LECTURE NOTES

ON

CONCEPTUAL DESIGN OF FLIGHT VEHICLES

Prepared by

Ms. M Snigdha
Assistant Professor

Ms. G Swathi
Assistant Professor

Department of Aeronautical Engineering
INSTITUTE OF AERONAUTICAL ENGINEERING
(AUTONOMOUS)
Dundigal, Hyderabad - 500 043

UNIT 1
OVERVIEW OF THE DESIGN PROCESS, SIZING FROM A CONCEPTUAL SKETCH AIRFOIL GEOMETRY SELECTION, THRUST TO WEIGHT RATIO, WING LOADING

1.1 DESIGN:

Design is a creation of plan or convention for the construction of an object or a system as in architectural blueprints, engineering drawings, business processes, circuit diagrams and sewing patterns.

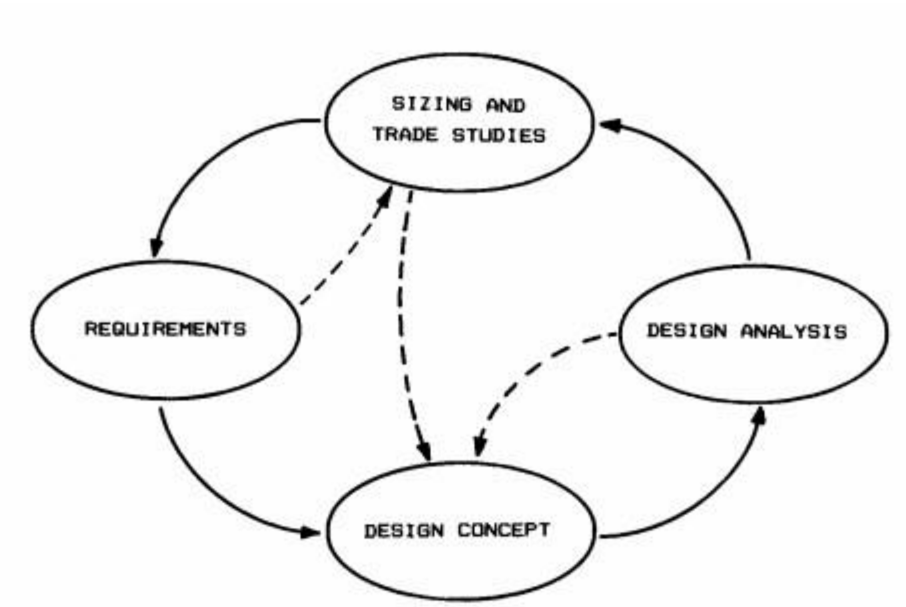


Fig 1.1 Design Wheel

1.2 DESIGN OVERVIEW:

The aircraft is essentially a payload compartment with wings. This focus was derived from early scoring analysis which identified system weight and aircraft loading time as the two key design parameters. Since the aircraft must be capable of carrying a variety of payload sizes and weights, the structure required to achieve that objective is potentially the heaviest element of the aircraft. As the design of the payload system also has a direct impact on loading time, preliminary design focused on the development of a fast and light weight payload system capable of meeting restraint requirements with the minimum aerodynamic features needed to complete lap requirements for the flight mission.

The design takes advantage of a high tensile strength fabric for the primary payload system structure. Individual fabric pockets are attached to a central carbon-fiber spar, eliminating the need for a structural payload bay floor. This innovative fabric payload system is enclosed by a sixty nine inch span, twin tractor, low wing monoplane with tricycle landing gear. The aircraft sits diagonally within the 4 ft x 5 ft plan form limits, maximizing aspect ratio and providing additional length for the fuselage fairing, thus maximizing aerodynamic efficiency.

The aircraft utilizes mold less, foam/fiberglass/carbon-fiber composite construction for the wing, tail and fuselage internal structure. As the external fuselage takes no structural loads, significant weight savings were achieved by vacuum-forming a thin, foam shell designed only for aerodynamic loads. The foam fuselage fairing has a full-length top hatch which, combined with a low-wing, and allows rapid access to the payload. This payload-focused configuration minimizes the key parameters of system weight and loading time through its structural efficiency and access to payloads, while providing sufficient aerodynamic performance and propulsive power density.

1.3 DESIGN OF AN AIRCRAFT STARTS OUT IN THREE PHASES:

1.3.1 CONCEPTUAL DESIGN PHASE:

The 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.

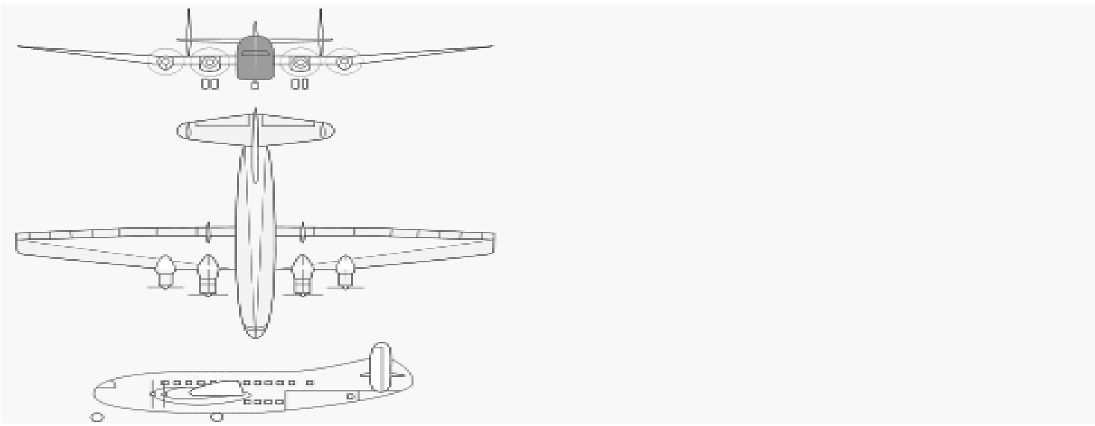


Fig 1.2 Conceptual Sketch

1.3.2 PRELIMINARY DESIGN PHASE:

The design configuration arrived at in the conceptual design phase is then tweaked and remodeled to fit into the design parameters. In this phase, wind tunnel testing and computational fluid dynamic 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.

1.3.3 DETAILED 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.

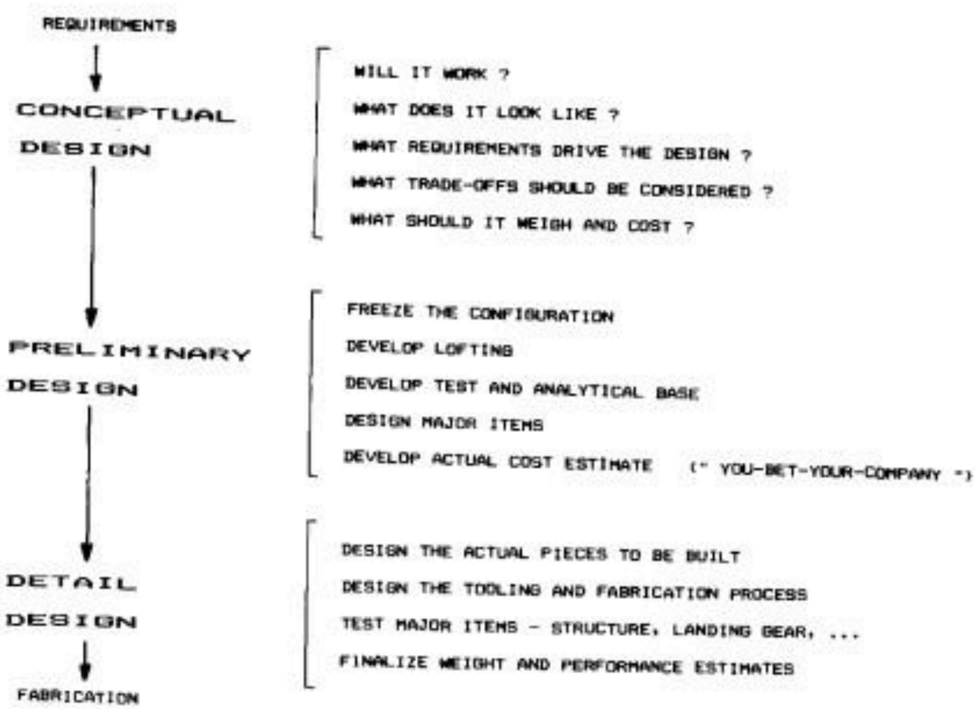


Fig 1.3 Three phases of aircraft design

1.4 CONCEPTUAL DESIGN:

1.4.1 MISSION REQUIREMENTS:

As a requirement, each participating team has to simulate the delivery of an emergency package (payload) to stranded crew of an Antarctic expedition on a blended wing/body radio controlled transport vehicle. The aircraft can be of any configuration of a blended wing/body and the wing aspect ratio must be greater than or equal to 2.0. The payloads to be used are four 400-gram Dick Smith's Chicken Gravy packets each with dimension 180mm x 130mm x 40mm.The ultimate aim is to achieve highest wing loading with best payload weight-to- take-off weight ratio.

1.4.2 TIME CONSIDERATION:

A total time allowance of 30 minutes is given to perform 3 flights, which does not include refueling. As soon as the payloads are given, timer starts. Within 30 minutes, the team needs to install the given payload, start the engine and complete an allocated circuit (approx. 600m per circuit) before landing and such process is to be repeated twice more.

In order to complete the mission as quickly as possible, the following points are taken into consideration:

1) Ease of payload accessibility
2) Height lift coefficient
3) Short take off and landing
4) Capability of rapid turn

1.5 PRELIMINARY DESIGN:

A careful look at the commercial transport jets in use, points out many common features amongst them. Some of these are:

1) Medium bypass turbofan

   This choice regarding the type of engine is due to the following reasons. 10 In the flight regime of Mach number between 0.6-0.85, turbofans give the best efficiency and moreover reduction in thrust output with speed is not rapid. Also, the noise generated by a medium bypass turbofan engine is considerably low. Following this trend a medium bypass turbofan is chosen as the power plant.

2) Wing mounted engines

   Though not a rule, wing mounted engines dominate the designs of top aircraft companies like Boeing and Airbus. Alternative designs could be adopted. But, given the experience gained with the wing mounted engines and the large data available a configuration with two wing mounted engines is adopted. Swept back wings and a conventional tail configuration is chosen. Again, this choice is dictated by the fact that a large amount of data (is available) for such configurations.

1.6 DESIGN PROJECT:

After the preliminary design has been approved by the manufacture / customer, the detailed design studies are carried out. These include the following.

1) Wind tunnel and structural testing on models based on the preliminary design. These test serve as a check on the correctness of the estimated characteristics and assessment of new concepts proposed in the design.
2) Mock-up: This is a full scale model of the airplane or its important sections.

This helps in

a) Efficient lay-out of structural components and equipments.
b) Checking the clearances, firing angles of guns, visibility etc. Currently this stage can be avoided by the use of CAD packages.

3) Complete wind tunnel testing of the approved configuration.

   Currently CFD (Computational Fluid Dynamics) plays an important role in reducing the number of test to be carried out.

4) Preparation of detailed drawings.
5) Final selection of power plant, c.g. calculations, performance & stability calculations.

6) Fabrication of prototypes. Generally six prototypes are constructed. Some of them are used for verifying structural integrity and functioning of various systems. Others are used for flight testing to evaluate performance and stability.
7) Pre-production manufacture and flight testing to ensure that the defects in the prototype (s) have been corrected.
8) Series production and flight testing to meet specified operational and airworthiness requirements.

1.7 CLASSIFICATION OF AIRPLANES:

Before discussing further about airplane design, it is helpful to know about different types of airplane. These are classified according to

a) Purpose
b) Configuration

1.7.1 CLASSIFICATION OF AIRPLANE ACCORDING TO PURPOSE:

There are two main types of airplanes viz. civil and military.

The category of civil airplanes includes passenger, cargo, agricultural, sports and ambulance. The category of military airplanes includes fighter, bomber, interceptor, reconnaissance, and aircraft for logistic support like troop-carriers and rescue aircraft. Military aircraft are often designed to cater to more than one role e.g. fighter bomber or interceptor fighter. The purpose of an airplane dictates its specifications. For example a passenger airplane should have,

a) High level of safety
b) High payload carrying capacity
c) Economy in operation
d) Comforts
e) Ability to fly in any weather and
f) Ability to use aerodromes of respective classes

A bomber should have,

a) Long range
b) High load carrying capacity
c) High speed
d) High endurance
e) High ceiling and
f) Adequate fire protection

An interceptor should have,

a) High rate of climb
b) High ceiling (3 to 4 km above contemporary bombers)
c) High speed
d) High maneuverability
e) Ability to fly in any weather and
f) Appropriate armament

1.7.2 CLASSIFICATION ACCORDING TO CONFIGURATION:

This classification is carried out according to the following features of the configuration.

a) Shape, number and position of wing
b) Type of fuselage
c) Location of horizontal tail
d) Location and number of engines

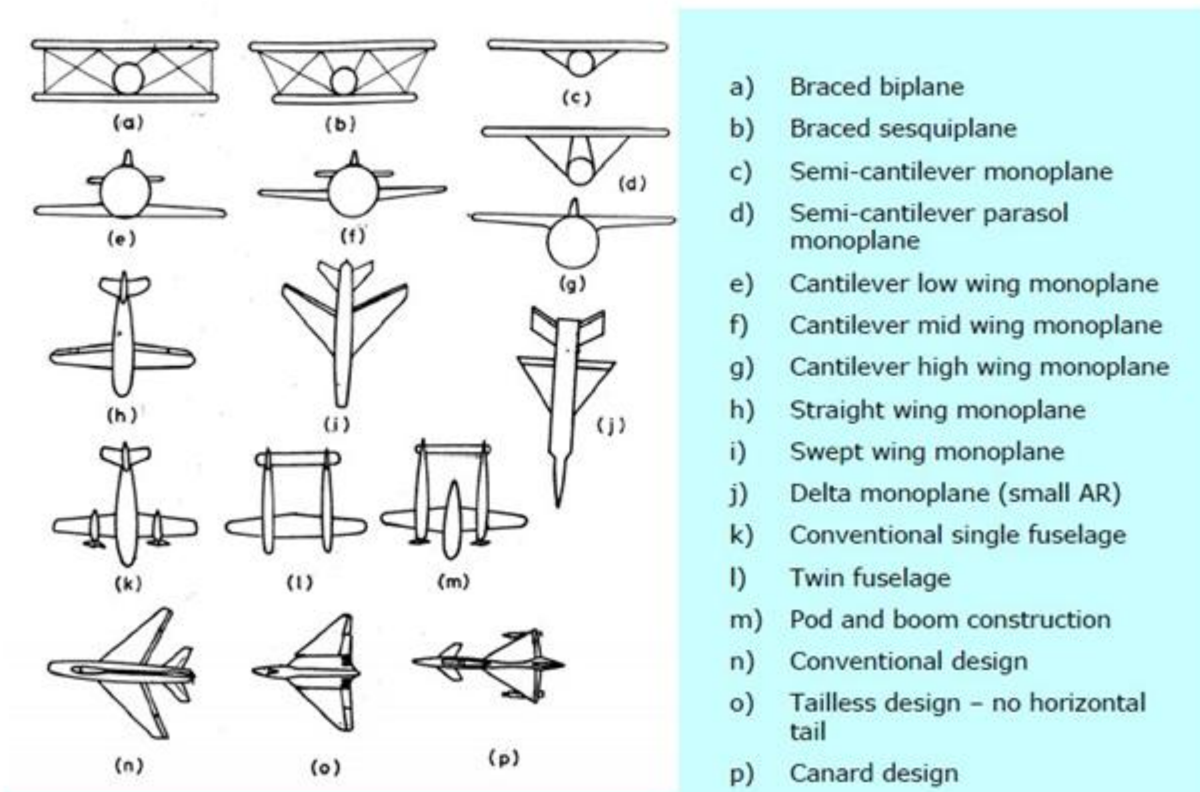


Fig 1.4 Types of airplane

1.8 STAGES IN PRELIMINARY DESIGN AS PART OF THIS COURSE:

The aim of this course is to enable the student to appreciate the aerodynamics aspects of preliminary design. The following topics are covered and illustrated through an example on preliminary design of a jet airplane

