Toronto Metropolitan University

AER621 Aerospace Structural Design

Aerospace Structural Design and Analysis of a Landing Gear Structure and a Wing Box

Final Report

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Abstract

This is a design project that applies knowledge from a wide range of subjects covered during the semester. The project is divided into two design tasks. The first design task pertains specifically to the design of the aircraft's landing gear structure. A section is also dedicated towards the material selection of the landing gear and their economic and environmental impacts. The second task pertains specifically to the structural design of a 1-meter section of the aircraft's wing box starting 6 meters outboard of the wing root. A section is also dedicated towards a detailed weight distribution and stress analysis of the given aircraft. Starting with the landing gear design task, the applied weight on the wheels was determined using the maximum landing weight. A tire type and size were selected based on this applied loading. Along with it, calculations for the stroke and oleo strut preliminary sizing were also based on this applied loading. After determining all the dimensions, a CAD model of the landing gear was developed on Catia. The material selection was performed for the designed landing gear according to the set constrains such as weight and strength. An economical and an environmental impact analysis was completed for the selected materials.

For the second design task, the design of the wing box section was completed to meet the weight and stress criteria. The material selection was performed for the wing box section keeping in mind the weight constrain. A method of structural idealization along with an iterative process was adopted to design the wing box such that the stringers carry all the direct stress. These direct stresses were compared with the critical buckling stress to check if the wing box fails under buckling. Finally, a weight analysis was completed for the wing box making sure that the total weight for the entire wingspan lies between 50-60 percent of the total wing structural weight. The stringer shapes and dimensions are determined using the preliminary sizing equations and a CAD model was developed on Catia. In the end, a detailed weight analysis and a detailed stress analysis was conducted for the given aircraft.

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Introduction

The objective of this project is to complete two structural design tasks related to the aircraft provided. The first design task pertains specifically to the structural design of the aircrafts landing gear and its material selection. The second design task pertains specifically to the structural design of aircraft wing box. Different methods of structural design and analysis were considered to complete both tasks. These methods are discussed further in the introduction section for each design task. General aircraft specifications were given and are presented in the next section, but to carry out the analysis, important assumptions and decisions were made and are discussed in the introduction section for each component design.

Aircraft Specifications

The following information about the aircraft is provided in the project guidelines which are used to complete the two design tasks. Table 1 tabulates the aircraft dimensions, and the aircraft mass targets. Figure 1 illustrates the distribution of wing structures loads, fuel loads, engine, and landing gear loads over the span of the wing.

Table 1: Given Aircraft Specifications.

Aircraft Overall Dimensions	
Overall A/C Length	31 m
Overall A/C Height	10 m
Total Wingspan	34 m
Wing Surface Area	99 m^2
Taper Ratio	0.16
Fuselage Diameter	3.9 m
Aircraft Mass Targets	
Maximum Take-off Mass	52,800 Kg
Operating Empty Mass	23,760 Kg
Maximum Fuel Mass	15,840 Kg
Maximum Payload Mass	13,200 Kg
Maximum Cargo Mass	3,168 Kg
Wing Structure and Systems Mass	5,280 Kg
Each Engine Mass	1,936 Kg
Total Mass for Tricycle Landing Gear	2,640 Kg

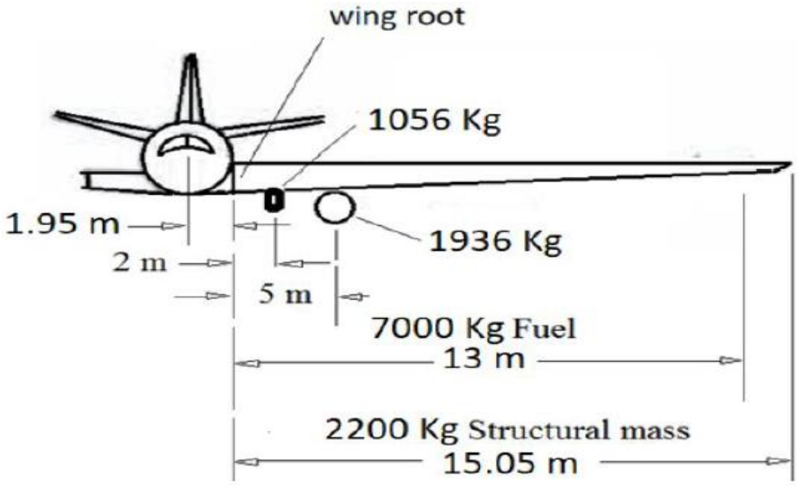


Figure 1: Aircraft Weight Distribution and Wing Dimensions.

The cruise speed of the aircraft was given as Mach 0.8 at an altitude of 12,000 m and the maximum design speed(dive speed) was given as Mach 0.85. The clean $C_{l,max}$ of the aircraft was given as 1.5. The propulsion system consists of two wing-mounted engines, each weighing 1936 Kg and delivering 70 KN of static sea level thrust. As seen in the figure the engines are located at 5 meters outboard of the wing root. The total fuel mass is 15840 Kg. As seen in the figure, a total of 7000 Kg of fuel is placed on each wing distributed over the inner 13 meters of the exposed wing section. The remaining amount is placed in the centre box section. The main landing gear weighs 1056 Kg each. The wing mounted landing gears are 2 meters from the root of the wing on each side. The nose landing gear is 528Kg and is located 4 meters aft of the aircraft nose.

First Design Task – Landing Gear Design

Introduction

The primary objective of the first task is to do a structural design and analysis of the landing gears and do the material selection for the same. The structural design is focused on determining the preliminary sizing of the main landing gear(MLG) Oleo struts, including stroke, shock strut diameter and the overall strut length. The secondary objective is to choose the appropriate tire type and its size based on the ground load requirements for the aircraft discussed previously. In the end, a section is dedicated towards material selection for the design. This section will compare various materials by estimating the weight of the landing gear load bearing components. The materials are also compared through their environmental and economic aspects.

Summary of Analysis and Calculations

This section is a detailed overview of the assumptions made and calculations done to complete the structural design of the main landing gear. One of the factors while making the assumptions was to compare the given aircraft data to an actual aircraft. An Embraer E195-AR aircraft was discovered to have the closest specifications in terms of its weights and dimensions.

From the given data on aircraft weights, an assumption was made that the maximum landing weight would equal 90% of the maximum take-off weight of the aircraft. This is a very generous assumption as it considers situations when the aircraft must do a rejected take-off or return to the departure airport. This assumption was proved valid after verifying with the Embraer's take-off and landing weight. The following calculations are based on this landing weight and the assumption that the maximum static load on main landing gears is 90% of the aircraft weight while landing.

$$M_{max,landing} = (90\%) * M_{max,takeoff} = 47{,}520\ Kg$$

$$Maximum\ Static\ load\ on\ Main\ Landing\ Gear = 0.9 * M_{max,landing} = 42{,}768\ Kg$$
$$= 94{,}287\ lbs$$

Based on the above maximum landing mass, minimum of 2 wheels per landing gear are required for a reliable design.

The weight on each wheel is calculated from: $W_W = \frac{Max\ Static\ Load\ on\ MLG}{no.of\ Tires} = \frac{94287}{4} \approx$ $23{,}600\ lbs$

Based on the load carried by each wheel, a Type VIII tire and a Type VII tire were selected for the MLG wheel design from the tire selection figure provided in the appendix section(Figure 12). The following table compares important parameters such as the tire's diameter, width, rolling radius and wheel diameter.

Table 2: Tire Selection.

Choice 1: Type VIII Tire – 37X14.0 – 14.	**Choice 2:** Type VII Tire – 36X11.
Maximum Load Sustained: 25,000 lbs	Maximum Load Sustained: 26,000 lbs
Pressure: 160 Psi	Pressure: 235 Psi
Maximum Width: 14 in	Maximum Width: 11.5 in
Maximum Diameter: 37 in	Maximum Diameter: 35.1 in
Rolling Radius: 15.1 in	Rolling Radius: 14.7 in
Wheel Diameter: 14 in	Wheel Diameter: 16 in

To make a definitive decision on the tire type, the kinetic breaking energy is calculated to find the optimal wheel diameter. The kinetic energy was found using the landing weight, no. of tires and the stall speed.

$$KE_{breaking}\ per\ tire = \frac{\frac{1}{2} M_l * V_{stall}^2}{no.of\ tires} = \frac{\frac{1}{2} 47520 * 72^2}{4} = 30.8\ MJ$$

Based on the above calculated kinetic breaking energy per tire, if using expensive and complicated breaks the required wheel diameter is 15 inches, and if using the wheels for a large transport aircraft the required wheel diameter is 22 inches. The kinetic energy Vs wheel diameter graph is provided in the appendix section.

For a good design, tires are usually selected based on the smallest tire for the given load (W_W). For the given design requirements, a Type VII – 36X11 tire was selected with a modified wheel diameter of 22 inches. The type VII tire has higher internal pressure with smaller dimensions compared to the type VIII tire. The type VII tire is designed for higher speeds and is used by most jet aircrafts. After selecting the tire, the range for the energy absorption efficiency of tire was found to be 0.45 to 0.5. This is needed for the calculations of the stroke of the landing gear strut.

The Oleo pneumatic shock strut combines the spring effect using compressed air with damping effect. For the preliminary design of the MLG oleo struts, FAR empirical equations and regulations were used. According to FAR, the sink speed is equal to 10 fps(3.05 m/s), the minimum tire deflection is 1.2 in (0.03 m). The efficiency for an oleo pneumatic shock absorber is 80% and the energy absorption efficiency for the tires is 0.47. For a large aircraft, a reaction factor of 1.5 is recommended.

