DESIGN OF SINGLE ENGINE DELTA WING FIGHTER AIRCRAFT

AIRCRAFT DESIGN PROJECT II REPORT

Submitted by

SAI KISHORE REDDY SEELAM (18101011)

PUSHADAPPU BALAJI BHAVESH (18101022)

PALLA SAI KIRAN YADAV (18101023)

In partial fulfilment for the award of the degree

of

BACHELOR OF TECHNOLOGY

in

AERONAUTICAL ENGINEERING

DEPARTMENT OF AERONAUTICAL ENGINEERING

HINDUSTAN INSTITUTE OF TECHNOLOGY AND SCIENCE

PADUR, CHENNAI- 603103

OCTOBER 2021

HINDUSTAN INSTITUTE OF TECHNOLOGY AND SCIENCE

PADUR, CHENNAI- 603103

BONAFIDE CERTIFICATE

Certified that this project report titled **“Design of single Engine Delta wing Fighter Aircraft” is the bonafide work of “SAI KISHORE REDDY SEELAM (18101011), PUSHADAPPU BALAJI BHAVESH (18101022) and PALLA SAI KIRAN YADAV (18101023)”** who carried out the project work under my supervision. Certified further that to the best of my knowledge the work reported here does not form part of any other project / research work on the basis of which a degree or award was conferred on an earlier occasion on this or any other candidate.

HEAD OF DEPARTMENT
Dr. R. ASOKAN
Head of the Department
Department of Aeronautical Engineering
Hindustan Institute of Technology
and Sciences

SUPERVISOR
Mr. E S ELUMALAI
Assistant Professor\
Department of Aeronautical Engineering
Hindustan Institute of Technology
and Sciences

The Project Viva -Voice Examination is held on ____________

INTERNAL EXAMINER

EXTERNAL EXAMINER

ACKNOWLEDGEMENT

We would like to place on record our sincere thanks to all those who contributed to the successful completion of our final year project work.

It's a matter of pride and privilege for us to express our deep gratitude to the management of Hindustan Institute of Technology and Science for providing us with the necessary facilities and support.

We express our deep sense of gratitude to our respected Chairperson **Dr. ElizabethVerghese** and Pro-Chancellor **Dr. Anand Jacob Verghese** for giving us an opportunity to do the project.
We would like to thank our Director **Dr. Ashok Verghese** and Vice Chancellor
Dr. S. N. Sridhara for giving us moral support to complete this project.
We would like to express our grateful thanks to Dean (E&T) **Dr. Angelina Geetha** and Registrar **Dr. Pon. Ramalingam** for support and encouragement.

We extend our sincere thanks to our Head of the Department **Dr. R Asokan** for inspiring and motivating us to complete this project.
We would like to thank our internal guide **Mr. E. S. Elumalai,** for continually guiding and actively participating in our project, giving valuable suggestion to complete our project.

We would like to thank all the faculty members of the School of Aeronautical Sciences, who have directly or indirectly extended their support.
Last, but not least, we are deeply indebted to our parents who have been our greatest support while we worked day and night for the project to make it a success.

TABLE OF CONTENTS

NOMENCLATURE:

A.R. - Aspect Ratio

b - Wing Span (m)

C - Chord of the Airfoil (m)

C root - Chord at Root (m)

C tip - Chord at Tip (m)

C - Mean Aerodynamic Chord (m)

Cd - Drag Co-efficient

$C_{d,0}$ - Zero Lift Drag Co-efficient

C_p - Specific fuel consumption (lbs/hp/hr)

C_L - Lift Co-efficient

D - Drag (N)

E - Endurance (hr)

e - Oswald efficiency

L - Lift (N)

$(L/D)_{loiter}$ - Lift-to-drag ratio at loiter

$(L/D)_{cruise}$ - Lift-to-drag ratio at cruise

M - Mach number of aircraft

M_{ff} - Mission fuel fraction

R - Range (km)

Re - Reynolds Number

S - Wing Area (m^2)

T - Thrust (N)

V_{cruise} - Velocity at cruise (m/s)

V_{stall} - Velocity at stall (m/s)

V_t - Velocity at touch down (m/s)

W_{crew} - Crew weight (kg)

W_{empty} - Empty weight of aircraft (kg)

W_{fuel} - Weight of fuel (kg)

$W_{payload}$ - Payload of aircraft (kg)

W0 - Overall weight of aircraft (kg)

W/S - Wing loading (kg/m²)

- Density of air (kg/m³)

$A_{stringer}$ - Cross sectional area of stringers

A - Total cross-sectional area

A_{spar} - Cross sectional area of spar

a_t-Slope of the CL vs. α curve for a horizontal tail

a-Distance of the front spar from the nose of the aircraft b_w-Width of the web

b_f-Width of the flange

I_{xx} - Second moment of area about X axis

I_{zz} - Second moment of area about Z axis

K - Gust alleviation factor

n_{max} - Maximum load factor

t_w - Thickness of the web

t_f - Thickness of the flange

T - Torque

U - Gust velocity

V_{cruise} - Cruise velocity

V_s - Stalling velocity.

<u>**LIST OF TABLES**</u>

LIST OF FIGURES

ABSTRACT

Fighter aircraft, aircraft designed primarily to secure control of essential airspace by destroying enemy aircraft in combat. The opposition may consist of fighters of equal capability or of bombers carrying protective armament. For such purposes fighters must be capable of the highest possible performance in order to be able to outfly and outmanoeuvre opposing fighters. Above all, they must be armed with specialized weapons capable of hitting and destroying enemy aircraft.

Fighter airplanes have been described by a variety of labels. Early in World War I they were used as scout planes for artillery spotting, but it was quickly discovered that they could be armed and do combat with one another, shoot down enemy bombers, and conduct other tactical missions. Since that time fighters have assumed various specialized combat roles. An interceptor is a fighter whose design and armament best fit it for intercepting and defeating or routing invading fighters. A night fighter is one equipped with sophisticated radar and other instruments for navigating in unfamiliar or hostile territory at night. A day fighter is an airplane in which weight and space are saved by eliminating the special navigational equipment of the night fighter. The air supremacy, or air superiority, fighter must have long-range capability, to enable it to travel deep into enemy territory to seek out and destroy enemy fighters. Fighter-bombers fill the dual role suggested by their name.

During the Korean War jet fighters, notably, the U.S. F-86 and the Soviet MiG-15, were extensively used. The U.S. F-100 and F-4; the Soviet MiG-21; and the French Mirage III saw combat service in the Middle East and in Vietnam in the 1960s and ’70s.

Modern supersonic jet fighters can fly at more than 1,000 miles (1,600 km) per hour. They have fast rates of climb, great manoeuvrability, and heavy firepower, including air-to-air missiles. The U.S. F-16 and the Soviet MiG-25 are among the most advanced jet fighters in the world.

At the speeds and altitudes at which such aircraft can operate, the problem of striking and destroying enemy aircraft becomes extremely complicated and requires an array of electronic, navigational, and computational gear. A single-seated, high-performance fighter of the 1980s might weigh as much as, and be vastly more complicated than, one of the multiengine bombers of World War II. In many cases the search and attack functions are completely automatic, the pilot’s role in combat being virtually reduced to monitoring the operation of the equipment. Indeed, with modern jet-powered fighter airplanes, a point has been reached where the performance capabilities of the machine exceed the capabilities of a human pilot to control it.

1.INTRODUCTION

FIGHTER AIRCRAFT

Fighter aircraft or military cargo aircraft are typically fixed and rotary wing cargo aircraft which are used to deliver troops, weapons and other military equipment by a variety of methods to any area of military operations around the surface of the planet, usually outside of the commercial flight routes in uncontrolled airspace. Originally derived from bombers, Fighter aircraft were used for delivering airborne forces during the Second World War and towing military gliders. Some Fighter aircraft are tasked to perform multi-role duties such as aerial refuelling and, tactical, operational and strategic airlifts onto unprepared runways, or those constructed by engineers.

What is an Airlift?
An airlift is the organized delivery of supplies or personnel primarily via aircraft. Airlifting consists of two distinct types, strategic and tactical airlifting. Typically, strategic airlifting involves moving material long distances (such as across or off the continent or theatre), whereas a tactical airlift focuses on deploying resources and material into a specific location with high precision. Depending on the situation, airlifted supplies can be delivered by a variety of means. When the destination and surrounding airspace is considered secure, the aircraft will land at an appropriate airport or airbase to have its cargo unloaded on the ground. When landing the craft, or distributing the supplies to a certain area from a landing zone by surface transportation is not an option, the cargo aircraft can drop them in mid-flight using parachutes attached to the supply containers in question. When there is a broad area available where the intended receivers have control without fear of the enemy interfering with collection and/or stealing the goods, the planes can maintain a normal flight altitude and simply drop the supplies down and let them parachute to the ground. However, when the area is too small for this method, as with an isolated base, and/or is too dangerous to land in, a Low Altitude Parachute Extraction System drop is used.

CLASSIFICATION OF AIRLIFTS

- STRATEGIC AIRLIFT
- TACTICAL AIRLIFT

STRATEGIC AIRLIFT
Strategic airlift is the use of cargo aircraft to transport materiel, weaponry, or personnel over long distances. Typically, this involves airlifting the required items between two airbases which are not in the same vicinity. This allows commanders to bring items into a combat theatre from a point on the other side of the planet, if necessary. Aircraft which perform this role are considered strategic airlifters. This contrasts with tactical airlifters, which can normally only move supplies within a given theatre of operations.
TACTICAL AIRLIFT
Tactical airlift is a military term for the airborne transportation of supplies and equipment within a theatre of operations (in contrast to strategic airlift). Aircraft which perform this role are referred to as tactical airlifters. These are typically turboprop aircraft, and feature short landing and take-off distances and low-pressure tires allowing operations from small or poorly-prepared airstrips. While they lack the speed and range of strategic airlifters (which are typically jet-powered), these capabilities are invaluable within war zones. Helicopters have the advantage that they do not require a landing strip and that equipment can often be suspended below the aircraft allowing it to be delivered without landing but are highly inefficient. Tactical

airlift aircraft are designed to be manoeuvrable, allowing low-altitude flight to avoid detection by radar and for the airdropping of supplies. Most are fitted with defensive aids systems to protect them from attack by surface-to-air missiles.

