ASE 451 - AERONAUTICAL ENGINEERING DESIGN

Final Report

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Introduction

The aviation industry is one of the areas that have had a massive increment in the development field over decades. It predictably has years to grow due to technological developments and combinations of discoveries and inventions. The high demand for aviation opens new areas in the aviation industry, such as making an airplane for company use, like business jets. The other reality is that a successful aircraft design project lays under extensive work. In this report, a business jet can fly from Paris Charles de Gaulle Airport to New York John F. Kennedy Airport with up to 50 passengers is designed. The distance between these destinations is 5830 km, and 7.5 hours of flight time is required. This project includes aspects of a design project, precisely the designed airplane. It contains all conceptual and preliminary design stages and geometrical and aerodynamical performance parameters. Preliminary design includes essential steps such as weight estimation, fuselage layout, airfoil selection, wing geometry, performance parameter estimations, refined weight estimation, and center of gravity positions. Additionally, cost analysis, environmental issues, landing gear, and airplane layout are over consideration in this report. The design process is based on similar types of aircraft chosen concerning the requirements like range, jet engine, and so on mentioned above. The project resulted successfully acceptable under consideration of analysis. The reasons are explained with the calculations and choices in detail. A further 3D and 2D engineering CAD drawings of the airplane with dimensions and other parameters are included in the report's last chapter.

The airplane is designed to provide comfort for passenger while traveling. The seats are with a pitch of 1 meter with suitable legroom, with the baggage capacity optimization. That makes the airplane to 'sell itself' easily, that means higher demand for the airplane, by selling the comfort to passengers. In the first chapter, six possible similar candidates are chosen are ERJ145 LR, Boeing 737-500 VIP, FOKKER 70, Bombardier CRJ550, and Antonov An-148 in order to get inspired for the new design parameter selection. The airplanes are analyzed in terms of geometric, aerodynamic and general parameters and positive effects on each feature is selected to optimize new design with help of historical data and trends. With these informations, maximum altitude is decided between 41000 and 45000 ft. An important aspect of designing aircraft is weight fraction. In this chapter, each weight fraction of the airplanes is calculated. In chapter 2, a simple mission profile is defined. The focus is on initial weight estimation with other similar results and curve-fit equation. Appropriate takeoff weight is selected by evaluating the data of airplanes. In calculation, range (5830 km) and loiter (20 minutes) time is used to determine cruise speed. With appropriate data of weight, thrust and thrust specific fuel consumption (TSFC) data comparison in the table with other 6 candidates, the best engine option for the design is CFM-56-3B-1. Chapter 3 is fuselage layout design including cabin arrangement, cabin deck flow line and aisle location, seating for economy and business class, floor thickness, standart cargo containers, and shape of nose and tail cones. All parameters are shown in one single table for understanding the concept. Th next chapter , chapter 4 is calculations and discussions on determining design parameters for the airplane. One of the most important parameter is lift coefficient. Determination of lift coefficient is focused. Concerning the C_L calculation, airfoil is selected as NACA 65(3)-618. In chapter 5, performance parameter estimation of thrust to weight ration and wing loading under the requirements of stall speed, landing distance, takeoff distance, maximum ceiling compared with historical trends. In chapter 6, weight is calculated with other parameters by MATLAB code to get accuracy less than 0.01 absolute error. Each segment weight fraction is calculated. Chapter 7 focuses on determining aircraft sizing and center of gravity location, Aircraft sizing includes calculation of wing, and fuel tanks with dimension and locations that contributes calculation of center of gravity of the airplane with whole pieces, firstly without wing and fuel tanks and then including wing and fuel tanks. Horizontal and vertical tail locations and lengths are calculated to obtain tail location in nose cone based coordinate system.Taking all information and data into consideration, cost analysis is made in chapter 8. The consideration includes the fields of RDT&E and production cost, operational and maintenance cost, field and oil costs , crew salaries, maintenance expenses, depreciation and insurance aspects. In chapter 9, environmental issue under concerns on engine type, fuel consumption& carbon dioxide emision, thrust resever , noise, engine efficiency and takeoff distance reduction is analyzed. The resulting analyze shows the airplane and therefore, the engine is environmentally efficient to save the nature. Lastly, in chapter 10, aircraft layout and landing gear design is made. Landing gear sizing and aircraft layout in 2-D drawing are provided. 3-D CAD drawing of the business jet figures is attached .

Chapter 1. Design Requirements and Competitor Study

The aim of this project is to create a business plane with a seating capacity of 50 people. The aircraft is designed for direct targeted flights as well as the 7.5 hour flight between Paris and New York to John F. Kennedy International Airport, USA. In addition to the literature review, an analysis of existing business jets similar to the design criteria of the project was also carried out. Using the findings of this study, efforts were made to develop a jet with improved economic and environmental performance. In addition, lightness, high cruising speed and performance, as well as passenger comfort in compliance with regulations, were prioritized in the design.

Table 1.1 *Design Requirements*

Number of Passengers	50
Takeoff Airport	Paris Charles de Gaulle Airport, France
Landing airport	John F. Kennedy International Airport, USA
Range	5830 km
Flight Duration	7 hours and 30 minutes

In Table 1.1, the design requirements are shown in detail and a competitor study was carried out in accordance with the requirements.

1.1 Competitor Study

Table 1.2 *General Characteristics*

	ERJ145 LR[1,2]	**Boeing 737-500 VIP[7,8,9]**	**FOKKER 70[4,5,11]**	**Bombardier CRJ550[3]**	**Antonov An-148[10,12,13]**	**Comac[15,16] ARJ21-700**
Airplane type	Regional Jet	Regional Jet	Regional Jet	Regional Jet	Regional Jet	Advanced Regional Jet
Number of seats	50	40-60	45-85	50	68-85	70-95
Crew	2 pilot + 1 cabin crew	2 pilots + 1 cabin crew	2 pilots	2 pilots+ 2 cabin crew	2 pilots	2 pilots
MTOW (W)	48,500 lb	133,500 lb	87,998 lb	65,000 lb	96,342 lb	89,287 lb
OEW	26,707 lb	70,438 lb	50,266 lb	44,244 lb	48,502 lb	55,016 lb
Fuel capacity	11,323 lb	44,322 lb	29,454 lb	25,035 lb	26,566 lb	22,897 lb
Maximum payload	12,756 lb	32,563 lb	24,008 lb	15,544 lb	19,842 lb	43,426 lb
Cruise speed	M 0.78	M 0.71	M 0.68	M 0.78	M 0.66	M 0.78
Maximum speed	870.33 ft/s	830.23 ft/s	859.39 ft/s	798.33 ft/s	792.86 ft/s	792.86 ft/s
Range	9,262 ft	14,597 ft	6,591 ft	6,076 ft	11,417 ft	12,139 ft
Ceiling	37,000 ft	37,001 ft	35,000 ft	40,997 ft	37,185 ft	39,042 ft

Takeoff (MTOW, SL, ISA)	7,448 ft	8,694 ft	4,265 ft	4,055 ft	6,194 ft	5,577 ft
Landing (MLW, SL, ISA)	4,593 ft	6,249 ft	3,969 ft	4,711 ft	5,459 ft	5,249 ft
Model, Type	2 Turbofan Engines, AE30O7 A1	2 turbofan Engines, CFM56-3C-1	2 Turbofan Engines Rolls-Royce Tay 620-15	2 x GE CF34-8C5B, Turbofan	2 Turbofan Progress D-436-148	2x General Electric CF34-10ATurbofan
Static thrust	14,882 lb	18,501.77 lb	13,623.42 lb	12,005 lb	13,839 lb	17,056 lb
Bypass ratio	NA	5.0	NA	5:1	NA	5:1
Thrust (takeoff)	14,882 lb	20,007 lb	13,623 lb	12,005 lb	13,839 lb	17,056 lb

The purpose of this section is to identify other aircraft with a similar capacity in terms of passenger numbers. In Table 1.2, 6 similar business jet planes are compared and many general characteristics are discussed while making comparisons, some of their number of seats, Maximum takeoff weight, and Cruise speed. These planes are ERJ 145 LR, Boeing 737-500 VIP, FOKKER 70, Bombardier CRJ550, Antonov An148, and Comac ARJ21 respectively.

Table 1.3 *Geometric Characteristics*

		ERJ145 LR[1,2]	**Boeing 737-500 VIP**[7,8,9]	**FOKKER 70**[4,5,11]	**Bombardier CRJ550**[3]	**Antonov An-148**[10,12,13]	**Comac**[15,16] **ARJ21-700**
Fuselage Geometry	**Length**	91.63 ft	97.76 ft	101.41ft	105.97 ft	95.57 ft	109.77 ft
	Height	7.48 ft	13.15 ft	27.88 ft	24.93 ft	26.87 ft	27.69 ft
	Width	7.48 ft	12.33 ft	10.82 ft	8.85 ft	8.90 ft	10.49 ft
	Cabin width	5.92 ft	11.58 ft	10.17 ft	7.08 ft	10.33 ft	10.30 ft
	Fineness ratio	12.32	8.02	9.97	11.9	NA	10.45
Wing Geometry	**Wing span**	65.74 ft	94.75 ft	92.19 ft	76.11 ft	94.84 ft	89.50 ft
	Wing area	550.89 ft^2	1134.51ft^2	1006.42 ft^2	759.93 ft^2	939.90 ft^2	859.60 ft^2
	Aspect ratio	7.8	9.16	8.45	7.6	9.57	9.31
	Taper ratio	NA	0.240	NA	NA	NA	N/A
	Wing sweep	NA	25.00	NA	NA	NA	25°
	Cargo volume	324.89 ft^3	822.83 ft^3	451.32 ft^3	547.37 ft^3	515.59 ft^3	711.23 ft^3

In Table 1.3, the geometric properties of 6 business jets are compared. These features consist of 3 main parts, the first part is the dimensions in relation to the fuselage, the second part is the dimensions in relation to the wing, and the last part is the cargo volume.

Table 1.4 *Important Performance and Design Characteristics*

	ERJ145 LR	Boeing737 − 500 VIP	FOKKER70	Bombardier CRJ550	Antonov An-148	Comac ARJ21-700
Wing Loading [psf]	88.07	117.71	87.46	85.54	102.53	88.61
Thrust-to-Weight Ratio	0.307	0.300	0.315	0.382	0.320	0.394
Empty Weight Fraction	0.551	0.528	0.571	0.67	0.503	0.616
Fuel Weight Fraction	0.233	0.332	0.335	0.391	0.276	0.256
MaximumLift Coefficient	0.458	0.553	0.400	0.422	0.557	0.547
MaximumDrag Coefficient	0.066	0.083	0.063	0.088	0.089	0.118

In Table 1.4, some important performance and design characteristics are shown. These parameters are compared for the same six business jets. These properties are respectively wing load, thrust-to-weight ratio, curb weight ratio, fuel-to-weight ratio, maximum lift coefficient, and maximum drag coefficient.

1.2 Discussing and comparing competitor study

As a result of competitor research for the business jet we will design, six different aircraft were examined, namely ERJ145 LR, Boeing737-500 VIP, FOKKER 70, Bombardier CRJ550, and Antonov An-148. There are three different research results tables regarding these business jets General Features, Geometric Features, and Performance and Design Features. When the General Characteristics table is examined, the maximum takeoff gross weight values are between 48,500 lb and 133,500 lb. Operational commodity weight is low in ERJ 145 LR and high in Boeing 737-500 VIP, but it is seen that this value is around 50,000 lb in the remaining aircraft. Cruise speed values vary between 0.66 and 0.75. When the geometric characteristic table is examined, we see that the length of each plane is 95 ft. Wing span values vary, with the exception of the smallest aircraft, the ERJ 145 LR, the others have a wingspan of approximately 90 ft. The business jet, which will be designed by taking advantage of the values of these 6 aircraft, is planned to be close to these values.

Chapter 2 Initial Weight Estimation

2.1 Calibration of the empty-weight fraction statistical curve-fit equation

Part I

The maximum takeoff gross weight and operating empty weight values of the three business jets "ERJ 145 LR, Boeing 37-500 VIP, and Bombardier CRJ550" selected by the competitor study are shown in Table 1.

Table 2.1.1 *MTOW and OEW values of different business jets from competitor study*

	ERJ 145 LR	**Boeing 37-500 VIP**	**Bombardier CRJ550**
MTOW	48,500 lb	133,500 lb	65,000 lb
OEW	26,707 lb	70,438 lb	44,244 lb

The weight fraction formula given by

$$\frac{W_e}{W_o} = \frac{\text{MTOW}}{\text{OEW}}, \tag{2.1}$$

Where, $\frac{W_e}{W_0}$ the empty weight fraction, MTOW is the maximum takeoff gross weight, and OEW is the operating empty weight.

Table 2.1.2 *Competitor study values for chosen business jets*

	ERJ 145 LR	**Boeing 37-500 VIP**	**Bombardier CRJ550**
$\boldsymbol{\frac{W_e}{W_o}}$	0.551	0.5276	0.586

Part II.

Statistical curve-fit equation based upon takeoff gross weight is given by

$$\frac{W_e}{W_0} = A{W_0}^C K_{vs}, \tag{2.3}$$

Where, A & C are the constant values, W_0 is the take off gross weight, and K_{vs} is the sweep constant.

Table 2.1.3 *The corresponding coefficients taken from table 3.1 (D. Raymer, "Aircraft Design: A Conceptual Approach") with respect to jet transport values.*

A (metric)	0.97
C	-0.06
$\boldsymbol{K_{VS}}$	1.00

Table 2.1.4 *Empty-weight fraction for each airplane using statistical curve-fit equation*

	ERJ 145 LR	Boeing 37-500 VIP	Bombardier CRJ550
$\frac{W_e}{W_o}$	0.532	0.501	0.523

ERJ 145 LR : $\frac{W_e}{W_o}$ = (0.97) *($22000^{-0.06}$)*(1.00) = 0.532

$$\textbf{Absolute Error} = \left|\frac{0.551 - 0.532}{0.551}\right| * 100 = 3.45\ \%$$

Boeing 737-500 VIP: $\frac{W_e}{W_o}$ = (0.97) *($60555^{-0.06}$)*(1.00) = 0.501

$$\textbf{Absolute Error} = \left|\frac{0.5276 - 0.501}{0.5276}\right| * 100 = 5.04\ \%$$

Bombardier-CRJ550: $\frac{W_e}{W_o}$= (0.97) *($29484^{-0.06}$)*(1.00) = 0.523

$$\textbf{Absolute Error} = \left|\frac{0.586 - 0.523}{0.586}\right| * 100 = 9.75\ \%$$

Part III.

Table 2.1.5 *Selecting different the corresponding coefficients for the statistical curve-fit equation*

A (metric)	**1.15**
C	**-0.07**
K_{VS}	**1.00**

Table 2.1.6 *Empty-weight fraction with different the corresponding coefficients*

	ERJ 145 LR	Boeing 37-500 VIP	Bombardier CRJ550
$\frac{W_e}{W_o}$	0.571	0.532	0.560

ERJ 145 LR : $\frac{\mathrm{W_e}}{\mathrm{W_o}}$ = (1.15) *($22000^{-0.07}$)*(1.00) = 0.571

Boeing 737-500 VIP : $\frac{\mathrm{W_e}}{\mathrm{W_o}}$= (1.15) *($60555^{-0.07}$)*(1.00) = 0.532

Bombardier-CRJ550: $\frac{\mathrm{W_e}}{\mathrm{W_o}}$= (1.15) *($29484^{-0.07}$)*(1.00) = 0.560

Table 2.1.7 *Accuracy estimation for Raymer with different constant values*

Aircraft	A	C	Accuracy for Raymer (%)
ERJ 145 LR	1.02	-0.05	99.586
Boeing 37-500 VIP	1.17	-0.06	99.874
Bombardier CRJ550	1.21	-0.07	99.985

Part IV.

For empty-weight fractions, all estimated values range between 0.5 and 0.6. For the selected sized takeoff weights W_o, it is evident the comparable empty-weight percent numbers for jet transport aircraft vary from 0.5 to 0.6. Therefore, it can be said that the results from Part I are compatible with trends in the empty-weight fraction. In Part II, the values of the empty-weight fractions are determined from Eqn. (2) change slightly from those discovered in Part I. To match the results from Part I with the results obtained using Eqn. (1), Part III adjusts the values of A & C. (2). As a result, the empty-weight fraction values we discovered in Part iii began to resemble those found in Part I more.

2.2 Design takeoff gross weight estimation

2.2.1 Determining Range, Loiter, and Cruise speed

Designing a business jet has a simple mission profile. Takeoff, climb, cruise, descent, loiter, and landing are in this mission. For this mission, range, loiter and cruise speed should be targeted. To find the range, two airports are chosen. These airports are John F. Kennedy International Airport (JFK) and Paris-Charles De Gaulle Airport (CDG). These two airports' distance is 5,830 km. Also, Loiter (endurance) time is assumed to be 20 minutes. Because of that, require time is calculated at 25800 seconds with loitering.

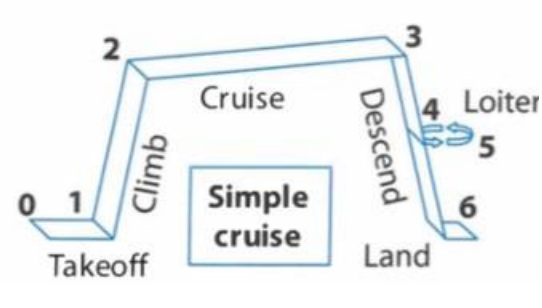


Figure 2.2.1 *Simple mission profile, including mission segments with takeoff, climb, cruise, descent, loiter, and landing.*

Also, cruise speed is calculated with this formula

$$V_{cruise} = \frac{Distance}{Required\ Time}, \tag{2.4}$$

$$V_{cruise} = \frac{5830\ km}{(7hours\ 30\ \text{min}) - \ 20\ min} = \frac{5830 * 10^3\ m}{25800\ s} = 225.97\frac{m}{s} \ = \ 741.37\ ft/s$$

2.2.2 Engine Selection

Table 2.2.1 *Different type of Engines*

ENGINE	THRUST	TSFC
CFM56-3B-1	20,000 lbf	0.667 lb/(lbf·h)
Rolls-Royce AE 3007-A1E	8,917 lbf	0.65 lb/(lbf·h)
General Electric CF34-3B1	8,729 lbf	0.69 lb/(lbf·h)
Progress D-436-148	14,000 lbf	0.617 lb/(lbf·h)
Rolls-Royce BR710A2-20	14,700 lbf	0.64 lb/(lbf·h)

Table 2.2.1 displays the values of thrust produced and the specific thrust of several engines. More thrust and less specific thrust are needed while selecting the best engine. As a result, it makes more sense to select the CFM-56-3B-1. It has 4276 lbs, and will be use in this project.

2.2.3 The design takeoff gross weight estimation with MATLAB code

In this section, to design take-off gross weight estimation, some parameters need to be found in Eqn. [8]. They are crew weight, payload, empty weight fraction, and fuel weight fraction. For this reason, crew weight and payload weights are shown in Table 2.1. Also, the lift/drag ratio is 15 and thrust-specific fuel consumption is assumed to be 0.36 for each engine. Mission segment weight fraction is required to find fuel weight fraction. The mission segment is calculated in Table 2.3, Eqn. [4], and Eqn. [5]. It is found with Eqn. [7] thanks to the fuel weight fraction mission segment. With these values we found, the take-off gross weight is found with the iterative method thanks to the MATLAB code.

Table 2.2.2 *Weight of Payloads*

2 pilots & 1 crew weight	240*3 [lb]
Passengers weight	195 [lb]
Baggage weight	16 [lb]

Table 2.2.3 *Assume Parameters for MATLAB code*

Lift/Drag coefficient Ratio	15
Thrust Specific Fuel Consumption “C_j”	2*0.36

Table 2.2.4 *Weight fractions for the mission segments*

W_{1-0}	0.995
W_{2-1}	0.98
W_{4-3}	0.99
W_{6-5}	0.992

The cruise segment mission fraction is given by

$$\frac{W_3}{W_2} = \exp\left(\frac{-R * C_j}{V * \frac{L}{D}}\right), \tag{2.6}$$

where R is the range, C_j is the thrust specific fuel consumption, V is the cruise speed, and $\frac{L}{D}$ is the lift-to-drag ratio. The cruise mission segment fraction calculated as 0.7079 with these data.

Loiter segment mission fraction is given by

$$\frac{W_5}{W_4} = \exp\left(\frac{-E * C_j}{\frac{L}{D}}\right), \tag{2.7}$$

where E is the endurance (loiter) time. The loiter segment mission fraction is calculated as 0.9841.

Mission segments weight fractions is given by

$$M_f = \frac{W_6}{W_0} = \frac{W_6}{W_5} * \frac{W_5}{W_4} * \frac{W_4}{W_3} * \frac{W_3}{W_2} * \frac{W_2}{W_1} * \frac{W_1}{W_0} = 0.6671 \tag{2.8}$$

Fuel Weight Fraction is given by

$$\frac{W_f}{W_0} = 1.06\left(1 - M_f\right) = 0.3528 \tag{2.9}$$

Design takeoff gross weight

$$W_0 = \frac{W_c + W_{pl}}{1 - \frac{W_f}{W_0} - \frac{W_e}{W_0}}, \tag{2.10}$$

where W_0 is the takeoff gross weight, W_c is the crew weight, and W_{pl} is the payload weight.

Table 2.2.5 *Result of initial takeoff weight and weight fraction*

Maximum Takeoff Gross Weight	95869 lb
Empty Weight Fraction	0.5259

According to the MATLAB code which can be seen on appendix, W_0 = 95869 lbs. Also, empty weight fraction is 0.5259. When the takeoff gross weight result compared with other competitor studies, it is seen clearly that values are appropriate.

2.2.4 The results of the Trade Studies and Plots

In Part 2.2, endurance time is determined as 20 minutes and takeoff gross weight estimation has been made accordingly with MATLAB code. In this section, how the takeoff gross weight changes in different endurance times are shown in the graph thanks to the MATLAB code. Endurance time is taken between 20 minutes and 3 hours as can be seen in Figure 2.2.1

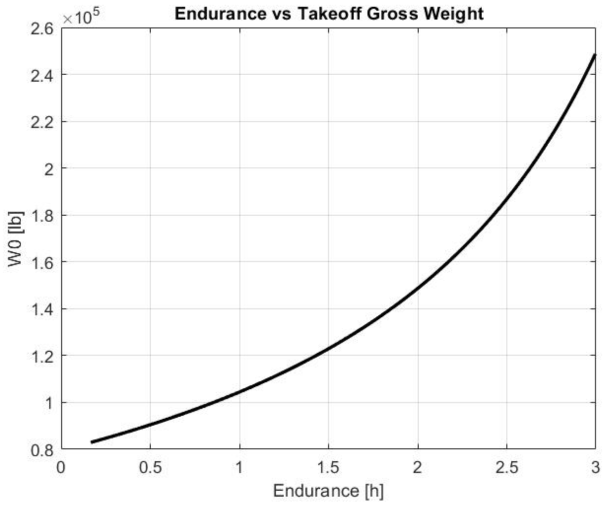


Figure 2.2.1 *Endurance vs Takeoff Gross Weight*

In addition, thanks to the MATLAB code, the change in takeoff gross weight has been observed in different ranges. The original distance between the airports of John F. Kennedy International Airport (JFK) and Paris-Charles De Gaulle Airport (CDG) is 5830 km or 3622.59 mi. In the MATLAB code, the range is calculated between 500 mi and 4000 mi. While the takeoff gross weight is increasing slightly at the beginning with a small range, it is clearly seen that it increases rapidly after 2500 mi.

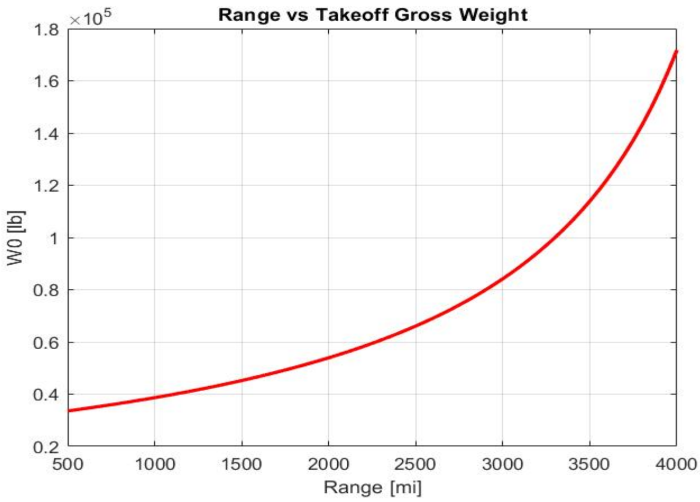


Figure 2.2.2 *Range vs Takeoff Gross Weight*

Moreover, the number of passengers to be used in this project has been determined as 50, but MATLAB code was used to observe how the takeoff gross weight increased in different passenger numbers. As shown in Figure 2.2.2, it is observed that the takeoff gross weight increases linearly as the number of passengers increases.

