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AIRCRAFT DESIGN PROJECT - I RECORD

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INTERNAL EXAMINER

EXTERNAL EXAMINER

CHAPTER 01

INTRODUCTION

1.1 DESCRIPTION OF AIRCRAFT:

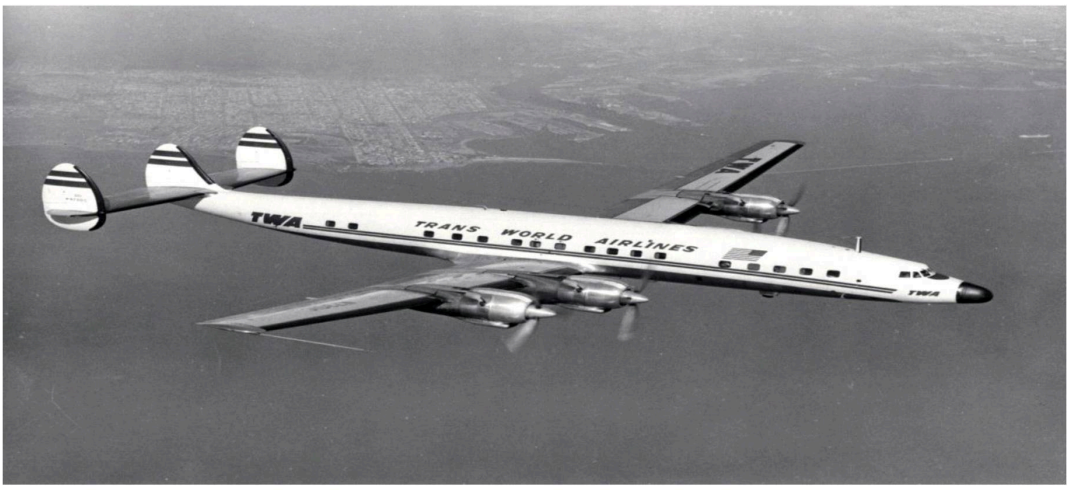
A non-stop flight is a flight by an aircraft with no intermediate stops. An ultra long-haul non-stop flight is a flight by a commercially operated airliner without any scheduled intermediate stop of any kind and route length using the higher end of the airliners' range. These flights usually follow a great circle route, often passing over the northern polar region.

History

During World War II, from 1943 to 1945, Qantas operated "The Double Sunrise", a weekly Consolidated PBY Catalina on a 5,650-kilometre (3,050 nmi) flight between Western Australia and Ceylon (now Sri Lanka), with average flight times of 27 hours.

On 1–2 October 1957 a Trans World Airlines Lockheed L-1649 Starliner, the ultimate piston-engine airliner in terms of range and endurance, flew the inaugural 8,640-kilometre (4,670 nmi) London–San Francisco polar route in 23 h 19 min

Fig:1.1 Lockheed L-1649 Aircraft



In June 1961 El Al extended the record for the world's longest scheduled non-stop commercial flight with a 9,270 kilometres (5,010 nmi) route from New York to Tel Aviv. Previously tested on trial runs with a Bristol Britannia in December 1957, the scheduled flight used a Boeing 707, averaging 9 hours 33 minutes.

In August 1967, Aerolineas Argentinas established their non-stop Boeing 707 service on a 10,062-kilometre (5,433 nmi) route between Madrid and Buenos Aires with a flight time of 12 h.

The Aerolineas Argentinas route was the longest scheduled flight by distance until 1976, when the Boeing 747SP entered service. In April 1976, Pan Am set a new record for longest non-stop scheduled flight with its 10,854-kilometre (5,861 nmi) New York–Tokyo route and in December the airline set another record with Sydney–San Francisco, covering 11,937 kilometres (6,445 nmi).

In May 1988 El Al introduced its long-haul passenger flight from Tel Aviv to Los Angeles, completing its 12,189-kilometre (6,582 nmi) journey in 13 h 41 min.

The fall of the Soviet Union eventually opened the airspace over Russia, allowing new circumpolar routes to come into use for commercial airlines. On March 1, 2001, Continental Airlines launched a 13,578-kilometre (7,332 nmi) non-stop service from Newark to Hong Kong—the first non-stop long-haul route for any airline with flying duration exceeding 16 hours. A few days later, United Airlines started their own New York to Hong Kong service, adding 3 kilometers to the distance.

On 3 February 2004, Singapore Airlines introduced a 14,113-kilometre (7,620 nmi) flight from Singapore to Los Angeles, scheduled for 16 h 30 min in summer and 15 h 35 min in winter ; 17 h 20 min in summer and 18 h 5 min in winter on the return trip.

In June 2004, Singapore Airlines introduced Flight SQ21, using the Airbus A340-500 on a 15,344-kilometre great circle route from Newyork to Singapore, passing within 130 kilo metres taking a little over 18 hours. This non-stop scheduled-commercial distance was immediately surpassed by return Flight SQ22, which flew the still current record 16,600-kilometre (9,000 nmi) back to Newark, on a route over Asia and Alaska. Despite the greater distance, Flight SQ22 averaged a slightly shorter 17 h 45 min due to assistance from prevailing high-altitude winds. In service for nine years, revenue was no longer high enough to sustain the service and the routes were dropped in November 2013.

Current

Since 1 March 2016, the longest non-stop scheduled airline flight is Emirates flight EK448 from Dubai, United Arab Emirates to Auckland, New Zealand, and its return flight EK449, at 14,200 kilometres (7,700 nmi). The inaugural flight was flown using an Airbus A380-800,[10][11] but subsequent regular service flights will be flown using a Boeing 777-200LR.

Future

United Airlines announced a daily Boeing 787-9 service from San Francisco to Singapore non-stop starting June 1, 2016. The 16 h 20m UA1 flight and its 15 h 30m return UA2 flight will cover 13,592 km (7,339 nmi).

On 3 December 2016, Qatar Airways will begin nonstop flights between Doha and Auckland, making this flight the longest in the world. The flight covers 14,536 kilometres (7,849 nmi) and will be operated by Boeing 777-200LR aircraft.

Fig:1.2 FIGURE OF BOEING 777

1.2 INTRODUCTION TO AIRCRAFT DESIGN:

1.2.1 PURPOSE AND SCOPE OF AIRPLANE DESIGN OBJECTIVES

- To meet the FUNCTIONAL, OPERATIONAL and SAFETY requirements set out OR acceptable to the USER.

1.2.2 ACTUAL PROCESS OF DESIGN

- Selection of aircraft type and shape
- Determination of geometric parameters
- Selection of power plant
- Structural design and analysis of various components
- Determination of aircraft flight and operational characteristics.

1.2.3 PRELIMINARY DESIGN

It consists of the initial stages of design, resulting in the presentation of a brochure containing preliminary drawings and clearly stating the operational capabilities of the airplane being designed. This Brochure has to be approved by the manufacturer and/or the customer.

The steps involved:

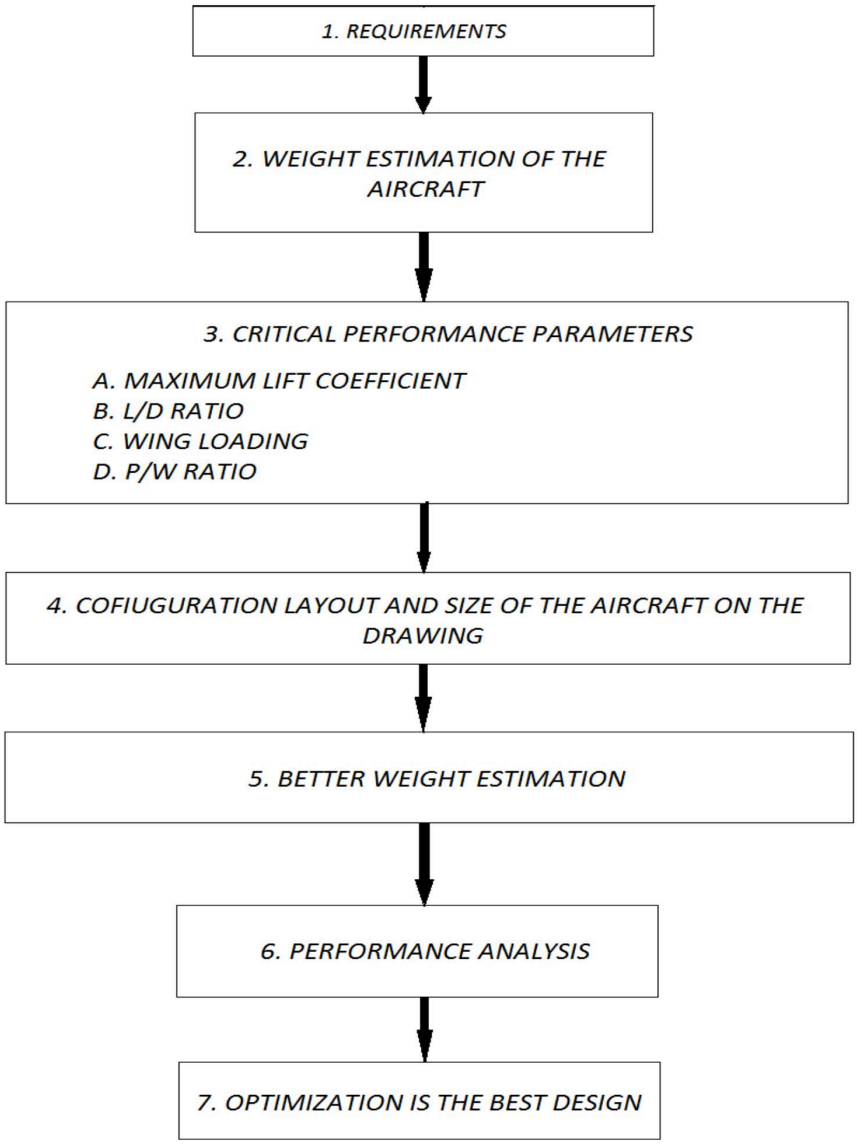
- Layout of the main components
- Arrangement of airplane equipment and control systems
- Selection of power plant
- Aerodynamic and stability calculations
- Preliminary structural design of MAJOR components
- Weight estimation and c.g. travel
- Preliminary and Structural Testing
- Drafting the preliminary 3-view Drawings

1.2.4 DESIGN PROJECT

- Internal discussions
- Discussions with prospective customers
- Discussions with Certification Authorities
- Consultations with suppliers of power plant and major accessories
- Deciding upon a BROAD OUTLINE to start the ACTUAL DESIGN, which will consist of Construction of Mock-up
- Structural layout of all the individual units, and their stress analysis
- Drafting of detailed design drawings
- Structural and functional testing
- Nomenclature of parts
- Supplying key and assembly diagrams
- Final power plant calculations
- Final weight estimation and c.g. limits
- Final performance calculation

1.2.5 CYCLES OF DESIGN PROCESS:

Fig: 1.3



Aircraft design can be broken into three major phases,

- Conceptual Design
- Preliminary Design
- Detail Design

1.2.5 CONCEPTUAL DESIGN:

Conceptual design is a very fluid process. New ideas and problems emerge as a design is investigated in ever increasing detail. Each time the latest design is analyzed and sized, it must be redrawn to reflect the new gross weight, fuel weight, wing size, engine size, and other changes.

Conceptual design will usually begin with either a specific set of design requirements established by the prospective customer or a company generated guess as to what future customers need. Design requirements include aircraft range and payload, take-off and landing distances, and maneuverability and speed requirements.

The actual design effort usually begins with conceptual sketch. A good conceptual sketch will include the approximate wing and tail geometries, the fuselage shape, and the internal locations of the major components such as the engine, cockpit, payload/passenger compartment, landing gear and fuel tanks.

1.2.6 PRELIMINARY DESIGN

It can be said to begin when the major changes are over. The big questions such as whether to use a canard or an aft tail have been resolved. At some point late in preliminary design, even minor changes are stopped when a decision is made to freeze the configuration. During this design the specialists in areas such as structures, landing gear, and control systems will design and analyze their portion of the aircraft. Testing is initiated in areas such as aerodynamics, propulsion, structures, and stability and control.

A key activity during this type of design is “LOFTING’. Lofting is the mathematical modeling of the outside skin of the aircraft with sufficient accuracy to insure proper fit between its different parts, even if they are designed by different designers and possibly fabricated in different locations. The ultimate objective during this design is to ready the company for the detail stage, also called “FULL-SCALE DEVELOPMENT”.

1.2.7 DETAILED DESINGNING

Assuming a favorable decision for entering full-scale development, the detail design phase begins in which the actual pieces to be fabricated are designed. During conceptual and preliminary design the wing box will designed and analyzed as whole. During detail design, that whole will be broken into individual ribs, spars and skins, each of which must be separately designed and analyzed.

Another important part of detail design is called production design. Specialists determine how the airplane will be fabricated, starting with smallest and simplest subassemblies and building upto the final assembly process. Production designers frequently wish to modify the design for ease of manufacture; that can have a major impact on performance or weight. Compromises are inevitable, but the design must still meet the original requirements

During detail design, the testing effort intensifies. Actual structure of the aircraft is fabricated and tested. Control laws for the flight control system are tested on an "iron-bird" simulator, a detailed working model of the actuators and flight control surfaces. Flight simulators are developed and flown by both company and customer test pilots.

Detail design ends with fabrication of the aircraft. Frequently the fabrication begins on part of the aircraft before the entire detail-design effort is completed.