DESIGN OF AN AIRPLANE

Airplane design is both an art and a science. It's the intellectual engineering process of creating on paper (or on a computer screen) a flying machine to

- meet certain specifications and requirements established by potential users (or as perceived by the manufacturer) and

- pioneer innovative, new ideas and technology

The design process is indeed an intellectual activity that is rather specified one that is tempered by good intuition developed via by attention paid to successful airplane designs that have been used in the past, and by (generally proprietary) design procedure and databases (hand books etc) that are a part of every airplane manufacturer.

PHASES OF AIRPLANE DESIGN

The complete design process has gone through three distinct phases that are carried out in sequence. They are

- Conceptual design

- Preliminary design

- Detailed design

CONCEPTUAL DESIGN

The design process starts with a set of specifications (requirements)for a new airplane, or much less frequently as the response to the desire to implement some pioneering, innovative new ideas and technology. In either case, there is a rather concrete good towards which the designers are aiming. The first steps towards achieving that goal constitute the conceptual design phase. Here, within a certain somewhat fuzzy latitude, the overall shape, size, weight and performance of the new design are determined.

The product of the conceptual design phase is a layout on a paper or on a computer screen) of the airplane configuration. But one has to visualize this drawing as one with flexible lines, capable of being slightly changed during the preliminary design phase. However, the conceptual design phase determines such fundamental aspects as the shape of the wings (swept back, swept forward or straight), the location of the wings related to the fuselage, the shape and location of the horizontal and vertical tail, the use of a engine size and placement etc, the major drivers during the conceptual design process are aerodynamics, propulsion and flight performance.

Structural and context system considerations are not dealt with in any detail. However, they are not totally absent. During the conceptual design phase, the designer is influenced by such qualitative as the increased structural loads imposed by a high horizontal tail location trough the fuselage and the difficulties associated with cut-outs in the wing structure if the landing gear are to be retracted into the wing rather than the fuselage or engine nacelle. No part of the design is ever carried out in a total vacuum unrelated to the other parts.

PRELIMINARY DESIGN

In the preliminary design phase, only minor changes are made to the configuration layout (indeed, if major changes were demanded during this phase, the conceptual design process have been actually flawed to begin with. It is in the preliminary design phase that serious structural and control system analysis and design take place. During this phase also, substantial wind tunnel testing will be carried out and major computational fluid dynamics (CFD) calculations of the computer flow fluid over the airplane configurations are done.

It's possible that the wind tunnel tests the CFD calculations will in cover some undesirable aerodynamic interference or some unexpected stability problems which will promote change to the configuration layout. At the end of preliminary design phase, the airplane configuration is frozen and preciously defined. The drawing process called lofting is carried out which mathematically models the precise shape of the outside skin of the airplane making certain that all sections of the aircraft property fit together

The end of the preliminary design phase brings a major concept to commit the manufacture of the airplane or not. The importance of this decision point for modern aircraft manufacturers cannot be understated, considering the tremendous costs involved in the design and manufacture of a new airplane.

DETAIL DESIGN

The detail design phase is literally the nuts-and-bolts phase of airplane design. The aerodynamic, propulsion, structures performance and flight control analysis have all been finished with the preliminary design phase. The airplane is now simply a machine to be fabricated. The pressure design of each individual rib, spar and section of skin now take place. The size of number and location of fastness are determined. At this stage, flight simulators for the airplane are developed. And these are just a few of the many detailed requirements during the detail design phase. At the end of this phase, the aircraft is ready to be fabricated.

OUTLINE AIRCRAFT DESIGN PROJECT 2:

The structural design of the aircraft which is done in aircraft design project 2 involves:

- Determination of loads acting on aircraft:

- V-n diagram for the design study
- Gust and maneuverability envelopes
- Schrenk‟s Curve
- Critical loading performance and final V-n graph calculation

- Determination of loads acting on individual structures:

- Structural design study – Theory approach
- Load estimation of wings
- Load estimation of fuselage.
- Material Selection for structural members
- Detailed structural layouts
- Design of some components of wings, fuselage

Table 1.1 **Parameters taken from aircraft design project 1**

S. No.	**Design Parameters**	**Value**	**Unit**
1.	Aspect Ratio	10.08	-
2.	Height	5	m
3.	Length	17	m
4.	MTOW	18000	Kg
5.	Payload	5600	Kg
6.	Max Range	2700	Km
7.	Rate Of Climb	300	m/s
8.	Service Ceiling	20000	m
9.	Wing Span	11	m
10.	Total Thrust	90	kN
11.	Wing Area	12	M^2
12	Crew	1	-
13	Empty weight	7000	kg
14	Wing Loading	1499.99999	Kg/m^2
15	Number of engines	2	-
16	Type of Engines	Turbojet	-
17	Maximum Speed	2250	Km/hr

2. V-n Diagram

INTRODUCTION:

Airplanes may be subjected to a variety of loading conditions in flight. The structural design of the aircraft involves the estimation of the various loads on the aircraft structure and designing the airframe to carry all these loads, providing enough safety factors, considering the fact that the aircraft under design is a commercial transport airplane. As it is obviously impossible to investigate every loading condition that the aircraft may encounter, it becomes necessary to select a few conditions such that each one of these conditions will be critical for some structural member of the airplane.

Velocity –Load Factor (V-n) diagram:

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. Excess 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 (Av.P.970 and BCAR) 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.

- **Proof load**, which is the product of the limit load and the **proof factor** (1.0-1.25), and

- **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 basic strength and fight performance limits for a particular aircraft are selected by the airworthiness authorities and are contained in the flight envelope or **V-n diagram**.

There are two types of V – n diagram for military airplanes:

- V–n manoeuvre diagram and

- V–n gust diagram

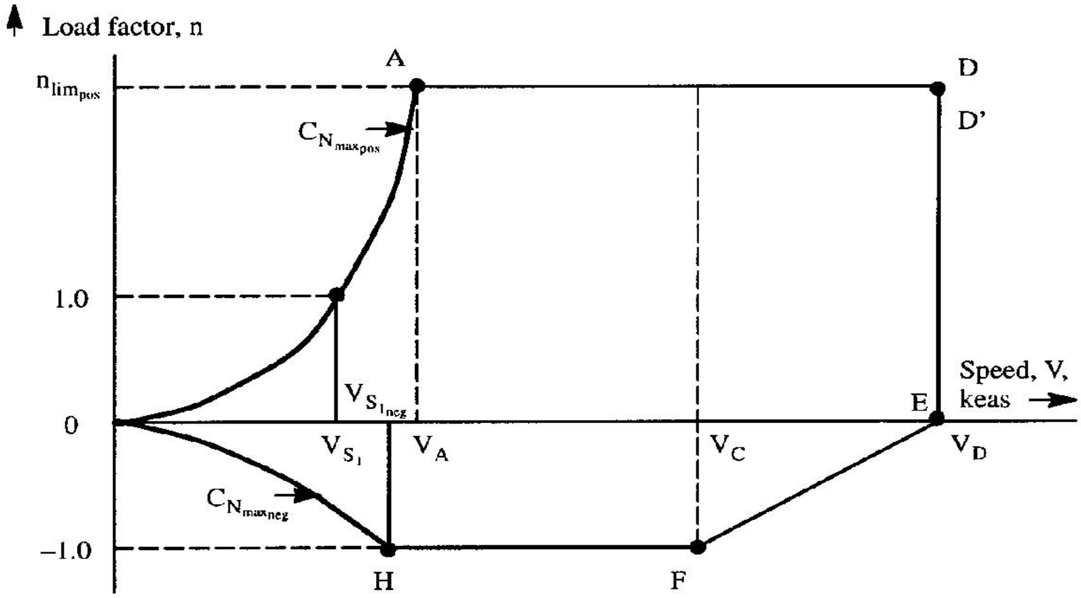

Figure 2.1 v – n manoeuvre diagram

The positive design limit load factor must be selected by the designer, but must meet the following condition
The maximum positive limit load factor for Fighter aircraft should be in the range 2 to 3.

<u>3. GUST V-n DIAGRAM</u>

Description:

Gust is a sudden, brief increase in the speed of the wind. Generally, winds are least gusty over large water surfaces and most gusty over rough land and near high buildings. With respect to aircraft turbulence, a sharp change in wind speed relative to the aircraft; a sudden increase in airspeed due to fluctuations in the airflow, resulting in increased structural stresses upon the aircraft**. Sharp-edged gust** (u) is a wind gust that results in an instantaneous change in direction or speed.

Derived gust velocity (U_g or U_{max}) is the maximum velocity of a sharp-edged gust that would produce a given acceleration on a particular airplane flown in level flight at the design cruising speed of the aircraft and at a given air density. As a result a 25% increase is seen in lift for a longitudinally disturbing gust.

The effect of turbulence gust is to produce a short time change in the effective angle of attack. These changes produce a variation in lift and thereby load factor. For V_B, a gust velocity of 20.1168 m/s is assumed. For V_C, a gust velocity of 15.24 m/s at sea level is assumed. For V_D, a gust velocity of 7.26 m/s is assumed.

Effective gust velocity: The vertical component of the velocity of a sharp-edged gust that would produce a given acceleration on a particular airplane flown in level flight at the design cruising speed of the aircraft and at a given air density.

4.CRITICAL LOADING PERFORMANCE AND FINAL V-n DIAGRAM

CRITICAL LOADING PERFORANCE:

The greatest air loads on an aircraft usually comes from the generation of lift during high-g manoeuvres. Even the fuselage is almost always structurally sized by the lift of the wings rather than by the pressures produced directly on the fuselage. Aircraft load factor (n) expresses the maneuvering of an aircraft as a standard acceleration due to gravity.

At lower speeds the highest load factor of an aircraft may experience is limited by the maximum lift available. At higher speeds the maximum load factor is limited to some arbitrary value based upon the expected use of the aircraft. The maximum lift load factor equals 1.0 at levels flight stall speed. This is the slowest speed at which the maximum load can be reached without stalling.

