

# **LAB MANUAL AIRCRAFT DESIGN LAB [23AE59]**

**Prepared by**

**Dr. Sreenadh Chevula & Mr. A.Pratyush**

**Department of Aerospace Engineering**



**Lakireddy Bali Reddy College of Engineering (A)**

**Mylavaram, NTR (Dt.), Andhra Pradesh, India - 521 230**

**Affiliated To  
Jawaharlal Nehru Technological University, Kakinada, Andhra Pradesh, India**

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To empower students to become technologically vibrant, innovative, and emotionally mature; and to train them to face the challenges of the quality-conscious globalized world economy.

### **LBRCE MISSION**

1. To provide an environment most conducive to learning and to create a stimulating intellectual atmosphere on the campus.
2. To achieve Academic Excellence.
3. To ensure a holistic development of personality.
4. To spread education to rural areas.
5. To establish partnership between Institution & Industry.

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1. To achieve Academic Excellence of a high order.
2. To nurture nation-building skills with emphasis on developing a well built in personality.
3. To ensure that education spreads to rural areas.
4. To establish partnership between Institute and Industry.
5. To explore the Innate Caliber of the students by providing a unique learning experience.
6. To inculcate acquisition of superior intellectual and interpersonal skills.
7. To improve analytical thinking and problem-solving skills.
8. To understand teamwork dynamics and decision-making process.
9. To enable to cope with intense pressure over an extended period.
10. To improve the communication skills so that the students communicate cogently, rationally, and convincingly through acceptable social behaviour.
11. To generate competence to build, motivate and lead project teams effectively.
12. To make learning highly interactive and object-oriented by adopting innovative methods.

## **DEPARTMENT OF AEROSPACE ENGINEERING**

### **VISION**

Department of Aerospace Engineering aims to be indispensable source in Aerospace Engineering which has a zeal to provide the value driven platform for the students to acquire knowledge and empower themselves to shoulder higher responsibility in building a strong nation.

### **MISSION**

The primary mission of the department is to promote engineering education and research. To strive consistently to provide quality education, keeping in pace with time and technology. Department passions to integrate the intellectual, spiritual, ethical, and social development of the students for shaping them into dynamic engineers.

### **QUALITY POLICY STATEMENT**

Impart up-to-date knowledge to the students in Aerospace area to make them quality engineers. Make the students experience the applications on quality equipment and tools. Provide systems, resources, and training opportunities to achieve continuous improvement. Maintain global standards in education, training, and services.

## **23AE59- AIRCRAFT DESIGN LAB**

Course Outcomes: At the end of the semester, the student will be able

**CO1: To estimate design parameters of an aircraft system, component, or process as per the requirement [Apply- L3]**

**CO2: To calculate design parameters of an aircraft as per the assigned specifications [Apply-L3]**

### **Experiments are to be performed:**

1. Aircraft conceptual sketch and its gross weight estimation algorithm
2. Preliminary weight estimation
  - a. Jet Fighter Aircraft (Pure Turbojet Engine)
  - b. Twin turboprop A/c
  - c. Sailed –Powered A/c
  - d. Jet trainer
3. Trade off study on range
4. Trade off study on payload
5. Fixed sizing
6. Load or Induced Drag Estimation
7. Preliminary design of load distribution on a fuselage
8. Preliminary design of an aircraft fuselage
9. Estimate the Critical Mach number for an Airfoil.
10. Static Performance: Thrust required curve.
11. Static Performance: Power required curve.
12. Drawing all the 3 views of a new Aircraft

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|           | Introduction                                                                                                           |         |
| <b>1</b>  | Aircraft conceptual sketch and its gross weight estimation algorithm                                                   |         |
| <b>2</b>  | Preliminary weight estimation                                                                                          |         |
|           | 2a. Jet Fighter Aircraft (Pure Turbojet Engine)<br>2b Twin turboprop A/c<br>2c. Sailed –Powered A/c<br>2d. Jet trainer |         |
| <b>3</b>  | Trade off study on range                                                                                               |         |
| <b>4</b>  | Trade off study on payload                                                                                             |         |
| <b>5</b>  | Fixed sizing                                                                                                           |         |
| <b>6</b>  | Load or Induced Drag Estimation                                                                                        |         |
| <b>7</b>  | Preliminary design of an aircraft fuselage                                                                             |         |
| <b>8</b>  | Preliminary design of load distribution on a fuselage                                                                  |         |
| <b>9</b>  | Estimate the Critical Mach number for an Airfoil                                                                       |         |
| <b>10</b> | Static Performance: Thrust required curve                                                                              |         |
| <b>11</b> | Static Performance: Power required curve                                                                               |         |
| <b>12</b> | Drawing all the 3D views of a new Aircraft                                                                             |         |

## **INTRODUCTION**

Aircraft design is both arts and science and is a separate discipline of aeronautical engineering different from the analytical discipline such as aerodynamics, structure, control and propulsion. It is the intellectual engineering process of creating on paper a flying machine to

- 1) Meet certain specification
- 2) Certain pioneer innovative, new ideas and technology

Aircraft design is by its nature an iterative process. This means that estimates and assumptions have sometimes to be made with inadequate data. Such 'guesstimates' must be checked when more accurate data on the aircraft is available.

Requirements are set by prior design trade studies. Concepts are developed to meet requirement, design analysis frequently points towards new concepts and technologies, which can initiate a whole new design effort. All of these activities are equally important in producing a good aircraft concept. The start of the design process requires the recognition of a 'need'. This normally comes from a 'project brief' or a 'request for proposals (RFP)'.

## **PURPOSE AND SCOPE OF AIRCRAFT DESIGN**

An airplane is designed to meet the functional and safety requirements. The actual process of design is a task involving.

- a. Selection of airplane type and data
- b. Determination of the geometric parameter
- c. Selection of power plant
- d. Structural design and the analysis of various component
- e. Determination of airplane flight and operational characteristics

Designing a new airplane starts from development of a concept - a general idea for airplane. The design of any aircraft starts out in three phases

## CONCEPTUAL DESIGN

- The first design step involves sketching a variety of possible aircraft configurations that meet the required design specifications. By drawing a set of configurations, designers seek to reach the design configuration that satisfactorily meets all requirements as well as go hand in hand with factors such as aerodynamics, propulsion, flight performance, structural and control systems. This is called design optimization.
- Fundamental aspects such as fuselage shape, wing configuration and location, engine size and type are all determined at this stage. Constraints to design like those mentioned above are all taken into account at this stage as well.
- The final product is a conceptual layout of the aircraft configuration on paper or computer screen, to be reviewed by engineers and other designers.

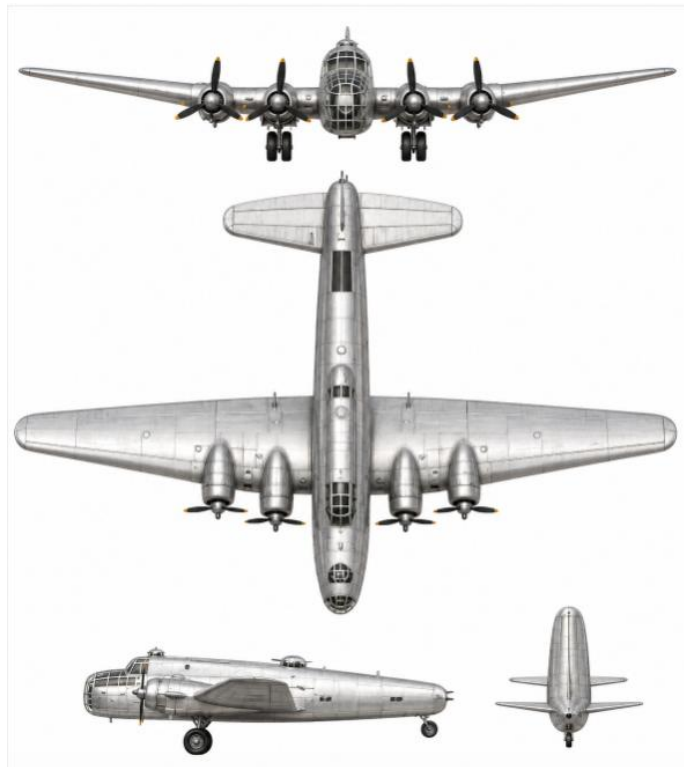


FIG 1: 3D Sketching

## 1. PRELIMINARY DESIGN PHASE

- The design configuration arrived at in the conceptual design phase is then tweaked and remodelled to fit into the design parameters.
  - In this phase, wind tunnel testing and computational fluid dynamics calculations of the flow field around the aircraft are done.
- Major structural and control analysis is also carried out in this phase. Aerodynamic flaws and structural instabilities if any are corrected and the final design is drawn and finalized. Then after the finalization of the design lies the key decision with the manufacturer or individual designing it whether to actually go ahead with the production of the aircraft. At this point several designs, though perfectly capable of flight and performance, might have been opted out of production due to their being economically nonviable.

## 2. DETAIL DESIGN PHASE

- This phase simply deals with the fabrication aspect of the aircraft to be manufactured.
- It determines the number, design and location of ribs, spars sections and other structural elements. All aerodynamic, structural, propulsion, control and performance aspects have already been covered in the preliminary design phase and only the manufacturing remains.
- Flight simulators for aircraft are also developed at this stage.



FIG2: Design Cycle

- The concept defines ways; means and parameters that should provide high efficiency and competitiveness of a future airplane, its superiority in comparison with airplanes that are already exploited or designed.
- The concept of a future airplane is determined by requirements for corresponding airplane functional and performance characteristics defined by a customer. Then a concept of a future airplane – its scheme and set of values for main parameters – is chosen.
- All main parameters that will be chosen and determined at these stages should be based on statistics and take into account the aviation development dynamics by forecasting changes of the most important airplane features and characteristics with time. It requires knowledge of the latest achievements in the field of aviation science and technology - aerodynamics, engine, equipment, weapon and airframe engineering, constructional materials, airplane operation and etc.
- Development of the airplane design concept while executing laboratory works, term and graduation design projects requires working up a list of new technical achievements in the field of aviation with estimation of its approximate positive influence on the main airplane parameters and characteristics: possible decrease of airframe mass, fuel and power plant; expected improvement of airplane performance. This list should contain specific technical innovations that should provide the improvement of efficiency ratios for the designed airplane.
- For example, in aerodynamics it can be using new supercritical Airfoil, utilizing tip vortex scatterers, wing boundary layer laminarization systems. Constructional perfection can be provided by decreasing the number of structural divisions, applying honeycomb structures, employing new materials such as aluminium, lithium and titanium alloys. While making a detailed list you may approximately estimate the influence of each innovation on corresponding airplane parameters and characteristics. This information can usually be found in dedicated literature, and it is recommended to use it while executing this stage of your project. An example of such estimation of positive influence of state-of-the-art solutions on advanced airplane performance is the fifteen - year scientific and technical forecast made by one of the foreign airplane production companies.
- According to it passenger planes designed and built with 2010th technologies have to weight on 23-35% less then airplanes of the year 1995. This significant take-off mass decrease is achieved by using following technological advances: wing boundary layer laminarization (4-6%), advanced aerodynamic schemes (6%), applying new materials (8– 10%), advanced power plants (3%), equipment and systems (1%). These numbers are just an example, and now it can be a little out of date. Fresh forecasts can be found in new publications, especially periodic, combining it with gathering statistics

## ANALYZING THE DESIGN SITUATION

- A design situation analysis is carried out on basis of statistics research and airplane development study. During this analysis the following problems should be solved:
  - 1) Future demand for airplanes should be assessed with the required number of airplanes in the certain time, for example, 5-10 years. This information can be found in publication from periodical editions, describing the airplanes of a given type, and in review magazines dedicated to international air shows.
  - 2) History of the development should be investigated, and achieved perfection degree of a given type of airplanes should be mentioned. The average statistical and maximum values of the most important performance parameters, geometric and weight parameters, the mileage rating and cost efficiency of these airplanes should be given. Also, maintenance and operating features - airplane cost, traffic handling cost, an airplane life, reliability indexes, comfort ratios and etc. should be included.
  - 3) Development prospects should be studied, and changes in main performance and relative airplane parameters should be forecasted for the near future. Thus, dynamic and static diagrams for the most important prototype parameters should be plotted using data from statistic tables; then their trend functions with an approximation error should be figured out and forecasted (extrapolated) parameter values should be found
  - 4) New engineering solutions that are expected to improve the value of each parameter should be mentioned.

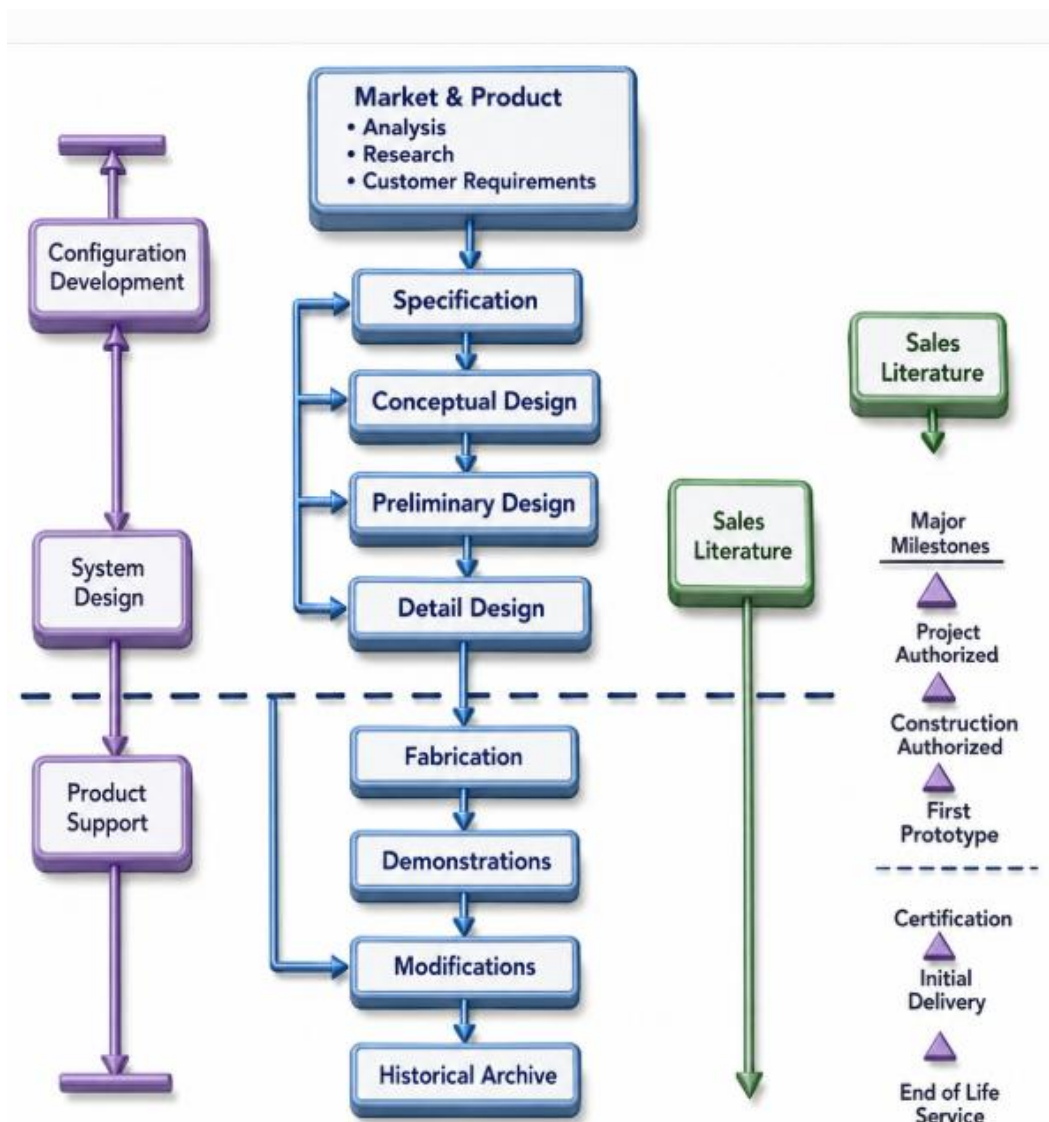


FIG 3: Design Phases

## EXPERIMENT 1

### Aircraft Conceptual Sketch and Its Gross Weight Estimation Algorithm

#### Aim

To prepare a conceptual sketch of an aircraft and estimate its gross take-off weight using an iterative weight estimation algorithm.

#### Objectives

- Understand the conceptual aircraft design process.
- Draw a preliminary aircraft configuration.
- Estimate payload, crew, fuel, and empty weight.
- Calculate the Gross Take-Off Weight (GTOW).
- Learn the iterative weight estimation procedure.

#### Theory

Aircraft conceptual design is the first stage of aircraft development where the designer determines the aircraft's general configuration based on mission requirements.

The major tasks are:

- Mission analysis
- Configuration selection
- Gross weight estimation
- Wing sizing
- Power plant selection
- Initial stability analysis

One of the most important parameters is the Gross Take-Off Weight (GTOW) because almost every aircraft component depends upon it.

#### Aircraft Weight Components

The gross weight consists of four main parts:

$$W_0 = W_e + W_f + W_p + W_c$$

where

- $W_0$  = Gross Take-Off Weight
- $W_e$  = Empty Weight
- $W_f$  = Fuel Weight
- $W_p$  = Payload Weight
- $W_c$  = Crew Weight

## Define Mission Requirements

Assume the following mission.

| Parameter       | Value            |
|-----------------|------------------|
| Aircraft Type   | Trainer Aircraft |
| Crew            | 2                |
| Passengers      | 2                |
| Cruise Speed    | 220 km/hr        |
| Range           | 900 km           |
| Service Ceiling | 5000 m           |
| Endurance       | 5 hr             |

## Select Aircraft Configuration

Prepare a conceptual sketch showing

- Nose
- Cockpit
- Wing
- Tail
- Horizontal stabilizer
- Vertical stabilizer
- Engine
- Landing gear

## Aircraft Configuration



**Estimate Payload Weight**

Payload includes passengers and baggage.  $W_p = n_p \times W_{person} + W_{baggage} =$  \_\_\_\_\_

Estimate Crew Weight  $W_c = n_c \times W_{crew} =$  \_\_\_\_\_

Empty Weight Fraction For light aircraft and Assume  $\frac{W_e}{W_0} = 0.60$  Therefore  $W_e = 0.60W_0$

**Estimate Fuel Weight**

Fuel weight is estimated using the fuel fraction.  $W_f = \left(\frac{W_f}{W_0}\right) W_0 =$  \_\_\_\_\_ the Typical value

$\frac{W_f}{W_0} = 0.15$  Hence  $W_f =$  \_\_\_\_\_  $W_0$

**Assume Initial Gross Weight** Start with  $W_0 = 1500 \text{ kg}$

**Calculate Empty Weight**  $W_e =$  \_\_\_\_\_

**Calculate Fuel Weight**  $W_f =$  \_\_\_\_\_

**Calculate Total Aircraft Weight**

Using  $W_0 = W_e + W_f + W_p + W_c$

Substitute

$$W_0 = \text{_____}$$

**Compare Estimated Weight with Assumed Weight**

Assumed Initial Gross Weight  $W_{0A} =$  1500 kg

**Estimated Initial Gross Weight**  $W_{0E} =$  \_\_\_\_\_

Difference  $W_{0A} - W_{0E} =$  \_\_\_\_\_  $\text{kg}$

Percentage error

$$\% \text{ Error} = \frac{|W_{0A} - W_{0E}|}{W_{0A}} \times 100 = \text{_____} \%$$

if the error is very small, the result is acceptable.

## Gross Weight Estimation Algorithm

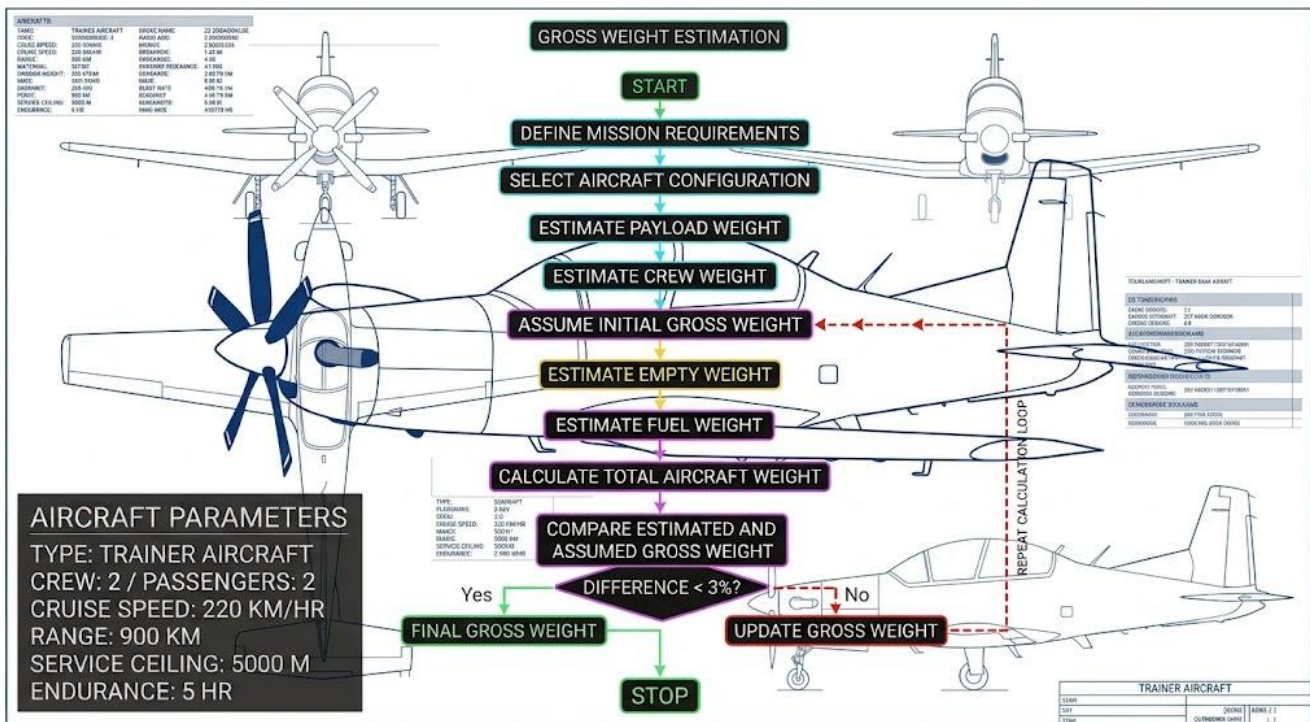


Fig. Gross Weight Estimation Algorithm

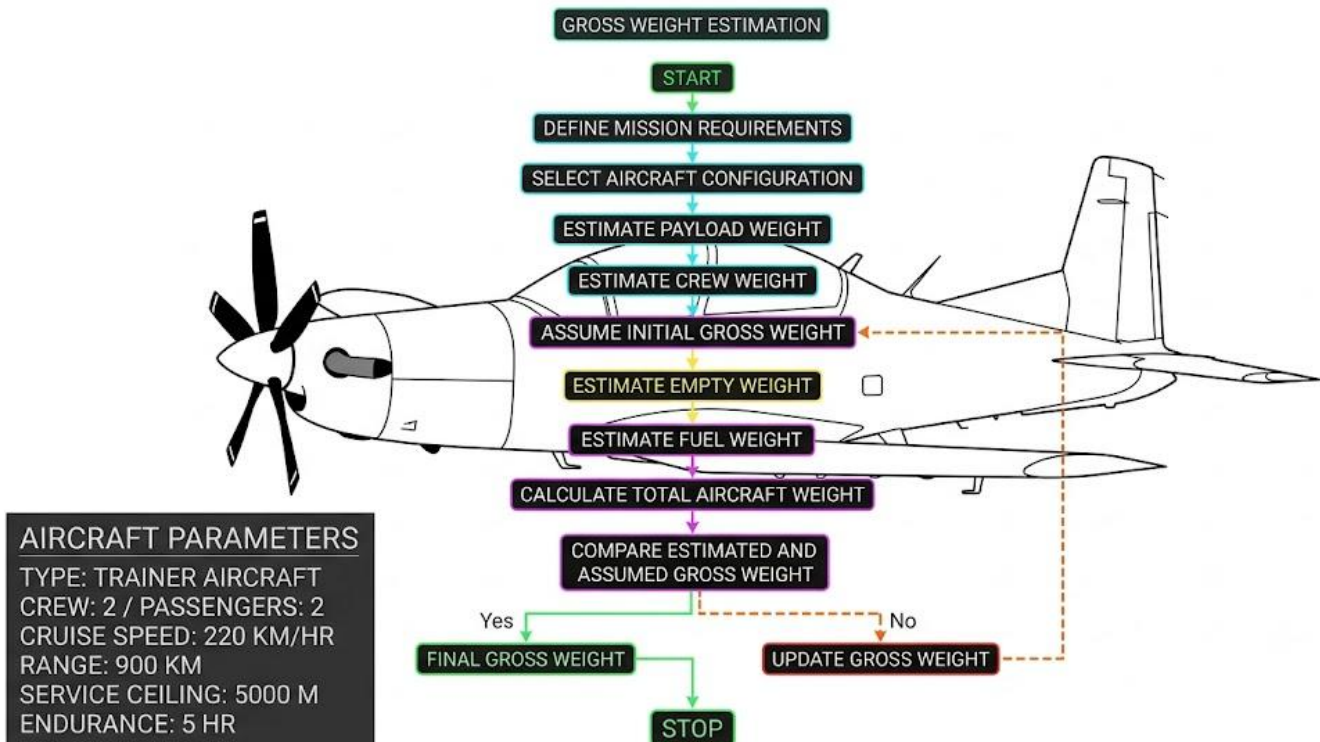


Fig. Complete Flow Chart

**Weight Estimation Summary**

| Component      | Formula                              | Value |
|----------------|--------------------------------------|-------|
| Payload Weight | $W_p = n_p W_{person} + W_{baggage}$ |       |
| Crew Weight    | $W_c = n_c W_{crew}$                 |       |
| Empty Weight   | $W_e = 0.60W_0$                      |       |
| Fuel Weight    | $W_f = 0.15W_0$                      |       |
| Gross Weight   | $W_0 = W_e + W_f + W_p + W_c$        |       |

**Precautions**

- Choose realistic mission requirements.
- Use appropriate empty and fuel weight fractions from aircraft design data.
- Ensure all weights are in the same unit (kg or N).
- Repeat the calculations until the estimated GTOW converges to the assumed value.