From the above data, the strut stroke is calculated as:

$$S = \frac{V_v^2}{2g\eta\lambda} - \frac{\eta_t}{\eta} S_t = \frac{3.05^2}{2 * 9.81 * 0.8 * 1.5} - \frac{0.47}{0.8} 0.03 = 37.75\ cm$$

Adding the 3 cm safety margin to the stroke: $S_{design} = 40.75\ cm = 16\ in$

A good estimation for the Strut length: $l_s = 2.5 * S_{design} \approx 102\ cm = 40.2\ in$

The shock strut diameter(d_s) can be calculated using the landing weight and number of struts.

$$P_m = \frac{w_l}{n_s} = \frac{466171}{2} = 52{,}391\ lbs$$

$$d_s = 0.041 + 0.0025\sqrt{P_m} = 18.7\ cm = 7.36\ in$$

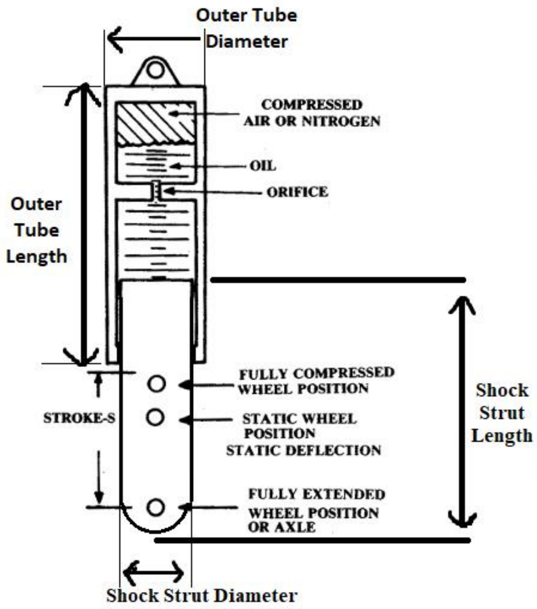
A good first estimation for the landing gear outer cylinder dimensions can be determined as,

$$Cylinder\ outer\ diamter\ \left(d_{cylinder}\right) = 1.2 * d_s = 22.5\ cm = 8.86\ in$$

$$Cylinder\ Length = l_{cylinder} = S_{design} + 3.25 * d_s = 102\ cm = 40.16\ in$$

A yield stress analysis is done for the landing gear oleo strut which will be an important criterion for its material selection. The load W_W is considered to act axially on the cross section of the strut. The calculation below is done with a safety factor of 1.5 to obtain the ultimate yield stress.

$$Ultimate\ yield\ Stress = 1.5 * \frac{W_W * 4}{\pi * d_s^2} = 832\ psi = 5.7\ Mpa$$



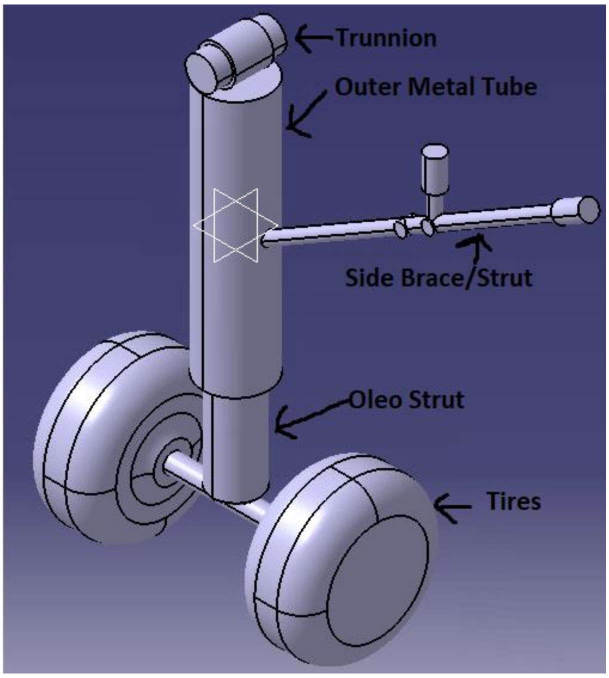


Figure 2: Main Landing Gear Oleo-Pneumatic Strut(Left) and Main Landing Gear CAD Design(Right).

The above figure illustrates the designed main landing gear with oleo pneumatic strut. The figure on the left shows the designed parameters for strut diameter, stroke, and overall strut length to be 18.7 cm, 40.75 cm, and 102 cm respectively. The outer cylinder dimensions such as cylinder diameter and cylinder length are 22.5 cm and 102 cm respectively. The figure on the right illustrates a CAD design for the main landing gear. The CAD design comprises of the oleo strut piston, the outer metal tube, the tires, the trunnion, and the side strut. The engineering drawing for the CAD design is included in the appendix section.

For simplicity of the design, the shock strut diameter is constant as seen in the figure on left. The volume of the load bearing component (shock strut) can be calculated as: $V = \pi r^2 h = 28014\ cm^3 = 0.028\ m^3 = 1710\ in^3$. The outer cylinder is considered hollow for it to accommodate the strut. The inner diameter is assumed to be equal to the piston diameter for simplicity. The volume for the outer tube is calculated as: $V = \pi(r_o^2 - r_i^2)h = 12542\ cm^3 = 0.0125\ m^3 = 765.4\ in^3$. The next section compares various pre-selected materials based on minimum weight and high strength for the landing gear load bearing component.

Material Selection

A pre-selected material list was created based on the most common materials used for landing gears manufacturing in the industry. These materials were chosen for their high specific strength and toughness and low density. For an initial analysis, the materials were tabulated and scored based on their geometric and mechanical properties. The objective of this initial analysis is to score materials (highest being 10 and least being 1) that prevents plastic deformation under ultimate loads whilst having low density.

Table 3: Material Mechanical Properties Comparison for Main Landing Gear.

Material	**Ultimate Tensile Strength (Mpa)**	**Density (Kg/m^3)**	**E (Gpa)**	**G (Gpa)**	**Score**
Aluminium Alloy 7010 T 736	500	2820	69	26	8
Titanium AMS 4967 TA13	830	4420	113	42	9
Steel BS S99	1080	7830	200	76	7

The above tabulated materials were selected to withstand the impact and shock resistance during landing, resist vibrations during the flight, withstand cold temperatures, humidity, and water. In the above table, steel was ranked the lowest as it would offer the highest weight of the landing gear. A Little higher than steel, the aluminium alloy was scored 8. Aluminium alloys are light weight with high specific strength, but they are very prone to stress concentration. The best scored material is the titanium alloy because of its light weight, high strength, and reduced corrosion sensitivity. Note that no material was scored 10 as there are some ultra-high-tensile steels available which have better performance. These materials are not considered because of their high weight and costs.

Table 4: Material Weight and Cost Comparison For MLG.

Material	Density (Kg/m^3)	Weight of the Shock Strut (Kg)	Weight of the Outer Cylinder (Kg)	Total Weight(Kg)	Cost ($/Kg)	Total cost ($)
Aluminium Alloy 7010	2820	80	35.4	115.4	1.6	184.64
Titanium AMS 4967	4420	125	55.4	180.4	25	4510
Steel BS S99	7830	220	98.2	318.2	0.9	286.38

The above table tabulates the weight (in Kg) and cost (in $) of the component for the materials considered. As expected, the aluminium alloy MLG has the least weight followed by titanium alloy MLG and the steel MLG having the highest weight. To make an optimal choice based on the components final weight and the material's mechanical and physical properties, aluminium alloy becomes a clear choice. When choosing aluminium, the main landing gear has a combined weight (from oleo strut and outer tube) of 115.4 Kg and an ultimate tensile strength of 500 Mpa with an ultimate critical yield stress of only 5.7 Mpa. The next section discusses the economic impacts of these materials which would also validate the material choice.

Economic Impacts

Material productions and component manufacturing have huge economic impacts on all stakeholders including the consumers, and the suppliers. Component design and manufacturing techniques are major aspects to reduce costs of common engineering materials such as steel alloys, aluminium alloys, and titanium alloys. Other significant costs include labour costs, insurance, profit, etc. The production and manufacturing of materials contribute largely towards the country's economy and the workers quality of life. To engineers, different materials costs variably due to their machinability. In our design case, even though titanium has the most desired properties, it also costs the most due to its low machineability and higher production costs. The cost for steel per unit kg is relatively low as compared to aluminium but being a heavier material, it increases the aircrafts overall structural weight resulting in more fuel being consumed in flight and therefore increasing the costs for operation. This also has an environmental impact as the increase in fuel consumption leads to more toxic emissions.

Environmental Impacts

A large amount of energy is required for material mining and production. This is a significant source of greenhouse gas emissions. The materials also produce large number of wastes during production as well as at end-of-life waste disposal. Concerned with materials used for the landing gear design, steel production has the highest impacts on the environment including air emissions, hazardous and solid wastes, and wastewater contaminants. A key element to produce primary aluminium used in aluminium alloy is aluminium oxide. The extraction of pure aluminium has major environmental emissions, mainly CO2 emissions. To reduce these emissions, industries are taking advantage of endless recyclability of aluminium and by using green electricity for its production. Out of all three, titanium is the most environment friendly. Even though titanium has low toxicity, when heated in air it has explosive hazardous. The use of titanium alloys in planes, trains, and ships improves their fuel economy. Since titanium is noncorrosive and non-rustic, it has longer life cycles but can also be easily recycled at the end of its life cycle.

Second Design Task – Wing Box Design

Introduction

The second design task pertains specifically to the structural design and analysis of a 1-meter section of the wing box for the given aircraft. The location of the wing box is 6 meters outboard of the wing root, i.e., the section contained within the 6^{th} and 7^{th} meter outboard of the wing root. For the stress analysis part of the task, the wing box is constrained to withstand the limit loads at the positive low angle of attack (PLAA) with a safety factor of 1.5. The wing box is designed between two ribs and two spars with thin skin panels covering top and bottom surface. An iterative method was adopted for the design process. For the 1^{st} iteration an educated guess is made for the structural idealization calculations such as, number of stringers, their initial areas, and panel and spar thickness. These assumptions are modified in each iteration to find an optimal combination of the wing box's structural weight while maintaining its structural integrity.

Material Selection

The specific objective of the second task is to design the wing section to optimize weight while meeting all structural integrity requirements. The choice of material is an important factor not only for the wing box's optimal weight but also for its strength, stiffness, fracture toughness, and resistance to cracking and corrosion. Therefore, it is important to choose a material, before the stress analysis, with less density and high stiffness and strength. There are many criteria to consider for the material selection. For this design task, the objective is to choose an appropriate light weight, high strength, and cost-effective material. The final material chosen for the wing box should be able to resist high compressive and tensile loading and fatigue loading. A similar set of materials as for the landing gear design are chosen and are evaluated based on the standards set above. The aim of this analysis is to score materials (highest being 10 and least being 1) that prevents panel buckling and yielding under ultimate loads whilst having low density.