The aircraft maximum speed, or dive speed at right of the V-n diagram represents the maximum dynamic pressure and maximum load factor is clearly important for structural sizing. At this condition, the aircraft is at fairly low angle of attack because of the high dynamic pressure, so the load is approximately vertical in the body axis. The most common manoeuvres that we focused are,

- Level turn
- Pull up
- Pull down
- Climb

5.STRUCTURAL DESIGN STUDY – THEORY APPROACH

THEORY APPROACH:

Aircraft loads are those forces and loadings applied to the airplanes structural components to establish the strength level of the complete airplane. These loadings may be caused by air pressure, inertia forces, or ground reactions during landing. In more specialized cases, design loadings may be imposed during other operations such as catapulted take-offs, arrested landings, or landings in water.

The determination of design loads involves a study of the air pressures and inertia forces during certain prescribed manoeuvres, either in the air or on the ground. Since the primary objective is an airplane with a satisfactory strength level, the means by which this result is obtained is sometimes unimportant. Some of the prescribed manoeuvres are therefore arbitrary and empirical which is indicated by a careful examination of some of the criteria.

Important consideration in determining the extent of the load analysis is the amount of structural weight involved. A fairly detailed analysis may be necessary when computing operating loads on such items as movable surfaces, doors, landing gears, etc. proper operation of the system requires an accurate prediction of the loads.

Aircraft loads is the science of determining the loads that an aircraft structure must be designed to withstand. A large part of the forces that make up design loads are the forces resulting from the flow of air about the airplane's surfaces-the same forces that enable flight and control of the aircraft.

Load factors

In normal straight and level flight the wing lift supports the weight of the airplane. During manoeuvres or flight through turbulent (gusty) air, however, additional loads are imposed which will increase or decrease the net loads on the airplane structure. The number of additional loads depends on the severity of the manoeuvres or the turbulence, and its magnitude is measured in terms of load factor.

The maximum manoeuvring load factor to which an airplane is designed depends on its intended usage. Fighters, which are expected to execute violent manoeuvres, are designed to withstand loads commensurate with the accelerations a pilot can physically withstand. Long range, heavily loaded bombers, on the other hand, are designed to low load factors and must be handled accordingly.

For a typical two spar layout, the ribs are usually formed in three parts from sheet metal by the use of presses and dies. Flanges are incorporated around the edges so that they can be riveted to the skin and the spar webs Cut-outs are necessary around the edges to allow for the stringers to pass through Lightening holes are usually cut into the rib bodies to reduce the rib weight and also allow for passage of control runs fuel electrics etc.,

STRUCTURAL DESIGN CRITERIA

The structural criteria define the types of manoeuvres, speed, useful loads, and gross weights which are to be considered for structural design analysis. These are items which are under the control of the airplane operator. In addition, the structural criteria must consider such items as inadvertent manoeuvres, effects of turbulent air, and severity of ground contact during landing. The basic structural design criteria, from which the loadings are determined, are based largely on the type of the airplane and its intended use.

6. LOAD ESTIMATION ON WINGS

Description

The solution methods which follow Euler's beam bending theory ($\sigma/y=M/I=E/R$) use the bending moment values to determine the stresses developed at a particular section of the beam due to the combination of aerodynamic and structural loads in the transverse direction. Most engineering solution methods for structural mechanics problems (both exact and approximate methods) use the shear force and bending moment equations to determine the deflection and slope at a particular section of the beam. Therefore, these equations are to be obtained as analytical expressions in terms of span wise location. The bending moment produced here is about the longitudinal (x) axis.

Loads acting on wing

As both the wings are symmetric, let us consider the starboard wing at first. There are three primary loads acting on a wing structure in transverse direction which can cause considerable shear forces and bending moments on it. They are as follows:

- Lift force (given by Schrenk's curve)
- Self-weight of the wing
- Weight of the power plant
- Weight of the fuel in the wing

Shear force and bending moment diagrams due to loads along transverse direction at cruise condition

Lift varies along the wing span due to the variation in chord length, angle of attack and sweep along the span. Schrenk's curve defines this lift distribution over the wing span of an aircraft, also called simply as Lift Distribution Curve.

Schrenk's Curve is given by
y=y1+y2
where,
y_1 is Linear Variation of lift along semi wing span also named as L1 y_2 is
Elliptic Lift Distribution along the wing span also named as L2

Linear lift distribution (trapezium):

Lift at root= ½*ρ*v^2*C_{root}*C_l

=1/2*1.225**625^2*1.0909*0.3699

L_{root}=96546.245N/m

Lift at tip= ½*ρ*v^2*C_{tip}*C_l

= ½*1.223*625^2 *0.3272*0.3699

=28957.6793N/m

By representing this lift at sections of root and tip we can get the equation for the wing.

Equation of linear lift distribution for starboard wing

y_1 = -12288.83067*(x)+96546.248

Equation of linear lift distribution for port wing we have to replace x by –x in general,

y_1 = 12288.83067*(x)-96546.248

For the Schrenk's curve we only consider half of the linear distribution of lift and hence we derive y*1/2

y_1 = -6144.4153*(x)+48228.124

Table 6.1 linear lift distribution

SPAN	LINEAR LIFT
0	48228.124
0.275	46538.40979
0.55	44848.69559
0.825	43158.98138
1.1	41469.26717
1.375	39779.55296
1.65	38089.83876
1.925	36400.12455
2.2	34710.41034
2.475	33020.69613
2.75	31330.98193
3.025	29641.26772
3.3	27951.55351
3.575	26261.8393
3.85	24572.1251
4.125	22882.41089
4.4	21192.69668
4.675	19502.98247
4.95	17813.26827
5.225	16123.55406
5.5	14433.83985

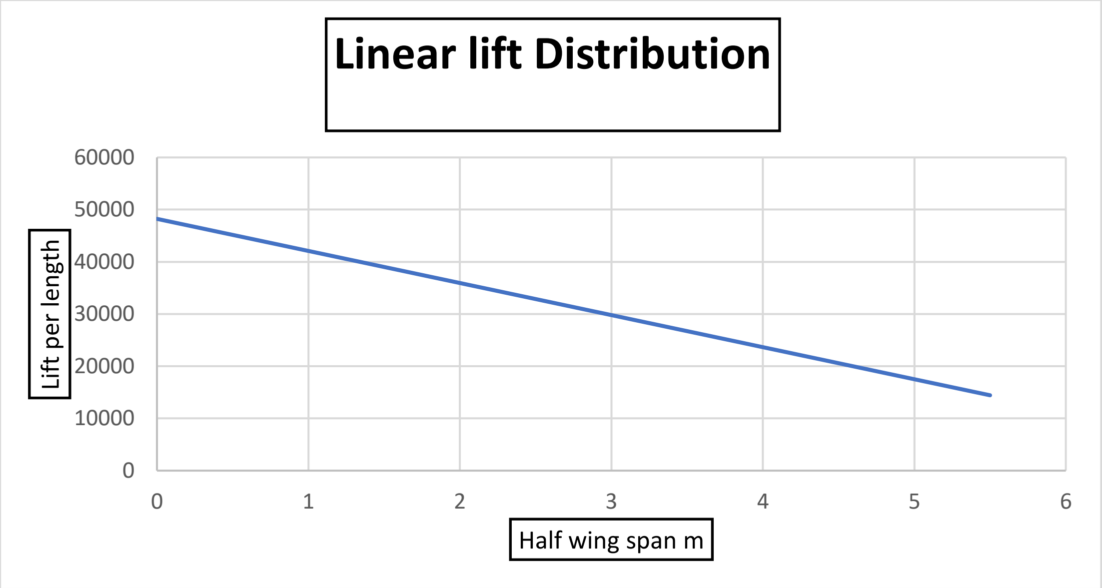

Figure 6.1 Linear lift Distribution

Elliptical Lift Distribution:

Twice the area under the curve or line will give the lift which will be required to overcome weight

Considering an elliptic lift distribution, we get

$X^2/a^2 + y^2/b^2 = 1$

Where,

b is actual lift at root and a is wing semi span

Lift at tip= $\frac{1}{2}*\rho*v^{\wedge}2*C_{tip}*C_l$

$= \frac{1}{2}*1.223*625^2 *0.3272*0.3699$

$=28957.6793 N/m$

Equation of elliptic lift,

$y2=8776.9215*\sqrt{30.25-x^2}$

Table 6.2 elliptical lift distribution

SPAN	ELLIPTICAL LIFT
0	48273.12325
0.275	48212.74409
0.55	48031.15118
0.825	47726.96097
1.1	47297.8081
1.375	46740.2506
1.65	46049.62464
1.925	45219.83371
2.2	44243.04827
2.475	43109.2775
2.75	41805.75105
3.025	40316.0063
3.3	38618.4986
3.575	36684.39767
3.85	34473.90547
4.125	31929.66978
4.4	28963.87395
4.675	25429.44561
4.95	21041.76659
5.225	15073.2779
5.5	0