CHAPTER 02

COMPARITIVE STUDY

2.1 COMPARITIVE DATA SHEET:

Table:2.1

S.NO	SPECIFICATION	BOEING 777	AIRBUS A340-600	BOEING 777-300ER	BOEING 747-8	AIRBUS A350-900
1	TYPE	Passenger	Passenger	Passenger	Passenger	Passenger
2	PAY LOAD	51,250kg	63,000kg	53,928kg	52,300kg	67,700g
3	CRUSING ALTITUDE	10,973m	10,668m	10,668m	10,668m	10,973m
4	OPERATING MACH NO	0.89	0.86	0.84	0.855	0.83
5	RANGE	14,260km	14,600km	14,490km	14,815km	14,350km
6	ENDURANCE	16hrs	19hrs	17hrs	18hrs	16hrs
7	CRUISE VELOCITY	905kmph	889kmph	901kmph	921kmph	910kmph
8	$(L/D)_{MAX}$	2.55	2.53	-	-	-
9	$(L/D)_{CRUISE}$	7.77	-	-	7.50	7.57
10	GROSS WEIGHT	1,68,598 kg	2,58,620 Kg	1,53,436 kg	3,47,300 kg	2,39,000 kg
11	FUEL WEIGHT	1,17,348 kg	1,95,620 Kg	1,18,436 kg	2,95,000 kg	2,03,000 kg
12	EMPTY WEIGHT	1,34,800 kg	2,40,000 kg	3,50,678 kg	2,13,000 kg	3,05,300 kg
13	$(C_L)_{MAX}$	-	0.025	-	0.025	0.015
14	$(C_L)_{CRUISE}$	-	0.027	-	0.024	0.015

15	**$(C_L)_{CLIMB}$**	-	1.625	-	-	-
16	**STALL VELOCITY**	247.43 m/s	266.67 m/s	-	-	-
17	**DENSITY AT SEALEVEL**	1.225 kg/m^3	1.225 kg/m^3	1.225 kg/m^3	1.225 kg/m^3	1.225 kg/m^3
18	**DENSITY AT MAXIMUM OPERATING ALTITUDE**	0.410 kg/m^3	-	-	-	-
19	**SFC**	44.1 g/KNs	53.8 g/KNs	33.8 g/KNs	47.1 g/KNs	51.9 g/KNs
20	**SWEPT ANGLE**	31.64°	-	31.64°	31.64°	-
21	**TAPER RATIO**	0.300	0.283	0.240	0.251	0.220
22	**WING AREA**	$460m^2$	$437.3m^2$	$425m^2$	$554m^2$	$442m^2$
23	**WING SPAN**	60.9m	73.5m	66.5m	59.9m	64.75m
24	**THRUST**	338KN	249KN	512KN	296KN	374KN
25	**ASPECT RATIO**	7.73	9.39	8.80	9.26	9.39
26	**LENGTH**	63.7m	74.96m	63.7m	63.7m	69.72m
27	**HEIGHT**	8.39m	5.64m	8.39m	8.39m	5.64m
28	**MAXIMUM TAKE OFF WEIGHT**	251.29 Tons	368 Tons	388 Tons	447.696 Tons	268.9 tons
29	**RATE OF CLIMB**	3m/s	2.8m/s	3m/s	3m/s	2.8m/s
30	**FUSELAGE DIAMETER**	6.20m	6.37m	5.6m	5.94m	5.72m
31	**MAXIMUM SPEED**	905km/h	611km/h	905km/h	905km/h	618km/h

32	**POWER PLANT**	PW-4084 RR877 two spool high bypass turbo fan engines	RR550 TRENT • Trent 500 is a three-shaft high bypass ratio engine	GE90-115B most powerful turbofan (having demonstrated over 127,000 pounds of thrust)	GENX-2B67 high-thrust jet engine blade count (from 22 to 18 fan blades	RR TRENT XWB Three-shaft high bypass ratio turbofan engine
33	**THRUST TO WEIGHT RATIO**	0.297	0.2568	0.2771	0.2884	0.2789
34	**CREW**	Two	Two	Two	Two	Two
35	**SERVICE CEILING**	13,140m	12,500m	13,140m	13,140m	12,500m

CHAPTER 03

SELECTION OF DESIGN PARAMETER

3.1 PAYLOAD:

Payload is the carrying capacity of an aircraft or launch vehicle, usually measured in terms of weight. Depending on the nature of the flight or mission, the payload of a vehicle may include cargo, passengers, flight crew, munitions, scientific instruments or experiments, or other equipment.

3.2 CRUISING ALTITUTE

The height in the sky at which an airplane stays for most of a flight.

3.3 CRUISING VELOCITY

A speed for a particular vehicle, ship, or aircraft, usually somewhat below maximum, that is comfortable, and economical.

3.4 OPERATING MACH NO

The Mach number is a dimensionless quantity representing the ratio of flow velocity past a boundary to the local speed of sound.

$$\mathrm{M} = \frac{u}{c},$$

Where

M is the Mach number,
u is the local flow velocity with respect to the boundaries (either internal, such as an object immersed in the flow, or external, like a channel), and
c is the speed of sound in the medium.
For subsonic upto 0.85
For supersonic upto 2.2

3.5 Range

The maximal total range is the maximum distance an aircraft can fly between takeoff and landing, as limited by fuel capacity in powered aircraft, or cross-country speed and environmental conditions in unpowered aircraft. The range can be seen as the cross-country ground speed multiplied by the maximum time in the air.

3.6 FERRY RANGE

Ferry range means the maximum range the aircraft can fly. This usually means maximum fuel load, optionally with extra fuel tanks and minimum equipment. It refers to transport of aircraft for use on remote location without any passengers or cargo.

3.7 ENDURANCE

Endurance is the maximum length of time that an aircraft can spend in cruising flight.

Endurance is different from range, which is a measure of distance flown.

3.8 LIFT TO DRAG RATIO

Lift the lift-to-drag ratio, or L/D ratio, is the amount of lift generated by a wing or vehicle, divided by the aerodynamic drag it creates by moving through the air. A higher or more favorable L/D ratio is typically one of the major goals in aircraft design; since a particular aircraft's required lift is set by its weight, delivering that lift with lower drag leads directly to better fuel economy in aircraft, climb performance, and glide ratio to drag ratio.

Lift to drag ratio is directly proportional to aerodynamic efficiency.When lift is high and drag is low,lift to drag ratio is maximum.

3.9 GROSS WEIGHT

An aircraft's gross weight will decrease during a flight due to fuel and oil consumption.

3.10 FUEL WEIGHT

The maximum zero fuel weight (MZFW) is the maximum weight allowed before usable fuel and other specified usable agents (engine injection fluid, and other consumable propulsion agents) are loaded in defined sections of the aircraft as limited by strength and airworthiness requirements.

3.11 EMPTY WEIGHT

Operating empty weight (OEW) or basic operating weight is the basic weight of an aircraft including the crew, all fluids necessary for operation such as engine oil, engine coolant, water, unusable fuel and all operator items and equipment required for flight but excluding usable fuel and the payload.

3.12 STALL VELOCITY

A stall is a condition in aerodynamics and aviation wherein the angle of attack increases beyond a certain point such that the lift begins to decrease. The angle at which this occurs is called the critical angle of attack.

3.13 WING AREA

The wing area is the projected area of the plan form and is bounded by the leading and trailing edges and the wing tips.

3.14 WING SPAN

The distance from the leading to trailing edges is called the chord. The ends of the wing are called the wing tips, and the distance from one wing tip to the other is called the span. The aspect ratio (AR) of a wing is defined to be the square of the span (s) divided by the wing area (A). Aspect ratio is a measure of how long and slender a wing is from tip to tip.

Tapered wing gives good performance.

3.15 ASPECT RATIO

The aspect ratio (AR) of a wing is defined to be the square of the span (s)divided by the wing area (A). Aspect ratio is a measure of how long and slender a wing is from tip to tip. For a rectangular wing, this reduces to the ratio of the span to the chord length (c):

$AR = s^2 / A = s^2 / (s * c) = s / c$

High aspect ratio wings have long spans (like high performance gliders), while low aspect ratio wings have either short spans or thick chords (like the Space Shuttle). Gliders have a high aspect ratio because the drag of the aircraft depends on this parameter. A higher aspect ratio gives a lower drag, a higher lift to drag ratio, and a better glide angle.

3.16 RATE OF CLIMB

The rate of climb (RoC) is an aircraft's vertical speed – the rate of altitude change with respect to time.

3.17 SERVICE CEILING

A service ceiling is defined as the maximum altitude at which the aircraft can maintain a specified rate of climb (typically 500 feet/minute for jets).

3.18 THRUST

Thrust is the force which moves an aircraft through the air. Thrust is used to overcome the drag of an airplane, and to overcome the weight of a rocket. Thrust is generated by the engines of the aircraft through some kind of propulsion system

3.19 SWEPT ANGLE

A swept wing is a wing which angles either backward or, occasionally, forward, from its root rather than in a straight sideways direction. Wing sweep has the effect of delaying the shock waves and accompanying aerodynamic drag rise caused by fluid compressibility near the speed of sound, improving performance. Swept wings are also sometimes used for other reasons, such as structural convenience or visibility.

3.20 POWERPLANT

An aircraft power plant is an engine. It is made up of many components, such as cylinders, pistons, and fans, which help produce the energy needed to propel an aircraft. The power plant may be a jet engine or a combination of propellers.

Scramjet/Ramjet: Missiles

Turbojet: Poor performance at low altitude(only for high speed aircraft

3.21 SPECIFIC FUEL CONSUMPTION

The fuel consumption of TSFC is "how much fuel the engine burns each hour." The specific of TSFC is a scientific term meaning "divided by mass or weight." In this case, specific means "per pound (Newton) of thrust." The thrust of TSFC is included to indicate that we are talking about gas turbine.

3.22 COEFFICIENT OF LIFT

The lift coefficient (C_L, C_N or C_z) is a dimensionless coefficient that relates the lift generated by a lifting body to the fluid density around the body, the fluid velocity and an associated reference area. A lifting body is a foil or a complete foil-bearing body such as a fixed-wing aircraft.

3.23 THRUST TO WEIGHT RATIO

It is defined as ratio between thrust and weight.

For fighter T/W >1

CHAPTER 04

MEAN DATA SHEET

Table:4.1

SPECIFICATIONS	MEAN VALUE
TYPE	Long Range Passenger
PAYLOAD	51,250 kg
CRUISING ALTITUDE	10,973 m
OPERATING MACH. NO	0.83
RANGE	14,000 KM
ENDURANCE	19 hr
CRUISE VELOCITY	256 m/s
$(L/D)_{MAX}$	2.55
$(L/D)_{CRUISE}$	7.56
NET WEIGHT	1,53,436 kg
FUEL WEIGHT	2,95,000 kg
EMPTY WEIGHT	1,34,800 kg
$(C_L)_{MAX}$	0.025
$(C_L)_{CRUISE}$	0.027
$(C_L)_{CLIMB}$	1.625
STALL VELOCITY	247.43m/s
DENSITY AT SEA LEVEL	1.225 kg/m³

DENSITY AT MAXIMUM OPERATING ALTITUDE	0.410 kg/m³
SFC	33.8 g/KNs
SWEPT ANGLE	31.64°
TAPPER RATIO	0.283
WING AREA	534 m^2
WING SPAN	73.5 m
THRUST	512 KN
ASPECT RATIO	9.39
LENGTH	74.96 m
HEIGHT	8.39 m
MAX. TAKE OFF WEIGHT	447.696 tons
RATE OF CLIMB	3.08 m/s
FUSELAGE DIAMETER	6.20
MAXIMUM SPEED	251m/s
POWERPLANT	High bypass turbo fan (RR Trent 1000)
THRUST TO WEIGHT RATIO	0.297
CREW	2
SERVICE CEILING	13,140 m

CHAPTER 05
PROFILE CALCULATION

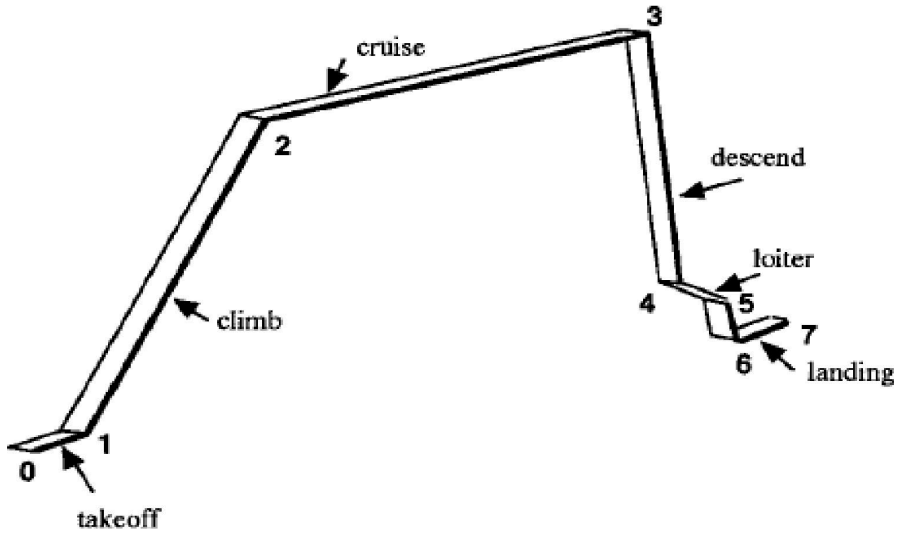
5.1 MISSION PROFILE:

The aircraft undergoes a simple mission. The need of fuel is necessary only for the shown cruise. The reserve fuel as mentioned above is needed during loiter.

- Mission profile includes take off from the runway.
- Climbing to a specified altitude
- Cruising and favorable atmospheric conditions
- Decent
- Loiter
- Landing
- For civil transports

- From point 0 to 1 - taxing and takeoff of aircraft
- From point 1 to 2 – Climb of aircraft
- From point 2 to 3 – Cruising of aircraft
- At point 4 – Loiter of aircraft
- From point 3 to 5 – Decent of aircraft
- From point 5 to 6 – Landing of aircraft

Fig:5.1 MISSION PROFILE



5.2 CERTIFICATION AGENCIES :

The two major civil aircraft certification agencies are as follows,

- DGCA – Director General for Civil Aviation.
- FAA – Federal Aviation Administration.

