STRUCTURE ANALYSIS ON DESIGN AND FABRICATION OF UAV CAPABLE OF CARRYING MAXIMUM PAYLOAD UNDER GIVEN CONSTRAINTS

A MAJOR PROJECT REPORT

Submitted by:

ABHAY GUPTA [R290209004]
MANGESH TAKKAR [R290209032]

Of

BACHELOR OF TECHNOLOGY

In

AERROSPACE ENGINEERING

Under the Supervision of Prof. Vijay Kumar Patidar

UNIVERSITY OF PETROLEUM & ENERGY STUDIES

विद्यया शक्ति वशम्

Harnessing Energy through Knowledge

DEPARTMENT OF AEROSPACE ENGINEERING
COLLEGE OF ENGINEERING STUDIES (CoES)
UNIVERSITY OF PETROLEUM & ENERGY STUDIES
DEHRADUN- 248007, INDIA

APRIL/MAY 2013

THESIS CERTIFICATE

I hereby certify that the work which is being presented in the project report entitled **"Structure analysis on design and fabrication of UAV capable of carrying maximum payload under given constraints"** in partial fulfilment of the requirements for the satisfactory performance for B.Tech Aerospace Engineering, Major Project submitted in the Department of Aerospace Engineering, University of Petroleum and Energy Studies, Dehradun is an authentic record of my own work carried out during a period from July 2012 to April 2013.

SUBMITTED BY:

Abhay Gupta [R20209004]
Mangesh Takkar [R290209032]

This is to certify that the above statement made by the candidate is correct to the best of our knowledge.

Prof. Vijay Kumar Patidar

Guide vkpatidar

Date: 19/4/2013

Panel members: 27/4/13

Mr.Sourabh Bhatt Mr. V.K Patidar

ACKNOWLEDGEMENT

I would sincerely like to extend my gratitude to my parents for their continued support and motivation. This project would not have completed without them.

I would also like to thank my mentor and guide, Professor Vijay Kumar Patidar for his professional advice and attitude on every aspect of this project and making this project a success. I would also not fail to thank Professor Saurabh Bhatt and Professor Karthik Sunderrraj for their assistance in CFD software which provides us with useful results. Their dedication towards the work is unmatchable.

I would also like to thank the HOD of aerospace department Dr. Om Prakash for his expert assistance on this project.

At last I also sincerely thank the CAD lab assistant for providing us the required system and my team mates for their complete dedication and support.

ABSTARCT

Aircraft are generally built up from the basic components of wings, fuselages, tail units and control surfaces. The structure of an aircraft is required to support two distinct classes of load; the first termed ground loads, including all loads encountered by the aircraft during movement or transportation on ground such as taxiing and landing loads, towing and hoisting loads; while the second air loads, comprises loads imposed on the structure during flight by manoeuvers and gusts. In addition, aircraft designed for a particular role encounter loads peculiar to their sphere of operation. Carrier borne aircraft, for instance, are subjected to catapult take-off and arrested landing loads; most large civil and practically all military aircraft have pressurized cabin for high altitude flying; amphibious aircraft must be capable of landing on water and aircraft designed to fly at high speed at low altitude.

The report presented contains the structural design of an UAV which is capable of carrying maximum payload under given constraints. The design of wing and its components like spar and ribs, vertical stabilizer and horizontal stabilizer are being covered in this report. While designing, basic formulae of bending and shear stresses are being used.

Table of Content

List of Illustrations

List of Tables

Nomenclature

AR	Wing aspect ratio	~
b	Wing span	in
c	Chord length	in
S	Wing area	in^2
c.g.	Centre of gravity	~
C_D	Airplane drag coefficient	~
C_L	Airplane lift coefficient	~
C_m	Pitching moment coefficient	~
C_{Do}	Drag coefficient at zero lift	~
D	Airplane drag	lbs
L	Airplane lift	lbs
e	Oswald's span efficiency factor	~
g	Acceleration due to gravity	ft/s^2
P	Power	hp
T	Thrust	lbs
S_{TO}	Take-off distance	ft
S_L	Landing distance	ft
V	Velocity	ft/s
V_{LO}	Liftoff Velocity	ft/s
α	Angle of attack	deg
ρ_∞	Ambient density	$slug/ft^3$
X_{cg}	Distance of individual components centre of gravity from aircraft centre of gravity	in
W	Gross weight of aircraft	lbs
μ_T	Coefficient of rolling friction	~
C_{mac}	Pitching moment coefficient about wing mean	

	aerodynamic chord	~
$C_{L\alpha w}$	Wing lift slope	rad^{-1}
C_{Low}	Zero lift coefficient	~
C_{Lmax}	Maximum wing lift coefficient	~
x_{cg}/c	Centre of gravity location(% of mean aerodynamic chord)	~
x_{ac}/c	Aerodynamic centre(% of mean aerodynamic chord)	~
x_{np}/c	Location of neutral point(% of mean aerodynamic chord)	~
C_{mact}	Pitching moment coefficient about mean	
	tail aerodynamic chord	~
C_{m0}	Zero angle moment coefficient	~
$C_{m\alpha}$	Pitching moment slope	rad^{-1}
$C_{L\alpha t}$	Horizontal tail lift slope	rad^{-1}
ε_o	Zero lift downwash angle	rad
ε	Downwash angle	rad
S_e	Elevator area	in^2
S_t	Horizontal tail area	in^2
τ_e	Elevator effectiveness parameter	~
l_t	Distance from centre of gravity to horizontal tail	
	aerodynamic centre	in
n	Load factor	~
η	Dynamic pressure ratio	~
V_H	Horizontal tail volume ratio	~
$C_{m\delta e}$	Elevator control power	rad^{-1}
S_r	Surface area of Rudder	in^2
S_a	Surface area of Aileron	in^2
S_v	Surface area of vertical tail	in^2
τ_a	Aileron effectiveness parameter	~
τ_r	Rudder effectiveness parameter	~
σ	Side wash angle	rad

β	Side slip angle	rad
V_v	Vertical tail volume ratio	~
$C_{n\beta v}$	Vertical tail directional stability derivative	rad^{-1}
$C_{l\delta a}$	Aileron control power	rad^{-1}
$C_{n\delta r}$	Rudder control power	rad^{-1}
A_o	Operational availability	~
PPB	Payload prediction bonus	~
EWB	Empty weight bonus	~

CHAPTER 1 Introduction

SAE Aero Design Competition is meant to give students the opportunity to demonstrate their ability to apply engineering knowledge and a dais to enhance their skills in the field of aviation. The engineering challenge is to design, conceive and implement their concepts of aircraft design and fabrication inclusive of all trade studies and making compromises to arrive at a design solution that will optimally meet the mission requirements while conforming to the dimensions limitations. The aim of the SAE Aero Design Competition is to design, fabricate and fly an aircraft that carries maximum possible payload while keeping in mind the available power and staying within the competition's limits. Following were the mission requirements for Regular Class that dictated terms in the design process:

- The aircraft should take-off within 200 feet.
- Aircraft shall not take more than 400 feet to come to a complete stop in landing.
- Sum of dimensions (i.e. Length + Width + Height) shall not exceed 225 inches.
- The aircraft shall not weigh more than 65 pounds with payload and fuel.
- The aircraft must be powered by a single, unmodified OS 61FX with E- 4010 Muffler.

The legacy of participating in the competition is a great responsibility and an immense challenge. Previous team participated from the university inspired & assisted the upcoming team to perform and excel in the competition. The Design strategy of previous year's team had certain loopholes which led to low overall score so a failure analysis was performed on their aircraft.

1.1 Failure Analysis: -

Design	High Wing with Struts Tricycle Landing Gear Conventional Tail Plane
Advantageous Designed Parts	Wings with Horner wing tips Light weight (10.8 lbs. excluding payload) Payload Compartment
Disadvantageous Designed Parts	Box-Shaped Fuselage Landing Gear Fatigue

Table 1 Previous year design analysis

Fuselage Design

- Frontal area was too large producing much of the drag.
- Fuselage was fabricated using plywood so it was quite heavy.

Tail Design

- Tail size was large compared to aircraft thus increasing weight and weathercock stability

Struts

- Struts used were made of Aluminium but were not properly oriented.
- Struts joints were weak.

Landing Gear

- Material selection was not proper which made landing gear assembly too heavy.
- Proper analysis was not done that lead to landing gear failure

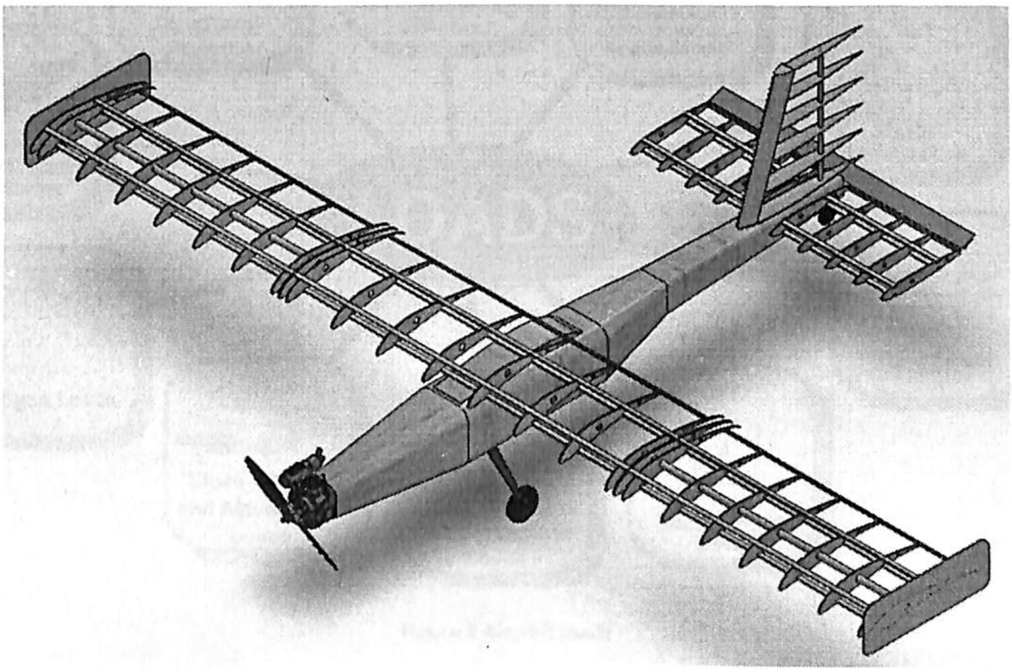


Figure 1 Present Aircraft

1.2 General loads on aircraft: -

Loads may accompany an aircraft's life from the cradle to the grave. Although the overall type and magnitude of major load sets remain the same; there is no fixed load set that is to be applied one aircraft model throughout the life and often identical airframes serving different roles within a fleet over time will be subjected to very different loads. To include as much as possible (or specified) of these loading scenarios in the early process of designing a new type of aircraft is the responsibility of the loads engineering department, while ensuring that these loads can be safely endured throughout the specified life is the task of the design and stress engineers. "New" load sets, developed later during usage of the aircraft are common tasks and handled similar as the "initial design loads" by the design authority with the constriction, that now the airframe is already build and deployed and the focus is on minimising changes though structural modifications to qualify the structure for its new environment either through analysis and / or test. In short, every major change in the aircraft's role, payloads or usage in principle influences the loads acting on the airframe or at least some components. Fig 1 gives an idea how loads are initially generated and how they are used throughout the design-, qualification- and usage process.

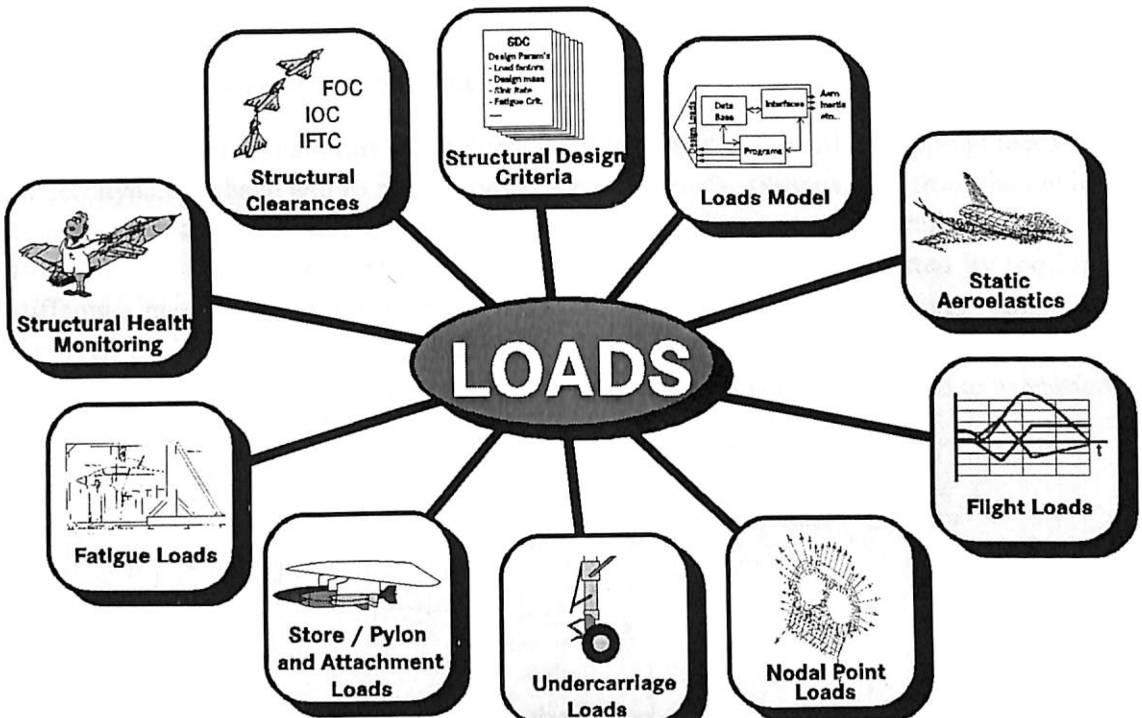


Figure 2 Aircraft loads

The types of loads can be further subdivide into surface forces which act upon the surface of the structure, e.g. aerodynamic and hydrostatic pressure, and body forces which act over the volume of the structure and are produced by gravitational and inertial effects. Calculation of the distribution of the aerodynamic pressure over the various surfaces of an aircraft's structure is presented in numerous texts on aerodynamics. Basically, all air loads are the resultants of the pressure distribution over the surfaces of the skin produced by steady flight, manoeuvre or gust conditions. Generally these resultants cause direct loads, bending, shear and torsion in parts of structure in addition to local, normal pressure loads imposed on the skin. Conventional aircrafts usually consists of fuselage, wings and tail plane. The fuselage contains crew and payload, the latter being passengers, cargo, weapons plus fuel, depending on the type of aircraft and its functions; the wing provide lift and the tail plane is the main contributor to the directional control. In addition, ailerons, elevators and the rudder enable the pilot to manoeuvre the aircraft and maintain its stability in flight, while wing flaps provide the necessary increase of lift for take-off and landing.

The force on an aerodynamic surface results from the differential pressure distribution caused by incidence, camber or the combination of both. Such a pressure distribution has vertical and horizontal resultants acting at a centre of pressure. Clearly the position of the CP changes as the pressure distribution varies with the speed or wing incidence. Wing, tail plane, fuselage are each subjected to direct, bending, shear and torsional loads and must be designed to withstand critical combinations of these.

1.3 Functions of structural components: -

The basic functions of aircraft's structures are to transmit and resist the applied loads, to provide an aerodynamic shape and to protect passengers, payloads, systems, etc. from the environmental conditions encountered during flight. These requirements, in most aircraft results in thin shell structure where the outer surface or skin of the shell is usually supported by the longitudinal stiffening members and transverse frames to enable it to resist bending, compressive and torsional loads without buckling. Such structures are called semi-monocoque, while thin shells which rely entirely on their skin for their capacity to resist loads and referred to as monocoque.

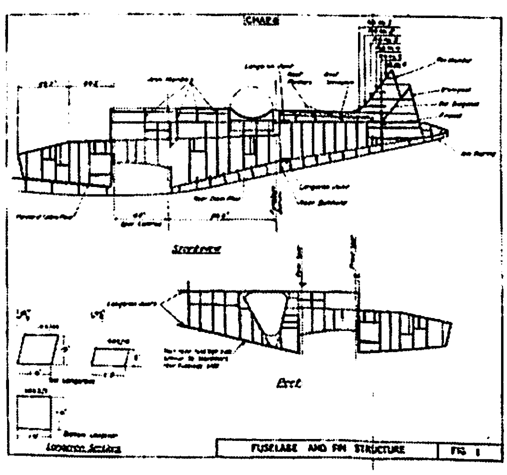


Figure 3 Fuselage Structure

1.4 Loads and Fatigue: -

The determination of loads together with the qualification for static strength and fatigue by calculation and test for all important structural components is a main prerequisite for successful design and safe operation of any aircraft. Whereas for transport aircraft with their rather limited range of operational manoeuvres and high number of flight hours / cycles fatigue is the main design driver for the airframe, fighter aircraft are predominantly designed to (static) limit load cases for the "corners" of the envisaged flight envelope, which in general cover a lot of strength required for fatigue of their comparatively short life. But this is only true as long as fighter life does not exceed the originally planned lifetime and the roles, missions etc. are compatible with the design criteria at the beginning. Aging aircraft in both cases does not only mean that an aircraft is getting older in terms of flight hours and flight cycles, it also means that some of the reference data for the basic design criteria have changed during time, i.e.:
- airframe and equipment mass growth

- enhancement of systems performance, especially engine thrust
- new configurations (stores)
- update of flight control systems (FCS) (electronically or hardware changes like added slats or enlarged ailerons)
- mission profiles and additional/changed roles
- actual usage spectrum

Most of these changes have an immediate impact on aircraft load scenarios, others will not change load levels but may change underlying statistic, e.g. fatigue spectra. Assessment of external loads is therefore a basic task throughout the life of a fleet. Admittedly in many cases there is no simple one to one relationship between "external" loads and local internal stresses, which after all are the basis for the assessment of "life consumption" or "remaining life" of structural components. But providing loads are known for a special structural interface or component, reliable conclusions can be drawn regarding local stresses relating to the manifold of load cases from experience, measurement and detailed FE analysis during design, qualification and test phases in many cases. In addition the comparison of load spectra alone may already be suitable for drawing conclusions without recourse to detail stress calculations of specific locations for components with limited load case variations i.e. landing gears.