1) Data collection and preparation of preliminary three-view.
2) Weight estimate in three stages
   a) Based on data collection
   b) Refinement of fuel fraction
   c) Assessment of empty weight
3) Choice of wing loading and thrust loading
4) Engine selection and adjustment of wing loading
5) Wing design
6) Fuselage design
7) Location of engine(s)
8) Layout, weight balance & c.g. calculations

9) Determination of tail & control surface areas
10) Final drag estimation, performance calculations and preparation of brochure

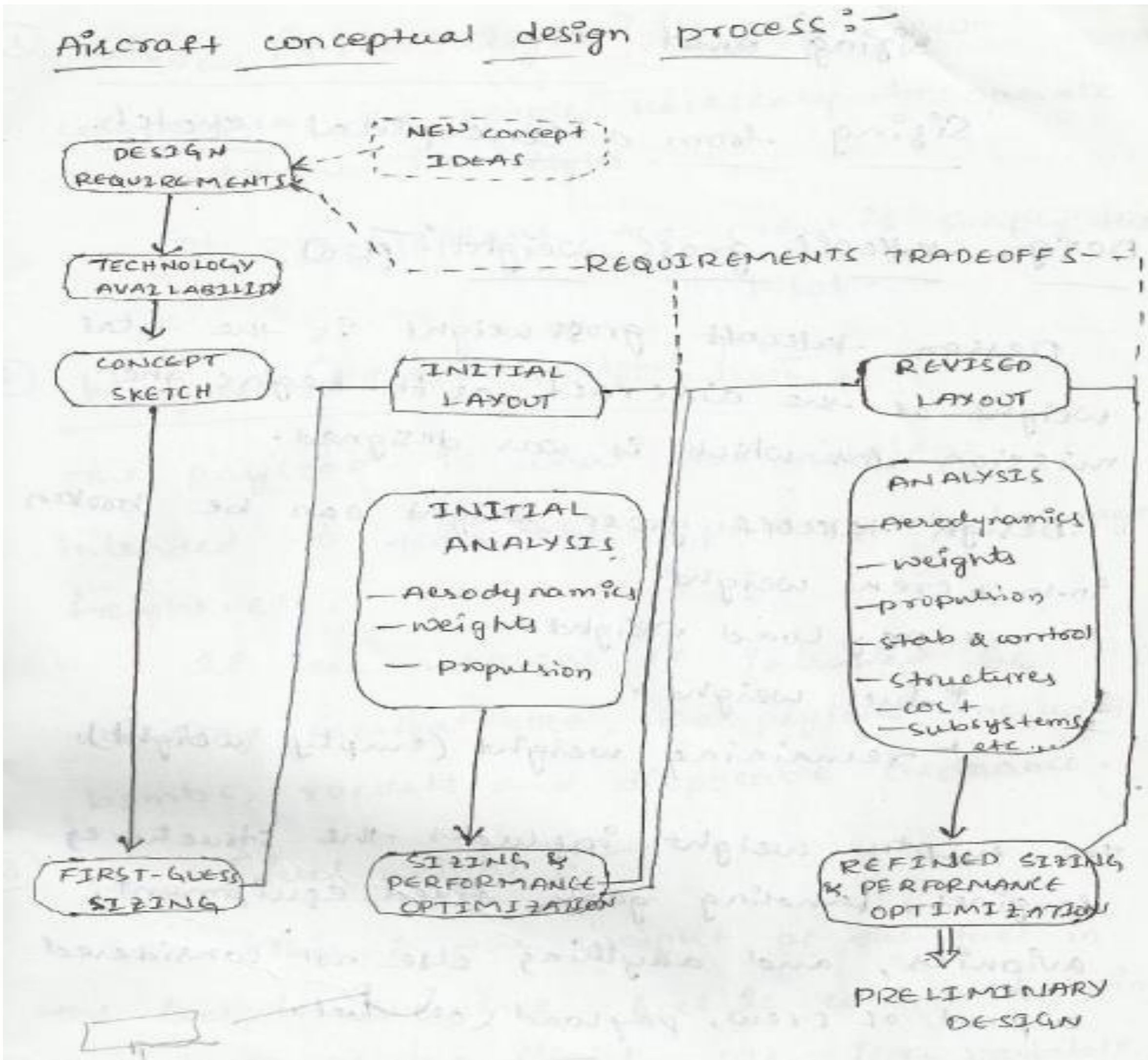


Fig 1.5 Aircraft Conceptual Design Process

1.9 EMPTY WEIGHT:

The last term in determining maximum take-off weight in equation 4.5 is the empty weight fraction ( ). At this moment (preliminary design phase), the aircraft has been design only conceptually, hence, there is no geometry or sizing. Therefore, the empty weight fraction cannot be calculated analytically. The only way is to past history and statistics. Table 4.7 shows the empty weight fraction for several aircraft. The only known information about the aircraft is the configuration and aircraft type based on the mission. According to this data, the author has developed a series of empirical equations to determine the empty weight fraction. The equations are based on the published data taken from Ref. [3] and other sources. In general, the empty weight fraction varies from about 0.2 to about 0.75. Figure 4.3 shows the human powered aircraft Daedalus with an empty-weight-to- take-off-weight ratio of 0.3.

$$W_e/W_0 = a\, W_0^{\,c}\, K_{vs}$$

Where a and b are found in Table 4.8. Note that the equation 4.26 is curve fitted in British units system. Thus the unit for maximum take-off weight and empty weight is lb. Table 4.8 illustrates statistical curve-fit values for the trends demonstrated in aircraft data as shown in Table 4.7. Note that the unit of WTO in Table is lb. This is included due to the fact that all data in FAR publications are in British units. In Table, the assumption is that the either the entire aircraft structure or majority of aircraft components are made up of aluminum. The preceding take-off weight calculations have thus

No	Aircraft	a	b
1	Hang glider	6.53×10^{-3}	-1.663
2	Man-powered	-1.05×10^{-5}	0.31
3	Glider/Sailplane	-2.3×10^{-4}	0.59
4	Motor-glider	-1.95×10^{-4}	1.12
5	Micro-light	-7.22×10^{-5}	0.481
6	Homebuilt	-4.6×10^{-5}	0.68
7	Agricultural	3.36×10^{-4}	-3.57
8	GA-single engine	1.543×10^{-5}	0.57
9	GA-twin engine	2.73×10^{-4}	-9.08
10	Twin turboprop	-8.2×10^{-7}	0.65
11	Jet trainer	1.39×10^{-6}	0.64
12	Jet transport	-7.754×10^{-8}	0.576
13	Business jet	1.13×10^{-6}	0.48
14	Fighter	-1.1×10^{-5}	0.97
15	Long range, long endurance	-1.21×10^{-5}	0.95

Table 1.1 Empty weight fraction table

1.10 ESTIMATION OF FUEL FRACTION:

1) Mission profile
2) Weight fractions for various segments of mission
3) Fuel fraction for warm up, taxing and take-off(W1/W0)
4) Fuel fraction for climb (W2/W1)
5) Fuel fraction during cruise – outline of approach
6) Fuel fraction during loiter – outline of approach
7) Estimation of (L/D) max – outline of approach
8) Drag polar of a typical high subsonic jet airplane

The weight of fuel needed depends on the following.

1) Fuel required for mission.
2) Fuel required as reserve.
3) Trapped fuel which cannot be pumped out.
4) The fuel required for the mission depends on the following factors.
5) Mission to be flown.
6) Aerodynamics of the airplane viz. (L / D) ratio.
7) SFC of the engine.

1.10.1 MISSION PROFILE:

Simple Mission: For a transport airplane the mission profile would generally consist of

(a) warm up and take off,

(b) climb,

(c) cruise,

(d) descent,

(e) loiter and

(f) landing

Sometimes the airplane may be required to go to alternate airport if the permission to land is refused. Allowance also has to be made for head winds encountered en-route.

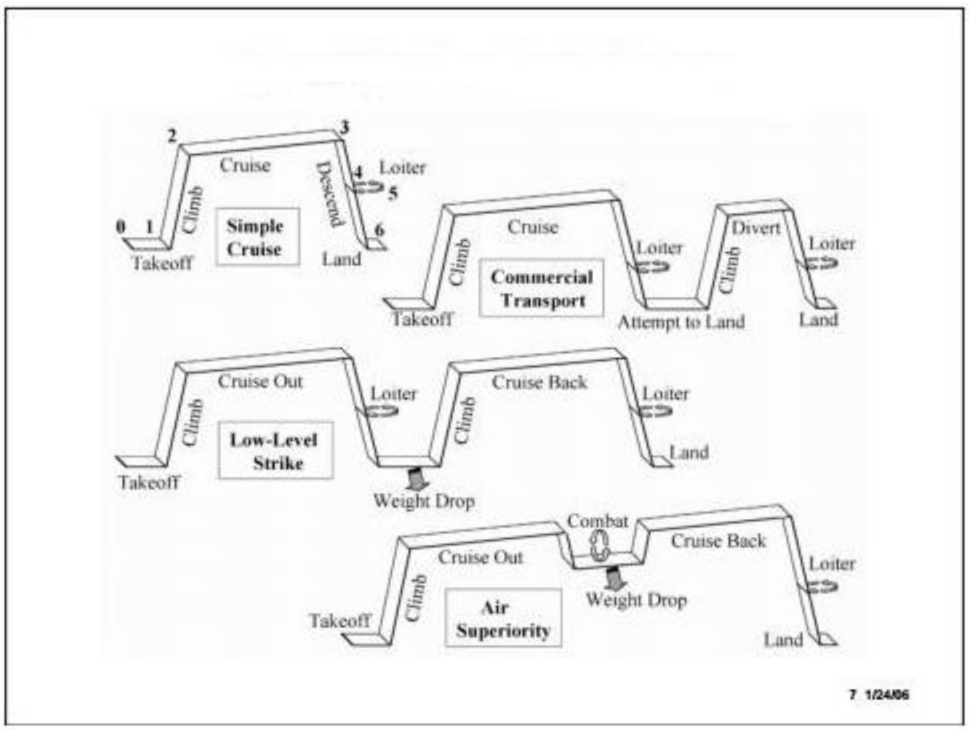


Fig 1.6 Typical mission Profile

1.10.2 WEIGHT FRACTIONS FOR VARIOUS SEGMENTS OF MISSION:

The fuel required in a particular phase of the mission depends on

(a) The weight of the airplane at the start of that phase and

(b) The distance covered or the duration of time for the phase. Keeping these in view, the approach to estimate fuel fraction for chosen mission profile is, as follows.

1) Let the mission consist of 'n' phases.

2) The fuel fractions for the phase 'i' is denoted as Wi / Wi-1.

3) Let W0 be the weight at the start of the flight (say warm up) and Wn be the weight at the end of last phase (say landing). Then, Wn/Wo is expressed as:

$$W_f/W_0=1.06(1-W_x/W_0)$$

1.11 FUEL FRACTION FOR WARM UP, TAXING AND TAKE-OFF (W1 / W0):

Based on the data, the rough guidelines are as follows.

For home built and single engine piston airplanes W1/W0 is 0.99. For twin engine turboprops, jet transports (both civil and military), flying boats and supersonic airplanes W1/W0 is 0.98. For military trainers and fighters W1/W0 is 0.97.

1.12 FUEL FRACTION FOR CLIMB (W2 / W1)

The following guidelines are given based on the data. The low speed airplanes including the twin-engine airplanes and flying boat cruise at moderate altitude (say 4 to 6 km) and hence W2/W1 is taken as 0.99.

The military and civil transport jets cruise around 11 km altitude and W2/W1 is taken as 0.98. The fighter airplanes have very powerful engines and attain supersonic Mach number at the end of the climb. In this case, W2/W1 is between 0.9-0.96. Similarly, the supersonic transport airplanes which cruise at high altitudes (15 to 18 km), W2/W1 is around 0.9. Reference 1.15, chapter 7 gives more elaborate procedure to estimate W2/W1 and is followed for fighter and supersonic cruise airplanes.

1.13 FUEL FRACTION DURING CRUISE – OUTLINE OF APPROACH:

Equations (3.8) and (3.10) present the Breguet formulae for range of airplanes with engine-propeller combination and with jet engine respectively. Consult books on performance analysis (e.g. section 7.4.2 of Ref.3.3) for the derivation of these equations. However, it may be pointed out that while deriving these formulae it is assumed that the following quantities remain constant during the flight.

a) Lift coefficient.
b) Specific fuel consumption (BSFC or TSFC)
c) Propeller efficiency for airplanes with engine-propeller combination and
d) Flight altitude

Equations for range can also be derived when the flight velocity remains constant instead of the lift coefficient.

The derivation is as follows.

In a flight at velocity V (in m/s), the distance dR (in km) covered when a quantity of fuel dWf (in N) is consumed in time dt, is given as :

dR = dWf x (km / N of fuel) (3.27)

Now, in a time interval dt, the distance covered in km is 3.6 V dt, where V is the flight speed in m/s; the factor 3.6 is to convert velocity to kmph. Note dt is in hrs.

Further, for jet engined airplanes the fuel consumed, dWf, in the time interval ‘dt‘ is :

dWf = TSFC x T x dt

Where T is in N, TSFC is N/N-hr or hr-1 and dt is in hrs.

Hence, (km / N of fuel) = $\frac{3.6V \times dt}{TFSC \times T \times dt}$

Substituting this in Eq. (3.27) gives :

$$dR = dW_f \frac{3.6V \times dt}{TFSC \times T \times dt} = \frac{3.6\,V}{TFSC \times T} dW_f \tag{3.28}$$

Noting that, $T = W\frac{C_D}{C_L} = \frac{W}{(L/D)}$ and dW_f = - dW, gives :

$$dR = \frac{-3.6 \times V \times (L/D)}{TFSC \times W} dW \tag{3.29}$$

Assuming V, TSFC and (L/D) to be constant and taking W_{i-1} and W_i as the weights of the airplane at the beginning and the end of the cruise, and integrating Eq.(3.29), yields :

$$R = \frac{-3.6 \times V \times (L/D)}{TFSC} \ln\left(\frac{W_{i-1}}{W_i}\right) \tag{3.30}$$

$$\text{Or} \frac{W_i}{W_{i-1}} = \exp\left\{\frac{-R \times TSFC}{3.6 \times V \times (L/D)}\right\};\ V \text{ in m/s.} \tag{3.31}$$

For an airplane with engine-propeller combination, the fuel consumed in the time interval 'dt' is :

$$BSFC \times BHP \times dt = BSFC \times \frac{THP}{\eta_p} \times dt = \frac{BSFC \times T \times V \times dt}{\eta_p \times 1000}$$

Hence, (km / N of fuel) = $\frac{3.6 \times V \times dt}{BSFC \times \frac{TV}{\eta_p 1000} \times dt}$

Substituting this in Eq.(3.27) yields:

$$dR = \frac{3.6 \times V \times dt \times dW_f}{BSFC \times \frac{TV}{\eta_p 1000} \times dt} = \frac{3600 \times \eta_p}{BSFC \times T} dW_f \tag{3.32}$$

Noting that, $T = W\frac{C_D}{C_L} = \frac{W}{L/D}$ and dW_f = -dW yields:

$$dR = \frac{-3600 \times \eta_p \times (L/D)}{BSFC} \frac{dW}{W} \tag{3.33}$$

Assuming η_p. BSFC and L/D to be constant and integrating Eq.(3.33) gives:

$$R = -\frac{3600 \times \eta_p}{BSFC} (L/D) \ln\left(\frac{W_i}{W_{i-1}}\right)$$

$$\text{Or } \frac{W_i}{W_{i-1}} = \exp\left\{\frac{-R \times BSFC}{3600\eta_p(L/D)}\right\} \tag{3.34}$$

UNIT 2

INTIAL SIZING, CONFIGURATION LAYOUT, CREW STATION, PASSENGERS AND PAYLOAD

2.1 INTRODUCTION:

In the conceptual phase, it shall be estimated the trust-to-weight ratio and the wing loading two fundamental parameters in aircraft design. This section presents and provides an interactive methodology to estimate the best suited

1) Takeoff mass, operating empty mass, fuel mass
2) Takeoff thrust
3) Wing area and wing loading

The following parts of Loftin's method will be presented:

a) landing field length
b) takeoff field length
c) climb gradient in the 2 segment
d) climb gradient during missed approach
e) Cruise speed.