The two major military aircraft certification agencies are as follows,

- CEMILAC – Centre for Military Airworthiness and Certification
- MIL-SPEC – Military Specification Certification

5.3 WEIGHT ESTIMATION:

1. $W_{OVERALL} = W_{CREW} + W_{PAYLOAD} + W_{FUEL} + W_{EMPTY}$
2. $W_{OVERALL} = \frac{W_{CREW} + W_{PAYLOAD}}{1 - \frac{W_{FUEL}}{W_{OVERALL}} - \frac{W_{EMPTY}}{W_{OVERALL}}}$
3. Take off $= \frac{\text{Takeoff Weight}}{\text{Overall Weight}} = \frac{W_1}{W_0} = 0.970$
4. Climb $= \frac{W_2}{W_1} = 0.98$

5. Cruise, R = $\frac{V_\infty}{SFC}(L/D)\ \ln(W_2/W_3)$

6. Endurance = $\frac{1}{SFC}(L/D)\ \ln(W_3/W_4)$

   $[W_4 = W_5]$

7. Estimation of Fuel Fraction,

$$\frac{W_{FUEL}}{W_{OVERALL}} = 1 - \frac{W_1}{W_0} - \frac{W_2}{W_1} - \frac{W_3}{W_2} - \frac{W_4}{W_3} - \frac{W_5}{W_4} - \frac{W_6}{W_5}$$

5.3.1 CALCULATION :

$$W_{OVERALL} = W_{CREW} + W_{PAYLOAD} + W_{FUEL} + W_{EMPTY}$$

$$W_{OVERALL} = 160 + 51250 + 295000 + 134800$$

$$W_{OVERALL} = 481210 \text{ kg}$$

$$W_{OVERALL} = \frac{160+51250}{1-\frac{295000}{481210}-\frac{134800}{481210}}$$

$$W_{OVERALL} = 481210 \text{ kg}$$

1. Take Off = 0.970 * 481210

   $W_1 = 466773.7$ kg

2. Climb ,W_2 / W_1 = 0.985

   W_2 = 0.985 (466773.7)

   W_2 = 459772 kg

3. Cruise,

$$14000 = \frac{256}{0.034}\ (7.46)\ \ln\ (459772/W_3)$$

$$e^{(0.249)} = \frac{459772}{W_3}$$

$$W_3 = \frac{459772}{1.283} = 358314 \text{ kg}$$

$$W_3 = 358314 \text{ kg}$$

4. Endurance $= \frac{1}{C}\ (L/D)_{CRUISE} \ln(W_3/W_4)$

$$19 = \frac{1}{0.034}\ [7.46]\ \ln\ (358314\ /W_4\ )$$

$$W_4 = W_5 = 328591.23 \text{ kg}$$

5. $W_6 / W_5 = 0.995$

$W_6 = 0.995 * 328591.23$

W_6= 326943.27 kg

6. $$\frac{W_{FUEL}}{W_{OVERALL}} = 1 - \frac{466773.7}{481210} - \frac{459772}{466773.7} - \frac{358314}{459772} - \frac{328591.23}{358314} - \frac{328591.23}{328591.23} - \frac{326943.27}{328591.23}$$

W_F/W_0 = - 4.62

CHAPTER 06
POWER PLANT SELECTION

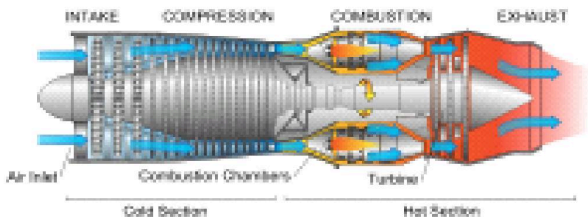
6.1 CLASSIFICATION OF POWER PLANTS:

- Piston power propeller engine
- Turbojet engine
- Turboprop engine
- Turbofan engine
- Turboshaft engine

6.1.1 JET ENGINE :

Jet engine is a reaction engine discharging a fast moving jet that generates thrust by jet propulsion in accordance with Newton's laws of motion. This broad definition of jet engines includes turbojets, turbofans, rockets, ramjets, and pulse jets. In general, jet engines are combustion engines but non-combusting forms also exists.

Fig : 6.1



A turbojet engine is a gas turbine engine that works by compressing air with an inlet and a compressor (axial , centrifugal or both), mixing fuel with the compressed air, burning the mixture in the combustor, and then passing the hot,

high pressure air through a turbine and a nozzle. The compressor is powered by the turbine, which extracts energy from the expanding gas passing through it. The engine converts internal energy in the fuel to kinetic energy in the exhaust, producing thrust. All the air ingested by the inlet is passed through the compressor, combustor, and turbine, unlike the turbofan engine described below.

6.1.2 TURBO FAN ENGINE:

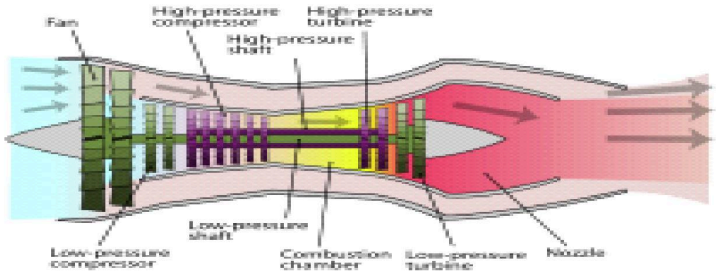


Fig : 6.2

Schematic diagram illustrating the operation of a low-bypass turbofan engine.

A turbofan engine is a gas turbine engine that is very similar to a turbojet. Like a turbojet, it uses the gas generator core (compressor, combustor, turbine) to convert internal energy in fuel to kinetic energy in the exhaust. Turbofans differ from turbojets in that they have an additional component, a fan. Like the compressor, the fan is powered by the turbine section of the engine. Unlike the turbojet, some of the flow accelerated by the fan bypasses the gas generator core of the engine and is exhausted through a nozzle. The bypassed flow is at lower velocities, but a higher mass, making thrust produced by the fan more efficient than thrust produced by the core. Turbofans are generally more efficient than turbojets at subsonic speeds, but they have a larger frontal area which generates more drag.

There are two general types of turbofan engines, low bypass and high bypass. Low-bypass turbofans have a bypass ratio of around 2:1 or less, meaning that for each kilogram of air that passes through the core of the engine, two kilograms or less of air bypass the core. Low-bypass turbofans often use a mixed exhaust nozzle meaning that the bypassed flow and the core flow exit from the same nozzle. High-bypass turbofans have larger bypass ratios, sometimes on the order of 5:1 or 6:1. These turbofans can produce much more thrust than low-bypass turbofans or turbojets because of the large mass of air that the fan can accelerate, and are often more fuel efficient than low-bypass turbofans or turbojets.

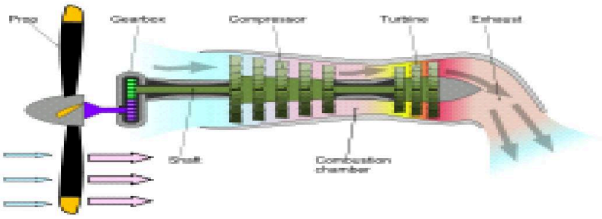


Fig : 6.3

In our design, we select 4× High Bypass Turbofan engine (Rolls Royce Trent1000) to meet the given design standards

6.1.3 TURBOPROP ENGINE:

Turboprop engines are jet engine derivatives, still gas turbines, that extract work from the hot-exhaust jet to turn a rotating shaft, which is then used to produce thrust by some other means. While not strictly jet engines in that they rely on an auxiliary mechanism to produce thrust, turboprops are very similar to other turbine-based jet engines, and are often described as such.

In turboprop engines, a portion of the engine's thrust is produced by spinning a propeller rather than relying solely on high-speed jet exhaust. As their jet thrust is augmented by a propeller, turboprops are occasionally referred to as a type of hybrid jet engine. They are quite similar to turbofans in many respects. Both fans and propellers are powered the same way, although most turboprops use gear reduction between the turbine and the propeller. While many turboprops generate the majority of their thrust with the propeller, the hot-jet exhaust is an important design point, and maximum thrust is obtained by matching thrust contributions of the propeller to the hot jet. Turboprops generally have better performance than turbojets or turbofans at low speeds where propeller efficiency is high, but become increasingly noisy and inefficient at high speeds.

Turboshaft engines are very similar to turboprops, differing in that nearly all energy in the exhaust is extracted to spin the rotating shaft, which is used to power machinery rather than a propeller, they therefore generate little to no jet thrust and are often used to power helicopters.

TABLE 6.1.1: SPECIFICATION CHART FOR POWERPLANT

1	Engine name	Rolls Royce Trent 1000
2	Manufacturer	Rolls Royce PLC
3	Engine type	Turbofan
4	Length	4.738 m (186.5 inches)
5	Diameter	112 inches (2.85m)
6	Dry weight	Approx 12,710 lb (5,765 kg)
7	Compressor	1- Stage LP, 8- Stage IP, 6- Stage HP Compressor
8	Combustors	Tiled Combustor
9	Turbine	• Stage HP Turbine, 1- Stage IP Turbine, 6- Stage LP Turbine
10	T / W	6.189
11	Maximum Thrust	53,000-78,000 lbf
12	Overall pressure ratio	52:1(Top To Climb)
13	Airflow	1290 kg/s

6.2 DESCRIPTON OF ROLLS ROYCE TRENT 1000:

The Rolls-Royce Trent 1000 is a turbofan engine, developed from earlier Trent series engines. The Trent 1000 powered the Boeing 787 Dreamliner on its maiden flight, and on its first commercial flight.

On 26 October 2011, the 787 flew its first commercial flight from Tokyo-Narita to Hong Kong on All Nippon Airways, powered by Trent 1000 engines.

FIG:6.4 ROLLS ROYCE TRENT 1000 ENGINE

6.2.1 LOACTION OF POWERPLANT:

The powerplant is located below the wing of an aircraft. In most of the aircraft aviation turbine engine is located below the wing for the additional stability.

6.2.2 ADVANTAGES:

- Greater due to elimination of wing-pylon.
- Less drag particularly in critical take off.
- It may be the fashion.

6.2.3 DISADVANTAGES:

- The centre of the gravity of the empty aircraft is moved aft-well behind the centre of gravity of payload.
- The wing weight advantage of wing mounted engines is lost.
- It may not be the fashion.

CHAPTER 07

AERODYNAMIC ANALYSIS

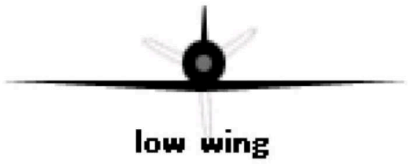
7.1 WING DESIGN:

BASED ON LOCATION:

Low wing configuration

Mid wing configuration

High wing configuration



BASED ON SHAPE:

- Straight wing
- Tapered wing
- Elliptical wing
- Trapezoidal wing
- Swept backward wing
- Swept forward wing
- Delta wing
- Blended wing

STRAIGHT WING:

Is the oldest aircraft wing design ever made, it first seen working in the wright flyer, though also failed-try aircraft that had wing featured this design.

It simply consists in a straight-line board structure that includes all the components.

BLENDED WING:

A Blended wing body is a fixed-wing aircraft having no clear dividing line between the wings and the main body of the aircraft the form is composed of distinct wing and body structures, through the wings are smoothly blended into the body, unlike a flying wing 1 has no distinct fuselage.

TAPERED WING:

Way to reduce drag while increasing strength is with trapezoidal shaped wing

ELLIPTICAL WING:

An elliptical wing is a wing platform whose leading and trailing edges each approximate to segments of an ellipse.

TRAPEZOIDAL WING:

A trapezoidal wing is a straight edged and tapered wing platform. Trapezoidal configuration offers some advantages for high-speed flight and has been used on a small number of aircraft types.

SWEPT BACKWARD WING:

Constant angle for cargo, civil, transport, military cargo

SWEPT FORWARD WING:

Angle between horizontal plane and wing airflow is in opposite direction.

DELTA WING:

A single wing configuration in the shape of an isosceles triangle used on certain aircraft.

LOCATION OF WING :

WING LOCATION = LOW WING

Low wing aircrafts are planes with the wing mounted below the main fuselage of the aircraft. Low wing aircrafts are the majority of both general aviation and commercial planes .

ADVANTAGES:

- Low wing aircraft can offer better visibility above the aircraft , as the wing remains mostly out of the field of view
- Its simple in design.
- It allow for a shortened takeoff roll and faster acceleration.
- It is easier to land in cross wind.

DISADVANTAGES:

- Taxi is high.
- Low wings plane may not be as safe as high wing models when landing on soft or on prepared fields.

WING SELECTION:

Type of wing: Low wing configuration

7.2 LIFT ESTIMATION:

1. LIFT DURING CLIMB:

$$V_{Takeoff} = 1.2 \times V_{Stall}$$

$$= 1.2 \times (247.43)$$

$$V_{Takeoff} = 296.92 \text{ m/s}$$

$$L = \frac{1}{2}\rho_\infty V_\infty^2 S C_L$$

$$L_{climb} = \frac{1}{2} \times 1.225 \times (296.92)^2 \times 534 \times 1.625$$

$$\mathbf{L_{climb} = 46857.54 \text{ KN}}$$

2. LIFT DURING CRUISE:

$$L = \frac{1}{2}\rho_\infty V_\infty^2 S C_L$$

$$L_{cruise} = \frac{1}{2} \times 0.410 \times (256)^2 \times 534 \times 0.027$$

$$\mathbf{L_{climb} = 937.04 \text{ KN}}$$

3. LIFT DURING LANDING:

$$L = \frac{1}{2}\rho_\infty V_\infty^2 S C_L$$

$$L_{landing} = \frac{1}{2} \times 1.225 \times (251)^2 \times 534 \times 1.3$$

$$\mathbf{L_{landing} = 26787.87 \text{ KN}}$$

7.3 DRAG ESTIMATION:

7.3.1 TYPES OF DRAG:

INDUCED DRAG:

The drag that results from the generation of a trailing vortex system downstream of a lifting surface with a finite aspect ratio. In another word, this type of drag is induced by the lift force.

