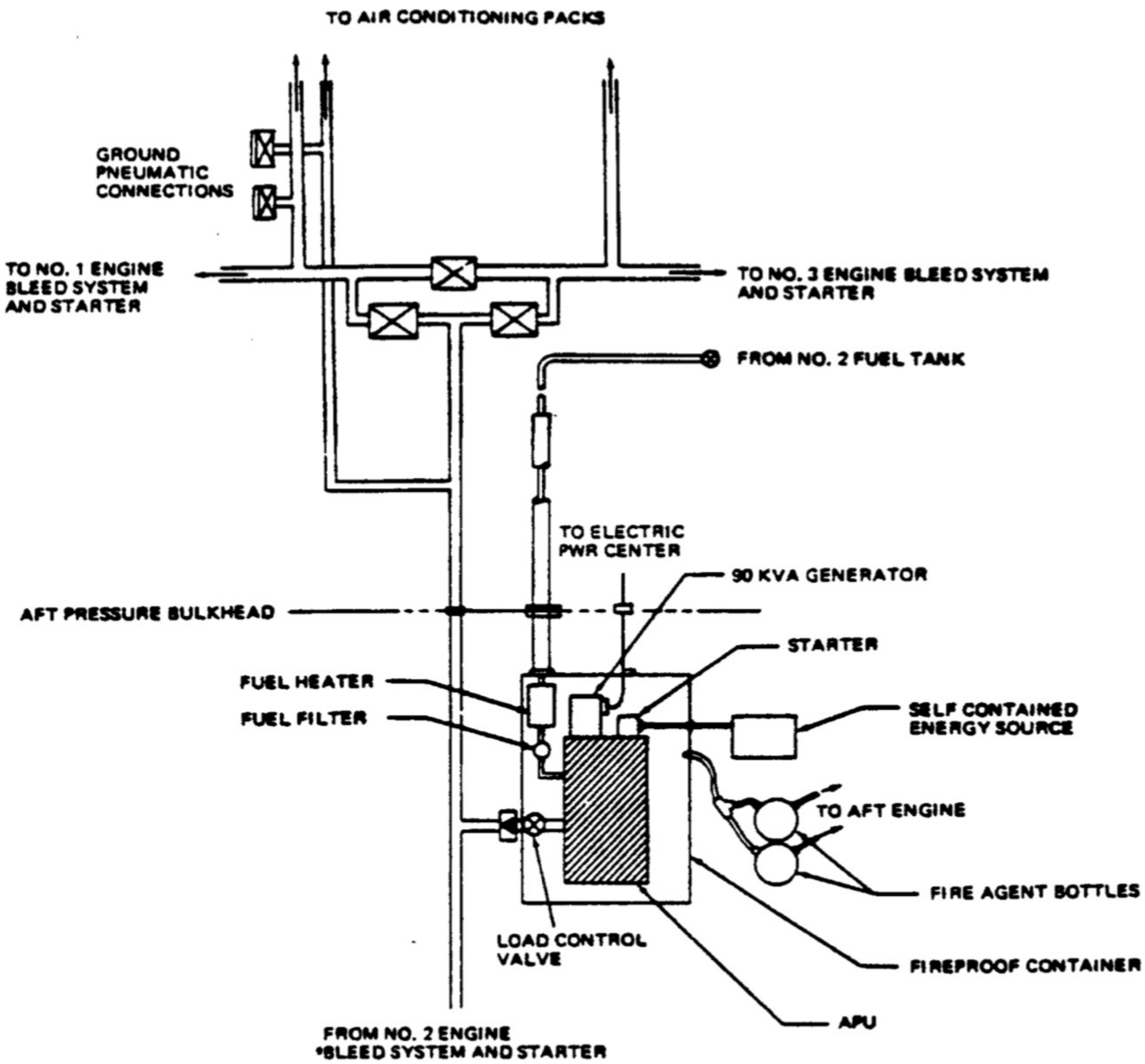


Figure 158: APU electrical system layout

14.6 ENVIRONMENTAL CONTROL SYSTEM

14.6.1 PRESSURIZATION SYSTEM

The purpose of the pressurization system is to maintain sufficient cabin air pressure at higher altitudes during the flight so that the passengers remain comfortable. Typical differential pressure capabilities in jet transports are designed to maintain a cabin altitude from 1000ft below sea level to 10,000 ft above sea level.

The cabin pressurization system needs the following components:

I. A source of high-pressure air. The air source is mainly the pneumatic system.
II. A control and metering system to: provide positive pressure relief to protect the structure. This pressure is typically set for a pressure differential larger than 9-10 psi. and negative pressure relief to let the air into the cabin when the outside pressure is more than the inside pressure.

The pressurization system is of great importance to the aircraft as if it fails, the passengers will have a problem breathing due to which an emergency oxygen system is installed in all jet transports. With the failure in the pressurization system while landing, it would be impossible to open the cabin doors.

Major problems can arrive if the cargo doors located below the cabin floor accidentally blow out causing the pressure in the cargo bleed off rapidly. The pressure difference then created causes the cabin floor to fail.

This can be avoided by providing proper location of essential controls, providing pressure relief for the cabins if the cargo door fails and a fail-safe design of a cargo door.

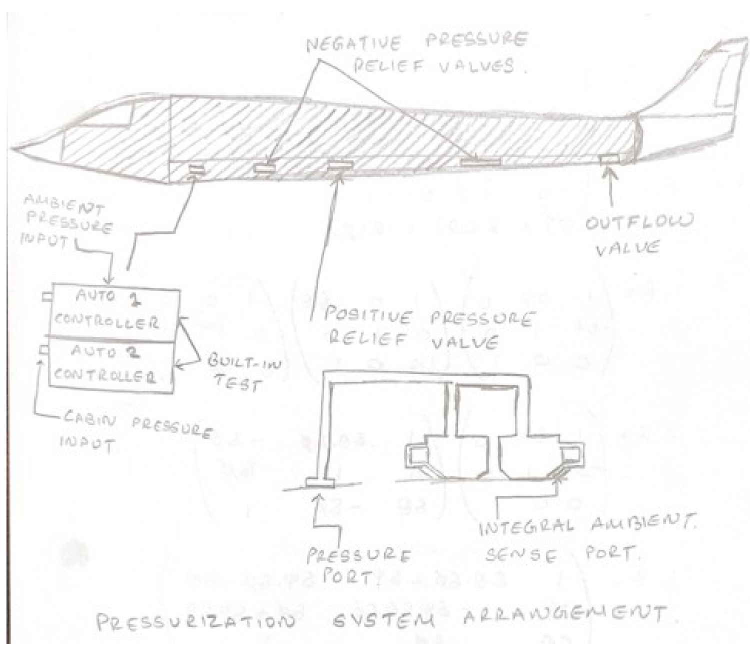


Figure 159: Pressurization System for the SSBJ

14.6.2 PNEUMATIC SYSTEM

The purpose of the pneumatic system is to supply air for the following uses:

- cabin pressurization and air conditioning
- Ice protection system
- Cross engine starting

14.6.3 AIR-CONDITIONING SYSTEM

The air-conditioning system is used to condition the air in terms of temperature and humidity. The overall efficiency of the cabin air conditioning system depends a great deal on the thermal insulation of the cabin walls.

The air coming from the air conditioning system must be distributed into the cabin. The amount of cabin air required in jet transports is typically 20 cubic feet per minute per passenger. The air conditioning system should be designed and distributed properly as improper designs can cause a lot of noise