Table 5: Material Mechanical Properties Comparison for the Wing Box.

Material	Yield Strength (Mpa)	Density (Kg/m^3)	E (Gpa)	G (Gpa)	Score
Aluminium Alloy 7010 T 736	503	2780	73	26	8
Titanium AMS 4967 TA13	1100	4420	113	42	7
Steel BS S99	470	7830	200	76	5

The above table lists some of the most common materials used in modern aircrafts along with their mechanical properties. The most important parameter, while scoring the materials, was that the material must be lightweight. Aluminium alloy was chosen to be the optimal material for wing box panels, spars, and stringers as it was scored highest at 8. It is the most widely used metal in the aircraft industry due to its high strength to weight ratio. In addition to its excellent mechanical properties, Aluminium alloy is easily manufactured and is available widely at lower costs. The above tabulated materials are also compared through their economic and environmental impacts in a subsequent section. Note that no composites are listed in the table as the use of a composite material comes with certain trade-offs mainly from an economic standpoint.

V-N Diagram and PLAA Moment Calculation

Before doing the Structural idealization of the wing box structure, it is important to determine the moment loads acting on the direct stress caring booms. To find the moment based on the forces acting in the positive low angle of attack(PLAA) section, a V-N diagram was created based on the given CL max, max take-off weight and cruise and dive Mach numbers. Figure 3 illustrates a V-N diagram for the given aircraft specification. The figure shows the PLAA region with limit load of 2.5 and dive speed of 130 m/s or 253 Kts. It also shows the ultimate limit load line which is found by multiplying the limit load factor by 1.5. The stall speed at level flight was calculated to be 76 m/s or 148 Kts. The following calculations are used to draw the V-N diagram.

$$V_{c,eq} = \sqrt{M_c^2 \frac{\alpha RT * \rho}{\rho_{sl}}} = 120 \frac{m}{s} = 233\ Kts$$

$$V_{D,eq} = \sqrt{M_D^2 \frac{\alpha RT * \rho}{\rho_{sl}}} = 130 \frac{m}{s} = 253\ Kts$$

$$At\ n = 1, V_{s1,eq} = \sqrt{\frac{2 * n * W}{\rho * s * Cl_{max}}} = 76 \frac{m}{s} = 148\ Kts$$

$$limit\ load\ factor(n) = 2.1 + \frac{24{,}000}{W + 10{,}000} = 2.3$$

Since the limit load factor (n) cannot be less than 2.5 according to FAR regulations, the modified limit load factor (n) = 2.5.

$$At\ n = 2.5, V_{A,eq} = \sqrt{\frac{2 * n * W}{\rho * s * Cl_{max}}} = 120 \frac{m}{s} = 233\ Kts = V_{C,eq}$$

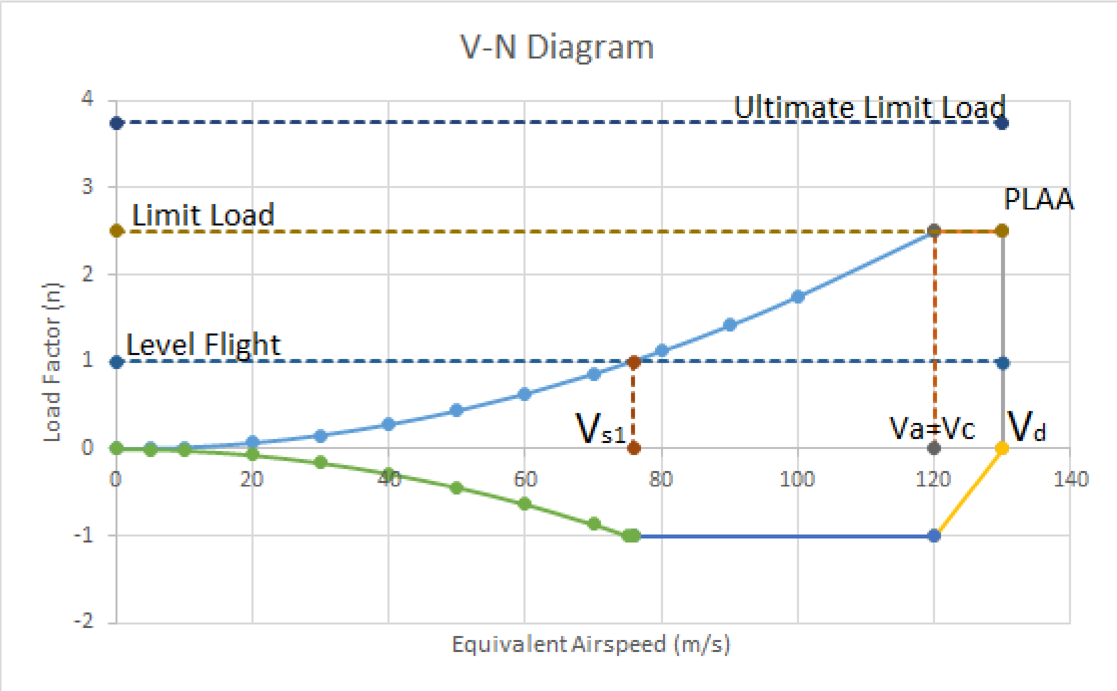


Figure 3: V-N Diagram.

From the above calculated limit load factor (n), At PLAA region $L_{lim} = n_{lim} * W = 1295\ KN$

The ultimate lift for each wing is calculated from, $L_{ult} = \frac{L_{lim}}{2} * 1.5 = 971.2\ KN$

The ultimate moment for each wing is calculated from, $M_{ult} = L_{ult} * \frac{17}{3} = 5500\ MN * mm$

This ultimate moment is used to calculate direct stress in each boom. The maximum direct stress is then compared to the critical buckling stress to check if the system fails under buckling. Note that this is not the actual moment acting over the wing box. The actual moment will be a certain percentage lower due to the wing structures, fuel, landing gear and engine weight acting on the wing opposite to the lift.

Panel Buckling Calculation

The buckling analysis can be applied for the thin wall panels. The critical buckling for the panels is calculated based on the assumption that the panels are curved. This is because curved panels have much higher resistance to buckling as compared to flat panels. The radius of curvature of the panel was found for the selected NACA 0012 Airfoil which was then used to calculate the buckling constant K_c. The buckling constant chart for curved panels is included in the appendix section. As discussed previously, an aluminium alloy is used for panels with E = 73 Gpa and ν = 0.33.

In the first iteration, the initial guess for the panel thickness and stringer spacing was 8 mm and 300 mm respectively. For radius of curvature (r) = 170 mm, $z = (1-\nu^2)^{0.5} * \frac{b^2}{r*t} = 62.5$

For K_c = 40 the critical stress is calculated as, $\sigma_{CR} = \frac{K_c*\pi^2*E}{12(1-\nu^2)} * \left(\frac{t}{b}\right)^2 = 1916.5\ Mpa$

In the final iteration, the modified panel thickness and stringer spacing are 5 mm and 250 mm respectively. For radius of curvature (r) = 170 mm, $z = (1-\nu^2)^{0.5} * \frac{b^2}{r*t} = 69.4$

For K_c = 50 the critical stress is calculated as, $\sigma_{CR} = \frac{K_c*\pi^2*E}{12(1-\nu^2)} * \left(\frac{t}{b}\right)^2 = 1348\ Mpa$

The above determined critical stress values are compared with maximum direct stress calculated in the next section. For the panels to not fail under buckling, it is important that the critical buckling stress is higher than the maximum direct stress.

Structural Idealization Calculation

The design of a wing box using the method of structural idealization is a very open-ended task as there are numerous free variables to be considered. For the purpose of simplicity for this design task, some assumptions are made related to the structural layout of the wing box ahead of the boom and stress calculations. For a preliminary analysis, it was assumed that the panel thickness is constant from root to tip of the wing. To use the method of structural idealization, it is necessary that the panels and spars are treated as a closed thin wall cell, with booms resisting all the direct stress and the panels only effective in shear stress. The wing box is assumed to have a rectangular cross section which allows for double symmetry with an axis of symmetry in the vertical direction and the other in the chord direction. A NACA 0012 symmetric Airfoil is selected for the wing box design. The Airfoil has the maximum thickness of 12% of the chord length, which is also considered for the wing box spar height calculation.

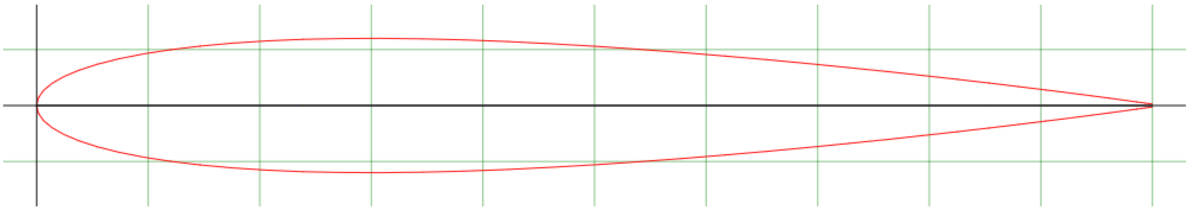


Figure 4: NACA 0012 Airfoil.

The front spar of the wing box is located at 21% of the chord length and the rear spar is located at the 65% of the chord length. From this constrain, the width of the wing box is calculated to be 1500 mm and the height of the wing box is calculated to be 400 mm. With the initial guess for the number of stringers to be 12, the spacing is calculated to be 300 mm. The initial guess for the panel thickness and spar thickness is 8 mm and 10 mm respectively and the stringer area is assumed to be 450 mm^2. These initial estimations are modified based on the resulting stress and weight analysis. The structural weight of the wing box is required to be 50% - 60% of the total wing structural mass. The maximum direct stress is required to be less than the critical buckling stress calculated above.

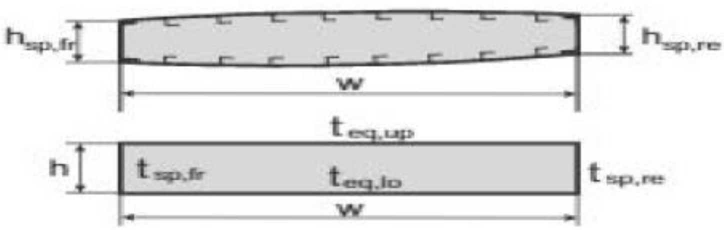


Figure 5: Simplified Rectangular Wing Box Design.