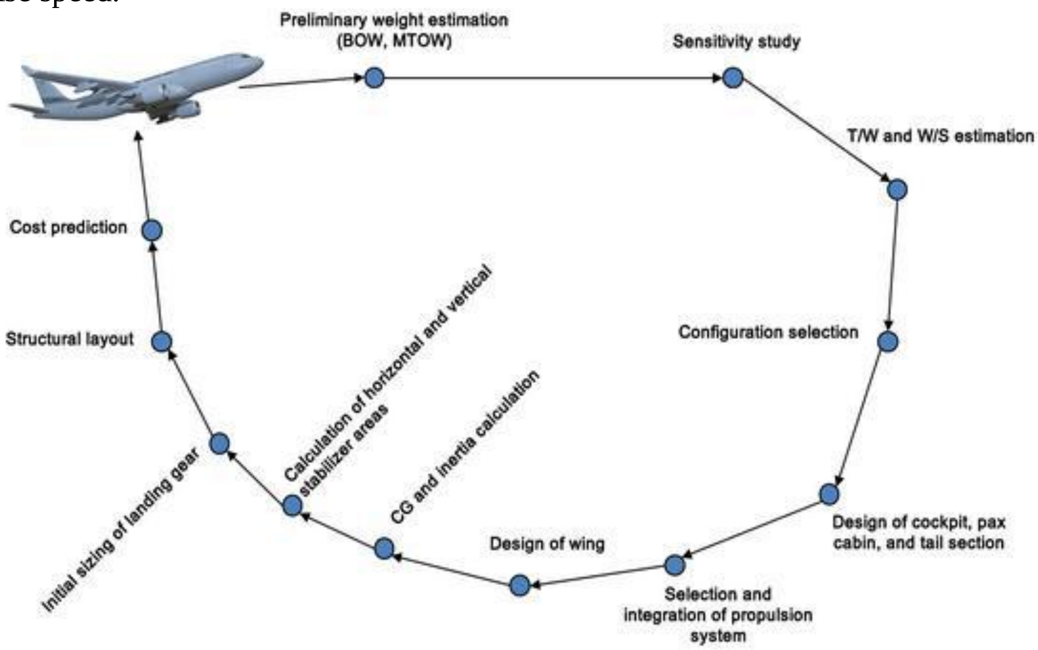


Fig 2.1 Loftin Method

2.2 SIZING METHODOLOGY:

The aim of the sizing exercise is to determine to within some limits the size and weight of the aircraft that will meet the mission objectives and to determine overall design parameters such as wing loading, disk loading, power required, rotor tip speed, wing aspect ratio, etc. At this stage, details of the construction are not known so that heavy reliance is placed on statistical formulas for component weights. Relatively simple analysis techniques are used to determine aerodynamic characteristics such as aircraft drag, maximum lift coefficient and lift-curve slope of wings and/or rotor blades, and stability and trim parameters. Often actual engine data (termed an engine deck) is not available since the engine selection has not yet taken place. In such cases, data for similar engine types is scaled to provide estimates of power or thrust available and fuel consumption as functions of altitude, velocity and throttle settingThe overall size and shape of the fuselage is determined by the payload. In the case of the Civil Transport Rotorcraft, the fuselage must hold 12 (or 19) passengers and their luggage along with two flight crew. A flight attendant is also required since the aircraft must be certified under FAR Part 25. Standards for aisle width, seat width and pitch and pilot seat size are available and can be used to estimate the interior dimensions of the fuselage. Decisions such as whether to place passengers two-abreast, three-abreast or in some other arrangement need also be made. Because of the nature of the current mission and payload, the fuselage will likely more or less resemble that of a typical commuter aircraft rather than that of the conventional helicopter or large transport. As an example, consider a 30‖ seat pitch and 18‖ seat width in a three-abreast configuration. The total aisle width is 20‖. The minimum fuselage inner diameter to accomodate this interior comes out to about 6 1 2 feet. According to Raymer, at least 2—8 inches are needed for insulation and other space between inner wall and outer skin so that an outer diameter of 7‗ is realistic. For the 12-passenger configuration, four rows of seats are necessary. These four rows plus additional space for an entry door, a small galley and, perhaps, a lavatory result in 13-foot-long cabin section. The luggage compartment will be located behind the passengers. That, enough space for the cockpit, and tapered sections at the tail and nose will result in a fuselage length of 41 ft. The fineness ratio (length/average diameter) of this configuration comes out to approximately 5.86, within the range for reasonable tradeoff between pressure drag and skin-friction drag. Some guidelines for the fuselage design include:

1) The overall fineness ratio should be at least 4.
2) The fineness ratio of the aft fuselage (converging section) should be approximately 3 with maximum upsweep angle less than 14◦.
3) The optimal fineness ratio of the nose section depends on the aircraft design Mach number. The fineness ratio should be at least 1 regardless of Mach number. For design Mach numbers between 0.76 and 0.9

The ideal nose fineness ratio can be obtained from:= $74.7714\ M^2 - 113.0765M + 43.7671$

As long as the above guidelines are met, the fineness ratio should be minimized if low drag is the only goal. Of additional primary concern is the moment arm from aircraft center of gravity to the horizontal and vertical tail surfaces. Short moment arms lead to large tail areas in order to meet stability and control-power requirements. Normally the minimum fineness ratio should be about 6 to ensure reasonably sized tails and control surfaces. For the 19-passenger version of the aircraft, a fuselage ―plug‖ of about 5 ft. can be added which extends the fineness ratio to a value of 6.57. The ―basic‖ fuselage would then have an equivalent flat-plate drag area of about 2.4 ft2 while the extended fuselage would have $f \approx 2.8$ ft2. Note that these estimates are based on a standard drag build-up method outlined in Chapter 1. Very little else about the aircraft is known at this point. Even the fuselage weight cannot yet be determined since the loads experienced by the fuselage structure (and therefore the weight of that structure) depend on the maximum weight of the aircraft. Consequently, the usual next step consists of guessing–experience will often be invoked at this time in order to guess values within reason. The objective of the succeeding sizing exercise is to refine the design to optimal values of the parameters, originally guessed, but now selected with conviction. The first guess to be made is of the aircraft design gross weight (GW).

UNIT - II . [Airfoil and Geometry Selection, Thrust to weight ratio, wing loading]. ①

Introduction: The no. of parameters are choosen before the design layout. These include, airfoils, wing and tail geometries, wing loading, thrust-to-weight or power-to-weight ratio, estimated takeoff gross weight and fuel weight, estimated wing, tail, engine sizes, fuselage size.

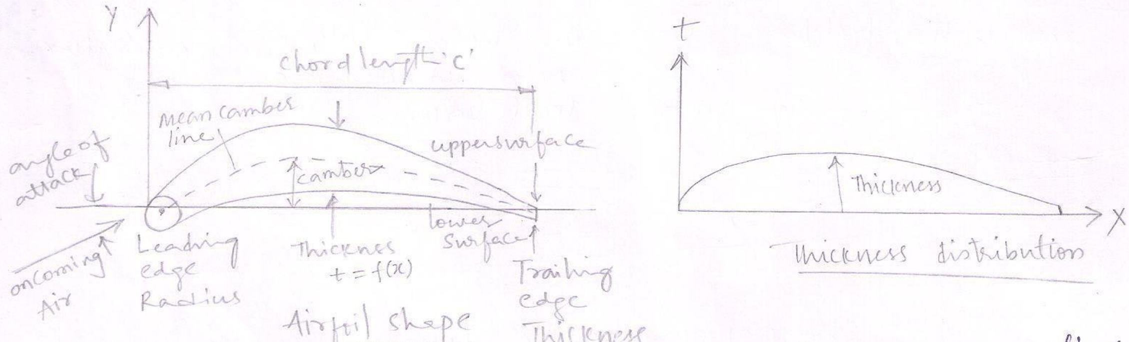
This chapter covers selecting the airfoil, wing, tail geometry.

Airfoil Selection:

→ The airfoil affects the cruise speed, takeoff, landing distances, stall speed, handling qualities, aerodynamic efficiency.

Recently, the low-speed airfoils developed by Peter Lissaman.

Airfoil Geometry:



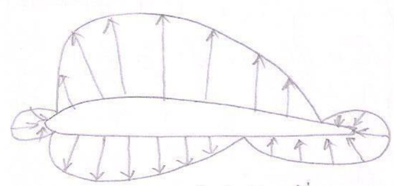
→ The front of the airfoil is defined by a Leading-edge radius.
→ The last of the airfoil is defied by a Trailing edge.
→ The chord of the airfoil is the straight line from the LE to TE.
→ Camber refers to curvature.
→ mean camber line, is the line equidistant from upper and lower surfaces.

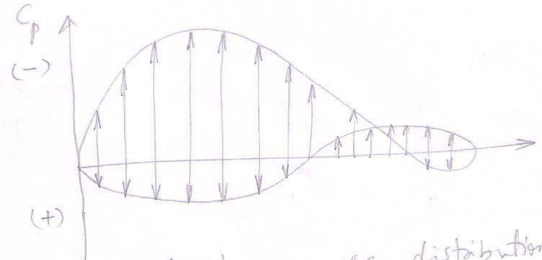
An airfoils with curved bottoms, they are known as "double-cambered".
An airfoil with a concave lower surface known as "under-cambered" airfoil.

→ The thickness is the distance from upper surface to the lower-surface, measured perpendicular to the mean camber line

The "airfoil thickness ratio (t/c)" refers to the maximum thickness of the airfoil divided by its chord.

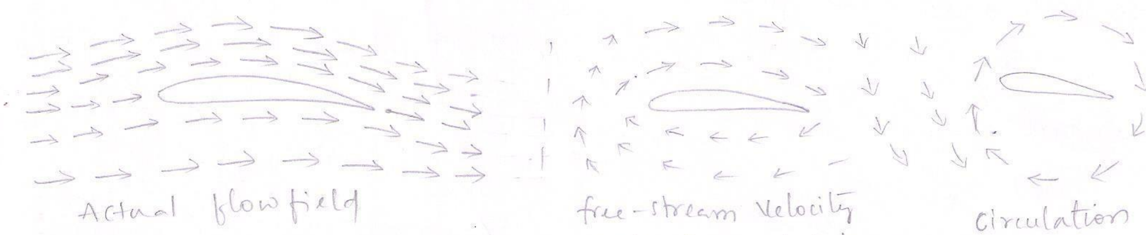
Airfoil Lift and Drag:

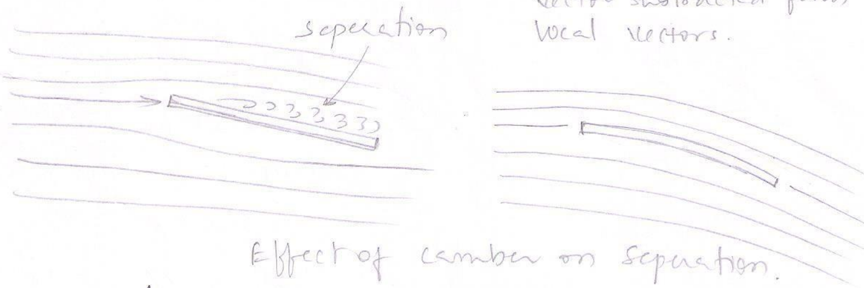




Typical airfoil pressure distribution.

"Circulation" is the theoretical basis for the classical calculation of lift and drag-due-to lift.





Effect of camber on separation.

$$C_l = \frac{L}{qc}, \quad C_d = \frac{D}{qc}, \quad C_m = \frac{m}{qc^2}.$$

where, c – chord length

$q = \frac{1}{2}\rho V^2$ dynamic pressure

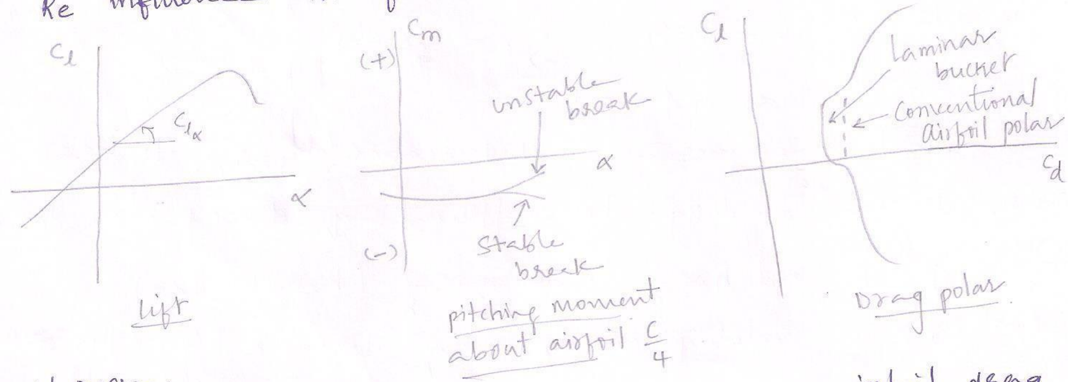
α – angle of attack

$\frac{\partial C_l}{\partial \alpha} = C_{l_\alpha}$ – slope of the lift curve $= 2\pi$ (theoretical thin airfoil)

→ Airfoil characteristics are strongly effected by the "Reynolds no."

$$\text{Reynolds number} = \frac{\text{Inertial force}}{\text{Viscous force}} = \frac{\rho V l}{\mu}$$

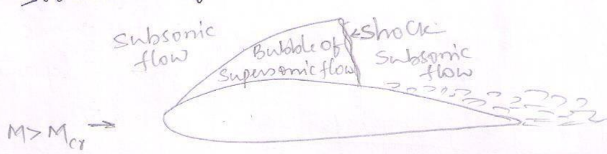
Re influence the flow will be laminar (or) turbulent.



<u>Airfoil Design:</u>

The airfoil design would consider factors as airfoil drag during cruise, stall, pitching-moment characteristics, thickness for structural ease of manufacture.

Modern airfoil design is based upon inverse Computational Solutions for desired pressure (or) velocity distributions on the airfoil.



As an airfoil generates lift the velocity of the air passing over its upper surface is increased. The speed at which Supersonic flow first appears on the airfoil is called the "critical Mach number".

The upper surface shock creates a large increase in drag, reduction in lift and a change in the pitching moment.

A "Supercritical" airfoil is one designed to minimize these effects.

<u>Design lift Coefficient:</u>

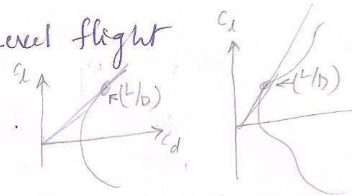
The first consideration in initial airfoil selection is the "design lift coefficient."

• In Subsonic flight a well-designed airfoil operating at its design lift Coefficient has C_D is little more than $D_{\text{skin friction}}$.

The aircraft should be designed so that the design lift Coefficient to Maximize the aerodynamic efficiency.

In a first approximation, assumed Level flight

$W = L = qSC_L \cong qSC_l$, $C_l = \frac{1}{q}\left(\frac{W}{S}\right)$.