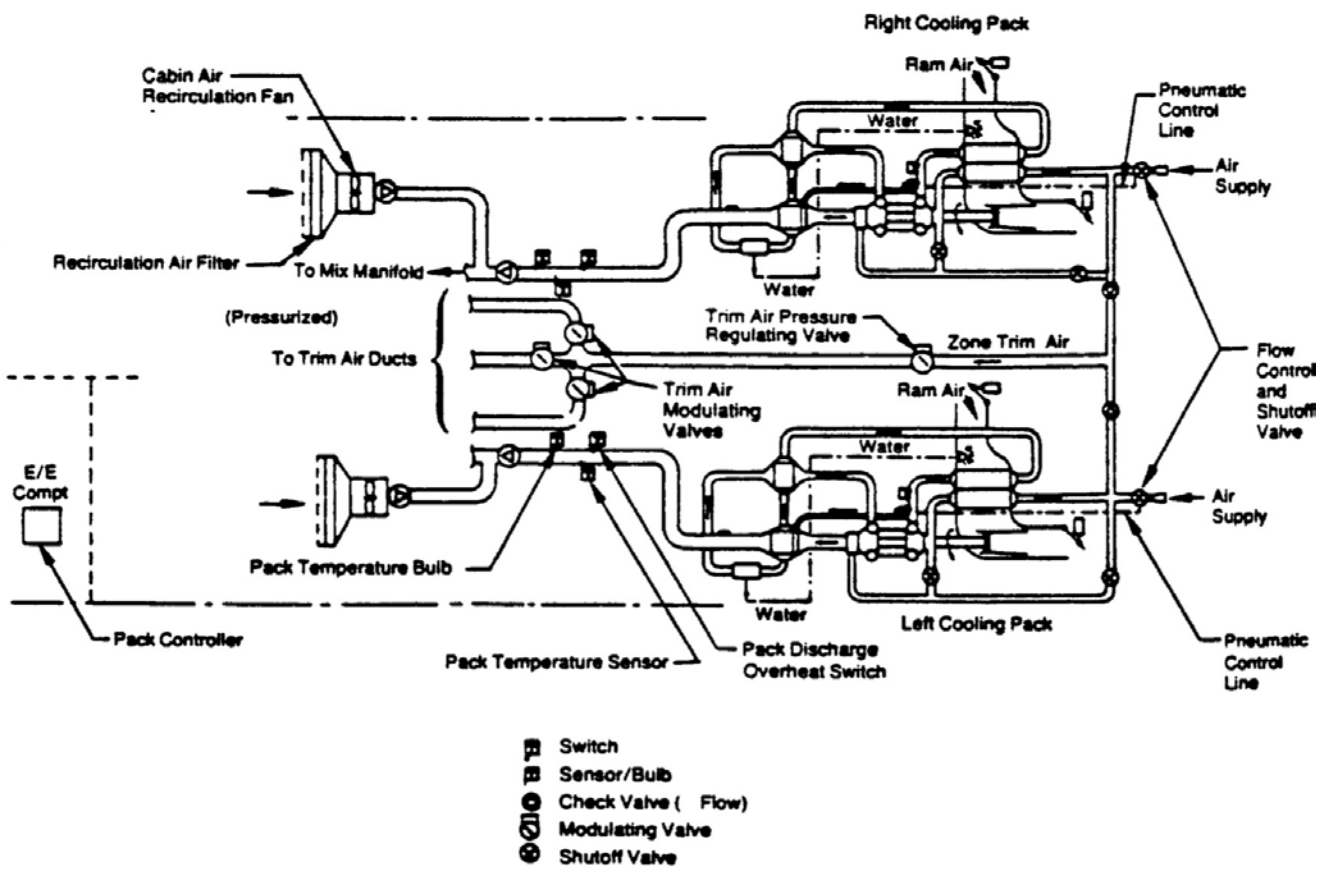


Figure 160: Air Conditioning System

14.6.4 OXYGEN SYSTEM

At higher altitudes, oxygen is required after the failure of the cabin pressurization system. The oxygen system's use usually gaseous oxygen or a chemically obtained oxygen. The crew oxygen system is normally supplied from a gaseous source whereas the passenger oxygen is supplied form a chemical source.

The disadvantage associated with the use of gaseous oxygen is that it presents a fire hazard during servicing and during cylinder replacement and main disadvantage being its larger weight.

14.7 COCKPIT INSTRUMENTATION, FLIGHT MANAGEMENT AND AVIONICS SYSTEM

14.7.1 COCKPIT INSTRUMENTATION LAYOUT

The layout of the cockpit instrumentation system should be uncluttered and functional. The crew must be able to see all flight crucial instruments, controls and warning devices. Since cockpit instrumentation and airplane avionics are evolving rapidly, changes occur almost each year.

A typical cockpit instrumentation panel layout is described in the figure below.

14.7.2 FLIGHT MANAGEMENT AND AVIONICS SYSTEM LAYOUT

Flight management and avionics systems are undergoing very rapid development. Also, the number of different systems available to the user is so large. In recently developed aircrafts, the pilot interfaces with the flight management system.

The flight management system is made up of a number of subsystems:

1. Flight control computer
2. Auto-pilot/ Auto-throttle controls
3. Thrust management computer
4. Inertial reference system
5. Flight data acquisition systems
6. Communication and advisory systems

14.7.3 ANTENNA SYSTEM LAYOUT

for communications between the aircraft and the ground a large number of antenna systems are required. An example of the antenna systems installed on an aircraft is described as below.

14.7.4 INSTALLATION, MAINTENANCE AND SERVICING CONSIDERATIONS

Many avionics equipment's in an aircraft consume a considerable amount of electric power. Most of the power is transformed into heat. This eventually leads to major malfunctions in the avionics equipment's due to which cooling is important. An assumption is that most of the electrical and electronic equipment's fail frequently due to which they should be easily accessible which shows a good design layout.

The radar system and the flight control antennae are accessed through the removal of the radome.