From the given aircraft specifications, the Airfoil chord length, and the wing box width and height can be calculated as a function of the distance of the wing box from its root. The wing box is located between the 6th and 7th meter outboard of the wing root. The chord length is calculated at the average location of 6.5 meters outboard of the wing root. As mentioned above the width of the wing box is 44% (65-21) of the chord length and the height is 12% of the chord length.

$$Chord\ C_y = \frac{2*s}{(1+\lambda)b}\left[1+\frac{2*y}{b}(\lambda-1)\right] = \frac{2475}{493} - \frac{2079}{8381}*y$$

$$w(y) = 0.44*C_y(y) = 0.44*\left[\frac{2475}{493} - \frac{2079}{8381}*y\right] = 1500\ mm$$

$$h(y) = 0.12*C_y(y) = 0.12*\left[\frac{2475}{493} - \frac{2079}{8381}*y\right] = 400\ mm$$

First iteration

For the first iteration, the panel thickness and the spar thickness are 8 mm and 10 mm respectively. There are 12 stringers each with an initial area of 450 mm^2. The boom area for each boom is calculated as, $B_n = 450 + \sum_m \frac{t_m * b_m}{6}(2+\frac{y_m}{y_n})$. The boom area calculations are included in the appendix section and are tabulated in Table 6. As seen from the figure below, the two axis of symmetry allows for a simpler approach where, $B_1 = B_6 = B_7 = B_{12}$, $B_2 = B_5 = B_8 = B_{11}$ and $B_3 = B_4 = B_9 = B_{10}$.

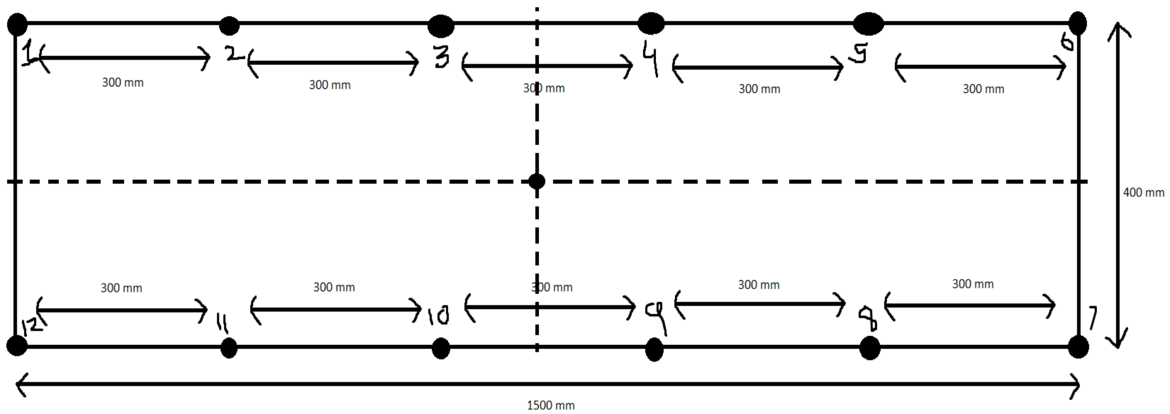


Figure 6: Wing Box - First Iteration.

Table 6: Structural Idealization – First Iteration.

Boom	Boom Area (mm^2)	Y (mm)	I_{xx} (mm^4)	σ_z (N/mm^2)
1	2317	200	$92.68*10^6$	857.55
2	2850	200	$114*10^6$	857.55
3	2850	200	$114*10^6$	857.55
4	2850	200	$114*10^6$	857.55
5	2850	200	$114*10^6$	857.55
6	2317	200	$92.68*10^6$	857.55
7	2317	-200	$92.68*10^6$	-857.55
8	2850	-200	$114*10^6$	-857.55
9	2850	-200	$114*10^6$	-857.55
10	2850	-200	$114*10^6$	-857.55
11	2850	-200	$114*10^6$	-857.55
12	2317	-200	$92.68*10^6$	-857.55
$\Delta\ I_{xx}$ =			$1282.72*10^6$	

The maximum direct stress calculated from the first iteration is 857.55 MPa is tension and -857.55 Mpa in compression. Comparing the maximum direct stress to the critical buckling stress of 1916.5 Mpa it can be said that the wing box will not fail under buckling. Since the main criteria for this design task is for the wing box to be light weight, the weight for the wing box is required to be between 50-60 percent of the total wing structural weight. The table below tabulates the volume for each component. The weight for each component and the overall weight is calculated by multiplying the volume by the density of the selected material.

Table 7: Weight Calculation - First Iteration.

Component	Volume for 1 Meter Section (m^3)	Density (Kg/m^3)	Weight for 1-meter wing box section (Kg)	Weight for the entire wingspan (15.05 m) (Kg)
Spars	0.008	2780	22.24	334.7
Panels	0.024	2780	66.72	1004.1
Stringers	0.0054	2780	15.012	226
Total	0.0374	2780	104	1565

As seen in the above table, the calculated weight for the wing box for the right wing is 1565 Kg, or 71% of the total wing structural mass. The total weight of the designed wing box is higher than the desired range discussed previously. This means that the designed parameters are 'over' the required limits. For the next iteration the number of stringers is increased to 14 while reducing the spacing between them to 250. The panel thickness is also reduced to 5 mm. This change will have an impact on the maximum direct stress and the structural weight of the wing box.

Final Iteration

For the final iteration, the panel thickness is modified to 5 mm. The number of stringers is increased to 14 and the spacing is reduced to 250 mm. Boom areas are calculated using the same equation as for the first iteration. The boom area calculations are included in the appendix section and are tabulated in Table 8. As seen from the figure below, the two axis of symmetry allows for a simpler approach where, $B_1 = B_7 = B_8 = B_{14}$, $B_2 = B_6 = B_9 = B_{13}$, $B_3 = B_5 = B_{10} = B_{12}$ and $B_4 = B_{11}$.

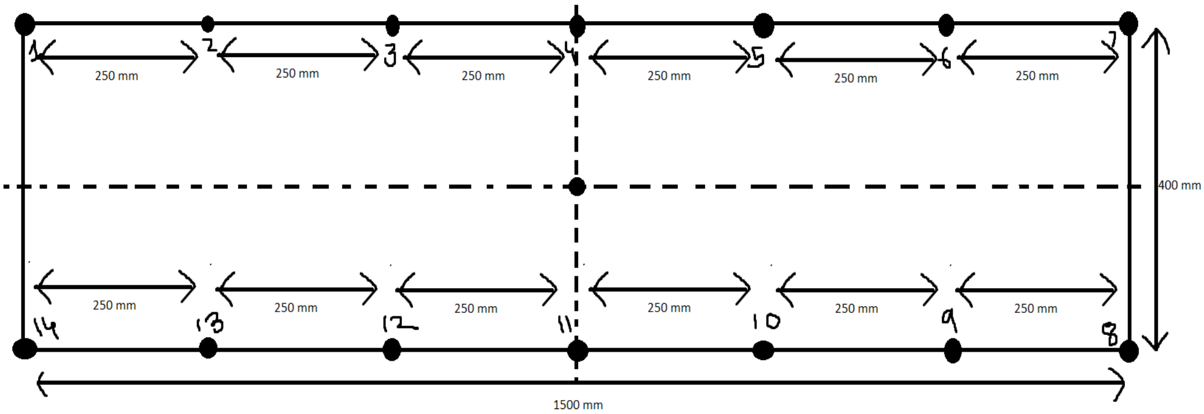


Figure 7: Wing Box - Final Iteration.

Table 8: Structural Idealization – Final Iteration.

Boom	Boom Area (mm²)	Y (mm)	I_{xx} (mm⁴)	σ_z (N/mm²)
1	1742	200	$69.68*10^6$	1147.36
2	1700	200	$68*10^6$	1147.36
3	1700	200	$68*10^6$	1147.36
4	1700	200	$68*10^6$	1147.36
5	1700	200	$68*10^6$	1147.36
6	1700	200	$68*10^6$	1147.36
7	1742	200	$69.68*10^6$	1147.36
8	1742	-200	$69.68*10^6$	-1147.36
9	1700	-200	$68*10^6$	-1147.36
10	1700	-200	$68*10^6$	-1147.36
11	1700	-200	$68*10^6$	-1147.36
12	1700	-200	$68*10^6$	-1147.36
13	1700	-200	$68*10^6$	-1147.36
14	1742	-200	$69.68*10^6$	-1147.36
ΔI_{xx} =			$958.72*10^6$	

The maximum direct stress calculated from the final iteration is 1147.36 MPa is tension and -1147.36 Mpa in compression. Comparing the maximum direct stress to the critical buckling stress of 1348 Mpa it can be said that the wing box will not fail under buckling. Therefore, under the ultimate loads from PLAA the wing box has a buckling safety factor of 1.17. Since the main criteria for this design task is for the wing box to be light weight, the weight for the wing box is required to be between 50-60 percent of the total wing structural weight. The table below tabulates the volume for each component. The weight for each component and the overall weight is calculated by multiplying the volume by the density of the selected material.

Table 9: Weight Calculation - Final Iteration.

Component	Volume for 1 Meter Section (m^3)	Density (Kg/m^3)	Weight for 1-meter wing box section (Kg)	Weight for the entire wingspan (15.05 m) (Kg)
Spars	0.008	2780	22.24	334.7
Panels	0.015	2780	41.7	627.6
Stringers	0.0063	2780	17.514	263.6
Total	0.0293	2780	81.45	1226

As seen in the above table the calculated weight for the wing box for the right wing is 1226 Kg, or 55.7% of the total wing structural mass. The total weight of the designed wing box is within the desired range discussed previously. This means that the designed wing box from this final iteration is an acceptable design with an optimal weight percentage which will not fail under panel buckling.