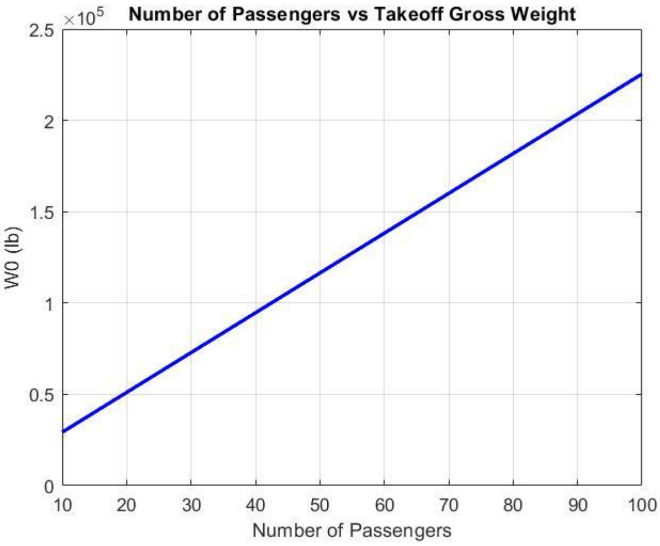


Figure 2.2.3 *Number of Passengers vs Takeoff Gross Weight*

2.2.5 Conclusion of different Range, Endurance time, and Number of passengers

As a result, changes in range and durability directly affect take-off gross weight. The increase in the number of passengers proportionally affects the weight of the aircraft. As the range is increased, the take-off gross weight also increases. The reason for the increase in take-off gross weight is that when the range is increased, the required fuel also increases. This fact also applies to durability, because the higher the durability, the higher the fuel needed and the direct increase in take-off gross weight. In light of these calculations in this study, it can be said that the aircraft we designed can carry 50 people in 7 hours and 30 minutes.

2.3 Discussing and comparing initial weight estimation

In Chapter 2.1, weight fractions for the ERJ 145 LR, Boeing 37-500 VIP, and Bombardier CRJ550 business jets were calculated using the competition study. In addition, new values of weight fractions were found using the statistical curve-fitting equation. The coefficients were then replaced with optimal values for the statistical curve-fitting equation. We have the necessary values to be able to interpret these values. Equivalent curb weight percentage numbers for jet transport aircraft range from 0.5 to 0.6 for selected size take-off weights Wo. In conclusion, their results appear to be consistent with current trends in the empty weight fraction. In Chapter 2.2, a simple mission profile is defined, which includes mission sections such as take-off, climb, and cruise, descent, hovering, and landing. Two airports are chosen to determine the range. John F. Kennedy International Airport (JFK) and Paris-Charles De Gaulle Airport are these airports (CDG). A delay period of 20 minutes is also assumed. As a result, idleness is included in the calculation of the required time. The cruise speed is then calculated. It was determined that the CFM-56-3B-1 engine was suitable for this project. Therefore, some parameters such as pilot, crew, passengers, and luggage were defined to design the take-off gross weight estimation. Using the numbers we found with the help of the MATLAB code, the take-off gross weight was 95,869 lb and the empty weight fraction was 0.5259. It was observed that these values were close to the values of the 6 planes examined in the competitor study. As an example, our takeoff weight is 95869lb, while the Fokker 70 is 87,998lb and the Antonov An-148 is 96,342lb.

Chapter 3. Fuselage Layout

3.1 Cabin Arrangement

The number of passengers on this mission is set at 50. For this reason, it was decided to leave 2 rows for business class and 4 seats for each row. This means that the total business class passengers are 8 people. It was also decided to leave 6 seats in 7 rows and 3 seats per row for economy classes. In total, 42 people belong to the economy class. The reason for choosing 3-3 seats in Economy class is to keep the fuselage length as short as possible.

3.2 For Cabin classes & configurations

3.2.1 Design of Economy and Business class seating

Using Table 1.1, seat width, aisle width and seat pitch have been selected for both economy and business.

Table 3.2.1 *Typical Seating Dimensions [14]*

Description	First Class (in.)	Coach (in.)
Seat width ws	19-21	17-18
Armrest witdh	2.75	2.5
Aisle width wi	18-20	16-18
Pitch P	37-42	32-36
Seat height	17	17

The parameters selected according to Table 1.1 for economy and business classes are shown in Table 1.2. In Table 1.1, the values are given in inches, although the values in the Table are given in millimeters.

Table 3.2.2 *Seat configuration parameters selected for Economy and Business classes*

[in.]	Seat width	Aisle width	Seat pitch
Economy Class	17	18	32
Business Class	21	20	37

3.2.2 The cabin deck floor line (cross-section) & the seats and aisle locations

The fuselage width is proportional to the number of seats abreast, seat width, and the width of the aisles between the seats,

$$w_{fus} \approx c_1 + c_2 w_s N_a, \tag{3.1}$$

Solving equation 1 with the help of Table 1.3, w_{fus} is found as 12.48599 feet. 12.48599 feet was used in 2d drawing as 3805.75 mm in the SI unit.

Table 3.2.3 *Average Fuselage Data for Typical Commercial Aircraft [14]*

Na	**c1**	**c2**	**c3**	***ws* (ft)**
6	2.49	1.24	1.08	1.43

Seat parameters, cabin deck ceiling height, fuselage diameter, and thickness figures 3.2.1 and 3.2.1 for business and economy classes are drawn in detail. Seats with an aisle with are also shown in the drawing. For the Economy class, the seats are arranged 3-3 in a single row. For Economy class, the seat width is 17 inches and the aisle width is 18 inches.

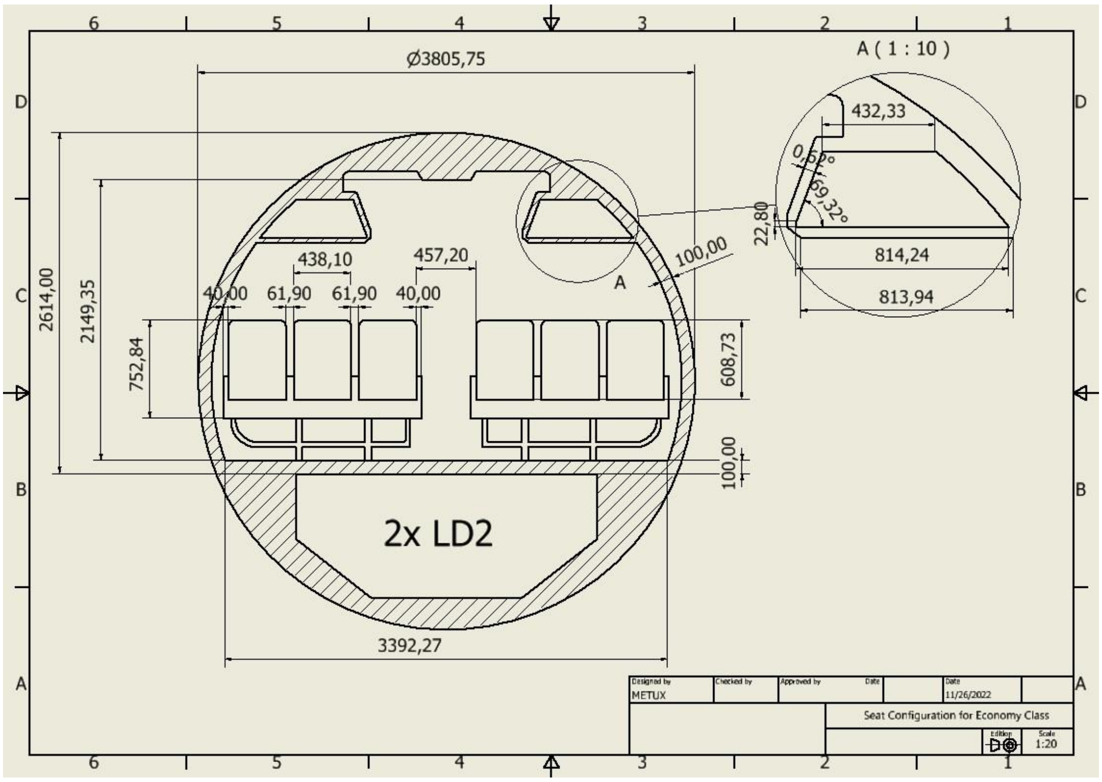


__Figure 3.2.1__ Economy class seat configuration on 2D drawing in SI units

For the Business class, 2-2 seats are arranged in a single row. For the Business class, the seat width is 21 inches and the aisle width is 20 inches.

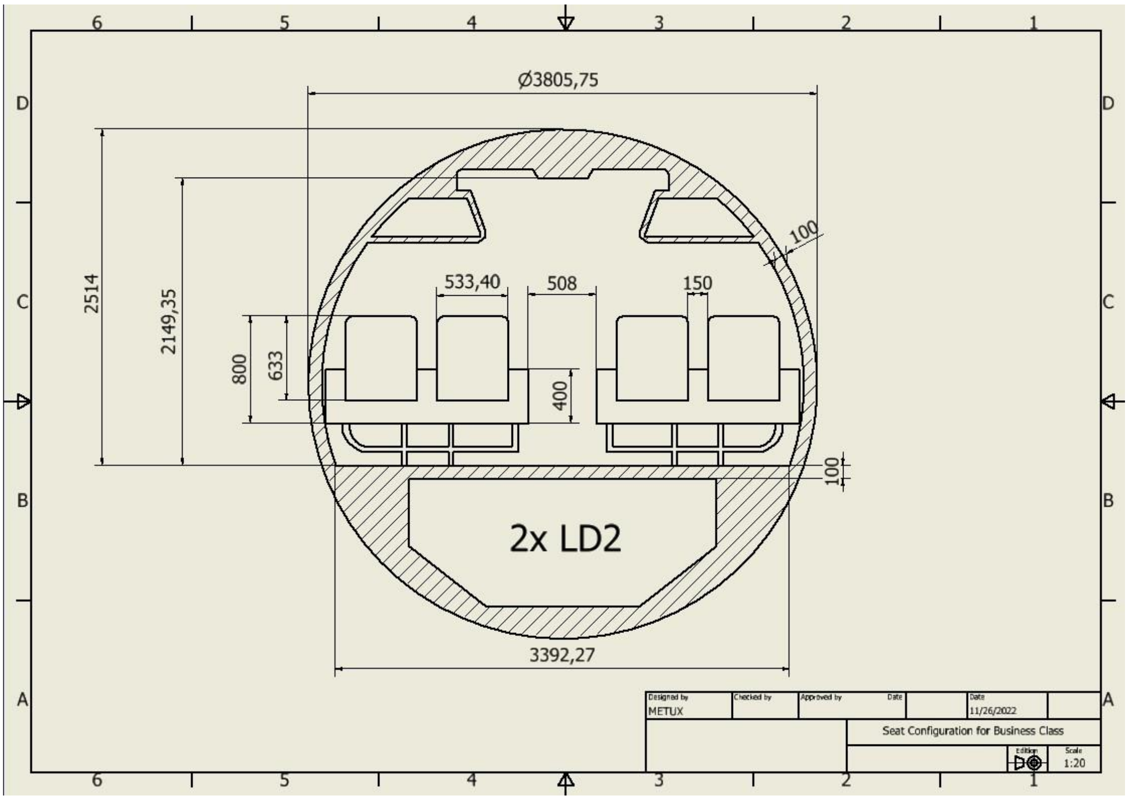


__Figure 3.2.2__ Business class seat configuration on 2D drawing in SI units

Cab ground clearance is 84.65 inches. Considering the size of an average human, this dimension appears to be sufficient. For business, the seat height is also given as 31.50 Inches. Also, economy class seat height is approximately 31.50 Inches.

3.2.3 A floor thickness to locate ceiling of cargo hold

Floor thickness is taken as 10 cm to locate the ceiling of the cargo hold. The cargo hold thickness is 10 cm also shown in 2D.

3.2.4 The standard cargo containers (ULD)

In assignment 2, the take-off gross weight was calculated as 85869 lb. Most passenger aircraft are able to carry 10% of their take-off gross weight.[1] Therefore, the aircraft which we have designed is capable to carry 8586.9 lb. To get a passenger baggage allowance, the density of baggage is chosen as 12.5 $\mathrm{lb/ft^3}$.[14] Therefore, the cargo compartment volume is calculated as 686.952 $\mathrm{ft^3}$. It's equal to 19.452 $\mathrm{m^3}$ in the SI unit.

Table 3.2.1 *Dimensions of Common Cargo Containers (ULD) for Airliners [14]*

Container Type	A (in.)	B (in.)	C (in.)	D (in.)	Volume ft^3
LD-1	61.5	92	60.4	64	173
LD-2	47	61.5	60.4	64	120
LD-3	61.5	79	60.4	64	159
LD-6	125	160	60.4	64	316
LD-8	96	125	60.4	64	243

To choose the best cargo container, LD-2 was selected as the container type after consulting Table 2.1. Because it needs 686.952 $\mathrm{ft^3}$ volume, there are 6 containers in the plane. 6 LD-2 are equals to 720 $\mathrm{ft^3}$.

3.3 Lay-out the floor plan of the cabin

The length of the cabin Lc is proportional to the product of the seat pitch P and the number of rows of seats Nr,

$$L_c \approx c_3(PN_r)^{1+\epsilon}, \tag{3.2}$$

Where L_c is the cabin length, P is the seat pitch, c_3 is the constant accounting for the presence of lavatories, galleys, etc. ,and ϵ is equal to 0.052

Table 3.3.1 *Average Fuselage Data for Typical Commercial Aircraft [14]*

Na	**c3**	***P* (ft)**	***Nr***
2	1.32	2.5	2
6	1.08	2.7	7

When the fuselage cabin length was calculated with the help of Table 3.3.1 and equation 3.2, the fuselage cabin length was found to be 44.467 feet.

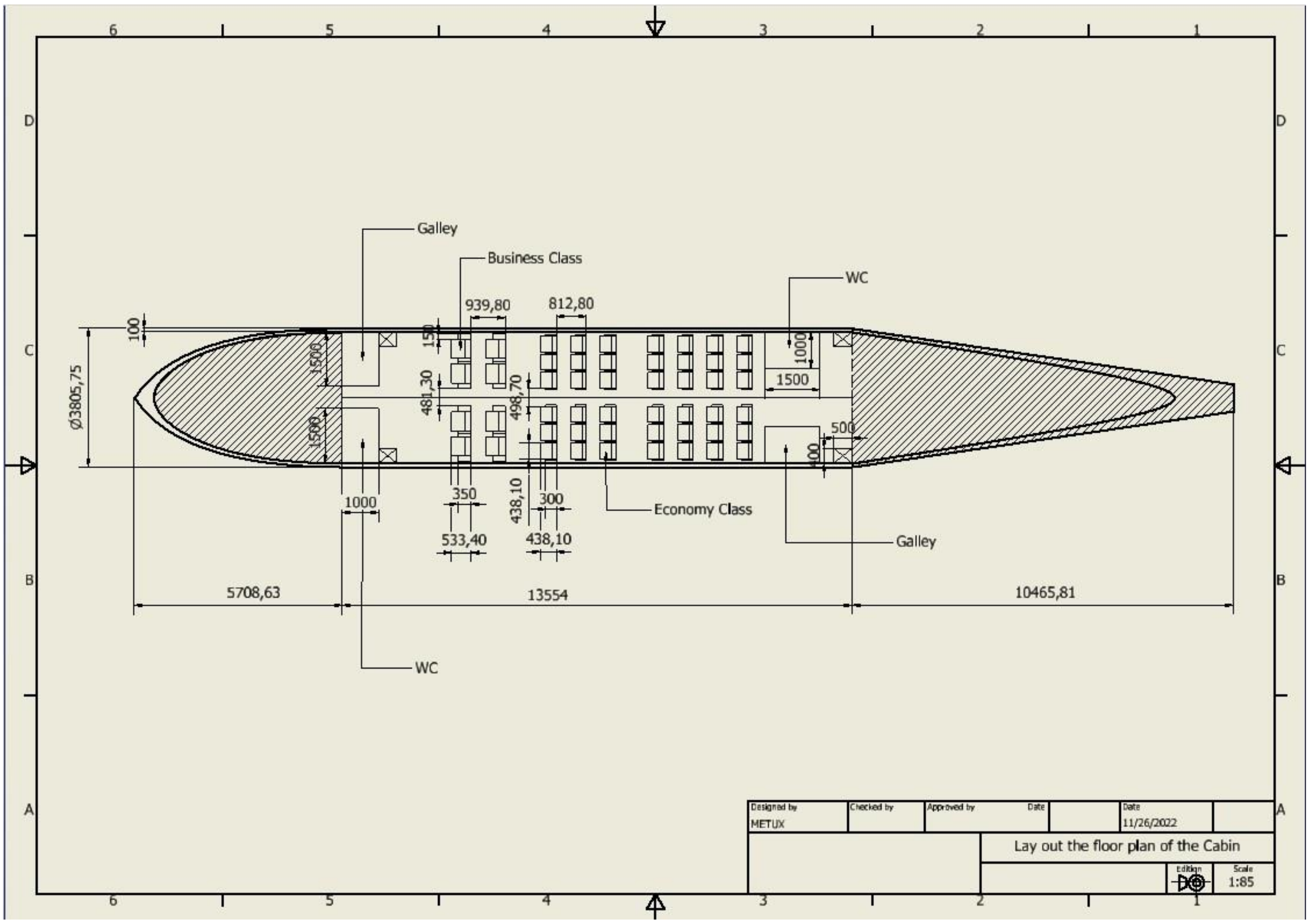


Figure 3.3.1 *Lay-out the floor plan of the cabin (top view of the cabin) in SI units*

While designing 8 seats for business class on this plane, it took place as 42 seats for economy class. There are 8 C-type emergency doors, 2 toilets, and 1 galley area. The length of the cabin section is 44.467 feet, while the diameter is about 12.46 feet. In addition, the maximum cabin height is 7.87 feet.

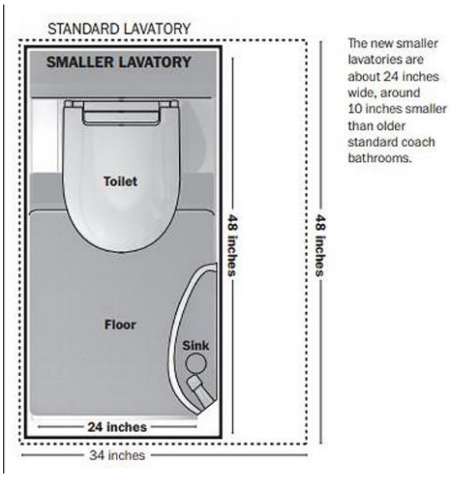


Figure 3.3.2 *Smaller Lavatory for a standard airplane [2]*

Looking at the dimensions of the standard small lavatory for a standard airplane when Figure 3.2 is examined, its length is 48 inches. Looking at its width, it appears to be 34 inches. However, since it is a business jet designed, the lavatory dimensions are kept larger due to comfort. The same is true for Galley.

The dimensions of the lavatories have 59.05 inches (1500 mm) length and 39.37 inches (1000 mm) width. Also, the space reserved for the galley is 59.05 inches (1500 mm) and the width is 39.37 inches (1000 m). The dimensions of emergency doors are the length and width respectively 19.69 inches (500 mm) and 15.74 inches (400 mm). All these dimensions are shown in Figure 3.1 in detail.

3.4 Designing the shape of the nose cone and tail cone

To design the nose and tail cone, the fuselage diameter has to be known. It is known that the ratio of nose cone length to fuselage diameter is between 1.0 and 2.0. [1] Choosing the ratio is equal to 1.5. The fuselage diameter is 121.50 inches (3085.75 mm) because of that the nose cone length is equal to 224.76 inches (5708.63 mm). In addition, the ratio of tail cone length to body diameter is between 2.0 and 3.5. [1] To calculate the tail cone length, the ratio is chosen to be 2.75. Therefore, the tail cone length is equal to 412.04 inches (10465.81 mm). In addition, the tail cone angle was adjusted to 16.84. All dimensions can be seen in Figure 3.1 again.

3.5 Important Parameters

Table 5.1 Important Parameters of Designed Aircraft

	SI unit	**Imperial unit**
Floor Thickness	100 mm	3.94 in.
Cargo Volume	20.39 m^3	720 ft^3
Cargo Container	6x LD2	6x LD2
Fuselage Diameter	3805.75 mm	10.12 ft
Cabin Width	3392.27 mm	11.12 ft
Maximum Height of Cabin	2149.35 mm	7.04 ft
Tail Cone Length	10465.81 mm	34.33 ft
Total Length	29728.44 mm	97.53 ft
Cabin Length	13554.00 mm	44.46 ft
Nose Cone Length	5708.63 mm	18.72 ft
Fineness Ratio	7.929	7.929
Number of Emergency Door	4	4
Number of Business Class Seats	8	8
Number of Economy Class Seats	42	42
Height of Economy seats	800.00 mm	31.50 in.
Height of Business seats	752.84 mm	29.65 in.
Width of Business Class Seats	533.30 mm	21.00 in.
Width of Economy Class Seats	431.80 mm	17.00 in.
Number of WC	2	2
Number of Galleys	2	2
Tail Cone Angle	16.84 °	16.84 °

It is explained how the data in Table 5.1 is found in the above section

Chapter 4. Airfoil and Initial Wing Geometry

4.1 Calculation the Design Lift Coefficient

The aspect of increasing aerodynamic efficiency involves the design lift coefficient. Velocity and altitude are important variables. Depending on the airfoil's mission, the design must be done at its design coefficient. The historical W/S trend by type of aircraft is depicted in the following graph.

Table 4.1 *Historical Trends for Wing Loading [15]*

Typical takeoff W/S		
Historical trends	**lb/ft^2**	**kg/m^2**
Sailplane	6	{30}
Homebuilt	11	{54}
General aviation-single engine	17	{83}
General aviation-twin engine	26	{127}
Twin turboprop	40	{195}
Jet trainer	50	{244}
Jet fighter	70	{342}
Jet transport/bomber	120	{586}

The following equation is used to determine C_l where L indicates lift, A indicates area and V shows freestream velocity, and $indicates$ density.

Since L=W in level flight we can say that (L/A) is equal to (W/S),

$$L = W = \frac{1}{2}\rho V^2 S C_L \tag{4.1}$$

$$\frac{W}{S} = \frac{1}{2}\rho V^2 C_L \tag{4.2}$$

$$C_L = 2\frac{W}{S\rho V^2} \tag{4.3}$$

$\frac{W}{S} = 120\frac{lb}{ft^2}$ from historical trends, $V_{cruise} = 741.14\ \frac{ft}{s}$, and $\rho = 8.02 * 10^{-4}\frac{slug}{ft^3}\ at\ 32.000\ ft.$ With these parameters, $C_L = 0.51$ is calculated.

4.2. Airfoil Selection

Four potential airfoils are chosen. The selection process takes into account the airfoil thickness, stall characteristics, Mach number, and Reynolds number. NACA 6-digit series are searched to order decrease drag and improve laminar flow over the airfoil. These NACAs are chosen: NACA 65(3)-618, NACA 16009, NACA 747A315, and NACA 64008A. Two of them are thin and are typically used at the tip of the plant, while the other two are thick and are typically utilized at the root. Additionally, the $Cl_{max}\ /\ Cd_{max}$ ratios play a crucial role in choosing. Selected airfoils match the $L_{max}/D_{max} = Cl_{max}\ /\ Cd_{max}$ ratio from our prior analysis.