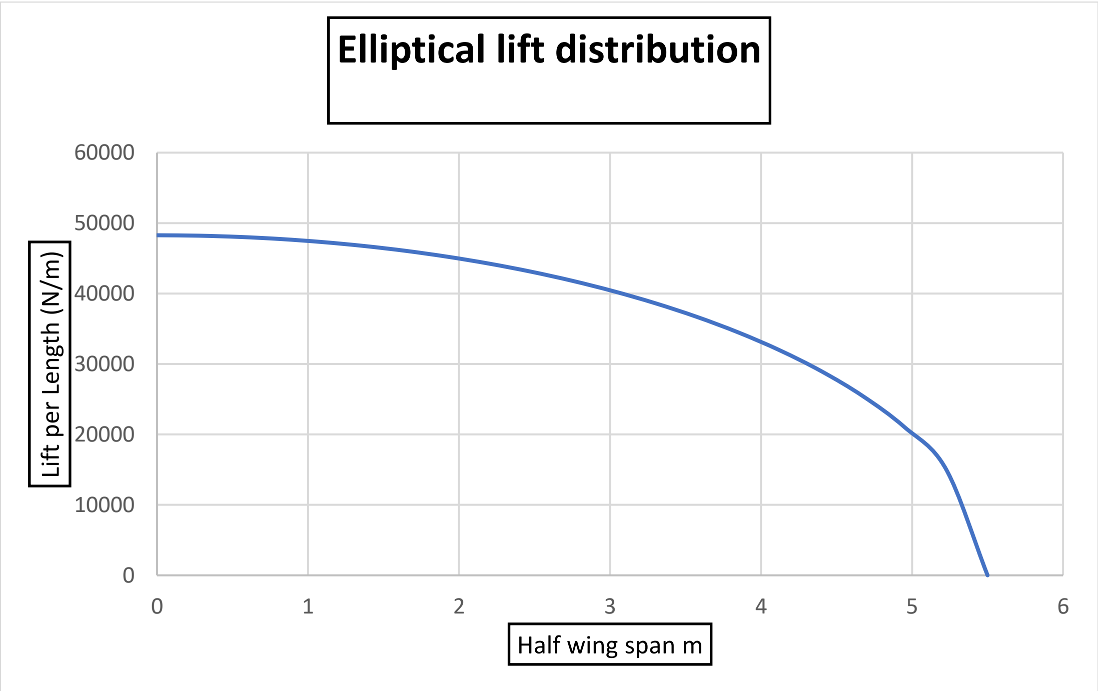

Figure 6.2 Elliptical lift distribution

Construction of Schrenk's Curve:

Schrenk's Curve is given by,

y=y1+y2

$y = -6144.4153*(x)+48228.124+8776.9215*\sqrt{30.25-x^2}$

Substituting different values for x we can get the lift distribution for the wing semi span

Table 6.3 schrenk's curve semi-span distribution

SPAN	LIFT
0	96501.24725
0.275	94751.15388
0.55	92879.84677
0.825	90885.94235
1.1	88767.07527
1.375	86519.80357
1.65	84139.4634
1.925	81619.95826
2.2	78953.45861
2.475	76129.97363
2.75	73136.73298
3.025	69957.27401
3.3	66570.05211
3.575	62946.23697
3.85	59046.03057
4.125	54812.08067
4.4	50156.57063
4.675	44932.42808
4.95	38855.03486
5.225	31196.83196
5.5	14433.83985

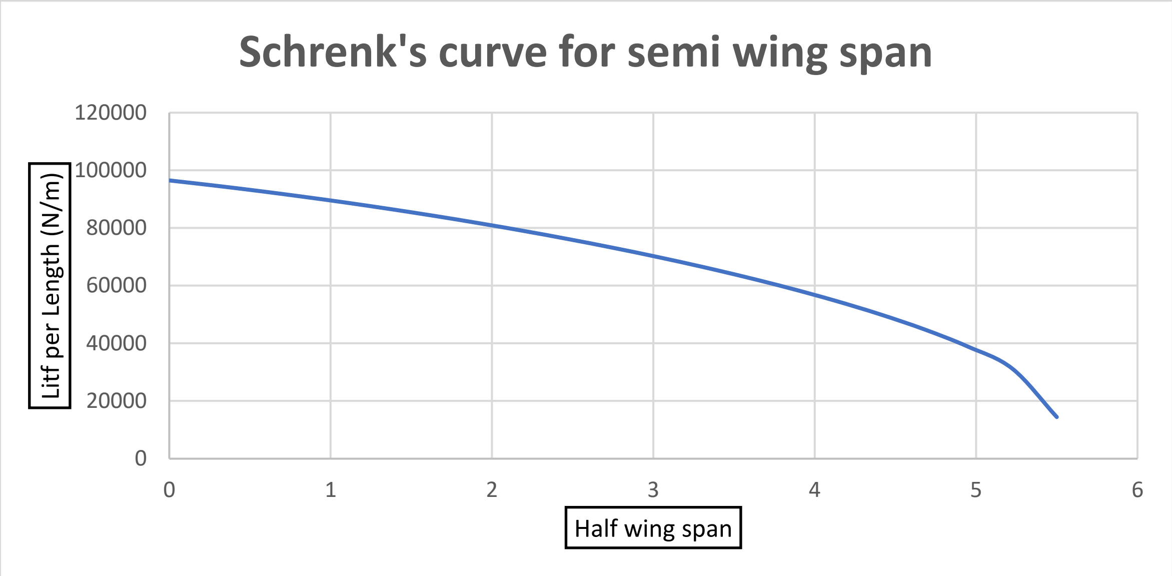

Figure 6.3 Schrenk's curve for semi wing

Table 6.4 schrenk's curve full-span distribution

SPAN	LIFT
-5.5	14433.83985
-5.225	31196.83985
-4.95	38855.03486
-4.675	44932.42808
-4.4	50156.57063
-4.125	54812.08067
-3.85	59046.03057
-3.575	62946.23697
-3.3	66570.05211
-3.025	69957.27401
-2.75	73136.73298
-2.475	76129.97363
-2.2	78953.45861
-1.925	81616.95826
-1.65	84139.95826
-1.375	86519.80357
-1.1	88767.07527
-0.825	90885.94235
-0.55	92879.84677
-0.275	94751.15388
0	96501.24725
0.275	94751.15388
0.55	92879.84677
0.825	90885.94235
1.1	88767.07527
1.375	86519.80357

1.65	84139.4634
1.925	81619.95826
2.2	78953.45861
2.475	76129.97363
2.75	73136.73298
3.025	69957.27401
3.3	66570.05211
3.575	62946.23697
3.85	59046.03057
4.125	54812.08067
4.4	50156.57063
4.675	44932.42808
4.95	38855.03486
5.225	31196.83196
5.5	14433.83985

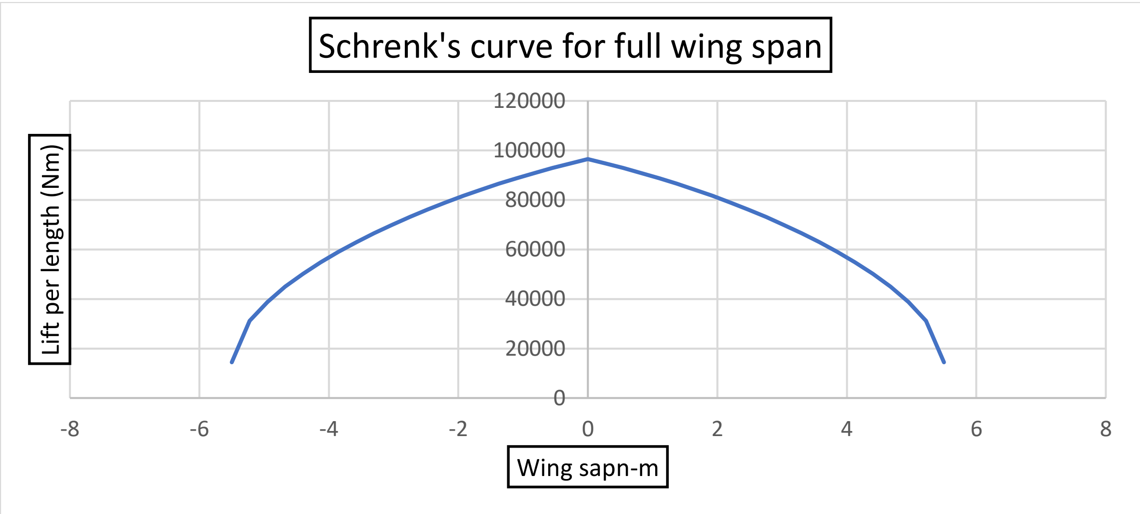

Figure 6.4 Schrenk's curve for full wing

SELF-WEIGHT OF WING (Y_3):

W_{wing}/ W_{TO} =7.5%

W_{wing}= 7.5/100*108000 Kg
= 8100

$W_{semiwing}$= 4050

At y= $W_{semiwing}$, x varies from 0 to b/2

- $W_{semiwing}= {}_0\int^{5.5} k(x-5.5)^2\, dx$

$-4050= {}_0\int^{5.5} k(x-5.5)^2\, dx$

K=-73.0299

By substituting k in y.
Then,
$y3= -73.0299(x-5.5)^2$

Table 6.5 self-weight distribution

SPAN	LIFT
0	-4049.991
0.275	-3655.116878
0.55	-3280.49271
0.825	-2926.118498
1.1	-2591.99424
1.375	-2278.119938
1.65	-1984.49559
1.925	-1711.121198
2.2	-1457.99676
2.475	-1225.122278
2.75	-1012.49775
3.025	-820.1231775
3.3	-647.99856
3.575	-496.1238975
3.85	-364.49919
4.125	-253.1244375
4.4	-161.99964
4.675	-91.1247975
4.95	-40.49991
5.225	-10.1249775
5.5	0

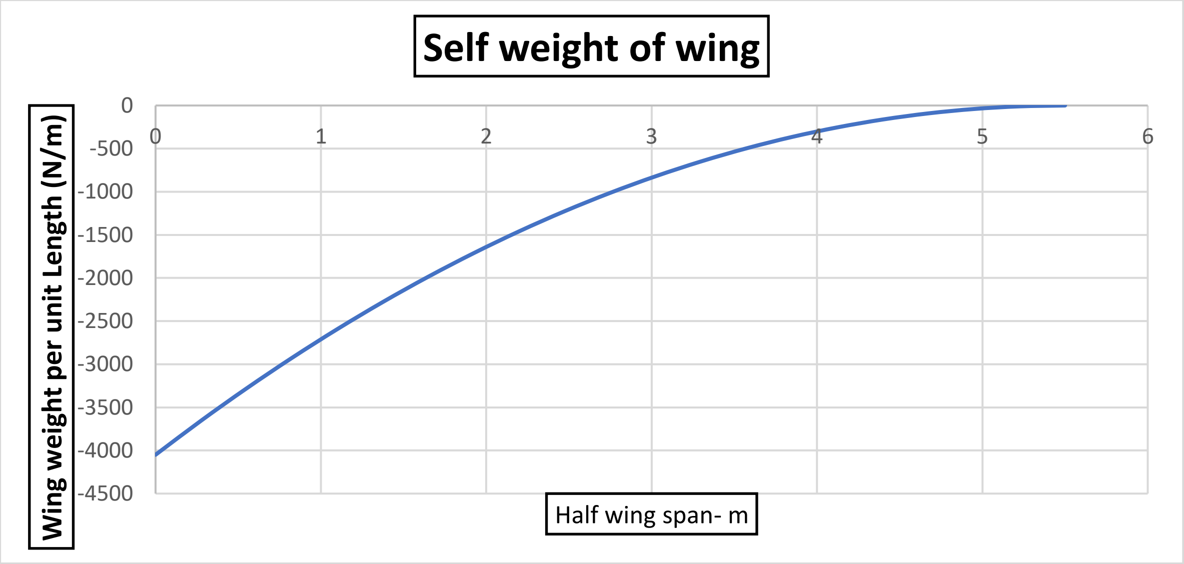

Figure 6.5 Self-weight of wing

FUEL WEIGHT IN THE WING:

This design has fuel in the wing so we have to consider the weight of the fuel in one wing.

