STRESS AND STRENGTH ANALYSES OF ULTRA-LIGHT AIRCRAFT FUSELAGE

by

MUHAMMAD ISA BIN SAIRONI

Thesis submitted in fulfilment of the requirements
for the Bachelor Degree of Engineering (Honours) (Aerospace Engineering)

27 June 2021

ENDORSEMENT

I, Muhammad Isa Bin Saironi, hereby certify that all supervisor and examiner adjustments and comments have been taken into account.

(Signature of Student)

Name: MUHAMMAD ISA BIN SAIRONI
Date: 27/06/2021

(Signature of Supervisor)

Name: DR. A HALIM KADARMAN
Date: 27/06/2021

(Signature of Examiner)

Name: IR. DR. HUSSIN MAMAT
Date: 27/06/2021

DECLARATION

Apart from where it is mentioned, this thesis is the outcome of my examinations. Different sources are distinguished by explicit references. This work has not lately been recognized in substance for any degree while also being submitted as a potential for a different degree.

(Signature of Student)

Name: MUHAMMAD ISA BIN SAIRONI
Date: 27/06/2021

ACKNOWLEDGEMENTS

The completion and fulfillment of this thesis necessitated a significant lot of assistance and help from a variety of people. Above all, I want to express my gratitude to my project supervisor, PE. Dr. A Halim Bin Kadarman, who has supported me throughout this inquiry with his continual tenacity, support, and vast expertise. This thesis would not have been completed or produced without his persistent assistance, crucial direction, thinking, and expertise in the structural system in the aerospace engineering sector. I thank him for allowing me to work on my Final Year Project under his supervision.

In addition, I would like to thank the School of Aerospace Engineering's staff member and assistant engineer for their assistance in establishing the project's fuselage model. Furthermore, I would want to express my gratitude to Zahidah Binti Hassim, my mother, for her unwavering emotional and motivational support to finish this Final Year Project. Without her, I would not have been able to solve the challenges that arose throughout the project's completion.

Finally, I would want to express my gratitude to my entire family, my classmates, and especially my course mates for their unrelenting physical and mental support.

Thank you all.

STRESS AND STRENGTH ANALYSES OF ULTRA-LIGHT AIRCRAFT FUSELAGE

ABSTRACT

Ultra-light aircraft is a special kind of low-speed general aircraft. It is low weighted and small-sized however can depart and land without changing the structure of any parts. This thesis is the case study on the stress and strength analyses of an ultra-light aircraft fuselage about its performance dependent on a guideline from the Federal Aviation Administration under Federal Aviation Regulation (FAR) that has been provided to the operators of ultra-lights in the United States. This archive additionally gave a detailed method in dissecting the stress and strength of the fuselage of an ultra-light aircraft and the point-by-point design of the fuselage. The fuselage structure is intended to invigorate better weight proportions as the focal part of the body of an ultra-light aircraft. This project utilizes composite material as the chose material. The determination of material readies to withstand the load applied to the fuselage structure. The simulation analysis is finished by utilizing one of the Computer-aided Design (CAD) which is SolidWorks 2020 ×64 Edition software. The examination is performed to see whether the structure of the fuselage can uphold the applied load or neglect to convey the required base on the few kinds of simulation testing. Other than the simulation analysis, the analytical methodology is additionally determined to look at and check the structure of the fuselage. Mostly, maximum stress and fatigue failure strength is calculated on two critical points on the ultra-light aircraft fuselage which is on the tail dragger of the ultra-light aircraft fuselage and at the bracket connector between the landing gear and mainframe fuselage. The maximum equivalent von Mises stress calculated on the fuselage model is 550.80 MPa whereas the maximum principal stress is 574.03 MPa happen in case study 2. Besides, the maximum compressive buckling stress calculated on the fuselage model is 709.77 MPa happen in case study 2. In finishing this project, the material and equipment utilized in fabricating this fuselage are accessible in the Aerospace Structure Lab, School of Aerospace Engineering, Universiti Sains Malaysia, Engineering Campus.

ANALISIS TEKANAN DAN KEKUATAN STRUKTUR BAGI BADAN PESAWAT ULTRA RINGAN

ABSTRAK

Pesawat ultra ringan adalah jenis khas pesawat am berkelajuan rendah. Beratnya rendah dan bersaiz kecil namun dapat berlepas dan mendarat tanpa mengubah struktur bahagian mana pun. Tesis ini adalah kajian kes mengenai analisis tekanan dan kekuatan pesawat ultra ringan mengenai prestasinya bergantung pada garis panduan dari Pentadbiran Penerbangan Persekutuan di bawah Peraturan Penerbangan Persekutuan (FAR) yang telah diberikan kepada pengendali pesawat ultra ringan di Amerika Syarikat. Arkib ini juga memberikan kaedah terperinci dalam membedah tekanan dan kekuatan pesawat ultra ringan dan reka bentuk titik demi titik pesawat. Struktur badan pesawat bertujuan untuk meningkatkan perkadaran berat badan yang lebih baik sebagai bahagian fokus badan pesawat ringan. Projek ini menggunakan bahan komposit sebagai bahan pilihan. Penentuan kesediaan bahan untuk menahan beban yang dikenakan pada struktur pesawat. Analisis simulasi selesai dengan menggunakan salah satu Reka Bentuk Berbantu Komputer (CAD) yang merupakan perisian SolidWorks 2020 × 64 Edition. Pemeriksaan dilakukan untuk melihat apakah struktur badan pesawat dapat menahan beban yang diaplikasikan atau diabaikan untuk menyampaikan seperti yang diperlukan berdasarkan beberapa jenis ujian simulasi. Selain daripada analisis simulasi, metodologi analisis juga ditentukan untuk melihat dan memeriksa struktur badan pesawat. Kebanyakannya, kekuatan kegagalan tekanan dan keletihan maksimum dikira pada dua titik kritikal pada pesawat ultra-ringan yang berada pada penyeret ekor pesawat ringan dan pada penyambung kurungan antara gear pendaratan dan badan pesawat utama. Tekanan von Mises setara maksimum yang dikira pada model pesawat adalah 550.80 MPa sedangkan tegasan utama maksimum ialah 574.03 MPa berlaku dalam kajian kes 2. Selain itu, tekanan tegangan mampatan maksimum yang dikira pada model pesawat adalah 709.77 MPa berlaku dalam kajian kes 2. Dalam menyelesaikan projek ini, bahan dan peralatan yang digunakan dalam pembuatan pesawat ini dapat diakses di Makmal Struktur Aeroangkasa, Pusat Pengajian Kejuruteraan Aeroangkasa, Universiti Sains Malaysia, Kampus Kejuruteraan.

