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MIDDLE EAST TECHNICAL UNIVERSITY
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ASE 451
Aeronautical Engineering Design

Project Report
Business Jet Design

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Introduction

The goal of this project is to create a modern business jet that can carry up to 50 passengers. The jet is designed for point-to-point travel, with a flight time of 7.5 hours from Paris to New York. A literature survey has been initiated in response to the design requirements, and a competitor study has been conducted among the existing business jets on the market. Based on this research, an attempt was made to design a jet with superior economic and environmental performance. Furthermore, in accordance with the requirements, the design is intended to be lightweight, have a cruise speed with strong performance characteristics, and place a high priority on passenger comfort.

Number of passengers	50
Number of crew members	2
Takeoff airport	Paris Charles de Gaulle Airport, France
Landing airport	John F. Kennedy International Airport, USA
Range	5850 km (Approximate distance between the airports)
Flight Duration	7.5 hours

Chapter 1. Design Requirements and Competitor Study

The goal of this chapter is to look for similar types of planes with roughly the same number of passengers. The results of this chapter are shown in the tables, and the values are used throughout the report to compare with our design values.

Table 1. General Characteristics

General Characteristics	**Boeing BBJ 787-800**	**Boeing 737-500 VIP**	**Bombardier CRJ 200**	**Mitsubishi MRJ90**	**Falcon 2000Dx**
Airplane Type	Private Jet	Private Jet	Regional Jet	Regional Jet	Private Jet
Number of Seats	45	50	50	96	18-30
Crew	2	2	2	2	2
MTOW (W)	502,494 lb (227,930 kg)	133,501 lb (60,555 kg)	53,000 lb (24,041 kg)	94,358 lb (42,700 kg)	41,000 lb (18600 kg)
OEW	277,200 lb (125,736 kg)	70,510 lb (31,983 kg)	30,900 lb (14,016 kg)	57,320 lb (25,999.9 kg)	20735 lb (9405 kg)
Fuel Capacity	33,340 gal (126,205 litre)	5,310.65 gal (20,103 litre)	1,714.1 gal (6489 litre)	3,200 gal (12,113 litre)	1,814 gal (6,866 litre)
Maximum Payload	77,822 lb (35,300 kg)	32,562 lb (14,770 kg)	13,100 lb (5942 kg)	79,697 lb (36,150 kg)	5700 lb (2585 kg)

Cruise Speed	Mach 0.85 (567 knots) (1050.08 km/h)	Mach 0.74 (493 knots) (913.04 km/h)	Mach 0.63 424 knots (785 km/h)	Mach 0.78 (520.02 knots) (963.5 km/h)	Mach 0.68 (459 knots) (851.2 km/h)
Maximum Speed	515 knots (954 km/h)	492 knots (911 km/h)	464 knots (860 km/h)	489 knots (905 km/h)	573 knots (1061 km/h)
Range	9,945 nmi (18,418 km)	2,402 nmi (4,449 km)	1,700 nmi (3,148 km)	2,040 nmi (3,230 km)	3,740 nmi (6,020 km)
Ceiling	43,000 feet	37,000 feet	41,000 feet	39,000 feet	50,900 feet
Takeoff (MTOW, SL, ISA)	9,186 feet (2,800 m)	8,249 feet (2,514 m)	5,800 feet (1,768 m)	5,710 feet (1,740.4 m)	5,436 feet (1,657 m)
Landing (MLW, SL, ISA)	4,987 feet (1,520 m)	6,249 feet (1,905 m)	4,850 feet (1,479 m)	4,860 feet (1,481 m)	2,579 feet (786 m)
Model, Type (x2)	Rolls-Royce Trent 1000	CFM56-3B1	General Electric CF34-3B1 Turbofan	MRJ90 2x Turbofan Engines PW1200G Pratt & Whitney	2 × Pratt & Whitney Canada PW308C turbofan engines
Static Thrust (x2)	64,100 lbf (285 kN)	18,500 lbf (82.3 kN)	9220 lbf (41.0 kN)	15,000-17,000 lbf (66.7 -75.7 kN)	7,000 lbf (31 kN)
Bypass Ratio	10:1	5:1	6.2:1	9:1	4:1
Thrust (takeoff) (x2)	64,000 lbf (280 kN)	20,100 lbf (89.4 kN)	8,729 lbf (38.84 kN)	34,200 lbf (15.21 kN)	7,000 lbf (31 kN)

As it can be seen from Table 1, reflecting the general characteristics for various airplanes, which are Boeing BBJ 787-800, Boeing 737-500 VIP , Bombardier CRJ 200, Mitsubishi MRJ90, Falcon 2000Dx [9-11].

Table 2. Geometric Charactheristics

General Characteristics	**Boeing BBJ 787-800**	**Boeing 737-500 VIP**	**Bombardier CRJ 200**	**Mitsubishi MRJ90**	**Falcon 2000Dx**
Fuselage Length (m)	56.72	31.10	26.77	35.8	20.23

Fuselage Height (m)	5.94	4.1	3.16	10.50	7.06
Fuselage Width (m)	5.77	3.76	2.69	3.0	-
Cabin Width (m)	5.49	3.54	2.53	2.76	2.34
Finess Ratio	9.54	7.58	8.47	-	-
Wing Span (m)	60.17	28.90	21.21	29.2	19.33
Wing Area (m^2)	377	105.4	48.35	89.8	49
Aspect Ratio	9.6	9.16	9.3	9.5	7.66
Taper Ratio	-	0.240	0.259	0.25	-
Wing Sweep	32.2°	25°	24.5°	24°	29°
Dihedral	-	6°	4°	-	-
Cargo Volume (m^3)	138.2	23	13.8	18.2	-

As it can be seen from Table 2, reflecting the geometric characteristics for various airplanes, which are Boeing BBJ 787-800, Boeing 737-500 VIP , Bombardier CRJ 200, Mitsubishi MRJ90, Falcon 2000Dx [9-11].

Table 3. Important Performance and Design Characteristics

Important Performance and Design Characteristics	**Boeing BBJ 787-800**	**Boeing 737-500 VIP**	**Bombardier CRJ 200**	**Mitsubishi MRJ90**	**Falcon 2000Dx**
Wing Loading (W/S)	123.83 lb/sq ft	101.803 lb/sq ft	101.84 53 lb/sq ft	-	89 lb/sq ft
Thrust-to-Weight Ratio (T/W)	0.255	0.320	0.348	0.360	0.341

Empty Weight Fraction (W_E/ W_O)	0.552	0.609	0.583	0.607	0.506
Fuel Weight Fraction (W_F/ W_O)	0.556	0.307	0.595	0.283	0.369
Maximum Lift Coefficient (for takeoff and landing)	-	-	-	-	-
Payload Fraction (W_P/ W_O)	0.155	0.239	0.247	0.845	0.072

As it can be seen from Table 2, reflecting the important performance and design characteristics characteristics for various airplanes, which are Boeing BBJ 787-800, Boeing 737-500 VIP , Bombardier CRJ 200, Mitsubishi MRJ90, Falcon 2000Dx.

Chapter 2. Preliminary Weight Estimation

This project's goal is to have an initial estimate of the weight fraction for each mission segment as well as the empty-weight to gross weight fraction.

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

The selected aircrafts are:

- Bombardier CRJ 200
- Boeing 737-500 VIP
- Falcon 2000DX

The empty weight fraction (W_E/W_0) for an airplane is given by

$$\frac{W_E}{W_0} = AW_0^C K_{VS} \ [1]$$

where W_E is the empty weight of an airplane, W_0 is the design takeoff gross weight. A and C are constants and can be estimated from Table.4. K_{VS} is the variable sweep constant. For variable sweep, K_{VS} is 1.04 and for fixed sweep, K_{VS}.

Table 4. Empty Weight Fraction

$W_e/W_0 = A W_0^C K_{vs}$	A	{A-metric}	C
Sailplane—unpowered	0.86	{0.83}	−0.05
Sailplane—powered	0.91	{0.88}	−0.05
Homebuilt—metal/wood	1.19	{1.11}	−0.09
Homebuilt—composite	1.15	{1.07}	−0.09
General aviation—single engine	2.36	{2.05}	−0.18
General aviation—twin engine	1.51	{1.4}	−0.10
Agricultural aircraft	0.74	{0.72}	−0.03
Twin turboprop	0.96	{0.92}	−0.05
Flying boat	1.09	{1.05}	−0.05
Jet trainer	1.59	{1.47}	−0.10
Jet fighter	2.34	{2.11}	−0.13
Military cargo/bomber	0.93	{0.88}	−0.07
Jet transport	1.02	{0.97}	−0.06
UAV—Tac Recce & UCAV	1.67	{1.47}	−0.16
UAV—high altitude	2.75	{2.39}	−0.18
UAV—small	0.97	{0.93}	−0.06

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

i) Bombardier CRJ 200: $\frac{W_E}{W_0} = \frac{30900\ lb}{53000\ lb} = 0.583$

Boeing 737-500 VIP: $\frac{W_E}{W_0} = \frac{70510\ lb}{133501\ lb} = 0.528$

Falcon 2000DX: $\frac{W_E}{W_0} = \frac{20735\ lb}{41000\ lb} = 0.507$

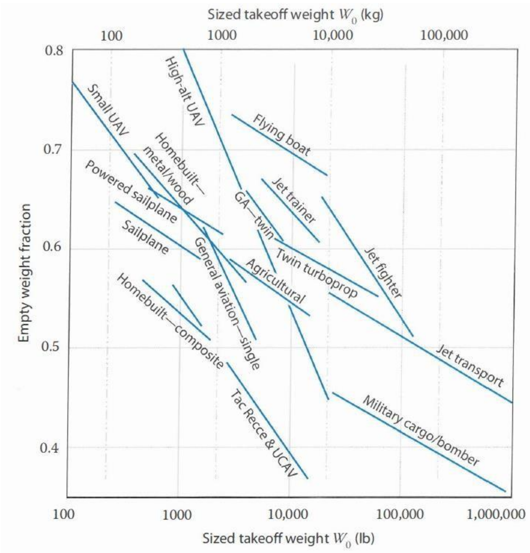

Figure 1. Empty weight fraction trend [14].

The relationship between empty-weight fraction and sized takeoff weight is shown in Figure 1 above. It can be observed that the empty-weight fraction ratio for Bombardier CRJ 200 it is 0.540, for Boeing 737-500 VIP it is 0.500 and for Falcon 2000DX it is 0.500.

ii) Equation [1] is used to calculate the empty weight fraction. Since the selected aircraft types are all jet transports, A is 1.02, C is -0.06 and K_{VS} is 1, as shown in Table 4.

Bombardier CRJ 200:

$$\frac{W_E}{W_0} = 1.02\ W_0^{-0.06} = 1.02\ (53000\ lb)^{-0.06} = 0.531,$$

Boeing 737-500 VIP:

$$\frac{W_E}{W_0} = 1.02\ W_0^{-0.06} = 1.02\ (133501\ lb)^{-0.06} = 0.502,$$

Falcon 2000DX:

$$\frac{W_E}{W_0} = 1.02\ W_0^{-0.06} = 1.02\ (41000\ lb)^{-0.06} = 0.539.$$

iii) Because the similarity between the values calculated in equation 1 and the values calculated in part I is high, the A and C values for the three selected aircrafts are not changed. They are 91%, 95%, and 94% correct, respectively.

iv) As can be seen from the calculations above, when estimating the empty-weight fractions of the chosen aircraft, equation 1 produced results that were very close to the original ones calculated using real values of W_0 and W_E. The real value for the Bombardier CRJ200, which is 0.583 as calculated in part i and the value found using equation [1], which is 0.531, differ by 6%. As a result, there is no need to change the values of constants A and C because doing so may result in significantly large or small values. Similarly, in the cases of the Boeing 737-500 VIP and the Falcon 2000DX, equation [1] yielded values that were 95 percent and 94 percent of the originals, respectively. As a result, varying the constants A or C is not recommended.

2.2. Design takeoff gross weight estimation

2.2.1.

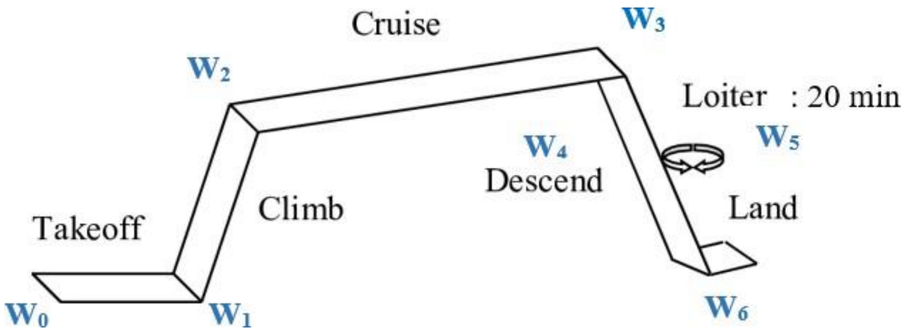

Figure 2. Mission Profile [14]

As shown in Figure 2, the mission profile is divided into six segments: takeoff (0-1), climb (1-2), cruise (2-3), descend (3-4), loiter (4-5), and landing (5-6).

Estimations:

- The contemporary business jet will fly from Paris to New York. The approximate distance between Paris Charles de Gaulle Airport and New York John F. Kennedy International Airport is 3158.75 nautical miles (5850 kilometers).
- Loiter time equals 2 hours.
- Cruise speed is 421 $\mathrm{nmi/h}$ (780 $\mathrm{km/h}$) which is defined as,

$$\text{Cruise speed} = \frac{\text{Distance btw airports}}{\text{average flight time}},$$

$$\text{Cruise speed} = \frac{3158.75\ \text{nmi}}{7.5\ \text{h}} = 421.12\frac{\text{nmi}}{\text{h}}.$$

2.2.2. Engine Selection

PW1400G is selected for turbofan engine from Pratt & Whitney.

Table 4. Engine Specifications [1],[15]

ByPass Ratio [BPR]	12:1
Length [inch]	133.9
Fan Diameter [inch]	81
Takeoff Thrust [lbf]	33,000
Specific Fuel Consumption [SFC][lb/lbfh]	0.51

2.2.3. Design Take-off Gross Weight Estimation:

The design takeoff gross weight estimation can be calculated by:

$$W_0 = \frac{W_{CREW} + W_{PAYLOAD}}{1 - \left(\frac{W_f}{W_0}\right) - \left(\frac{W_E}{W_0}\right)}, [2]$$

where W_{CREW} is the weight of the crew, $W_{PAYLOAD}$ is the weight of payload, W_f is the weight of fuel, W_0 is the design take off gross weight, W_E is the weight of empty aircraft.

