Preliminary (Class I & II) Transonic Business Jet (TSBJ) Design

a project presented to
The Faculty of the Department of Aerospace Engineering
San José State University

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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
$W_{f_{used}}$: Maximum Fuel Used
$W_{f_{res}}$: Maximum Fuel Reserved
W_f: Amount of Fuel in a journey
W_{PL}: Weight of Payload
$W_{passengers}$: Weight of the passengers
W_{crew}: Weight of the crew
$W_{E_{tent}}$: Tentative Empty Weight
R: Range
E: Endurance
S: Wing Area
f: Parasite Wing Area
S_{wet}: Wetted Area
$C_{L_{max}}$: Maximum Coefficient of Lift
$C_{L_{max_L}}$: Maximum Coefficient of Lift while Landing
$C_{L_{max_{TO}}}$: 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_{D_0}: Zero lift Coefficient of Drag
C_{D_i}: Induced Coefficient of Drag
C_f: Skin Friction Drag
RC: Rate of Climb
T_{reqd}: 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

CHATER 1: MISSION SPECIFICATION AND COMPARATIVE STUDY

1.1 INTRODUCTION

The design of the transonic jets is created in a way that enables them to travel above the speed of sound (Mach 0.8 to 1.0). Many leading aircraft companies have proposed different designs time and now to convert this into a reality but unfortunately, there are very limited transonic business jet that makes this a conceptual aircraft. New modifications in Transonic Business (TSBJ) jets have enabled them to fly at higher altitudes and at transonic speed.

The mission specifications for a conceptual design of a long-range business class luxury jet airplane holds the minimum basic requirements. The proposed design aircraft carries the similar features of business jet aircrafts available in the market such as Gulfstream G550, Gulfstream G650, Bombardier Global 5000, Bombardier Global 6000, and Bombardier Global 7000. This aircraft contains the basic requirements of the Gulfstream series and Bombardier Global series aircrafts with few modifications in the design to reach a long-range transonic business jet aircraft.

Most of the manufacturers believe that all the above issues can be solved if worked upon at small scales. Focusing on this, many companies have started to develop modified designs to develop transonic jets that overcome the shortcomings mentioned-above.

1.2 MISSION SPECIFICATION

1.2.1 Mission Specification:

Table 1.1: Aircraft Design Specifications

Payload Capacity	20-24 passengers
Number of crew members	2 pilots and 2 flight attendants
Range, R(nm)	6,000
Cruise Speed (Mach Number)	0.9
Takeoff field length(feet)	7,000
Landing field length(feet)	7,000
Cruise altitude (ft)	41,000
Holding fuel (min)	30

Reserve fuel(min)	45

1.2.2 Mission Profile:

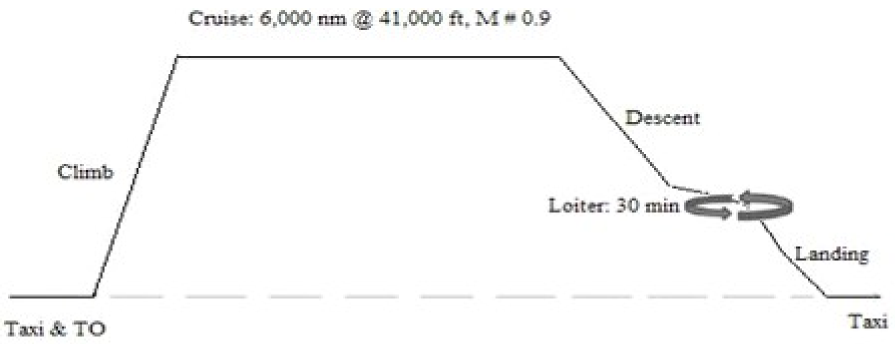


Figure 1.1: Flight mission profile of proposed aircraft

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:

Due to speed and services provided by the transonic jets they are in a great demand. As described by the flight International in the year 2012, high-value passengers like Prime ministers, executives, presidents etc. will find the value in higher speed transportation and will be the biggest consumers of the transonic business jets.

1.2.4 Technical and Economic Feasibility:
The engines providing speed up to Mach 0.9 can be installed in the transonic business jets and by making certain modification they can be made to run by a combustible and environment-friendly gas: hydrogen. Implementing the development in the transonic jets will initially cost more to the companies but will eventually provide cheaper transportation at high speed and at low cost benefitting the manufacturers and consumers.

Manufacturing companies have made enormous attempts to manufacture and install such an engine and have been successful to a certain extent, but it will still take some more time for them to design a powerful engine. Once, the engine is developed it will provide a huge boost to the aviation industry.

1.2.5 Critical Mission Requirements

Below are the critical mission requirements for the transonic jets:

- The takeoff distance for the jet is very high for shorter runways which may be a problem for the TSBJ's jet in case of the airports having shorter runways.
- The landing distance should be longer in order to land the jet safely.
- The other critical requirement is the usage of pressurized containers for the storing the fuel gas i.e. Hydrogen. These containers increase the overall weight of the aircraft.

1.3 COMPARATIVE STUDY OF SIMILAR AIRPLANES
1.3.1 COMPARATIVE STUDY

Some Aircrafts like the conventional TSBJ are:

The tables below presented the general design specifications for Gulfstream aircraft series and Bombardier Global series aircraft. This datum has been collected after the severe trade off study.

The table 1.2 reflects the design specification of Gulfstream G550. The table is condensed to the proposed design capabilities and configurations.

Table 1.2: Design Specification for Gulfstream G550

Maximum Takeoff Weight	91,000	lbs
Gross Weight	48,300	lbs
Landing Weight	75,300	lbs
Maximum Rated Thrust	30,770	lbf
Optimum Cruise Altitude	41,000	ft
Cruise Mach	0.85	mach
Fuel Weight	41,300	lbs
Used load	6,200	lbs

The figure 1.2 shows the three-view drawing of Gulfstream G550 aircraft.

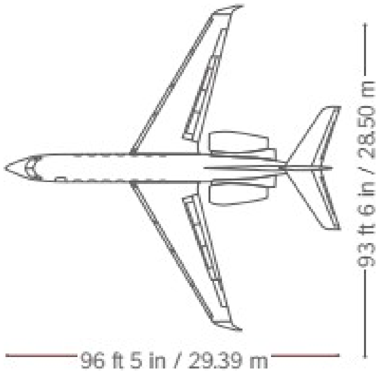


Figure 1.2: Gulfstream G550 Three View Drawing

Similarly, table 1.3 is a design specification for Gulfstream G650. The table 1.3 shows the various specification to design a Gulfstream G650. The given data are previously used specifications for Gulfstream G650.

Table 1.3: Design Specification for Gulfstream G650

Maximum Takeoff Weight	91,125	lbs.
Gross Weight	48,215	lbs.
Landing Weight	75,430	lbs.
Maximum Rated Thrust	32,200	lbf
Optimum Cruise Altitude	51,000	ft
Cruise Mach	0.85	Mach
Fuel Weight	41,550	lbs.
Used load	6,345	lbs.

The following pictures of the Gulfstream G650 listed below are at different views. The figure 1.3 is a front view of Gulfstream G650 whereas figure 1.4 is the top view and figure 1.5 is the side view of the Gulfstream G650.

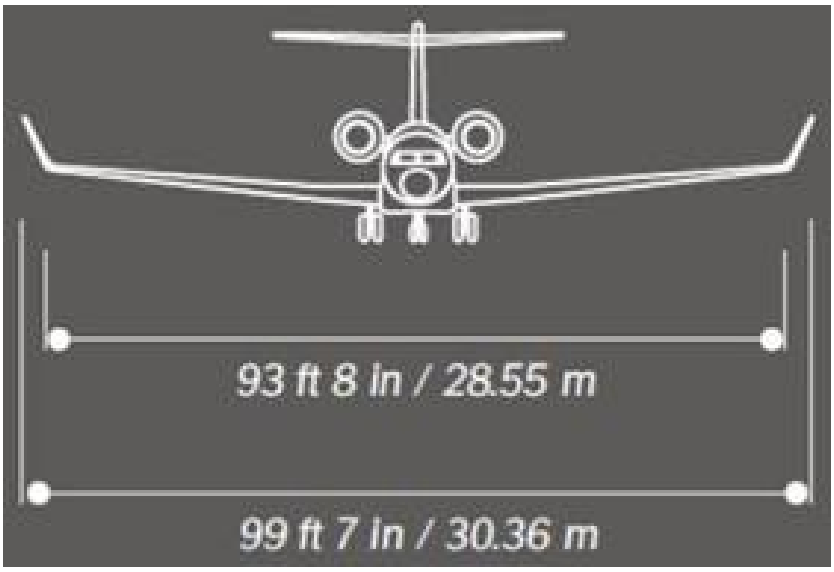


Figure 1.3: Front view of Gulfstream G650

Figure 1.4: Top view of Gulfstream G650

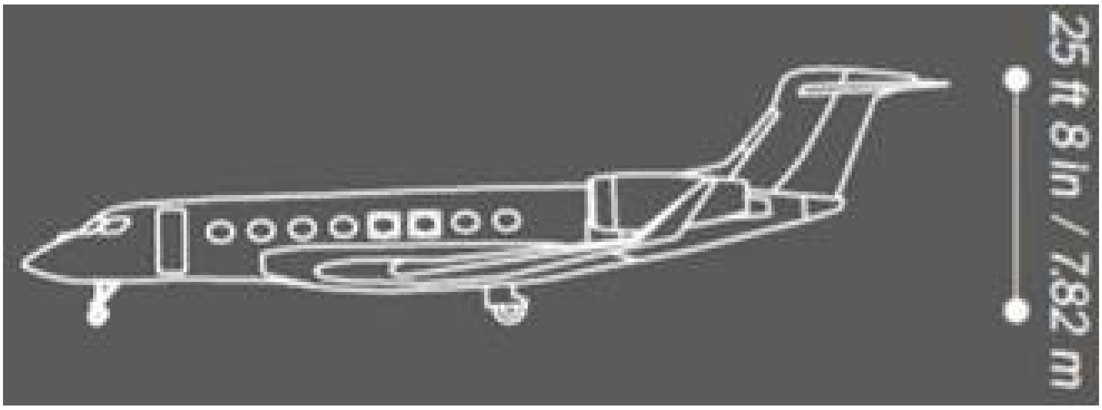


Figure 1.5: Side view of Gulfstream G650

The table 1.4 is a design specification of Bombardier Global 5000 aircraft. This table denotes the important specification of the aircraft.

Table 1.4: Design Specification for Bombardier Global 5000

Maximum Takeoff Weight	92,500	lbs.
Gross Weight	56,000	lbs.
Landing Weight	69,750	lbs.
Maximum Rated Thrust	29,500	lbf
Optimum Cruise Altitude	51,000	ft
Cruise Mach	0.85	Mach
Fuel Weight	39,250	lbs.
Used load	1,775	lbs.

The figure 1.6 shows the three views drawing of the Bombardier Global 5000.

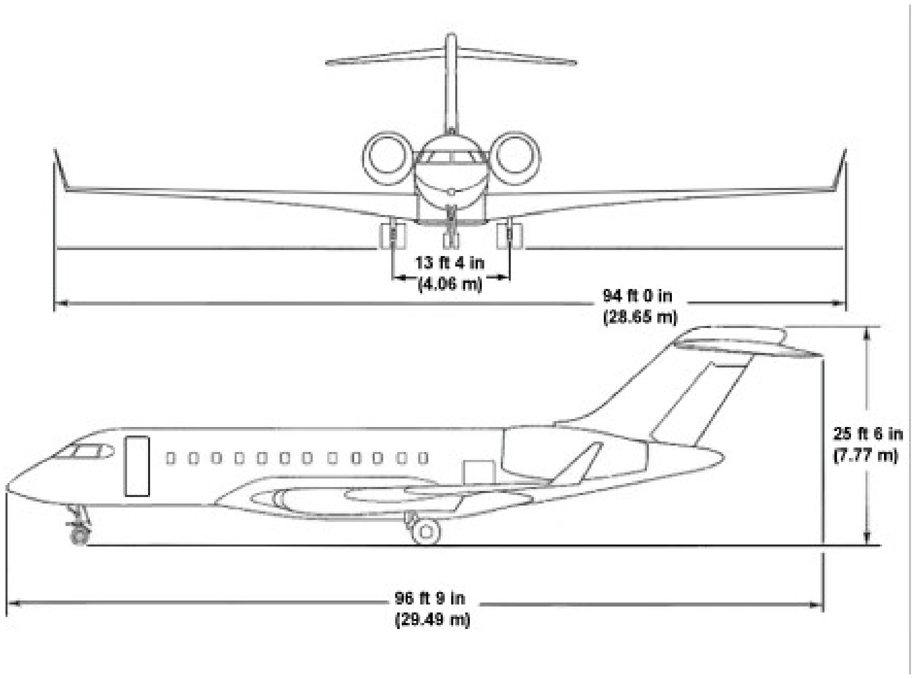


Figure 1.6: Bombardier Global 5000 Three Views Drawing

The table 1.5 includes the essential design specification of Bombardier Global 6000.

Table 1.5: Design Specification for Bombardier Global 6000

Maximum Takeoff Weight	92,440	lbs.
Gross Weight	55,400	lbs
Landing Weight	69,950	lbs
Maximum Rated Thrust	31,250	lbf
Optimum Cruise Altitude	51,000	ft
Cruise Mach	0.85	Mach
Fuel Weight	39,500	lbs
Used load	1,850	lbs

Similarly, the figure 1.7 resembles the three views drawing of Bombardier Global 6000 aircraft. The three views are side, top and front view respectively.

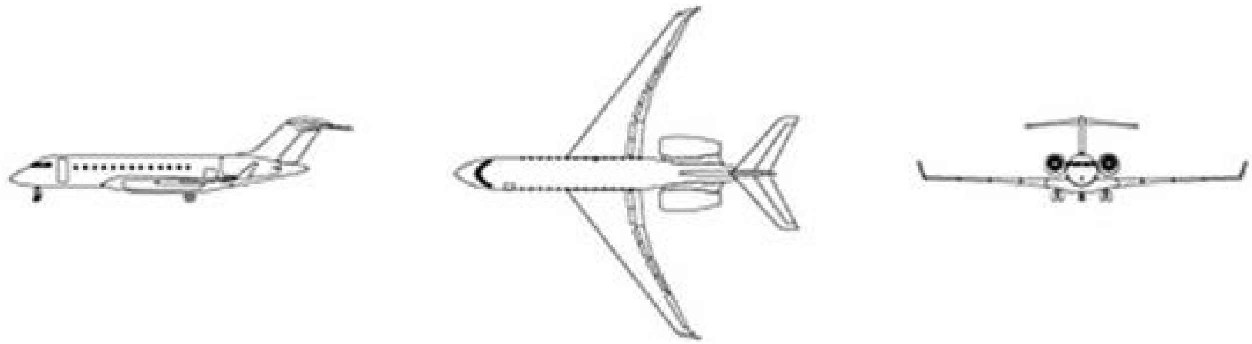


Figure 1.7: Bombardier Global 6000 Three Views Drawing

The design specifications of Bombardier Global 7000 are listed in the table 1.6 below. The following specifications are the previously published data that are imported from earlier design of Bombardier global 7000.

Table 1.6: Design Specification for Bombardier Global 7000

Maximum Takeoff Weight	93,000	lbs
Gross Weight	54,300	lbs
Landing Weight	68,850	lbs
Maximum Rated Thrust	32,000	lbf
Optimum Cruise Altitude	51,000	ft
Cruise Mach	0.85	Mach
Fuel Weight	39,495	lbs
Used load	1,770	lbs

The three views drawing for Bombardier Global 7000 is shown in figure 1.8 with side view, front view and top view respectively.

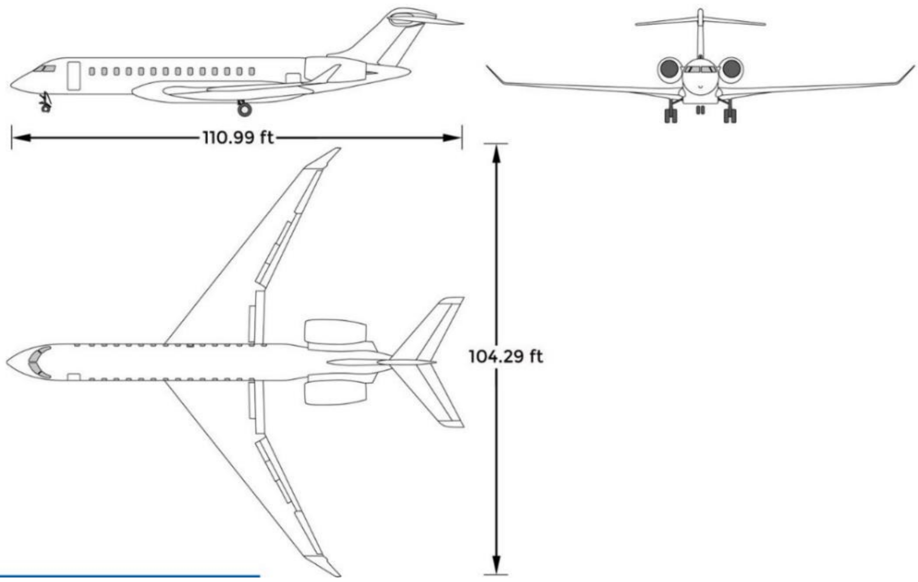


Figure 1.8: Bombardier Global 7000 Three Views Drawing

3.2 Comparison of Important Design Parameters

Similarly, the tables below are presenting the general design parameters of Gulfstream series and Bombardier Global series aircraft.

The table 1.7 represents the few important design parameters of Gulfstream G550. The Gulfstream G550 parameters are previously published data as well to design the G550.

Table 1.7: Design Parameters of Gulfstream G550

Wing Area	1,265	ft*2
Wing Loading	78.9	lb/ft*2
Aspect Ratio	7.5	
Wing Sweep	34	deg
Tail Configuration	T-tail	
Engine configuration	Twin Jets	

The essential design parameters of Gulfstream G650 are shown in the table 1.8.

Table 1.8: Design Parameters of Gulfstream G650

Wing Area	1,283	ft*2

Wing Loading	77.7	lb/ft*2
Aspect Ratio	7.7	
Wing Sweep	36	deg
Tail Configuration	T-tail	
Engine configuration	Twin Jets	

Similarly, the important design parameters of Bombardier Global 5000 are illustrated in the table 1.9.

Table 1.9: Design Parameters of Bombardier Global 5000

Wing Area	1,022	ft*2
Wing Loading	95.9	lb/ft*2
Aspect Ratio	7.8	
Wing Sweep	35	deg
Tail Configuration	T-tail	
Engine configuration	Twin jets	

The following table 1.10 is the design parameters of Bombardier Global 6000.

Table 1.10: Design Parameters of Bombardier Global 6000

Wing Area	1,110	ft*2
Wing Loading	94.7	lb/ft*2
Aspect Ratio	7.5	
Wing Sweep	36	deg
Tail Configuration	T-tail	
Engine configuration	Twin jets	

The table 1.11 below consists various design parameters of Bombardier Global 7000.

Table 1.11: Design Parameters of Bombardier Global 7000

Wing Area	1,300	ft*2

Wing Loading	95.7	lb/ft*2
Aspect Ratio	7.7	
Wing Sweep	35	deg
Tail Configuration	T-tail	
Engine configuration	Twin Jets	

1.3.2 DISCUSSION

The above table mentions different informative parameters to be considered while designing transonic business jets. The aircrafts selected for the study in the table have many parameters like the take-off weight, payload capacity of up to 12 passengers and 5 crew members to a range of 6000 nm with a cruising velocity of 0.8 Mach and a maximum cruising velocity of 0.9 Mach like the TSBJ being designed. In all the proposed prototypes, the main obstacle that the companies are facing to develop such aircraft is the high cost of production. In order to develop such an aircraft, the cost of production needs to be reduced to a certain extent.

1.4 **CONCLUSIONS AND RECOMMENDATIONS**

Transonic Business Jets are the highlight of the aviation world as they can provide fast transportation to the people. Hence, designing an aircraft that is economical, fuel-efficient, and ecofriendly while incorporating the latest available technology is the concept of this report. Presently, Transonic Business Jets are just a concept which has seen in-depth research by numerous researchers.

CHAPTER – 2: CONFIGURATION SELECTION

2.1 INTRODUCTION

This report aims to define the configuration of the transonic business jet which shall be designed at a point in future. The report defines each section of the aircraft and its configuration with respect to the needs of the designs. All of this is done while taking into consideration its limitations and the advantages-disadvantages of the configuration.

This report encompasses the comparison of the performance, weights, and geometry of the aircrafts that are a close match to my transonic business jet design. The report also presents sketches of my design which represents the various 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.

The table 1.2 shows the necessary design specification for Gulfstream G550. It also resembles the previously published datum.

Table 1.2: Design Specification for Gulfstream G550

Maximum Takeoff Weight	91,000	lbs
Gross Weight	48,300	lbs
Landing Weight	75,300	lbs
Maximum Rated Thrust	30,770	lbf
Optimum Cruise Altitude	41,000	ft
Cruise Mach	0.85	Mach
Fuel Weight	41,300	lbs
Used load	6,200	lbs

The table 1.3 demonstrates the design specification for Gulfstream G650 used in the previous design.

Table 1.3: Design Specification for Gulfstream G650

Maximum Takeoff Weight	91,125	lbs
Gross Weight	48,215	lbs
Landing Weight	75,430	lbs

Maximum Rated Thrust	32,200	lbf
Optimum Cruise Altitude	51,000	ft
Cruise Mach	0.85	Mach
Fuel Weight	41,550	lbs
Used load	6,345	lbs

The table 1.4 consists of the necessary design specification for Bombardier Global G5000 used in the design process.

Table 1.4: Design Specification for Bombardier Global 5000

Maximum Takeoff Weight	92,500	lbs
Gross Weight	56,000	lbs
Landing Weight	69,750	lbs
Maximum Rated Thrust	29,500	lbf
Optimum Cruise Altitude	51,000	ft
Cruise Mach	0.85	Mach
Fuel Weight	39,250	lbs
Used load	1,775	lbs

The table 1.5 consists of the necessary design specification for Bombardier Global G6000 used in the design process.

Table 1.5: Design Specification for Bombardier Global 6000

Maximum Takeoff Weight	92,440	lbs
Gross Weight	55,400	lbs
Landing Weight	69,950	lbs
Maximum Rated Thrust	31,250	lbf
Optimum Cruise Altitude	51,000	ft
Cruise Mach	0.85	Mach
Fuel Weight	39,500	lbs

Used load	1,850	lbs

The table 1.6 presents the important design specification for Bombardier Global G7000 used in the design process.

Table 1.6: Design Specification for Bombardier Global 7000

Maximum Takeoff Weight	93,000	lbs
Gross Weight	54,300	lbs
Landing Weight	68,850	lbs
Maximum Rated Thrust	32,000	lbf
Optimum Cruise Altitude	51,000	ft
Cruise Mach	0.85	Mach
Fuel Weight	39,495	lbs
Used load	1,770	lbs

Similarly, the tables below are presenting the general design parameters of Gulfstream series and Bombardier Global series aircraft.

The table 1.7 consists of the necessary design parameters of Gulfstream G550 used in the previous design process of similar aircraft.

Table 1.7: Design Parameters of Gulfstream G550

Wing Area	1,265	ft*2
Wing Loading	78.9	lb/ft*2
Aspect Ratio	7.5	
Wing Sweep	34	deg
Tail Configuration	T-tail	
Engine configuration	Twin Jets	

The table 1.8 provides the necessary design parameters of Gulfstream G650 used in the previous design process of similar aircraft.

Table 1.8: Design Parameters of Gulfstream G650

Wing Area	1,283	ft*2
Wing Loading	77.7	lb/ft*2
Aspect Ratio	7.7	
Wing Sweep	36	deg
Tail Configuration	T-tail	
Engine configuration	Twin Jets	

The table 1.9 consists of the essential design parameters of Bombardier Global G5000 used in the previous design process of similar aircraft.

Table 1.9: Design Parameters of Bombardier Global 5000

Wing Area	1,022	ft*2
Wing Loading	95.9	lb/ft*2
Aspect Ratio	7.8	
Wing Sweep	35	deg
Tail Configuration	T-tail	
Engine configuration	Twin jets	

The table 1.10 shows the design parameters of Bombardier Global G6000 used in the previous design process of similar aircraft.

Table 1.10: Design Parameters of Bombardier Global 6000

Wing Area	1,110	ft*2
Wing Loading	94.7	lb/ft*2
Aspect Ratio	7.5	
Wing Sweep	36	deg
Tail Configuration	T-tail	
Engine configuration	Twin jets	

The table 1.11 shows the design parameters of Bombardier Global G7000 used in the previous design process of similar aircraft.

Table 1.11: Design Parameters of Bombardier Global 7000

Wing Area	1,300	ft*2
Wing Loading	95.7	lb/ft*2
Aspect Ratio	7.7	
Wing Sweep	35	deg
Tail Configuration	T-tail	
Engine configuration	Twin Jets	

2.2 Configuration Comparison of Similar Airplanes

This section includes 3-views (photocopies) of the 5 airplanes similar to the proposed long-range business jet.

2.2.1 Gulfstream G550

The figure 2.1 represents the three-view drawing of Gulfstream G550 as shown below:

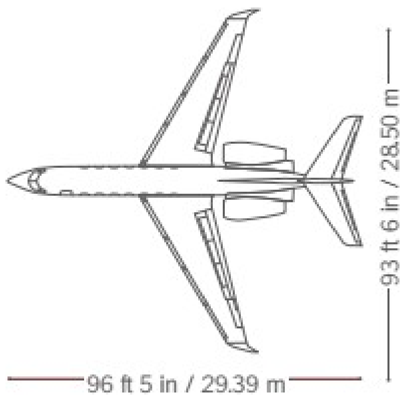


Figure 2.1: Gulfstream G550 Three View Drawing

The table 2.1 includes the interior specifications for Gulfstream G550.

Table 2.1: Interior Specifications for Gulfstream G550

Total Interior Length	50 ft 1 in | 15.27 m

Cabin Length (excluding baggage)	43 ft 11 in | 13.39 m
Cabin Height	6 ft 2 in | 1.88 m
Cabin Width	7 ft 4 in | 2.24 m
Cabin Volume	1,669 cu ft | 47.26 cu m
Baggage Compartment Usable Volume	170 cu ft | 4.81 cu m

The table 2.2 represents the exterior specifications for Gulfstream G550.

Table 2.2: Exterior Specifications for Gulfstream G550

Exterior Height	25 ft 10 in
Exterior Length	96 ft 5 in
Wingspan	93 ft 6 in
Baggage, External	0

2.2.2 Gulfstream G650

The figure 2.2, 2.3 and 2.4 represents the three-view drawing of Gulfstream G650 front view, top view and side view respectively.

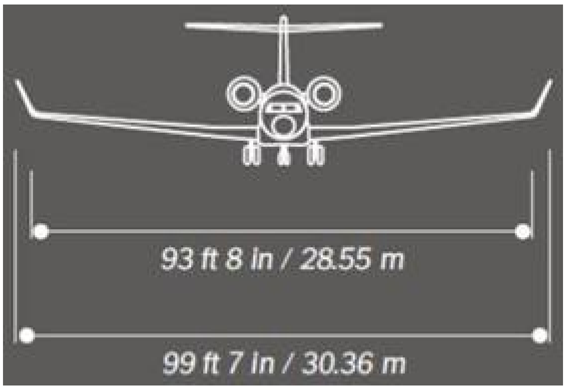


Figure 2.2: Front view of Gulfstream G650

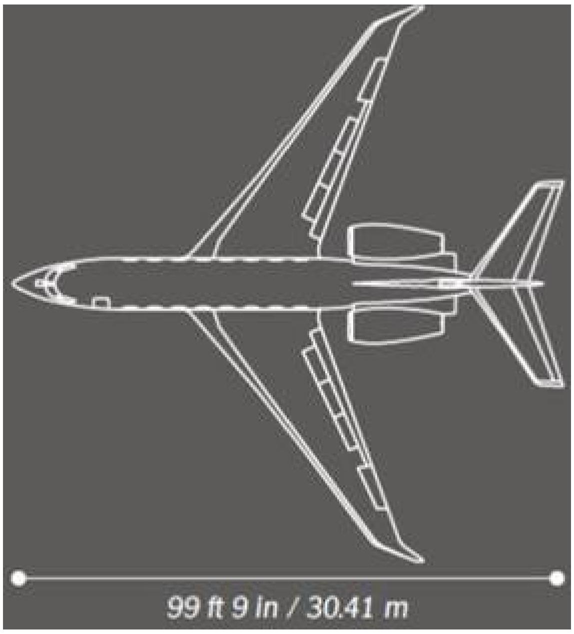


Figure 2.3: Top view of Gulfstream G650

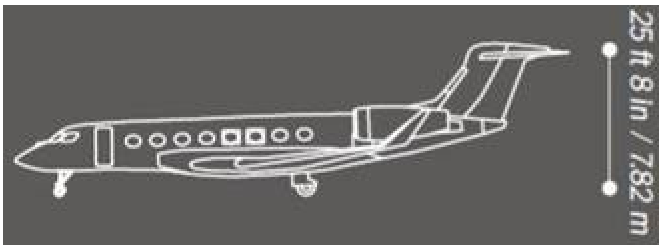


Figure 2.4: Side view of Gulfstream G650

The table 2.3 includes the interior specifications for Gulfstream G650 as below:

Table 2.3: Interior Specifications for Gulfstream G650

Total Interior Length	53 ft. 7 in. / 16.33 m
Cabin Length	46 ft. 10 in. / 14.27 m
Cabin Height	6 ft. 5 in. / 1.95 m
Cabin Width	8 ft. 6 in. / 2.59 m
Cabin Volume	2,138 cu. ft. / 60.54 cu. m.

Usable Baggage Compartment Volume	195 cu. ft. / 5.52 cu. m.

The table 2.4 includes the exterior specifications for Gulfstream G650.

Table 2.4: Exterior Specifications for Gulfstream G650

Height	25 ft. 4 in. / 7.72 m
Length	99 ft. 9 in. / 30.40 m
Fuselage Width	9 ft. / 2.74 m
Overall Span	99 ft. 7 in. / 30.36m
Wing Span	93 ft. 8 in. / 28.55 m
Wing Sweep	36 degrees
Wing Area	1,283 sq. ft. / 119.2 sq. m.
Aspect Ratio	7.7

2.2.3 Bombardier Global 500

The figure 2.5 demonstrates the three-view drawing of Bombardier Global 5000 as shown below:

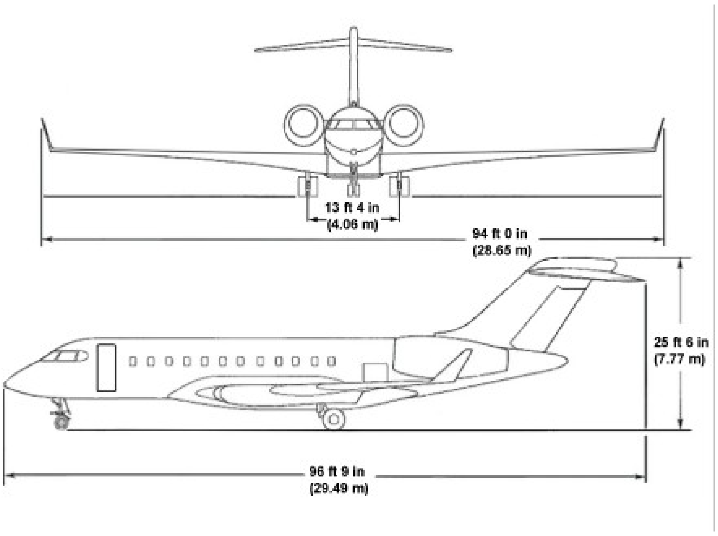


Figure 2.5: Bombardier Global 5000 Three View Drawing

The table 2.5 contains the interior specifications for Bombardier Global 5000.

Table 2.5: Interior Specifications for Bombardier Global 5000

Cabin Length	40 ft 9 in
Cabin Width	7 ft 11in
Cabin Height	6 ft 2 in
Baggage, Internal	195
Cabin Volume	1889 cu-ft

The table 2.6 includes the exterior specifications for Bombardier Global 5000.

Table 2.6: Exterior Specifications for Bombardier Global 5000

Exterior Height	25 ft 6 in
Exterior Length	96 ft 10 in
Wingspan	94 ft
Baggage, External	0

2.2.4 Bombardier Global 6000

The figure 2.6 demonstrates the three-view drawing as a side view, top view and front view respectively of Bombardier Global 6000 as below.

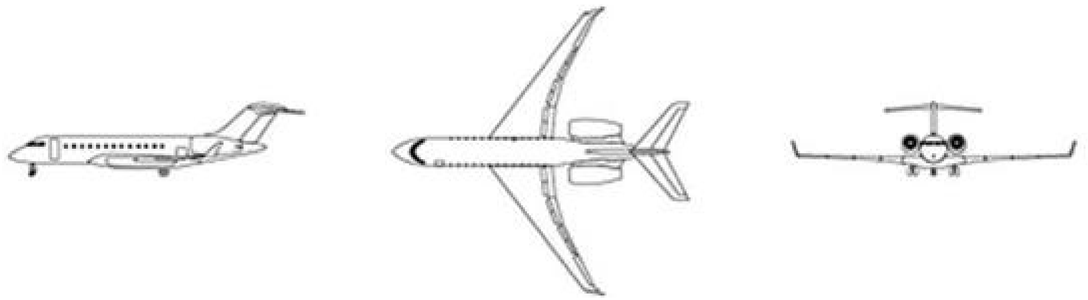


Figure 2.6: Bombardier Global 6000 Three View Drawing

The table 2.7 includes the interior specifications for Bombardier Global 6000.

Table 2.7: Interior Specifications for Bombardier Global 6000

Cabin Length	48.35 ft. / 14.7 m.
Max Cabin Width	8.17 ft. / 2.49 m.
Cabin Width	6.25 ft. / 2.11 m.
Cabin Height	6.25 ft. / 1.91 m.
Floor Area	335 sq. ft. / 31.1 sq. m.
Cabin Volume	2,140 cu. ft. / 60.6 cu. m.

The table 2.8 includes the exterior specifications for Bombardier Global 6000.

Table 2.8: Exterior Specifications for Bombardier Global 6000

Length	99.4 ft. / 30.3 m.

Wingspan	94 ft. / 28.7 m.
Height Overall	25.5 ft. / 7.8 m.

2.2.5 Bombardier Global 7000

The figure 2.7 demonstrates the three-view drawing of Bombardier Global G7000 with a side view, front view and top view respectively.

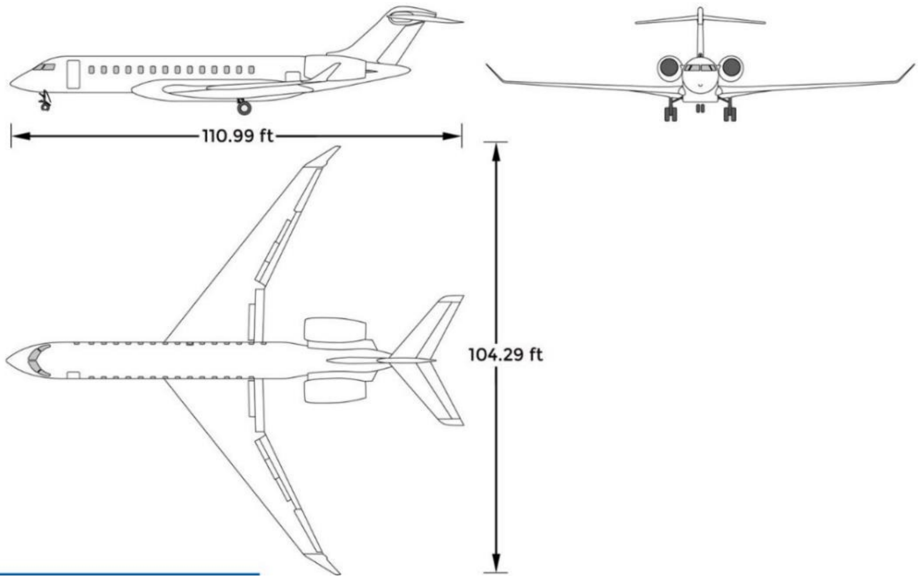


Figure 2.7: Bombardier Global 7000 Three View Drawing

The table 2.9 demonstrates the interior specifications for Bombardier Global 7000.

Table 2.9: Interior Specifications for Bombardier Global 7000

Cabin Length	54 ft 7 in
Cabin Width	8 ft 2 in
Cabin Height	6 ft 3 in
Cabin Volume	2,637 cu-ft

The table 2.10 demonstrates the exterior specifications for Bombardier Global 7000.

Table 2.10: Exterior Specifications for Bombardier Global 7000

Exterior Height	27 ft 0 in
Exterior Length	111 ft 2 in
Wingspan	104 ft

2.2.3 Discussion

Given above are the 3 views of those aircrafts which are similar to my proposed design. While considering the earlier designs, the aircrafts' configurations are given below:

Low-wing configuration and a less wing area are two important aspects of all suggested transonic business jet designs. The best fit for business jets is low wing configurations and hence these were chosen for better maneuverability in comparison to any high or mid-wing aircraft. Some of the other advantages of low wing configurations are reduced take-off and landing distances along with enhanced safety. Due to this, the pilot's load is reduced while landing the aircraft, as it ensures easier retraction of the landing gears. Hence, the low-wing configuration is the best configuration of the proposed business jet designs.

The above-mentioned jets which are similar to my proposed design possess a T-tail in the empennage section or a vertical tail. The fuselage is very thin airfoil shaped. This is done as the jets are transonic and they must travel at greater speeds with lesser weight. Greater speed is obtained by thinner airfoils and by reducing the wing area, but this is at the cost of reducing the aircraft's maneuverability, which undoubtedly is the most important characteristic of an aircraft. These also possess sharper nose tips as the angle formed while cutting the airflow is lesser, thereby increasing the speed of the aircraft. If the angle was wider while passing through a laminar flow, there would be a greater drag and the speed of the aircraft would eventually be lesser.

2.3 CONFIGURATION SELECTION

2.3.1 Overall Configuration

Airplanes can be classified into 3 categories:
1. Land Based
2. Water Based
3. Amphibious

The design of the transonic business jet is completely land based. Therefore, the overall configuration of the aircraft is designed in a way to satisfy all the requirements of a land-based aircraft. The requirements include the safety while take-off, landing and while in flight.

2.3.2 Wing Configuration

The wings can be classified into 2 broad categories from the structural point of view:

1. Cantilever wing
2. Strutted wing

The wings can be classified as:

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

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

1. Aspect ratio
2. Airfoil(s)
3. Thickness ratio
4. Twist
5. Taper ratio
6. Winglets
7. Dihedral angle
8. High lift and control surface requirements
9. Incidence angle

The business jet that is under designing phase will have a Cantilever wing with a low wing configuration and swept-back wing. In the later section, the characteristics of the wing configuration will be designed.

2.3.3 Empennage Configuration

Following are the parts of the aircraft that the empennage configuration contains:

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

The rules applied on the empennage section are same as the ones applied on the wing configuration. Following are the configurational choices that the empennage section must make:

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 transonic business jet will have a T-tail configuration in the Empennage section as it is the most suitable design for the business jets.

2.3.4 Integration of the Propulsion System

The aircraft engines can be arranged in following three ways:

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

The above-mentioned three basic configurations can be installed in following two ways:

1. Pods or Nacelles
2. Buried

The configurations can be dispositioned on or in the:

1. Wing
2. Empennage
3. Fuselage

The dispositioning of the engines creates many consequences and the major ones are:

1. Airplane weight
2. Engine efficiency
3. Handling characteristics
4. Airplane vibration and noise
5. Maintenance

The propeller will have pusher installation and will be located behind the center of gravity (CG) of the aircraft. Pusher configuration tends to be more stabilizing both in the static longitudinal and static directional stability and because of these features; it was selected to be used. This feature can also save the empennage area.
The engines will be installed in pods/ nacelles and on the fuselage in the far aft section of the fuselage. This installation makes the aircrafts run in the best conditions and save time during the maintenance of the engines.
This configuration was selected as it showed major consequences that were created due the engine's dispositioning. The design proposed is developed to overcome all these consequences and to create an efficient transonic business jet.

2.3.5 Landing Gear Disposition

The landing gears on an aircraft are classified into two categories:

1. Fixed or non-retractable

2. Retractable

As per the layout, the landing gears can be classified as:
1. Taildraggers
2. Outrigger
3. Tandem
4. Conventional or tricycle

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

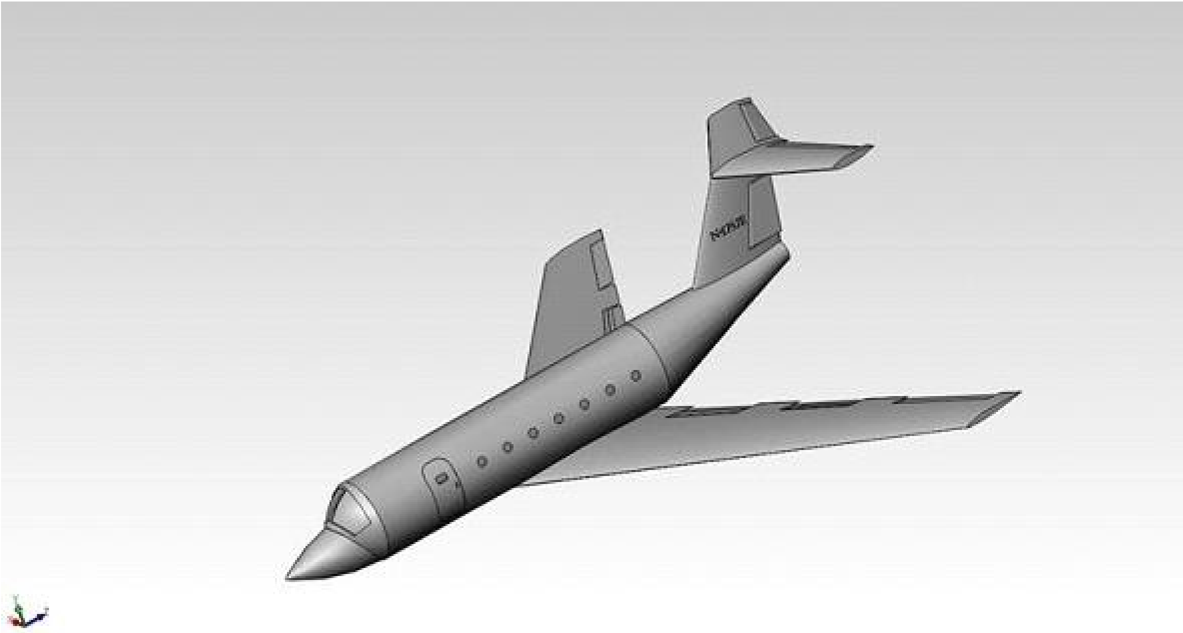
The proposed transonic jet will be having retractable type landing gear due to its advantages. This type of gear needs less power while flying the jet, prevents the aircraft from producing a lot of drag and gives longer ranges.
The aircraft will have a tricycle or conventional type configuration to balance the aircraft during take-off, landing and at rest.
The landing gears will be attached before the CG and will be mounted on the wing beyond the CG because of the attachment of the propulsion system behind the CG for balancing the aircraft.
The number of the number of tires and the mail gear struts to be installed will be decided during the designing of the landing gear configuration.

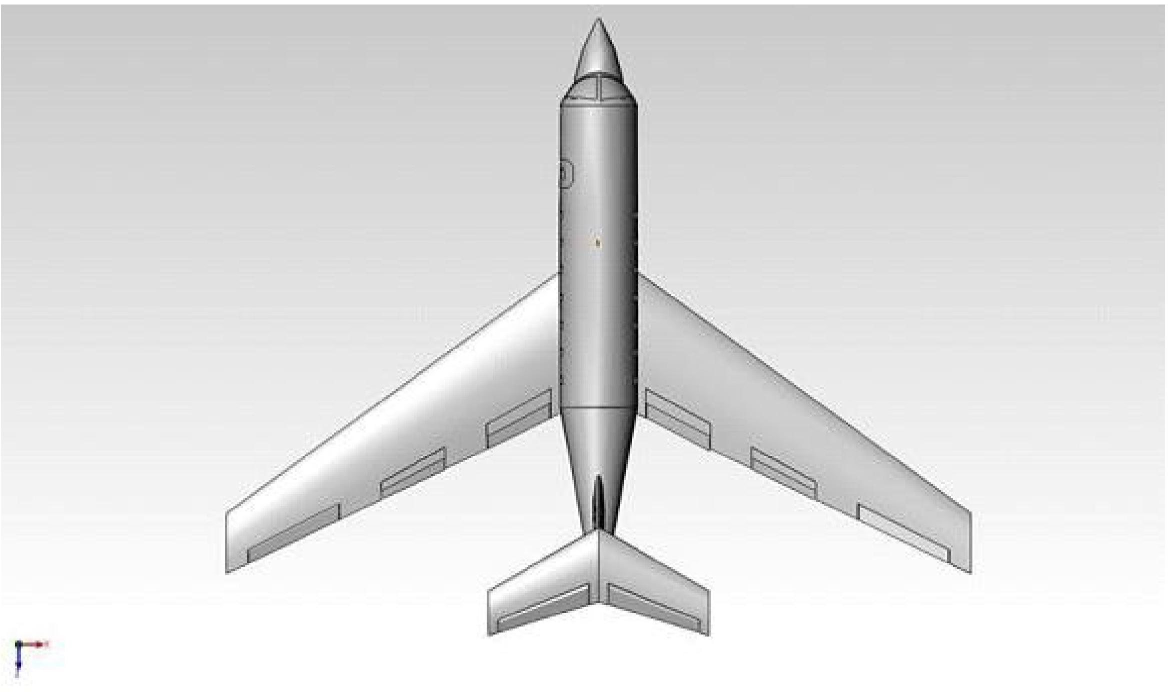
2.3.6 Proposed Configuration

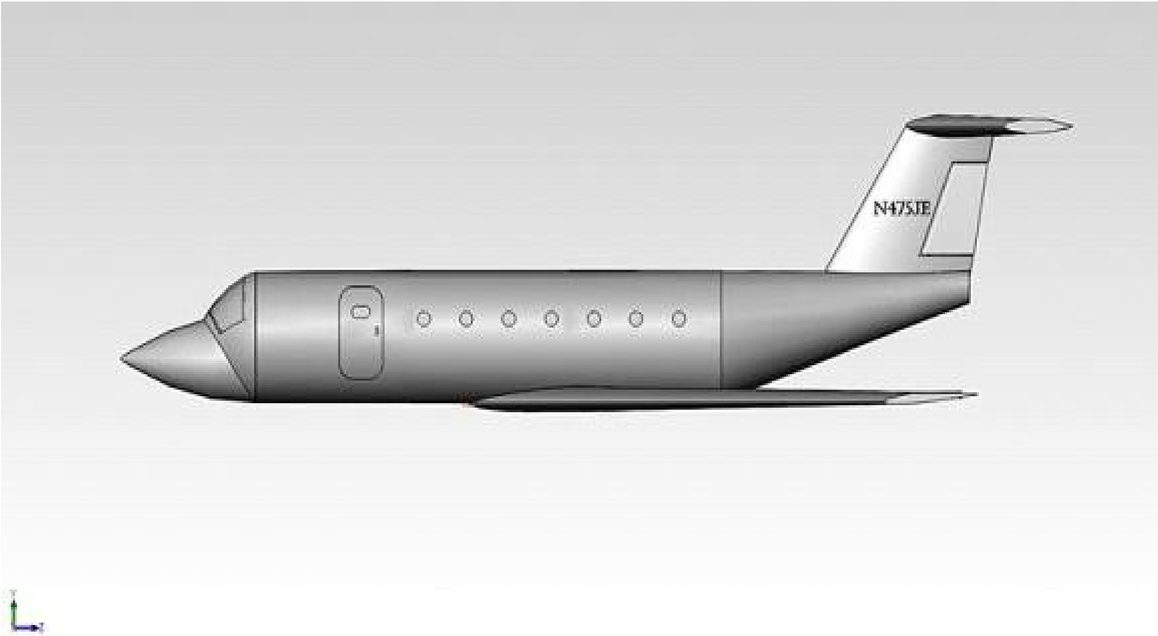
The proposed configuration is a land-based design with the engines attached using the pods mounted on the fuselage. It will have a cantilevered low wing configuration with a swept back angle and a T-tail configuration in the empennage section. The landing gear composition will be conventional with the engines attached far aft of the fuselage behind the CG of the aircraft.

A preliminary CAD three view drawing of the proposed airplane with a front view, top view and side view respectively is shown below:

Aircraft Model:







CHAPTER 3: WEIGHT SIZING AND WEIGHT SENSITIVITIES

3.1 INTRODUCTION

This is the third report in the series which is generated with a purpose to calculate the preliminary weight estimations. Here we begin the third report from the series reports, in this report we shall calculate the preliminary weight estimations with the use of the AAA program and hand-written calculations. One of the
most important processes while designing an aircraft is the preliminary weight estimation process. This is an extremely important process as this provides a clear estimate of the weight of the aircraft post every stage of the flight path. This includes, the amount of fuel that shall be required by the aircraft to make the journey, that is from take-off and right up to its landing. It also includes a rough estimate of the payload of the aircraft and its ability to lift it throughout the journey.

The report involves the regression points A and B, which are calculated of various aircrafts which have a similar design as that of my aircraft while using empty weights and the maximum take-off weights. The graph comprising of these regression points is constructed thereby forming an equation. Then the equation us solved, by making use of the estimated maximum takeoff weight of my proposed aircraft, this provides a value that shall define the permissible or the necessary empty weight of the aircraft. Upon obtaining this value, the empty weight of the aircraft shall be calculated, making use of manual calculations by the formulas obtained from Aircraft Design book by Jan Roskam. The calculations shall provide us with certain data, and this obtained data shall be compared with the data obtained from the AAA program. Therefore, the regression coefficients and the graphs which are obtained via the two methods shall be compared.

After acquiring the necessary data, the sensitivities of the different parameters with respect to the take-off weight shall be calculated through the AAA program and manually. The sensitivities are resourceful for obtaining different data about the aircraft being designed including its drawbacks and its advantages etc. Other parameters such as the payload, range, endurance, empty weight, the L/D ratio, the specific fuel consumption is all compared with the takeoff weight to determine the changes that shall be observed if the flight journey was increased or decreased by a mile throughout its 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. Upon considering the necessary sensitivities, trade studies shall be commissioned on the existing parameters of the aircraft to see the differences that shall occur on the mission of the aircraft and the design, if key values are decreased or increased.

3.2 MISSION WEIGHT ESTIMATES

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

Table 1: Weights of Similar Airplanes

Aircraft	Maximum Takeoff Weight (MTOW)	Empty Weight (W_E)	Landing Weight	Units
Gulfstream G550	91,000	48,300	75,300	lbs
Gulfstream G650	92,125	48,215	75,430	lbs
Bombardier Global 5000	92,500	56,000	69,750	lbs
Bombardier Global 6000	92,440	55,400	69,950	lbs
Bombardier Global 7000	93,000	54,300	68,850	lbs

3.2.2 Determination of Regression Coefficients A and B.

3.2.2.1 Determination of Regression Coefficients Using Manual Calculations.

Using the natural log of the Maximum Takeoff Weights and the Empty Weights of the aircrafts similar to my aircraft, the regression coefficients A and B are obtained and are mentioned in the following table:

Table 2: Weights of similar aircraft after taking natural log of takeoff and empty weight

Aircraft	Log_{10}(MTOW)	$Log_{10}(W_E)$
Gulfstream G550	4.959041392	4.683947131
Gulfstream G650	4.964377501	4.683182171
Bombardier Global 5000	4.966141733	4.748188027
Bombardier Global 6000	4.965859937	4.743509765
Bombardier Global 7000	4.968482949	4.73479983

The graph below is obtained by comparing these equations with the natural log of the Maximum Takeoff Weight on the X-axis and the Empty Weight on the Y-axis.