1.5 The Determination of Design Loads: -

Design loads, better "Initial Design Loads" are the first step in the loads history of an airframe that influences the detail design of a component (i.e. wing or fuselage structure) or, at a later stage in the design process, a part (i.e. wing spar cap or fuselage skin panel) in many details. Since not every load is determining these design tasks, establishment and identification of the "design load cases" is important. The following is a summary on the methods how design load cases are determined, with special attention to points where an immediate context with fatigue calculations exists. Figure below shows a typical "loads loop" which usually is repeated several times in the different phases of the aircraft design. First of all the Structural Design Criteria (SDC) are prepared as a basis for design, specifying the basic performance and flight parameters, then a Loads Model (LM) is built, based on the SDC's, the aerodynamic, flight mechanic and weight and balance data of the aircraft.

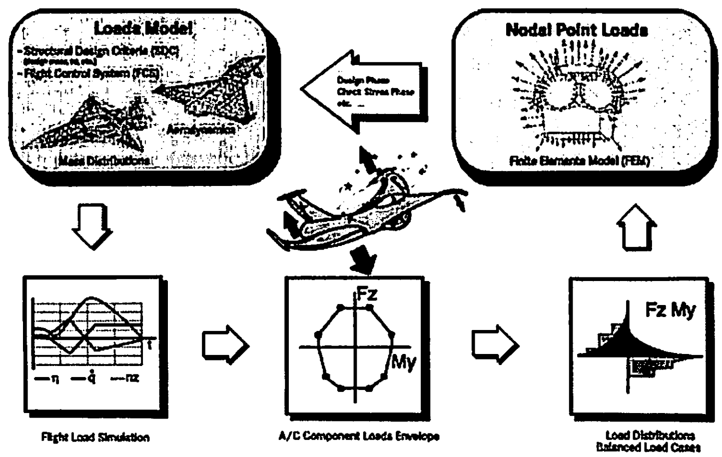


Figure 4 Design strategy

1.6 The Fuselage: -

The fuselage should carry the payload, and is the main body to which all parts are connected. It must be able to resist bending moments (caused by weight and lift from the tail), torsional loads (caused by fin and rudder) and cabin pressurization. The structural strength and stiffness of the fuselage must be high enough to withstand these loads at the same time; the structural weight must be kept to a minimum. In transport aircraft, the majority of the fuselage is cylindrical or near-cylindrical, with tapered nose and tail sections. The semi-monocoque construction, which is virtually standard in all modern aircraft, consists of a stressed skin with added stringers to prevent buckling, attached to hoop-shaped frames. The fuselage also has members perpendicular to the skin, that supports it and helps keep its shape. These supports are called frames if they are open or ring-shaped or bulkheads if they are closed. Disturbances in the perfect cylindrical shell, such as doors and windows, are called cut outs. They are usually unsuitable to carry many of the loads that are present on the surrounding structure. The direct load paths are interrupted and as a result the structure around the cut-out must be reinforced to maintain the required strength. A typical freighter aircraft will have a much larger door than a passenger aircraft. It is therefore necessary for them to transmit some of the loads from the frames and stringers. Where doors are smaller, the surrounding structure is reinforced to transmit the loads around the door. In aircraft with pressurized fuselages, the fuselage volume both above and below the floor is pressurized, so no pressurization loads exist on the floor. If the fuselage is suddenly de-pressurized, the floor will be loaded because of the pressure difference. The load will persist until the pressure in the plane has equalized, usually via floor-level side wall vents. Sometimes different parts of the fuselage have different radii. This is termed a double-bubble fuselage Pressurization can lead to tension or compression of the floor-supports, depending on the design. Frames give the fuselage its cross-sectional shape and prevent it from buckling, when it is subjected to bending loads. Stringers give a large increase in the stiffness of the skin under torsion and bending loads, with minimal increase in weight. Frames and stringers make up the basic skeleton of the fuselage. Pressure bulkheads close the pressure cabin at both ends of the fuselage, and thus carry the loads imposed by pressurization. They may take the form of flat discs or curved bowls. Fatigue is a phenomenon caused by repetitive loads on a structure. It depends on the magnitude and frequency of these loads in combination with the applied materials and structural shape. Fatigue-critical areas are at the fuselage upper part and at the joints of the fuselage frames to the wing spars.

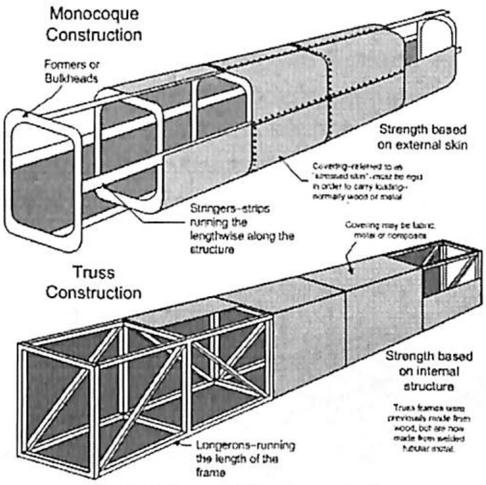


Figure 5 Types of fuselage design

1.7 Wing Contents: -

Providing lift is the main function of the wings of an aircraft. The wings consist of two essential parts. The internal wing structure, consisting of spars, ribs and stringers, and the external wing, which is the skin. Ribs give the shape to the wing section, support the skin (prevent buckling) and act to prevent the fuel surging around as the aircraft manoeuvres. They serve as attachment points for the control surfaces, flaps, undercarriage and engines. The ribs need to support the wing-panels, achieve the desired aerodynamic shape and keep it, provide points for conducting large forces, add strength, prevent buckling, and separate
the individual fuel tanks within the wing. There are many kinds of ribs. Form ribs consist of a sheet of metal, bent into shape. Plate-type ribs consist of sheet-metal, which has upturned edges and weight-saving holes cut out into it. These ribs are used in conditions of light to medium loading. Truss ribs consist of profiles that are joined together. These ribs may be suitable for a wide range of load-types. Closed ribs are constructed from profiles and sheet-metal, and are suitable for closing off sections of the wing. This rib is also suitable for a variety of loading conditions. Forged ribs are manufactured using heavy press-machinery, and are used for sections where very high loads apply. Milled ribs are solid structures, manufactured by milling away excess material from a solid block of metal, and are also used where very high loads apply. The stringers on the skin panels run in the length of the wing, and so usually need to bridge the ribs. There are several methods for dealing with this problem. The stringers and ribs can both be uninterrupted. The stringers now run over the rib, leaving a gap between rib and skin. Rib and skin are indirectly connected, resulting in a bad shear load transfer between rib and skin. The stringers can be interrupted at the rib. Interrupting the stringer in this way certainly weakens the structure, and therefore extra strengthening material, called a doubler, is usually added. Naturally, the stringers can also interrupt the rib. The stringers now run through holes cut into the

rib, which also causes inevitable weakening of the structure. The ribs also need to be supported, which is done by the spars. These are simple beams that usually have a cross-section similar to an I-beam. The spars are the most heavily loaded parts of an aircraft. They carry much more force at its root, than at the tip. Since wings will bend upwards, spars usually carry shear forces and bending moments. Aerodynamic forces not only bend the wing, they also twist it. To prevent this, the introduction of a second spar seems logical. Torsion now induces bending of the two spars, which is termed differential bending. Modern commercial aircrafts often use two-spar wings here the spars are joined by a strengthened section of skin, forming the so-called torsion-box structure. The skin in the torsion-box structure serves both as a spar-cap (to resist bending), as part of the torsion box (to resist torsion) and to transmit aerodynamic forces.

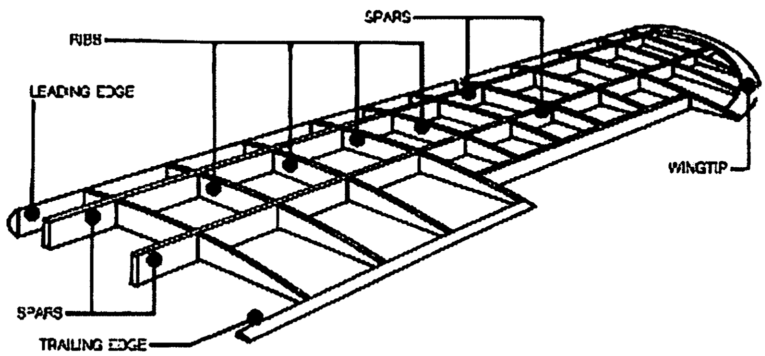


Figure 6. Wing internal structure

1.8 The Tail: -

In most aircraft, the sole function of the tail unit is to provide the required stability and control. Stability is the tendency of the aircraft to return to its original attitude by itself.
Since an aircraft flies in three-dimensional space, stability and control are required in three directions. These axes are lateral (left and right), vertical (up and down) and longitudinal (fore and aft). For aircraft turns, three manoeuvre cases are used. For pitch, which is rotation about the lateral axis, the horizontal tail with elevators is used. For yaw, which is rotation about the vertical axis, the vertical fin with rudder is used. For roll, which is rotation about the longitudinal axis, the ailerons are used. The fin provides stability in yaw. When the aircraft is required to yaw, the rudder is deflected. The tail plane provides stability in pitch. When the aircraft is required to climb or descend, the elevators are deflected. If the position of the centre of gravity varies, or if the aircraft speed is changed, the elevator position necessary to maintain level flight will change. Therefore a small extra control surface is added to each main surface to allow the pilot to trim the aircraft.

CHAPTER 2 Conceptual Design

Aircraft is a pinnacle of engineering science and it is designed to withstand immense stress and loads with the vision to minimize the weight. Thus the Design Strategy is broken down into following sequential and interrelated steps:

- Design Configuration Selection
- Study of components to meet requirements and limitations
- Wing Sizing and Aerofoil
- Fuselage Dimensions and rigidity
- Drag Estimation
- Aircraft Stability and Control Parameters
- Engine Specifications and Lift optimization
- Payload Prediction
- Flight Tests

2.1 Configuration Selection:

Configuration of an aircraft is based on the type of mission and performance it is required to portray under operational grounds.

- Conventional Plane (Tail Aft)
- Conventional Plane with Sweep Wings
- Delta Wings
- Stacked Wings (Bi/tri-plane)

The study of different configuration of the airplane describes the various advantages and disadvantages which help to select particular design according to mission objective. The delta wing was eliminated because it is ineffective at low speeds and generates relatively higher drag at lower altitudes. The biplane/triplane configuration was considered initially due to the high lift produced while maintaining a strong structure. But for a given wing area biplane produces more drag and less lift than a monoplane. Whereas the Sweep forward/back angles are applied on the aircraft ranging for subsonic speeds & in consideration to delta wings, it provides greater plan-form area of the wings which prove to be very significant at the high speeds and high profile drag at the low speeds. Hence these designs were not considered. Conventional plane is the simplest design and easy to fabricate. Also historical data shows conventional configuration as a successful configuration in this competition. Conventional design reinforced with Semi-Monocoque fuselage with high wing and tail dragger landing gear was selected for this year's competition. Key factors in favour of a conventional configuration were experiences from last year's performance and pilot's ease of handling.

2.2 Aerofoil Selection: -

The main objective of the team is to design an aircraft with sufficiently high lifting capacity at low speeds. Different airfoils were considered for study and analysis like FX 63-137, Selig

S1210, and Selig S1223. In which S1210 is specially designed for heavy lift competition aircrafts whereas S1223 aerofoil is most uniquely engineered and perform 30% more than FX 63-137 aerofoil. Moreover previous team's supported S1223 for best flight performance. The Selig 1223 low Reynolds number high-lift aerofoil was chosen. The S1223 achieves over 30% more lift than conventional aerofoils with a maximum lift coefficient of **2.25**.

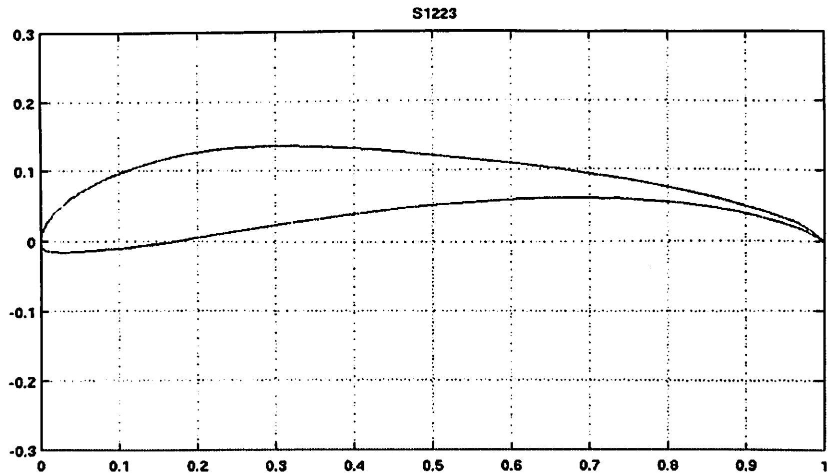


Figure 7 Selig 1223 aerofoil

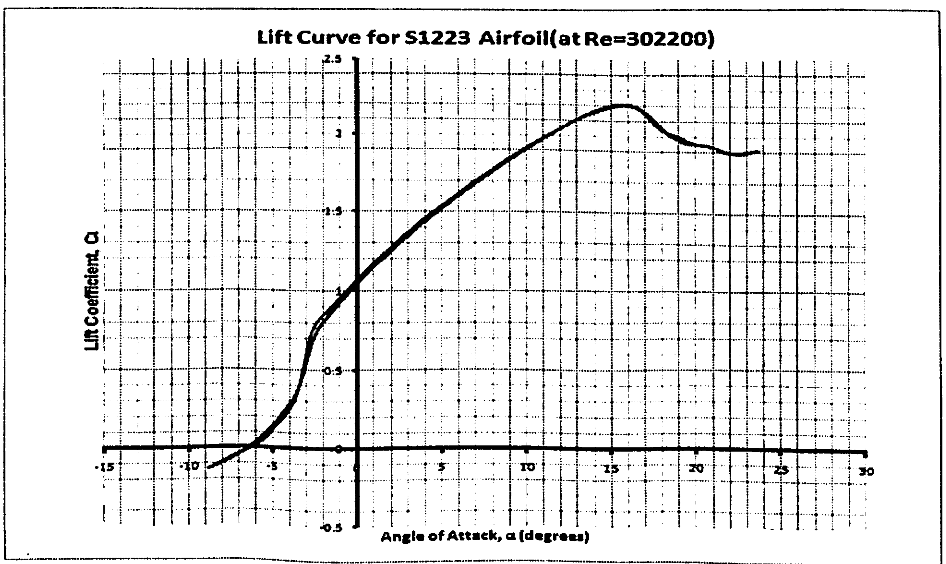


Figure 8 Lift Curve

2.3 Wing Layout and Design: -

Most important part of an aircraft is its wing which provides lift and transfer the force of lift to the fuselage. Wing's lifting capability is depended on aerofoil selection and its dimensions. Wing design holds the key to carrying maximum payload. Rectangular wings were considered as they are simple to fabricate and more structurally efficient in low speed applications. End plate wing tip is introduced to reduce induced drag and increase the lateral stability of the airplane.

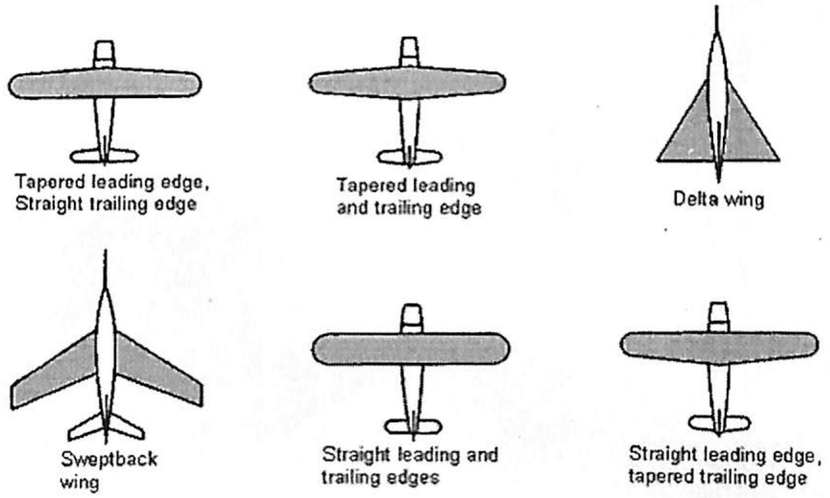


Figure 9 Wing configuration

2.4 Empennage: -

The main objective for the tail is to exert control over the aircraft. Drag should be kept to minimum therefore; the vertical stabilizer should be only as thick as necessary to hold up to cross winds and support the horizontal stabilizer. Since no lift is produced, the vertical stabilizer was chosen to be a 12% thick **NACA 0012** symmetrical aerofoil.

2.5 Fuselage: -

The fuselage is the main assembly & strongest part of the aircraft which holds the wings, tail plane and the power plant. As the aircraft is designed to be lightweight, the material selection of the fuselage is of prime importance. Different fuselage Designs:

- Monocoque Fuselage
- Semi Monocoque Fuselage
- Truss –type Fuselage

As the maximum capacity to lift the weight have increased from 55lbs to 65 lbs., the Semi-Monocoque fuselage is the answer to increased strength and capacity to withstand loads and aerodynamic loads generated due to aircraft manoeuvres.