TABLE OF CONTENTS

LIST OF FIGURES

LIST OF TABLES

LIST OF ABBREVIATIONS

FAA	Federal Aviation Administration
FAR	Federal Aviation Regulation
CAD	Computer-aided Design
MTOW	Maximum Take-Off Weight
IAS	Indicated Airspeed
POH	Pilot’s Operating Handbook
AFM	Airplane Flight Manual
FEA	Finite Element Analysis
FEM	Finite Element Method
FS	Safety Factor
JAR VLA	Joint Aviation Requirements – Very Light Airplane
EASA CS VLA	European Aviation Safety Agency Certification Specification for Very Light Aircraft

LIST OF SYMBOLS

$\overline{w}$	Average limit control surface loading
n_1	Limit maneuver load factor
$\frac{W}{S}$	Wing loading
σ_{CR}	Compressive buckling stress
τ_{CR}	Shear buckling stress
k_σ	Compressive buckling coefficient
k_τ	Shear buckling coefficient
E	Young Modulus
v	Poisson’s ratio
t	thickness
b	width
FS	Safety factor
S_{al}	Allowable strength
σ_{ap}	Maximum stress

CHAPTER 1

INTRODUCTION

1.1 Research Background

Microlight aircraft are ultra-light aircraft with different interpretations, weight, and speed restrictions from nation to nation (Ultralight and light aircrafts, 2021). In recent years, Europe has been enthralled by the growing popularity of ultra-light aircraft in general. Only within recent years has the maximum fly speed of ultra-light aircraft been increased due to the deployment of more powerful engines and improved aerodynamics. It is hardly surprising that a top speed of around 280 km/h has been achieved. Because the weight constraints are so stringent, it is critical to look for ways to improve structural design, safety, flying characteristics, aeroelastic behaviour, and system authenticity without incurring additional expenditures (Green, J. 2002). CGS Hawk Arrow II, which is manufactured by CGS Hawk Aviation, is one of the ultra-light aircraft available on the market. CGS Hawk Arrow II has attained a maximum rate of climb of roughly 6.096 m/s as shown in Figure 1.1 below.

Figure 1.1: CGS Hawk Arrow II Ultra-light Aircraft (CGS Hawk Aviation, 2020).

1.2 Control System

Push-pull cables, sheathed cables, and aluminium or steel tubes are two options for the control system for light sport and ultra-light aircraft. Aluminum is chosen over steel because it is lighter. This is especially true when the cost is not an issue. These two methods are combined and integrated into a hybrid control system that will use push-pull cables to move the elevator (Christopher King, David Roncin, Nathan Swanger, 2007).

1.3 Problem Statement

The Malaysian aviation sector is now growing at a glacial pace due to the high cost of making aircraft and the limited availability of manufacturing technologies. These constraints are the root of investors' reluctance to invest in the aeronautics industry. However, in other nations, particularly Europe, a large number of investors and small-scale manufacturers are working hard to meet the demand for ultra-light aircraft. To build an aircraft, a significant volume of money must be committed, as well as a significant amount of time (Airplane Flying Handbook, published by FAA). Furthermore, a lack of expertise and limited resources are the two most significant factors limiting Malaysia’s aircraft production.

Dr. A. Halim Kadaraman of Universiti Sains Malaysia's School of Aerospace Engineering designed an ultra-light aircraft project to develop and build a one-man ultra-light aircraft. Because it is a new design, it must be thoroughly examined to ensure that it complies with airworthiness regulations – (Hu Jizhong, Wu Dawei, Wu Zheng, 2011).

1.4 Objectives of Project

The following objectives serve as the foundation for the project effort reported in this thesis:

1. To design ultra-light aircraft fuselage mainframe using CAD software.
2. To perform the stresses and strengths analyses on the fuselage structure model.
3. To fabricate the fuselage if the condition permits.

1.5 Scope of Works

The development of the new fuselage structure for an ultralight aircraft was a particularly challenging project that took a large number of cycles to complete. As a result, the project's scope of work is as follows:

1. Research the concept of fuselage structure configuration for the ultra-light aircraft's flight control system.
2. Research the design requirements for the ultra-light aircraft's fuselage structure.
3. Design and perform stress and strength analyses and fabricate the fuselage structure using composite material if condition permits.

1.6 Thesis Layout

This thesis has 5 main chapters that cover all aspects of the project. The ultra-light aircraft is introduced in Chapter 1. It depicts the foundation of the ultra-light aircraft project. This part discusses the ultra-light aircraft's actual design and application. The fuselage structure design has been defined for the time being. Then, in this chapter, the problem statement, project objectives, and project scope of work and study are described.

Chapter 2 reviews all of the literature and theoretical background associated with this study and project. This chapter focuses on the discovery and investigation of a journal that was prepared by others and is directly related to the fuselage structure project that will be produced. The purpose of this chapter is to acquire project-related knowledge and information. This segment clarified the hypothesis as well as the critical understanding of the project's components.

The approach and procedure used in this project are detailed explained in Chapter 3. Before delving into the details, this chapter began with a broad overview of the project flow. The methodology depicts an effective strategy, a successful activity, simulation, and theoretical analysis.

The results and discussion in Chapter 4 are based on the investigation of the fuselage structure. This chapter presented the study findings and conclusions, as well as explanations for particular outcomes obtained through simulation and analytical analysis, which were supported by the methodology. In order to arrive at a valid conclusion, the connection between simulation and analytical data was investigated in this chapter.

Chapter 5 is the research project's conclusion and suggestion. The whole project's detailed breakdown has been completed, which mostly represents the interaction in developing this project. A few ideas and future work for the future advancement of this ultra-light aircraft project are integrated towards the end of this thesis.

CHAPTER 2

LITERATURE REVIEW

Ultra-light aircraft are really a popular and rapidly growing kind of diversion all over the world. More than 20 000 ultralights are estimated to be in service in the United States and Canada alone. When compared to traditional flying, a portion of the sport's popularity can be attributed to its comparatively low cost. Approximately 30 different ultralight versions range in price from $5000.00 to $10,000.00 (The Baltimore Sun, 1998). Most ultralights may be constructed by their owners, which contributes to the appeal of owning and flying ultralights while maintaining their cost.