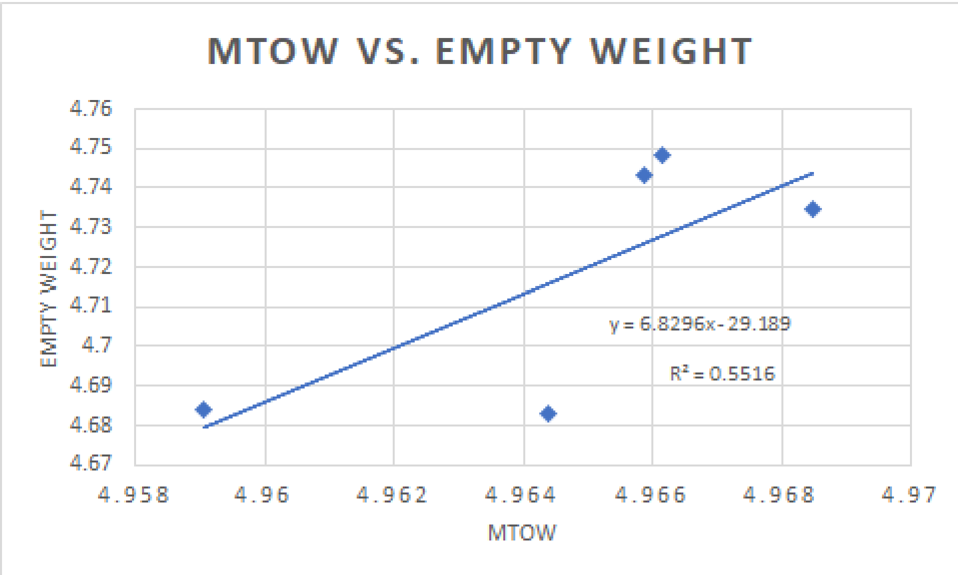


Figure 2: Graph of the MTOW vs Empty weight

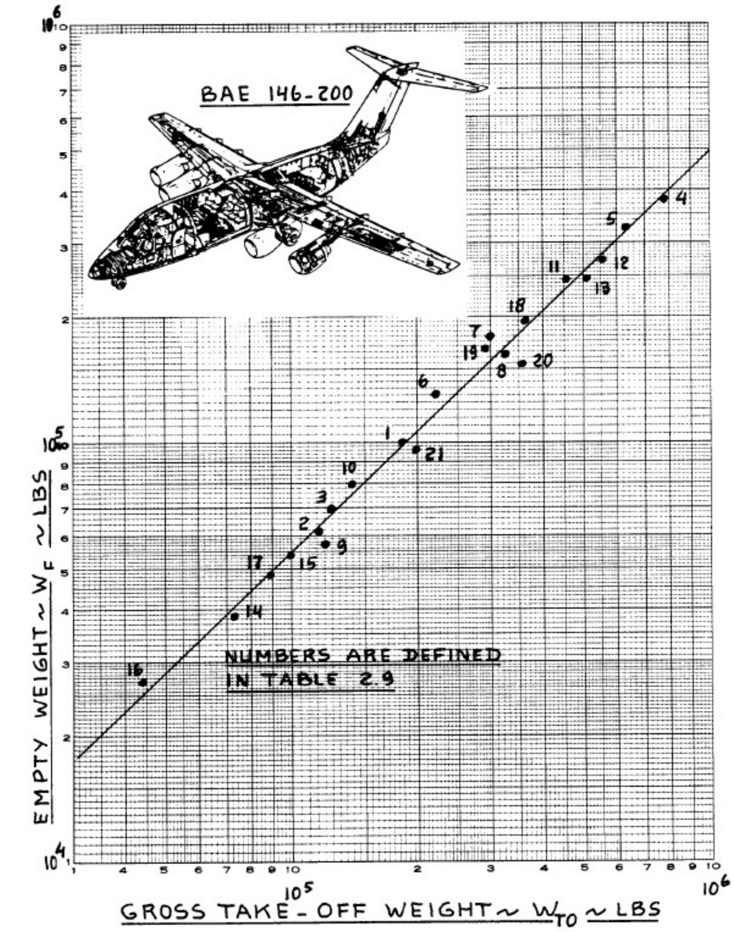


Figure 2.9 Weight Trends for Transport Jets

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The above graph in the figure 2 generates a linear equation in the form of "y=mx+c" as

$y=6.8296x-29.189$. Equation 1

This linear equation 1 is compared with the natural log equation that is provided in the Jan Roskam book and the equation is given as

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

Here W_{TO} means the Maximum Takeoff Weight which will be assumed during designing the preliminary design of aircraft. By comparing the equation 1 and equation, we obtained the Empty Weight.

$\log_{10}W_e = (1/B)*\log_{10}W_{TO} -(A/B)$ Equation 3

From equation 1,

$A/B = 29.189$

1/B = 6.8296

A= 4.2738

B= 0.1464

$$\log_{10}W_e = 6.8296*\log_{10}(90{,}000) –(29.189)$$

$$\log_{10}W_e = 6.8296*4.9542 –(29.189)$$

$$W_e = 44309.27 \text{ lbs}$$

W_e = 44309.27 lbs Equation 4

From equation 3, we get the value for the Empty Weight of the airplane. Thus, the variation between the manual calculations of the Empty weight estimation using the formulas provided in the Book should vary by about +/-5 the Empty Weight obtained through the Regression coefficients.

3.2.2.2 Determination of the Regression Coefficients using the AAA Program.

The regression coefficient is obtained after entering the takeoff and empty weights of the similar aircrafts in the AAA program as shown in the figure 2.1

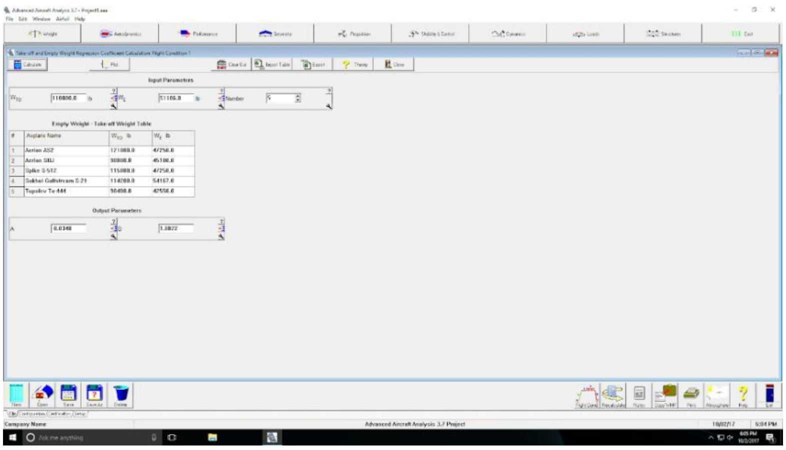


Figure 2.1: Table of Similar Airplanes obtained Using the AAA Program to calculate the regression coefficients.

The regression coefficients obtained from the AAA program are:

A= -0.0348

B= 1.0822

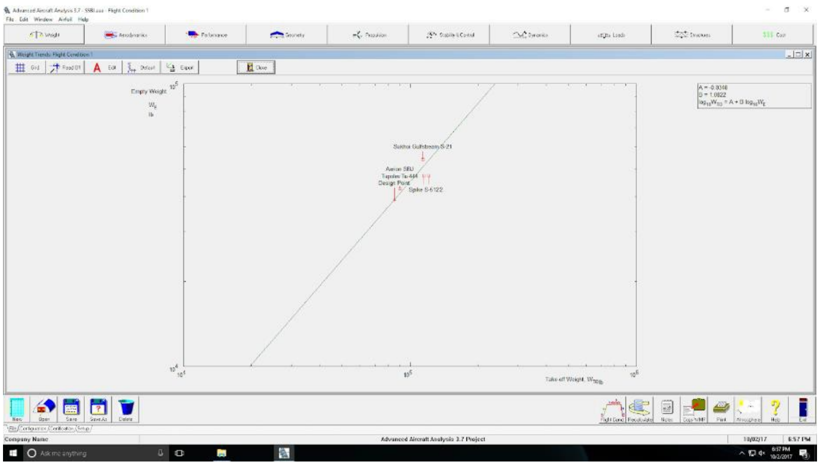


Figure 2.2: Graph of the regression coefficients obtained using AAA program

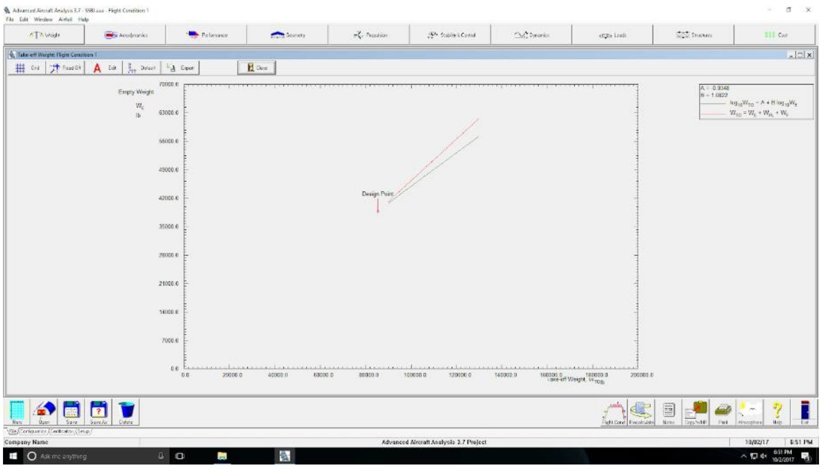


Figure 2.3: Design Point of the TSBJ Obtained from the AAA Program.

Regression Coefficients are obtained by entering the take-off and empty weights of the similar airplanes in the AAA program. There is a difference between the Regression Coefficients A and B obtained from the AAA program and from the manual calculations.

Following are the regression coefficients obtained from the AAA program:
A= -0.0348
B= 1.0822
On obtaining the above data, graph is plotted, and the design point is defined. It is the point where the TSBJ stands in comparison with similar aircrafts.

3.2.3 Determination of Mission Weights

3.2.3.1 Manual Calculation for the Mission Weights.

For the manual calculation of the weight estimation, the proposed range of the aircraft, maximum takeoff weight of the aircraft, the weight fractions of the aircraft at every stage of the flight path, the cruising speed, the flight path and the loiter time of the aircraft are required which can be obtained from the book.

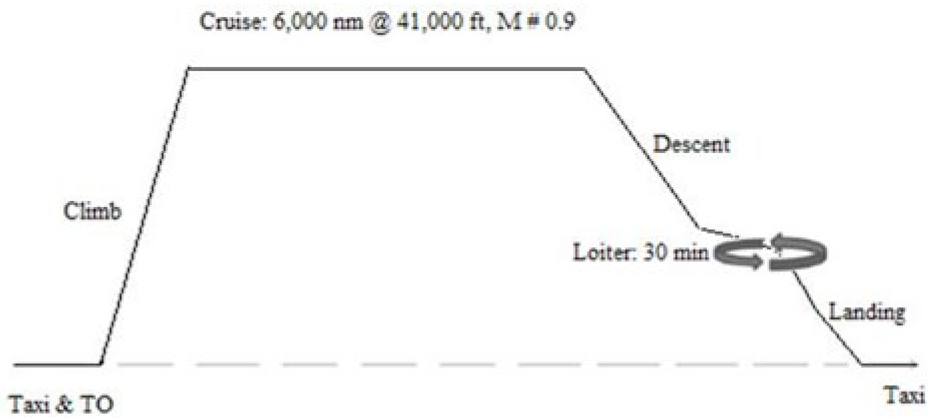


Figure 4: Mission Profile of long-range business jet

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

Hence, 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 can be calculated by using these formulas:

$$\text{Cruise:} \quad R_{cr} = \left(\frac{V}{C_j}\right)_{cr} \left(\frac{L}{D}\right)_{cr} \ln\left(\frac{W_4}{W_5}\right) \tag{4}$$

$$\text{Loiter:} \quad E_{ltr} = \left(\frac{1}{C_{j_{ltr}}}\right) \left(\frac{L}{D}\right)_{ltr} \ln\left(\frac{W_7}{W_8}\right) \tag{5}$$

Below are the fuel fractions at different stages of the flight:

Engine Start, Warmup: $\frac{W_1}{W_{TO}} = 0.990$ 6

Taxi: $\frac{W_2}{W_1} = 0.995$ 7

Takeoff: $\frac{W_3}{W_2} = 0.995$ 8

Climb: $\frac{W_4}{W_3} = 0.92$ 9

Cruise for 5000nm (using eqn 4): $\frac{W_5}{W_4} = \frac{W_{10}}{W_9}$

- $5753.897 = \frac{1227.63}{0.9} * 12 * \ln\left(\frac{W_4}{W_5}\right)$
- $\frac{W_4}{W_5} = 1.4212$
- $\frac{W_5}{W_4} = 0.7036$ 10

Descent: $\frac{W_6}{W_5} = \frac{W_{10}}{W_9} = 0.99$ 11

Cruise for 300nm (using eqn 4): $\frac{W_7}{W_6}$ and $\frac{W_9}{W_8}$

- $345.234 = \frac{613.815}{0.9} * 12 * \ln\left(\frac{W_5}{W_6}\right)$
- $\frac{W_5}{W_6} = 1.043$
- $\frac{W_6}{W_5} = \frac{W_8}{W_7} = 0.9586$ 12

Loiter (for 1 hour) (using eqn 5): $\frac{W_8}{W_7}$

- $1 = \frac{1}{0.8} * 14 * \ln\left(\frac{W_7}{W_8}\right)$
- $\frac{W_7}{W_8} = 1.0588$
- $\frac{W_8}{W_7} = 0.9444$ 13

Landing, Taxi, Shutdown: $\frac{W_{11}}{W_{10}} = 0.992$ 14

The M_{ff} (Maximum Fuel Fraction) can be found by taking the product of equations 6 through 14 As shown below:

$$M_{ff} = \frac{W_1}{W_{TO}} * \frac{W_2}{W_1} * \frac{W_3}{W_2} * \frac{W_4}{W_3} * \frac{W_5}{W_4} * \frac{W_6}{W_5} * \frac{W_7}{W_6} * \frac{W_8}{W_7} * \frac{W_9}{W_8} * \frac{W_{10}}{W_9} * \frac{W_{11}}{W_{10}} \quad 15$$

$$M_{ff} = 0.99 * 0.995 * 0.995 * 0.95 * 0.7036 * 0.99 * 0.9586 * 0.9444 * 0.9586 * 0.99 * 0.992$$

$$M_{ff} = 0.5353 \quad 16$$

The amount of Fuel used ($M_{f_{used}}$)

$$W_{f_{used}} = \left(1 - M_{ff}\right) * M_{TO} \quad 17$$

$$W_{f_{used}} = (1 - 0.5353) * 110000$$

$$W_{f_{used}} = 51115.55\ lbs \quad 18$$

$$W_{f_{res}} = 5\%\ of\ W_{f_{used}}$$

$$W_{f_{res}} = 2555.77\ lbs \quad 19$$

$$W_f = W_{f_{used}} + W_{f_{res}} \quad 20$$

$$W_f = 53671.33\ lbs \quad 21$$

From the equation 21, we get the weight of fuel that is required to complete the entire journey.
The weight of the payload is calculated as:

$$W_{PL} = W_{passengers} + W_{crew} \quad 22$$

$$W_{passengers} + W_{crew} = 20 * 205 + 4 * 205$$

$$W_{PL} = 4920\ lbs \quad 23$$

The total weight of the payload is calculated by adding the weight of the crew, weight of the passengers (estimated to be 175 lbs. / person) and the baggage weight (30 lbs. / person). The total weight per person comes out to be 205 lbs. per person which when further multiplied with the number of passengers gives the total payload of the aircraft i.e. 4920 lbs. By removing the fuel weight and the payload from the maximum takeoff weight, the total empty weight of the aircraft can be calculated as shown below:

$$W_{E_{tent}} = W_{TO} - W_f - W_{PL} \quad 24$$

$$W_{E_{tent}} = 110000 - 53671.33 - 4920$$

$$W_{E_{tent}} = 51408.66\ lbs \quad 25$$

The equation 25 shows the empty weight of the aircraft calculated manually.
Below is the difference obtained by comparing the empty weights of the aircraft obtained from the regression coefficients and manual calculations of the weight estimation process:

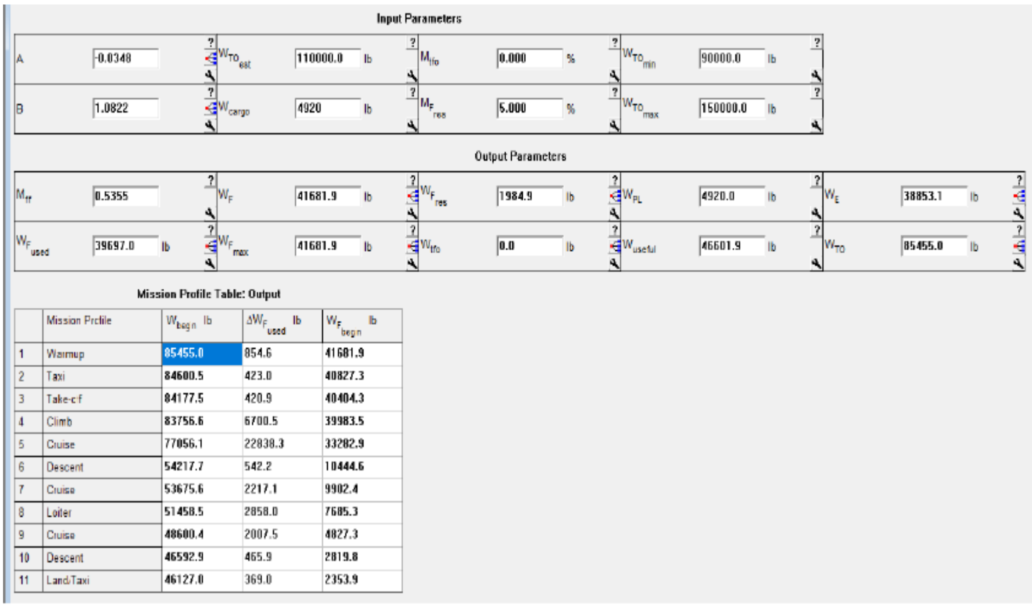
From equation 3 and 25:

$$Difference\ in\ empty\ weights = W_{E_{tent}} - W_E = 51408.66 - 47966.57$$

$$Difference\ in\ empty\ weights = 3442.09 = \frac{3442.09}{47966.57} = 0.071 \quad 26$$

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

3.2.3.2 Calculations of Mission Weights using the AAA Program.


Input Parameters

A	-0.0348	$W_{TO_{est}}$	110000.0 lb	M_{tfo}	0.000 %	$W_{TO_{min}}$	90000.0 lb
B	1.0822	W_{cargo}	4920 lb	$M_{F_{res}}$	5.000 %	$W_{TO_{max}}$	150000.0 lb

Output Parameters

M_{ff}	0.5355	W_F	41681.9 lb	$W_{F_{res}}$	1984.9 lb	W_{PL}	4920.0 lb	W_E	38853.1 lb
$W_{F_{used}}$	39697.0 lb	$W_{F_{max}}$	41681.9 lb	W_{tfo}	0.0 lb	W_{useful}	46601.9 lb	W_{TO}	85455.0 lb

Mission Profile Table: Output

	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.

For manually calculating the takeoff weight sensitivities, values of C and D needs to be calculated which constants similar to the A and B regression coefficients. Using the following equation, the value of C and D is found:

$$W_E = W_{TO}\left\{1-(1+M_{res})\left(1-M_{ff}\right)-\ M_{tfo}\right\}-(W_{PL}+W_{crew})$$

where,

$$C = \left\{1-(1+M_{res})\left(1-M_{ff}\right)-\ M_{tfo}\right\} \quad 27$$

$$D = \ (W_{PL}+W_{crew}) \quad 28$$

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 sensitivity study is conducted to find out the parameters which drive the design, to determine the changes to be made in future in case a new mission capability is to be achieved and to calculate the estimate of the impact caused when changes are made to the design.

➢ **Takeoff weight sensitivities:**

$$log_{10}W_{TO} = A + Blog_{10}(CW_{TO} - D) \tag{30}$$

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

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

After solving the above equation, the allowable takeoff weight i.e. 99263.5 lbs. is obtained.

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

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

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

Following data is obtained from the preliminary design:
A= -6.069
B= 2.3736
C= 0.5122
D= 4920 lbs.
W_{TO}= 1, 10,000 lbs.
By substituting the data in equation 31, sensitivity of W_{TO} and W_{PL} is obtained.

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

Thus, according to the calculations, for every pound of payload added the take-off gross weight of the aircraft needs to be increased byy 3.60 lbs. Therefore, the factor 3.60 is termed as the growth factor due to the payload for TSBJ. The mission performance remains the same according to the results obtained.

➢ **Sensitivity of Takeoff Weight to Empty Weight:**

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

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

The derivative of the eqn 32 is as follows:

$$\frac{\partial W_{TO}}{\partial W_E} = BW_{TO}\left[invlog_{10}\left(\frac{\log_{10} W_{TO} - A}{B}\right)\right]^{-1} \quad 33$$

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

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

The above calculations show that to increase each pound in the empty weight, the take-off weight must be increased by 5.44 lbs. Here the factor 5.44 is the growth factor due to empty weight.

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

For the TSBJ, 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:

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

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:

$$\frac{\partial W_{TO}}{\partial R} = Fc_j \left(V \frac{L}{D}\right)^{-1}$$

$$\frac{\partial W_{TO}}{\partial R} = 13.788\ lbs/nm \qquad 36$$

$$\frac{\partial W_{TO}}{\partial E} = Fc_j \left(\frac{L}{D}\right)^{-1}$$

$$\frac{\partial W_{TO}}{\partial E} = 11{,}206.79\ lbs/hr \qquad 37$$

The above sensitivities show that when the mission specification is decreased by 1 nm, then the gross weight can be decreased by 13.788 lbs. Also, 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.

$$\frac{\partial W_{TO}}{\partial V} = -FRc_j \left(V^2 \frac{L}{D}\right)^{-1}$$

$$\frac{\partial W_{TO}}{\partial V} = -74.965\ lbs/kt \qquad 38$$

The above parameters show that when the cruise speed is increased without changing any other parameter, the gross weight gradually decreases.

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

With respect to the range requirement:

$$\frac{\partial W_{TO}}{\partial c_j} = FR \left(V \frac{L}{D}\right)^{-1} \qquad 39$$

$$\frac{\partial W_{TO}}{\partial c_j} = 88{,}856.96\ lbs/lbs/lbs/hr$$

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.

$$\frac{\partial W_{TO}}{\partial \left(\frac{L}{D}\right)} = -FRc_j \left(V \left(\frac{L}{D}\right)^2\right)^{-1} \qquad 40$$

$$\frac{\partial W_{TO}}{\partial\left(\frac{L}{D}\right)} = -6664.27\ lbs.$$

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:

$$\frac{\partial W_{TO}}{\partial c_j} = \ FE\left(\frac{L}{D}\right)^{-1} \quad 41$$

$$\frac{\partial W_{TO}}{\partial c_j} = 14{,}008\ lbs/lbs/lbs/hr$$

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.

$$\frac{\partial W_{TO}}{\partial\left(\frac{L}{D}\right)} = \ -FEc_j\left(\frac{L}{D}\right)^{-2} \quad 42$$

$$\frac{\partial W_{TO}}{\partial\left(\frac{L}{D}\right)} = \ -800.48\ lbs$$

To improve the lift to drag ration during loiter is to be improved from 14 to 15 then, the gross takeoff weight needs to be reduced by 800.48 lbs.

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

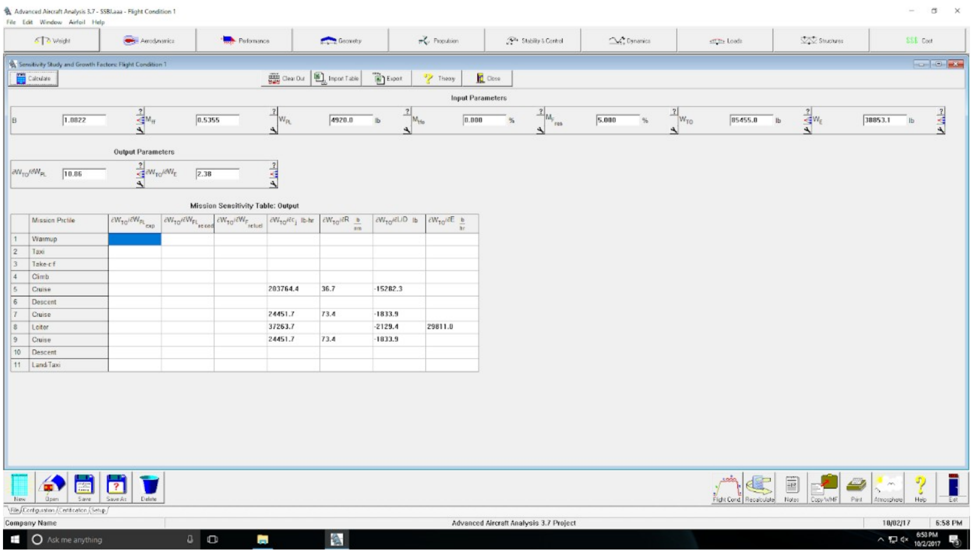


Figure 2.5: Weight Sensitivities using the AAA Program.

3.3.3 Trade Studies

The main objective behind the trade study is to get the best design point for the transonic jet being designed. The study is done over the parameters considered for the weight estimation. The first trade study is done between the range and the payload, keeping the maximum takeoff weight constant throughout the process. For the transonic aircraft being designed, the best design point is the 5000-nm range carrying a payload weight of 4920 lbs. The graph below shows that the range of the aircraft is inversely proportional to the payload. Profiles of any mission whether short range-more payload or long range-less payload can be created using this design.

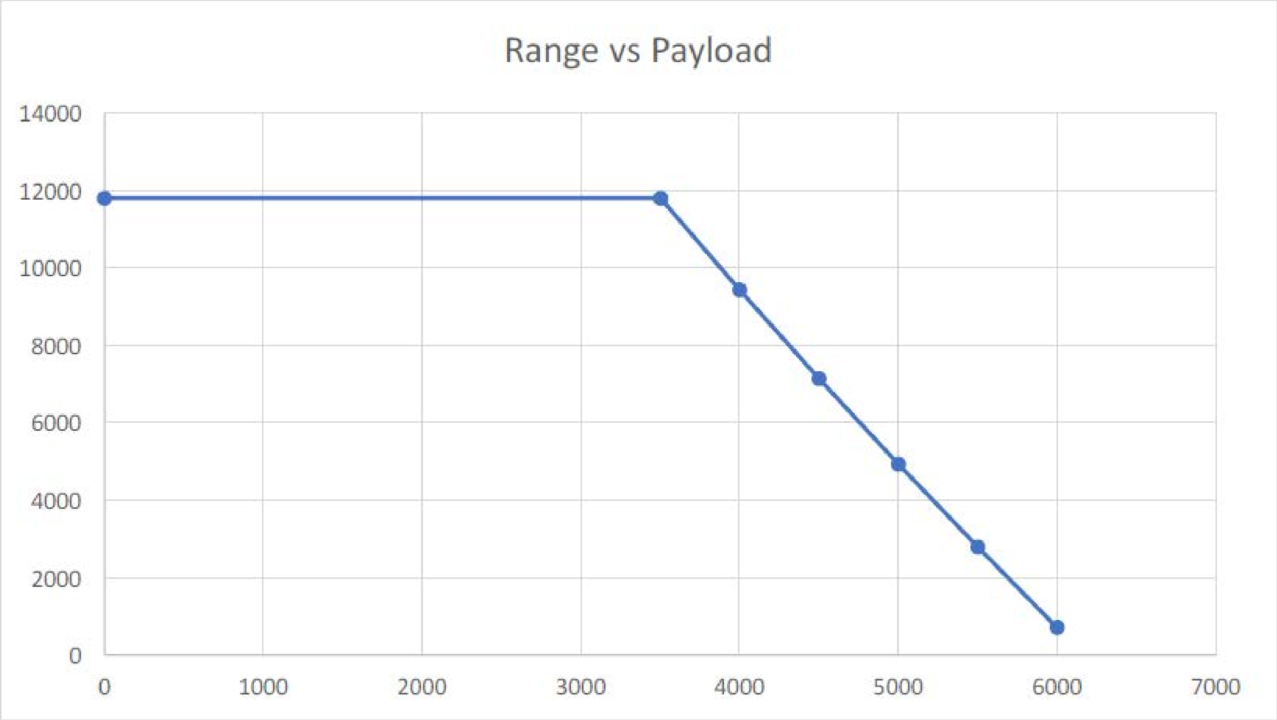


Figure 21: Trade study between the Range vs Payload

The second trade study is done between the maximum takeoff weight and the lift-to-drag ratio considering the empty weight of the aircraft to be constant. With the increase in the Lift-to-Drag ratio the weight of the fuel will decrease which in turn will decrease the maximum takeoff weight of the aircraft. Whereas, decreasing the L/D ratio will increase the take-off weight of the aircraft. According to the Aircraft Design book by Jan Roskam, 12 best value to be considered for the lift-to-drag ratio.

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

3.4 Discussion

This is the third report of the series and in this report the weight sizing and mission requirements are discussed. The regression points were calculated using the data of other similar airplanes at the first and after obtaining these points, the Empty weight and the Takeoff weights were calculated. Initially some values were assumed like maximum takeoff weight was assumed to be 1, 10,000 lbs. After calculating manually, the regression coefficients, AAA program was used to calculate the same. The points obtained through manual calculations are:

A= -6.069
B= 2.3736
Whereas the points obtained through the AAA program are:
A= -0.0348
B= 1.0822

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

On obtaining all the data regarding the weights, sensitivities of the parameters are conducted. It is conducted by changing the values of the parameters to see what changes affect the design of the aircraft. Manual calculations and AAA program are used to conduct the sensitivities and minor difference between the values was obtained using these two methods.

Trade studies and the takeoff weight sensitivities can be found almost similar to each other. The sensitivities show the difference between the designs upon changing the values by one unit. Whereas the trade studies show what parameters can be traded to obtain the other. For example: the trade study for range vs. payload proves that to increase the range, the payload must be decreased keeping constant the maximum takeoff weight.

3.5 CONCLUSION AND RECOMMENDATIONS

3.5.1 Conclusion

The primary aim of this report is to calculate the weight sizing of the Transonic Business Jet by making use of the manual calculation which itself is done by using the formulas obtained from the Airplane design book by Jan Roskam and by making use of the AAA program. Upon obtaining all the data, they shall be compared between the two methods of calculating the weight estimations to derive the most accurate results which shall be used to design the aircraft. It is of immense importance to compare the aircraft being designed with airplanes similar to it, to examine if the calculations or the weight estimations are comparable or not.

The outcomes obtained from the above-mentioned calculations can be summarized in the following ways:

•The empty weight and the maximum takeoff weight are assumed to be constant. Hence, the parameters that will vary for the different kind of missions are the range, payload, total weight of the fuel required.

•The Regression coefficients which are derived from the AAA program starkly vary from the coefficients which are derived via manual calculations. The coefficients derived from the AAA program are A= -0.0348 and B= 1.0822 while the coefficients derived from the manual calculations are A= -6.069 and B= 2.3736.

•The sensitivity study highlights that the minor changes to the parameters can affect design of the airplane. Even if one parameter is improved, the requirements of the other parameter are disturbed.

3.5.2 Recommendations

Various lessons were learnt via this report. Lessons such as the weight sizing of the aircraft. The lack of freedom of changing the parameters was also learnt from the results of the trade studies. Also, the future works describe the designing of the other parts of the aircraft in a detailed manner.

CHAPTER 4: PERFORMANCE CONSTRAINT ANALYSIS

4.1 INTRODUCTION

This is the fourth report, and the primary objective of the report is to prepare a list of the performance constraints and to highlight all the calculation which are essential in determining the size of the airplane. The airplanes are specifically designed so that they meet performance objectives in the categories given below:

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

Multiple calculations shall be performed, these shall enable the rapid estimation of the airplane design parameters which shall have a primary impact in the design of the airplane. Some of these parameters are listed below:

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 host of values of wing loading (W/S), thrust loading (T/W) and the maximum lift coefficient (CLMAX are considered and it is within these values that certain performance requirements are to be met. Considering the data obtained, the airplane with the lowest possible weight and the lowest possible cost can be obtained upon considering the lowest possible thrust loading and the highest possible wing loading while continuing to meet all the necessary performance requirements. The Wing Area (S) and the Takeoff Thrust (TTO) can be derived from these calculations.

Upon obtaining the manual calculations, the AAA program shall be used to perform the calculations and then derive the values of the performance Constraints. These derived values shall then be compared with the calculations obtained from the manual calculations. Then when the results are compared, they shall be summarized.

The propulsion system that shall be selected, must be sufficient and one that matches the requirements along with the number of engines to be used in accordance to the design specification. This is to derive the suitable thrust for the airplane to fly in transonic speeds. The propulsion system shall be specified along with its performance, its sizing, and the components of the engine. It shall also depend upon how it fulfils the necessities of the airplane.

4.2 MANUAL CALCULATION OF PERFORMANCE CONSTRAINTS

4.2.1 Stall Speed

In the case of majority of the airplanes, a stall speed that is not higher than some minimum value is required. The airplanes which are certified under the FAR 25 categories do not have a minimum stall speed requirement. The stall speed can be calculated via the following equation:

$$V_s = \left[\frac{2*\frac{W}{S}}{\rho * C_{L_{MAX}}}\right]^{\frac{1}{2}} \quad 1$$

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

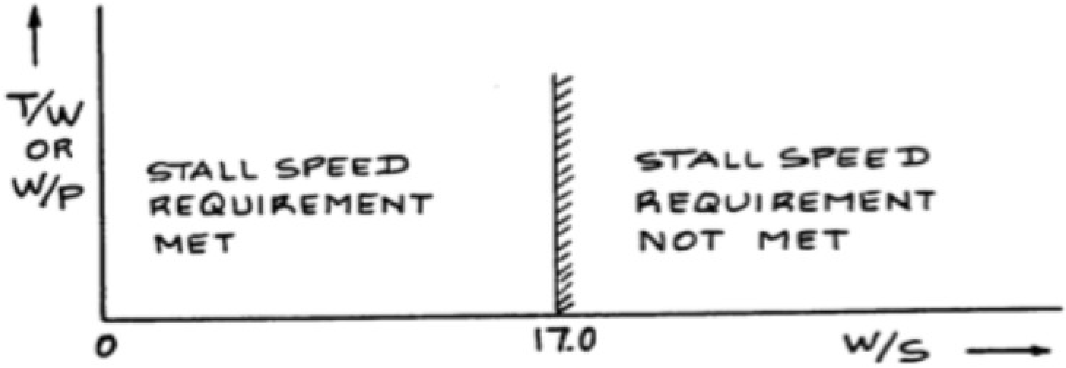


Figure 23: Example of the Stall speed sizing

The values of Maximum Coefficient of Lift can be obtained from the below table:

	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 maximum lift coefficient ranges from 1.2 to 2.2 for transonic 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 $(V_s) = 190\ Knots$

$$190 = \left[\frac{2*(\frac{W}{S})}{0.002377*1.8}\right]^{\frac{1}{2}}$$

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

similarly, the values of the Wing Loading W/S for all the values of $C_{L_{MAX}}$ in all 3 states, clear 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

As per the table above, the values of the wing loading vary at different stages and conditions of the flights path. As per the Aircraft design book by Jan Roskam, the actual

value of the W/S considered is the value at maximum coefficient of lift during the clean stage that is 77.22 lb. /ft2.
The wing loading of the aircraft are like my design is:

Table 3.3: Wing Loading for similar aircrafts:

Aircraft	W/S (lb/ft^2)
Gulfstream G550	78.9
Gulfstream G650	77.7
Bombardier Global G5000	95.9
Bombardier Global G6000	94.7
Bombardier Global G7000	95.7

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

There is a wide difference in take-off field lengths and they primarily depend upon the type of airplane which is being considered. For civil airplanes, the requirements for FAR 23 and FAR 25 must be met. The transonic 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_{LMAX}. This can be defined from the equation below:

$$s_{TOFL} = \frac{\left(\frac{W}{S}\right)_{TO}}{\sigma * C_{L_{MAX TO}} * \left(\frac{T}{W}\right)_{TO}} = TOP_{25}. \quad 3$$

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:

$$s_{TOFL} = 37.5 * \frac{\left(\frac{W}{S}\right)_{TO}}{\sigma * C_{L_{MAX TO}} * \left(\frac{T}{W}\right)_{TO}} = 37.5 * TOP_{25}. \quad 5$$

Hence from the above equation, we can say that

$$TOP_{25} = \frac{\left(\frac{W}{S}\right)_{TO}}{\sigma * C_{L_{MAX TO}} * \left(\frac{T}{W}\right)_{TO}} \quad 6$$

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

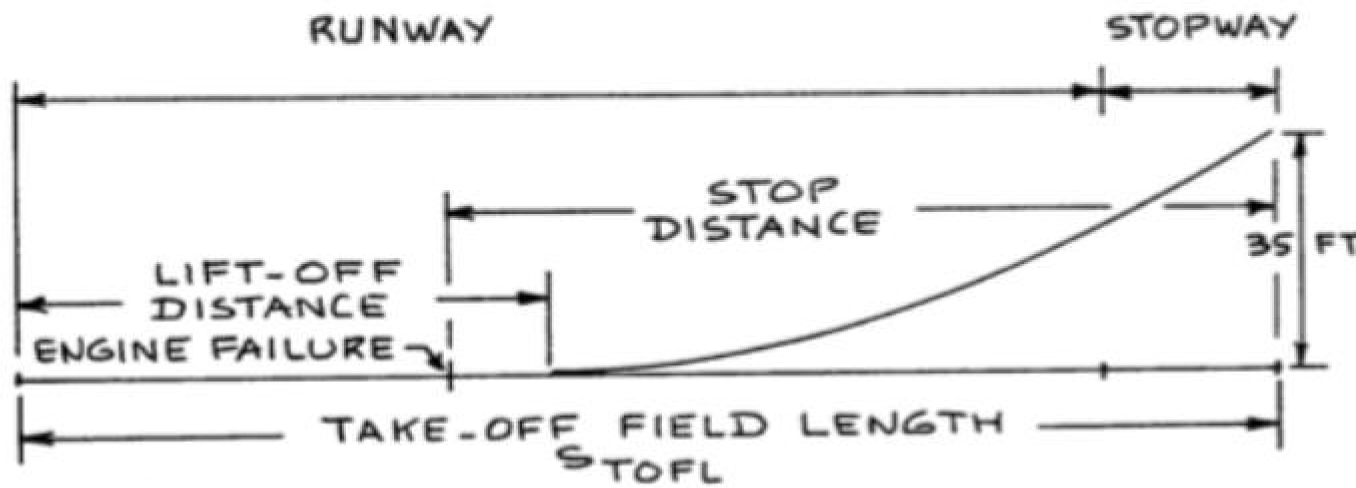


Figure 25: Definition of FAR 25 take-off distance

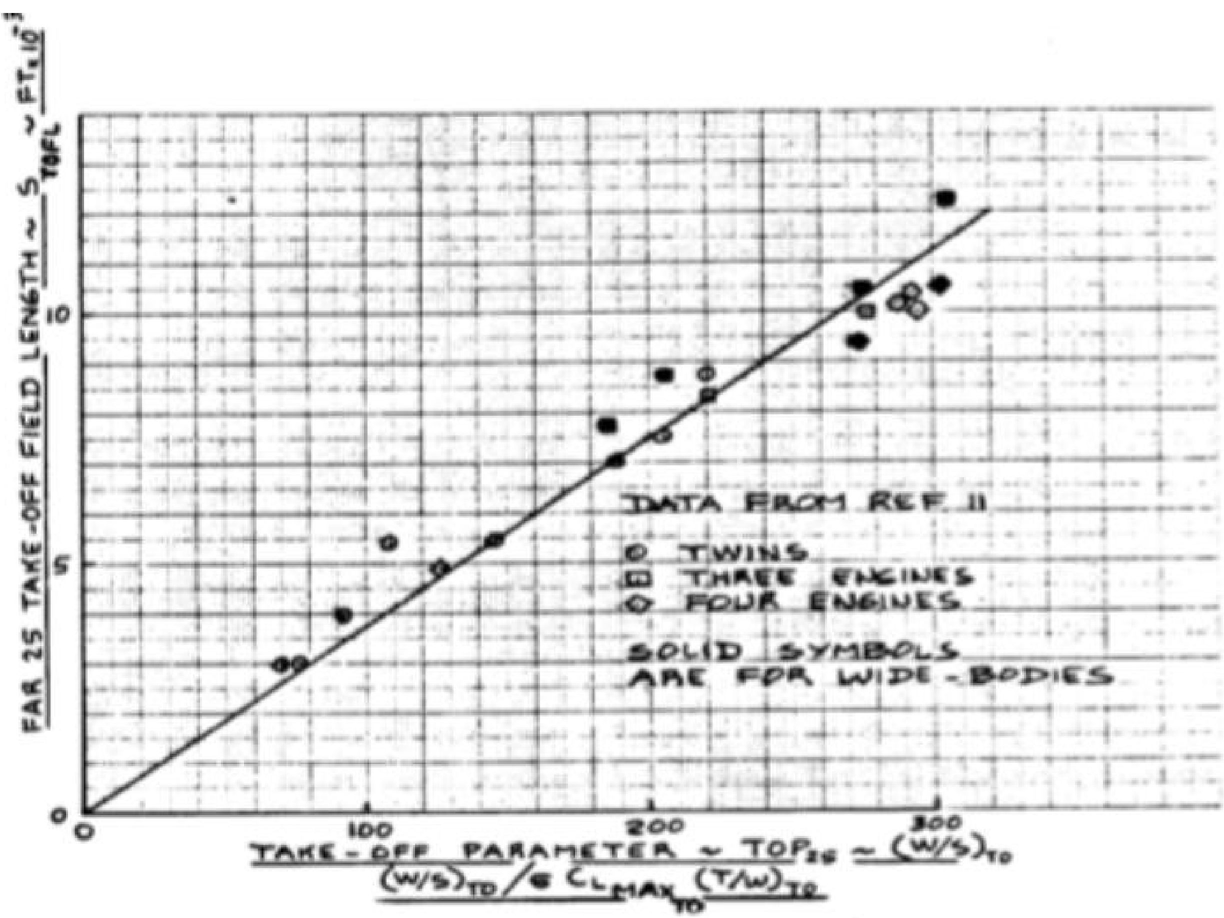


Figure 26: Effect of take-off parameter, TOP 25 on FAR 25 take-off field length

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

$$TOP_{25} = \frac{s_{TOFL}}{37.5} = \frac{5000}{37.5}$$

$$TOP_{25} = 133.34$$

Substituting the value of TOP_{25} , s_{TOFL}, $C_{L_{MAX_{TO}}}$, 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

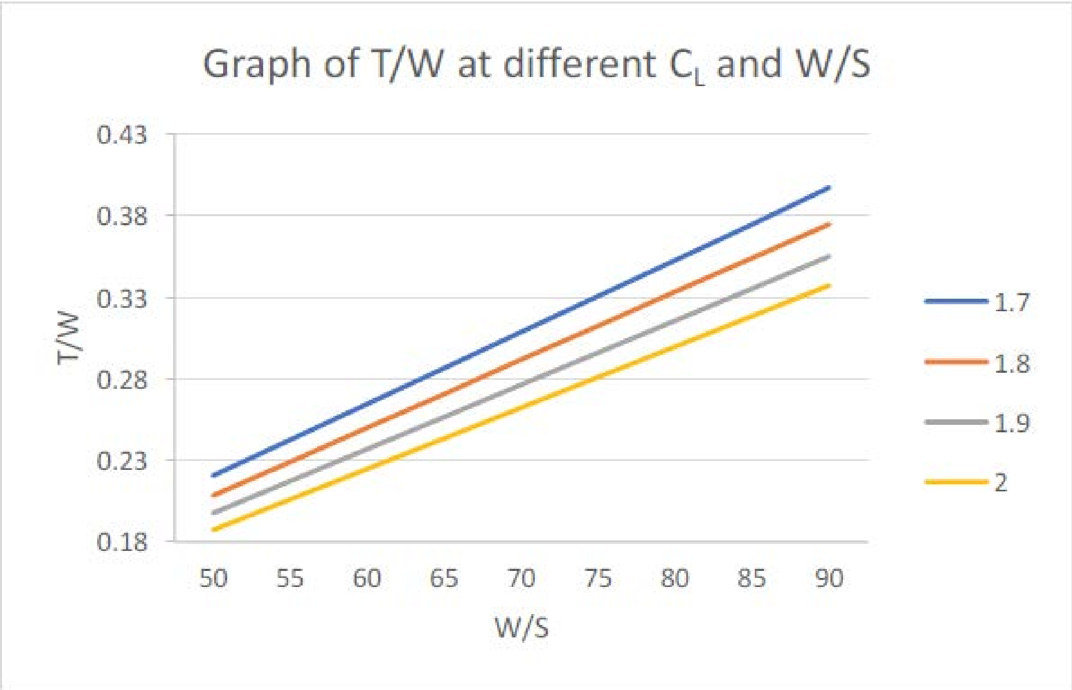


Figure 27: Graph of T/W at different C_L and W/S

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

There are five factors upon which landing distances of airplanes are dependent. Those factors are listed 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. When the kinetic energy is considered, the approach speed should have a ‘square effect’ on the total landing distance. When 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:

W_L / W_{TO}

	Airplane Type	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 transonic business jets are obtained from the figure above:

$$\left(\frac{W_L}{W_{TO}}\right)_{min} = 0.63$$
$$\left(\frac{W_L}{W_{TO}}\right)_{Avg} = 0.75$$
$$\left(\frac{W_L}{W_{TO}}\right)_{max} = 0.88$$

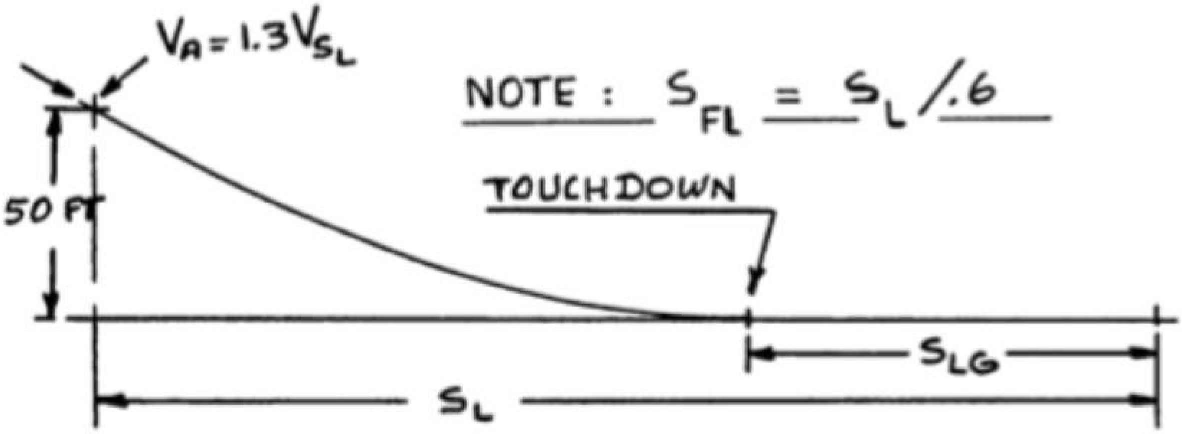


Figure 29: Definition of FAR 25 Landing Distance

The calculations to landing distance sizing is as follows:

$$V_{S_L} = \left[\frac{2*\frac{W}{S}}{\rho*C_{L_{MAX_L}}}\right]^{\frac{1}{2}} \quad 7$$

$$V_{S_L} = 136.41\ knots$$

$$V_A = 1.3 * V_{S_L} = 1.3 * 136.41 = 177.33\ knots \quad 8$$

$$s_{FL} = 0.3\, V_A^2 = 9434.09\ ft$$

$$s_L = s_{FL} * 0.6 = 5660\ ft \quad 9$$

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.

$$2\frac{\left(\frac{W}{S}\right)_L}{0.002377C_L} = 36099\frac{ft^2}{s^2}$$

$$\left(\frac{W}{S}\right)_L = 42.90\ C_{L_{max_L}} \qquad 10$$

$$\left(\frac{W}{S}\right)_{TO} = \frac{42.9}{0.85}\ C_{L_{max_{TO}}} = 50.47\ C_{L_{max_{TO}}} \qquad 11$$

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 determine the size of an airplane for climb requirements, it is of utmost importance 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 a function of the equivalent skin friction coefficient of an airplane, C_f. By estimating the drag, prediction of S_{wet} that correlates with the W_{TO} for wide range of aircrafts becomes convenient. The wetted area of an aircraft can be determined by the following equation:

$$\log_{10} S_{wet} = c + d log_{10} W_{TO} \qquad 13$$

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

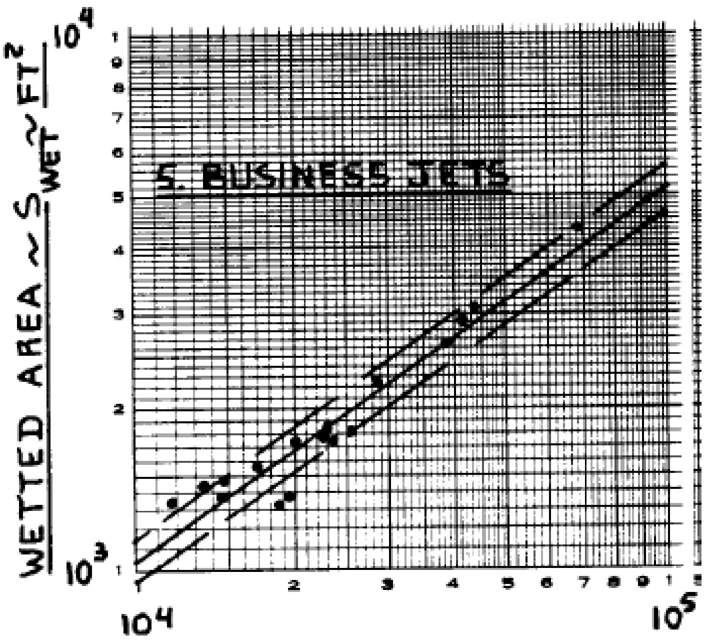


Figure 3.10: Graph between the wetted area and Maximum takeoff weight for business jet

Configuration	ΔC_{D_0}	e
Clean	0	0.80 - 0.8
Take-off flaps	0.010 - 0.020	0.75 - 0.8
Landing Flaps	0.055 - 0.075	0.70 - 0.7
Landing Gear	0.015 - 0.025	no effect

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

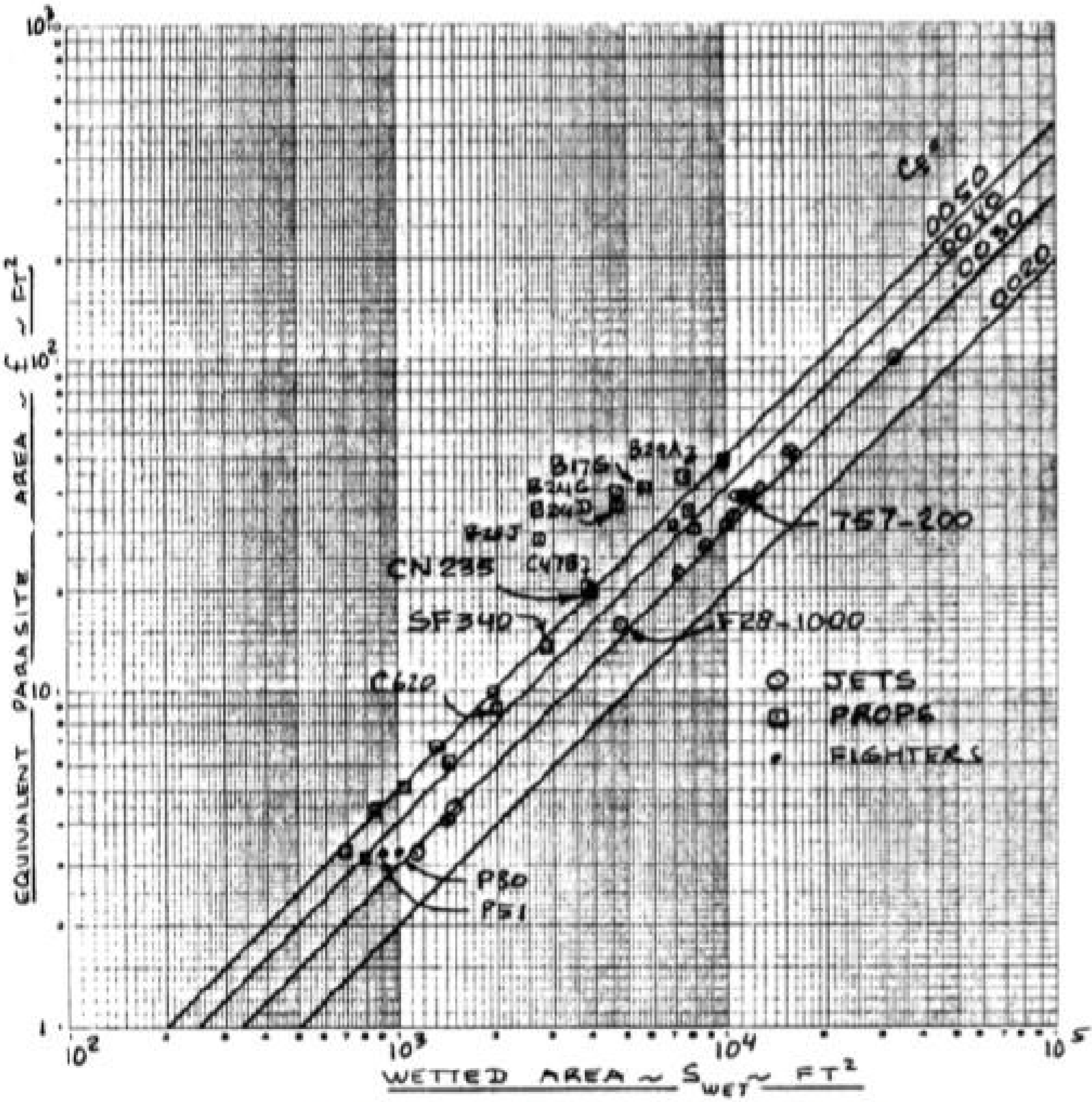


Figure 34: Relation between the wetted area and the equivalent parasite area

The values of transonic aircrafts are:

c = -1.1868

d = 0.9609

WTO = 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 cf is equal to 0.0030 hence from figure 8, we can get the value of a and b as:

Cf = 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:

$$C_{D_0} = 0.00299$$

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_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 35: estimates for Δ 〖C_D〗 _0 and e

Thus, upon substituting all the values obtained in equation 10 we derive 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: $C_D = 0.00299 + 0.0499\ C_L^2$ 14

Take-off, Gear up: $C_D = 0.01799 + 0.0530\ C_L^2$ 15

Take-off, Gear down: $C_D = 0.03799 + 0.0530\ C_L^2$ 16

Landing, Gear up: $C_D = 0.06299 + 0.0566\ C_L^2$ 17

Landing, Gear down: $C_D = 0.08299 + 0.0566\ C_L^2$ 18

4.2.5 Climb Constraints

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

The take-off climb requirements for FAR 25.111 OEI can be summarized 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

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

Following are the configurations of the initial climb segment requirements:

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 are as follows:

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

The above-mentioned requirements are for the following conditions:

1. Take-off flaps
2. Landing gear down
3. Remaining engines at take-off thrust
4. Between VLOF and V2
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 V2
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 1.25VS
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 VS
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 1.5 V_{SL}.
4. V_{SA} must not be more than 1.1 V_{SL}
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):

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

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

$$\left(\frac{T}{W}\right) = \left[\left(\frac{L}{W}\right)^{-1} + CGR\right] \quad 20$$

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):

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

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

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

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

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

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

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

$\left(\frac{T}{W}\right)_{TO} = 2\left[\frac{1}{\frac{L}{D}} + 0.024\right], at\ 1.2\ V_{S_{TO}}$ 24

C_D= 0.1202

L/D = 11.55

$\left(\frac{T}{W}\right)_{TO} = 0.2211$

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

$\left(\frac{T}{W}\right)_{TO} = 2\left[\frac{1}{\frac{L}{D}} + 0.012\right], at\ 1.25\ V_S$ 25

$C_L = 1.8 = \frac{1.8}{1.25^2} = 1.152$

$C_D = 0.069$

L/D = 16.695

$\left(\frac{T}{W}\right)_{TO} = 0.1438$

For FAR 25.119 (AEO): balked landing

$$\left(\frac{T}{W}\right)_L = 2\left[\frac{1}{\frac{L}{D}} + 0.032\right], at\ 1.3\ V_{s_L}$$ 26 C_L

= 1.301

C_D = 0.1688

L/D = 7.707

$$\left(\frac{T}{W}\right)_L = 0.3235$$

For FAR 25.121 (AEO): balked landing

$$\left(\frac{T}{W}\right)_L = 2\left[\frac{1}{\frac{L}{D}} + 0.021\right], at\ 1.5\ V_{s_A}$$ 27

C_L = 0.933

C_D = 0.09975

L/D = 9.353

$$\left(\frac{T}{W}\right)_L = 0.2558$$

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:

$$RC = V\left[\left(\frac{T}{W}\right) - \frac{1}{\frac{L}{D}}\right] \quad 28$$

where, the velocity V = $\left[\frac{2\left(\frac{W}{S}\right)}{\rho\left(C_{D_0}\pi Ae\right)^{\frac{1}{2}}}\right]^{\frac{1}{2}}$ 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

These constraints are specifically for agriculture, utility, aerobatic and military airplanes.