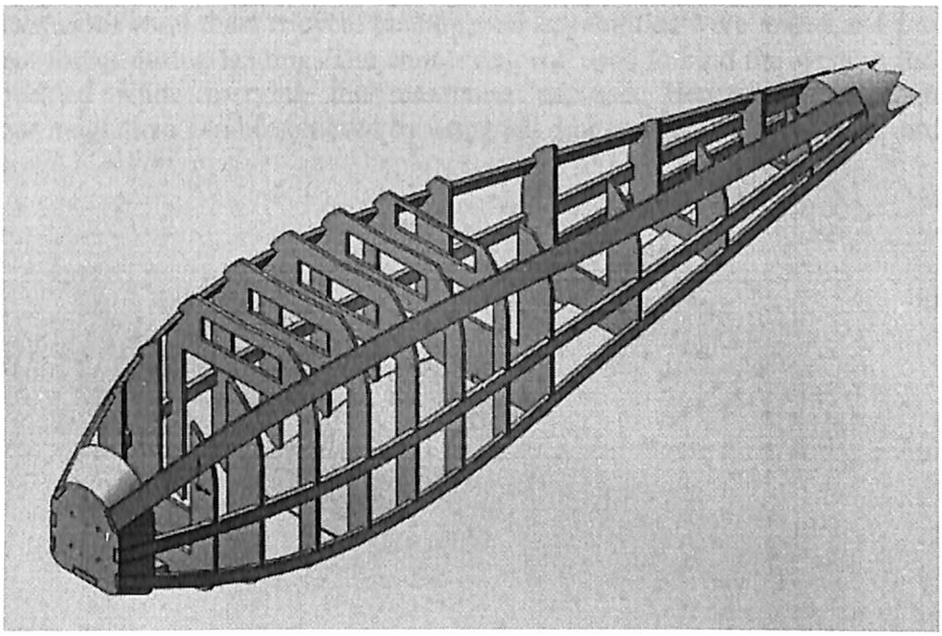


Figure 10 Semi-monocoque fuselage

2.6 Landing Gear Design: -

Landing gears are the structural integral part (SIP) of an aircraft. It is made up of one of the strongest material in the material science department to withstand impact force during landing. Landing gears are of different types according to the requirement and loading conditions:

- Tricycle landing gear
- Tail Dragger landing Gear

Figure 11 Tail dragger landing gear

As per the previous team data, tricycle landing gear capabilities were tested and proved to fail under impact forces during landing. The connecting rod used to bind the tyres to the calculated distance buckled while carrying the maximum payload. Hence increased strength and advantageous height loss can be achieved by using tail dragger landing gear on the aircraft.

Chapter 3 Literature Review

3.1 Airworthiness: -

The airworthiness of an aircraft is concerned with the standards of safety incorporated in all aspects of its construction. These range from structural strength to the provision of certain safeguards in the event of crash landings, and include design requirements relating to aerodynamics, performance and electrical and hydraulic systems. The selection of minimum standards of safety is largely the concern of 'national and international' airworthiness authorities who prepare handbooks of official requirements. The handbooks include operational requirements, minimum safety requirements, recommended practices and design data, etc. In this we shall concentrate on the structural aspects of airworthiness which depend chiefly on the strength and stiffness of the aircraft. Strength problems arise, as we have seen, from ground and air loads, and their magnitudes depend on the selection of manoeuvring and other conditions applicable to the operational requirements of a particular aircraft.

3.2 Factors of safety-flight envelope: -

The control of weight in aircraft design is of extreme importance. Increases in weight require stronger structures to support them, which in turn lead to further increases in weight and so on. Excesses of structural weight mean lesser amounts of payload, thereby affecting the economic viability of the aircraft. The aircraft designer is therefore constantly seeking to pare his aircraft's weight to the minimum compatible with safety. However, to ensure general minimum standards of strength and safety, airworthiness regulations lay down several factors which the primary structure of the aircraft must satisfy. These are the *limit load*, which is the maximum load that the aircraft is expected to experience in normal operation, *the proof load*, which is the product of the limit load and the proof factor (1.0–1.25), and the *ultimate load*, which is the product of the limit load and the ultimate factor (usually 1.5). The aircraft's structure must withstand the proof load without detrimental distortion and should not fail until the ultimate load has been achieved. The proof and ultimate factors may be regarded as factors of safety and provide for various contingencies and uncertainties.

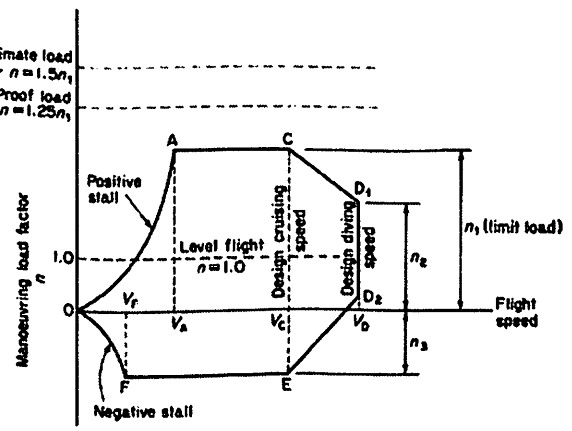


Figure 12 The flight Envelope

The basic strength and flight performance limits for a particular aircraft are selected by the airworthiness authorities and are contained in the *flight envelope* or V-n diagram shown in figure above. The curves OA and OF correspond to the stalled condition of the aircraft and are obtained from the well-known aerodynamic relationship.

$$\text{Lift} = nW = \tfrac{1}{2}\rho V^2 S C_{L,max}$$

Therefore, for speeds below V_A (positive wing incidence) and V_F (negative incidence) the maximum loads which can be applied to the aircraft are governed by Clmax. As the speed increases it is possible to apply the positive and negative limit loads, corresponding to n_1 and n_3, without stalling the aircraft so that AC and FE represent maximum operational load factors for the aircraft. Above the design cruising speed V_C, the cut-off lines CD_1 and D_2E relieve the design cases to be covered since it is not expected that the limit loads will be applied at maximum speed. Values of n_1, n_2 and n_3 are specified by the airworthiness authorities for particular aircraft. A particular flight envelope is applicable to one altitude only since Clmax is generally reduced with an increase of altitude, and the speed of sound decreases with altitude thereby reducing the critical Mach number and hence the design diving speed V_D. Flight envelopes are therefore drawn for a range of altitudes from sea level to the operational ceiling of the aircraft.

3.3 Load factor determination: -

Several problems require solution before values for the various load factors in the flight envelope can be determined. The limit load, for example, may be produced by a specified manoeuvre or by an encounter with a particularly severe gust. Clearly some knowledge of possible gust conditions is required to determine the limiting case. Furthermore, the fixing of the proof and ultimate factors also depends upon the degree of uncertainty of design, variations in structural strength, structural deterioration, etc. We shall now investigate some of these problems to see their comparative influence on load factor values.

3.3.1 Limit load: -

An aircraft is subjected to a variety of loads during its operational life, the main classes of which are: manoeuvre loads, gust loads, undercarriage loads, cabin pressure loads, buffeting and induced vibrations. Of these, manoeuvre, undercarriage and cabin pressure loads are determined with reasonable simplicity since manoeuvre loads are controlled design cases; undercarriages are designed for given maximum descent rates and cabin pressures are specified. The remaining loads depend to a large extent on the atmospheric conditions encountered during flight. Estimates of the magnitudes of such loads are only possible therefore if in-flight data on these loads is available. It obviously requires a great number of hours of flying if the experimental data are to include possible extremes of atmospheric conditions. In practice, the amount of data required to establish the probable period of flight time before an aircraft encounters, say, a gust load of a given severity, is a great deal more than that available. It therefore becomes a problem in statistics to extrapolate the available data and calculate the probability of an aircraft being subjected to its proof or ultimate load during its operational life. The aim would be for a zero or negligible rate of occurrence of its ultimate load and an extremely low rate of occurrence of its proof load. Having decided on an ultimate load, then the limit load may be fixed as defined earlier. Although the value of the ultimate factor includes, as we have already noted, allowances for uncertainties in design, variation in structural strength and structural deterioration.

3.3.2 Uncertainties in design and structural deterioration: -

Neither of these presents serious problems in modern aircraft construction and therefore do not require large factors of safety to minimize their effects. Modern methods of aircraft structural analysis is refined and, in any case, tests to determine actual failure loads are carried out on representative full scale components to verify design estimates. The problem of structural deterioration due to corrosion and wear may be largely eliminated by close inspection during service and the application of suitable protective treatments.

3.3.3 Variation in structural strength: -

To minimize the effect of the variation in structural strength between two apparently identical components, strict controls are employed in the manufacture of materials and in the fabrication of the structure. Material control involves the observance of strict limits in chemical composition and close supervision of manufacturing methods such as machining, heat treatment, rolling, etc. In addition, the inspection of samples by visual, radiographic and other means, and the carrying out of strength tests on specimens, enable below limit batches to be isolated and rejected. Thus, if a sample of a batch of material falls below a specified minimum strength then the batch is rejected. This means of course that an actual structure always comprises materials with properties equal to or better than those assumed for design purposes, an added but un allowed for 'bonus' in considering factors of safety. Similar precautions are applied to assembled structures with regard to dimension tolerances, quality of assembly, welding, etc. Again, visual and other inspection methods are employed and, in certain cases, strength tests are carried out on sample structures.

3.3.4 Fatigue: -

Although adequate precautions are taken to ensure that an aircraft's structure possesses sufficient strength to withstand the most severe expected gust or manoeuvre load, there still remains the problem of fatigue. Practically all components of the aircraft's structure are subjected to fluctuating loads which occur a great many times during the life of the aircraft. It has been known for many years that materials fail under fluctuating loads at much lower values of stress than their normal static failure stress. A graph of failure stress against number of repetitions of this stress has the typical form shown in Figure below. For some materials, such as mild steel, the curve (usually known as an S–N curve or diagram) is asymptotic to a certain minimum value, which means that the material has an actual infinite-life stress. Curves for other materials, for example aluminium and its alloys, do not always appear to have asymptotic values so that these materials may not possess an infinite-life stress. We shall discuss the implications of this a little later. Prior to the mid-1940s little attention had been paid to fatigue considerations in the design of aircraft structures. It was felt that sufficient static strength would eliminate the possibility of fatigue failure. However, evidence began to accumulate that several aircraft crashes had been caused by fatigue failure. The seriousness of the situation was highlighted in the early 1950s by catastrophic fatigue failures of two Comet airliners. These were caused by the once-per-flight cabin pressurization cycle which produced circumferential and longitudinal stresses in the fuselage skin. Although these stresses were well below the allowable stresses for single cycle loading, stress concentrations occurred at the corners of the windows and around rivets which raised local stresses considerably above the general stress level. Repeated cycles of pressurization produced fatigue cracks which propagated disastrously, causing an explosion of the fuselage at high altitude.

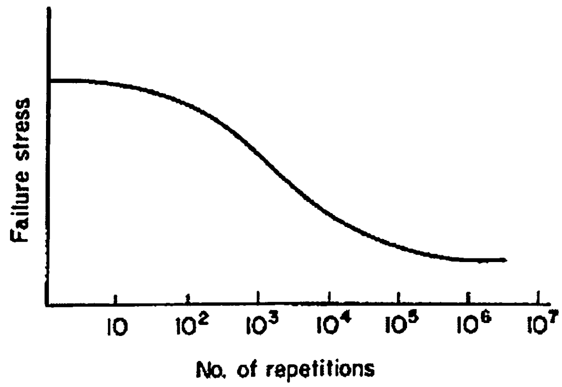


Figure 13 The S-N curve

3.4 Aircraft inertia loads: -

The maximum loads on the components of an aircraft's structure generally occur when the aircraft is undergoing some form of acceleration or deceleration, such as in landings, take-offs and manoeuvres within the flight and gust envelopes. Thus, before a structural component can be designed, the inertia loads corresponding to these accelerations and decelerations must be calculated. For these purposes we shall suppose that an aircraft is a rigid body and represent it by a rigid mass, m, as shown in figure below. We shall also, at this stage, consider motion in the plane of the mass which would correspond to pitching of the aircraft without roll or yaw. We shall also suppose that the centre of gravity (CG) of the mass has coordinates $\bar{x}$, $\bar{y}$ referred to x and y axes having an arbitrary origin O; the mass is rotating about an axis through O perpendicular to the xy plane with a constant angular velocity ω.

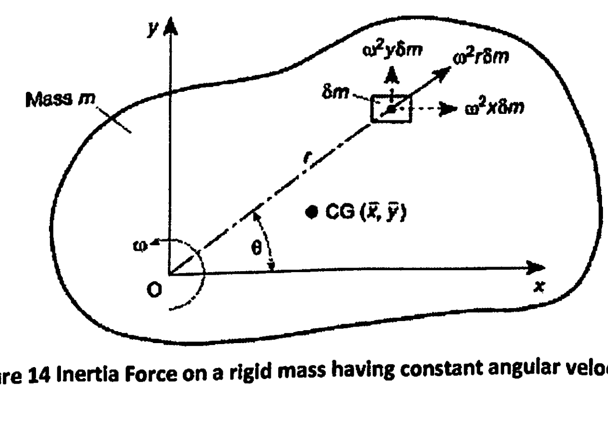


Figure 14 Inertia Force on a rigid mass having constant angular velocity

The acceleration from any point, a distance r from O, is $\omega^2 r$ and is directed towards O. Thus the inertia force acting on the element, δm, is $\omega^2 r\delta m$ in a direction opposite to the acceleration as shown above. The components of this force parallel to x and y axis are, $\omega^2 r\delta m\cos\theta$ and $\omega^2 r\delta m\sin\theta$ respectively. The resultant inertia force F_x and F_y are given by: -

$$F_x = \int \omega^2 x \, dm = \omega^2 \int x \, dm$$

$$F_y = \int \omega^2 y \, dm = \omega^2 \int y \, dm$$

In which we note that the angular velocity ω is constant and therefore can be taken outside the integration. In the above equation $\int xdm$ and $\int ydm$ are the moments of the mass, m, about the x and y axis respectively so that;

$$F_x = \omega^2 \bar{x} m$$

And

$$F_y = \omega^2 \bar{y} m$$

Suppose now that the rigid body is subjected to an angular acceleration (or deceleration) α in addition to the constant angular velocity, ω, as shown in figure below. An additional inertia force, $\alpha r\delta m$, acts on the element δm in a direction perpendicular to r and in the opposite sense to the angular acceleration. This inertia force has components $\alpha r\delta m\cos\theta$ and $\alpha r\delta m\sin\theta$. Thus the resultant force F_x and F_y are given by: -

$$F_x = \int \alpha y \, dm = \alpha \int y \, dm$$

And

$$F_y = -\int \alpha x \, dm = -\alpha \int x \, dm$$

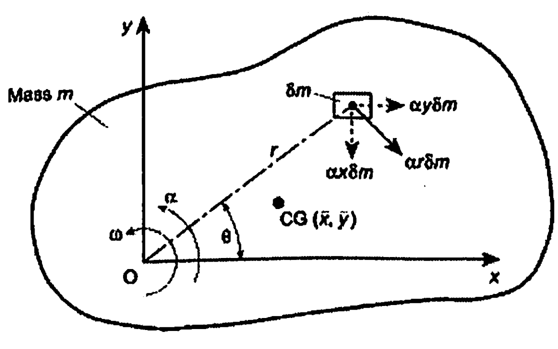


Figure 15 Inertia force on a rigid mass subjected to constant angular acceleration

The torque about the axis of rotation produced by the inertia force corresponding to the angular acceleration on the element dm is given by

$$\delta T_O = \alpha r^2 \delta m$$

Thus for complete mass: -

$$T_O = \int \alpha r^2 \, dm = \alpha \int r^2 \, dm$$

3.5 Symmetric manoeuvre loads: -

We shall now consider the calculation of aircraft loads corresponding to the flight conditions specified by flight envelopes. There are, in fact, an infinite number of flight conditions within the boundary of the flight envelope although, structurally, those represented by the boundary are the most severe. Furthermore, it is usually found that the corners A, C, D_1 ,D_2 , E and F (as shown in the flight envelope) are more critical than points on the boundary between the corners so that, in practice, only the six conditions corresponding to these corner points need be investigated for each flight envelope. In symmetric manoeuvres we consider the motion of the aircraft initiated by movement of the control surfaces in the plane of symmetry. Examples of such manoeuvres are loops, straight pull-outs and bunts, and the calculations involve the determination of lift, drag and tail plane loads at given flight speeds and altitudes.

3.5.1 Level flight: -

Although steady level flight is not a manoeuvre in the strict sense of the word, it is a useful condition to investigate initially since it establishes points of load application and gives some idea of the equilibrium of an aircraft in the longitudinal plane. The loads acting on an aircraft in steady flight are shown in following figure, with the following notation:

L is the lift acting at the aerodynamic centre of the wing.
D is the aircraft drag.
M_0 is the aerodynamic pitching moment of the aircraft less its horizontal tail.
P is the horizontal tail load acting at the aerodynamic centre of the tail, usually taken to be at approximately one-third of the tailplane chord.
W is the aircraft weight acting at its CG.
T is the engine thrust, assumed here to act parallel to the direction of flight in order to simplify calculation.