**Result**

A conceptual sketch of a light trainer aircraft was prepared, and the Gross Take-Off Weight (GTOW) was estimated using an iterative conceptual design algorithm. Based on the assumed mission requirements, the estimated GTOW is 1465 kg, demonstrating the standard preliminary weight estimation procedure used in aircraft conceptual design.

**EXPERIMENT 2a****PRELIMINARY WEIGHT ESTIMATION (Jet Fighter Aircraft (Pure Turbojet Engine))**

Aim: Preliminary Weight Estimation of a Jet Fighter Aircraft (Pure Turbojet Engine)

Software used: Mat lab / Excel.

Theory:

| Empty Weight Fraction vs. $W_0$ , $A$ , $T/W_0$ , $W_0/S$ , and $M_{\max}$ |       |      |       |      |      |       |      |
|----------------------------------------------------------------------------|-------|------|-------|------|------|-------|------|
| Aircraft type                                                              | (a)   | (b)  | (C1)  | (C2) | (C3) | (C4)  | (C5) |
| Jet trainer                                                                | 0     | 4.28 | -0.10 | 0.10 | 0.20 | -0.24 | 0.11 |
| Jet fighter                                                                | -0.02 | 2.16 | -0.10 | 0.20 | 0.04 | -0.10 | 0.08 |
| Military cargo/bomber                                                      | 0.07  | 1.71 | -0.10 | 0.10 | 0.06 | -0.10 | 0.05 |
| Jet transport                                                              | 0.32  | 0.66 | -0.13 | 0.30 | 0.06 | -0.05 | 0.05 |

**Variable sweep factor**

- $K_{vs} = 1.04$  for variable-sweep wings
- $K_{vs} = 1.00$  for fixed wings

Given warm-up/take-off, climb and landing weight fractions from historical trends: The warmup, take-off and landing weight fractions can be estimated historically from Table

$$\frac{W_f}{W_0} = 0.97-0.99$$

where:

- $W_f$  = final (or landing) weight
- $W_0$  = initial takeoff gross weight

**Requirements:**

Aircraft type, engine type, wing sweep type, mission profile, crew weight, specific fuel consumption, L/D ratio, different range values, different payload weight values.

**Procedure:**

1. Draw the mission profile as per the given aircraft requirements and decide the weight fractions in the mission.
2. Calculate the value of takeoff weight from table 1 as per your aircraft requirements.
3. Calculate the climb weight fraction by considering the following equation.

$$\text{Subsonic} \quad \frac{W_i}{W_{i-1}} = 1.0065 - 0.0325 M$$

$$\text{Supersonic} \quad \frac{W_i}{W_{i-1}} = 0.991 - 0.007 M - 0.01 M^2$$

where:

- $W_i$  = aircraft weight at the end of mission segment  $i$
- $W_{i-1}$  = aircraft weight at the beginning of mission segment  $i$
- $M$  = Mach number

**4. Cruise Weight Segment (Jet)**  $\frac{W_i}{W_{i-1}} = \exp\left(-\frac{RC}{V(L/D)}\right)$

where:

- $R$  = cruise range
- $C$  = specific fuel consumption
- $V$  = cruise velocity
- $L/D$  = lift-to-drag ratio

**5. Loiter Weight Fraction (Jet)**  $\frac{W_i}{W_{i-1}} = \exp\left(-\frac{EC}{L/D}\right)$

where:

- $E$  = loiter endurance
- $C$  = specific fuel consumption
- $L/D$  = lift-to-drag ratio

**6. Combat Weight Fraction**  $\frac{W_i}{W_{i-1}} = 1 - C\left(\frac{T}{W}\right)d$

**7. Descent Weight Segment**  $\frac{W_i}{W_{i-1}} = 0.990 \text{ to } 0.995$

**8. Landing Weight Segment**  $\frac{W_i}{W_{i-1}} = 0.992 \text{ to } 0.997$

**9. Fuel Weight Fraction**  $\frac{W_f}{W_0} = 1.06\left(1 - \frac{W_x}{W_0}\right)$

**10. Gross Take-off Weight**  $W_0 = \frac{W_{crew} + W_{payload}}{1 - \left(\frac{W_f}{W_0}\right) - \left(\frac{W_e}{W_0}\right)}$

where:

- $W_0$  = gross takeoff weight
- $W_{crew}$  = crew weight
- $W_{payload}$  = payload weight
- $W_f/W_0$  = fuel weight fraction
- $W_e/W_0$  = empty weight fraction

where:

- $W_f$  = fuel weight
- $W_0$  = gross takeoff weight
- $W_x$  = aircraft weight at the end of the mission (product of all mission weight fractions)

where:

- $C$  = combat fuel consumption coefficient
- $T/W$  = thrust-to-weight ratio
- $d$  = combat duration

### 11. Iterative Determination of Gross Takeoff Weight

Since both the **empty weight fraction**  $\left(\frac{W_e}{W_0}\right)$  and the **fuel weight fraction**  $\left(\frac{W_f}{W_0}\right)$  are functions of the **gross takeoff weight**  $W_0$ , the value of  $W_0$  must be determined by an iterative process.

The procedure is as follows:

1. Assume an initial value of  $W_0$ .
2. Using the assumed value, calculate the empty weight fraction  $\left(\frac{W_e}{W_0}\right)$  and the fuel weight fraction  $\left(\frac{W_f}{W_0}\right)$ .
3. Substitute these values into the gross takeoff weight equation:

$$W_0 = \frac{W_{\text{crew}} + W_{\text{payload}}}{1 - \left(\frac{W_f}{W_0}\right) - \left(\frac{W_e}{W_0}\right)}$$

4. Compare the calculated value of  $W_0$  (right-hand side) with the assumed value.
5. If the difference is significant, revise the assumed value of  $W_0$  and repeat the calculations.
6. Continue the iteration until the absolute difference between the assumed and calculated values of  $W_0$  is minimized (within the desired tolerance).

The converged value is taken as the estimated **gross take-off weight**.

### 12. Iteration Table

The iterative solution is carried out using an **iteration table**, in which the assumed and calculated values of  $W_0$  are compared at each step until convergence is achieved.

| S.no | W0 | W <sub>e</sub> /W <sub>0</sub> | RHS |
|------|----|--------------------------------|-----|
| 1    |    |                                |     |
| 2    |    |                                |     |
| 3    |    |                                |     |
| 4    |    |                                |     |
| 5    |    |                                |     |
| 6    |    |                                |     |

**Given Data****Aircraft type: Jet Fighter Aircraft (Pure Turbojet Engine)**

Range 1 = 1000 N miles

Endurance 1 = 1 hr 20 min

Range 2 = 500 N miles

Endurance 2 = 25 min

Range 3 = 1500 N miles

Endurance 3 = 2hr

SFC (cruise) = 0.9/ hr

SFC (LOITER) = 0.8/hr

 $W_c + W_p = 9000 \text{ kg}$  $(L/D)_{\text{cruise}} = 32$ 

Mach = 0.8

A = 2.34

 $c = -0.13$ **Covert the Given Data into SI Units**

N miles to meters Conversion : 1 N mile = 1852 m

1 hr = 3600 seconds

1 min = 60 seconds.

 $(L/D)_{\text{max}}$  \_\_\_\_\_ $W_c + W_p$  \_\_\_\_\_ kg

Range1 \_\_\_\_\_ m

Endurance1 \_\_\_\_\_ sec

Range2 \_\_\_\_\_ m

Endurance2 \_\_\_\_\_ sec

Range3 \_\_\_\_\_ m

Endurance3 \_\_\_\_\_ sec

SFC<sub>cruise</sub> \_\_\_\_\_ (sec)<sup>-1</sup> SFC<sub>loiter</sub> \_\_\_\_\_ (sec)<sup>-1</sup>

Velocity V = \_\_\_\_\_ m/s

**PROCEDURE TO BE FOLLOWED:****Open excel sheet.****Historical data (Raymer BOOK):** $W_1/W_0 = 0.97$  (take-off) $W_2/W_1 = 0.985$  (Climb) $W_9/W_8 = 0.995$  (Landing)

**FORMULAE:**

**BREGUT RANGE:**  $\frac{W_i}{W_{i-1}} = \exp \left[ -\frac{R_j \text{SFC}_{\text{cruise}}}{V \times 0.866 \times \left(\frac{L}{D}\right)_{\text{cruise}}} \right]$

where:

- $W_i$  = aircraft weight at the end of the cruise segment
- $W_{i-1}$  = aircraft weight at the beginning of the cruise segment
- $R_j$  = cruise range, where  $j = 1, 2, 3$  Range (given data)
- $\text{SFC}_{\text{cruise}}$  = cruise specific fuel consumption
- $V$  = cruise velocity
- $\left(\frac{L}{D}\right)_{\text{cruise}}$  = cruise lift-to-drag ratio
- 0.866 = correction factor (typically used for average cruise conditions)

**BREGUT ENDURANCE EQ**  $\frac{W_i}{W_{i-1}} = \exp \left[ -\frac{E_j \text{SFC}_{\text{loiter}}}{\left(\frac{L}{D}\right)_{\text{cruise}}} \right]$

where:

- $W_i$  = aircraft weight at the end of the loiter segment
- $W_{i-1}$  = aircraft weight at the beginning of the loiter segment
- $E_j$  = loiter endurance, where  $j = 1, 2, 3$  Endurance (given data)
- $\text{SFC}_{\text{loiter}}$  = loiter specific fuel consumption
- $\left(\frac{L}{D}\right)_{\text{cruise}}$  = cruise lift-to-drag ratio (or average lift-to-drag ratio during loiter, if applicable)

**Next Need to find the values for**

BREGUT RANGE EQ

BREGUT ENDURANCE EQ

$(W_3/W_2) = \underline{\hspace{2cm}}$

$(W_4/W_3) = \underline{\hspace{2cm}}$

$(W_5/W_4) = \underline{\hspace{2cm}}$

$(W_6/W_5) = \underline{\hspace{2cm}}$

$(W_7/W_6) = \underline{\hspace{2cm}}$

$(W_8/W_7) = \underline{\hspace{2cm}}$

$(W_9/W_0) = (W_1/W_0) * (W_2/W_1) * (W_3/W_2) * (W_4/W_3) * (W_5/W_4) * (W_6/W_5) * (W_7/W_6) * (W_8/W_7) * (W_9/W_8)$

$(W_f/W_0) = (((1 - (W_9/W_0)) * 1.06) \text{-----}(1)$

$(W_e/W_0) = A * (W_0^c) \text{-----}(2)$

$\text{RHS} = \frac{(W_c + W_p)}{1 - \left( \left( \frac{W_f}{W_0} \right) - \left( \frac{W_e}{W_0} \right) \right)} \text{-----}(3)$

Create the excel sheet with the following tables.

| Iteration | $W_0$ | $W_e/W_0$ | RHS |
|-----------|-------|-----------|-----|
| 1         | 50000 |           |     |
| 2         |       |           |     |
| 3         |       |           |     |
| 4         |       |           |     |
| 5         |       |           |     |
| 6         |       |           |     |
| .         |       |           |     |
| .         |       |           |     |
| .         |       |           |     |
| .         |       |           |     |

The initial value and 1<sup>st</sup> iteration of the  $W_0$  will be 50000 the for the given problem.

Calculate the value of  $W_e/W_0$  in the above table by using Equation (2)

Calculate the value of RHS in the above table by using Equation (3)

For the 2<sup>nd</sup> iteration the value of the  $W_0$  will be average value of 1<sup>st</sup> iteration's  $W_0$  and RHS

Calculate the value of  $W_e/W_0$  in the above table by using Equation (2) For the 2<sup>nd</sup> iteration

Calculate the value of RHS in the above table by using Equation (3) For the 2<sup>nd</sup> iteration.

For the 3<sup>rd</sup> iteration the value of the  $W_0$  will be average value of 2<sup>nd</sup> iteration's  $W_0$  and RHS

Calculate the value of  $W_e/W_0$  in the above table by using Equation (2) For the 3<sup>rd</sup> iteration

Calculate the value of RHS in the above table by using Equation (3) For the 3<sup>rd</sup> iteration

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Similarly continue the iteration until the values of  $W_0$  and RHS will be mostly similar or Equals

Then stop the iteration and that is the final obtained value of  $W_0$  n which is the initial weight of the Aircraft.

Draw a graph in excel for the obtained values of  $W_0$  and RHS

Finally, we find the weight of the given Aircraft.

**EXPERIMENT 2b****PRELIMINARY WEIGHT ESTIMATION (Twin turboprop A/c)**

Aim: Preliminary Weight Estimation of a Twin turboprop A/c

Software used: Mat lab / Excel.

Theory:

| Empty Weight Fraction vs. $W_0$ , $A$ , $T/W_0$ , $W_0/S$ , and $M_{\max}$ |       |      |       |      |      |       |      |
|----------------------------------------------------------------------------|-------|------|-------|------|------|-------|------|
| Aircraft type                                                              | (a)   | (b)  | (C1)  | (C2) | (C3) | (C4)  | (C5) |
| Jet trainer                                                                | 0     | 4.28 | -0.10 | 0.10 | 0.20 | -0.24 | 0.11 |
| Jet fighter                                                                | -0.02 | 2.16 | -0.10 | 0.20 | 0.04 | -0.10 | 0.08 |
| Military cargo/bomber                                                      | 0.07  | 1.71 | -0.10 | 0.10 | 0.06 | -0.10 | 0.05 |
| Jet transport                                                              | 0.32  | 0.66 | -0.13 | 0.30 | 0.06 | -0.05 | 0.05 |

**Variable sweep factor**

- $K_{vs} = 1.04$  for variable-sweep wings
- $K_{vs} = 1.00$  for fixed wings

Given warm-up/take-off, climb and landing weight fractions from historical trends: The warmup, take-off and landing weight fractions can be estimated historically from Table

$$\frac{W_f}{W_0} = 0.97-0.99$$

where:

- $W_f$  = final (or landing) weight
- $W_0$  = initial takeoff gross weight

**Requirements:**

Aircraft type, engine type, wing sweep type, mission profile, crew weight, specific fuel consumption, L/D ratio, different range values, different payload weight values.

**Procedure:**

4. Draw the mission profile as per the given aircraft requirements and decide the weight fractions in the mission.
5. Calculate the value of takeoff weight from table 1 as per your aircraft requirements.
6. Calculate the climb weight fraction by considering the following equation.

$$\text{Subsonic} \quad \frac{W_i}{W_{i-1}} = 1.0065 - 0.0325 M$$

$$\text{Supersonic} \quad \frac{W_i}{W_{i-1}} = 0.991 - 0.007 M - 0.01 M^2$$

where:

- $W_i$  = aircraft weight at the end of mission segment  $i$
- $W_{i-1}$  = aircraft weight at the beginning of mission segment  $i$
- $M$  = Mach number

**4. Cruise Weight Segment (Jet)**  $\frac{W_i}{W_{i-1}} = \exp\left(-\frac{RC}{V(L/D)}\right)$

where:

- $R$  = cruise range
- $C$  = specific fuel consumption
- $V$  = cruise velocity
- $L/D$  = lift-to-drag ratio

**5. Loiter Weight Fraction (Jet)**  $\frac{W_i}{W_{i-1}} = \exp\left(-\frac{EC}{L/D}\right)$

where:

- $E$  = loiter endurance
- $C$  = specific fuel consumption
- $L/D$  = lift-to-drag ratio

**6. Combat Weight Fraction**  $\frac{W_i}{W_{i-1}} = 1 - C\left(\frac{T}{W}\right)d$

**7. Descent Weight Segment**  $\frac{W_i}{W_{i-1}} = 0.990 \text{ to } 0.995$

**8. Landing Weight Segment**  $\frac{W_i}{W_{i-1}} = 0.992 \text{ to } 0.997$

**9. Fuel Weight Fraction**  $\frac{W_f}{W_0} = 1.06\left(1 - \frac{W_x}{W_0}\right)$

**10. Gross Take-off Weight**  $W_0 = \frac{W_{crew} + W_{payload}}{1 - \left(\frac{W_f}{W_0}\right) - \left(\frac{W_e}{W_0}\right)}$

where:

- $W_0$  = gross takeoff weight
- $W_{crew}$  = crew weight
- $W_{payload}$  = payload weight
- $W_f/W_0$  = fuel weight fraction
- $W_e/W_0$  = empty weight fraction

where:

- $W_f$  = fuel weight
- $W_0$  = gross takeoff weight
- $W_x$  = aircraft weight at the end of the mission (product of all mission weight fractions)

where:

- $C$  = combat fuel consumption coefficient
- $T/W$  = thrust-to-weight ratio
- $d$  = combat duration

### 11. Iterative Determination of Gross Take-off Weight

Since both the **empty weight fraction**  $\left(\frac{W_e}{W_0}\right)$  and the **fuel weight fraction**  $\left(\frac{W_f}{W_0}\right)$  are functions of the **gross takeoff weight**  $W_0$ , the value of  $W_0$  must be determined by an iterative process.

The procedure is as follows:

4. Assume an initial value of  $W_0$ .
5. Using the assumed value, calculate the empty weight fraction  $\left(\frac{W_e}{W_0}\right)$  and the fuel weight fraction  $\left(\frac{W_f}{W_0}\right)$ .
6. Substitute these values into the gross take-off weight equation:

$$W_0 = \frac{W_{\text{crew}} + W_{\text{payload}}}{1 - \left(\frac{W_f}{W_0}\right) - \left(\frac{W_e}{W_0}\right)}$$

7. Compare the calculated value of  $W_0$  (right-hand side) with the assumed value.
8. If the difference is significant, revise the assumed value of  $W_0$  and repeat the calculations.
9. Continue the iteration until the absolute difference between the assumed and calculated values of  $W_0$  is minimized (within the desired tolerance).

The converged value is taken as the estimated **gross take-off weight**.

### 12. Iteration Table

The iterative solution is carried out using an **iteration table**, in which the assumed and calculated values of  $W_0$  are compared at each step until convergence is achieved.

| S.no | $W_0$ | $W_{e/0}$ | RHS |
|------|-------|-----------|-----|
| 1    |       |           |     |
| 2    |       |           |     |
| 3    |       |           |     |
| 4    |       |           |     |
| 5    |       |           |     |
| 6    |       |           |     |

**Given Data**

|                |            |                     |                |          |                     |
|----------------|------------|---------------------|----------------|----------|---------------------|
| (L/D) max      | 16         |                     |                |          |                     |
| $W_c + W_p$    | 10000      | kg                  |                |          |                     |
| Range1         | 740000     | m                   | Endurance1     | 2400     | sec                 |
| Range2         | 900000     | m                   | Endurance2     | 4800     | sec                 |
| Range3         | 850000     | m                   | Endurance3     | 3600     | sec                 |
| $SFC_{cruise}$ | 0.00028417 | (sec) <sup>-1</sup> | $SFC_{loiter}$ | 0.000341 | (sec) <sup>-1</sup> |
| M= 0.8         | A= 0.96    | c = -0.05           |                |          |                     |

Velocity V= \_\_\_\_\_ m/s

**PROCEDURE TO BE FOLLOWED:**

Open excel sheet.

**Historical data (Raymer BOOK):**

$W_1/W_0 = 0.97$  (take-off)

$W_2/W_1 = 0.985$  (Climb)

$W_9/W_8 = 0.995$  (Landing)

**FORMULAE:**

$$\text{BREGUT RANGE: } \frac{W_i}{W_{i-1}} = \exp \left[ - \frac{R_j SFC_{cruise}}{V \times 0.866 \times \left( \frac{L}{D} \right)_{cruise}} \right]$$

where:

- $W_i$  = aircraft weight at the end of the cruise segment
- $W_{i-1}$  = aircraft weight at the beginning of the cruise segment
- $R_j$  = cruise range, where  $j = 1, 2, 3$  Range (given data)
- $SFC_{cruise}$  = cruise specific fuel consumption
- $V$  = cruise velocity
- $\left( \frac{L}{D} \right)_{cruise}$  = cruise lift-to-drag ratio
- 0.866 = correction factor (typically used for average cruise conditions)

$$\text{BREGUT ENDURANCE EQ } \frac{W_i}{W_{i-1}} = \exp \left[ - \frac{E_j SFC_{loiter}}{\left( \frac{L}{D} \right)_{cruise}} \right]$$

where:

- $W_i$  = aircraft weight at the end of the loiter segment
- $W_{i-1}$  = aircraft weight at the beginning of the loiter segment
- $E_j$  = loiter endurance, where  $j = 1, 2, 3$  Endurance (given data)
- $SFC_{loiter}$  = loiter specific fuel consumption
- $\left( \frac{L}{D} \right)_{cruise}$  = cruise lift-to-drag ratio (or average lift-to-drag ratio during loiter, if applicable)

**Need to find the values for****BREGUT RANGE EQ**

$$(W_3/W_2) = \underline{\hspace{2cm}}$$

$$(W_5/W_4) = \underline{\hspace{2cm}}$$

$$(W_7/W_6) = \underline{\hspace{2cm}}$$

**BREGUT ENDURANCE EQ**

$$(W_4/W_3) = \underline{\hspace{2cm}}$$

$$(W_6/W_5) = \underline{\hspace{2cm}}$$

$$(W_8/W_7) = \underline{\hspace{2cm}}$$

$$(W_9/W_0) = (W_1/W_0) * (W_2/W_1) * (W_3/W_2) * (W_4/W_3) * (W_5/W_4) * (W_6/W_5) * (W_7/W_6) * (W_8/W_7) * (W_9/W_8)$$

$$(W_f/W_0) = (((1 - (W_9/W_0)) * 1.06) \text{-----}(1)$$

$$(W_e/W_0) = A * (W_0^c) \text{-----}(2)$$

$$\text{RHS} = \frac{(W_c + W_p)}{1 - \left( \left( \frac{W_f}{W_0} \right) - \left( \frac{W_e}{W_0} \right) \right)} \text{-----}(3)$$

Create the excel sheet with the following tables.

| Iteration | $W_0$ | $W_e/W_0$ | RHS |
|-----------|-------|-----------|-----|
| 1         | 50000 |           |     |
| 2         |       |           |     |
| 3         |       |           |     |
| 4         |       |           |     |
| 5         |       |           |     |
| 6         |       |           |     |
| .         |       |           |     |
| .         |       |           |     |
| .         |       |           |     |
| .         |       |           |     |

For the 1<sup>st</sup> iteration of the  $W_0$  will be 50000 the for the given problem.