$q = \frac{1}{2}\rho V^2$, as $h\uparrow$, $\rho\downarrow$

as flight time increases "W" decreases (W_{fuel}), to have Study & stay of design lift Coefficient, q should also decrease

i.e., $q \propto \rho$ and $h \propto \frac{1}{\rho}$ then $q \propto \frac{1}{h}$ (h = altitude)

(or) $q \propto V$, velocity ~~many~~ may decrease (slows down the aircraft).

Stall: Stall characteristics play an important role in airfoil selection. By rapid change in pitching moment, some airfoils exhibit a gradual reduction in lift others exhibit a violent loss of lift during stall.

This effect due to three different type of airfoil stall.

→ "Fat" airfoils (round Leading edge and t/c greater than about 14%), stall from a Leading edge. The turbulent boundary layer increases with angle of attack at around 10°, the boundary layer begins to seperate.

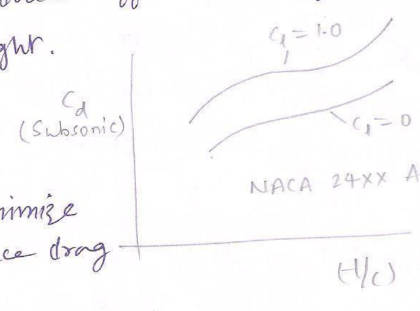
→ Thinner airfoils stall from the leading edge. airfoil is moderate thickness (about 6-14%), the flow seperates near the nose at a very small angle of attack.

→ Very thin airfoils, the flow separates from the nose at a small angle of attack and reattaches almost immediately.

Airfoil thickness Ratio: (t/c) has a direct effect on drag, maximum lift, stall characteristics, structural weight.

The drag increases with increasing thickness due to increased separation.

A Super-critical airfoil tends to minimize shock formation and can be used to reduce drag for a given thickness ratio.



Wing Geometry

The reference (trapezoidal) wing is the basic wing geometry.

S – Reference wing Area

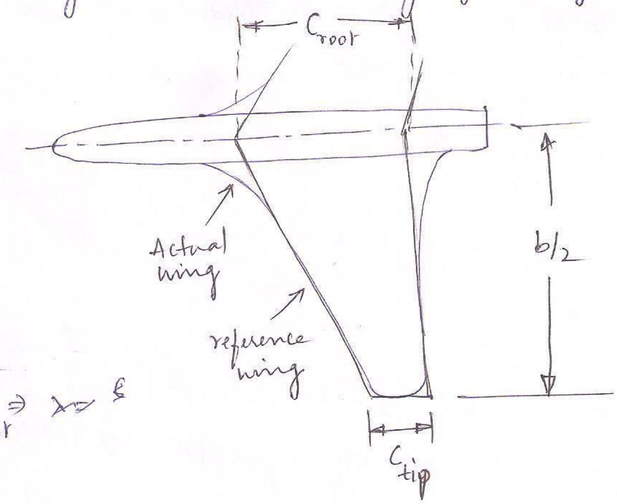
C – chord

A – Aspect ratio $= \frac{b^2}{S}$

t/c – Airfoil thickness ratio

λ – Taper ratio $= \frac{C_t}{C_r}$

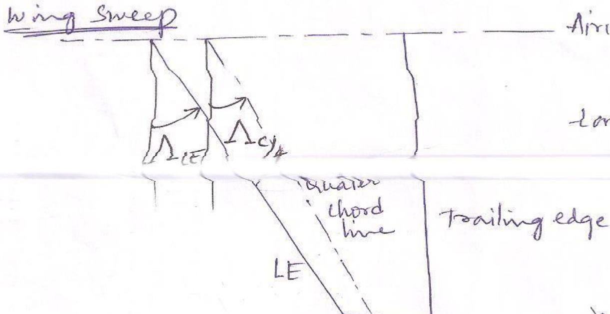
b – Span



$C_{root} = \frac{2 \cdot S}{b(1+\lambda)}$, $C_{tip} = \lambda \cdot C_{root}$ ⇒ ~~λ~~

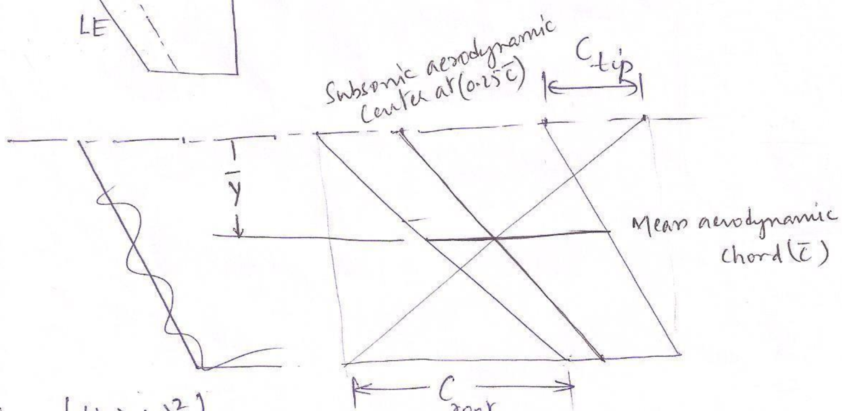
$S = \frac{W}{(W/S)}$, $b = \sqrt{A \cdot S}$

Wing Sweep



$$\tan(\Lambda_{LE}) = \tan \Lambda_{c/4} + \left[\frac{1-\lambda}{A(1+\lambda)}\right]$$

Aspect Ratio

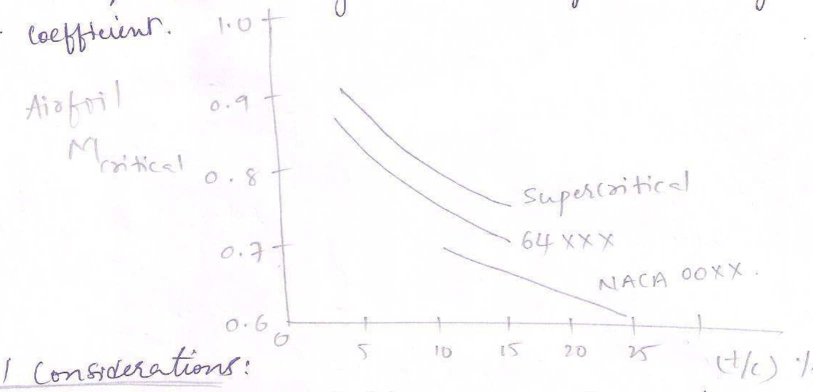


$$\bar{c} = \left(\frac{2}{3}\right) C_{root} \frac{(1+\lambda+\lambda^2)}{(1+\lambda)}$$

$$\bar{y} = \left(\frac{b}{6}\right)\left[\frac{1+2\lambda}{1+\lambda}\right] ,$$

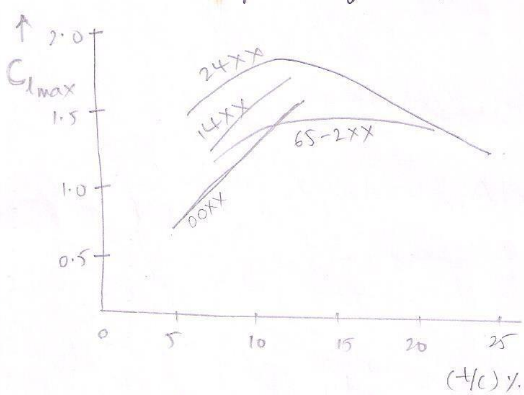
Typical wing aerodynamic center $= 0.25\bar{c}$ (Subsonic)
$= 0.4\bar{c}$ (supersonic)

The thickness ratio affects the max Lift, Stall effect upon nose shape. (3)
wing of high AR (Aspect Ratio $= \frac{b^2}{S} = \frac{b^2}{b \cdot c} = \frac{b}{c}$), moderate sweep, large nose radius provides a higher Stall angle, and a greater maximum lift coefficient.

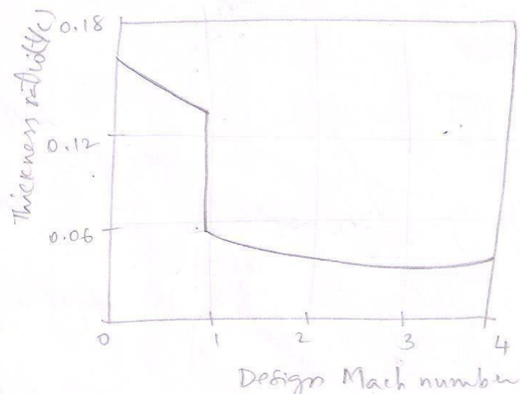


Other Airfoil Considerations:

Another aspect of airfoil selection is Reynolds number.

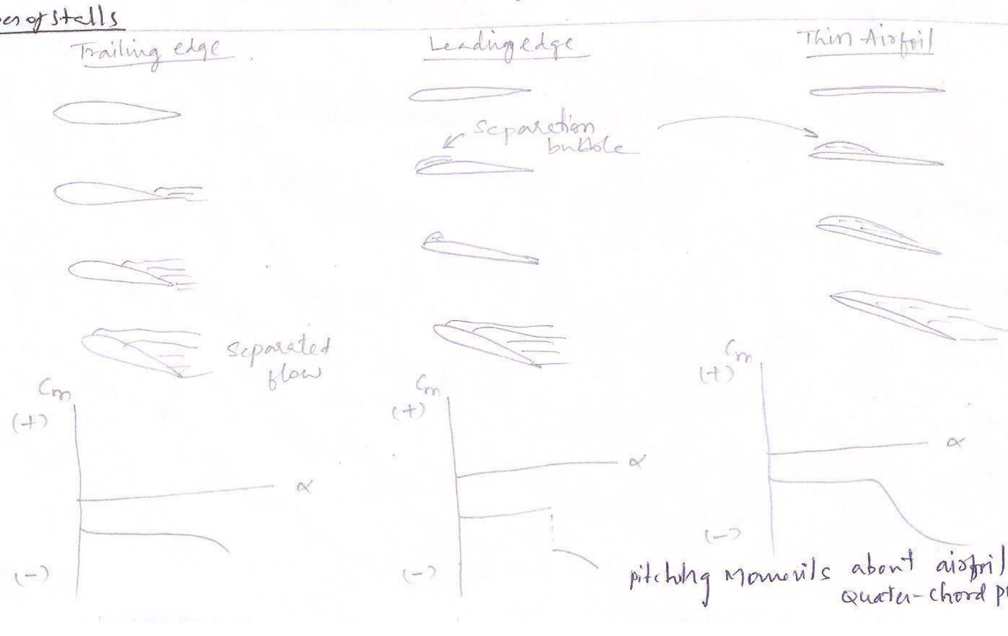


Effect of (t/c) on maximum Lift



Thickness ratio historical trend.

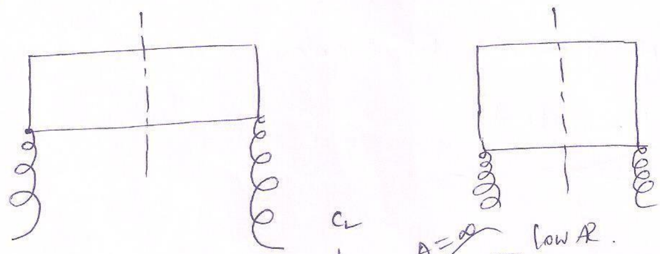
Types of Stalls



pitching Moments about airfoil Quarter-chord point.

Wing tip Vortices

- A wing with high AR has tips farther apart than an equal Area wing with a low AR. Therefore the amount of the wing affected by the tip vortex is less for a high AR wing than for a low AR wing.

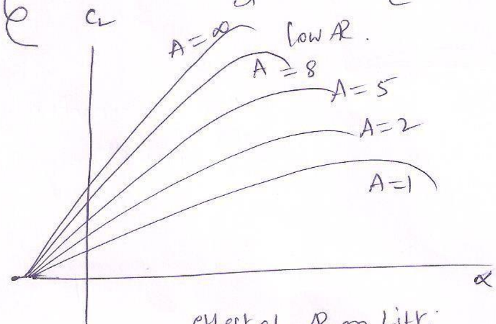


high AR

low AR.

$$C_L^2 = \frac{1}{\pi e AR}.$$

$$C_D = C_{D,0} + K C_L^2$$

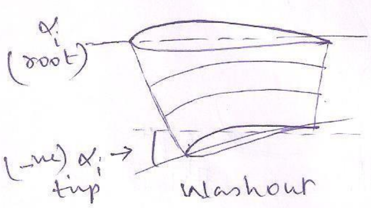


effect of AR on Lift.

Twist

Wing twist is used to prevent tip stall and to revise the lift distribution to approximate an ellipse. typical wing twist 0° to 5°.

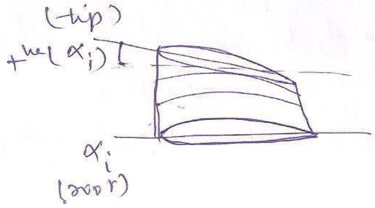
Geometric twist; is the actual change in airfoil angle of incidence, measured withrespect to C_{root}.



Washout

A wing whose tip airfoil is at a negative angle (nose-down) compared to the root airfoil is said to be washout.

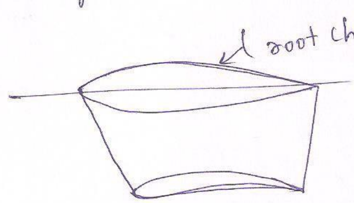
Wing with "washout", C_{root} stall first than C_{tip}



Aerodynamic twist is the angle between $\alpha_{L=0}$ angle of an airfoil and $\alpha_{L=0}$ of the root airfoil

if identical airfoil is used from root to tip, the aerodynamic twist and geometric twist is same.

a wing with No geometric twist can have aerodynamic twist.



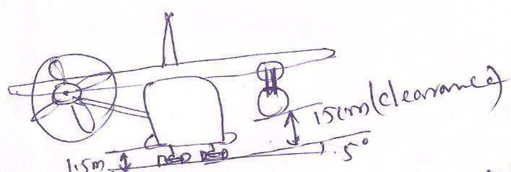
Total wing aerodynamic twist = (wing geometric twist) + (root airfoil Zero-lift angle) − (tip airfoil Zero lift angle)

Wing incidence

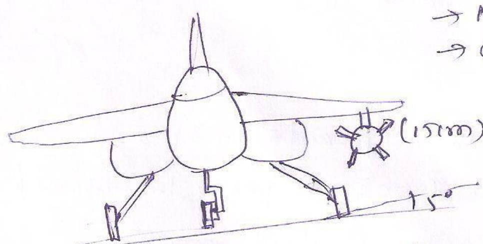
Dihedral

Anhedral

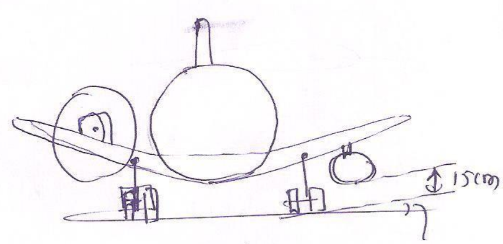
Wing location < High / Mid / low



→ Mostly used for military purpose
→ Maintenance problem
→ cargo/military, → Subsonic



→ Mainly for fighter aircraft
→ Supersonic
→ Aerobatic performances.



→ High stability compared to other type
→ mainly for passenger aircrafts
→ fuselage divided into two compartments
→ ~~first~~ upper compartment for passengers
→ lower compartment for cargo.