PARASITE DRAG:

The total drag of an airplane minus the induced drag. Thus, it is the drag not directly associated with the production of lift. The parasite drag is composed of drag of various aerodynamic components; the definitions of which follow.

SKIN FRICTION DRAG:

The drag on a body resulting from viscous shearing stresses (i.e., friction) over its contact surface (i.e., skin). The drag of a very streamlined shape such as a thin, flat plate is frequently expressed in terms of a skin friction drag. This drag is a function of Reynolds number. There are mainly two cases where the flow in the boundary layer is entirely laminar or entirely turbulent over the plate. The Reynolds number is based on the total length of the object in the direction of the velocity. In a usual application, the boundary layer is normally laminar near the leading edge of the object undergoing transition to a turbulent layer at some distance back along the surface.

A laminar boundary layer begins to develop at the leading edge and its thickness grows in downstream. At some distance from the leading edge the laminar boundary becomes unstable and is unable to suppress disturbances imposed on it by surface roughness or fluctuations in the free stream. In a distance the boundary layer usually undergoes a transition to a turbulent boundary layer. The layer suddenly increases in thickness and is characterized by a mean velocity profile on which a random fluctuating velocity component is superimposed. The distance, from the leading edge of the object to the transition point can be calculated from the transition Reynolds number. Skin friction factor is independent of surface roughness in laminar flow, but is a strong function of surface roughness in turbulent flow due to boundary layer.

FORM DRAG (sometimes called Pressure Drag):

The drag on a body resulting from the integrated effect of the static pressure acting normal to its surface resolved in the drag direction. Unlike the skin friction drag that results from viscous shearing forces tangential to a body's surface, form drag results from the distribution of pressure normal to the body's surface. In an extreme case of a flat plate normal to the flow, the drag is totally the result of an imbalance in the pressure distribution. As with skin friction drag, form drag is generally dependent on Reynolds number. Form drag is based on the projected frontal area. As a body begins to move through the air, the vorticity in the boundary layer is shed from the upper and lower surfaces to form two vortices of opposite rotation. A number of symmetrical shapes having drag. The drag coefficient values in this table are based on the frontal area. In this table, the flow is coming from left to the right.

DRAG CALCLATION:

AIRCRAFT TYPE	C_{D_0}	e
Single Engine Light Aircraft	0.026	0.8
Multi Engine Wide Body Aircraft	0.019	0.84
Twin Engine Wide Body Aircraft	0.017	0.85
Military Aircraft With External Sources	0.028	0.70
Twin Passenger Aircraft	0.021	0.85

TABLE:7.3.1 DRAG COEFFICIENT

DRAG DURING CLIMB:

$$D = \frac{1}{2}\rho_\infty V_\infty^2 S C_D$$

$$C_d = C_{d_0} + KC_L^2$$

C_{d_0} is constant for aircraft based on the NACA series

$$K = \frac{1}{\pi \times e \times AR}$$

$$K = \frac{1}{\pi \times .84 \times 8.39}$$

$$K = 0.04$$

$$C_{d_0} = 0.019$$

FOR CLIMB,

$$D = \frac{1}{2}\rho_\infty V_\infty^2 S C_D$$

$$C_d = C_{d_0} + KC_L^2$$

$$= 0.019 + 0.04(1.625)^2$$

$$C_d = 0.124$$

$$D = \frac{1}{2} \times 1.225 \times (296.92)^2 \times 534 \times 0.124$$

D = 3575.59 KN

FOR CRUISE,

$$D = \frac{1}{2}\rho_\infty V_\infty^2 S C_D$$

$$C_d = C_{d_0} + KC_L^2$$

$$= 0.019 + 0.04(0.027)^2$$

$$C_d = 0.019$$

$$D = \frac{1}{2} \times 0.410 \times (256)^2 \times 534 \times 0.019$$

D = 136.31 KN

FOR LANDING,

$$D = \frac{1}{2}\rho_\infty V_\infty^2 S C_D$$

$$C_d = C_{d_0} + K C_L^2$$

$$= 0.019 + 0.04(1.3)^2$$

$$C_d = 0.0866$$

$$D = \frac{1}{2} \times 1.225 \times (251)^2 \times 534 \times 0.0866$$

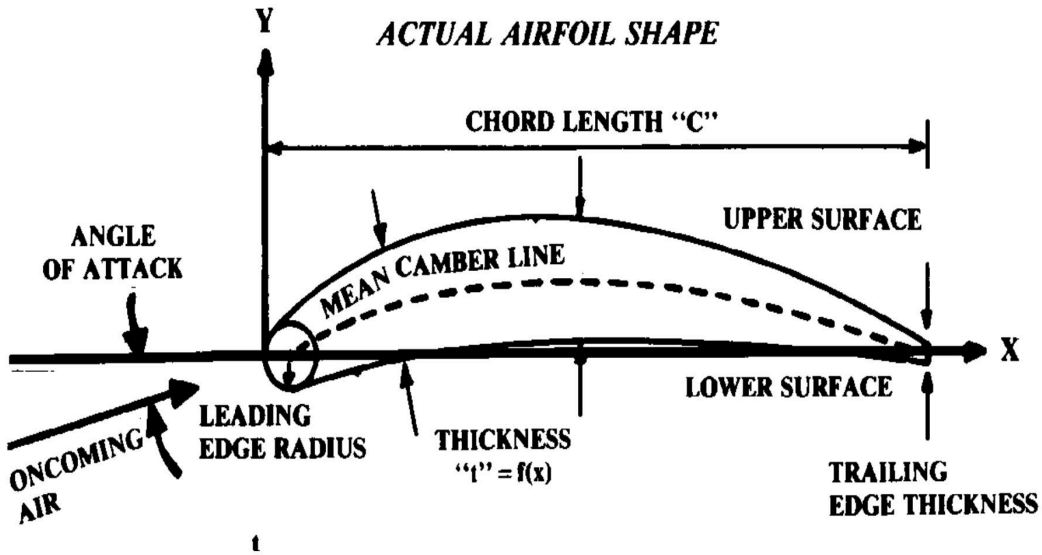
D = 1784.866 KN

7.4 AIRFOIL SELECTION:

7.4.1 INTRODUCTION:

After the estimation of weight, a compatible airfoil for the type of aircraft that would be able to generate enough lift to overcome the weight efficiently has to be selected. First, a study of the Existing airfoils is suggested before proceeding to airfoil selection. NACA provides a wide variety of airfoils, each having its own different merits and demerits.

FIG: 7.4.1 SCHEMATIC DIAGRAM OF AIRFOIL



7.4.2 CLASSIFICATION OF AEROFOIL

BASED ON SHAPE:

- Symmetrical Airfoil
- Cambered Airfoil
- Semi-Symmetrical Airfoil
- Modified Flat Bottom
- Under-Cambered
- Reflexed Airfoil-Convex Concave

BASED ON NACA NOMENCLATURE

- 4- digits series
- 5- digits series
- 6- digits series

BASED ON APPLICATION

- Subsonic airfoil-Cambered airfoil with curved top & bottom surface with sharp trailing edge.
- Supersonic airfoil-Sharp leading and trailing edge.

7.4.3 AIRFOIL TERMINOLOGIES

1) Mean camber line
2) Chord line
3) Chord
4) Max thickness
5) Aerodynamics center
6) Center of pressure

MEAN CAMBER LINE:

A line joining leading edge and trailing edge from upper and lower surface is called as mean camber line.

CHORD LINE:

It refers to the imaginary straight line joining the leading and trailing edges of the airfoil.

CHORD:

It refers to the imaginary straight line joining the leading and trailing edges of the airfoil.

MAXIMUM THICKNESS:

Thickness close to the leading edge is called maximum thickness.

AERODYNAMIC CENTER:

It is point at which the pitching moment co efficient doesn't vary with lift co efficient

CENTER OF PRESSURE:

The centre of pressure is the point where the total sum of a pressure field acts on a body, causing a force to act through that point.

7.4.4. NACA AIRFOILS:

NACA FOUR DIGIT SERIES :

Around 1932, NACA tested a series of airfoil shapes known as the four-digit sections. The four-digit airfoil geometry is defined, as the name implies, by four digits; the first gives the maximum camber in percent of chord, the second the location of the maximum camber in tenths of chord, and the last two the maximum thickness in percent of chord. For Example, the 2412 airfoil is a

12% thick airfoil having a 2% camber located 0.4 from the leading edge.

ADVANTAGES:

- Good stall characteristics
- Small centre of pressure movement across large speed range
- Roughness has little effect

DISADVANTAGES:

- Low C_{Lmax}
- Relatively high drag
- High pitching moment

NACA FIVE DIGIT SERIES :

The NACA five-digit series developed around 1935 uses the same thickness distribution as the four-digit series.

The numbering system for the five-digit series is not as straightforward as for the four-digit series. The first digit multiplied by 3/2 gives the design lift coefficient of the airfoil. The next two digits are twice the position of maximum camber in percent of chord. The last two digits give the percent thickness. For Example, the 23012 airfoil is a 12% thick airfoil having a design C_l of 0.3 and a maximum camber located 15% of c back from the leading edge.

ADVANTAGES:

- Higher C_{Lmax}
- Low pitching moment
- Roughness has little effect

DISADVANTAGES:

- Poor stall behaviour
- Relatively high drag

NACA 1 SERIES (SERIES 16):

The NACA 1-series of wing sections developed around 1939 was the first series based on theoretical considerations. The most commonly used 1-series airfoils have the minimum pressure located at the 0.6 and are referred to as series-16 airfoils. The camber line for these airfoils is designed to produce a uniform chord wise pressure difference across it. In the thin airfoil theory to follow, this corresponds to a constant chord wise distribution of vortices.

Thus the airfoil has been applied extensively to both marine and aircraft propellers. In the former application, Low-pressure regions are undesirable from the standpoint of cavitation (the formation of vaporous cavities in a flowing liquid). In the latter, the use of series-16 airfoils delays the onset of deleterious effects resulting from shock waves being formed locally in regions of high velocities.

Series-1 airfoils are also identified by five digits as, for Example, the NACA 16212 section. The first digit designates the series; the second digit designates the location of the minimum pressure in tenths of chord. Following the dash, the first number gives the design C_l in tenths. As for the other airfoils, the last two digits designate the maximum thickness in percent of chord.

ADVANTAGES:

- Avoids low pressure peaks.
- Low drag at high speed.

DISADVANTAGES:

- Relatively low lift.

NACA 6 SERIES:

The 6 series airfoils were designed to achieve desirable drag, compressibility, and C_l, performance. These requirements are somewhat conflicting, and it appears that the motivation for these airfoils was primarily the achievement of low drag. The chord wise pressure distribution resulting from the combination of thickness and camber is conducive to maintaining Extensive laminar flow over the leading portion of the airfoil over a limited range of G values. Outside of this range, C_d and C_{lm} values are not too much different from other airfoils.

The mean lines used with the 6-series airfoils have a uniform loading back to a distance of x/c = 2. Aft of this location the load decreases linearly.

There are many perturbations on the numbering system for the 6 series airfoils. The later series is identified, for Example, as

NACA 65_1-212 a = 0.6

Here 6 denotes the series; the numeral 5 is the location of the minimum pressure in tenths of chord for the basic thickness and distribution; and the subscript 1 indicates that low drag is maintained at C_l, values of 0.1 above and below the design C_l, of the 0.2, denoted by the 2 following the dash. Again, the last two digits specify the percentage thickness. If the fraction, a, is not specified, it is understood to equal unity.

ADVANTAGES:

- High C_{Lmax}
- Very low drag over a small range of operating conditions
- Optimized for high speed

DISADVANTAGES:

- High drag outside of optimum range of operating conditions
- Higher pitching moment
- Poor stall behaviour
- Very susceptible to roughness

CALCULATION:

WING ROOT CHORD: NACA63215

Maximum thickness in terms of chord is 0.3c Coefficient of lift is 0.2Minimum pressure point in terms of chord0.2. 6-denotes the series.

Root Chord length $= \frac{2s}{b(1+\lambda)}$

Taper ratio for wing, $\lambda = 0.3$

Root chord $= \frac{2\times219}{73.5(1+0.3)}$

$= 11.17$ m

WING TIP CHORD: NACA2410

Maximum thickness in terms of chord is 0.1c. Minimum pressure point in terms of chord 0.1c .

Tip chord = Root chord × λ = 4.583×0.3 = 3.351m.