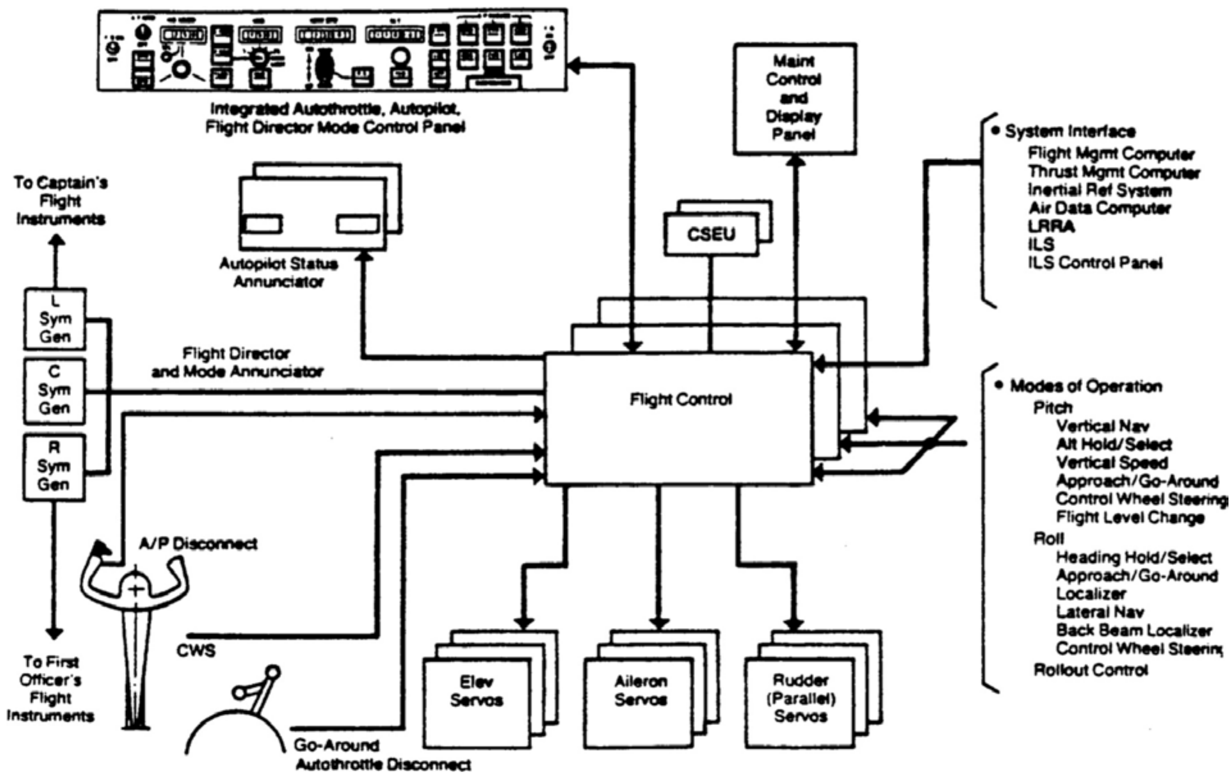


Figure 161: Flight Control Computer Functions

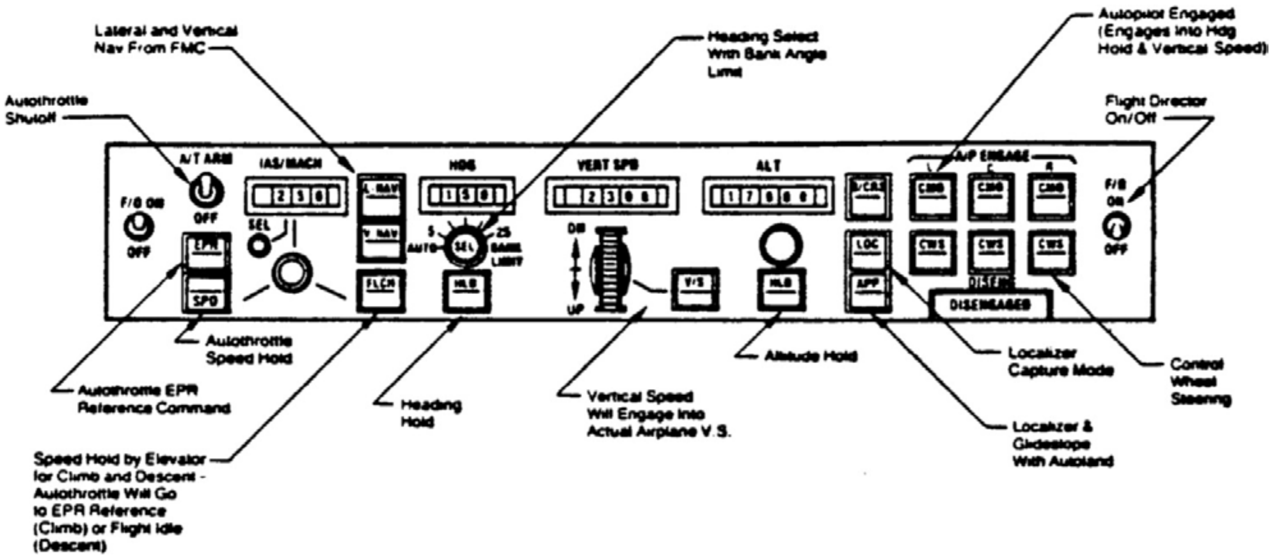


Figure 162: Autopilot/ Auto throttle Control Panel

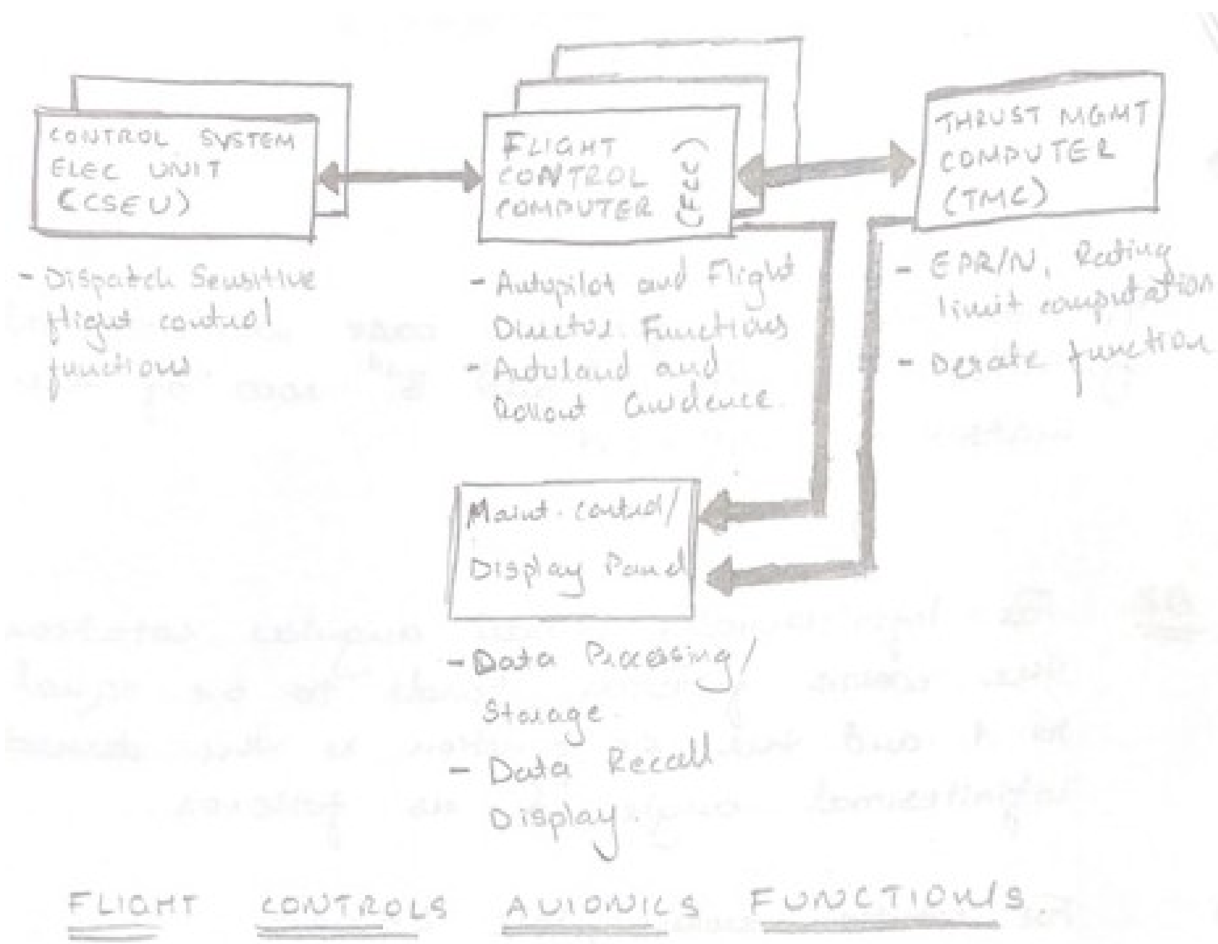


Figure 163: Flight Controls Avionics Functions

14.8 DE-ICING, ANTI-ICING, RAIN REMOVAL AND DEFOG SYSTEM

Whenever an airplane can be expected to be operated under icing conditions, certain special systems must be installed to prevent or remove the ice from the surface of the airplane.

In addition, when the airplane is flying under rainy conditions, the water accumulating on top of the wind shields which causes a serious visibility problem a rain removal system must also be equipped to the airplane.

Under certain combinations of humidity and temperature fog tends to form on the windshields which also affects the visibility. Hence to avoid this a defog system must be made available for the safety of the flight.

14.8.1 DE-ICING AND ANTI-ICING SYSTEMS

Ice formation on the wind shields can cause the following consequences:

1) Ice formed on the wings and tails can distort the aerodynamic contours such that:
    a. The drag increases which causes the airplane to slow down and may even loose its climb capability.
    b. The lift decreases due to a sharp drop in the maximum lift coefficient which leads to early stall when the pilots maneuver the airplane.
    c. There is a change in the pitching moment which leads to unexpected trim changes causing changes in the speed of the stick-force or the stick-force per ‘g’ gradients.
2) Ice formed at the engine inlets results in serious degradation of the engine performance and in the case when the ice breaks loose from the inlet can lead to serious damage to the engine.
3) Ice formed on pitot inlets, stall vanes or any sensor critical to the safe operation of an airplane can result in accidents

14.8.1.1 DE-ICING SYSTEMS

There are 2 types of the de-icing systems, the De-icing boots and the Electro-impulse systems. For the SSBJ design as the engines are attached at the rear of the fuselage, the Electro-impulse system will be used. Using the De-icing boots will tend to increase the weight of the aircraft with added devices to use the engine bleed air to heat the rubber boots. An example of the Electro-impulse system is as shown below:

(Part 4: ch-10: pg 361)

The Electro-impulse system is operated by delivering mechanical impulse to the surfaces where the ice has been formed. These impulses are delivered by electromagnetic coils installed on these surfaces. The above figure depicts the cross-section of a leading edge with an electro-impulse system installed.

14.8.1.2 ANTI-ICING SYSTEMS

The main reason to include an anti-icing system to the design is to avoid the formation of ice on the surface of the airplane. These systems are to be turned on as soon as the crew suspects that their flight will encounter conditions favorable to the formation of ice.

The following anti-icing systems are available to be used in an aircraft:

i. Thermal anti-icing system
ii. Chemical system
iii. Carburetor heating system
iv. Inertial anti-ice system

Considering all the above anti-icing systems available for an aircraft, a conclusion is made considering safe to move forward with the thermal anti-icing system for the Supersonic Business Jet design as it can also be used as a de-icing system for the aircraft.

Using the same system for both of the purposes De-Icing and Anti-Icing helps to obtain the following:

1) Reduce the number of systems and make the design less complex
2) With the reduced number of systems, we reduce the cost and use of materials and also reduce the total weight which is an important aspect in the design of an aircraft flying at supersonic speeds.

Two types of thermal anti-icing systems are used,

- Air heated systems
- Electrically heated systems.

Air heated anti-icing system blows hot air through the surfaces where there is a possibility for the formation of ice. The thermal anti-icing systems are sometimes also used to de-ice the surface of the aircraft. Hence it is suggested to avoid the formation of ice on the surface rather than removing it later on after it has formed.

Air heated systems are used to avoid the formation of ice on the leading-edge slats as well as the engine nose cowls.

In electrically heated systems, electrical resistances are used to heat the surfaces where there are high chances of the ice to be formed. It is also used to prevent ice formation at the pitot tube inlets, stall vanes and the total temperature probes.