2.2.4. Empty-Weight Estimation:

Table 5. Average Crew Member Weights [14]

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 kit	10	NA

Table 5 shows the average weights of crew members in two categories: with and without bags (lb). Table 2 shows how to calculate the average weight of a crew member.

$$Wc = 240 * 2 = 480 \text{ lb}$$

The weight (Wc) includes 2 flight crew members and one attendant with one luggage for each.

$$Wc = 2 * (\text{Flight Crew Member}) + 1 * \text{Flight Attendant}$$

$$Wc = 2 * (240) + 220 = 700 \text{ lb}$$

Table 6. Average Crew Member Weights [14]

Standard Average Passenger Weight	Weight per Passenger (lb)
Summer weights	
Average adult passenger weight	190
Average adult male	200
Average adult female	179
Average child (2–13 years old)	82
Winter weights	
Average adult passenger	195
Average adult male	205
Average adult female	184
Average child (2–13 years old)	87

Table 6 shows the standard average passenger weights (lb). Look at table 4 to get the standard average passenger weight. It has been determined that the weight (Wpl) consists of 25 male and 25 female passengers. In addition, each passenger has one piece of luggage. Weight per person is taken as 70 kg + 15 kg baggage weight.

$$\mathrm{Wpl} = 50 * (\text{Average adult passenger})$$

$$\mathrm{Wpl} = 50 * (33 + 154) = 9.365\ \mathrm{lb}$$

2.2.5. Fuel-Weight Estimation:

For any mission segment i, the mission segment weight fraction can be expressed as $\frac{W_i}{W_{i-1}}$. Therefore, a simple mission profile is defined as

$$\mathrm{M_F} = \frac{\mathrm{W_6}}{\mathrm{W_0}} = \frac{\mathrm{W_6}}{\mathrm{W_5}} \frac{\mathrm{W_5}}{\mathrm{W_4}} \frac{\mathrm{W_4}}{\mathrm{W_3}} \frac{\mathrm{W_3}}{\mathrm{W_2}} \frac{\mathrm{W_2}}{\mathrm{W_1}} \frac{\mathrm{W_1}}{\mathrm{W_0}} \quad [3],$$

where $\mathrm{M_F}$ is the mission profile fraction rate.

Table 7. Weight fractions for the mission segments for turbofan aircraft [14]

Stage	Description	W_i/W_{i-1}
1	Engine start and warm-up	0.990
2	Taxi	0.990
3	Takeoff	0.995
4	Climb	0.980
5	Cruise to full range	$\exp[-RC_j/V(L/D)]$
6[a]	One hour additional flight at cruise conditions	$\exp[-C_j/(L/D)]$
6[b]	Ten percent nominal flight time additional	$\exp[-RC_j/10V(L/D)]$
7[a]	Descent to destination and refused landing	0.990
8[a]	Climb	0.980
9[a]	Diversion to alternate airport 200 nm distant	$\exp[-200C_j/V(L/D)$
9[b]	Diversion to alternate airport 200 nm distant plus 0.5 h hold at 15,000 ft altitude at alternate airport	$\exp[-200C_j/V(L/D)$ $+\exp[0.5C_j/V(L/D)]$
10	Descent	0.990
11	Landing	0.992

[a] *Flight diversion for domestic flights.*
[b] *Flight diversion for international flights.*

The following values are used in calculations:

- Takeoff: $\frac{W_1}{W_0} = 0.995$
- Climb: $\frac{W_2}{W_1} = 0.980$
- Descend: $\frac{W_4}{W_3} = 0.990$

- Landing: $\frac{W_6}{W_5} = 0.992$

The fuel weight fraction for the cruise segment mission fraction is given by

$$\frac{W_3}{W_2} = \exp\frac{(-RC_J)}{V\,(\frac{L}{D})}\ [4],$$

The values listed below are used in the subsequent calculations:

- Range: 3158.75 nmi (5850 km)
- Cruise Velocity: 421.12
- Thrust Specific Fuel Consumption: $\mathrm{C_J} = 2 * 0.51$ 1/hr (for two engines, 1.02 [lbm/(lbf*h)], imperial unit has been considered instead metric unit).
- $\left(\frac{\mathrm{L}}{\mathrm{D}}\right)_{\mathrm{max}}$: 13 (Estimated)
- $\left(\frac{\mathrm{L}}{\mathrm{D}}\right)_{\mathrm{cruise}} = 0.866\left(\frac{\mathrm{L}}{\mathrm{D}}\right)_{\mathrm{max}}:\ 11.258$

After plugging in all of the known values into the equation [4],

$$\frac{\mathrm{W_3}}{\mathrm{W_2}} = 0.5551$$

The loiter segment mission fraction can be calculated by

$$\frac{W_5}{W_4} = \exp\frac{\left(-EC_J\right)}{\frac{L}{D}}, [5]$$

- Endurance: 20 min
- Thrust Specific Fuel Consumption: $C_J = 2 * 0.51$ 1/hr (for two engines, 1.02 [lbm/(lbf*h)], imperial unit has been considered instead metric unit).
- $\left(\frac{L}{D}\right)_{max}$: 13 (Estimated)

After plugging in all of the known values into the equation [5],

$$\frac{\mathrm{W_5}}{\mathrm{W_4}} = 0.9742$$

Substituting all the known values in equation [6],

$$\mathrm{M_F} = \frac{\mathrm{W_6}}{\mathrm{W_0}} = \frac{\mathrm{W_6}}{\mathrm{W_5}}\frac{\mathrm{W_5}}{\mathrm{W_4}}\frac{\mathrm{W_4}}{\mathrm{W_3}}\frac{\mathrm{W_3}}{\mathrm{W_2}}\frac{\mathrm{W_2}}{\mathrm{W_1}}\frac{\mathrm{W_1}}{\mathrm{W_0}},$$

$$M_F = \frac{W_6}{W_0} = 0.5179$$

The fuel weight fraction is calculated using by

$$\frac{W_f}{W_0} = 1.06\ (1 - M_F)\ \ [6]$$

where M_F is the mission segments weight fraction. 1.06 is used for allowing for 6% reserve of trapped fuel.

$$\frac{W_f}{W_0} = \ 0.5110$$

Using the MATLAB program, the design takeoff gross weight (W_0) is calculated based on the estimates. The MATLAB code is used to solve Equation (1) by iteration (for MATLAB code, see Appendix A.2).

$$W_0 \ = 207248\ lb$$

Trade Studies:

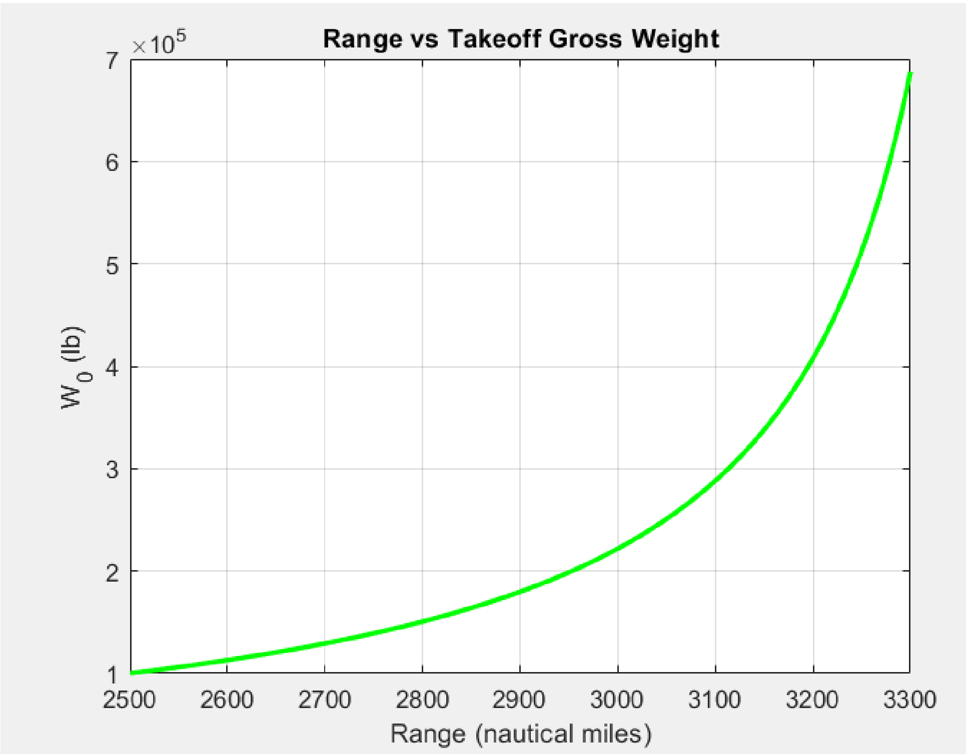

Figure 3. Range vs Wo

The figure 3 shows the range versus design gross weight relationship. It can be seen that this is an exponential growth. Range is 4630 km to 6111.6 km. (for MATLAB code, see Appendix A.3)

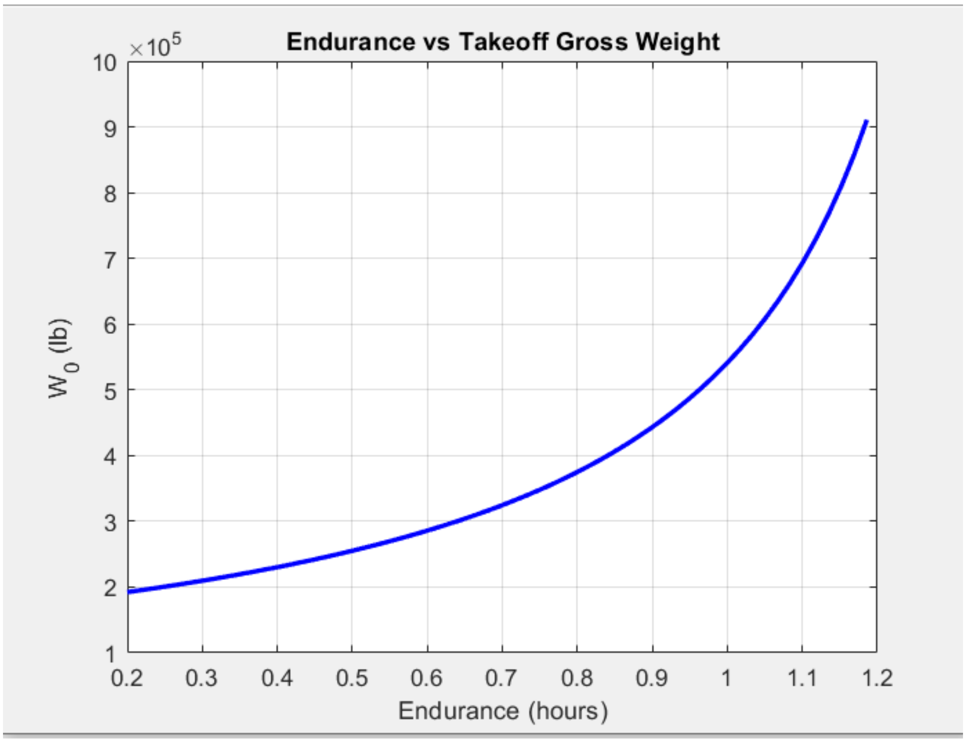

Figure 4. Endurance vs Takeoff Gross Weight

Number of passenger increased from 5 to 75 person as shown below in Figure 4. (for MATLAB code, see Appendix A.4).

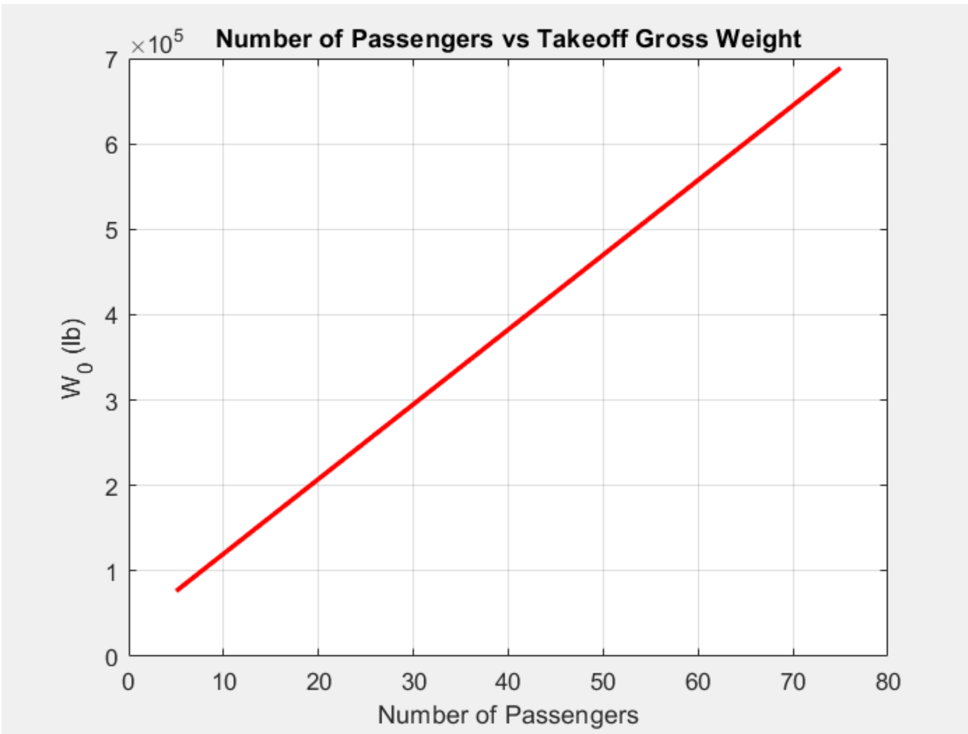

Figure 5. Wo (lb) vs Number of Passengers

The Figure 5 shows the linear relationship between payload (number of passengers) and design gross weight (lb) (for MATLAB code, see Appendix A.5).

In conclusion, looking at the graphs above, it is clear that loiter time and payload weight have a very close linear relationship with gross weight. The range vs. gross weight graph, on the other hand, clearly shows that as the range increases, the gross weight increases exponentially. As a result, we can conclude that range has the greatest influence on the design takeoff gross weight. This is the expected result because payload weight has a linear relationship with gross weight in the formula, and loiter time is very short compared to total flight time.

Chapter 3. Fuselage Layout

The purpose of this project chapter is to calculate the dimensions of the fuselage and all cabin components.