Mean aerodynamic chord $= \frac{2}{3}$[root chord + tip chord - $\frac{(root\ chord \times tip\ chord)}{(root\ chord + tip\ chord)}$

$= \frac{2}{3}$ [11.47+3.351 - $(11.17 \times 3.351)/(11.17 + 3.351)$]

$= 6.12$

Mean aerodynamic chord = 6.12

FIG:7.4.2 AIRFOIL DIAGRAM NACA 63215

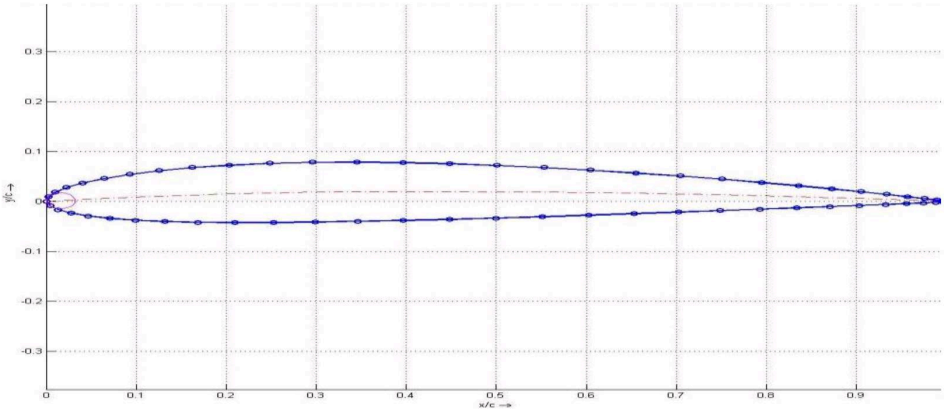


FIG:7.4.3 VELOCITY DISTRIBUTIONS FOR NACA63215

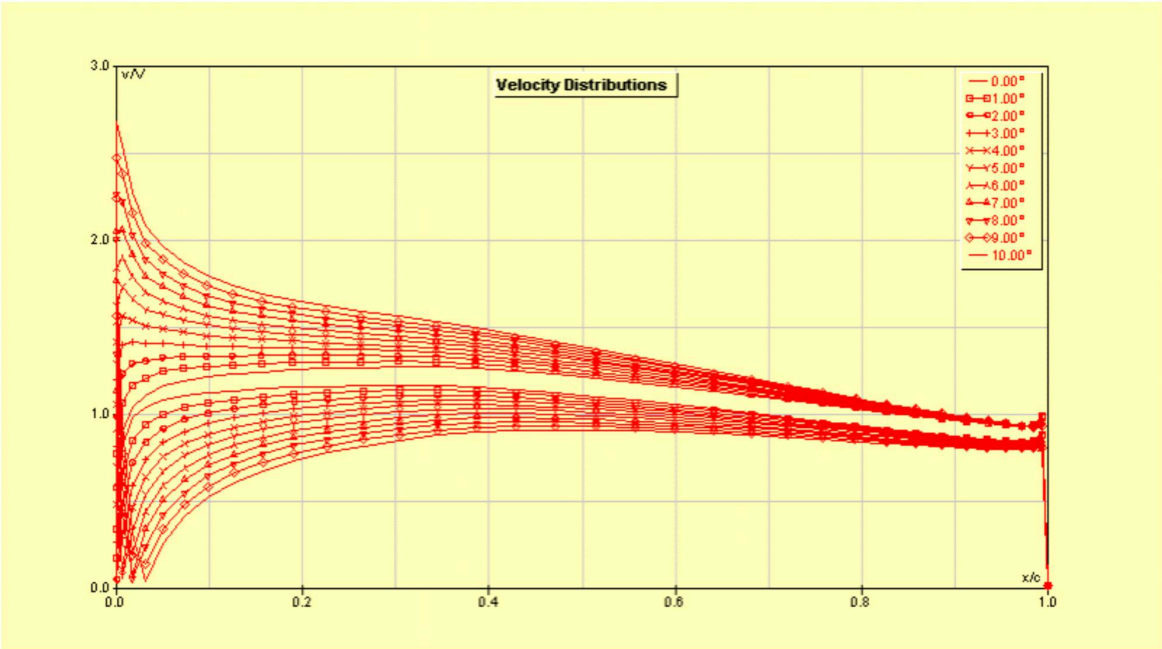


TABLE 7.4.1 VELOCITY DISTRIBUTION FOR NACA63215

A	Cl	Cd	Cm 0.25	Cp*	M cr.
[°]	[-]	[-]	[-]	[-]	[-]
0.0	0.121	0.01284	-0.015	-0.601	0.689

FIG:7.4.4 PRESSURE DISTRIBUTION FOR NACA63215

TABLE 7.4.2 PRESSURE DISTRIBUTION FOR NACA63215

A	Cl	Cd	Cm 0.25	Cp*	M cr.
[°]	[-]	[-]	[-]	[-]	[-]
0.0	0.121	0.01284	-0.015	-0.601	0.689

FIG:7.4.5 BOUNDARY LAYER COORDINATES FOR NACA63215

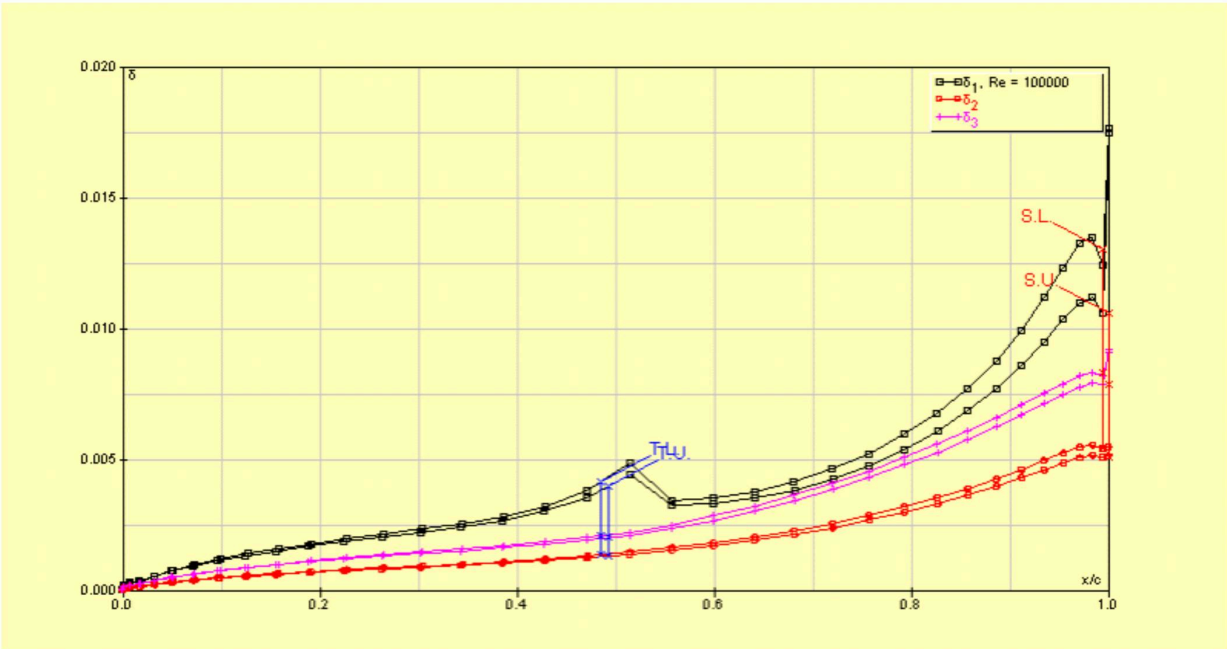
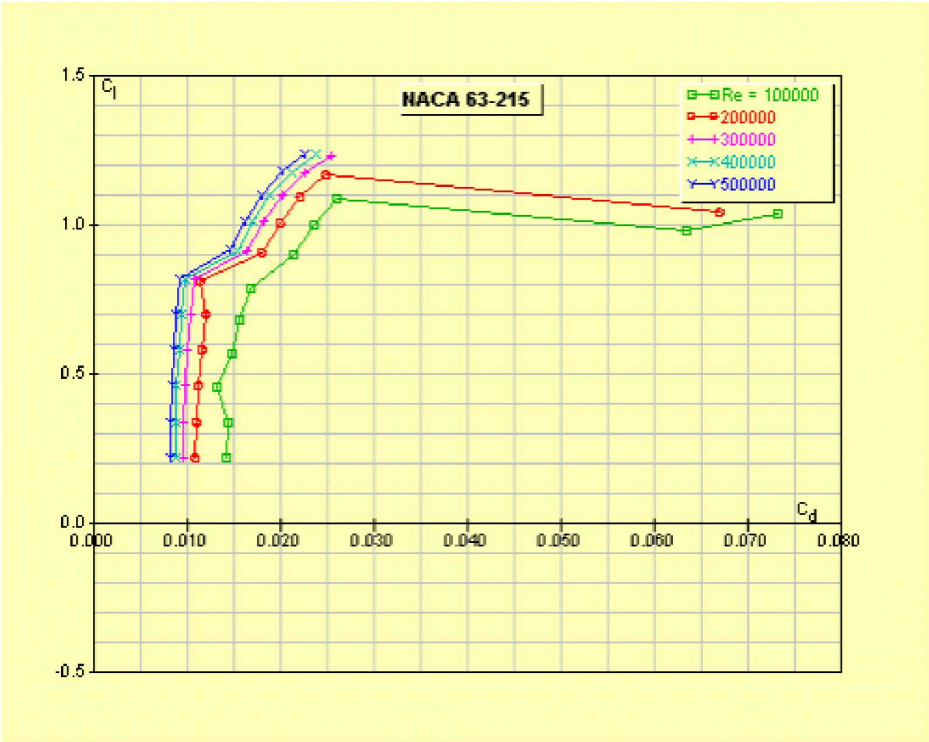


TABLE 7.4.3 BOUNDARY LAYER COORDINATES FOR NACA63215

A	Re	Mach	Λ	Cl	Cd	Cm 0.25
[°]	[-]	[-]	[-]	[-]	[-]	[-]
0.000	100000	0.000	∞	0.121	0.01284	-0.015

FIG:7.4.6 DRAG POLAR ESTIMATION FOR NACA63215



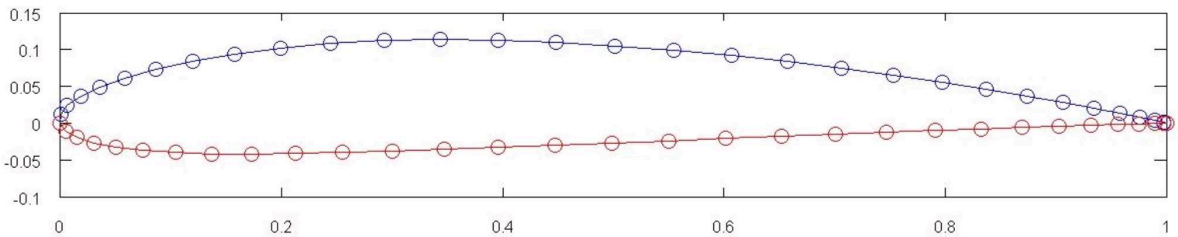


FIG:7.4.7 AIRFOIL DIAGRAM FOR NACA2410

TABLE 7.4.4 VELOCITY DISTRIBUTION FOR NACA 2410

A	Cl	Cd	Cm 0.25	Cp*	M cr.
[°]	[-]	[-]	[-]	[-]	[-]
0.0	0.346	0.01465	-0.068	-0.589	0.692

FIG:7.4.8 VELOCITY DISTRIBUTIONS FOR NACA2410

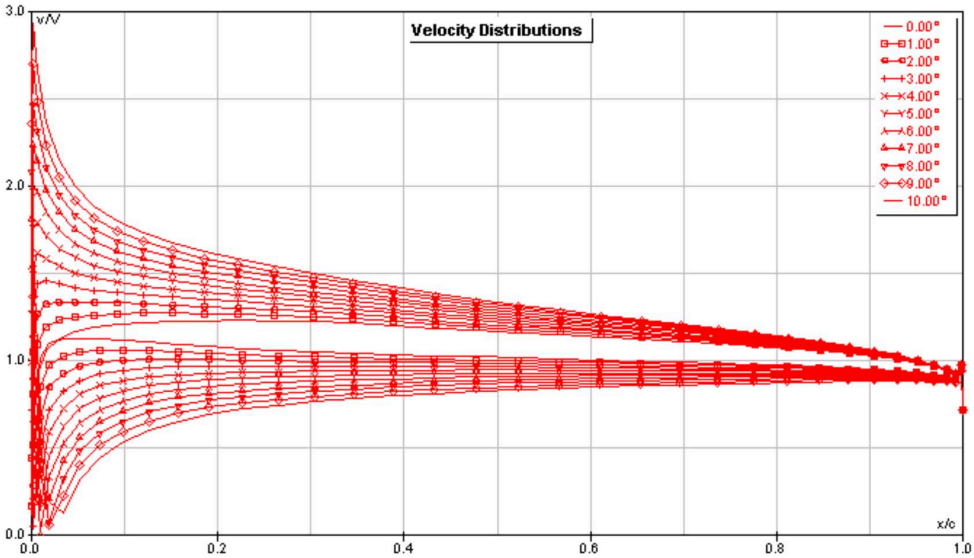


TABLE 7.4.5 PRESSURE DISTRIBUTION FOR NACA2410

A	Cl	Cd	Cm 0.25	Cp*	M cr.
[°]	[-]	[-]	[-]	[-]	[-]
0.0	0.346	0.01465	-0.068	-0.589	0.692