The Anti-Icing system for the Supersonic Business Jet is as described below:

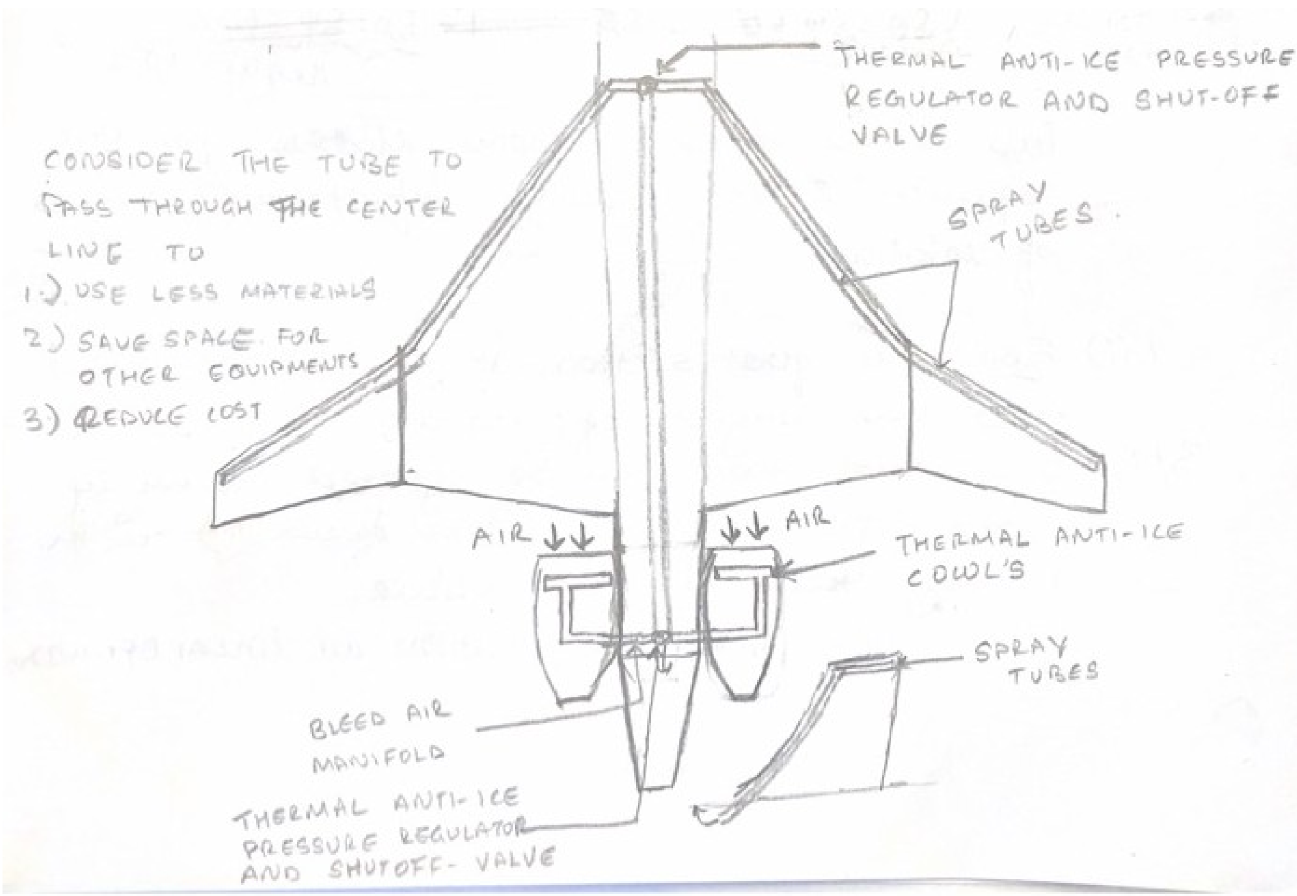


Figure 164: Air Heated Anti-Icing/De-Icing System for the SSBJ

The defog system is similar to the anti-icing / de-icing system which uses hot engine air to defog the windshields.

14.8.2 RAIN REMOVAL AND DEFOG SYSTEMS

The rain removal and Defog systems are important as they help to obtain proper visibility of what's ahead during rains and temperature differences which causes fog on the windshields of the airplane.

The rain removal system is usually satisfied by the use of windshield wipers similar to those used in cars. It is also important to add rain repellant into the wiper paths.

To prevent the windshield from fogging from inside and/or outside, a defog system can be added to the windshields. This can be achieved by installing electrical wirings inside the windshield material.

The Rain removal and defog system to be used in the SSBJ are as follows:

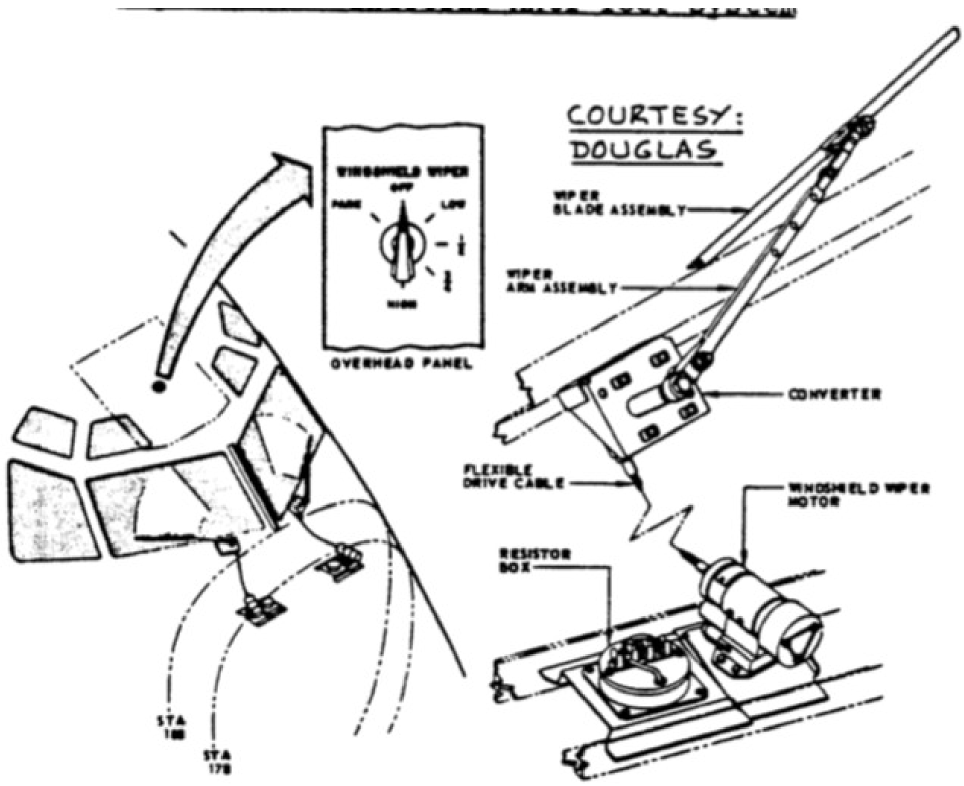


Figure 165: Rain removal system using wipers similar to those in cars.

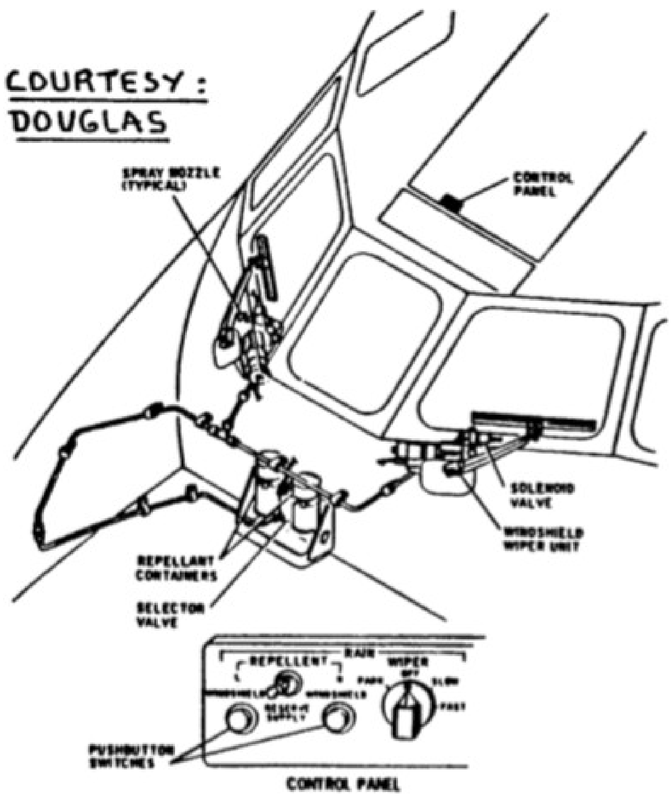


Figure 166: Rain repellant system added to the wiper paths

14.9 ESCAPE SYSTEM

The purpose of the escape system is to provide emergency exits to the passengers for their safety in the case of emergency situations. All airplanes must have emergency exits for the safety of the passengers according to the FAR-25 requirements, the aircraft should satisfy the number of exit routes required according to its size and all the exit routes should be marked properly and should also indicated self-illuminating signs. The operation of each emergency exit must be prominently displayed on the exit.

All passenger transports must be equipped with life jackets and all life jackets must be within reach to the passengers due to which they are installed under the seats of the passengers.

For over water flights (for more than 30 minutes), the aircrafts must carry emergency rafts. There should be sufficient space to carry the passengers as well as the crew members.

14.10 WATER AND WASTE SYSTEM

All passenger airplanes are equipped with water and waste systems. They occupy a large amount of weight and volume in an aircraft. It is important to include these systems in the preliminary design considerations.

The water systems are typically sized as 0.3 gallons per passenger. The system is usually pressurized with air from the airplane pneumatic system. These systems contain drain masts which must be heated to avoid them from freezing. A large concern is the location of the drain masts, if not heated large blobs of ice may be formed. If these blobs of ice break, they should not be ingested by the engines.

In some flights, both hot and cold water is available, the hot water is obtained by running cold water through an electrically heated heat exchanger.

Waste systems in an airplane are self-contained, they have waste tanks and flushing units which mix the waste materials with chemicals contained in the flushing unit.