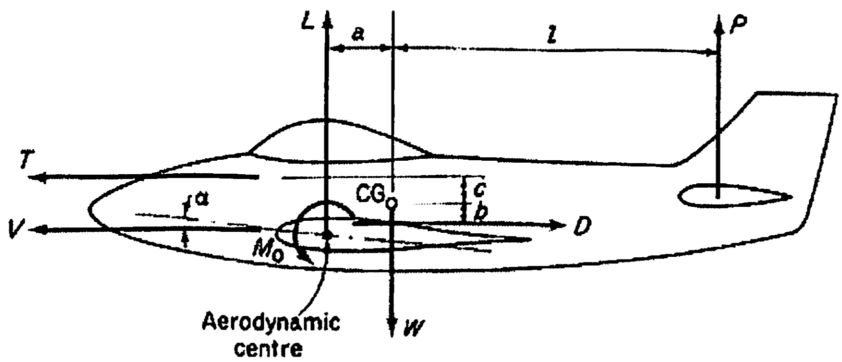


Figure 16 Aircraft loads in level flight

The loads are in static equilibrium since the aircraft is in a steady, un-accelerated, level flight condition. Thus for vertical equilibrium: -

$$L + P - W = 0$$

And for the horizontal equilibrium: -

$$T - D = 0$$

Taking moment about the centre of gravity of the airplane in plane of symmetry: -

$$La - Db - Tc - M_0 - Pl = 0$$

For a given aircraft weight, speed and altitude, the above equations may be solved for the unknown lift, drag and tail loads. However, other parameters in these equations, such as M_0 , depend upon the wing incidence α which in turn is a function of the required wing lift so that, in practice, a method of successive approximation is found to be the most convenient means of solution. As a first approximation we assume that the tail load P is small compared with the wing lift L so that, from above equation, $L \approx W$. From aerodynamic theory with the usual notation

$$\tfrac{1}{2}\rho V^2 S C_L \approx W$$

3.5.2 General case of a symmetric manoeuvre: -

In a rapid pull-out from a dive a downward load is applied to the tailplane, causing the aircraft to pitch nose upwards. The downward load is achieved by a backward movement of the control column, thereby applying negative incidence to the elevators, or horizontal tail if the latter is all-moving. If the manoeuvre is carried out rapidly the forward speed of the aircraft remains practically constant so that increases in lift and drag result from the increase in wing incidence only. Since the lift is now greater than that required to balance the aircraft weight the aircraft experiences an upward acceleration normal to its flight path. This normal acceleration combined with the aircraft's speed in the dive results in the curved flight path shown in figure below. As the drag load builds up with an increase of incidence the forward speed of the aircraft falls since the thrust is assumed to remain constant during the manoeuvre. It is usual, as we observed in the discussion of the flight envelope, to describe the manoeuvres of an aircraft in terms of a manoeuvring load factor n. For steady level flight n =1, giving 1 g flight, although in fact the acceleration is zero. What is implied in this method of description is that the inertia force on the aircraft in the level flight condition is 1.0 times its weight. It follows that the vertical inertia force on an aircraft carrying out an ng manoeuvre is nW. We may therefore replace the dynamic conditions of the accelerated motion by an equivalent set of static conditions in which the applied loads are in equilibrium with the inertia forces. Thus, n is the manoeuvre load factor while f is a similar factor giving the horizontal inertia force. Note that the actual normal acceleration in this particular case is (n -1)g.

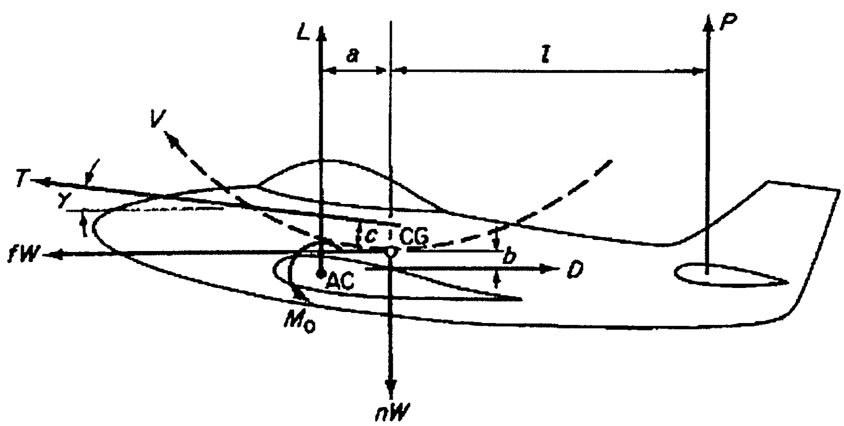


Figure17 Aircraft loads in a pull-out from a dive

For the vertical equilibrium, consider the above shown aircraft which depicts the lowest point of dive we have: -

$$L + P + T\sin\gamma - nW = 0$$

And for horizontal equilibrium: -

$$T\cos\gamma + fW - D = 0$$

The pitching moment equation about aircraft's CG is given by: -

$$La - Db - Tc - M_0 - Pl = 0$$

3.5.3 Steady pull-out: -

Let us suppose that the aircraft has just begun its pull-out from a dive so that it is describing a curved flight path but is not yet at its lowest point. The loads acting on the aircraft at this stage of the manoeuvre are shown in the figure below, where R is the radius of curvature of the flight path. In this case the lift vector must equilibrate the normal (to the flight path) component of the aircraft weight and provide the force producing the centripetal acceleration V^2/R of the aircraft towards the centre of curvature of the flight path. Thus

$$L = \frac{WV^2}{gR} + W\cos\theta$$

Since L=nW so the above equation can be written as: -

$$n = \frac{V^2}{gR} + \cos\theta$$

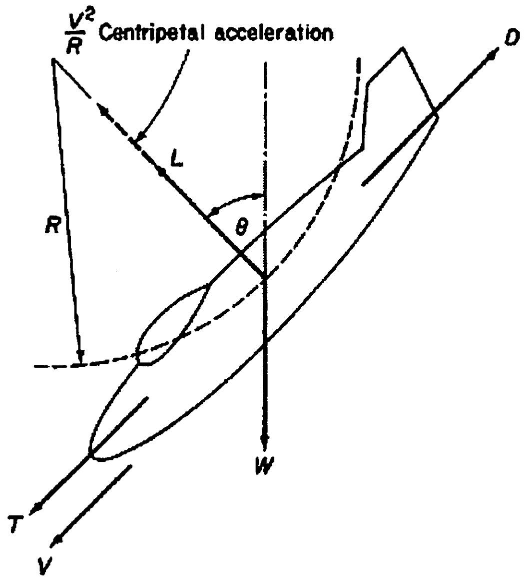


Figure 18 Aircraft loads and acceleration during a steady pull-out

At the lowest point of pull-out, θ=0 and thus the above equation can be written as: -

$$n = \frac{V^2}{gR} + 1$$

We see from either Eq. (14.15) or Eq. (14.16) that the smaller the radius of the flight path, that is the more severe the pull-out, the greater the value of n. It is quite possible therefore for a severe pull-out to overstress the aircraft by subjecting it to loads which lie outside the flight envelope and which may even exceed the proof or ultimate loads. In practice, the control surface movement may be limited by stops incorporated in the control circuit. These stops usually operate only above a certain speed giving the aircraft adequate manoeuvrability at lower speeds. For hydraulically operated controls 'artificial feel' is built in to the system whereby the stick force increases progressively as the speed increases; a necessary precaution in this type of system since the pilot is merely opening and closing valves in the control circuit and therefore receives no direct physical indication of control surface forces. Alternatively, at low speeds, a severe pull-out or pull-up may stall the aircraft. Again safety precautions are usually incorporated in the form of stall warning devices since, for modern high speed aircraft, a stall can be disastrous, particularly at low altitude.

3.5.4 Correctly banked turn: -

In this manoeuvre the aircraft flies in a horizontal turn with no sideslip at constant speed. If the radius of the turn is R and the angle of bank φ, then the forces acting on the aircraft are those shown in figure below. The horizontal component of the lift vector in this case provides the force necessary to produce the centripetal acceleration of the aircraft towards the centre of the turn. Then

$$L \sin\phi = \frac{WV^2}{gR}$$

For vertical equilibrium: -

$$L = W \sec\phi$$

From the above equations we get

$$\tan\phi = \frac{V^2}{gR}$$

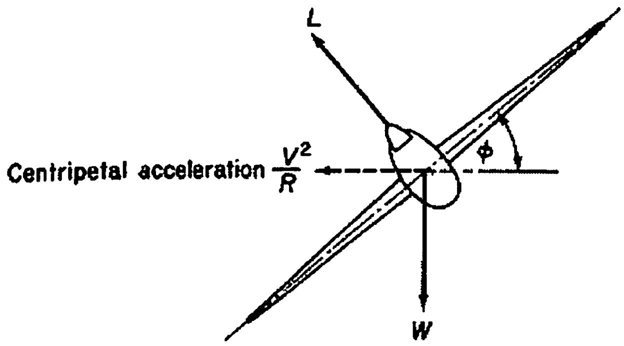


Figure 19 correctly banked turn

CHAPTER 4 Aircraft Sizing

With the increase of the payload to 65 pounds a more lift generating wing is required. So the historical wing data of previous year's team was taken and modified for the required lift. Also to maintain L+B+H = 225 inches the fuselage length was reduced to compensate for the increase in wing's length. Accordingly the sizes of elevator, aileron, flaps and, rudder were obtained to generate the appropriate moments about the aircraft centre of gravity. The final dimensions of all the aircraft components are listed below:-

4.1 Wing Sizing: -

Airfoil	Selig 1223
Span	9.83 ft
Chord	1.416 ft
Planform Area	13.93 ft^2
Maximum Wing Loading	4.66 lb/ft^2
Aspect Ratio	6.9411

Table 2 Wing Sizing

4.2 Fuselage sizing: -

Fuselage length	6.11 ft
Nose Length	1.445 ft
Fineness Ratio	10.474

Table 3 Fuselage Sizing

4.3 Horizontal Stabilizer sizing: -

Airfoil	NACA 0012
Span	3.31 ft
Chord	1.092 ft
Area	3.58 ft^2

Table 4 Horizontal Stabalizer Sizing

4.4 Vertical Stabilizer sizing: -

Aerofoil	NACA 0012
Mean Chord	1.284 ft

Vertical height	1.31 ft

Table 5 Vertical Stabalizer Sizing

4.5 Control Surface Sizing: -

Control Surface	Area
Flaps	96 in^2
Aileron	72 in^2
Elevator	72 in^2
Rudder	32 in^2

Table 6 Control Surface Sizing

4.6 Propeller Selection: -

The team decided on the **OS 61 FX** as the primary engine to be used, keeping a Magnum 61 XLS as a reserve. Both engine manufacturers recommend using propellers of size 12×6, 12×7, 12×8, 13×6 13×7 and 14 x 7. Thrust was measured through a strain-gauge test bench setup placed at the exhaust of a low subsonic open-looped 0.5m x0.5m wind tunnel. The consistent and high values of thrust across operational speeds (40 to 65 ft/s) as well as lesser ground clearance requirement (0.5 inch less) made the **14 x 7** propeller an outright choice for both main and reserve engines.

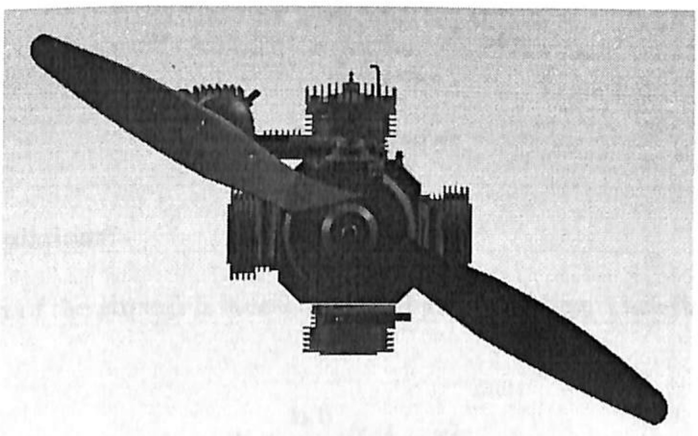


Figure 20 Propeller

CHAPTER 5 Structural Analyses and Optimization

5.1 V-n Diagram: -

The V-n diagram was constructed between velocity and load factor calculated at maximum lift. The diagram was used as a reference for all structural calculations. The theoretical velocity limit for structural safety was found to be 88 ft/s, with the positive limit load factor intersecting at 3. The gust load factors were calculated according to FAR Section 23.341.

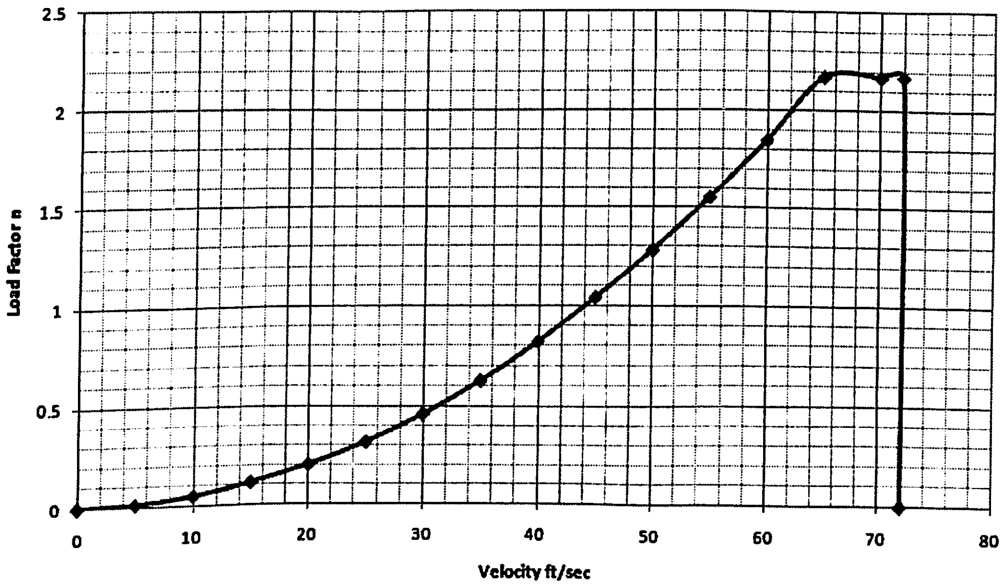


Figure 21 V-N diagram

5.2 Structural Calculations: -

The structural design of the aircraft is based on the elliptical loading. The elliptical load on the wing is given by:

$$W = \frac{W0}{L}\sqrt{(L^2 - X^2)}$$

The value of major and minor axis is 59 inch and 1.89 lbs. respectively. The value of major and minor axis is calculated from the following formulae.

$$\frac{\pi ab}{2} = 175$$

Where a=major axis=59 inches
b = minor axis

W_o = maximum load

L = Semi span

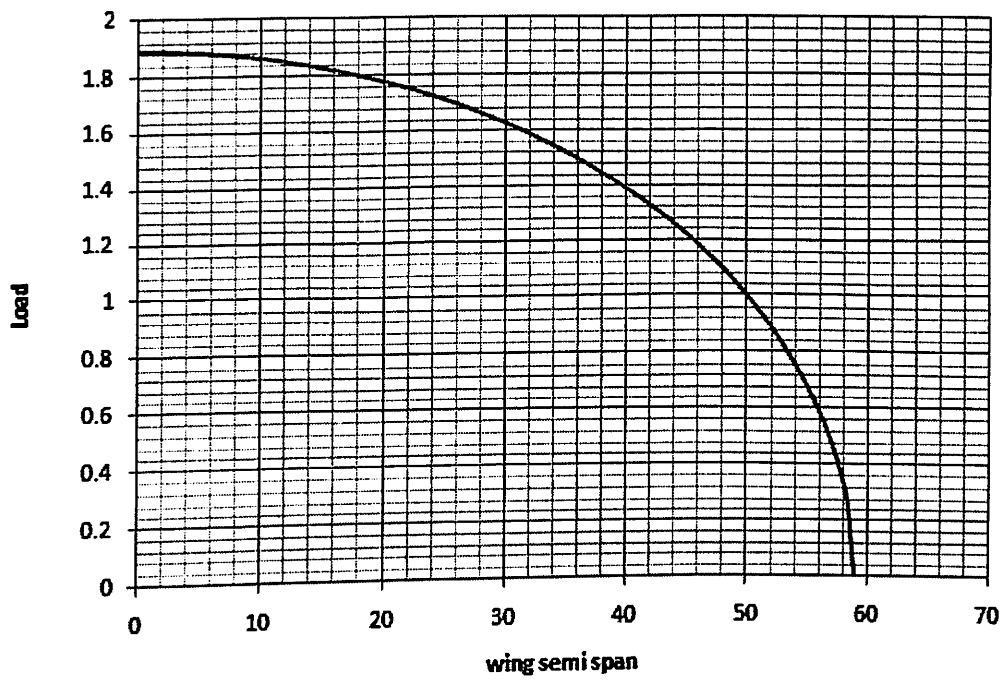


Figure 22 Elliptic loading Curve

Shear stress of elliptical loading is given by:

$$EI\frac{d^3y}{dx^3} = -\int Wdx$$

$$\tau = -\frac{Wo}{l}\left[\frac{x}{2}\sqrt{L^2 - x^2} + \frac{L^2}{2}\sin^{-1}\frac{x}{L} + \frac{PiL^2}{4}\right]$$

The value of maximum shear stress is 87.57 lb.-inch at the root i.e. at x=0 while at the wing tip(x=59) it is zero.