W_{fuel}= 7357.562 kg

$W_{semifuel}$= W_{fuel}/2
=3678.781 kg

V_{fuel}= $W_{semifuel}$/ ρ
=3678.781/800
=4.5987 m^3

V_{fuel}= C_{mean}*h*t_{mean}

h= V_{tank}/ C_{mean}*t_{mean}
h=3.2939m

Again by using general formula for straight line y= mx + c we get,

Slope, m= $W_{semifuel}/h^2$
=338.24
At x=2.75m , y=3678.78

y=mx+c
3678.78=(338.24*2.25)+c

C=-29147.74

y_f = 338.24x- 2917.74

Table 6.6 fuel-weight distribution

SPAN	LIFT
0.601	-2714.45776
0.76585	-2658.698896
0.9307	-2602.940032
1.09555	-2547.181168
1.2604	-2491.422304
1.42525	-2435.66344
1.5901	-2379.904576
1.75495	-2324.145712
1.9198	-2268.386848
2.08465	-2212.627984

2.2495	-2156.86912
2.41435	-2101.110256
2.5792	-2045.351392
2.74405	-1989.592528
2.9089	-1933.833664
3.07375	-1878.0748
3.2386	-1822.315936
3.40345	-1766.557072
3.5683	-1710.798208
3.73315	-1655.039344
3.898	-1599.28048

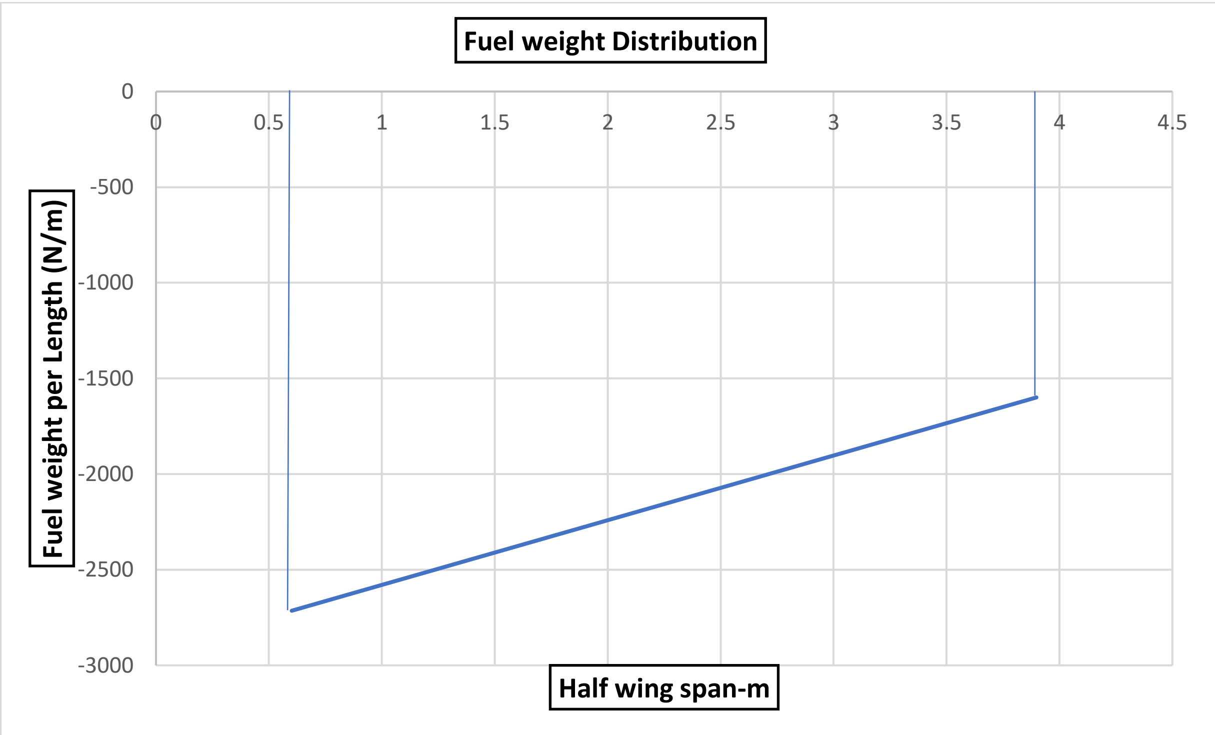

Figure 6.6 Fuel weight distribution

Table 6.7: Loads simplified as point loads:

Curve / component	**Area enclosed / structural weight (N)**	**Centroid (from wing root)**
$y_1/2$	172320.40059	2.0251m
$y_2/2$	208524.9218	2.3342 m
Wing	7424.9835	1.35 m
Fuel	7110.9837	1.5603 m

Reaction force and Bending moment calculations:

The wing is fixed at one end and free at another end.

Then,

V_A= -363609.66N

$\sum$,

Then,

M_A =814529.58 N/m

Now we know V_A and M_A, using this we can find out shear force and Bending moment.

SHEAR FORCE

$y1=-6144.4153x+48228.124$
$y2=8776.9345(30.25-x^2)^{0.5}$
$y3=-73.0299(x-5.5)^2$
$yf=338.211x-2917.74$
$\int y1 =-3072.2x^2+8228.12x$
$\int y2 =x/2*(30.25-X^2)^{0.5}+15.1125*\sin^{-1}(x/5.5)$
$\int y3 =-73.02(x^3/3-5.5x^2+x*5.5^2)$
$\int yf =169x^2-2917.74x$

SBC=-30.72x^2+ 48228.12x+ 4388.46x$(30.25x^2)^{0.5}$+ 132751*$\sin^{-1}$(x/5.5)+ 73.02(x^3/3-5.5x^2+30.25x)-VA

At x= 5.5 m
SB=748504.62N

At x=3.898m
SC=679813.3N

At x=0.601m
SD=678120.78N

By using the corresponding values of x in appropriate equations we get the plot

of shear force.

Table 6.8: Shear force

x	y
0	-363609.66
0.601	678120.78
3.898	679813.3
5.5	748504.62

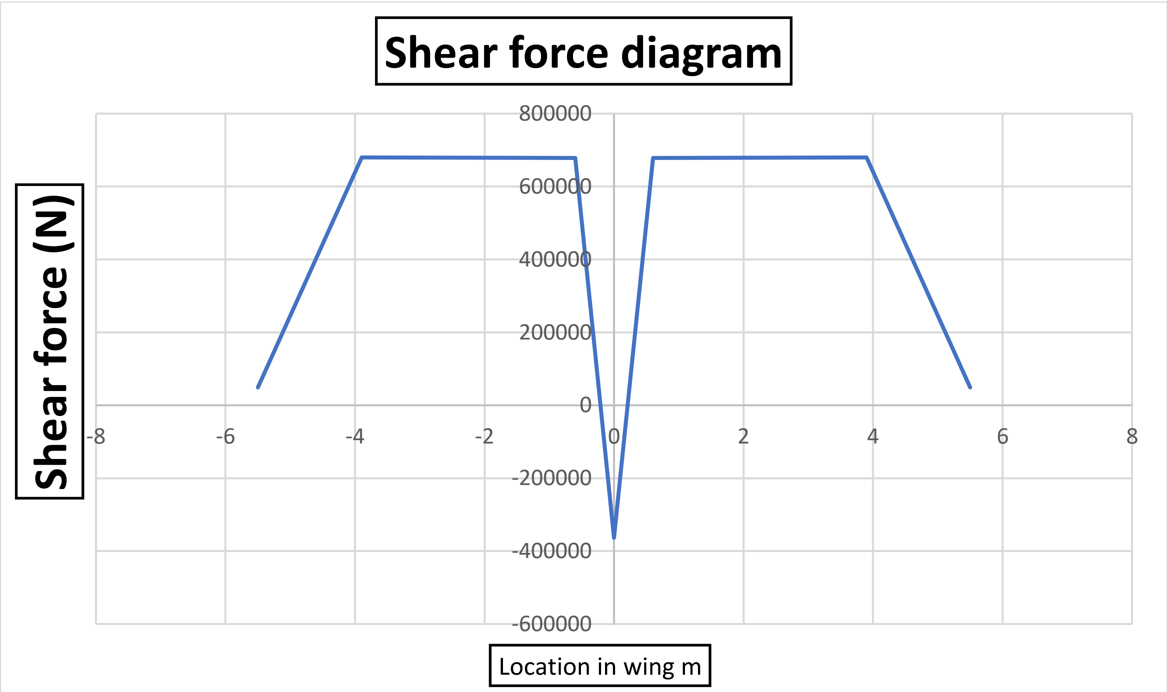

Figure 6.7 Shear force

BENDING MOMENT:

$\iint y1= -1024.066x^3+24114.05x^2$
$\iint y2= 730131.05(1-4x^2/121)^{0.5}-1462.8333(30.25-x^2)^{(3/2)}$
$\iint y3= -6.085825x^4+133.88815x^3-1104.5772x^2$
$\iint VA= -181804.83x^2$

at x=5.5
BMBC=13687.69Nm
At x=3.898
BMCD=385685.3Nm
At x=0.601
BMAD=438175.5Nm

By substituting the values of x for the above equations of bending moments obtained we can get a continuous bending moment curve for the port wing.