The number of lavatories required in an aircraft varies directly with the number of passengers in the aircraft. Usually 1 lavatory per 30 passengers is a necessity. Both the water and waste systems need to be serviced after each flight. The access to service these systems should not interfere with the access to service the other components of the aircraft while loading or unloading.

The service of both the water and waste systems is an important parameter to be considered during the preliminary design of the aircraft to avoid leakage as the water/waste leakage can form ice which when breaks should not be ingested by the engines of the aircraft.

For the SSBJ, the vacuum waste system will be used for the water and waste management as it is less in weight due to lower number of components on it. Which helps to maintain the overall weight of the design.

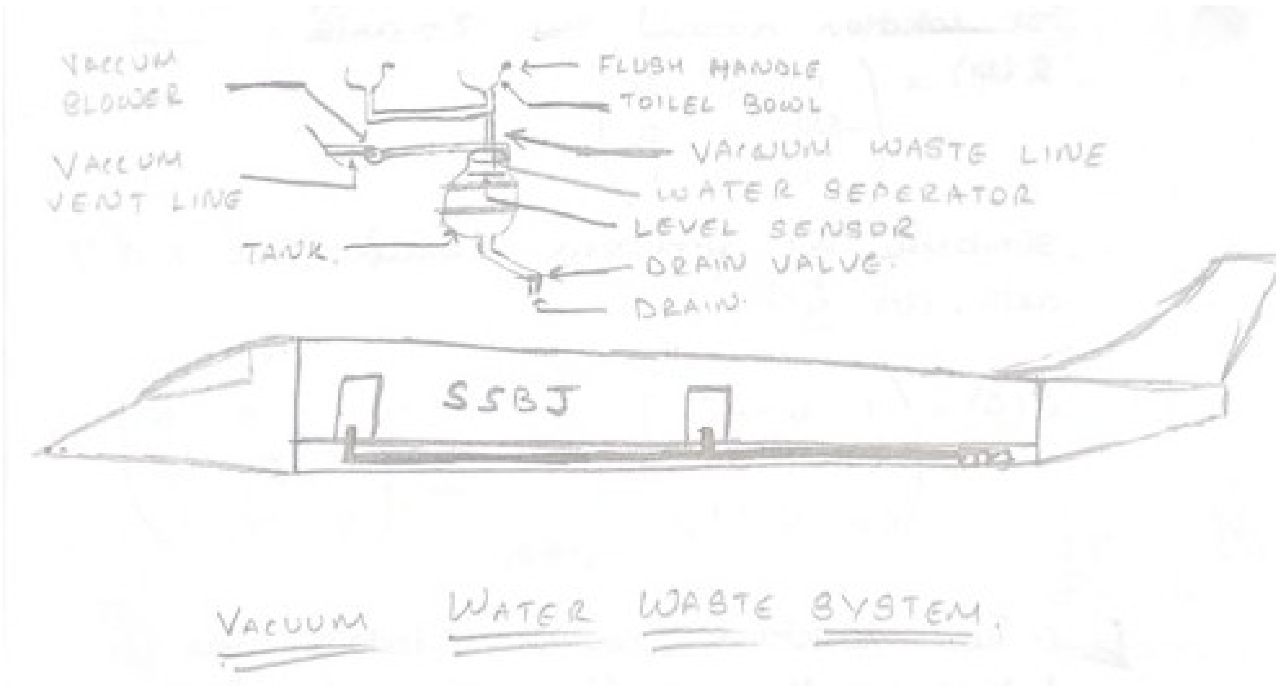


Figure 167: Vacuum Water Waste System for the SSBJ

14.11 SAFETY AND SURVIVABILITY

The purpose of this chapter is to provide details on the design insight of safety and survivability considerations. The main problem associated with the safety of the aircraft is how safe is safe enough. It is impossible to design airplanes such that the probability of fatalities is zero. Acceptable levels of safety will be discussed considering the trade-off of cost incurred to reduce the number of fatalities over the time.

The FAA is responsible for setting up the Airworthiness regulations, carrier operating rules and enforcing these rules and regulations. It is also responsible for the operation of air traffic control system.

The Airplane system and associated components should be considered separately. They must be designed such that any occurrence of failure would prevent and continue the safe flight and landing of the aircraft is extremely improbable.

The factors that contribute to the safety of Aviation are as described in the table below:

Topic	Contributing Factors
Aircraft design and manufacturing	Airworthiness requirements: - performance and flying qualities - aircraft structure and loads - powerplants - aircraft systems - crashworthiness Aircraft production control
Aircraft flight operations	Flightplanning Air Traffic Control and Air Navigation Airport lay-out and facilities Aircraft maintenance
Personnel	Selection, training and licensing of - operational staff - technical staff
Abnormal events	Occurrence reporting and accident investigation

Figure 168: Factors affecting the safety of Aviation

in accidents of aircrafts, two types of accidents are considered:

- Predominantly Airworthiness
- Predominantly operation

Over half of the accidents are caused due to human factor. More accidents were occurred due to the errors made by the flight crew compared to the errors made by the ground crew.

In designing for safety and survivability, the following factors are to be considered:

I. Preventive factors
   a. Benign flying qualities: plenty control power with moderate cockpit control forces, certainly in engine out emergencies. Changes in the flap setting and power setting should be easily controlled.
   b. Easy inspectability of the structure for fatigue crack detection.
   c. Production and materials quality control.
   d. Design systems for the ease of operation and prevent any mistakes attached to the design.

II. Post-crash factors

Despite the best of care in preventive designs crashes do occur. The cabin environment is supposed to be survivable to survive during crash conditions.

a. The structure and seats should not fail in a hazardous manner under g loadings which are survivable by the human body.
b. Prevent fires by the safe fuel system design.
c. Prevent the use of materials which generate toxic fumes when ignited by fire.
d. Arrange emergency exits so people have a chance of getting out in case of emergency situations.

CHAPTER 15: CLASS II CG LOCATIONS

This chapter shows the locations of the center of gravity of each and every component of the supersonic business jet for the class II design of the aircraft. The locations of the CG can be obtained by initially dividing the components into the following parts and then calculating the CG locations with the equations provided.

15.1 C.G. locations of Structural Components
15.2 C.G. locations of Power plant locations
15.3 C.G. locations of Fixed Equipment

15.1 C.G. locations of Structural Components

Most likely C.G. locations for the major structural components of the SSBJ are as defined below:

Wing (Swept wing): 70% of the distance between the front and rear spar behind the front spar at 35 percent of the semi-span.

As explained in the class I design of the airplane, the C.G. of the wing is located at 38%-42% of the mean aerodynamic chord. But due to the modified cranked arrow wing, the CG of the wing was obtained at a distance of 47.1 ft from the tip of the aircraft in the positive X-direction.

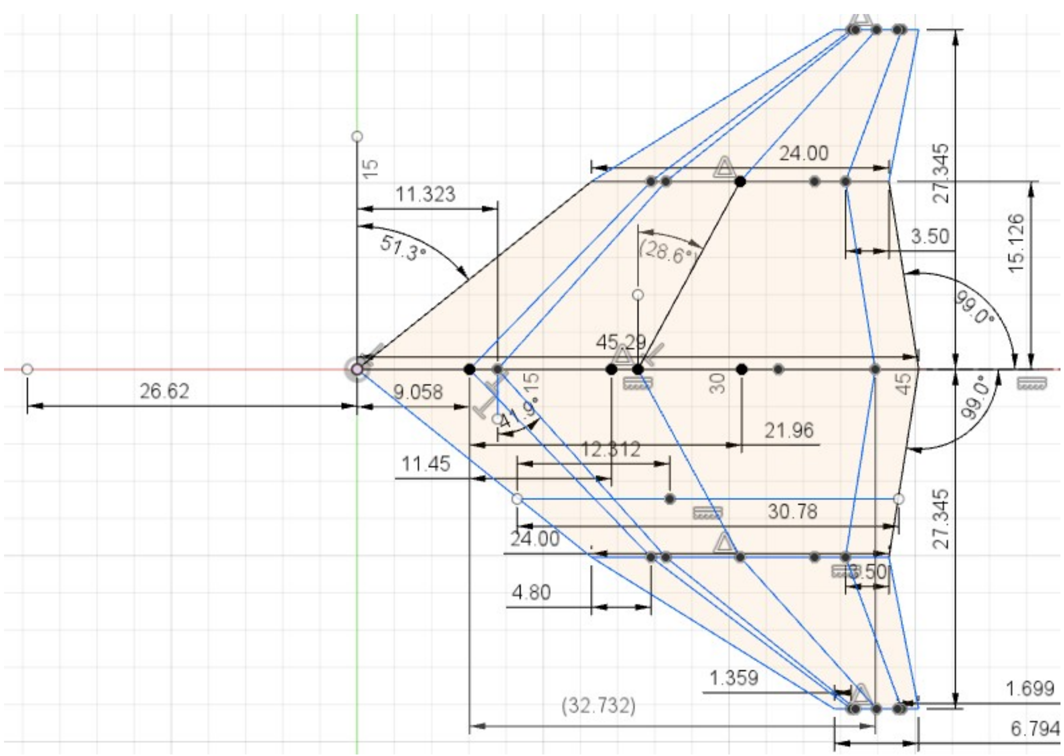


Figure 169: CG location on the swept wing

Horizontal Tail: 42% chord from the Leading Edge at 38% of the semi-span.