Calculate the value of  $W_e/W_0$  in the above table by using Equation (2)

Calculate the value of RHS in the above table by using Equation (3)

For the 2<sup>nd</sup> iteration the value of the  $W_0$  will be average value of 1<sup>st</sup> iteration's  $W_0$  and RHS.

Calculate the value of  $W_e/W_0$  in the above table by using Equation (2) For the 2<sup>nd</sup> iteration

Calculate the value of RHS in the above table by using Equation (3) For the 2<sup>nd</sup> iteration.

For the 3<sup>rd</sup> iteration the value of the  $W_0$  will be **average value of 2nd** iteration's  $W_0$  and RHS.

Calculate the value of  $W_e/W_0$  in the above table by using Equation (2) For the 3<sup>rd</sup> iteration

Calculate the value of RHS in the above table by using Equation (3) For the 3<sup>rd</sup> iteration

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Similarly continue the iteration until the values of  $W_0$  and RHS will be mostly similar or Equals.

Then stop the iteration and that is the final obtained value of  $W_0$  which is the initial weight of the Aircraft.

Draw a graph in excel for the obtained values of  $W_0$  and RHS

Finally, we find the primary weight of the given Aircraft.

**EXPERIMENT 2c****PRELIMINARY WEIGHT ESTIMATION (Sailed –Powered A/c)**

Aim: Preliminary Weight Estimation of a Sailed –Powered A/c

Software used: Mat lab / Excel.

Theory:

| Empty Weight Fraction vs. $W_0$ , $A$ , $T/W_0$ , $W_0/S$ , and $M_{\max}$ |       |      |                   |                   |                   |                   |                   |
|----------------------------------------------------------------------------|-------|------|-------------------|-------------------|-------------------|-------------------|-------------------|
| Aircraft type                                                              | (a)   | (b)  | (C <sub>1</sub> ) | (C <sub>2</sub> ) | (C <sub>3</sub> ) | (C <sub>4</sub> ) | (C <sub>5</sub> ) |
| Jet trainer                                                                | 0     | 4.28 | -0.10             | 0.10              | 0.20              | -0.24             | 0.11              |
| Jet fighter                                                                | -0.02 | 2.16 | -0.10             | 0.20              | 0.04              | -0.10             | 0.08              |
| Military cargo/bomber                                                      | 0.07  | 1.71 | -0.10             | 0.10              | 0.06              | -0.10             | 0.05              |
| Jet transport                                                              | 0.32  | 0.66 | -0.13             | 0.30              | 0.06              | -0.05             | 0.05              |

**Variable sweep factor**

- $K_{vs} = 1.04$  for variable-sweep wings
- $K_{vs} = 1.00$  for fixed wings

Given warm-up/take-off, climb and landing weight fractions from historical trends: The warmup, take-off and landing weight fractions can be estimated historically from Table

$$\frac{W_f}{W_0} = 0.97-0.99$$

where:

- $W_f$  = final (or landing) weight
- $W_0$  = initial takeoff gross weight

**Requirements:**

Aircraft type, engine type, wing sweep type, mission profile, crew weight, specific fuel consumption, L/D ratio, different range values, different payload weight values.

**Procedure:**

7. Draw the mission profile as per the given aircraft requirements and decide the weight fractions in the mission.
8. Calculate the value of takeoff weight from table 1 as per your aircraft requirements.
9. Calculate the climb weight fraction by considering the following equation.

$$\text{Subsonic} \quad \frac{W_i}{W_{i-1}} = 1.0065 - 0.0325 M$$

$$\text{Supersonic} \quad \frac{W_i}{W_{i-1}} = 0.991 - 0.007 M - 0.01 M^2$$

where:

- $W_i$  = aircraft weight at the end of mission segment  $i$
- $W_{i-1}$  = aircraft weight at the beginning of mission segment  $i$
- $M$  = Mach number

**4. Cruise Weight Segment (Jet)**  $\frac{W_i}{W_{i-1}} = \exp\left(-\frac{RC}{V(L/D)}\right)$

where:

- $R$  = cruise range
- $C$  = specific fuel consumption
- $V$  = cruise velocity
- $L/D$  = lift-to-drag ratio

**5. Loiter Weight Fraction (Jet)**  $\frac{W_i}{W_{i-1}} = \exp\left(-\frac{EC}{L/D}\right)$

where:

- $E$  = loiter endurance
- $C$  = specific fuel consumption
- $L/D$  = lift-to-drag ratio

**6. Combat Weight Fraction**  $\frac{W_i}{W_{i-1}} = 1 - C\left(\frac{T}{W}\right)d$

**7. Descent Weight Segment**  $\frac{W_i}{W_{i-1}} = 0.990 \text{ to } 0.995$

**8. Landing Weight Segment**  $\frac{W_i}{W_{i-1}} = 0.992 \text{ to } 0.997$

**9. Fuel Weight Fraction**  $\frac{W_f}{W_0} = 1.06\left(1 - \frac{W_x}{W_0}\right)$

**10. Gross Take-off Weight**  $W_0 = \frac{W_{crew} + W_{payload}}{1 - \left(\frac{W_f}{W_0}\right) - \left(\frac{W_e}{W_0}\right)}$

where:

- $W_0$  = gross takeoff weight
- $W_{crew}$  = crew weight
- $W_{payload}$  = payload weight
- $W_f/W_0$  = fuel weight fraction
- $W_e/W_0$  = empty weight fraction

where:

- $W_f$  = fuel weight
- $W_0$  = gross takeoff weight
- $W_x$  = aircraft weight at the end of the mission (product of all mission weight fractions)

where:

- $C$  = combat fuel consumption coefficient
- $T/W$  = thrust-to-weight ratio
- $d$  = combat duration

### 11. Iterative Determination of Gross Take-off Weight

Since both the **empty weight fraction**  $\left(\frac{W_e}{W_0}\right)$  and the **fuel weight fraction**  $\left(\frac{W_f}{W_0}\right)$  are functions of the **gross takeoff weight**  $W_0$ , the value of  $W_0$  must be determined by an iterative process.

The procedure is as follows:

7. Assume an initial value of  $W_0$ .
8. Using the assumed value, calculate the empty weight fraction  $\left(\frac{W_e}{W_0}\right)$  and the fuel weight fraction  $\left(\frac{W_f}{W_0}\right)$ .
9. Substitute these values into the gross take-off weight equation:

$$W_0 = \frac{W_{\text{crew}} + W_{\text{payload}}}{1 - \left(\frac{W_f}{W_0}\right) - \left(\frac{W_e}{W_0}\right)}$$

10. Compare the calculated value of  $W_0$  (right-hand side) with the assumed value.
11. If the difference is significant, revise the assumed value of  $W_0$  and repeat the calculations.
12. Continue the iteration until the absolute difference between the assumed and calculated values of  $W_0$  is minimized (within the desired tolerance).

The converged value is taken as the estimated **gross take-off weight**.

### 12. Iteration Table

The iterative solution is carried out using an **iteration table**, in which the assumed and calculated values of  $W_0$  are compared at each step until convergence is achieved.

| S.no | $W_0$ | $W_{e/0}$ | RHS |
|------|-------|-----------|-----|
| 1    |       |           |     |
| 2    |       |           |     |
| 3    |       |           |     |
| 4    |       |           |     |
| 5    |       |           |     |
| 6    |       |           |     |

**Given Data****Aircraft type: Sailed –Powered A/c**

|                       |      |                    |                       |       |                    |
|-----------------------|------|--------------------|-----------------------|-------|--------------------|
| $(L/D)_{\max}$        | 32   |                    |                       |       |                    |
| $W_c + W_p$           | 9000 | kg                 |                       |       |                    |
| Range1                | 555  | KM                 | Endurance1            | 150   | min                |
| Range2                | 1110 | KM                 | Endurance2            | 240   | min                |
| Range3                | 1665 | KM                 | Endurance             | 60    | min                |
| $SFC_{\text{cruise}}$ | 0.9  | $(\text{hr})^{-1}$ | $SFC_{\text{loiter}}$ | 0.8   | $(\text{hr})^{-1}$ |
| $A =$                 | 0.91 |                    | $c =$                 | -0.05 |                    |
|                       |      |                    | $M =$                 | 0.8   |                    |

**Covert the Given Data into SI Units**

N miles to meters Conversion: 1 N mile = 1852 m  
 1 hr = 3600 seconds  
 1 min = 60 seconds.

$(L/D)_{\max}$  \_\_\_\_\_  
 $W_c + W_p$  \_\_\_\_\_ kg  
 Range1 \_\_\_\_\_ m      Endurance1 \_\_\_\_\_ sec  
 Range2 \_\_\_\_\_ m      Endurance2 \_\_\_\_\_ sec  
 Range3 \_\_\_\_\_ m      Endurance3 \_\_\_\_\_ sec  
 $SFC_{\text{cruise}}$  \_\_\_\_\_  $(\text{sec})^{-1}$      $SFC_{\text{loiter}}$  \_\_\_\_\_  $(\text{sec})^{-1}$   
 Velocity  $V =$  \_\_\_\_\_ m/s

**PROCEDURE TO BE FOLLOWED:**

**Open excel sheet.**

**Historical data (Raymer BOOK):**

$W_1/W_0 =$  0.97 (take-off)  
 $W_2/W_1 =$  0.985 (Climb)  
 $W_9/W_8 =$  0.995 (Landing)

$$\text{BREGUT RANGE: } \frac{W_i}{W_{i-1}} = \exp \left[ - \frac{R_j SFC_{\text{cruise}}}{V \times 0.866 \times \left( \frac{L}{D} \right)_{\text{cruise}}} \right]$$

where:

- $W_i$  = aircraft weight at the end of the cruise segment
- $W_{i-1}$  = aircraft weight at the beginning of the cruise segment
- $R_j$  = cruise range, where  $j = 1, 2, 3$  Range (given data)
- $SFC_{\text{cruise}}$  = cruise specific fuel consumption

- $V$  = cruise velocity
- $\left(\frac{L}{D}\right)_{\text{cruise}}$  = cruise lift-to-drag ratio
- 0.866 = correction factor (typically used for average cruise conditions)

**BREGUT ENDURANCE EQ**  $\frac{W_i}{W_{i-1}} = \exp \left[ - \frac{E_j \text{SFC}_{\text{loiter}}}{\left(\frac{L}{D}\right)_{\text{cruise}}} \right]$

where:

- $W_i$  = aircraft weight at the end of the loiter segment
- $W_{i-1}$  = aircraft weight at the beginning of the loiter segment
- $E_j$  = loiter endurance, where  $j = 1, 2, 3$  Endurance (given data)
- $\text{SFC}_{\text{loiter}}$  = loiter specific fuel consumption
- $\left(\frac{L}{D}\right)_{\text{cruise}}$  = cruise lift-to-drag ratio (or average lift-to-drag ratio during loiter, if applicable)

**Next Need to find the values for**

**BREGUT RANGE EQ**

**BREGUT ENDURANCE EQ**

$(W_3/W_2) =$  \_\_\_\_\_

$(W_4/W_3) =$  \_\_\_\_\_

$(W_5/W_4) =$  \_\_\_\_\_

$(W_6/W_5) =$  \_\_\_\_\_

$(W_7/W_6) =$  \_\_\_\_\_

$(W_8/W_7) =$  \_\_\_\_\_

$(W_9/W_0) = (W_1/W_0) * (W_2/W_1) * (W_3/W_2) * (W_4/W_3) * (W_5/W_4) * (W_6/W_5) * (W_7/W_6) * (W_8/W_7) * (W_9/W_8)$

$(W_f/W_0) = (((1 - (W_9/W_0)) * 1.06) \text{-----}(1)$

$(W_e/W_0) = A * (W_0^c) \text{-----}(2)$

$\text{RHS} = \frac{(W_c + W_p)}{1 - \left( \left( \frac{W_f}{W_0} \right) - \left( \frac{W_e}{W_0} \right) \right)} \text{-----}(3)$

Create the excel sheet with the following tables.

| Iteration | $W_0$ | $W_e/W_0$ | RHS |
|-----------|-------|-----------|-----|
| 1         | 30000 |           |     |
| 2         |       |           |     |
| 3         |       |           |     |
| 4         |       |           |     |
| 5         |       |           |     |
| 6         |       |           |     |
| .         |       |           |     |
| .         |       |           |     |
| .         |       |           |     |
| .         |       |           |     |

The initial value and 1<sup>st</sup> iteration of the  $W_0$  will be 30000 the for the given problem.

Calculate the value of  $W_e/W_0$  in the above table by using Equation (2)

Calculate the value of RHS in the above table by using Equation (3)

For the 2<sup>nd</sup> iteration the value of the  $W_0$  will be average value of 1<sup>st</sup> iteration's  $W_0$  and RHS

Calculate the value of  $W_e/W_0$  in the above table by using Equation (2) For the 2<sup>nd</sup> iteration

Calculate the value of RHS in the above table by using Equation (3) For the 2<sup>nd</sup> iteration.

For the 3<sup>rd</sup> iteration the value of the  $W_0$  will be average value of 2<sup>nd</sup> iteration's  $W_0$  and RHS

Calculate the value of  $W_e/W_0$  in the above table by using Equation (2) For the 3<sup>rd</sup> iteration

Calculate the value of RHS in the above table by using Equation (3) For the 3<sup>rd</sup> iteration

·  
·  
·  
·  
·  
·

Similarly continue the iteration until the values of  $W_0$  and RHS will be mostly similar or Equals

Then stop the iteration and that is the final obtained value of  $W_0$  n which is the initial weight of the Aircraft.

Draw a graph in excel for the obtained values of  $W_0$  and RHS

Finally, we find the weight of the given Aircraft.

**EXPERIMENT 2d****PRELIMINARY WEIGHT ESTIMATION (Jet trainer))**

Aim: Preliminary Weight Estimation (Jet Trainer))

Software used: Mat lab / Excel.

Theory:

| <b>Empty Weight Fraction vs. <math>W_0</math>, <math>A</math>, <math>T/W_0</math>, <math>W_0/S</math>, and <math>M_{\max}</math></b> |            |            |                        |                        |                        |                        |                        |
|--------------------------------------------------------------------------------------------------------------------------------------|------------|------------|------------------------|------------------------|------------------------|------------------------|------------------------|
| <b>Aircraft type</b>                                                                                                                 | <b>(a)</b> | <b>(b)</b> | <b>(C<sub>1</sub>)</b> | <b>(C<sub>2</sub>)</b> | <b>(C<sub>3</sub>)</b> | <b>(C<sub>4</sub>)</b> | <b>(C<sub>5</sub>)</b> |
| Jet trainer                                                                                                                          | 0          | 4.28       | -0.10                  | 0.10                   | 0.20                   | -0.24                  | 0.11                   |
| Jet fighter                                                                                                                          | -0.02      | 2.16       | -0.10                  | 0.20                   | 0.04                   | -0.10                  | 0.08                   |
| Military cargo/bomber                                                                                                                | 0.07       | 1.71       | -0.10                  | 0.10                   | 0.06                   | -0.10                  | 0.05                   |
| Jet transport                                                                                                                        | 0.32       | 0.66       | -0.13                  | 0.30                   | 0.06                   | -0.05                  | 0.05                   |

**Variable sweep factor**

- $K_{vs} = 1.04$  for variable-sweep wings
- $K_{vs} = 1.00$  for fixed wings

Given warm-up/take-off, climb and landing weight fractions from historical trends: The warmup, take-off and landing weight fractions can be estimated historically from Table

$$\frac{W_f}{W_0} = 0.97-0.99$$

where:

- $W_f$  = final (or landing) weight
- $W_0$  = initial takeoff gross weight

**Requirements:**

Aircraft type, engine type, wing sweep type, mission profile, crew weight, specific fuel consumption, L/D ratio, different range values, different payload weight values.

**Procedure:**

10. Draw the mission profile as per the given aircraft requirements and decide the weight fractions in the mission.
11. Calculate the value of takeoff weight from table 1 as per your aircraft requirements.
12. Calculate the climb weight fraction by considering the following equation.

$$\text{Subsonic} \quad \frac{W_i}{W_{i-1}} = 1.0065 - 0.0325 M$$

$$\text{Supersonic} \quad \frac{W_i}{W_{i-1}} = 0.991 - 0.007 M - 0.01 M^2$$

where:

- $W_i$  = aircraft weight at the end of mission segment  $i$
- $W_{i-1}$  = aircraft weight at the beginning of mission segment  $i$
- $M$  = Mach number

**4. Cruise Weight Segment (Jet)**  $\frac{W_i}{W_{i-1}} = \exp\left(-\frac{RC}{V(L/D)}\right)$

where:

- $R$  = cruise range
- $C$  = specific fuel consumption
- $V$  = cruise velocity
- $L/D$  = lift-to-drag ratio

**5. Loiter Weight Fraction (Jet)**  $\frac{W_i}{W_{i-1}} = \exp\left(-\frac{EC}{L/D}\right)$

where:

- $E$  = loiter endurance
- $C$  = specific fuel consumption
- $L/D$  = lift-to-drag ratio

**6. Combat Weight Fraction**  $\frac{W_i}{W_{i-1}} = 1 - C\left(\frac{T}{W}\right)d$

**7. Descent Weight Segment**  $\frac{W_i}{W_{i-1}} = 0.990 \text{ to } 0.995$

**8. Landing Weight Segment**  $\frac{W_i}{W_{i-1}} = 0.992 \text{ to } 0.997$

**9. Fuel Weight Fraction**  $\frac{W_f}{W_0} = 1.06\left(1 - \frac{W_x}{W_0}\right)$

**10. Gross Take-off Weight**  $W_0 = \frac{W_{crew} + W_{payload}}{1 - \left(\frac{W_f}{W_0}\right) - \left(\frac{W_e}{W_0}\right)}$

where:

- $W_0$  = gross takeoff weight
- $W_{crew}$  = crew weight
- $W_{payload}$  = payload weight
- $W_f/W_0$  = fuel weight fraction
- $W_e/W_0$  = empty weight fraction

where:

- $W_f$  = fuel weight
- $W_0$  = gross takeoff weight
- $W_x$  = aircraft weight at the end of the mission (product of all mission weight fractions)

where:

- $C$  = combat fuel consumption coefficient
- $T/W$  = thrust-to-weight ratio
- $d$  = combat duration

### 11. Iterative Determination of Gross Take-off Weight

Since both the **empty weight fraction**  $\left(\frac{W_e}{W_0}\right)$  and the **fuel weight fraction**  $\left(\frac{W_f}{W_0}\right)$  are functions of the **gross takeoff weight**  $W_0$ , the value of  $W_0$  must be determined by an iterative process.

The procedure is as follows:

10. Assume an initial value of  $W_0$ .

11. Using the assumed value, calculate the empty weight fraction  $\left(\frac{W_e}{W_0}\right)$  and the fuel weight fraction  $\left(\frac{W_f}{W_0}\right)$ .

12. Substitute these values into the gross take-off weight equation:

$$W_0 = \frac{W_{\text{crew}} + W_{\text{payload}}}{1 - \left(\frac{W_f}{W_0}\right) - \left(\frac{W_e}{W_0}\right)}$$

13. Compare the calculated value of  $W_0$  (right-hand side) with the assumed value.

14. If the difference is significant, revise the assumed value of  $W_0$  and repeat the calculations.

15. Continue the iteration until the absolute difference between the assumed and calculated values of  $W_0$  is minimized (within the desired tolerance).

The converged value is taken as the estimated **gross take-off weight**.

### 12. Iteration Table

The iterative solution is carried out using an **iteration table**, in which the assumed and calculated values of  $W_0$  are compared at each step until convergence is achieved.

| S.no | $W_0$ | $W_{e/0}$ | RHS |
|------|-------|-----------|-----|
| 1    |       |           |     |
| 2    |       |           |     |
| 3    |       |           |     |
| 4    |       |           |     |
| 5    |       |           |     |
| 6    |       |           |     |

**Given Data****Aircraft type: Jet trainer A/c**

(L/D) max 16

 $W_c + W_p$  5000 kg

Range1 1852 KM Endurance1 80 min

Range2 926 KM Endurance2 20 min

Range3 400 KM Endurance 120 min

 $SFC_{cruise}$  0.9 (hr)<sup>-1</sup>  $SFC_{loiter}$  0.8 (hr)<sup>-1</sup>

A= 1.59 c = -0.1 M= 0.6

**Covert the Given Data into SI Units**

N miles to meters Conversion: 1 N mile = 1852 m

1 hr = 3600 seconds

1 min = 60 seconds.

(L/D)<sub>max</sub> \_\_\_\_\_ $W_c + W_p$  \_\_\_\_\_ kg

Range1 \_\_\_\_\_ m Endurance1 \_\_\_\_\_ sec

Range2 \_\_\_\_\_ m Endurance2 \_\_\_\_\_ sec

Range3 \_\_\_\_\_ m Endurance3 \_\_\_\_\_ sec

 $SFC_{cruise}$  \_\_\_\_\_ (sec)<sup>-1</sup>  $SFC_{loiter}$  \_\_\_\_\_ (sec)<sup>-1</sup>

Velocity V= \_\_\_\_\_ m/s

**PROCEDURE TO BE FOLLOWED:****Open excel sheet.****Historical data (Raymer BOOK):** $W_1/W_0$ = 0.97 (take-off) $W_2/W_1$ = 0.985 (Climb) $W_9/W_8$ = 0.995 (Landing)**FORMULAE:**

**BREGUT RANGE:** 
$$\frac{W_i}{W_{i-1}} = \exp \left[ - \frac{R_j SFC_{cruise}}{V \times 0.866 \times \left( \frac{L}{D} \right)_{cruise}} \right]$$

where:

- $W_i$ = aircraft weight at the end of the cruise segment
- $W_{i-1}$ = aircraft weight at the beginning of the cruise segment
- $R_j$ = cruise range, where  $j=1,2,3$  Range (given data)

- $SFC_{\text{cruise}}$  = cruise specific fuel consumption
- $V$  = cruise velocity
- $\left(\frac{L}{D}\right)_{\text{cruise}}$  = cruise lift-to-drag ratio
- 0.866 = correction factor (typically used for average cruise conditions)

**BREGUT ENDURANCE EQ**  $\frac{W_i}{W_{i-1}} = \exp \left[ -\frac{E_j SFC_{\text{loiter}}}{\left(\frac{L}{D}\right)_{\text{cruise}}} \right]$

where:

- $W_i$  = aircraft weight at the end of the loiter segment
- $W_{i-1}$  = aircraft weight at the beginning of the loiter segment
- $E_j$  = loiter endurance, where  $j=1,2,3$  Endurance (given data)
- $SFC_{\text{loiter}}$  = loiter specific fuel consumption
- $\left(\frac{L}{D}\right)_{\text{cruise}}$  = cruise lift-to-drag ratio (or average lift-to-drag ratio during loiter, if applicable)

**Next Need to find the values for**

**BREGUT RANGE EQ**

$$(W_3/W_2) = \underline{\hspace{2cm}}$$

$$(W_5/W_4) = \underline{\hspace{2cm}}$$

$$(W_7/W_6) = \underline{\hspace{2cm}}$$

$$(W_9/W_0) = (W_1/W_0) * (W_2/W_1) * (W_3/W_2) * (W_4/W_3) * (W_5/W_4) * (W_6/W_5) * (W_7/W_6) * (W_8/W_7) * (W_9/W_8)$$

$$(W_f/W_0) = (((1 - (W_9/W_0)) * 1.06) \text{ -----(1)}$$

$$(W_e/W_0) = A * (W_0^c) \text{ -----(2)}$$

$$\text{RHS} = \frac{(W_c + W_p)}{1 - \left( \left( \frac{W_f}{W_0} \right) - \left( \frac{W_e}{W_0} \right) \right)} \text{ -----(3)}$$

Create the excel sheet with the following tables.

| Iteration | $W_0$ | $W_e/W_0$ | RHS |
|-----------|-------|-----------|-----|
| 1         | 30000 |           |     |
| 2         |       |           |     |
| 3         |       |           |     |
| 4         |       |           |     |
| 5         |       |           |     |
| 6         |       |           |     |
| .         |       |           |     |
| .         |       |           |     |
| .         |       |           |     |

|   |  |  |  |
|---|--|--|--|
| . |  |  |  |
|---|--|--|--|

The initial value and 1<sup>st</sup> iteration of the  $W_0$  will be 30000 the for the given problem.