Initial Sizing

→ Aircraft can be sized < fixed size-engine (or) fixed engine
Rubber engine

Rubber Engine Sizing

$$W_0 = \frac{W_{crew} + W_{payload}}{1 - \left(\frac{W_f}{W_0}\right) - \left(\frac{W_e}{W_0}\right)} \quad , \quad \left(\frac{W_f}{W_0}\right) = 1.06\left(1 - \frac{W_x}{W_0}\right)$$

Refined Sizing equation

$$W_0 = W_{crew} + W_{\substack{fixed\\payload}} + W_{\substack{dropped\\payload}} + W_{fuel} + W_{empty}$$

(or) $$W_0 = W_c + W_{fixed\,p} + W_{dropped\,p} + W_{fuel} + \left(\frac{W_e}{W_0}\right) \cdot W_0 .$$

Empty - weight fraction $\left(\frac{W_e}{W_0}\right)$

from table 6.1, $\left(\frac{W_e}{W_0}\right) = \left(a + b\, W_0^{c_1} A^{c_2} \left(\frac{T}{W_0}\right)^{c_3} \left(\frac{W_0}{S}\right)^{c_4} M_{max}^{c_5}\right) K_{vs}$. for jet

K_{vs} = variable Sweep Constant = 1.04 if variable sweep
= 1.00 if fixed sweep

from table 6.2

$$\left(\frac{W_e}{W_0}\right) = a + b\, W_0^{c_1} A^{c_2} \left(hp/W_0\right)^{c_3} \left(\frac{W_0}{S}\right)^{c_4} V_{max}^{c_5} \quad \text{for propeller.}$$

Fuel weight

$$W_{f_i} = \left(1 - \frac{W_i}{W_{i-1}}\right) W_{i-1} \, , \quad \text{total mission fuel } W_{f_m} = \sum_1^n W_{f_i}$$

the total aircraft fuel, $W_f = 1.06\left(\sum_i^n W_{f_i}\right)$

Engine start, Taxi and Takeoff

$$\frac{W_1}{W_0} = 0.97 - 0.99$$

Climb and Accelerate $\quad \dfrac{W_i}{W_{i-1}} = 1.0065 - 0.0325\,M \quad \text{(Subsonic)}$

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

Cruise $\quad \frac{W_i}{W_{i-1}} = \exp\left[\frac{-RC}{V(L/D)}\right]$ (Jet)

$$\frac{W_i}{W_{i-1}} = \exp\left(\frac{-RC_{power}}{\eta_p (L/D)}\right) = \exp\left[\frac{-RC_{bhp}}{550\eta_p(\frac{L}{D})}\right] \quad \text{(propeller)}$$

during cruise and loiter, $L = W$, $\frac{L}{D} = \frac{1}{\frac{qC_{D,O}}{(W/S)} + \left(\frac{W}{S}\right)\frac{1}{q\pi Ae}}$

Loiter $\quad \frac{W_i}{W_{i-1}} = \exp\left(\frac{-EC}{L/D}\right)$ (Jet)

$$\frac{W_i}{W_{i-1}} = \exp\left[\frac{-EVC_{power}}{\eta_p(\frac{L}{D})}\right] = \exp\left[\frac{-EVC_{bhp}}{550\,\eta_p(\frac{L}{D})}\right]$$

Combat - known - time Fuel Burn

Combat missions normally specified as either a time duration at max power or as a certain number of combat turns at max power, at some altitude and Mach number.

d - ~~time~~ duration

$$\frac{W_i}{W_{i-1}} = 1 - C\left(\frac{T}{W}\right)d \quad , \quad \frac{T}{W} \rightarrow \text{Combat weight and Thrust. (Not takeoff weight)}$$

If the Combat is defined by some no. of turns, the duration of Combat (d) must be calculated.

The time to complete 'x' turns is total number of radians to turn divided by the turn rate. $\quad d = \frac{2\pi x}{\dot\psi} = \frac{2\pi V x}{g\sqrt{n^2-1}}$

$$n = \frac{L}{W} = \left(\frac{T}{W}\right)\left(\frac{L}{D}\right), \quad n \le n_{max}, \; n \le \frac{qC_{L max}}{(W/S)}$$

max structural load factor and max available lift.

$$\left(\frac{L}{D}\right) = \frac{1}{q\frac{C_{D,O}}{n\left(\frac{W}{S}\right)} + \left(\frac{W}{S}\right)\frac{n}{q\pi Ae}}$$

Descent for Landing $\quad \frac{W_i}{W_{i-1}} = 0.990$ to 0.995

Landing and Taxi Back $\quad \frac{W_i}{W_{i-1}} = 0.992$ to 0.997

Fixed-Engine Sizing , $W_0 = \dfrac{N\, T_{per\,engine}}{(T/W)}$, N - no. of engines

$W_f = CTd$ (weight of fuel calculated)

Geometric Sizing

Fuselage ; after the takeoff weight has been estimated, the fuselage, wing and tail can be sized.

once the fuselage length, diameter are known (that depend upon the seating capacity, compartment no.s)

(f) Fuselage fineness ratio = $\dfrac{\text{Fuselage length}}{\text{Fuselage max diameter}}$.

Table 6.3 length $= a W_0^c$

~~Wing~~

Tail volume Coefficient

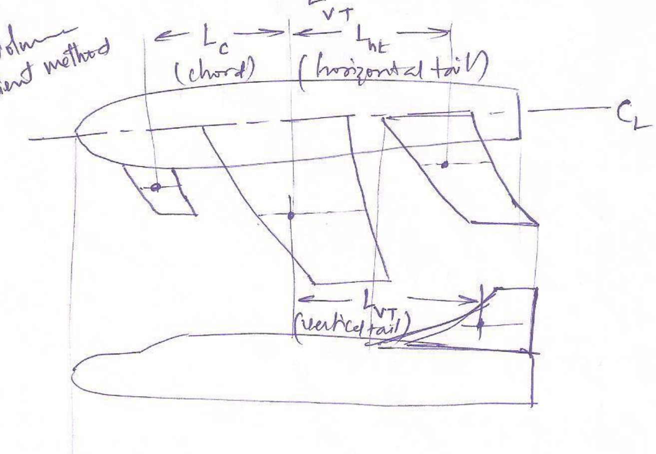
Tail effectiveness $\propto$ tail Area $\times$ tail moment arm.

tail volume coefficient — vertical tail volume coefficient
— Horizontal tail volume coefficient.

$$C_{VT} = \frac{L_{VT} S_{VT}}{b_W S_W}, \quad C_{HT} = \frac{L_{HT} S_{HT}}{\bar{C}_W S_W}, \quad \bar{C}_W \text{ - horizontal tail volume coefficient.}$$

$$\Rightarrow S_{VT} = C_{VT}\, b_W S_W / L_{VT}, \quad S_{HT} = C_{HT}\, \bar{C}_W S_W / L_{HT}$$

Tail Volume Coefficient method



S_W - wing area

b_W - wing span

$\bar{C}_W$ - wing mean chord

Geometric Sizing ; Fuselage

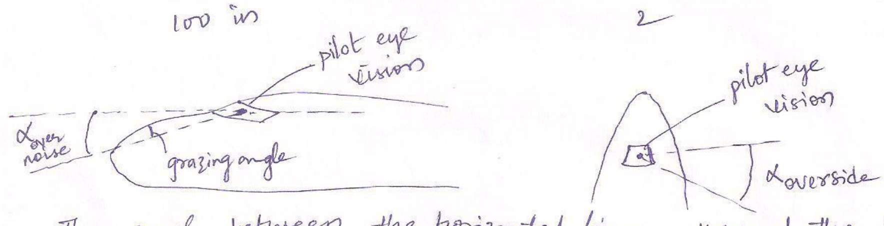
Volume Consideration

(i) passenger requirements, (a) → coach class
→ passenger compartment requirement,
→ passenger seating arrangement
→ cargo container (long Range Aircrafts)
→ placement and type of emergency exit.

(ii) crew requirements

The size of the crew compartment will vary depending on the Aircraft, with long range military | commercial, transport, passenger Aircraft, it should be designed for two to four members.

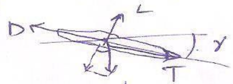
crew compartment length	crew members
150 in	4
130 in	3
100 in	2



The angle between the horizontal line through the pilots vision down to the point of the heighest vision abstraction is called overnoise angle ($\alpha_{overnoise}$)

The proper overnoise angle depend upon the Landing approach angle

$\gamma_{approach} = \sin^{-1}\left(\frac{D}{W}\right)$.



$D = W\sin\gamma \Rightarrow \boxed{\gamma = \text{Arc}\sin\left(\frac{D}{W}\right)}$

an important factor that impacts the shape of the Forward section of the fuselage is the pilots unabstracted Front view, which can be obtained by a proper amount of overnoise angle.

(iii) Fuel Storage Requirements : In long range aircraft, a large percentage of the ~~weight~~ "takeoff weight" is due to the "Fuel weight".

The volume required to hold this fuel can be allocated to Fuselage, wing, both.

The actual volume available in locations in the Fuselage, wing depends on the type of fuel tank used.

→ Discrete fuel tank (for small range aircrafts): for small general aviation Aircrafts consists of seperately manufactured containers, mounted in the aircraft wing span portion, behind the engine in the fuselage.

→ Bladder fuel tank (made up of Rubber): Thick Rubber bags are placed into the cavities in the wing, fuselage. They can be made to be self-sealing improving aircraft servivability in the event of uncontrolable, enemy fire.

→ Integral Fuel tank (it is integral part of the main body): These are cavities with in the Airframe structure that are seal to form fuel tanks. ie; wing Box areas, b/w wing spars, Bulkheads

Configuration layout and loft

output of the configuration layout task will be design drawings of several types aswell as the geometric information required for further analysis.

once the design hasbeen analyzed, optimized and redrawn for a number of iterations of the conceptual design process and more detailed drawings can be prepared, called the "inboard profile".

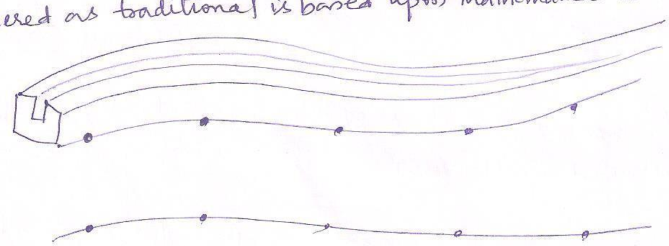
The inboard profile is farmore detailed than the initial layout.

The inboard profile drawings is being prepared "a line control" drawing may be refinee and details the external geometry, definition provided on the initial layout.

after inboard profile drawing has been prepared, an "inboard isometric drawing".

Conic Lofting : lofting is the process of defining the external geometry of the aircraft.

A new method of lofting were used for D-15 mustang, This method is now considered as traditional is based upon mathematical curve from known as the "conic".



Spline lofting

Generalized

A Conic equation $C_1X^2+C_2XY+C_3Y^2+C_4X+C_5Y+C_6=0$.

Advantages using conic lofting

→ A wide variety of curves that can be represent.

→ The ease with which it can be constructed on the drafting table.

Dis Advantages

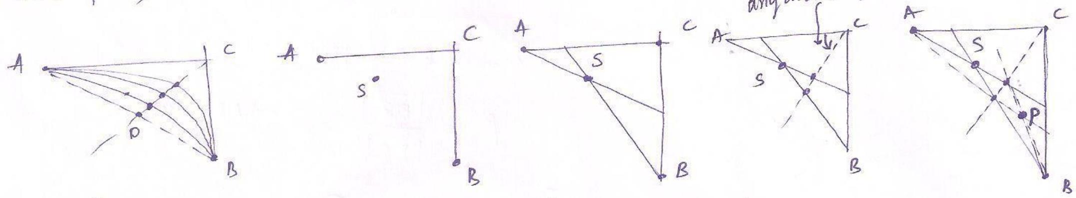
→ It requires lot of trail and error to achieve a smooth surface both in cross-section and longitudinal.

→ It doesn't provide a unique mathematical definition of the surface, i.e., to create new cross-section it requires a tremondous amount of Drafting effort, Especially for the canted surfaces (i.e. a cross-section cut at some angle other than ⊥r to the centerline of the aircraft).

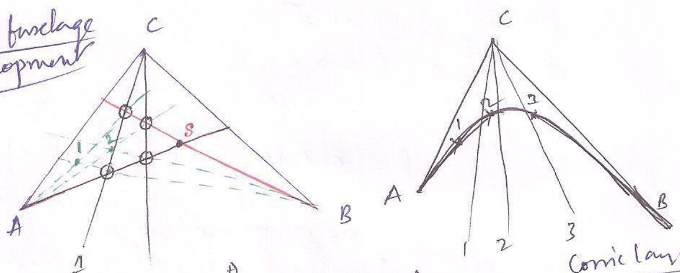
→ Make a points A and B and draw tangents from A and B, then they intersect at C.

→ 'S' is any point, and Draw the lines from A and B to meet the point 'S'.

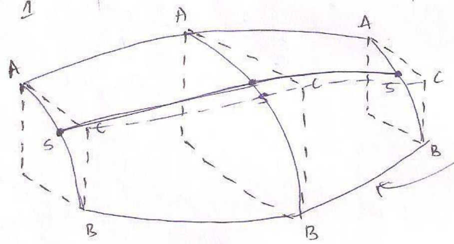
→ Draw a line with any acute angle to meet $\overline{AS}$ and $\overline{BS}$ line. Now connect 'B' to that point, The ~~line~~ cuts the line $\overline{AS}$ and connect A to the point on



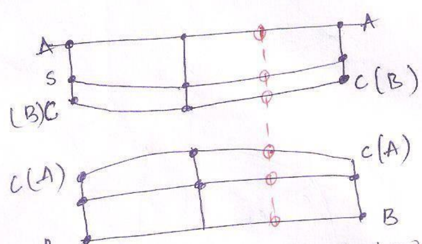
Conic fuselage Development



Conic layout example

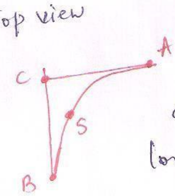


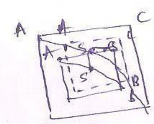
Longitudinal Control lines (lines which connects, ABCS points in three sections, as shown)



Front view

Top view



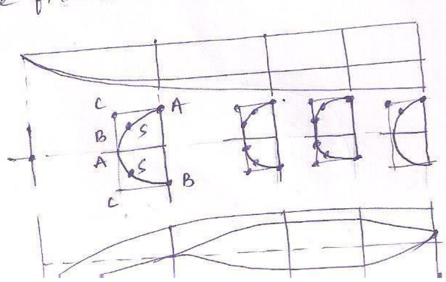


Front view

cross-section development from longitudinal Control lines.

Fuselage lofting example (application of lofting).

Take five Control stations, Station '0' is the nose.



Fuselage reference line

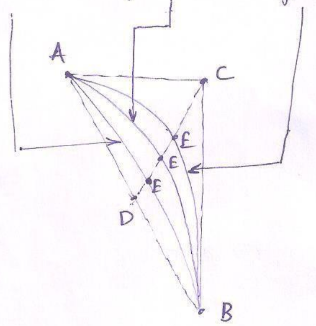
(Typical fuselage lofting for Fighter Aircraft).

Conic shape parameter: (ρ)

In initial lofting problems to select the shoulder point (S). The locations of the shoulder points (S) can be difficult to control.