Table 8. Cabin Dimensions

Type of Class	Business Class	Economy Class
Number of seats	10	40
Seat width	0.6096 m	0.4572 m
Aisle width	1.3188 m	0.700 m
Seat pitch	1 m	0.85 m
Number of rows	5	10
Number of seats abreast	2	4
Number of seats on each side	1-1	2-2
Seat length	1.8 m	1.8 m
Type of emergency exit	Type-C	Type-C
Number of emergency exit	2	2

Fuselage's cabin width can be calculated as

$$W_{fus} \approx c_1 + c_2 w_s N_a \text{ [7]},$$

where W_{fus} is fuselage width, w_s is the seat width, N_a is the number of seats abreast (in a row), and c_1, c_2 are constants.

Fuselage's cabin length can be calculated as

$$L_C \approx c_3 (PN_r)^{1+\varepsilon} \text{ [8]},$$

where P is the seat pitch, N_r is the number of rows of seats, ε is a constant that accounts for any non-linearity (= 0.052) and c_3 is constant.

Number of passengers assuming number of seats in a row is constant and can be used as

$$N_p \approx N_a N_r \ [9].$$

The internal width (diameter) can be calculated as

$$d_f \approx w_s N_a + w_i N_i \ [10],$$

where w_s is the seat width, N_a is the number of seats abreast (in a row), w_i is the aisle width and N_i is the number of aisle. Table in the above is used for determine fuselage's cabin width and cabin length.

Table 9. Average Fuselage Data (Commercial Aircraft)

N_a	c_1	c_2	c_3	$c_{3,pax}$	P (ft)	w_s (ft)	h (ft)	N_r
1	0.71	1.09	1.32	1.08	2.5	1.43	4	1–4
2	2.49	1.24	1.32	1.08	2.5	1.43	5	6–10
3	2.49	1.24	1.32	1.08	2.5	1.43	5.5	10–15
4	2.49	1.24	1.32	1.08	2.5	1.43	6	16–18
5	2.49	1.24	1.08	1.08	2.7	1.43	6	19–31
6	2.49	1.24	1.08	1.08	2.7	1.43	6	31–37
7	0	1.3	1.08	1.08	3	1.5	6	33–37
8	0	1.3	1.08	1.08	3	1.5	6	37–45
9	0	1.3	1.08	1.08	3	1.5	6	45
10	0	1.3	1.08	1.08	3	1.5	6	

Using Table 2 , c_1, c_2, c_3 determined as,

Table 10. Constants for the Cabin Dimensions

Type of Class	Business Class	Economy Class
c_1	2.49	2.49
c_2	1.24	1.24
c_3	1.32	1.32
ε	0.052	0.052

Table 11. Yielded values for the Fuselage

Cabin length	17.73775 m
Cabin length for business class	5.21223 m
Cabin length for economy class	8.38164 m
Cabin diameter	2.9788 m
Nose length	5.45 m
Tail length	3 m
Emergency exit width	0.6096 m

3.1. Cargo Type

ULD-2 cargo-type is used.

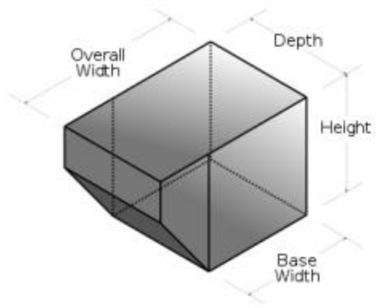

Figure 6. Layout of ULD

Figure 6 depicts the ULD's overall width, base width, depth, and height. When designing the fuselage cargo compartment, the layout of the ULD is taken into account.

Table 12. Cargo Unit Dimensions

ULD-2 Cargo Type	Base width	Overall width	Height	Depth	Volume
Dimensions	1.194 m	1.56 m	1.62 m	1.56 m	3.4 m^3

3.2. Emergency Exits

For emergency exits, according to maximum number of passenger seats permitted for each exit type table shown below, since there would be 50 passengers, Type-C is used which is a floor-level exit with a rectangular opening of not less than 30 in wide by 48 in high, with corner radii not greater than 10 in.

Table 13. Number of passenger seats permitted for each exit

Exit Type	Number of Seats
A	110
B	75
C	55
I	45
II	40
III	35
IV	9

3.3. Fuselage and Cabin Layout

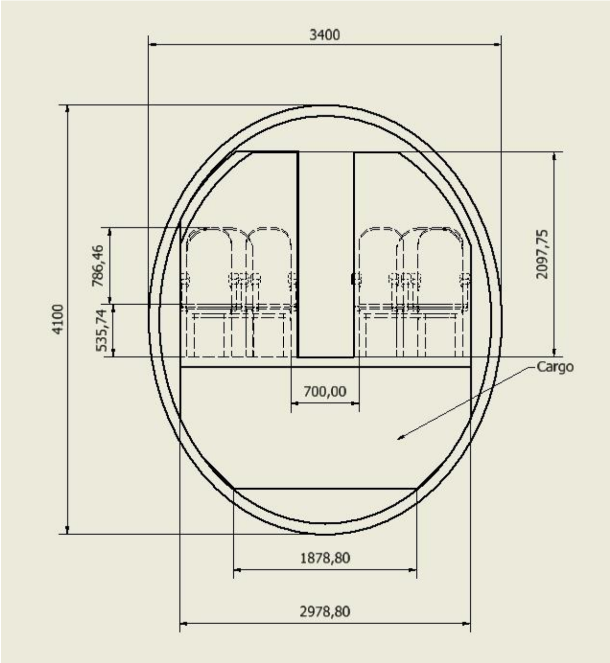

Figure 7. The view of fuselage cross section for business class

The Figure 7 shows the dimensions of the fuselage cross section for business class. All dimensions are in mm.

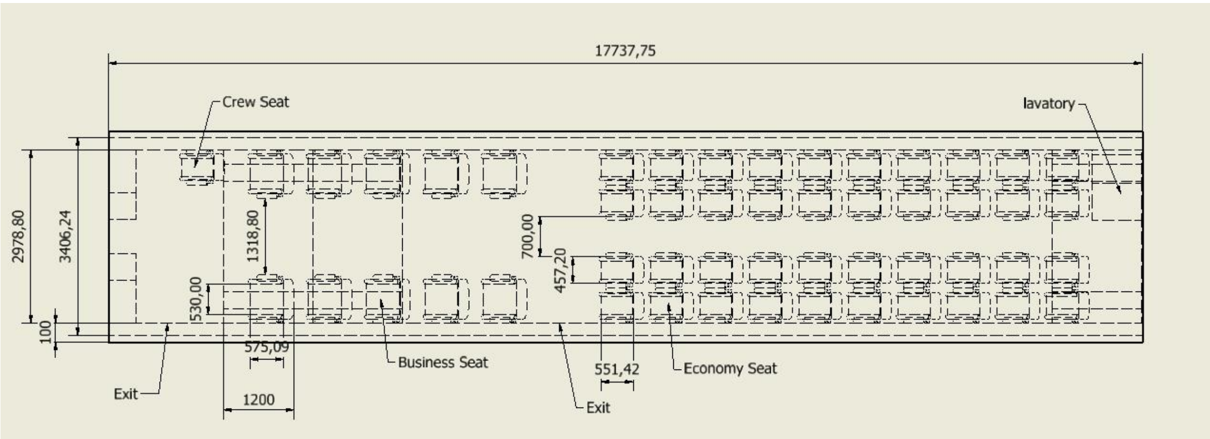

Figure 8. The top view of fuselage

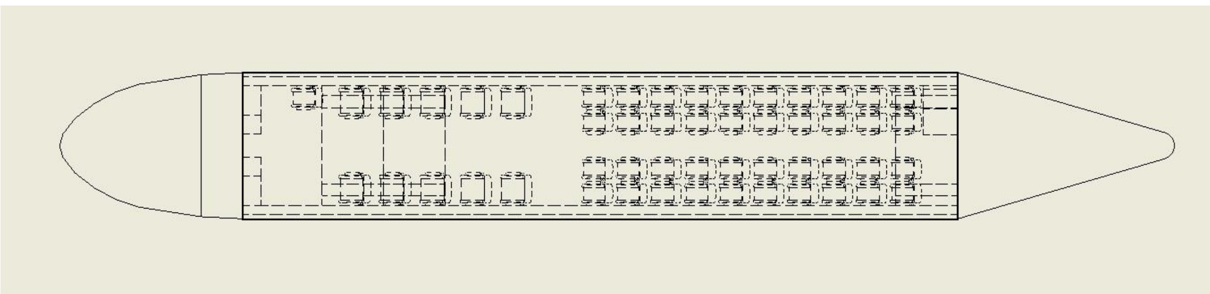

Figure 9. Top view of the fuselage with nose and tail

Chapter 4. Airfoil and Initial Wing Geometry

4.1. Calculation the Design Lift Coefficient

The airplane's design lift coefficient for cruise flight can be calculated by using

$$C_L = \frac{2W}{\rho V_{CR}^2 S} g \ [11],$$

where C_L is the lift coefficient, ρ is the air density, V_{CR} is the cruise velocity, g is the gravity force and $\frac{W}{S}$ is wing loading.

- $\rho = 0.365 \ kg/m^2$
- $V_{CR} = 780 \ km/h \ = 216.67 \ m/s$ from assignment 2.
- $\frac{W}{S} = \frac{24{,}041 \ kg}{48.35 \ m^2} = \ 497.23 \ kg/m^2$ from competitor study [Bombardier CRJ200].
- $g \ = \ 9.81 \ m/s^2$

Therefore, C_L is found as 0.57 by plugging these values into equation [11].

4.2. Airfoil Selection

The properties of various airfoil series are shown in Table 1. In terms of stall angle and moment coefficient, the NACA-6 series of swept-wing subsonic aircraft can be a good candidate.

Table 14. Properties of candidate airfoils [2]

Airfoils	$C_{L,max}$	$C_{L,\alpha}$	$C_{m,0}$	α_{stall}
NACA 63A010	1	0.10983	0	10.5°
NACA 64-215	1.3288	0.10	-0.04	16.750°
Falcon (falcon-il)	1.3584	1.4245	-0.0062	15°
NACA 63-015A	1.2465	0.11	0	16.250°

When obtaining the data in Table 1, an airfoil database [2] is used. Although typical Reynolds numbers (Re) for subsonic jet aircraft range from $3x10^6$ to $9x10^6$, data for this range is not available in the database. As a result, during the research, $Re \ = 1x10^6$ is used. $C_{L,\alpha}$ values are calculated by using C_L vs. α graphs as shown below.

<u>NACA 63A010:</u>

Figure 10. Cl vs Alpha, Cm vs Alpha, Cl/Cd Graphs for NACA 63A010 [2]

Important data can be obtained from these graphs. $C_{L,max}$= 1, α_{stall}= 10.5°, $C_{m,0}$= 0 (symmetric airfoil), and $C_{L,\alpha}$ = 0.10983 are the values (slope of the first graph).

<u>NACA 64-215:</u>

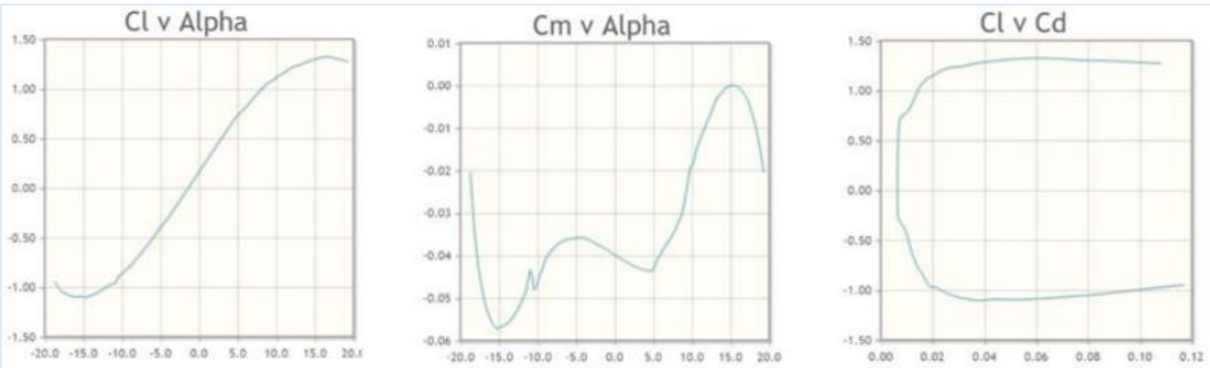

Figure 11. Cl vs Alpha, Cm vs Alpha, Cl/Cd Graphs for NACA 64-215 [2]

Figure 11 shows the aerodynamic parameter relationships of NACA 64-215. $C_{L,max}$= 1.3584, α_{stall}= 15°, $C_{m,0}$= −0.04, and $C_{L,\alpha}$ = 0.10 are the values (slope of the first graph).

<u>Falcon (falcon-il):</u>

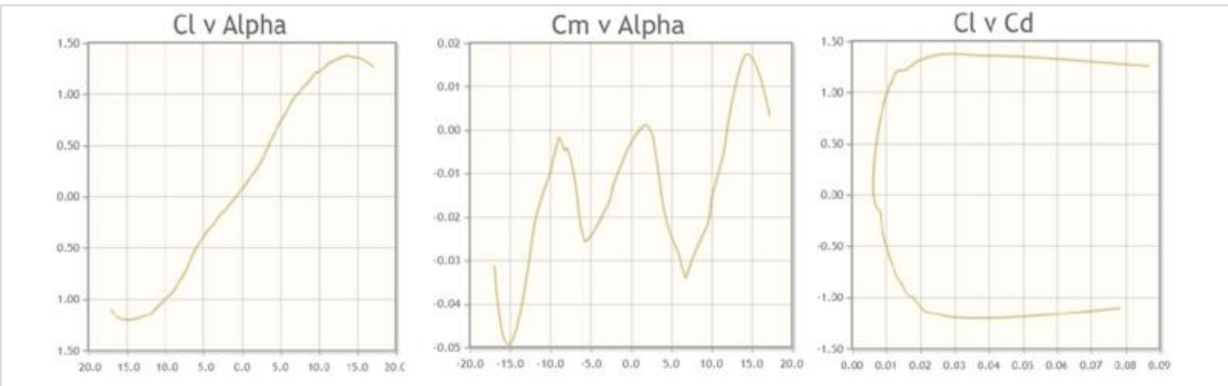

Figure 12. Cl vs Alpha, Cm vs Alpha, Cl/Cd Graphs for Falcon [2]

Figure 2 shows the aerodynamic parameter relationships of NACA 64-215. $C_{L,max}$= 1.3288, α_{stall}= 16.75°, $C_{m,0}$= −0.0062, and $C_{L,\alpha}$ = 1.4245 are the values (slope of the first graph).