2.1 Ultra-light Aircraft

The Canadian Aviation Regulations divide ultra-light aircraft into two categories: basic ultra-light aircraft and advanced ultra-light aircraft as shown in Figures 2.1 and 2.2 (Transport Canada, 1996). A basic ultra-light aircraft is any aircraft that fulfils one of the three definitions stated below (Transport Canada, 2007):

1. A single-seat aircraft with a maximum takeoff weight (MTOW) of 165 kg. The wing area (S) can be no less than 10 m^2 and no more than the MTOW minus 15 divided by 10.
2. A two-seat instructional aircraft with two MTOW of 195 kg. The wing area can be at least 10 m^2 and the wing loading ($\frac{MTOW}{S}$) must not exceed 25 kg/m^2.
3. An aircraft with up to two seats having a MTOW of 544 kg and a landing stall speed no greater than 72 km/h (39 knots) indicated airspeed (IAS). The uncorrected value from the airspeed indicator measured by a pitot-static probe is referred to as the indicated airspeed. Temperature, density, and instrumentation error are not taken into account while calculating the IAS.

Figure 2.1: Basic Ultra-light Aeroplane (Microlight Aviation, 2008).

Figure 2.2: Advanced Ultra-light Aeroplane (Golden Horseshoe Advanced Ultralights, 2020).

2.2 Weight Changes

According to Weight and Balance Handbook, (2016), one critical preflight application is the distribution of the load in the aircraft. Loading the aircraft so the gross weight is less than the maximum acceptable is not sufficient. This weight must be dispersed to control the center gravity within the limits stated in the Pilot’s Operating Handbook (POH) or Airplane Flight Manual (AFM).

2.3 Equilibrium and Stability

The location of an aircraft's Center Gravity (CG) is indicated via balance control. This is a matter of constitutional importance for aircraft stability, which is an important part of flight safety. The center of gravity (CG) is the point where the total weight of the aircraft is assumed to be concentrated. For safe flight, the CG must be kept within specific limits (Weight and Balance Handbook, 2016).

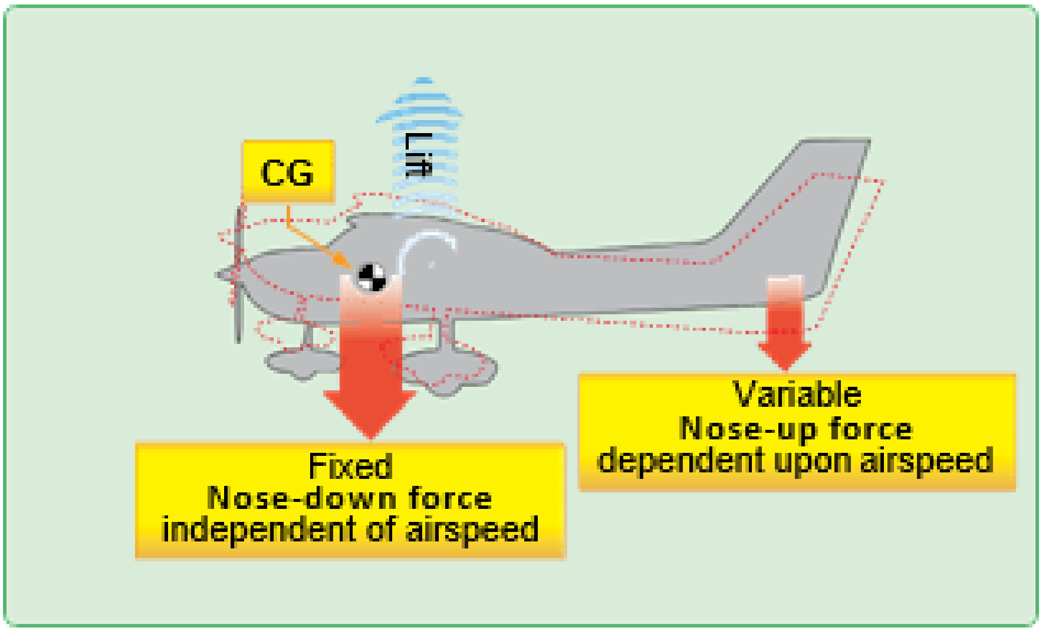


Figure 2.3: Longitudinal forces acting on an airplane in flight (Weight and Balance Handbook, 2016).

The study of stability and control may be viewed as a study of how to achieve and maintain equilibrium (Michael V. Cook, 2012). For example, in steady level flight or steady rise, the net force and moment on an aircraft are zero, and the aircraft moves at a constant speed.

Stability refers to a system's tendency to return to equilibrium after being perturbed in some way (Donald Luwing, Brian Walker, Crawford S. Holling, 1997). When a system is disrupted, static stability refers to the system's immediate response: a statically stable system will go back to its equilibrium state. A dynamically stable system will eventually return to equilibrium, though not always quickly (C. Nelson R., 1998). Figure 2.4 illustrates the two cases and furthermore that of neutral stability, where the system remains in the state to which it has been perturbed.

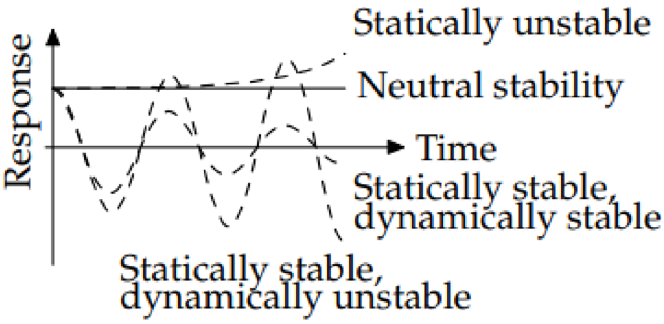


Figure 2.4: Equilibrium and static and dynamic stability (Aircraft Stability and Control, 2020).

The notation for the aircraft stability study is shown in Figure 2.5. Incidence α and the inclination θ are the two angles depicted. The second of these is the angle between a reference line on the aircraft and the horizontal and though it is of little relevance to us when studying stability and control, however, it is critical to a pilot, to whom it is known as “attitude” (Bandu N. Pamadi, 2004). The angle between the reference line and the direction of flight, on the other hand, is quite interesting. We use the zero-lift line (ZLL) as a reference, which is the angle of attack at which the lift is zero. This option reduces the need for future analysis since CL = aα, nevertheless, be cautious in advising additional work because the reference system may be extraordinary (Michael Carley, 2020).