→ creating conics ~~either~~ too square (shoulder point too close to 'C')

→ creating conics too flat (shoulder point faraway from 'C')



So, conic shape parameter

$$\rho = \frac{|\overline{DE}|}{|\overline{DC}|}, \quad |AD| = |BD|$$

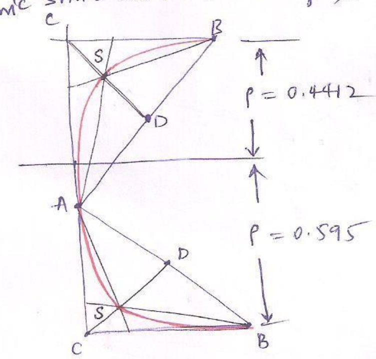
Hyperbola, $\rho > 0.5$

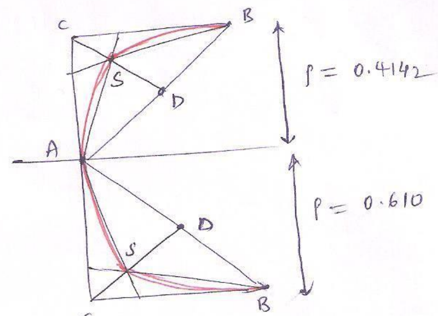
parabola, $\rho = 0.5$

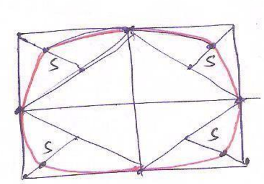
Ellipse, $\rho < 0.5$

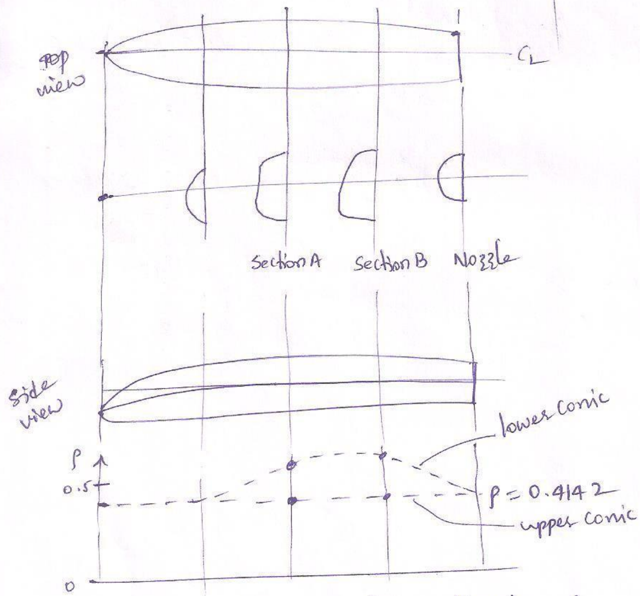
circle, $\rho = 0.4142$ and $|\overline{AC}| = |\overline{BC}|$

Example of the conic-developed loft lines for supersonic SAAB J35 Draken (Dragon).









conic fuselage development using ρ
ρ (conic shape parameter).

Flat-wrap Fuselage Lofting

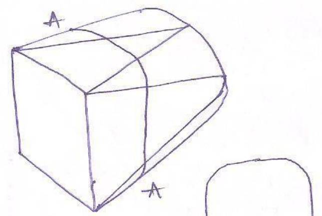
→ Compound curvature is important for aircraft fabrication.
→ Compound curvature implies the existence of surface curvature in all directions for some point on the surface.

Ex: a ball is entirely composed of Compound curvature surfaces.
a flat sheet has no curvature, compound or any other curve.
a cylinder is curved but only in one direction. so, a surface with curvature in only one direction is said to be "flat-wrapped".

→ flat-wrap lofting is defined in the initial Loft definition ~~to~~ used for the Conceptual layout.

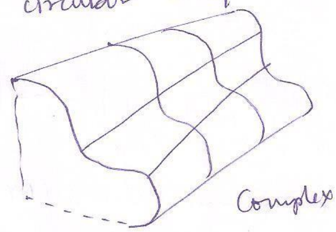
Circle to square Adapter

→ common problem in lofting is the "circle to square adapter".
for Ex: The inlet duct of supersonic jet aircraft is approximately square at the air inlet, and attain circular shape at the engine ~~front~~ front face.



Circle to square adapter

Section A-A



Complex flat wrapped surface

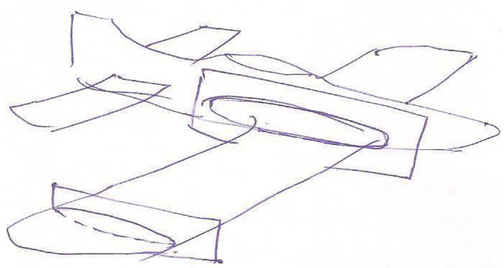
→ flat-wrap can be attained for a circle to square adapter by constructing the adapter of interlocking, V-shaped segments.

Fuselage Loft verification

The use of smooth longitudinal control lines defining Conic cross-sections gives smooth fuselage, but sometimes the definition deviates.

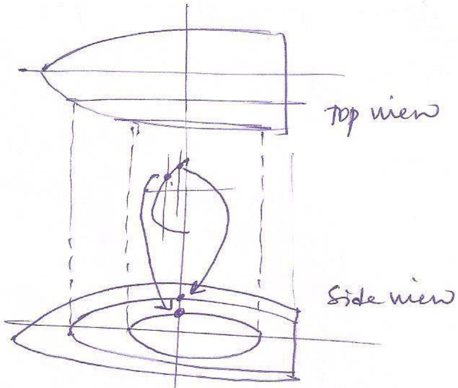
example, if a part of the fuselage is to be flat wrapped, it may be difficult to smoothly connected straight control lines for the flat-wrap part of the fuselage with the curved control lines for the rest of the fuselage.

→ The horizontal ~~lines~~ waterlines cuts can be used for the evaluation of the smoothness of an aircraft fuselage. but, common to use vertically oriented cuts known as "buttock plane cuts".



airfoil is a cut of the wing

"Buttock-plane cut"



Buttock-plane cut layout

Wing / Tail Layout and Loft

: Reference wing / Tail layout

The basic geometric parameters are Aspect ratio (A), taper ratio (λ), sweep, dihedral, thickness.

$$b = \sqrt{AS}\ ,\quad C_{root} = \frac{2S}{b(1+\lambda)}\ ,\quad C_{tip} = \lambda C_{root}$$

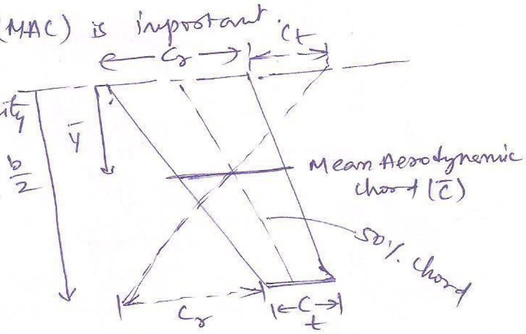
$$\bar{C} = \left(\frac{2}{3}\right) C_{root}\left(\frac{1+\lambda+\lambda^2}{1+\lambda}\right),\quad \bar{Y} = \left(\frac{b}{6}\right)\left(\frac{1+2\lambda}{1+\lambda}\right).$$

Wing location with respect to Fuselage

here, location of mean Aerodynamic Chord (MAC) is important.

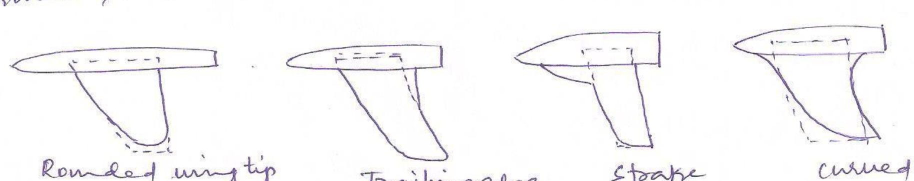
for stable, unstable aircraft, centre of gravity will change with the effect of wing alone, and combination of fuselage etc.,

50%, to 40% to 25% of $\bar{C}$.

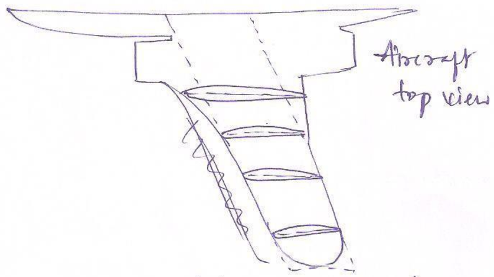


Wing / Tail lofting

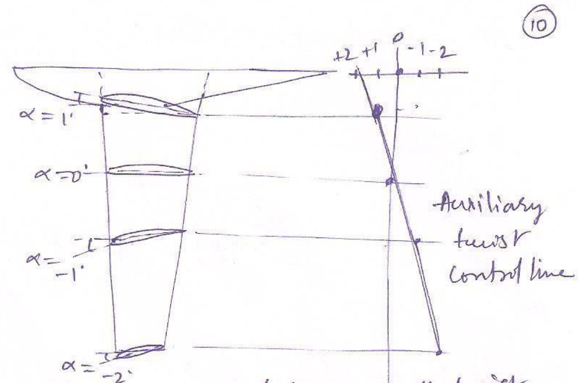
The reference wing is defined to the aircraft centreline, and is based upon the projected area.



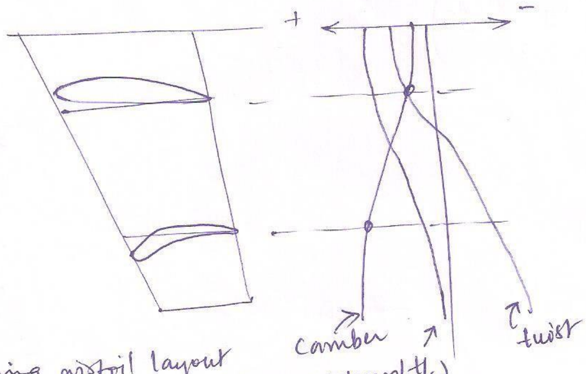
Non-trapezoidal wings



Airfoil layout on wing planform



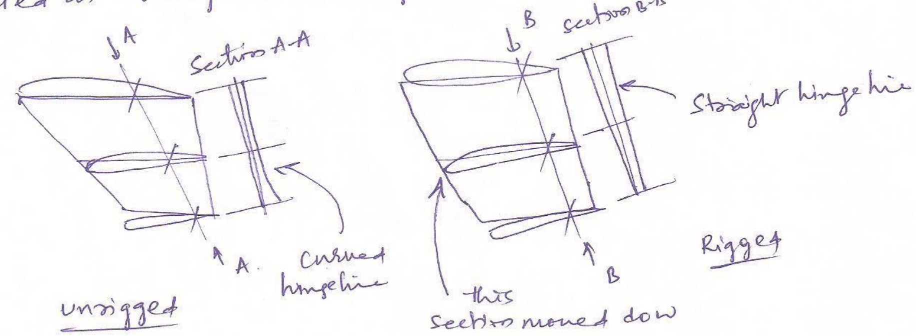
Airfoil layout with twist



wing airfoil layout
- non-linear variations

Airfoil linear interpolation

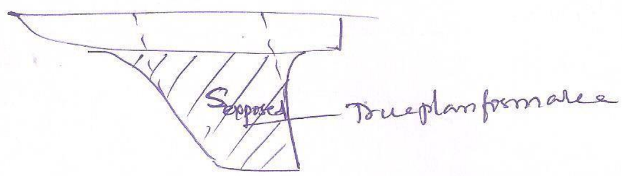
linear interpolation, easiest method, here new airfoils are created as "weighted averages" of the root and tip airfoils.



Unrigged

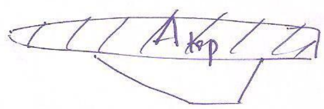
Rigged

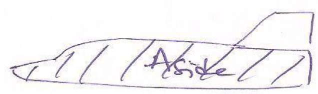
Wetted Area Determination



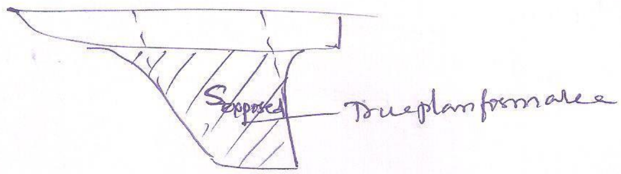
if $t/c < 0.05$, $S_{wet} = 2.003\, S_{exposed}$, if $t/c > 0.05$, $S_{wet} = S_{exposed}\left(1.977 + 0.52\,(t/c)\right)$.

$$S_{wet} \simeq 3.4\left(\frac{A_{top} + A_{side}}{2}\right).$$





Wetted Area Determination



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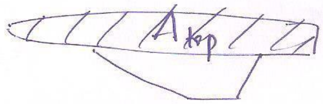
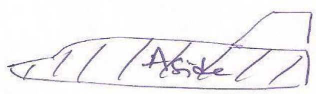


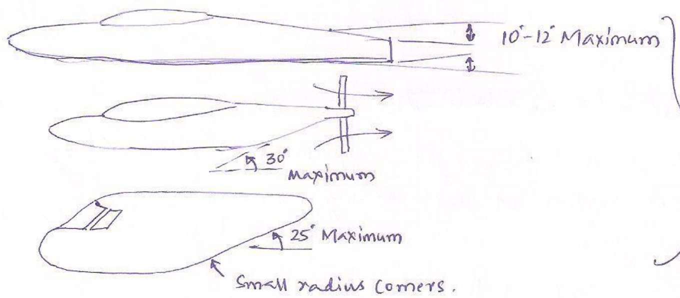


Aerodynamic Considerations

(i) Design arrangement:

- poorly designed aircraft can have more flow separation, rise in transonic drag and supersonic wave drag.
- poor arrangement of wing-fuselage cause lift loss and Elliptic lift distribution.
- Minimization of wetted Area is the most powerfull aerodynamic consideration

wetted Area $\propto$ skin friction Drag.



prevent flow seperation and reduces Drag penalty

(ii) Isobar Tailoring: (Isobar – constant pressure)

- shocks are formed over the top of the wing due to the increased velocity (the velocity which is perpendicular to the leading edge), sweeping the wing causes this velocity to be slower, Shock formation is delayed.
- Isobars are lines connecting regions with same pressure.

Wing sweep is also based upon the wing pressure

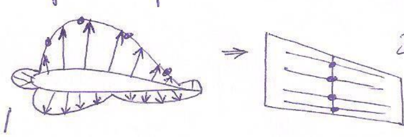


fig ①

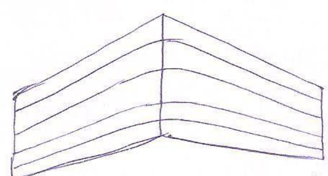


fig ② Isobars unsweep at root and tips

fig ③ Restore isobar sweep with 'pesky' root airfoil.

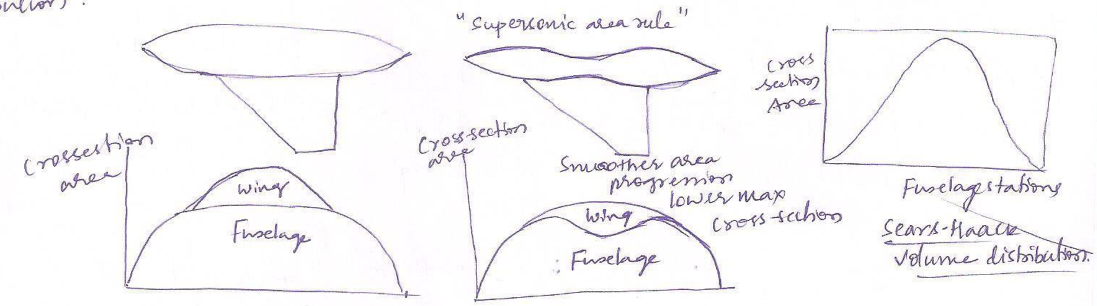
- Here, ~~two~~ common problems, with "real wing", from fig ①, isobar lines are joined by a rounded-off corners, due to this shocks form near the wing root.

To solve this, two Aerodynamic strategies can be employed. one is to exaggerate the wing sweep near the root, blending the wing in a smooth fashion.

Another approach, use specially designed airfoils to have its pressure peak very near its nose (airfoil preferebly have, Large nose radius, fairly flat top, enough negative camber)

(iii) Supersonic Area rule :

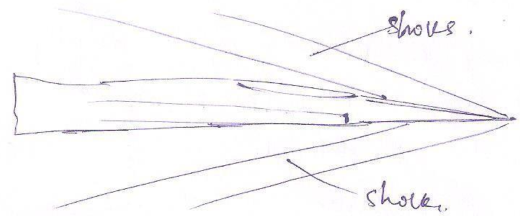
- Mainly to minimize supersonic wave Drag (pressure drag due to shock).
- Sears-Haack body has the lowest wave Drag, because it had good Volume distribution.