FIG:7.4.9 PRESSURE COEFFICIENT DISTRIBUTIONS FOR NACA2410

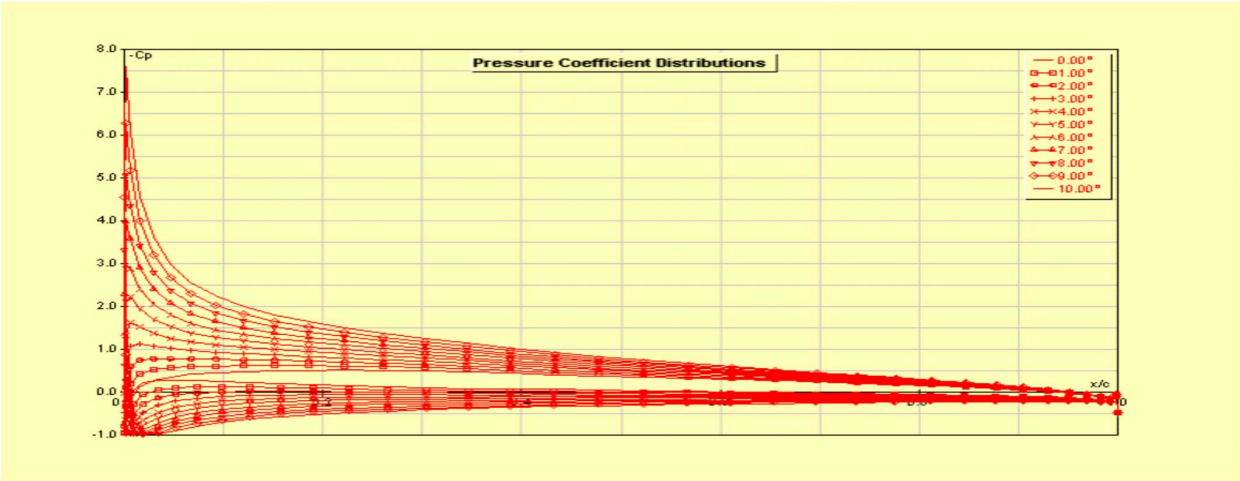


FIG:7.4.10 BOUNDARY LAYER COORDINATES FOR NACA2410

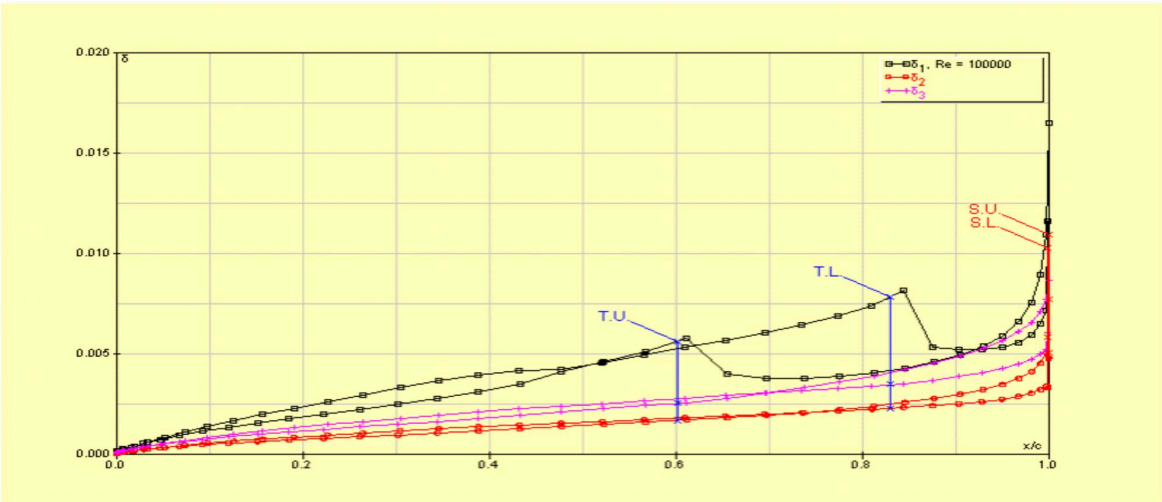


TABLE 7.4.6 BOUNDARY LAYER COORDINATES FOR NACA2410

A	Re	Mach	Λ	Cl	Cd	Cm
[°]	[-]	[-]	[-]	[-]	[-]	[-]
0.000	100000	0.000	∞	0.346	0.01465	-0.068

FIG:7.4.11 DRAG POLAR ESTIMATION FOR NACA2410

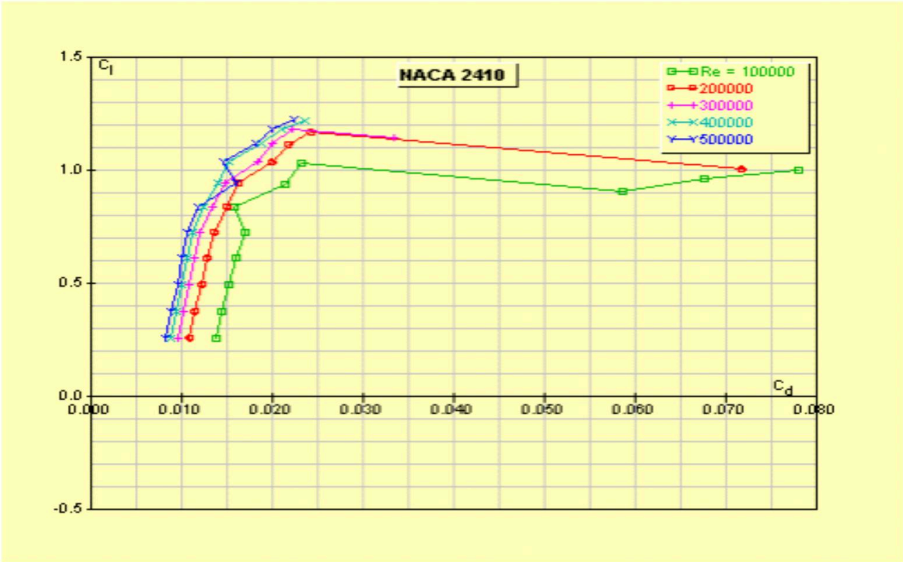


TABLE:7.5.1,7.5.2 ASPECT RATIO FOR HORIZONTAL AND VERTICAL TAIL

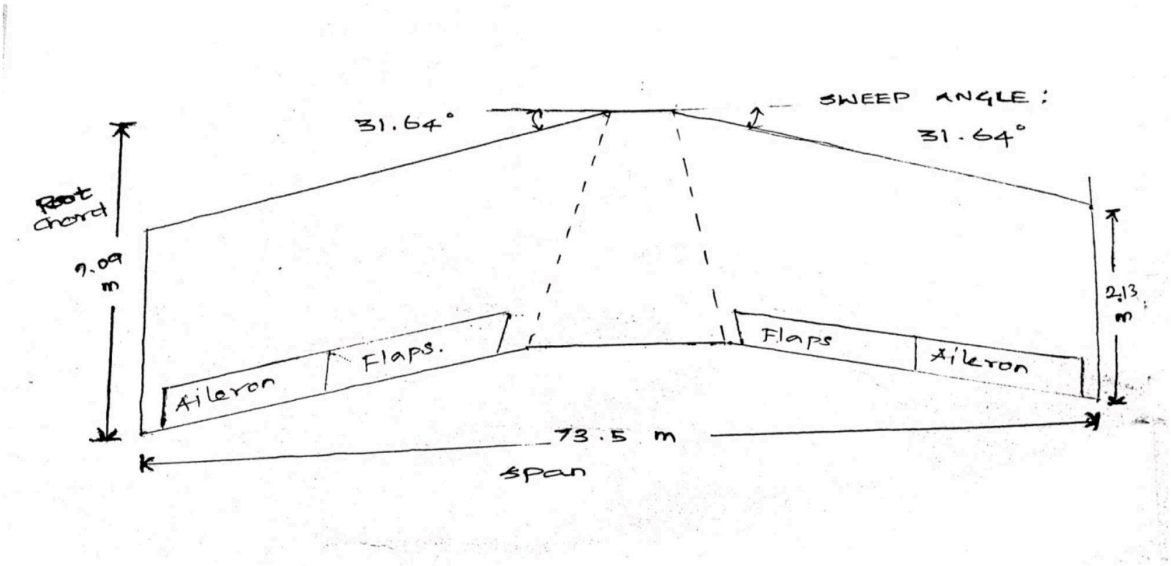
TYPES	HORIZONTAL TAIL		VERTICAL TAIL	
	ASPECT RATIO	λ TAPPER RATIO	ASPECT RATIO	λ TAPPER RATIO
FIGHTER	3 – 4	0.2 – 0.4	0.6 – 1.4	0.2 – 0.4
SAIL PLANE	6 – 10	0.3 – 0.5	1.5 – 2	0.4 – 0.6
OTHERS	3 – 5	0.3 – 0.6	1.3 – 2	0.3 – 0.6

7.5 SELECTION OF CONTROL SURFACE

TABLE:7.5.1 ΔC_L Condition for high lift devices:

S.NO	HIGH LIFT DEVICES	ΔC_L
1	Plain Flap	0.7 – 0.9
2	Split Flap	0.7 – 0.9
3	Fowler Flap	1 – 1.3
4	Slotted Flap	1.3 cf/c
5	Double slotted Flap	1.9 cf/c
6	Triple slotted Flap	1.9 cf/c
7	Loading Edge	0.2 – 0.3
8	Loading edge slot	0.3 – 0.4
9	Kruger Flaps	0.3 – 0.4

7.5.1 CONTROL SURFACE SELECTION IN WING:



FLAT TYPES: Double Slotted Flap

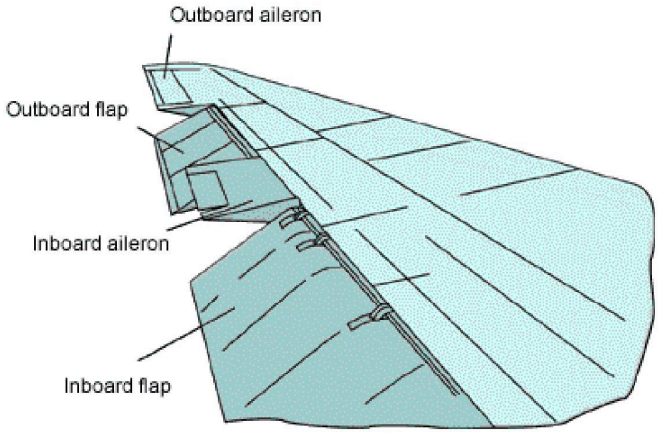
ΔC_L : 1.6 Cf/C

AILERON TYPE: Standard type effective aileron

AILERON:

It is located on the ends of the wings on the straight edge. The main purpose of the ailerons is to roll the plane, which helps it turn. When the left aileron goes down, right one up causes roll to the right.

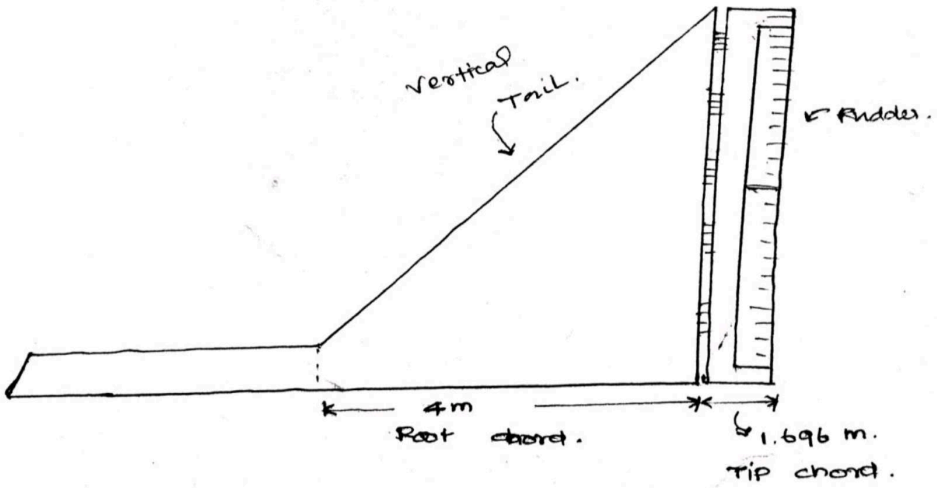
FIG:7.5.1, AILERON



FLAPS:

Flaps are important for landing. They go down to help the plane slow down. They also go up to about degrees when taking off for extra lift, so they have time to recover if something goes wrong.