NACA 63-015A:

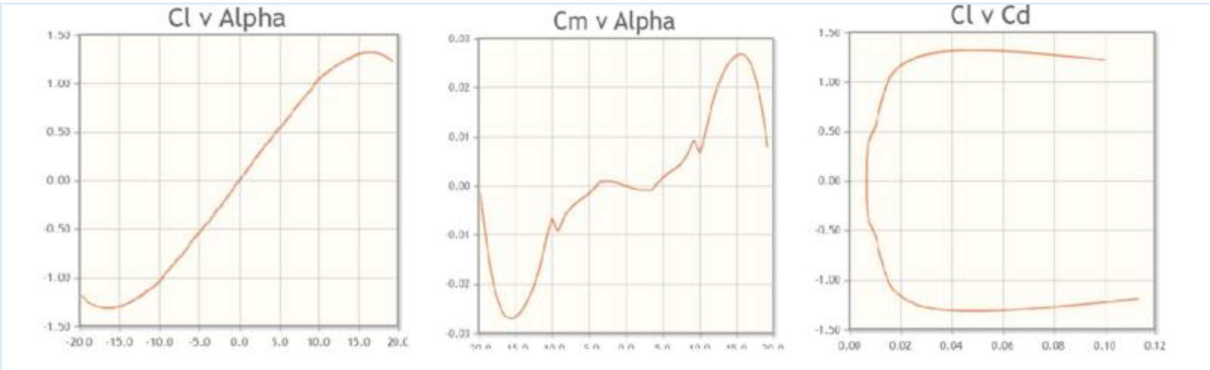

Figure 13. Cl vs Alpha, Cm vs Alpha, Cl/Cd Graphs for NACA 63-015A [2]

It can be seen from the Figure 4, these three graphs shows the relationship of aerodynamic parameters of NACA 63-015A airfoil. Significant values which are gathered from them and used in this project are$C_{L,max}$ = 1.2465, α_{stall} = 16.25°,$C_{m,0}$ = 0 (symmetric airfoil) and $C_{L,\alpha}$ = 0.11 (slope of the first graph).

4.3. Initial Wing Geometry

The aspect ratio of jet transports has to be between 7.5 to 10. In addition to that, our competitive study reveals all of the aircraft that we analyzed have an aspect ratio higher than 9. Moreover, our base aircraft, which are Bombardier CRJ-200 & Mitsubishi MRJ90 use 9,3 & 9,5 aspect ratio, respectively. Therefore, our aspect ratio has to be 9,3.

When it comes to sweep angle, we also used historical data, especially data of CRJ200. As our design project very close CRJ200, we use some parameters of CRJ200. More specifically, we used quarter-chord line sweep angle, which is $24{,}5°$. To define our leading-edge sweep, we must use the equation given below.

$$\tan \Lambda_{LE} = \tan \Lambda_{C/4} + \frac{1-\boldsymbol{\lambda}}{AR(1+\boldsymbol{\lambda})} \quad [12]$$

Despite this, we have yet to define the taper ratio. To do so, we examine the historical trend as well as our competitive analysis. We discovered that as the size of the body increases, the taper ratio tends to decrease. It should be noted, however, that the taper ratio of the subsonic wing should be between 0.2 and 0.3. As a result, we chose a taper ratio of 0.265.

$$\Lambda_{C/4} = 24.5°$$

$$\lambda = 0.265$$

As a result, we obtain $\Lambda_{LE} = 27.49°$. However, as shown in the graph of NACA-TN-1093, this design slightly increases the pitch-up risk. To compensate for this issue, we intend to reduce pitch-up risk to a manageable level by selecting a suitable horizontal wing. Because there is no aerodynamic twist, the airfoil has the same shape from root to tip.

For the initial study, a geometrical twist is a washout mechanism with an angle of 3° is considered to avoid having a stall at the tip for better control at the high angle of angle. For this design, the angle of incidence between the fuselage and the wing root is set to $2°$. Because the aircraft will have a lower low wing position, a wing dihedral of 3° should be enough to provide extra safety space for the engines.

4.3.1. High Lift Devices

Leading-edge slats:

These are used to increase the maximum lift coefficient of the profile as well as the stall angle. Powered slats are primarily used during landing and maneuvering. They are also used for takeoff because they produce less lift with lower drag.

Krueger flaps:

Krueger flaps are high-lift devices that attach to the entire or a portion of the leading edge of certain types of aircraft wings. The aerodynamic effect of the Krueger flap is comparable to that of the slats. The method of provision, however, differs. The Krueger flap is hinged at the leading edge and attached to the underside of the wing. The actuator moves the flap from the underside of the wing to the front, increasing the wing's curvature. This will result in more lift. As a result, the area of the wing increases, and thus the lift, particularly at low speeds.

Trailing edge flaps:

In the form of slotted flaps, they are used to reduce the stall speed of aircraft wings. This is due to the upward flow of air beneath the wings, which strengthens the boundary layer against separation with the help of the vortex (Turbulent flow). As a result, they reduce flow separation.

Fowler flaps:

They consist of a split flap that slides back slightly before folding. This increases the tendons (and thus the surface of the wing), followed by the curvature. This can improve both takeoff (partial extension for optimal lift) and landing performance (full extension for optimal lift and drag). This type of flap, or one of its variants, can be found on the majority of large aircraft.

Therefore, after weighing the benefits and drawbacks of each high-lift device, we decided on the Fowler Flap and Krueger Flap.

Table 15. the parameters chosen for our design aircraft

AR	**S**	**λ**	**Λ_{LE}**	**$\Lambda_{C/4}$**	**Washout twist angle**	**Fuselage-wing root angle**	**Dihedral**
9.3	$90.43\ m^2$	0.265	27.49°	24.5°	3°	2°	3°

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

The goal of this chapter is to calculate the thrust-to-weight ratio and wing loading, as well as other parameters like takeoff and landing distances and maximum ceiling. First, a thrust to weight ratio value is estimated using historical trends and competitor research. Following that, wing loading for the designed aircraft is calculated using several parameters such as landing distance, stall speed, and takeoff conditions. At the end of this chapter, a value for the wing loading (W/S) and thrust to weight ratio (T/W) is chosen for the business jet design.

5.1. Establish necessary requirements

5.1.1 Stall Speed

Using FAR/EC design specifications and noting that our aircraft falls into the category of transport jet aircraft, the takeoff speed, $V_{takeoff}$, and stall speed, V_{stall}, are related by

$$V_{takeoff} = 1.08\ V_{stall} \quad [13],$$

Checking the historical trends, which states the typical takeoff speed values varies from 240 km/h to 285 km/h, and referring to the competitor study $V_{takeoff}$ is established as 260 km/h for our design.

$$V_{takeoff} = 260\ km/h$$

$$V_{stall} = \frac{V_{takeoff}}{1.08} = \frac{260\ km/h}{1.08} = 240.74\ km/h = 66.87\ m/s$$

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 14. According to Figure 1, for aircraft with W_{TO} > 100,000 lb (45,360 kg) , 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 in the figure as the correlation $x_l = 816W_{TO}^{0.15}$.

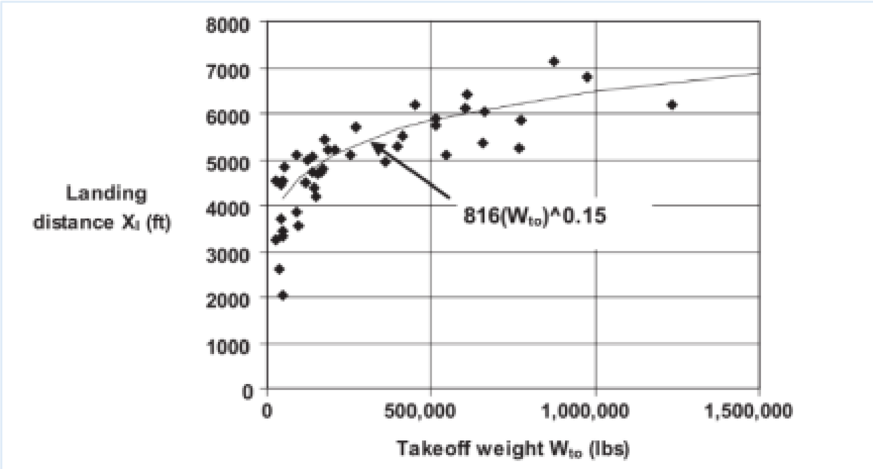

Figure 14. Nominal landing distances for 50 commercial airliners [4]

The landing distance can be calculated using by

$$x_{land} = 816 * W_{TO}^{0.15} \text{ [14]},$$

where x_{land} is the landing distance and W_{TO} is the takeoff weight (207,248 lb from previous calculations).

$$x_{land} = 816 * (207{,}248)^{0.15} = 5118.74 \text{ ft} = 1560.2 \text{ m}.$$

5.1.3. Takeoff Distance

Takeoff Parameter for jet aircraft is calculated using by

$$\text{TOP} = \frac{\left(\frac{W}{S}\right)_{\text{Takeoff}}}{\sigma C_{L,TO}\left(\frac{T}{W}\right)} \quad \text{[15]},$$

where $\left(\frac{W}{S}\right)_{\text{Takeoff}}$ 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.

- $\left(\frac{W}{S}\right)_{\text{Takeoff}} = 497.23 \text{ kg/m}^2 = 101.84 \text{ lb/ft}^2$ from assignment 4.
- 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 1.3288 from assignment 4. So, $C_{L,TO} = \frac{1.3288}{1.21} = 1.0981$.

- $\left(\frac{T}{W}\right) = 0.35\ \mathrm{m/s^2} = 1.15\ \mathrm{ft/s^2}$ from previous assignment.

$$\mathrm{TOP} = \frac{101.84}{(0.2978)(1.0981)(1.15)} = 270.8$$

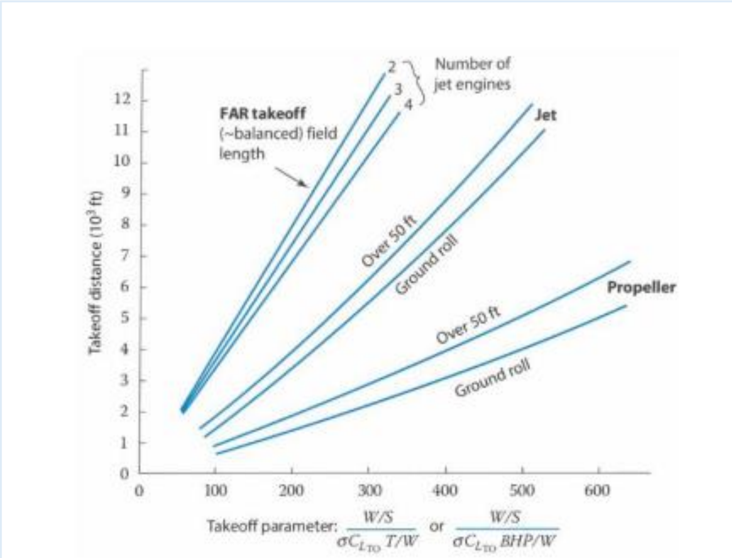

Figure 15. Takeoff Distance vs. TOP [14]

As can be seen from Figure 15, using TOP, the landing distance is found to be approximately 8000 ft or 2438.4 m.

5.1.4. Maximum Ceiling

The typical ceiling values for jetliners and business jets are around 13000 m and 16000 m, respectively, according to FAR/EC regulations and typical historical trends in standard weather conditions. Taking these factors into account, as well as our competitor research, we have determined that the maximum ceiling value for our design is around 13000 m (42000 ft).

5.2. Choosing thrust-to-weight ratio (T/W) using historical trend and historical data

Table 16. Typical Values of Thrust-to-Weight Ratio [14]

Aircraft Type	Typical Installed T/W
Jet trainer	0.4
Jet fighter (dogfighter)	0.9
Jet fighter (other)	0.6
Military cargo/bomber	0.25
Jet transport (higher value for fewer engines)	0.25–0.4

Taking into account the jet transport category in Table 16 and the competitor study in Chapter 1, the value of T/W is set at 0.35.

5.3. Calculating the wing loading, (W/S) according to necessary requirements, and choosing a wing loading value from historical trends

5.3.1. Historical Trends

Table 17. Historical Trends for Wing Loading [14]

Typical Takeoff W/S		
Historical Trends	lb/ft²	{kg/m²}
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}

Taking into consideration the jet transport/bomber category in Table 2, the W/S value is chosen to be 586 $\mathrm{kg/m^2}$.

5.3.2. Stall speed in landing configuration

$(W/S)_{stall}$ is given by

$$(W/S)_{stall} = \frac{1}{2}\rho V_{stall}^2 C_{L,max} \text{ [16]},$$

where ρ is the density at sea level, V_{stall} is the stall velocity, and $C_{L,max}$ is the maximum lift coefficient.

- $\rho = 1.225\ \mathrm{kg/m^3}$ @ sea level
- $V_{stall} = 66.87\ \mathrm{m/s}$ from part 1.1
- $C_{L,max} = 2.4$ from chosen airfoil which is NACA 64-15 by considering the flap and slat configuration.