Vertical Tail: 42% chord from the leading edge at 55% vertical tail span from the root chord.

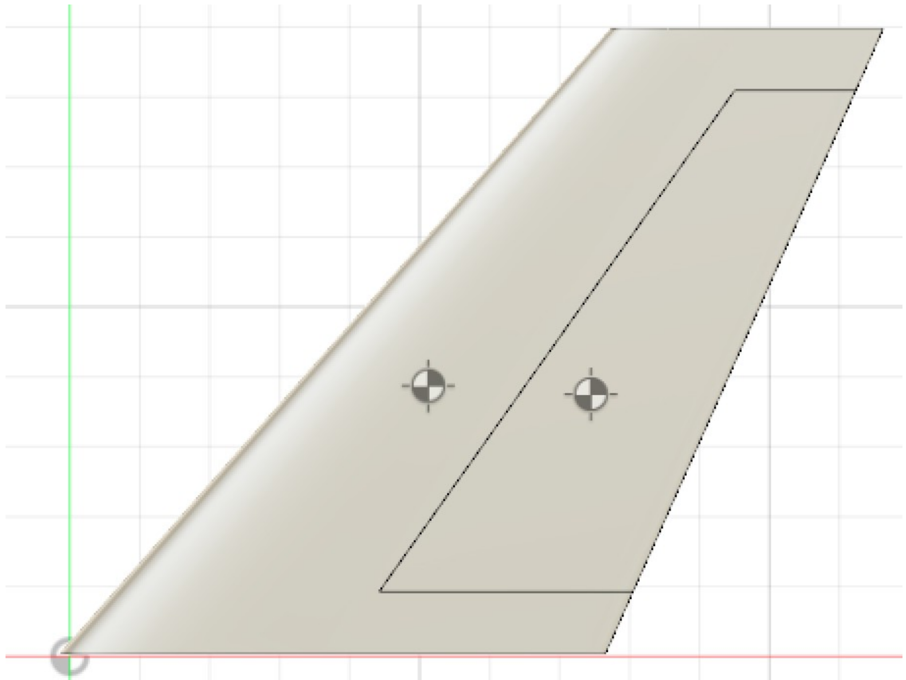


Figure 170: CG's of the vertical stabilizer and the rudder

Fuselage: 47% - 50% of the fuselage length for rear fuselage mounted engines.

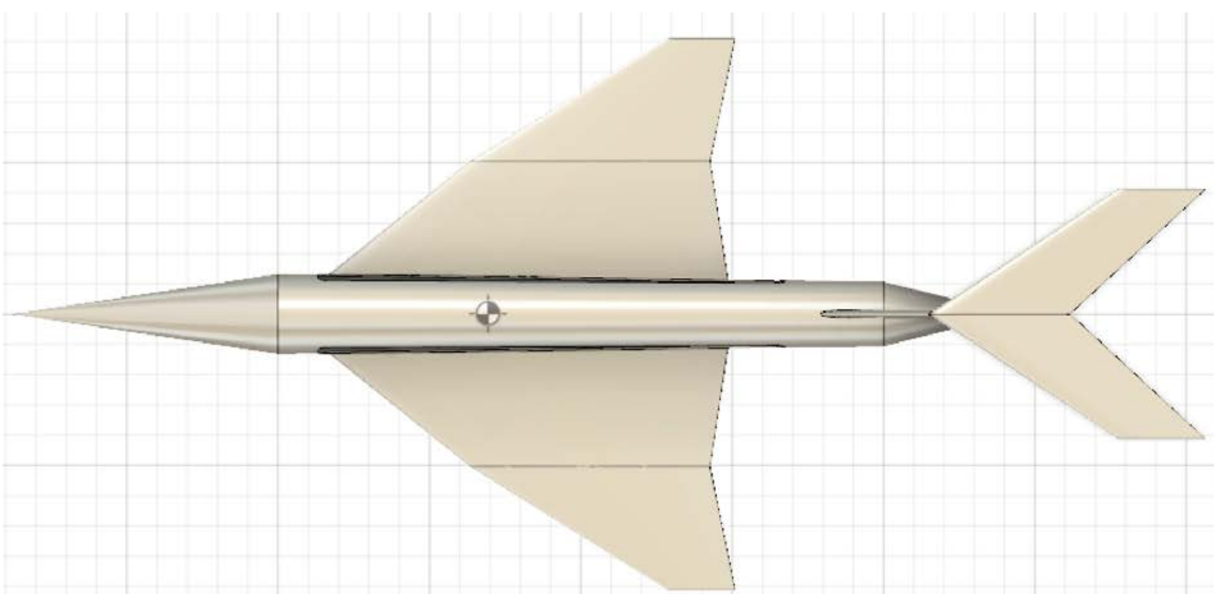


Figure 171: Top view of the airplane C.G (3D representation)

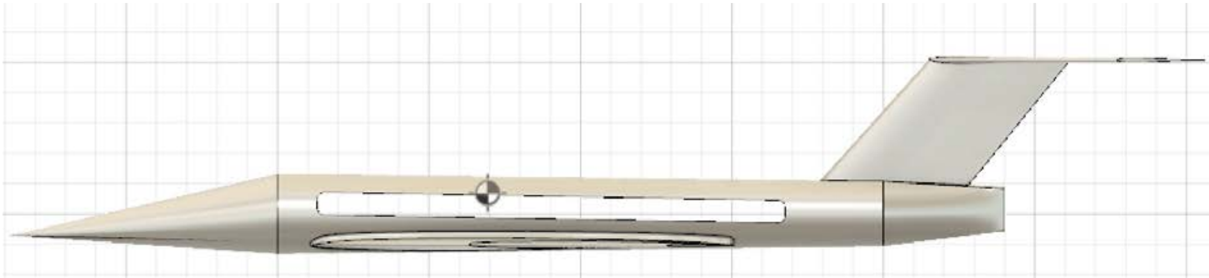


Figure 172: Side view of the Airplane C.G (3D representation)

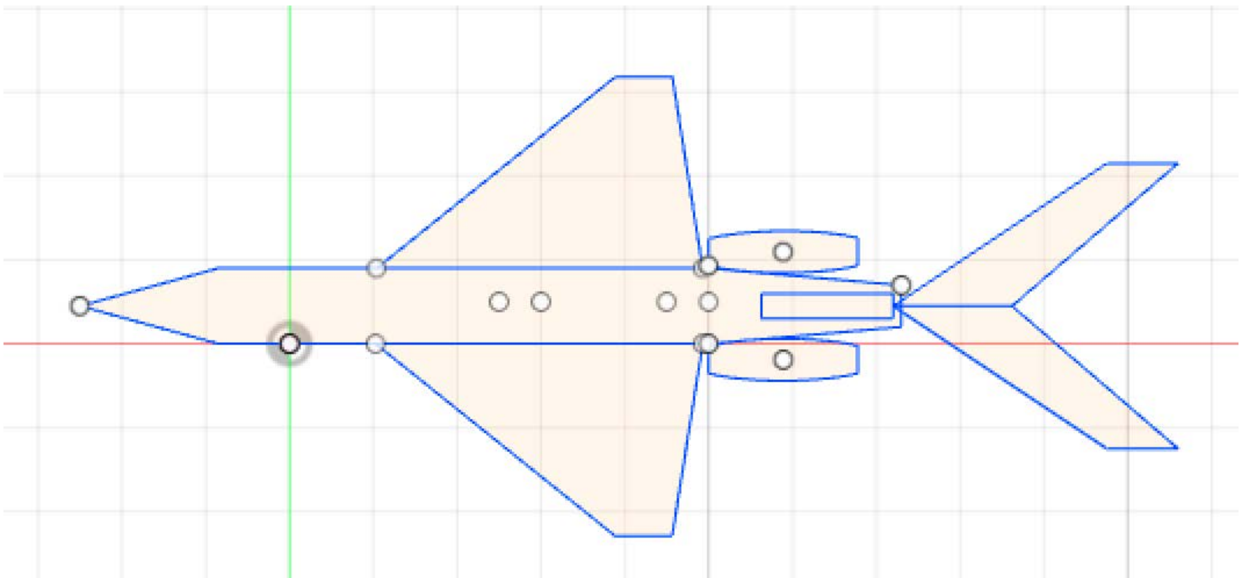


Figure 173: CG's of all major components (2D representation)

Nacelles: 40% of the nacelle length from nacelle nose.

Tail booms: 0.4% - 0.45% of the boom length starting from the most forward structural attachment of the boom.

Landing Gears: 50% of the strut length for gears with mostly vertical struts.