Calculate the value of  $W_e/W_0$  in the above table by using Equation (2)

Calculate the value of RHS in the above table by using Equation (3)

For the 2<sup>nd</sup> iteration the value of the  $W_0$  will be average value of 1<sup>st</sup> iteration's  $W_0$  and RHS

Calculate the value of  $W_e/W_0$  in the above table by using Equation (2) For the 2<sup>nd</sup> iteration

Calculate the value of RHS in the above table by using Equation (3) For the 2<sup>nd</sup> iteration.

For the 3<sup>rd</sup> iteration the value of the  $W_0$  will be average value of 2<sup>nd</sup> iteration's  $W_0$  and RHS

Calculate the value of  $W_e/W_0$  in the above table by using Equation (2) For the 3<sup>rd</sup> iteration

Calculate the value of RHS in the above table by using Equation (3) For the 3<sup>rd</sup> iteration

.

.

.

.

.

Similarly continue the iteration until the values of  $W_0$  and RHS will be mostly similar or Equals

Then stop the iteration and that is the final obtained value of  $W_0$  n which is the initial weight of the Aircraft.

Draw a graph in excel for the obtained values of  $W_0$  and RHS

Finally, we find the weight of the given Aircraft.

## EXPERIMENT 3

### Trade-Off Study on Aircraft Range Using the Breguet Range Equation

#### Experiment Title

Trade-Off Study on Aircraft Range Using the Breguet Range Equation

#### Aim

To study the effect of various aircraft design parameters on aircraft range and understand the trade-offs between fuel weight, lift-to-drag ratio, specific fuel consumption, cruise speed, and aircraft weight.

#### Learning Objectives

After completing this experiment, students will be able to:

1. Understand the concept of aircraft range.
2. Apply the Breguet Range Equation.
3. Analyze the effect of design parameters on range.
4. Perform sensitivity analysis.
5. Recommend design improvements based on engineering trade-offs.

#### Theory

The range of a jet aircraft is estimated using the **Breguet Range Equation**.  $R = \frac{V}{c} \left( \frac{L}{D} \right) \ln \left( \frac{W_i}{W_f} \right)$

Where

- **R** = Aircraft range (km)
- **V** = Cruise speed (km/hr)
- **c** = Specific fuel consumption (1/hr)
- **L/D** = Lift-to-drag ratio
- **Wi** = Initial aircraft weight (kg)
- **Wf** = Final aircraft weight (kg)

#### Factors Affecting Range

- Cruise speed
- Aerodynamic efficiency (L/D)
- Fuel consumption
- Fuel fraction
- Aircraft structural weight

#### Apparatus / Software Required

- Calculator or Excel
- MATLAB/Python (optional)
- Aircraft Design Handbook

**Record Baseline Aircraft Data**

| Parameter                   | Symbol | Value                 |
|-----------------------------|--------|-----------------------|
| Cruise Speed                | V      | 850 km/hr             |
| Lift-to-Drag Ratio          | L/D    | 16                    |
| Specific Fuel Consumption c |        | 0.60 hr <sup>-1</sup> |
| Initial Weight              | Wi     | 70,000 kg             |
| Final Weight                | Wf     | 50,000 kg             |

**Calculate Baseline Range**

Substitute the values into the Breguet equation.  $R = \frac{850}{0.60} \times 16 \times \ln\left(\frac{70000}{50000}\right)$

Calculate

- Wi/Wf \_\_\_\_\_
- ln(Wi/Wf) \_\_\_\_\_
- Final range \_\_\_\_\_

Answer  $R =$  \_\_\_\_\_

**Trade-Off Study**

Change only one parameter at a time while keeping the others constant.

a) Effect of Cruise Speed

| Speed (km/hr) | Range (km) |
|---------------|------------|
| 750           |            |
| 800           |            |
| 850           |            |
| 900           |            |
| 950           |            |

**Observation:**

- Does increasing speed always improve range? \_\_\_\_\_

## b) Effect of Lift-to-Drag Ratio

| L/D | Range (km) |
|-----|------------|
| 12  |            |
| 14  |            |
| 16  |            |
| 18  |            |
| 20  |            |

Observation:

- Identify the percentage improvement.

## c). Effect of Specific Fuel Consumption

| SFC (1/hr) | Range (km) |
|------------|------------|
| 0.70       |            |
| 0.65       |            |
| 0.60       |            |
| 0.55       |            |
| 0.50       |            |

Observation:

- Lower SFC results in \_\_\_\_\_ range.

## d) Effect of Fuel Fraction

| Wi (kg) | Wf (kg) | Range |
|---------|---------|-------|
| 65000   | 50000   |       |
| 70000   | 50000   |       |
| 75000   | 50000   |       |
| 80000   | 50000   |       |

Observation:

- More fuel increases range but also increases take-off weight.

**e) Combined Trade-Off**

Prepare the following combinations.

| Case | Speed | L/D | SFC | Range |
|------|-------|-----|-----|-------|
| 1    |       |     |     |       |
| 2    |       |     |     |       |
| 3    |       |     |     |       |
| 4    |       |     |     |       |

**Data Analysis**

Students should plot the following graphs:

1. Range vs Cruise Speed
2. Range vs Lift-to-Drag Ratio
3. Range vs Specific Fuel Consumption
4. Range vs Fuel Fraction

Interpret the trends observed from each graph.

**Precautions**

- Keep all other parameters constant while varying one parameter.
- Use consistent units throughout the calculations.
- Verify logarithmic calculations carefully.
- Round off results to three decimal places where appropriate.

**Result**

Students should summarize:

- Baseline aircraft range.
- Parameter with the highest sensitivity.
- Best design configuration obtained.
- Engineering trade-offs between speed, fuel, weight, and aerodynamic efficiency.

**Viva Questions**

1. Define aircraft range.
2. State the Breguet Range Equation.
3. What is lift-to-drag ratio?
4. What is specific fuel consumption?
5. Why is logarithmic weight ratio used in the Breguet equation?
6. What is fuel fraction?

## EXPERIMENT 4

### Trade-Off Study on Aircraft Payload Using Payload–Range Analysis

#### Aim

To study the effect of payload variation on aircraft performance and range, and to understand the engineering trade-offs between payload, fuel weight, and aircraft take-off weight.

#### Objectives

After completing this experiment, students will be able to:

- Understand the concept of aircraft payload.
- Analyze the relationship between payload, fuel weight, and aircraft range.
- Perform a payload trade-off study using aircraft weight constraints.
- Interpret payload–range characteristics.
- Recommend an optimum payload configuration for different mission requirements.

#### Theory

##### Payload

Payload is the useful load carried by an aircraft for commercial or military purposes.

Examples include:

- Passengers
- Cargo
- Baggage
- Mail
- Mission equipment

Payload excludes:

- Fuel
- Crew
- Empty aircraft structure

#### Aircraft Weight Equation

$$W_{TO} = W_E + W_P + W_F$$

Where

- $W_{TO}$  = Maximum Take-Off Weight (kg)
- $W_E$  = Empty Weight (kg)
- $W_P$  = Payload Weight (kg)
- $W_F$  = Fuel Weight (kg)

Since Maximum Take-Off Weight (MTOW) is fixed,  
Increasing payload reduces the fuel that can be carried.  
Therefore,

**Higher Payload → Lower Fuel → Lower Range**

**Apparatus / Software Required**

- Calculator
- Microsoft Excel
- MATLAB/Python (optional)
- Aircraft Design Handbook
- Laboratory Manual

**Aircraft Data**

| Parameter               | Symbol  | Value     |
|-------------------------|---------|-----------|
| Maximum Take-Off Weight | MTOW    | 70,000 kg |
| Empty Weight            | WE      | 40,000 kg |
| Maximum Fuel Weight     | WF(max) | 20,000 kg |
| Maximum Payload         | WP(max) | 15,000 kg |

Calculate the available fuel weight using

$$W_F = W_{TO} - W_E - W_P$$

*Assume different payload values.*

**Case    Payload (kg)**

|   |       |
|---|-------|
| 1 | 5000  |
| 2 | 7000  |
| 3 | 9000  |
| 4 | 11000 |
| 5 | 13000 |
| 6 | 15000 |

Calculate Fuel Weight

For each payload,

$$Fuel = MTOW - Empty - Payload$$

Observation Table

| Payload (kg) | Fuel Weight (kg) |
|--------------|------------------|
| 5000         |                  |
| 7000         |                  |
| 9000         |                  |
| 11000        |                  |
| 13000        |                  |
| 15000        |                  |

Estimate Aircraft Range

Assume

- Cruise Speed = 850 km/hr
- Lift-to-Drag Ratio = 16
- Specific Fuel Consumption =  $0.60 \text{ hr}^{-1}$

Use the **Breguet Range Equation**  $R = \frac{V}{c} \left( \frac{L}{D} \right) \ln \left( \frac{W_i}{W_f} \right)$

where:

- $W_i = MTOW$
- $W_f = MTOW - \text{Fuel Weight}$

Calculate the range for each payload case.

Observation Table

| Payload (kg) | Fuel Weight (kg) | Range (km) |
|--------------|------------------|------------|
| 5000         |                  |            |
| 7000         |                  |            |
| 9000         |                  |            |
| 11000        |                  |            |
| 13000        |                  |            |
| 15000        |                  |            |

## **Trade-Off Study**

### **Case 1: Payload vs Fuel Weight**

Observe how increasing payload affects fuel weight.

### **Case 2: Payload vs Range**

Observe how aircraft range changes with payload.

### **Case 3: Payload vs Revenue**

Assume

Revenue = ₹600 per kg of payload.

Calculate total revenue.

| Payload | Revenue |
|---------|---------|
| 5000    |         |
| 7000    |         |
| 9000    |         |
| 11000   |         |
| 13000   |         |
| 15000   |         |

Discuss the balance between higher payload revenue and reduced operational range.

### **Graphs to Plot**

Students should plot:

1. Payload vs Fuel Weight
2. Payload vs Aircraft Range
3. Payload vs Revenue
4. Fuel Weight vs Aircraft Range

Interpret each graph and explain the observed trends.

### **Precautions**

- Maintain a constant MTOW for all calculations.
- Vary only the payload while keeping other parameters fixed.
- Use consistent units throughout.
- Verify calculations before plotting graphs.

**Result**

Students should report:

- Fuel weight corresponding to each payload.
- Estimated range for each payload.
- Payload configuration that provides the best balance between range and carrying capacity.
- Key trade-offs identified between payload, fuel, and mission performance.

**Viva Questions**

1. What is aircraft payload?
2. What is Maximum Take-Off Weight (MTOW)?
3. Why does increasing payload reduce fuel capacity?
4. What is the difference between payload and useful load?
5. Why is payload important in airline economics?
6. What is payload–range trade-off?
7. Why do cargo aircraft have different payload characteristics than passenger aircraft?
8. What limits the maximum payload an aircraft can carry?
9. How does payload influence aircraft performance?
10. Why is payload optimization essential during aircraft design?

## EXPERIMENT 5

### Fixed Sizing of an Aircraft Using Preliminary Design Methodology

#### Aim

To perform the preliminary (fixed) sizing of an aircraft by estimating the Maximum Take-Off Weight (MTOW), Empty Weight, Fuel Weight, Wing Area, Thrust-to-Weight Ratio, and Wing Loading based on specified mission requirements.

#### Learning Objectives

After completing this experiment, students will be able to:

- Understand the concept of preliminary aircraft sizing.
- Estimate the aircraft take-off weight using empirical methods.
- Calculate empty weight and fuel weight fractions.
- Determine wing loading and wing area.
- Estimate engine thrust requirements.
- Analyze the influence of design requirements on aircraft size.

#### Theory

Aircraft sizing is the first stage of aircraft design, where the principal dimensions and weights of the aircraft are estimated to satisfy specified mission requirements.

The sizing process determines:

- Maximum Take-Off Weight (MTOW)
- Empty Weight
- Fuel Weight
- Payload Weight
- Wing Area
- Engine Thrust
- Wing Loading
- Thrust-to-Weight Ratio

These parameters are interrelated and significantly affect aircraft performance, safety, and economics.

#### Design Requirements

Assume the following design specifications:

| Parameter          | Symbol            | Value     |
|--------------------|-------------------|-----------|
| Payload Weight     | (W <sub>P</sub> ) | 12,000 kg |
| Cruise Speed       | V                 | 850 km/h  |
| Cruise Altitude    | H                 | 11 km     |
| Design Range       | R                 | 3,000 km  |
| Number of Engines  | N                 | 2         |
| Passenger Capacity | —                 | 120       |

### **Apparatus / Software Required**

- Calculator
- Microsoft Excel
- MATLAB/Python (optional)
- Aircraft Design Data Book
- Laboratory Manual

### **Formulae Used**

1. Aircraft Weight Equation  $W_{TO} = W_E + W_F + W_P$

Where:

- $W_{TO}$  = Maximum Take-Off Weight
- $W_E$  = Empty Weight
- $W_F$  = Fuel Weight
- $W_P$  = Payload Weight

### **Empty Weight Fraction**

Assume  $\frac{W_E}{W_{TO}} = 0.55$  Therefore,  $W_E = 0.55W_{TO}$

### **Fuel Weight Fraction**

Assume  $\frac{W_F}{W_{TO}} = 0.20$  Therefore,  $W_F = 0.20W_{TO}$

### **Estimation of MTOW**

Using  $W_{TO} = \frac{W_P}{1 - \left(\frac{W_E}{W_{TO}}\right) - \left(\frac{W_F}{W_{TO}}\right)} =$  \_\_\_\_\_

**Wing Loading :** Assume  $\frac{W}{S} = \frac{600 \text{ kg}}{\text{m}^2}$  Wing Area  $S = \frac{W_{TO}}{W/S} = \underline{\hspace{2cm}}$

### Thrust-to-Weight Ratio

Assume  $\frac{T}{W} = 0.30$  Required Thrust  $T = 0.30 \times W_{TO} = \underline{\hspace{2cm}}$

### Experimental Procedure

Assume

- Empty Weight Fraction = 0.55
- Fuel Weight Fraction = 0.20

Calculate Maximum Take-Off Weight  $W_{TO} = \underline{\hspace{2cm}}$

Calculate Empty Weight  $W_E = 0.55W_{TO} = \underline{\hspace{2cm}}$

Calculate Fuel Weight  $W_F = 0.20W_{TO} = \underline{\hspace{2cm}}$

Verify  $W_{TO} = W_E + W_F + W_P \underline{\hspace{2cm}}$

Calculate Wing Area Wing Loading = 600 kg/m<sup>2</sup>,  $S = \frac{W_{TO}}{600} = \underline{\hspace{2cm}}$

Calculate Engine Thrust : Thrust-to-Weight Ratio = 0.30,  $T = 0.30W_{TO}$  for twin engines,  $T_{engine} = \frac{T}{2} = \underline{\hspace{2cm}}$

| Parameter         | Symbol       | Calculated Value      |
|-------------------|--------------|-----------------------|
| Payload Weight    | $W_P$        |                       |
| MTOW              | $W_{TO}$     |                       |
| Empty Weight      | $W_E$        |                       |
| Fuel Weight       | $W_F$        |                       |
| Wing Loading      | W/S          | 600 kg/m <sup>2</sup> |
| Wing Area         | S            |                       |
| Total Thrust      | T            |                       |
| Thrust per Engine | $T_{engine}$ |                       |

### Graphs to Plot

Students should plot:

1. Payload vs MTOW
2. Payload vs Wing Area
3. Payload vs Fuel Weight
4. Payload vs Engine Thrust

Interpret the trends observed in each graph.

**Precautions**

- Use consistent SI units throughout the calculations.
- Assume realistic weight fractions based on the aircraft category.
- Verify that the sum of empty weight, fuel weight, and payload equals MTOW.
- Round off results appropriately.

**Result**

Students should summarize:

- Estimated Maximum Take-Off Weight (MTOW).
- Empty Weight and Fuel Weight.
- Wing Area required.
- Total Engine Thrust and Thrust per Engine.
- The effect of payload on aircraft sizing parameters.

**Viva Questions**

1. What is preliminary aircraft sizing?
2. Define Maximum Take-Off Weight (MTOW).
3. What are empty weight and fuel weight fractions?
4. What is wing loading, and why is it important?
5. Define thrust-to-weight ratio.
6. Why is iterative sizing often required in aircraft design?
7. How does payload affect MTOW?
8. Why is wing area a critical design parameter?
9. What assumptions are commonly made during preliminary sizing?
10. How can preliminary sizing influence aircraft performance and economics?

## EXPERIMENT 6

### Load (Induced) Drag Estimation of an Aircraft

#### Aim

To estimate the induced drag coefficient and induced drag force of an aircraft at different lift coefficients using aerodynamic theory and compare the results.

#### Objectives

After completing this experiment, students will be able to:

- Understand the concept of induced drag.
- Calculate the lift coefficient.
- Estimate induced drag coefficient.
- Calculate induced drag force.
- Study the effect of aspect ratio on induced drag.
- Analyze the influence of Oswald efficiency factor.

#### Apparatus / Software Required

- Calculator
- Microsoft Excel / MATLAB / Python (optional)
- Aircraft geometric data
- Atmospheric data sheet
- Lab manual

#### Theory

An aircraft experiences two major types of drag:

1. Parasite Drag
2. Induced (Load) Drag

Induced drag is produced because lift is generated by finite wings. Wingtip vortices create downwash, which tilts the lift vector rearward, producing drag.

The induced drag becomes maximum during:

- Take-off
- Landing
- Low-speed flight
- High angle of attack

It decreases significantly during high-speed cruise.

**Formulae**

**1. Lift Equation**  $L = \frac{1}{2} \rho V^2 S C_L$

where

- $L$  = Lift (N)
- $\rho$  = Air density (kg/m<sup>3</sup>)
- $V$  = Velocity (m/s)
- $S$  = Wing area (m<sup>2</sup>)
- $C_L$  = Lift coefficient

**2. Lift Coefficient**  $C_L = \frac{2W}{\rho V^2 S}$

where

- $W$  = Aircraft weight

**3. Aspect Ratio**  $AR = \frac{b^2}{S}$

where

- $b$  = Wing span (m)

**4. Induced Drag Coefficient**  $C_{Di} = \frac{C_L^2}{\pi e AR} = \underline{\hspace{2cm}}$

Where  $e$  = Oswald efficiency factor and Typical value  $e = 0.75 - 0.9 = \underline{\hspace{2cm}}$

**5. Dynamic Pressure**  $q = \frac{1}{2} \rho V^2$

**6. Induced Drag**  $D_i = q S C_{Di}$

**Given Aircraft Data**

| Parameter         | Value                   |
|-------------------|-------------------------|
| Weight            | 60,000 N                |
| Wing Span         | 16 m                    |
| Wing Area         | 32 m <sup>2</sup>       |
| Air Density       | 1.225 kg/m <sup>3</sup> |
| Flight Speed      | 60 m/s                  |
| Oswald Efficiency | 0.8                     |

Calculate Aspect Ratio  $AR = \underline{\hspace{2cm}}$

Calculate Lift Coefficient  $C_L = \underline{\hspace{2cm}}$

Calculate Induced Drag Coefficient  $C_{Di} = \underline{\hspace{2cm}}$

Calculate Dynamic Pressure  $q = \frac{1}{2} \times \text{Air Density} \times \text{flight speed}^2 = \underline{\hspace{2cm}}$

Calculate Induced Drag  $D_i = \underline{\hspace{2cm}}$

### Observation Table

| Trial | Velocity<br>(m/s) | Lift<br>Coefficient | Aspect<br>Ratio | Oswald<br>Factor | Induced Drag<br>Coefficient | Induced Drag<br>(N) |
|-------|-------------------|---------------------|-----------------|------------------|-----------------------------|---------------------|
| 1     | 50                |                     | 8               | 0.8              |                             |                     |
| 2     | 60                |                     | 8               | 0.8              |                             |                     |
| 3     | 70                |                     | 8               | 0.8              |                             |                     |
| 4     | 80                |                     | 8               | 0.8              |                             |                     |
| 5     | 90                |                     | 8               | 0.8              |                             |                     |

### Graphs to Plot

Students should plot:

1. Lift Coefficient vs Induced Drag Coefficient
2. Velocity vs Induced Drag
3. Aspect Ratio vs Induced Drag
4. Oswald Efficiency vs Induced Drag
5. Lift Coefficient vs Velocity

### Result

The induced drag coefficient and induced drag force were estimated using aerodynamic relations. The results show that:

- Induced drag increases with increasing lift coefficient.
- Induced drag decreases as aspect ratio increases.
- Higher Oswald efficiency reduces induced drag.
- Induced drag is significant at low flight speeds and high lift conditions.

### Precautions

- Use consistent SI units throughout the calculations.
- Assume steady and level flight ( $L = W$ ).
- Use the correct value of air density for the operating altitude.
- Verify aircraft geometry before calculations.
- Avoid rounding intermediate values excessively.

### **Viva Questions**

1. What is induced drag?
2. Why does induced drag occur?
3. How does aspect ratio affect induced drag?
4. What is Oswald efficiency factor?
5. Why is induced drag maximum during take-off and landing?
6. What are wingtip vortices?
7. How is induced drag different from parasite drag?
8. Why do gliders have long wings?
9. How can induced drag be reduced?
10. Why are winglets used on modern aircraft?

## EXPERIMENT 7

### Preliminary design of an aircraft fuselage

#### Aim

To perform the preliminary design of an aircraft fuselage by estimating its dimensions, layout, structural arrangement, and weight based on mission and payload requirements.

#### Objectives

After completing this experiment, students will be able to:

- Understand the function of the fuselage.
- Estimate fuselage dimensions from design requirements.
- Prepare a preliminary fuselage layout.
- Identify major structural members.
- Estimate fuselage weight using empirical methods.

#### Theory

The fuselage is the main body of an aircraft that accommodates:

- Cockpit
- Crew and passengers
- Payload
- Fuel (in some aircraft)
- Landing gear attachment
- Wing and tail attachments
- Aircraft systems

A good fuselage should possess:

- High strength
- Low weight
- Aerodynamic shape
- Adequate internal space
- Ease of manufacturing and maintenance

**Design Inputs**

| Parameter     | Value                    |
|---------------|--------------------------|
| Aircraft Type | Light Transport Aircraft |
| Passengers    | 6                        |
| Crew          | 2                        |
| Payload       | 800 kg                   |
| Cruise Speed  | 250 km/h                 |
| Range         | 1000 km                  |
| MTOW          | 3500 kg                  |

**Select Fuselage Configuration**

Choose the aircraft configuration.

Example

- Conventional fuselage
- Low-wing aircraft
- Tricycle landing gear
- Circular cross-section

Reason:

Circular sections have better pressure distribution and structural efficiency.

**Estimate Cabin Dimensions**

Determine the seating arrangement.