Resolving forces and moments from Figure 1.2,

$T - D - W \sin\theta = 0$; (1.1a)

$L - W \cos\theta = 0$; (1.1b)

$Mcg = 0$, (1.1c)

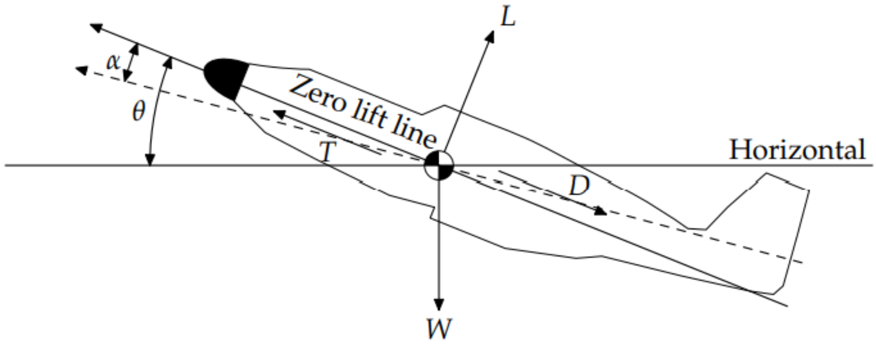


Figure 2.5: Notation for longitudinal stability: the dashed line indicates the flight direction (Aircraft Stability and Control, 2020).

where c.g. refers to "center of gravity" and coordinates are taken in a frame of reference connected to the aircraft. By taking moments about the center of gravity, the effects of the aircraft's mass distribution are eliminated, and (1.1c) is only a statement about the balance of aerodynamic moments (Louis Melville Milne-Thomson, 1966). This is especially handy if we want to compare scale-test results to full-size aircraft. A pilot brings an aircraft into, or out of, trim by adjusting the aerodynamic moments through the utilization of the control surfaces; our analysis allows us to manage the things a pilot changes without agonizing over subtleties of the mass distribution (Michael Carley, 2020).

Having discovered an equilibrium, we might want to know whether it is stable. In aeronautical terms, this can be expressed as the necessity that when an aircraft is pitched nose-up (nose-down) by a perturbation, the adjustment at the moment should be such as to pitch it nose-down (nose-up). In other words, $\partial Mcg/\partial\alpha < 0$ means that the adjustment at the moment is opposite to the change in incidence as shown in Figure 2.6 (Mao Sun, Yan Xiong, 2005).

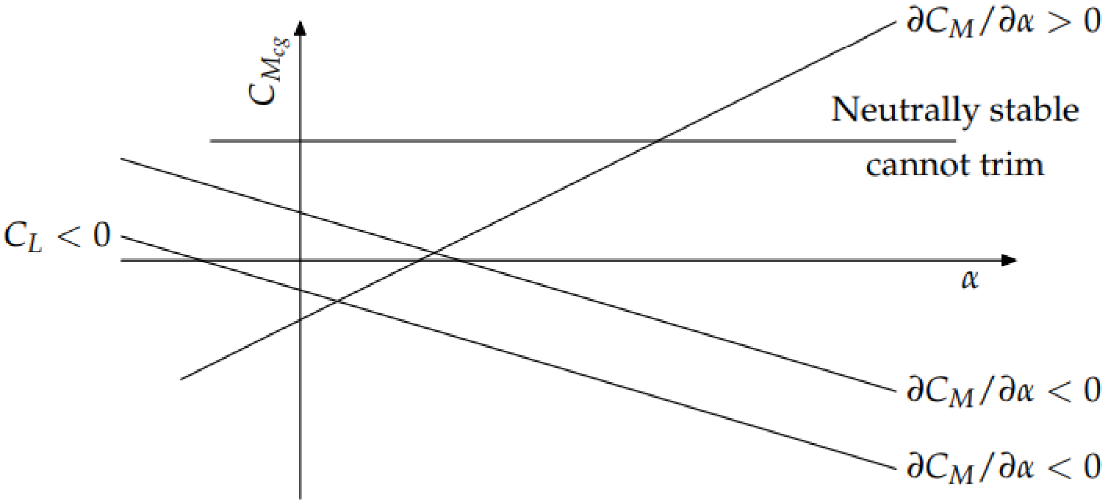


Figure 2.6: Trim and stability behavior (Aircraft Stability and Control, 2020).

2.4 Aerodynamics of wings and controls

To make the challenge manageable, we reduce the system's constituents to a minimal number of parameters when developing any tolerably complicated system. We do not look at the finer points of how a wing works or how the pressure distribution changes when control is deflected in this case; instead, we focus on the overall effect on forces and moments (Rauno Cavallaro, Luciano Demasi, 2016).

As consistently in aerodynamics, we bargain in non-dimensional quantities, normalized on air density ρ, velocity V, wing planform area S and, where vital, wing mean chord $\bar{c}$, or root chord c_0 for tailless aircraft. The ‘mean chord,' also known as the ‘mean aerodynamic chord,' or MAC, is a way of addressing the wing that produces a force and moment on the aircraft that is similar to that of the real wing (Pasquale Sforza, 2014). Figure 2.7 depicts various typical planforms together with their MACs. It is worth noting that the length and position of the mean chord are both important when calculating moments and forces (Michael Carley, 2020). As a result, the coefficients of lift, drag, and moment are given by:

$$C_L = \frac{L}{\rho V^2 S/2}, C_D = \frac{D}{\rho V^2 S/2}, C_M = \frac{M}{\rho V^2 S/2}. \tag{1.2}$$

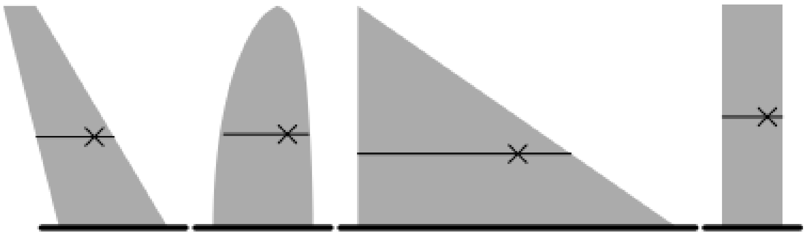


Figure 2.7: Wing planforms (rectangular, delta, bi-elliptical, and swept and tapered) with their mean aerodynamic chords and neutral point (Aircraft Stability and Control, 2020).

The aerodynamic properties of an aircraft are fixed after it is assembled, mostly through the choice of wing section and planform, although diverse impacts must be addressed. With the exception of control surfaces elevator, rudder, and aileron, which may be moved to vary the aerofoil section and hence its properties, the lift curve slope of the wing is constant (Charles E. Dole *et al.*, 2016).

Figure 2.8 shows these essential control surfaces on a conventional aircraft, together with the associated sign conventions. A deflection is considered positive if it results in a positive increase in force. Ailerons act in a differential manner, so the deflection is equal on both surfaces, with a positive deflection producing a positive rolling moment (C. Nelson R., 1998).