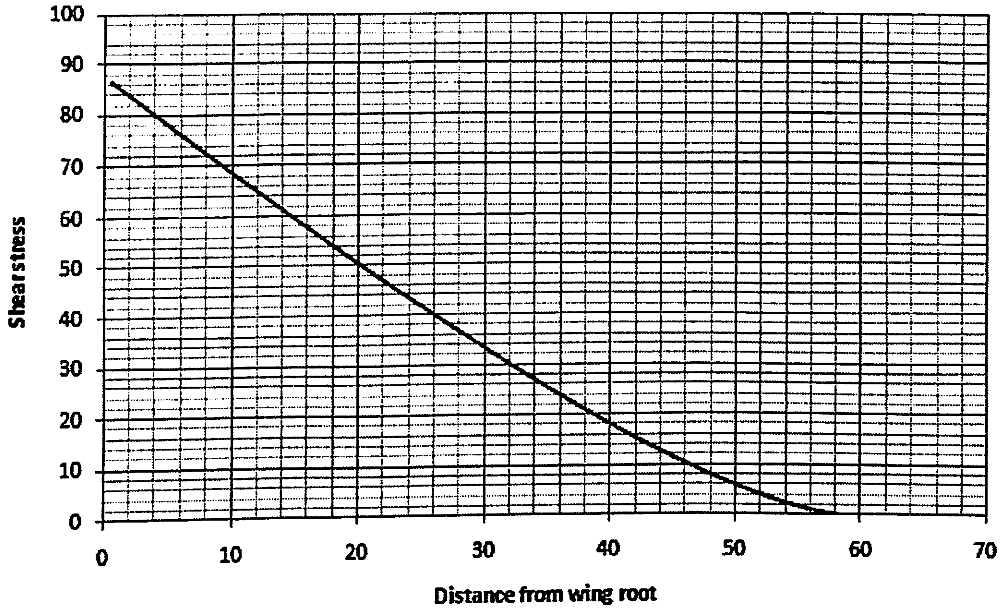


Figure 23 Shear stress variation along the wing

The bending moment equation is given by: -

$$EI\frac{d^2y}{dx^2} = -\int \tau dx$$

$$M = \frac{W0LPix}{4} - \frac{W0}{L}\left[-\frac{(L^2 - x^2)^{\frac{3}{2}}}{6} + \frac{L^2}{2}\left(x\sin^{-1}\frac{x}{L} + (L^2 - x^2)\right)\right]$$

The value of maximum bending moment is 2193.03 at the root and zero at the tip.

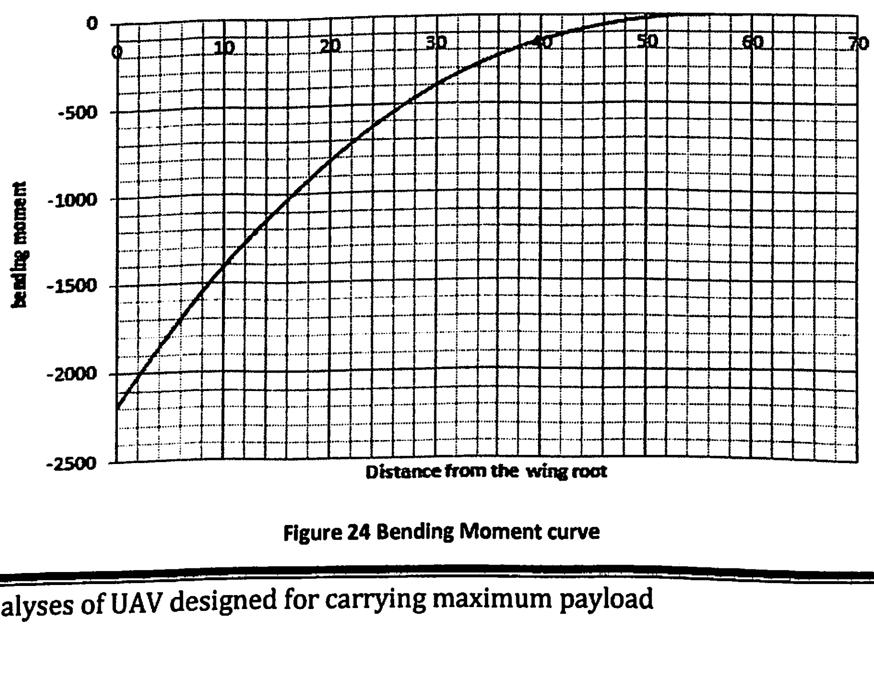


Figure 24 Bending Moment curve

5.3 Wing design:-

Most important part of an aircraft is its wing which provides lift. Wing design holds the key to carrying maximum payload. Rectangular wings were considered as they are simple to fabricate and more structurally efficient in low speed applications. The wing is divided into three parts, the middle part of 3.8 feet while the left and right parts are of three feet each. The solid circular spar is used as the joiner between the wings. A number of stringers are also used to stiffen the structure. Wing span 118 inch and chord 17 inch. Rib thickness is 1/8 inch and kept at a distance of 5.48 inches. The spacing at the tips is of 1 inch. Material used is balsa. Two hollow circular spars are used one having its centre at 1/3 of the chord and other at 2/3 of the chord. The dimension of the rear spar is .6inch outer diameter and 0.55 inch inner diameter while the dimension of front spar is outer diameter 1inch and inner diameter 0.85 inch. The spar is made of aluminium alloy 2024-T3. The wing is stiffened using a number of rectangular members in between the ribs. These members have slots being cut in them to reduce the weight. The members are covered using balsa stringers across the span thus creating an I- section which further resists bending of the structure. The formula used for bending moment is:-

$$\frac{M}{I} = \frac{\sigma}{y}$$

M= maximum bending moment= 1293.03

I= Moment of inertia= $\frac{\pi}{4}(Do^4 - Di^4)$

D_o= outer diameter

Di= Inner diameter

σ= Ultimate tensile strength of the aluminium

y= distance of outer most fibre from the neutral axis

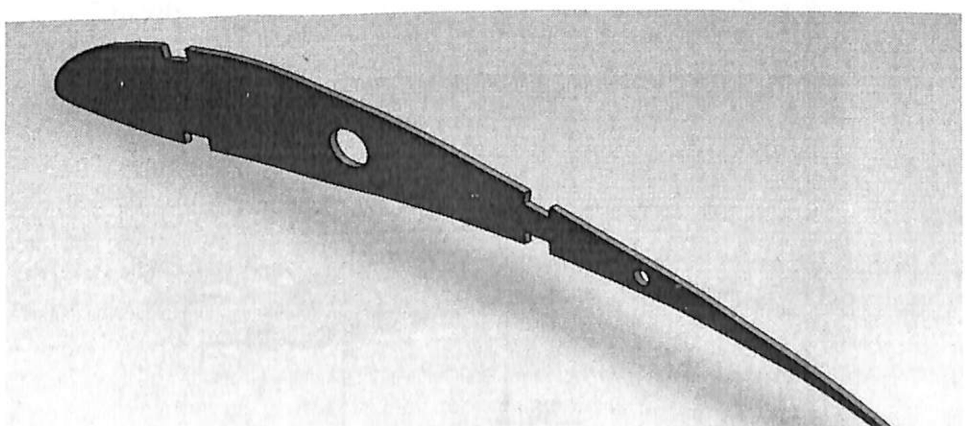


Figure 25 S1223 Aerofoil

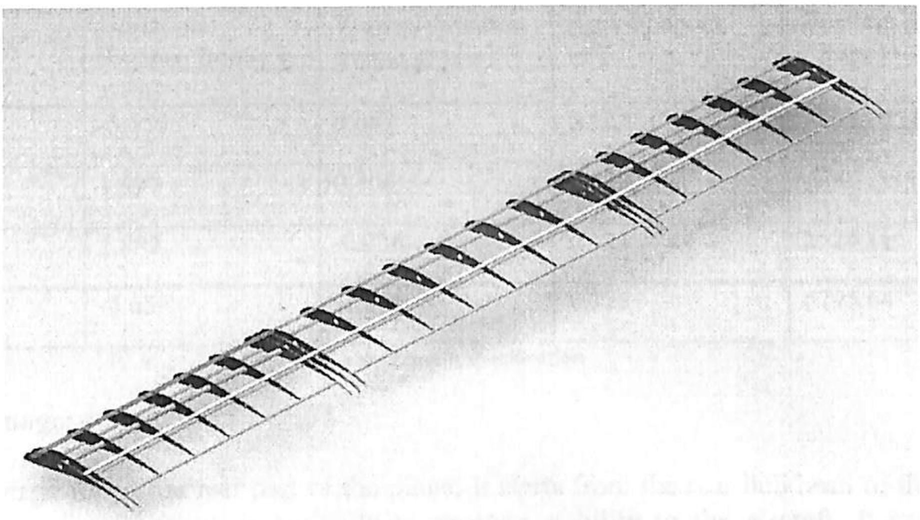


Figure 26 Wing structural model

5.4 Idealization Technique: -

In this method every stringer is considered as a boom and then the bending stress which every stringer is resisting is calculated. Centre of Gravity of rib is at a distance of 7.44 in from the leading edge and 1.18 in from the chord line.

$$\sigma_z = \left(\frac{MyIxx - MxIxy}{IxxIyy - Ixy^2}\right)x + \left(\frac{MxIyy - MyIxy}{IxxIyy - Ixy^2}\right)y$$

$M_x = 2193.03$ lb-in, $I_{xx} = 0.328$ in^4, $I_{yy} = 6.93$ in^4, $I_{xy} = 0.412$ in^4

$\boldsymbol{\sigma_z = 430.25x - 7236.99y}$

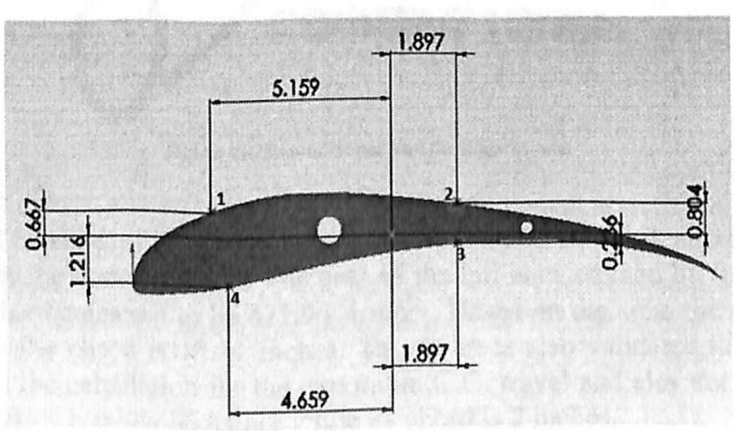


Figure 27 Boom Location

Boom	Horizontal distance from c.g, x (in)	Vertical distance from c.g, y(in)	Area of boom (in^2)	Bending Stress (lb/in^2)
1.	-5.159	0.667	0.125	- 7046.732
2.	1.897	0.804	0.125	-5002.355
3.	1.897	-0.236	0.125	2524.11
4.	-4.659	-1.216	0.125	6795.64

Table 7 Boom specification

5.5 Empennage:-

The empennage forms the rear part of the plane. It starts from the rear bulkhead of the fuselage. The function of empennage is basically to provide stability to the aircraft. It comprises of horizontal stabilizer and the vertical tail. The horizontal stabilizer is used to provide longitudinal stability whereas vertical tail is used for lateral stability. In our design we have used a conventional tail configuration for the ease of fabrication and affordability as mentioned in the systems report.

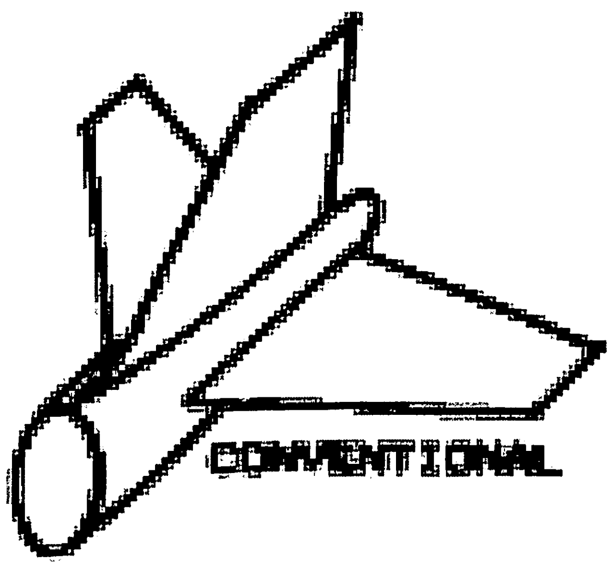


Figure 28 Conventional Tail Configurations

The horizontal tail consists of NACA 0012 symmetrical aerofoil. The leading edge spar is used to avoid twisting of the horizontal tail. The area of the tail is calculated by the tail volume ratio formula and its value comes out to be 471.96 $inch^2$. Based on the area, the span is taken equal to 36 inches while the chord is 13.11 inches. The result is also validated through the moment equation. Based on the calculation for the maximum C.G. travel and elevator effectiveness ratio the span of the elevator is taken 36 inches while its chord is 2 inches.

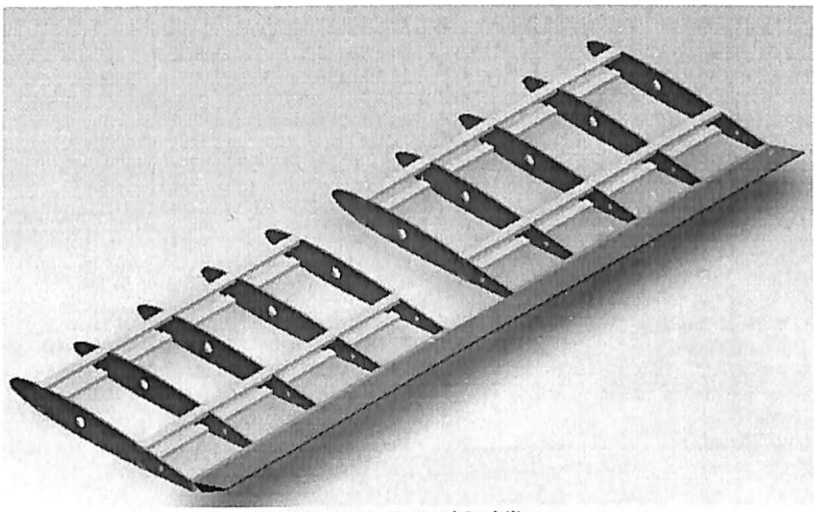


Figure 29 Horizontal Stabilizer

The vertical tail is also made of NACA 0012 aerofoil. The height of the vertical stabilizer is 1.31 feet while its mean aerodynamic chord is 1.284 feet. The leading edge spar is inclined at an angle of 108.76 degrees

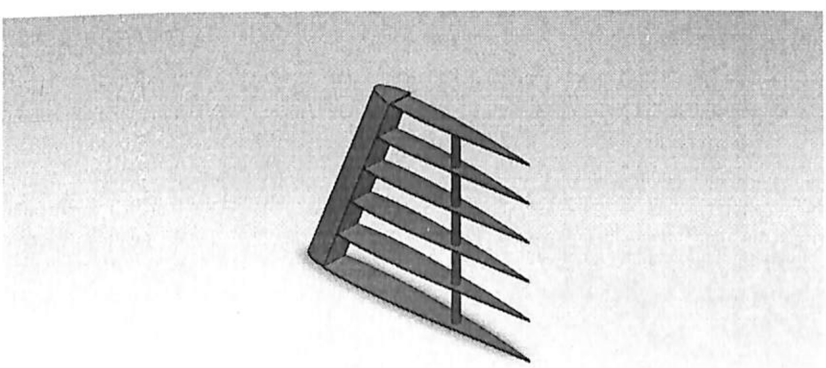


Figure 30 Vertical Stabilizer

5.6 The fuselage: -

Various types of fuselage designs were considered which includes monocoque, semi-monocoque, truss structure etc. The analysis and the literary review suggest that a semi-monocoque structure solves the purpose of the design as it is lighter and stronger than the other structures mentioned above. The fuselage has a total length of 73 inches while its maximum frontal area is 49 inches. The fineness ratio is 10.42. The skinned structure of the fuselage is shown below.

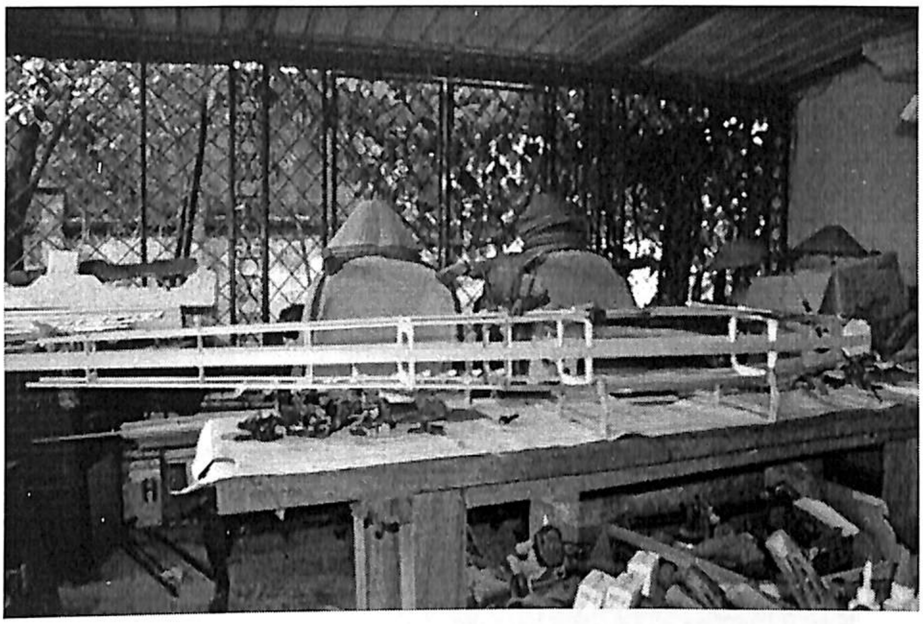


Figure 31 Fuselage Structure

Chapter 6 Fabrication and final model

Though the team has prepared the initial prototype of the plane based on the calculations and aerodynamics, but still we were open to new ideas and challenges that might be faced during fabrication and test flight. The subsequent paragraphs demonstrate the design changes and the fabrication procedure of each part.