4.2.7 Speed Constraints

Following equations are used to attain the maximum speed:

$$T_{reqd} = C_D qS \quad 31$$

$$W = C_L qS \quad 32$$

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

$$T_{reqd} = C_{D_0} qS + \frac{C_L^2 qS}{\pi Ae} \quad 33$$

dividing this equation by the weight, we get:

$$\left(\frac{T}{W}\right)_{reqd} = \frac{C_{D_0} qS}{W} + \frac{W}{qS\pi Ae} \quad 34$$

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

$$\left(\frac{T}{W}\right)_{reqd} = \frac{C_{D_0} q}{\frac{W}{S}} + \frac{\frac{W}{S}}{q\pi Ae} \quad 35$$

On substituting the values, the equation obtained is:

Substituting the values, we get the equation as:

$$\left(\frac{T}{W}\right)_{reqd} = \frac{4.044}{\frac{W}{S}} + \frac{\frac{W}{S}}{27074.39} \qquad 36$$

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

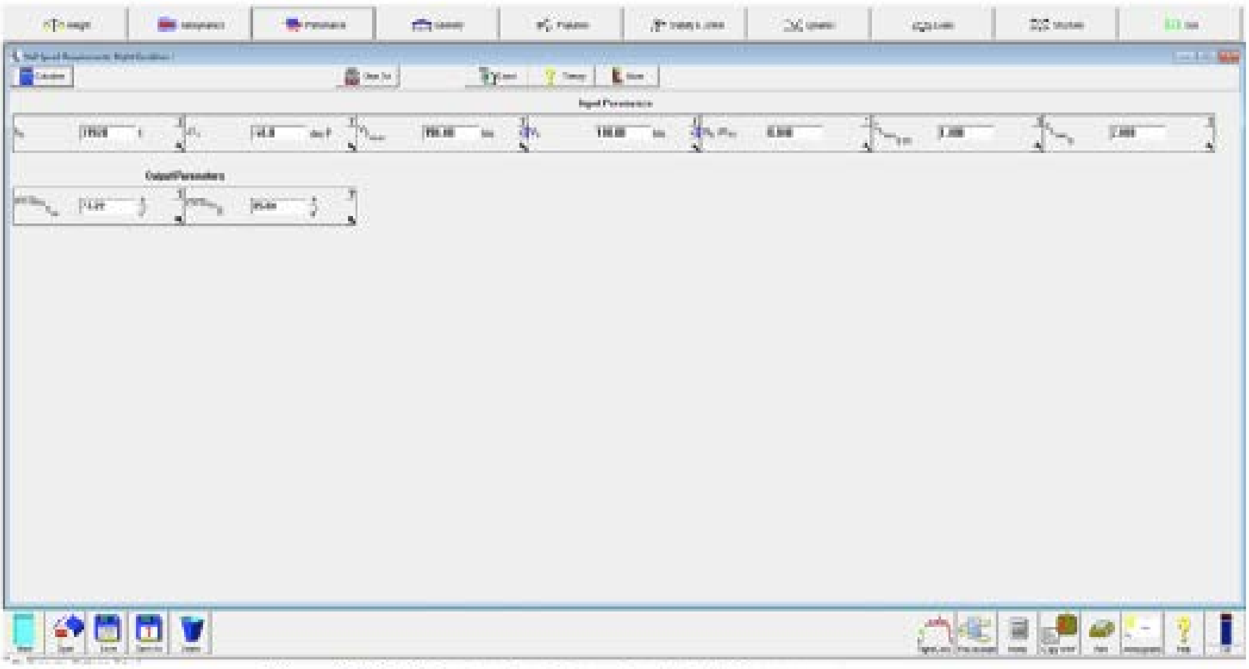


Figure 37: Stall Speed calculation using the AAA program

The stall speed assumed here is 190 knots. This 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. At the height of 31920 ft from the sea level the stall is

obtained, with temperature difference being 50 deg F less than that at sea level.

4.3.2 Takeoff Distance

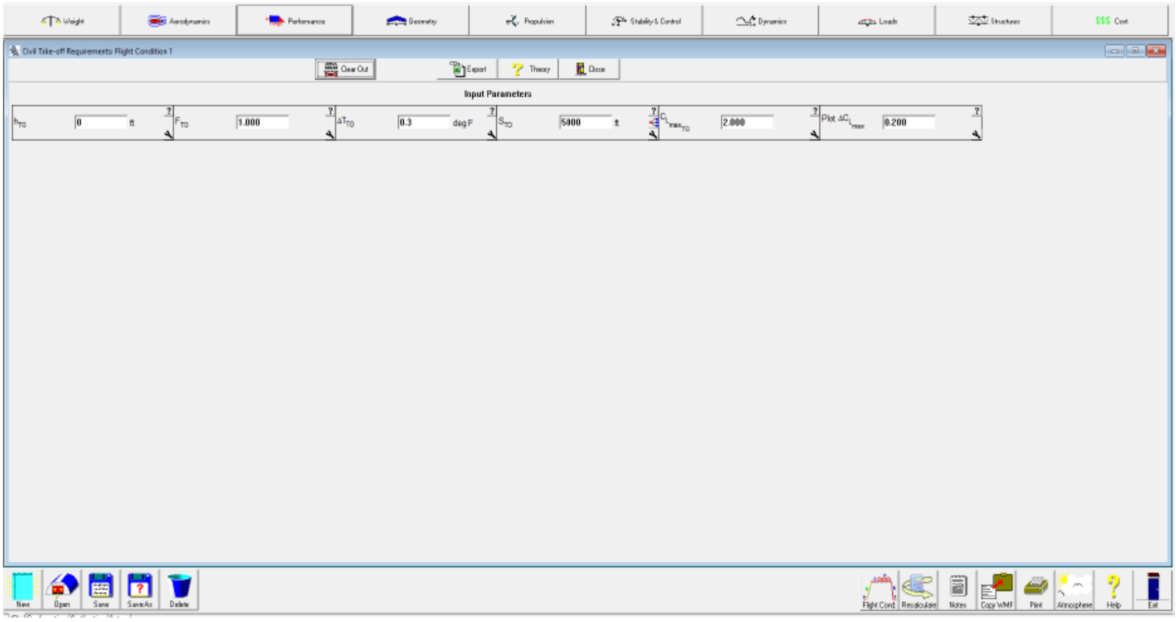


Figure 38: Take-off distance parameters using the AAA program

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

STO = 5000 ft

4.3.3 Landing Distance

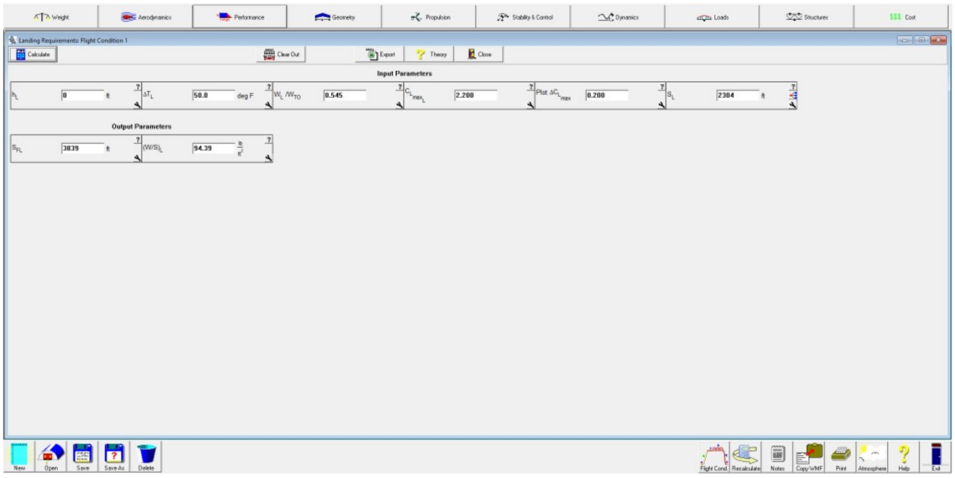


Figure 39: Landing distance calculations using the AAA program

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:

SFL = 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:

$$B_{DP_{clean}} = 0.0499$$

$$B_{DP_{TO\_down}} = 0.0531$$

$$B_{DP_{L\_down}} = 0.0566$$

4.3.5 Climb Constraints

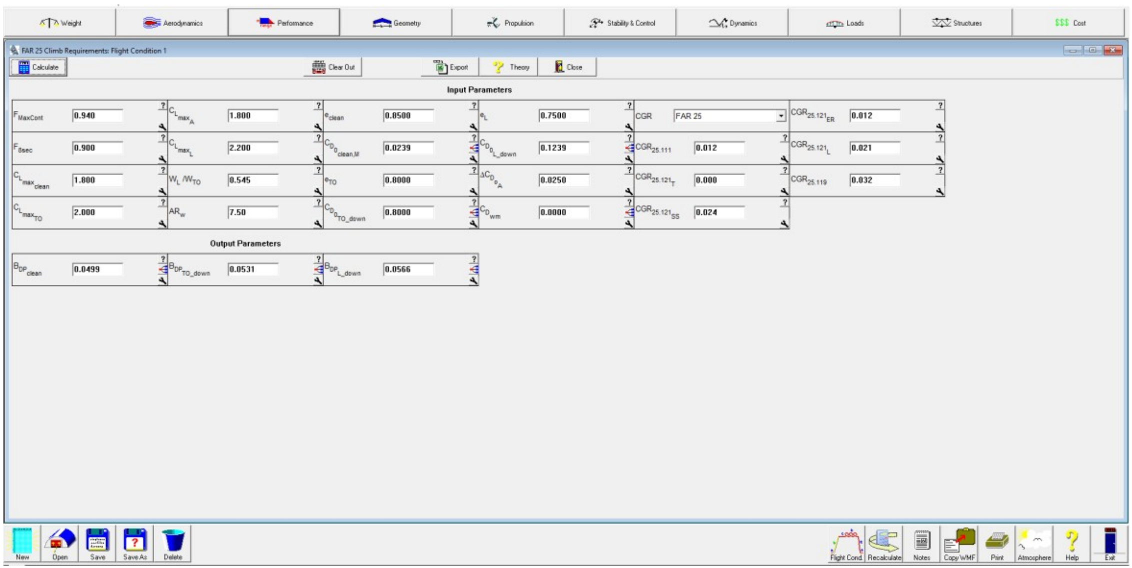


Figure 40: Climb constraint calculations using the AAA program

4.3.6 Maneuvering Constraints

Maneuvering Constraints are specifically for the Agricultural, military and training airplanes and not for the transonic jet airplanes.

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4.3.7 Speed Constraints

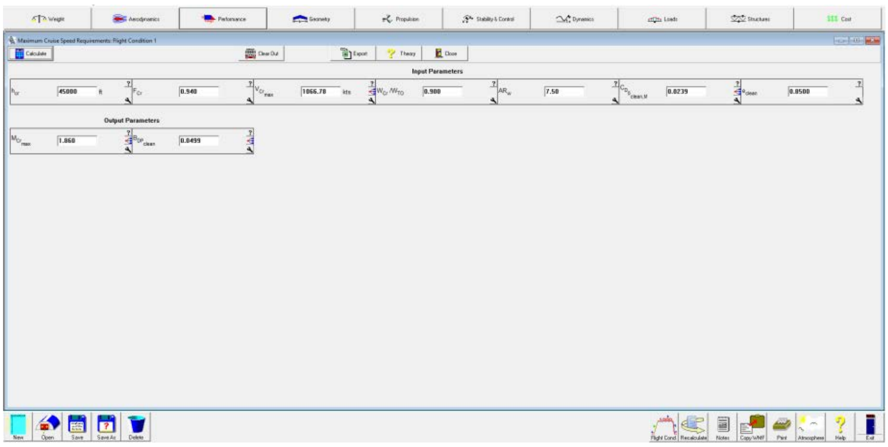


Figure 41: Cruise speed calculations using the AAA program

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
E_{clean} = 0.8500
Aspect ratio = 7.5

Providing this data to the software, the Maximum Cruising Mach number is obtained:

$$M_{cr_{max}} = 1.860\ Mach$$

4.3.8 Summary of Performance Constraints

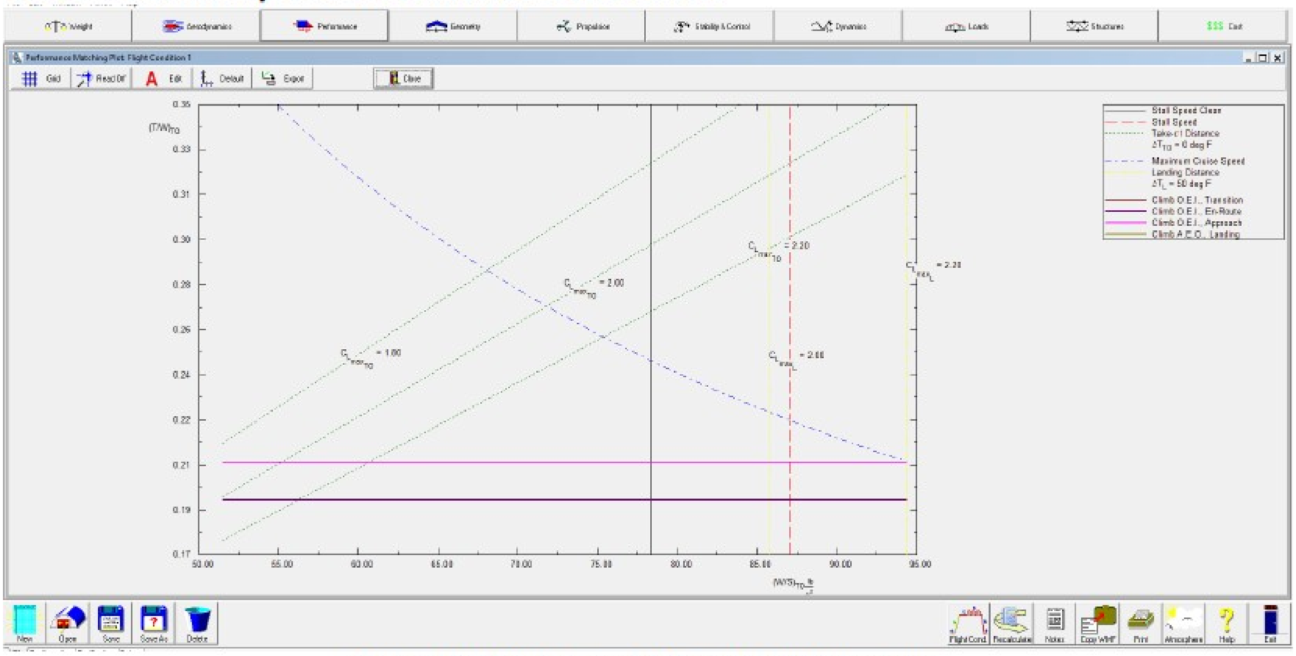


Figure 42: Matching graph from the AAA Program

4.4 SELECTION OF PROPULSION SYSTEM

4.4.1 Selection of Propulsion System Type

Propulsion systems used for similar airplanes are described as follows:
The 2 x Pratt and Whitney J58 Engines produces 25,000 pound-force (110 kN) of thrust without the after burner and 34,000 (150 kN) of thrust with the after burner. The thrust obtained is enough for the transonic jets and because of these features; these engines will be used in the transonic jet being designed.

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.

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

Air flow: 300 lb. /s at take-off.

Specific Fuel Consumption: 1.9 lb. / (lbf.h)

Thrust to weight ratio: approximately 6

4.5 DISCUSSION

This being the fourth report, has the specifics of the performance sizing of the airplane. At the beginning, the wing loading was calculated using the stall speed requirements. 190 knots were the stall speed that was considered in comparison to similar airplanes. The wing loading (W/S) which was obtained from the considered stall speed and the provided coefficient of lifts was then compared to the wing loadings of airplanes of a similar kind.

The data which was obtained above, was later used to calculate the take- off parameters which would satisfy the FAR 25 requirements. Upon obtaining the take-off distance and landing distance, the relationship 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.

Upon obtaining the final data for the wing loading, the drag polar distance is then calculated, this in turn gives the parasite area, wing area, the wetted area, weight during take-off, weight while landing and the equivalent skin friction coefficient. Making use of all these terms, in accordance with the aspect ratio which is 7.5, we can calculate the coefficient of drag at different stages of flight with different configurations of the airplane. Again, by making use of this coefficient of drag, the lift to drag ratio required to the airplane at different stages of flight can be calculated.

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

Once we are done covering the climb configurations, we move towards the maneuvering constraint which is not applicable to the Transonic Business Jet as the data provided is satisfactory for the agricultural and military aircraft requirements. Later the speed constraints section provides the cruising speed of the airplane at the necessary altitude of 45000-55000 ft.

After the calculations, the AAA program was utilized, and these provided the data which was similar to the one obtained via manual calculations. In the end a matching graph was obtained, and this gave the entire data pertaining to the airplane performance constraints while showing the area where the airplane shall have the best performance according to the data acquired.

Later, it was specified so as to the number of engines to be used on the airplane. Also, the Pratt and Whitney J58 Engines was selected as it satisfied all the necessities of the airplane and later its specifications were discussed.

4.6 CONCLUSIONS AND RECOMMENDATIONS

4.6.1 Conclusions

Calculating the performance sizing of the transonic business jet using AAA program and was the aim of this report. When hand calculations data is derived it shall be compared with the data acquired via the AAA program to determine how valid the data obtained is in both the cases in comparison to airplanes of similar type. The results acquired in the report can be summarized in the following way:

- The thrust-to-weight ratios and the wing-loading ratios are inversely proportional to one another. Hence, it is necessary to maintain the ratios, or it could lead to complications in the journey of the airplane.
- The necessities for FAR 23 and FAR 25 are completely different. Therefore, it is important to design the parameters in accordance with the requirements in order to acquire near accurate data.
- If there are minor changes in the aircraft configurations it could make a big difference in the airplane right from the clean state to the flaps state, and the Lift to drag ratio (L/D) could be altered by a big difference.
- The airplane could crash if it is flying below the Stall Speed thereby making it is essential to fly above the given stall speed.
- In accordance with the altitude, each airplane has different performances thereby making it essential to fly at the satisfactory altitude to gain the best possible output from the specified design.
- During different stages of the flight wing loading varies, that is why the thrust to weight ratios vary in accordance with the wing loading within a certain defined range.

4.6.2 Recommendations

With the help of this report, immense knowledge has been acquired. the knowledge was pertaining to the performance of the airplane under numerous conditions, also the kind of changes which occur if there are minute changes to the design. The work ahead involves designing the other parts of the airplane.

CHAPTER 5: FUSELAGE DESIGN

5.1 INTRODUCTION

This being the fifth report of the preliminary design of the transonic business jet, it describes the cockpit and fuselage design. It was in the previous reports that the weight sizing, wing loading, performance parameters were calculated. This report specifically defines the design of the cockpit and the fuselage along with its dimensions. It is upon the following parameters that the design of the fuselage typically depends upon:

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:

1. Number and weight of cockpit crew members
2. Number and weight of cabin crew members
3. Number and weight of special duty crew members
4. Number and weight of passengers
5. Weight and volume of 'carry-on' baggage
6. Weight and volume of 'check-in' baggage
7. Weight and volume of cargo
8. Number, weight and size of cargo containers
9. Weight and volume of special operational equipment
10. Weight and volume of military payload
11. Weight and volume of fuel carried in fuselage
12. Radar equipment
13. Auxiliary power unit

The second section of this report details the preliminary design of the cockpit including the cockpit crew requirement, pilot visibility requirements, reachability of the pilot to the essential cockpit controls.

The third and the final section of the report describes the preliminary design of the fuselage section keeping in mind the number of passengers and the thickness of the fuselage.

5.2 LAYOUT DESIGN OF THE COCKPIT

It is usually with small or medium sized airplanes that the term cockpit is usually associated. The parameters listed below help in designing the layout of the cockpit of an airplane:

I. The positioning of the pilot and the cockpit crew members should be done in such a way that they can easily reach all the controls, without too much effort from their designated position.
II. It is without undue effort that all the essential instruments to the flight must be visible.
III. The pilot should be able to communicate without extra effort and by simple use of touch and voice.
IV. Minimum required visibility standards must be met from the cockpit.
V. It is essential to take into consideration the dimensions and weights of the crew members while designing the cockpit as the leg and arm motion required to be carried out for control manipulation of stick, throttles or wheel, rudder pedals and side arm controller should be ensured to be feasible.

The height of the crew members must also be factored in, while designing the cockpit of an airplane, the dimensions can be established according to the figure described below. The total height of the male crew member is described by A as shown below in figure 2.

To obtain the weights and dimensions of female crew members, the heights of the male crew members are to be multiplied by a factor of 0.85.

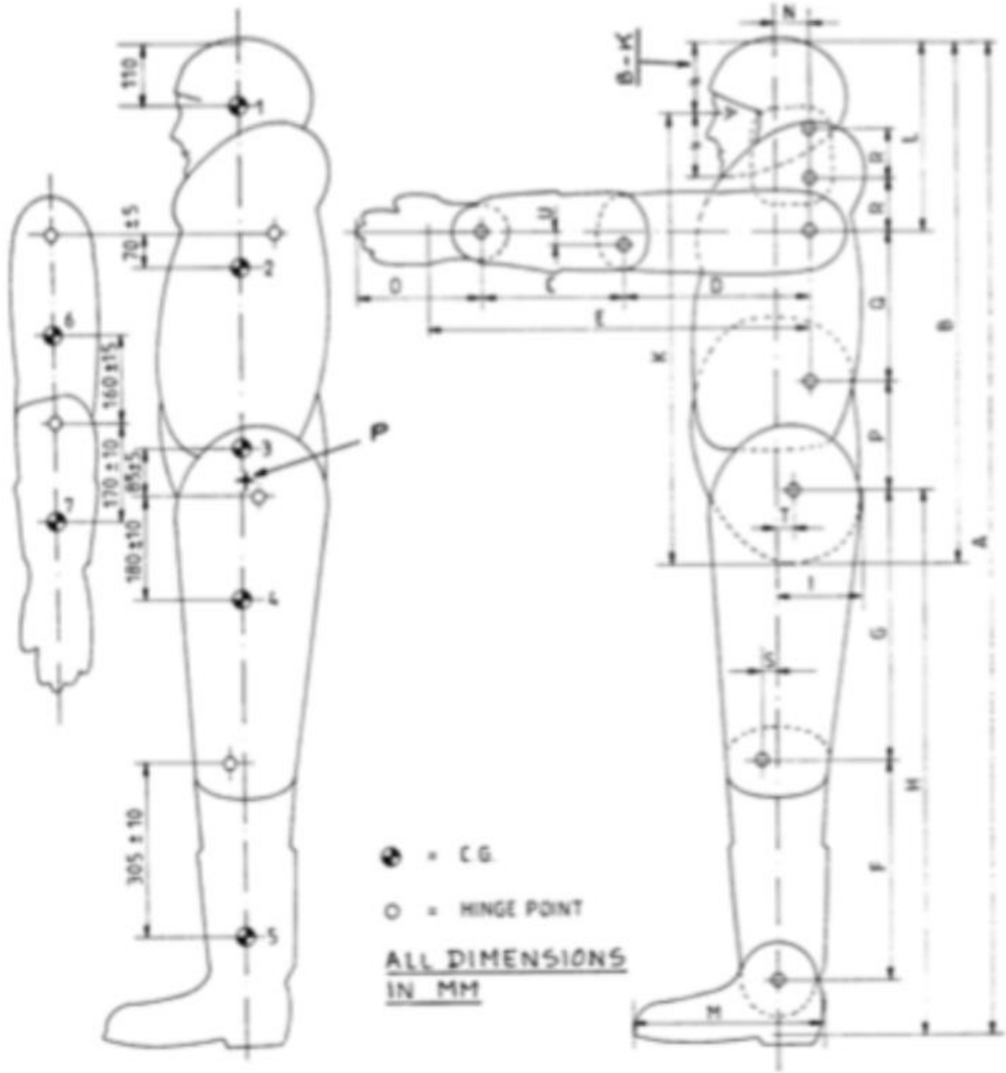


Figure 43: Dimensions of a standing male crew member

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 layout of the cockpit should have certain variations in the dimensional limitations of the human body. Every human comes in widely different sizes. Hence, the cockpit should be able to accommodate these variations. This can be obtained by arranging the rudder paddle adjustments and the seat position adjustments also.

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

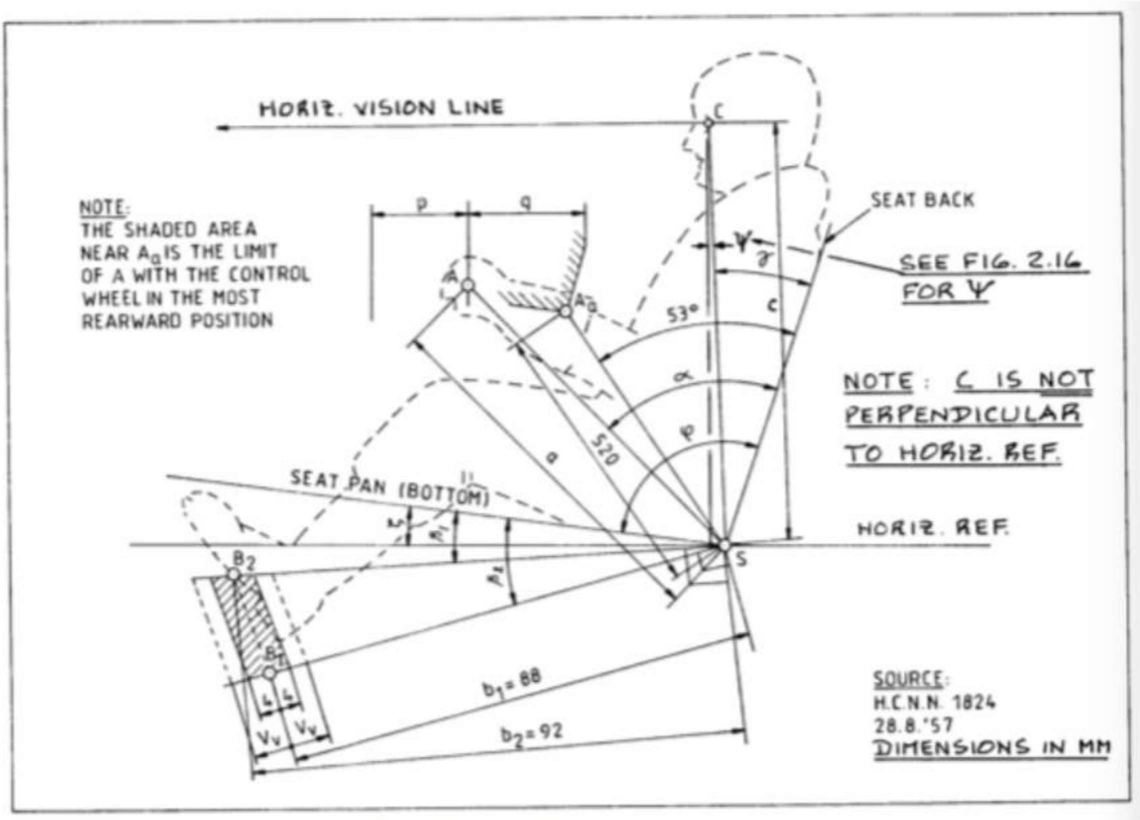


Figure 45: Pilot seat and controls arrangement

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

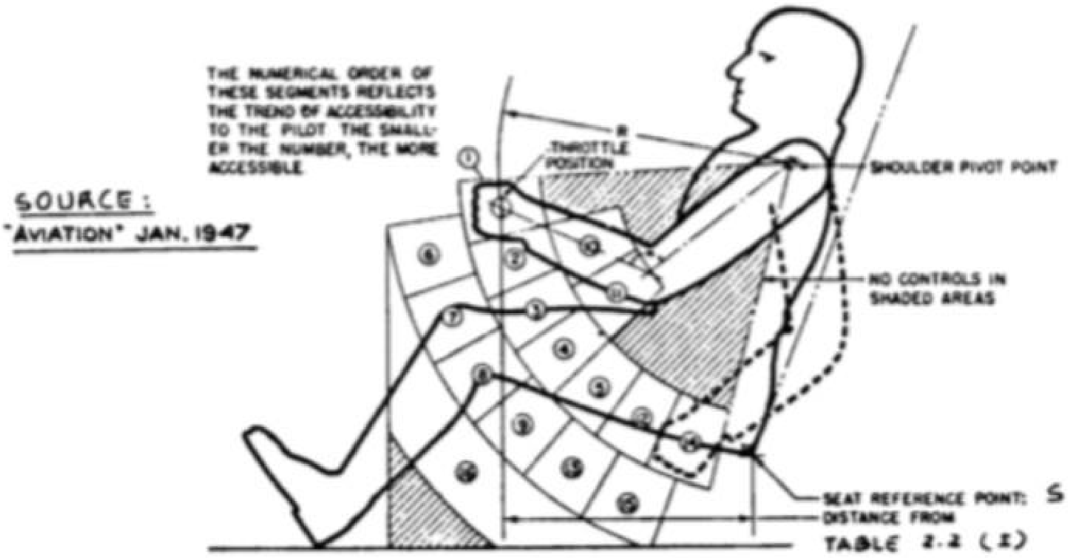


Figure 47: Areas of good and Poor accessibility

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

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

As there is no systematic relationship between each of these points, it is implied that several adjustments must be designed into cockpits. The above figure 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. Within the 5-degree arcs, flight essential crew members and their primary cockpit controls should not be located.
II. This are requirement must be met for propeller driven airplanes only according to the 23.771 and FAR 25.771.

The dimensions for seat adjustments and for the civil cockpit controls 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 several reasons:

- The pilot must have a good view of the immediate surroundings during take-off and landing operations.
- The pilot must be able to observe conflicting traffic during en-route operations.
- The success in combat depends upon good visibility and in fighters, formation flying is impossible without it.

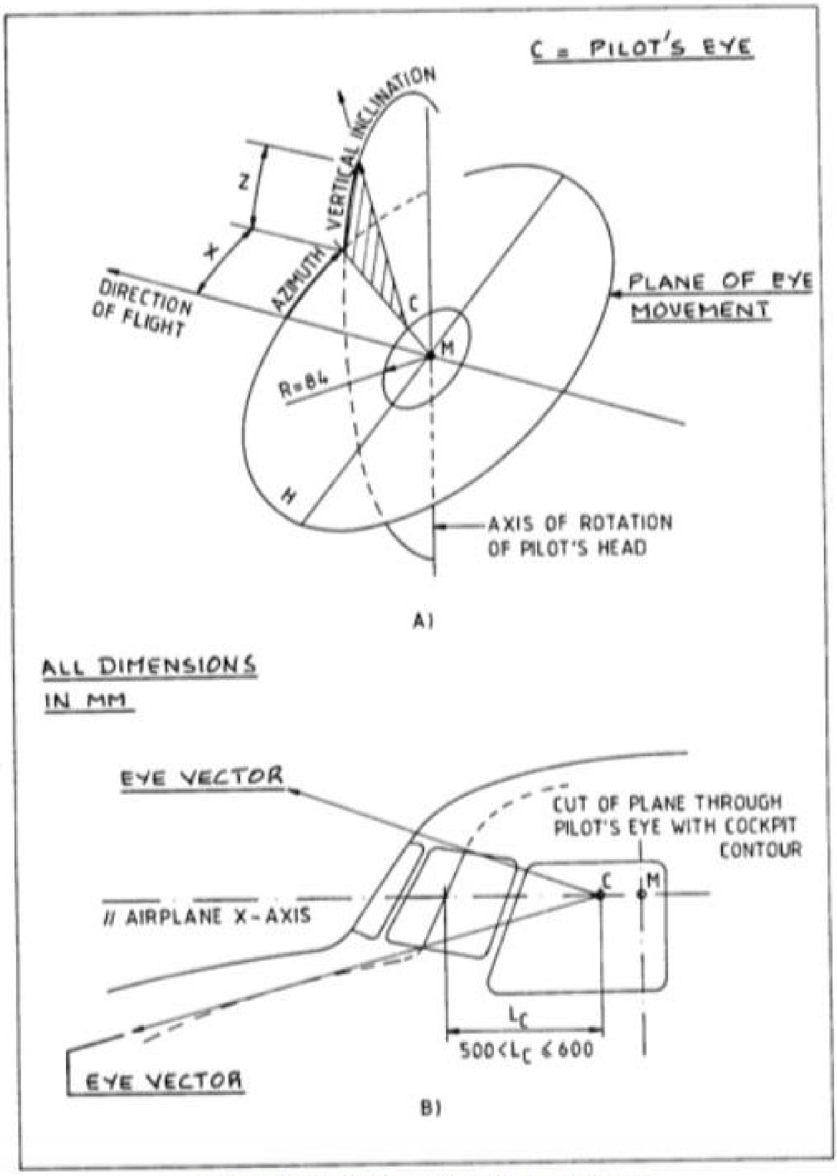


Figure 49: Definition of radial eye vectors

It was for the civil and military airplanes that minimum cockpit visibility rules were brought. Various types of airplanes have numerous types of requirements of visibility and these are designed in accordance to the customer demands. 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 is define as the required cockpit visibility. 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 is of utmost importance to locate the point C making use of 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 detailed 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 Y = 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 which have side-by-side pilot seating arrangements, there shall be no window frames will be in the area from 30 degrees starboard to 20-degree port.

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

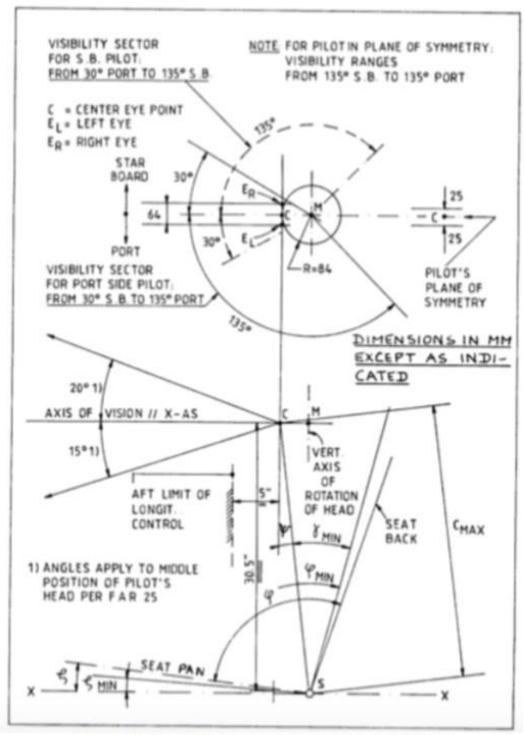


Figure 50: Visibility requirements for the port and starboard side.

Thus, stiff frames are required for larger windows require. The windows and the frames must comply with the bird strike requirements and this causes an increase in weight of the airplane. Another issue is the increase in drag, this is caused due to flat windows, it can be solved by using curved windows, but it results in image distortions.

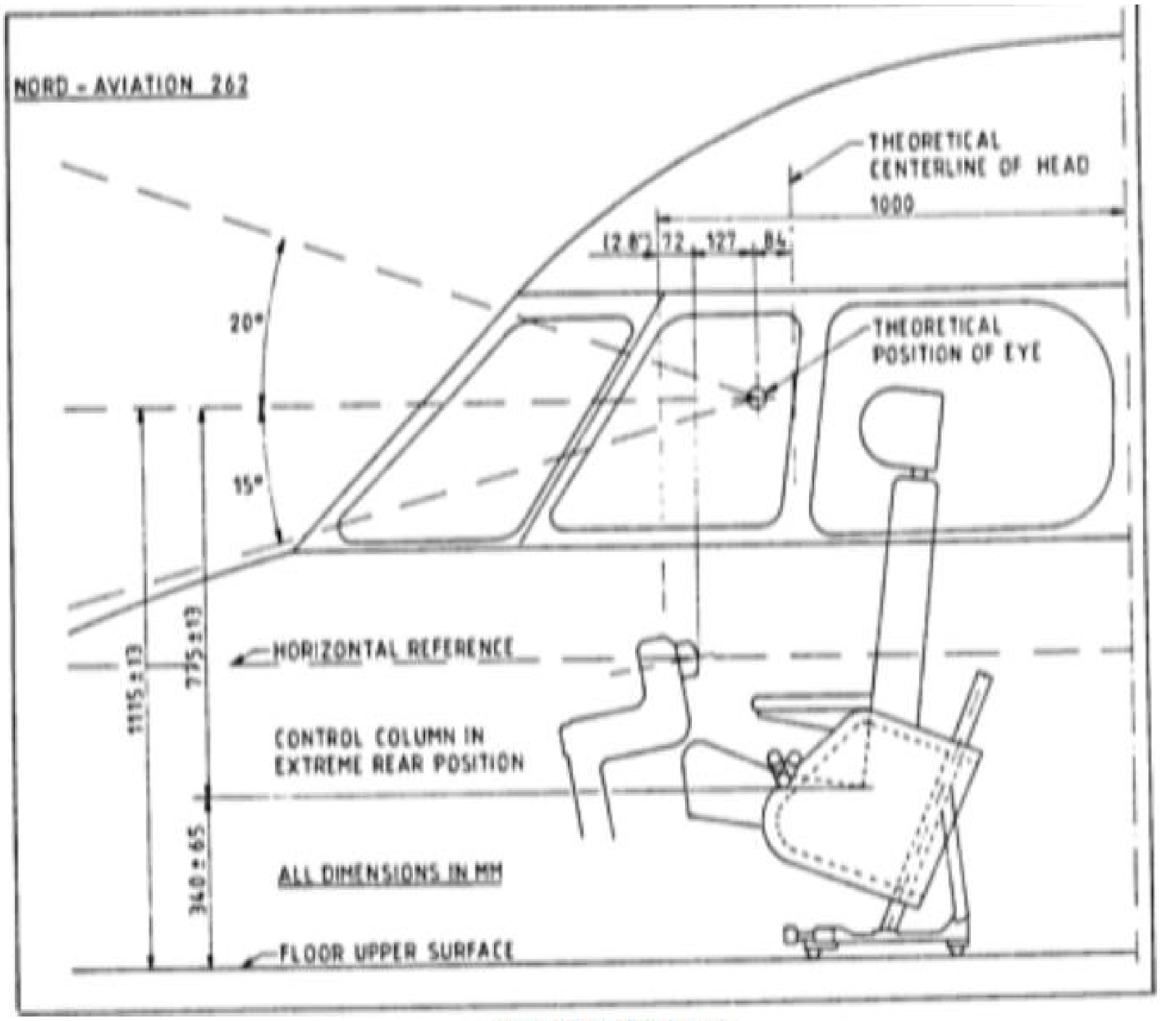


Figure 51: Cockpit layout

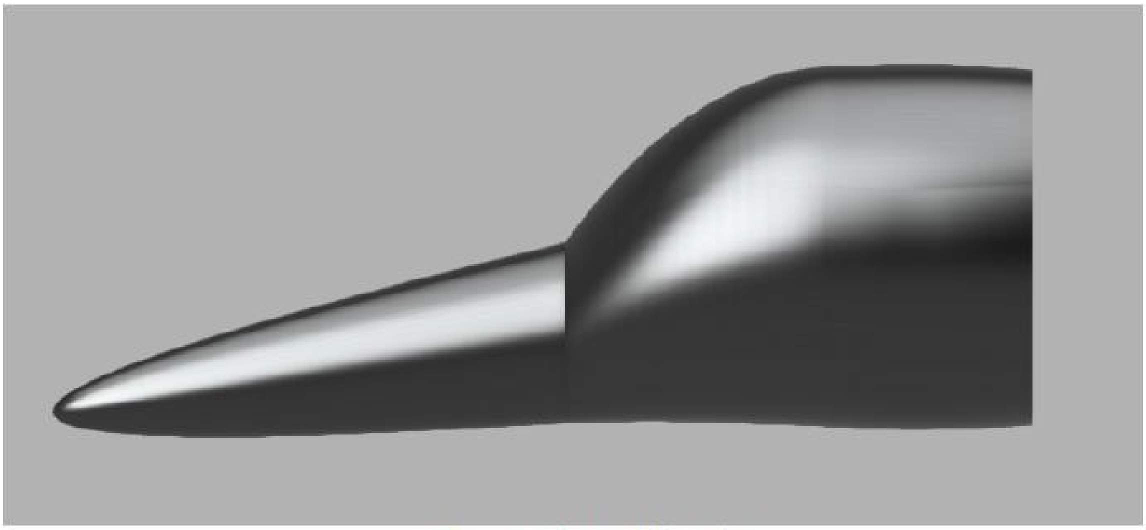


Figure 52: Right view of the cockpit

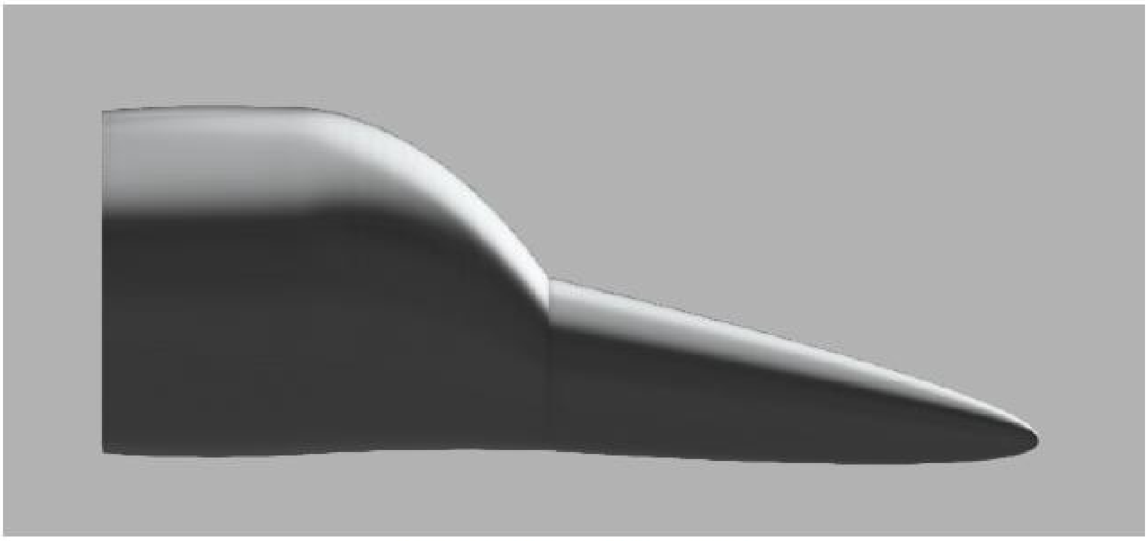


Figure 53: Left View of the Cockpit

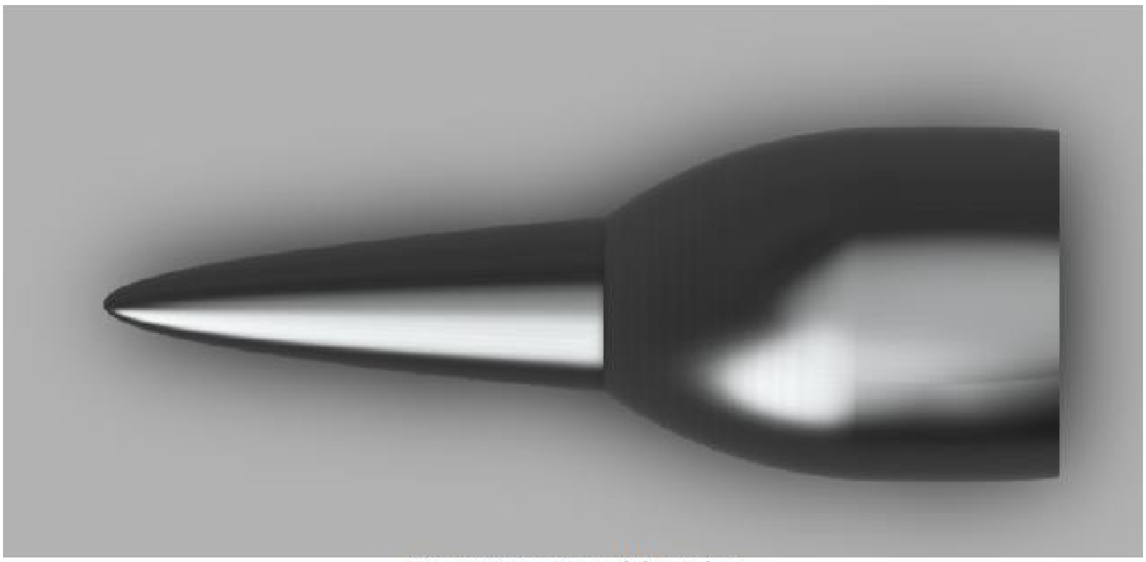


Figure 54: Top View of the cockpit

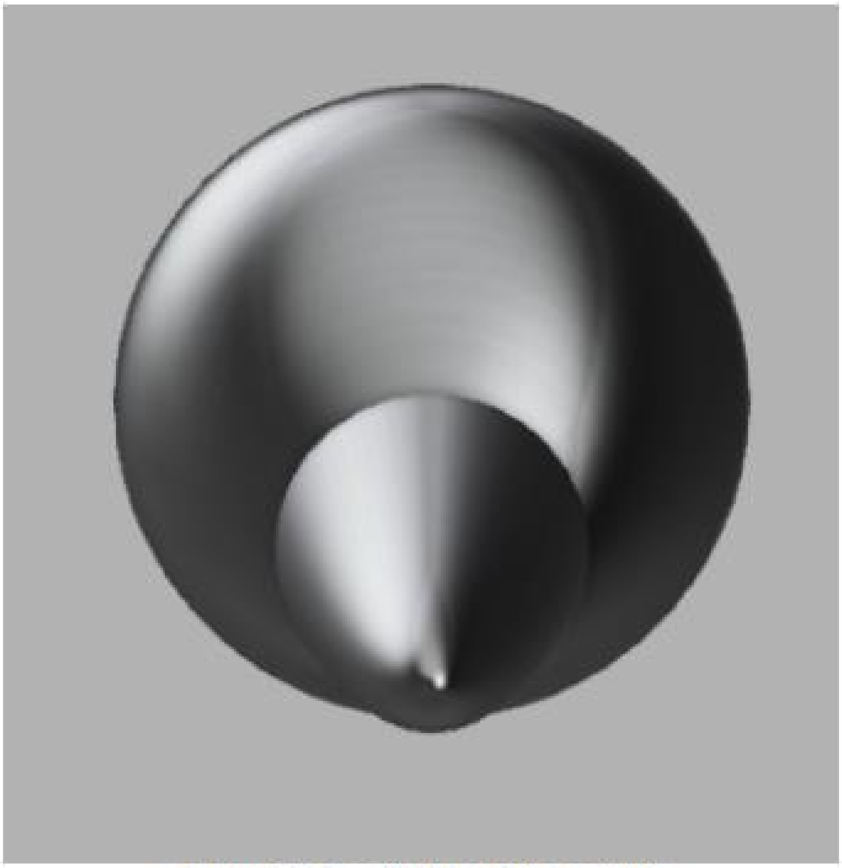


Figure 55: Front view of the cockpit

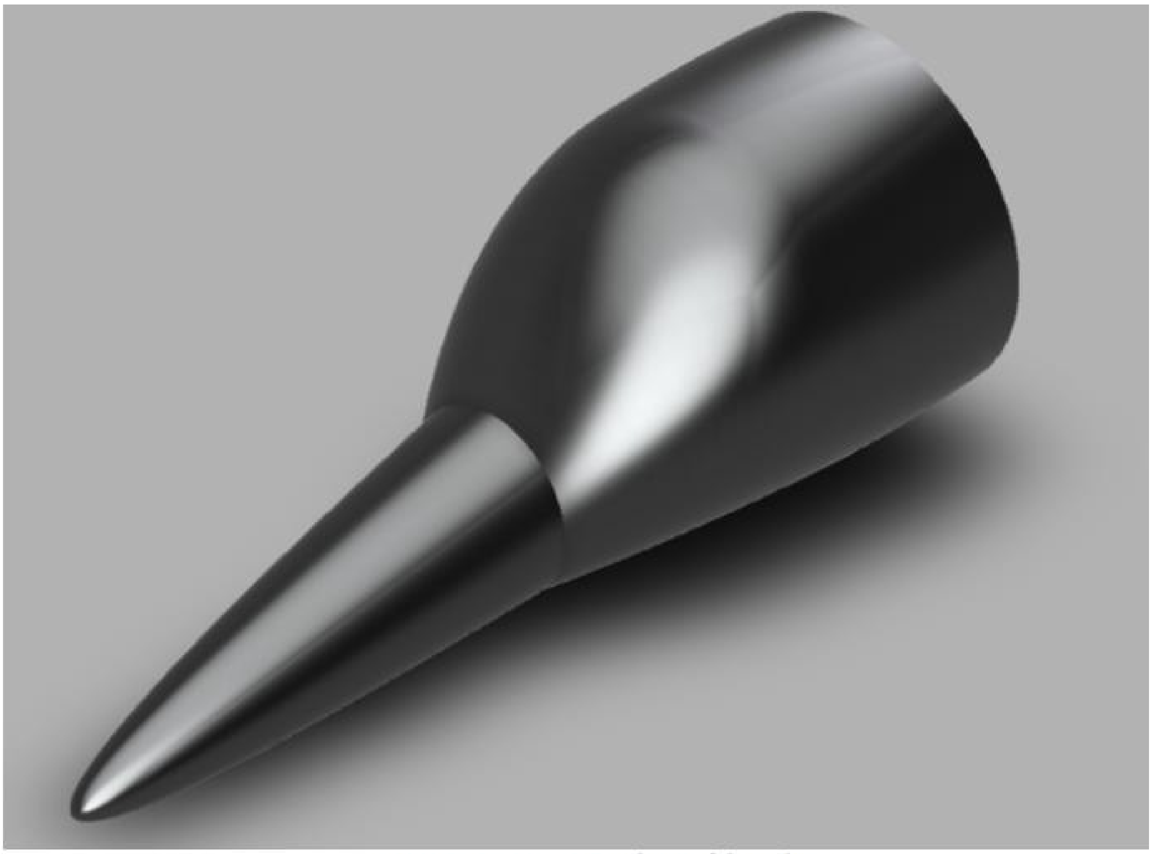


Figure 56: 3D cross-sectional view of the cockpit

5.3 LAYOUT DESIGN OF THE FUSELAGE

When the fuselage of a business jet is being designed, it is of utmost importance 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.