Airfoil 1 NACA 63210 (Thin)

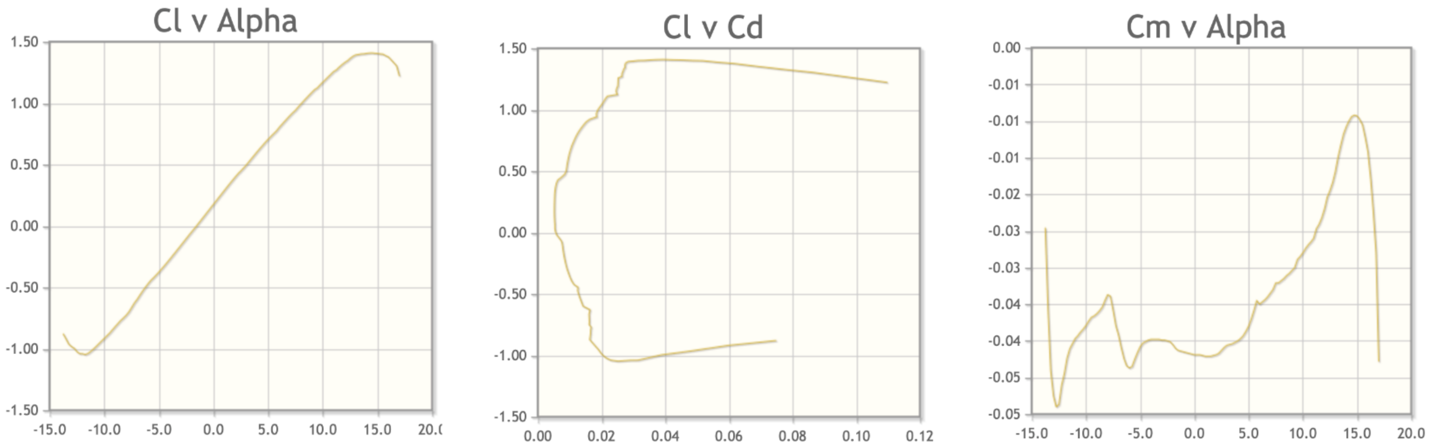


Figure 4.2.1 C_l *vs* α, C_l *vs* C_d, and C_m *vs* α *graph for NACA 63210 [16]*

Airfoil 2 NACA 65(3)-618 (Thick)

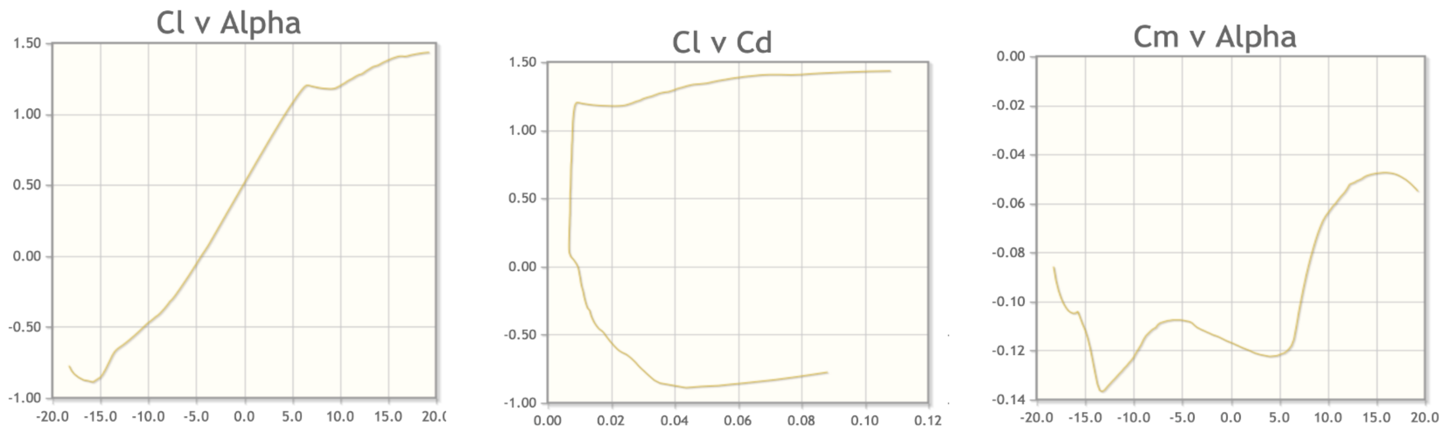


Figure 4.2.2 C_l *vs* α, C_l *vs* C_d, and C_m *vs* α *graph for NACA 65(3)-618 [16]*

Airfoil 3 NACA 64-012A (Thin)

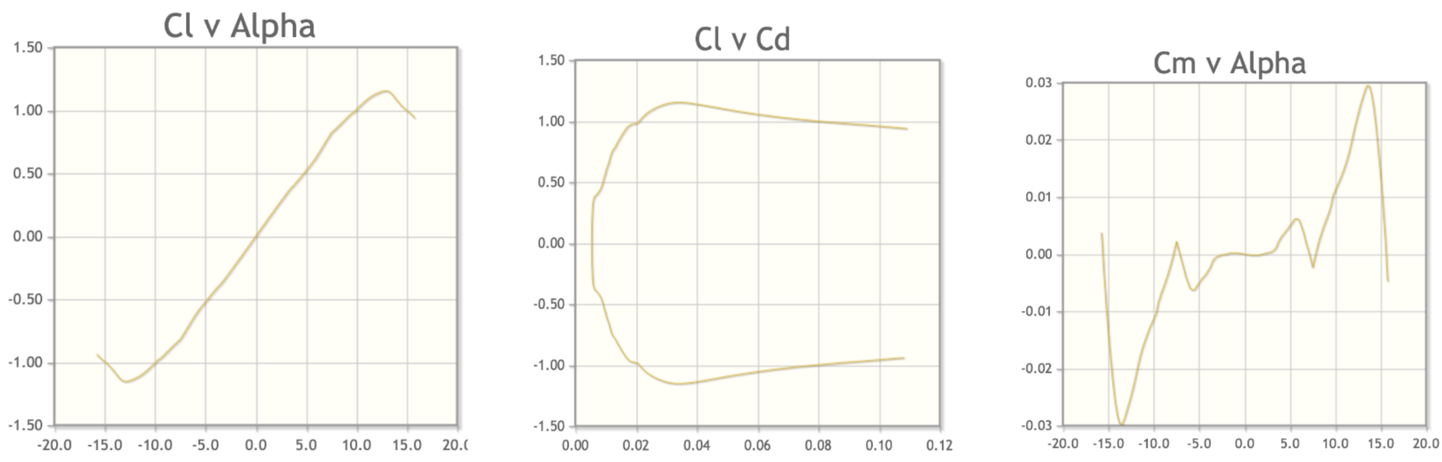


Figure 4.2.3 C_l *vs* α, C_l *vs* C_d, and C_m *vs* α *graph for NACA 64-012A [16]*

Airfoil 4 NACA 747A315 (Thick)

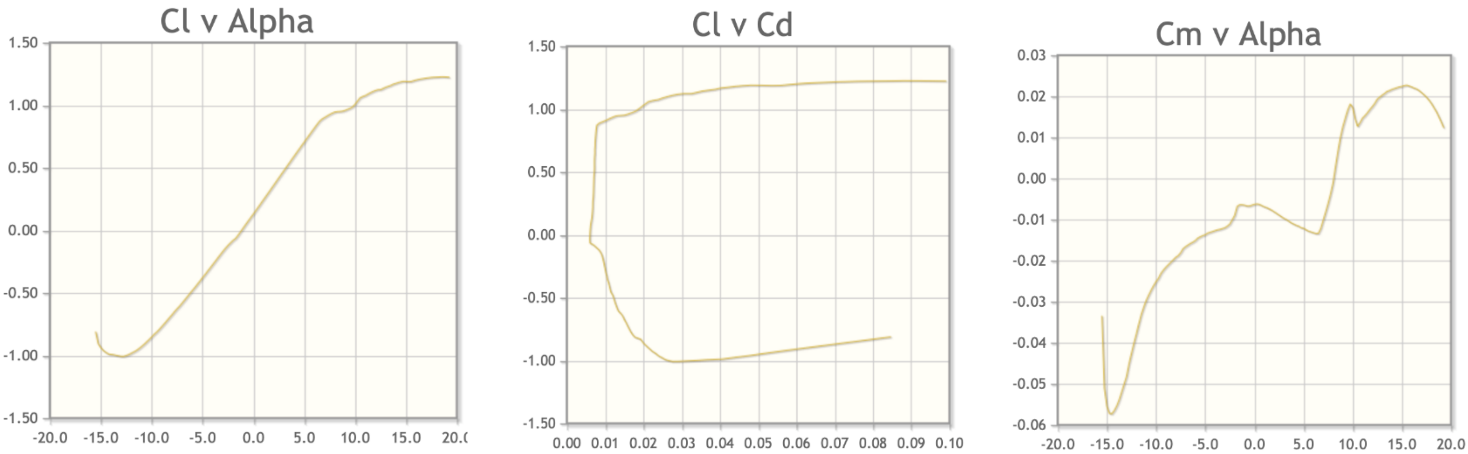


Figure 4.2.3 *C_l vs α, C_l vs C_d,* and *C_m vs α graph for NACA 747A315 [16]*

Table 4.2.1 *Chosen Airfoil Characteristics Parameters*

Airfoil	$C_{l,max}$	$C_{l\alpha}$	C_{m0}	α_{stall}
NACA 63210	1.410	0.110	-0.044	14.55
NACA 65(3)-618	1.430	0.114	-0.115	19.30
NACA 64-012A	1.155	0.110	0	13.10
NACA 747A315	1.125	0.105	-0.005	18.52

First, the Reynolds number is taken as 1,000,000 for all types of airfoil analysis. To determine $C_{l,max}$, $C_{l\alpha}$, $C_{m0}$and α stall values graphs which are C_l vs Alpha and C_m vs Alpha are used. Stall angle is a limitation to get maximum lift coefficient so $C_{l,max}$ can be determined by $C_{l\alpha}$ Alpha graph where an airfoil starts to stall. Secondly, $C_{l\alpha}$ vs lift curve slope and it can be calculated thanks to C_l vs Alpha. If the tangent line is drawn from 0 angles of attack, a lift curve slope can be obtained. Thirdly, C_{m0} where the angle of attack equals zero on the C_m vs alpha graph. Lastly, the stall angle is gained at the point where the lift coefficient begins to drop down.

4.3 Initial Wing Geometry

The design will use two airfoils. It is chosen to use the NACA 65(3)-618 airfoil at the wing's root. First of all, compared to other airfoils, it has the greatest maximum lift coefficient. The second explanation has to do with its thickness. In other words, by utilizing this thicker airfoil, fuel tanks can be placed inside the wing. It is chosen to choose NACA 63210, a comparable thinner airfoil, for the wing's tip. To determine the aspect ratio, historical trends and data from competitor aircraft are used. The following graph shows historical aspect ratio trends for several aircraft types.

Table 4.3.1 *Aspect ratio vs aircraft [15]*

	Equivalent Aspect Ratio = αM_{max}^{C}	
Jet Aircraft	α	C
Jet trainer	4.737	-0.979
Jet fighter (dogfighter)	5.416	-0.622
Jet fighter	4.110	-0.622
Military cargo/bomber	5.570	-1.075
Jet transport	7.50 to 10	0

Given that the intended aircraft is a jet transport, the values for and C obtained from the figure are, respectively, 7.50 to 10 and 0. And from the previous assignment, the M max for the intended airplane is estimated as 0.76.

$$M = \frac{v}{a} \tag{4.4}$$

where $a = 294.9\ m/s\ at\ 40.000\ ft$, and v= 225.9 m/s

$$M = \frac{225{,}9}{294{,}9} = 0.766$$

$$AR = a * M_{max}^{C} = 7.50 * 0.77^{0} = 7.50 \qquad (2)$$

$$AR = a * M_{max}^{C} = 8.00 * 0.77^{0} = 8.00 \qquad (3)$$

$$AR = a * M_{max}^{C} = 9.00 * 0.77^{0} = 9.00 \qquad (4)$$

$$AR = a * M_{max}^{C} = 10 * 0.77^{0} = 10.00 \qquad (5)$$

So, the aspect ratio range is between 7.5 and 10.

Table 4.3.2 *Aspect ratio of the competitor aircraft from chapter 1*

	Embraer ERJ-145 LR	**Fokker 70 Fellowship**	**Bombardier CRJ550**	**Boeing 737-500**
AR	7.8	8.45	7.6	9.16

With respect to airplanes that are like the plane designed, the optimum value for the aspect ratios chosen is 8.25. The estimation of aspect ratio for desired aircraft competitor aircraft and historical data are used. To get a proper guess, the aspect ratio of aircraft should be reasonable for both data. Since competitor aircraft are in the same class as the desired aircraft, the average aspect ratio that is calculated as 8.25 can be taken as an aspect ratio for design. According to the historical trends, the jet transport class's C value equals zero so the aspect ratio only depends on a value. Therefore, the aspect ratio range is between 7.5 and 10. Also, as the aspect ratio decreases, lift loss, and wing tip vortex will be decreased. On the other hand, the wing stall angle decreases due to the high aspect ratio. As a result, historical trends and competitor aircraft supply a considerable range for aspect ratio estimation and the aspect ratio is taken at 8.25 because it is a proper value to have a high stall angle and stability control for an aircraft. Sweep angle provides lateral stability and delays the sweep back profile chosen for jet transport. Below graph which is aspect ratio vs quarter chord sweep is used to determine the sweep angle for designing aircraft.

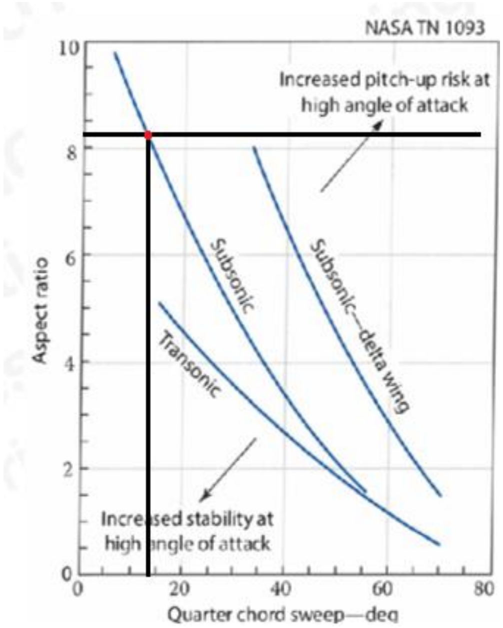


Figure 4.3.2 *Aspect ratio vs quarter chord sweep [15]*

The aspect ratio is determined as 8.25 for desired aircraft and Mach number equals to 0.77 for cruise so it is in subsonic flow region. According to figure 4.3.2, quarter chord sweep angle is found 14 degrees with respect to 8.25 aspect ratio. Since it is quarter chord aspect ratio, aspect ratio of aircraft is determined thanks to below formula.

$$tan\,(\wedge) = tan(\wedge_{\frac{c}{4}}) + [\frac{(1-\lambda)}{AR(1+\lambda)}] \tag{4.5}$$

where $\lambda = 0.25$, $\wedge_{\frac{c}{4}}$ *=14 degrees, AR=8.25*

$$\Lambda = tan^{-1}\left(tan(14) + \left[\frac{(1-0.25)}{8.25(1+0.25)}\right]\right) = 18.18°$$

As a result, it is not satisfied for the performance and appearance for design. To find effective sweep angle, competitor aircrafts' sweep angles is considered.

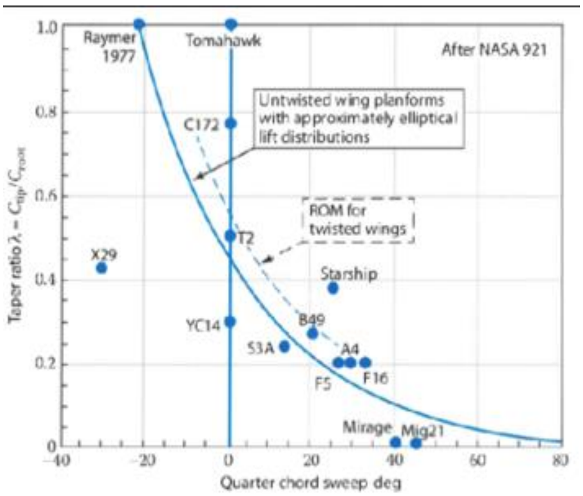
Table 4.3.3 *Sweep angle of the competitor aircraft*

	Boeing 737-500 VIP	**Comac ARJ21-700**
Wing sweep	25 degrees	25 degrees

According to Table 4.3.3, aircraft up in the jet transport class generally have a sweep angle of around 25 degrees. Therefore, picking up sweep angles close to the aircraft is more effective for design. However, the pitch pitch-up characteristics of an aircraft are directly related to the sweep angle. The pitch-up risk at a high angle of attack is increasing as the aspect ratio increases with respect to figure 15. Moreover, while determining the aircraft sweep angle, take the aspect ratio smallest value and high sweep angle to have low pitch-up risk. To sum up, with respect to the found angle which is 18.18 degrees, and competitor aircraft, the sweep angle is determined as 22 degrees.

One of the main aspects of the taper ratio is the determination of lift distribution along the span of the wing, shown as b. When the lift distribution along the span shapes elliptically, drag due to the lift decreases until the minimized limit value. This case of drag specification is called induced drag. In order to reach the induced drag existence, there are requirements for the determination of elliptical lift distribution. These are firstly that all chord lines are to be parallel to each other and the second requirement is that the lift coefficient is to be constant over the span.

The taper ratio can be found in figures 17&18, the relation between quarter chord sweep angle and taper ratio. Lower taper ratio has strengths and weaknesses. Lighter wing structure potentiality, decrement of bending moment, and movement of the center of pressure to root can be counted as the beneficial aspect of lower taper ratio, λ. On the other hand, it costs undesirable stall characteristics. Therefore, taper ratio can be defined as 0.25 corresponding the sufficient sweep angle of 22 degree, shown in the Figure 4.3.3



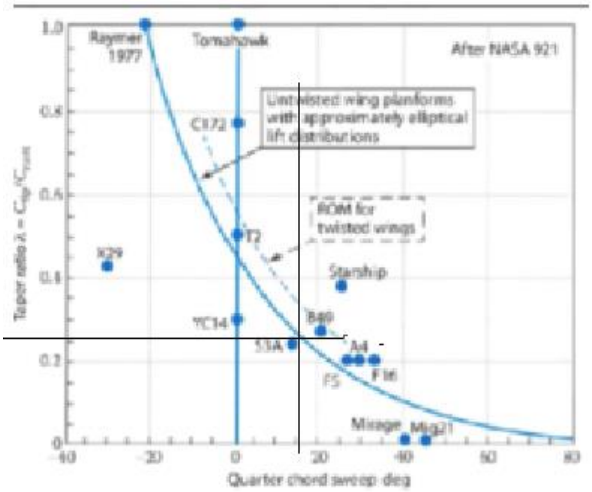


Figure 4.3.3 *Quarter chord sweep vs taper ratio graph* ***Figure 4.3.4*** *Taper ratio in chosen sweep angle*

Figures 4.3.3 & 4 shows the relation of taper ratio with quarter chord sweep in graph. It can be observed that taper ratio and sweep angle has transversely proportional to each other.

4.4 Twist value, Wing incidence angle & Dihedral angle

4.4.1 Twist Angle

Several twist types such as geometric twist, aerodynamic twist, and linear twist. Mainly, wing twist has the advantage of prevention of wingtip stall. Geometric twist, specifically, has two terms to know to assign the twist angle. First, wash-in means that the angle of attack at tip is bigger than the one in the root of the wing, whereas washout is transversely the meaning of wash-in is that angle of attack at the root is bigger than the angle at the tip. Washout provides significant improvement for performance, so it is chosen to design the wing in negative twist, which is a washout. According to Raymen (1992), twist angle is to be between 0-5 degrees. Hence, 3 degree of taper ratio seems sufficient choice to reach safer stall characteristics.

4.4.2 Wing Incidence Angle

Wing incidence angle is the pitch angle the of wing with respect to the fuselage. In other words, it is the difference as the angle between the root and tip of the wing. It is aimed to minimize drag during cruise and incidence angle has a great role in the achievement of the aim. Raymen (1992) mentions that wing incidence angle can be assumed 2 degrees for the most initial design work of general aviation and homebuilt aircraft. Therefore, 2 degrees of incidence angle is determined.

4.4.3 Dihedral Angle

The most important effect of dihedral is to provide lateral stability, also called as roll stability due to production of inverse roll moment. Also, the dihedral is more effective in higher wings. On the other hand, excessive dihedral causes 'Dutch Roll' is that roll, and yaw motions are repeatedly continued.

Table 4.4.1 *Dihedral Guideline in several wing positions [15]*

	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 4.4.1 shows dihedral angle values in unswept, subsonic and supersonic swept wings in low, mid and high wing positions. In this report, designed wing is swept wing in subsonic region and positioned in the LOWER side of the fuselage. Hence, dihedral angle range is 3 to 7 degrees. 22 degree of sweep angle corresponds as +2.2 degrees this comes the conclusion of 5.2 degrees of dihedral angle. It can be approximately to 5 degrees of dihedral angle.

4.5 High Lift Devices

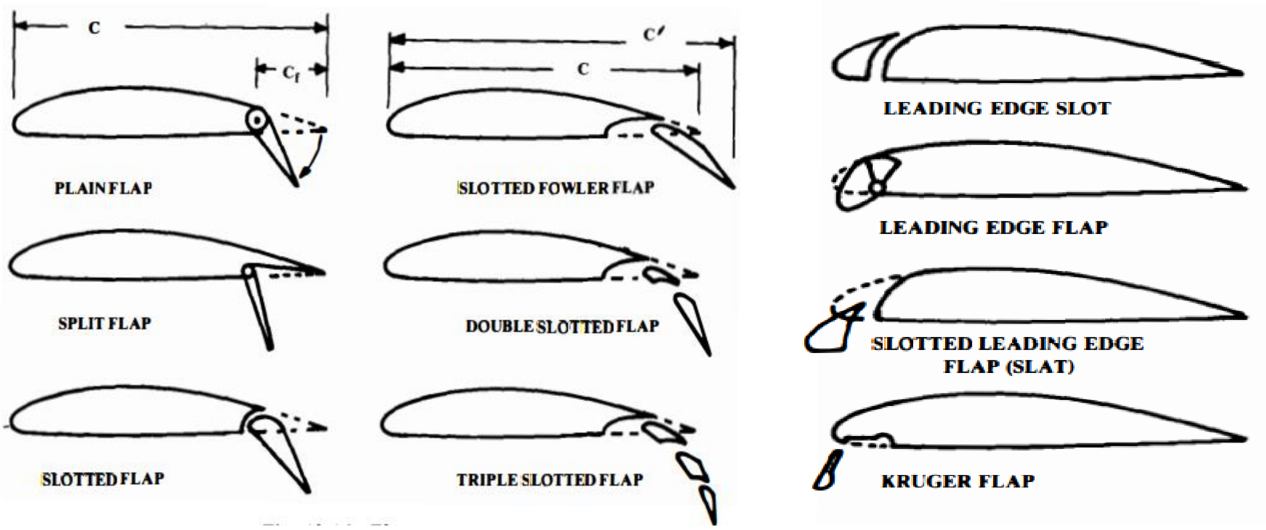


Figure 4.5.1 *Leading and Trailing edge Configurations [15]*

Leading-edge slats; these are employed to raise the profile's maximum lift coefficient and stall angle. The main use of powered slats is for landing and maneuvering. Since they provide less lift and less drag, they are also employed for takeoff.

Krueger flaps; they are high-lift components that can be attached to all or part of the leading edge of some types of aircraft wings. The Krueger flap has a similar aerodynamic effect to the slats. However, the method of provision varies. The Krueger flap is affixed to the underside of the wing and hinged at the leading edge. The actuator causes the flap to travel from the front of the wing to the underside, causing the wing to become more curved. More lift will occur from this. As a result, especially at low speeds, the wing's area, and lift increase.

Trailing edge flaps; they are applied to lower the stall speed of an aircraft's wings in the form of slotted flaps. This is because air is forced upward beneath the wings, and the vortex helps to improve the boundary layer's ability to resist separation (Turbulent flow). They lessen flow separation as a result.

Fowler flaps; they include a split flap that folds after sliding back slightly. Following the curve, this results in an increase in the tendons (and subsequently the wing's surface). Both takeoff (half extension for the best lift) and landing performance (full extension for ideal lift and drag) can be enhanced by doing this.

Most large aircraft have this kind of flap or one of its variations. Therefore, having considered the advantages and disadvantages of each high-lift device, we chose the Fowler Flap and Krueger Flap.

4.6 Summary of Wing geometry parameters

Table 4.6.1 *Wing geometry parameters of the designed aircraft*

Wing geometry parameters	Values
Aspect ratio	8.25
Wing sweep	22
Taper ratio	0.25
Twist value	3
Wing incidence	2
Dihedral angle	5
High Lift Devices	the Fowler Flap and Krueger Flap

Table 4.6.1 shows the values of the important parameters for the blade. Some of these values are aspect ratio "AR", wing sweep, and taper ratio. In this project, the Fowler flap and Krueger flap were used as high-lift devices.

2.3 Discussing and comparing airfoil and initial wing geometry

In Chapter 4, parameters and design data are calculated and discussed. One of the most parameters to understand the air flight is lift coefficient, shown as C_L. With design data and historical trends, C_L is found as 0.51 and the value is in sufficient region by the result of comparison with similar airplanes. Airfoil selection is made under the consideration of stall characteristics, weight, critical Mach number and the Reynolds number. These parameters are taken into account and 4 NACA 65(3)-618 airfoil is chosen. It has a lift coefficient of 1.430 and is a thick airfoil. The reason of the selection is that the airfoil for the root has the biggest maximum lift coefficient compared with other airfoil candidates. The tip airfoil is selected depending on thickness due to the advantage of extra fuel tank capacity is provided by thicker airfoil. Next, aspect ratio is estimated as 8.25. This value is selected due to calculation depending on historical trends and indirectly to Mach number. For the high-lift devices part, after weighing the benefits and drawbacks of each high-lift mechanism, we decided on the Fowler Flap and Krueger Flap. As a result, the wing's area, and lift increase, particularly at low speeds.

Chapter 5 Performance Parameters Estimation (T/W, W/S)

5.1 Establish necessary requirements

5.1.1 Stall Speed

Firstly, to determine stall speed requirements, FAR/EC specifications are used. FAR/EC specifications states that $V_{approach}$ should be 1,3 times bigger than V_{stall}. To calculate V_{stall}, Jet Transport $V_{approach}$ was researched and it is between 240-285 km/h. $V_{approach}$ is taken as a maximum value which is 260 km/h.

$$V_{approach} = 1.3 * V_{stall} \tag{5.1}$$

$$V_{stall} = 260/1.3 = 200 \text{ km/h} = 182.27 \text{ ft/s}$$

5.1.2 Landing Distance

Nominal landing distances for aircraft in the database of 50 commercial airliners as a function of takeoff weight is shown in Figure 5.1.1. According to Figure 1, for aircraft with WTO > 100,000lb the landing distance lies in the range from 4000 ft (1220 m) to 7500 ft (2286 m). An approximation to the behavior of the landing distance is shown.