Stringer Shape and Dimension Determination

The designed wing box comprises of a curved panel sheet with uniformly spaced Z-section stringers along the panel and L shaped spar caps at the four corners joining the spars and panels. The Z-section stringer type were chosen as they provide high resistance under buckling stress and bending moments. Using the methodology developed for stiffener sizing in class, the dimensions for the z-section stringer are calculated as follows:

Table 10: Guidelines for Preliminary Sizing of Compressive Skin-Stiffener Panels.

t_a	$\frac{b_a}{t_a}$	$\frac{b_w}{t_w}$	$\frac{b_f}{t_f}$	$\frac{b_f}{b_w}$	$\frac{A_{st}}{A_{sk}}$
0.7t	<10	18-22	6-8	0.4	0.5

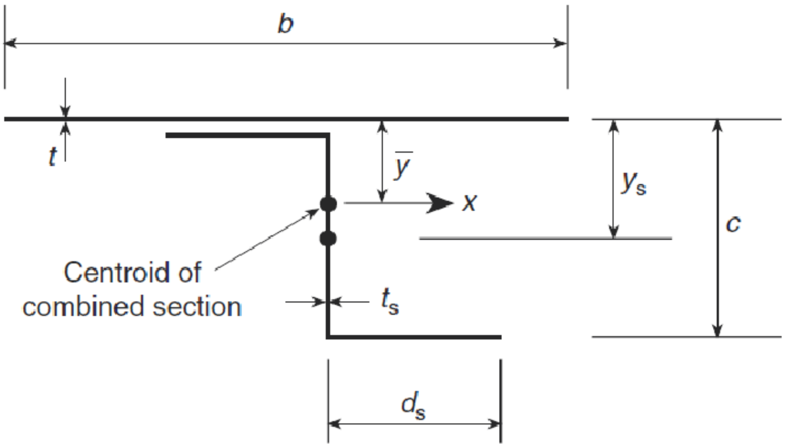


Figure 8: Common Z-Shape Stringer for Preliminary Sizing.

Based on the above figure, for the Z-Section stringer, $b_f = d_s$, $t_w = t_f = t_a = t_s$, and $b_w = c$. Therefore using $t_a = t_s = 0.7 * t = 3.5\ mm$

$$\frac{b_w}{t_w} = 20 = \frac{c}{t_s} \Rightarrow c = 20 * t_s = 70\ mm$$

$$\frac{b_f}{b_w} = 0.4 = \frac{d_s}{c} \Rightarrow d_s = 0.4 * c = 28\ mm$$

$$Area = t_s(2 * d_s + c) = 441\ mm^2$$

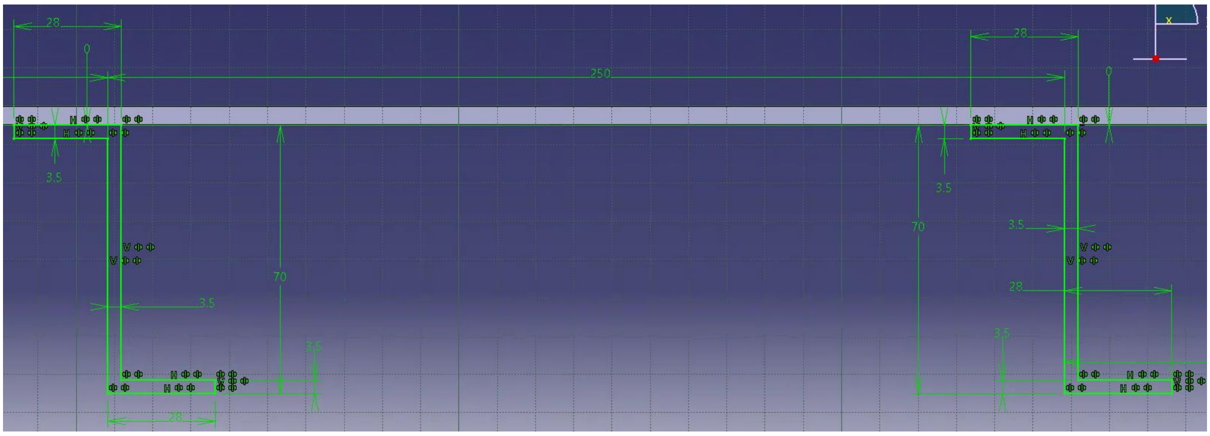


Figure 9: Z-Shape Stringer CAD Model.

As seen from the actual stringer area above, it can be confirmed that the initially guessed stringer area is a good approximation and the calculations for the direct stress carrying booms are valid.

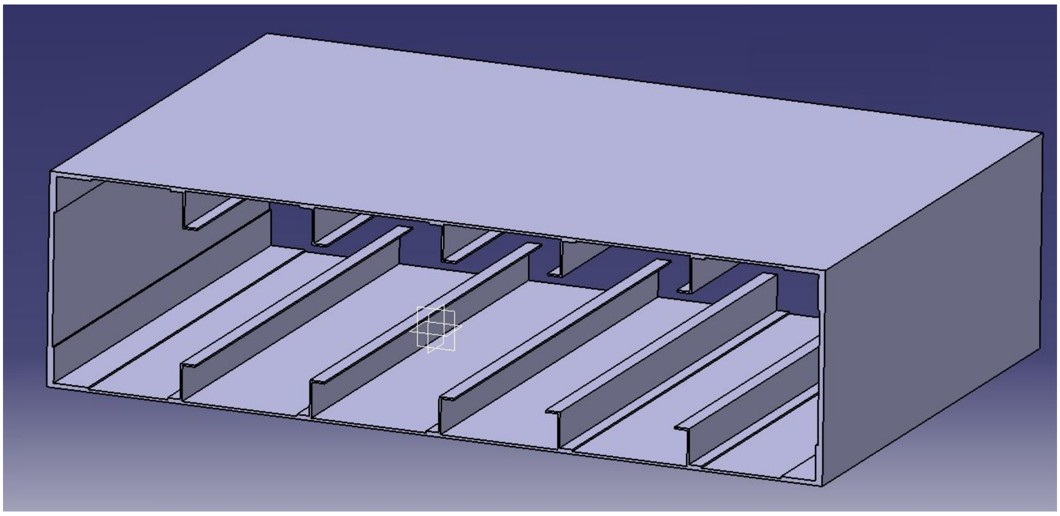


Figure 10: Wing box CAD Design.

The figure above shows the CAD model designed on Catia for the wing box section between the 6th and 7th meter outboard of the wing root. The CAD model is based on the final iteration and illustrates a wing box with 14 stringers – 10 Z-Shape stringers and 4 L-shaped stringers. The engineering drawing and the front view for this CAD design are included in the appendix section.

Wing Loading Distribution

The loading over the given wing was calculated using Schrenk's method which essentially states that the resultant load distribution is an arithmetic mean of a load distribution representing the actual planform shape and an elliptical distribution of the same span and area. The formulation of this method is as follows,

$$W_{y,eliptical} = \frac{4 * L}{\pi * b}\sqrt{1 - \left(\frac{2 * y}{b}\right)^2}$$

$$W_{y,taper} = \frac{2 * L}{(1 + \lambda)b}\left[1 + \frac{2 * y}{b}(\lambda - 1)\right]$$

$$W_y = \frac{W_{y,eliptical} + W_{y,taper}}{2}$$

The lift (L) used in the above equation was found from the ultimate limit loads in the PLAA section. The total weight distribution of the wing includes the weight of fuel(assumed constant throughout the span until y = 13m), weight of the wing structures(assumed constant throughout the wingspan), weight of the landing gear and engine and the air loads.

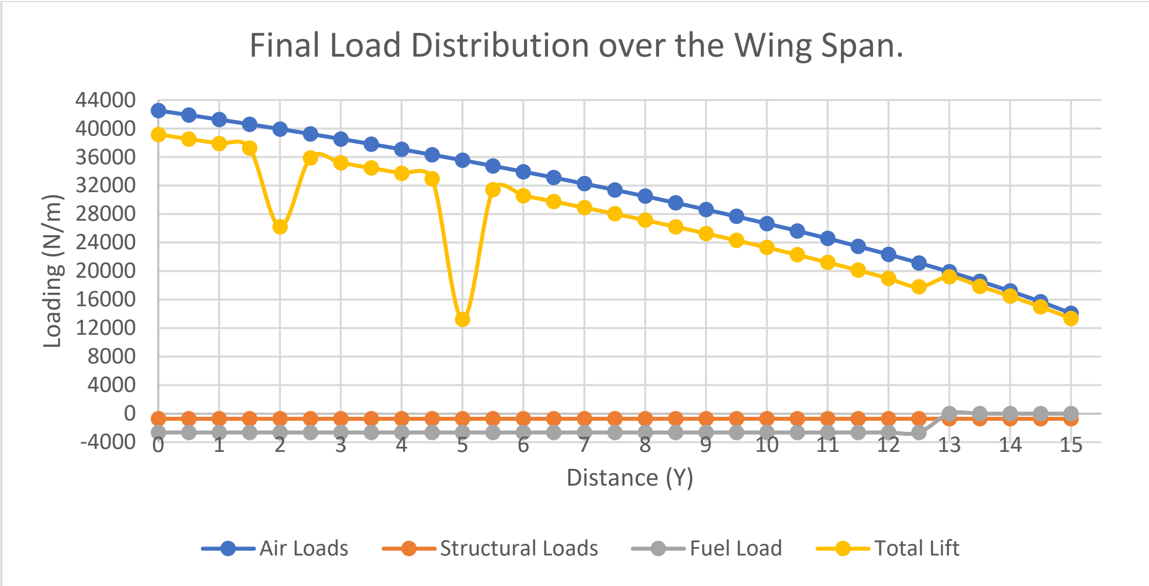


Figure 11: Detailed Load Distribution over the Wing.

The above figure illustrates a detailed load distribution of all the loadings applied on the wing over its span. The method of Schrenk's approximation was used to find an elliptical and tapered distribution of all loads. The yellow line represents the total lift distribution calculated from subtracting the structural, fuel, engine, and landing gear weights from the air loads. The two big dips for the total lift graph are accounting for the weight of the landing gear and the weight of the engine respectively.