2 seats per row

Passenger seat width = 0.50 m

Aisle width = 0.40 m

Wall clearance = 0.10 m

Cabin Width  $W_c = 2(0.50) + 0.40 + 2(0.10)$  ,  $W_c = 1.60 \text{ m}$

Choose

**Cabin Width = 1.6 m**

**Cabin Height**

Seat height = 0.90 m

Head clearance = 0.40 m

Floor thickness = 0.20 m

Cabin Height  $H_c = 1.50 \text{ m}$

Choose **Cabin Height = 1.6 m**

### **Estimate Cabin Length**

Passengers = 6

Rows = 3

Seat pitch = 0.80 m

Cabin Length  $L_c = 3 \times 0.80 = L_c = 2.4 \text{ m}$

Add

Cockpit = 1.3 m

Rear baggage = 1.0 m

Total Cabin  $2.4 + 1.3 + 1.0 = L = 4.7 \text{ m}$

### **Nose Length**

Generally  $L_n = (0.12 - 0.18)L_f$

Assume 15%

Initially assume  $L_f = 8.5\text{m}$

Therefore  $L_n = 1.3\text{m}$

### ***Tail Cone Length***

*Typically 20–25% of fuselage length*

Choose  $L_t = 2.0\text{m}$

### ***Total Fuselage Length***

$$L_f = L_n + L_c + L_t$$

$$L_f = 1.3 + 4.7 + 2.0$$

$$L_f = 8.0\text{m}$$

### ***Select Fuselage Diameter***

*Choose larger of Cabin width*

*or*

*Cabin height Therefore*

$$D = 1.6\text{m}$$

### **Calculate Fineness Ratio**

$$FR = \frac{L_f}{D} = \frac{8.0}{1.6} \text{ so } FR = 5$$

Typical values

- Transport aircraft: 5–8
- Fighter aircraft: 8–12
- UAV: 6–10

Hence acceptable.

**Select Structural Arrangement**

Typical semi-monocoque structure consists of

- Skin
- Frames
- Bulkheads
- Longerons
- Stringers
- Floor beams

**Locate Major Components**

Prepare the layout.

| Component       | Location   |
|-----------------|------------|
| Cockpit         | Nose       |
| Cabin           | Center     |
| Baggage         | Rear cabin |
| Wing attachment | CG region  |
| Landing gear    | Below wing |
| Tail attachment | Tail cone  |

**Estimate Wetted Area**

Approximate equation  $S_w = \pi DL = \pi \times 1.6 \times 8$  so,  $S_w = 40.2 \text{ m}^2$

**Estimate Fuselage Weight**

Empirical relation  $W_f = 0.12 \times MTOW$

For

MTOW

3500 kg so  $W_f = 420 \text{ kg}$

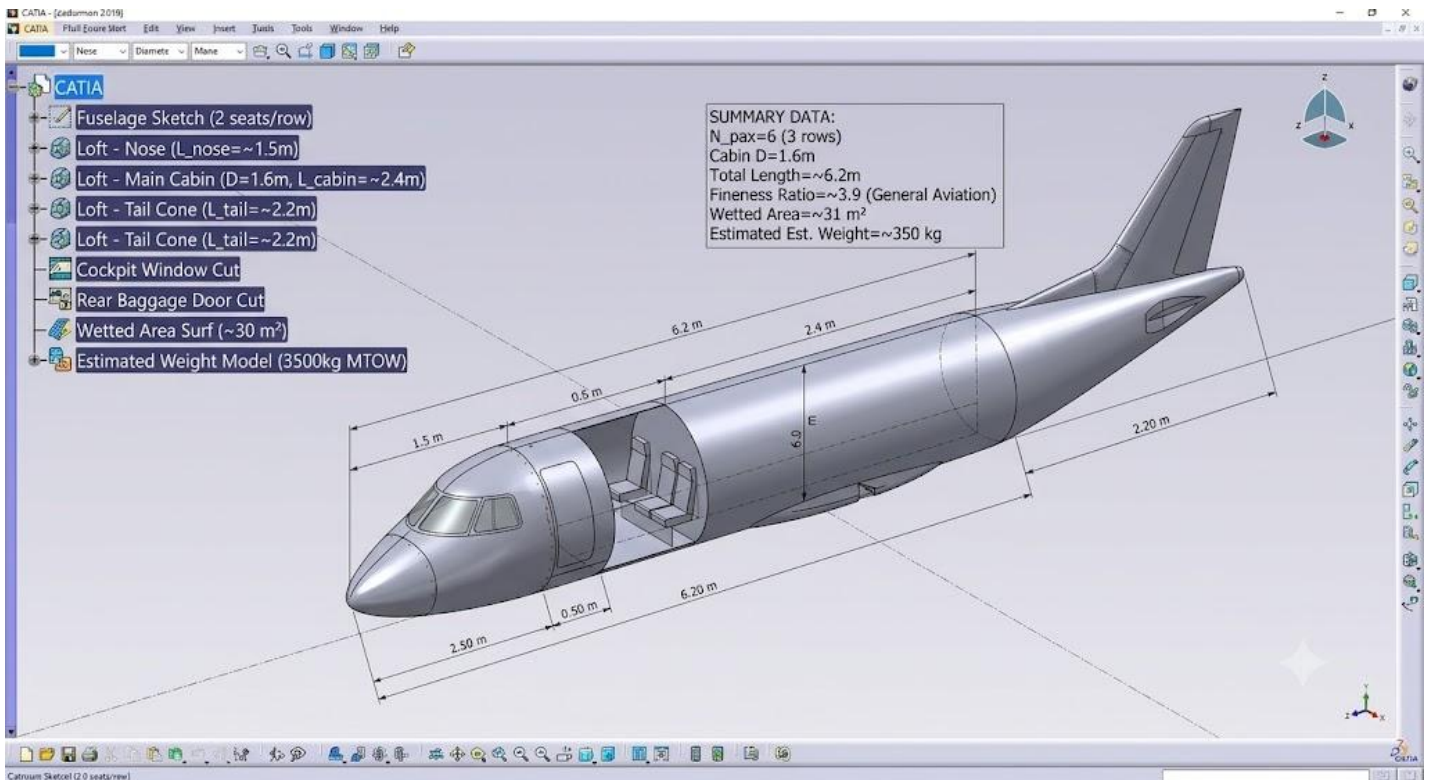


Fig. Expected Outcome with Given parameters in Catia

Or

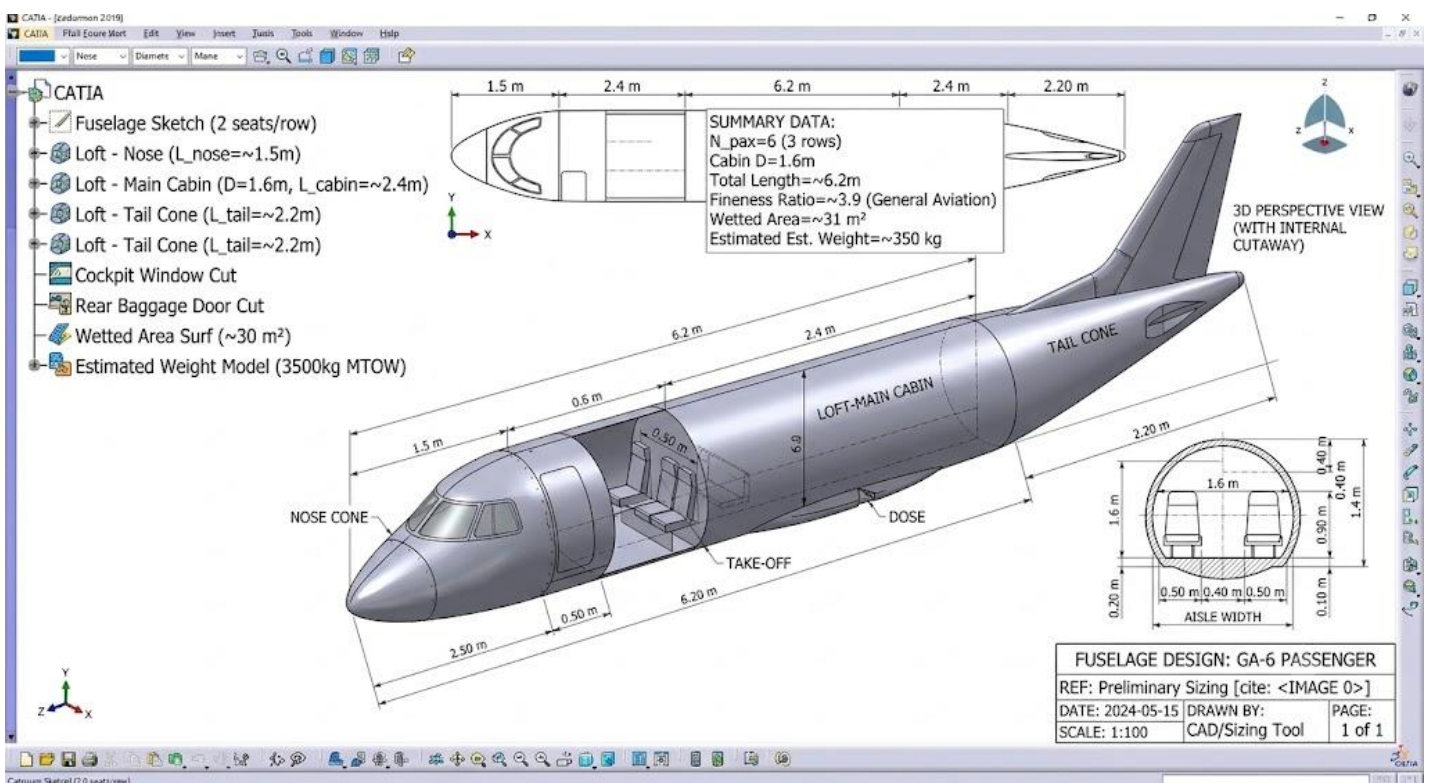


Fig. Expected Outcome with Given parameters in Catia

## Results

| Parameter                 | Value               |
|---------------------------|---------------------|
| Fuselage Length           | 8.0 m               |
| Diameter                  | 1.6 m               |
| Cabin Length              | 4.7 m               |
| Nose Length               | 1.3 m               |
| Tail Cone Length          | 2.0 m               |
| Wetted Area               | 40.2 m <sup>2</sup> |
| Estimated Fuselage Weight | 420 kg              |

Students should submit:

1. Design calculations.
2. Side, top, and front view sketches (to scale).
3. Fuselage general arrangement drawing.
4. Structural member identification.
5. Estimated fuselage weight and dimensions.
6. Brief discussion on design assumptions and limitations.

## Viva Questions

1. What are the primary functions of an aircraft fuselage?
2. Why is a circular cross-section commonly used for fuselages?
3. Define the fuselage fineness ratio. Why is it important?
4. What is the difference between monocoque and semi-monocoque construction?
5. What are the roles of frames, stringers, and longerons?
6. How is the fuselage length estimated during preliminary design?
7. What factors influence fuselage diameter?
8. Why is the wing typically attached near the aircraft's center of gravity?
9. What is fuselage wetted area, and why is it significant?
10. Why are empirical weight estimation methods used during preliminary aircraft design?

## EXPERIMENT 8

### Preliminary Design of Load Distribution on an Aircraft Fuselage

#### Aim

To estimate and illustrate the preliminary load distribution acting on an aircraft fuselage under static flight conditions and identify the major structural load paths.

#### Objectives

After completing this experiment, students will be able to:

- Understand different loads acting on a fuselage.
- Identify load paths in aircraft structures.
- Estimate distributed and concentrated loads.
- Draw free-body diagrams (FBDs) of the fuselage.
- Understand bending moment and shear force distributions.

#### Theory

The fuselage acts as the primary structural member connecting the aircraft's wings, empennage, landing gear, payload, and propulsion systems. During flight and ground operations, it is subjected to several external and internal loads.

#### Major Loads Acting on the Fuselage

- Self-weight
- Payload weight
- Crew weight
- Fuel weight (if located in fuselage)
- Wing lift reaction
- Tail loads
- Landing gear loads
- Aerodynamic drag
- Engine thrust (for fuselage-mounted engines)
- Pressurization loads (for pressurized aircraft)

**Aircraft Data and Assume** a light transport aircraft.

| Parameter       | Value         |
|-----------------|---------------|
| MTOW            | 3500 kg       |
| Crew            | 2             |
| Passengers      | 6             |
| Payload         | 800 kg        |
| Fuselage Length | 8 m           |
| Wing Position   | 4 m from nose |
| Tail Position   | 7.5 m         |

**Draw the Aircraft Side View**

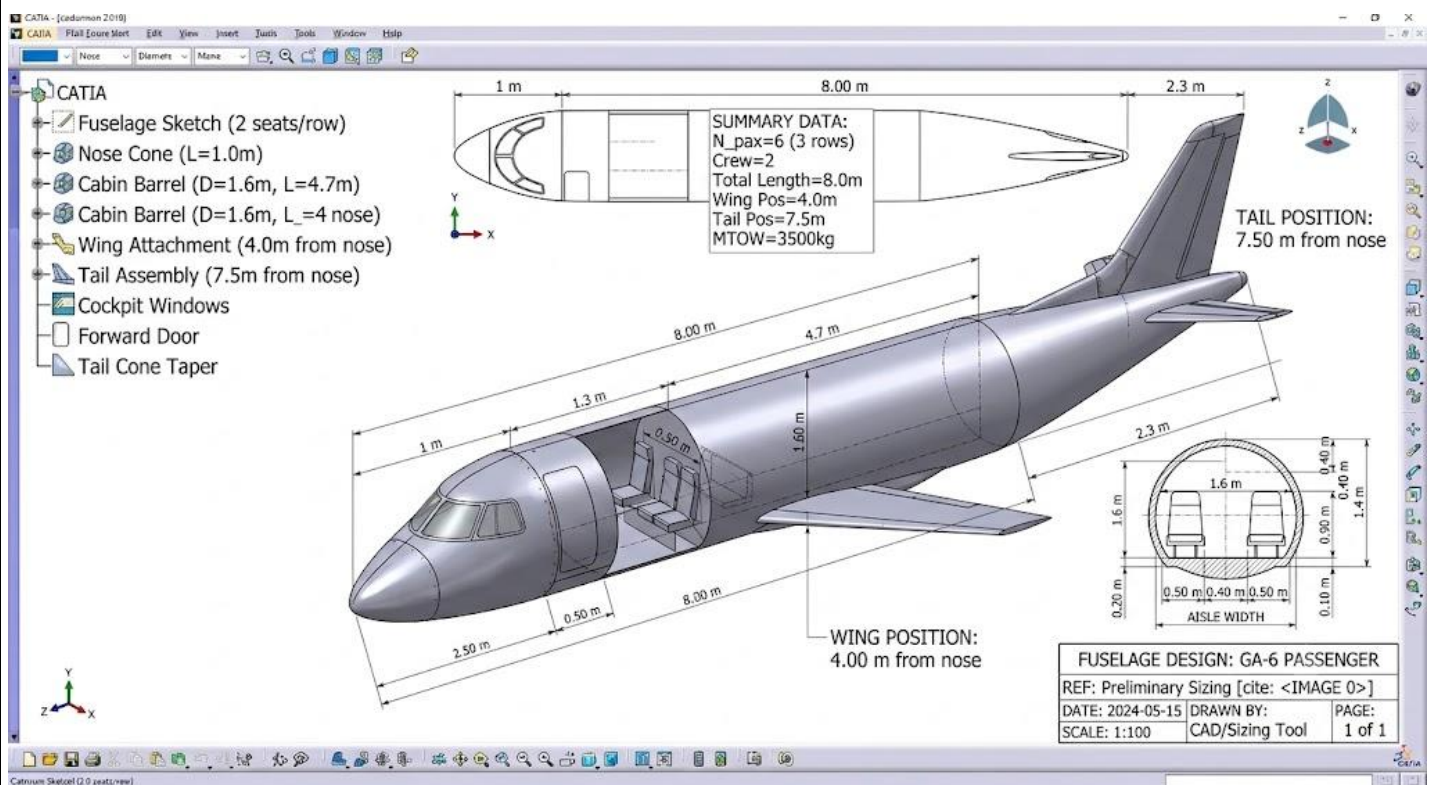


**Fig: Aircraft Side View with given data in Catia**

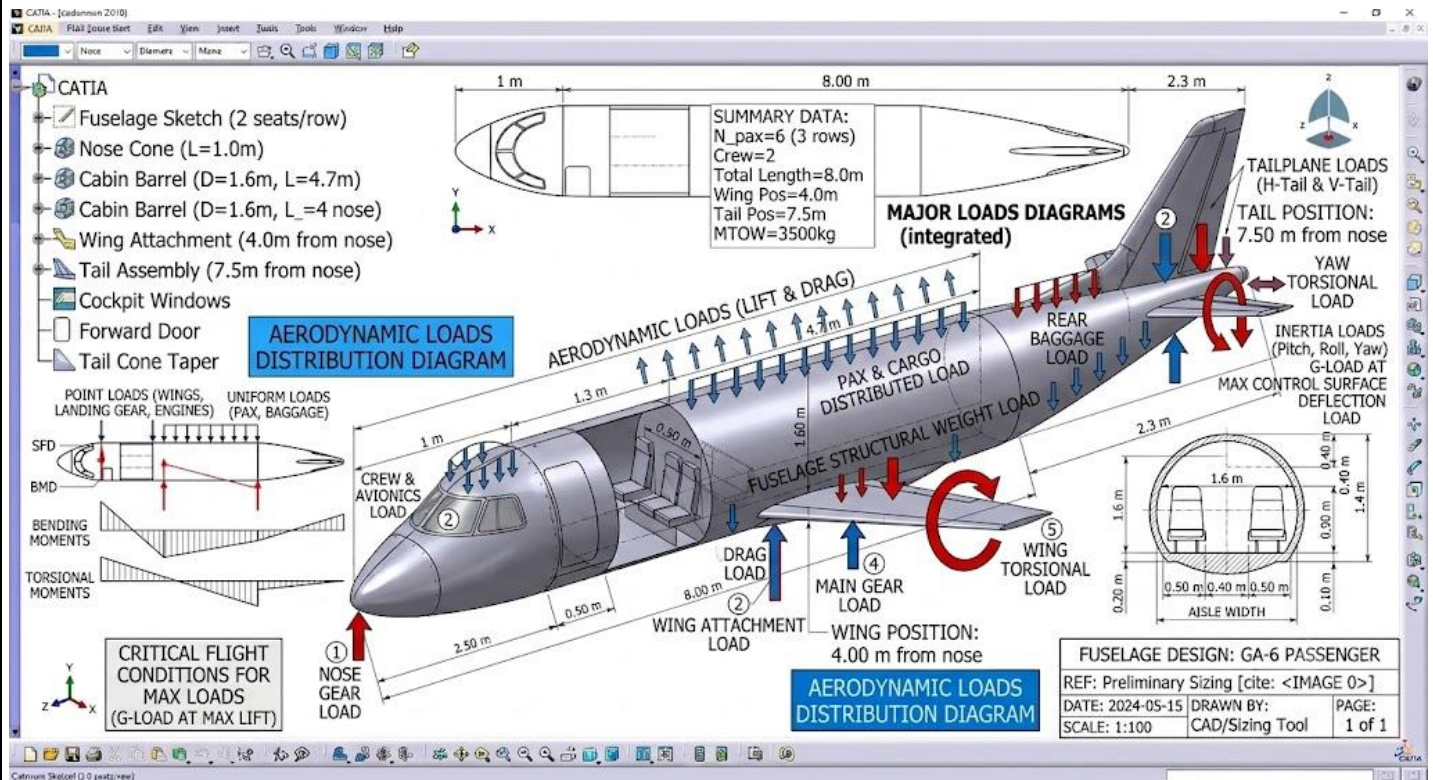


Fig: Major Loads on Fuselage with given data in Catia

## Calculate Distributed Weight

Fuselage weight = 420 kg

Length = 8 m then Distributed load  $w = \frac{420}{8}$  and  $w = \underline{\hspace{2cm}} \text{ kg/m}$

Convert to force  $w = \underline{\hspace{2cm}} \times 9.81$  so  $w = \underline{\hspace{2cm}} \text{ N/m}$

## Passenger Loads

Passenger weight = 80 kg and Passengers = 6

Total passenger weight  $6 \times 80 = \underline{\hspace{2cm}} \text{ kg}$

Force =  $480 \times 9.81 = 4709 \text{ N}$

## Crew Load

Crew 2 persons with Weight 75 kg each Total 150 kg

Force  $150 \times 9.81 = 1472 \text{ N}$

and Applied near cockpit.

## Payload Load

Payload = 800 kg and the Force = Payload x g =                      and applied in cargo compartment.

## Wing Lift Reaction

During level flight

Lift approximately equals aircraft weight. Aircraft weight =  $3500 \times 9.81 = 34335 \text{ N}$  and Wing carries almost all lift. And Reaction acts upward.

### Tail Load

Tail produces balancing force and the Typical value 5–10%. Choose Aircraft weight = 2500 N downward.

### Sample Calculations Summary

| Load                      | Value      |
|---------------------------|------------|
| Aircraft Weight           | 34.34 kN   |
| Distributed Fuselage Load | 0.515 kN/m |
| Passenger Load            | 4.71 kN    |
| Crew Load                 | 1.47 kN    |
| Payload Load              | 7.85 kN    |
| Tail Load                 | 2.50 kN    |
| Wing Lift Reaction        | 34.34 kN   |

### Results

Students should obtain:

- Side-view fuselage loading diagram.
- Free-body diagram.
- Distributed load diagram.
- Shear force diagram (qualitative).
- Bending moment diagram (qualitative).
- Structural load path diagram.

### Viva Questions

1. What are the primary loads acting on an aircraft fuselage?
2. Distinguish between concentrated and distributed loads.
3. Why is the wing attachment region critical for fuselage design?
4. Explain the purpose of longerons, frames, and stringers in load transfer.
5. How does the tail load affect fuselage bending?
6. Why is a free-body diagram essential before structural analysis?
7. What is the significance of shear force and bending moment diagrams?
8. How do passenger and payload locations influence fuselage loading?
9. Why is static equilibrium important in preliminary design?
10. How would fuselage load distribution change during landing compared with steady level flight?

## EXPERIMENT 9

### Estimate the Critical Mach number for an Airfoil.

Aim: Preliminary Estimate the Critical Mach number for an Airfoil

Software used: Mat lab / Excel.

Theory:

The Critical Mach number,  $M_{cr}$ , is the freestream Mach number at which sonic flow is first achieved on the airfoil.

#### Calculating the critical Mach number

The critical Mach number is determined by equating two pressure coefficients:  $(C_p)_{min}$  and  $C_{p,cr}$ .

$(C_p)_{min}$  is given by

$$(C_p)_{min} = \frac{(C_{p,0})_{min}}{\sqrt{1 - (M_\infty)^2}} \quad \text{-----} \quad (1)$$

where  $(C_{p,0})_{min}$  is the minimum pressure coefficient for the airfoil at a  $0^\circ$  angle of attack.

It is usually supplied for a particular airfoil, determined experimentally (in a wind tunnel).

In turn,  $C_{p,cr}$  is calculated from an equation:

$$C_{p,cr} = \frac{2}{\gamma M_{cr}^2} \left[ \left( \frac{1 + [(\gamma - 1)/2] M_{cr}^2}{1 + (\gamma - 1)/2} \right)^{\gamma/\gamma-1} - 1 \right] \quad \text{-----} \quad (2)$$

The critical Mach number is then calculated by setting.

$$(C_p)_{min} = C_{p,cr} \quad \text{-----} \quad (3)$$

Which can be rearranged as

$$(C_p)_{min} - C_{p,cr} = 0 \quad \text{-----} \quad (4)$$

Expanding the terms yields:

$$\frac{(C_{p,0})_{min}}{\sqrt{1 - (M_\infty)^2}} - \frac{2}{\gamma M_{cr}^2} \left[ \left( \frac{1 + [(\gamma - 1)/2] M_{cr}^2}{1 + (\gamma - 1)/2} \right)^{\gamma/\gamma-1} - 1 \right] = 0$$

Note that all instances of  $M_{cr}$  were replaced by  $M_\infty$ ; for the desired condition, they are the same.