In the case of small commercial airplanes, like a 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. Long fuselage cones cause an increase in the tail moment arm thereby reducing the tail area and vice versa. In case of low fineness ratio, there shall be a large base drag penalty even if the fuselage weight might be reduced. However, if the fineness ratio of this cone is large, there shall be a large penalty in the friction drag of the airplane as well as a large weight penalty.

Figure 57 describes the typical layout of the exterior of the fuselage:

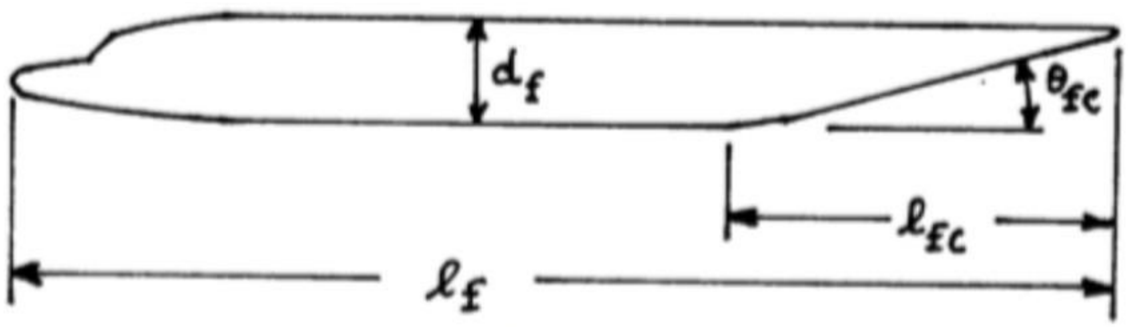


Figure 57: Dimensions of an aircraft

The following parameters can be used to design the fuselage, 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,

$$Length = a * W_0^C = 0.67 * (1{,}10{,}000)^{0.43} = 98.6\,ft \qquad 1$$

Figure 60: Right view of the fuselage and cockpit

Figure 61: Top view of the fuselage and cockpit

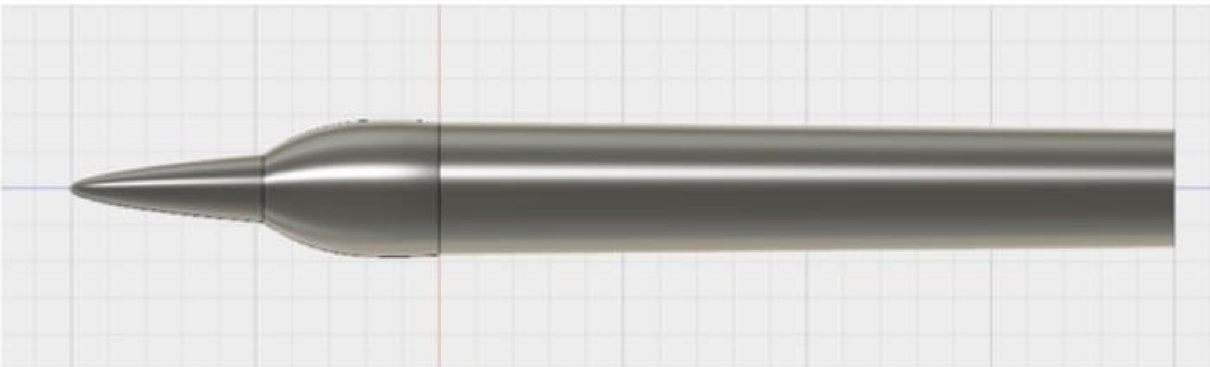


Figure 62: Bottom view of the fuselage and cockpit

Figure 63: Front view of the aircraft

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

AERODYNAMIC DRAG CONSIDERATIONS

A large percentage of the overall drag of an airplane is due to the fuselage. Most of the airplanes produce a drag in the range of 25 to 50 percent. As it is essential for an airplane to produce as minimum drag as possible, the fuselage should be sized and shaped 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. 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 friction drag can be reduced.

An important role in determining the fuselage friction drag is played by the fuselage fineness ratio. As the cruise speed increases the fineness ratio of an airplane gradually 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. If the windshields are integrated smoothly into the fuselage only then ideal 'streamline' nose shape can be obtained only.

The drag increases due to the unsweep in the aft-body of the fuselage large with the fuselage fineness ratio. The unsweep could result in vortex induced separations. These vortices can increase the drag and tend to amplify the issues caused by lateral oscillations. These problems can be stabilized using sharp corners. The sharp corners solve the problems caused by lateral oscillations apart from reducing the drag issues. The unsweep is applied to the airplane:

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

At times, a bulge is necessarily added to the upper rear section of the fuselage if large unsweep angles are detected by rear loading considerations. The bulge is required to acquire enough structural depth in the fuselage in order to resist tail loads.

Compressibility drags are seen at extremely high subsonic Mach numbers. These arise from the existence of shocks on the fuselage. The above-mentioned shocks are strong because of the sweep and thickness of the wing in the region of wing/fuselage juncture. To minimize compressibility, drag, the area rule concept should be used. Because of the adverse effect on the wings span load distribution, the fuselage contributes the most to induced drag.

Figure 65: Effect of fuselage on wing span loading

INTERIOR LAYOUT DESIGN OF THE FUSELAGE

The fuselage is responsible for carrying the passengers, the crew, the payload and several key systems required for the functioning of an airplane. The interior of the airplane design shows the compromise between the level of creature comforts and the sizes and weights required to sustain these creature comforts.

An important role is played by the ability to load and unload cargo plays. The issues associated with maintenance and servicing dictate where access must be designed into the fuselage. The design for good access, inspect ability and maintenance usually is in direct contrast with design for low complexity weight, low structural 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 creature comfort considerations, drag, and systems and weight.

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)

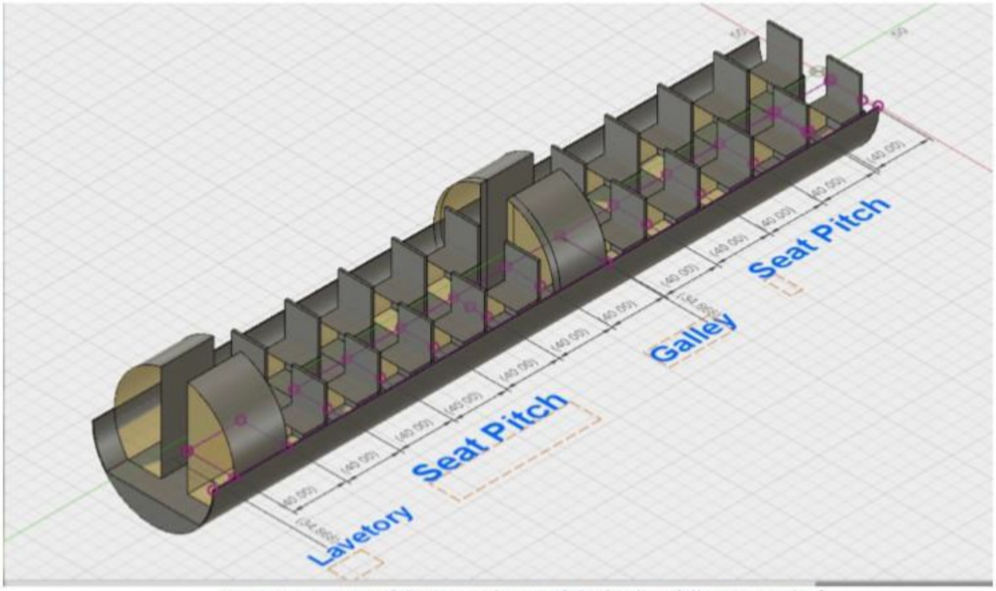


Figure 70: 3D view of the interior layput of the fuselage (all units in inches)

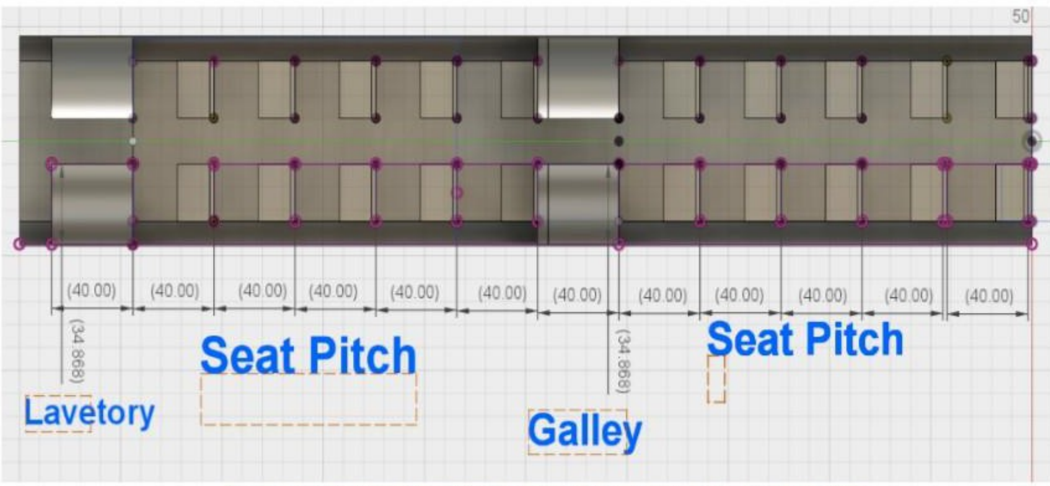


Figure 71: Top view of the interior layout of the fuselage (all units in inches)

5.4 DISCUSSION

The 5th report of the transonic business jet describes in immense detail the fuselage and cockpit design. The formula for calculating the length of the fuselage was derived from the book airplane design: A Conceptual design, which was written by Daniel P Raymer. The first section is an introduction part which explains the parameters needed to acquire the design of the fuselage and the cockpit.

The next section describes the cockpit design section that details the design requirement of the cockpit. It lays out the dimensions of both, the male and the female crew members' seat arrangement requirements, also the visibility requirements in accordance with the FAR 25 requirements. Wheel/stick control dimensions and requirements and the design of the cockpit. Different views of the cockpit are attached in the above section, the front view, left view, top view, right view, and a 3D view of the cockpit. The 3D geometry of the cockpit was designed on Fusion 360.

The final section explains the design of the fuselage. The fuselage was designed in accordance with the requirements within the acquired dimensions of the length of the airplane as seen in the section above. Different views of the fuselage are attached in the fuselage section, the cross- sectional view which explains the dimensions of the inner area of the fuselage, the left view, the top view, the right view, the bottom view, the back view and the front view. The overall design until now is also attached in the fuselage section that comprises of all the progress in the design section until now.

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

6.1 INTRODUCTION

The sixth report specifies the wing design of the airplane. In the previous reports, the weight sizing, performance parameters, fuselage and cockpit were designed and calculated. The report defines the design and dimensions of the airfoil which must 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:

1. Size (area)
2. Aspect ratio
3. Sweep angle
4. Thickness ratio
5. Airfoils
6. Taper ratio
7. Incidence angle and twist angle
8. Dihedral angle
9. Lateral control surface size and layout

A few parameters were discussed in the sections preceding this, including the aspect ratio of the wing and the wing area. The design of the wing for the transonic business jet includes a low-wing configuration.

First the taper ratio, sweep angle, dihedral angle, and the thickness ratio shall be calculated assuming the minimum requirements in accordance with the design of the airplane. All the results shall be plotted on a graph and shall be justified. In the next section of the report, the airfoil shall be selected and discussed there after the C_{LMAX}. of the wing is verified using the AAA program.

In the 5th section of this report, high lift devices shall be designed and determined in accordance with the acquired dimensions. The 6th section of this report shall specify the lateral control surfaces and their sizes considering that these are compatible with the high-lift devices designed.

The final section of this report details the calculations and the designs of the parameters of utmost importance 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})

6.2 WING PLANFORM DESIGN

The wing planform of the transonic business jet is a cranked arrow wing. The wing size directly affects the following characteristics of the airplane:

a. Take-off/ Landing field length
b. Cruise performance (L/D)
c. Ride through turbulence
d. 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 Transonic Business Jet is 1424.34 ft^2

$$S = 1424.34\ ft^2 \qquad 1$$

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 transonic business jet is obtained to be 2.1 from the previous report

$$A = 2.1 \qquad 2$$

Taper Ratio (l): 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 TSBJ is transonic, 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 equal to 0.15

$\lambda = 0.15$ **3**

$\lambda = \frac{c_t}{c_r}$

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 Transonic 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 (G**):** 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 transonic business jet has a lot of sweep, a dihedral, won't be added to the TSBJ design. The dihedral will be equal to 0.

$\Gamma = 0$ 4

$\Gamma = 0$ 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

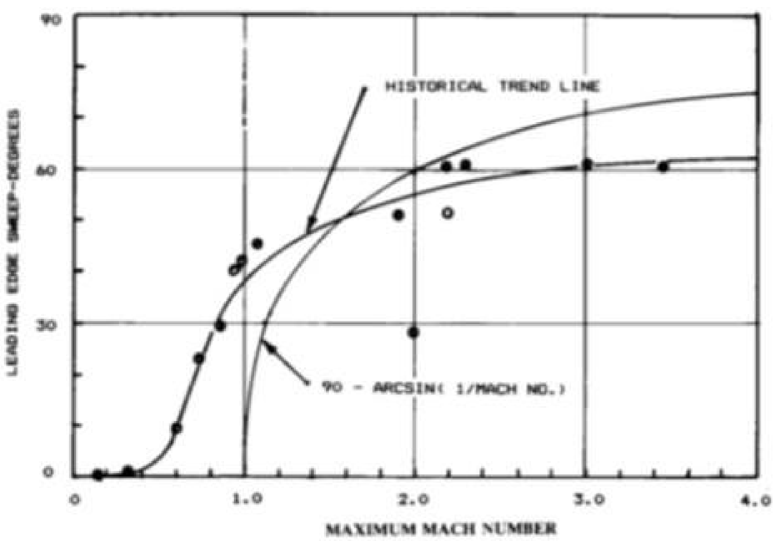


Figure 76: Mach number v/s leading edge sweep

The sweep angle of a transonic airplane can be acquired by the formula as seen above in the figure, obtained via the book by Raymer.

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

For Mach 1.6,

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

$$\Lambda = 51.32^{\circ} \qquad 5$$

The sweep angle of the transonic business jet obtained is equal to 51.32°.

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

$$tan\,\Lambda_{\mathrm{LE}} = \tan\Lambda_{\frac{c}{4}} + \left[\frac{1-\lambda}{A(1+\lambda)}\right]$$

$$\Lambda_{\frac{c}{4}} = 41.59^{\circ} \qquad 6$$

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

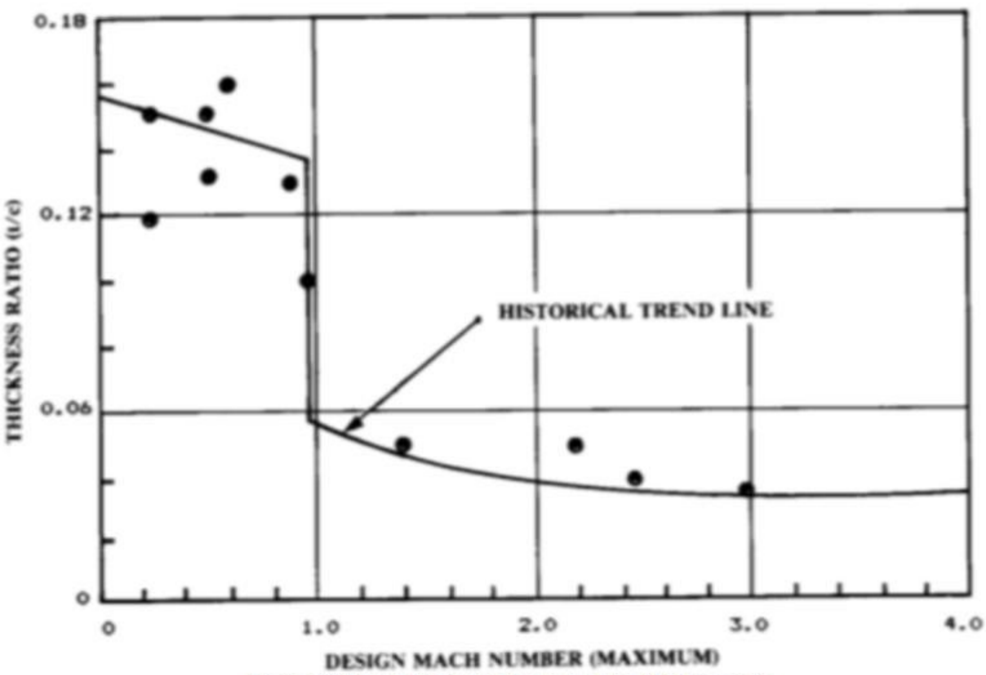


Figure 77: Design Mach number verses the thickness ratio

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,

$$R_{n_r} = \frac{\rho V c_r}{\mu} = \frac{4.62*10^{-4}*1066.78*45.29}{2.969*10^{-7}} = 75.18 * 10^6 \quad 11$$

$$R_{n_t} = \frac{\rho V c_t}{\mu} = \frac{4.62*10^{-4}*1066.78*6.79}{2.969*10^{-7}} = 11.0 * 10^6 \quad 12$$

$$C_{L_{max_w}} = 0.95 \frac{\left(C_{L_{max_r}} + C_{L_{max_t}}\right)}{2} = 0.95 \frac{2.52+2.32}{2} = 2.3 \quad 13$$

$$C_{L_{max_w}} = 2.3 \sin(51.32) = 1.8 \quad 14$$

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.

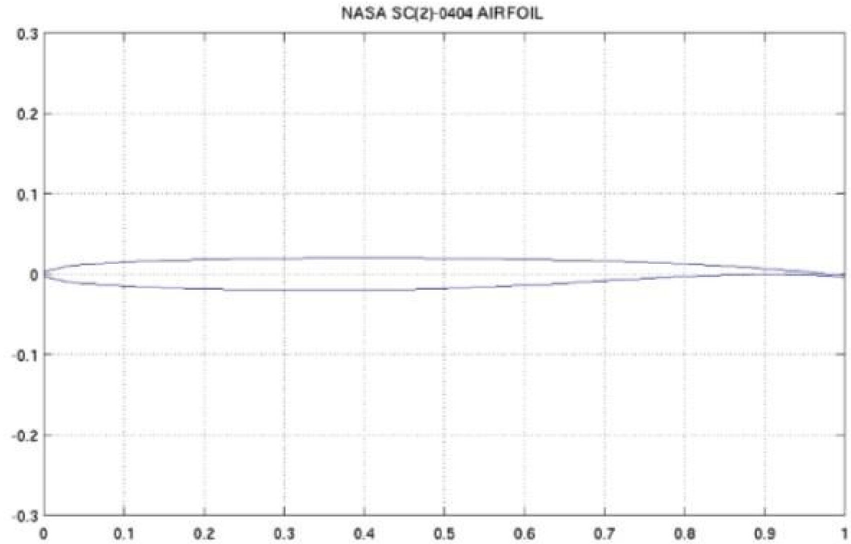


Figure 78: NASA SC(2)-0404

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 transonic business jet hence no incidence will be applied to the wing of the transonic 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.

$$\varepsilon_t = 0 \tag{16}$$

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.

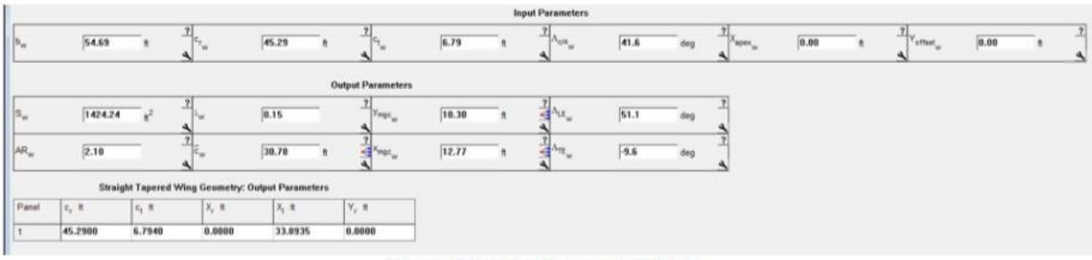


Figure 80: Wing Parameters AAA

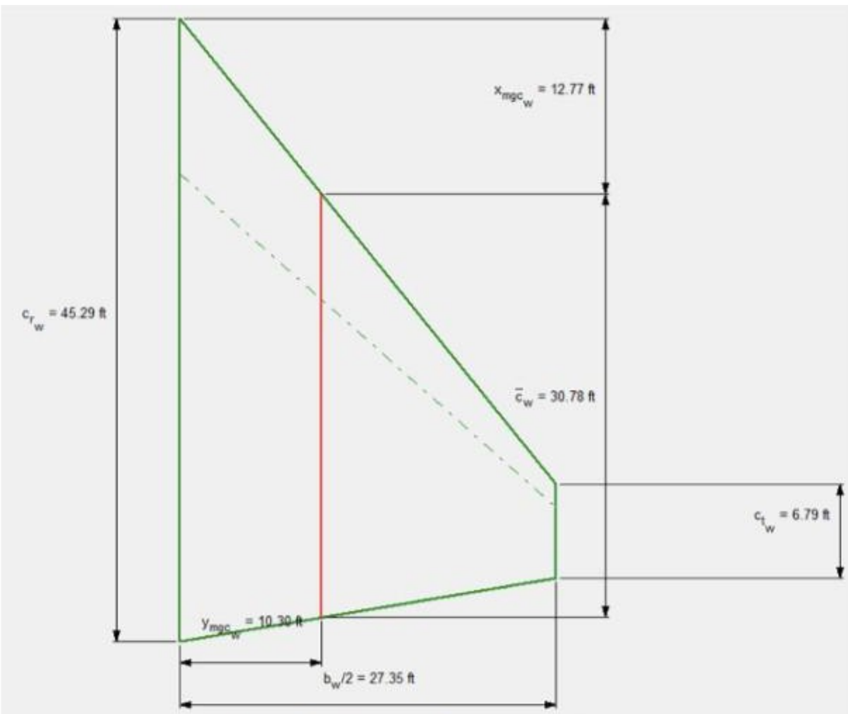


Figure 81: Wing Design Obtained from AAA

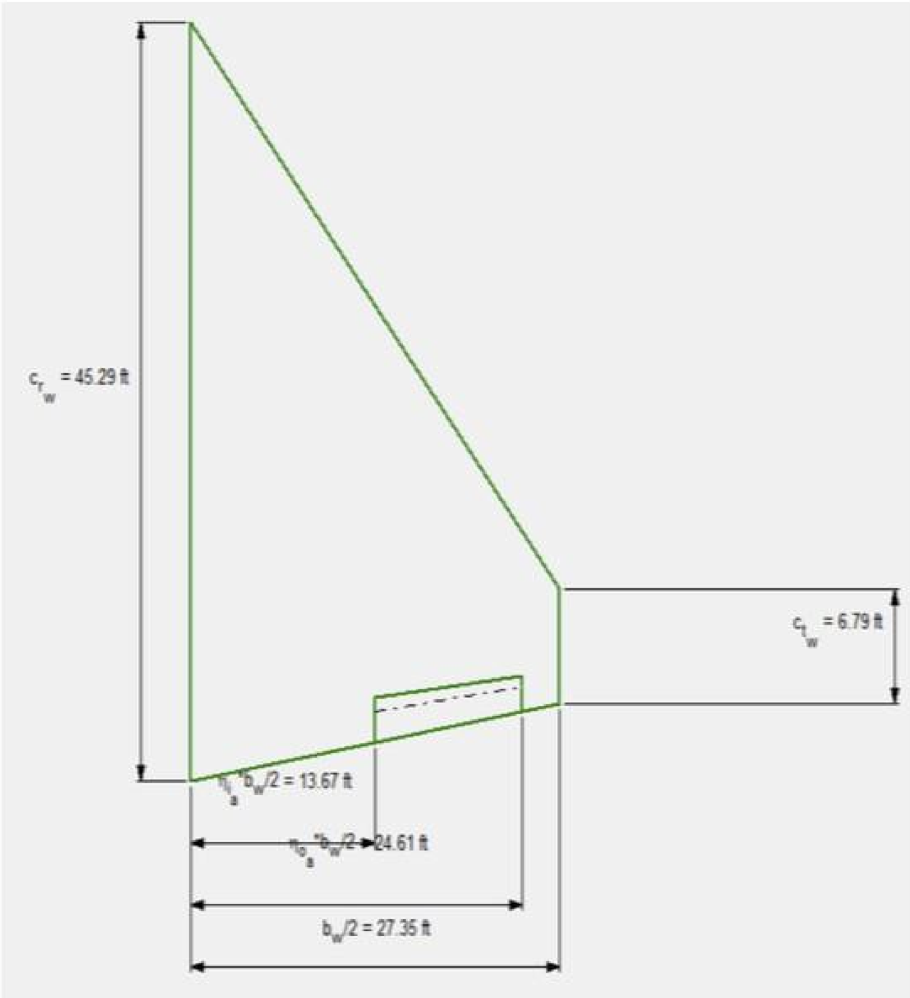


Figure 83: Aileron sizing obtained from AAA


Figure 84: Flap Geometry Sizing

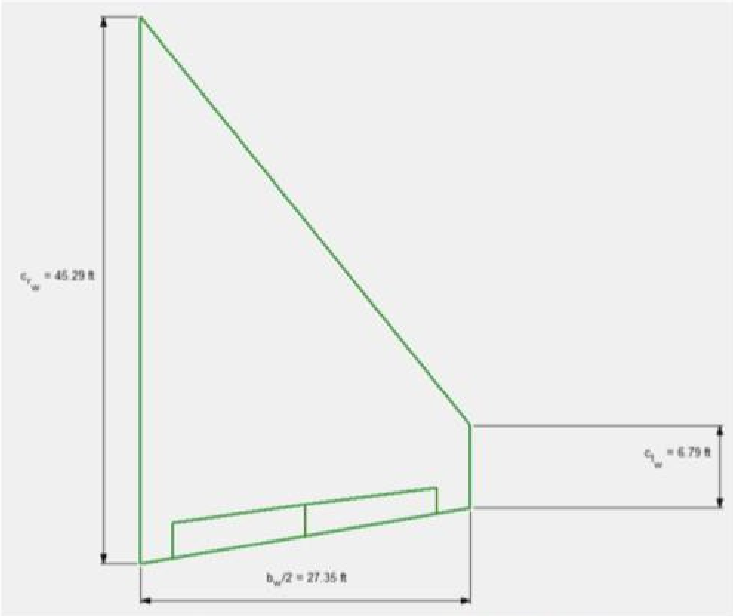


Figure 85: Aileron and Flap Locations Obtained from AAA

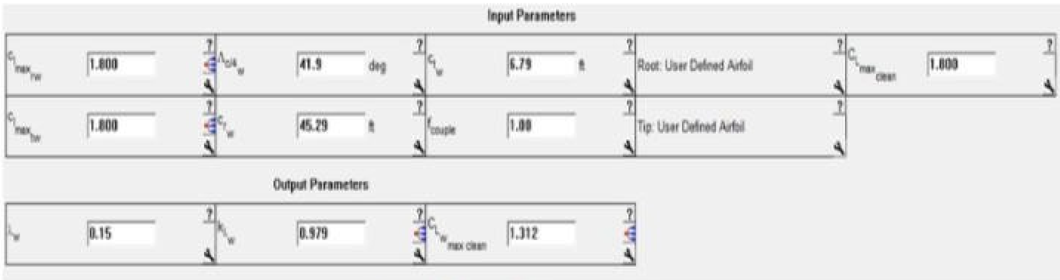


Figure 86: Cl_max requirements

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:

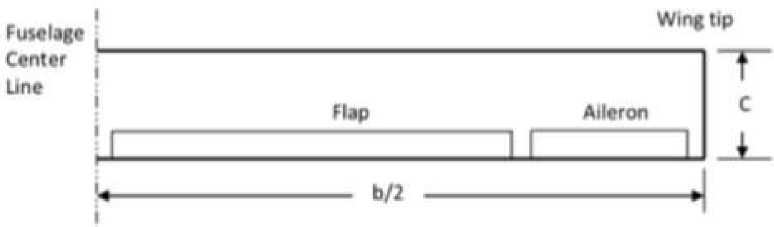


Figure 87: location of high-lift devices on the wing.

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,

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

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

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

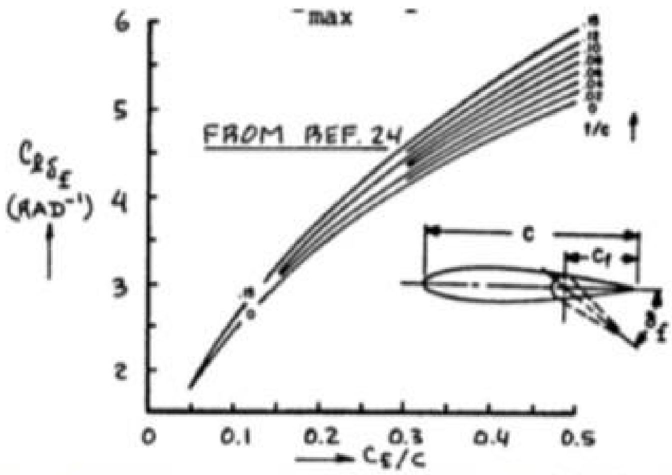


Figure 88: Effect of thickness ratio and chord ratio on 〖〖C_l〗_δ〗_f.

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

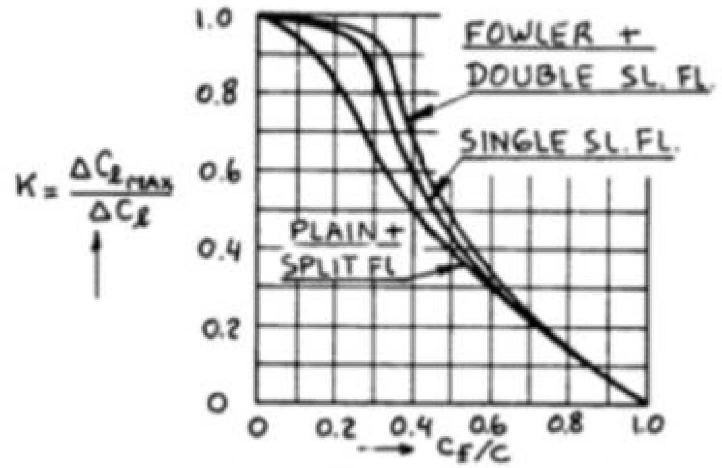


Figure 89: Effect of Flap chord ratio and Flap type on K.

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

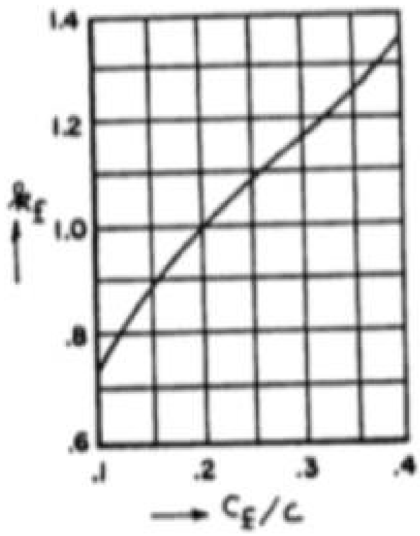


Figure 90: Relation between kf and cf/c

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

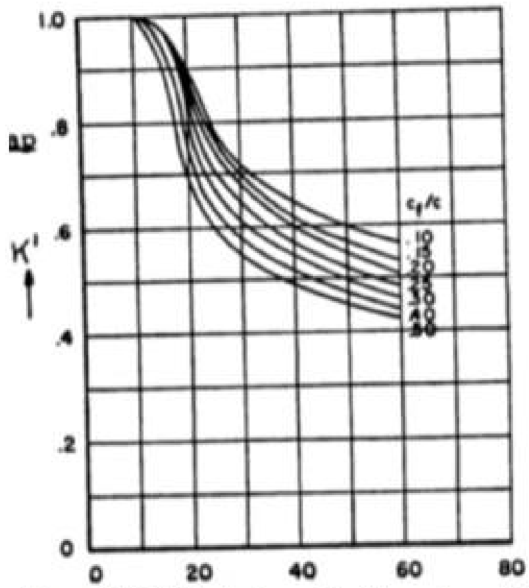


Figure 91: Relation between the flap angle and K'

While landing: $\Delta C_l = C_{l_{\delta_f}} * \delta_f * K' = 4.5 * rad(40) * 0.55 = 1.728$ 26

While Take-off: $\Delta C_l = C_{l_{\delta_f}} * \delta_f * K' = 4.5 * rad(20) * 0.75 = 1.178$ 27

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

$$\Delta C_{l_{max}} = \Delta C_{L_{max}}\left(\frac{S}{S_{wf}}\right)K_\Lambda = 0.765 * 2.0 * 0.77 = 1.178 \quad 29$$

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.

$$\Delta C_l = \left(\frac{1}{K}\right)\Delta C'_{l_{max}} = \left(\frac{1}{0.85}\right) * 1.8 = 2.11 \quad 30$$

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

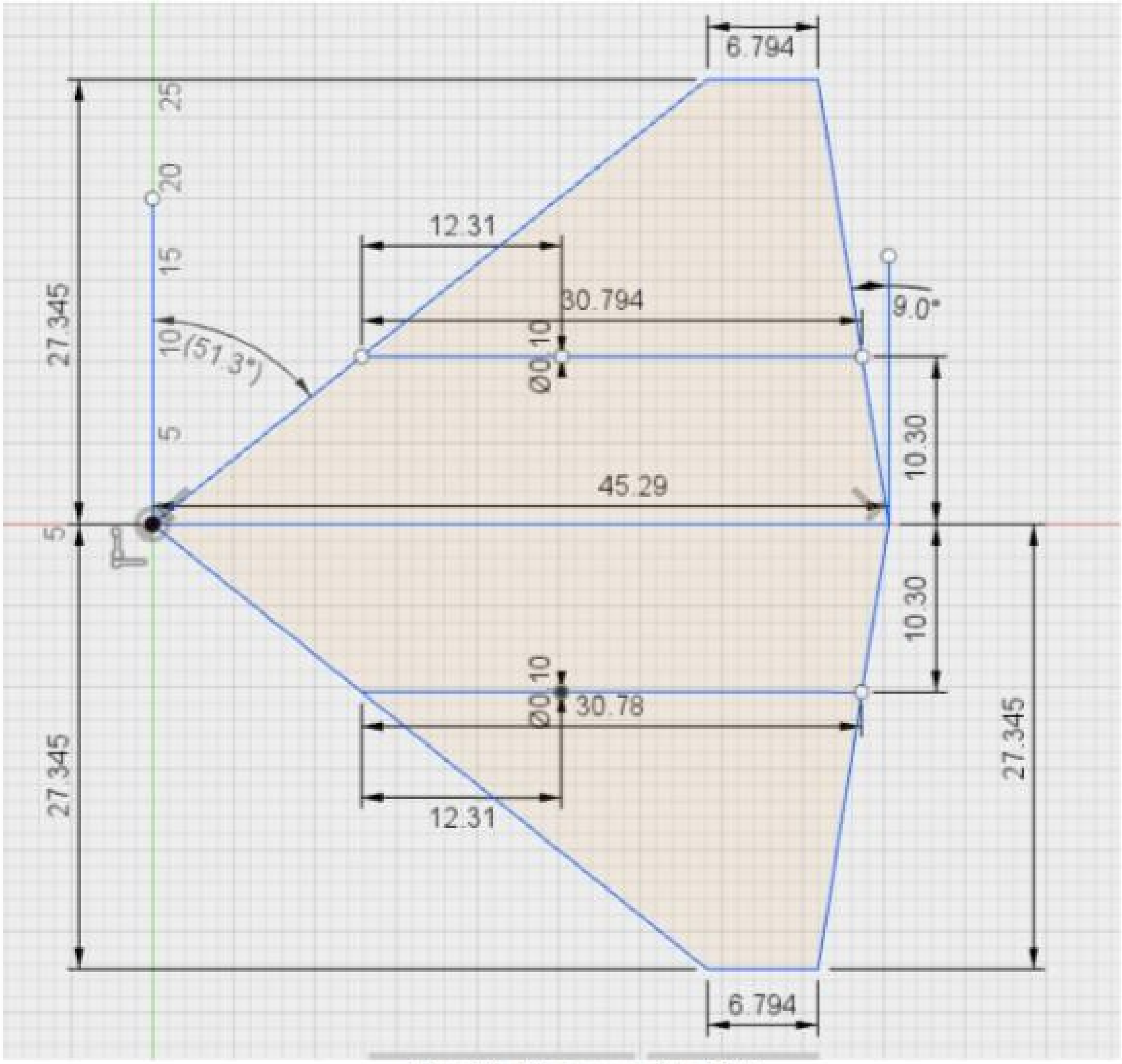


Figure 93: Wing Layout of the SSBJ

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:

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

ii) **Root chord, c_r**

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

iii) Tip chord, c_t

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

$$c_t = \lambda * c_r = 0.15 * 45.29 = 6.794\,ft \qquad 35$$

iv) Mac (mean aerodynamic chord)

$$Mac(c) = \frac{2}{3} c_r \frac{1+\lambda+\lambda^2}{1+\lambda} = 30.78\,ft \qquad 36$$

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

$$Y = \frac{b}{6}[(1+2\lambda)(1+\lambda)] = 13.62\,ft \qquad 37$$

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

v) Mgc (mean geometric chord)

$$Mgc = \frac{S}{b} = \frac{1424.34}{54.69} = 26.04\,ft \qquad 38$$

vi) Leading-edge sweep angle

$$\Lambda_{LE} = 90 - \arcsin\left(\frac{1}{M}\right) = 51.32^{\circ} \qquad 39$$

vii) Trailing edge sweep angle

$$\Lambda_{TE} = -9^{\circ} \qquad 40$$

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

6.8 DISCUSSION

The 6th report of the entire series, describes the design of the wing. At the beginning of the report, in the first section, the parameters which were needed for designing the wing area, were derived from the 4th report such as the aspect ratio and the wing area. Later, some parameters, such as the taper ratio, incidence angle, dihedral angle, thickness ratio, etc. were calculated by making use of the provided formulas and certain parameters were taken from similar aircrafts data.

Upon obtaining data for the wing, an airfoil was selected, one that satisfied all the necessary conditions, like the coefficient of lift. The airfoil chosen here was the NASA SC (2)-0404. Post choosing the airfoil, the AAA program was utilized to determine the wing parameters and compare these parameters with the obtained results to verify the accuracy of these results. Post comparison of the results, the high-lift devices were designed in such a manner that they match the requirements to facilitate the required lift. The high lift devices contain the flaps and ailerons. The high lift devices acquired are displayed in the AAA section figures alongside the drawings and the parameters.

Once all the data is obtained, the wing was designed as shown in the above sections, making use of the tip chord, root chord, wing area, sweep angle, quarter chord angle, span, taper ratio, and lastly, the aspect ratio. The aerodynamic center was situated in the design of the wing along with the CG of the wing.

6.9 CONCLUSIONS

For the transonic business jet, a cranked arrow wing is a necessary configuration instead of a delta wing. Thus, for future design, a cranked arrow wing shall be designed because of which the weight of the wing can be reduced to a certain extent and the addition of the control surfaces on the wing surface, thereby neglecting the use of horizontal on the empennage section ultimately reducing cost, weight, maintainability etc.

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

7.1 INTRODUCTION

This is the 7th report of the entire series of the preliminary design of the Transonic Business Jet. This report defines the design of the longitudinal & directional stability controls and the empennage. The weight sizing, fuselage, performance parameters, and cockpit, along with the wing design were designed and calculated 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. Airfoils
III. Thickness ratios
IV. Sweep angles
V. Control surface sizes and their layouts
VI. Taper ratios
VII. Incidence angles
VIII. Aspect ratio
IX. Wing span
X. Dihedral angles

The various parameters, including the wing aspect ratio, wing area, wing mean aerodynamic chord length, wing sweep angles, thickness ratios of the wing airfoil, etc. were discussed in the previous report, number 6, that specifies the wing design.The empennage area of the transonic airplane comprises of a T-tail configuration, being the suggested configuration for business jets to obtain enhanced directional and longitudinal stability.

The empennage area of the transonic business jet shall be worked on as mentioned in report number 2, later the calculations of the vertical and horizontal stabilizers shall be based upon certain considerations made from aircrafts of similar type and the acquired data from the preceding reports and later, the CAD drawings shall be designed based on the calculations acquired. Upon designing the horizontal and vertical stabilizers, the directional and longitudinal control surfaces shall be worked upon to define the size and the layout in accordance with the requirements of the transonic business jet.

Another process of similar kind shall be worked on the AAA program. The data and calculations obtained shall then be compared with the data obtained from the manual calculations and eventually, the final plan forms will be created according to the data obtained.

7.2 OVERALL EMPENNAGE DESIGN

As described in the 2nd report, the empennage area shall have a T-tail configuration as the T-tail is the best preferred configuration for business jets.

As the T-tail configuration gets a direct clean flow over the horizontal stabilizer rather than the turbulent flow created after passing the wing section of the aircraft, it is selected. Because of the direct clean flow obtained, a great amount of induced drag is created on the surface of the horizontal stabilizer because of the turbulent flow from the wing being decreased in the case of a T-tail configuration.

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

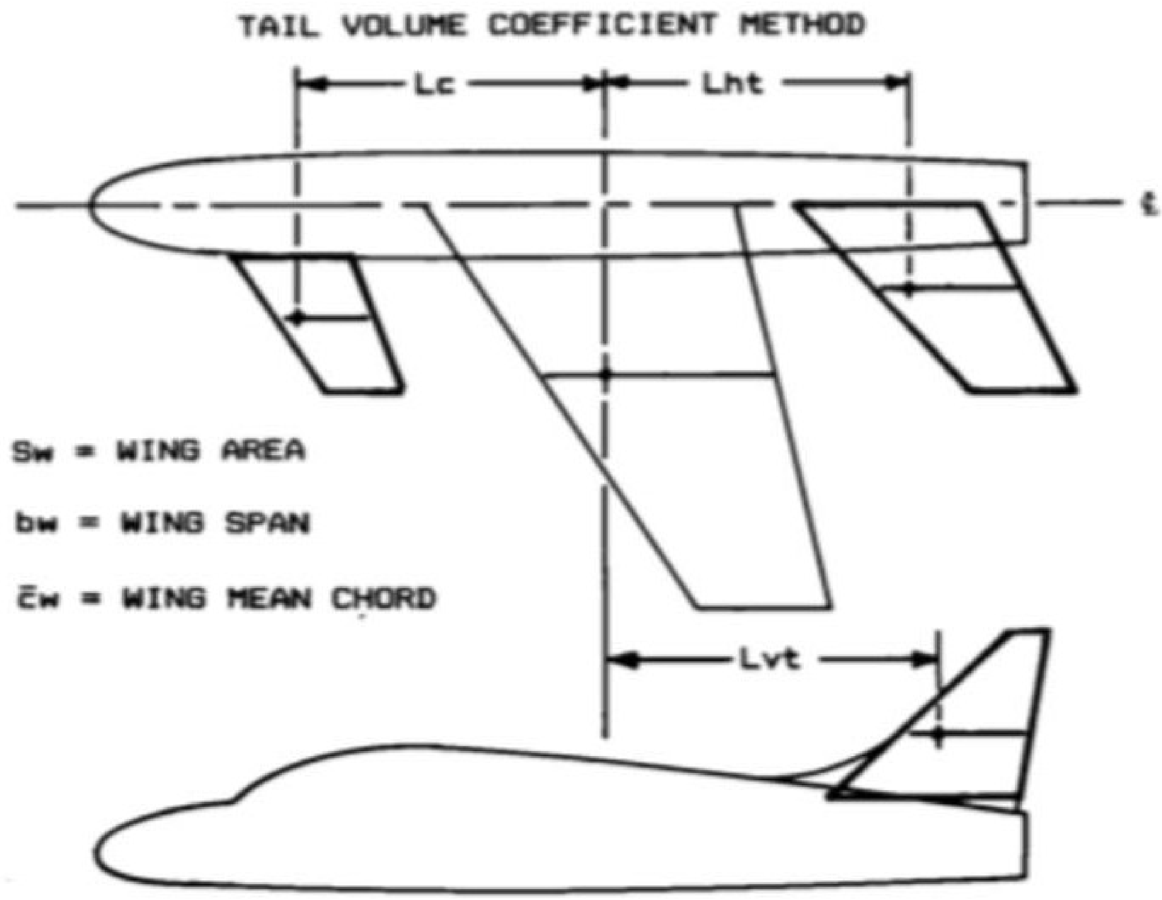


Figure 94: Tail volume coefficient method

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

The aircrafts which have aft mounted engines, the vertical moment arm is between 45% - 50% the fuselage length. Therefore, 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 Roskam.

L_{HT}= 40.78 ft. 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 transonic 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.

$$S_{VT} = \frac{c_{VT} * b_W * S_W}{L_{VT}} = \frac{0.0855 * 54.69 * 1424.34}{25} = 266.4 ft^2 \qquad 4$$

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,

$$\bar{c}_V = \frac{2}{3} * c_{r_V}\left(\frac{1+\lambda_V+\lambda_V^2}{1+\lambda_V}\right) = 14.979\ ft \quad 8$$

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:

$$\bar{Y}_V = 2 * \left(\frac{b}{6}\right)\left(\frac{1+2\lambda_V}{1+\lambda}\right) = 8.783\ ft \quad 9$$

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:

$$c_{HT} = \frac{L_{HT} * S_{HT}}{C_W * S_W} \quad 10$$

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

$$S_{HT} = \frac{c_{HT} * C_W * S_W}{L_{HT}} = \frac{0.359 * 30.78 * 1424.34}{30.78} = 385.94ft \quad 11$$

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

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

$$\frac{b_{VT}}{2} = 17.0\ ft \quad 13$$

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:

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

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

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

$$\bar{c}_H = \frac{2}{3} * c_{r_H} * \left(\frac{1+\lambda_H+\lambda_H^2}{1+\lambda_H}\right) = 11.5778\ ft \quad 16$$

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

$$\bar{Y}_H = \left(\frac{b}{6}\right)\left(\frac{1+2\lambda_H}{1+\lambda_H}\right) = 7.7978\ ft \quad 17$$

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.

$\Lambda_{LE} = 56.32°$ 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.

$\left(\frac{t}{c}\right)_{HT} = 2\%$ 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.

$$\Lambda_{\frac{c}{4}VT} = 40° \qquad 24$$

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

$$tan\Lambda_{VT} = tan\Lambda_{\frac{c}{4}VT} + \left[\frac{1-\lambda_{VT}}{A(1+\lambda_{VT})}\right] = 0.8829 \qquad 25$$

$$\Lambda_{VT} = 41.44° \qquad 26$$

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%.