Table6.9: Bending Moment

x	y
0	449873.76
0.601	438175.5
3.898	385685.25
5.5	13687.39

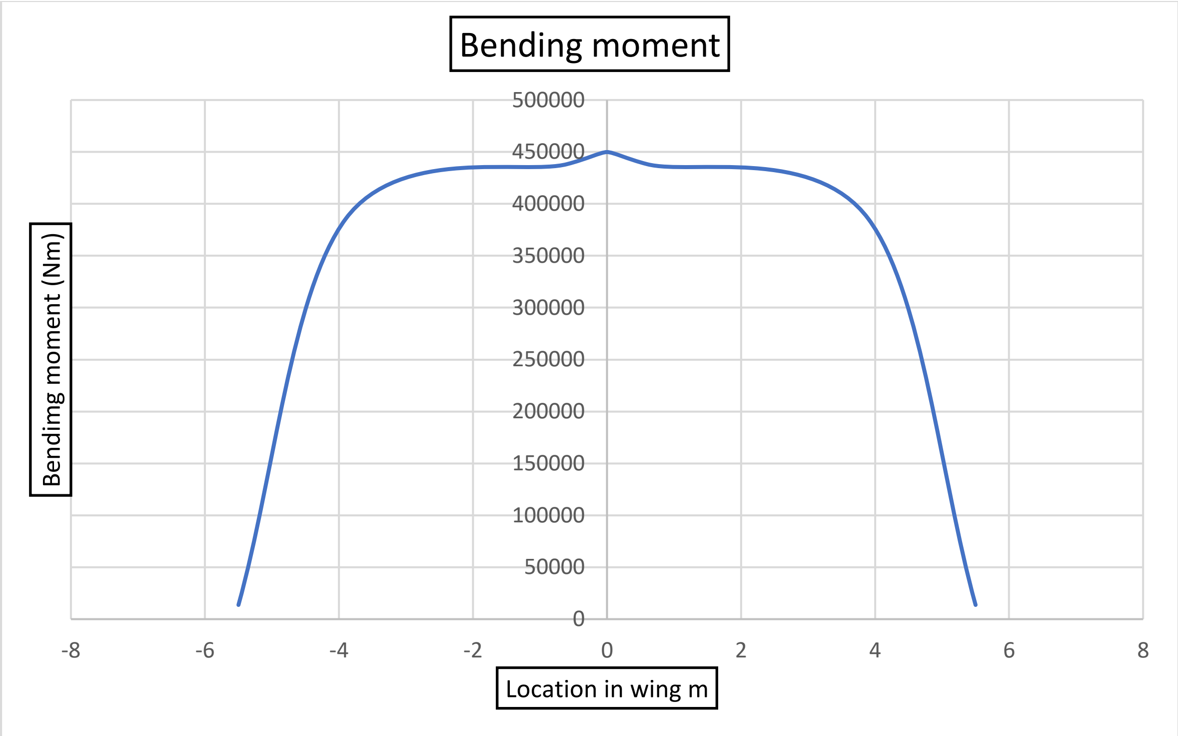

Figure 6.8 Bending moment

7. LOAD ESTIMATION ON FUSELAGE

LOAD ESTIMATION ON FUSELAGE

Fuselage contributes very little to lift and produces more drag but it is an important structural member/component. It is the connecting member to all load producing components such as wing, horizontal tail, vertical tail, landing gear etc. and thus redistributes the load. It also serves the purpose of housing or accommodating practically all the equipments, accessories and systems in addition to carrying the payload. Because of large amount of equipment inside the fuselage, it is necessary to provide sufficient number of cutouts in the fuselage for access and inspection purposes. These cutouts and discontinuities result in fuselage design being more complicated, less precise and often less efficient in design. As a common member to which other components are attached, thereby transmitting the loads, fuselage can be considered as a long hollow beam. The reactions produced by the wing, tail or landing gear may be considered as concentrated loads at the respective attachment points. The balancing reactions are provided by the inertia forces contributed by the weight of the fuselage structure and the various components inside the fuselage. These reaction forces are distributed all along the length of the fuselage, though need not be uniformly .Unlike the wing, which is subjected to mainly unsymmetrical load, the fuselage is much simpler for structural analysis due to its symmetrical cross-section and symmetrical loading. The main load in the case of fuselage

is the shear load because the load acting on the wing is transferred to the fuselage skin in the form of shear only. The structural design of both wing and fuselage begin with shear force and bending moment diagrams for the respective members

To find out the loads and their distribution, consider the different cases. The main components of the fuselage loading diagram are:

- Weight of the fuselage
- Engine weight
- Weight of the horizontal and vertical stabilizer

8. BALANCING AND MANEUVERING LOADS ON TAIL PLANE, RUDDER AND AILERON LOADS

MANOEUVRING LOADS:

Each horizontal surface and its supporting structure, and the main wing of a canard or tandem wing configuration, if that surface has pitch control, must be designed for the manoeuvring loads imposed by the following conditions:

- A sudden movement of the pitching control, at the speed V_A, to the maximum aft movement, and the maximum forward movement, as limited by the control stops, or pilot effort, whichever is critical.

- A sudden aft movement of the pitching control at speeds above V_A, followed by a forward movement of the pitching control resulting in the following combinations of normal and angular acceleration. At speeds up to V_A, the vertical surfaces must be designed to withstand the following conditions. In computing the loads, the yawing velocity may be assumed to be zero:

- With the airplane in unaccelerated flight at zero yaw, it is assumed that the rudder control is suddenly displaced to the maximum deflection, as limited by the control stops or by limit pilot forces.

- With the rudder deflected, it is assumed that the airplane yaws to the over swing sideslip angle. In lieu of a rational analysis, an over swing angle equal to 1.5 times the static sideslip angle may be assumed.

- A yaw angle of 15 degrees with the rudder control maintained in the neutral position (except as limited by pilot strength)

- The airplane must be yawed to the largest attainable steady state sideslip angle, with the rudder at maximum deflection caused by any one of the following:

1. Control surface stops

2. Maximum available booster effort

3. Maximum pilot rudder force

The rudder must be suddenly displaced from the maximum deflection to the neutral position.

The yaw angles may be reduced if the yaw angle chosen for a particular speed cannot be exceeded in:

1. Steady slip conditions
2. Uncoordinated rolls from steep banks or
3. Sudden failure of the critical engine with delayed corrective action.

The ailerons must be designed for the loads to which they are subjected:

- In the neutral position during symmetrical flight conditions; and
- By the following deflections (except as limited by pilot effort), during unsymmetrical flight conditions
- Sudden maximum displacement of the aileron control at V_A. Suitable allowance may be made for control system deflections.
- Sufficient deflection at V_C, where V_C is more than V_A, to produce a rate of roll not less than obtained
- Sufficient deflection at V_C to produce a rate of roll not less than one-third of that obtained

(a)Symmetric manoeuvring conditions:

Where sudden displacement of a control is specified, the assumed rate of control surface displacement may not be less than the rate that could be applied by the pilot through the control system. In determining elevator angles and chord wise load distribution in the manoeuvring conditions, the effect of corresponding pitching velocities must be taken into account. The in-trim and out-of-trim flight conditions must be considered.

(b)Manoeuvring balanced conditions:

Assuming the airplane to be in equilibrium with zero pitching acceleration, the manoeuvring conditions on the manoeuvring envelope must be investigated.

(c)Pitch manoeuvre conditions:

The movement of the pitch control surfaces may be adjusted to take into account limitations imposed by the maximum pilot effort, control system stops
and any indirect effect imposed by limitations in the output side of the control system (for example, stalling torque or maximum rate obtainable by a power control system.

Maximum pitch control displacement at V_A:

The airplane is assumed to be flying in steady level flight and the cockpit pitch control is suddenly moved to obtain extreme nose up pitching acceleration. In defining the tail load, the response of the airplane must be taken into account. Airplane loads that occur subsequent to the time when normal acceleration at the c.g. exceeds the positive limit manoeuvring load or the resulting tail plane normal load reaches its maximum, whichever occurs first, need not be considered.

Specified control displacement:

A checked manoeuvre, based on a rational pitching control motion vs. time profile, must be established in which the design limit load factor will not be exceeded. Unless lesser values cannot be exceeded, the airplane response must result in pitching accelerations not less than the following:
A positive pitching acceleration (nose up) is assumed to be reached

concurrently with the airplane load factor of 1.0. The positive acceleration must be equal to at least 39n(n-1)/v, (rad/sec)

Where 'n' is the positive load factor at the speed under consideration; and V is the airplane equivalent speed in knots.

A negative pitching acceleration (nose down) is assumed to be reached on currently with the positive manoeuvring load factor. This negative pitching acceleration must be equal to at least -26n(n-1)/v, (rad/sec)

Where 'n' is the positive load factor at the speed under consideration; and V is the airplane equivalent speed in knots.

Balancing loads:

A horizontal surface balancing load is a load necessary to maintain equilibrium in any specified flight condition with no pitching acceleration.
Horizontal balancing surfaces must be designed for the balancing loads occurring at any point on the limit manoeuvring envelope and in the flap conditions

It is not required to balance the rudder because it will not deflect due to gravity.
Aileron will defect in vice versa direction so it doesn't require balancing load.

9.DETAILED STRUCTURAL LAYOUTS
FUNCTION OF THE STRUCTURE:

The primary functions of an aircraft's structure can be basically broken down into the following:

- To transmit and resist applied loads.
- To provide and maintain aerodynamic shape.
- To protect its crew, passenger, payload, systems, etc.

For the vast majority of aircraft, this leads to use of a semi-monocoque design (i.e., a thin, stressed outer shell with additional stiffening members) for the wing, fuselage & empennage. These notes will discuss the structural layout possibilities for each of these main areas, i.e., wing, fuselage & empennage.

WING STRUCTURAL LAYOUT:

Specific Roles of Wing (Main wing) Structure:

The specified structural roles of the wing (or main plane) are:

- To transmit:

1. wing lift to the root via the main span wise beam
2. Inertia loads from the power plants, undercarriage, etc., to the main beam.
3. Aerodynamic loads generated on the aerofoil, control surfaces & flaps to the main beam.