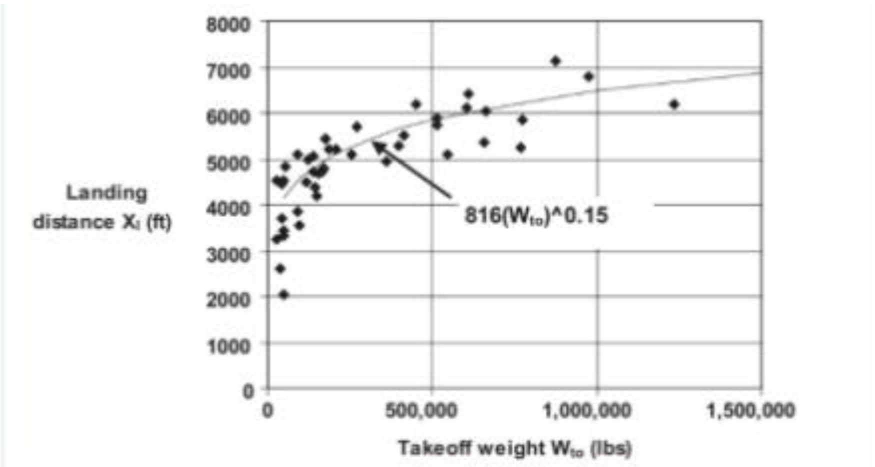


Figure 5.1.1 *Nominal landing distances for 50 commercial airliners [17]*

$$X_{Landing} = 816 * W_0^{0.15} \tag{5.2}$$

$$X_{Landing} = 816 * 85869^{0.15} \tag{5.3}$$

$$X_{Landing} = 4485.03\ ft$$

5.1.3 Takeoff Distance

Takeoff Parameter for jet aircraft is calculated using by where $\left(\frac{W}{S}\right)_{Takeoff}$ is takeoff wing loading, σ is the ratio of the local density to sea-level density, $C_{L,TO}$ is the takeoff lift coefficient and $\frac{T}{W}$ is thrust to weight ratio.

$$TOP = \frac{\left(\frac{W}{S}\right)_{Takeoff}}{\sigma C_{L,TO}\left(\frac{T}{W}\right)}, \tag{5.4}$$

where $\left(\frac{W}{S}\right)_{Takeoff}$ is takeoff wing loading, σ is the ratio of the local density to sea-level density, $C_{L,TO}$ is the takeoff lift coefficient and $\frac{T}{W}$ is thrust to weight ratio. $\left(\frac{W}{S}\right)_{Takeoff}$ is $120\ lb/ft^2$, the ratio of the local density to sea-level density, $\sigma = 0.2978$. $C_{L,TO} = \frac{C_{L,max}}{1.21}$ where $C_{L,max}$, the maximum lift coefficient is 1.430 from Chapter 4.

So, $C_{L,TO} = \frac{1.430}{1.21} = 1.18$, and $\left(\frac{T}{W}\right) = 1.15\ ft/s^2$ from previous assignment.

$$TOP = \frac{120}{(0.2978)(1.18)(1.15)} = 215$$

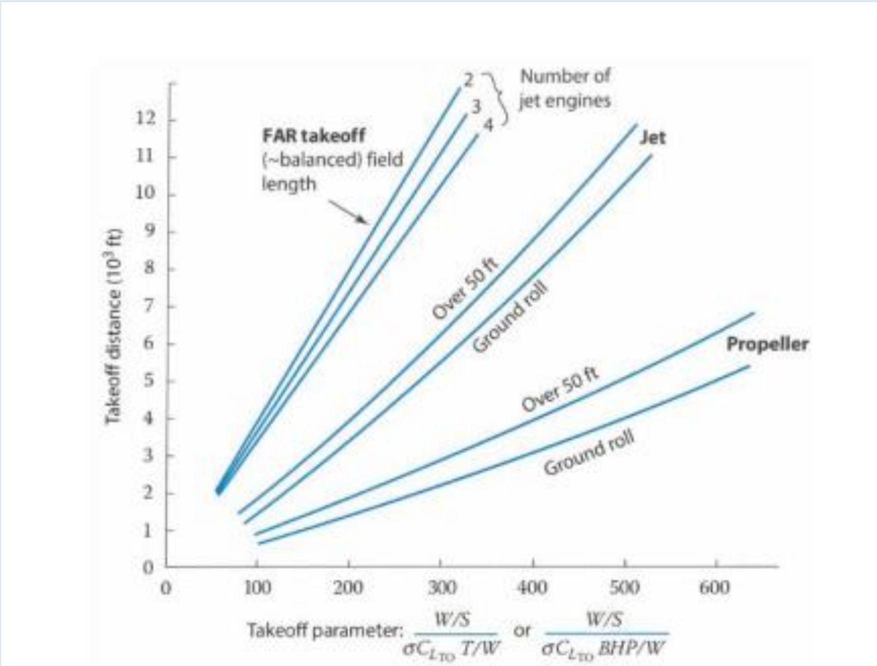


Figure 5.1.2 *Takeoff Distance vs. TOP [17]*

The landing distance is determined by TOP to be roughly 9,000 feet as displayed in Figure 5.1.2

5.1.4 Maximum Ceiling

According to FAR/EC standards and normal historical trends in typical weather situations, the typical ceiling values for jetliners and business jets are approximately 13000 m and 16000 m, respectively. Considering these elements and the study we conducted on our competitors, we came to the conclusion that our design's ceiling value may go as high as 13000 m. (42000 ft).

Table 5.1.1 *Values of Requirements of Designed Aircraft [17]*

Parameters	Values
Stall Speed	182.27 ft/s
Landing Distance	4485.03 feet
Take-off Distance	9000 feet
Maximum Ceiling	42650.92 feet

5.2 Thrust-to-weight ratio (T/W) using historical trend and historical data

Table 5.2.1 *Typical Values of Thrust-to-Weight Ratio [17]*

Aircraft type	Typical Installed (T/W)
Jet Trainer	0.4
Jet Fighter(dogfighter)	0.9
Jet Fighter(other)	0.6
Jet Cargo/Bomber	0.25
Jet Transport (higher value for fewer engines)	0.25-0.40

The value of T/W is chosen at 0.35 after accounting for the jet transport category in Table 5.2.1 and the competitor study in Chapter 1.

5.3 Wing loading (W/S) according to the following requirements

5.3.1 Historical Trends

Table 5.3.1 *Historical Trends for Wing Loading [17]*

	Typical Take-off W/S	
Historical Trends	lb/ft^2	kg/m^2
Jet trainer	50	224
Jet fighter	70	342
Jet transport/bomber	120	586

Considering the historical trend in Table 5.3.1, the W/S value has been chosen as 120 lb/ft^2 in the jet transport/bomber category.

5.3.2 Stall speed in landing configuration

Stall speed is given by following formula,

$$(W/S)_{stall} = \frac{1}{2}\rho V_{stall}^2 C_{L,max}, \tag{5.5}$$

where ρ is the density at sea level, V_{stall} is the stall velocity, and $C_{L,max}$ the maximum lift coefficient. According to the sea level standard, the density is equal to $1.225 kg/m^3$. The stall velocity taken from section 1.1 is calculated as 66.87 m/s. NACA 65(3)-618 was chosen as the aerofoil in Assignment 4. $C_{L,max}$ for this airfoil was found to be 1.430 with plain wing, no flaps.

When all its values are replaced in Equation 5.5,

$$(W/S)_{stall} = \frac{1}{2}(1.225)(66.87)^2(1.430) = 399.24\ kg/m^2 = 81.77\ lb/ft^2$$

$C_{L,max}$ is between 2-3 (~2.4) for a commercial transport aircraft with flaps and slats.[17]
When $C_{L,max}$ is equal to 2.4,

$$\left(\frac{W}{S}\right)_{stall} = \frac{1}{2}(1.225)(66.87)^2(2.4) = 670.056\frac{kg}{m^2} = 137.23\ lb/ft^2$$

5.3.3 Landing Distance

Wing loading for landing requirements is calculated as

$$(W/S)_{landing} = \frac{1}{2}\rho V_{landing}^2 C_{L,landing}, \tag{5.6}$$

where $Cl_{L,land}$ is the lift coefficient in the landing configuration, and $V_{E,land}$ is the equivalent landing speed.
The landing speed depends on the stall speed,

$$V_{E,land} = k\ V_{E,stall} \tag{5.7}$$

k is between 1.2 to 1.3. In this project, it is chosen 1.25.

$$V_{E,land} = 1.25\ V_{E,stall} = 1.25\ (66.87) = 83.59\ m/s.$$

The landing speed depends with equivalent landing speed,

$$V_{E,land} = V_{land}\sqrt{\sigma} = V_{land}\sqrt{\rho/\rho_{sl}} \tag{5.8}$$

where σ is ratio of the local density to sea level density. σ is assumed almost 1 for landing.

$C_{L,takeoff}$ depends on $C_{L,max}$,

$$C_{L,takeoff} = \frac{C_{L,max}}{1.21} = \frac{2.4}{1.21} = 1.9835\ . \tag{5.9}$$

The maximum lift coefficient on takeoff is approximately 80% of the landing value. Then,

$$C_{L,landing} = \frac{C_{L,takeoff}}{0.8} = \frac{1.983}{0.8} = 2.48.$$

Solving wing loading for landing with Equation 5.6,

$$(W/S)_{landing} = \frac{1}{2}(1.225)(83.59)^2(2.48) = 1081.93\frac{kg}{m^2} = 221.60\ lb/ft^2.$$

5.3.4 Cruise Speed

Another wing loading value to consider is in cruise flight conditions. Under cruise conditions, lift equals weight and thrust equals drag.

Dynamic pressure formula is given by,

$$\bar{q} = \frac{1}{2}\rho V_{cruise}{}^{2} \tag{5.10}$$

where ρ is density at cruise, and V_{cruise} is cruise speed. The density at cruise is equal to 0.3639 kg/m^3 at 11,000 km. The cruise speed is equal to 225.97 m/s from assignment 2.

Wing loading for cruise is following formula,

$$(W/S)_{cruise} = \bar{q}\sqrt{\pi A e C_{D0}/3} \tag{5.11}$$

where C_{D0} is parasite drag, form drag or zero-lift drag, e is the Oswald span efficiency factor, and $\bar{q}$ is dynamic pressure.

C_{D0} is the zero lift drag coefficient. Its approximate values are 0.015 for a jet aircraft. Also, 'e' is the Oswald span efficiency which is approximately 0.6 to 0.8 for a fighter and 0.8 for other aircraft. [17]

When all the parameters found in Equation 5.10 and 5.1.1 are written,

$$(W/S)_{cruise} = \left(\frac{(0.5)(0.369)(225.97)^2}{9.81}\right)\sqrt{\frac{\pi(8.25)(0.8)(0.015)}{3}} = 328.30\frac{kg}{m^2} = 67.24 lb/ft^2$$

5.3.5 Sustained turn with the positive limit load factor (n)

In sustained turn, the speed is maintained,

$$n = (\frac{T}{W})(\frac{L}{D}) \tag{5.12}$$

T/W is found as 0.35 from the historical trend. Also, L/D is considered as 14 for the maximum. Because of that n has been found 4.9.

The wing loading during a sustained turn is given by

$$(W/S)_{sustained} = \frac{\bar{q}}{n}\sqrt{\pi A e C_{D0}/3} \tag{5.13}$$

Solving equation 5.13,

$$(W/S)_{sustained} = \left(\frac{(0.5)(0.369)(225.97)^2}{9.81 * 4.9}\right)\sqrt{\frac{\pi(8.3)(0.8)(0.015)}{3}} = 66.07\frac{kg}{m^2} = 13.53\ lb/ft^2.$$

5.3.6 Summary of the wing loading with different situation

Table 5.3.2 *Summary of the wing loading with different situation*

	Historical Trends	**Stall Speed**	**Landing Distance**	**Cruise Flight**	**Sustained Turn**
$(\boldsymbol{W/S})$ $[\boldsymbol{lb/ft^2}]$	120	137.23	221.60	67.24	13.53

Table 5.3.2 shows the wing loading calculations for Historical Trends, Stall Speed Landing Distance, Cruise Flight, and Sustained Turn.

5.4 Wing Loading for Take-off Conditions

5.4.1 Historical Trends

Considering the historical trend in Table 5.1.1, the W/S value has been chosen as 120 lb/ft^2 in the jet transport/bomber category.

5.4.2 Stall speed

Converting landing wing load to takeoff wing load, the equation given below is used.

$$\left(\frac{W}{S}\right)_{(takeoff)} = \left(\frac{W_0}{S}\right) = \left(\frac{W}{S}\right)_{landing}\left(\frac{W_0}{W_{landing}}\right) \tag{5.14}$$

$$\left(\frac{W}{S}\right)_{(takeoff)} = \left(\frac{W}{S}\right)_{stall}\left(\frac{W_0}{W_{landing}}\right) \tag{5.15}$$

It can be concluded that

$$\left(\frac{W}{S}\right)_{landing} = \left(\frac{W}{S}\right)_{stall} \tag{5.16}$$

Wing loading at stall condition is calculated in Part 5.3 $\left(\frac{W}{S}\right)_{stall} = 137.23\ lb/ft^2$.

Where

$$(M_f)^{-1} = \left(\frac{W_0}{W_6}\right) \tag{5.17}$$

In Chapter 2, M_f is calculated as 0.532. By using this calculation, $\frac{W_0}{W_{landing}}$ is founded as 1.879.

As the complete computation of the equation, wing loading of takeoff is to be 257.87 lb/ft^2.

5.4.3 Landing Distance

In the equation (5.18), the unknown term can be find by relation with entire mission fraction,$(M_f)^{-1}$.

$$\frac{W_0}{W_{landing}} = (M_f)^{-1} \tag{5.18}$$

And the equation becomes

$$C = \left(\frac{W}{S}\right)_{landing} (M_f)^{-1} \tag{5.19}$$

In part 5.3, wing loading for landing is founded as $1081.93\ kg/m^2$. The result of wing loading in the effect of landing distance, in landing condition is $2032.95\ kg/m^2$ by the calculation shown below.

$$(W/S)_{landing-takeoff} = 1081.93 * 1.879 = 2032.95\ kg/m^2 = 416.38\ lb/ft^2.$$

5.4.4 Cruise flight effect

Takeoff wing loading and cruise wing loading relation is

$$(\frac{W}{S})_{takeoff} = \left(\frac{W}{S}\right)_{cruise}\left(\frac{W_0}{W_{cruise}}\right) \tag{5.20}$$

Where

$$\left(\frac{W_0}{W_{cruise}}\right) = \frac{W_0}{W_3} = \ (\frac{W_3}{W_2} * \frac{W_2}{W_1} * \frac{W_1}{W_0})^{-1} \tag{5.21}$$

And

$$\frac{W_0}{W_3} = (0.532 * 0.980 * 0.995)^{-1} = 1.928$$

Wing loading at cruise flight equals to $328.30\ kg/m^2$, as it is founded in Part 5.3. As a result, takeoff wing loading, $\left(\frac{W}{S}\right)_{cruise}$, equals to 632.96 $kg/m^2 = 129.64\ lb/ft^2$. .

5.4.5 Sustained Turn

The difference with sustained turn and cruise is that loiter time should be included. Thus, weight fraction of additive loiter time period must be added to weight ratio. This refers to (W_5/W_0) and the values are assigned by assignment 2.

$$\left(\frac{W_5}{W_0}\right) = (\frac{W_0}{W_{cruise}}) \tag{5.22}$$

Wing loading in sustained turn condition during takeoff can be found by the equation below by using the relation shown above.

$$(W/S)_{sustained\ turn-takeoff} = \left(\frac{W}{S}\right)_{sustained\ turn} (\frac{W_0}{W_{cruise}}) \tag{5.23}$$

Wing loading for sustained turn is equals to 259.65 $\frac{kg}{m^2} = 259.65\ lb/ft^2$.

5.4.6 Summary of Takeoff Wing Loading

Table 5.4.1 *Summary of the wing loading with different situation*

	Historical Trends	**Stall Speed**	**Landing Distance**	**Cruise Flight**	**Sustained Turn**
$(\boldsymbol{W/S})$ $[\boldsymbol{lb/ft^2}]$	120	257.87	416.38	129.64	53.18

Table 5.4.1 shows the wing loading in takeoff calculations for Historical Trends, Stall Speed Landing Distance, Cruise Flight, and Sustained Turn. The lowest value should be selected due to the fact that the wing would be large enough to carry all flight condition of weights.

5.5 Checking the suitability of T/W

5.5.1 Takeoff Distance

First of all, take off distance can be calculated in a different way. The formula given below is used to check whether chosen ($\frac{W}{S}$) is suitable for ($\frac{T}{W}$).

$$(\frac{W}{S})_{takeoff} = (TOP) * \sigma * C_{L,takeoff} * \left(\frac{T}{W}\right). \tag{5.24}$$

Where, the σ density at 11,000 km altitude ,the takeoff parameter (TOP) was obtained in part 1.3 as 297, the takeoff lift coefficient is 1.9835 $from\ part$ 3.3., and wing loading is seleceted as 58 lb/ ft^2. As a result thrust to weight ratio can be determined with all these values as it follows below.

$$\left(\frac{T}{W}\right) = \frac{58}{297 * 0.2978 * 1.98} = 0.33$$

As a result, the thrust the weight ratio is in the range of jet transport class. Also, it can be concluded that thrust weight ratio is taken 0.35 in the beginning and calculated value is 0.33. Therefore, it is suitable for design.

5.5.2 Maximum Ceiling

Maximum ceiling formula is written below and it is used to calculate the wing loading to get a certain maximum ceiling.

$$\frac{W}{S} = \frac{\left(\left[\left(\frac{T}{W}\right) - G\right] \pm \sqrt{\left[\left(\frac{T}{W}\right) - G\right]^2 - \frac{4C_{D0}}{\pi Ae}}\right)}{\frac{2}{q\pi Ae}} \tag{5.25}$$

$$\frac{W}{S} = 5765\ \mathrm{lb/ft^2}$$

Where the climbing gradient, represented by "G," is the ratio of the climb rate to the forward velocity. It's important to note that for the service ceiling, the climb rate is 1.667 ft/s and the forward velocity is 710.849 ft/s, resulting in a climbing gradient of approximately 0.0023. "A" represents the aspect ratio and has a value of 8.25. "e" represents the Oswald span efficiency and has a value of 0.8. The parasite drag and has a value of 0.015. "q" represents the dynamic pressure at an altitude of 36089 ft and has a value of 5765 $\mathrm{lb/ft^2}$.

5.6 T/W versus W/S plots

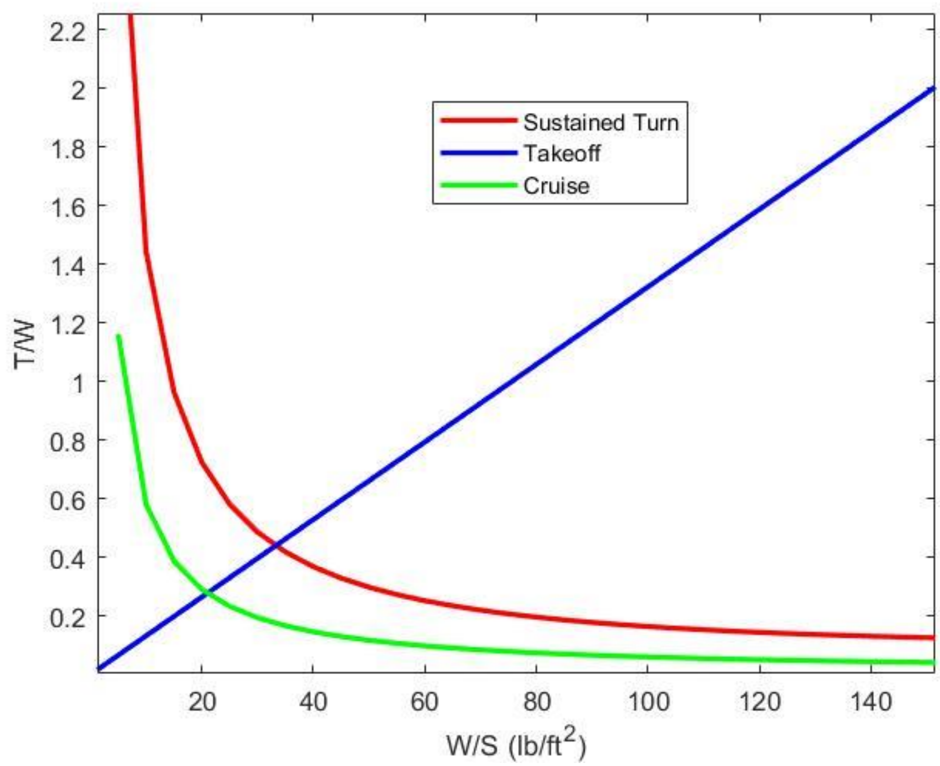


Figure 5.6.1 *Thrust to weight ratio versus Wing loading*

Chapter 6. Refined Weight Estimation

6.1 Mission Profile

A simple mission segment was used in this project. Segments are indicated as 0-1: Takeoff, 1-2: Climb, 2-3: Cruise, 3-4: Descend, 4-5: Loiter, and 5-6: Landing, respectively.

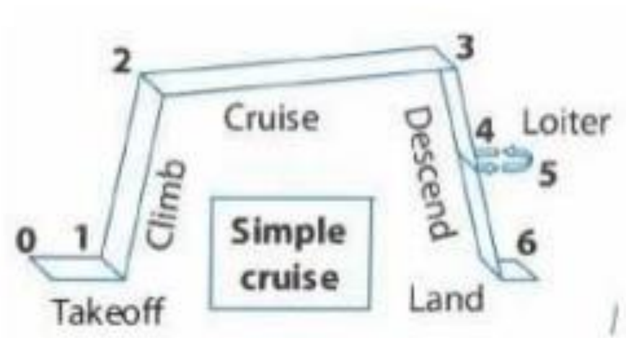


Figure 6.1 *Simple Mission Profile*

6.2 The Weight of the Crew and Weight of the Fixed Payload

Table 6.2.1 *Average Crew Member Weights [18]*

Crew Member	Average Weight [lb]	Average Weight with Bags [lb]
Flight Crew Member	190	240
Flight Attendant	170	220
Male Flight Attendant	180	220
Female Flight Attendant	160	200
Crew Member Roller Bag	30	NA
Pilot Flight Bag	20	NA
Flight Attendant Kid	10	NA

This plane had a crew of two pilots and two flight attendants. Using the crew member weight table, the W_c may be determined. The allocated weights for a pilot and a flight attendant, with their luggage, are 240 lb and 220 lb, respectively. As a result, the crew weight for this aircraft is determined to be:

$$Wc = 2(240) + 2(220) = 920\ lb$$

Table 6.2.1 *Standard Average Passenger Weights [18]*

Standard Average Passengers Weight	Weight per Passengers [lb]
Average Adult Passenger’s Weight	190
Average Adult Male	200
Average Adult Female	179
Average Child	82
Average Adult Passengers	195
Average Adult Male	205
Average Adult Female	184
Average Child	87

The average passenger weight table is used to calculate the payload's weight. The weight, luggage, and hand luggage allowances for each passenger are given as follows: 195 lb, 30 lb, and 16 lb.

$$Wpayload = 50(195 + 16 + 30) = 12050\ lb$$

6.3 Empty Weight Fraction

The improved statistical equation below is used to obtain the empty weight fraction of aircraft,

$$\frac{W_e}{W_0} = (a + bW_0^{C1} A^{C2} (\frac{T}{W_0})^{C3} (\frac{W_0}{S})^{C4} M_{Max}^{C5}) K_{VS} \quad 6.1$$

where a, b,C1,C2,C3,C4 and C5 are constant values, M_{Max} is the maximum Mach number , $\frac{T}{W_0}$ is the thrust the weight ratio, $\frac{W_0}{S}$ is the wing loading ,and $\frac{W_e}{W_0}$ is the empty weight fraction. The constants are taken from below table.

Table 6.3.1 *Improved statistical equations (jet aircraft) [18]*

$\frac{W_e}{W_0} = (a + bW_c^{C1} A^{C2} (T/W_0)^{C3} (W_0/S)^{C4} M_{max}^{C5}) K_{vs}$							
Fps units	a	b	C1	C2	C3	C4	C5
Jet trainer	0	4.28	-0.10	0.10	0.20	-0.24	0.11
Jet fighter	-0.02	2.16	-0.10	0.20	0.04	-0.10	0.08
Military Cargo/bomber	0.07	1.71	-0.10	0.10	0.06	-0.10	0.05
Jet transport	0.32	0.66	-0.13	0.30	0.06	-0.05	0.05

K_{vs}= variable sweep constant = 1.04 if variable sweep
= 1.00 if fixed sweep

The constants are taken with respect to the jet transport class to determine the empty weight fraction. According to the table, a=0.32, b=0.66, C1=-0.13, C2=0.3, Ce=0.06, C5=0.05 and Kvs=1 due to fixed sweep. Also, the maximum Mach number, thrust-to-weight ratio, and wing loading are found to respectively 0.77, 1.15, and 20.50.

6.4 Fuel Weight (W_f)

To find the fuel burning for each mission, below formula is used.

$$W_{fi} = (1 - \frac{W_i}{W_{i-1}}) W_{i-1} \quad (6.2)$$

The fuel weight is calculated for each mission segment. For this mission, N equals to 6, and trapped fuel is taken as % 6. Thanks to the below formula, fuel weight can be determined.

$$W_f = (1.06) \sum_{i=1}^{N} W_{fi} \quad (6.3)$$

6.4.1 Engine Start, Taxi and Take-off Segment

The weight fraction is between 0.97 and 0.98 with respect to historical trends. It is taken maximum value for calculation as 0.98. The burned fuel weight for this mission;

$$W_{f1} = (1 - \frac{W_1}{W_0})W_0 \tag{6.4}$$

The burned fuel determined for the take-off segment is 1003.9 lb. Also, the weight at the end of the take-off segment as it follows down.

$$W_1 = W_0 - W_{f\imath} \tag{6.5}$$

As a result, $W_1 = 97{,}557\ lb$.