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

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

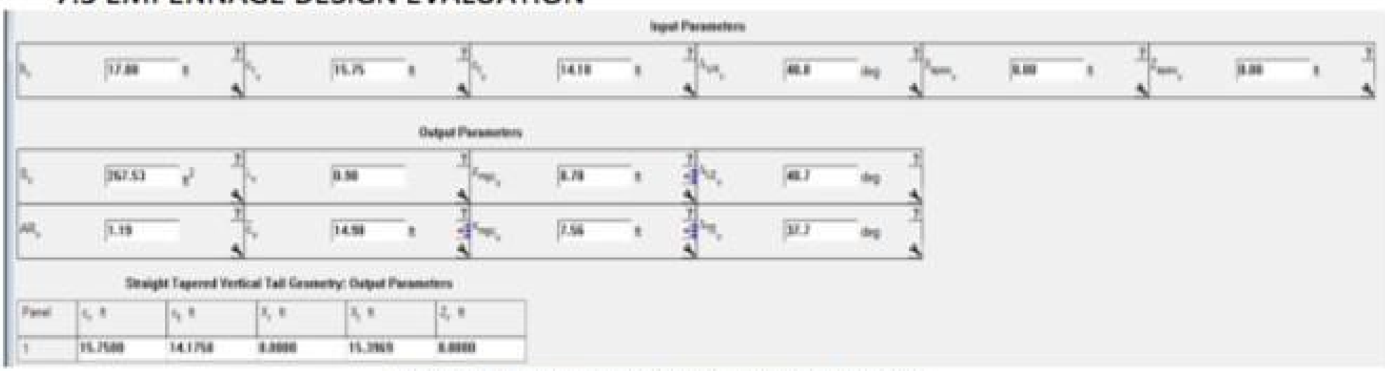


Figure 99: Parameters to Design a Vertical Stabilizer

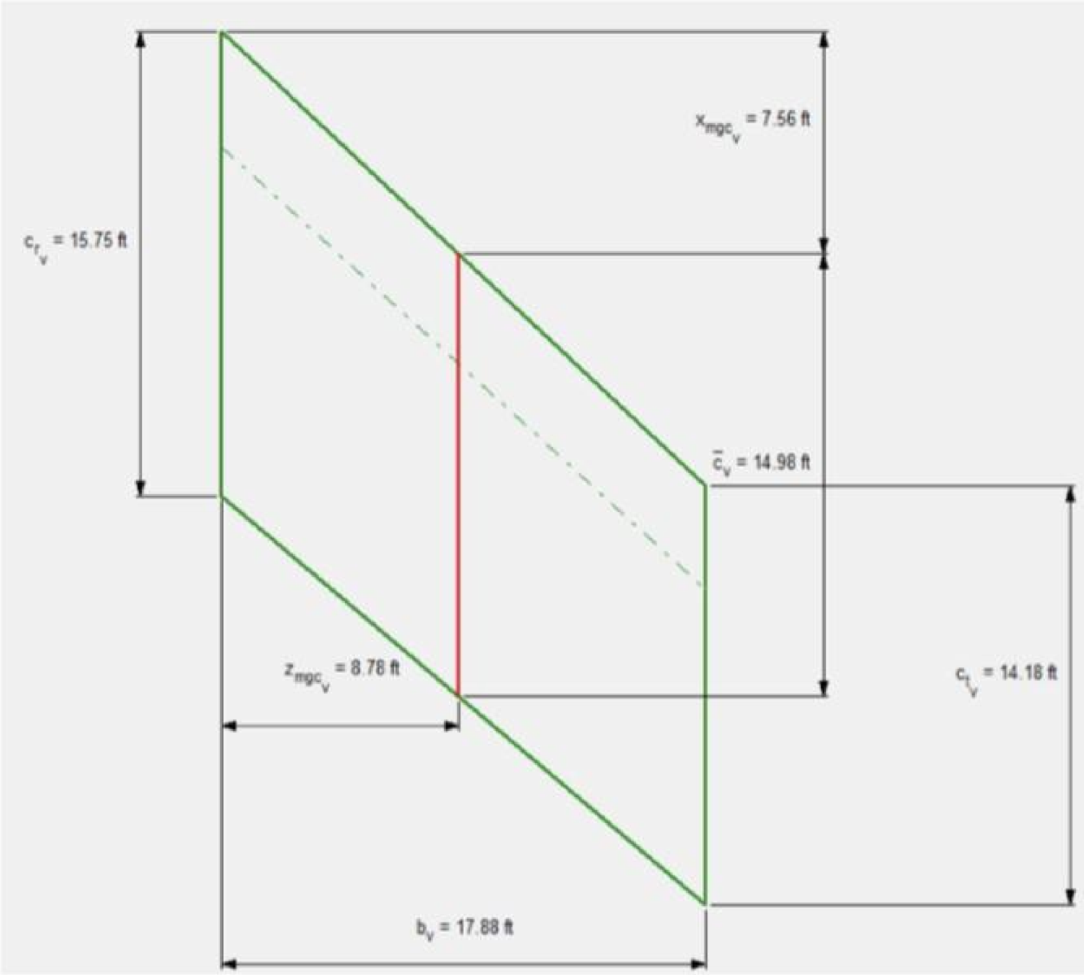


Figure 100: Vertical Tail Obtained From AAA

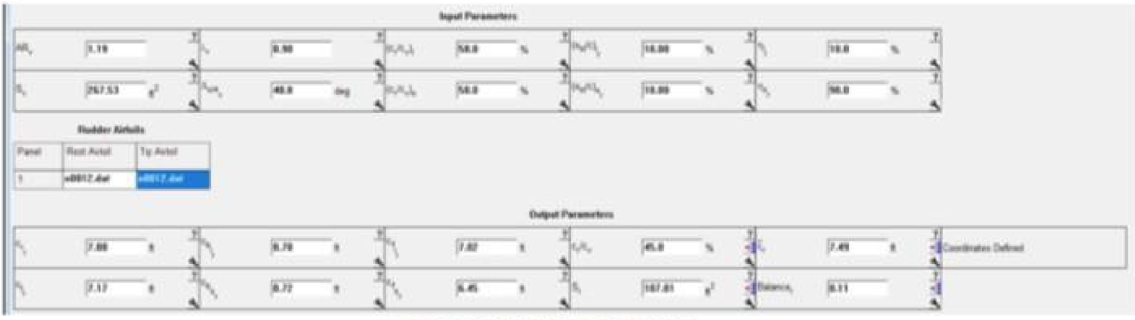


Figure 101: Rudder Sizing Parameters

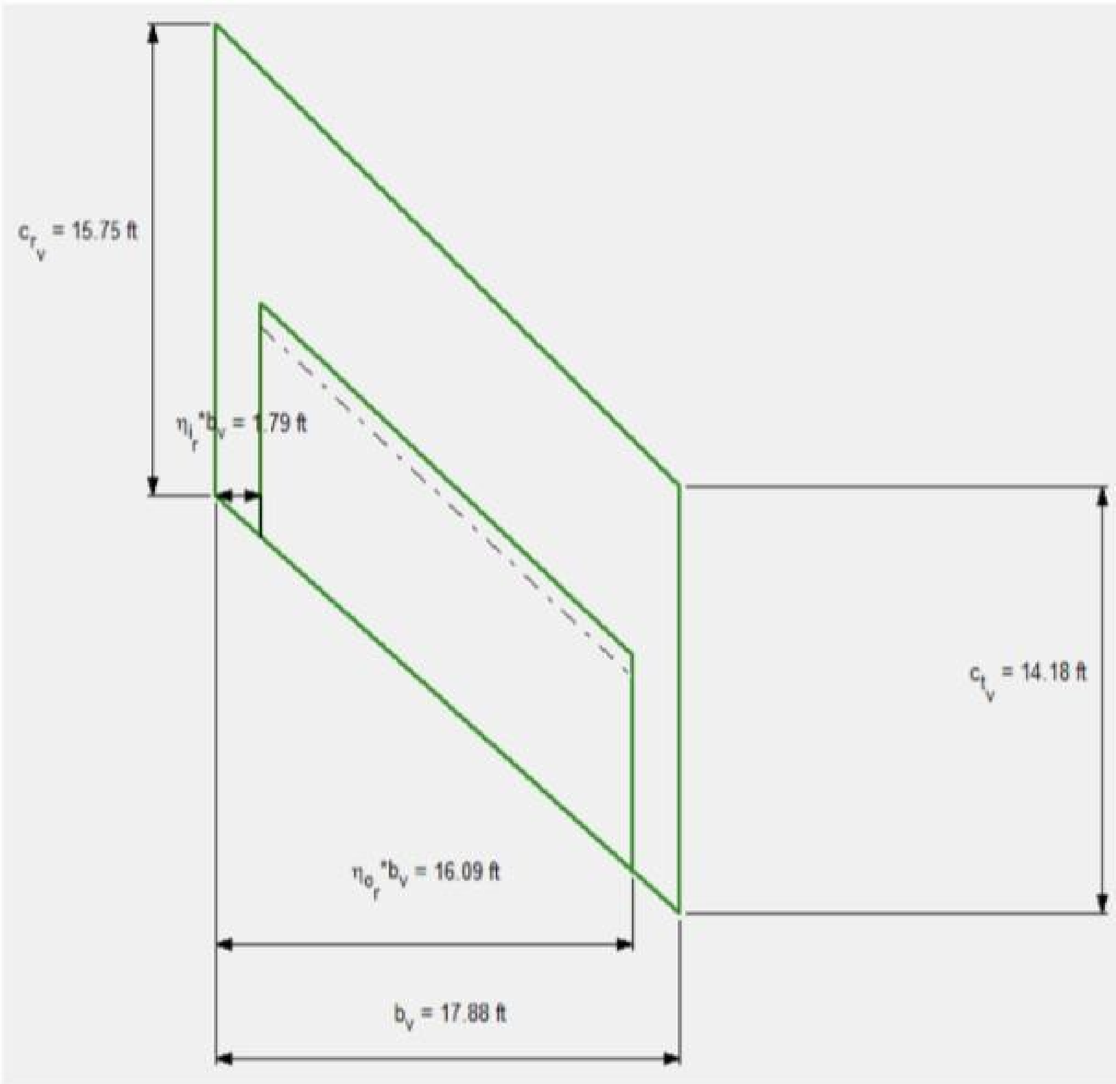


Figure 102: Rudder Location

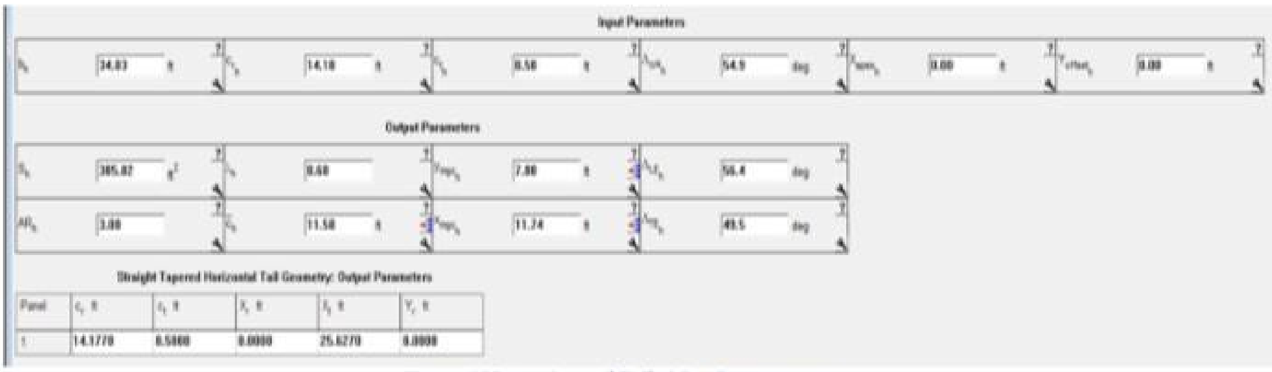


Figure 103: Horizontal Tail sizing Parameters

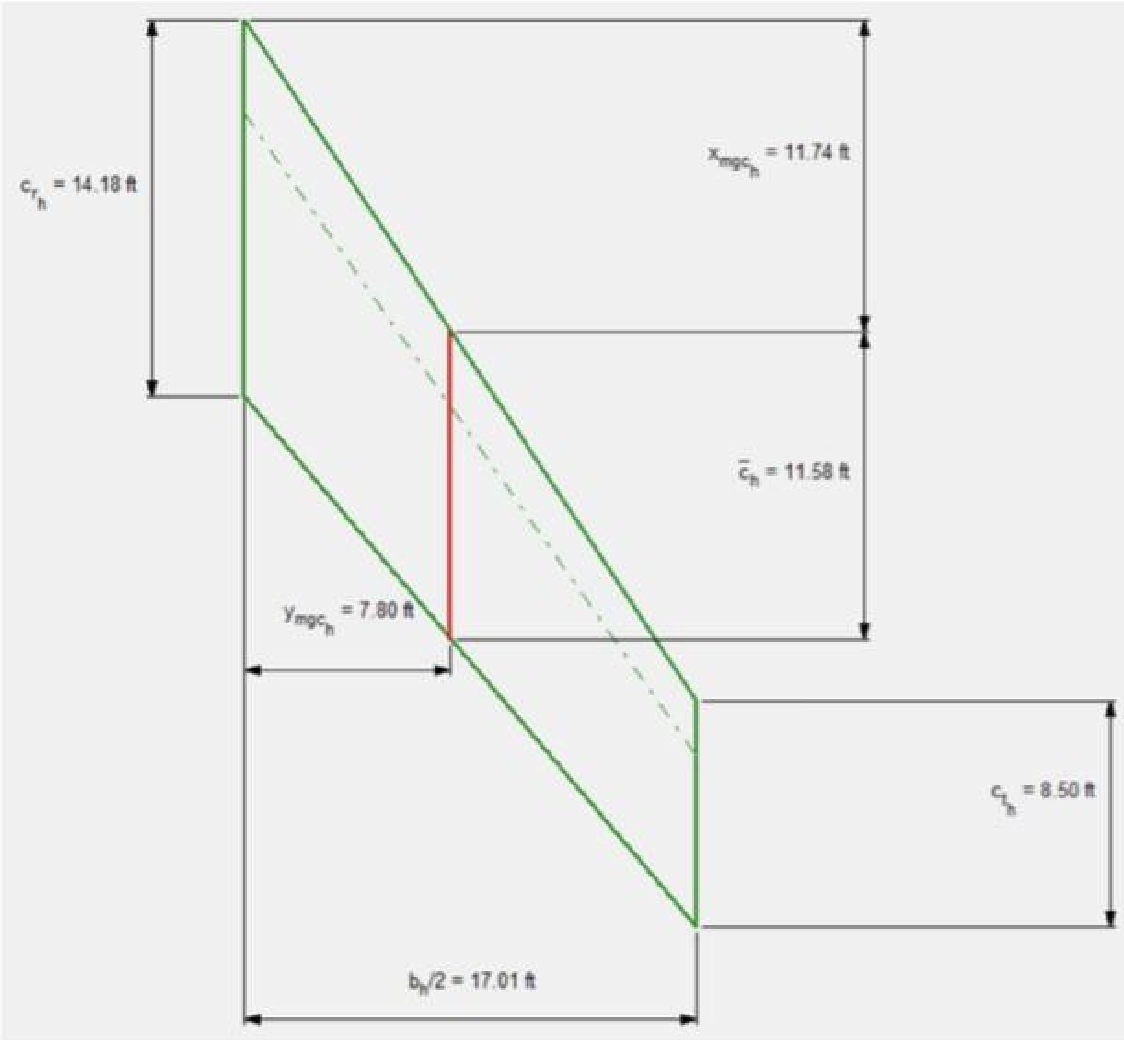


Figure 104: Horizontal Tail Obtained From AAA

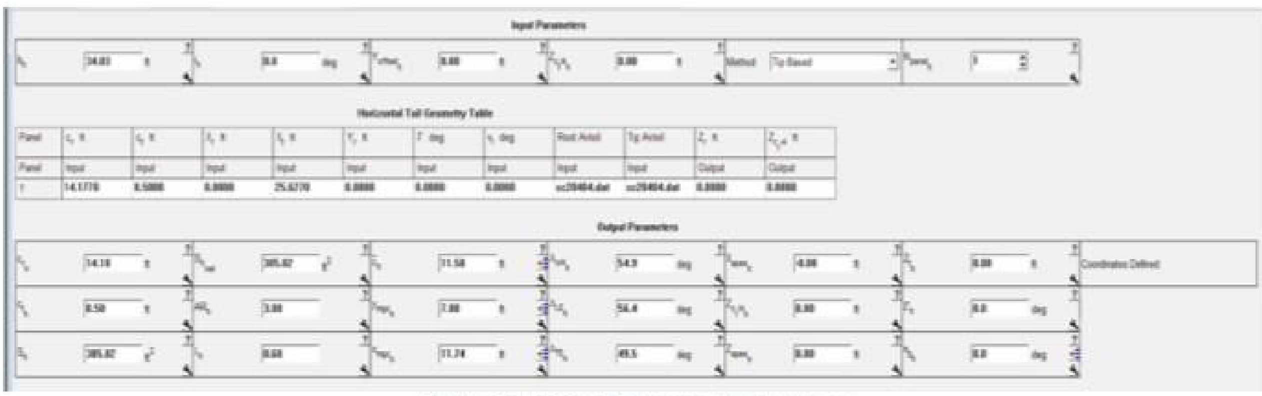


Figure 105: Sizing Parameters for Elevators

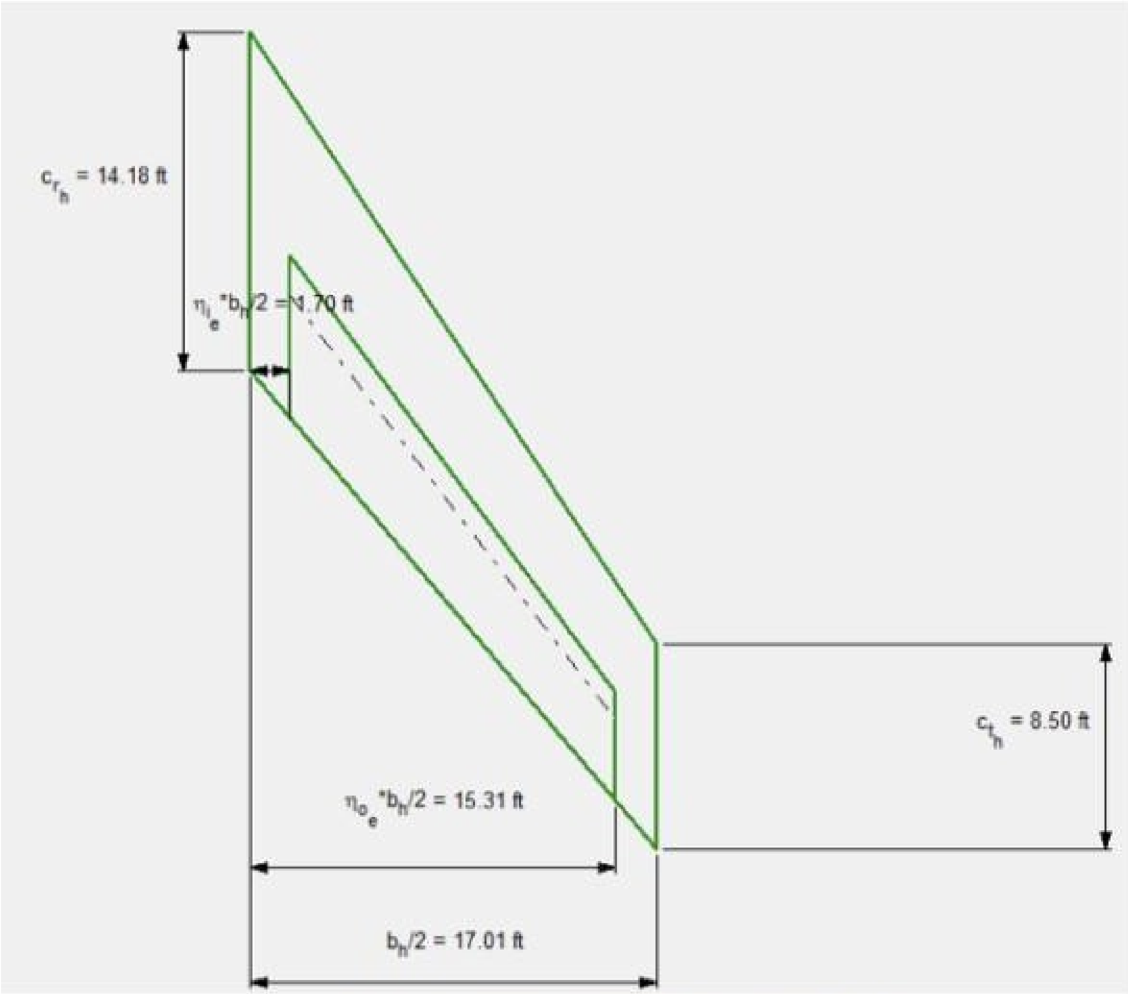


Figure 106: Elevator location and sizing obtained from AAA

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.

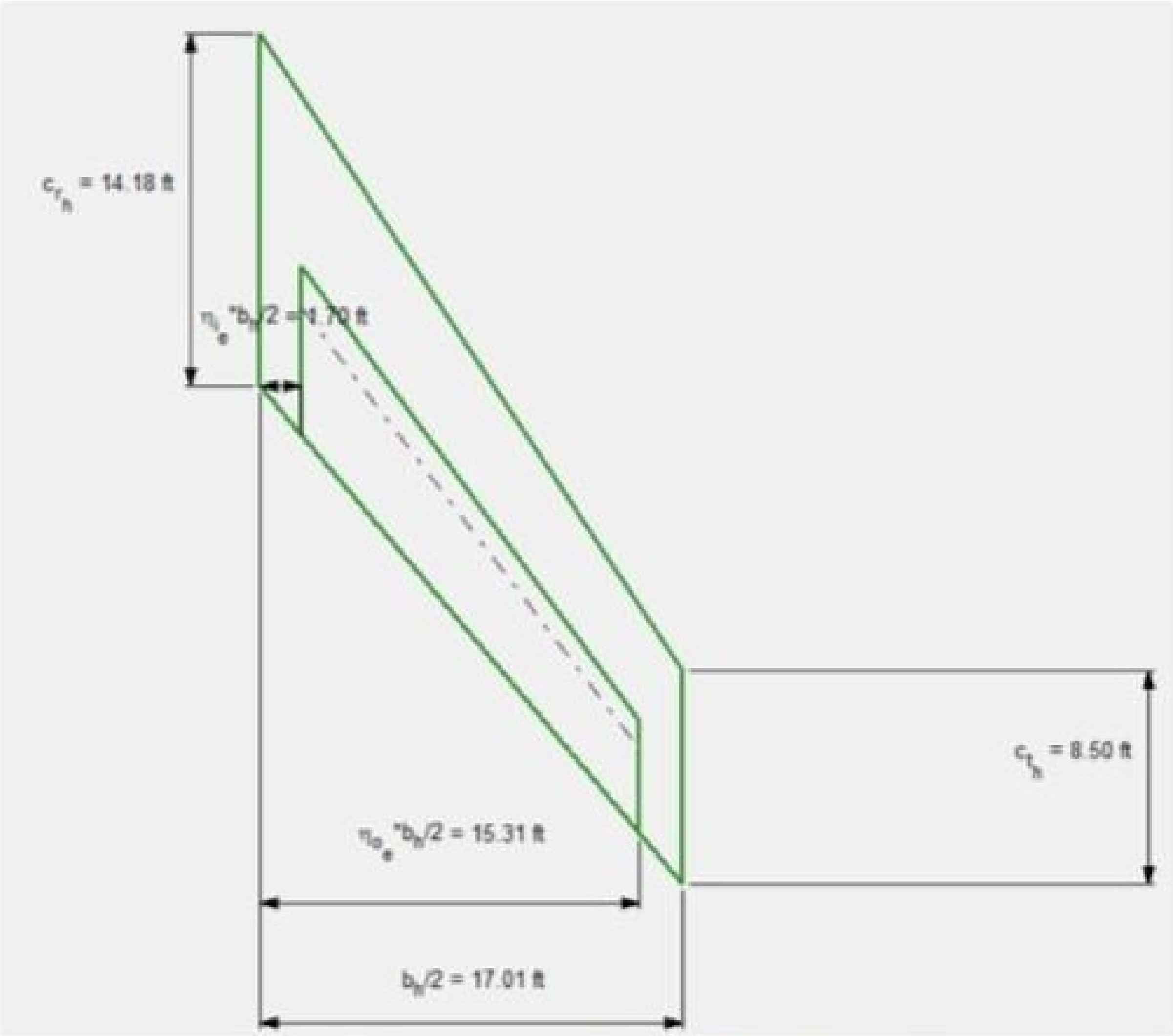


Figure 106: Elevator location and sizing obtained from AAA

7.6 DESIGN OF THE LONGITUDINAL AND DIRECTIONAL CONTROLS

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

For the transonic business jet design, the elevators and the rudder require large area to provide better performance at transonic speeds. So, the rudders and the elevator extend to 50% of the chord. As the design has a T- tail configuration, greater strength is required at the roots of both the horizontal and vertical stabilizers, therefore, the span of the elevators and the rudders shall begin at 10% from the fuselage and go 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 transonic 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

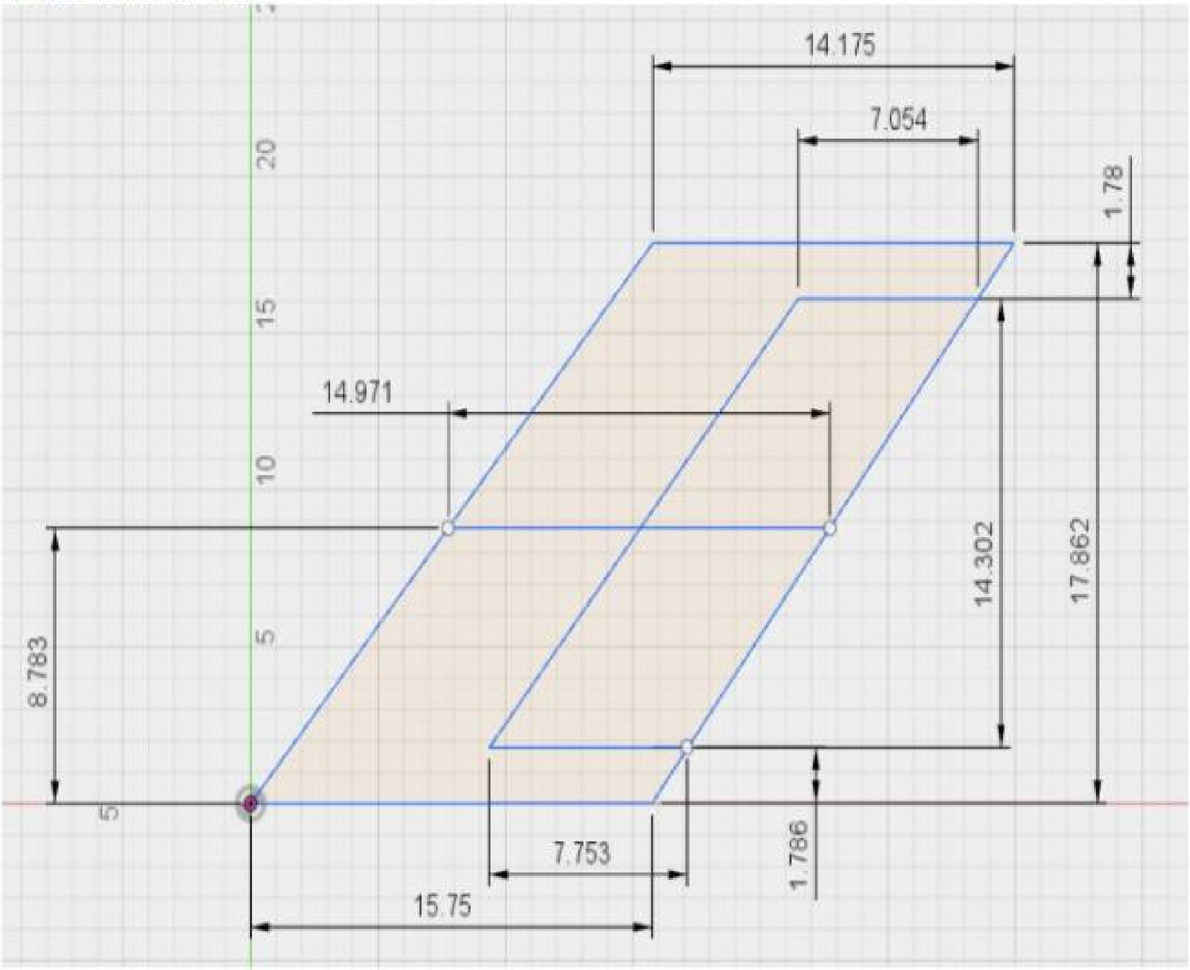


Figure 107: Rudder and vertical stabilizer dimensions (all dimensions in fts)

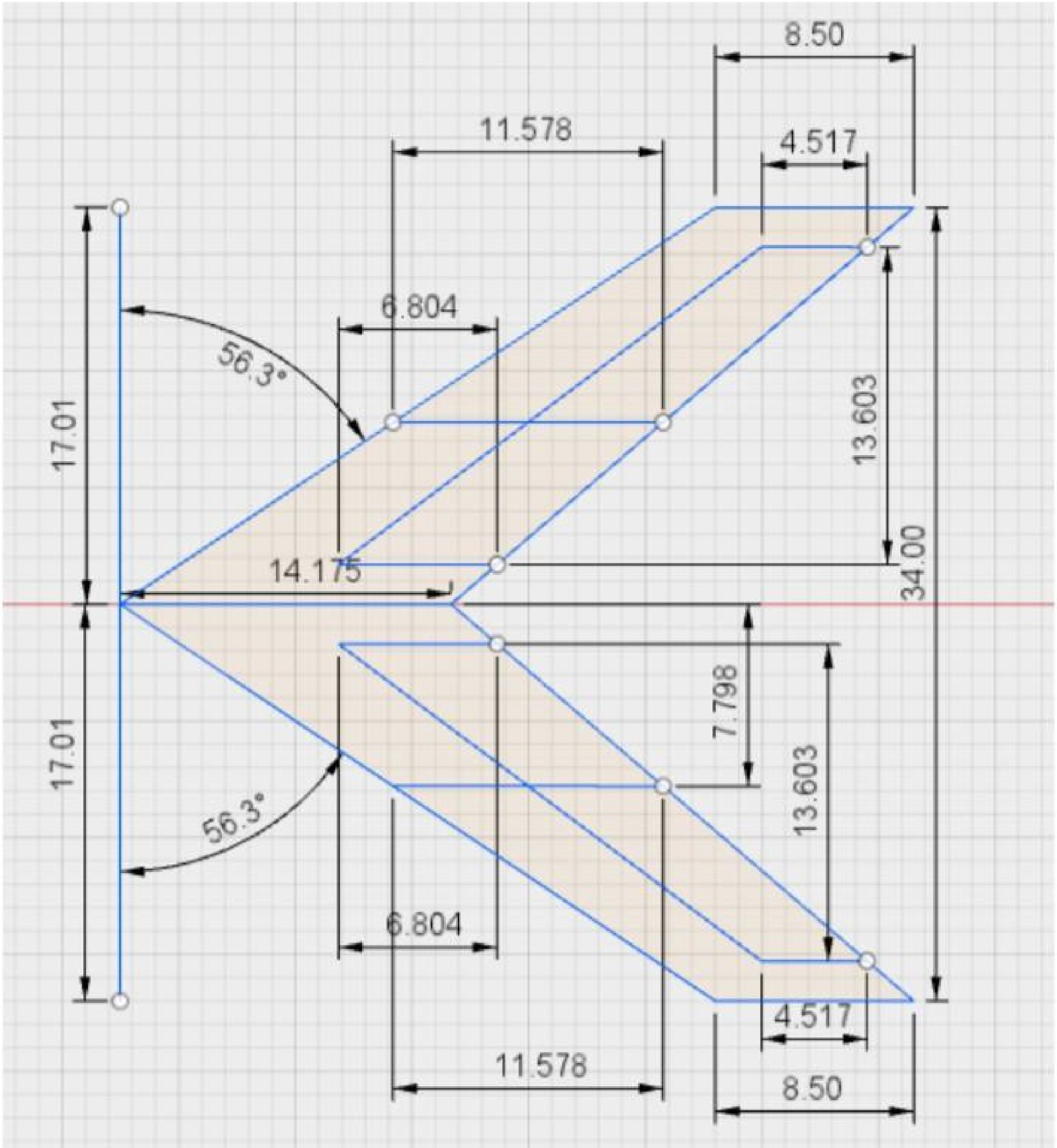


Figure 108: Elevators and horizontal stabilizer dimensions (all dimensions in fts)

7.8 DISCUSSION

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

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

Upon obtaining the necessary parameters, we shall proceed with the designing of the empennage area along with the directional and longitudinal stabilizers. The data which is acquired by using the manual calculations is later compared with the data acquired from the AAA program. The differences in the data acquired through both the process is negligible. Therefore, no changes are made to 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

Seeing the results acquired from the sections given above, we can conclude that no changes are needed in the design of the empennage area in accordance to class-1 Design. The given data satisfies the necessities according to class-1 design. Further calculations along with minute changes shall be carried out post obtaining the stability and control design report.

CHAPTER 8: LANDING GEAR DESIGN, WEIGHT AND BALANCE ANALYSIS

8.1 INTRODUCTION

This is the eighth report of the series for the design of the transonic business jet. This report specifies the landing gear design of the transonic business jet. For designing the landing gears, a suitable configuration needs to be chosen at first. The parameters given below are essential 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

Upon obtaining the above, choose the landing gear system which has to be installed. A retractable tricycle landing gear configuration shall be used for the transonic business jet, as described in the first report. At speeds greater than 150 knots, an extremely high drag penalty is observed in fixed landing gears. Therefore, Retractable landing gears are used as these helps to avoid the same drag penalty.

Two main criterions should be satisfied while designing the landing gears. Those are:

I. Tip-over Criteria.
II. Ground Clearance Criteria.

The design of the landing gear needs an iteration process to acquire the actual CG location of the airplane in order to place the landing gears. This is done to maintain the center of gravity of the airplane. The entire process which is mentioned above is explained in the coming sections:

8.2 ESTIMATION OF THE CENTER OF GRAVITY LOCATION OF THE AIRPLANE

The weights of the components determine the center of gravity and thus, it is essential to breakdown the maximum takeoff weight into parts to acquire the weights of all the components. In table 1, which is given below, the ratio of each component was acquired from similar airplanes. The weights of all the components along with the center of gravity distance is given in the table. While designing the landing gear, the above given step is the primary one. Upon obtaining the new CG, the landing gear shall be re- designed complying to the new CG.

The method to acquire the Center of gravity of the major components is listed in the figure below and the locations of CG of the transonic business jet are detailed in the table given below:

Table 14: List of CG of Major Components

Components	Equation	CG Location (from the nose)
Wing (transonic)	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

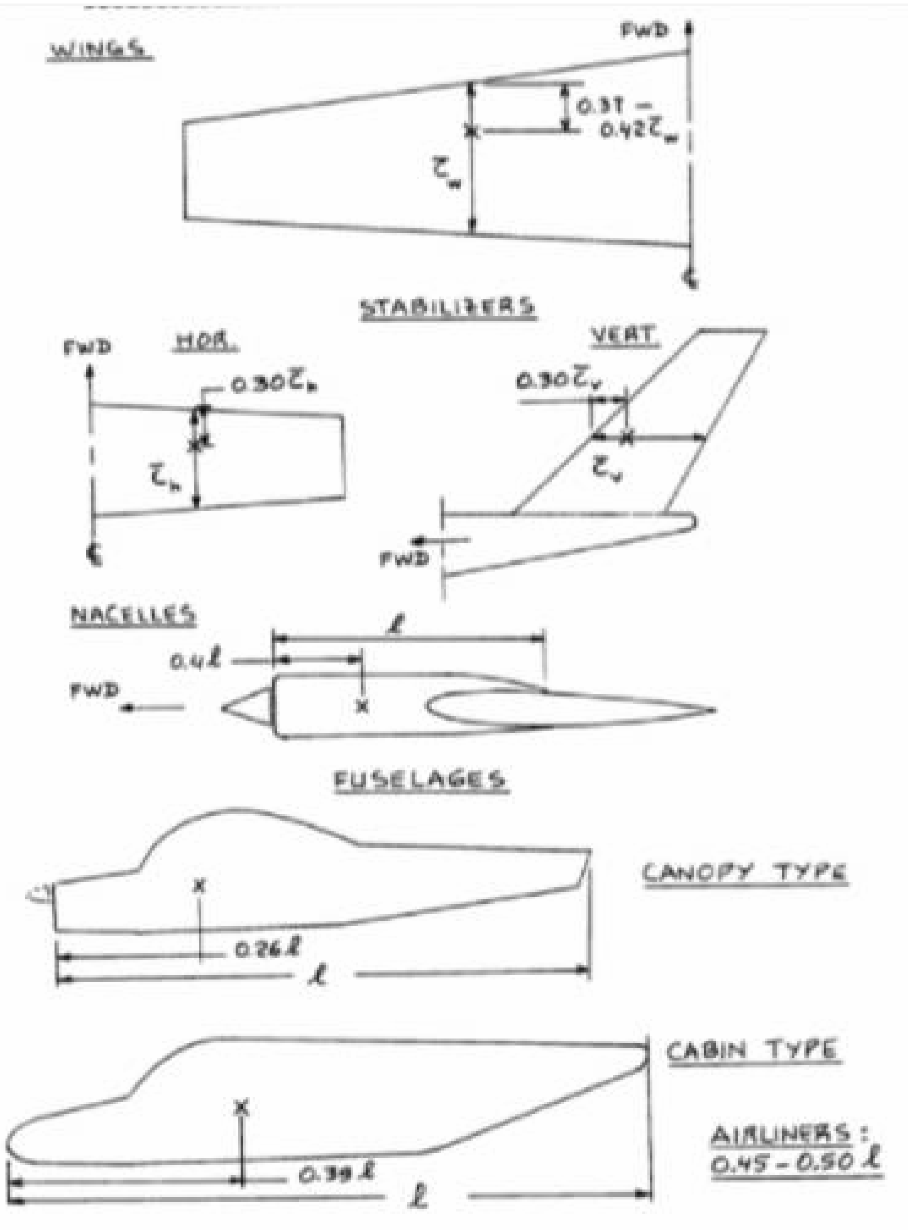


Figure 109: Locations of CG's of major components

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

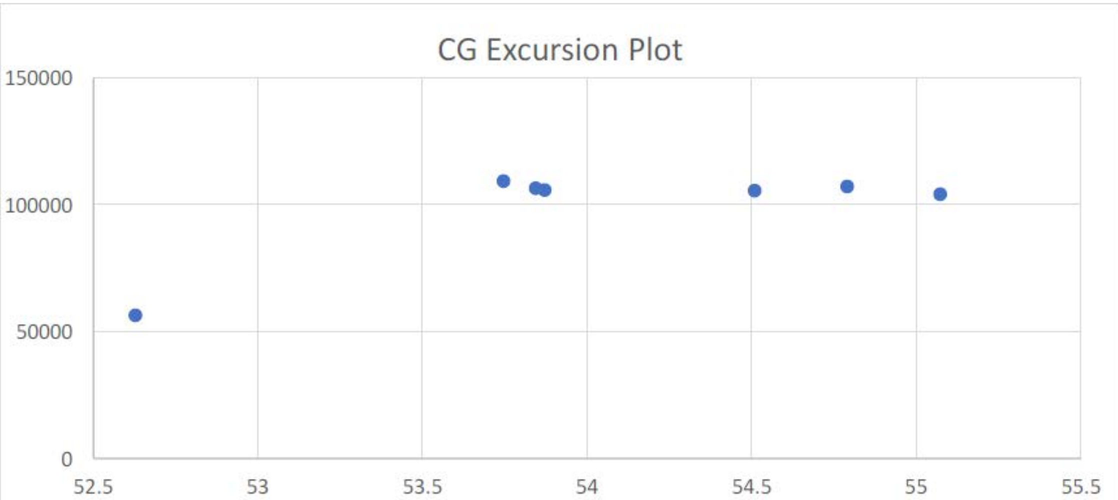


Figure 110: Initial CG Excursion Plot

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 Transonic Business Jet is 2.45 ft (29.4 in) from the allowable range of 20 to 100 ft.

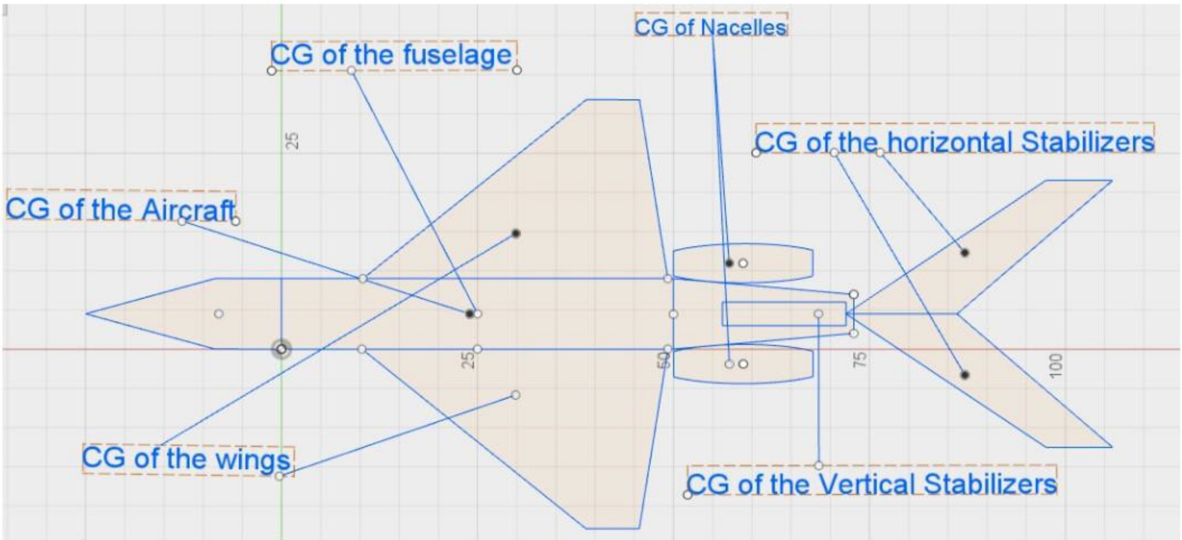


Figure 111: CG Locations of Major Components of the SSBJ

8.3 LANDING GEAR DESIGN

The Transonic Business Jet has landing gears in the form of a retractable landing gear with a tricycle configuration. In general aviation this is the most commonly used configuration. The reason behind this is that the configuration provides a better inclination to the airplane, as it has a low wing configuration. It is between the nose and the main landing gears that the load of the airplane is distributed. The nose landing gear can resist a maximum load of 20% of the total load while 10% of the entire load is ideal for an airplane. It is the remaining 90% of the load that is upon the main landing gears. The Nose landing gear is designed over the CG of the cockpit section while the main landing gears are designed behind the main CG of the aircraft obtained by the iteration process that is carried in the report which is given below. By placing the landing gears 7-15 degrees behind the main CG of the airplane, the tip- over criteria is fulfilled.

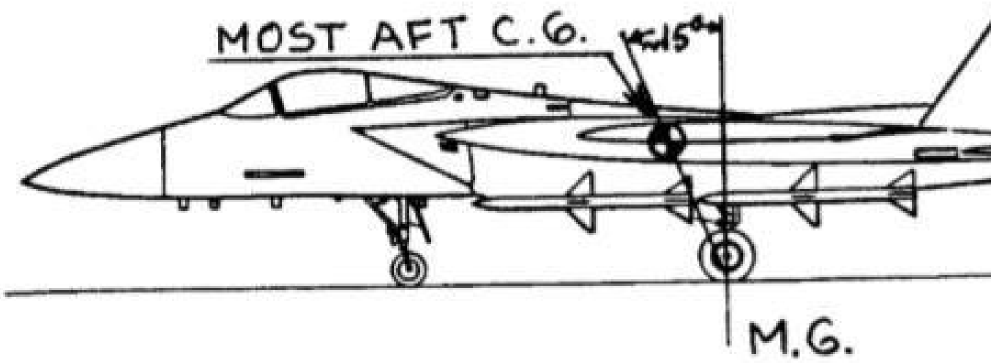


Figure 112: Longitudinal Tip over Criteria

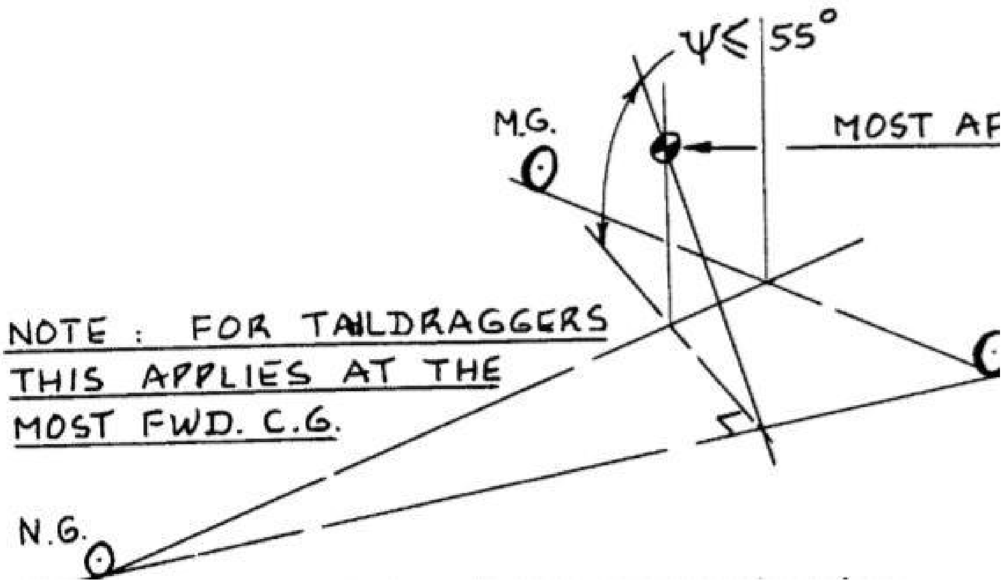


Figure 113: Lateral Tip-Over Criteria

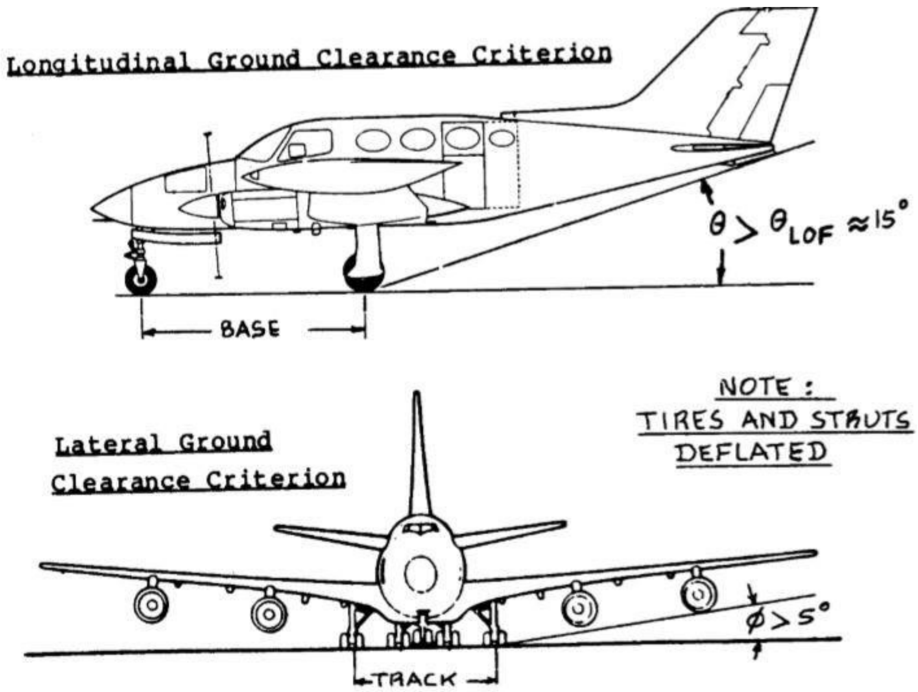


Figure 114: Longitudinal Ground Clearance Criterion

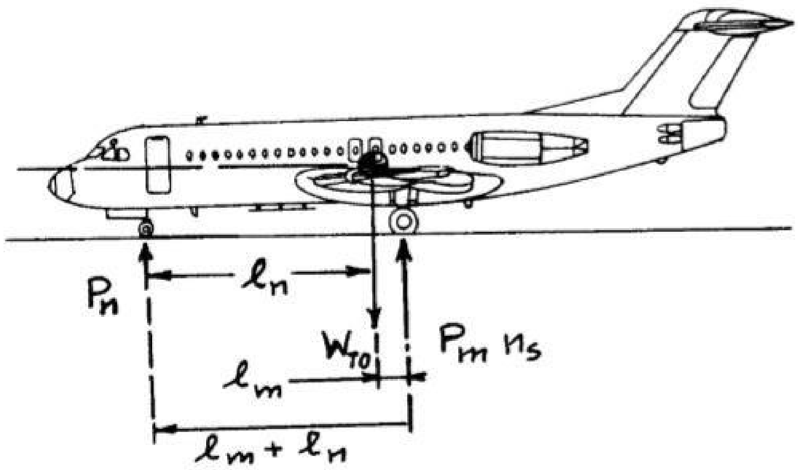


Figure 115: Geometric Wheel location Definition

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: $P_n = \frac{W_{TO} * l_m}{l_m + l_n} = 101065.33 \text{ lbs}$

Main gear strut: $P_m = \frac{W_{TO} * l_n}{number\ of\ strut * (l_m + l_n)} = 48914 \text{ lbs}$

The tire contact area can be obtained using the following equations

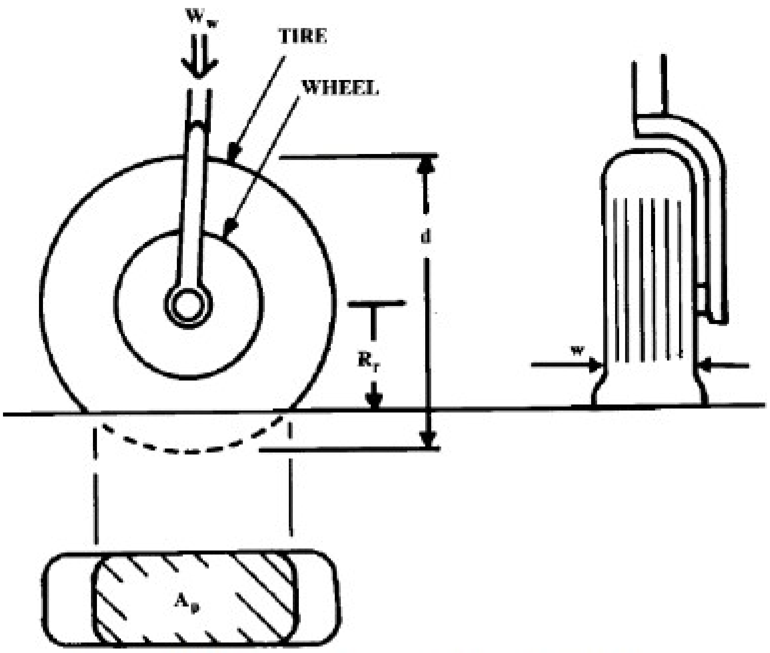


Figure 117: Tire Contact Area Definition

Main Wheel:

$$W_w = PA_P = 24750 \text{ lbs}$$

$$A_P = 2.3\sqrt{wd}\left(\frac{d}{2} - R_r\right) = 190.5\ in^2$$

Nose Wheel:

$$W_w = PA_P = 5500 \text{ lbs}$$

$$A_P = 2.3\sqrt{wd}\left(\frac{d}{2} - R_r\right) = 36.994\ in^2$$

The placements of the landing gears are shown in the figure 10 below:

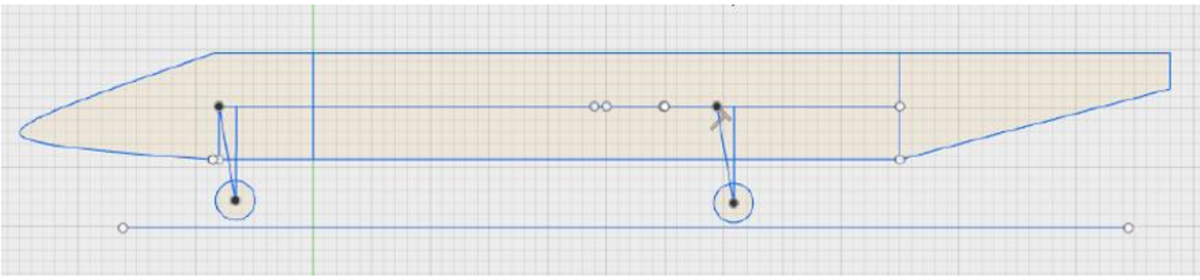


Figure 118: Side View of the TSBJ showing the landing gear Placements

8.4 WEIGHT AND BALANCE

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

Table 18: Parameters Deciding of CG of the TSBJ

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

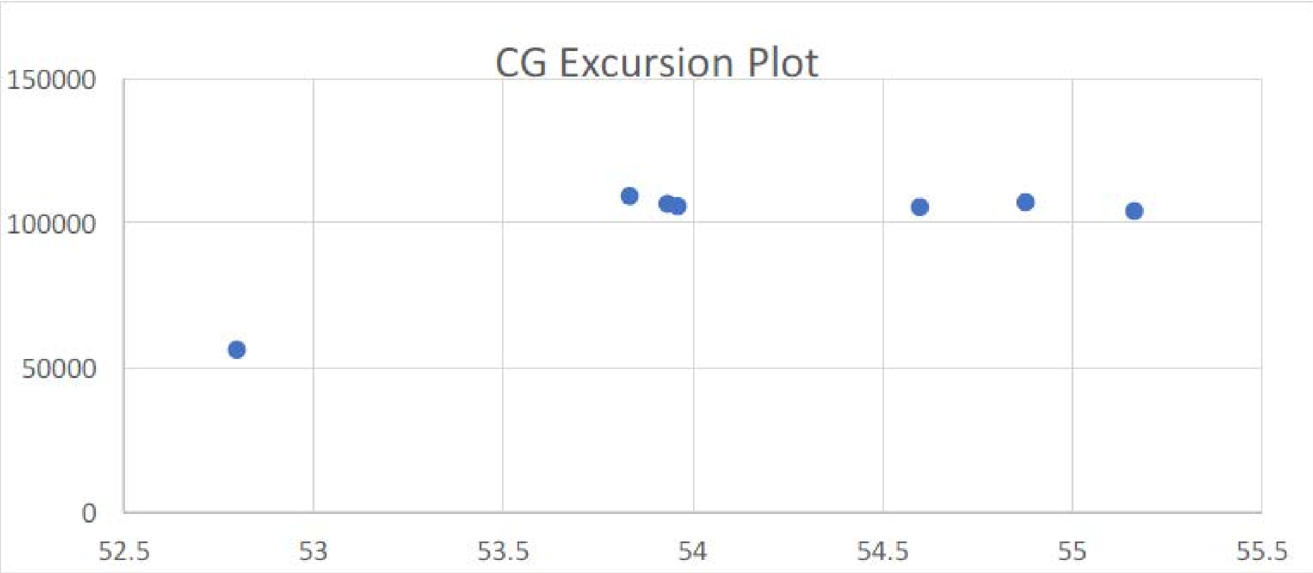


Figure 119: Final CG Excursion Plot

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

This provides a range of 2.37 ft. (28.44 inches) which is an acceptable value from the provided range of 20-100 inches for transonic airplane.

8.5 DISCUSSION

In this report the explanation of the locations of the Center of gravity of numerous components of the airplane, the weights of various sections of the airplane and the location of the landing gears is given. The retractable tri-cycle landing gear configuration is the landing gear configuration which is chosen for the transonic business jet. To pin point the CG of the whole airplane, considering the distance from the nose tip and the weight, an iterative process is to be worked upon to obtain a range of CG's from which a final CG of the airplane shall be determined. Upon obtaining the CG, the landing gears shall be designed behind the most aft CG obtained in order to maintain the balance of the airplane and ensuring that the tip over and the ground clearance criterion is met. The loads acting on each tyre is also tabulated via the equations as mentioned in the nose landing gear wheel dimensions' and the main landing gear table. Using the tables and equations obtained from Roskam and Raymer, the landing gear data is then calculated. Then via the dimensions obtained, the CAD is designed as seen in the above sections.

8.6 CONCLUSION

The CG range acquired is 28.7 inches, this is absolutely within the permissible range of 20 – 100 inches for transonic airplanes. Thus, it can be determined that the airplane is stable with such a low variation in the range of the CG of the airplane. For future work, a greater number of iterations shall be calculated to determine the most exact CG, also the place of the landing gears, to gain accurate results while designing the Transonic Business Jet.

CHAPTER 9: STABILITY AND CONTROL ANALYSIS

9.1 INTRODUCTION

The 9th report of the preliminary airplane design describes the stability and control analysis of the Transonic Business Jet. The process of designing shall be carried via the following the steps as given by Roskam in the book Airplane design part II, these satisfy the class 1 design requirements. The longitudinal and directional stabilities shall be calculated in the report. There are two types of stability:

I. Static Stability
II. Dynamic stability

Static stability is the type of stability that deals with the initial tendency of the object to return to the equilibrium position when it is disturbed while dynamic stability is the type of stability that deals with the time history of the vehicles motion after it initially responds to its own static stability. Static stability is not always enough to ensure dynamic stability. Hence, a dynamically stable airplane should always be statically stable.

Control is defined as the study of deflections of the high lift devices. It also includes the directional and longitudinal devices essential to make the aircraft under a controlled situation and thereby obtaining the desired output as needed by the pilot.

The 9th chapter explains the x-plots to determine, it also defines changes in the horizontal area with respect to the center of gravity and the aerodynamic centers, this also changes in the vertical tail area with respect to the side-slip. In this process, iteration processes are essential to determine the exact values which are to be used whilst designing the empennage section and the landing gears of the airplane.