Therefore,

$$(W/S)_{stall} = \frac{1}{2}(1.225)(66.87)^2(2.4) = 670.1\ \mathrm{kg/m^2}$$

5.3.3. Landing Distance

Landing velocity is calculated using $V_{E,land} = kV_{E,stall}$ where $V_{E,stall}$ is equivalent airspeed at stall and k=1.2-1.3. Taking k=1.2, landing velocity can be found as,

$$V_{E,land} = 1.2\ V_{E,stall} = 1.2\ (66.87) = 80.244\ \mathrm{m/s}.$$

The takeoff lift coefficient is calculated using by

$$C_{L,takeoff} = \frac{C_{L,max}}{1.21} = \frac{2.4}{1.21} = 1.983.$$

The flaps are only deployed to about half their maximum angle during takeoff. As a result, the maximum lift coefficient at takeoff is approximately 80% of the landing value. Therefore,

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

Wing loading at landing is calculated as

$$(\mathrm{W/S})_{\mathrm{landing}} = \frac{1}{2}\rho \mathrm{V}_{\mathrm{landing}}^{2}\mathrm{C_{L,landing}} \text{ [17]},$$

$$(\mathrm{W/S})_{\mathrm{landing}} = \frac{1}{2}(1.225)(80.24)^2(2.48) = 996.94 \text{ kg/m}^2.$$

$(W/S)_{cruise}$ is found by applying

$$(W/S)_{cruise} = \bar{q}\sqrt{\pi AeC_{D0}/3} \text{ [17].}$$

where $\bar{q}$ is dynamic pressure at 11000 m, A is the aspect ratio, e is the Oswald span efficiency which is is approximately 0.6 to 0.8 for a fighter and 0.8 for other aircraft, and C_{D0} is the zero-lift drag coefficient. Also, C_{D0} is 0.015 for a jet aircraft. Using $V_{cruise} = 780\ km/h\ (216.67\ m/s)$ from chapter 2.

$$\bar{q} = \frac{\frac{1}{2}\,(0.4135)(216.67)}{9.81} = 990.32\ kg/m^2,$$

$$(W/S)_{cruise} = 990.32\sqrt{\pi(9.3)(0.8)(0.015)/3} = 338.59\ kg/m^2.$$

$(W/S)_{sustained}$ is found by applying

$$(W/S)_{sustained} = \frac{\bar{q}}{n}\sqrt{\pi AeC_{D0}/3} \text{ [18],}$$

where n is the load factor $= \left(\frac{T}{W}\right)\left(\frac{L}{D}\right) = 0.35 * 14 = 4.9$ that $(L/D)_{max}$ is estimated 14. Because the sustained-turn load factor is optimized by maximizing T/W and L/D, maximum L/D is chosen.

$$(W/S)_{sustained} = \frac{990.32}{4.9}\sqrt{\pi(9.3)(0.8)(0.015)} = 119.67\ kg/m^2$$

Table 18. Summary of the results for W/S value in different flight segments

	Historical Trends	**Stall Speed**	**Landing Distance**	**Cruise Flight**	**Sustained Turn**
$(\boldsymbol{W/S})$ $[\boldsymbol{kg/m^2}]$	586	670.1	996.94	338.59	119.67

Considering the fact that from the calculated wing loadings, the lowest value should be selected to ensure that the wing is large enough for all flight conditions [ ], the W/S value is chosen to be 119.67 kg/m^2.

5.4. Converting all the wing loadings obtained in Part 3 to the takeoff condition

5.4.1. Historical Trends

Because historical trends are based on takeoff conditions (5.4), it is at 586 kg/m^2.

5.4.2. W/S Value at Stalling Speed and Landing Distance

To convert $(W/S)_{landing}$ to $(W/S)_{takeoff}$ we must use equation given below

$$\left(\frac{W_0}{S}\right) = (\frac{W}{S})_{takeoff} = \left(\frac{W}{S}\right)_{land} \left(\frac{W_0}{W_{landing}}\right) \ \& \ \left(\frac{W}{S}\right)_{takeoff} = \left(\frac{W}{S}\right)_{stall} \left(\frac{W_0}{W_{landing}}\right) \text{ [19].}$$

Here, $(\frac{W_0}{W_{landing}})$ equal to inverse of entire mission fraction, M_f, and we found M_f =0.5179 at second assignment. As calculations made, result yielded to 1,293.88 kg/m^2 for stalling & 1,924.96 kg/m^2 for landing.

5.4.3. W/S Value at Cruise Flight

To convert cruise wing load to takeoff condition, we must use similar equation given at stalling and landing part.

$$(\frac{W}{S})_{takeoff} = \left(\frac{W}{S}\right)_{cruise} \left(\frac{W_0}{W_3}\right) \text{ [20].}$$

The only problem here is $(\frac{W_0}{W_3})$ unknown, yet from the information that we get at assignment 2, we can find $(\frac{W_0}{W_3})$ by taking inverse of result.

$$\frac{W_3}{W_0} = \frac{W_1}{W_0}\frac{W_2}{W_1}\frac{W_3}{W_2} = (0.995)(.980)0.5551)$$

As a result, we found it $\frac{W_3}{W_0}$ is equal to 0.541, which is 1.848 for $\frac{W_0}{W_3}$, and when we use our new result with $\left(\frac{W}{S}\right)_{cruise}$, gives us $\left(\frac{W}{S}\right)_{cruise} = 625.54\ kg/m^2$, taken at takeoff condition.

5.4.4. W/S Value at Sustained Turn

The sustained turn can be obtained using the same formula as the cruise. The only difference is that (W_5/W_0) should be used because of loiter time. From the assignment 2, (W_5/W_0) can be found as 0.522. And W/S value at sustained turn can be determined as 216.73 kg/m^2.

Table 19. Summary of the results for W/S values converted into take-off condition

	Historical Trends	Stall Speed	Landing Distance	Cruise Flight	Sustained Turn
(W/S) $[kg/m^2]$	586	1293.88	1924.96	625.54	216.73

Considering the fact that from the calculated wing loadings, the lowest value should be selected to ensure that the wing is large enough for all flight conditions [5], the W/S value is chosen to be 216.73 $\mathrm{kg/m^2}$.

5.5. Checking the suitability of T/W

Firstly, take-off distance requirement should be checked, and T/W can be recalculated by using

$$(\mathrm{W/S})_{\mathrm{takeoff}} = (\mathrm{TOP})\sigma \mathrm{C_{L,TO}} \left(\frac{\mathrm{T}}{\mathrm{W}}\right) \quad [21].$$

Where TOP is take-off parameter which is 270.8 in british unit from part 1.3, σ is the ratio of the local density at 11000 meters to sea level density which is 0.2987, the take-off lift coefficient is 1.98 from part 3.3, and the wing loading can be chosen as 229.25 $\mathrm{kg/m^2}$[46.95 $\mathrm{lb/ft^2}$]. Therefore, T/W can be calculated as 0.278, which is less than 0.35. It means that aircraft design can handle the sustained maneuver.

The suitability T/W can be also checked using by

$$\frac{W}{S} = \frac{\left(\left[\left(\frac{T}{W}\right) - G\right] \pm \sqrt{\left[\left(\frac{T}{W}\right) - G\right]^2 - \frac{4C_{D0}}{\pi Ae}}\right)}{\frac{2}{q\pi Ae}} \quad [22]$$

Where G stand for climbing gradient and it is ratio of climb rate to forward velocity. It should be noted that climb rate for service ceiling equal 100 ft/min≈1.667 ft/s and forward velocity is 780 km/h=216.667 m/s≈710,849 ft/s. Therefore, climbing gradient,G, equal to 0.00235≈0.0023. A stand for Aspect Ratio and equal to 9.3, e stand for Oswald span efficiency and equal to 0.8, $\mathrm{C_{D_0}}$ stand for parasite drag and equal to 0.015, q stands for dynamic pressure at 11,000 meters ≈ 36089 ft, and it is equal to 5,811.02 $\mathrm{lb/fts^2}$

Chapter 6. Refined Weight Estimation

Redefined take-off gross weight estimation is an important stage for the design in order to achieve a more accurate value for our takeoff gross weight W_0 while taking into account cruise conditions, which are the most important parts of the mission profile. Improved statistical equations are used to estimate refined weight. These equations can also be used to calculate the empty weight fraction. Fuel weight is estimated for each segment, and all values are added together to get the total fuel weight.

6.1. Defining Mission Profile

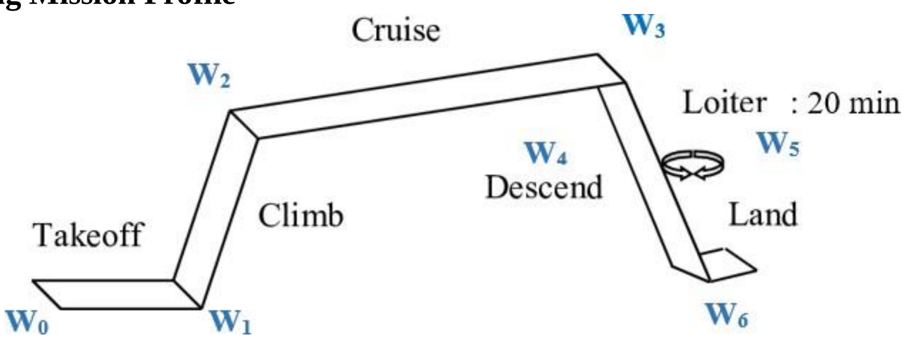

Figure 16. Mission Profile

As shown in Figure 16, the mission profile is divided into six segments: takeoff (0-1), climb (1-2), cruise (2-3), descend (3-4), loiter (4-5), and landing (5-6).

6.2. Determining the Crew and Fixed Payload Weight

Takeoff gross weight can be calculated using by

$$W_0 = W_{crew} + W_{fixed\ payload} + W_{dropped\ payload} + W_{fuel} + W_{empty}$$

where W_{crew} is the crew weight, $W_{fixed\ payload}$ is the fixed payload weight, $W_{dropped\ payload}$ is the dropped payload weight, W_{fuel} is the fuel weight and W_{empty} is the empty weight. Although the empty weight is calculated using an empty-weight fraction, the fuel is determined directly. Therefore, from assignment 2,

$$W_0 = W_{crew} + W_{fixed\ payload} + W_{dropped\ payload} + W_{fuel} + \left(\frac{W_E}{W_0}\right) W_0 \quad [23]$$

There is no payload dropped during flight for our business jet design. So, $W_{dropped\ payload}$ can be considered as zero. Previous chapters are used to make an educated guess about the takeoff weight. It is also used to calculate the takeoff weight.

In the following calculations, the following values are used:

- $W_{Crew} = 700$ lb includes two flight crew members and one flight attendant with one baggage for each.
- $W_{Payload} = 9{,}365$ lb includes 50 passengers with each passengers has one baggage.

Empty-Weight Fraction (W_E/W_0) can be determined using a statistical equation shown as

$$\frac{W_E}{W_0} = \left(a + bW_0^{C1} A^{C2} \left(\frac{T}{W_0}\right)^{C3} \left(\frac{W_0}{S}\right)^{C4} M_{max}^{C5}\right) K_{vs} \quad [24]$$

where are a, b, C1, C2, C3, C4, C5 are constants, and taken from table as shown below.

Table 20. Constants for empty weight fraction equation [14]

$W_e/W_0 = [a + b\, W_0^{C1} A^{C2} (T/W_0)^{C3} (W_0/S)^{C4} M_{max}^{C5}] K_{vs}$							
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 and 1.00 if fixed sweep.

W_0 is the takeoff gross weight which is 207248 lb from chapter 2, A is the aspect ratio which is 9.3, $\frac{T}{W_0}$ is the thrust to weight ratio which is $0.35\ m/s^2$ in metric, and $1.15\ ft/s^2$ in British unit, $\frac{W_0}{S}$ is the wing loading for takeoff condition which is taken from assignment 5 cruise flight as $625.54\ kg/m^2$ in metric, $128.12\ lb/ft^2$ in British unit. M_{max} is the maximum Mach number which is 0.8 taken from competitor study, and K_{vs} is the variable for sweep constant which is 1 because of fixed sweep. Therefore, from equation [24],

$$\frac{W_E}{W_0} = 0.5367$$

For each segment of the mission profile, the weight of consumed fuel can be calculated by using

$$W_{f_i} = \left(1 - \frac{W_i}{W_{i-1}}\right) W_{i-1} \quad [25]$$

6.2.1. Takeoff and Taxi

By modifying Equation [3], the fuel consumed during the takeoff and taxi segment is calculated using by

$$W_{f_1} = \left(1 - \frac{W_1}{W_0}\right) W_0$$

where W_0 is the takeoff gross weight. W_0 is calculated by using MATLAB program (See Appx. 1). $\frac{W_1}{W_0}$ is taken 0.99, and the gross weight at the takeoff and taxi conditions is determined using by $W_1 = W_0 - W_{f_1}$.

6.2.2.Climb

By modifying Equation [25], the burned fuel weight in the climb segment is calculated using by

$$W_{f_2} = \left(1 - \frac{W_2}{W_1}\right) W_1$$

The subsonic climb weight fractions are calculated using by

$$\frac{W_i}{W_{i-1}} = 1.0065 - 0.0325M \ [26]$$

where $\frac{W_i}{W_{i-1}}$ is the weight fraction, and M is the Mach number. The climb weight fraction is obtained for the initial estimation by modifying [26]

$$\frac{W_2}{W_1} = 1.0065 - 0.0325M$$

where Mach number is taken 0.8 for the subsonic condition. Therefore, $\frac{W_2}{W_1}$ is calculated as 0.9830, and the gross weight at climb condition is obtained by $W_2 = W_1 - W_{f_2}$. The previous weight fraction is multiplied by the take-off and taxi weight fractions to get the climb weight fraction for the following iterations.

6.2.3. Cruise

By modifying Equation [25], the burned fuel weight in the cruise segment is calculated using by

$$W_{f_3} = \left(1 - \frac{W_3}{W_2}\right) W_2 \,,$$

The first iteration value of the fuel weight fraction for the mission profile's cruise segment is given by

$$\frac{W_3}{W_2} = \exp \frac{(-RC_J)}{V\left(\frac{L}{D}\right)}$$

The values listed below are used in the subsequent calculations:

- Range : 3158.75 nmi (5850 km)
- Cruise Velocity : 216.67 m/s
- Thrust Specific Fuel Consumption, C_J = 2x 0.51 1/hr ((for two engines, 0.545 [lbm/(lbf*h)], imperial unit has been considered instead metric unit).
- $\left(\frac{L}{D}\right)_{max} = 17.095$ (estimated)
- $\left(\frac{L}{D}\right)_{cruise} = 0.866 \left(\frac{L}{D}\right)_{max} = 19.74$

After entering all of the values, the fuel weight fraction for the cruise segment of the mission profile is calculated as shown below.

$$\frac{W_3}{W_2} = 0.6632$$

The total gross weight at the cruise segment is found by modifying Equation [26] as $W_3 = W_2 - W_{f_3}$.

6.4. Descent

By modifying Equation [25], the burned fuel weight in the descent segment is calculated using by

$$W_{f_4} = \left(1 - \frac{W_4}{W_3}\right) W_3\,,$$

The first descent weight fraction estimation is set to $\frac{W_4}{W_3}$ = 0.995 from historical trends, and the gross weight at descent condition is obtained by $W_4 = W_3 - W_{f_4}$.

6.5. Loiter

By modifying Equation [25], the burned fuel weight in the descent segment is calculated using by

$$W_{f_5} = \left(1 - \frac{W_5}{W_4}\right) W_5$$

The first iteration value of the fuel weight fraction for the mission profile's loiter segment is given by

$$\frac{W_5}{W_4} = \exp\frac{(-EC_J)}{\left(\frac{L}{D}\right)}$$

The values listed below are used in the subsequent calculations:

- Endurance: 0.5 h (30 min loiter time)
- Thrust Specific Fuel Consumption, C_J = 2x 0.51 1/hr ((for two engines, 0.545 [lbm / (lbf *h)], imperial unit has been considered instead metric unit).
- $\left(\frac{L}{D}\right)_{max} = 17.095$ (estimated)
- $\left(\frac{L}{D}\right)_{loiter} = 17.095$

After entering all of the values, the fuel weight fraction for the loiter segment of the mission profile is calculated as shown below.

$$\frac{W_5}{W_4} = 0.9843$$

The total gross weight at the loiter segment is found by modifying Equation [4] as $W_5 = W_4 - W_{f_5}$.