- To react against:

1. Landing loads at attachment points
2. Loads from pylons/stores
3. Wing drag and thrust loads

- To provide:

1. Fuel tank age space

2. Torsional rigidity to satisfy stiffness and aero-elastic requirements.

To fulfill these specific roles, a wing layout will conventionally compromise:

1. Span wise members (known as spars or booms)
2. Chord wise members(ribs)
3. A covering skin
4. Stringers

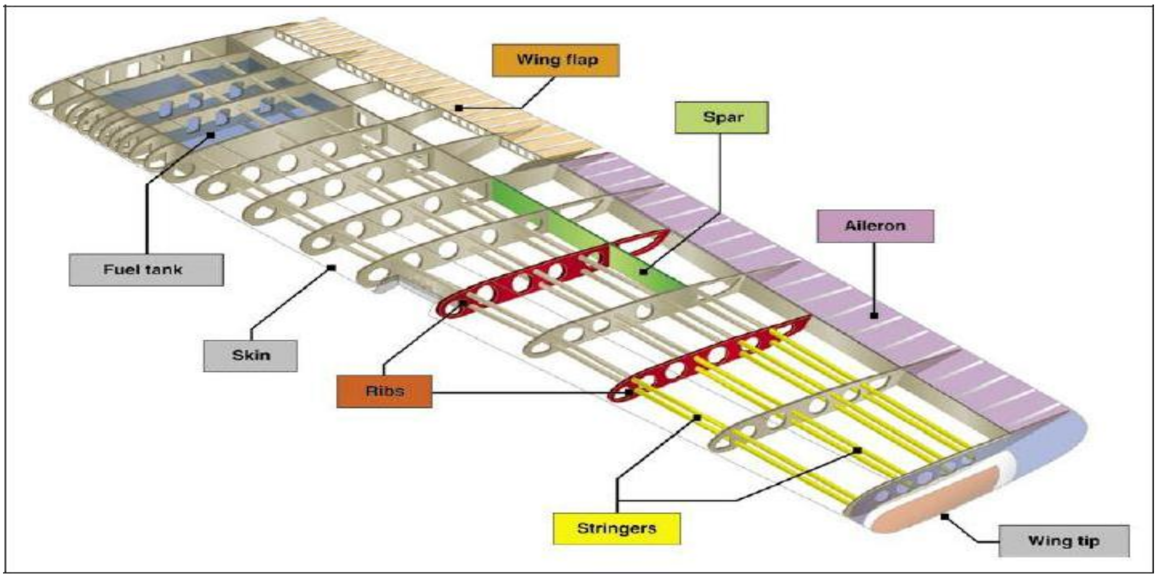

Figure 9.1 wing

Basic Functions of Wing Structural Members

The structural functions of each of these types of members may be considered independently as:

SPARS

- Form the main span wise beam
- Transmit bending and torsional loads
- Produce a closed-cell structure to provide resistance to torsion, shear and tension loads.

In particular:

- Webs – resist shear and torsional loads and help to stabilize the skin.

- Flanges - resist the compressive loads caused by wing bending.

SKIN

- To form impermeable aerodynamics surface

- Transmit aerodynamic forces to ribs & stringers
- Resist shear torsion loads (with spar webs).

- React axial bending loads (with stringers).

STRINGERS

- Increase skin panel buckling strength by dividing into smaller length sections.
- React axial bending loads

RIBS

- Maintain the aerodynamic shape

- Act along with the skin to resist the distributed aerodynamic pressure loads

- Distribute concentrated loads into the structure & redistribute stress around any discontinuities

- Increase the column buckling strength of the stringers through end restraint

- Increase the skin panel buckling strength**.**

SPARS

These usually comprise thin aluminium alloy webs and flanges, sometimes with separate vertical stiffeners riveted on to the webs.

Types of spars:

In the case of a two or three spar box beam layout, the front spar should be located as far forward as possible to maximize the wing box size, though this is subject to there being:

- Adequate wing depth for reacting vertical shear loads.

- Adequate nose space for LE devices, de-icing equipment, etc.

This generally results in the front spar being located at 12% to 18% of the chord length. For a single spar D-nose layout, the spar will usually locate at the maximum thickness position of the aerofoil section (typically between 30% & 40% along the chord length). For the standard box beam layout, the rear spar will be located as for aft as possible, once again to maximize the wing box size, but positioning will be limited by various space requirements for flaps, control surfaces, spoilers etc. This usually results in a location somewhere between about 55%and 70% of the chord length. If any intermediate spars are used, they would tend to be spaced uniformly unless there are specific pick-up point requirements.

RIBS

For a typical two spar layout, the ribs are usually formed in three parts from sheet metal by the use of presses &dies. Flanges are incorporated around the edges so that they can be riveted to the skin and the spar webs. Cut-out are necessary around the edges to allow for the stringers to pass through. Lightening holes are usually cut into the rib bodies to reduce the rib weight and also to allow for the passage of control runs, fuel, electrics, etc.

Rib bulkheads do not include any lightening holes and are used at fuel tank ends, wing crank locations and attachment support areas. The rib should be ideally spaced to ensure adequate overall buckling support to spar flanges. In reality, however, their positioning is also influenced by:

- Facilitating attachment points for control surfaces, flaps, slats, spoiler hinges, power plants, stores, undercarriage attachment etc.

- Positioning of fuel tank ends, requiring closing ribs.

- A structural need to avoid local shear or compression buckling; there are several different possibilities regarding the alignment of the ribs on swept-wing aircraft is a hybrid design in which one or more inner ribs are aligned with the main axis while the remainders are aligned perpendicularly to the rear spar and usually the preferred option but presents several structural problems in the root region also Gives good torsional stiffness characteristics but results in heavy ribs and complex connections.

SKIN

The skin tends to be riveted to the rib flanges and stringers, using countersunk rivets to reduce drag. It is usually pre-formed at the leading edges, where the curvature is large due to aerodynamic considerations.

FUSELAGE STRUCTURE

- The fundamental purpose of the fuselage structure is to provide an envelope to support the payload, crew, equipment, systems and (possibly) the power-plant. Furthermore, it must react against the in-flight manoeuvre, pressurisation and gust loads; also the landing gear and possibly any power-plant loads. It must be also be able to transmit control and trimming loads from the stability and control surfaces throughout the rest of the structure

- Fuselage contributes very little to lift and produces more drag but it is an important structural member/component. It is the connecting member to all load producing components such as wing, horizontal tail, vertical tail, landing gear etc. and thus redistributes the load. It also serves the purpose of housing or accommodating practically all equipment, accessories and systems in addition to carrying the payload. Because of large amount of equipment inside the fuselage, it is necessary to provide sufficient number of cutouts in the fuselage for access and inspection purposes. These cutouts and discontinuities result in fuselage design being more complicated, less precise and often less efficient in design.

- As a common member to which other components are attached, thereby transmitting the loads, fuselage can be considered as a long hollow beam. The reactions produced by the wing, tail or landing gear may be considered as concentrated loads at the respective attachment points. The balancing reactions are provided by the inertia forces contributed by the weight of the fuselage structure and the various components inside the fuselage. These reaction forces are distributed all along the length of the fuselage, though need not be uniformly. Unlike the wing, which is subjected to mainly unsymmetrical load,
- the fuselage is much simpler for structural analysis due to its symmetrical cross-section and symmetrical loading. The main load in the case of fuselage is the shear load because the load acting on the wing is transferred to the fuselage skin in the form of shear only. The structural design of both wing and fuselage begin with shear force and bending moment diagrams for the respective members. The maximum bending stress produced in each of them is checked to be less than the yield stress of the material chosen for the respective member.

Fuselage Layout Concepts

There are two main categories of layout concept in common use;

- Mass boom and longeron layout

- Semi-monocoque layout

Mass Boom & Longeron layout

This is fundamentally very similar to the mass-boom wing-box concept discussed in previous section. It is used when the overall structural loading is relatively low or when there are extensive cut-outs in the shell. The concept comprises four or more continuous heavy booms (longeron), reacting against any direct stresses caused by applied vertical and lateral bending loads. Frames or solid section

Semi-Monocoque layout

The semi-monocoque is the most often used construction for modern, high-performance aircraft. **Semi-monocoque** literally means half a single shell. Here, internal braces as well as the skin itself carry the stress. The vertical structural members are referred to as bulkheads, frames, and formers. The heavier vertical members are located at intervals to allow for concentrated loads. These members are also found at points where fittings are used to attach other units, such as the wings and stabilizers.

Primary bending loads are taken by the longerons, which usually extend across several points of support. The longerons are supplemented by other longitudinal members known as stringers. Stringers are more numerous and lightweight than longerons. The stringers are smaller and lighter than longerons and serve as fill-ins. They have some rigidity but are chiefly used for giving shape and for attachment of skin.

The strong, heavy longerons hold the bulkheads and formers. The bulkheads and formers hold the stringers. All of these joins together to form a rigid fuselage framework. Stringers and longerons prevent tension and compression stresses from bending the fuselage. The skin is attached to the longerons, bulkheads, and other structural members and carries part of the load. The fuselage skin thickness varies with the load carried and the stresses sustained at particular location.

10.DESIGN OF SOME COMPONENTS OF WING AND FUSELAGE

DESIGN OF WING COMPONENT

SPAR:

Wing is the major lift producing surface. Therefore, the analysis has to be very accurate. The structural analysis of the wing by defining the primary load carrying member Spars is done below.

Spars are members which are basically used to carry the bending and shear loads acting on the wing during flight. There are two spars, one located at 15-20% of the chord known as the front spar, the other located at 60-70% of the chord known as the rear spar. Some of the functions of the spar include: They form the boundary to the fuel tank located in the wing.

- The spar flange takes up the bending loads whereas the web carries the shear loads.
- The rear spar provides a means of attaching the control surfaces on the wing.

Considering these functions, the locations of the front and rear spar are fixed at 0.17c and 0.65c respectively. The spar design for the wing root has been taken because the maximum bending moment and shear force are at the root. It is assumed that the flanges take up all the bending and the web takes all the shear effect. The maximum bending moment for high angle of attack condition is Nm.