Table 30: Moment calculations for the structural components (feet)

CG LOCATIONS In Feet							
	(lbs)	(feet)	(ft.lbs)	(feet)	(ft.lbs)	(Feet)	(ft.lbs)
	Wi	Xi	WiXi	Yi	WiYi	Zi	WiZi
Structural Components							
Wing (Swept Wing)	11696.3	47.1	551224.9	17.2	201410.9	3.5	40937.2
Horizontal Tail	9603.5	103.0	843807.5	6.2	59407.0	21.7	208750.3

Vertical Tail	5352.1	87.9	470258.7	0.0	0.0	15.1	80816.1
fuselage	14672.7	50.0	733637.5	0.0	0.0	6.5	95372.9
Nacelles	1701.5	83.9	142767.6	3.3	5614.9	8.5	14462.6
Landing Gears	2959.0	55.0	162745.6	0.0	0.0	0.0	0.0
SUM	45985.1		2904441.8		266432.8		440339.0
CG			63.2		5.8		9.6

Table 31: Moment calculations for the structural components (inches)

CG LOCATIONS in Inches							
	(lbs)	(inches)	(in.lbs)	(inches)	(in.lbs)	(inches)	(in.lbs)
	Wi	Xi	WiXi	Yi	WiYi	Zi	WiZi
Structural Components							
Wing (Swept Wing)	11696.3	565.5	6614699.1	206.6	2416930.9	42.0	491246.1
Horizontal Tail	9603.5	1235.4	11864453.4	74.2	712883.6	260.8	2505003.5
Vertical Tail	5352.1	1054.4	5643104.0	0.0	0.0	181.2	969793.1
fuselage	14672.7	600.0	8803650.0	0.0	0.0	78.0	1144474.5
Nacelles	1701.5	1006.9	1713210.8	39.6	67378.5	102.0	173550.7
Landing Gears	2959.0	660.0	1952947.0	0.0	0.0	0.0	0.0
SUM	45985.1		36592064.3		3197193.0		5284067.9
CG			795.7		69.5		114.9

15.2 C.G. Locations of Power Plant Components

The locations for the power plant components are as follows:

Engines: (Using the manufacturers data is suggested) In this case the engine chosen for the design that satisfies the requirements of the SSBJ

Air Induction System: C.G. of the gross shell area of the inlets is used.

Propellers: The spin axis of the propeller spin plane is considered as the C.G. of the propellers.

Fuel System: (Fuel System Layout Diagram)

Filled Fuel Tank: the C.G. is located at

$$l_{31} = \mathrm{F}^{\mathrm{Z}}_{\grave{e}}\mathrm{H}\,\frac{*æ\hat{}‡pæ_{d}‡m(æ\hat{}æ_{d})\hat{\bar{d}}+}{æ\hat{}‡æ_{d}‡(æ\hat{}æ_{d})\hat{\bar{d}}} \qquad (x)$$

Trapped Fuel and Oil: Located at the bottom of the Fuel tanks and Fuel lines.

Propulsion System: (This contains the list of components that contribute to the propulsion system)

Table 32: Moment calculations for the powerplant components (feet)

CG LOCATIONS In Feet							
	(lbs)	(feet)	(ft.lbs)	(feet)	(ft.lbs)	(Feet)	(ft.lbs)
	Wi	Xi	WiXi	Yi	WiYi	Zi	WiZi
Powerplant Components							
Engines	10000.0	78.2	781720.0	6.0	60000.0	8.5	85000.0
Air Induction System	1200.0	77.0	92400.0	4.5	5400.0	7.0	8400.0
Propellers	500.0	76.2	38086.0	6.0	3000.0	8.5	4250.0
Fuel System	1003.0	55.2	55324.5	17.2	17271.7	3.5	3510.5
Filled Fuel Tank	53671.3	55.2	2960456.9	17.2	924220.3	3.5	187849.7
Trapped Fuel and Oil	536.7	75.0	40253.5	3.0	1610.1	3.5	1878.5
Propulsion System	393.3	77.2	30349.7	6.0	2359.6	7.0	2752.9
SUM	67304.3		3998590.6		1013861.7		293641.6
CG			59.4		15.1		4.4

Table 33: Moment calculations for the powerplant components (Inches)

CG LOCATIONS in Inches							
	(lbs)	(inches)	(in.lbs)	(inches)	(in.lbs)	(inches)	(in.lbs)
	Wi	Xi	WiXi	Yi	WiYi	Zi	WiZi
Powerplant Components							
Engines	10000.0	938.1	9380640.0	72.0	720000.0	102.0	1020000.0
Air Induction System	1200.0	924.0	1108800.0	54.0	64800.0	84.0	100800.0

Propellers	500.0	914.1	457032.0	72.0	36000.0	102.0	51000.0
Fuel System	1003.0	661.9	663893.5	206.6	207259.9	42.0	42126.0
Filled Fuel Tank	53671.3	661.9	35525482.7	206.6	11090643.6	42.0	2254195.9
Trapped Fuel and Oil	536.7	900.0	483042.0	36.0	19321.7	42.0	22542.0
Propulsion System	393.3	926.1	364196.9	72.0	28315.7	84.0	33035.0
SUM	67304.3		47983087.1		12166340.9		3523698.8
CG			712.9		180.8		52.4

15.3 C.G. location of Fixed Equipment

Flight Control System: According to the layout design, the CG of the flight control system was estimated to be at a distance of 45 ft in the X-direction, both the systems in the Y-direction will be neglected due to which the distance estimated in the Y-direction is 0 ft and 3 ft in the Z-direction as explained in the tables below.

Hydraulic and Pneumatic System: According to the layout design, the CG of the hydraulic and pneumatic system was estimated to be at a distance of 46 ft in the X-direction, both the systems in the Y-direction will be neglected due to which the distance estimated in the Y-direction is 0 ft and 3 ft in the Z-direction as explained in the tables below.

Electrical System: According to the layout design, the CG of the electric system was estimated to be at a distance of 25.7 ft in the X-direction, both the systems in the Y-direction will be neglected due to which the distance estimated in the Y-direction is 0 ft and 3 ft in the Z-direction as explained in the tables below.

Instrumentation, Avionics and Electronics: According to the layout design, the CG of the Instrumentation, Avionics and Electronics was estimated to be at a distance of 20.6 ft in the X-direction, both the systems in the Y-direction will be neglected due to which the distance estimated in the Y-direction is 0 ft and 3 ft in the Z-direction as explained in the tables below.

Air-conditioning, Pressurization, Anti-Icing and De-icing Systems: According to the layout design, the CG of the Air-conditioning, Pressurization, Anti-Icing and De-icing system was estimated to be at a distance of 46 ft in the X-direction, both the systems in the Y-direction will be neglected due to which the distance estimated in the Y-direction is 0 ft and 7.5 ft in the Z-direction as explained in the tables below.

Oxygen Systems: From the weight estimation report, the weight of the oxygen system was estimated to be 68.8 lbs and According to the layout design, the CG of the hydraulic and pneumatic system was estimated to be at a distance of 31.2 ft in the X-direction, both the systems in the Y-direction will be neglected due to which the distance estimated in the Y-direction is 0 ft and 7.5 ft in the Z-direction as explained in the tables below.

Auxiliary Power Unit: According to the book, manufacturer's data is to be used on the APU used for the design, according to the calculations performed, the weight estimated for the design is around 440 lbs. The CG of the APU from the layouts of the design is proposed to be at 31.2 ft in the X-direction from the tip of the aircraft, 0 ft in the Y-direction and 3.0 ft in the Z-direction as shown in the table below.

Furnishings: From the weight estimations, the total weight of the furnishings was obtained to be 864.8 lbs and the CG of the furnishings according to the layout was obtained to be placed at 40 ft in the X-direction and 4.0 ft in the Z-direction as shown in the table below.

Baggage and Cargo Handling: The SSBJ will not require any cargo handling compartment as not much weight will be carried by the passengers flying due to which the carry on's can be adjusted with them in the fuselage.

Paint: The weight of the paint required was calculated to be 330 lbs and as the paint will be distributed evenly though out the design, hence a midpoint was selected to place the CG which is 45 ft in the X-direction behind the tip of the aircraft.

Table 34: Moment calculations for the Fixed equipment components (feet)

CG LOCATIONS In Feet							
	(lbs)	(feet)	(ft.lbs)	(feet)	(ft.lbs)	(Feet)	(ft.lbs)
	Wi	Xi	WiXi	Yi	WiYi	Zi	WiZi
Fixed Equipment Components							
Flight Control System	1780.5	45.0	80124.4	15.2	27083.8	3.0	5341.6
Hydraulic System	1500.0	46.0	69000.0	2.5	3750.0	3.0	4500.0
Electrical System	1902.2	25.7	48830.7	2.5	4755.6	3.0	5706.7
Instrumentation, Avionics and Electronics	1974.6	20.6	40720.0	0.0	0.0	3.0	5923.8
A/C, Pressurization, Anti & De-icing System	427.7	46.0	19672.5	0.0	0.0	7.5	3207.5
Oxygen System	68.8	31.2	2148.8	0.0	0.0	7.5	516.0
Auxiliary Power Unit	440.0	31.2	13742.1	0.0	0.0	3.0	1320.0
Furnishings	864.8	40.0	34592.0	3.0	2594.4	4.0	3459.2
Baggage and Cargo Handling	0.0	0.0	0.0	3.0	0.0	0.0	0.0
Operational Items	0.0	0.0	0.0	0.0	0.0	0.0	0.0
Flight Test Instruments	0.0	0.0	0.0	2.0	0.0	0.0	0.0
Auxiliary Gear	0.0	0.0	0.0	2.0	0.0	0.0	0.0
Paint	330.0	45.0	14850.0	13.8	4537.5	0.0	0.0
SUM	9288.6		323680.4		42721.3		29974.8
CG			34.8		4.6		3.2