6.1 The wing and winglets: -

The fabrication of wings was mainly according to the design. The ribs of thickness .125 inches were chosen which were kept at a distance of 5.5 inches. The rib spacing at the wing tips and the places where parts of the wing need to be joined, are kept 1.125 inches to provide extra strength to the structure. The leading as well as trailing edge of the wing was covered with 1/16 inch balsa sheet to prevent the twisting in the wing and for the ease of covering the wing with monocoque sheet. The wing was monted on the fuselage with the help of four nuts.

Figure 32 Final Wing Design

Though the span and thus the aspect ratio (6.94) of the wing was quite large to account for the induced drag but still team decided to reduce it further by incorporating the winglets which will provide an advantage over the other teams. It was firstly decided to use end plate winglets of rectangular cross-section made of plywood. Apart from increasing the weight it was also creating drag which was not beneficial over the induced drag. Thus team decided to go with the balsa winglet which was lighter and also serves the purpose.

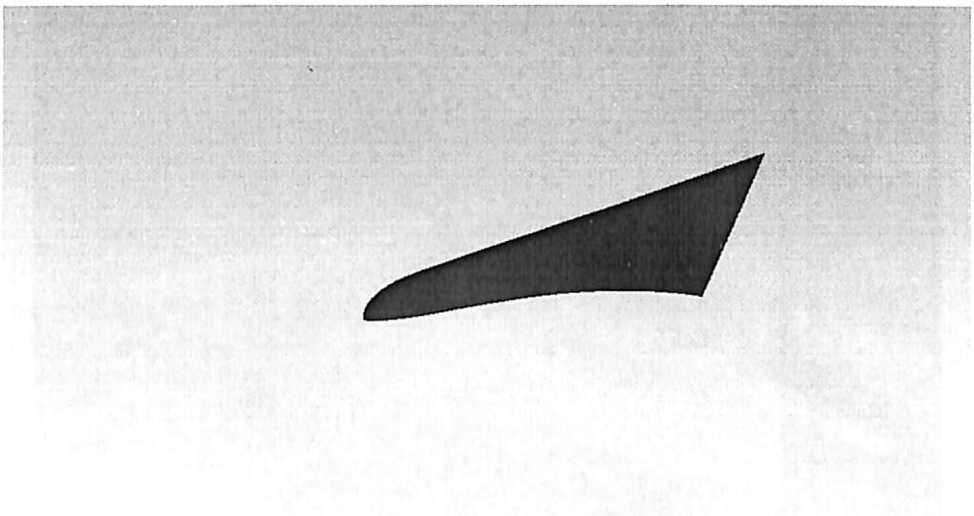


Figure 33 the Winglet

6.2 Struts: -

Initially the strut based design was not thought to be an option since strut being an outer member not only produces unnecessary drag but also increases the weight. Rather the idea of the design was to make the wing strong and stiff enough to restrict the bending moment. But the initial flight test suggests that due to the large span of the wing, the tip of the wing tend to touch the ground while encountering gust during landing. Thus to ensure the safety of the structure, struts are also incorporated in the design so that the aircraft can have safe flight at all atmospheric conditions. Two hollow aluminium rods are attached on either side of the wing to reduce the deflection of wing in case of slid-slip landing and gusts.

6.3 The fuselage: -

The dimensions of the fuselage are according to the initial design. There is a little variation in the shape of the fuselage. The previous fuselage design has multiple tapers which was difficult to fabricate with the available material. Another disadvantage of the multiple taper was that it was obstructing the smooth flow over the horizontal stabilizer by creating wake aft of the region of the taper. All these results were computed using CFD and thus finally a reconsideration of the design was done and the final design has only single taper aft of the mid-section. The fabrication of the fuselage was done using longerons and formers of balsa and plywood respectively. This is similar to the semi-monocoque construction. This type of structure helps to reduce weight and provide excess of space to keep the payload and other equipment like batter, receiver, fuel tank etc. The complete structure was then covered with balsa sheet to restrict the twisting of the structure.

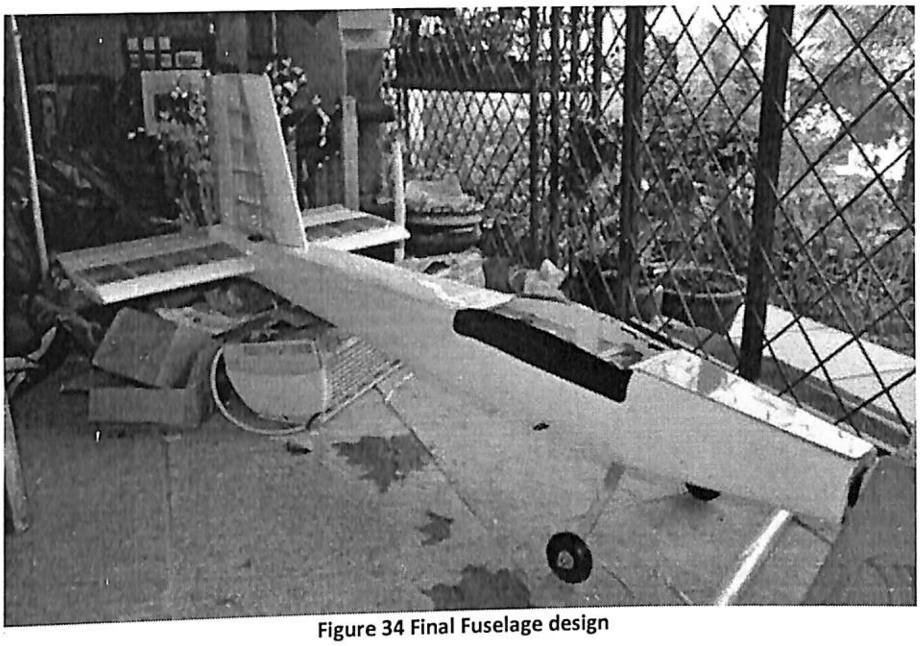


Figure 34 Final Fuselage design

6.4 Landing Gear: -

The landing gear is an important feature of the structure. Since it is placed close to the centre of gravity of the aircraft, it takes maximum impact load during the landing. Though the design of the structure is such that the load is distributed among the frames of fuselage, but the structural analysis of the landing gear shows that it still has tendency to expand under impact load. A stronger element would have solved the purpose but will add to the weight of the plane. So the team had redesigned the landing gear using two aluminium frames as shown below. The frames are such joined that they form two triangles which is a redundant structure. This will not only provide resistance to the extension of the landing gear but also will provide an extra strength to take impact loads during landing.

Figure 35 the Landing Gear

6.5 Payload bay access: -

The team has previously decided to cut a slot in the fuselage just below the wing to put the payload into the aircraft. But at later stage, this design concept was discarded since a slot in the fuselage would mean the removal of the material and thus decreasing the strength of the structure. Thus an alternative to this was found out. The payload would be kept in the fuselage from the slot being cut on the top rear part of the wing. The opening is being covered using balsa block incorporated with spring loaded lock. The payload would be kept close to the centre of gravity and the slot will be covered and locked during flight.

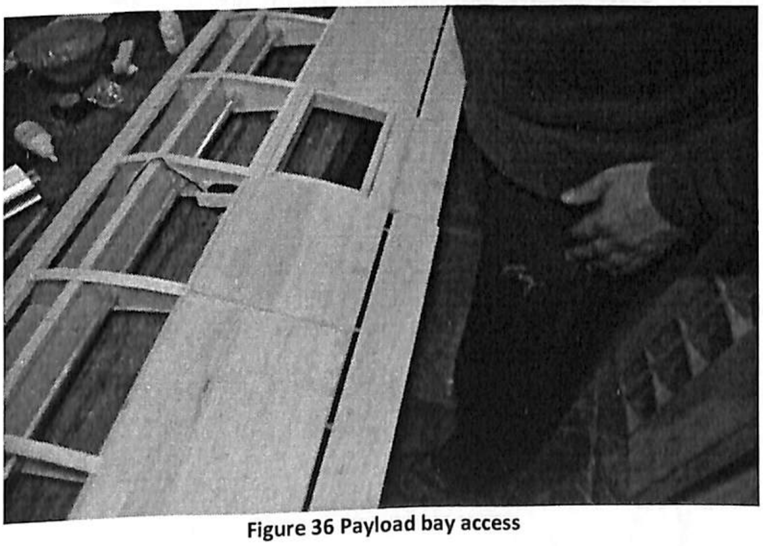


Figure 36 Payload bay access

6.6 Software analysis: -

The analysis of the solid wing is done using ANSYS workbench. A static structural analysis is performed. The maximum load of 175 lbs. was applied on the upper surface of the wing which is the maximum value of lift obtained for this wing. The maximum deformation is at the wing tips and its value is .000727mm which is acceptable.

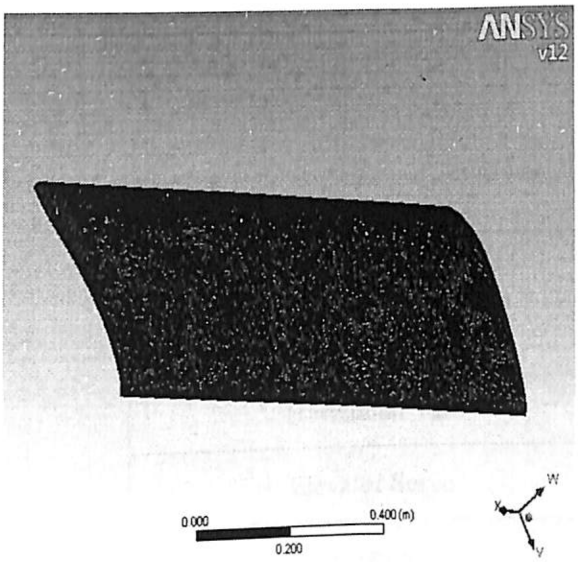


Figure 37 Meshed Geometry

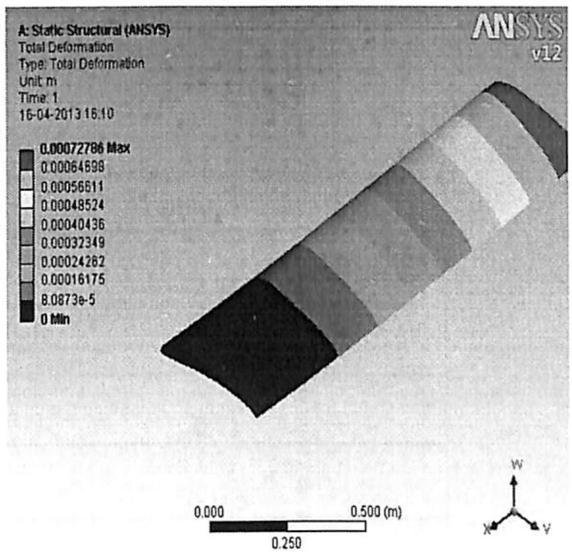


Figure 38 Total Deformation

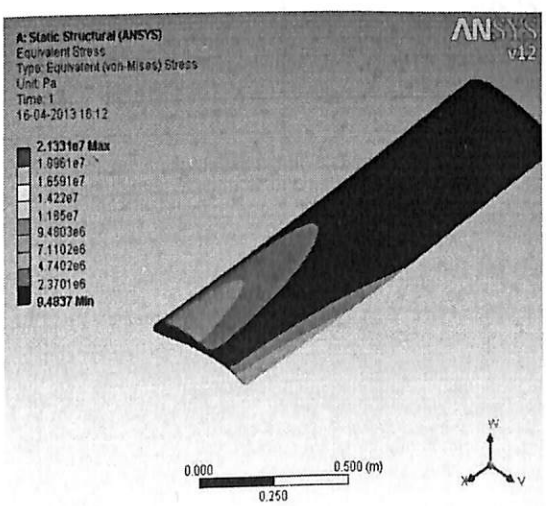


Figure 39 Von misses stress

The von misses stress contour has no critical points hence the wing structure is safe for flight.

CHAPTER 7 Weight and Balance

The centre of gravity of the whole aircraft was balanced at **42.17%** of wing MAC to give pitch authority to the aircraft, with a static margin of **20%.** To ease out longitudinal trimming, the payload box was placed with its centre of gravity coinciding with that of the aircraft. The gross weight of the aircraft is **9.95 lbs**, and the distance of the aircraft C.G from propeller hub towards the tail is **27.12 inches.**

Component	**Weight, W (lb)**
Fuselage	2.75
Wing	2.32
Horizontal Tail	0.71
Elevator Servo	0.165
Engine	1.47
Avionics	0.511
Vertical Tail	0.55
Rudder Servo	0.71
Fuel Tank	0.154
Main Landing Gear + Nose Wheel	0.61
TOTAL	**9.95**

Table 8 Weight balance

CHAPTER 8 Fabrication Scheduling and Budgeting

8.1 Manufacturing: -

The fabrication process is a very prominent and requires great precision while scaling down according to the calculated dimensions. Hence it is done according to the steps which did not require further amendments or resizing.

- Wing Structure
- Fuselage
- Horizontal and Vertical Tail
- Landing Gears
- Push-pull Rods

8.2 Material Selection: -

For fabrication of the aircraft capable of carrying maximum weight with respect to the minimum empty weight, Balsa wood is regarded as the main constituent in the aircraft structure. Density of Balsa wood varies from 8 to 12 lb/Ft^3 . Lathe machinery and drill tools were used for precise cutting of the wing structure. Fuselage comprises of plywood at the payload bay and rest is fused with balsa wood. SemiMonocoque structure with formers and bulkheads are applied in order to provide strength. Fabrication was monitored to minimize wastage and reduce cost of production.

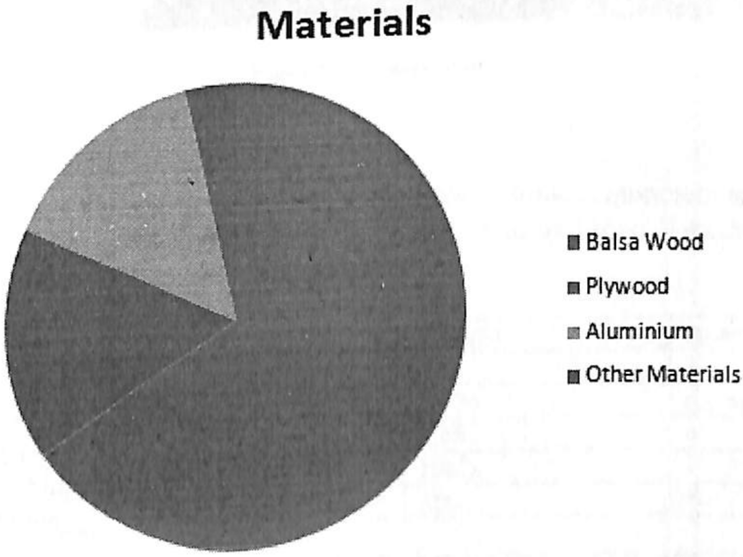


Figure 40 Material Proportion

8.3 Work Flow Planning: -

Fabrication is an end product of calculations and estimations where we are actually able to visualize that what was on computer or drawing board has received its shape. It was started in first week of December to the mid-week of February, totally closing to three months in prototyping, testing and reviews.

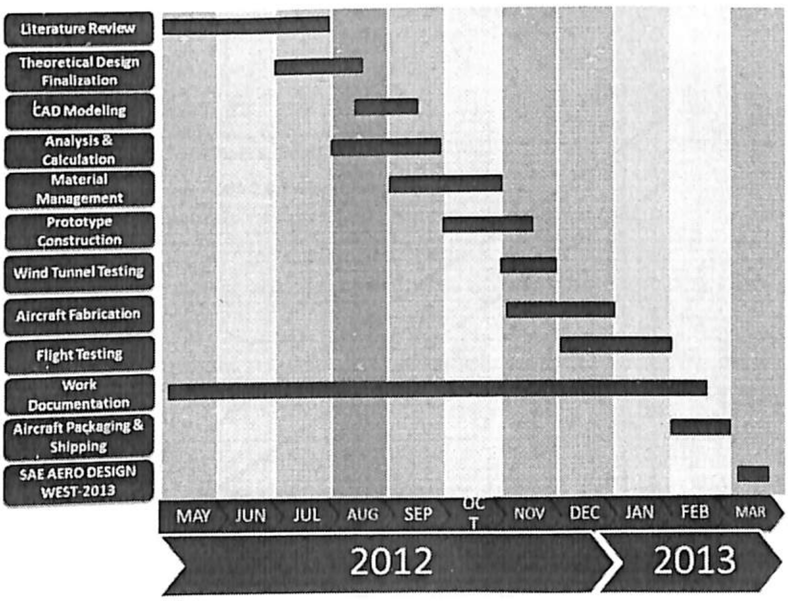


Figure 41 Schedule chart

8.4 Cost Estimation: -

For optimum results and best-in class performance all the avionics equipment and components are newly purchased from authorized dealers. The engine required special procurement & was done from abroad to ensure its quality and performance.

S.No.	Expense	Cost($)	Quantity	Cost($)
1.	SAE Registration Fee	650	1	650
2.	Aircraft Fabrication[Engine, Tools]	2,728	Assorted	5,770
3.	Travel Expense	1,682	4	6,728
4.	Packaging & Shipping	1,442	1	1,442
5.	Accommodation	560	4	2240
	Total ($)			**16,830**

Table 9 Overall budget

8.5 Scoring Analysis: -

Understanding the competition's scoring mechanics gives a team an insight into how and where to optimize performance. There are three scoring areas, the design report (50 points), an oral presentation (50 points) and flight score. The flight score has no upper limit, thus accounting for a majority of the total score, thus determining a team's in-class competition ranking. The flight score is affected by the team's operational availability (success rate), maximum weight lifted in a single round over the course of the Aero Design East weekend, accuracy in predicting maximum payload (maximum of 20 bonus points) and penalties (for which points are deducted). The team was placed 9 th last year, with an adjusted flight score of 81.20, leading to an adjusted overall score of 165.17, while the winners in the Regular Class got a flight score of 112.400 and an overall score of 221.31. It is apparent that in order to be favourably placed in the competition, a flight score between 140 and 170 is required along with a score of close to 85 points (report + presentation).