9.2 STATIC LONGITUDINAL STABILITY

This section of the 9th report describes the static longitudinal stability. In order to obtain the directional x-plot, the desired parameters are the center of gravity of the airplane and the aerodynamic center, both of which are directly proportional to the change in the horizontal tail area. The aerodynamic center is obtained from the following equation:

$$\bar{x}_{ac_A} = \frac{\bar{x}_{ac_{wf}} + \frac{C_{L_{ah}}\left(1-\frac{\partial\epsilon_h}{\partial\alpha}\right)\left(\frac{S_h}{S}\right)\bar{x}_{ac_h}}{C_{L_{awf}}}}{F}$$

Where,

$$F = 1 + \frac{C_{L_{ah}}\left(1-\frac{\partial\epsilon_h}{\partial\alpha}\right)\left(\frac{S_h}{S}\right)}{C_{L_{awf}}}$$

The CG of the aircraft is acquired by varying the area of the horizontal tail with respect to the change in its weight. When we compare 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 that is prepared is known as the longitudinal X-plot. The horizontal tail's area can be determined at the point where there is a variation of 10% in the graph after the interaction of the points.

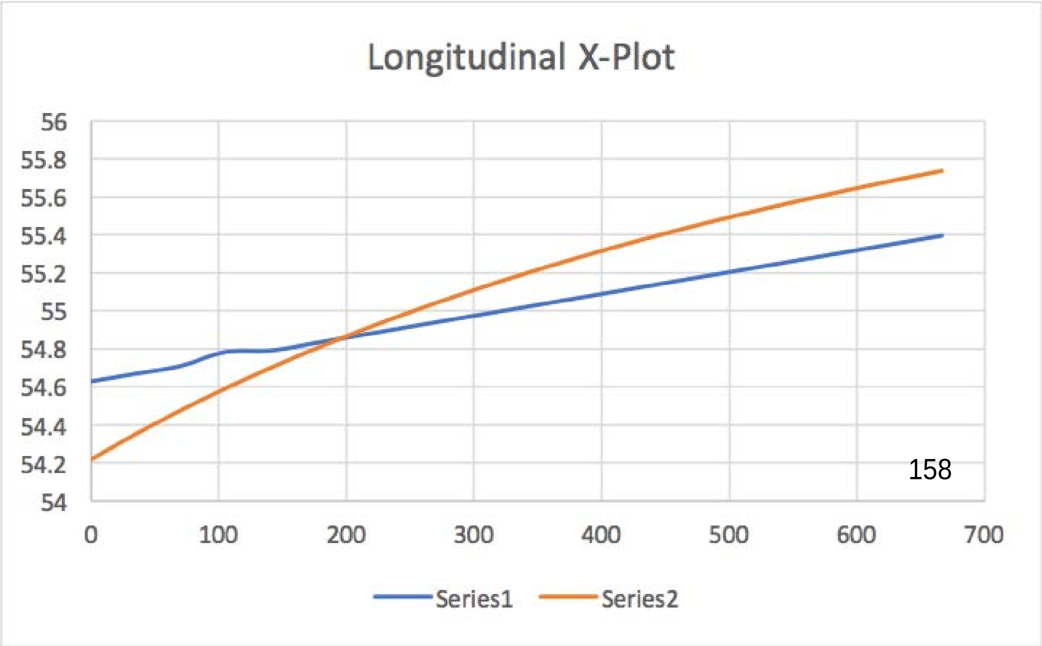


Figure 120: Longitudinal X-Plot

The horizontal stabilizer’s area in the transonic business jet is pegged at 385.94 ft^2 . As seen from the above X-plot, the 10% variation is acquired around 380 ft^2. Therefore, there isn’t a significant change in the area of the horizontal tail thereby concluding that the aircraft is longitudinally stable according to class-1 stability requirements and the horizontal tail area shall be maintained at 385.94 ft^2.

9.3 STATIC DIRECTIONAL STABILITY

This section of the report defines the Static directional stability of the aircraft. The vertical tail area is directly proportional to the side-slip feedback system in order to ensure the static directional stability. The side- slip of the wing is assumed to be zero for the preliminary design purposes, whereas the side-slip of the fuselage and the vertical tail can be obtained from the equations described below:

$$C_{n_\beta} = C_{n_{\beta_w}} + C_{n_{\beta_f}} + C_{n_{\beta_v}}$$

Where,

$$C_{n_{\beta_w}} = 0\ (\textit{For priliminary Sizing})$$

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

$$C_{n_{\beta_v}} = -\left(C_{Y_{\beta_v}}\right)\left(\frac{l_v \cos(\alpha) + Z_v \sin(\alpha)}{b}\right)$$

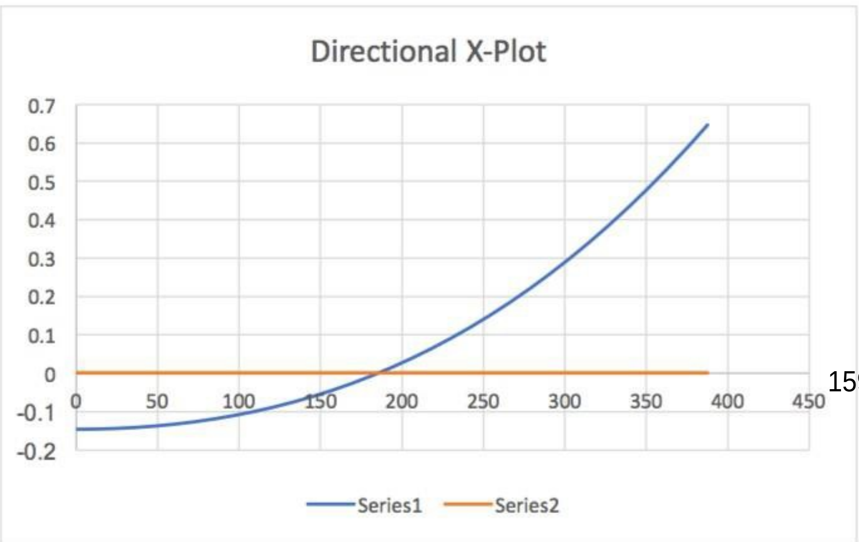


Figure 121: Directional X-Plot

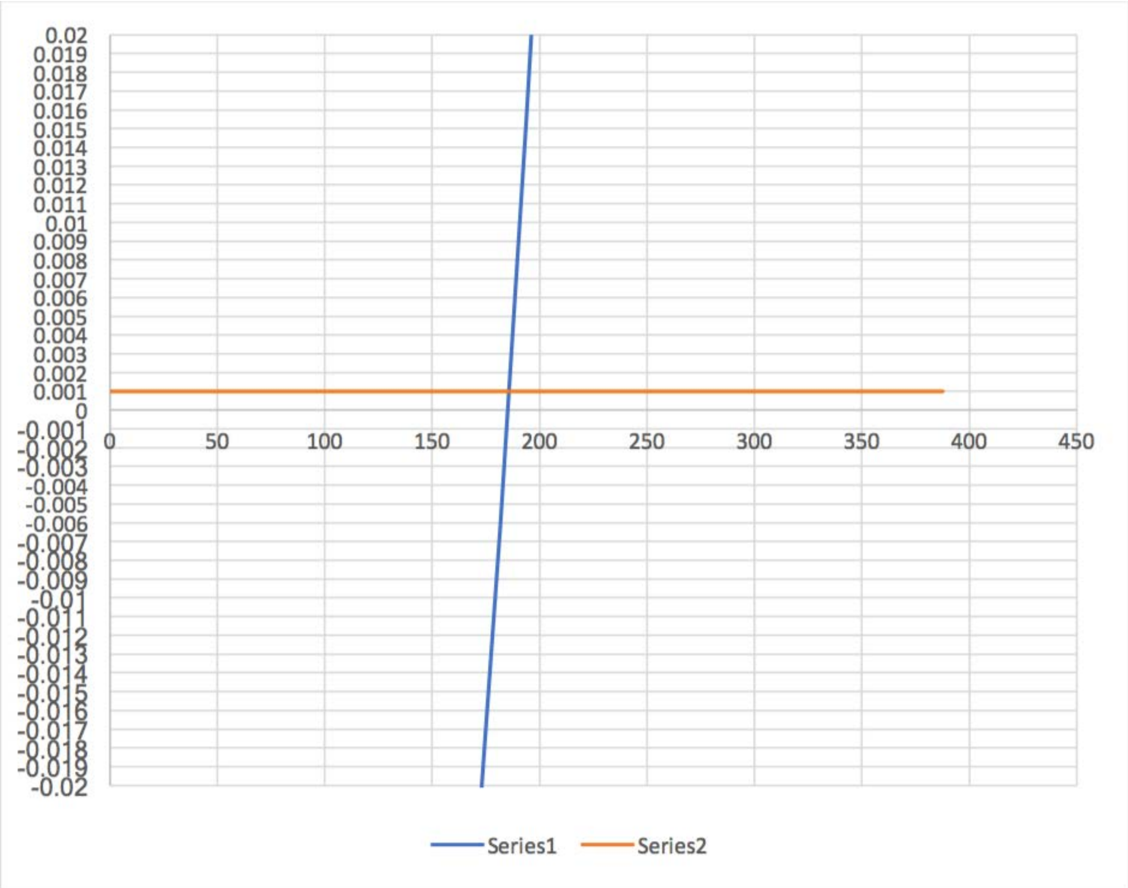


Figure 122: Zoomed view of the directional X-plot

At side-slip C_0 = 0.001, the area of the vertical stabilizer acquired is 190 ft this value is 2

slightly lesser than the area which is used to design the
vertical stabilizer. Therefore, it wouldn't make too much of a difference on the design of the transonic business jet. This concludes that such small differences are slightly negligible in the preliminary design process. According to class 1 design process, the current design is directionally stable, and the deflection obtained on the vertical is:

$$k_\beta = -\frac{1.6436196}{-1.6426196} = 1\ deg/deg$$

$$k_\beta = 1.$$

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 transonic business jet is 190 knots hence for one engine out requirements to satisfy, the minimum required speed is tabulated to be 1.2* 190 = 228 knots.

From the vertical tail and the rudder geometry, the value for rudder control power derivative is computed to the value of $-1.6436196\ deg^{-1}$
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

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

9.5 DISCUSSION

According to the above report, the static longitudinal and the static directional stability is discussed in detail, along with the one engine stall speed requirements for satisfying the stability and control of the transonic business jet. For obtaining the static longitudinal stability, the X-plot was generated by making use of 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 acquired from the different weights and horizontal tail area which proves that the transonic business jet is statically longitudinally stable according to class 1 preliminary design as is satisfies all the requirements with in the provides range.

In the next step, the static directional stability was calculated by making use of the sideslip with respect to the vertical tail areas along with their respective weights. The condition was satisfied by acquiring the required area of the vertical tail, making it longitudinally stable according to the preliminary design requirements by providing a deflection angle of 1 degree from the maximum allowable 5 degrees. Upon satisfying the stability criterion, the one engine out requirements were calculated to obtain the minimum requirements to satisfy the controls requirements.

9.6 CONCLUSION

Post calculating the static longitudinal and static directional X-plots, it can be determined that the transonic business jet is both directionally and longitudinally stable according to class 1 requirements. For future work, the areas of the vertical and horizontal tails shall be sized accurately to overcome the minute changes in the areas.

CHAPTER 10: DRAG POLAR ESTIMATION

10.1 INTRODUCTION

The drag polars of the transonic business jet have already been calculated in the performance sizing report which was report number 4. In the 10th report of the proposed aircraft we compare those results after all the changes made in the airplane beyond all the iteration process considering the CG locations and the required parameters. The 10th report specifies the drag polar estimation of the airplane. Every component of the airplane produces drag, however, this report specifies the drag produced using high lift devices and landing gears during 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

By adding up the wetted area of different parts of the airplane the overall wetted area of the airplane can be calculated. The wetted area of different parts can be calculated by the help of 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:

$$S_{wet_{plf}} = 2 * S_{exp.plf}\left(1 + \frac{0.25\left(\frac{t}{c}\right)_r(1+\tau\lambda)}{1+\lambda}\right)$$

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

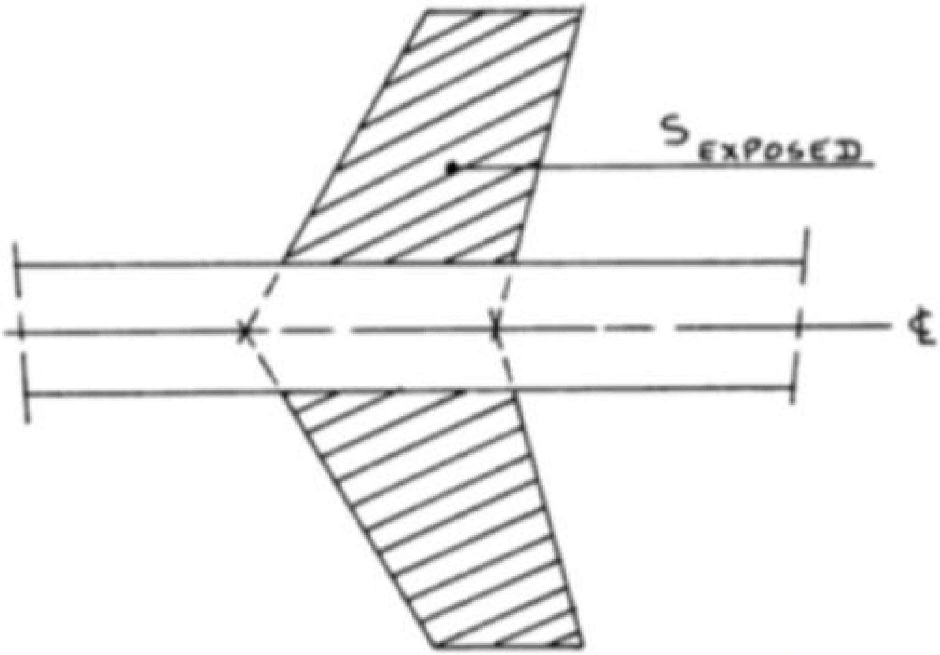


Figure 123: Definition of the Exposed Planform

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

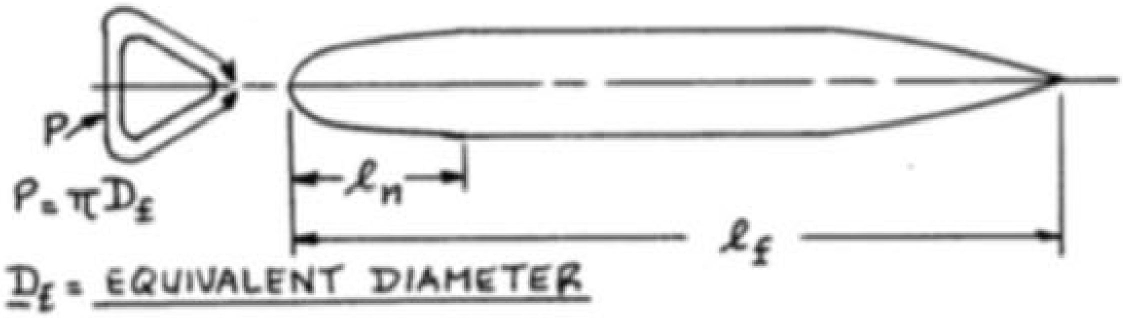


Figure 124: Fuselage Wetted Area Parameters

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:

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

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

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

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:

$S_{Wet} = 5789.67\ ft^2$

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:

$C_{D_0} = \frac{f}{S} = \frac{13.6}{1424.34} = 0.00954$

$C_{D_0} = 0.00954$

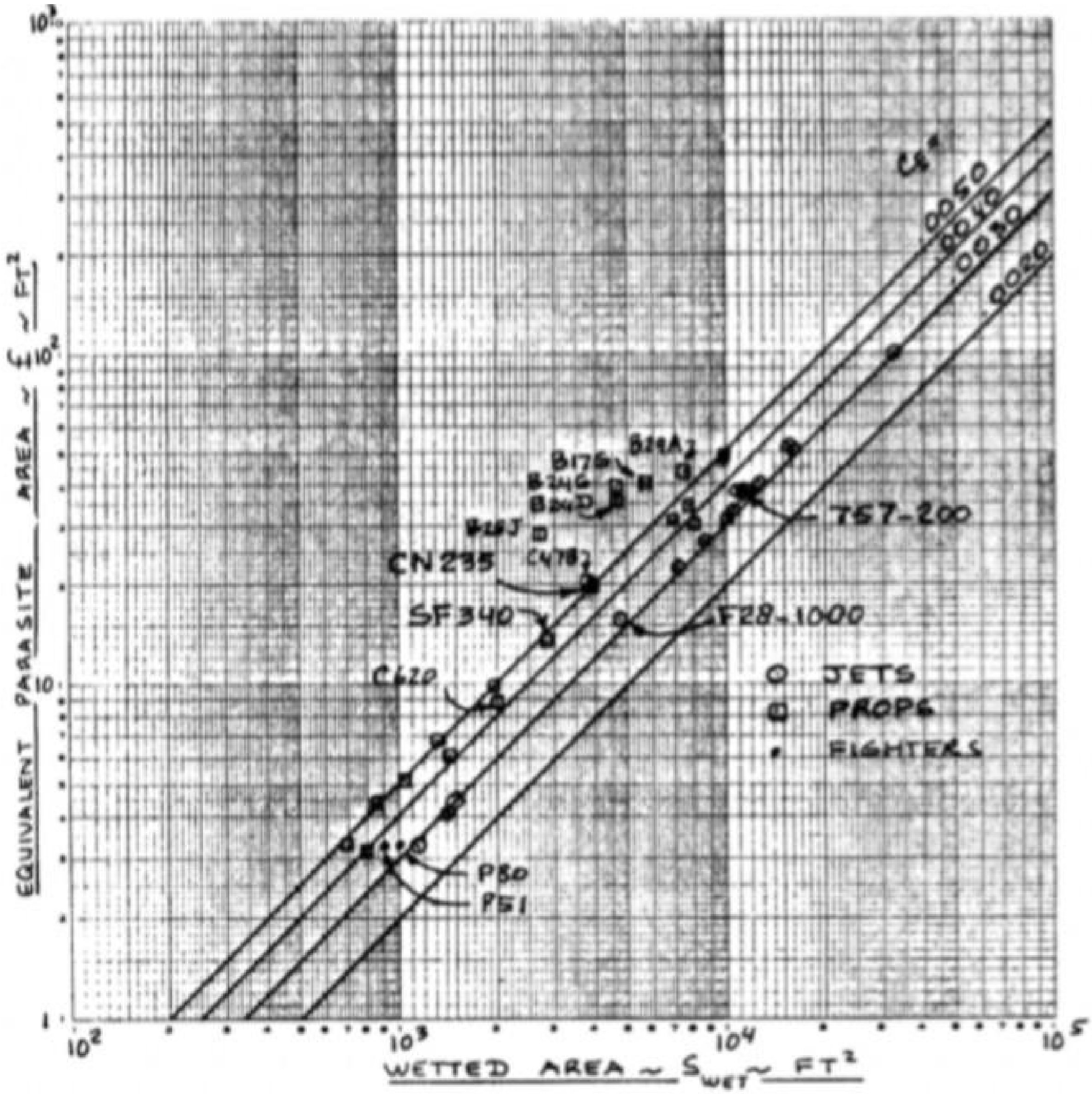


Figure 125: Equivalent Parasite Area v/s Wetted Area

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 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	ΔC_{D_0}	e	Aspect ratio	Drag Polar
Take-off Flaps	0.020	0.8	2.1	$C_D = 0.0419 + 0.1895\,C_L^2$
Landing-Flaps	0.075	0.75	2.1	$C_D = 0.0969 + 0.2021\,C_L^2$

10.3.2 Landing Gear Drag

Table 26: Drag Increment Due to the Landing- Gear

Component	ΔC_{D_0}	e	Aspect ratio	Drag Polar
Landing Gears	0.025	No Effect	2.1	$C_D = 0.03454$

10.4 COMPRESSIBILITY DRAG

The compressibility drag section is defined for the airplanes flying at subsonic speeds. The TSBJ cruises at transonic speed therefore the compressibility drag section does not 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_{D_0}
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	$C_D = 0.1219 + 0.1783\ C_L^2$
TO Flaps-Gear Down	0.8	2.1	0.045	$C_D = 0.0669 + 0.1895\ C_L^2$
TO Flaps-Gear Up	0.8	2.1	0.020	$C_D = 0.0419 + 0.1783\ C_L^2$
Landing Flaps Gear-Up	0.75	2.1	0.075	$C_D = 0.0969 + 0.2021\ C_L^2$
Landing Flaps Gear- Down	0.75	2.1	0.1	$C_D = 0.1219 + 0.2021\ C_L^2$

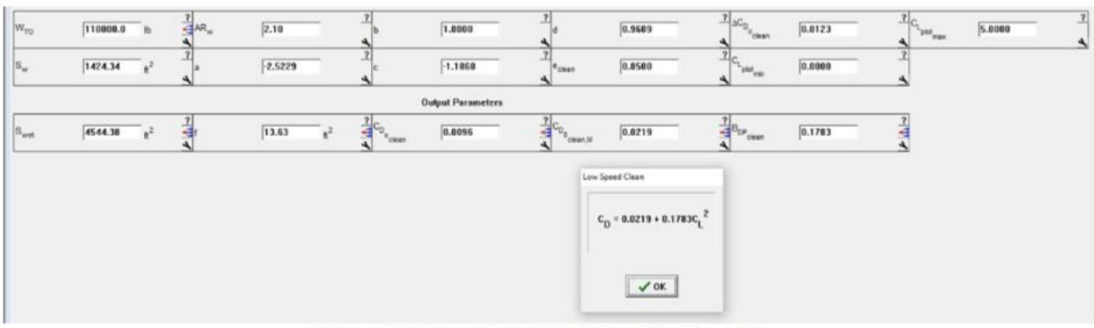


Figure 127: Drag Polar Calculation for Clean Stage

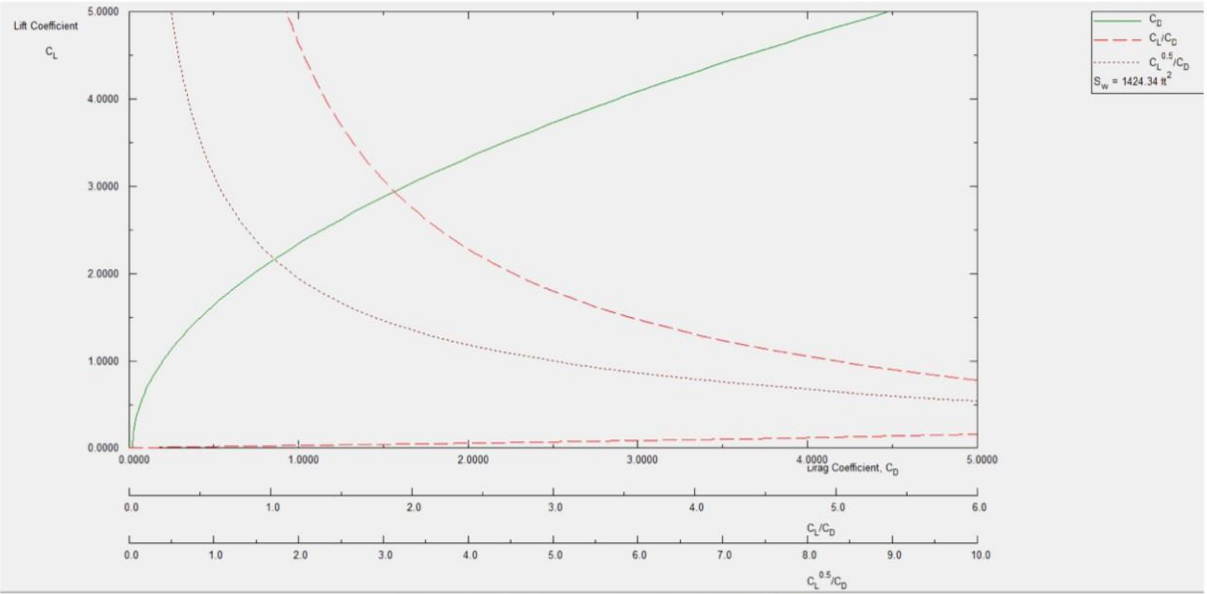


Figure 128: Plot of Clean stage CL v/s CD

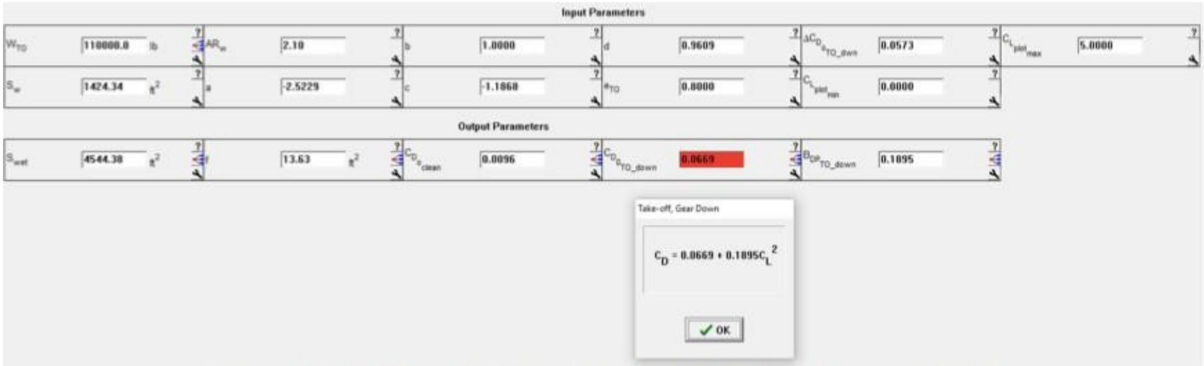


Figure 129: Drag Polar Calculations for Take-off-flaps- Gear down configuration

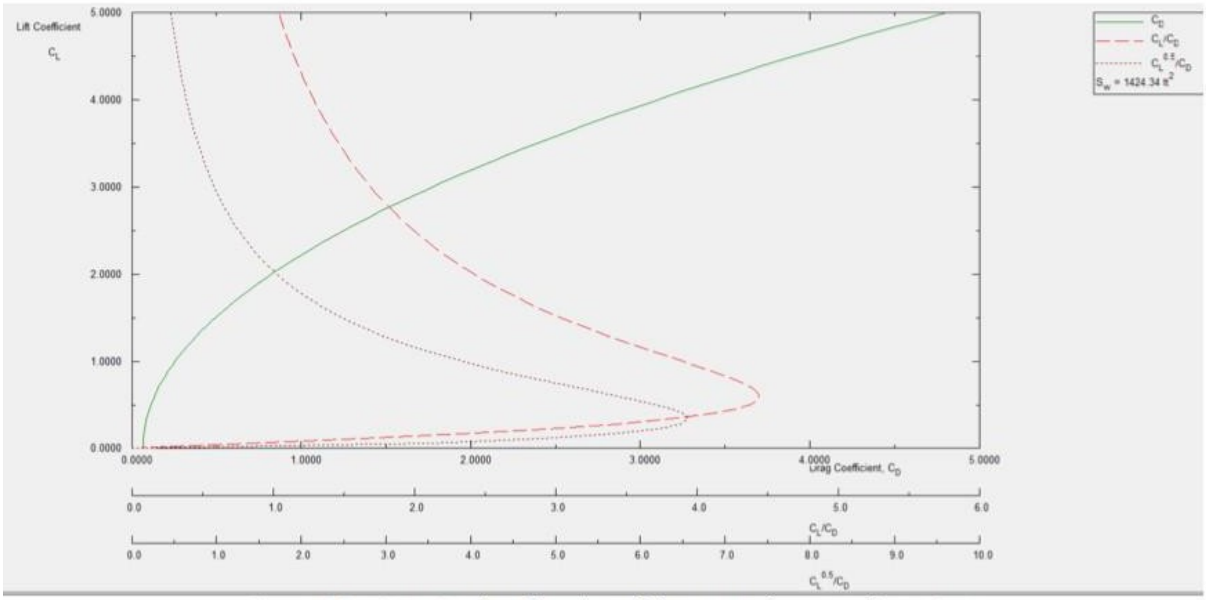


Figure 130: CL vs CD plots for take-off flaps-gear down configuration

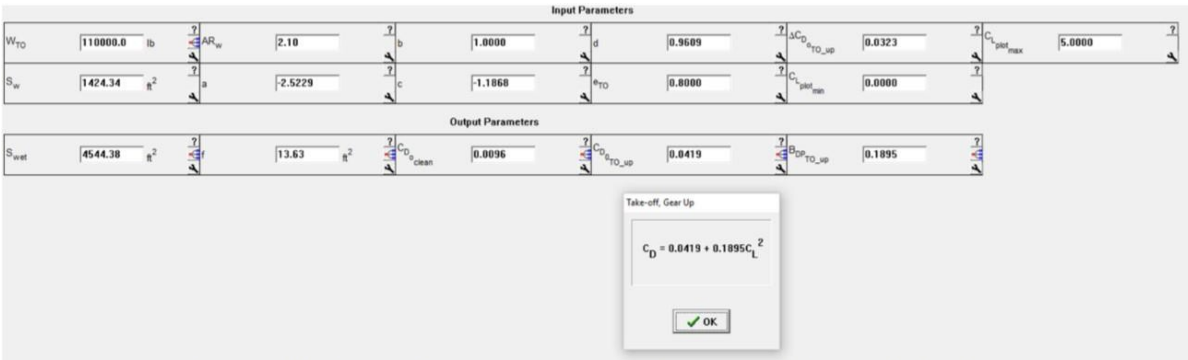


Figure 131: Drag Polar Calculations for Take-off flaps - Gear-Up Configuration

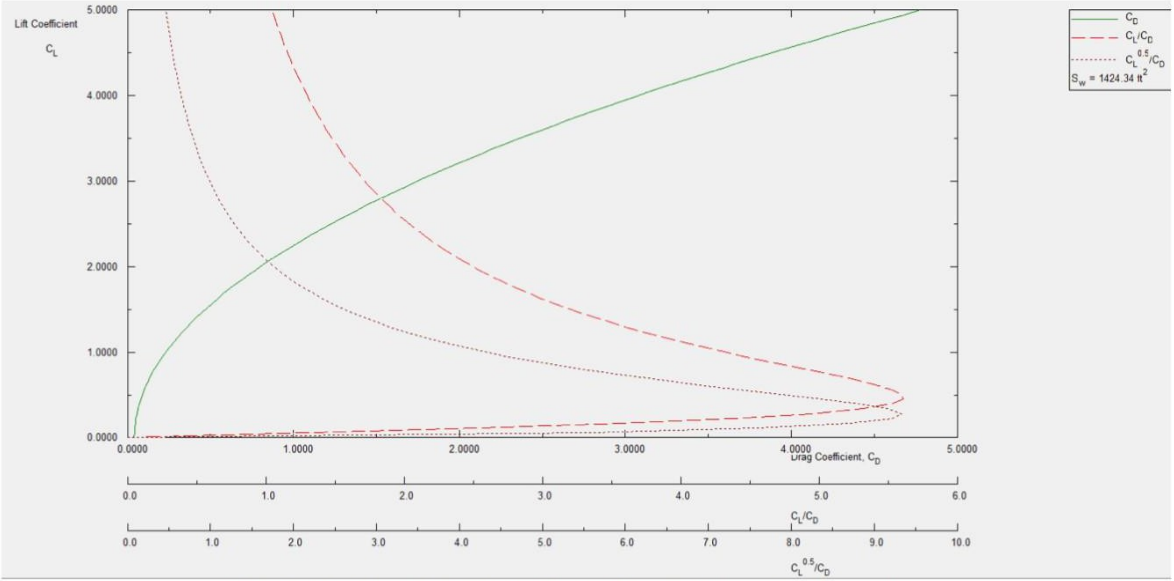


Figure 132: CL vs CD Plots for Take-off Flaps- Gear-up Configuration

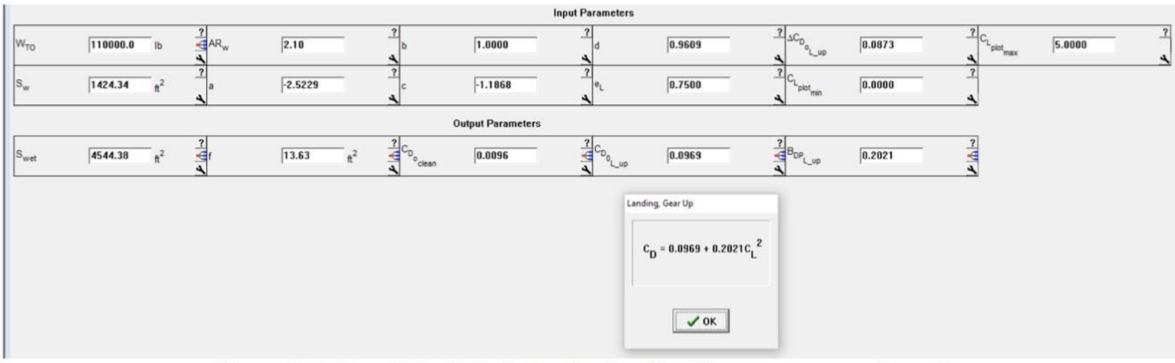


Figure 133: Drag Polar Calculation for Landing-Flaps gear-up configuration

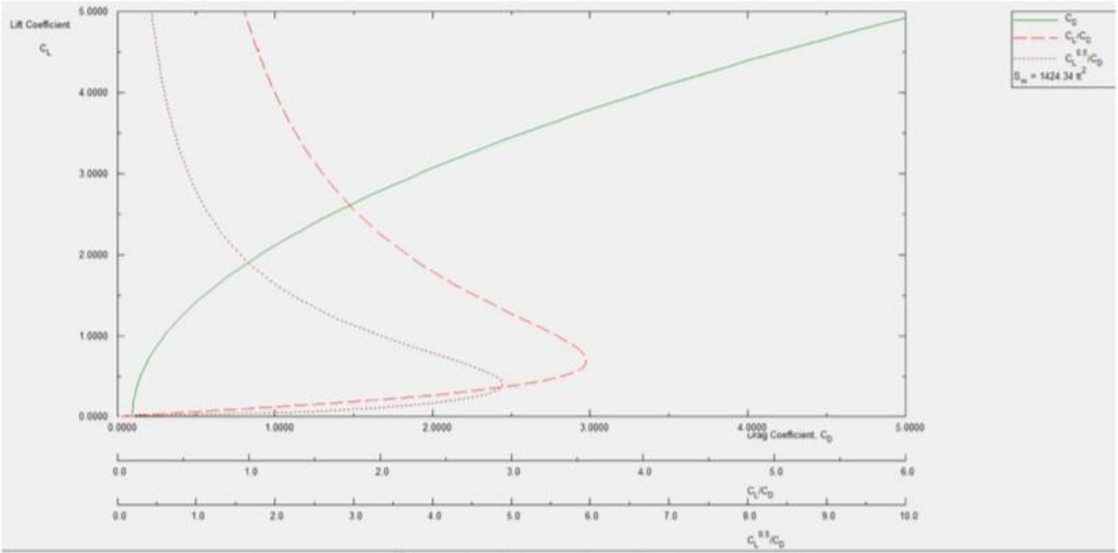


Figure 134: CL v/s CD plots for Landing-Flaps gear up configuration

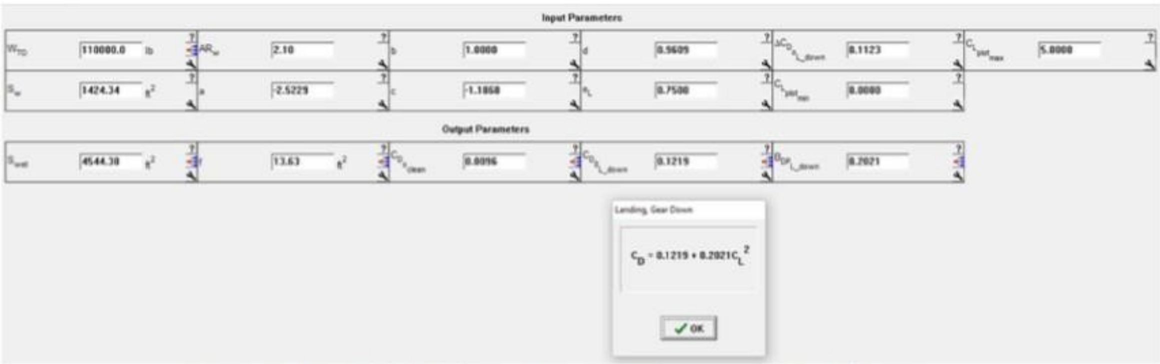


Figure 135: Drag Polar Calculation For Landing-Flaps gear-down configuration

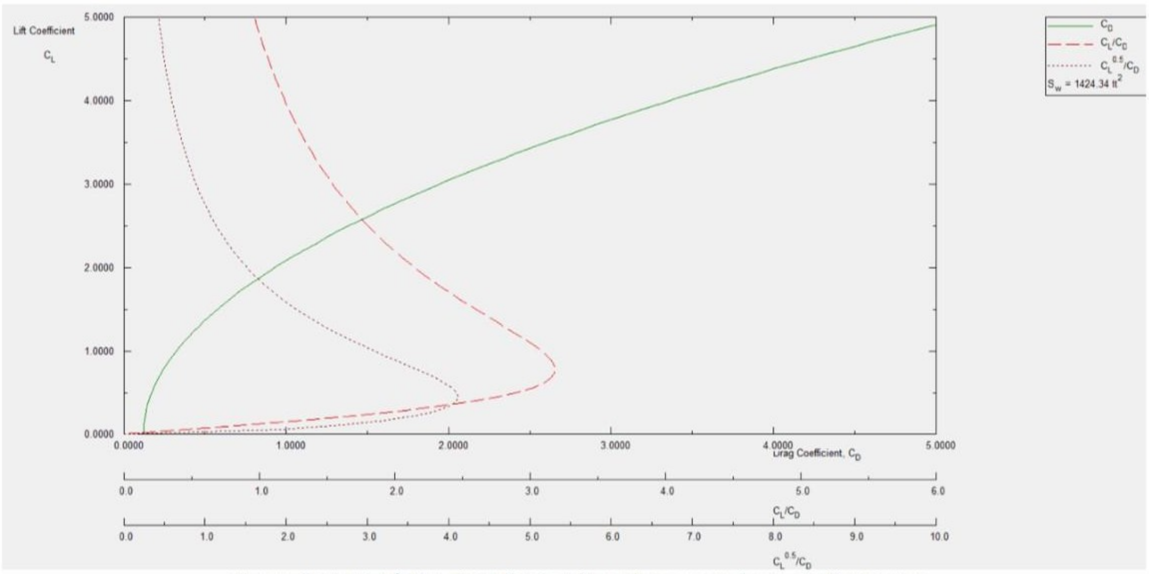


Figure 136: CL v/s CD plots for Landing-Flaps gear down configuration

10.7 DISCUSSION

In the 10th report of the proposed aircraft the drag polars of the landing gears and high lift devices were calculated and obtained by making use of the manual calculations also by making use of the AAA program. The results acquired from both the process provide the exact 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. Therefore, no further change is required in the design of the transonic business jet.

10.8 CONCLUSION

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

CLASS II TSBJ DESIGN

CHAPTER 11: LANDING GEAR DESIGN

11.1 Introduction

The landing gears which are part of an aircraft are used for the reasons given below:

- The landing gears absorb landing shocks and taxing shocks.
- The landing gears provide enhanced ground maneuvering that includes: taxi, take-off roll, landing roll and steering.
- The landing gears provide improved braking capability.
- The landing gears enable airplane towing.
- The landing gears help in protecting the ground surface.

The landing gears must be designed in a manner such that the zero touch down rate must be able to absorb landing and taxi loads along with being able to transmit a part of such loads to the airframe. The magnitude of these loads depends upon the kind of airplane and the airplane's mission profile. The kinds of loads which must be taken into consideration 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.

11.2 Vertical Landing Gear Loads:

The magnitude of the loads acting on the vertical landing gears depends directly on the touchdown rate. For the business Jet, the touch down rate should be,

$$w_t = 17 \text{ fps.} \quad (1)$$

Landing Gear types:

Below are the two main considerations to be kept in mind while designing the type of the landing gear:

a. Fixed of retractable landing gear.

b. Deciding the configuration of the landing gear (tricycle, bicycle, tailwheel or unconventional gear.

As per the earlier discussion, the transonic 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.

The reason for selecting the retractable landing gears was the minimal aerodynamic drag provided by them.

11.3 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

11.4 Nose gear steering loads:

A minimum normal force must act on the nose gear to generate appropriate levels of friction forces needed for steering in order to obtain adequate nose wheel 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:

$$0.08 * W_{TO} = 0.08 * 110{,}000 = 8800\ lbs\ (3991.6\ kg) \tag{2}$$

11.5 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**: these are the surfaces with rigid pavement (concrete). The thickness of these surfaces is half of the thickness of the flexible pavement.

11.6 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 TSBJ has a twin- tricycle configuration, the equivalent single wheel load will be:

$$ESWL = \frac{P_n}{2} \; or \frac{P_m}{2} \qquad (3)$$

where P_n and P_m are the loads acting on the nose wheel and the main landing gear wheels.

$$P_n + n_s P_m = W \qquad (4)$$

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_0 or D_t : outside diameter

W or W_t: maximum width

D: the tire rim diameter

The structure if tires get affected due to the severe static and dynamic load during taxing, during take-off roll and during landing roll. The tires also absorb shock during touchdown and the amount of shock tires can absorb directly depends upon 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/CG. 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 = $f_{dyn}(static\ load)$
For tires for new designs: $f_{dyn} = 1.50$ (5)
The allowable tire deflection may be computed from: $s_t = D_o - 2$ (loaded radius)

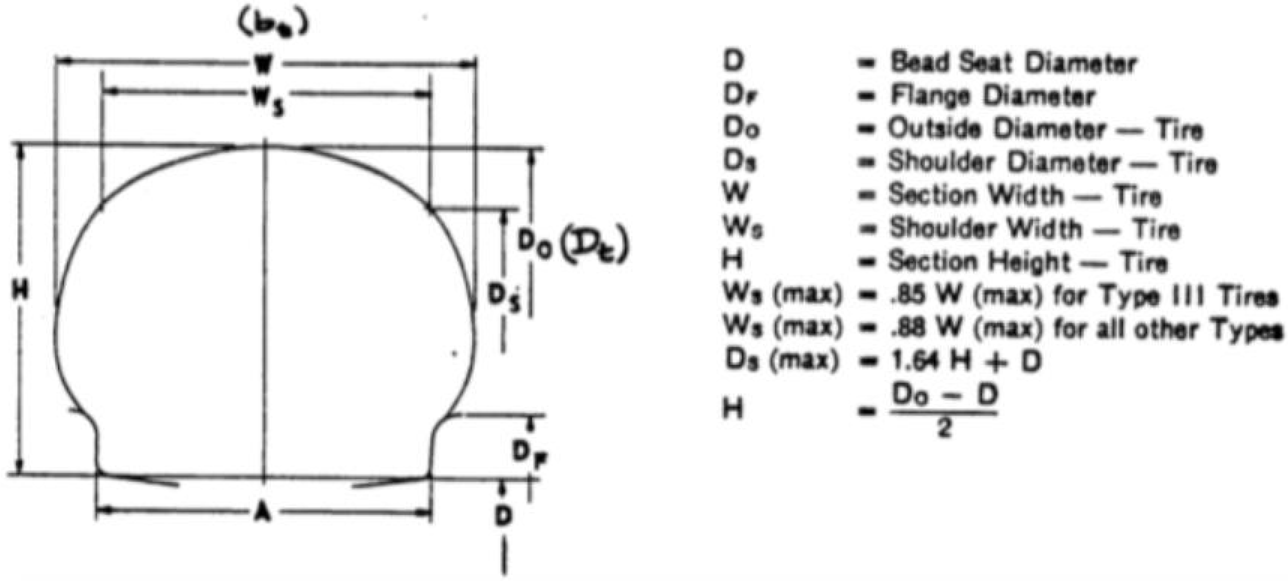


Figure 144: Tire Geometry parameters

Figure 144: Tire Geometry parameters

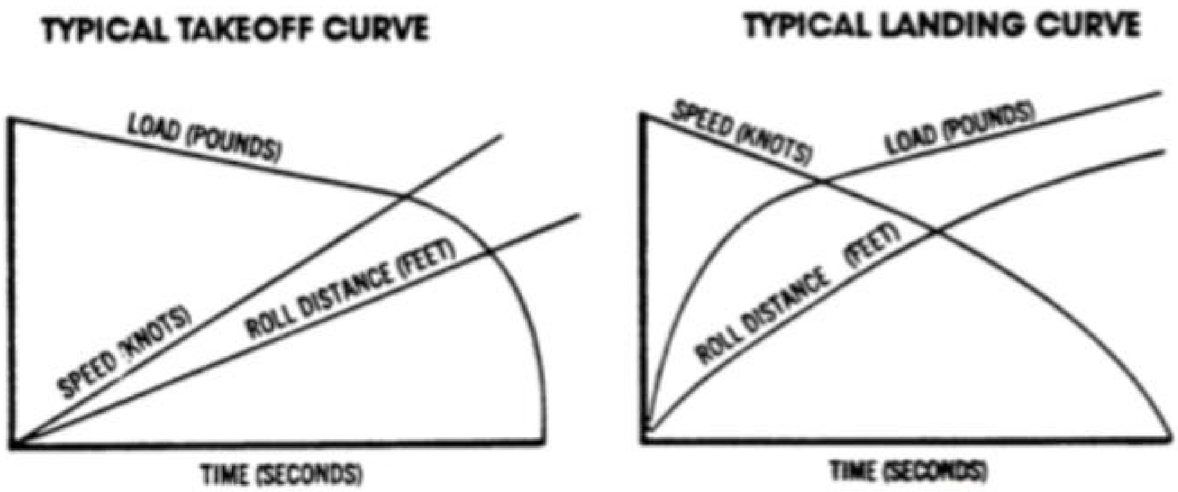


Figure 145: Tire Performance requirements while take-off and landing

11.7 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 during its service life; 4 percent in width and 10 percent in diameter. It grows under the influence of centrifugal forces that 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: 0.1Do + radial clearance due to centrifugal forces + 1 inch

11.7.1 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 TSBJ 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 = $\frac{135174.88}{4}$‰= 33793.7 lbs (6)

11.7.2 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 TSBJ 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 $= \frac{16284.2}{2} = 8142\ lbs$ (7)

The maximum static load per nose gear tire can be determined from:

$$P_{n\,dyn_t} = \frac{W_{TO}\left(l_m + \frac{a_x}{g(h_{cg})}\right)}{n_t(l_m + l_n)}$$

a_x can be obtained from the following:

$\frac{a_x}{g} = 0.35\ (for\ dry\ concrete\ with\ simple\ brakes)$

$\frac{a_x}{g} = 0.45\ (for\ dry\ concrete\ with\ anti-skid\ brakes)$

The TSBJ design has anti-skid brakes installed due to which ax/g = 0.45

Hence, the maximum dynamic load per nose gear tire is:

$$P_{n\,dyn_t} = 110000 * \frac{\left(2.761 + \frac{0.45}{10.32}\right)}{2(2.761+31.22)} = 4539.39\ lbs \quad (8)$$

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: $V_{tire_{max}} = 1.2\ V_{s_L} = 1.2 * 190 = 228\ knots$ (9)
For take-off: $V_{tire_{max}} = 1.1\ V_{s_{TO}} = 1.1 * 190 = 209\ knots$ (10)

11.8 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:

$$E_t = 0.5 * W_L * w_t^{\frac{2}{g}} \tag{11}$$

where, W_L = landing weight
w_t = design vertical touchdown rate

11.8.1 For the main landing gear:

$$E_t = n_s P_m N_g (\eta_t s_t + \eta_s s_s) \tag{12}$$

where, $W_L = n_s P_m$ (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.

$$s_s = \frac{\left[\left(\frac{0.5\left(\frac{W_L}{g}\right)(w_t)^2}{n_s P_m N_g}\right) - \eta_t s_t\right]}{\eta_s} \tag{14}$$

$$s_{s_{design}} = s_s + \frac{1}{12} \tag{15}$$

The diameter of the shock absorber (strut) may be estimated from:

$$d_s = 0.041 + 0.0025(P_m)^{\frac{1}{2}} \tag{16}$$

Element:	Energy Absorption Efficiency:
Tires:	$\eta_t = 0.47$
Shock absorbers:	
air springs	$\eta_s = 0.60$ 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

$$W_L = 2 * 67587.43 = 135174.879\ lbs \tag{17}$$

using equation 11, we get the maximum kinetic energy to be absorbed as,

$$E_t = 0.5 * W_L * (w_t)^{\frac{2}{g}} = 0.5 * 135174.879 * (17)^{\frac{2}{9.81}} = 120426.494\ lbs.ft \tag{18}$$

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,

$$s_s = \frac{\left[\left\{\frac{0.5\left(\frac{W_L}{g}\right)(w_t)^2}{n_s P_m N_g}\right\} - \eta_t s_t\right]}{\eta_-} = \frac{\left[\frac{\left\{0.5 * \frac{135174.879}{9.81} * (17)^2\right\}}{2 * 67587.43 * 2} - 0.47 * 38\right]}{0.8} = -13.12$$

$$s_s = -13.12 \tag{19}$$

$$E_t = n_s P_m N_g(\eta_t s_t + \eta_s s_s) = 2 * 67587.4394 * 2 * (0.47 * 38 + 0.8 * 13.12)$$

$$E_t = 1991108.05\ lbs \tag{20}$$

The diameter of the shock absorber (strut) can be obtained from the following equation,

$$d_s = 0.041 + 0.0025(P_m)^{\frac{1}{2}} = 0.69\ ft \tag{21}$$

11.8.2 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

$$s_S = \frac{\left[\left\{\frac{0.5\left(\frac{Pn}{g}\right)(w_t)^2}{n_S P_{ndyn_t} N_g}\right\} - \eta_t s_t\right]}{\eta_s} = \frac{\left[\left\{\frac{\left(0.5*\left(\frac{16284.19}{9.81}\right)*(17)^2\right)}{1*9078.79*2}\right\} - 0.47*22\right]}{0.8}$$

$$s_S = 3.58 \qquad (22)$$

$$d_s = 0.041 + 0.0025\left(2 * P_{ndyn_t}\right)^{\frac{1}{2}} = 0.28\,ft \qquad (23)$$

11.9 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 and this limitation must be accounted for in the design of the wheel and tire.

The rolling friction generated between the rolling tires and the runway 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 follows:

$$Slip\ Ratio = \left\{1 - \frac{Wheel\ RPM\ brakes\ on}{Wheel\ RPM\ brakes\ off}\right\}$$

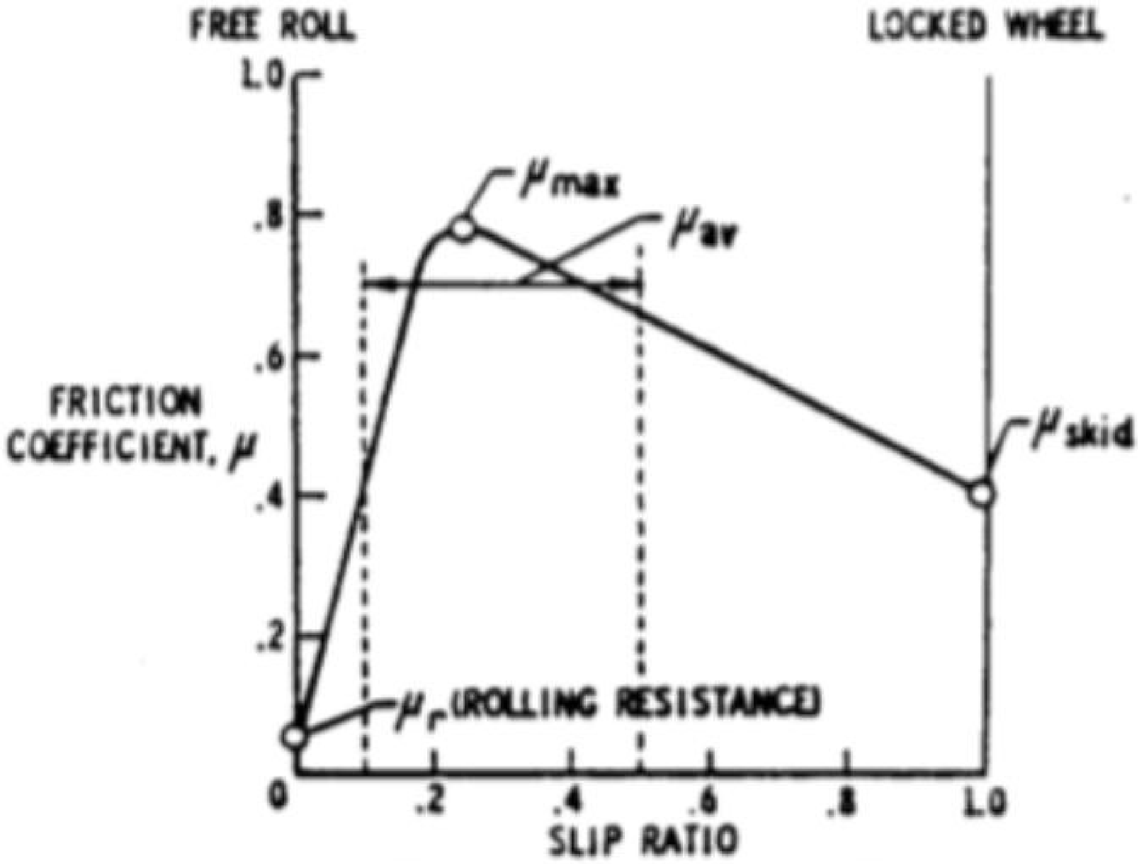


Figure 148: Effect of slip ratio on ground friction coefficient

The zero-slip ratio 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.). On using an anti-skid system is used to control wheel RPM during braking; the average value of friction coefficient that can be attained is about 0.70.