6.4.2 Climb and Accelerate Segment

The flow is considered subsonic due to the 0.77 maximum Mach number. To determine the mission segment weight fraction below formula can be used.

$$\frac{W_i}{W_{\text{i}-1}} = 1.00065 - 0.0325M_{max} \tag{6.6}$$

As a result, $\frac{W_2}{W_1}$ = 0.9756

Since weight fraction for the climb and accelerate segment is found. Therefore, W_{f2} can be calculated as it follows below.

$$W_{f2} = (1 - \frac{W_2}{W_1})W_1 \tag{6.7}$$

As a result, W_{f2} =1828.2 lb.

$$W_2 = W_1 - W_{f2} \tag{6.8}$$

The weight at the end of the climb and accelerate segment is 97,557 lb .

6.4.3 Cruise Segment

The weight fraction for the cruise segment can be calculated by below equation

$$\frac{W_3}{W_2} = exp\frac{-RC_j}{V(\frac{L}{D})} \tag{6.9}$$

Where R represents range of aircraft, C_j is the thrust specific fuel consumption, and V is cruise speed. From above the weight fraction equation lift to drag ratio can be calculated thanks to the Breguet Range equation below.

$$\frac{L}{D} = \frac{1}{\frac{qC_{D0}}{(W/S)} + (W/S)\frac{1}{q\pi Ae}} \tag{6.10}$$

Where C_{D0} is the parasite drag, A is the aspect ratio, e is the Oswald span efficiency factor, (W/S) is wing loading at cruise, and q is the dynamic pressure. Dynamic pressure is calculated from the following equation.

$$q = \frac{1}{2}\, x\, \rho\, x\, V_{cruise}^2 \tag{6.11}$$

Since air density 1.225 kg/m^3 and cruise speed is 225.9 m/s, dynamic pressured obtained as 3186.2N. The lift to drag ratio is 15.7258 and obtained by substation q=3186.2 N, C_{D0} =0.015, (W/S) = 3220.6 N, A= 8.25, and e=0.8. Therefore,

$$\frac{W_3}{W_2} = 0.7202$$

After found weight fraction for this segment, the burned fuel is calculated as below.

$$W_{f3} = (1 - \frac{W_3}{W_2})W_2 \tag{6.12}$$

As a result of calculation above, $W_{f3} = 27296$ lb. Then, the weight at the end of the segment becomes;

$$W_3 = W_2 - W_{f3} \tag{6.13}$$

The weight at the end of cruise is 70,261 lb

6.4.5 Descent for Landing Segment

The descent for landing weight fraction segment value varies from 0.99 to 0.995. For this segment $\frac{W_4}{W_3} = 0.995$ is picked up. The burned fuel for descent segment.

$$W_{f4} = (1 - \frac{W_4}{W_3})W_3 \tag{6.14}$$

As a result, $W_{f4} = 351.3034\ lb$.

$$W_4 = W_3 - W_{f4} \tag{6.15}$$

Thus, the weight at the end of descending is 69,909 lb

6.4.6 Loiter mission segment fuel weight fraction

Loiter mission segment weight fraction can be determined by using endurance equation as it written below.

$$\frac{W_5}{W_4} = exp\frac{-EC_j}{(\frac{L}{D})} \tag{6.16}$$

Where E is the loiter time 0.333 h, C_j is specific fuel consumption obtained from competitor study as 0.667 1/h, and (L/D) for loiter mission can be found by below formula for jet transport.

$$\left(\frac{L}{D}\right)_{loiter} = \frac{\left(\frac{L}{D}\right)_{cruise}}{0.866} \tag{6.17}$$

Therefore, the lift and drag ratio is 15.72 and weight fraction for loiter segment becomes 0.9869. Then, the fuel burned can be obtained.

$$W_{f5} = (1 - \frac{W_5}{W_4})W_4 \tag{6.18}$$

Thanks to above formula, $W_{f5} = 917.8782$ lb

$$W_5 = W_4 - W_{f5} \tag{6.19}$$

As a result, the weight at the end of the loiter mission is 68,991 lb

6.4.7 Landing and Taxi Back Segment

The landing and taxi back segment weight fraction value varies from 0.992 to 0.997 and selected value is 0.997 for this mission. The burned fuel is calculated as it follows below.

$$W_{f6} = (1 - \frac{W_6}{W_5})W_5 \tag{6.20}$$

Thus, $W_{f6} = 206.9745$ lb

$$W_6 = W_5 - W_{f6} \tag{6.21}$$

Subsequently, the weight at the end of the landing is equals to 68,784 lb

$$W_f = (1.06)\sum_{i=1}^{N} W_{fi} \tag{6.22}$$

The total fuel weight is calculated by considering 6% reserve and trapped fuel

$$W_f = (1.06)\ (W_{f1} + W_{f2} + W_{f3} + W_{f4} + W_{f5} + W_{f6}) \tag{6.23}$$

As a result, the total aircraft fuel 31604 lb.

6.5 Results of Refined weight estimation

The takeoff gross weight of aircraft is found by MATLAB code with 1000 lb initial guess and %0.1 tolerance. MATLAB does iterations to get exact result takeoff gross weight is 100390 lb.

Table 6.5.1 *All Parameters and Results*

Parameters	Results
W_{crew}	720 lb
$W_{Fix\ payload}$	12050 lb
(W_e/W_0)	0.5580
Take-off Weight Fraction (W_1/W_0)	0.9900
Climb Weight Fraction (W_2/W_1)	0.9816
Cruise Weight Fraction (W_3/W_2)	0.7202
Descend Weight Fraction (W_4/W_3)	0.9950
Loiter Weight Fraction (W_5/W_4)	0.9869
Landing Weight Fraction (W_6/W_5)	0.9970
Total Fuel Weight (W_f)	31604 lb
Take-off Fuel Weight (W_{f1})	1003.9 lb
Climb Fuel Weight (W_{f2})	1828.2 lb
Cruise Fuel Weight (W_{f3})	27296 lb
Loiter Fuel Weight (W_{f4})	351.30 lb
Descent Fuel Weight (W_{f5})	917.88 lb
Landing Fuel Weight (W_{f6})	206.97 lb
Total Take-off Gross Weight (W_0)	88365 lb
W_1	85432 lb
W_2	85430 lb

W_3	60261 lb
W_4	59909 lb
W_5	58991 lb
W_6	58784 lb

All values shown in Table 6.5.1 were found with the help of equations, and MATLAB code and are shown here.

Table 6.5.2 *Initial and Refined Weight Estimation Comparison*

	Initial Estimation	**Better Estimation**
$(\mathrm{W}_e/W_\mathbf{0})$	0.5259	0.5340
$W_\mathbf{0}$	95869 lb	88365 lb

$$\left(\frac{\mathrm{W}_e}{W_\mathbf{0}}\right) \textbf{Absolute Error} \;=\; \left|\frac{0.5340-0.5259}{0.5340}\right| * 100 = 1.52\ \%$$

$$(W_\mathbf{0})\ \textbf{Absolute Error} \;=\; \left|\frac{95869-85365}{95869}\right| * 100 = 7.83\ \%$$

The value of empty weight fraction and take-off gross weight was calculated in previous assignment by using MATLAB. Now, better estimation of values are shown above table.

6.6 Discussing and comparing refined weight estimation

In Chapter 6, basically shows that to estimate the refined weight estimate method more accurately, several parameters must be taken into account. The take-off gross weight has been computed using MATLAB code with less than 0.01 absolute error using the improved statistical equation, and the total fuel weight has been determined by computing the weight of burned fuel for each segment. The improved weight estimation is predicted to result in a lower take-off gross weight than the initial weight estimation, which was determined to be and, respectively. Additionally, the lift-to-drag ratio, which is 15.73, one of the most important effective metrics, has a greater value despite having a higher empty weight percentage. According to previous estimations in Chapter 2, the value of take-off gross weight and empty weight fraction value found as $\mathrm{W}_\mathbf{0}$ = 95869 lb and $\mathrm{W}_\mathbf{e}/\mathrm{W}_\mathbf{0}$ = 0.5259 respectively. Thus, for the conceptual design process of the airplane, this estimation method produces better and more trustworthy results. After all the calculations, the code generates a precise estimate of our business jet's takeoff gross weight ($\mathrm{W}_\mathbf{0}$) of around 88365 lb. Considering all factors and estimates, this weight is more accurate for the business jet and is closer to the take-off gross weight of our aircraft rather than determined during the initial weight estimation process.

Chapter 7. Aircraft Sizing and Center of Gravity Location

7.1 Wing Sizing

One of the most important components of the aircraft is the wings. The wings provide lift and aerodynamic stability to the aircraft. In this project, the wing profile was chosen as trapezoidal. The reason for this is that when the trapezoidal wing is used, it is exposed to less coefficient of friction and provides better aerodynamic stability. When we look at the competitor studies in Assignment 1, it is seen that the wingspan of the 50-seat business plane is approximately 24 meters. In Assignment 4, our aspect ratio was 8.25 and our wingspan was 20.24 meters. In Assignment 6, the take-off gross weight was calculated as 88365 lb (45536.13 kg) and in Assignment 5, the wing loading at cruise speed was calculated as 670.056 $\mathrm{kg/m^2}$.The following equation gives wing area with take-off gross weight and wing loading,

$$S = W_0 * \left(\frac{W_0}{S}\right)^{-1} \tag{7.1}$$

$$S = \frac{45536.13}{670.056} = 67.959\ m^2 = 731.50\ ft^2$$

As a result of these calculations, the wing area "S" was calculated as $731.50\ ft^2$. After finding the wing area, the following equation is used to find the wingspan.

$$b = \sqrt{S * AR}, \tag{7.2}$$

$$b = \sqrt{67.959 * 8.25} = 23.171\ m = 76.02\ ft$$

where AR is the aspect ratio is equal to 8.25. From Equation 7.2, the wingspan was calculated to be 76.02 ft.

The root chord and tip chord can be calculated using by

$$S = b\left(\frac{C_{tip} + C_{root}}{2}\right), \tag{7.3}$$

where S is wing reference area, $Croot$ is root chord $Ctip$ is tip root and b is wing span.

$$\lambda = \frac{C_{tip}}{C_{root}}, \tag{7.4}$$

where λ is the taper ratio. Most swept wings have a taper ratio of about 0.2-0.3. [19] For this reason, the taper ratio was chosen as 0.25. When we combine equation 7.3 and 7.4, C_{tip} is calculated as 3.848 feet (1.173 m) and C_{root} as 15.394 feet (4.692 m).

The average aerodynamic chord, c^{-}, can be determined as

$$\bar{c} = \frac{\left(\frac{2}{3}\right) \mathrm{C_{root}}(1 + \lambda + \lambda^2)}{1 + \lambda}. \tag{7.5}$$

$$\bar{c} = \frac{\left(\frac{2}{3}\right) 4.692(1 + 0.25 + 0.25^2)}{1 + 0.25} = 3.284\ m = 10.774\ ft$$

Table 7.1.1 *Summary of the Wing Geometries*

	Span, b (ft)	Area, S (ft^2)	Root Chord C_{root} (ft)	Tip Chord C_{tip} (ft)	Mean Aerodynamic Chord, $\bar{c}$ (ft)
Wing	76.02	731.50	15.394	3.848	10.774

7.2 Fuel Tanks Dimensions and Locations

7.2.1 Fuel Volume Calculations

In Assignment 6, fuel weight is founded as 14315.83 kg (31604 lb) thanks to iteration with Matlab code. Knowing that most airplanes and jets use kerosene as fuel, its fuel density is 0.813 kg/lt and 813 kg/m^3 [19].

$$Weight = Volume * (Density). \tag{7.6}$$

The fuel tank volume is calculated as 17.609 m^3 with equation 1.2.1. In imperial system, it's equal to 621.85ft^3.

7.2.2 Size of the Fuel Tanks

According to the calculations in the fuel volume, the volume of our fuel is 17.609m^3. Fuel must be evenly distributed on both wings to ensure aerodynamic stability. Therefore, each wing has to carry 8.804m^3.

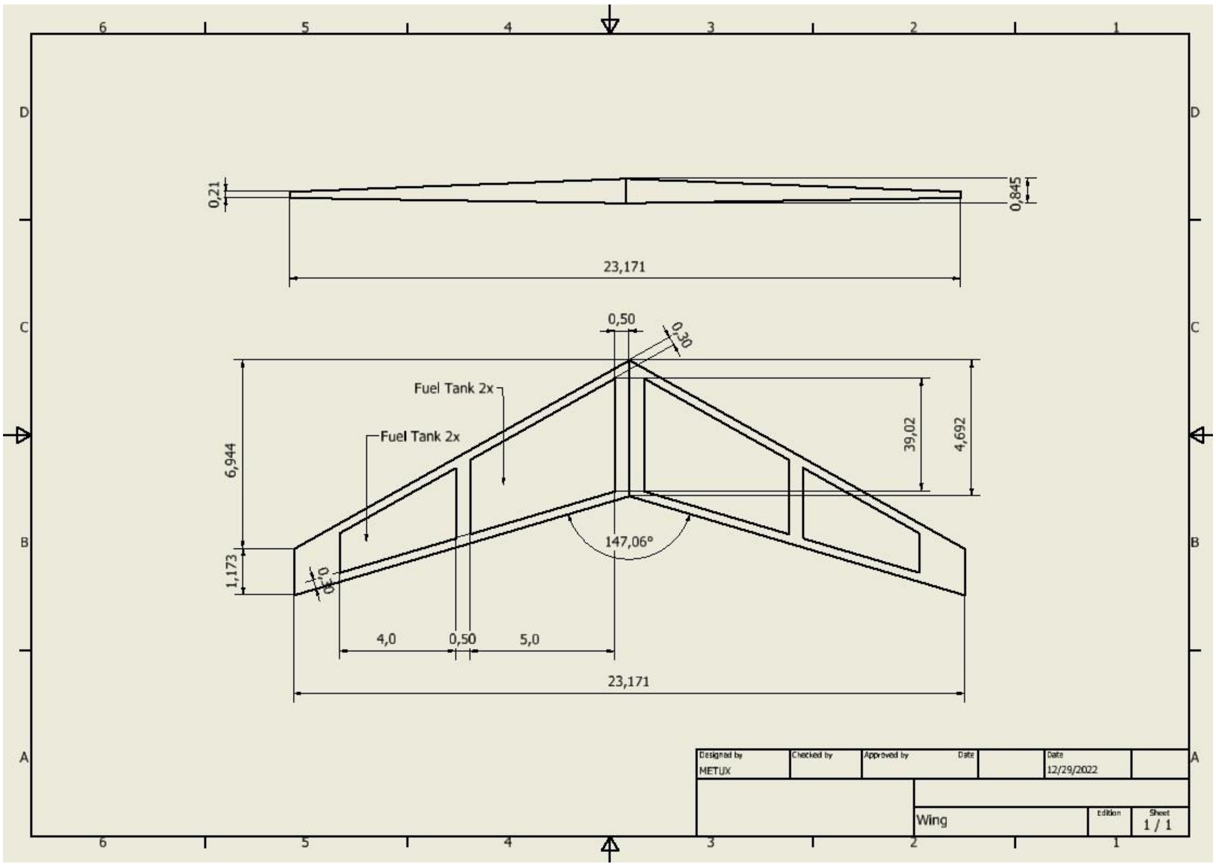


Figure 7.2.1 *Fuel Tanks Layout, all dimension in meter*

Figure 7.2.1 shows how the fuel tanks look and the dimensions inside the wing. Autodesk Inventor Professional program is used for the drawings of the fuel tank. All dimensions are given in meters. There are a total of 4 fuel tanks, 2 small and 2 large, on the entire wing. The area of the large fuel tank is 16.1725 m^2 and its height is 0.4 m on average. According to this calculation, the volume of the large single fuel tank is 6.469m^3. Since there are 2 large fuel tanks, the total volume is 12.938m^3. When we examine the small fuel tank, its area is 8.982 m^2 and its height is 0.3 m. The volume of this tank is 2.6946m^3. Since there are 2 small fuel tanks, the volume becomes 5.3892m^2. The total volume accordingly makes 18.3272 m^3(647.22ft^3). It adequately meets the amount of fuel we need.

7.3. Horizontal and Vertical Tail Configuration

Using historical trend tables and our gross takeoff weight 100,390 lbs., we can calculate vertical and horizontal tail configurations such as aspect ratio, taper ratio, and sweep angle.

Table 7.3.1 *Horizontal tail properties of different airplanes with historical trends [20]*

Aircraft	Gross Weight W_g (lb)	Wing Area S (sq.ft.)	S_{HT}/S	H.T. Span b_{HT}	Aspect Ratio A_{HT}	Taper λ_{HT}	Sweep Λ_{HT} (deg)
Dash 8 Q100	36,300	585	0.28	26.8	4.43	0.80	6.5
XAC MA60	48,050	810	0.22	31.0	5.49	0.44	20.0
ATR72-500	48,500	657	0.20	24.2	4.37	0.56	8.0
ERJ145LR	46,275	551	0.24	24.8	4.68	0.59	20.0
CRJ200LR	53,000	587	0.19	20.5	3.72	0.50	34.5
CRJ700ER	75,250	739	0.24	28.0	4.36	0.40	33.0
E175	85,517	783	0.32	32.8	4.29	0.50	35.0
B737-700ER	154,500	1341	0.33	47.1	5.04	0.33	36.0
A320-200	169,800	1320	0.26	40.9	4.84	0.31	32.5

The table 7.3.1 and 7.3.2 display several aircraft's horizontal and vertical tail characteristics. The optimum values for the aspect ratio, taper ratio, and sweep angle of the horizontal and vertical tail of a design aircraft are chosen using this table, which is based on historical trends. When we look at Tables 7.3.1 and 7.3.2, we see that the weight of our aircraft is between E175 and B737-700ER. Therefore, using these tables, the aspect ratio 4.67, taper ratio 0.415, and sweep angle 34-degree are assumed for the horizontal tail from historical trends.

Table 7.3.2 *Vertical tail properties of different airplanes with historical trends [20]*

Aircraft	Gross Weight W_g (lb)	Wing Area S (sq.ft.)	SV_T/S	V.T. Span b_{HT}	Aspect Ratio A_{VT}	Taper λ_{VT}	Sweep Λ_{VT} (deg)
Dash 8 Q100	36,300	585	0.28	14.0	1.20	0.67	32.0
XAC MA60	48,050	810	0.18	15.2	1.57	0.40	27.0
ATR72-500	48,500	657	0.26	15.8	1.48	0.50	38.5
ERJ145LR	46,275	551	0.18	11.4	1.29	0.64	36.0
CRJ200LR	53,000	587	0.19	11.4	1.16	0.69	44.5
CRJ700ER	75,250	739	0.17	12.0	1.16	0.71	40.0
E175	85,517	783	0.29	20.5	1.84	0.30	40.0
B737-700ER	154,500	1341	0.22	26.8	2.45	0.20	37.0
A320-200	169,800	1320	0.24	23.8	1.77	0.30	39.0

Table 7.3.2 also shows parameters like table 7.3.2 for vertical tail. At the same time, when we looked at the vertical tail using these tables 7.3.1 and 7.3.2, the taper ratio, aspect ratio, and sweep angle were found to be 0.25, 2.145 and 43.5 °respectively.

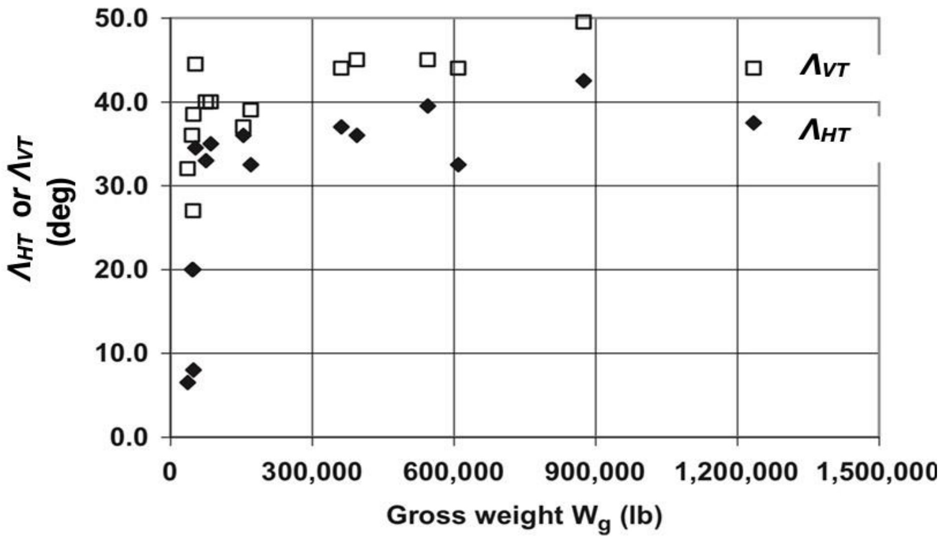


Figure 7.3.1 *Aspect ratios of horizontal tail (solid symbols) and of the vertical tail (open symbols) with gross weight (for airliners) [20]*

The aspect ratio of the aircraft designed according to Figure 7.3.1 was chosen as 4.5 for the horizontal tail and 1.6 for the vertical tail.

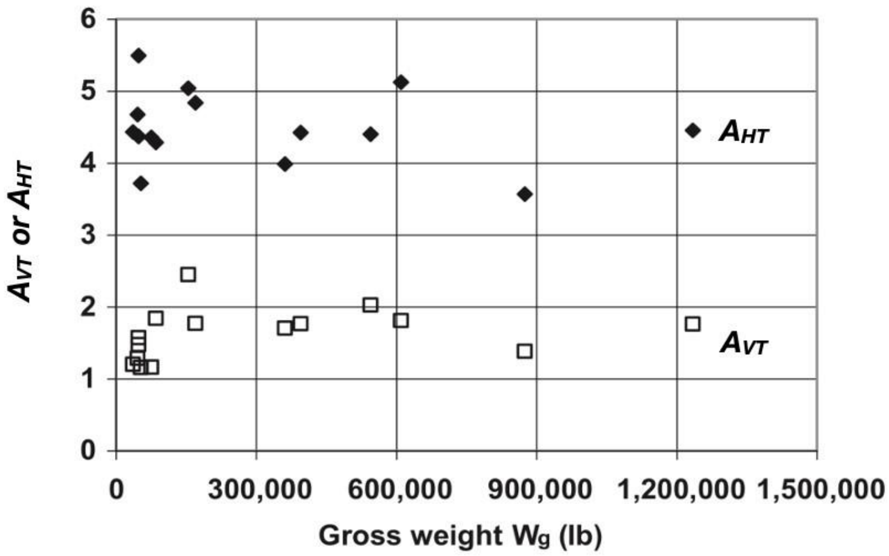


Figure 7.3.2 *Leading edge sweepback of the horizontal tail (solid symbols) and of the vertical tail (open symbols) with gross weight (for airliners) [20]*

Vertical tails for the turbofan aircraft have sweepback angles about 5-10 degree larger than the horizontal tails (both larger than those of their respective wings). Vertical tails for the turboprop aircraft have large sweepback and little or none sweepback on horizontal tail. [2] Looking at Figure 1.3.2, the sweep angle was chosen as 30° for the horizontal tail and 36° for the vertical tail.

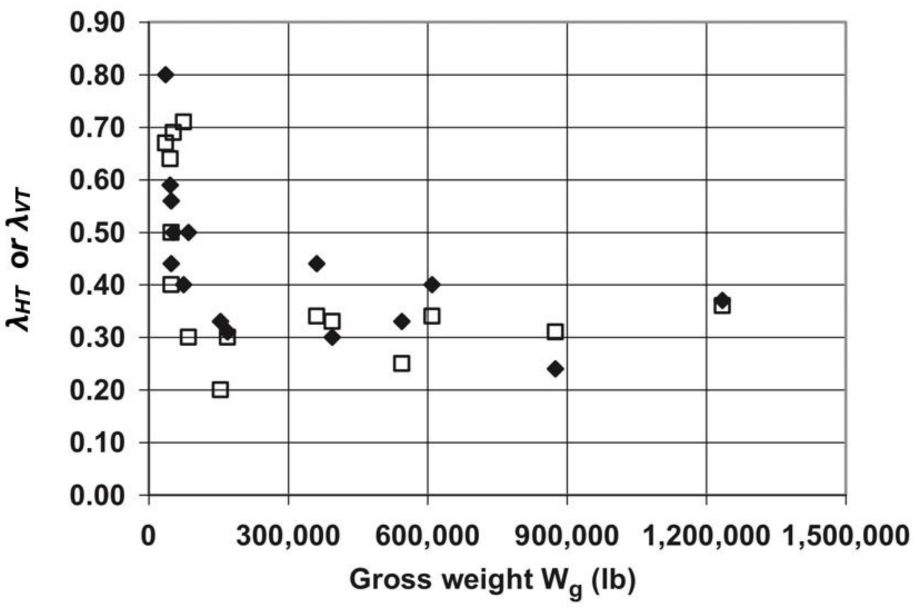


Figure 7.3.3 *Taper of the horizontal tail (solid symbols) and of the vertical tail (open symbols) with gross weight (for airliners) [20]*

Taper ratios for the horizontal and vertical tail surfaces are about the same. Taper ratios are smaller for the heavier jetliners and have greater variability for the lighter aircraft. [2] According to Figure 7.3.3, the optimal taper ratio is 0.6 for the horizontal tail and 0.6 for the vertical tail.