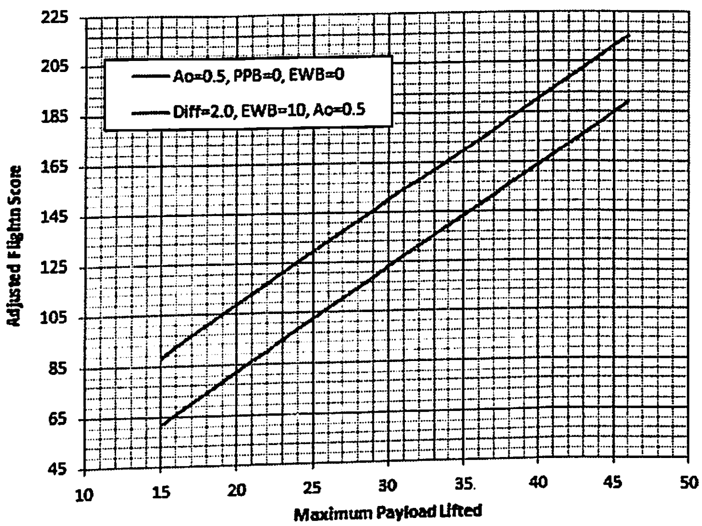


Figure 42 Scoring analysis

CHAPTER 9 CONCLUSION

The University of Petroleum & Energy Studies 2013 SAE Aero Design team has developed a robust aircraft that meets all the competition requirements. Team SkyHawks has predicted the maximum payload of 28.39 lbs under sea level condition with sum of dimensions 224 inches. In this year's competition there is an increase in maximum weight of aircraft from 55 lbs to 65 lbs, so the 9.83ft wingspan has given more freedom in the design process than the previous year and also aided in enhancing maximum payload carrying capacity. Adding more wing area increases the payload carrying capacity due to increase in lift, but takeoff speed decreases as drag increases. Also to reduce gross weight of aircraft a semi monocoque fuselage is used. In order to design a successful aircraft, an engineering trade-off was employed. The result of this competition was not as expected. The team secured an international rank of 25 and came second in India. The plane which was built for the competition was crashed due to unexpectedly high gust velocity of 40km/hr. which led to the loss of stability and thus crash. Another major cause of crash was the excess use of cyno on the hinges of the control surface rendering them ineffective. The plane was not built as designed due to fabrication constraint and unavailability of proper machines.

References

- Roskam, Jan. (1985). *Airplane Design.* Ottawa, Kansas: Roskam Aviation and Engineering Corporation.
- Lennon, A. (1996). *Basics of R/C Model Aircraft Design.* Ridgefield, CT: Air Age Media Inc.
- Raymer, D. (1992). *Aircraft Design, A Conceptual Approach.* Washington D.C: AIAA Inc.
- Nelson, C. Robert. (1998). *Flight Stability and Automatic Control.* New York: McGraw-Hill Companies, Inc.
- Anderson, J. D., Jr. (1979). *Introduction to Flight.* New York: McGraw-Hill Companies, Inc.
- Clancy, L. J. (2006). *Aerodynamics.* D.N. Road, Mumbai: Sterling Book House.
- Anderson, J. D., Jr. (2010). *Aircraft Performance and Design:* McGraw-Hill Companies, Inc.
- Selig, Michael. (2006).UIUC Aerofoil Data Site. "Coordinates Database." http://www.ae.uiuc.edu/m-selig/ads/coord_database.html
- Anderson, J.D., Jr (1979) *Computational Fluid Dynamics:* McGraw-Hill Companies, Inc.

Appendix A
Performance, Stability and Control

A.1 Drag Estimation: -

For most conventional aircraft, we divide drag into two main parts, lift dependent drag and non-lift dependent drag. The first part is called induced drag (Di) because this drag is induced by lift. The second part is referred to as zero-lift drag (Do), since it does not have any influence from lift. In order to calculate zero-lift drag coefficient of an aircraft, we must include every contributing item. The C_{Do} of an aircraft is simply the summation C_{Do} of all contributing components.

$$C_{Do} = C_{Dowing} + C_{Dofuselage} + C_{DoHT} + C_{DoVT} + C_{DoLG} + C_{Doint}$$

So zero lift drag was calculated for all major components like fuselage, wing, horizontal tail, vertical tail which is mentioned below. The zero lift drag was taken to be 0.025 to account for landing gear and interference drag. This is a conservative assumption but will not have large effect on payload prediction, so it is an acceptable assumption.

Components	Estimated Drag Coefficient
Fuselage	0.0016
Wing	0.010
Horizontal Tail	0.002
Vertical Tail	0.00108
LG + Interference	0.025

Table 10 Drag estimation

The total wetted area of aircraft is **48.257 ft**

The drag polar equation is shown below:

$$C_d = C_{do} + \frac{1}{\pi e AR} C_l^2 = 0.04 + 0.057\, C_l^2$$

The maximum lift to drag ratio is **10.**

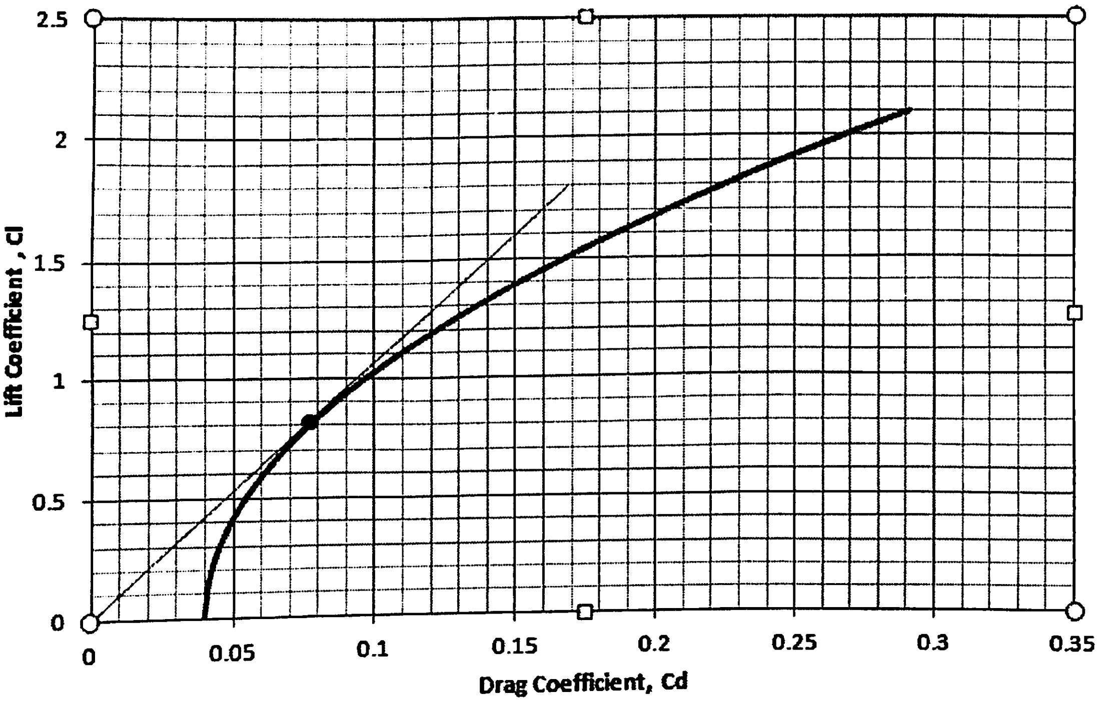


Figure 43 Drag Polar

A.2 Power Required: -

The maximum horsepower of engine is 1.9hp at 16000 rpm. The efficiency of engine was tested and was calculated as 60%. Using this efficiency the actual output of the engine comes out to be 1.14 hp. From the drag polar obtained the power required curve was obtained as in fig . The maximum velocity of aircraft is **72 ft/sec**

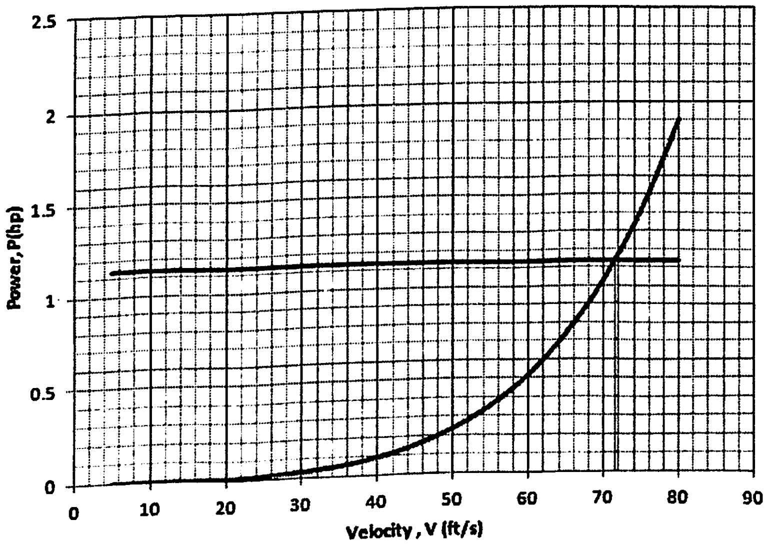


Figure 44 Power Required Curve

A.3 Take off Distance: -

The take-off distance as mentioned in rules was not to exceed 200 ft, keeping it in mind and considering various other parameters. Take off distance was calculated using formulae

$$S_{to} = \frac{1.44\, W^2}{g\rho S Clmax\, T}$$

- Take off distance varies directly proportional with W^2
- Take off distance is proportional to $1/\rho$
- From the formulae we can observe that the take-off distance can be decreased by increasing wing area, Clmax, and thrust.

A graph as shown is plotted between Gross Take-off Weight and Take-off Distance.

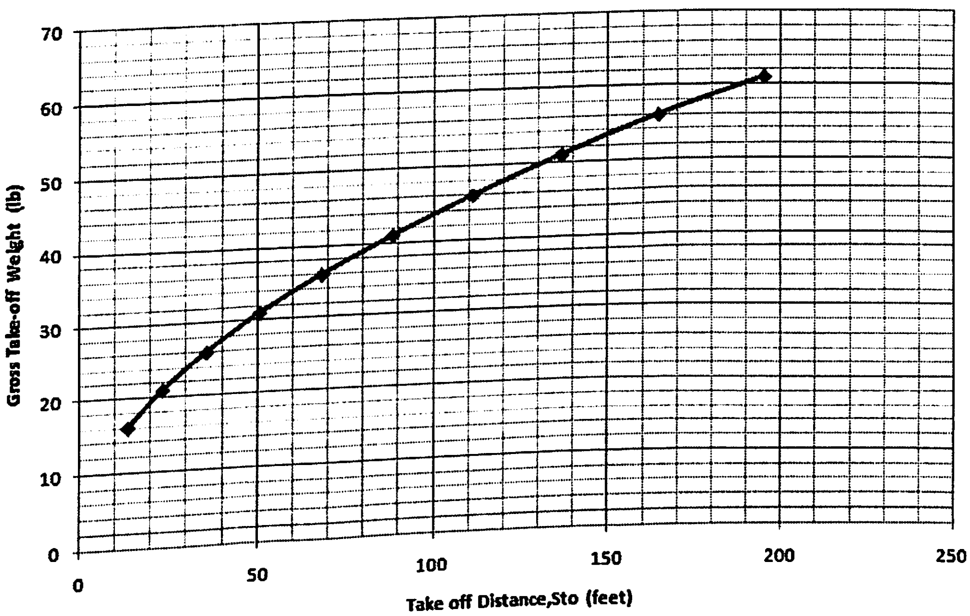


Figure 45 Take-off distance curve

A.4 Landing Distance: -

The landing distance as mentioned in rules was not to exceed 400 ft, keeping it in mind and considering various other parameters landing distance was calculated using formulae

$$S_{landing} = \frac{1.69\ W^2}{g\rho S Clmax\,(D+\mu\,(W-L))}$$

- Landing distance is directly proportional to W^2 i.e. weight of airplane

- Landing distance can be decreased by increasing wing area and C lmax
- Landing distance can also be reduced by using ground spoilers which reduces speed after touchdown and hence decrease landing run.

Landing distance was calculated and represented on graph between gross weight of airplane and landing distance.

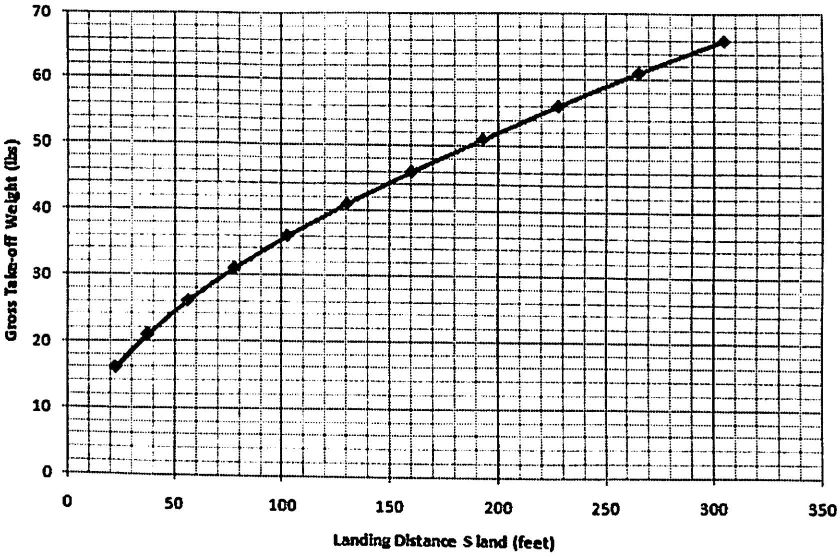


Figure 46 Landing Distance Graph

A.5 Longitudinal Static Stability: -

To determine the longitudinal stability of the aircraft, the pitching moment and static stability characteristics of the wing and tail were calculated theoretically. The location of neutral point and centre of gravity were determined for the static margin of 20% of the mean aerodynamic chord. The neutral point was found to be 8.738 inches aft of the wing leading edge and centre of gravity is at a distance of 5.338 inches aft of the wing leading edge.

- **Wing contribution:**

The static stability characteristics of the wing after theoretical calculations were obtained as listed below:-

Parameter	Value	Unit
$C_{L\alpha W}$	4.019	rad^{-1}
C_{LoW}	0.4208	-
$C_{m\alpha W}$	0.2572	rad^{-1}
C_{moW}	-0.263	-

Table11 Wing stability characteristics

The wing pitching moment equation is thus given as:-

$$C_{mcgw} = -0.263 + 0.2572\alpha_W$$

As C_{moW} is negative and $C_{m\alpha W}$ is positive the equation satisfies the destabilizing nature of the wing, as expected.

- **Tail contribution:-**

The static stability characteristics of the tail after theoretical calculations were obtained as listed below:-

Parameter	Value	Unit
V_H	0.764	-
$C_{L\alpha T}$	4.75	rad^{-1}
$C_{m\alpha T}$	-2.2943	rad^{-1}
C_{moT}	0.2349	-

Table12 Tail stability characteristics

The tail pitching moment equation is thus given as:-

$$C_{mcgT} = 0.2349 - 2.2943\alpha_T$$

As C_{moT} is positive and is $C_{m\alpha T}$ negative the equation satisfies the stabilizing nature of the tail, as expected. Graphically the static stability characteristics of the aircraft can be represented with the help of a plot between C_m and α of the aircraft. The plot produced from the theoretical calculation is shown below:-

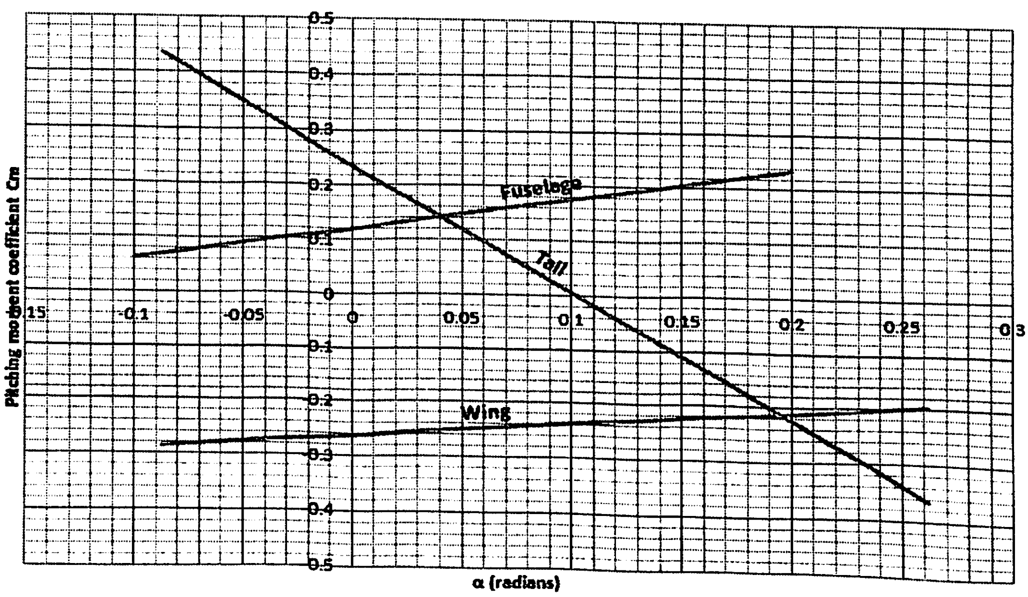


Figure 47 Longitudinal stability characteristic curve

By adding the equations of the three curves as shown above the complete pitching moment equation of the aircraft is given as:-

$$C_m = 0.09 - 1.466\alpha$$

With the positive value of C_{mo} and negative value of $C_{m\alpha}$ the aircraft is stable at a positive angle of attack as expected. Neutral Point is the point at which $C_{m\alpha}$ is zero i.e. C_m is constant. It is calculated using formula:

$$\frac{Xnp}{c} = \frac{Xcg}{c} + SM$$

This gave the Neutral Point as **8.27 inches** from leading edge.