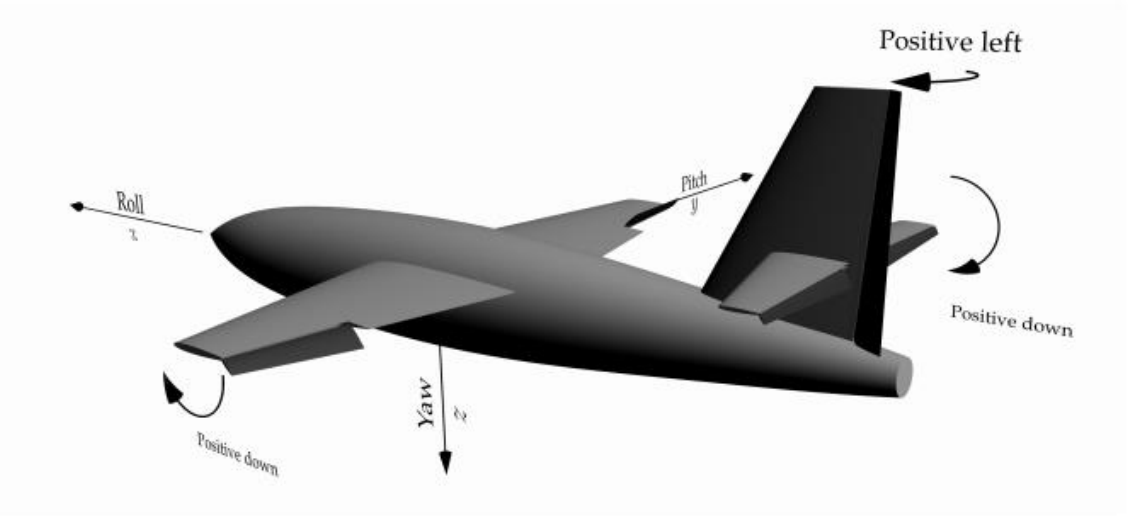
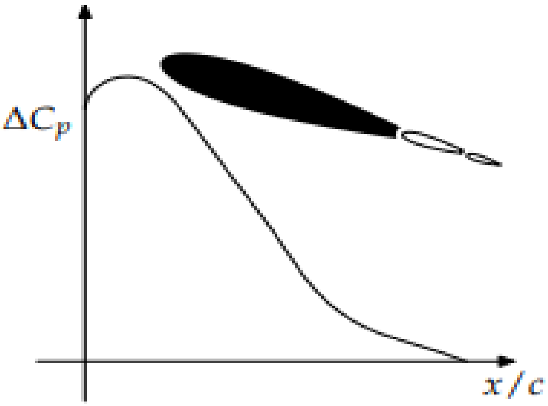
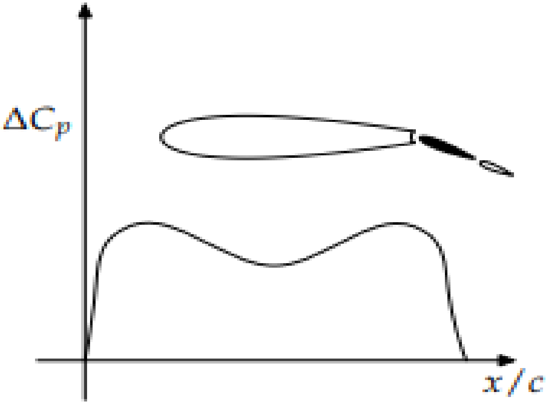


Figure 2.8: Axes and sign conventions for control deflections (Aircraft Stability and Control, 2020).

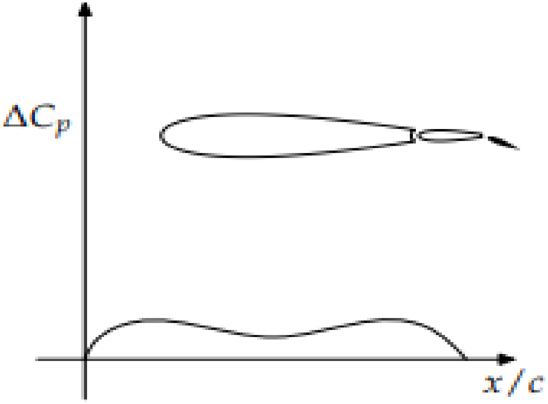
To build a tailplane, we don't need to understand the complexities of control surface aerodynamics, but we do need to understand how they operate (Etkin B, Reid L.D., 1959). Figure 2.9 demonstrates how variations in incidence, elevator deflection, and tab angle affect the pressure coefficient over a surface. We can observe that changes in α_T or η give extremely large changes in pressure distribution, compared to extremely large changes in C_{L_T}, making the tailplane useful for controlling moments on the aircraft. The tab appears to have no effect on pressure distribution or tailplane lift once more.



Tailplane, α_T



Elevator, η



Tab, β

Figure 2.9: Pressure distribution changes with control deflection (Aircraft Stability and Control, 2020).

2.5 Aerodynamic center

When it comes to determining the moments on an aircraft, it is not enough to know what lift and moment are generated by a wing or control surface; we also need to make a choice about where they operate. The moment formed by a lifting body is then computed by considering it as a force and a moment positioned at some reference point so that the moment about another point at a distance x is computed (Antonio Viviani *et al.*, 2020):

$$\mathrm{M(x)} = M_0 + \mathrm{Lx}. \tag{1.3}$$

We have some flexibility in terms of where we get our references, and we should make use of it. The center of pressure, or the point around which the aerodynamic moment is zero, is an irrefutable reference point on a wing. At that point, $M_0 = 0$ and $M = Lx$, which makes life easy. The problem with this reference point is that the center of pressure shifts when the incidence varies. Because controlling α is a big part of stability and control, the center of pressure is not very useful as a reference point because it shifts when our plane pitches (Michael V. Cook, 2012).

We use an alternate reference point termed the aerodynamic center or, for a whole aircraft, the neutral point at that location instead of the center of pressure. This is a point about which the moment is independent of incidence, dM/dα = 0, and

$$\mathrm{M(x)} = M_0 + \mathrm{L(x - xn)}, \tag{1.4}$$

where now M_0 is the zero-lift pitching moment, while n stands for “neutral point”. According to the perspective view of the flight, the neutral point of the entire aircraft is quite probably the most fundamental property (C. Nelson R., 1998).