6.6 Landing and Taxi Back

By modifying Equation [25], the burned fuel weight in the landing and taxi back segment is calculated using by

$$W_{f_6} = \left(1 - \frac{W_6}{W_5}\right) W_5$$

The first landing and taxi back weight fraction estimation is set to $\frac{W_6}{W_5}$ = 0.9970 from historical trends, and the gross weight at landing and taxi back condition is obtained by $W_6 = W_5 - W_{f_6}$.

The first estimation of the total aircraft fuel (+6% for reserve and trapped fuel) is calculated using by

$$W_f = 1.06 \sum_{i=1}^{n} W_{f_i}$$

where W_{f_i} is the fuel weight for each part of the mission profile segment. Total W_f can be written as

$$W_f = 1.06 \left(W_{f_1} + W_{f_2} + W_{f_3} + W_{f_4} + W_{f_5} + W_{f_6}\right)$$

The iteration process is carried out using MATLAB (App., Appendix B) to determine the more accurate values of W_f and W_0.

The total weight of the aircraft at the end of each mission segment is shown in both British and metric units in the table above. This table also shows the empty weight fraction and the fuel weight fraction.

Table 21. The weights of the mission segments and final iteration values

	W_0	W_1	W_2	W_3	W_4	W_5	W_6	W_E/W_0	W_F/W_0
British (lb)	107589.9	106509.7	104699.7	69438.9	69092.9	68008.2	67805.4	0.5367	0.3919
Metric (kg)	48802	48312	47491	31497	31340	30848	30756	0.5367	0.3919

The total weight of the aircraft at the end of each mission segment is shown in both British and metric units in the table above. This table also displays the empty weight fraction and the fuel weight fraction. By subtracting the weight of this segment from the weight of the previous segment, the total fuel burned during each mission segment can be calculated. This table shows the total gross weight as W_0 = 48802 kg (107589.9 lb). By multiplying the fuel weight fraction by the total gross weight, the total fuel weight is 18044 kg (39780.2 lb).

Chapter 7. Aircraft Sizing and Center of Gravity Location

The goal of this chapter is to figure out how big and where the wings, horizontal and vertical tails, and fuel tanks should be. Then, without the wings, locate the center of gravity, then after the wings are in place; locate the center of gravity once more. Then examine the results for accuracy.

7.1. Wing Sizing

The taper was chosen for the wings. The reason for this selection is that trapezoidal sweptbackward shaped wings produce less fraction, resulting in greater aerodynamic efficiency and a good stability characteristic. The blended-winglet wing tip was chosen because it reduces drag caused by turbulence at the tip and increases the lift/drag ratio. To calculate the planform area of our design aircraft's wing, we need our design wing loading value as well as our estimated W_0. From previous calculations:

- $\frac{W}{S} = 625.54\ \mathrm{kg/m^2}$
- $W_0 = 107{,}589.9$ lb = 48,802 kg

Substituting these values to get planform area S

$$S = 78.01\ \mathrm{m}^2$$

Since we calculated our planform area, we can also calculate our wing span using the following formula, where the aspect ratio was taken from a competitor study and set to 9.3.

$$b^2 = \sqrt{AR(S)}\ ,\ \ b^2 = \sqrt{9.3(78.01)} = 26.936\ \mathrm{m}$$

Taking into account the values of planform area and wingspan, as well as competitor jet aircraft, it is decided that the wing shape would be trapezoidal as previously stated.

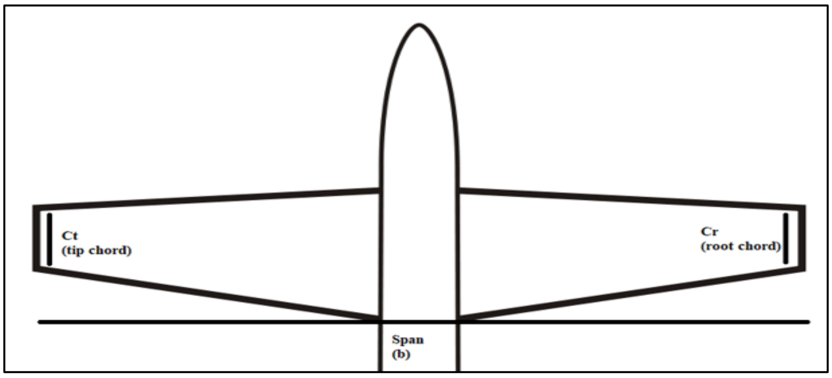

Figure 17. Geometry of Trapezoidal Wing [16]

The geometrical properties of the wing are represented on an example wing in the figure above. Because tip and root chord lengths and span are frequently used in this assignment, this figure is useful for visualizing the geometry.

The root chord and tip chord can be calculated using by

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

where S is wing reference area, C_{root} is root chord C_{tip} is tip root and b is wing span.

$$\text{Taper Ratio}(\gamma) = \frac{C_{tip}}{C_{root}} = 0.265$$

Most swept wings have a taper ratio of about 0.2-0.3. Therefore, the taper ratio is chosen to be 0.3 for our design as mentioned previous assignments. Therefore,

$$78.016 = 26.936\left(\frac{C_{root} + 0.265C_{root}}{2}\right),$$

As a result, C_{root} is found to be 4.579 m, and C_{tip} is found to be 1.213 m.

The mean aerodynamic chord, $\overline{c}$, can be determined using by

$$\overline{c} = \frac{\left(\frac{2}{3}\right) C_{root}(1 + \gamma + \gamma^2)}{(1 + \gamma)},$$

$$\overline{c} = \frac{\left(\frac{2}{3}\right) 4.579(1 + 0.265 + 0.265^2)}{(1 + 0.265)} = 3.222$$

7.2 Fuel Tanks Dimensions and Locations

7.2.1. Fuel Volume Calculations

Obtaining the fuel weight calculated by our iteration code, W_f = 19,125 kg from assignment 6. Knowing that most aircraft and jets use kerosene as fuel, we calculated the fuel density to be 0.813 kg/lt and 813 kg/m^3 [1] using the density table provided in the slides.

As shown below, there are three fuel tanks inside the wing, as well as a fuel tank under the passenger cabin, between the nose and the wing, which holds the remainder of the fuel during flight. Because weight, volume, and density are all related, fuel volume can be calculated using by

$$\text{Weight} = \text{Volume (Density)},$$

$$19{,}125 = \text{Volume } (0.813),$$

$$\text{Volume} = 23524.6 \text{ lt} = 23.5 \text{ m}^3$$

7.2.2. Fuel Tank Sizing

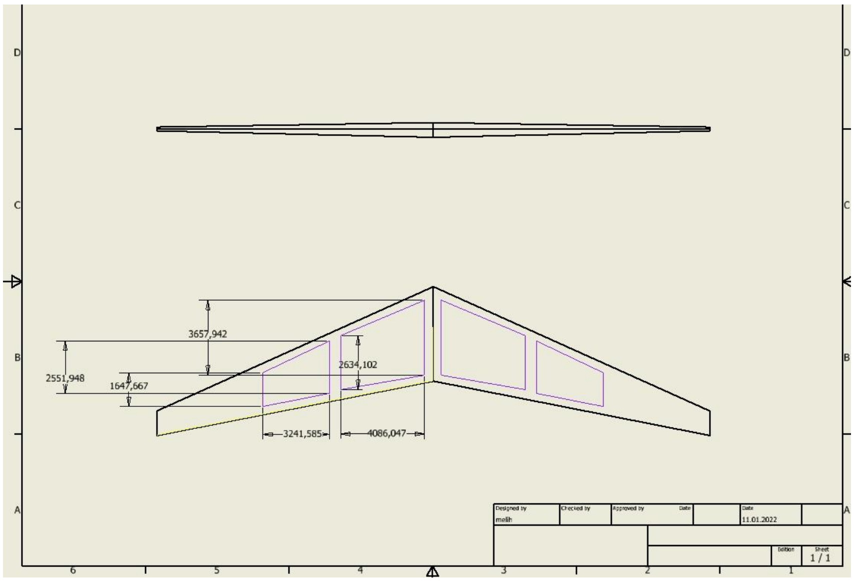

Figure 18. Fuel Tanks Layout

Figure 18 depicts how fuel tanks appear inside the wing when they are installed. Because the locations of the fuel tanks are critical, inventor software is used.

7.2.3 Horizontal and Vertical Tail Configuration

Using historical trend graphs and our gross take-off weight value W_0 = 107,589.9 lb = 48,802 kg, we can calculate vertical and horizontal taper ratios.

Table 22. Historical data for vertical tail properties of different airliners. [4]

Vertical Tail Properties for Airliners							
Aircraft	Gross Weight W_g (lb)	Wing Area S (sq.ft.)	SV_T/S	V.T. Height b_{VT}	Aspect Ratio A_{VT}	Taper λ_{VT}	Sweep Λ_{VT} (deg)
Dash 8 Q100	36,300	585	0.23	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

The vertical tail properties of some of the airplanes are shown in the table above. This table is based on historical trends and is used to determine the best values for the aspect ratio, taper ratio, and sweep angle of the vertical tail of a design aircraft.

Table 23. Historical data for horizontal tail properties of different airliners. [4]

Horizontal Tail Properties for Airliners							
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

The horizontal tail properties of the same airplanes as in Table 23 are shown in this table. This table is based on historical trends and is used to determine the best values for the aspect ratio, taper ratio, and sweep angle of the horizontal tail of a design aircraft.

Tail geometries are defined using Table [22-24] as follows,

$$AR_{VT} = 1.82 \quad , AR_{HT} = 4.54$$

$$\lambda_{VT} = 0.26 \quad , \lambda_{HT} = 0.44$$

Vertical tail and horizontal tail volume coefficients are obtained using the figures below in order to calculate planform area S and span b. Using W_0 and referring to Figure 24-25, the mean value of horizontal and vertical tail volume coefficients is V'_{HT} = 1.2, V'_{VT} = 0.11.

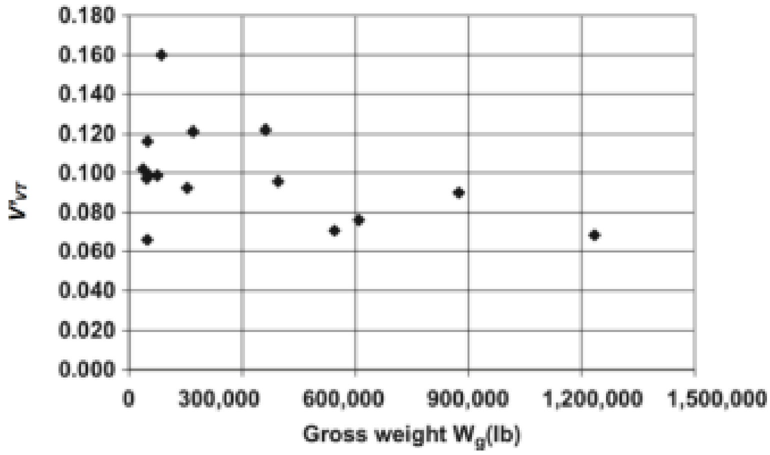

Figure 19. Historical data of horizontal tail coefficients for vertical tail vs. gross weight [4]

The vertical tail volume coefficient vs. gross weight relationship for various airplanes is depicted in the figure above. To calculate this coefficient, the average value of V_{VT} for the gross weight of the design airplane is used.

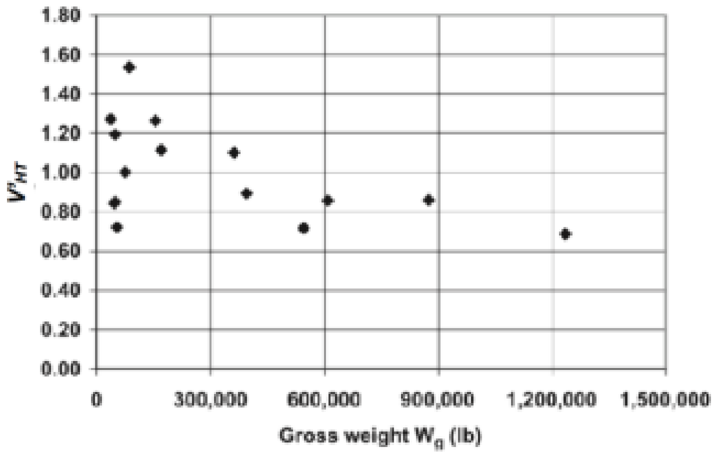

Figure 20. Historical data of vertical tail coefficients for horizontal tail vs. gross weight [4].

Figure 20 depicts the horizontal tail volume coefficient vs. gross weight relationship for the same planes as in Figure 2. To be able to determine this average coefficient value of V_{HT} for the design airplane's gross weight, the same method as before is used.

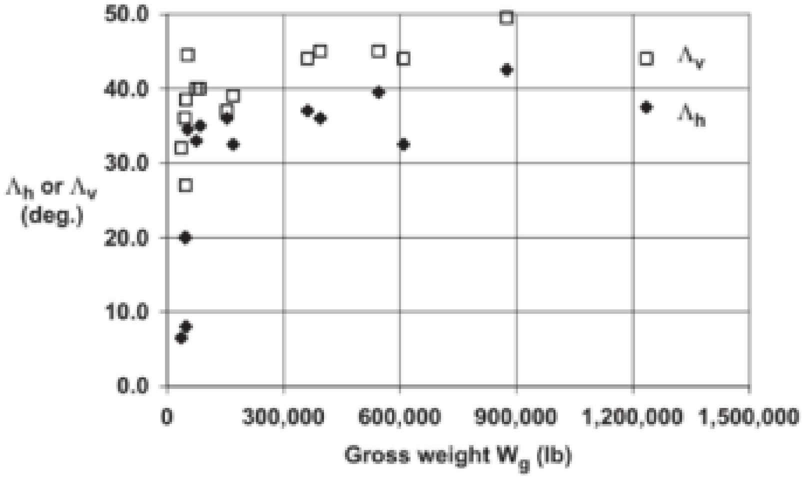

Figure 21. Historical data of sweep angle for vertical and horizontal tails vs. gross weight [4]

The graph above depicts the sweep angle vs. gross weight relationship for both horizontal and vertical tails of the same airplanes depicted in Figures 23 and 24. This figure, like the previous ones, is used to calculate the sweep angles for the design airplane by taking the mean value of the same gross weighted airplanes. Using W_0 and referring to Figure 25, the mean horizontal and vertical tail sweep angles are Λ_{HT}=35 ° and Λ_{VT} =40 ° , respectively.