DESIGN OF FUSELAGE COMPONENT

STRINGER:

The circumference of the fuselage is 43.102 m. To find the area of one stringer, number of stringers per quadrant is assumed to be 4. i.e., the total number of stringers in the fuselage is 16. The stringers are equally spaced around the circumference of the fuselage.

11. MATERIAL SELECTION

DESCRIPTION:

Aircraft structures are basically unidirectional. This means that one dimension, the length, is much larger than the others - width or height. For example, the span of the wing and tail spars is much longer than their width and depth; the ribs have a much larger chord length than height and/or width; a whole wing has a span that is larger than its chords or thickness; and the fuselage is much longer than it is wide or high. Even a propeller has a diameter much larger than its blade width and thickness, etc.... For this simple reason, a designer chooses to

use unidirectional material when designing for an efficient strength to weight structure.

Unidirectional materials are basically composed of thin, relatively flexible, long fibers which are very strong in tension (like a thread, a rope, a stranded steel wire cable, etc.). An aircraft structure is also very close to a symmetrical structure. Those mean the up and down loads are almost equal to each other. The tail loads may be down or up depending on the pilot raising or dipping the nose of the aircraft by pulling or pushing the pitch control; the rudder may be deflected to the right as well as to the left (side loads on the fuselage). The gusts hitting the wing may be positive or negative, giving the up or down loads which the occupant experiences by being pushed down in the seat or hanging in the belt.

Because of these factors, the designer has to use a structural material that can withstand both tension and compression. Unidirectional fibers may be excellent in tension, but due to their small cross section, they have very little inertia (we will explain inertia another time) and cannot take much compression. They will escape the load by bucking away. As in the illustration, you cannot load a string, or wire, or chain in compression.

In order to make thin fibers strong in compression, they are "glued together" with some kind of an "embedding". In this way we can take advantage of their tension strength and are no longer penalized by their individual compression
weakness because, as a whole, they become compression resistant as they help each other to not buckle away. The embedding is usually a lighter, softer "resin" holding the fibers together and enabling them to take the required compression loads. This is a very good structural material.

WOOD

Historically, wood has been used as the first unidirectional structural raw material. They have to be tall and straight and their wood must be strong and light. The dark bands (late wood) contain many fibres, whereas the light bands (early wood) contain much more "resin". Thus, the wider the dark bands, the stronger and heavier the wood. If the dark bands are very narrow and the light bands quite wide, the wood is light but not very strong. To get the most efficient strength to weight ratio for wood we need a definite numbers of bands per inch. Some of our aircraft structures are two-dimensional (length and width are large with respect to thickness). Plywood is often used for such structures. Several thin boards (foils) are glued together so that the fibres of the various layers cross over at different angles (usually 90 degrees today years back you could get them at 30 and 45 degrees as well). Plywood makes excellent "shear webs" if the designer knows how to use plywood efficiently. (We will learn the basis of stress analysis sometime later.)

Today good aircraft wood is very hard to come by. Instead of using one good board for our spars, we have to use laminations because large pieces of wood are practically unavailable, and we no longer can trust the wood quality. From an availability point of view, we simply need a substitute for what nature has supplied us with until now.

ALUMINIUM ALLOYS

So, since wood may not be as available as it was before, we look at another material which is strong, light and easily available at a reasonable price (there's no point in discussing Titanium - it's simply too expensive). Aluminium alloys are certainly one answer. We will discuss the properties of those alloys which are used in light plane construction in more detail later. For the time being we will look at aluminium as a construction material.

EXTRUDED ALUMINIUM ALLOYS

Due to the manufacturing process for aluminium, we get a unidirectional material quite a bit stronger in the lengthwise direction than across. And even better, it is not only strong in tension but also in compression. Comparing extrusions to wood, the tension and compression characteristics are practically the same for aluminium alloys so that the linear stress analysis applies. Wood, on the other hand, has a tensile strength about twice as great as its compression strength; accordingly, special stress analysis methods must be used and a good understanding of wood under stress is essential if stress concentrations are to be avoided!

Aluminium alloys, in thin sheets (.016 to .125 of an inch) provide an excellent two dimensional material used extensively as shear webs - with or without stiffeners - and also as tension/compression members when suitably formed (bent).It is worthwhile to remember that aluminium is an artificial metal. There is no aluminium ore in nature. Aluminium is manufactured by applying electric power to bauxite (aluminium oxide) to obtain the metal, which is then mixed with various strength-giving additives. (In a later article, we will see which additives are used, and why and how we can increase aluminium's strength by cold work hardening or by tempering.) All the commonly used aluminium alloys are available from the shelf of dealers. When requested with the

purchase, you can obtain a "mill test report" that guarantees the chemical and physical properties as tested to accepted specifications.
As a rule of thumb, aluminium is three times heavier, but also three times stronger than wood. Steel is again three times heavier and stronger than aluminium.

STEEL

The next material to be considered for aircraft structure will thus be steel, which has the same weight-to-strength ratio of wood or aluminium.

Apart from mild steel which is used for brackets needing little strength, we are mainly using a chrome-molybdenum alloy called AISI 413ON or 4140. The common raw materials available are tubes and sheet metal. Steel, due to its high density, is not used as shear webs like aluminium sheets or plywood. Where we would need, say.100" plywood, a .032 inch aluminium sheet would be required, but only a .010 steel sheet would be required, which is just too thin to handle with any hope of a nice finish. That is why a steel fuselage uses tubes also as diagonals to carry the shear in compression or tension and the whole structure is then covered with fabric (light weight) to give it the required aerodynamic shape or desired look. It must be noted that this method involves two techniques: steel work and fabric covering. .

COMPOSITE MATERIALS

The designer of composite aircraft simply uses fibres in the desired direction exactly where and in the amount required. The fibres are embedded in resin to hold them in place and provide the required support against buckling. Instead of plywood or sheet metal which allows single curvature only, the composite designer uses cloth where the fibres are laid in two directions. (The woven thread and weft) also embedded in resin. This has the advantage of freedom of shape in double curvature as required by optimum aerodynamic shapes and for very appealing look (importance of aesthetics).

Today's fibers (glass, nylon, Kevlar, carbon, whiskers or single crystal fibres of various chemical compositions) are very strong, thus the structure becomes very light. The drawback is very little stiffness. The structure needs stiffening which is achieved either by the usual discreet stiffeners, -or more elegantly with a sandwich structure: two layers of thin uni- or bi-directional fibres are held apart by a lightweight core (foam or "honeycomb"). This allows the designer to achieve the required inertia or stiffness.

From an engineering standpoint, this method is very attractive and supported by many authorities because it allows new developments which are required in case of war. But this method also has its drawbacks for homebuilding: A mold is needed, and very strict quality control is a must for the right number of fibres and resin and for good adhesion between both to prevent too "dry" or "wet" a structure. Also, the curing of the resin is quite sensitive to temperature, humidity and pressure. Finally, the resins are active chemicals which will not only produce the well-known allergies but also the chemicals that attack our body (especially the eyes and lungs) and they have the unfortunate property of being cumulatively damaging and the result (in particular deterioration of the eye) shows up only years after initial contact.

Another disadvantage of the resins is their limited shelf life, i.e., if the resin is not used within the specified time lapse after manufacturing, the results may be unsatisfactory and unsafe.

HEAVY AIRCRAFT RAW MATERIALS

1. **Magnesium:** An expensive material. Castings are the only readily available forms. Special precaution must be taken when machining magnesium because this metal burns when hot.

2. **Titanium:** A very expensive material. Very tough material and difficult to machine.

3. **Carbon Fibbers:** Still very expensive materials.

4. **Kevlar fibres:** Very expensive and also critical to work with because it is hard to "soak" in the resin.

A number of properties are important to the selection of materials for an aircraft structure. The selection of the best material depends upon the application. Factors to be considered include yield and ultimate strength, stiffness, density, fracture toughness, fatigue, crack resistance, temperature limits, producibility, repairability, cost and availability. The gust loads, landing impact and vibrations of the engine and propeller cause fatigue failure which is the single most common cause of aircraft material failure.

For most aerospace materials, creep is a problem only at the elevated temperature. However, some titanium plastics and composites will exhibit creep at room temperatures.

Taking all the above factors into considerations, the following aluminium alloys which have excellent strength to weight ratio and are abundant in nature are considered.

12.DESIGN REPORT

Table 12.1: design report

Parameters	**Values**
Span	11 m
Planform area	12 m^2
Aspect ratio	10.08
Empty weight	7000 kg
Maximum takeoff weight	18000 kg
Chord at root	1.0909 m
Chord at tip	0.3272m
Taper ratio	0.3
Wing loading	kg/m2
Thrust-to-weight ratio	
Rate of climb	300 m/s
Range	2700 km
Stall speed	120.81 m/s
Landing distance	490 m
Takeoff distance	401.72m

13.CONCLUSION:

The structural design of the Fighter aircraft which is a continuation of the aerodynamic design part carried out last semester is completed satisfactorily. The aeroplane has gone through many design modifications since its early conceptual designs expected, among these was a growth in weight.

To ensure continued growth in payload and the reduced cost of cargo operations, improvements in methods, equipment and terminal facilities will be required in order to reduce cargo handling costs and aircraft ground time and to provide improved service for the shippers.

We have enough hard work for this design project. A design never gets completed in a flutter sense but it is one step further towards ideal system. But during the design of this aircraft, we learnt a lot about aeronautics and its implications when applied to an aircraft design.

14.BIBLIOGRAPHY:

1. *Raymer, D.P,* ***Aircraft Design - a Conceptual Approach ,*** *AIAA educational series second edition 1992.*

2. *T.H.G.Megson* ***, Aircraft Structures for engineering students,*** *4th Edition Elsevier Ltd USA 2007.*

3. *E.F.Bruhn* ***, Analysis and design of flight vehicle structures,****1st Edition, tri-state offset company,USA,1973.*

4. *Micheal Chun-Yung Niu****, Airframe structural design,*** *2nd Edition, Hong Kong Conmilit Press Ltd, Hong Kong, 2001.*

5. *Anderson, John D ,* ***Aircraft design and performance by Anderson****, 3rd Edition , Tata Mc Graw-Hill, New York , 2010.*