We may draw some potential links between the center of gravity location and static stability if we consider the overall lift on the aircraft operating at the neutral point, as shown in Figure 2.10. Remember that when the aircraft pitches nose-up, the lift rises, causing a pitching moment around the center of gravity (Mao Sun, Yan Xiong, 2005). The relative locations of the neutral point and the center of gravity are used to determine that moment (Barret, C., 1997). The first figure in Figure 2.10 depicts a stable aircraft because an increase in lift tends to push the nose down: $\partial Mcg/\partial \alpha < 0$; the second is neutrally stable because changes in lift have no effect on the moment, and the third is unstable because an increase in incidence induces an increase in pitching moment which keeps pushing the nose up, $\partial Mcg/\partial \alpha > 0$ (C. Nelson R., 1998).

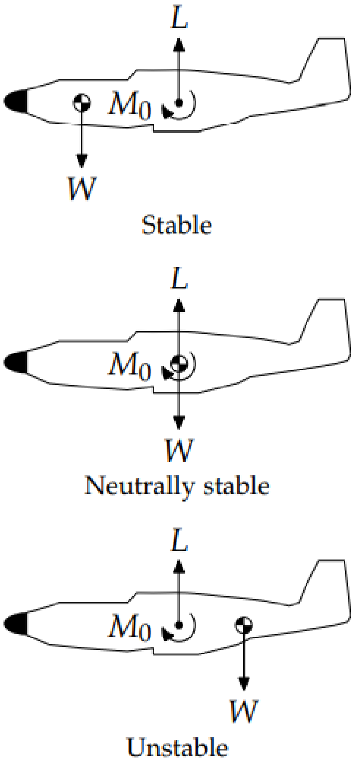


Figure 2.10: Center of gravity and neutral point position (Aircraft Stability and Control, 2020).

2.6 Composite Material

Composite materials have played a significant part in weight reduction, and they are now used in structural applications and components on many types of spacecraft and aircraft, from gliders and hot air balloon gondolas to combat planes, space shuttles, and passenger airliners. Composites are materials made up of at least two phases or constituent elements, the most common of which are plastics bonded with carbon fibers. They may be shaped to boost their strength and stacked with fibers running in different directions to allow designers to create structures with unique features (Jared W. Nelson, 2010).

Increasingly, laminated fiber-reinforced composite materials are being used in engineering applications. These materials have exceptional strength-to-weight ratios, and the manufacturing processes that produce them have improved dramatically since they were first introduced (Ashik, K.P., Sharma, R.S., 2015). The popularity of these materials may be observed in Figure 2.11, which shows the many composites utilized to construct airframes in modern aircraft.

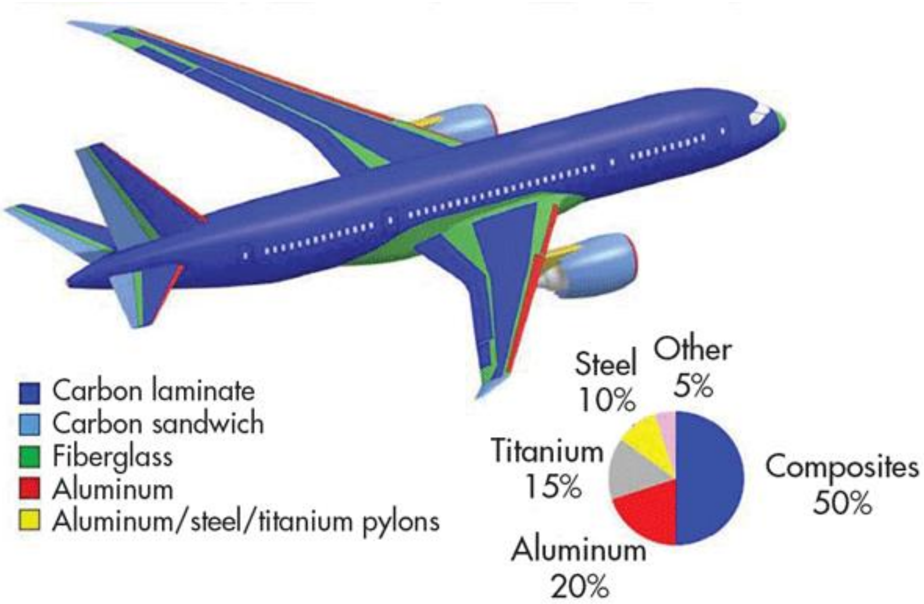


Figure 2.11: The various composites used to construct airframes in modern aircraft (Victor Giurgiutiu, 2016).

2.7 Stress and Strength Analysis of Ultralight Aircraft

The Finite Element Analysis (FEA) is used to conduct the stress and strength analysis of the composite ultralight aircraft fuselage for this project. When a load is applied, SolidWorks Simulation uses FEA (Finite Element Analysis) techniques to identify the behavior of parts or assemblies/parts. Pressure, force, temperature, gravity, centrifugal, and even loads transferred from prior simulation studies may all be used as loads. The ultimate outcome might be expressed as stress, displacement, or strain (Bathe, K.-J., 2016).

The FEM software utilized was SolidWorks Simulation. SolidWorks Simulation, according to Kuang-Hua Chang (2015), uses a generative methodology to help with design optimization. The lower and upper bounds for design variables are different. Individual design scenarios are created by combining these design factors. All of the situations are subjected to a finite element analysis. Feasible designs are obtained from the scenarios evaluated, and the best design that produces the lowest value in the objective function is identified as the solution to the design problem.

SolidWorks Simulation works in tandem with the design process, allowing us to execute linear stress analysis directly from our SolidWorks CAD model, resulting in less expensive prototypes, less effort and delays, and more time and cost-efficiency (SolidWorks, & Education, 2011).

CHAPTER 3

METHODOLOGY

In this chapter, the detailed methodology clarifies how the proposed research will be carried out. The flow chart in Figure 3.1 depicts the methodology in its entirety. Starting with the conceptual design of the fuselage structure, the methodology progressed through detailed design, simulation analysis, analytical analysis, and finally fabrication process.