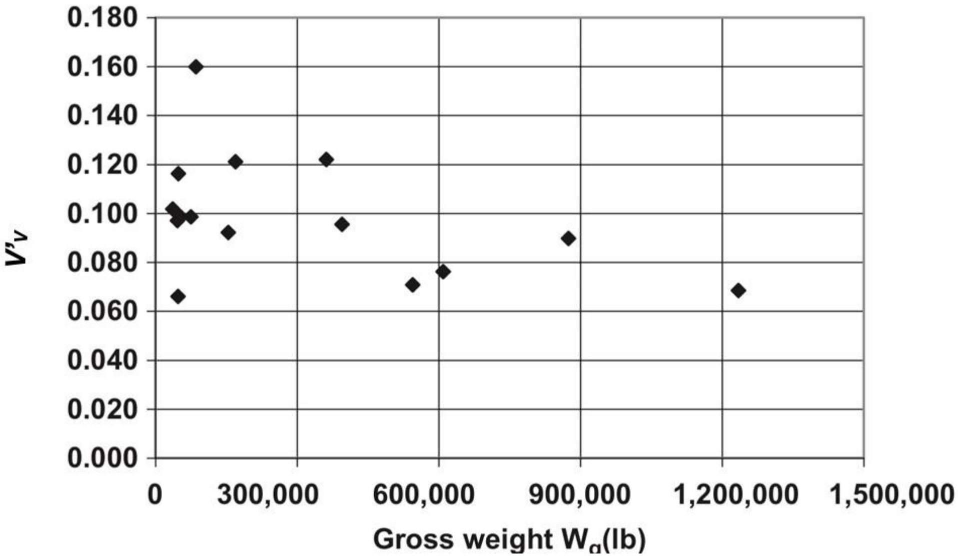


Figure 7.3.4 *Approximate vertical tail volume coefficient V'V as a function of gross weight (for airliners) [20]*

Figure 7.3.4 shows that the vertical tail volume coefficient $'V_V'$ is seen as approximately 0.11. Using the equations below, determine the tail and tail lengths while keeping in mind that $L_{fuselage}$ is 29.73 m from Assignment 3.

A common approximation for commercial airliners,

$$L_H = 0.49 * Lf\ , \qquad L_V = 0.45 * Lf \tag{7.7}$$

where Lf is the length of the fuselage. Thanks to Equation 7.3.1, the length of horizontal tail L_H was calculated as 47.794 ft (14.5677 m) and the length of vertical L_V tail as 43.898 ft (13.3785 m).

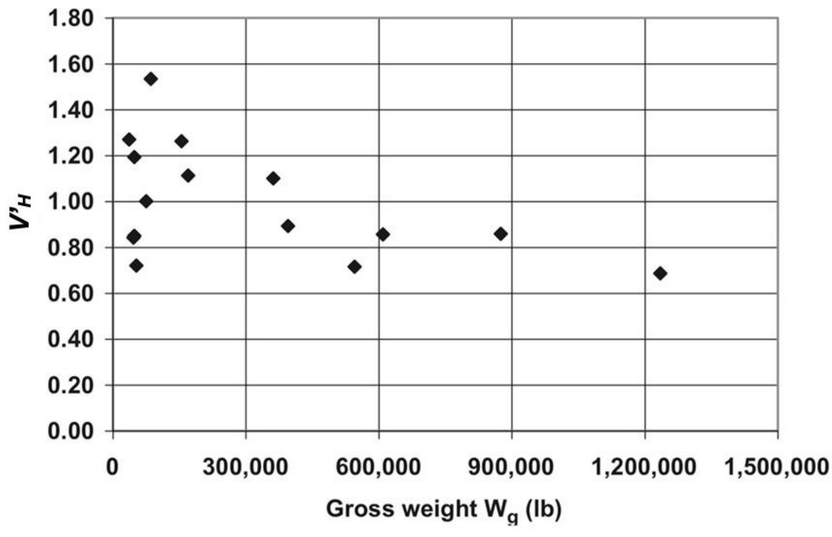


Figure 7.3.5 *Approximate horizontal tail volume coefficient V'H as a function of gross weight (for airliners) [20]*

When we look at Figure 1.3.5, the horizontal tail volume coefficient $'V_H'$ is seen as approximately 1.2.

The vertical tail area can be determined as follows,

$$S_V = \frac{V_V bS}{L_V}, \tag{7.8}$$

where S_V is the vertical wing area, V_V is the vertical tail volume coefficient, L_V is the distance between the CG and the average quarter chord positions of the vertical tail surface. Thanks to Equation 1.3.2, the vertical wing area of the tail S_V is 12.847m^2.

$$S_V = \frac{(0.11)(23.171)(67.595)}{(13.3785)} = 12.847\, m^2 = 138.28\, ft^2$$

The vertical tail area can be determined as follows,

$$\mathrm{S_H} = \frac{\mathrm{V_H}\, \bar{\mathrm{c}}\, \mathrm{S}}{L_{\mathrm{H}}} \tag{7.9}$$

where $\mathrm{S_H}$ is the wing area of tail, $\mathrm{V_H}$ is the tail volume coefficient, L_{H} is the distance between the CG and the average quarter chord positions of the tail. Thanks to Equation 1.3.3, the horizontal wing area of tail S_H is 18.384m^2.

$$\mathrm{S_H} = \frac{(1.2)(3.284\,)67.595)}{14.5677} = 18.384\, m^2 = 197.88\, ft^2$$

After finding the horizontal and vertical wing areas of tail, the following equation is used to find the wingspan of the horizontal and vertical wing tail.

$$b_H = \sqrt{S_H * AR_H}, \quad b_V = \sqrt{S_V * AR_V} \tag{7.10}$$

$$b_H = \sqrt{18.384 * 4.670} = 9.2657\, m = 30.40\, ft$$

$$b_V = \sqrt{12.847 \ * 2.145} = 5.2495\, m = 17.22\, ft$$

where AR_H and AR_V are the aspect ratio of the horizontal and vertical tail, and S_V and S_H are the wing areas of vertical and horizontal tail .

The root chord and tip chord for vertical and horizontal can be calculated using by

$$S_V = b_V\left(\frac{C_{tipV} + C_{rootV}}{2}\right), \tag{7.11}$$

$$S_H = b_H\left(\frac{C_{tipH} + C_{rootH}}{2}\right), \tag{7.12}$$

where $Croot$ is root chord, and $Ctip$ is tip root.

$$\lambda_V = \frac{C_{tipV}}{C_{rootH}}, \tag{7.13}$$

$$\lambda_H = \frac{C_{tipH}}{C_{rootH}}, \tag{7.14}$$

where λ_v and λ_H are the taper ratios of vertical and horizontal wing tail. λ_v is equal to 0.25 and, λ_H is equal to 0.415. Combining equations from1.3.5 to1.3.8 we get C_{tipV}, C_{tipH}, C_{rootV} and C_{rootH} as 0.979 m, 2.804 m, 3.916 m, 1.1638, respectively.

The average aerodynamic chord, c^{-}, can be determined as

$$\overline{c_V} = \frac{\left(\frac{2}{3}\right)\mathrm{C_{rootV}}(1 + \lambda_V + \lambda_V^2)}{1 + \lambda_V}. \tag{7.15}$$

$$\overline{c_H} = \frac{\left(\frac{2}{3}\right) \mathrm{C_{rootH}}(1+\lambda_H+\lambda_H^2)}{1+\lambda_H}. \tag{7.16}$$

As a result of that calculations, the average aerodynamic chords of tail,

$$\overline{c_H} = \frac{\left(\frac{2}{3}\right) 2.804(1+0.415+0.415^2)}{1+0.415} = 2.0969\ m = 6.88\ ft$$

$$\overline{c_V} = \frac{\left(\frac{2}{3}\right) 3.916(1+0.25+0.25^2)}{1+0.25} = 2.7412\ m = 8.99\ ft$$

Table 7.3.3 *Summary of the Tail Geometries*

	Span, b (ft)	**Area, S (ft^2)**	**Root Chord C_{root} (f)**	**Tip Chord C_{tip} (f)**	**Mean Aerodynamic Chord, $\bar{c}$ (f)**	**Sweep angle (deg)**
Horizontal Tail	30.40	197.88	9.20	3.82	6.88	34
Vertical Tail	17.22	138.28	12.85	3.21	8.99	43.5

The tail design used in this project is like the traditional tail designs because the calculations were made thanks to the data from the historical trends.

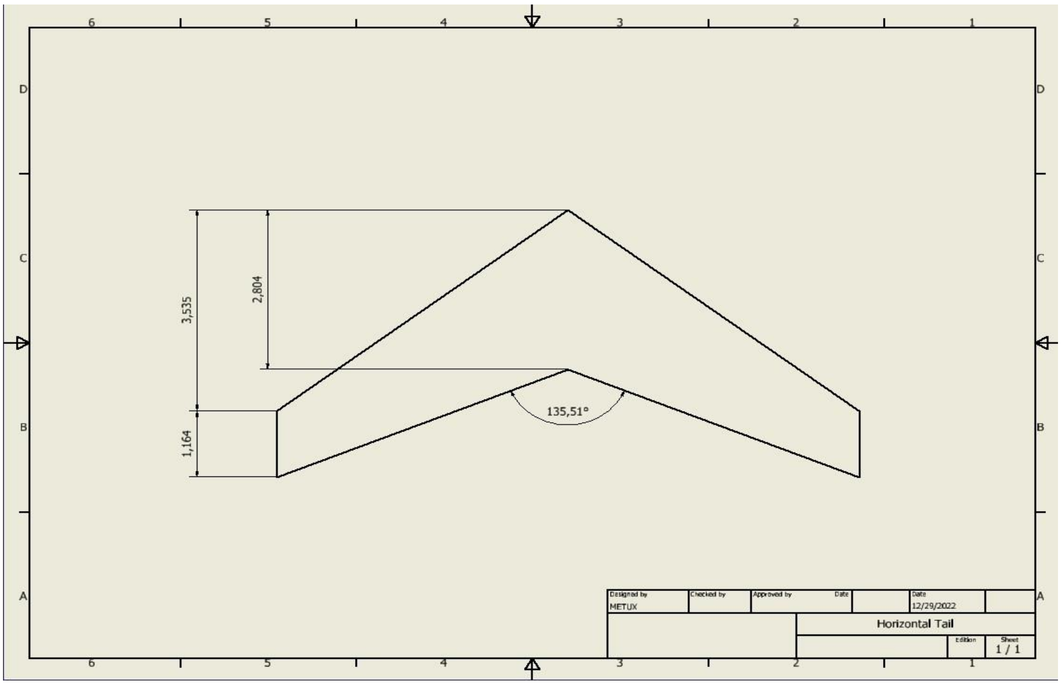


Figure 7.3.6 *Horizontal Tail, all dimension in meter*

The horizontal tail was drawn with the help of Autodesk Inventor and all dimensions are given in meters

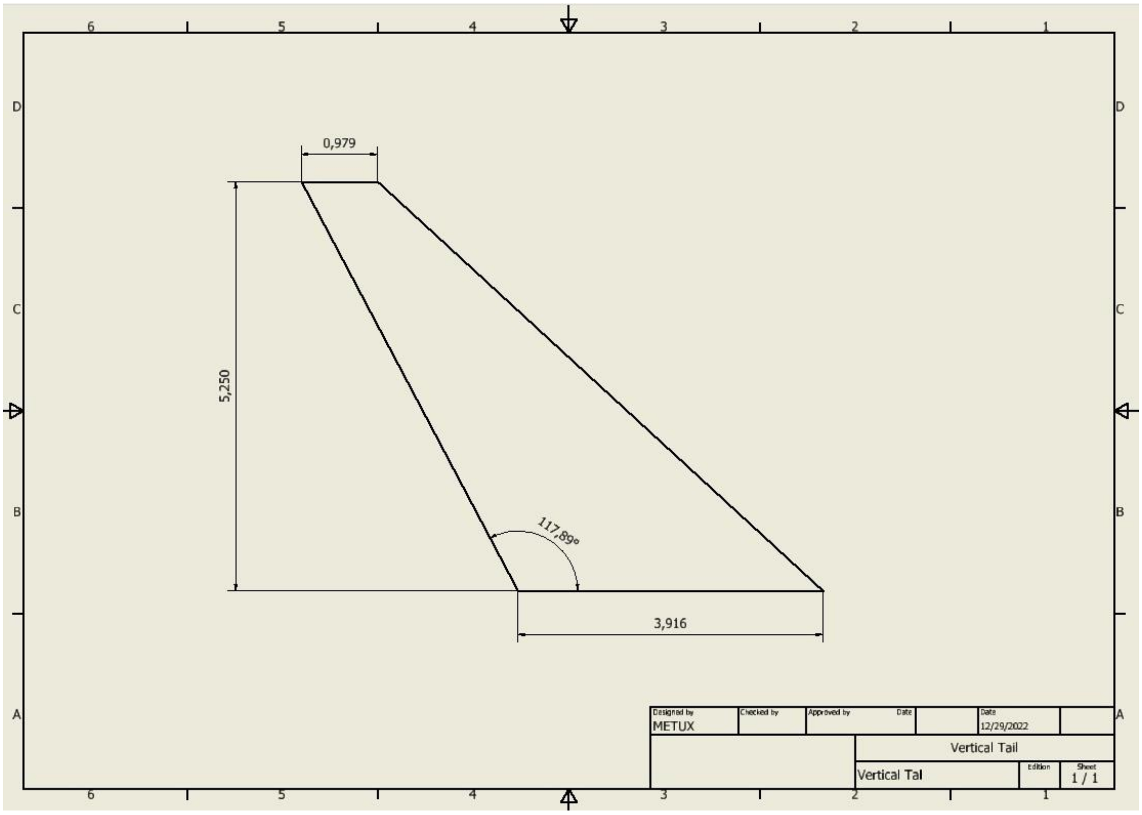


Figure 7.3.7 *Vertical Tail, all dimension in meter*

The vertical tail was drawn in Figure 7.3.7 with the help of Autodesk Inventor and all dimensions are given in meters.

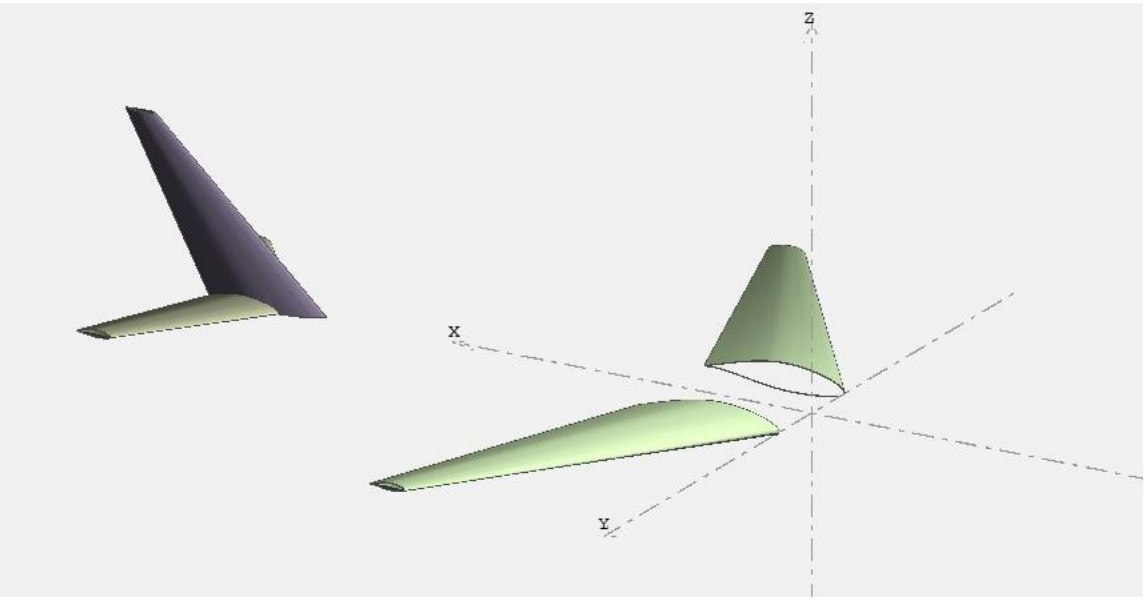


Figure 7.3.8 *Wing and tail configuration drawn with the help of XFLR5*

In Figure 7.3.8, the wing and tail configurations are drawn to give information about how our aircraft looks without a fuselage.

7.4 Weight and Location Components

7.4.1 Weights and Corresponding Locations of the Major Components

In this section, it is aimed to determine the designed aircraft's major components that are mainly wing, tail, engine, fuselage, passengers, payload, crew and fuel tanks. There are four ways to estimate weight components. Historical analogy, statistics, component selection, and structural analysis are the four of them. In our calculation, it is assumed fuel tanks to be empty, landing gear is retracted or extended.

Table 7.4.1 *Approximate Empty Weight Buildup [21]*

	Fighters		**Transport and bomber**		**General Aviation**			
	lb/ft^2	kg/m^2	lb/ft^2	kg/m^2	lb/ft^2	kg/m^2	**Multiplier**	**Approximate Low**
Wing	9	44	10	49	2.5	12	$S_{exposed\ planform}$	40% MAC
Horizontal Tail	4	20	5.5	27	2	10	$S_{exposed\ planform}$	40% MAC
Vertical Tail	5.3	26	5.5	27	2	10	$S_{exposed\ planform}$	40% MAC
Fuselage	4.8	23	5	24	1.4	7	$S_{wetted\ area}$	40-50% L

Table above shows data of approximate empty weight and calculations are done with respect to the table.

Wing weight:

$$W_{wing} = (10\ lb/ft^2)(S_{wing\ exposed\ planform}) \tag{7.17}$$

Data of S= 731.50 ft^2 is from Table 7.1.1

So,

$$W_{wing} = 7315 \text{ lb}$$

Horizontal tail weight:

$$W_{HT} = (5.5\ lb/ft^2)(S_{HT\ exposed\ planform}) \tag{7.18}$$

And $S_{HT\ exposed\ planform} = 197.88\ ft^2$ is from Table 7.3.3.

So,

$$W_{HT} = 1088.34\ lb$$

Vertical tail weight:

$$W_{VT} = (5.5\ lb/ft^2)(S_{VT\ exposed\ planform}) \tag{7.19}$$

And $S_{VT\ exposed\ planform} = 138.28\ ft^2$ is from Table 7.3.3.

So,

$$W_{VT} = 760.54\ lb$$

Fuselage weight:

$$S_{wetted\ area} = K\left(\frac{A_{top} + A_{side}}{2}\right) \tag{7.20}$$

(K=3.14 is a typical value)

$$A_{\ top} = A_{fuselage\ top} + A_{nose\ ellipse} + A_{tail\ ellipse} = A_{side} = \ 916.88\ ft^2.$$

As a result,

$$S_{wetted\ area} = 2879\ ft^2.$$

And,

$$W_{fuselage} = (5\ lb/ft^2)(S_{wetted\ area}) \tag{7.21}$$

Therefore $W_{fuselage} = 14395.02\ lb$

Engine weight:

CFM56-3B-1 has 4276 lbs, and it will be use in this project.

$$W_{engine} = 1.3(W_{dry\ engine}) \tag{7.22}$$

$$W_{dry\ engine} = 4276\ \text{lb} * 1.3 = 5558.8\ \text{lb}$$

The aircraft has two engine so the value is multiplied by two and is founded as

$$2 \times W_{dry\ engine} = 11117.6\ lb$$

So dry engine weight is found as 11117.6 lb.

Table 7.4.2 *Weight of some parameters from Chapter 6*

Weight [lb]	
Fuel tanks	$W_{fuel\ tanks} = 31604$ lb
Crew	$W_{crew} = 720$ lb
Passengers	$W_{passengers} = 9500$ lb
Payload	$W_{payload} = 12050$ lb

Fuel tanks weight:

To sum up, all weight and locations are written in Table 7.4.2 below.

Table 7.4.2 *Parameters and Locations of the aircraft components*

Aircraft component	Weight (lb)	Location(ft)
Wing	7315	53.261
Horizontal tail	1088.34	104.903
Vertical tail	760.54	104.450
Fuselage	14395.02	44.242
2x Engines	11117.6	14.764
Fuel tanks	31604	13.123
Crew	720	3.281
Navigation satellites and instruments [x]	3670	0.000
Passenger	9500	56.355
Payload	12050	65.322

The reference point for the locations of all components of the aircraft is taken relative to the nose of the airplane. It is shown in detail in Figure 7.5.1.

7.5 Center of Gravity Location

7.5.1 Estimation of Center Of Gravity

For this design, wings, fuel tanks and engines are not included in center of gravity estimation because they are located on wings.

$$\bar{x}_{cg1} = \frac{W_{crew}x_{crew} + W_F x_F + W_{payload}x_{payload} + W_P x_P + W_{VT}x_{VT} + W_{HT}x_{HT}}{W_{crew} + W_F + W_{payload} + W_P + W_{VT} + W_{HT}} \tag{7.23}$$

Putting all constants in the table 7.4.2 to the equation above gives the answer of $\bar{x}_{cg1} = 49.914$ ft.

7.5.2 Updated Center of Gravity with located wings

In this part, wings and fuel tanks are included to center of gravity (cg) estimation. Addition constants to the equation above and recalculating $\bar{x}_{cg2}$

$$\bar{x}_{cg2} = \frac{W_{crew}x_{crew} + W_F x_F + W_{payload}x_{payload} + W_P x_P + W_W x_W + W_{fuel}x_{fuel} + W_{VT}x_{VT} + W_{HT}x_{HT} + W_E x_E}{W_{crew} + W_F + W_{payload} + W_P + W_W + W_{fuel} + W_{VT} + W_{HT} + W_E} \tag{7.24}$$

Putting all constants in the table 7.24 to the equation above gives the answer of 52.849 ft for $\bar{x}_{cg2}$. To see the view clearly, location of the estimated center of gravities are shown in figure below.

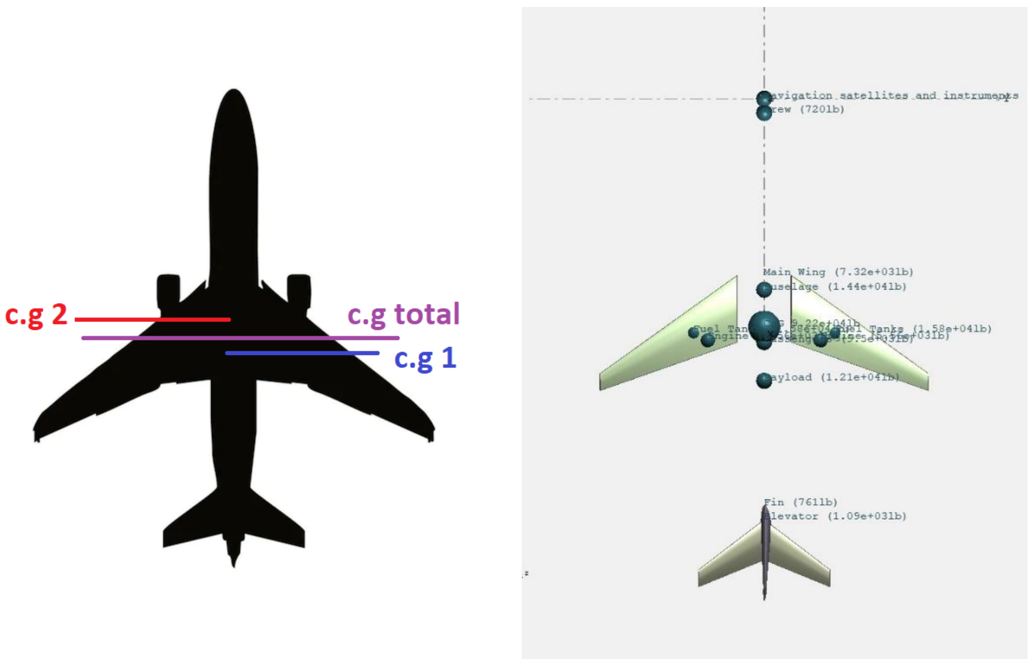


Figure 7.5.1 *Locations of the estimated center of gravities from xflr5*

The Center of gravity in the case of wings and fuel tank, engines are not included in consideration is

49.914 ft whereas the center of gravity in the case of the two is included is 52.849 ft. As a result, the overall center of gravity is located in the coordinate of 51.3815 ft away from the nose, which is chosen as the coordinate of zero.

$$\bar{x}_{cg} = 51.3815 \text{ ft}$$

7.5.3 Determination of moment arms for horizontal and vertical tails

The location of the center of gravity and the location where the weight force is acting is defined as the tail moment arm. Horizontal and vertical tail moment arms are computed as

$$l_{HT} = x_{HT} - \bar{x}_{cg} \tag{7.25}$$

$$l_{HT} = 104.903 - 51.3815 = 53.522\, ft$$

$$l_{VT} = x_{VT} - \bar{x}_{cg} \tag{7.26}$$

$$l_{VT} = 104.450 - 51.3815 = 53.069\, ft$$

In order to provide stability, tail size should be increased a little more with respect to historical trends.