After determining the lift loading, the moment and shear stress along the wingspan from tip to root are calculated using the following equation. For all the moment and shear calculations, the angle of attack was assumed 0.

$$V_{zi} = V_{zi+1} + N_T$$

$$M_{xi} = M_{xi+1} + N_T * \frac{y_{i+1} - Y_i}{2} + V_{zi+1} * (y_{i+1} - Y_i)$$

$$where, N_T = N_i - N_{fuel} - N_{structure} - N_{LG} - N_{engine}$$

All the calculations for the weight distribution figure and for the shear and moment distribution figure are tabulated in table 12, 13 which are included in the appendix section.

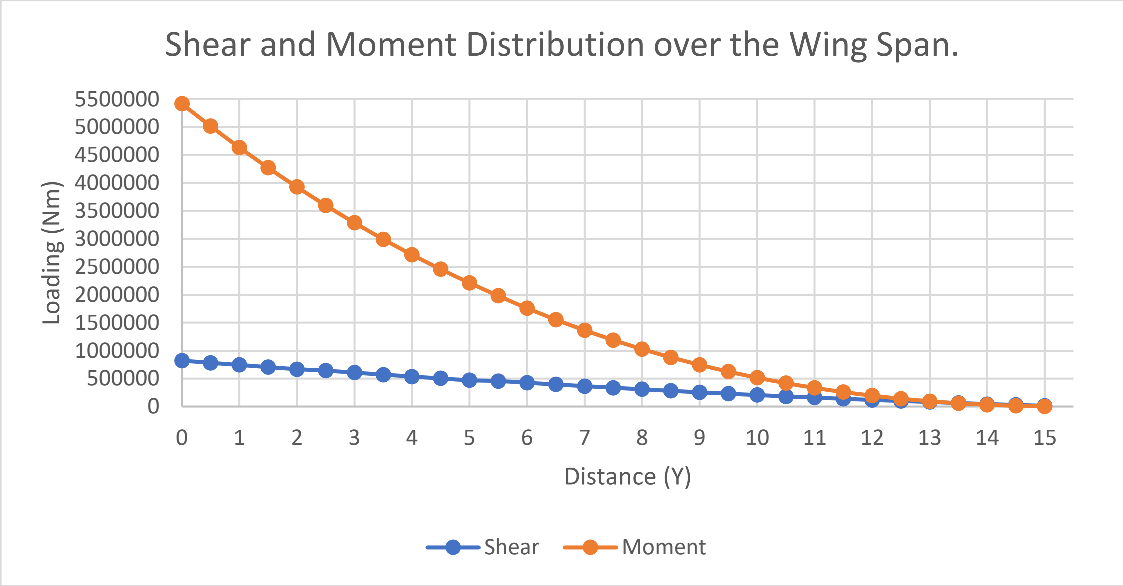


Figure 12: Bending Moment and Shear Stress Distribution over the Wing.

Economic Impact

This section will discuss the economic impact for the material selection of the wing box. The table below tabulates the materials considered for the wing box design. It compares the weight and cost of manufacturing for the wing box based on materials, such as, aluminium alloy, titanium alloy and steel alloy. Minimizing the manufacturing and production cost for any component while selecting a material is an important aspect of the design analysis. In addition to the manufacturing costs, labour and insurance costs should also be considered when selecting an optimal material.

Table 11: Material Weight and Cost Comparison for the Wing Box.

Material	**Density (Kg/m^3)**	**Weight of the Wing Box (Kg)**	**Cost($/Kg)**	**Total cost ($)**
Aluminium Alloy 7010 T 736	2780	1226	1.6	1962
Titanium AMS 4967 TA13	4420	1950	25	48,750
Steel BS S99	7830	3453	0.9	3108

For this design task, an aluminium alloy was chosen as an optimal material based on its mechanical properties. As seen in the table above, even though steel has the lowest of the costs per kg, the density of steel is the highest thus giving high weights. Titanium having the best mechanical properties amongst the three has very high costs per unit kg, which is why aluminium alloy was considered as a balanced material with optimal cost and weight.

Environmental Impact

This section discusses and compares the environmental aspects of the materials considered for the wing box design. The production and manufacturing of the materials both have an impact on the environment. Considering the materials related to the design of this wing box, steel production and manufacturing has number of impacts on the environment including air emissions, wastewater contaminants, hazardous wastes, and solid wastes. Titanium alloy is one of the greenest materials used in the industry as it can easily be recycled due to its high resistance to corrosion. A primary disadvantage of using titanium is its high reactivity and special care is needed during different stages of production. Environmental effects of aluminium are mainly a result of acidic precipitation largely affecting the aquatic and terrestrial ecosystems. Since aluminium ore is so stable, large amounts of electricity, water and other resources are needed to produce aluminium.

Conclusion

For this project, the two design tasks were successfully completed. Each task used different methods of analysis and design. Knowledge and understandings from a wide range of subjects were covered throughout the course of this project. For the first design task, the landing gear tire type and size selection was completed using the provided charts and resources during the lecture. The preliminary sizing of the MLG oleo stut and the sizing of the outer tube was completed using the FAR empirical equations and resources. A material selection analysis was completed for this task, to choose an optimal material with low density and high strength. The materials were also compared with their economic and environmental impacts. For the second design task, a 1-meter wing box section was designed that could resist the ultimate loads at PLAA conditions. This task was completed using the method of structural idealization with an iterative approach. The goal was to select a set of geometric properties for the wing box, such that the resulting overall weight lies between 50 to 60 % of the wing structural mass and the system doesn't fail under panel buckling. The first and the final iterations are included and discussed briefly in the report. The shape of the stringer was decided based on the highest resistance to bending moment and their dimensions were calculated from the empirical equations shared during the lecture. At the end a detailed shear stress, bending moment and weight analysis were conducted for the given aircraft. A CAD model was developed in CATIA V5 for the two design tasks to illustrate the designed parameters.

Appendix

Size	Speed mph	Max load, lb	Press psi	Max width, in	Max diam, in	Rolling radius	Wheel diam	Number of plies
				Type III				
5.00-4	120	1,200	55	5.05	13.25	5.2	4.0	6
5.00-4	120	2,200	95	5.05	13.25	5.2	4.0	12
7.00-8	120	2,400	46	7.30	20.85	8.3	8.0	6
8.50-10	120	3,250	41	9.05	26.30	10.4	10.0	6
8.50-10	120	4,400	55	8.70	25.65	10.2	10.0	8
9.50-16	160	9,250	90	9.70	33.35	13.9	16.0	10
12.50-16	160	12,800	75	12.75	38.45	15.6	16.0	12
20.00-20	174kt	46,500	125	20.10	56.00	22.1	20.0	26
				Type VII				
16 × 4.4	210	1,100	55	4.45	16.00	6.9	8.0	4
18 × 4.4	174kt	2,100	100	4.45	17.90	7.9	10.0	6
18 × 4.4	217kt	4,350	225	4.45	17.90	7.9	10.0	12
24 × 5.5	174kt	11,500	355	5.75	24.15	10.6	14.0	16
30 × 7.7	230	16,500	270	7.85	29.40	12.7	16.0	18
36 × 11	217kt	26,000	235	11.50	35.10	14.7	16.0	24
40 × 14	174kt	33,500	200	14.00	39.80	16.5	16.0	28
46 × 16	225	48,000	245	16.00	45.25	19.0	20.0	32
50 × 18	225	41,770	155	17.50	49.50	20.4	20.0	26
				Three-Part Name				
18 × 4.25-10	210	2,300	100	4.70	18.25	7.9	10.0	6
21 × 7.25-10	210	5,150	135	7.20	21.25	9.0	10.0	10
28 × 9.00-12	156kt	16,650	235	8.85	27.60	11.6	12.0	22
37 × 14.0-14	225	25,000	160	14.0	37.0	15.1	14.0	24
47 × 18-18	195kt	43,700	175	17.9	46.9	19.2	18.0	30
52 × 20.5-23	235	63,700	195	20.5	52.0	21.3	23.0	30

Figure 13: Tire Selection.

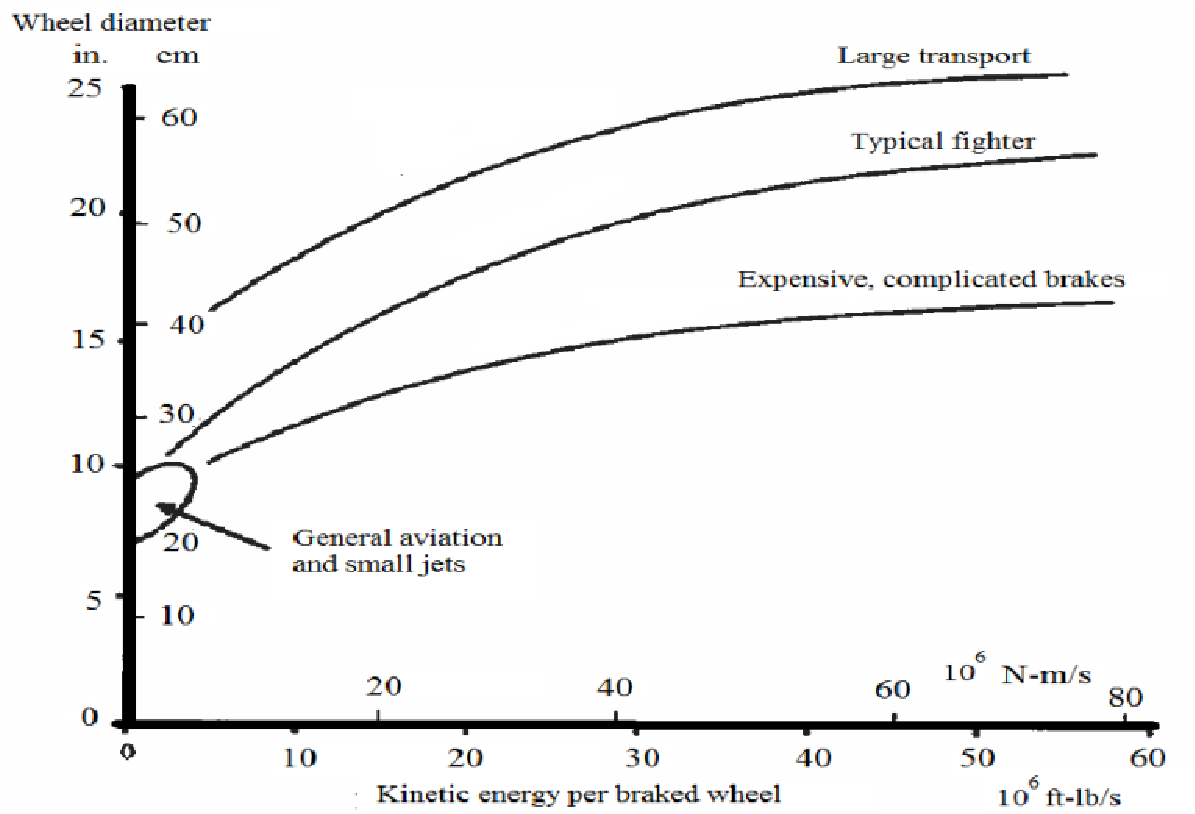


Figure 14: Kinetic Energy Vs Wheel Diameter.