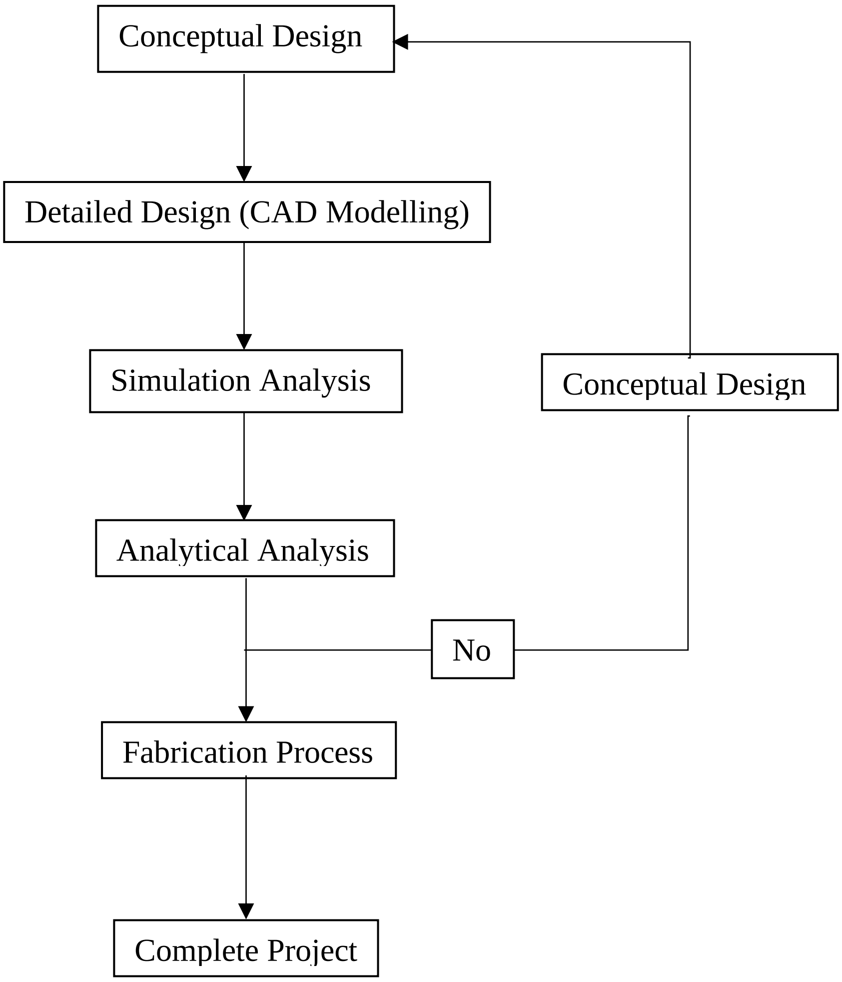


Figure 3.1: Illustrate the flow chart for the outline methodology.

3.1 Conceptual Design

The initial step includes developing the structure of the fuselage is designing. Conceptual design is a preliminary phase of the design process. The general knowledge about the ultra-light aircraft fuselage structural design should be introduced at this point. The primary goal of this conceptual design phase is to finalize all of the specifications and criteria for the fuselage structure. The criteria and specifications for this project, such as detail measurement, weight, the material used, improvement cost, and time consumption, should be examined (Cavagna, L., Ricci, S., Riccobene, L., 2011).

The first schematic of the fuselage structure to be designed once all the precise characteristics have been determined in conceptual design as shown in Figure 3.2. The fuselage structure should be sufficient due to the detail since it is the major structure or body of the ultra-light aircraft (Yuvraj, S.R., Subramanyam, P., 2015).

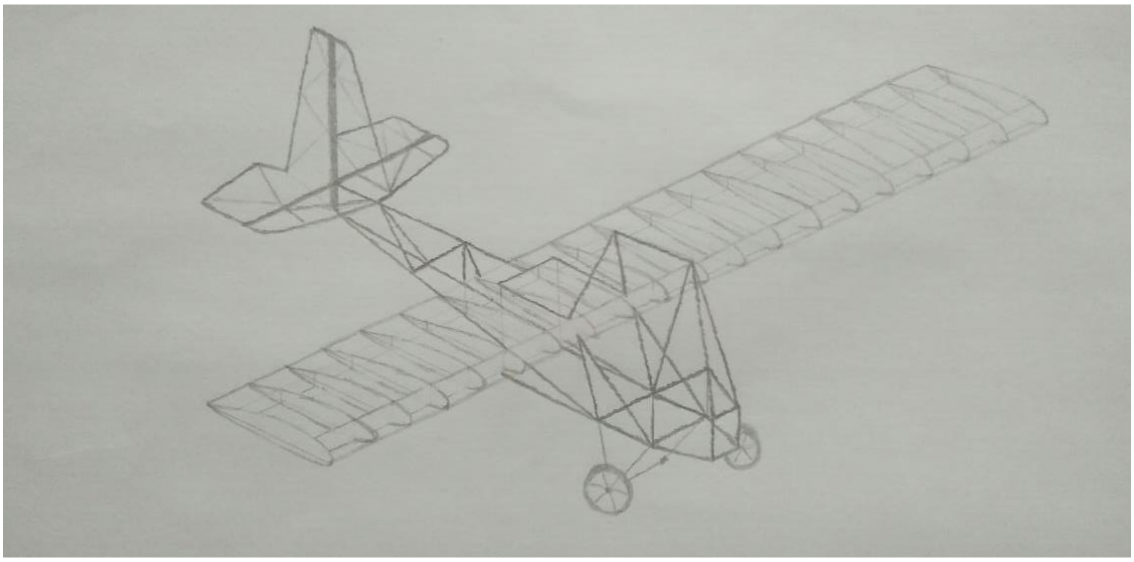


Figure 3.2: The manual conceptual design sketch.

The design starts with a preliminary drawing of the fuselage structure, which must be put together with the landing gear, cockpit, seat location, fuel tank, and tail dragger. As a central component of the ultra-light aircraft's body, the fuselage construction must give improved strength-to-weight ratios (Abzug, M.J., Larrabee, E.E., 2005). Table 3.1 shows the overall dimensions associated with the fuselage construction.

Table 3.1: General dimension related to the fuselage structure.

Description	**Value**	**Units**
Length of the fuselage	183.06	in
Width of the fuselage	52.00	in
Height of the fuselage	64.42	in
Thickness of Aluminium plate, t	0.13	in

3.2 Detailed Design

The ultra-light aircraft's detailed fuselage structural design is the next step forward. The design is improved and sketched at this step, and fuselage structure requirements are created. After all of the detail parameters and requirements for the fuselage structure have been determined and confirmed during the conceptual design process, the fuselage structure's detail and actual design are created utilizing design software. The purpose of this project was accomplished by utilizing SolidWorks 2020 x64 Edition CAD software to create the entire structure drawing and examine the fuselage structure.

Furthermore, SolidWorks software allows a user to easily adjust and revise the design idea at any point during the design process, including part, assembly, and drawing. Users may utilize the mass characteristics of the optimal material determination in SolidWorks software to check for any obstructions in any area of the design (SolidWorks, & Education, 2011). Figure 3.2.1 depicts the first model of the fuselage variations created with SolidWorks 2017 x64 Edition. Meanwhile, Figure 3.2.2 depicts the second design of the fuselage variations that were modelled, and more specifically, the fuselage that was examined throughout this project.