**Given Data :**

Coefficient of pressure  $C_{p,o} = -0.7$        $\gamma = 1.5$

Specific heat at constant pressure = 1.14 KJ/kg-K

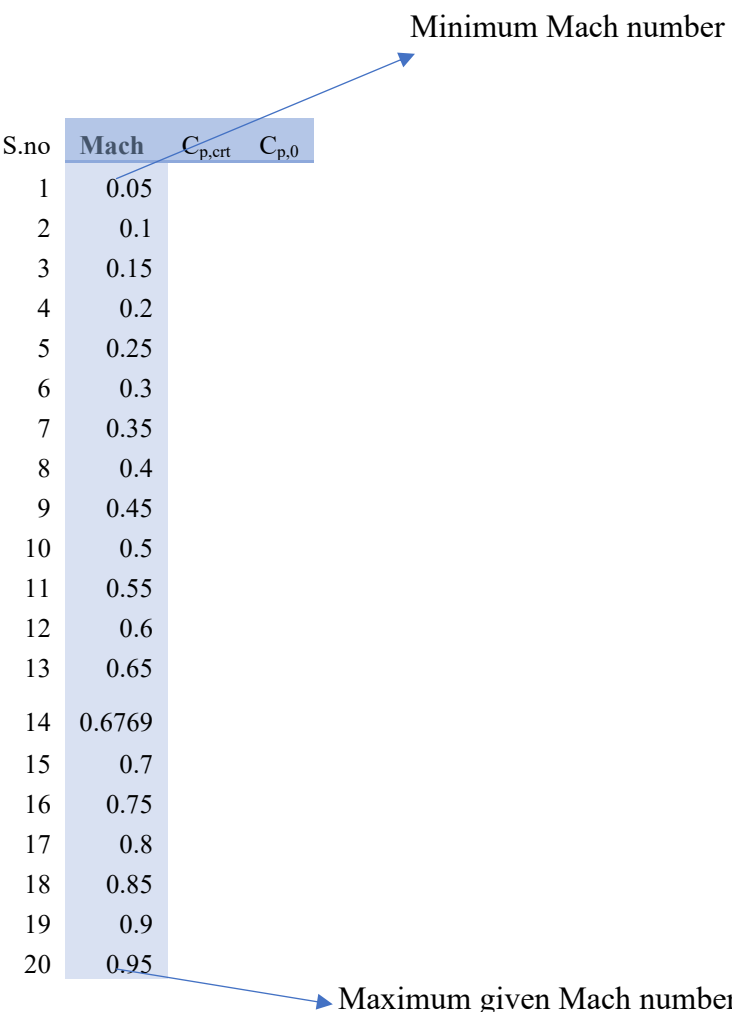
Specific heat at constant Volume = 0.95 KJ/kg-K

**Mach number range= 0.3 to 0.95**

**Procedure**

Open Excel sheet

Create a table like below.



| S.no | Mach   | $C_{p,crt}$ | $C_{p,0}$ |
|------|--------|-------------|-----------|
| 1    | 0.05   |             |           |
| 2    | 0.1    |             |           |
| 3    | 0.15   |             |           |
| 4    | 0.2    |             |           |
| 5    | 0.25   |             |           |
| 6    | 0.3    |             |           |
| 7    | 0.35   |             |           |
| 8    | 0.4    |             |           |
| 9    | 0.45   |             |           |
| 10   | 0.5    |             |           |
| 11   | 0.55   |             |           |
| 12   | 0.6    |             |           |
| 13   | 0.65   |             |           |
| 14   | 0.6769 |             |           |
| 15   | 0.7    |             |           |
| 16   | 0.75   |             |           |
| 17   | 0.8    |             |           |
| 18   | 0.85   |             |           |
| 19   | 0.9    |             |           |
| 20   | 0.95   |             |           |

- From the range of given Mach number (from the given data) generate the Mach numbers like shown in the table.
- For each Mach numbers in the table, Calculate the values of  $C_{p,crt}$  and  $C_{p,0}$  from the given formulas in Eq (1) and Eq (2).
- Compare the obtained values for  $C_{p,crt}$  and  $C_{p,0}$  for each Mach numbers in the table.
- Where ever the values of both  $C_{p,crt}$  and  $C_{p,0}$  are equal and near by values then the relevant Mach number is consider as a the Critical Mach number,  $C_{p,cr}$  of the Airfoil at given flow conditions.
- Plot the graphical representation of  $C_{p,crt}$  Vs  $C_{p,0}$  so that will make easy to identify the Critical Mach number  $C_{p,cr}$ .

**Results :** Finally, the Critical Mach number,  $C_{p,cr}$  of the Airfoil at given flow conditions has been obtained

## EXPERIMENT 10

### Static Performance: Thrust required curve.

**Aim: Static Performance: Thrust required curve for an Aircraft**

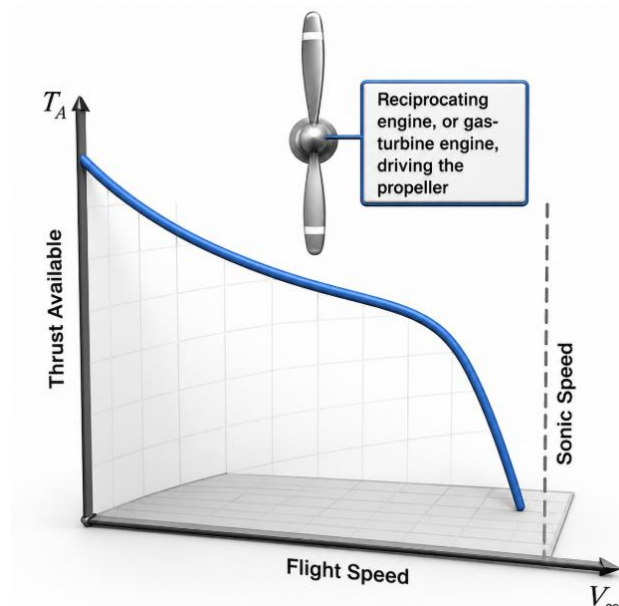
**Software used: Mat lab / Excel.**

#### Theory:

Thrust available, denoted by  $T_A$ , is the thrust provided by the power plant of the airplane. Unlike the thrust required  $T_R$  which has almost everything to do with the airframe (including the weight) of the airplane and virtually nothing to do with the power plant, the thrust available  $T_A$  has almost everything to do with the power plant and virtually nothing to do with the airframe. This statement is not completely true; there is always some aerodynamic interaction between the airframe and the power plant. For conventional, low-speed airplanes, this interaction is usually small. However, for modern transonic and supersonic airplanes, it becomes more of a consideration.

#### Propeller-Driven Aircraft

An aerodynamic force is generated on a propeller that is translating and rotating through the air. The component of this force in the forward direction is the thrust of the propeller. For a propeller/reciprocating engine combination, this propeller thrust is the thrust available. For a turboprop engine, the propeller thrust is augmented by the jet exhaust, albeit by only a small amount (typically almost 5%). The combined propeller thrust and jet thrust is the thrust available  $T_A$  for the turboprop.



The qualitative variation of  $T_A$  with  $V$ , for propeller-driven aircraft is sketched in Figure. The thrust is highest at zero velocity (called the static thrust) and decreases with an increase in  $V$ . The thrust rapidly decreases as  $V$ , approaches sonic speed; this is because the propeller tips encounter compressibility problems

at high speeds, including the formation of shock waves. It is for this reason (at least to the present) that propeller-driven aircraft have been limited to low to moderate subsonic speeds.

The propeller is attached to a rotating shaft which delivers power from a reciprocating piston engine or a gas turbine (as in the case of the turboprop). For this reason, power is the more appropriate characteristic of such power plants rather than thrust. The values of TA for a propeller-driven airplane can be readily obtained from the power ratings as follows. The power available from a propeller/reciprocating engine combination is

$$P_A = \eta_{pr} P$$

where  $\eta_{pr}$  is the propeller efficiency and P is the shaft power from the piston engine. Since power is given by

$$P_A = T_A V_\infty \longrightarrow$$

$$T_A = \frac{\eta_{pr} P}{V_\infty}$$

force times velocity

$$P_A = \eta_{pr} P_{es} \longrightarrow T_A = \frac{\eta_{pr} P_{es}}{V_\infty}$$

Similarly, for a turboprop,

both P and  $P_{es}$ , are relatively constant with  $V_{inf}$ . By assuming a variable-pitch propeller such that the variation of the efficiency, with V, is minimized, TA decreases as V, increases.

## Jet-Propelled Aircraft

For subsonic speeds Ta is about constant with the velocity

for supersonic speeds For subsonic speeds

$$\frac{T_A}{(T_A)_{Mach\ 1}} = 1 + 1.18(M_\infty - 1)$$

The effect of the altitude is

$$\frac{T_A}{(T_A)_0} = \frac{\rho}{\rho_0}$$

For the high-bypass-ratio turbofans commonly used for civil transports, thrust decreases with increasing velocity

$$\frac{T_A}{(T_A)_{V=0}} = AM_\infty^{-n}$$

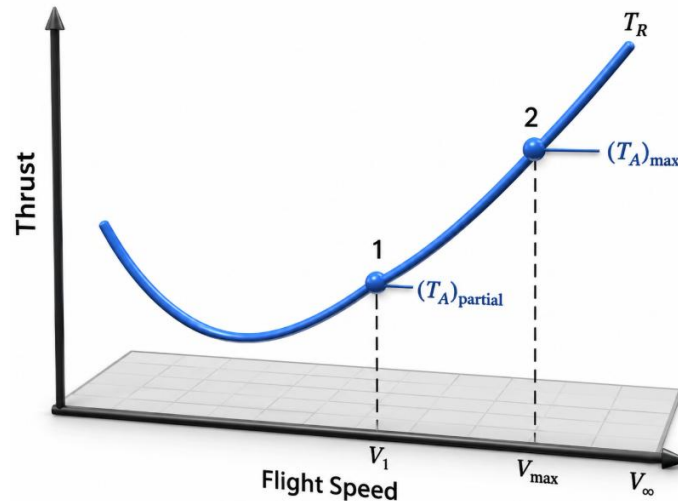
where  $(T_A)_{V=0}$  is the static thrust available (thrust at zero velocity) at standard sea level, and A and n are functions of altitude, obtained by correlating the data for a given engine. On the other hand, for a low-bypass-ratio turbofan, the thrust variation with velocity is much closer to that of a turbojet, essentially constant at subsonic speeds and increasing with velocity at supersonic speeds.

The altitude variation of thrust for a high-bypass-ratio civil turbofan is correlated where  $(T_A)_0$  is the thrust available at sea level and  $\rho_0$  is standard sea-level density

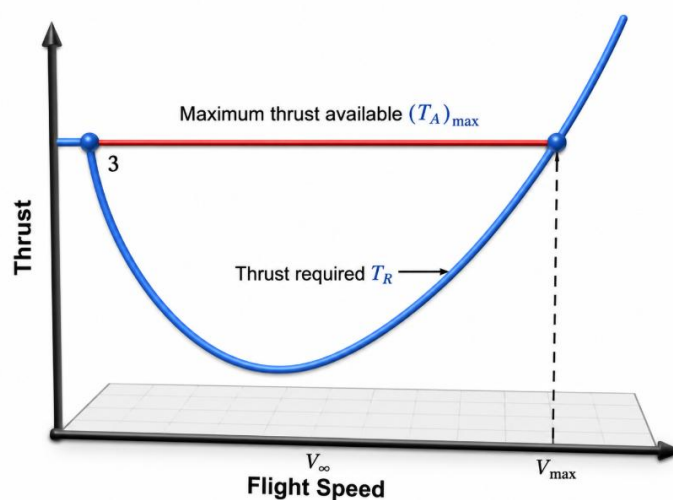
$$\frac{T_A}{(T_A)_0} = \left[ \frac{\rho}{\rho_0} \right]^m$$

## Maximum Velocity

Consider a given airplane flying at a given altitude, with a TR curve as sketched in Figure. For steady, level flight at a given velocity  $V_1$ , the value of TA is adjusted such that  $TA = TR$  at that velocity. This is denoted by point 1. The pilot of the airplane can adjust TA by adjusting the engine throttle in the cockpit. For point 1 the engine is operating at partial throttle, and the resulting value of TA is denoted by  $(TA)_{\text{partial}}$ .



The thrust available curve is the variation of TA with velocity at a given throttle setting and altitude. For the throttle full forward,  $(TA)_{\text{max}}$  is obtained. The maximum thrust available curve is the variation of  $(TA)$  with velocity at a given altitude. For turbojet and low-bypass-ratio turbofans, we have seen that at subsonic speeds, the thrust is essentially constant with velocity. Hence, the thrust available curve is a horizontal line. In steady, level flight, the maximum velocity of the airplane is determined by the high speed intersection of the thrust required and thrust available curves.



Note that there is a low-speed intersection of the  $(TA)$  and  $TR$  curves, point 3. At first glance, this would appear to define the minimum velocity of the airplane in steady, level flight. However, what is more usual

is that the minimum velocity of the airplane is determined by its stalling speed, which depends strongly on  $CL_{max}$ , and wing loading.

The generic solution of the velocity as a function of  $T_R$  is

$$V_{\infty} = \left[ \frac{(T_R/W)(W/S) \pm (W/S)\sqrt{(T_R/W)^2 - 4C_{D,0}K}}{\rho_{\infty}C_{D,0}} \right]^{1/2}$$

replacing  $V_{inf}$ , with  $V_{max}$ , and  $T_R$  with  $(T_A)_{max}$ , and taking the plus sign in the quadratic expression because we are interested in the highest velocity, we have the direct calculation of the maximum velocity:

$$V_{max} = \left\{ \frac{[(T_A)_{max}/W](W/S) + (W/S)\sqrt{[(T_A)_{max}/W]^2 - 4C_{D,0}K}}{\rho_{\infty}C_{D,0}} \right\}^{1/2}$$

$V_{max}$ , depends on the maximum thrust-to-weight ratio  $(T_A)_{max}/W$ ,

(2) wing loading  $W/S$ ,

(3) the drag polar via  $C_{D0}$  and  $K$ , and

(4) altitude via  $\rho$ .

1.  $V_{max}$ , increases as  $(T_A)_{max}/W$  increases.
2.  $V_{max}$ , increases as  $W/S$  increases.
3.  $V_{max}$ , decreases as  $C_{D0}$  and/or  $K$  increases.

#### Given Data:

Height = 10 Km      Weight = 7500 = (7500\*9.81) = 73575      Density of Air  $\rho$  at 10 KM = 0.4135 kg/m<sup>3</sup>  
 Span Area = 90 m<sup>2</sup>      Span efficiency ( $e$ ) = 0.75      Aspect ratio = 5       $C_{d0}$  = 0.015

#### Procedure

- Open a new Excel sheet.
- Calculate the value of  $K$ ,  $A$  and  $B$  from the following Eq (1), Eq (2), and Eq (3)

$$K = \frac{1}{\pi * \text{Span efficiency} * \text{Aspect ratio}} \quad \text{-----}(1)$$

$$A = \frac{1}{2}(\rho * \text{Span area} * C_{d0}) \quad \text{-----}(2)$$

$$B = \frac{(2 * K * W^2)}{\rho * \text{Span area}} \quad \text{-----}(3)$$

- Now Estimate the  $D_p$  and  $D_i$  from the following Eq (4) and Eq (5)

$$D_p = A * V^2 \quad \text{-----}(4)$$

$$D_i = \frac{B}{V^2} \quad \text{-----}(5)$$

- Create a table in the excel sheet like below.

| V(KMPH) | V(M/S) = 5/18* V(KMPH) | Dp | Di |
|---------|------------------------|----|----|
| 200     |                        |    |    |
|         |                        |    |    |
|         |                        |    |    |
|         |                        |    |    |
|         |                        |    |    |
|         |                        |    |    |
|         |                        |    |    |
|         |                        |    |    |
|         |                        |    |    |
|         |                        |    |    |
|         |                        |    |    |
|         |                        |    |    |

The initial value of V(KMPH) is assumed to be 200 the for the given problem.

Calculate the value of V(M/S), Dp and Di in the above table by using Equation (4) and (5)

The 2<sup>nd</sup> value of V(KMPH) is assumed to be a gradual increment of 25 to the previous Velocity i.e 225

Calculate the value of V(M/S), Dp and Di in the above table by using Equation (4) and (5) in respective cells

The 3<sup>rd</sup> value of V(KMPH) is assumed to be a gradual increment of 25 to the previous Velocity i.e 250

Calculate the value of V(M/S), Dp and Di in the above table by using Equation (4) and (5) in respective cells

.

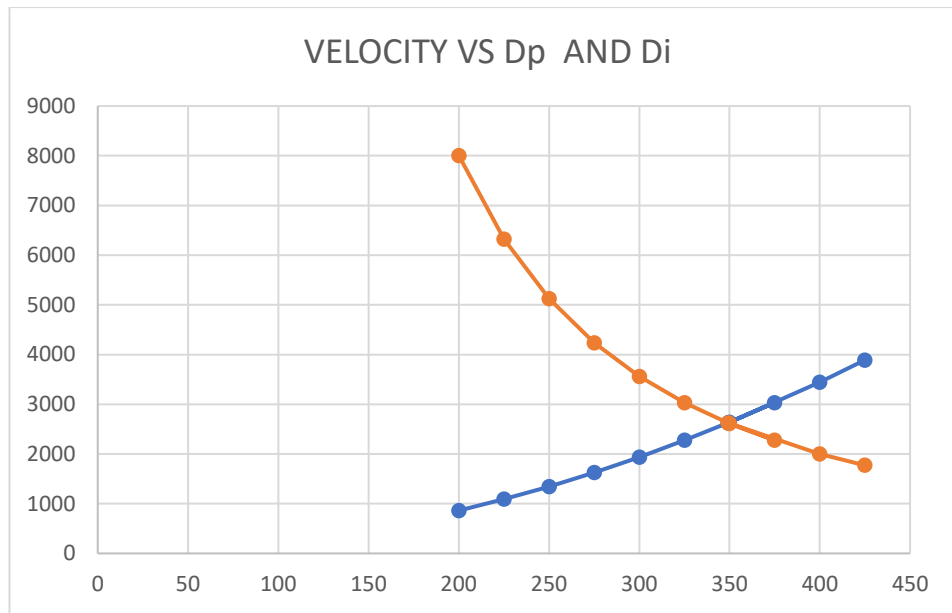
.

Similarly continue the process until the values of Dp and Di until both value will be mostly similar or Equals

Then stop the process and that is the final obtained Velocity value with respective to Dp and Di will shows the maximum thrust /Velocity required for the aircraft to generate the lift.

Draw a graph in excel for the obtained values of Dp and Di

Finally, we find the Thrust required curve for the given aircraft profile.



## EXPERIMENT 11

### Static Performance: Power Required curve.

**Aim: Static Performance: Power Required curve**

**Software used: Mat lab / Excel.**

Theory:

(Same as Experiment 4)

The propulsion system requirements to maintain steady level flight since

Thrust Requirements  $T_{req}$  has been obtained from previous experiment and

$$T_{req} = D \quad \text{and} \quad P_{req} = T_{req} V = DV \quad \text{----- (a)}$$

Where  $P_{req}$  is the Power requirements to maintain steady level flight.

$$P_{req} = \frac{1}{2} \rho V^3 S C_{D_o} + \frac{W^2}{\frac{1}{2} \rho V S} \left( \frac{1}{\pi e A R} \right) \quad \text{----- (b)}$$

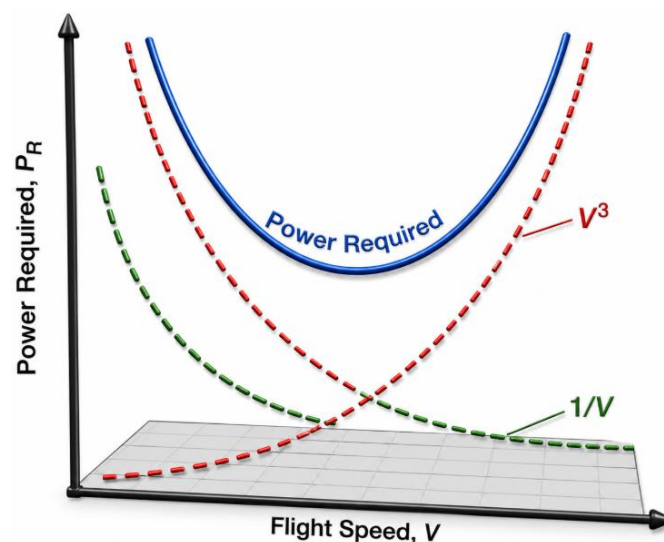


Figure Typical power required curve for an aircraft.

#### Given Data:

Height = 15 Km

Air Density  $\rho$  at 15 KM = 0.1948 kg/m<sup>3</sup>

Span Area = 90 m<sup>2</sup>

Span efficiency ( $e$ ) = 0.75

Aspect ratio = 5

$C_{D_o}$  = 0.015

Velocity = 30 m/s to 450 m/s (30,60,90,120,150.....up to 450 m/s)

**Procedure**

- Open a new Excel sheet.
- Calculate the value of K, A and B from the following Eq (1), Eq (2), and Eq (3)

$$K = \frac{1}{\pi * \text{Span efficiency} * \text{Aspect ratio}} \text{-----(1)}$$

$$A = \frac{1}{2} (\rho * \text{Span area} * C_{do}) \text{-----(2)}$$

$$B = \frac{(2 * K * W^2)}{\rho * \text{Span area}} \text{-----(3)}$$

- Now Estimate the  $D_p$  and  $D_i$  from the following Eq (4) and Eq (5)

$$D_p = A * V^2 \text{-----(4)}$$

$$D_i = \frac{B}{V^2} \text{-----(5)}$$

- Create a table in the excel sheet like below.

| V(M\S) | Dp | Di |
|--------|----|----|
| 30     |    |    |

The initial value of V(KMPH) is assumed to be 30 for the given problem.

Calculate the value of V(M\S),  $D_p$  and  $D_i$  in the above table by using Equation (4) and (5)

The 2<sup>nd</sup> value of V(KMPH) is assumed to be a gradual increment of 15 to the previous Velocity i.e 45

Calculate the value of V(M\S),  $D_p$  and  $D_i$  in the above table by using Equation (4) and (5) in respective cells

The 3<sup>rd</sup> value of V(KMPH) is assumed to be a gradual increment of 55 to the previous Velocity i.e 60

Calculate the value of V(M\S),  $D_p$  and  $D_i$  in the above table by using Equation (4) and (5) in respective cells

.

.

.