7.5 Discussing and comparing refined weight estimation

Chapter 7 focuses on determining the appropriate size for the airplane's wing, drawing the fuel tank, and determining the location of the center of gravity. The first part includes the calculation of the required wing size and fuel volume based on data from previous assignments using geometric and aerodynamic parameters and selected wing load and aspect ratio. Thanks to the computations, the wing length is 76.02 ft , the wing area is 731.50 ft^2, the root chord is 15.394 ft, the tip chord is 3.848 ft and the Mean Aerodynamic Chord is 10.774 ft. These results are consistent with the values of similar aircraft found in the competitor study. As a result of the calculations made later, the fuel volume of the aircraft was calculated as $621.85 ft^3$.In order to provide this volume, the fuel volume drawn with the help of the Autodesk Inventor program is designed as $647.22 ft^3$. Also, wing and tail configurations including fuel tanks were drawn. The second part involves determining the weights of the main components, including the wing, tail, engine, fuselage, passengers, payload, crew, and fuel tanks. All calculations for this part are described in Table 7.4.2 and the positions of the main components and their corresponding moment arm positions are also calculated. In order to make these calculations, the wing and tail configurations were drawn from the xflr5 and then the locations of the weights were determined. In the determination phase, the center of gravity was determined by considering the weight of all components except the wing with fuel tanks and engines. In the calculation made without adding the fuel tanks and engines, $\bar{x}_{cg}$ was calculated as49.914 ft. When the fuel tank and engines is added, this value is seen as 52.849 ft. As a result, the overall center of gravity is located in the coordinate of 51.3815 ft away from the nose, which is chosen as the coordinate of zero at the x-axis.

Chapter 8 Cost Analysis

8.1 RDT&E and Production Cost

Firstly, to calculate RDT&E and production costs of designed aircraft, the designer calculates how many hours are required for RDT&E. The following equations are used to calculate the hours necessary for engineering, tooling, manufacturing, and quality control.

$$H_e = 4.86 W_e^{0.777} V^{0.894} Q^{0.163} = 18{,}461{,}028.43 \text{ hours} \tag{8.1}$$

$$H_t = 5.99 W_e^{0.777} V^{0.696} Q^{0.263} = 11{,}092{,}592.64 \text{ hours} \tag{8.2}$$

$$H_m = 7.37 W_e^{0.82} V^{0.484} Q^{0.641} = 40{,}060{,}892.32 \text{ hours} \tag{8.3}$$

$$H_q = 0.133 H_m = 5{,}328{,}098.81 \text{ hours} \tag{8.4}$$

Where We represents the aircraft's empty weight, V is the maximum velocity the aircraft can achieve, and Q value is the production quantity selected 200 according to the graph below.

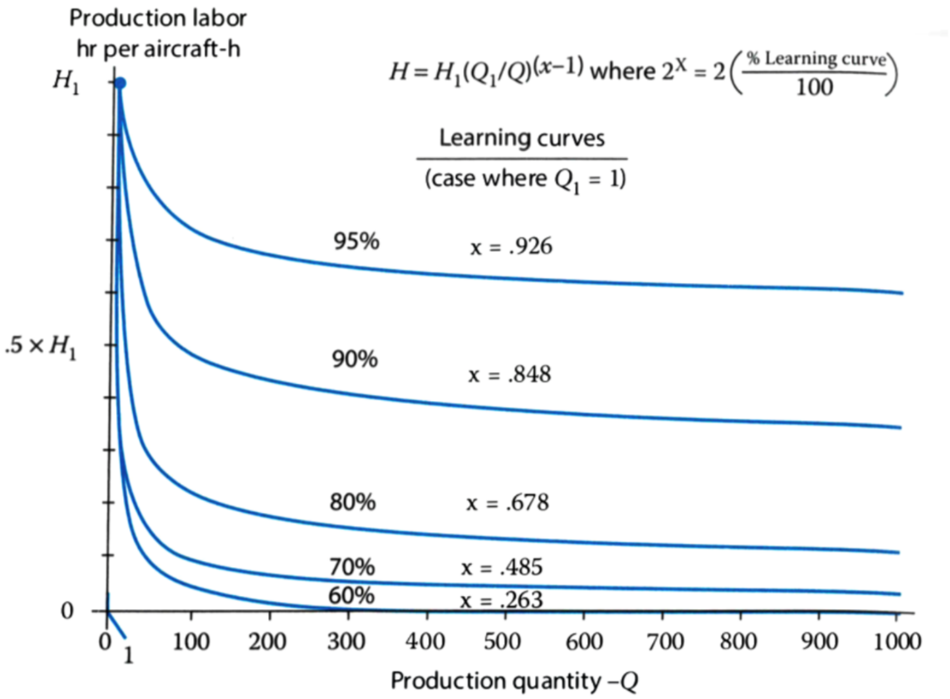


Figure 8.1.1 *Production quantity vs production labor hr per aircraft-h [22]*

Production quantity decreases as the number of airplanes produced increases in five years. To have lower RDT&E and production costs, Q is picked up as 200 in five years.

Table 8.1.1 *Fudge factors*

Alumium	1.0
Graphite-epoxy	1.1-1.8
Fiberglass	1.1-1.2
Steel	1.5-2.0
Titanium	1.1-1.8

According to the table above, the fudge factor of tooling is approximately 1.1 because the designed aircraft consists not only of one element, and the average of it is almost 1.1. Also, RDT&E and production costs include development support, flight test, engine production, and manufacturing material costs. They are calculated using the equations below, and they directly give the labor costs, so there is no need to multiply them by the appropriate hourly rates.

$$C_D = 91.3W_e^{0.630}V^{1.3} = 351{,}647{,}226\ \$ \quad (8.5)$$

$$C_F = 2498W_e^{0.325}V^{0.822}FTA^{1.21} = 111{,}065{,}411\ \$ \quad (8.6)$$

$$C_M = 22.1W_e^{0.921}V^{0.621}Q^{0.799} = 1{,}962{,}783{,}775\ \$ \quad (8.7)$$

$$C_{ENG} = 3112[0.043T_{max} + 243.25M_{max} + 0.969T_{Turbine\ inlet} - 2228] = 4{,}750{,}483.56\ \$ \quad (8.8)$$

$$C_{avionics} = 4000\$ * 85869\ \text{lb} = 343{,}476{,}000\ \$ \quad (8.9)$$

where we represent the aircraft's empty weight, V is the maximum velocity the aircraft can achieve, and the Q value is the production quantity. FTA shows how many flight tests are needed to complete the design, which is in the range of 2 two to six. For this design, FTA is taken 4 to have a safe and cheaper cost because as flight test numbers increases, price increases too. Moreover, the engine production cost is calculated by the maximum Mach number that the engine can achieve, turbine inlet temperature, and maximum thrust. As a result, the maximum Mach number is 0.82, the turbine inlet temperature is 1200 K, and the maximum thrust equals 34,000. Lb for the selected engine. Finally, there are two ways of calculating avionics cost, which are 5-25 % of the flyaway price and $4000-$8000 per pound. The second approximation is selected, and a minimum value of $4000 is taken for a lower cost.

Table 8.1.2 *Average Wrap Rate*

Engineering	R_E=$115
Tooling	R_T=$118
Quality Control	R_Q=$108
Manufacturing	R_M=$98

Finally, the cost of engineering, tooling, quality control, and manufacturing can be calculated by multiplying the average wrap rate of each.Thanks to below formula RDT&E + Production cost can be determined.

$$\text{RDT\&E + Poduction Cost} = H_eR_e + H_tR_t + H_mR_m + H_qR_q + C_D + C_F + C_M + C_{ENG}N_{ENG} + C_{avionics} \quad (8.10)$$

Where N_{ENG} represents number of engine and fort his desing it is two. The above equation is calculated by Matlab as $12,283,000,000. This result gives the total cost of 200 airplanes produced in five years. Therefore, one of the cost of airplane can be determined by dividing 200 to the total cost. As a result, the price of one airplane is calculated as $62,826,000 and with %20 profits 75,391,200 $ is the selling price.

8.2 Operational and Maintanence Cost (O&M)

Operational and Maintanence Cost for civil aircrafts it basically consist fuel cost , crew salaries, maintanance, depreciation and insurance.

8.2.1 Fuel and Oil Costs

The simple mission segment is selected to calculate fuel cost. For this aircraft design, total fuel consumption is determined as 31,604 lb, and the flying hour is 7.5 hours. In order to calculate the average fuel burned per hour, full fuel burned has to be divided by the duration of the flight.As a result, fuel burned per hour is calculated as 4,213.86 lb/h (628 gal). According to the table of LCC Parameters table below, the flight hours per year can be selected.

Table 8.2.1 *LCC Parameter Approximations*

Aircraft Class	FH/YR/AC
Light Aircraft	500-1000
Business Jet	500-2000
Jet Trainer	300-500
Fighter Modern	300-500
Bomber	300-500
Military Transport	700-1400
Civil Transport	2500-4500

Since the designed aircraft is in the business jet class, flight hours per year are selected at 700 hours, and the average jet fuel per gallon is 4.77 $. Therefore, the fuel cost per year can be estimated as follows below.

Fuel cost per year = 700 h x 628 gal/h x 4.77 $ = 2,096,892 $ (8.11)

Since the oil cost is almost half the fuel price, it can be ignored.

8.2.2 Crew Salaries

The crew salaries can be determined by using yearly block hours that include basically time spent for one flight mission. Fort his estimation again 700 hours of flight time can be taken. To estimate yearly cost of crew salaries, the total flight hours need to be multiplied by crew cost per block hour and crew number. Two- man crew cost is calculated as it follows below.

Two-man crew cost = $70.4(V_C \frac{W_o}{10^5})^{0.3}$+168.8 = 606.21 per block hours (8.12)

Where takeoff gross weight (W_o) is 100,390 lb and cruise speed (V_C) equals to 439.25 knots.

The total crew salaries cost = 606.21 per block hours x 700 hours x 4 crew = 1,697,388 $ (8.13)

8.2.3 Maintenance Expenses

Maintenance Man-hours per Flight Hour (MMH/FH) is a mainly quota to determine aircrafts maintenance quality cost and it is a one of the important design goal to decrease MMH/FH.According to the table below maintance man-hours per flight hour is selected 3.5 to achive good design goal.

Table 8.2.2 LCC Parameter Approximations for MMH/FH

Aircraft Class	MMH/FH
Light Aircraft	0.25 - 1
Business Jet	3-6
Jet Trainer	6-10
Fighter Modern	10-15
Bomber	25-50
Military Transport	20-40
Civil Transport	5-15

According to the Bureau of Labor Statistics (May 2018), the maintance salary per hour is 88.7 $. Therefore, the total cost of maintance per year can be calculated as it follows;

$$\text{The maintance cost} = 700 \text{ hours x MMH/FH x } 88.7\ \$ = 217{,}315\ \$ \qquad (8.14)$$

Also material cost is found by using below formula.

$$\frac{\text{Material Cost}}{\text{Flight hour}} = 3.3\left(\frac{C_a}{10^6}\right) + 14.2 + \left[58\left(\frac{C_e}{10^6}\right)\right] - 26.1]N_e = 704.71\ \$ \qquad (8.15)$$

$$\text{Material Cost} = 700 \text{ x } 704.71 = 493{,}297\ \$ \qquad (8.16)$$

Where C is the aircraft cost less engine, C_e represents the cost per engine and N is the number of engine. As a result, the material cost found 493,297 $ for a year with 700 flight hours.

8.2.4 Depreciation

Depreciation is the distribution of an aircraft's sales price over it's own operational lifespan. The airframe depreciation is calculated If the depreciation time is 12 years and the resale value is 10% of the sales price,as below

$$\text{Yearly depreciation} = \text{Airframe cost x } \frac{0.9}{12} = 58{,}076{,}100 \text{ x } \frac{0.9}{12} = 4{,}355{,}707.5\ \$ \qquad (8.17)$$

Engine yearly depreciation can be calculated by the cost of the engine divided by the used year of the engine. If it is assumed 4 years then it can be calculated as it follows.

$$\text{Engine Yearly Depreciation} = \frac{4{,}750{,}500}{4} = 1{,}187{,}635\ \$ \qquad (8.18)$$

8.2.5 Insurance

It is approximately 1-3 % cost of operations for commercial aircrafts.

8.3 Cost Results and Discussion

The costs of the designed business jet are shown in Table 8.3.1. The most cost is available at RDT&E and production. This amount is approximately 63 million dollars. The higher ones among the other costs were the amount spent on the airframe and engine. The approximate total amount of these values is 5.5 million dollars. Many other costs are shown in the table. With this cost, it was decided that the sales amount of the aircraft would be approximately 75 million dollars. Although this fee may seem high, the other 6 aircraft in the competitor study in chapter 1 are sold at approximately these prices. For these reasons, it can be said that the price is reasonable.

Table 8.3.1 *Resulf of all Cost*

RDT&E and Production Cost	62,826,000 $
Selling Price	75,391,200 $
Fuel & Oil Cost	2,096,892 $
Crew Salaries Cost	1,697,388 $
Maintance Cost	217,315 $
Material Cost	493,297 $
Depreciation of Airframe Cost	4,355,707.5 $
Depreciation of Engine Cost	1,187,635 $
Insurance	100,482.34 $

Chapter 9 Environmental Issue

In environmental consideration, several factors show if the engine is environmentally friendly. The factors are engine type, fuel consumption & carbon dioxide emission, thrust reverser, noise, engine efficiency, and takeoff distance reduction. The engine is selected as CFM56-3. It is high-bypass turbofan, BPR is 6.0, and thrust range is 18500 to 23500 lbf with dry weight of 4276 lb. The engine has a high-pressure compressor, combustor, and high-pressure turbine. One factor that directly has an impact on engine performance is fan diameter. Compared with other engines and CFM56 model 2, the engine has a smaller diameter, lowering the fan blades' tip speed. That allows fan blades to operate at over 5.5% efficiency and improve specific fuel consumption by about 3%, improving the fuel efficiency. For Boeing 737, the engine burns 5000 pounds an hour. That number has positively affected the environment and significantly reduced carbon dioxide emissions. The engine reduces CO2 emission by 234 tones per airplane per year. Additional exhaust gas temperature (EGT) improvement up to 22 degrees reduces maintenance costs, so it longer wing-life cycle. Another aspect of decreasing fuel consumption during landing and takeoff is the way of the thrust reverser. It consists of sleeves that slide back to block bypass airflow to reduce thrust so that the aircraft slows down. The engine decreases takeoff distance by as much as 3500 ft, 1100 m, decreasing fuel usage by 25%. The other factor for environmental protection is the fact of noise. Shortened takeoff distance reduces noise by nearly 24dB by using chevrons nozzle. The nozzle reduces jet noise by 1.3% during takeoff conditions. Engine efficiency affects the environment. Thrust is 75 to 65 kN. With high-pressure compressor (HPC) operation, engine efficiency reaches up to 87%.

Chapter 10 Aircraft Layout and Landing gear

10.1 Landing Gear Sizing Airplane Tires

Landing gear is the system of wheels, struts, and brakes that allow an aircraft to take off, land, and move on the ground. It typically includes a main gear and a nose gear, and may also include additional wheels for larger aircraft. The main gear is located under the wings, while the nose gear is located at the front of the aircraft. The landing gear must be able to support the weight of the aircraft during takeoff, landing, and ground operations, and must also be able to absorb the shock of landing. For commercial airliners, the main gear tires are usually between 28 and 37 inches in diameter and can support up to 110,000 pounds of weight. [23]

Table 10.1 *Wheels diameter or width [23]*

Aircraft	**Diameter**		**Width**	
	A	B	A	B
British units : Main wheels diameter or width (in) = AW_W^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

$$\text{Main wheels diametre or width (in.)} = \mathrm{AW_W^B} \qquad (10.1)$$

Where W is takeoff gross weight, A&B for constant for different aircraft types. W is equal to 88365lbs., A is 1.63 and 0.1043 for diameter and with respectively. B is also 0.315 and 0.480 with respectively.

Table 10.1 *Landing Gear Sizing Airplane Tires*

[in]	**Wheels diameter**	**Wheels width**
Tires on wing and nose	38.45	12.45

Considering that our aircraft weight is approximately 88365lbs., when it is calculated the equation 10.1, the tires on wing and nose has 38.45 inches diameter and 12.45 inches width. The front and wing wheels are of equal size.

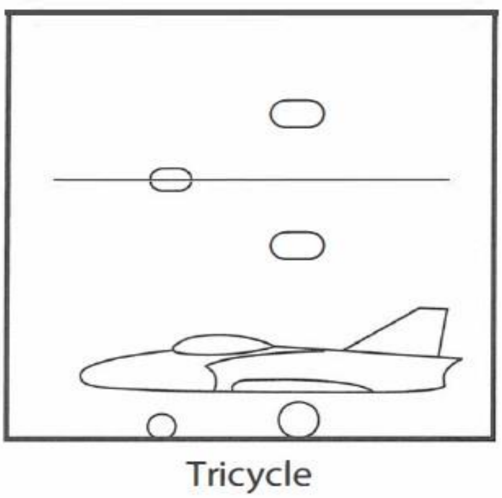


Figure 10.1 *Tricycle - Landing-gear arrangements [23]*

Today, the most popular type of landing gear is the "tricycle" configuration, which features two main wheels located behind the aircraft's center of gravity and an additional wheel positioned in front of it. This design allows for greater stability when landing, even at a significant angle to the runway, and also offers better visibility for the pilot when the aircraft is on the ground. [23] For these reasons, the tricycle configuration is used in this project.

10.2 Aircraft Layout

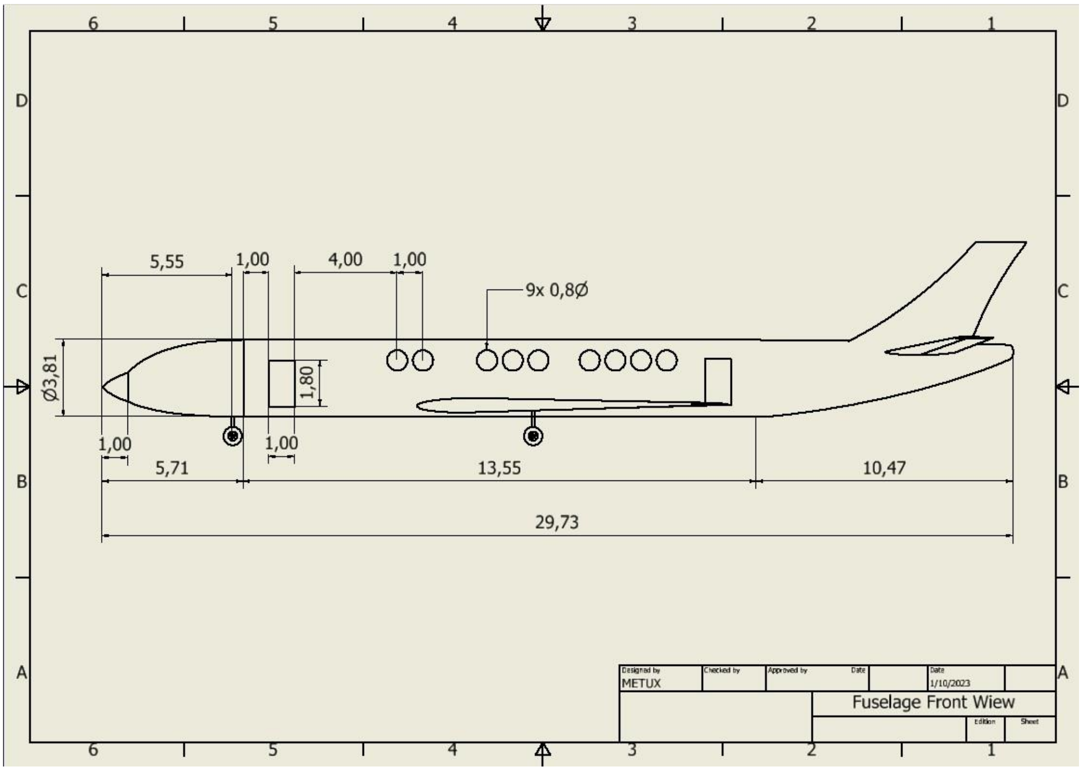


Figure 10.2.1 *Front view of Business jet aircraft, all dimension in meters*

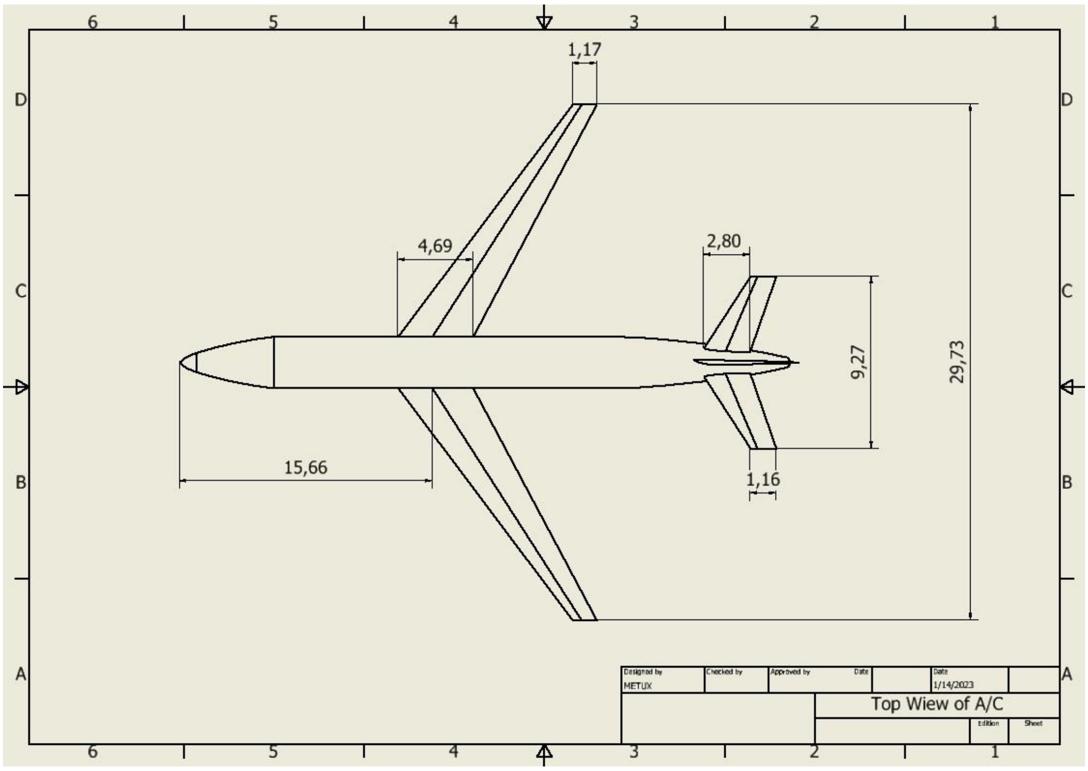


Figure 10.2.2 *Top view of Business jet aircraft, all dimension in meters*

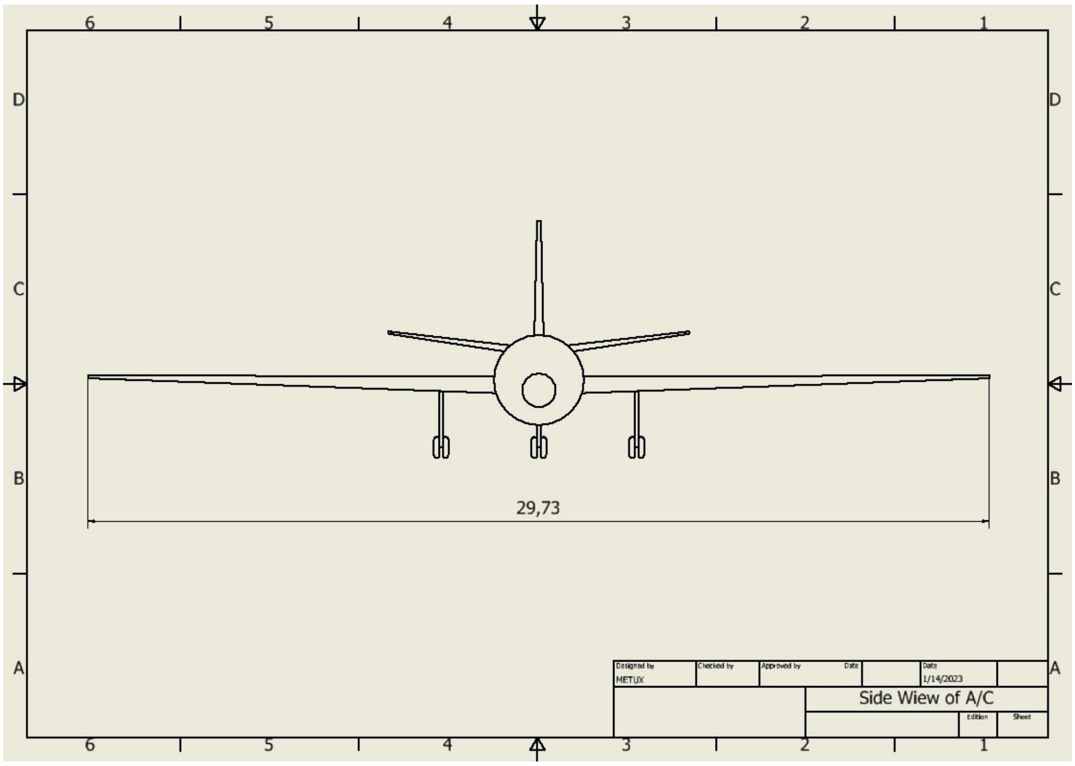


Figure 10.2.3 *Side view of Business jet aircraft, all dimension in meters*

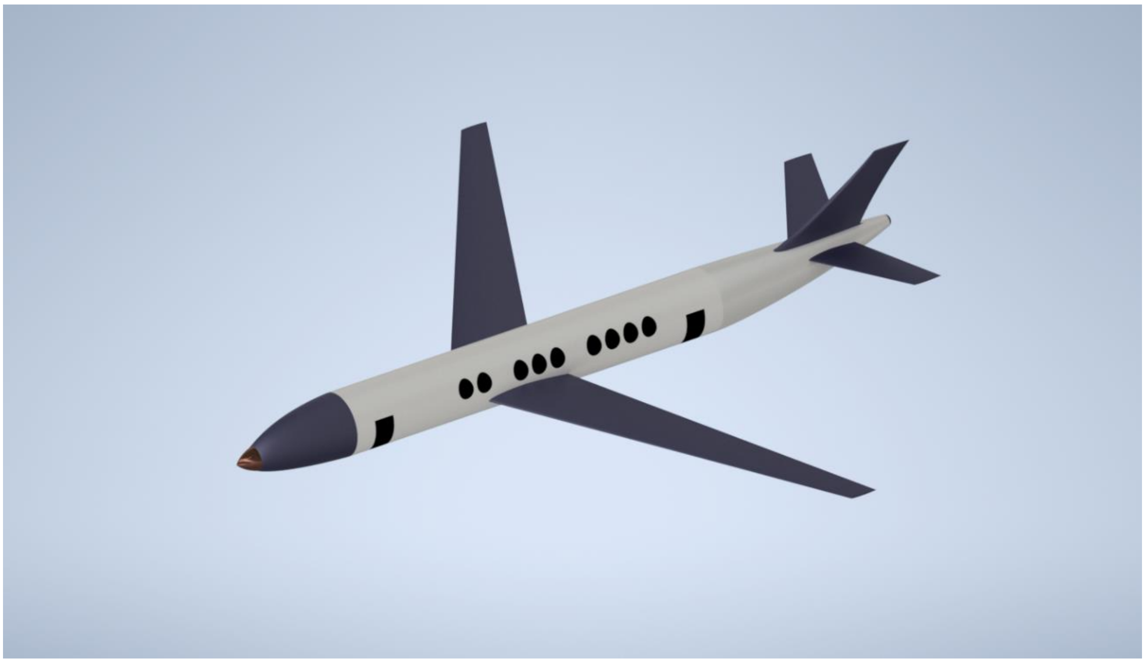


Figure 10.2.4 *Business jet aircraft with 3D*

Chapter 11 Summary of all parameters for the business jet

Table 11.1 *Summary of all parameters for the business jet*

External geometric parameters		
Parameter	**Units**	**Value/ Type/Name**
Total length	ft	97.53
Wingspan	ft	76.02
Outer cabin diameter	ft	12.49
Inner cabin diameter	ft	11.10
Wings		
Wing reference area	ft^2	731.50
Root airfoil	---	NACA 65(3)-618
Tip airfoil	---	NACA 63210
Aspect ratio	---	8.25
Wing sweep	degrees	22
Dihedral angle	degrees	5
Angle of incidence	degrees	2
Twist angle	Degrees	3
Taper ratio	---	0.25
Horizontal tail		
Airfoil	---	NACA 65(3)-618
Surface area	ft^2	197.88
Wing sweep	degrees	34
Aspect ratio	---	4.67
Vertical tail		
Airfoil	---	NACA 65(3)-618
Surface area	ft^2	138.28
Wing sweep	degrees	43.5
Aspect ratio	---	2.14
Cabin		
Number of seats	---	50+2
Business Seat width	inch	21
Business Seat height	inch	17
Business Seat pitch	inch	37
Business Aisle width	inch	20
Economy Seat width	inch	17
Economy Seat height	inch	17
Economy Seat pitch	inch	32
Economy Aisle width	inch	18
Performance parameters		
Maximum level speed	Mach	0.82
Stall speed	ft/s	182.27
Cruise speed	ft/s	744.48
Endurance	minutes	20
Service ceiling	ft	32000

Landing distance	ft	4485.03
Takeoff distance	ft	9000
Maximum payload	lb	12,970
Gross takeoff weight	lb	88,365
Engines (x2)	---	CFM56-3B-1

Table 11.1 shows all the values calculated in this project. These values are compatible with the values of 6 business jets in Chapter 1.