Design for low wave Drag

(iv) Compression Lift :

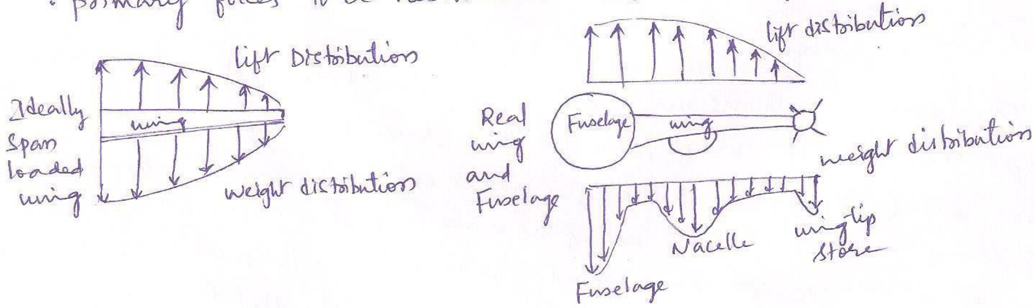
- Any body shape will create shock waves at supersonic speeds at the nose, or any place where the cross-section is increasing.
- By placing the wing above these shocks, the increased pressure beneath the wing provides free lift i.e., 30% of the total lift is produced.

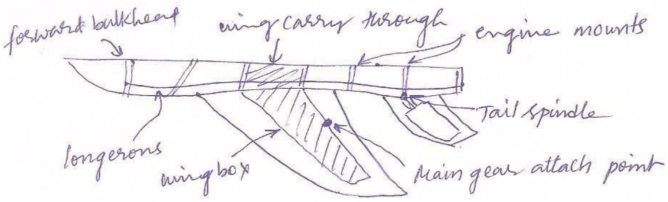


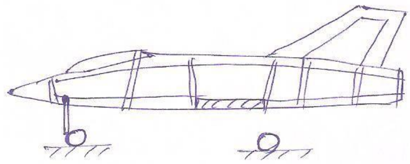
Structural Considerations

(i) load paths :

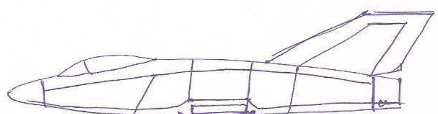
- Good Configuration designer will consider structural impacts of the arrangement of the aircraft.
- primary forces to be resolved are lift and weight.



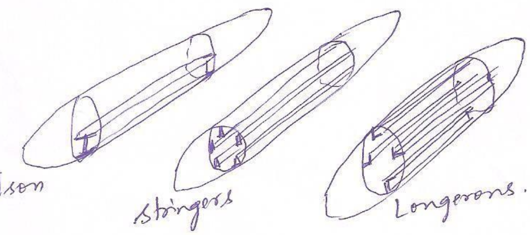




Structural arrangement



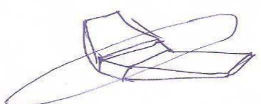
kinked lower longerons



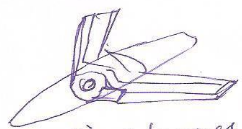
Keelson   Stringers   Longerons.

(ii) Carry through structure :

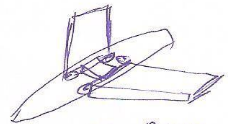
- lift Force produces tremendous bending moment at the junctions of wing and Fuselage.



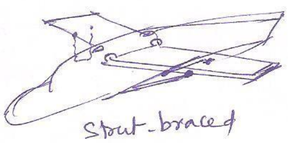
wing box carry through



Ring frames



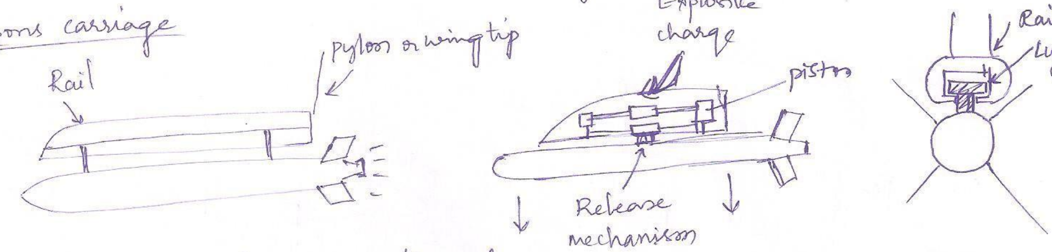
Bending Beam



Strut-braced

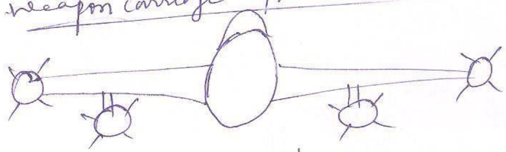
Fuselage Design

Weapons carriage



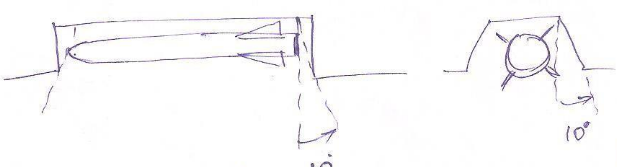
Missile carriage/launch.

weapon carriage options

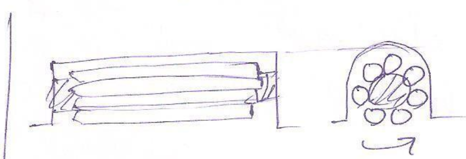


External

Semi submerged   Internal   Conformal.



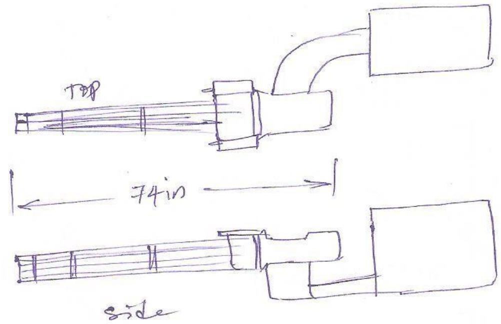
Weapon release clearance.



Rotary weapons bay

Gun installation

primary weapon of the air-to air fighter.
The standard U.S. air-to-air gun ~~tot~~ is the M61A1 Vulcan Six barrel gatling gun.



M61 Vulcan gun

→ when a gun is fired, it produces a bright flash and a large cloud of smoke this should not obscure the pilot vision.
→ Being very noisy, smoke, a gun should be located away from cockpit.

Wing tips,

Tail Geometry and arrangement

Tail arrangement

* Tail Configuration

Tail arrangement for Spin Recovery.

UNIT 3

PROPULSION AND FUEL SYSTEM INTEGRATION, LANDING GEAR AND SUBSYSTEMS

3.1 ENGINE PERFORMANCE:

As a rule, engine performance (thrust or power and fuel flow vs altitude and Mach number) should be determined using engine decks provided by the engine manufacturer. If these are not available, calculations based on thermodynamic and gas-dynamic analysis can be used. The guidelines given in Chapter 5 are limited, but can be used if other information is not available. One new type of engine which has been tested but which has never been in production is the convertible‖ engine. This power plant can act as both turbo-shaft and turbofan. The conversion between them is enacted through a series of clutches and shutters. As a rule of thumb, the thrust available under a set of atmospheric and operating conditions is approximately 1.2 times the available horsepower (i.e., T (lbs) = 1.2 (lb/hp) × hp). Thus the thrust-specific fuel consumption becomes the power-specific fuel consumption divided by 1.2. (Thrust-specific fuel consumption has units of lbs fuel/ lb-hr.)

The landing gear in aviation is the structure that supports an aircraft on the ground and allows it to taxi, takeoff and to land. It is a complex structure capable of reaching the largest local loads on the aircraft. In one brief moment, the landing gear must make the best of returning the aircraft from its natural environment to hostile environment-the earth.

3.2 TYPES OF LANDING GEARS:

The basic types of landing gear are:

1) Triangulated
2) Trailing link or levered
3) Cantilever
4) Articulated
5) Oleo-pneumatic

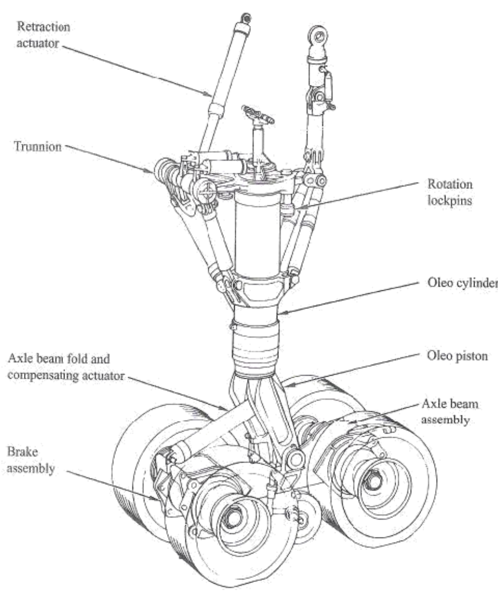


Fig 3.1 Landing Gear

3.3 PURPOSE:

The purpose of landing gear is

1) To provide smooth ride and steering the aircraft on taxiing
2) To absorb kinetic shocks during landing touchdown
3) To stop the aircraft with wheel brakes during runway operation
4) Provide adequate tail down angle for takeoff rotation &
5) To provide the stable support while ground parking

3.4 LOCATION OF LANDING GEARS ON THE AIRCRAFT:

Main gear is located at the best position for structural pick up. It is located depending on the stability criteria i.e. if the wing span is long then main gear is located on the wings, if the wing span is less it is located on the fuselage. the main gear tires cannot be selected until the static load has been determined, and to do this nose landing gear must be selected. It should be place as far as possible to minimize its load, minimize elevator power required for takeoff minimize main gear load for maximum retardation during braking, maximize flotation, and maximize stability. Conversely, the load should not be too light either, in that event steering would be difficult. The static and dynamic loads acting are calculated using the following formulae.

Let W= maximum gross weight

Maximum static main gear load (per strut)=W (F-M) / 2F

Maximum static nose gear load=W (F-L) / F

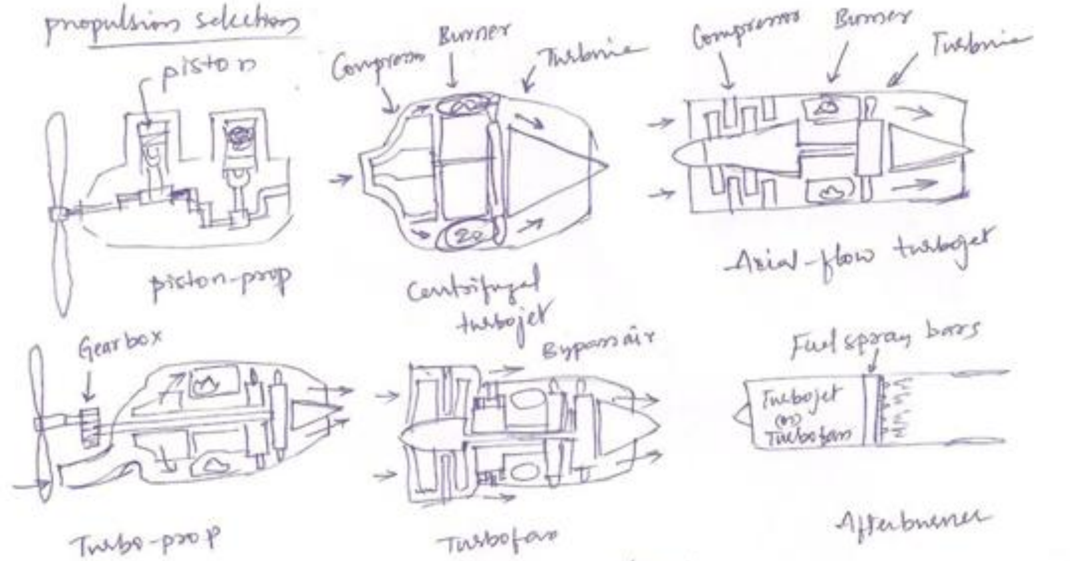
Minimum static nose gear load=W (F-N) / F

Maximum braking nose gear load=Max. Static load + 10H .W / 32.2 F (With 10 ft/sec^2 deceleration)

If the minimum static nose is too gear load small, the nose gear must be moved aft or the main gear must be moved aft. If the maximum static nose gear load is too high, the reverse procedure must be used i.e., move the nose gear forward or move the main gear forward. Hence, it is necessary to move both nose and main gear to obtain a satisfactory overall compromise in the overall loading.

propulsion and fuel system integration

propulsion selection



propulsion system options

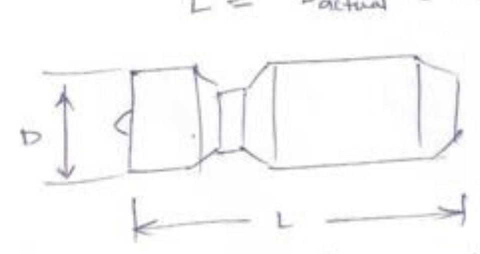
→ Bypass ratio $= \dfrac{\dot{m}_{air}}{\dot{m}_{in}} = 6$ high,

→ High temperature turbine blades are used upto 2000–2500°F, which require $\dfrac{\dot{m}_{air}}{\dot{m}_{fuel}} = 60$ to 1, only 25% of the total air is used in the combustion chamber,

Engine Dimensions

The scale factor "SF" is the ratio between the required thrust and the actual thrust of the nominal engine.

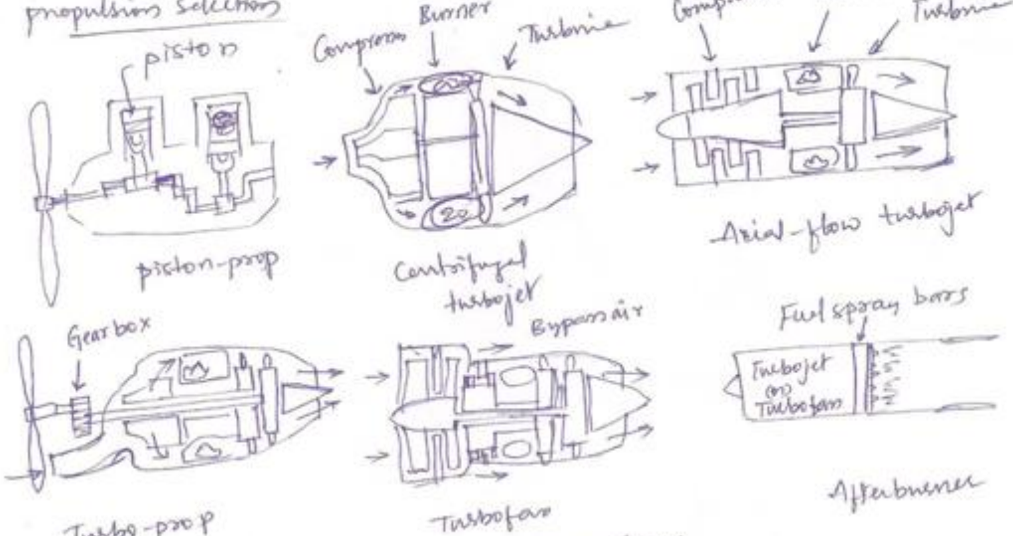
$$L = L_{actual}\,(SF)^{0.4},\quad D = D_{actual}\,(SF)^{0.5},\quad W = W_{actual}\,(SF)^{1.1}.$$



Engine scaling, Scale factor $= \dfrac{T_{required}}{T_{actual}} = (SF)$.

propulsion and Fuel system integration

propulsion selection



propulsion system options

→ Bypass ratio $= \frac{\dot{m}_{out}}{\dot{m}_{in}} = 6$ high,

→ High temperature turbine blades are used upto 2000–2500°F, which require $\frac{\dot{m}_{air}}{\dot{m}_{fuel}} = 60$ to 1, only 25% of the total air is used in the combustion chamber.