7.5.2 CONTROL SURFACE SELECTION FOR VERTICAL TAIL:

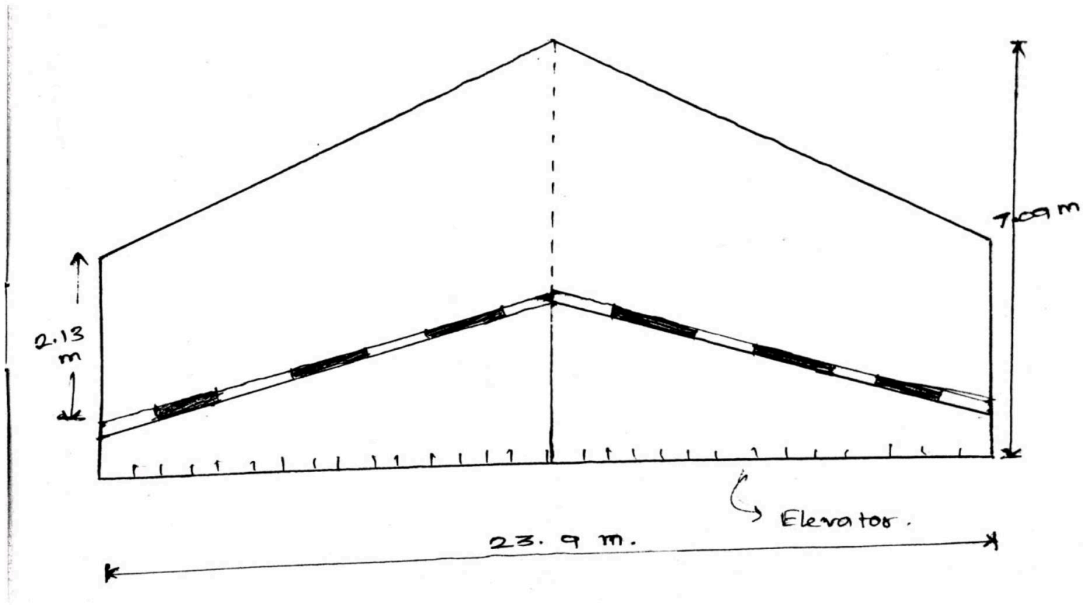


TAPPER RATIO: 0.290

RUDDER TYPE: Plain flap rudder

ΔC_L: 0.7 – 0.9

7.5.3 CONTROL SURFACE SELECTION FOR HORIZONTAL TAIL:



TAPPER RATIO: 0.300

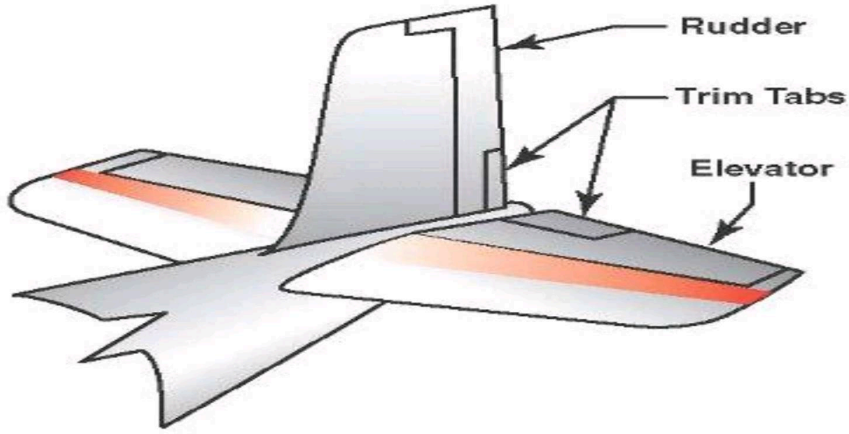
ELEVATOR TYPE: Plain flap elevator

ΔC_L: 0.7 – 0.9

RUDDER:

A rudder is a primary control surface used to steer a aircraft. It is attached to a vertical stabilizer, located at the rear of an airplane and used along with the ailerons, to turn the airplane. It is primarily used to encounter the adverse yaw effect and p-factor.

FIG:7.5.2,RUDDER



TRIM TABS:

It is small, secondary that are attached to the trailing edge of a larger, primary control surface such as an elevator.

CHAPTER 08

LANDING GEAR SELECTION

8.1 SELECTION OF LANDING GEARS:

Type of Landing gear : Retractable Landing gear.

1. Landing gear positioning

   ➔ Tricycle retractable landing gear

2. No. of axel in nose landing gear

   ➔ 2 (two)

3. No. of axel in rear landing gear

   ➔ 8 (eight)

4. Total no. of axel in both nose and rear landing gear:

   ➔ 10 (ten)

TABLE: 8.1.1 APPROXIMATE VALUES FOR LANDING GEARS:

DIAMETER			WIDTH	
TYPE	A	B	A	B
GENERAL AVIATION	1.51	0.349.	0.7150	0.312
TWIN	2.69	0.251	1.170	0.216
TRANSPORT/BOMBER	1.63	0.315	0.1043	0.480
TRANSPORT/TRAINER	1.59	0.302	0.0980	0.467

8.2 RETRACTABLE LANDING GEAR:

FIG:8.2.1 RETRACTABLE LANDING GEAR

Fixed and Retractable Landing Gear Further classification of aircraft landing gear can be made into two categories: fixed and retractable. Many small, single engine light aircraft have fixed landing gear, as do a few light twins. This means the gear is attached to the airframe and remains exposed to the slipstream as the aircraft is flown. As discussed in Chapter 2 of this handbook, as the speed of an aircraft increases, so does parasite drag. Mechanisms to retract and stow the landing gear to eliminate parasite drag add weight to the aircraft. On slow aircraft, the penalty of this added weight is not overcome by the reduction of drag, so fixed gear is used. As the speed of the aircraft increases, the drag caused by the landing gear becomes greater and a means to retract the gear to eliminate parasite drag is required, despite the weight of the mechanism.

A great deal of the parasite drag caused by light aircraft landing gear can be reduced by building gear as aerodynamically as possible and by adding fairings or wheel pants to streamline the airflow past the protruding assemblies. A small, smooth profile to the oncoming wind greatly reduces landing gear parasite drag. Figure 13-11 illustrates a Cessna aircraft landing gear used on many of the manufacturer's light planes. The thin cross section of the spring steel struts combines with the fairings over the wheel and brake assemblies to raise performance of the fixed landing gear by keeping parasite Retractable landing gear stow in fuselage or wing compartments while in flight. Once in these wheel wells, gear is out of the slipstream and do not cause parasite drag. Most retractable gear

have a close fitting panel attached to them that fairs with the aircraft skin when the gear is fully retracted. [Figure 13-12] Other aircraft have separate doors that open, allowing the gear to enter or leave, and then close again.

NOTE: The parasite drag caused by extended landing gear can be used by the pilot to slow the aircraft. The extension and retraction of most landing gear is usually accomplished with hydraulics.

8.3 TRICYCLE RETRACTABLE LANDING GEAR:

The most commonly used landing gear arrangement is the tricycle-type landing gear. It is comprised of main gear and nose gear.

Tricycle-type landing gear is used on large and small aircraft with the following benefits:

1. Allows more forceful application of the brakes without nosing over when braking, which enables higher landing speeds.

2. Provides better visibility from the flight deck, especially during landing and ground maneuvering.

3. Prevents ground-looping of the aircraft. Since the aircraft center of gravity is forward of the main gear, forces acting on the center of gravity tend to keep the aircraft moving forward rather than looping, such as with a tail wheel-type landing gear.

The nose gear of a few aircraft with tricycle-type landing gear is not controllable. It simply casters as steering is accomplished with differential braking during taxi. Heavy aircraft typically utilize hydraulic power to steer the nose gear. Control is achieved through an independent tiller in the flight deck.

The main gear on a tricycle-type landing gear arrangement is attached to reinforced wing structure or fuselage structure. The number and location of wheels on the main gear vary. Many main gear have two or more wheels.

Multiple wheels spread the weight of the aircraft over a larger area. They also provide a safety margin should one tire fail. Heavy aircraft may use four or more wheel assemblies on each main gear. When more than two wheels are attached to a landing gear strut, the attaching mechanism is known as a bogie

CHAPTER 09
PERFORMANCE CALCULATIONS

9.1 MINIMUM RADIUS OF TURN AT SEA LEVEL:

R_{MIN} at sea level = $\frac{4K(W/S)}{g\rho_{\infty(T/W)}\sqrt{1-4KC_{D_0}(T/W)^2}}$

Where, K = $\frac{1}{\pi e \times AR}$

K = $\frac{1}{\pi \times 0.84 \times 9.39}$ = 0.04

R_{MIN} at sea level = $\frac{(4\times0.04\times287.33)}{9.81\times1.225\times0.297\sqrt{1-4(0.04)(0.019)(0.297^2)}}$

R_{MIN} at sea level = 12.88 m

9.2 MINIMUM RADIUS OF TURN AT MAXIMUM ALTITUDE:

R_{MIN} at max altitutde = $\frac{4\times0.04\times(287.33)}{1.195\sqrt{0.99973}}$

R_{MIN} at max altitutde = 38.48 m/s

9.3 LOAD FACTOR FOR MINIMUM RADIUS OF TURN:

$$n_{MIN} = \sqrt{2 - \frac{4KC_{D_0}}{(T/W)^2}}$$

$$n_{MIN} = \sqrt{2 - \frac{4(0.04)(0.019)}{(0.297)^2}} = 1.402$$

.4 MAXIMUM TURN RATE AT SEA LEVEL ,

$$w_{MAX} = q_\infty\sqrt{\frac{\rho_\infty}{(W/S)}\left[\frac{T}{W}/2K\right] - (C_{D_0}/K)^{\frac{1}{2}}}$$

$$w_{MAX} = (0.5\times1.225\times256^2)\sqrt{\frac{1.225}{287.33}\left(\frac{0.297}{2\times0.04}\right) - (\frac{0.019}{0.04})^{0.5}}$$

$$(w_{MAX})_{SEA} = 2689.433\ (rad/sec)$$

9.5 MAXIMUM TURN RATE AT MAX.OPERATING ALTITUDE:

$$(w_{MAX})_{ALTITUDE} = q_\infty \sqrt{\frac{\rho_\infty}{(W/S)}\left[\frac{T}{W}/2K\right] - (C_{D_0}/K)^{\frac{1}{2}}}$$

$$w_{MAX} = (0.5\times0.410\times256^2)\sqrt{7.355\times10^{-4}\times1.81}$$

$$(w_{MAX})_{ALTITUDE} = 283.72\ (rad/sec)$$

9.6 LOAD FACTOR FOR MAXIMUM TURN RATE:

$$n_{MAX} = [(T/W)/\sqrt{KC_{D_0}} - 1]$$

$$n_{MAX} = [(0.297)/\sqrt{(0.04)(0.019)} - 1]$$

$$n_{MAX} = 9.77$$

9.7 PULL UP MANEUVER :

$$R = \frac{V_\infty^2}{g(n-1)}$$

n = take off load factor

V_∞ = take off velocity

$$V_\infty = \sqrt{\frac{2W}{\rho_\infty S(C_L)_{CLIMB}}}$$

$$V_\infty = \sqrt{\frac{2\times466773.7}{1.225\times534\times1.625}}$$

$$V_\infty = 29.63\ (m/s)$$

n = 2.55 (from v-n diagram)

$$R = \frac{(29.63^2)}{9.81(2.55-1)}$$

R = 57.73 m

9.8 PULL DOWN MANEUVER:

$$V_\infty = \sqrt{\frac{2W}{\rho_\infty S C_L}}$$

$$V_\infty = \sqrt{\frac{2\times358313.99}{1.225\times534\times0.410}}$$

$$V_\infty = 51.69\ (m/s)$$

n = 2.64 (from v-n diagram)

$$R = \frac{(51.69^2)}{9.81(3.64)}$$

R = 74.82 m

9.9 TAKE OFF DISTANCE:

$$\text{Takeoff Parameter} = \frac{(W_1/S)}{\rho_\infty (C_L)_{TAKE\ OFF}\ (T/W)}$$

$$= \frac{(\frac{466773.7}{534})}{1.225\times1.65\times0.297}$$

Takeoff Parameter = 1456.07 m

9.10 LANDING DISTANCE:

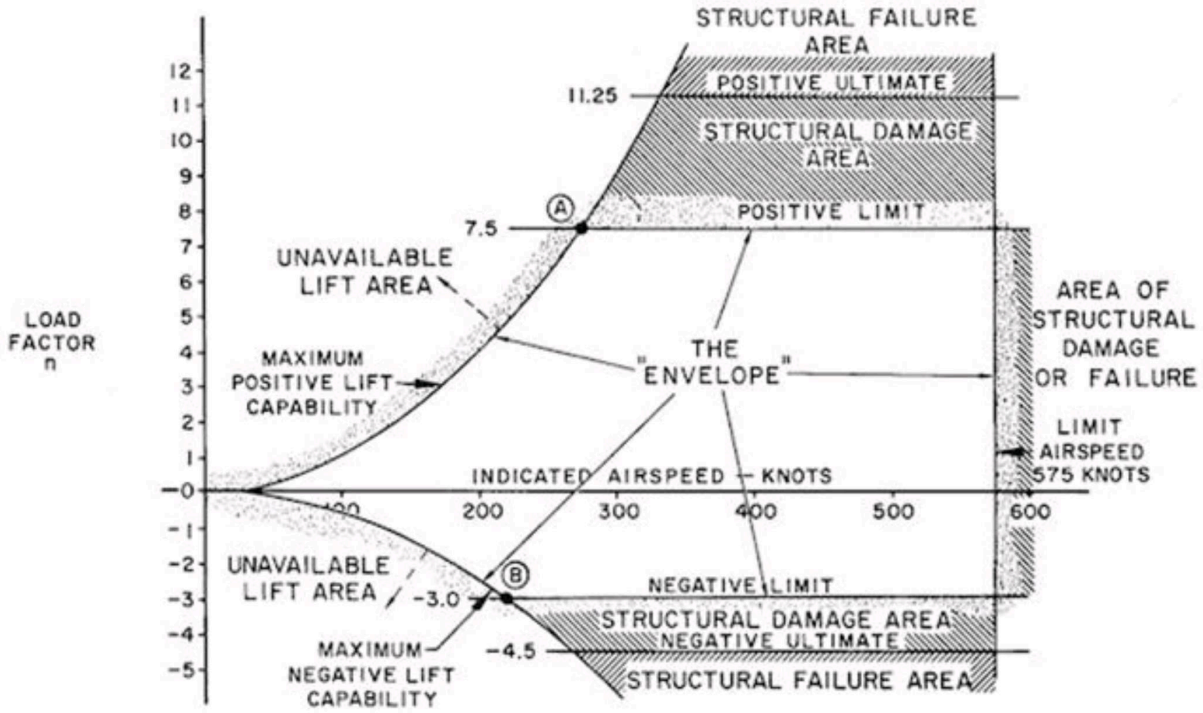
$$S_{LAND} = 80(W/S)\frac{1}{\rho_\infty (C_L)_{MAX}} + 1000\text{ft}$$

$$= 80(612.25)(1/1.225 \times 1.3) + 1000\text{ft}$$

Landing Parameter = 1237.56 m

CHAPTER 10
V-N DIAGRAM

10.1 INTRODUCTION:



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

LOAD FACTOR (V-n) DIAGRAM:

The control of weight in aircraft design is of extreme importance. Increases in weight require stronger structures to support them, which in turn lead to further increases in weight and so on. Excess of structural weight mean lesser amounts of payload, thereby affecting the economic viability of the aircraft.