Conclusion

In conclusion, the goal of this project was to design an aircraft that can transport customers from Paris to the east coast of North America in less than 7.5 hours, taking into account various requirements. The design objective was achieved using contemporary technology and tools such as MATLAB and Inventor Professional, as well as publicly available data. The main objective in mind was to make this aircraft modern and acceptable, and to use as much logic as possible when designing each component. All components were designed piece by piece and then combined to create an amazing aircraft. Materials relevant to each component were assigned to calculate the aircraft's center of gravity correctly. To simplify the analysis, many logical assumptions were made throughout the design process and algorithms were integrated into MATLAB to achieve fast iterative results. In terms of today's standards, this aircraft is a success. It meets all the pre-determined requirements and stands out from its competitors with its superior range, wing load, and wing sweep. This aircraft has a modern interior with ample baggage space and legroom in the cabin and luggage compartment, and comfortable wide seats with a pitch of 1 meter. The wide seat arrangement allows for comfortable flight, making this aircraft a good candidate among business class aircraft.In Chapter 1, a study was carried out on six different types of business jets (ERJ145 LR, Boeing 737-500 VIP, FOKKER 70, Bombardier CRJ550, and Antonov An-148) to develop a new jet. Three tables were compiled to display the research findings, which included information on the general features, geometric features, and performance and design characteristics of the aircraft. The maximum takeoff gross weight of the jets ranges from 48,500 lb to 133,500 lb and the weight of the jet being designed was found to be 88,000 lb, similar to the other aircraft. The geometric characteristics table reveals that the length of each plane is 95 ft, while the wing span values vary except for the ERJ 145 LR, which has the smallest wingspan. The wingspan of the aircraft being designed was determined to be around 76 ft. The new business jet will take into account the positive attributes of these six aircraft and will be similar in terms of its features. Furthermore, by examining the performance and design characteristics table, it can be seen that the range of the jets varies between 1,500 and 3,000 miles, and the maximum altitude they can reach varies between 41,000 and 45,000 ft, similar to the jet being designed. It is evident from this research that there is a wide range of features and capabilities among the six aircraft studied. In Chapter 1, weight fractions for the ERJ 145 LR, Boeing 37-500 VIP and Bombardier CRJ550 business jets were calculated using the competition study. In addition, new values of weight fractions were found using the statistical curve fitting equation. The coefficients were then replaced with optimal values for the statistical curve fitting equation. We have the necessary values to be able to interpret these values. Equivalent curb weight percentage numbers for jet transport aircraft range from 0.5 to 0.6 for selected size take-off weights. Wo. In conclusion, their results appear to be consistent with current trends in the empty weight fraction. In section 2.2, a simple mission profile is defined, which includes task segments such as take-off, climb and cruise, descent, hover. Two airports were chosen to determine the range. These airports are John F.Kennedy International Airport (JFK) and Paris-Charles De Gaulle Airport (CDG). As a result, idleness is included in the calculation of the required time. Cruising speed is then calculated. The CFM-56-3B-1 engine was determined to be suitable for this project. Therefore, some parameters such as pilot, crew, passengers and baggage are defined to design the take-off gross weight estimation. From the figures we found with the help of the MATLAB code, the take-off gross weight was 95,869 lb, and the curb weight ratio was 0.5259. These values were found to be close to the values of the 6 aircraft examined in the competitor study. As an example, our takeoff weight is 95,869 lb, while the Fokker 70 is 87,998 lb and the Antonov An-148 is 96,342 lb. In Chapter 4, calculations and discussions are focused on determining design parameters for the aircraft. A key factor in understanding flight is the lift coefficient, represented as C_L. Based on design data and past trends, C_L is determined to be 0.51, and is found to be sufficient after comparing to similar aircraft. Airfoil selection is made by taking into account factors such as stall characteristics, weight, critical Mach number, and Reynolds number. The 4 NACA 65(3)-618 airfoil is chosen for its lift coefficient of 1.430 and thick profile. This airfoil is chosen because it has the highest maximum lift coefficient among the options. The wing's aspect ratio is estimated at 8.25, which is selected based on historical trends and its relation to the Mach number. For the high-lift devices, the Fowler Flap and Krueger Flap are chosen after evaluating the pros and cons of each option. This results in an increase in the wing's area and lift, particularly at low speeds. In chapter 6, it is shown that a few parameters need to be taken into account to estimate the refined weight method more accurately. Take-off gross weight is calculated with a developed statistical equation using MATLAB code with less than 0.01 absolute error, and the total fuel weight is determined by calculating the weight of fuel burnt in each segment. It is estimated that the improved weight estimate will result in a lower take-off gross weight than the first weight estimate determined in sequence and proportionally. Furthermore, one of the most important effective metrics, lift-drag ratio of 15.73, has a higher empty weight percentage but still has a larger value. According to previous estimates in chapter 2, the take-off gross weight and empty weight ratio values were found to be 95,869lb and 0.5259 respectively. Therefore, this weight estimation method for the conceptual design process of the aircraft is expected to produce better and more reliable results. After all calculations, the code produces an exact estimate of the take-off gross weight of 88,365l lb for our business jet. When all factors and estimates are taken into account, this weight is more accurate for the business jet and closer to the gross take-off weight. The weight of our aircraft is not determined in the first weight estimation process. Chapter 7 focuses on determining the appropriate size for the airplane's wing, drawing the fuel tank, and determining the location of the center of gravity. The first part includes calculating the required wing size and fuel volume based on data from previous assignments using geometric and aerodynamic parameters and selected wing load and aspect ratio. Calculations show that the wing length is 76.02 ft, the wing area is $731.50 \mathrm{ft}^2$, the root chord is 15.394 ft, the end chord is 3.848 ft, and the Average Aerodynamic Chord is 10.774 ft. These results are consistent with the number of similar aircraft found in the competitor study. Later calculations determined that the fuel volume of the aircraft was $621.85 \mathrm{ft}^3$. The wing and tail configurations were also drawn, including fuel tanks. The second part involves determining the weights of the main components, including the wing, tail, engine, fuselage, passengers, payload, crew and fuel tanks. All calculations for this section are described in Table 7.4.2 and the positions of the major components and their corresponding moment arm positions are also calculated. Determining the center of gravity was done by considering the weights of all components except the fuel tanks and the wing where the engines are located. The overall center of gravity is the coordinate 51.3815 ft from the nose chosen as the coordinate of zero on the x-axis. For cost analysis, Table 8.3.1 displays the costs of the newly designed business jet. Most of the costs are for Research, Development, Test and Evaluation (RDT&E) and production, which amount to approximately 63 million dollars. The other significant costs are for the airframe and engine, which total around 5.5 million dollars. There are also many other costs listed in the table. Based on these costs, it was determined that the sales price of the aircraft would be approximately 75 million dollars. While this price may seem high, it is comparable to the prices of the other six aircraft in the competitor study discussed in chapter 1. Therefore, it can be argued that the price is reasonable. As a result, approximate values of designed bussines jet external geometric parameters, performance and performance were found. For this reason, it is predicted that the plane can go between Paris Charles de Gaulle Airport, France and John F. Kennedy International Airport, USA in 7 hours and 30 minutes.

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Appendix

MATLAB Code for Chapter 2

Initial Takeoff Weight Estimation

```
clc
clear
close all;
format short

E          = 1/3;                      %% Loiter time 20 minutes = 0.333 h
R          = 3622.59;                  %% Paris to Newyork distance [mi]
Wc         = 240*3;                   %% 2 pilot and 1 crew weight [lb]
Wadult     = 195;                      %% Passegers weight [lb]
Wbag       = 16;                       %% baggage weight [lb]
Pass       = 50;                       %% Number of passengers
Wp         = (Wadult+Wbag)*Pass;       %% Payload weight [lb]
LD         = 15;                       %% Lift/Drag coeffient from competitor study
Cj         = 2*0.36;                   %% Thrust Spesific Fuel Consuption
V          = 503.311;                  %% Cruise speed [mi/h]

%%% Weight fractions for the mission segments %%%
W1_0 = 0.995;                          %% Take off
W2_1 = 0.98;                           %% Climb
W3_2 = exp((-R*Cj)/(V*LD));            %% Cruise
W4_3 = 0.99;                           %% Descend
W5_4 = exp((-E*Cj)/LD);                %% Loiter
W6_5 = 0.992;                          %% Land
Mf   = W1_0*W2_1*W3_2*W4_3*W5_4*W6_5;  %% Mission segments weight fractions
Wf   = 1.06*(1-Mf);                    %% Fuel weight fraction

%%% Statistical curve-fit equations
A          = 1.02;                    %% A constant for jet transport
Kvs        = 1.00;                    %% Kvs constant for jet transport
C          = -0.06;                   %% C constant for jet transport
Wo_initial = 1000;                    %% Initial Guess take off gross weight [lb]
tol        = 0.02;                    %% Tolerans of 2%
W0         = 0;

while abs(Wo_initial-W0)>tol
    Wfract = A*(Wo_initial^C)*Kvs;
    W0 = (Wc+ Wp) / (1-Wf-Wfract);
    Wo_initial = Wo_initial + 0.01;
end

display(W0)
display(Wfract)
```

Different Endurance Time

```
clc
clear
close all;
format long

E          = [1/6:0.01:3];          %% Loiter time between 10 minutes to 3 hours
R          = 3622.59;               %% Paris to Newyork distance [mi]
Wc         = 240*3;                %% 2 pilot and 1 crew weight [lb]
Wadult     = 195;                   %% Passegers weight [lb]
Wbag       = 16;                    %% baggage weight [lb]
Pass       = 50;                    %% Number of passengers
Wp         = (Wadult+Wbag)*Pass;    %% Payload weight [lb]
LD         = 15;                    %% Lift/Drag coeffient from competitor study
Cj         = 2*0.36;                %% Thrust Spesific Fuel Consuption
V          = 503.311;               %% Cruise speed [mi/h]

%%% Weight fractions for the mission segments %%%
W1_0 = 0.995;                       %% Take off
W2_1 = 0.98;                        %% Climb
W3_2 = exp((-R*Cj)/(V*LD));         %% Cruise
W4_3 = 0.99;                        %% Descend
W5_4 = exp((-E*Cj)/LD);             %% Loiter
W6_5 = 0.992;                       %% Land
Mf   = W1_0*W2_1*W3_2*W4_3*W5_4*W6_5; %% Mission segments weight fractions
Wf   = 1.06*(1-Mf);                 %% Fuel weight fraction

%%% Statistical curve-fit equations
A          = 1.02;                 %% A constant for jet transport
Kvs        = 1.00;                 %% Kvs constant for jet transport
C          = -0.06;                %% C constant for jet transport
Wo_initial = 10000;                %% Initial Guess take off gross weight [lb]
tol        = 0.02;                 %% Tolerans of 2%
W0         = 0;

while abs(Wo_initial-W0)>tol
    Wfract = A*(Wo_initial^C)*Kvs;
    W0 = (Wc+ Wp) ./ (1-Wf-Wfract);
    Wo_initial = Wo_initial + 0.01;
end

plot(E,W0,'k','LineWidth', 2 )
xlabel('Endurance [h]')
ylabel('W0 [lb]')
title('Endurance vs Takeoff Gross Weight')
grid on
```

Different Number of Passengers

```
clc
clear
close all;
format long

E          = 1/3;                   %% Loiter time 20 minutes = 0.333 h
R          = 3622.59;               %% Paris to Newyork distance [mi]
Wc         = 240*3;                %% 2 pilot and 1 crew weight [lb]
Wadult     = 195;                   %% Passegers weight [lb]
Wbag       = 16;                    %% baggage weight [lb]
Pass       = [10:100];              %% Number of passengers
Wp         = (Wadult+Wbag)*Pass;    %% Payload weight [lb]
LD         = 15;                    %% Lift/Drag coeffient from competitor study
Cj         = 2*0.36;                %% Thrust Spesific Fuel Consuption
V          = 503.311;               %% Cruise speed [mi/h]

%%% Weight fractions for the mission segments %%%
W1_0 = 0.995;                       %% Take off
W2_1 = 0.98;                        %% Climb
W3_2 = exp((-R*Cj)/(V*LD));         %% Cruise
W4_3 = 0.99;                        %% Descend
W5_4 = exp((-E*Cj)/LD);             %% Loiter
W6_5 = 0.992;                       %% Land
Mf   = W1_0*W2_1*W3_2*W4_3*W5_4*W6_5; %% Mission segments weight fractions
Wf   = 1.06*(1-Mf);                 %% Fuel weight fraction

%%% Statistical curve-fit equations
A          = 1.02;                 %% A constant for jet transport
Kvs        = 1.00;                 %% Kvs constant for jet transport
C          = -0.06;                %% C constant for jet transport
Wo_initial = 10000;                %% Initial Guess take off gross weight [lb]
tol        = 0.02;                 %% Tolerans of 2%
W0         = 0;

while abs(Wo_initial-W0)>tol
    Wfract = A*(Wo_initial^C)*Kvs;
    W0 = (Wc+ Wp) / (1-Wf-Wfract);
    Wo_initial = Wo_initial + 0.01;
end

plot(Pass,W0,'b','LineWidth', 2 )
xlabel('Number of Passengers')
ylabel('W0 (lb)')
title('Number of Passengers vs Takeoff Gross Weight')
grid on
```

Different Ranges

```
clc
clear
close all;
format long

E          = 1/3;                    %% Loiter time 20 minutes = 0.333 h
R          = [500:50:4000];          %% Different distance between 500 to 4000 [mi]
Wc         = 240*3;                 %% 2 pilot and 1 crew weight [lb]
Wadult     = 195;                    %% Passegers weight [lb]
Wbag       = 16;                     %% baggage weight [lb]
Pass       = 50;                     %% Number of passengers
Wp         = (Wadult+Wbag)*Pass;     %% Payload weight [lb]
LD         = 15;                     %% Lift/Drag coeffient from competitor study
Cj         = 2*0.36;                 %% Thrust Spesific Fuel Consuption
V          = 503.311;                %% Cruise speed [mi/h]

%%% Weight fractions for the mission segments %%%
W1_0 = 0.995;                        %% Take off
W2_1 = 0.98;                         %% Climb
W3_2 = exp((-R*Cj)/(V*LD));          %% Cruise
W4_3 = 0.99;                         %% Descend
W5_4 = exp((-E*Cj)/LD);              %% Loiter
W6_5 = 0.992;                        %% Land
Mf   = W1_0*W2_1*W3_2*W4_3*W5_4*W6_5; %% Mission segments weight fractions
Wf   = 1.06*(1-Mf);                  %% Fuel weight fraction

%%% Statistical curve-fit equations
A          = 1.00;                  %% A constant for jet transport
Kvs        = 1.04;                  %% Kvs constant for jet transport
C          = -0.06;                 %% C constant for jet transport
Wo_initial = 10000;                 %% Initial Guess take off gross weight [lb]
tol        = 0.02;                  %% Tolerans of 2%
W0         = 0;

while abs(Wo_initial-W0)>tol
    Wfract = A*(Wo_initial^C)*Kvs;
    W0 = (Wc+ Wp) ./ (1-Wf-Wfract);
    Wo_initial = Wo_initial + 0.01;
end

display(Wfract)
plot(R,W0,'r','LineWidth', 2 )
xlabel('Range [mi]')
ylabel('W0 [lb]')
title('Range vs Takeoff Gross Weight')
grid on
```

MATLAB code for Chapter 5

T/W versus W/S

```
clc
clear
close all

q     = 960.35;
CD0   = 0.015;
n     = 2;
K     = 0.05;
e     = 0.8;
AR    = 8.25;
TOP   = 215;
sigma = 0.2978;
CLtakeoff = 1.18;
Vstall = 55.56;
rho = 1.225;
CLmax = 1.430;
S = 67.64;
C_V= 1;


W_S = 0:5:200;
T_W = 0:0.01:0.5;

T_W_sustained = q*CD0./(W_S) + (W_S)*n.^2./(q*pi*AR*e);
T_W_takeoff = W_S./((TOP)*sigma*CLtakeoff);
W_S_stall = 0.5*rho*(Vstall^2)*CLmax;
T_W_cruise = (sigma.^(3/4))*(((CD0*q)./W_S + W_S./(pi*AR*e*q)));
T_W_climb = (q*S*CD0./W_S + W_S./(q*pi*AR*e));
%T_W_climb = (sigma.^0.75)+2*((C_V) + 2*sqrt(CD0./(AR*e*pi)));


plot(W_S, T_W_sustained,'r', 'LineWidth', 2)
hold on
plot(W_S, T_W_takeoff,'b', 'LineWidth', 2)
hold on
plot(W_S, T_W_cruise,'g', 'LineWidth', 2)
hold on

xlabel('W/S (lb/ft^2)')
ylabel('T/W')
legend('Sustained Turn', 'Takeoff','Cruise')
hold on
```

MATLAB Code for Chapter 6

Refined Weight Estimation

```
clc
clear
close all
format short

%% Parametres for Lift/Drag %%
e       = 0.8;                             %% Oswald span efficiency
CDo     = 0.015;                           %% Zero lift drag coefficient
W_S     = 328.30*9.81;                     %% Wing Loading [N] at Cruise speed from
Assigment 5
AR      = 8.25;                            %% Aspect Ratio
V       = 225.9;                           %% Cruise speed [m/s]
rho     = 1.225;                           %% Air density kg/m3
q       = 0.5*rho*V^2/9.81;                %% Dynamic pressure [N]
LD      = 1/((q*CDo/W_S)+(W_S/(q*pi*AR*e))); %% Lift/Drag coeffient
LD_max  = LD/0.866 ;                       %% Lift/Drag max coeffient
%% Mission Segment Weight Fractions Parametres %%
Cj    = 2*0.36;                     %% Thrust Spesific Fuel Consuption
V     = 505.324;                    %% Cruise speed [mi/h]
R     = 3622.59;                    %% Paris to Newyork distance [mi]
E     = 1/3;                        %% Loiter time 20 minutes = 0.333 h
M     = 0.766;                      %% Mach Number
%% Weights %%
Wc      = 240*3;                        %% 2 pilot and 1 crew weight [lb]
Wadult = 195;                       %% Passegers weight [lb]
Wbag    = 16;                       %% baggage weight [lb]
Wroller= 30;                        %% Crew member roller bag weight [lb]
Pass    = 50;                       %% Number of passengers
WfixP   = (Wadult+Wbag+Wroller)*Pass; %% Fix Payload weight [lb]
Wd      = 0;                        %% Dropped Payload [lb]
%% Mission segments Weight ratios %%
W1_0 = 0.99;                        %% Take off
W2_1 = 1.0065 - 0.0325*M;           %% Climb
W3_2 = exp((-R*Cj)/(V*LD));         %% Cruise
W4_3 = 0.995;                       %% Descend
W5_4 = exp((-E*Cj)/LD_max);         %% Loiter
W6_5 = 0.997;                       %% Land
%% Improved statistical equations (jet aircraft) parametres %%
a      = 0.32;
b      = 0.66;
C1     = -0.13;
C2     = 0.3;
C3     = 0.06;
C4     = -0.05;
C5     = 0.05;
K_vs   = 1;
Mmax   = 0.766;
%% Some important Parameters for Improved statistical equations %%
W_S         = 20.4950;      %% Wing Loading [lb/ft3] at Cruise speed from Assigment 5
Wo_initial = 1000;          %% Initial Guess take off gross weight [lb]
tol         = 0.01;         %% Tolerans of 0.1%
TW          = 1.15;         %% Thrust/Weight ratio from histrocial trends
W0          = 0;
```

```matlab
while abs(Wo_initial-W0)>tol
    Wfract= (a + b*((W0)^C1)*(AR^C2)*(TW^C3)*(W_S^C4)*(Mmax^C5))*K_vs;
    Wf1   = (1 - W1_0)*Wo_initial;
    W1    = W1_0*Wo_initial;
    Wf2   = (1 - W2_1)*W1;
    W2    = W2_1 *W1;
    Wf3   = (1 - W3_2)* W2;
    W3    = W3_2 * W2;
    Wf4   = (1 - W4_3)* W3;
    W4    = W4_3 * W3;
    Wf5   = (1 - W5_4)* W4;
    W5    = W5_4 * W4;
    Wf6   = (1 - W6_5)*W5;
    Wfuel = Wf1 + Wf2 + Wf3 + Wf4 + Wf5 + Wf6 ;
    W0 = Wc + WfixP + Wd + Wfuel + (Wfract*Wo_initial);
    Wo_initial = Wo_initial + 0.01;
end
display(Wf1)
display(Wf2)
display(Wf3)
display(Wf4)
display(Wf5)
display(Wf6)
display(W0)
display(Wfract)
```

MATLAB Code for Chapter 8

Cost Analysis

```
clc; clear; close
W_e=85869 ; %lb
V=473 ;%knot
Q= 200 ; %quantity factor
FTA= 4; %flight test number
T_max= 34000 ;%lb
M_max= 0.82 ;% max mach number
T_inlet=2160 ; %R
V_C= 439.25 ;
W_0=100390 ;
HE= 4.86*W_e^0.777*V^0.894*Q^0.163 ;
HT= 1.2*(5.99*W_e^0.777*V^0.696*Q^0.263 );
HM= 7.37*W_e^0.82*V^0.484*Q^0.641 ;
HQ  = 0.133*HM ;
CD= 91.3*W_e^0.630*V^1.3 ;
CF= 2498*W_e^0.325*V^0.822*FTA^1.21 ;
CM= 22.1*W_e^0.921*V^0.621*Q^0.799 ;
CENG=3112*[0.043*T_max+243.25*M_max+0.969*T_inlet-2228]
CAV=4000*85869 ;
Aircraft_Cost=(HE*115+HT*118+HQ*108+HM*98+CD+CF+CM+CENG*2+CAV)/200

twomancrew_cost=70.4*(V_C*(W_0/10^5))^(0.3)+168.8
```