The vertical tails of turbofan aircraft have sweepback angles ranging from 5 to 10 degrees. greater in size than the horizontal tails (both larger than those of their respective wings)[5].

Using equations shown below to calculate vertical and horizontal tail lengths, and knowing that our fuselage length is $L_{fuselage} = 26.188$ m from chapter 3,

$$L_{HT} = 0.49\ L_{fuselage},$$

$$L_{HT} = 0.49(26.188) = 12.832 \text{ m}$$

where L_{HT} is the length of horizontal tail.

$$L_{VT} = 0.45\ L_{fuselage},$$

$$L_{VT} = 0.45(26.188) = 11.785 \text{ m}$$

where L_{VT} is the length of horizontal tail.

Area for horizontal & vertical tail calculation can be determined using by

$$S_V = \frac{V_V bS}{l_v},$$

where S_V is the area of the vertical wing, V_V is the vertical tail volume coefficient, l_v is the distance between the CG and the average quarter-chord locations of the vertical tail surface.

Therefore,

$$S_V = \frac{(0.11)(26.936)(78.016)}{(11.784)} = 19.615 \text{m}^2$$

$$S_H = (V_H \bar{c} S)/l_H,$$

where S_H is the area of horizontal wing, V_H is the horizontal tail volume coefficient, l_H is the distance between the CG and the average quarter-chord locations of the horizontal tail surface. Therefore,

$$S_H = (1.2)(3.222)(78.016)/(12.832) = 23.507 \text{m}^2$$

Because the aspect ratios for both vertical and horizontal tails are derived from historical trends, the span for horizontal and vertical tails can be calculated using by,

$$b_H^2 = AR_{HT}(S_H),$$

where b_H is the span of the horizontal tail, AR_{HT} is the aspect ratio of the horizontal tail, and S_H is area of the horizontal tail.

$$b_H = \sqrt{(4.54)(23.507)} = 10.331 \text{ m}$$

$$b_V^2 = AR_{VT}(S_V)$$

where b_V is the span of the vertical tail, AR_{VT} is the aspect ratio of the vertical tail, and S_V is area of the vertical tail.

$$b_V = \sqrt{(1.82)(19.615)} = 5.975 \text{ m}$$

Vertical Tail

$$\text{Taper Ratio}(\gamma) = \frac{C_{tip}}{C_{root}} = 0.26$$

$$S_V = b_V \left(\frac{C_{root} + C_{tip}}{2}\right),$$

$$(19.615) = (5.975)\left(\frac{C_{root} + C_{tip}}{2}\right),$$

$$C_{root} = 5.211 \;\; m, \;\; C_{tip} = 1.355 \;\; m \,.$$

$$\bar{c} \;= \frac{\left(\frac{2}{3}\right) C_{root}(1 + +\gamma^2)}{(1+\gamma)} = 3.660\, m$$

Horizontal Tail

$$\text{Taper Ratio}(\gamma) = \frac{C_{tip}}{C_{root}} = 0.44$$

$$S_H = b_H \left(\frac{C_{root} + C_{tip}}{2}\right),$$

$$(23.507) = (10.331)\left(\frac{C_{root} + C_{tip}}{2}\right),$$

$$C_{root} = 3.160 \;\; m, \;\; C_{tip} = 1.390 \;\; m \,.$$

$$\bar{c} \;= \frac{\left(\frac{2}{3}\right) C_{root}(1 + \gamma + \gamma^2)}{(1+\gamma)} = 2.390\, m$$

Table 24. Summary of the Wing and Tail Geometries

	Span, b (m)	**Area, S (m^2)**	**Root Chord C_{root} (m)**	**Tip Chord C_{tip} (m)**	**Mean Aerodynamic Chord, $\bar{c}$ (m)**	**Sweep angle (deg)**
Wing	**26.936**	**78.016**	**4.579**	**1.213**	**3.222**	**27.49**
Horizontal Tail	**10.331**	**23.507**	**3.160**	**1.390**	**2.390**	**35**
Vertical Tail	**5.975**	**19.615**	**5.211**	**1.355**	**3.660**	**40**

The airplane tail design is based on the traditional tail layout. This is due to the fact that the traditional tail provides adequate stability and control while also being the lightest in most circumstances.

7.3. Weight and Location Components

Table 25. The relation between exposed area and the approximate empty weight buildup of each airplane component & The approximate location of the center of gravity of the components and weight ratio of the gears and engine [14]

	Fighters		Transport & Bomber		General Aviation		Multiplier	Approximate Location
	lb/ft²	kg/m²	lb/ft²	kg/m²	lb/ft²	kg/m²		
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% length
	Weight Ratio		Weight Ratio		Weight Ratio			
Landing gear*	0.033		0.043		0.057		TOGW	centroid
Landing gear—Navy	0.045		—		—		TOGW	centroid
Installed engine	1.3		1.3		1.4		Engine weight	centroid
"All-else empty"	0.17		0.17		0.1		TOGW	40-50% length

Table 25 shows the historical values for various aircraft classes, as well as their approximate locations on the component. General aviation is chosen for the airplane design.

Wing Weight

The exposed wing planform area was discovered to be $S_{exposed} = 350\ m^2$ using Inventor. The wing's weight can be calculated using by

$$W_{wing} = 12(S_{exposed}) = 4200\ kg$$

Horizontal Tail Weight

The exposed wing planform area was discovered to be $S_{exposed,HT} = 90\ m^2$ using Inventor. The wing's weight can be calculated using by

$$W_{wing} = 10(S_{exposed,HT}) = 900\ kg$$

Vertical Tail Weight

The exposed wing planform area was discovered to be $S_{exposed,VT} = 79.5\ m^2$ using Inventor. The wing's weight can be calculated using by

$$W_{wing} = 10(S_{exposed,VT}) = 795\ kg$$

Payload Weight

The previously calculated payload weight is W_{pl} = 9,365 lb = 4247.9 kg. The payload weight includes the weight of 50 passengers plus their luggage.

Crew Weight

Calculated previously the crew weight is W_C = 700 lb = 317.5 kg. The crew weight includes the weight of 2 crew members and the weight of their baggage.

Fuel Weight

Fuel weight calculated from the Chapter 6 to be W_f = 39,780.2 lb = 18,044 kg.

Engine Weight

The weight of the engine can be calculated using the below relation. The engine which is chosen as PW1400G in assignment 2, and its dry weight is 2177 kg [6]. The weight of the engine is calculated by substituting the value in the relation shown below.

$$W_{engine} = 2(1.4)(\text{dry weight}) = 6095.6 \text{ kg}$$

Fuselage Weight

The exposed area of the fuselage can be found using

$$S_{fuselage,wetted} = \pi\left(\frac{A_{top} + A_{side}}{2}\right)$$

where A_{top} denotes the top exposed area and Aside denotes the side exposed area However, because it may produce better results, the method used for other components is also used for the fuselage. So, using Inventor Professional, the exposed fuselage area is $S_{fuselage} = 745 \text{ m}^2$.

Weight of the fuselage is found using

$$W_{fuselage} = 7\left(S_{fuselage}\right) = 5215$$

Table 26. Aircraft major components' weights and locations. [Inventor Professional]

Components	Weight (kg)	Location of c.g. from nose (m)
Wing	4200	16.4
Horizontal Tail	900	28.7
Vertical Tail	795	27.5
Payload	4247	13.7
Crew	317.5	2.9
Fuel	18044	10
Engine	6095.6	15.7
Fuselage	4700	16.5

7.4. Center of Gravity Location
7.4.1. Center of gravity location first estimation without wings

The moment of the major components of the aircraft with respect to the nose is calculated by dividing the moments by their weight using the locations of the aircraft components excluding the weights and locations of fuel tanks, wings, and engines. The weights of aircraft components are located using our aircraft drawing and Inventor professional.

The first estimation of the center of gravity is found using by

$$X_{c.g,1} = \frac{W_{crew}X_{crew} + W_{payload}X_{payload} + W_{fuselage}X_{fuselage} + W_{HT}X_{HT} + W_{VT}X_{VT}}{W_{crew} + W_{payload} + W_{fuselage} + W_{HT} + W_{VT}} = \frac{M}{W}$$

$$X_{c.g,1} = \frac{\sum_1^N M_i}{\sum_1^N W_i} = 16.8$$

7.4.2 Center of gravity location estimation with wings

After locating the wings at the c.g. obtained from the first estimate, it is assumed that the wing, engines, and fuel tanks will be located using the first c.g. estimation plus 15% of the mean aerodynamic chord.

$$X_{w,f,e} = X_{c.g,1} + 0.15(MAC)$$

The mean aerodynamic chord, according to our drawing is 3.222 m. Substituting the first c.g. estimation and the MAC, the location of the engines, fuel tanks, and wings are obtained.

$$X_{w,f,e} = 17.28$$

Table 27. Modified engines and fuel tanks and wing locations.

Components	Weight (kg)	Location (m)
Wing	4200	17.5
Engines	6095.6	17
Fuel Tanks	18044	12

$$X_{c.g,1} = \frac{W_{crew}X_{crew} + W_{payload}X_{payload} + W_{fuselage}X_{fuselage} + W_{HT}X_{HT} + W_{VT}X_{VT}}{W_{crew} + W_{payload} + W_{fuselage} + W_{HT} + W_{VT}} = \frac{M}{W}$$

The new value for the center of gravity is $X_{c.g,1}$ = 15.4 m because of discovering wings. The difference in center of gravity between without and with wing location obtained is less than 10%. Therefore, no need to refine the location of wings.

7.4.3. Moment Arms for the Horizontal and Vertical Tails

The moment arm for the horizontal tail L_{HT} can be calculated using by

$$L_{HT} = X_{HT} - X_{c.g.}$$

where X_{HT} is the horizontal tail location and $X_{c.g.}$ is the center of gravity location.

$$L_{HT,before} = 12.83 \text{ m}$$

$$L_{HT,after} = 15.1 \text{ m}$$

The final value of the moment arm of the horizontal tail is found to be %15 greater than the initially estimated value. Therefore, the horizontal tail should be resized.

The moment arm for the horizontal tail L_{VT} can be calculated using by

$$L_{VT} = X_{VT} - X_{c.g.}$$

where X_{HT} is the horizontal tail location and $X_{c.g.}$ is the center of gravity location.

$$L_{VT,before} = 11.78 \text{ m}$$
$$L_{VT,after} = 13.27$$

The final value of the moment arm of the horizontal tail is found to be %11 greater than the initially estimated value. Hence, the vertical tail should be resized to be larger than initially estimated.