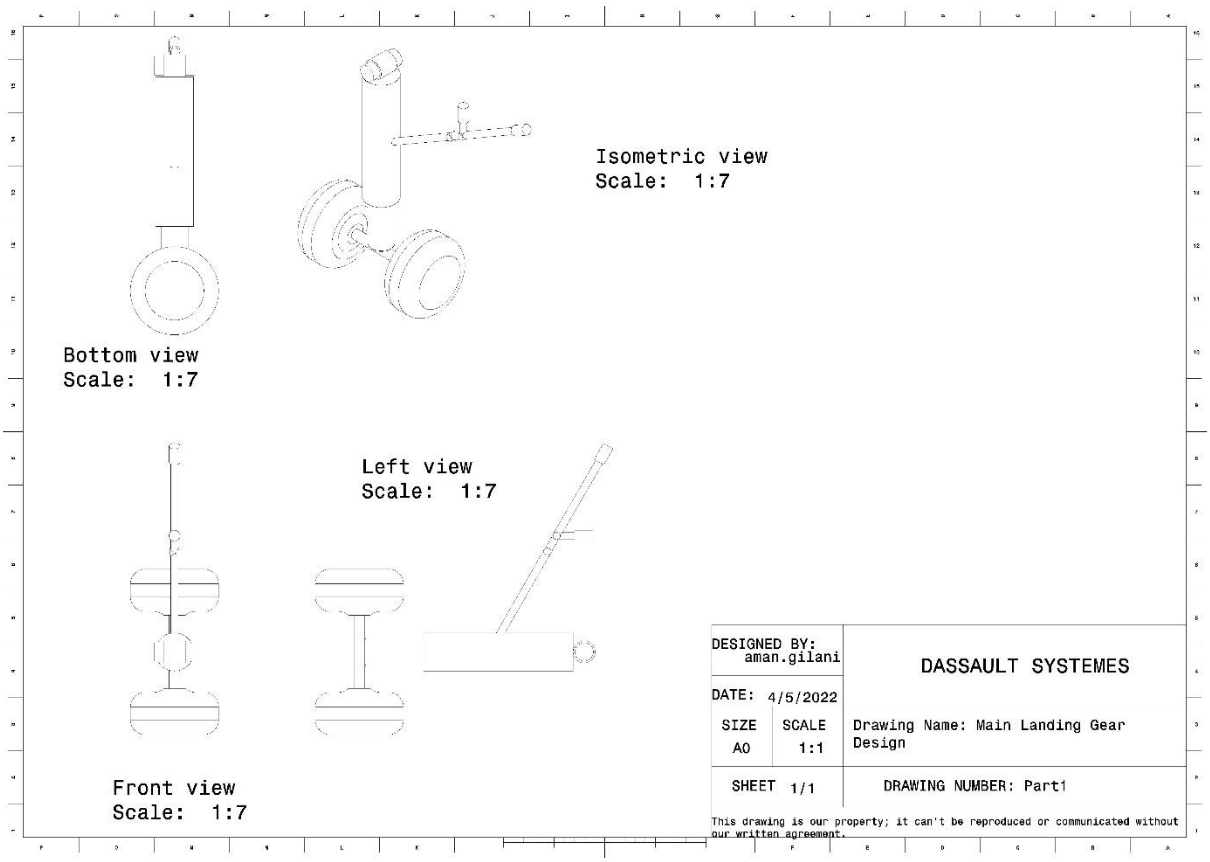


Figure 15: Engineering Drawing - Landing Gear Design.

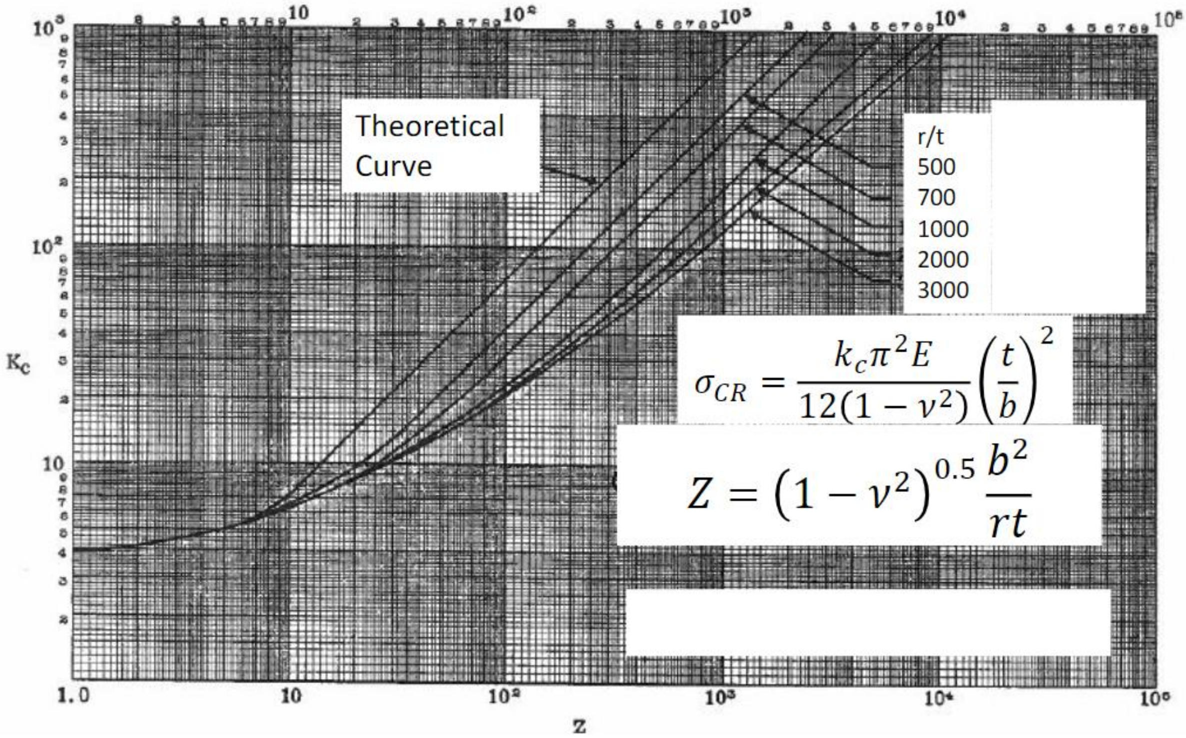


Figure 16: Buckling Constant Chart for Curved Panels.

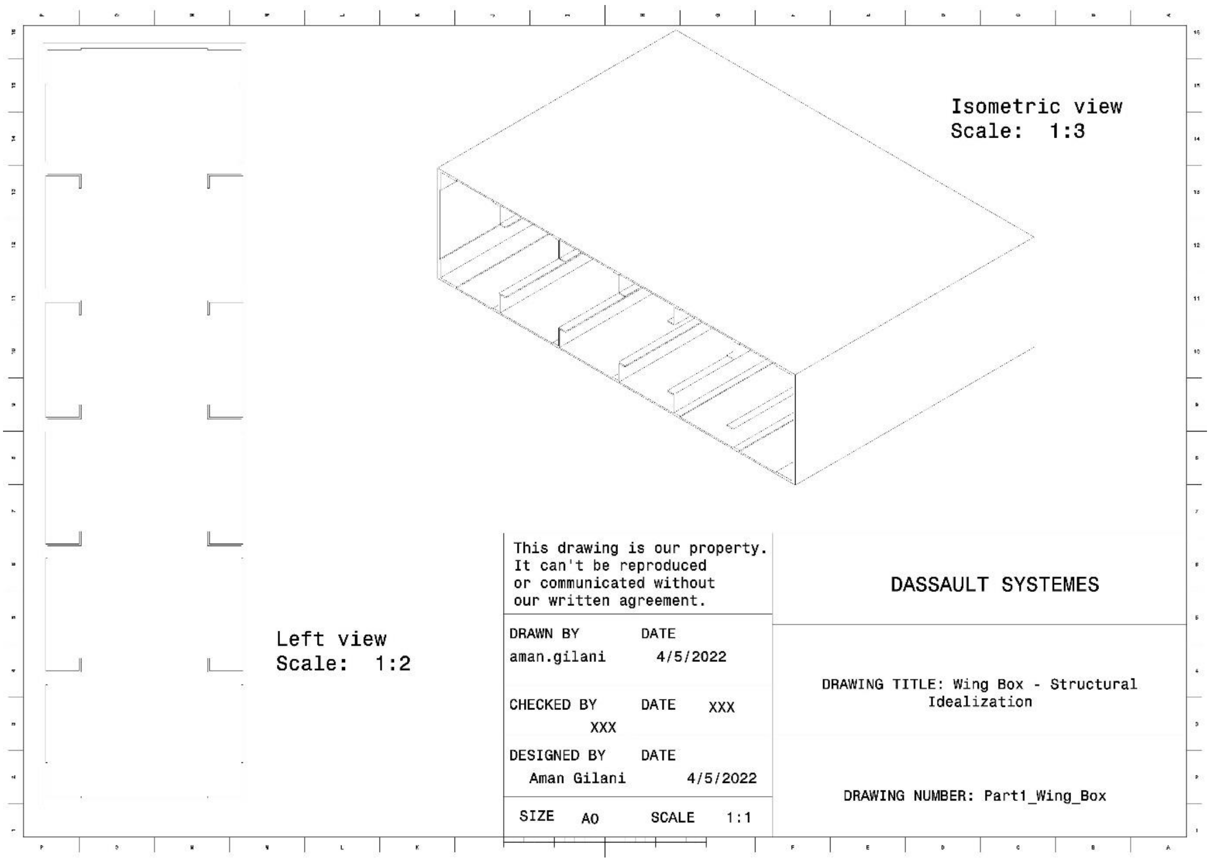


Figure 17: Engineering Drawing - Wing Box Design.