Table 35: Moment calculations for the Fixed equipment components (inches)

CG LOCATIONS in Inches							
	(lbs)	(inches)	(in.lbs)	(inches)	(in.lbs)	(inches)	(in.lbs)
	Wi	Xi	WiXi	Yi	WiYi	Zi	WiZi
Fixed Equipment Components							
Flight Control System	1780.5	540.0	961492.2	182.5	325005.7	36.0	64099.5
Hydraulic System	1500.0	552.0	828000.0	30.0	45000.0	36.0	54000.0
Electrical System	1902.2	308.0	585968.1	30.0	57067.4	36.0	68480.9
Instrumentation, Avionics and Electronics	1974.6	247.5	488640.2	0.0	0.0	36.0	71085.3
A/C, Pressurization, Anti & De-icing System	427.7	552.0	236070.5	0.0	0.0	90.0	38489.8
Oxygen System	68.8	374.8	25785.1	0.0	0.0	90.0	6192.0
Auxiliary Power Unit	440.0	374.8	164905.0	0.0	0.0	36.0	15840.0
Furnishings	864.8	480.0	415103.4	36.0	31132.8	48.0	41510.3
Baggage and Cargo Handling	0.0	0.0	0.0	36.0	0.0	0.0	0.0
Operational Items	0.0	0.0	0.0	0.0	0.0	0.0	0.0
Flight Test Instruments	0.0	0.0	0.0	24.0	0.0	0.0	0.0
Auxiliary Gear	0.0	0.0	0.0	24.0	0.0	0.0	0.0
Paint	330.0	540.0	178200.0	165.0	54450.0	0.0	0.0
SUM	9288.6		3884164.4		512655.9		359697.7
CG			418.2		55.2		38.7

As Seen above the CG for Structural Components were obtained from the 2D/3D model of the design, The CG locations of the Powerplant Components and the fixed equipment components were estimated from the designs made in the previous chapters. The weights of the components were obtained in the earlier chapters from the equations obtained from Roskam. The Moments of the individual group of components were calculated earlier and the CG was evaluated for the components. The Total CG of the aircraft is as described in the tables below. They are explained in both feet and inches respectively.

Table 36: Moment calculations for every component of the aircraft (feet)

CG LOCATIONS In Feet							
	(lbs)	(feet)	(ft.lbs)	(feet)	(ft.lbs)	(Feet)	(ft.lbs)
	Wi	Xi	WiXi	Yi	WiYi	Zi	WiZi
Structural Components							
Wing (Swept Wing)	11696.3	47.1	551224.9	17.2	201410.9	3.5	40937.2
Horizontal Tail	9603.5	103.0	843807.5	6.2	59407.0	21.7	208750.3
Vertical Tail	5352.1	87.9	470258.7	0.0	0.0	15.1	80816.1
fuselage	14672.7	50.0	733637.5	0.0	0.0	6.5	95372.9
Nacelles	1701.5	83.9	142767.6	3.3	5614.9	8.5	14462.6
Landing Gears	2959.0	55.0	162745.6	0.0	0.0	0.0	0.0
Powerplant Components							
Engines	10000.0	78.2	781720.0	6.0	60000.0	8.5	85000.0
Air Induction System	1200.0	77.0	92400.0	4.5	5400.0	7.0	8400.0
Propellers	500.0	76.2	38086.0	6.0	3000.0	8.5	4250.0
Fuel System	1003.0	55.2	55324.5	17.2	17271.7	3.5	3510.5
Filled Fuel Tank	53671.3	55.2	2960456.9	17.2	924220.3	3.5	187849.7
Trapped Fuel and Oil	536.7	75.0	40253.5	3.0	1610.1	3.5	1878.5
Propulsion System	393.3	77.2	30349.7	6.0	2359.6	7.0	2752.9
Fixed Equipment Components							
Flight Control System	1780.5	45.0	80124.4	15.2	27083.8	3.0	5341.6
Hydraulic System	1500.0	46.0	69000.0	2.5	3750.0	3.0	4500.0
Electrical System	1902.2	25.7	48830.7	2.5	4755.6	3.0	5706.7
Instrumentation, Avionics and Electronics	1974.6	20.6	40720.0	0.0	0.0	3.0	5923.8
A/C, Pressurization, Anti & De-icing System	427.7	46.0	19672.5	0.0	0.0	7.5	3207.5
Oxygen System	68.8	31.2	2148.8	0.0	0.0	7.5	516.0
Auxiliary Power Unit	440.0	31.2	13742.1	0.0	0.0	3.0	1320.0
Furnishings	864.8	40.0	34592.0	3.0	2594.4	4.0	3459.2
Baggage and Cargo Handling	0.0	0.0	0.0	3.0	0.0	0.0	0.0
Operational Items	0.0	0.0	0.0	0.0	0.0	0.0	0.0
Flight Test Instruments	0.0	0.0	0.0	2.0	0.0	0.0	0.0
Auxiliary Gear	0.0	0.0	0.0	2.0	0.0	0.0	0.0
Paint	330.0	45.0	14850.0	13.8	4537.5	0.0	0.0
TOTAL	122578.0		7226712.7		1323015.8		763955.4
FINAL CG LOCATION		**(X_cg)**	59.0	**(Y_cg)**	10.8	**(Z_cg)**	6.2

Table 37: Moment calculations for every component of the aircraft (inches)

CG LOCATIONS in Inches							
	(lbs)	(inches)	(in.lbs)	(inches)	(in.lbs)	(inches)	(in.lbs)
	Wi	Xi	WiXi	Yi	WiYi	Zi	WiZi
Structural Components							
Wing (Swept Wing)	11696.3	565.5	6614699.1	206.6	2416930.9	42.0	491246.1
Horizontal Tail	9603.5	1235.4	11864453.4	74.2	712883.6	260.8	2505003.5
Vertical Tail	5352.1	1054.4	5643104.0	0.0	0.0	181.2	969793.1
fuselage	14672.7	600.0	8803650.0	0.0	0.0	78.0	1144474.5
Nacelles	1701.5	1006.9	1713210.8	39.6	67378.5	102.0	173550.7
Landing Gears	2959.0	660.0	1952947.0	0.0	0.0	0.0	0.0
Powerplant Components							
Engines	10000.0	938.1	9380640.0	72.0	720000.0	102.0	1020000.0
Air Induction System	1200.0	924.0	1108800.0	54.0	64800.0	84.0	100800.0
Propellers	500.0	914.1	457032.0	72.0	36000.0	102.0	51000.0
Fuel System	1003.0	661.9	663893.5	206.6	207259.9	42.0	42126.0
Filled Fuel Tank	53671.3	661.9	35525482.7	206.6	11090643.6	42.0	2254195.9
Trapped Fuel and Oil	536.7	900.0	483042.0	36.0	19321.7	42.0	22542.0
Propulsion System	393.3	926.1	364196.9	72.0	28315.7	84.0	33035.0
Fixed Equipment Components							
Flight Control System	1780.5	540.0	961492.2	182.5	325005.7	36.0	64099.5
Hydraulic System	1500.0	552.0	828000.0	30.0	45000.0	36.0	54000.0
Electrical System	1902.2	308.0	585968.1	30.0	57067.4	36.0	68480.9
Instrumentation, Avionics and Electronics	1974.6	247.5	488640.2	0.0	0.0	36.0	71085.3
A/C, Pressurization, Anti & De-icing System	427.7	552.0	236070.5	0.0	0.0	90.0	38489.8
Oxygen System	68.8	374.8	25785.1	0.0	0.0	90.0	6192.0
Auxiliary Power Unit	440.0	374.8	164905.0	0.0	0.0	36.0	15840.0
Furnishings	864.8	480.0	415103.4	36.0	31132.8	48.0	41510.3
Baggage and Cargo Handling	0.0	0.0	0.0	36.0	0.0	0.0	0.0
Operational Items	0.0	0.0	0.0	0.0	0.0	0.0	0.0
Flight Test Instruments	0.0	0.0	0.0	24.0	0.0	0.0	0.0
Auxiliary Gear	0.0	0.0	0.0	24.0	0.0	0.0	0.0
Paint	330.0	540.0	178200.0	165.0	54450.0	0.0	0.0
TOTAL	122578.0		88459315.9		15876189.8		9167464.5
FINAL CG LOCATION		**(X_cg)**	721.7	**(Y_cg)**	129.5	**(Z_cg)**	74.8

15.4 Conclusions:

The Final CG Location of the entire aircraft obtained is as follows:
X-direction: 59 ft, 721.7 in
Y-direction: 0 ft, 0 in
Z-direction: 6.2 ft, 74.8 in

The tables show a different value for the Location in the Y-Direction, that is because components of only one direction were considered in this case which in actual will be neglected considering the opposite direction as well.

The final location of the airplane CG in ft's is (59, 0, 6.2).
The final location of the airplane CG in in's is (721.7, 0, 74.8).

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