Chapter 8. Aircraft Layout

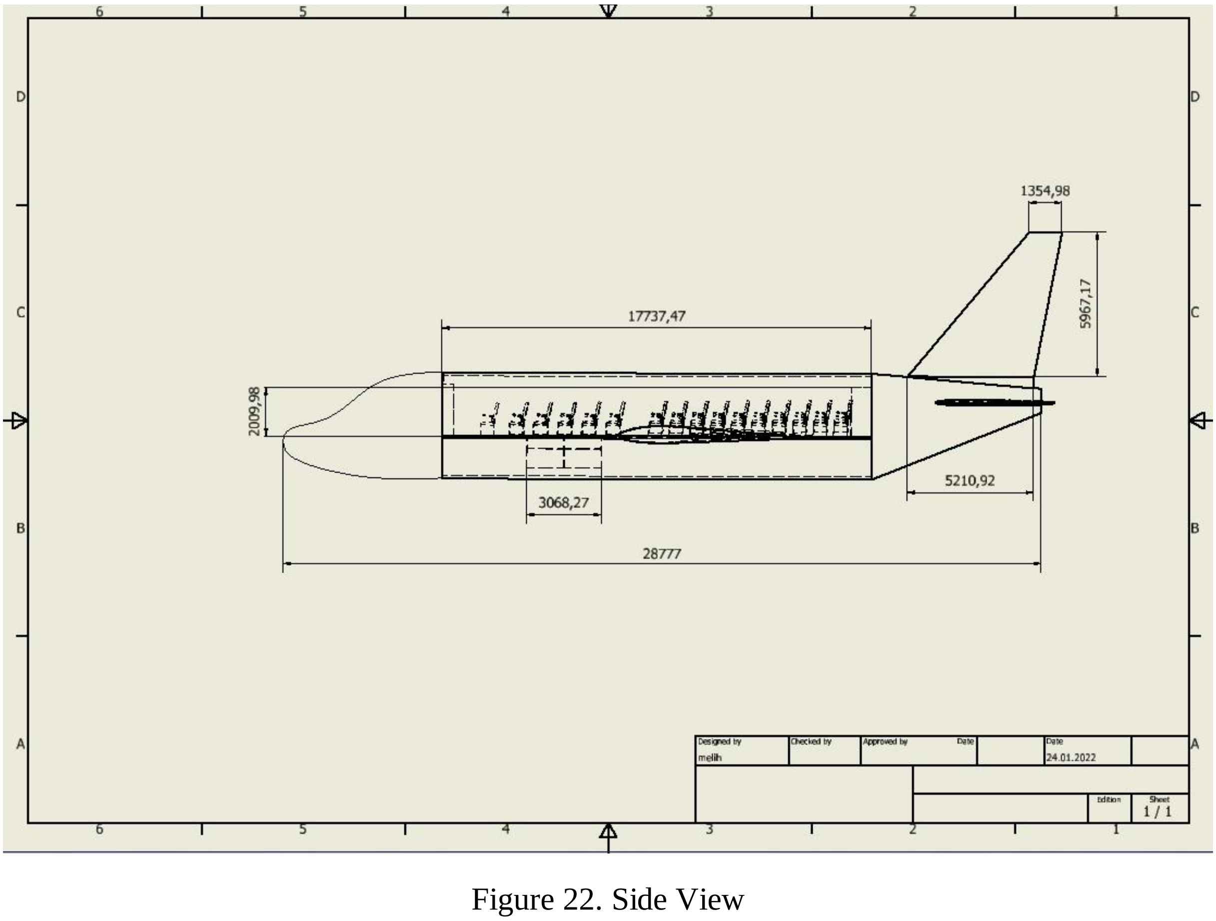

Figure 22. Side View

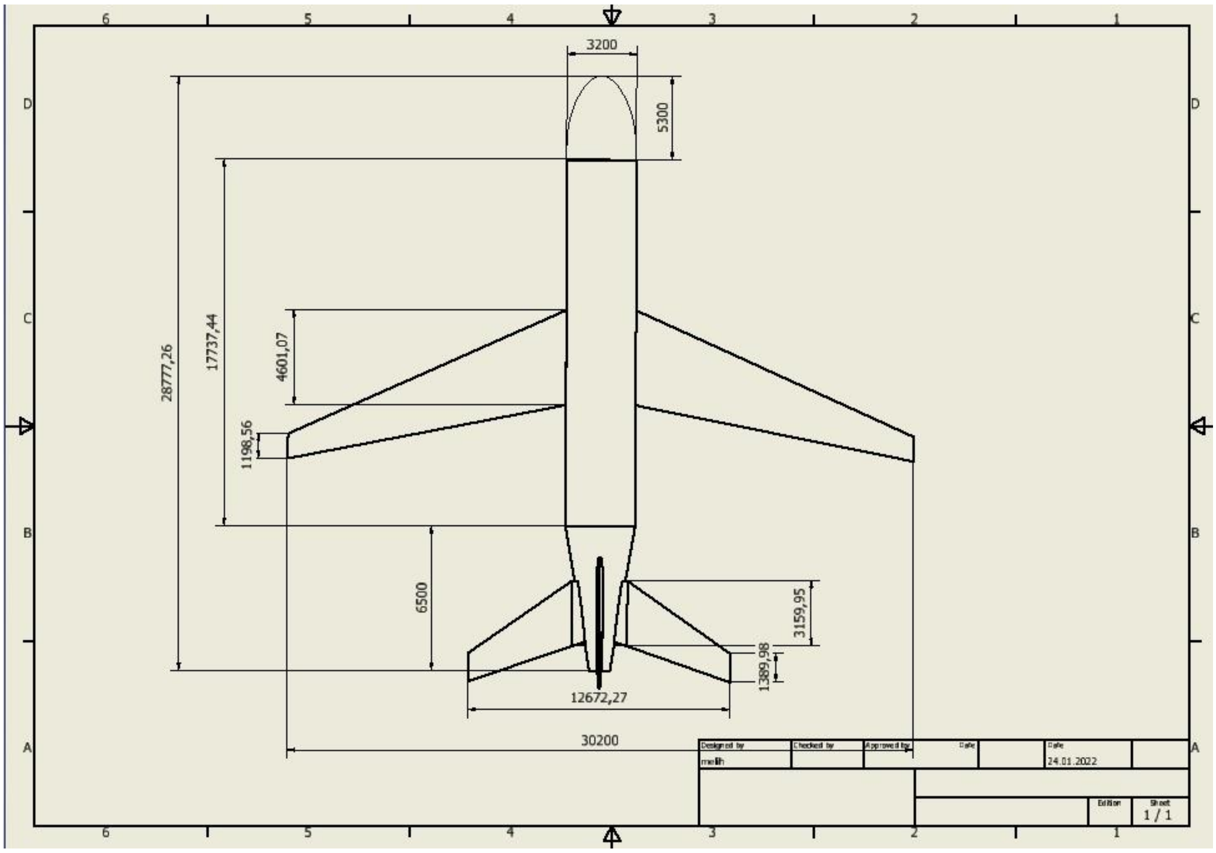

Figure 23. Top View

Figure 24. Designed Business Jet Aircraft

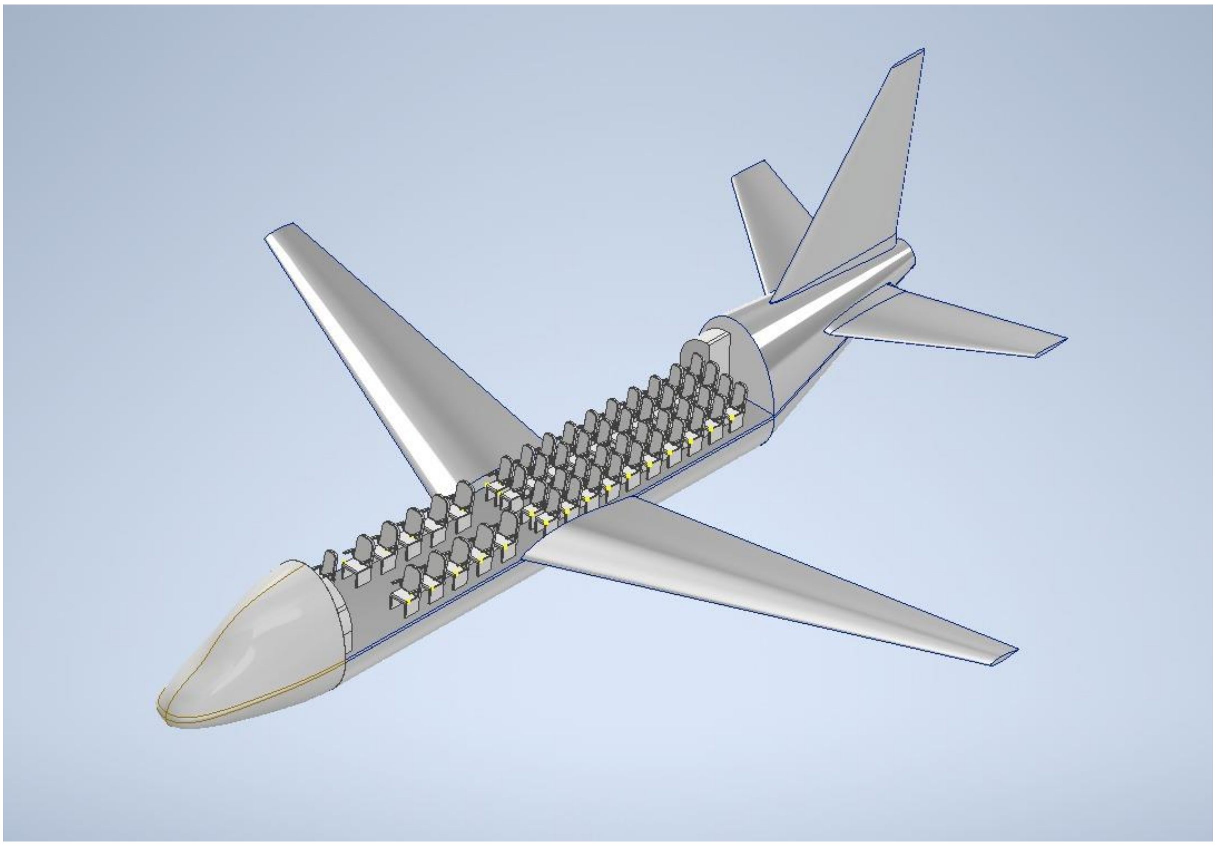

Figure 25. Passenger Compartment of a designed business jet aircraft

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Appendices

Appendix. A

Appendix.1

```
%Part 1 ii)
clc
clear all
close all
A = [1.02 1.02 1.02 ]
C = [-0.06 -0.06 -0.06 ]
W_o = [53000 133501 41000 ]
ks = [1 1 1 ]
emptyweightfract = A.*((W_o).^C).*ks

%Part 1 iii)
clc
clear all
close all
A = [1.1197 1.0720 0.9589 ]
C = [-0.06 -0.06 -0.06 ]
W_o = [53000 133501 41000 ]
ks = [1 1 1 ]
emptyweightfract = A.*((W_o).^C).*ks
```

Appendix.2

```
clc

clear all
passengers = 50
```

```
E = 1/3
R = 3158.75
Wpl = (33+154.3)*passengers %payload weight
Wc = 700 %crew weight
w10 = 0.995
w21 = 0.98
w43 = 0.99
w65 = 0.992
%Cj = specific fuel consumption
%LD = Lift / drag ratio
%R = range
%V = cruise speed
LD = 13
Cj = 2*0.51
V = 421.12
w32 = exp((-R*Cj)/(V*LD))
w54 = exp((-E*Cj)/LD);
Mf = w10*w21*w32*w43*w54*w65
fuelfract = 1.06*(1-Mf)
Kvs = 1
A = 1.02
c = -0.06
W_0_init = 2
W0 = 0
while abs(W_0_init-W0)>1
weightfract = (0.9).*( A*(W_0_init^c))*Kvs;
W0 = (Wc+ Wpl) / (1-fuelfract-weightfract);
W_0_init = W_0_init+1;
end
W0
W_0_init
```

Appendix 3

```
clc
clear all
close all
passengers = 50
E = 1/3
R = [2500:1:3300]
Wpl = (33+154.3)*passengers %payload weight
Wc = 700 %crew weight
w10 = 0.995
w21 = 0.98
```

```
w43 = 0.99
w65 = 0.992
%Cj = specific fuel consumption
%LD = Lift / drag ratio
%R = range
%V = cruise speed
LD = 13
Cj = 2*0.51
V = 421.12
w32 = exp((-R*Cj)/(V*LD))
w54 = exp((-E*Cj)/LD);
Mf = w10*w21*w32*w43*w54*w65
fuelfract = 1.06*(1-Mf)
Kvs = 1
A = 1.02
c = -0.06
W_0_init = 2
W0 = 0
while abs(W_0_init-W0)>1
weightfract = (0.9).*( A*(W_0_init^c))*Kvs;
W0 = (Wc+ Wpl) ./ (1-fuelfract-weightfract);
W_0_init = W_0_init+1;
end
W0
W_0_init
figure(1)
plot(R,W0 ,'g','LineWidth', 2)
xlabel('Range (nautical miles)')
ylabel('W_0 (lb)')
title('Range vs Takeoff Gross Weight')
grid on
```

Appendix 4

```
clc
clear all
close all
passengers = 50
E = [ 0.2 :0.017 : 1.2]
R = 3158.75
Wpl = (33+154.3)*passengers %payload weight
Wc = 700 %crew weight
w10 = 0.995
w21 = 0.98
```

```
w43 = 0.99
w65 = 0.992
%Cj = specific fuel consumption
%LD = Lift / drag ratio
%R = range
%V = cruise speed
LD = 13
Cj = 2*0.51
V = 421.12
w32 = exp((-R*Cj)/(V*LD))
w54 = exp((-E*Cj)/LD);
Mf = w10*w21*w32*w43*w54*w65
fuelfract = 1.06*(1-Mf)
Kvs = 1
A = 1.02
c = -0.06
W_0_init = 2
W0 = 0
while abs(W_0_init-W0)>1
weightfract = (0.9).*( A*(W_0_init^c))*Kvs;
W0 = (Wc+ Wpl) ./ (1-fuelfract-weightfract);
W_0_init = W_0_init+1;
end
W0
W_0_init
figure(1)
plot(E,W0,'b','LineWidth', 2 )
xlabel('Endurance (hours)')
ylabel('W_0 (lb)')
title('Endurance vs Takeoff Gross Weight')
grid on
```

Appendix 5

```
clc
clear all
close all
passengers = [5:1:75]
E = 1/3
R = 3158.75
Wpl = (33+154.3)*passengers %payload weight
Wc = 700 %crew weight
w10 = 0.995
w21 = 0.98
```

```
w43 = 0.99
w65 = 0.992
%Cj = specific fuel consumption
%LD = Lift / drag ratio
%R = range
%V = cruise speed
LD = 13
Cj = 2*0.51
V = 421.12
w32 = exp((-R*Cj)/(V*LD))
w54 = exp((-E*Cj)/LD);
Mf = w10*w21*w32*w43*w54*w65
fuelfract = 1.06*(1-Mf)
Kvs = 1
A = 1.02
c = -0.06
W_0_init = 2
W0 = 0
while abs(W_0_init-W0)>1
weightfract = (0.9).*( A*(W_0_init^c))*Kvs;
W0 = (Wc+ Wpl) ./ (1-fuelfract-weightfract);
W_0_init = W_0_init+1;
end
W0
W_0_init
figure(1)
plot(passengers,W0,'r','LineWidth', 2 )
xlabel('Number of Passengers')
ylabel('W_0 (lb)')
title('Number of Passengers vs Takeoff Gross Weight')
grid on
```

Appendix B

```
clc

close all

clear all

format short

a = 0.32;

b = 0.66;

C1 = -0.13;

C2 = 0.3;
```

```
C3 = 0.06;
C4 = -0.05;
C5 = 0.05;
K_vs = 1;
M_max =0.8;
A = 9.3; %aspect ratio
T_W0 = 0.278; %fix
W_S = 338.59;
S = 90.43 ;
R = 5850*1000; %meters
E = 1200; %seconds
C_J = 0.00013*2 %1/s ;
CD_0 = 0.015;
V =216.67; %meters per second
M = 0.723 %10000m altitude
L_D =17.095; %changed
L_Dmax = 19.74;
e = 0.8;
q = 990.4;
W_0_assumed = 3
W_crew = 317.5; %kg
W_fixed_payload = 4247.9; %kg
W_dropped_payload = 0; %kg
W_0 = 1
while abs(W_0 - W_0_assumed)> 1
W_e_to_W_0 = (a + b*((W_0_assumed)^C1) * (A^C2) * (T_W0^C3)
(W_S^C4)(M_max^C5))*K_vs;
%% define segment weight ratios
W1_to_W0 = 0.99 ; %takeoff
W2_to_W1 = 1.0065 - 0.0325*M ; %climb
W3_to_W2 = exp(-(R * C_J)/(V * L_D)) ;%cruise
```

```
W4_to_W3 = 0.995 ; %descent

W5_to_W4 = exp(-(E*C_J)/L_Dmax) ; %loiter

W6_to_W5 = 0.997 ; %landing

%% define fuel weight fractions

W_f1 = (1 - (W1_to_W0)) * W_0_assumed;

W1 = W1_to_W0 * W_0_assumed;

W_f2 = (1 - (W2_to_W1)) * W1;

W2 = W2_to_W1 * W1;

W_f3 = (1 - (W3_to_W2)) * W2;

W3 = W3_to_W2 * W2;

W_f4 = (1 - (W4_to_W3)) * W3;

W4 = W4_to_W3 * W3;

W_f5 = (1 - (W5_to_W4)) * W4;

W5 = W5_to_W4 * W4;

W_f6 = (1-(W6_to_W5))*W5;

W_fuel = W_f1+W_f2+W_f3+W_f4+W_f5+W_f6 ;

W_0 = W_crew + W_fixed_payload + W_dropped_payload + W_fuel + (W_e_to_W_0
*W_0_assumed);

W_0_assumed = W_0_assumed + 1;

end

W_0_assumed

Refined_takeoff_gross_weight = W_0 %(kg)

W_fuel
```