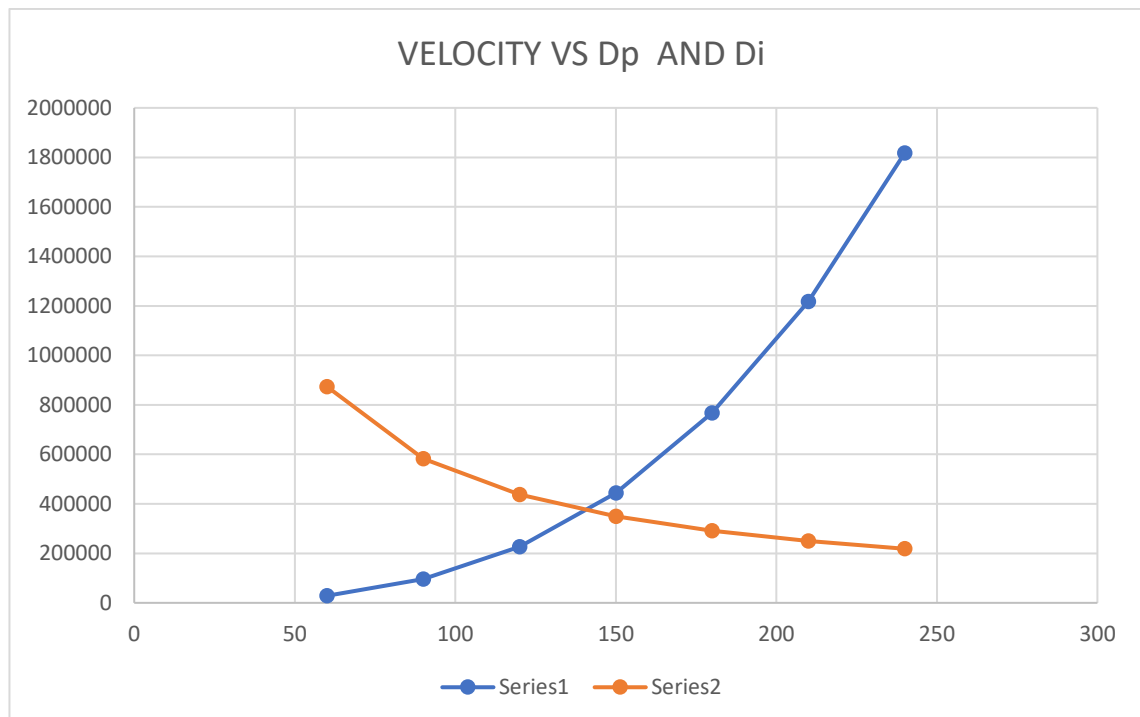
Similarly continue the process until the values of  $D_p$  and  $D_i$  until both value will be mostly similar or Equals. Then stop the process and that is the final obtained Velocity value with respective to  $D_p$  and  $D_i$  will shows the maximum thrust /Velocity required for the aircraft to generate the lift.

Draw a graph in excel for the obtained values of  $D_p$  and  $D_i$  which shows the value of  $T_{req}$

Now the estimate the required power to be maintain steady level flight from Eq (a) I,e

$$P_{req} = T_{req} V = D V = \underline{\hspace{2cm}}$$

Finally, we find the required power to be maintain steady level flight.



## EXPERIMENT 12

### Drawing all the 3D views of a new Aircraft

**Aim:** Write the request for proposal for the aircraft, Develop the conceptual sketch of the aircraft for given requirements.

**Software's used: Catia / Engineering Drawing Theory:**

Conceptual design begins with a specific set of design requirements established from customer or a company-generated guess what future customers may need. Design requirements include:

- a) Aircraft range
- b) Payload
- c) Take-off distance
- d) Landing distance
- e) Manoeuvrability and speed requirements

Design begins with innovative idea rather than as a response to a given requirement. Before design a decision is made to what technologies to incorporate, it must use only currently available technologies as well as existing engines and avionics. If designed to build in more distant future, then an estimate technological state of the art must be made to determine which emerging technologies will be ready for use at that time.

Design begins drawing with a conceptual sketch like shown in

**Procedure:**

1. Write the request for proposal for the given aircraft. It should be in the form of parameters and requirements for the aircraft.
2. Draw the conceptual sketch of the aircraft as explained in theory.
3. Draw the mission profile for the aircraft.

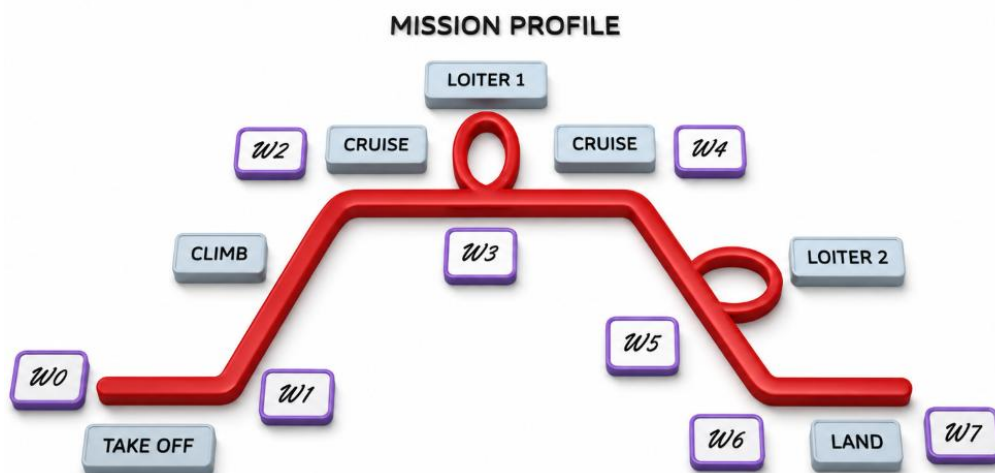


FIG 4: Mission Profile

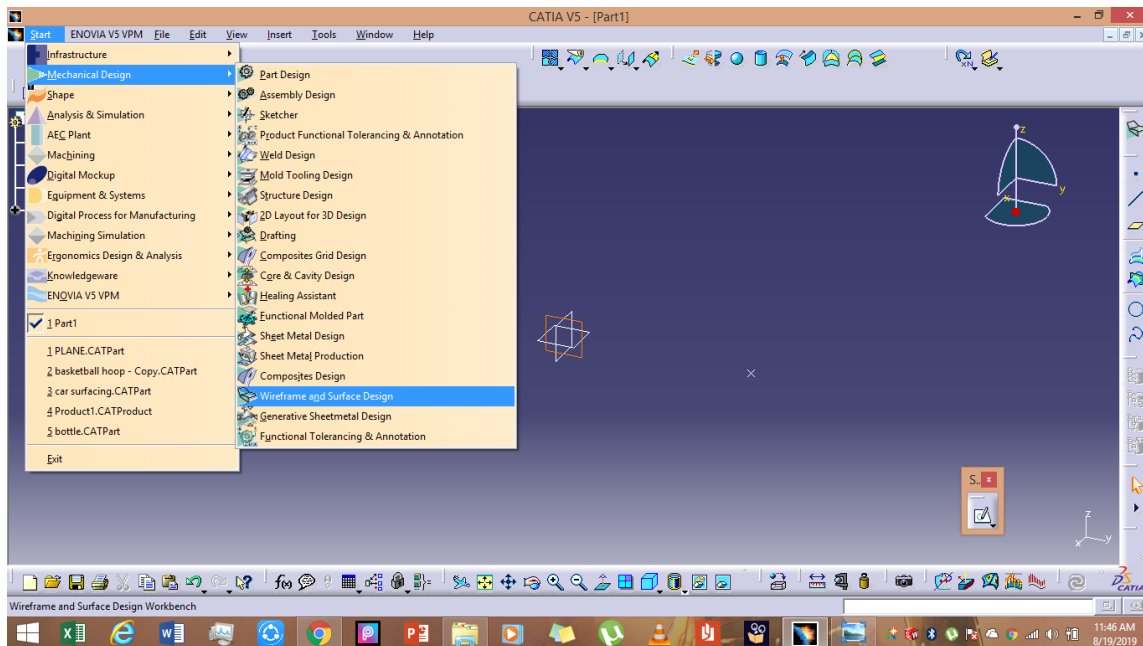
The steps to be followed:

## Step 1: Open CATIA

Open CATIA V5

Go to START--Mechanical Design--Wireframe and Surface Design.

You can open "MULTI SECTIONS SOLID" if you want to create the model using surfacing, if you want to create a solid body for the model, you may go to part design.

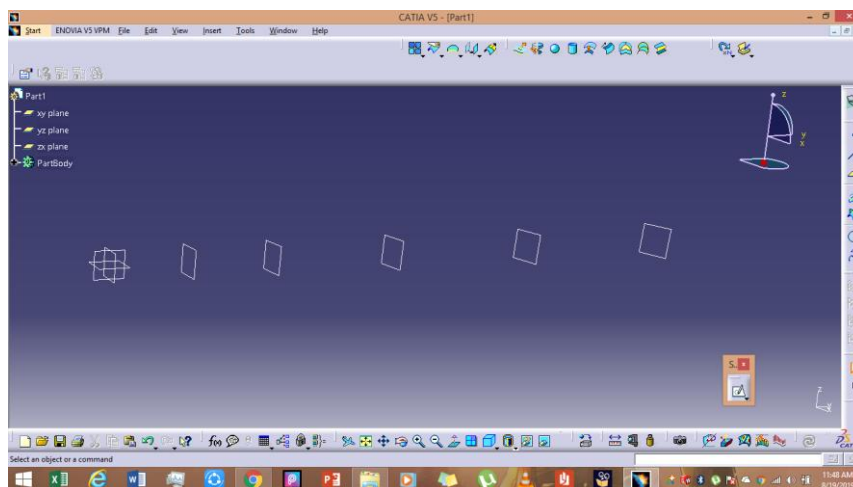


## Step 2: CREATE PLANES

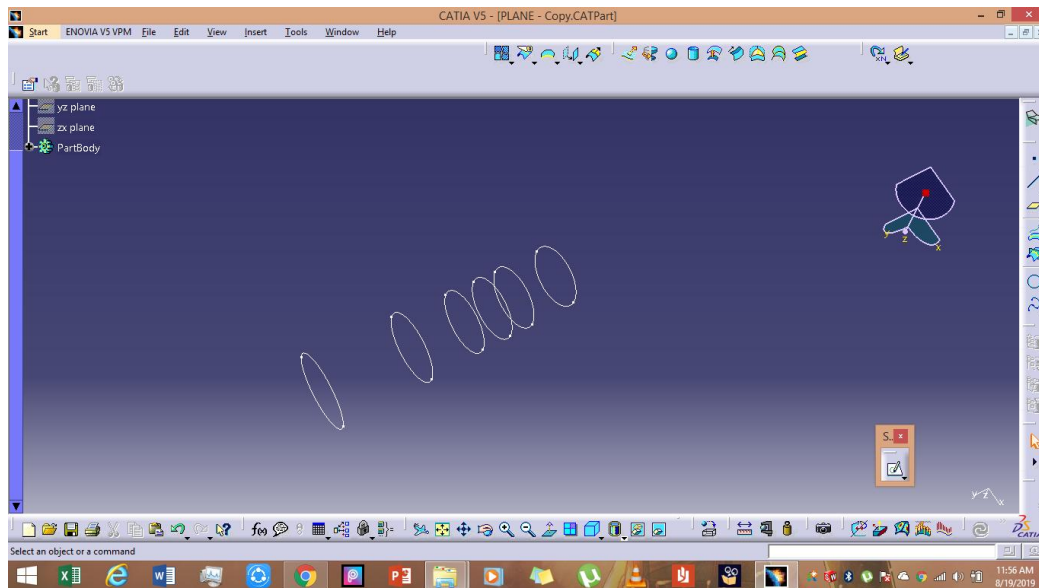
Now you need to create multiple planes in the direction you want the body of the aeroplane to be in , take ZX plane for reference.

You can also take multiple points , this will also work.

Take the planes at distance you want the body to change shapes at.

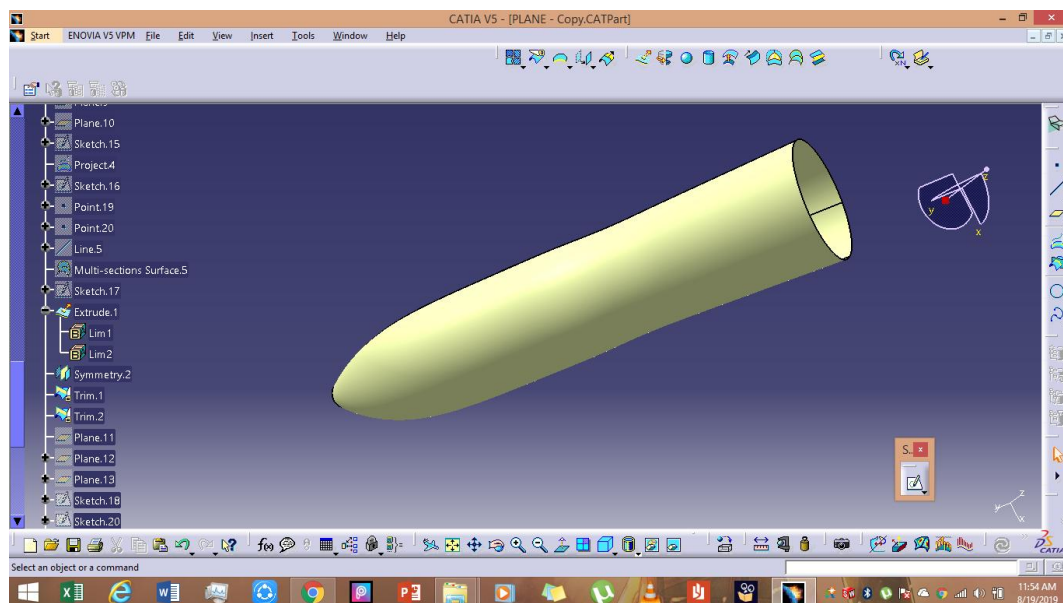


Create the shapes according to the flow of structure you want.



### Step 3: MULTI SECTIONS SURFACE

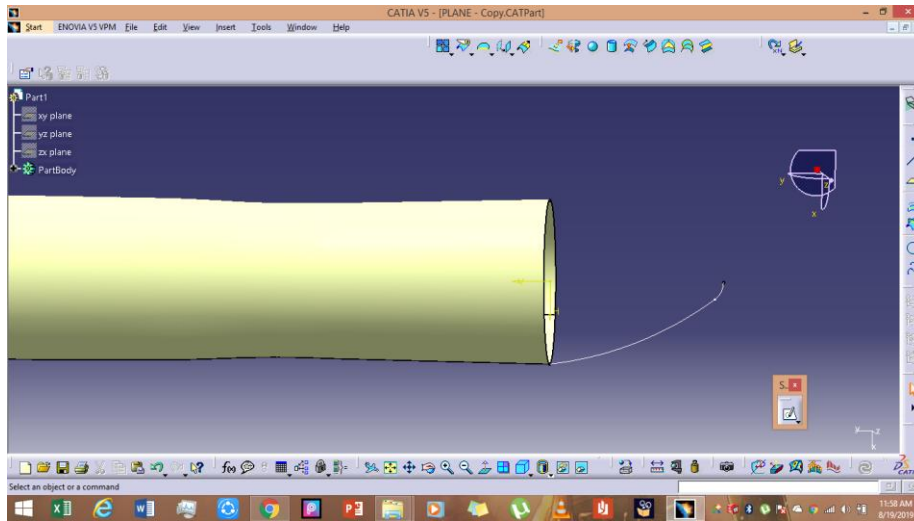
Select "MULTI SECTIONS SURFACE" and create surface of the various shapes you drew in the previous step.



The basic body of the Airplane is ready.

## Step 4: CREATING FRONT SECTION OF THE PLANE

Select the XY plane and draw a profile for the front of the areoplane.



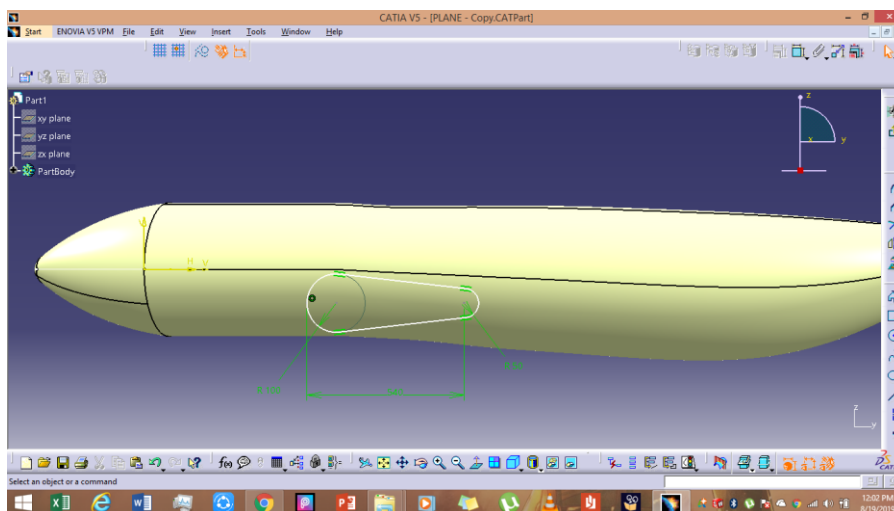
Now as you have drawn a profile accordingly, Revolve it around the Axis.

The front section of the plane is ready.

## Step 5: CREATING WINGS

Take a plane at some distance from YZ plane (At least at the distance of the body, It can be more than it) and draw the profile you want the wings of your Airplane to take.

Project the drawn profile on aerolane's body.



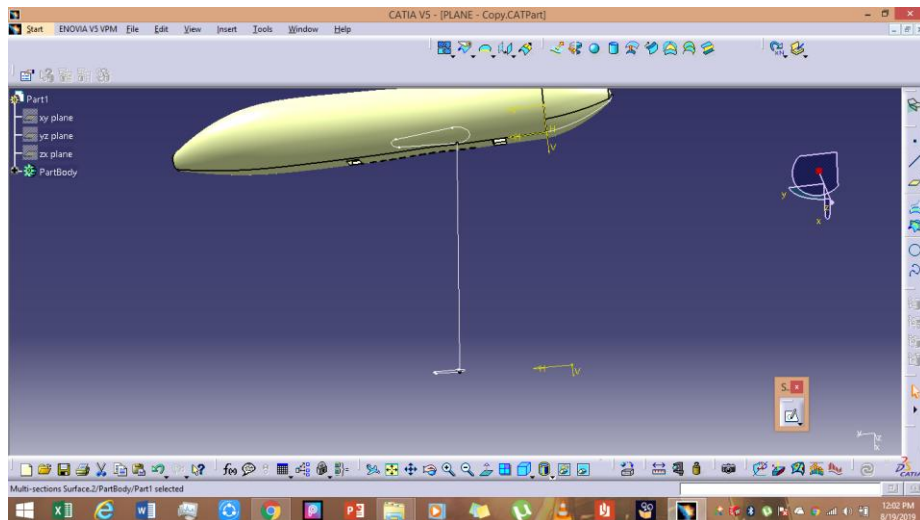
Take another plane at some distance to the previous plane you took in this step.

The plane must be at the distance you want your wing to be.

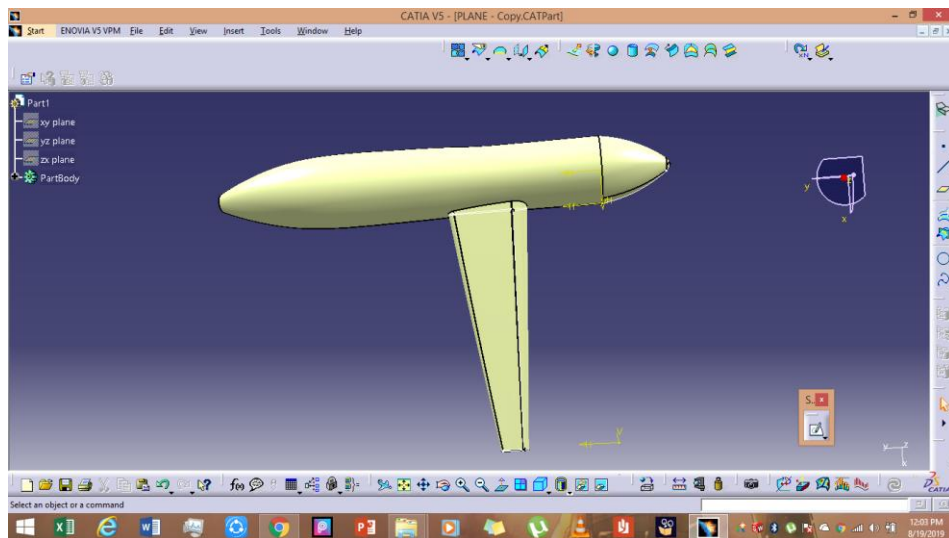
Draw a similar profile but smaller.

Take points on the new created sketch and on the projected image of previous sketch.

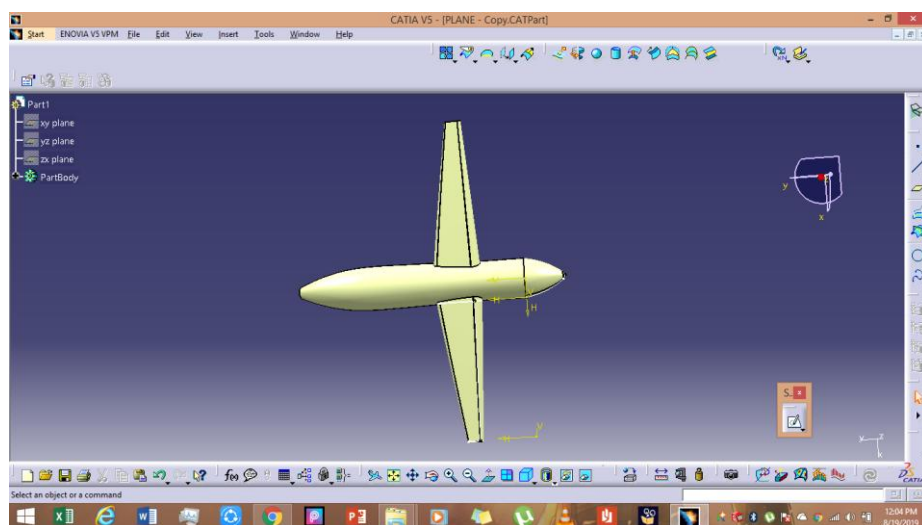
Draw a line between these two points



Select "MULTI SECTIONS SURFACE" command And make a surface following the line as the spline.



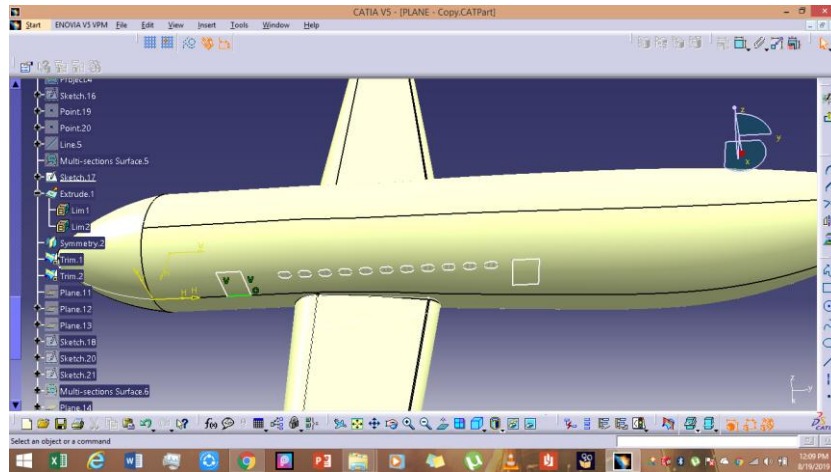
Use "SYMMETRY" for wings on other side



Wings are ready.

## Step 6: CREATING WINDOWS AND DOORS

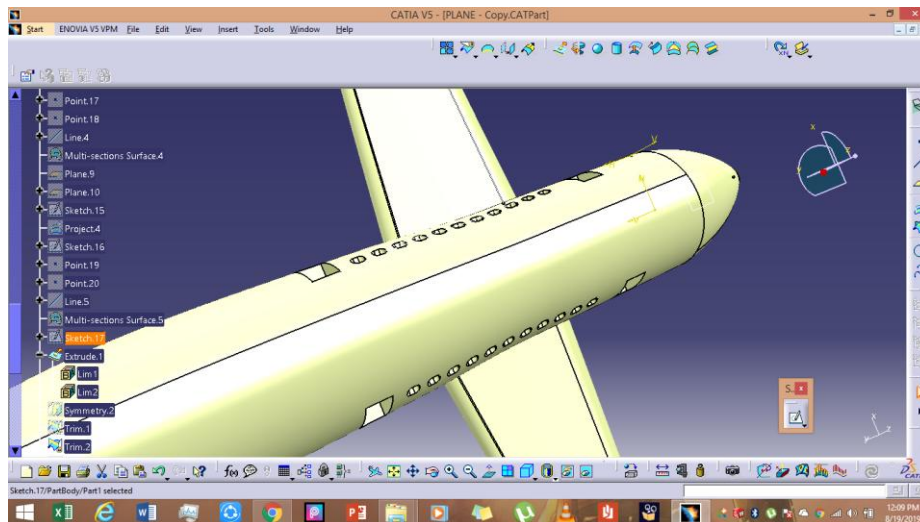
Make a plane and draw Doors and windows at the places you want them to be



Extrude The Surface to wards.