Since 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 and 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 12: FIXED EQUIPMENT LAYOUTS

12.1 INTRODUCTION

The main objective of this report is to define the fixed equipment's attached to the TSBJ and study in detail their configuration and location on the design of the airplane. As per the weight requirements obtained in chapter 13, the layouts of the systems for the TSBJ design are developed and described in the report.

The Equipment's attached to the TSBJ 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

12.2 FLIGHT CONTROL SYSTEM

Flight control system is divided into following two sections:

1. Primary Flight Control system
2. Secondary Flight Control System

The primary flight controls are:

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

12.2.1 REVERSIBLE FLIGHT CONTROL SYSTEM

The reversible flight control system is generally mechanized with cables, push-rods or a combination of both. In a reversible flight control system, every movement made in the cockpit controls changes the positions of the aerodynamic controls and vice-versa.
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

12.2.2 IRREVERSIBLE FLIGHT CONTROL SYSTEM

The irreversible flight control system is hydraulic and/or electrical and this makes it an irreversible process. In an irreversible flight control system, every movement made in the cockpit controls changes the aerodynamic control surfaces and not vice-versa. In this system, the aerodynamic control surfaces are moved by the actuator.
Below are major design problems associated with the irreversible flight control system:

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)

12.2.3 DESIGN FLIGHT CONTROL SYSTEM

After considering the flight control systems, design and specification of the airplane; the irreversible flight control system was the best-fit and therefore, will be used to design the flight control system in TSBJ.

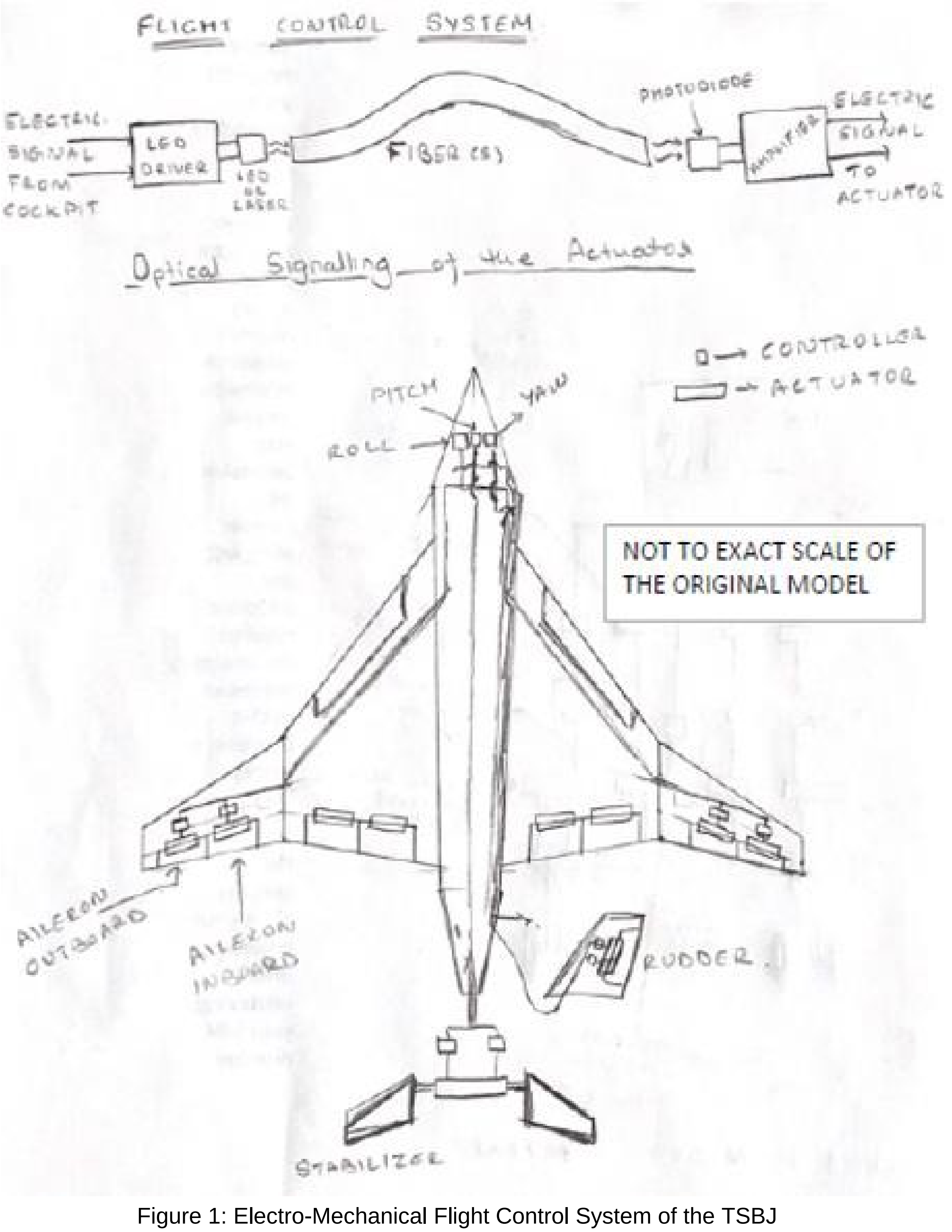


Figure 1: Electro-Mechanical Flight Control System of the TSBJ

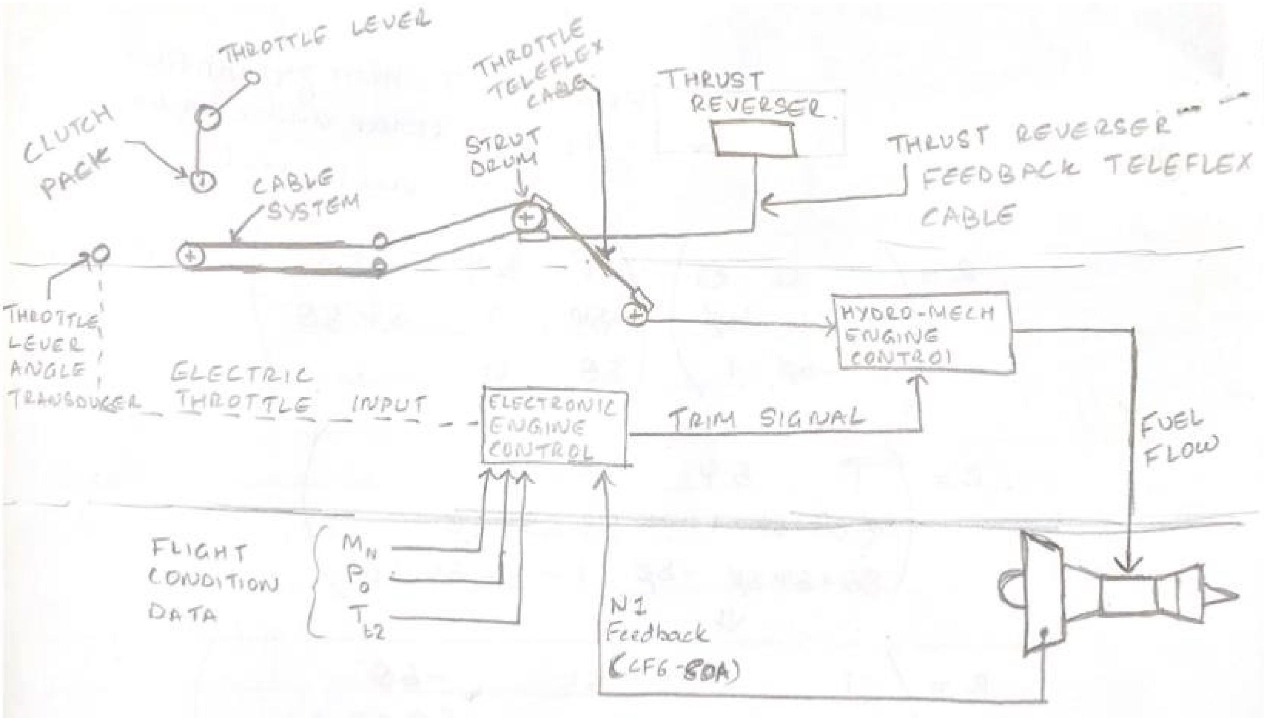


Figure 2: Thrust Control System for the TSBJ

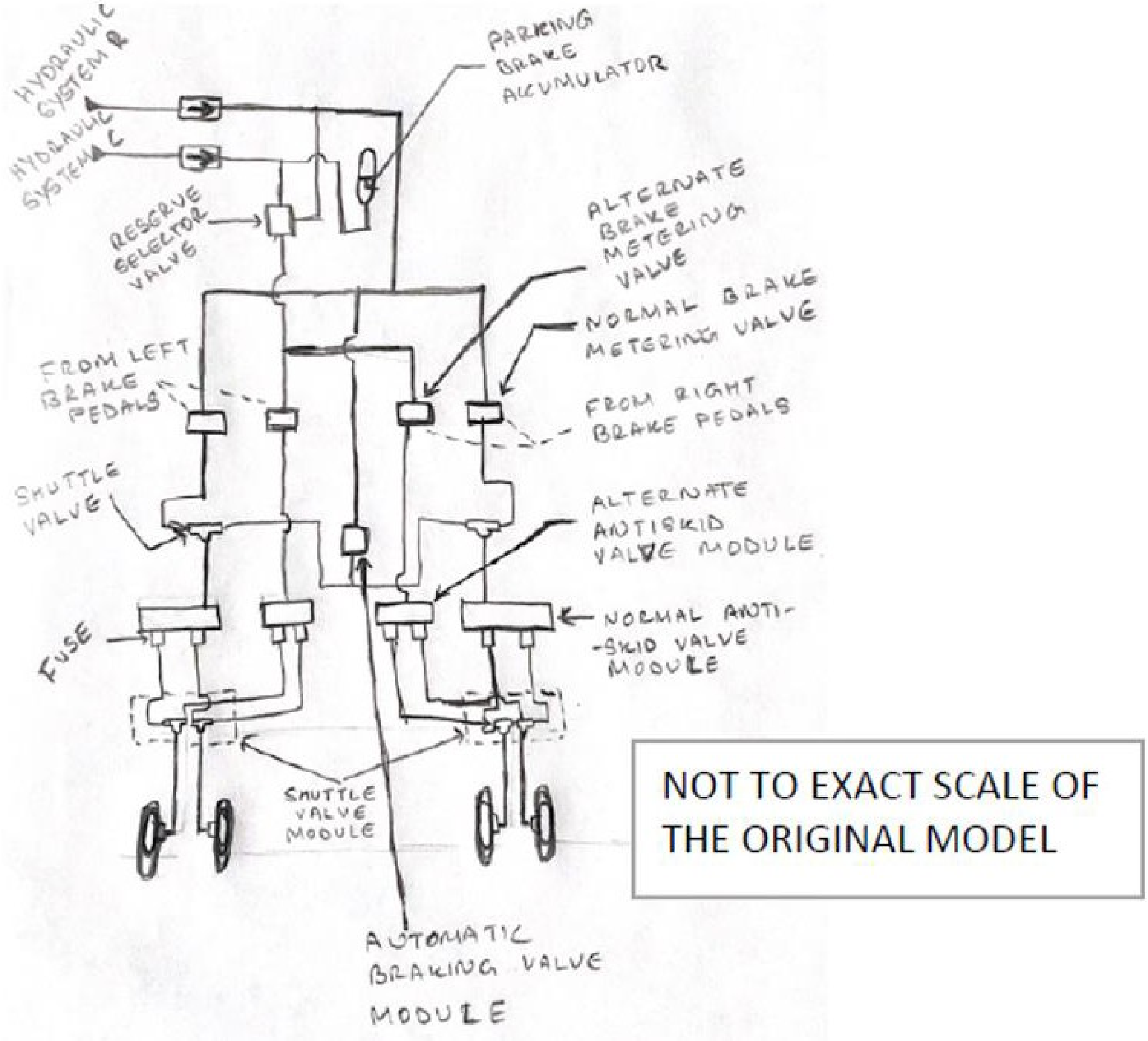


Figure 3: TSBJ Main Landing Gear Control System

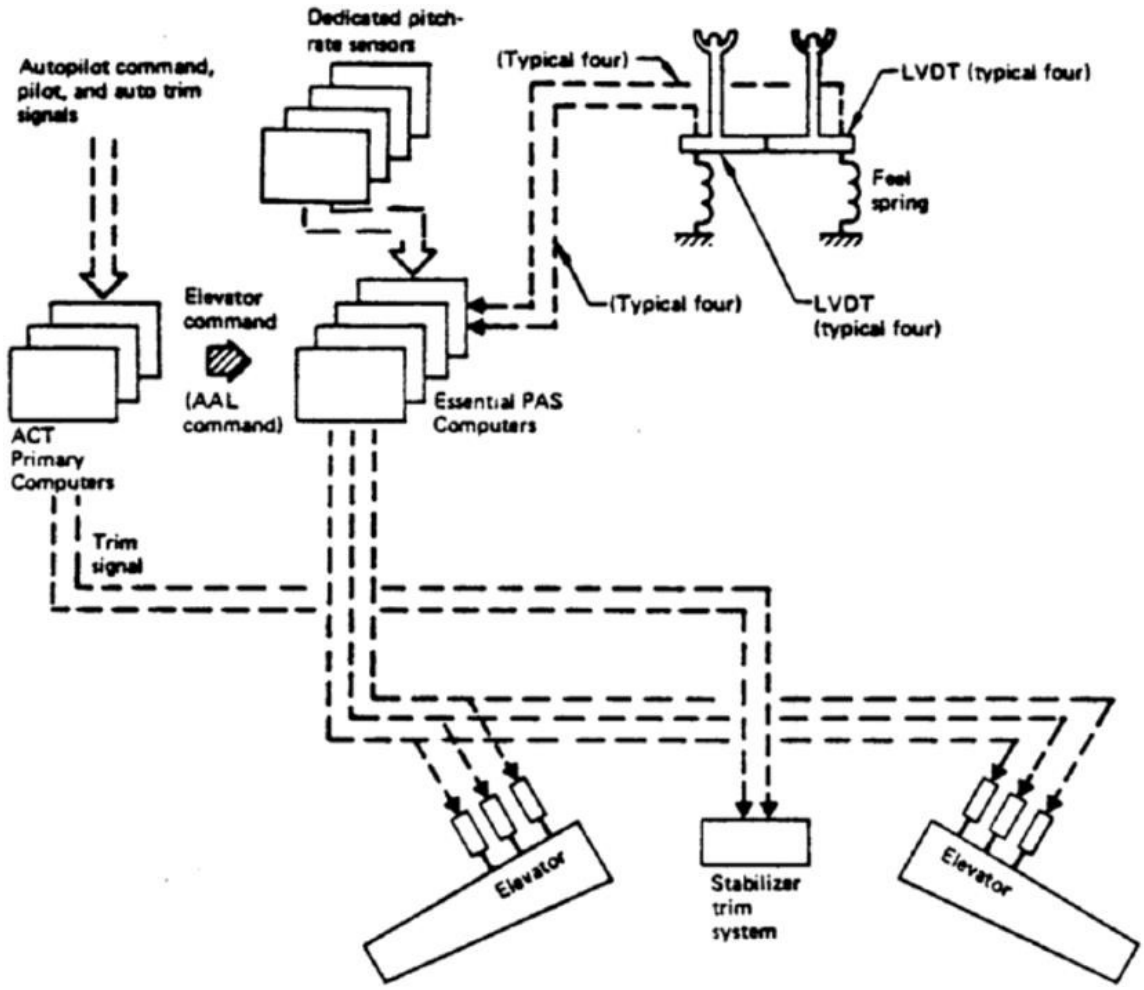


Figure 4: Fly-by-wire Hydraulic Elevator control System of the TSBJ

12.3 FUEL SYSTEM

This section of the report specifies and discusses the fundamental principles for the fuel system layout design. As airplane fuels are very combustible liquids; the design, operation, location, accessibility and maintenance aspects are important for the aircraft safety and economy.

Most fuel systems need the following components to operate properly:

I. Fuel tanks that can carry a enough 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 and to 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.

12.3.1 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).

$$Max. Fuel\ Flow = T_{TO}(c_j)$$

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.

12.3.2 GUIDELINES FOR FUEL SYSTEM LAYOUT DESIGN

Airplane's fuel systems should be equipped with fuel vents and fuel sump systems. The fuel vent system prevents excessive pressure from building up in the fuel tanks and also serves to maintain the ram-air pressure in the tanks during the flight. And, the surge tanks are used to collect and condense any excess fuel vapor before it exits through the overboard fuel vents.

The following points 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.

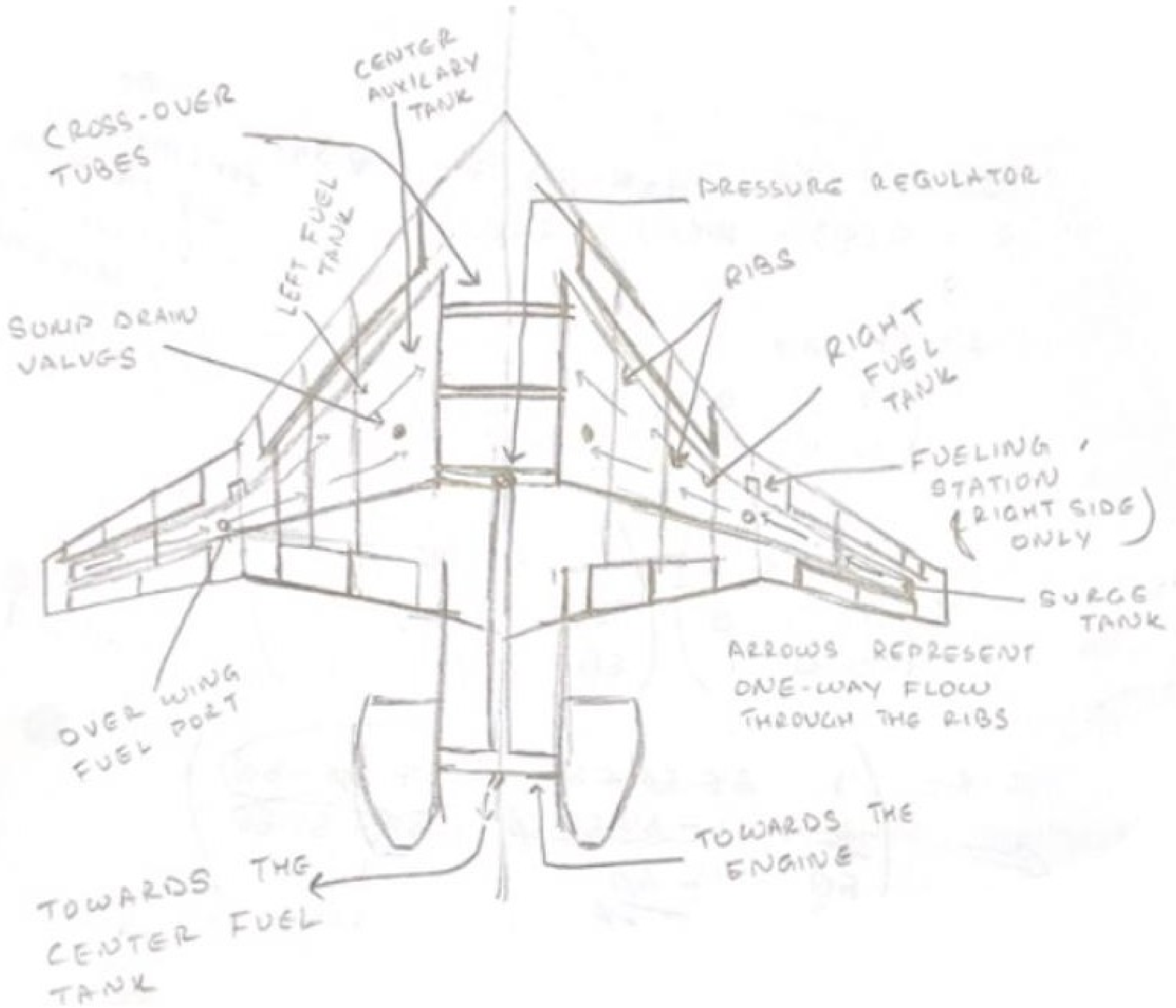


Figure: TSBJ Fuel System

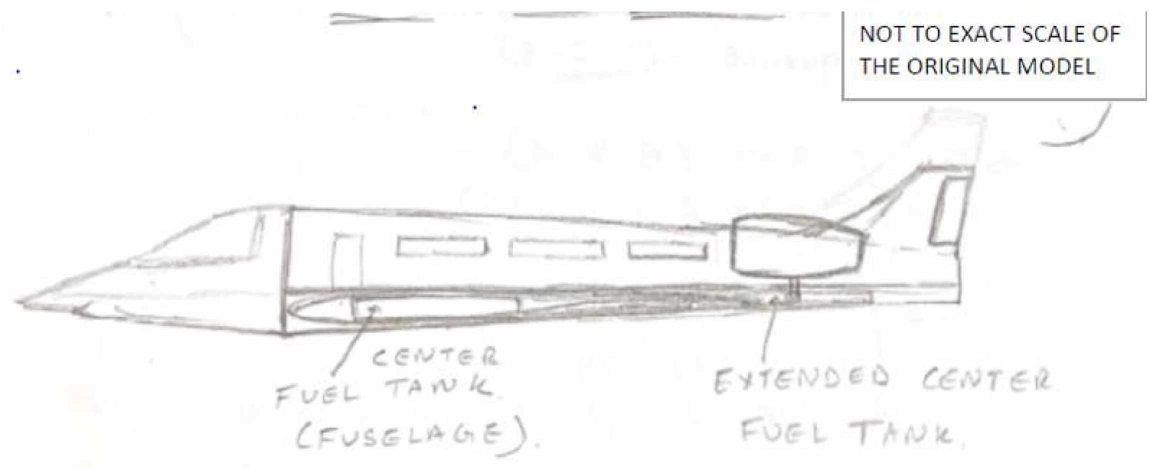


Figure 5: TSBJ Fuel System Layout

12.4 HYDRAULIC SYSTEM

12.4.1 FUNCTIONS OF HYDRAULIC SYSTEMS

The functions of a hydraulic system vary with the aircraft. The main 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.

Following components make up the hydraulic system:

- 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.

12.4.2 SIZING OF HYDRAULIC SYSTEMS

The landing phase needs to operate both the primary and secondary flight controls, landing gears and speed brakes and during this, the maximum amount of hydraulic fluid flow is required.

By creating a list of actuator rate and force requirements from which the gallons per minute flow requirements is obtained, the total fluid flow requirements can be easily calculated.

The power requirements for the hydraulic system for the TSBJ are usually up to the range of 700 hp.

12.4.3 GUIDELINES OF HYDRAULIC SYSTEM DESIGN

Below are the guidelines for designing the 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.

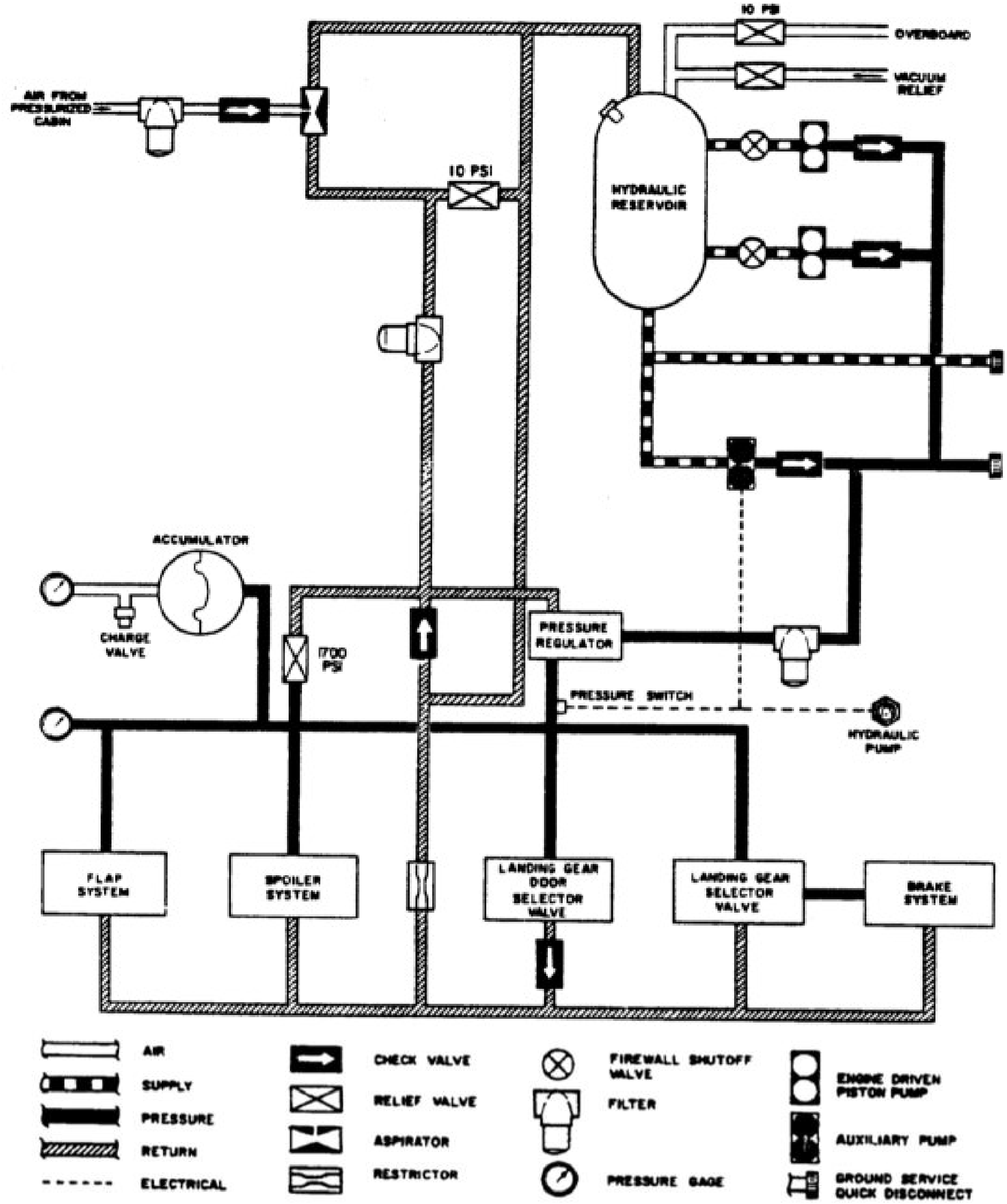


Figure 6: Hydraulic Reservoir of the TSBJ

12.5 ELECTRICAL SYSTEM

Electric power is required to operate various systems in an airplane and for transonic TSBJ airplane, electric power will be used to perform the following operations:

• Internal and External lighting.
• Flight instruments and avionics systems.
• Food and beverages heating system.
• Engine starting system.
• Flight control system (primary and secondary)

Below are the two main systems that generate the electrical power:

1. Primary power generating system: engine driven generators
2. Secondary power generating system: Battery, APU, RAT (Ram-air Turbine)

12.5.1 MAJOR COMPONENTS OF THE ELECTRICAL SYSTEMS

Engine driven generators or the alternators generate the electricity usually for the operation and are 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.

12.5.2 SIZING OF ELECTRICAL SYSTEMS

Construction of an electric power load profile is important to determine the electrical power requirements of the aircraft. These requirements should be determined 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 below-mentioned points 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.

12.5.3 TSBJ ELECTRICAL SYSTEM LAYOUT

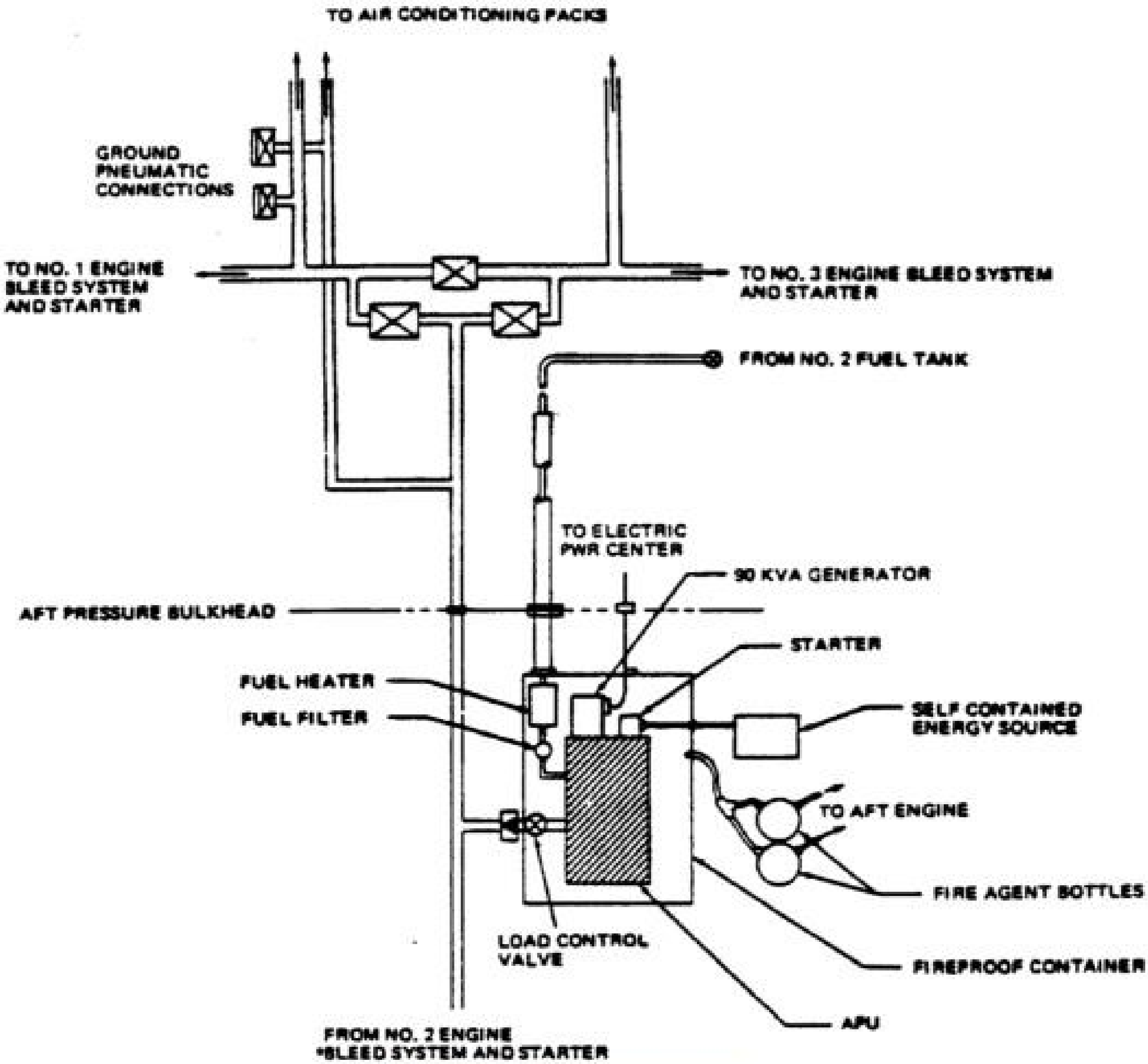


Figure 7: APU electrical system layout

12.6 ENVIRONMENTAL CONTROL SYSTEM
12.6.1 PRESSURIZATION SYSTEM

The main purpose of the pressurization system is to maintain sufficient cabin air pressure at higher altitudes during the flight in order to keep the passengers comfortable. Typical differential pressure capabilities in jet transports are designed to maintain a cabin altitude from 1000ft below sea level to 10,000ft 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 holds a great importance for the airplanes as a failure in it makes it difficult for the passengers to breathe. And, if it fails during landing, then there is great possibility that the cabin doors will get open.

If the cargo doors located below the cabin floor accidentally blow out causing the pressure in the cargo bleed off rapidly, major problems can arrive. The pressure difference then created causes the cabin floor to fail.

All these problems 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.

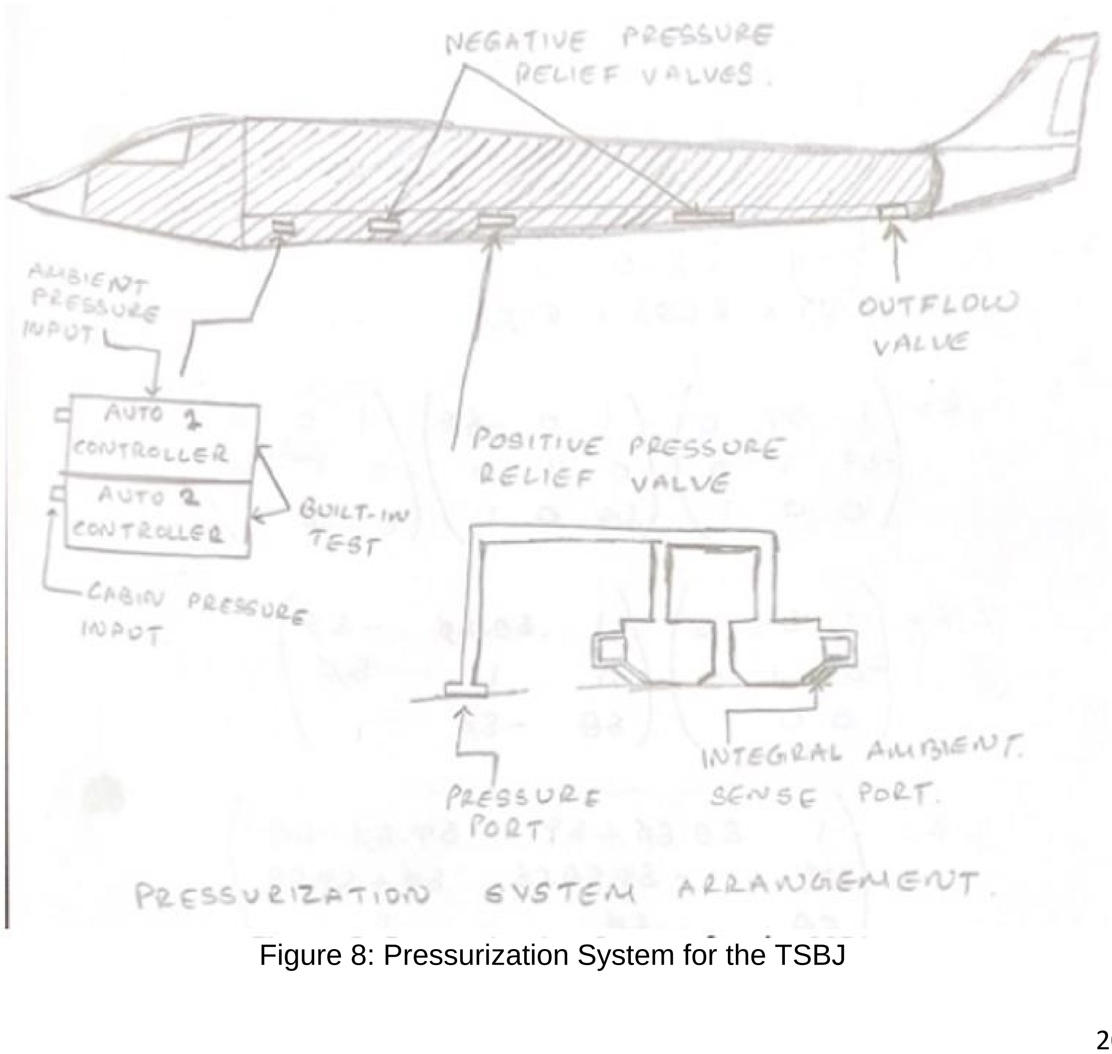


Figure 8: Pressurization System for the TSBJ

12.6.2 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

12.6.3 AIR-CONDITIONING SYSTEM

The air-conditioning system is used to condition the air in terms of temperature and humidity. The air coming from the air conditioning system must be evenly distributed into the cabin. The overall efficiency of the cabin air conditioning system depends a great deal on the thermal insulation of the cabin walls.

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 to avoid the noise produced by the improper design.

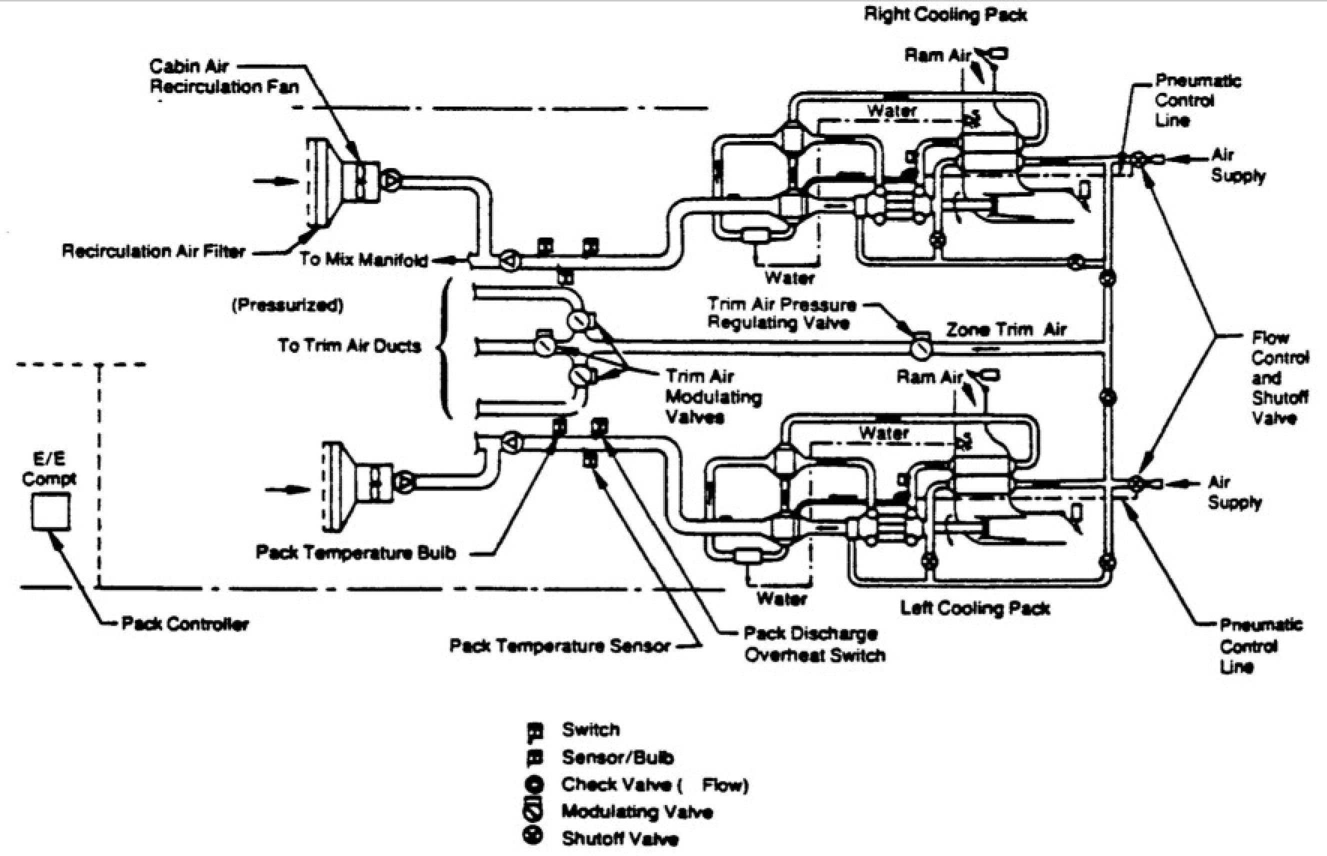


Figure 9: Air Conditioning System

12.6.4 OXYGEN SYSTEM

Oxygen is required at higher altitudes after the failure of the cabin pressurization system. The oxygen system's use usually gaseous oxygen or chemically obtained oxygen. Oxygen for the crew members is normally supplied from a gaseous source whereas the passenger oxygen is supplied form a chemical source.

Gaseous oxygen comes with a disadvantage and that is it presents a fire hazard during servicing and during cylinder replacement and main disadvantage being its larger weight.

12.7 COCKPIT INSTRUMENTATION, FLIGHT MANAGEMENT AND AVIONICS SYSTEM

12.7.1 COCKPIT INSTRUMENTATION LAYOUT

The layout of the cockpit instrumentation system should be uncluttered and functional. It should enable the crew to see all flight crucial instruments, controls and warning devices.

Due the advancements made in the cockpit instrumentation and airplane avionics, changes occur almost each year.

A typical cockpit instrumentation panel layout is described in the figure below:

12.7.2 FLIGHT MANAGEMENT AND AVIONICS SYSTEM LAYOUT

Flight management and avionics systems are undergoing very rapid development and in the recently developed aircrafts, the pilot is able to interface 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

12.7.3 ANTENNA SYSTEM LAYOUT

Many antenna systems are required to enable the communication between the aircraft and the ground.
An example of the antenna systems installed on an aircraft is described as below.

12.7.4 INSTALLATION, MAINTENANCE AND SERVICING CONSIDERATIONS

Many avionics equipment's in an aircraft consume a considerable amount of electric power and 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 random.

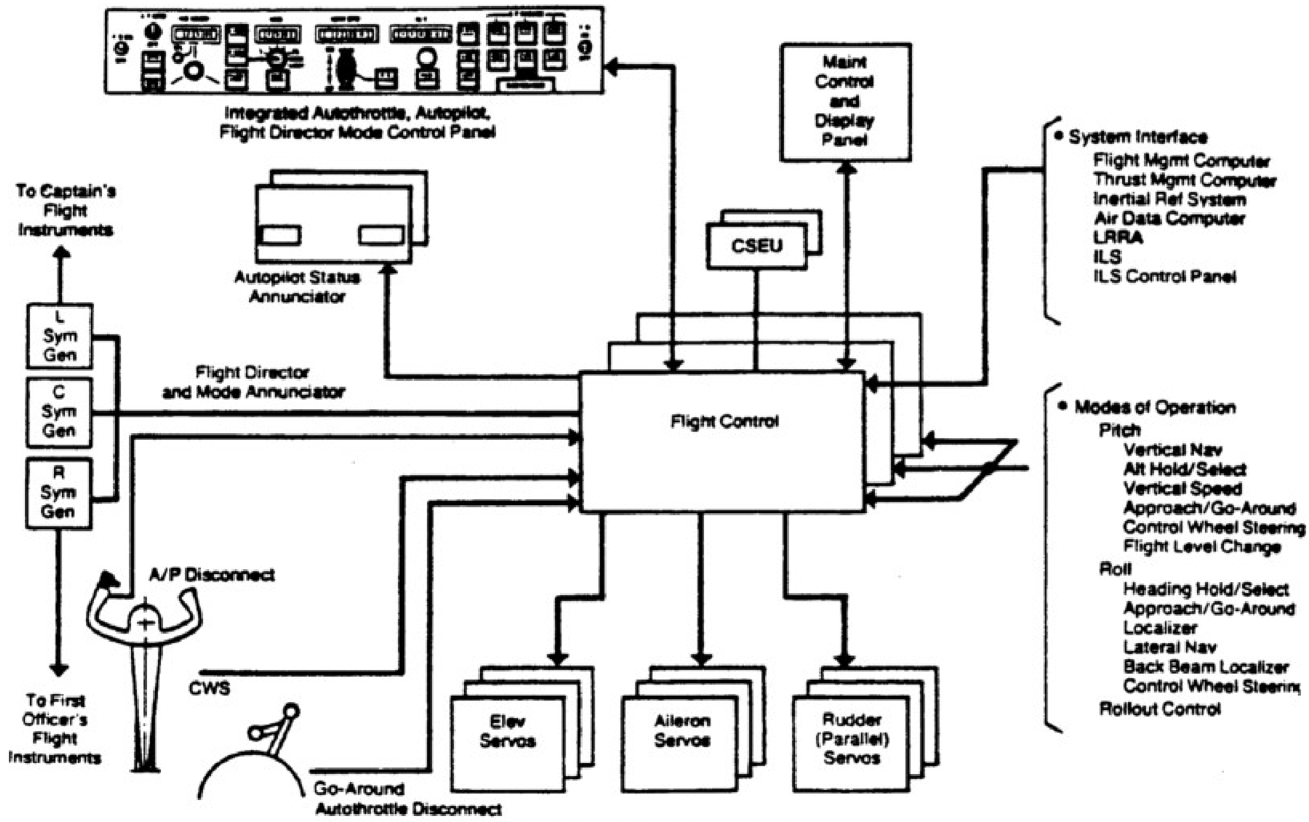


Figure 10: Flight Control Computer Functions

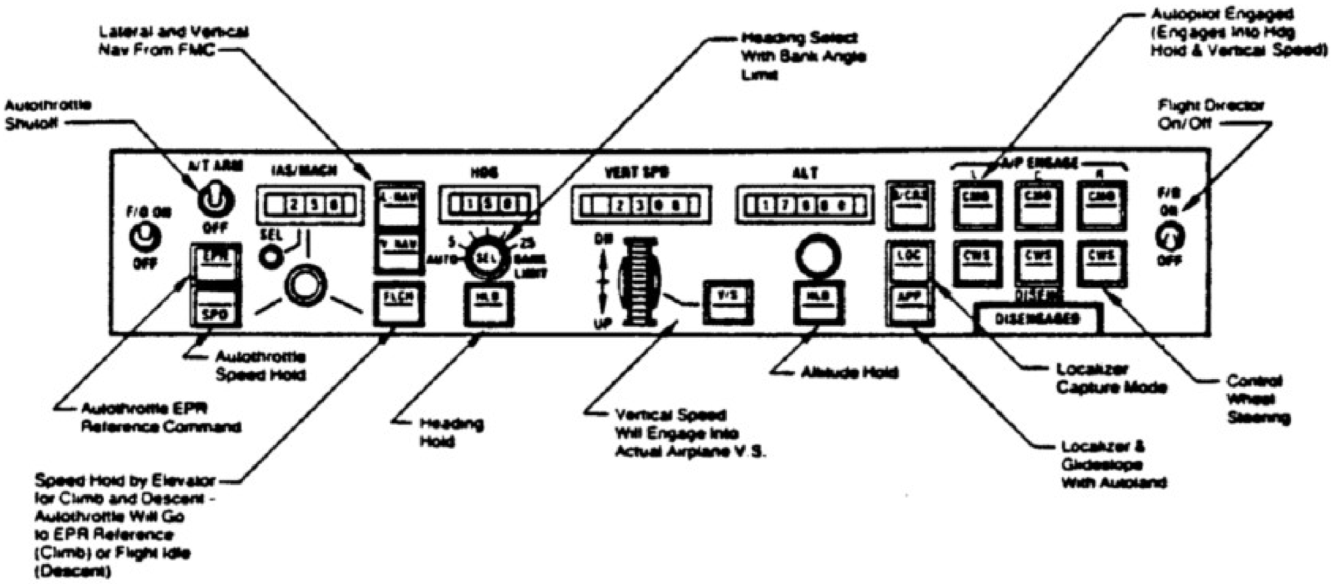


Figure 11: Autopilot/ Auto throttle Control Panel

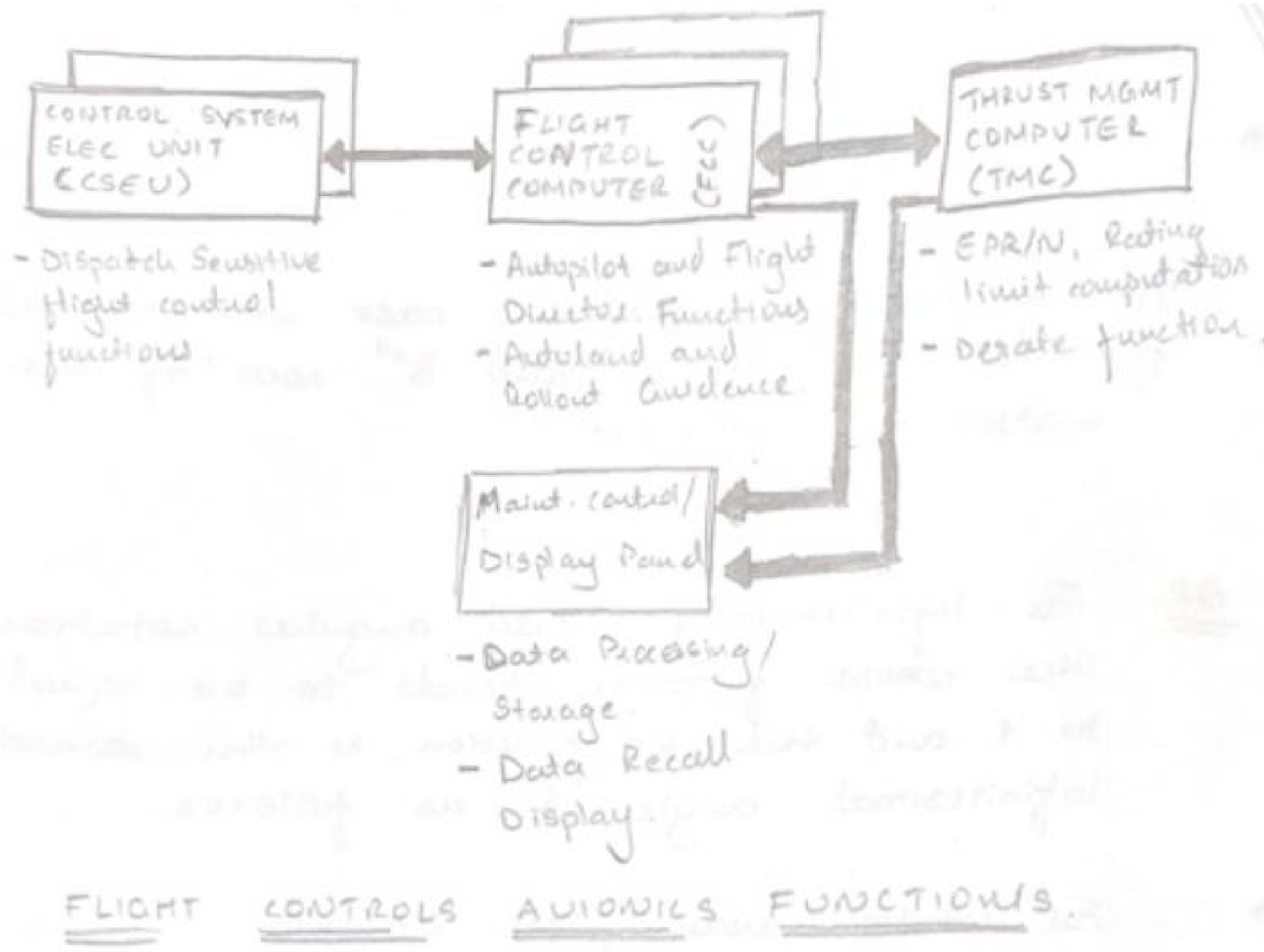


Figure 12: Flight Controls Avionics Functions

12.8 DE-ICING, ANTI-ICING, RAIN REMOVAL AND DEFOG SYSTEM

For an airplane to be operated under icing conditions, certain special systems must be installed to prevent or remove the ice from the surface of the airplane.

Also, when the airplane is flying under rainy conditions, a rain removal system must also be installed in the airplane to remove the water accumulating on top of the wind shields which causes a serious visibility problem.
Under certain combinations of humidity and temperature fog tends to form on the windshields which also affect the visibility. Therefore, to avoid this, a defog system must be made available for the safety of the flight.

12.8.1 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 ofthe 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.

12.8.1.1 DE-ICING SYSTEMS

2 types of the de-icing systems are used, and these are:

- the De-icing boots and
- the Electro-impulse systems.

As the engines are attached at the rear of the fuselage of an TSBJ, the Electro-impulse system will be used. The De-icing boots if installed 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.