The aircraft designer is therefore constantly seeking to pare his aircraft‟s weight to the minimum compatible with safety. However, to ensure general minimum standards of strength and safety, airworthiness regulations (Av.P.970 and BCAR) lay down several factors which the primary structure of the aircraft must satisfy.

LIMIT LOAD FACTOR:

- It is the maximum load that the aircraft is expected to experience in normal operation.
- It load factor is less than limit load factor the structure may deflect and return to its original position when the load factor = 1
- It the load factor > limit load factor structure will have a permanent deformation.

PROOF LOAD:

It is the product of the limit load and the proof factor (1.0- 1.25)

ULTIMATE LOAD:

- It is the product of the limit load and the ultimate factor (usually 1.5). The aircrafts structure must withstand the proof load without detrimental distortion and should not fail until the ultimate load has been achieved.
- If the load factor > ultimate load factor, the a/c will experience structural failure.

TABLE:10.1.1 LOAD FACOR FOR VARIOUS TYPES OF AIRCRAFT:

AIRCRAFT TYPE	+VE LOAD FACTOR	-VE FACTOR
Fighter	6.5 – 9	-3 to 6
General aircraft	2.5 – 3.8	-1 to -1.5
Acrobatic	6	3

LOAD FACTOR:

The load factor is defined as rate of lift to weight.

$$n = \frac{L}{W} = \frac{L}{\frac{1}{2}\rho_\infty V_\infty^2 \, SC_L}$$

$$n = \frac{2W_0}{\rho_\infty V_\infty^2 \, SC_L}$$

$$V_\infty = \sqrt{\frac{2W_0}{\rho_\infty V_\infty^2 \, SC_L}}$$

The main velocities that are plotted in Vn diagram are

1. 1 g stall velocity
2. Design manuvering velocity
3. Design cruise velocity
4. Design Dive velocity

1_g STALL VELOCITY:

Stall velocity = $\sqrt{\frac{2W}{\rho C_n S}}$

$$C_n = \sqrt{C_L^2 + C_d^2}$$

$(C_n)_{climb} = \sqrt{1.625^2 + 0.124^2} = 1.63$

$(C_n)_{cruise} = \sqrt{0.027^2 + 0.019^2} = 0.033$

$(C_n)_{landing} = \sqrt{1.3^2 + 0.0866^2} = 1.303$

- Stall velocity at climb = $\sqrt{\frac{2W_2}{\rho_\infty C_n S}}$ $=\sqrt{\frac{2\times459772}{1.225\times1.63\times534}}$ = 29.37 m/s

- Stall velocity at cruise = $\sqrt{\frac{2W_3}{\rho_\infty C_n S}}$ $=\sqrt{\frac{2\times358313.99}{0.410\times0.033\times534}}$ = 314.94 m/s

- Stall velocity at landing = $\sqrt{\frac{2W_6}{\rho_\infty C_n S}}$ $=\sqrt{\frac{2\times326943.27}{1.225\times1.303\times534}}$ = 27.68 m/s

DESIGN MANUVERING VELOCITY:

$$V_A \geq V_S \sqrt{(n)_{+VE\ LIMIT}}$$

$$(n)_{+VE\ LIMIT} = 2.1 + \frac{W_0}{W_{2,3,6}+W_6}$$

$$V_S = \sqrt{\frac{2W}{\rho S C_L}}$$

- V_S at climb $= \sqrt{\frac{2\times459772}{1.225\times534\times1.625}}$ = 29.41 m/s

- V_S at cruise $= \sqrt{\frac{2\times358313.99}{0.410\times534\times0.027}}$ = 348.18 m/s

- V_S at landing = $\sqrt{\frac{2\times326943.27}{1.225\times534\times1.3}}$ = 27.73 m/s

- $V_\infty = \sqrt{\frac{2\times481210}{1.225\times534\times0.410}}$ = 59.903 m/s

- n = $\frac{2\times481210}{1.225\times0.410\times534\times59.903^2}$ = 1

- $(n)_{+VE\ LIMIT}$ at climb = 2.1+$\frac{481210}{459772+326943.27}$ = 2.712

- $(n)_{+VE\ LIMIT}$ at cruise = 2.1+$\frac{481210}{358313.99+326943.27}$ = 2.802

- $(n)_{+VE\ LIMIT}$ at landing = 2.1+$\frac{481210}{326943.27+326943.27}$ = 2.836

- V_A at climb $= 29.41\sqrt{2.712} = 48.43\ m/s$
- V_A at cruise $= 348.18\sqrt{2.802} = 582.825\ m/s$
- V_A at landing $= 27.73\sqrt{2.836} = 46.699\ m/s$

DESIGN CRUISE SPEED:

$$V_C \geq K_C\sqrt{\frac{W_{EMPTY}}{S}}$$

For supersonic aircraft , $K_C = 30$

$$V_C = 30\sqrt{\frac{134800}{534}} = 476.65\ m/s$$

$$(V_C)_{climb} = 30\sqrt{\frac{459772}{534}} = 880.28\ m/s$$

$$(V_C)_{cruise} = 30\sqrt{\frac{358313.99}{534}} = 777.11\ m/s$$

$$(V_C)_{landing} = 30\sqrt{\frac{326943.27}{534}} = 742.31\ m/s$$

DESIGN DIVING SPEED:

$$V_D \geq 1.05V_C$$

$$V_C = 1.05\times476.65$$

$$V_C = 500.48\ m/s$$

$(V_C)_{climb} = 1.05 \times 880.28 = 924.29\ m/s$

$(V_C)_{landing} = 1.05 \times 742.31 = 779.43\ m/s$

10.2 V-N DIAGRAMS FOR CLIMB, CRUISE,LANDING:

FIG: 6.1 SCHEMATIC V-n DIGRAM FOR CLIMB :

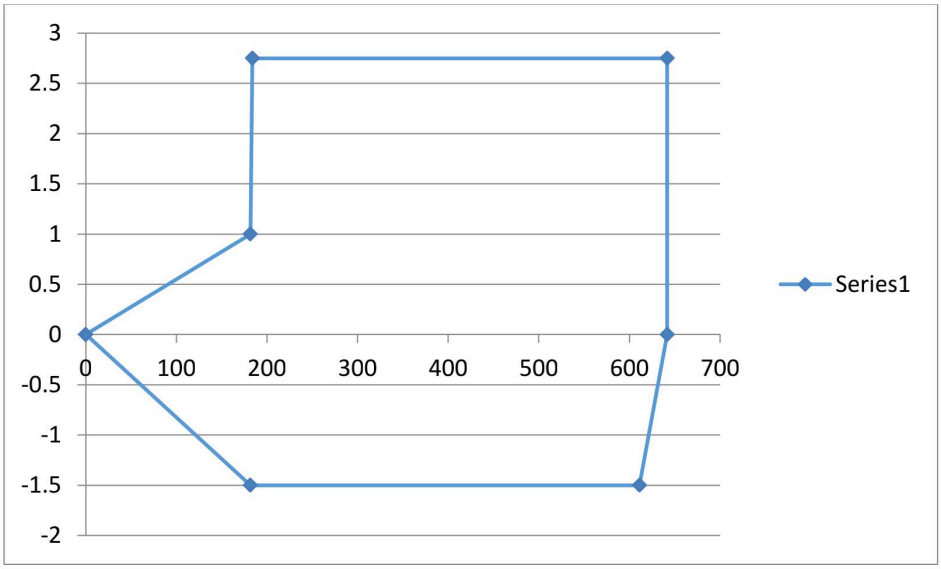


FIG:10.2 SCHEMATIC V-n DIGRAM FOR CRUISE ALTITUDE :

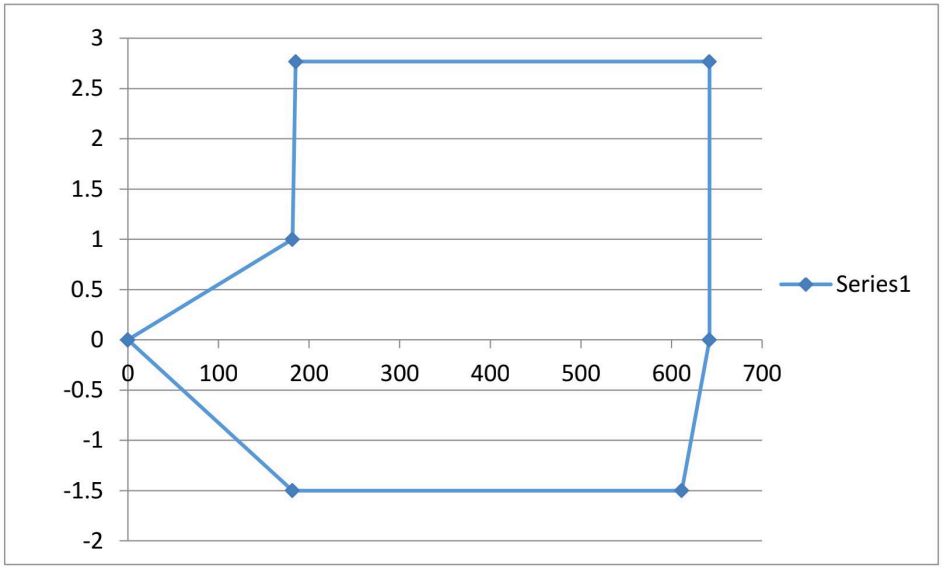
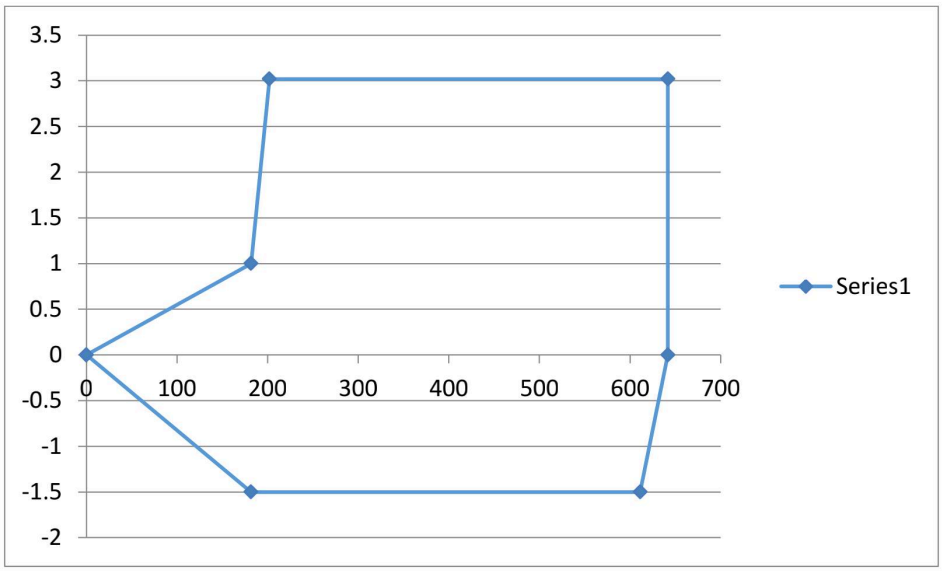


FIG:10.3 SCHEMATIC V-n DIGRAM FOR LANDING:



CHAPTER 11
THREE VIEW DIAGRAM OF AN AIRCRAFT

FIG:11.1 FRONT VIEW OF AN AIRCRAFT

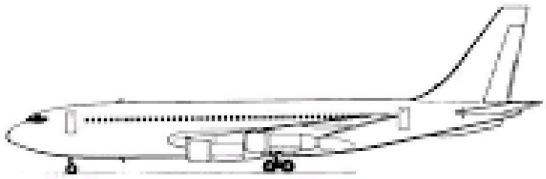


FIG:11.2 SIDE VIEW OF AN AIRCRAFT

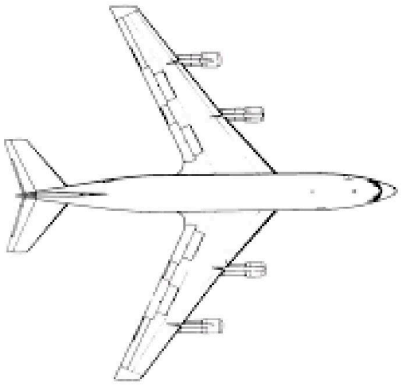


FIG:11.3 TOP VIEW OF AN AIRCRAFT

CHAPTER 12
CONCLUSION

The design of the long range passenger aircraft was done in a step by step method sticking to the basic rules. Calculations were performed with respect to data and formulae obtained from available design books. Some problems aroused during the process. However, they were solved by good teamwork. The conceptual design done in this project meets the initial requirements that were set by us.. It also helped us to understand some basic things about the aircraft and its design. We saw how the weight of an aircraft plays an important role in the design process. All the other following parameters vary with weight. The estimation of weight was a very crucial step and was also interesting. Then followed by the selection of an appropriate airfoil and estimating the lift that could be obtained using it. Then the estimation of some critical performance parameters to finalize the conceptual design process and selection of an appropriate powerplant. We would like to conclude saying that the experience that we got during the process will be helpful and the moments will always be remembered by us.

FUTURE WORK:

1. Preliminary design of an aircraft wing – Shrenck's curve, structural load distribution, shear force, bending moment and torque diagrams

2. Detailed design of an aircraft wing – Design of spars and stringers, bending stress and shear flow calculations – buckling analysis of wing panels

3. Preliminary design of an aircraft fuselage load distribution on an aircraft fuselage

4. Detailed design of an aircraft fuselage – design of bulkheads and longerons –

bending stress and shear flow calculations – buckling analysis of fuselage panels

5. Design of control surfaces - balancing and maneuvering loads on the tail plane and aileron, rudder loads

6. Design of wing-root attachment

7. Landing gear design

8. Preparation of a detailed design report with CAD drawings

CHAPTER 13

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