Make a Symmetry On the other side or you may also draw if you want some change in the layout of both the sides.

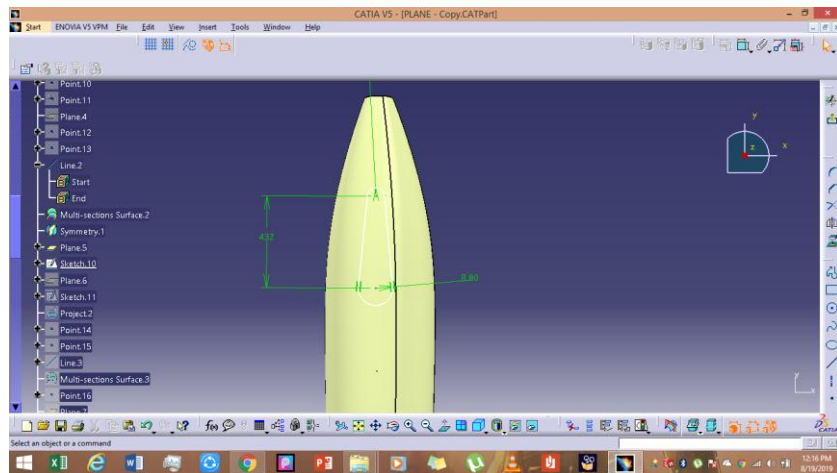
Now Trim The Extruded Surface.



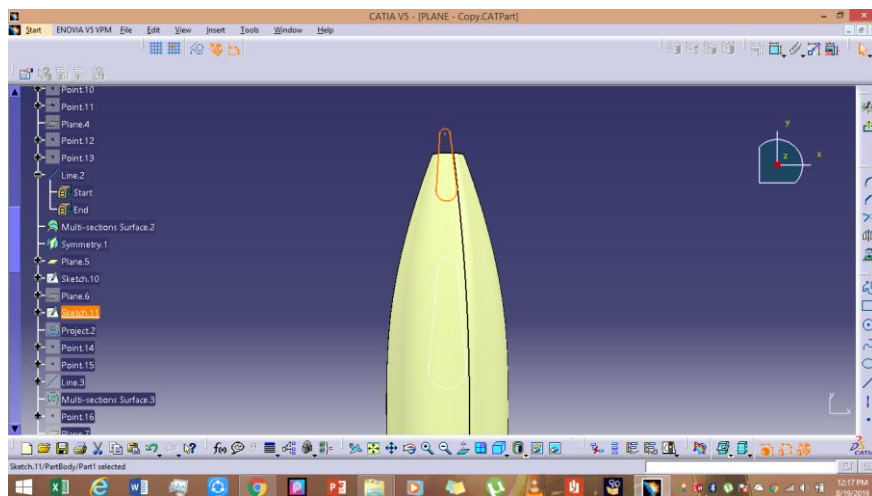
## Step 7: CREATING VERTICAL STABILIZER

Create a plane with respect to XY plane.

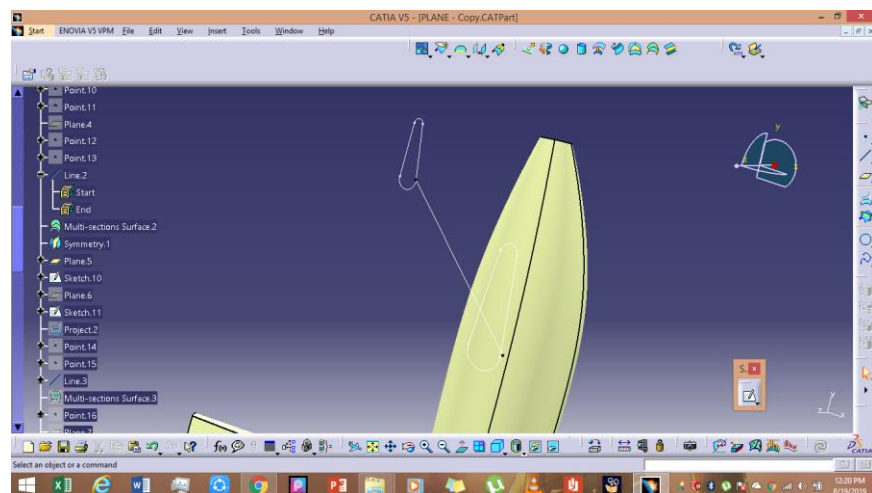
Sketch on the surface at the location you want your stabilizer to be at and project it on the body of the airplane.



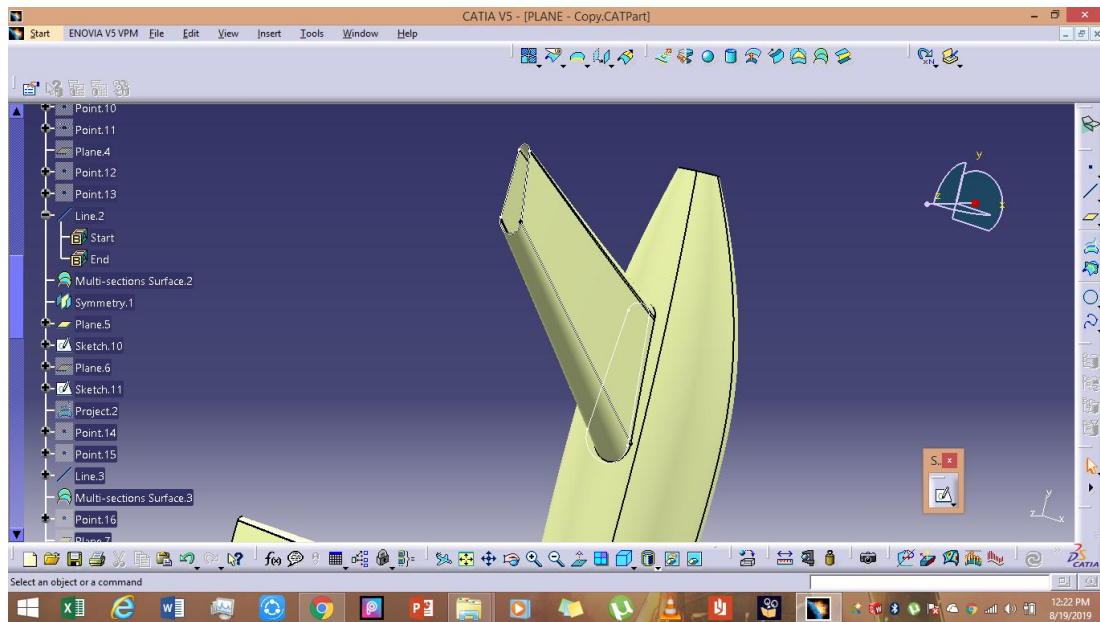
Create another plane and draw similar structure but smaller.



Create points on both the sketches and create a line.



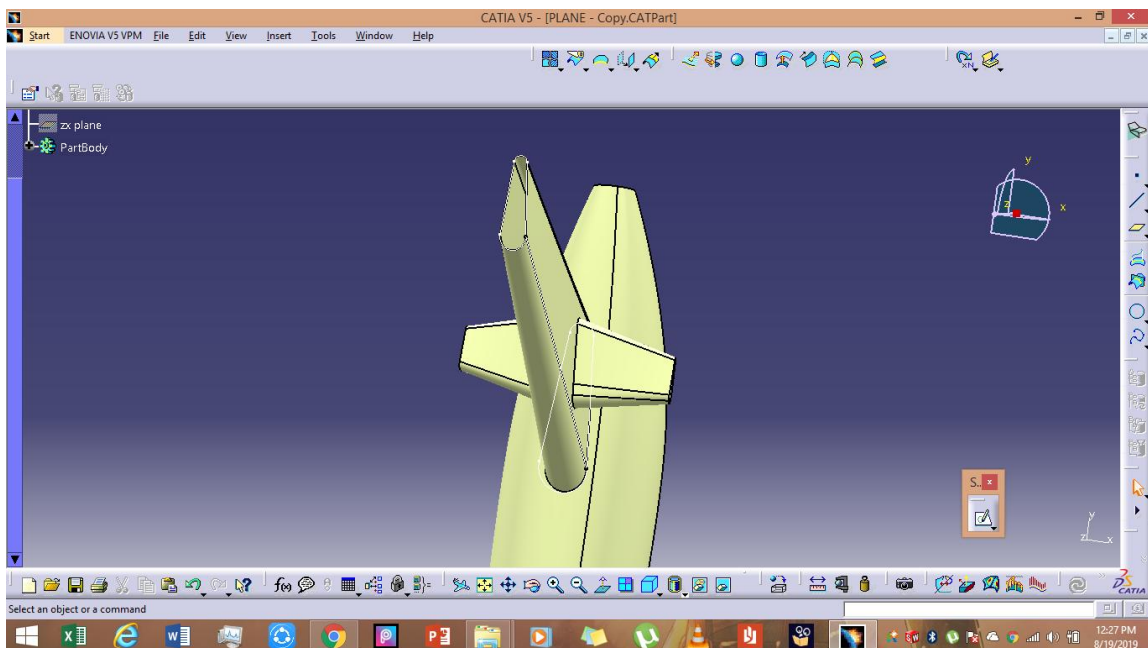
Using "MULTI SECTIONS SURFACE" create a surface with both the sketches and line as reference spline



## Step 8: CREATING RUDDER

Using the same steps as while creating wings create Rudders on the Stabilizer.

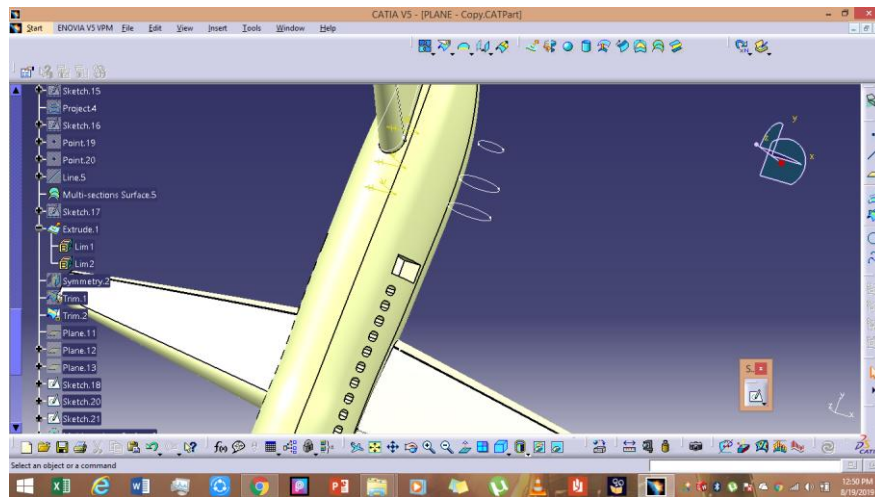
You can also create Rudders at other places on the body of Aeroplane.



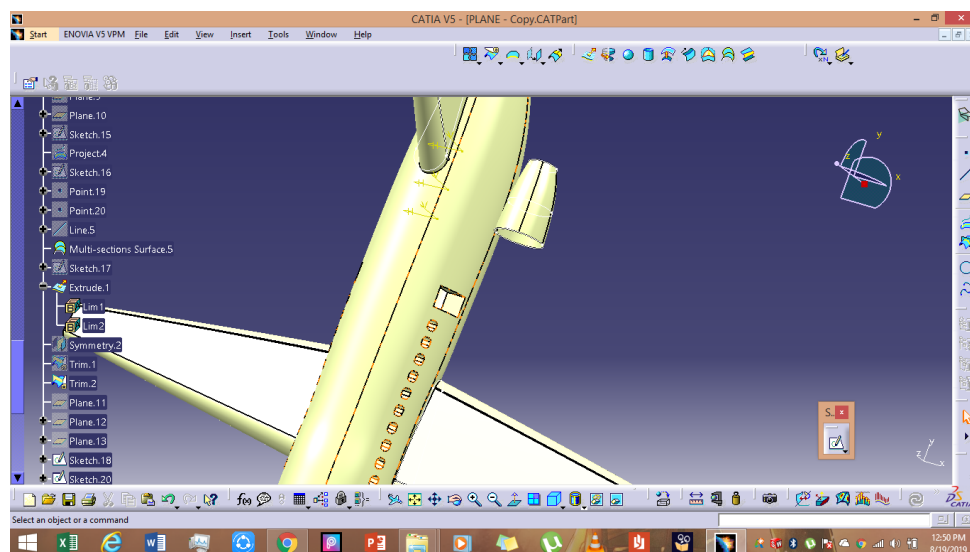
## Step 9: CREATING TURBINE

Take multiple planes according to the places you want the turbine of the Aeroplane to be at.

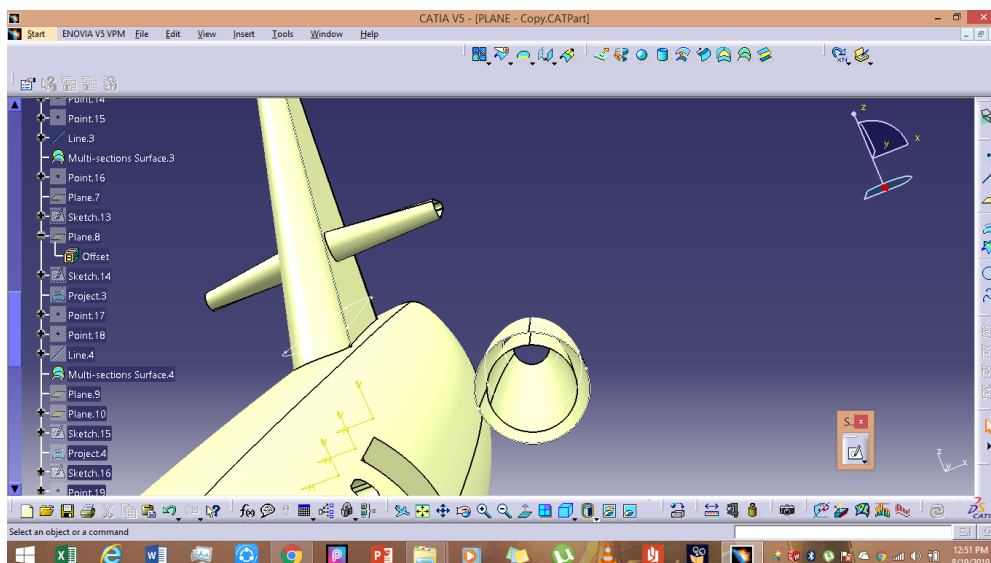
Create the profile you want the turbine to have.



Using "MULTI SECTIONS SURFACE" create Turbine



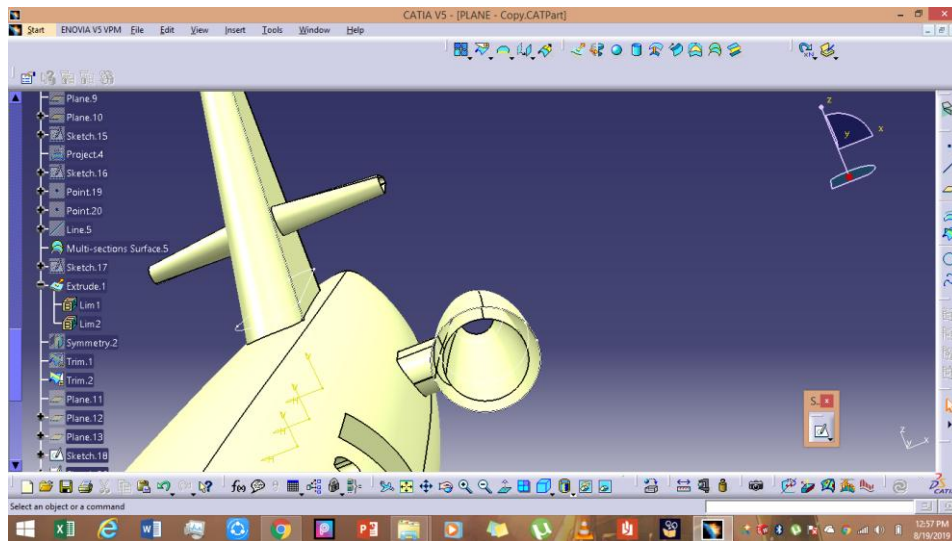
Take a Offset and create inner layer for the turbine



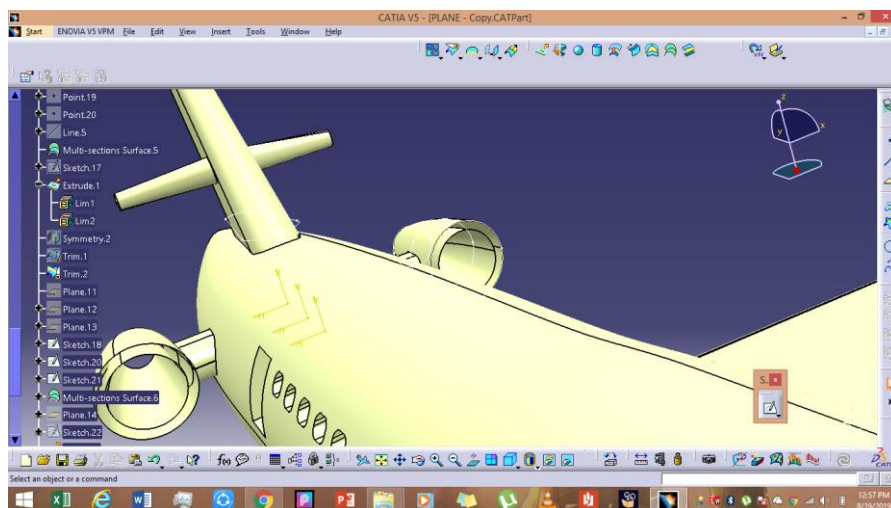
Create a plane Between the Turbine and the Body.

Draw a linking surface between body and turbine.

Extrude the profile "Up to the surface"

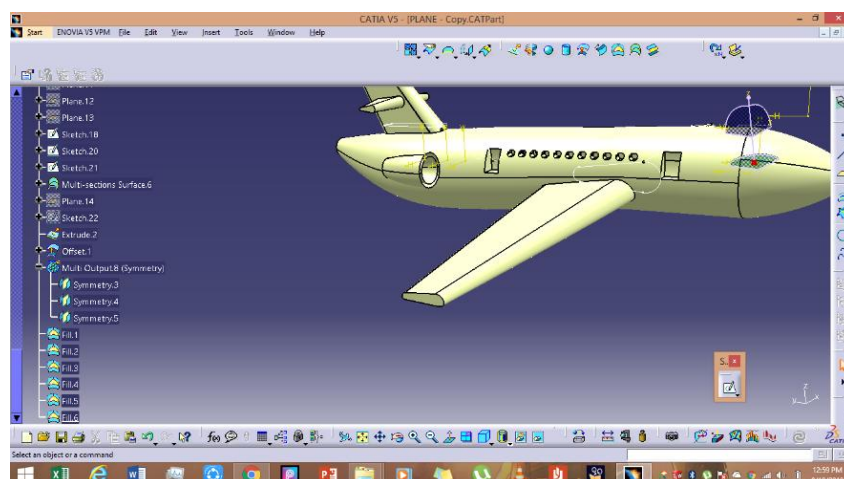


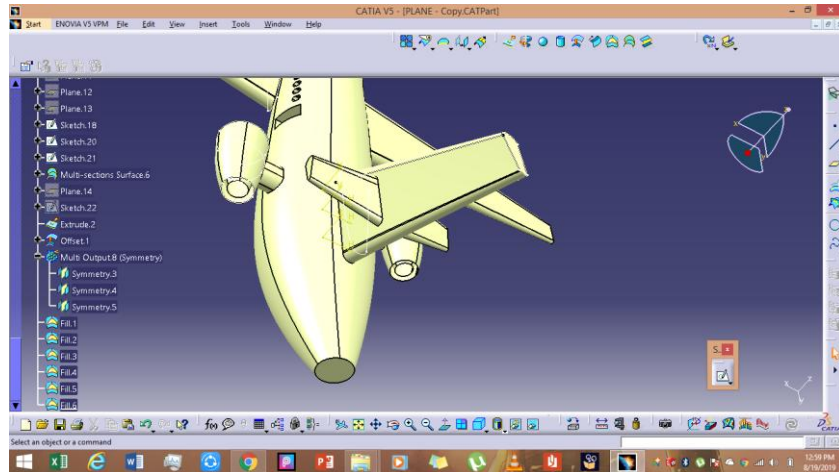
Create a Symmetry for the other side.



## Step 10: FILLING THE GAPS

Using the "FILL" command fill all the open surfaces

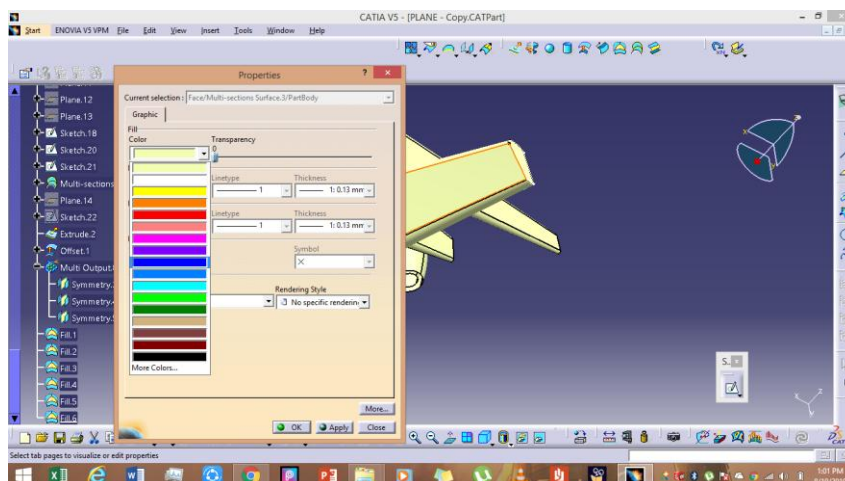
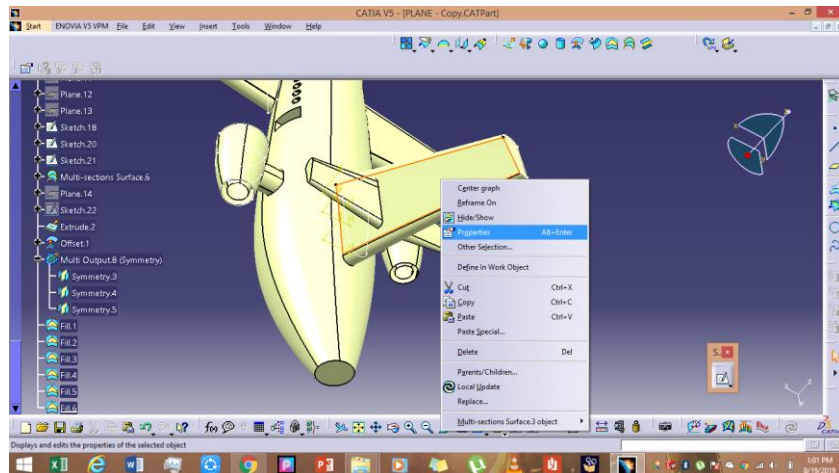




### Step 11: FINISHING

Add colours to the surfaces if you want.

Colours can be added by-----Right click on the surface----Go to properties----Change the colour



Click on OK.

### Validation:

1. Model 3D model of the given aircraft in Catia / Creo and obtain the 3D views compare with the hand sketch done.

**Result:** The 3D views of given aircraft configuration by hand sketch and Catia are done.