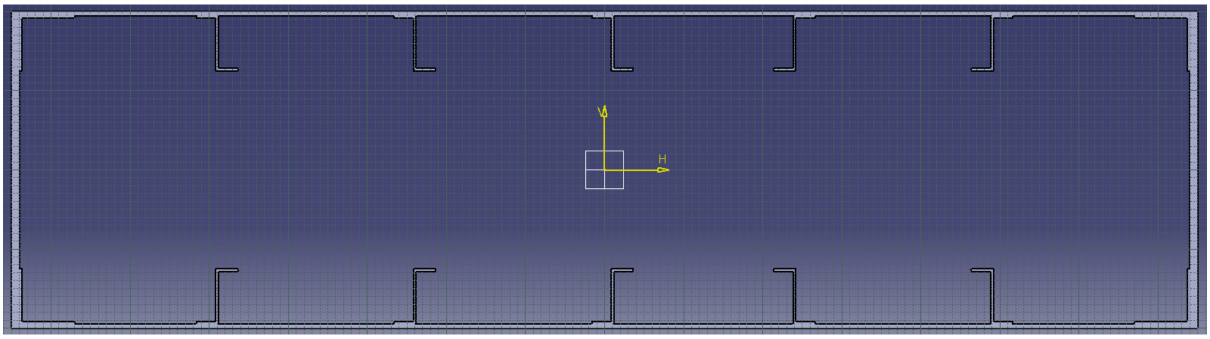


Figure 18: Wing Box Front View.

Boom Calculations – First Iteration

$$B_1 = B_6 = B_7 = B_{12} = 450 + \frac{10 * 400}{6}\left(2 + \frac{-200}{200}\right) + \frac{8 * 300}{6}\left(2 + \frac{200}{200}\right) = 2317\ mm^2$$

$$B_2 = B_5 = B_8 = B_{11} = 450 + \frac{8 * 300}{6}\left(2 + \frac{200}{200}\right) + \frac{8 * 300}{6}\left(2 + \frac{200}{200}\right) = 2850\ mm^2$$

$$B_3 = B_4 = B_9 = B_{10} = 450 + \frac{8 * 300}{6}\left(2 + \frac{200}{200}\right) + \frac{8 * 300}{6}\left(2 + \frac{200}{200}\right) = 2850\ mm^2$$

Boom Calculations – First Iteration

$$B_1 = B_7 = B_8 = B_{14} = 450 + \frac{10 * 400}{6}\left(2 + \frac{-200}{200}\right) + \frac{5 * 250}{6}\left(2 + \frac{200}{200}\right) = 1742\ mm^2$$

$$B_2 = B_6 = B_9 = B_{13} = 450 + \frac{5 * 250}{6}\left(2 + \frac{200}{200}\right) + \frac{5 * 250}{6}\left(2 + \frac{200}{200}\right) = 1700\ mm^2$$

$$B_3 = B_5 = B_{10} = B_{12} = 450 + \frac{5 * 250}{6}\left(2 + \frac{200}{200}\right) + \frac{5 * 250}{6}\left(2 + \frac{200}{200}\right) = 1700\ mm^2$$

$$B_4 = B_{11} = 450 + \frac{5 * 250}{6}\left(2 + \frac{200}{200}\right) + \frac{5 * 250}{6}\left(2 + \frac{200}{200}\right) = 1700\ mm^2$$

Wing Weight Distribution Excel Tables

Table 12: Wing Weight and Load Distribution.

y(m)	Wy(N/m)	Delta Air load	Structures Load	Structures Load Delta	Fuel Load	Fuel Load Delta	Engine Load Delta	Landing Gear Delta
0	85664.10641	42523.83442	-1434	-717	-5282.308	-2641.15385	0	0
0.5	84431.23129	41899.51308	-1434	-717	-5282.308	-2641.15385	0	0
1	83166.82104	41259.39333	-1434	-717	-5282.308	-2641.15385	0	0
1.5	81870.75229	40603.39248	-1434	-717	-5282.308	-2641.15385	0	0
2	80542.81762	39931.38511	-1434	-717	-5282.308	-2641.15385	0	-10359
2.5	79182.72282	39243.20143	-1434	-717	-5282.308	-2641.15385	0	0
3	77790.0829	38538.62492	-1434	-717	-5282.308	-2641.15385	0	0
3.5	76364.41678	37817.3893	-1434	-717	-5282.308	-2641.15385	0	0
4	74905.14041	37079.17466	-1434	-717	-5282.308	-2641.15385	0	0
4.5	73411.55823	36323.60266	-1434	-717	-5282.308	-2641.15385	0	0
5	71882.8524	35550.23053	-1434	-717	-5282.308	-2641.15385	-18992.2	0
5.5	70318.0697	34758.54376	-1434	-717	-5282.308	-2641.15385	0	0
6	68716.10533	33947.94708	-1434	-717	-5282.308	-2641.15385	0	0
6.5	67075.683	33117.75336	-1434	-717	-5282.308	-2641.15385	0	0
7	65395.33045	32267.16986	-1434	-717	-5282.308	-2641.15385	0	0
7.5	63673.34898	31395.28113	-1434	-717	-5282.308	-2641.15385	0	0
8	61907.77555	30501.02762	-1434	-717	-5282.308	-2641.15385	0	0
8.5	60096.33495	29583.17849	-1434	-717	-5282.308	-2641.15385	0	0
9	58236.37903	28640.29686	-1434	-717	-5282.308	-2641.15385	0	0
9.5	56324.80842	27670.69468	-1434	-717	-5282.308	-2641.15385	0	0
10	54357.9703	26672.37326	-1434	-717	-5282.308	-2641.15385	0	0
10.5	52331.52274	25642.94348	-1434	-717	-5282.308	-2641.15385	0	0
11	50240.2512	24579.51654	-1434	-717	-5282.308	-2641.15385	0	0
11.5	48077.81496	23478.55063	-1434	-717	-5282.308	-2641.15385	0	0
12	45836.38757	22335.62982	-1434	-717	-5282.308	-2641.15385	0	0
12.5	43506.13171	21145.13391	-1434	-717	-5282.308	-2641.15385	0	0
13	41074.40392	19899.7252	-1434	-717	-5282.308	0	0	0
13.5	38524.49688	18589.509	-1434	-717	0	0	0	0
14	35833.53914	17200.56888	-1434	-717	0	0	0	0
14.5	32968.73639	15712.18278	-1434	-717	0	0	0	0
15	29879.99471	14090.85916	-1434	-717	0	0	0	0
15.5	26483.44192	0	-1434	0	0	0	0	0

Table 13: Total Lift, Shear and Moment Distribution over the Wing Span.

Total Lift	Shear	Moment
39165.681	821750.1	5418360
38541.359	782584.4	5017276
37901.239	744043.1	4635619
37245.239	706141.8	4273073
26213.871	668896.6	3929314
35885.048	642682.7	3601419
35180.471	606797.7	3289049
34459.235	571617.2	2994445
33721.021	537158	2717251
32965.449	503436.9	2457102
13199.917	470471.5	2213625
31400.39	457271.6	1981690
30589.793	425871.2	1760904
29759.6	395281.4	1555616
28909.016	365521.8	1365415
28037.127	336612.8	1189881
27142.874	308575.7	1028584
26225.025	281432.8	881082
25282.143	255207.8	746921.9
24312.541	229925.6	625638.5
23314.219	205613.1	516753.9
22284.79	182298.9	419775.9
21221.363	160014.1	334197.7
20120.397	138792.7	259496
18977.476	118672.3	195129.7
17786.98	99694.83	140537.9
19182.725	81907.85	95137.28
17872.509	62725.12	58979.04
16483.569	44852.61	32084.6
14995.183	28369.04	13779.19
13373.859	13373.86	3343.465
0	0	0

References

1. Z. Fawaz, Class Lecture, Topic: “Design Project-W2021.” AER621, Department of Aerospace Engineering, Ryerson University, Toronto, 2021.
2. “Landing Gear Shock Absorber Design Concordia University mech,” SlideToDoc.com. [Online]. Available: https://slidetodoc.com/landing-gear-shock-absorber-design-concordia-university-mech/. [Accessed: 01-Apr-2022].
3. Norman S. Currey, Aircraft Landing Gear Design: Principles and Practices, AIAA, 1
4. “Landing Gear Shock Absorber - 123seminarsonly.com.” [Online]. Available: http://www.123seminarsonly.com/Seminar-Reports/2013-02/94681148-Landing-Gear-Shock-Absorber.pdf. [Accessed: 01-Apr-2022].
5. MatWeb, “Material Property Data”, Data Sheets
6. “Airport Planning Manual - Embraer.” [Online]. Available: https://www.embraercommercialaviation.com/wp-content/uploads/2017/02/APM_E195.pdf. [Accessed: 01-Apr-2022].
7. Z. Fawaz, Class Lecture, Topic: “Design Project Details.” AER621, Department of Aerospace Engineering, Ryerson University, Toronto, 2021.
8. E. Worrell, “The role of material efficiency in environmental stewardship,” Annual Reviews. [Online]. Available: https://www.annualreviews.org/doi/10.1146/annurev-environ-110615085737#:~:text=Materials%20production%20requires%20a%20large,end%2Dof%2Dlife%20disposal. [Accessed: 01-Apr-2022].
9. “Steel Production & Environmental Impact,” Greenspec. [Online]. Available: https://www.greenspec.co.uk/building-design/steel-products-and-environmental-impact/. [Accessed: 01-Apr-2022].
10. “The environmental impact of aluminium,” Reynaers Aluminium, 24-Sep-2020. [Online]. Available: https://www.reynaers.com/en/environmental-impact-aluminium. [Accessed: 01-Apr-2022].
11. “Titanium is environmentally friendly,” Titanium is environmentally friendly | Natori Manufacturing Co., Ltd. [Online]. Available: https://www.natori-mnf.co.jp/english/titanenvironment/index.html. [Accessed: 01-Apr-2022].