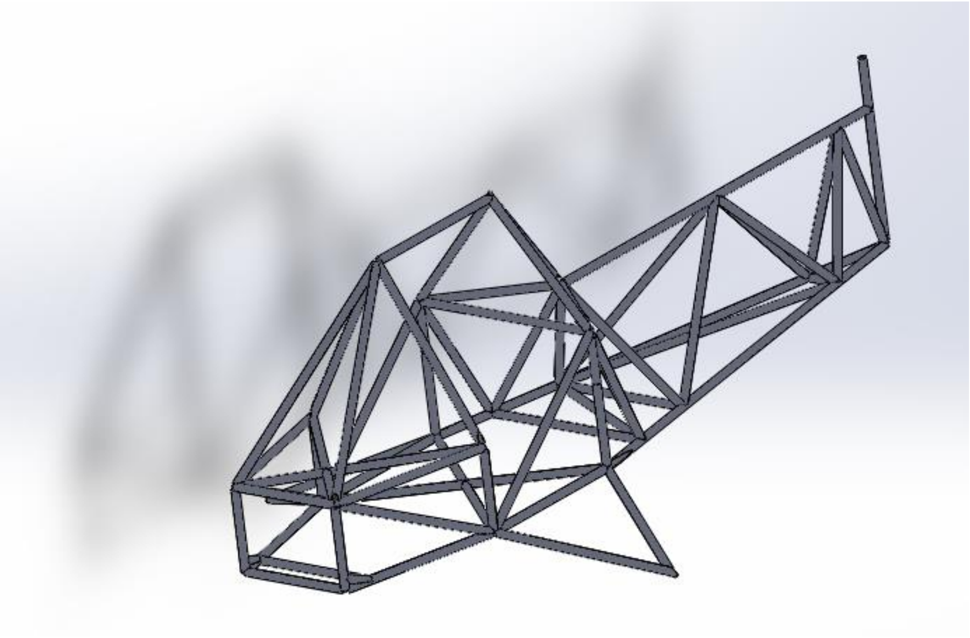


Figure 3.3: Detailed design of the first design of the fuselage structure.

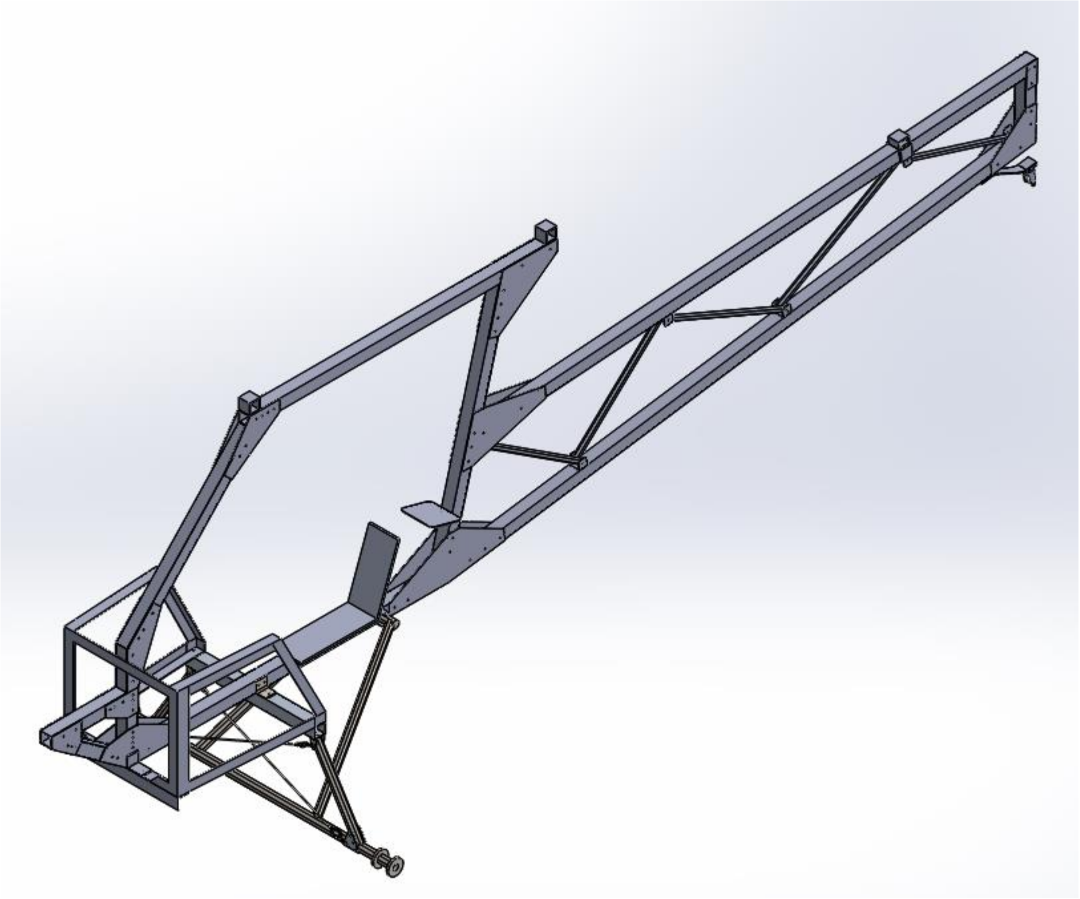


Figure 3.4: Detailed design of the second design of the fuselage structure.

Based on the FAR 23 (2021) assessments, the structure of the fuselage will be examined to see if it can withstand the requisite forces. Appendix A contains the fuselage structure's exact dimensions. The unit of measurement used in this drawing is the inch.

3.3 Simulation Analysis

The stress and strength of the fuselage structure in relation to the applied loads were investigated during the simulation phase. In Chapter 4, the results of this structural study will be discussed.

3.3.1 Finite Element Analysis

Simulation analysis includes the stress and strength analysis of the fuselage structure comparative with the loads applied. In this stage, the stress and strength analysis are conducted using the Finite Element Analysis (FEA). The FEA is the simulation of any given physical phenomenon utilizing the numerical technique called Finite Element Method (FEM). The finite element analysis can be applied in structural analysis, solid mechanics, dynamics, thermal analysis, electrical analysis, and biomaterials (Pravin Pawar *et al.*, 2016). FEA is amazingly valuable for settling complicated geometries, loading, and material properties where analytical solutions cannot be obtained (Madenci, E., Guven, I., 2015). To accomplish the targets of this project, the SolidWorks Simulation programming has been decided to be utilized to build up the simulation and analysis of the whole fuselage structure.