Appendix B

CFD Analysis

Aircraft configuration is validated using CFD tool and results were used to further modify the design to attain optimum performance. CFD analysis was carried out using ANSYS.

B.1 Pre-processing: -

Designing of CAD model was done in SolidWorks design software. The model was then saved in IGES format which was then imported to ANSYS 14.0. An outer domain was created which is 15 times the length of the airplane. Using ANSYS ICEM Unstructured mesh was generated on the airplane. Prism mesh is created up to 7.14 mm (boundary layer effects are significant) and thereafter tetrahedral mesh was created.

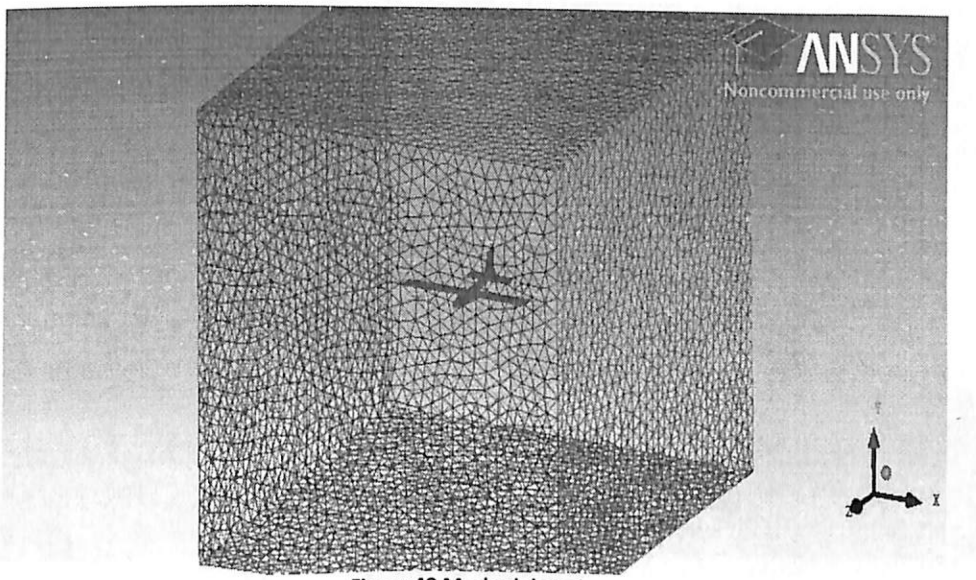


Figure 48 Meshed domain

B.2 Solver: -

Problem was solved in CFX-Expert tool of ANSYS. While solving boundary conditions, initial conditions and flow properties were assigned to the software.

Flow Conditions	
Velocity	15 m/s
Static Temperature	288 K

Turbulence	Medium intensity and Eddy Viscosity Ratio
Reynolds Number	2.151e+6
Turbulence Model	Steady State Turbulent
Pressure	101325 Pa

Table 13 Flow conditions

B.3 Post-processing: -

In Post Processing results were analysed and different colour plots were obtained like pressure plot, velocity plot, streamline pattern etc. and further Lift force was calculated which came out to be 35.67 lbs. In the velocity contour we can observe the velocity variation around the aircraft with maximum velocity over the top surface of aerofoil as expected.

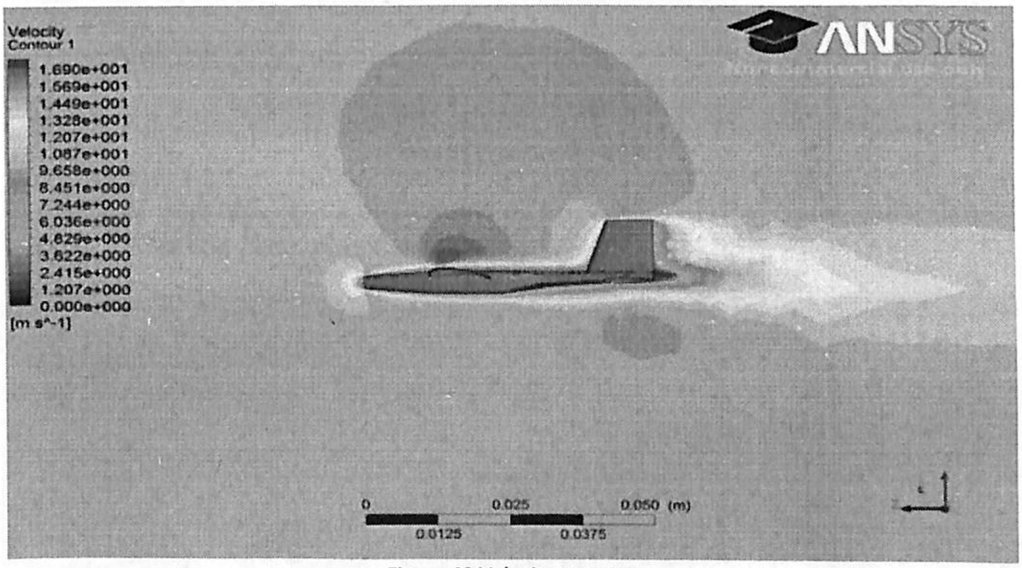


Figure 49 Velocity contour

In the pressure contour a high pressure zone is observed on the bottom surface of aircraft and low pressure zone on the upper surface, which is the cause of lift generation.

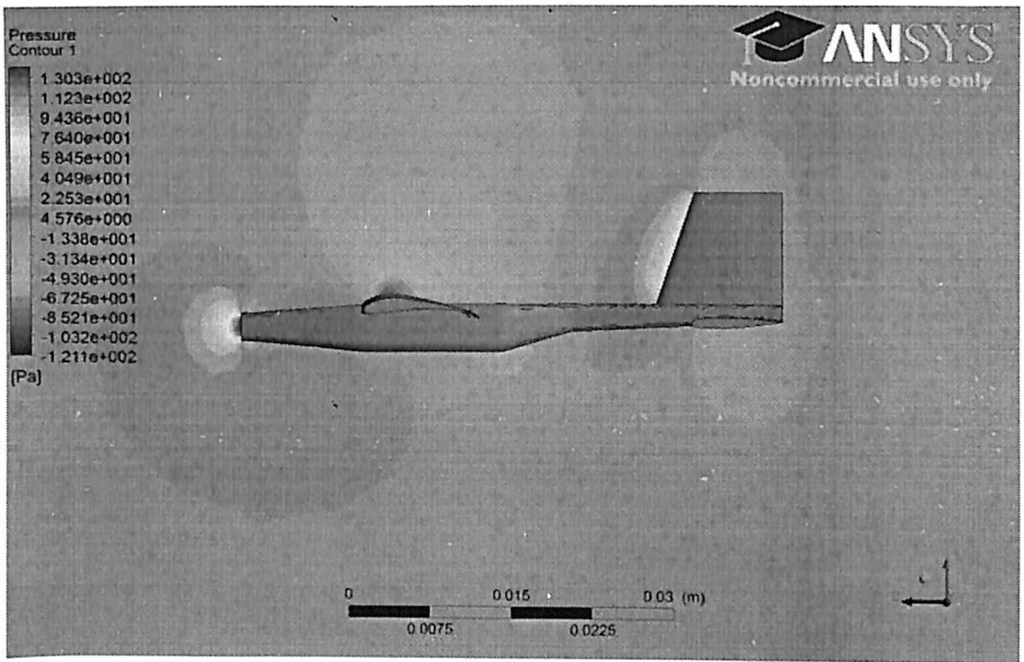


Figure 50 Pressure contour

Strong tip vortices were observed around the wing tips of aircraft that could lead to large induced drag. So to avoid this end plate wing tips were used.

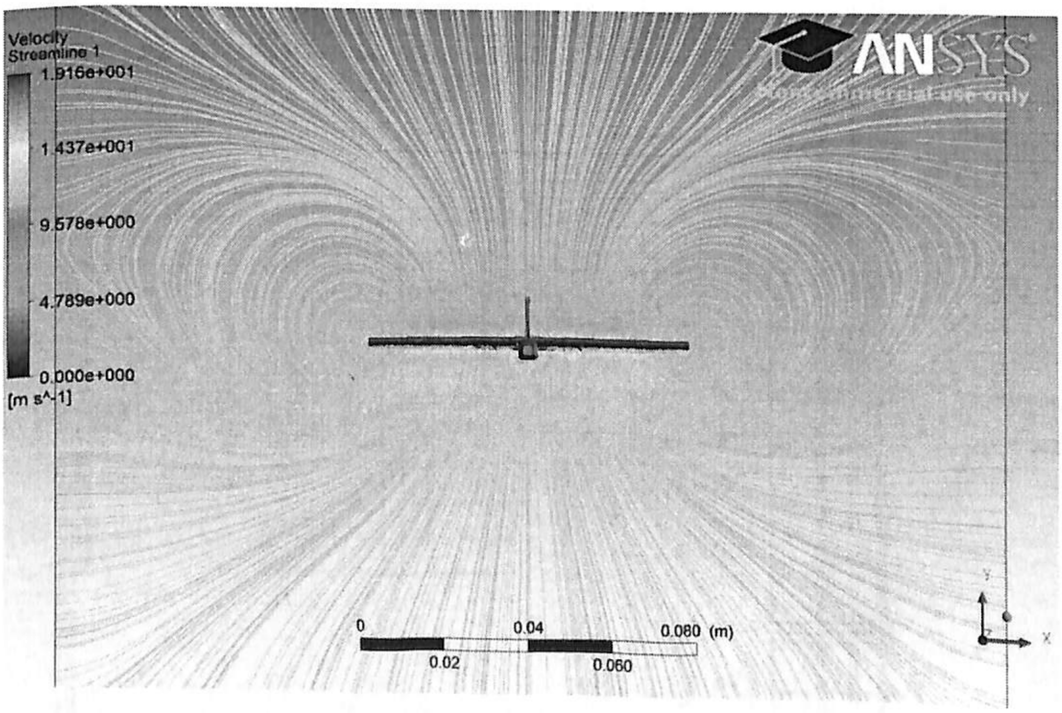
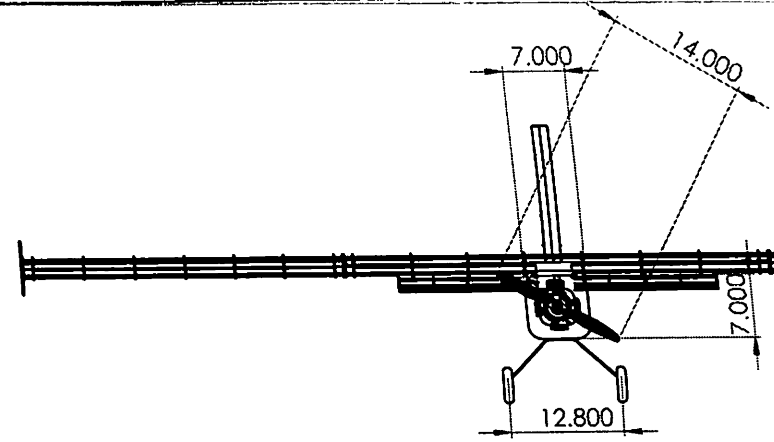
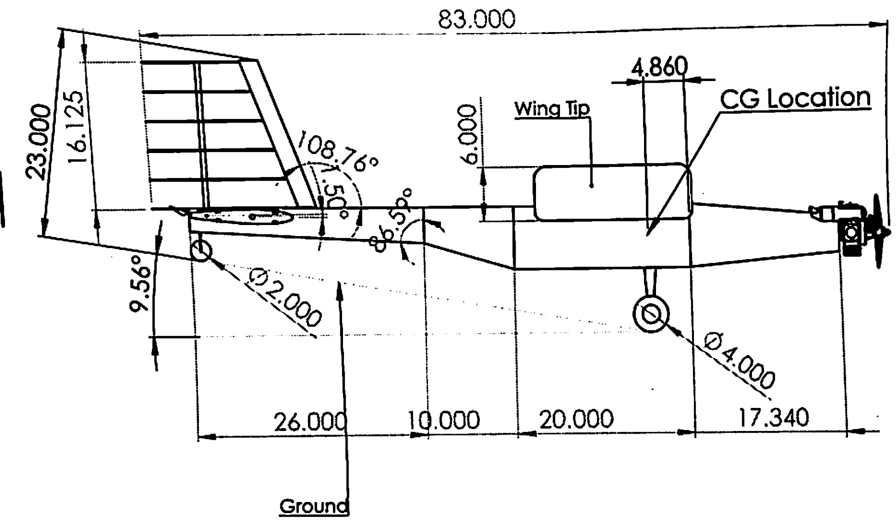
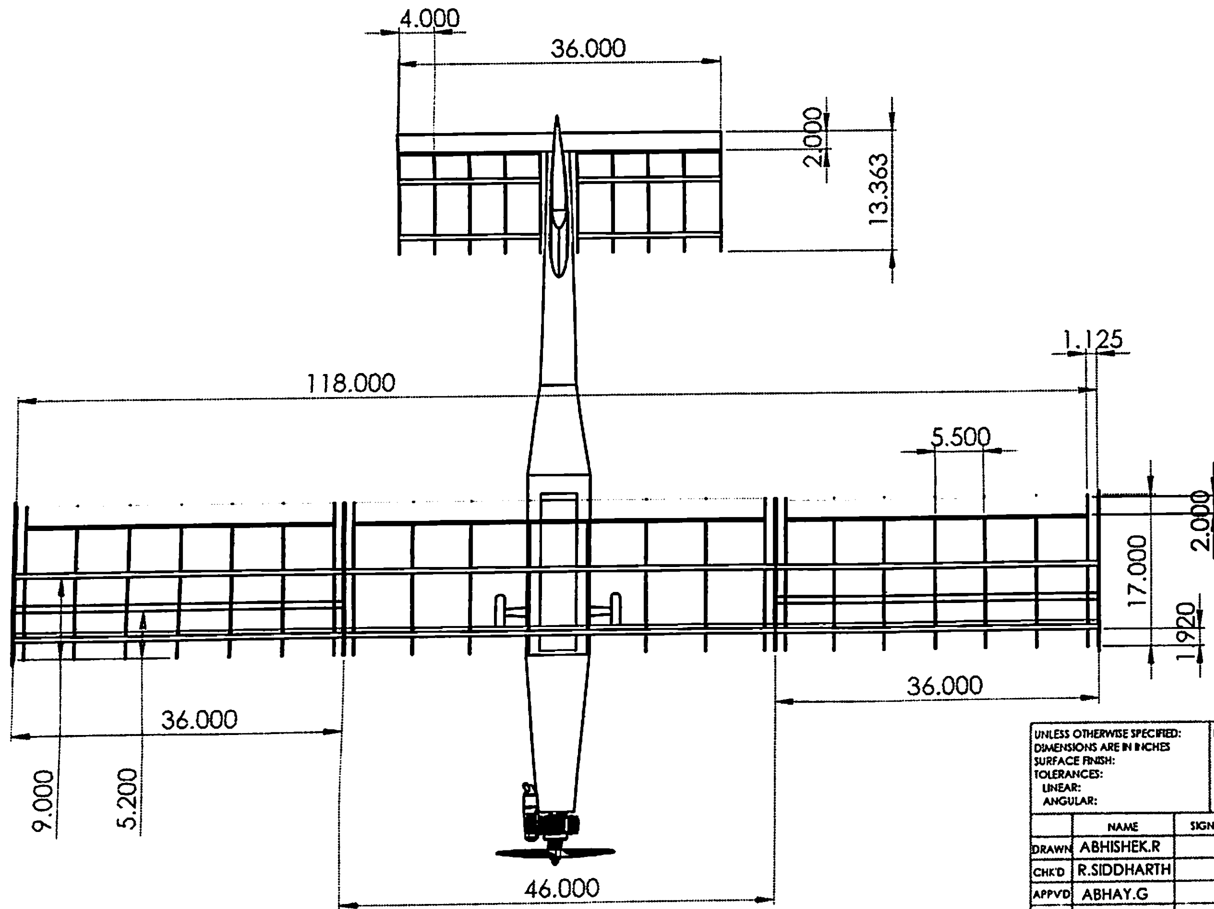


Figure 51 Streamline pattern







SPECIFICATIONS

Aircraft Length	83 in
Wing Span	118 in
Aircraft Height	23 in
L+B+H	224
Planform Area	2006 in^2
Engine	OS 61FX
Engine Capacity	0.61 cu in
Propeller Size	14 X 7
Wing Chord	17 in

UNLESS OTHERWISE SPECIFIED:
DIMENSIONS ARE IN INCHES
SURFACE FINISH:
TOLERANCES:
LINEAR:
ANGULAR:

FINISH:

DEBUR AND BREAK SHARP EDGES

DO NOT SCALE DRAWING

REVISION

	NAME	SIGNATURE	DATE
DRAWN	ABHISHEK.R		
CHK'D	R.SIDDHARTH		
APPV'D	ABHAY.G		
MFG			
Q.A			

MATERIAL:

WEIGHT:

TITLE:

SkyHawks
Team #030
University of Petroleum
& Energy Studies

DWG NO.

SkyHawks

A3

SCALE1:15

SHEET 1 OF 1

APPENDIX A: PAYLOAD PREDICTION GRAPH