12.8.1.2 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 need to be turned on immediately on realizing that the 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

After considering all the above anti-icing systems available for an aircraft, we concluded that it safe to move forward with the thermal anti-icing system for the Transonic 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 transonic 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 if the ice is formed. The thermal anti-icing systems are sometimes also used to de-ice the surface of the aircraft. Therefore, it is suggested to avoid the formation of ice on the surface rather than removing it later 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. Electrical resistances are used to heat the surfaces where there are high chances of the ice to be formed in electrically heated systems. 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 Transonic Business Jet is as described below:

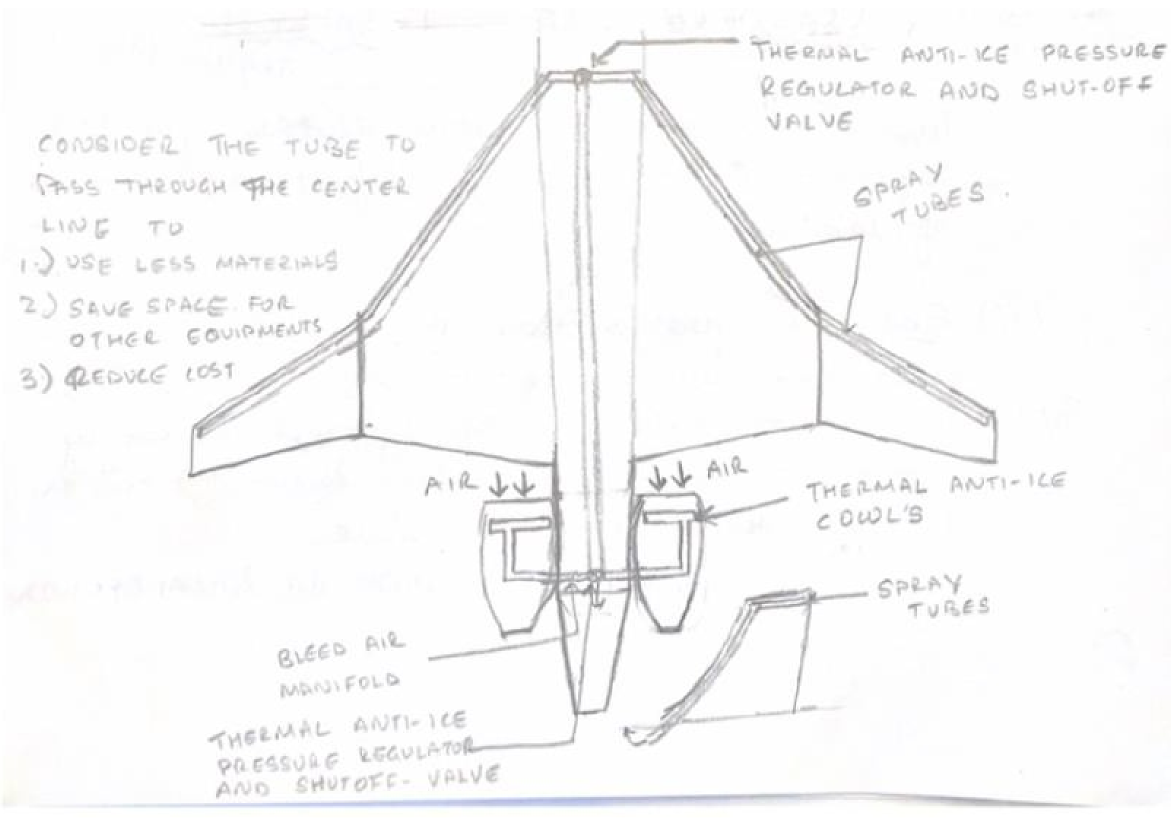


Figure 13: Air Heated Anti-Icing/De-Icing system for the TSBJ

The defog system is similar to the anti-icing / de-icing system which uses hot engine air to defog the windshields.

12.8.2 RAIN REMOVAL AND DEFOG SYSTEMS

It is important to install the rain removal and Defog systems in the airplane as they help to obtain proper visibility during rains and temperature differences which cause fog on the windshields of the airplane.

The rain removal system usually uses the 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. It can be achieved by installing electrical wirings inside the windshield material.

The Rain removal and defog system to be used in the TSBJ are as follows:

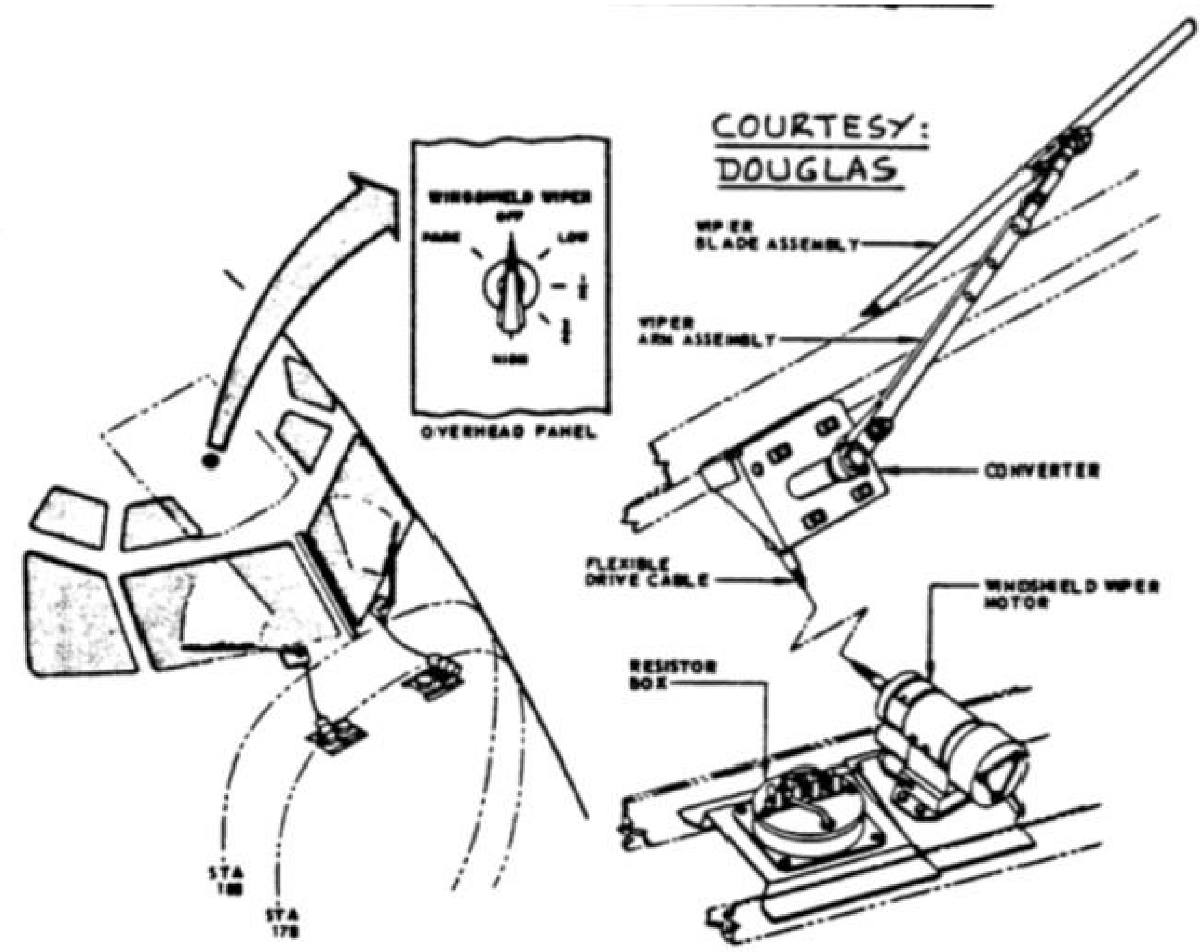


Figure 14: Rain removal system using wipers similar to those in cars.

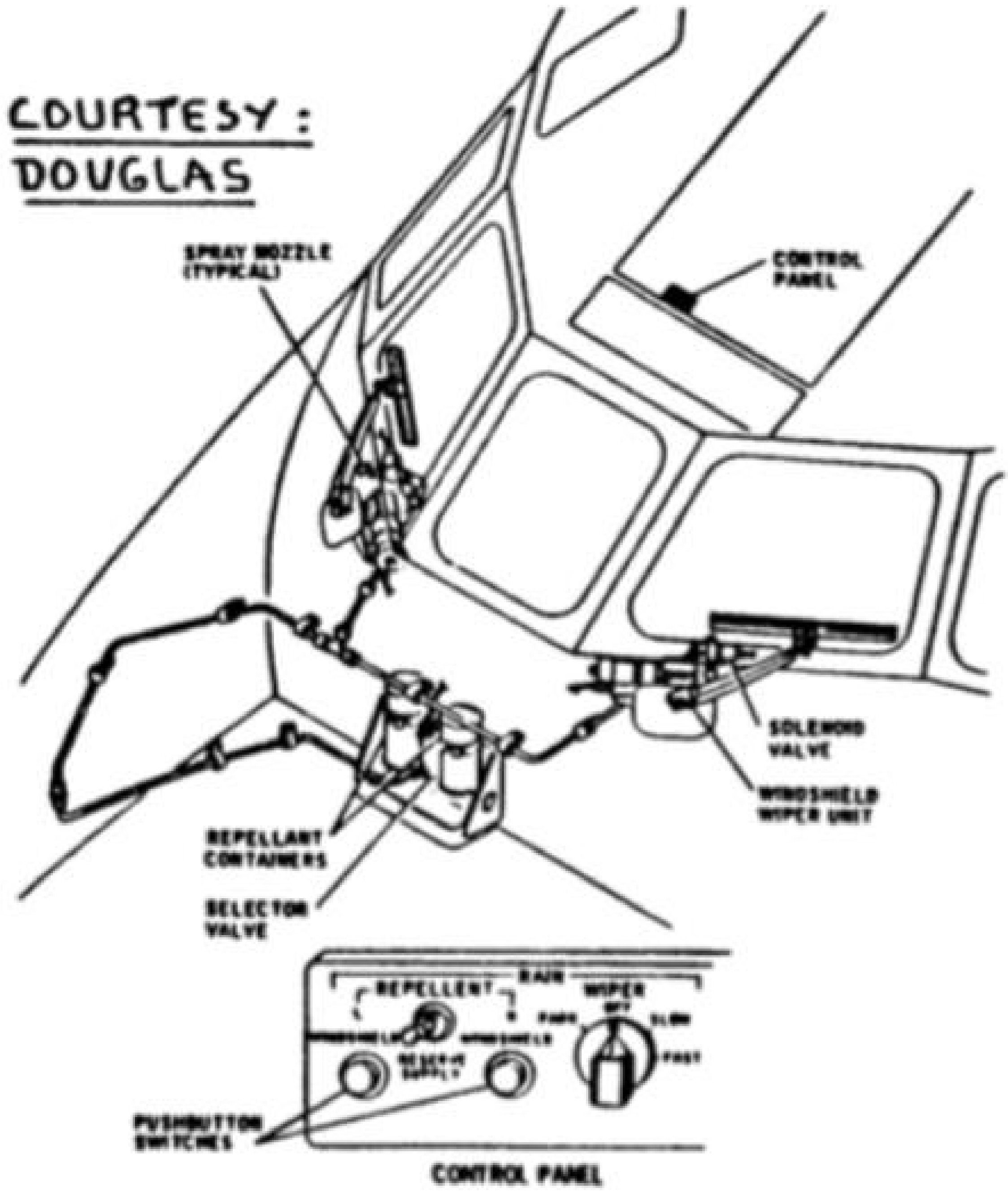


Figure 15: Rain repellant system added to the wiper paths

12.9 ESCAPE SYSTEM

The purpose of installing escape system in the airplane is to provide emergency exits to the passengers for their safety in the case of emergency situations. According to the FAR-25 requirements, all airplanes must have emergency exits for the safety of the passengers. The aircraft should have the number of exit routes required according to its size and the entire exit routes should be marked properly and should also indicated by the 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 the reach of the passengers. Therefore, theyare installed under the seats of the passengers. For over water flights (for more than 30 minutes), the aircrafts must carry emergency rafts. Also, there should be sufficient space to carry the passengers as well as the crew members.

12.10 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 and therefore, important to consider these in the preliminary design.

The water systems are typically sized as 0.3 gallons per passenger and are usually pressurized with air from the airplane pneumatic system. These systems contain drain masts which must be heated to avoid them from freezing.

A major concern while designing is the location of the drain masts. Large blobs of ice may be formed when it is not heated and if in case these blobs of ice break, they should not be ingested by the engines.

In some flights, both hot and cold water are 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 thewaste 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 service of both the water and waste systems should be considered during the preliminary design of the aircraft to avoid leakage as the water/waste leakage can form ice. And, this ice on breaking should not be ingested by the engines of the aircraft. For designing the TSBJ, 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. This will also help in maintaining the overall weight of the design.

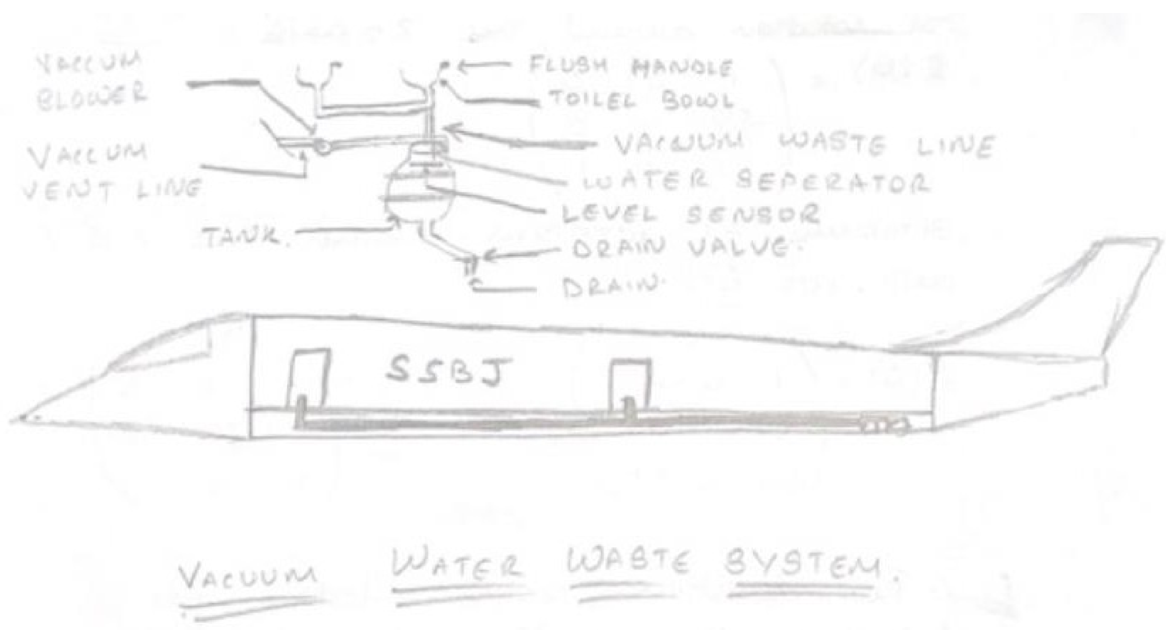


Figure 16: Vacuum Water System for the TSBJ

12.11 SAFETY AND SURVIVABILITY

The objective of this chapter is to provide details on the design insight of safety and survivability considerations. Designing the airplanes with the probability of fatalities is zero is impossible. 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. The design of both should be that any occurrence of failure would prevent and continue the safe flight. 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 17: Factors affecting the safety of Aviation

Following two types of accidents are considered in case of airplanes:

- Predominantly Airworthiness
- Predominantly operation

More than half of the airplane accidents are caused due to human factor. More accidents occur due to the errors made by the flight crew compared to the errors made by the ground crew.
While 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 inspect ability 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.

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 13: CLASS II AIRPLANE WEIGHT COMPONENTS/VN-Diagram

13.1 INTRODUCTION

To calculate the Class II weight of the TSBJ, the following data is required:

i. Airplane take-off gross weight

ii. Wing and empennage design parameters such as:

- Area

- Sweep angle

- Taper ratio, l

- Thickness ratio, t/c

iii. Load factor, nœîú or náUF

iv. Design cruise and dive speed, VK or V=

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 TSBJ described in the previous reports.

Some basic weight definitions useful as obtained in class 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,

$$W_E = W_{strut} + W_{pwr} + W_{feq} \quad (2)$$

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: $W_{PL} = 4220\ lbs$

Crew Weight: $W_{crew} = 700\ lbs$

Fuel weight: $W_{fuel} = 53671.33\ lbs$

Trapped fuel and oil: $W_{tfo} = 1073.4\ lbs$

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_{s_1}= +1-g stall speed or the minimum steady flight speed which can be obtained.

V_C= Design cruising speed

V_D= Design diving speed

V_A= Design maneuvering speed

V_B= Design speed for maximum gust intensity

13.1.1 Calculating the Stall Speed (V_S):

$$V_{S_{pos}} = \left\{\frac{2*\left(\frac{GW}{S}\right)}{\rho * C_{N_{max}}}\right\}^{\frac{1}{2}} \quad (3)$$

GW = Design Gross Weight
S = Wing Area, ft^2
ρ = air density in slugs, ft^3
$C_{N_{max}}$ = maximum normal force coefficient

The maximum normal force coefficient can be obtained from the equation,

$$C_{N_{max}} = \left\{\left(C_{L_{max}}\right)^2 + \left(C_{D_{at\,C_{L_{max}}}}\right)^2\right\}^{\frac{1}{2}} \quad (4)$$

$$C_{N_{max}} = \ 2.86 \quad (5)$$

therefore, we can get the stall speed with the help of the maximum normal force coefficient

$$V_{S_{pos}} = \left\{\frac{2*\left(\frac{110000}{1424.34}\right)}{0.0023769*1.8075}\right\}^{\frac{1}{2}} = 420.45\ knots \quad (6)$$

the negative stall speed line can be determined from the following equation:

$$V_{S_{neg}} = \left\{ \frac{2\left(\frac{GW}{S}\right)}{\rho C_{N_{max_{neg}}}} \right\}^{\frac{1}{2}} \tag{7}$$

$$C_{N_{max_{neg}}} = \left\{ \left(C_{L_{max_{neg}}} \right)^2 + \left(C_{D_{at\, C_{L_{max_{neg}}}}} \right)^2 \right\}^{\frac{1}{2}} \tag{8}$$

in equation 8, $C_{L_{max_{neg}}} = -1.0$ will be assumed

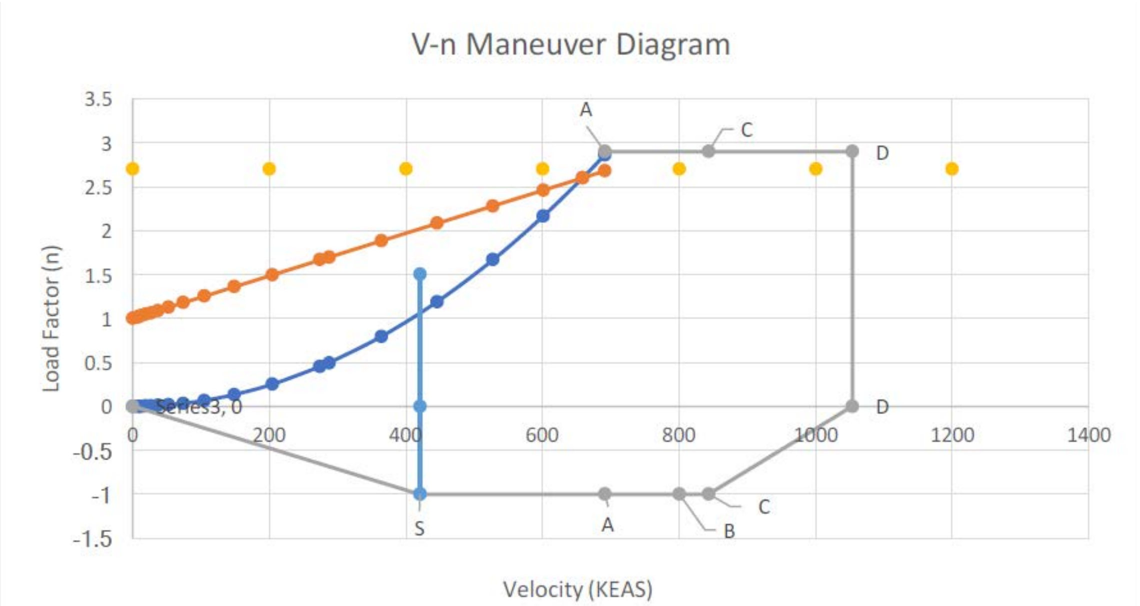
substituting that in Equation 8, we get

$$C_{N_{max_{neg}}} = 1.00279735 \tag{9}$$

hence, $V_{S_{neg}}$ can be obtained as,

$$V_{S_{neg}} = \left\{ \frac{\left(2*\left(\frac{110000}{1424.34}\right)\right)}{0.0023769*1.00279735} \right\}^{\frac{1}{2}}$$

$$V_{S_{neg}} = 704.9\ knots \tag{10}$$



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 $C_{N_{max}}$ line and the gust line marked V_B.

The V_B marked gust line intersects the $C_{N_{max}}$ line at around 210 knots

$$V_B = 800\ knots \qquad (11)$$

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.

$$V_C \geq V_B + 43\ kts \qquad (12)$$

$$V_C \geq 800 + 43 = 843\ knots \qquad (13)$$

13.2.3 Calculating the design diving speed, V_D

$$V_D \geq 1.25\ V_C \qquad (14)$$

$$V_D \geq 1.25 * 843 = 1053.75\ knots$$

13.2.4 Calculating the design maneuvering speed, V_A

$$V_A(+ve) \geq V_S\ (n_{\lim})^{\left(\frac{1}{2}\right)} \tag{15}$$

where, n_{lim} is the limit maneuvering load factor at V_C.

$$V_A(+ve) \geq 420.45\ * 2.3^{\frac{1}{2}}$$

$$V_A(+ve) \geq 690.87\ knots \tag{16}$$

$$V_A(-ve) \geq\ 704.90 * -1^{\frac{1}{2}}$$

$$V_A(-ve) \geq\ 704.90\ knots \tag{17}$$

13.2.5 Calculating the design limit load factor, n_{lim}.

$$n_{\lim_{pos}} \geq 2.5 + \left[\frac{24{,}000}{W+10{,}000}\right] \tag{18}$$

$$n_{lim_{pos}} \geq 2.5 + \left[\frac{24{,}000}{110{,}000+10{,}000}\right]$$

$$n_{lim_{pos}} \geq 2.7 \tag{19}$$

the negative, design limit load factor is determined from:

$$n_{lim_{neg}} \geq\ -1.0\ up\ to\ V_C \tag{20}$$

$$n_{lim_{neg}}\ varies\ linearly\ from\ the\ value\ at\ V_C\ to\ zero\ at\ V_D$$

13.2.6 Construction of gust load factor lines

For the gust line marked V_B:

$$U_{de} = 84.67 - 0.000933h = 84.67 - 0.000933(50{,}000) =\ 34.288 \tag{21}$$

For the gust line marked V_C:

$$U_{de} = 66.67 - 0.000833h = 66.67 - 0.000833(54{,}000) =\ 21.688 \tag{22}$$

For the gust line marked V_D:

For the gust line marked V_D:

$$U_{de} = 33.34 - 0.000417h = 33.34 - 0.000417(54{,}000) = 10.822 \qquad (23)$$

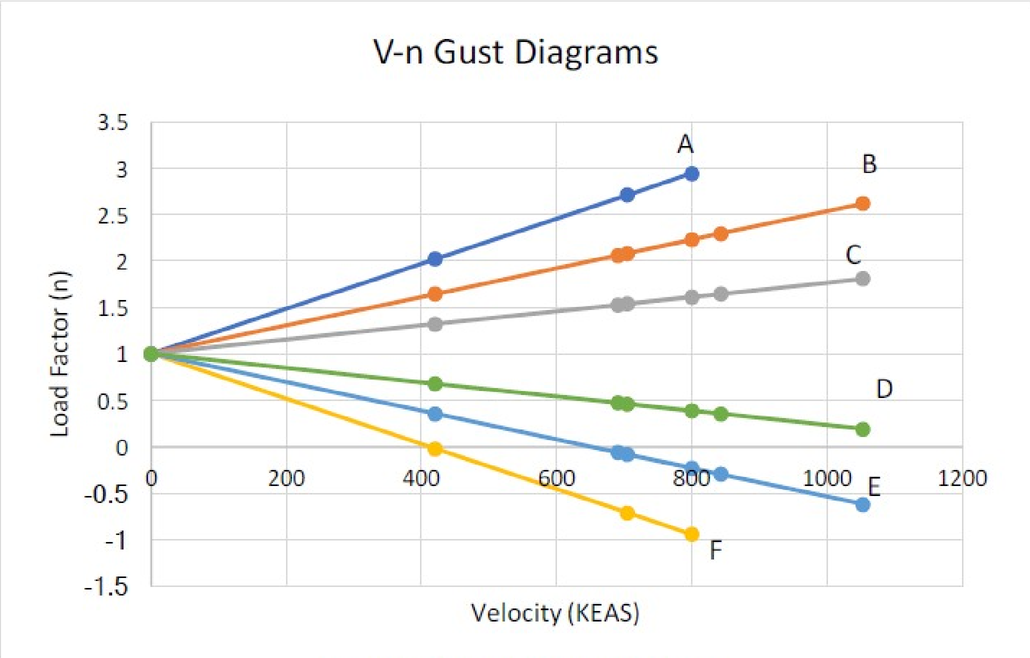


Figure 150: V-n Gust Diagram (in Knots)

13.3 METHODS FOR ESTIMATING THE STRUCTURE WEIGHTS

The airplane structure weight, W_{struct} consists of the following component weights:

i. Wing, W_W
ii. Empennage, W_{emp}
iii. Fuselage, W_f
iv. Nacelles, W_n
v. Landing gears, W_g

$$W_{struct} = W_W + W_{emp} + W_f + W_n + W_g \qquad (19)$$

i. Weight of the Wing:

$$W_W = 0.0017\, W_{MZF} \left(\frac{b}{coz\Lambda_{\frac{1}{2}}}\right)^{0.75} \left[\left(1 + \left(\frac{6.3\cos\left(\Lambda_{\frac{1}{2}}\right)}{b}\right)^{\frac{1}{2}}\right) (n_{ult})^{0.55} \left(\frac{bS}{t_r W_{MZF} cos\Lambda_{\frac{1}{2}}}\right)^{0.30}\right] \qquad (20)$$

where; W_{MZF} = maximum zero fuel weight = $W_{ro} - W_F$

$$W_{MZF} = 110000 - 53671.33 = 56328.67\ lbs \tag{21}$$

sU1bstituting the values of the terms obtained into equation 20, we get the weight of the wing to be equal to:

$$W_w = 6387.4\ lbs \tag{22}$$

ii. Weight of the Empennage:

$$W_{emp} = W_h + W_v \tag{23}$$

where; the weight of the horizontal stabilizer can be obtained using Torenbeeks method as follows,

$$W_h = K_h S_h \left[\frac{3.81\left((S_h)^{0.2} V_D\right)}{1000\left(cos\Lambda_{\frac{1}{2}h}\right)^{\frac{1}{2}}} - 0.287 \right] \tag{24}$$

Kh "' 1.1 variable incidence stabilizers

Substituting the values in the equation, we obtain the weight ohhe horizontal equal to:

$$W_{1r} = 7125.13\ lbs \tag{25}$$

The weight of the vertical stabilizer can be obtained using Torenbeeks method as follows,

$$W_V = K_v S_V \left[3.81 \left\{ \frac{(S_V)^{0.2} V_D}{1000\left(\cos\Lambda_{\frac{1}{2}V}\right)^{\frac{1}{2}}} \right\} - 0.287 \right] \quad (26)$$

$K_V = 1$ $(for\ fuselage\ mounted\ horizontal\ tails)$

$$\boldsymbol{W_V = 3605.2\ lbs} \quad (27)$$

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.

$$W_f = 0.021 K_f \left(\frac{V_D l_h}{w_f + h_f}\right)^{\frac{1}{2}} \left(S_{fgs}\right)^{1.2} \quad (29)$$

where, $S_{fgs} = fuselage\ gross\ shell\ area\ in\ ft^2$

the fuselage gross shell area is obtained from the previous sections as

$$S_{fgs} = 2204.67832 \frac{lb}{ft^2}$$

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 transonic business jet has 2 low by pass turbofan engines due to which the weight of the nacelles can be obtained from the equation

$$W_n = 0.055\, T_{TO} \quad (31)$$

where T_{TO} is, the total required take-off thrust, hence the equation accounts for all the nacelles.

$Trn = 56933.02 \frac{lbs}{ft^2}$

there the total weight of the nacelles is equal to:

ii½ $= 0.055 * 56933.02$

$$\boldsymbol{W_n = 3131.3161\ lbs} \quad (32)$$

v. Weight of the Landing Gears

The weight of the landing gears for a bus in ess jet with main landing gears attached to the wing and the nose gear mounted to the fuselage, the following equation is used:

$$W_g = K_{g_r}\left(A_g + B_g(W_{TO})^{\left(\frac{3}{4}\right)} + C_g W_{TO} + D_g (W_{TO})^{\frac{3}{2}}\right) \quad (33)$$

where,

$K_r = 1$ *(tor low wing aircrafts)*

the constants Ag *through* Dg 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.				D
Jet Trainer and Business Jets	Retr.	Main	33.0	[illegible]	0.021	
		Nose	12.0	[illegible]	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 Ag through Dg for the business jets with retractable landing gears are obtained as folllows:

Main: $A_g = 33.0$

$B_g = 0.04$

$C_g = 0.021$

$D_g = 0.0$

$$W_{g_{m}} = K_{g_r}\left(A_{g_m} + B_{g_m}(W_{TO})^{3/4} + C_{g_m} W_{TO} + D_{g_m}(W_{TO})^{3/2}\right) \quad (34)$$

$$W_{g_m} = 2584.6 \; lbs \quad (35)$$

Nose: $A_g = 12.0$

$B_g = 0.06$

$C_g = 0.0$

$D_g = 0.0$

$$W_{g_n} = 374.4 \; lbs \quad (36)$$

The total weight of the landing gears can be obtained by adding equation (35) and (36).

$$W_g = W_{g_m} + W_{g_n} = 2959 \; lbs \quad (37)$$

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

$$W_{struct} = 34600.3 \; lbs \quad (38)$$

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_{fsys}
iii. Propulsion system, $W_{p sys}$
iv. Accessory drives, Starting and Ignition system, $W_{Ast.sys}$
v. Thrust reversers, W_{thr}

Hence,

$$W_{pwr} = W_e + W_{fsys} + W_{prop.sys} + W_{ASt.sys} + W_{thr} \quad (39)$$

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 TSBJ uses two Pratt and Whitney J58 Engines that weight around 6000 lbs each. The exact values can be obtained directly from the manufacturers but for an estimate, these values can be considered.

$$W_e = W_{eng} + N_{eng} \tag{40}$$

where,
$W_{eng} \, is \; the \; weight \; of \; each \; engine \sim 6000 \; lbs$
$N_{eng} \, is \; the \; no \; of \; engines = 2$

$$W_e = 12000 \; lbs \tag{41}$$

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.

$$W_e = W_{eng} + N_{eng} \tag{40}$$

where,
$W_{eng} \, is \; the \; weight \; of \; each \; engine \sim 6000 \; lbs$
$N_{eng} \, is \; the \; no \; of \; engines = 2$

$$W_e = 12000 \; lbs \tag{41}$$

ii Fuel System

The TSBJ has internally integrated fuel tanks due to which to calculate the weight of the fuel system, the following equation will be used:

$$W_{fsys} = 80(N_e + N_t - 1) + 15(N_t)^{0.5}\left(\frac{W_F}{K_{fsp}}\right)^{0.333} \tag{42}$$

where,
$N_{eng} = No. of\ engines$
$N_t = No. of\ tanks$
$W_F = Weight\ of\ fuel$
$K_{fsp} = 6.55\frac{lbs}{gal}\ (constant\ for\ JP - 4\ Fuel)$

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, propeller controls, the engine starting system, the oil system and oil cooler.
The weight of the propulsion system is the summation of the above systems.

$$W_{prop.sys} = W_{ec} + W_{ess} + W_{pc} + W_{osc} \tag{44}$$

where,
The weight for the engine controls for an aircraft using fuselage mounted jet engines can be obtained from the following equation:

$$W_{ec} = K_{ec}\left(l_f N_e\right)^{0.792} \tag{45}$$

$K_{ec} = 1.080\ \ (for\ afterburning\ engines)$
$l_f = length\ of\ the\ fuselage$
$N_e = No. of\ engines$

$$W_{ec} = 1.080 * (2 * 98.6)^{0.792}$$
$$W_{ec} = 70.95\ lbs \tag{46}$$

The weight for the engine starting system can be determined using the equation:

$$W_{ess} = 9.33\left(\frac{W_e}{1000}\right)^{1.078} \quad (for\ engines\ using\ Pneumatic\ Starting\ Systems) \quad (47)$$

$$W_{ess} = 38.93\left(\frac{W_e}{1000}\right)^{0.918} \quad (for\ engines\ using\ electric\ starting\ system) \quad (48)$$

As it is not decided upon the system being used for the transonic 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:

$$W_{ess} = 135.9\ \ lbs\ (for\ engines\ having\ pneumatic\ starting\ systems) \quad (49)$$

$$W_{ess} = 381\ \ lbs\ (for\ engines\ having\ electric\ starting\ systems) \quad (50)$$

Weight of the oil system and oil cooler can be obtained using:

$$W_{osc} = K_{osc}W_e \quad (51)$$

$$K_{osc} = 0.000\ (for\ Jet\ Engines)$$

hence that gives us,

$$W_{osc} = \ 0.00\ lbs \quad (52)$$

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:

$$W_{prop.sys} = 452\ lbs \tag{53}$$

iv Thrust Reversers

An estimate of the weight of the thrust reversers can be made using the equation:

$$W_{thr} = 0.18\ W_e \tag{54}$$

$$W_{thr} = 15910\ lbs \tag{55}$$

hence, the total weight of the power plant system can be obtained as explained in equation (39) by adding the terms above:

$$W_{pwr} = 15910\ lbs \tag{56}$$

13.5 METHOD FOR ESTIMATING FIXED EQUIPMENT WEIGHT

The TSBJ fixed equipment's contain the following components:

i. Flight control systems, W_{fc}
ii. Electrical systems, W_{els}
iii. Instrumentation, avionics and electronics, W_{iae}
iv. Air-conditioning + deicing, W_{api}
v. Oxygen system, W_{ox}
vi. Furnishings, W_{fur}
vii. Paint, W_{pt}
viii. APU, W_{apu}
ix. Baggage and cargo handling, W_{bc}

The weight of the fixed equipment's of the SSBJ can be obtained by the following equation,

$$W_{feq} = W_{fc} + W_{els} + W_{iae} + W_{api} + W_{ox} + W_{fur} + W_{pt} + W_{apu} + W_{bc} \tag{57}$$

i. Flight Control System

The weight of the flight control system for an TSBJ can be estimated from the following equation:

$$W_{fc} = 56.01 \left(\frac{(W_m)(q_D)}{100000} \right)^{0.576} \quad (58)$$

where, *qD 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.

$$q_D = \frac{1}{2}\rho_\infty V_\infty^2 \quad (59)$$

$$qD = 169.6\ psf$$

$$W_{fc} = 1138.34\ lbs \quad (60)$$

ii. Electrical System Weight Estimation

The weight of electrical systems in a jet transport can be derived from the

$$W_{eis} = 10.8(V_{pax})^{0.7}(1 - 0.018(V_{pax})^{0.35}) \quad (61)$$

here, V_{pax} *is the passenger cabin volume in* ft^3

$$W_{eis} = 1902\ lbs \quad (62)$$

Hi. Instrumentation, Avionics and Electronics

The weight for the instrumentation, avionics and Electronics can be calculated using Torenbeek's method for jet transport.

$$W_{iae} = 0.575(W_E)^{0.556}(R)^{0.25} \quad (63)$$

where, *WE = empty weight in lbs and R = maximum range in nautical miles*

$$W_{iae} = 1974.6\ lbs \quad (64)$$

iv. Weight estimation for Air-Conditioning, and De-icing

For pressurized airplanes flying at subsonic speeds

$$W_{apt} = 469 \left(\frac{V_{pax}(N_{cr} + N_{pax})}{10000} \right)^{0.419} \quad (65)$$

Substituting the values in the equation gives the total weight of the air-conditioning and Deicing systems being equal to:

$$W_{api} = 972.8\ lbs \tag{66}$$

For Pressurized airplanes flying at transonic speeds, the weight for the air-conditioning and De-icing systems can be obtained using the equation:

$$W_{api} = 972.8\ lbs \tag{66}$$

For Pressurized airplanes flying at Supersonic Speeds, the weight for the air-conditioning and De-icing systems can be obtained using the equation:

$$W_{api} = 202\left(\frac{W_{iae}+200N_{cr}}{1000}\right)^{0.735} \tag{67}$$

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

$$W_o x = 83.2\ lbs \tag{70}$$

vi. Furnishin gs

$$W_{fur} = SSN_{fdc} + 32N_{pax} + 1SN_{cc} + K_{1av}(N_{pax})^{1.33} + K_{buf}(N_{pax})^{1.12} +$$

$$109\left(\frac{N_{pax}(1 + P_c)}{100}\right)^{0.505} + 0.771\left(\frac{W_{ro}}{1000}\right) \tag{71}$$

where,

N_{dfc} = No. of flight deck crew

N_{pax} = No. of passengers

N_{ee} = No. of Cabin Crew

K_{1av} = 3.90 (tor business Air planes)

K_{buf} = 5.68 (tor long range air planes)

P_c = Design U lt im ate cabin pressure

W_{ro} = Max. Take - off weight

substit uting the values of the following terms in equation 71 we get the weight of furnishings equal to:

$$W_{fur} = 1281.5 \; lbs \tag{72}$$

vii. Paint

The weight of paint is generally considered t o be between 0.003 t o 0..006 t imes t he max imum take-off weight

$$W_{pt} = (0.003 \; to \; 0.006) W_{TO} \tag{73}$$

hence to ut ilize the minimum weight for every compone nt, the weight of t he paint will in iti ally he consi dered to be equal to 0.003 times the maximum take-off weight.

$$W_{pt} = 0.003 * W_{TO} = 330 \; lbs \tag{74}$$

viii. APU

The weight of t he APU (aux i liary power unit) ranges t ypically between 0.004 to 0..013 times the maximum t ake -off weight. To obtain the list possi ble we ight est imat e fo r a component in the init ial stage of this design, the minimum limit will be cons idered.

$$W_{APU} = 0.004 * W_{TO} \tag{75}$$

$$W_{APU} = 440 \; lbs \tag{76}$$

ix. Baggage and cargo handling

For the SSBJ, the cargo will be carr ied alo ng wit h the passe ngers in to the passe nger cab in hence cargo han dling will hav e no extra weight s other than t he over-head baggage compartments.

$$W_{bc} = 0 \; lbs \tag{77}$$

hence accord ing to equat ion (57), the tot al weight of the fu rnishings can be obt ained by adding t he above individual t erms which provide the total weight equa l to

$$W_{feq} = 7577.6 \; lbs \tag{78}$$

CHAPTER 14: FINAL DESIGN REPORT – ENVIRONMENTAL/ ECONOMIC TRADEOFFS; SAFETY/ ECONOMIC TRADEOFFS

14.1 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)-	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
	Fuselag		
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

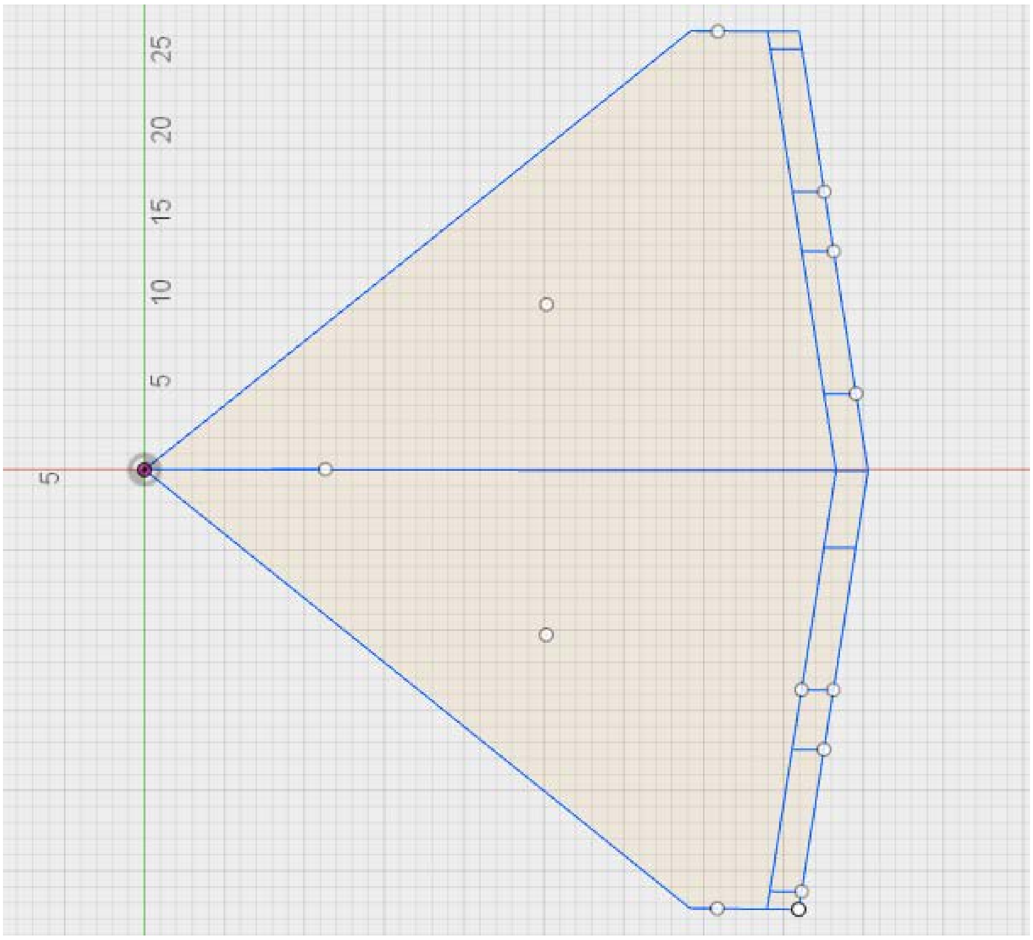


Figure 138: 2D view of the Designed Wing

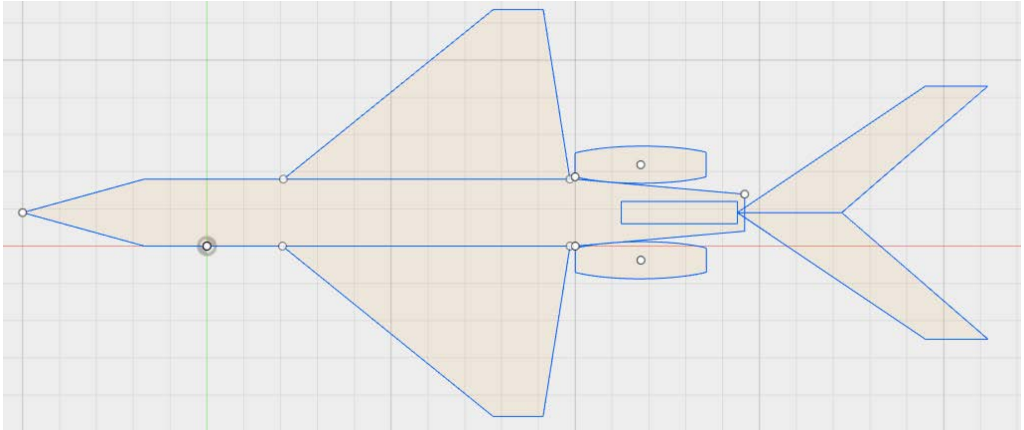


Figure 139: Top view of the designed TSBJ

Figure 140: Side view of the Designed TSBJ

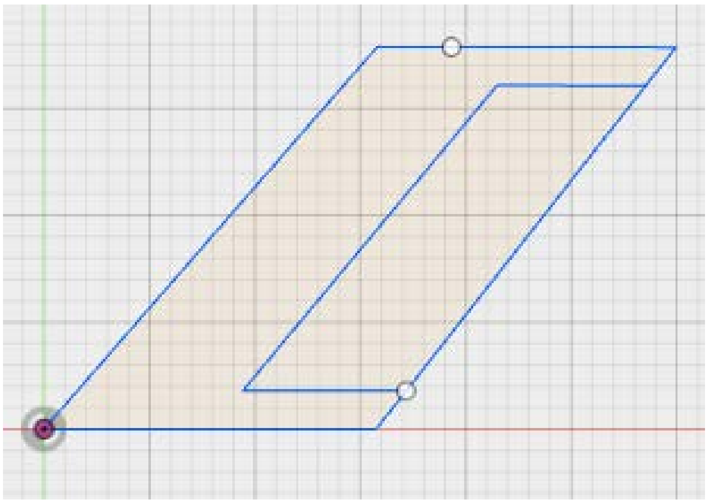


Figure 141: Side view of the vertical stabilizer

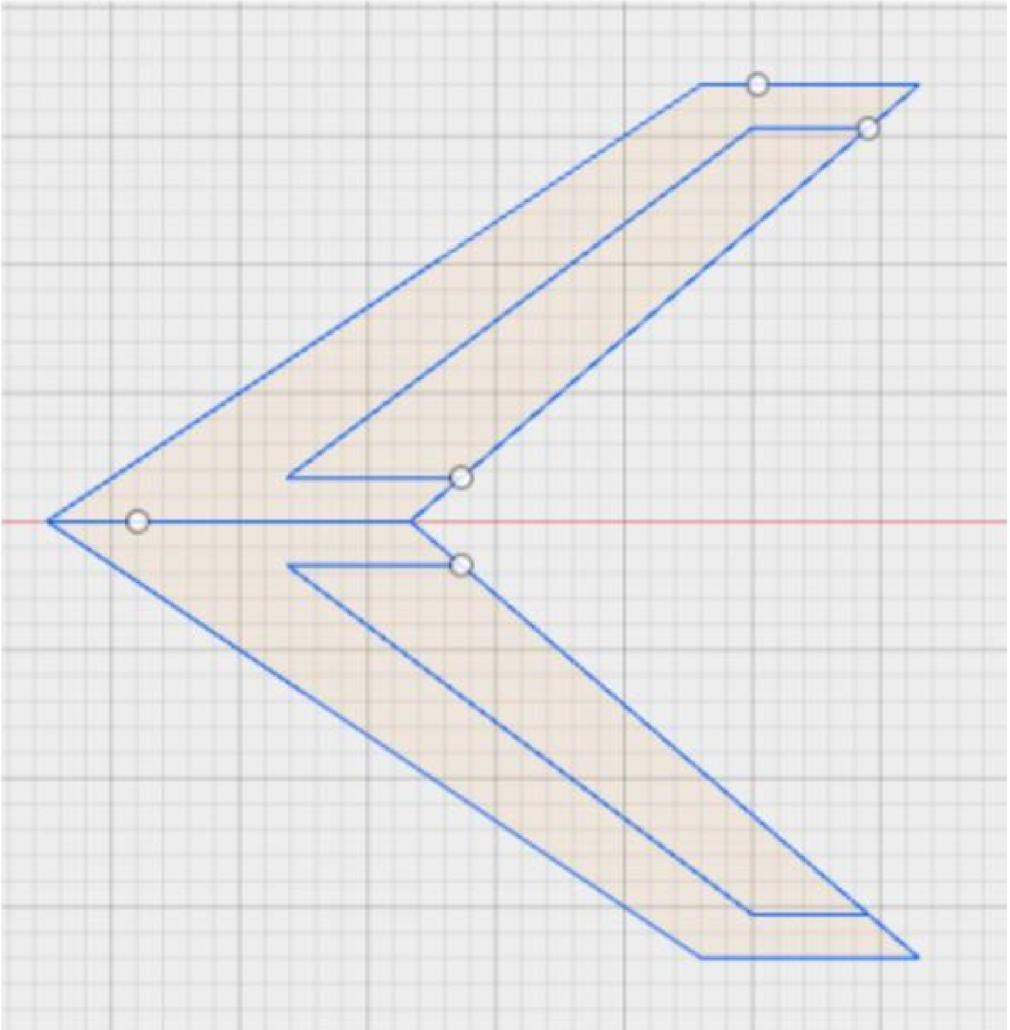


Figure 142: Top view of the horizontal Staibilizer

14.2 RECOMMENDATIONS

Once the process of designing the Transonic Business Jet is done, it can be determined via all the previous reports that it can be concluded that the proposed jet fulfils all the critically required parameters of the Preliminary design process. Thus, the design can be modified and taken further while keeping in mind all the characteristics and the parameters of the Business Jet. For further design, it is recommended that various changes are bound to be made on the design of the transonic business jet. As these changes were worked upon, the list of changes are recommended as follows:

1. When the initial sizing is being worked upon, it is of utmost importance that the number of passengers on-board should be reduced from 20 to just 12 passengers. The reduction in the number of passengers is not going to affect the design of the plane, rather it is done to provide greater comfort to the passengers, ensuring they get what they pay for. Most of the transonic business jets are designed to carry 12 passengers, and not more than 16, therefore, the lowest number of passengers is chosen as the other components of the plane are adjusted for. An increase in the range is also recommended.

2. A cranked arrow wing in the design of the airplane is recommended for the configuration design, as it shall help in reducing the weight of the airplane by a large extent. Also, by installing the elevators on the cranked part of the wing helps in removing the horizontal stabilizer and in turn reducing the weight of the vertical stabilizer.

3. It is always recommended to make use of composites on key large components of the airplane, this is done for the weight sizing of the airplane. This helps in reducing the weight of the entire aircraft and provides better performances, low maintenance, higher speeds, longer life cycles, and more strength etc.

4. For better performance of the airplane, the stall speed should be reduced to provide better low speed performance of the airplane. The process also aids in reducing wing loading on the airplane, reduces the drag produced, increases the lift coefficient and provides optimum take-off and landing performances for the airplane.

5. When the cockpit/fuselage of an airplane is being designed it is always recommended to further reduce the maximum height as this helps in providing better aerodynamic performances.

6. When considering the wing design, it is suggested to use the cranked arrow wing instead of a delta wing as mentioned earlier for the configuration design.

7. When it comes to the empennage section, it is suggested to reduce the weights after multiple iterations in the weight and balance section keeping in mind every minute change in the area of the horizontal and vertical design in accordance with the CG obtained that satisfies the stability requirements of the airplanes.

8. It is suggested to make use of the retractable landing gears which a tricycle configuration for the landing gear design, this is already used in the current design at

greater speeds, as fixed landing gears result in a lot of drag which is not optimal for transonic airplanes. One must also consider various iterations so as to acquire the exact CG location of the airplane and keep in mind every small factor that affects the change in weight of the airplane. The place of the landing gears behind the CG, is to maintain the stability of the airplane and satisfy the ground clearance criterion and the nose over criterion as the wing configuration is a mid-wing configuration. Hence, this should provide better ground clearance to obtain better ground effects while taking-off the airplane.

9. About the stability and control, it is suggested to iterate the longitudinal and directional X-plots for several times until the perfect configuration that is feasible for the design of the airplane is obtained. Hence, it should fulfil every critical error in the design that can affect the design of the airplane.

10. It is recommended to consider the highest possible zero lift drag incremental factors to acquire better performances to the design for the drag polar estimation.

14.3 ENVIRONMENTAL/ECONOMIC TRADEOFFS

- The biggest environmental issue which is associated with an airplane is it's fuel consumption. As a Business Jet cruises at transonic speeds, it consumes a lot of fuel, this produces immense noise within the engine section and this adds to the air and noise pollution. Hence, some research on the engine sections to reduce the usage of fuel and alternatives such as hydrogen powered engines which reduce the air pollution up to a great extent are suggested. As for the noise pollution, it is recommended to make use of spikes in the outlet sections of the engines as this aid in reducing noise and the vibrations which are produced.

- These were some of the prominent issues which were addressed while doing research work in this area. To combat air pollution battery powered airplanes could also be introduced while to reduce the noise effects spikes can be introduced and beyond this in-depth research work has been done on the noise reducers.

- The best solution for air pollution by planes is the use of hydrogen powered airplanes, as the fuel costs reduces by a great amount too. Hydrogen is profound for being the most combustible gas, and it also provides better performances and reduces air pollution as well.

- The biggest stumbling block in resolving the environmental issues is the immense costs involved, right from the research to the cost of manufacturing the proposed design, as even a minor change in the design can cost a fortune for redesigning and manufacturing.

14.4 SAFETY/ECONOMIC TRADEOFFS

- The control of the airplane at transonic speeds is the biggest safety issue related to the airplane. It is always essential to keep in the airplane under the control of the pilot.

- To ensure the safety requirements are met, high speed controllers should be designed. These can maintain the aircraft steady even without the assistance of pilots. It should be able to operate individually in such a way that it can handle every minor change needed to keep the aircraft steady in any sort of condition.

- Again, cost is an issue while designing such a system, as it costs a fortune to build such a controller.

- Only research work has been executed on transonic business jets as none are available in the market. The proposed trade-offs are presumptions made from the problems that airplanes which fly at similar speeds face.

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