Engine Dimensions

The scale factor "SF" is the ratio between the required thrust and the actual thrust of the nominal engine.

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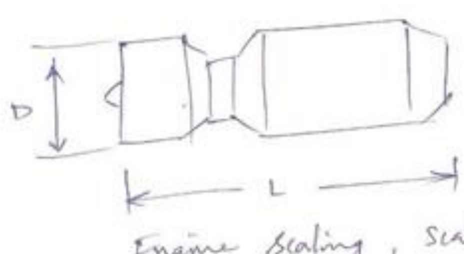
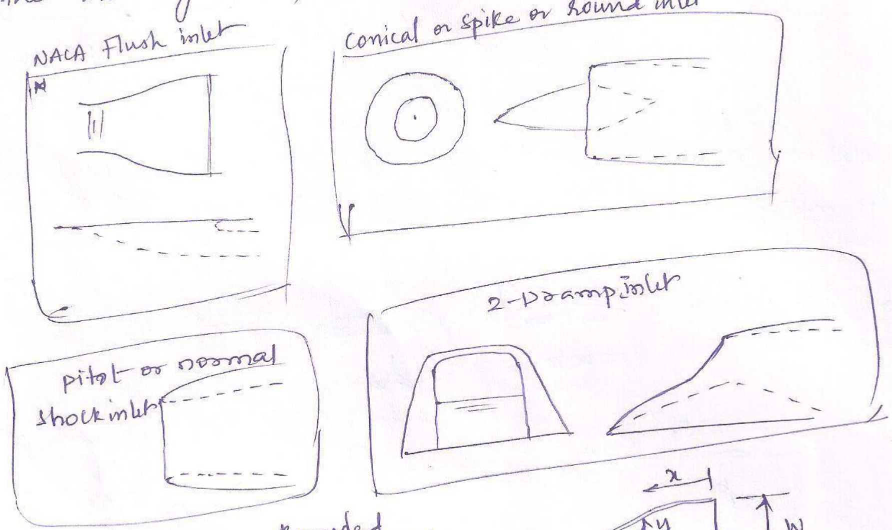


Engine scaling, Scale factor $= \frac{T_{required}}{T_{actual}} = (SF)$.

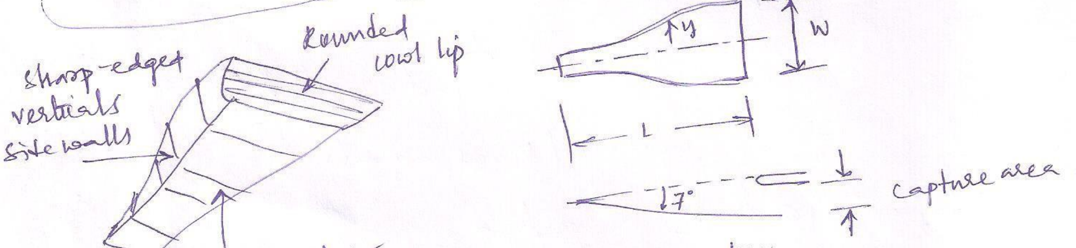
Inlet Geometry

→ for Turbojet, Turbofan engines, the air inlet should be slowed down after entering the engine inlet duct, for Compression.

- Tip speed of the compressor blades below sonic speed relative to the incoming air, slows down the

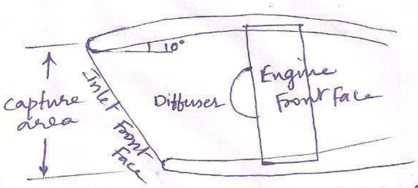


Inlet types:

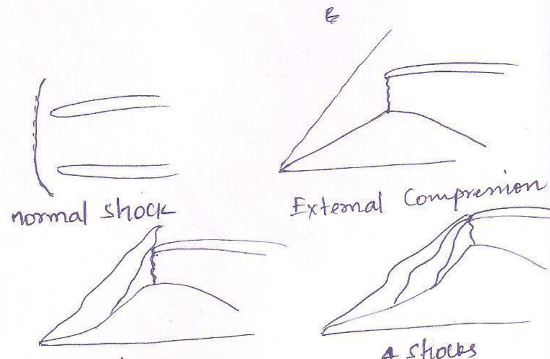


Flush inlet geometry

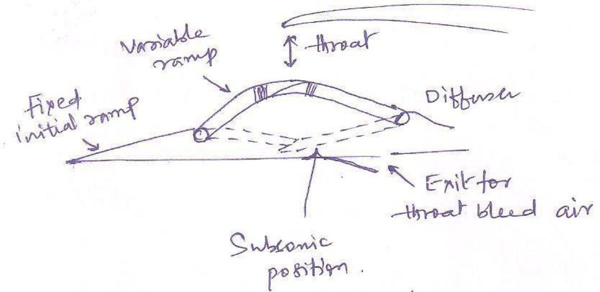
→ This flush inlet will provide 92% pressure recovery when operating at mass flow ratio 0.5.



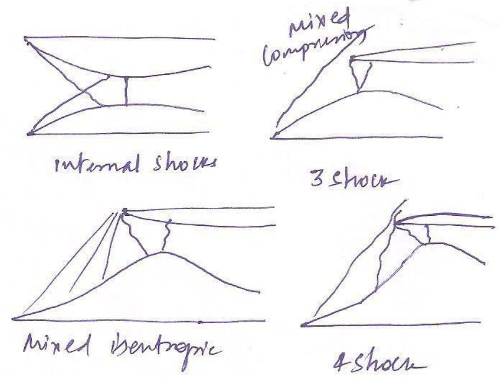
pit normal shock inlet layout



Supersonic inlets – External shocks.



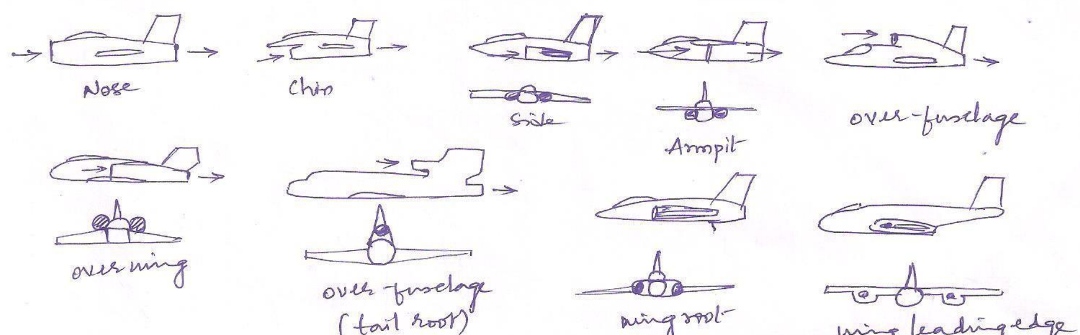
- Variable inlet geometry



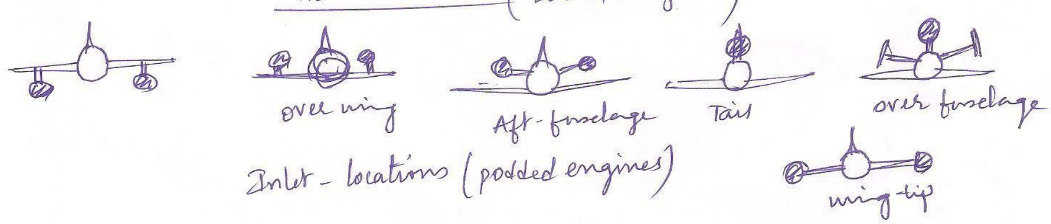
Supersonic inlets - internal shock

Inlet locations

→ The inlet locations can have almost great effect on engine performance as the inlet geometry.



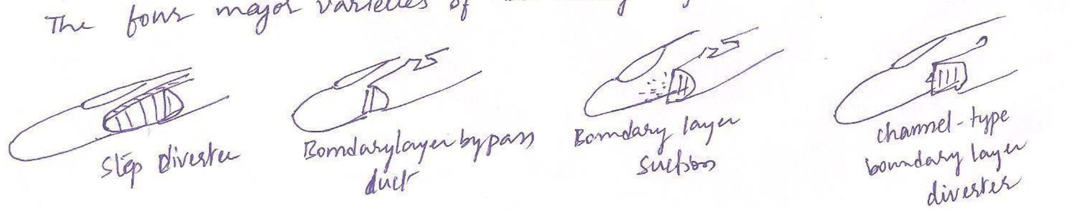
Inlet - locations (Buried engines)



Inlet - locations (podded engines)

Boundary layer Diverters

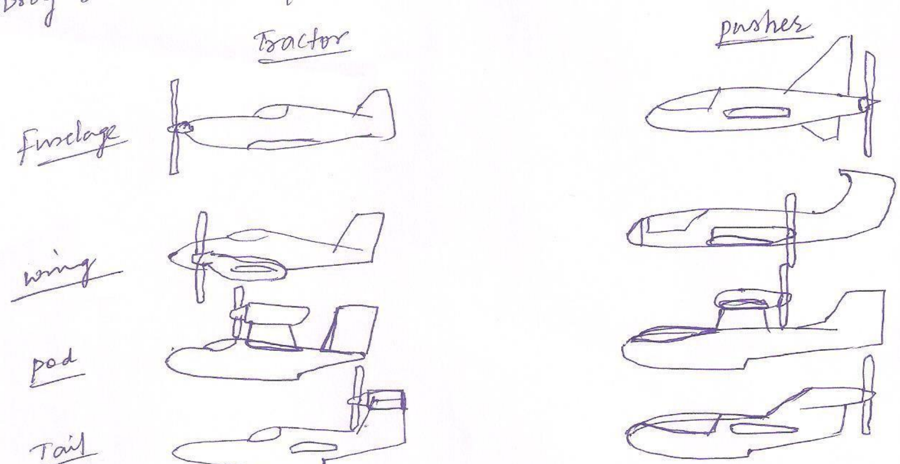
→ Boundary layer bleed was used to remove the low-energy boundary layer air from the compression ramps, to prevent shock-induced separation. The four major varieties of Boundary layer diverters are.



propeller - Engine integration

→ A pusher location has the propeller behind the attachment point.
→ A tractor installation has the propeller infront.

The pusher location has been important, it can reduce aircraft skin friction drag because the pusher location allows the aircraft to fly in undisturbed air.



propeller locations matrix

UNIT 4
BASELINE DESIGN- AERODYNAMICS AND PROPULSION, STRUCTURES, WEIGHT AND BALANCES

4.1 LIFT TO DRAG RATIO:

There are several parameters that are fundamental to understanding performance. These parameters do not necessarily improve our understanding of how or why airplanes fly but are a useful aid to understand airplane performance. The most important aerodynamic parameter is the lift-to-drag ratio, often referred to as L over D and written L/D. Anyone interested in airplanes has likely heard these words at one time or another. The L/D combines lift and drag into a single number that can be thought of as the airplane's efficiency for flight. Since lift and drag are both forces, L/D has no dimensions. A higher value of L/D means that the airplane is producing lift more efficiently.

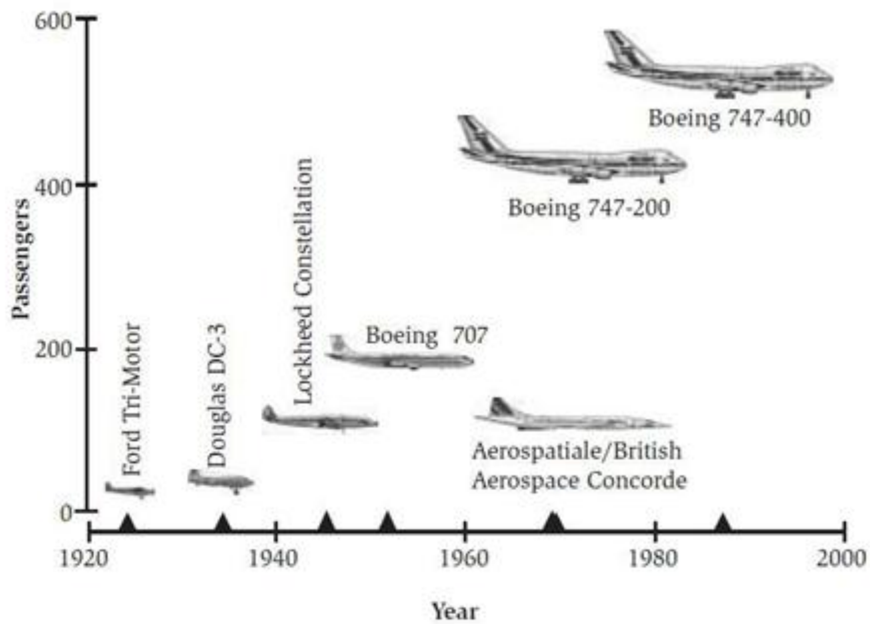


Fig 4.1 Increase in no of passengers carried

4.2 AERODYNAMIC FORCES ON A WING:

Lift is the force that holds an aircraft in the air. How is lift generated? There are many explanations and a bit of controversy on this subject. Many of the common explanations given for how airplanes or birds attain lift are incorrect or misleading. There are two accepted descriptions, namely, the Bernoulli and Newton positions, which arise from the Bernoulli equation and Newton's second law. Both are discussed in this section, and you will see that both explanations have merit. The Bernoulli equation describes lift as generated by a pressure difference across the wing, and Newton's position states that lift is the reaction force on a body caused by deflecting a flow of gas. A body moving through a fluid generates a force by the pressure variation around the body. Turning a moving fluid also generates a force on a solid body, although you must be careful to consider the flow turning on both surfaces of an airfoil, for otherwise an incorrect theory results.

The dynamic wing is a lifting surface that generates lift as a reaction to the airflow passing over it. Lift L signifies aerodynamic forces perpendicular to velocity v, or free stream (often written as v∞, while drag D represents aerodynamic forces parallel to the free stream. The angle at which the chord c of the airfoil moves in relation to the free stream is known as the angle of attack a.

Lift and drag are seen to ―act‖ through the quarter chord, or c/4, of an airfoil.

The lift (and drag) of a given airfoil shape can be complex to calculate due to the multiple factors involved, such as airfoil geometry, flow speed, and flow type.

The non dimensional lift and drag terms are called the coefficient of lift and drag, respectively. The lift coefficient and the drag coefficient are

$$C_L = \frac{L}{\frac{1}{2}\rho v^2 S}$$

$$C_D = \frac{D}{\frac{1}{2}\rho v^2 S}$$

Note that rv2 has units of pressure [(kg/m3)(m2/s2)_N/m2]. As in buoyancy, lift is still a pressure difference, but here it reveals the pressure above and below a lifting surface such as a wing. This pressure rv2 is called dynamic pressure and is represented by the symbol q.

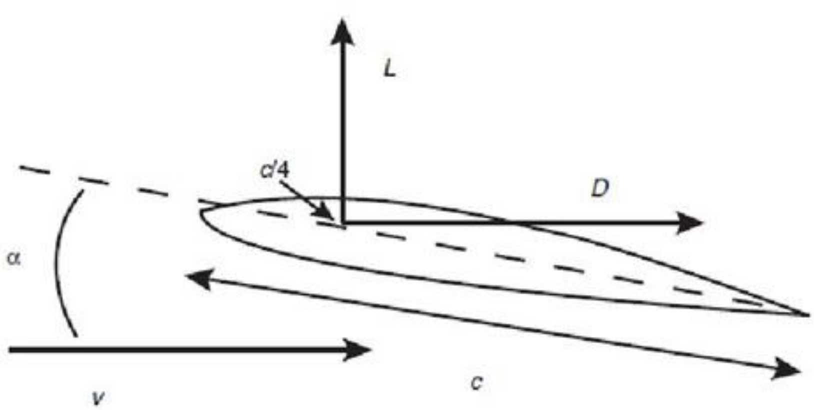


Fig 4.2 Aerofoil section with lift and drag forces depicted through quarter chord point

4.3 V-n DIAGRAM:

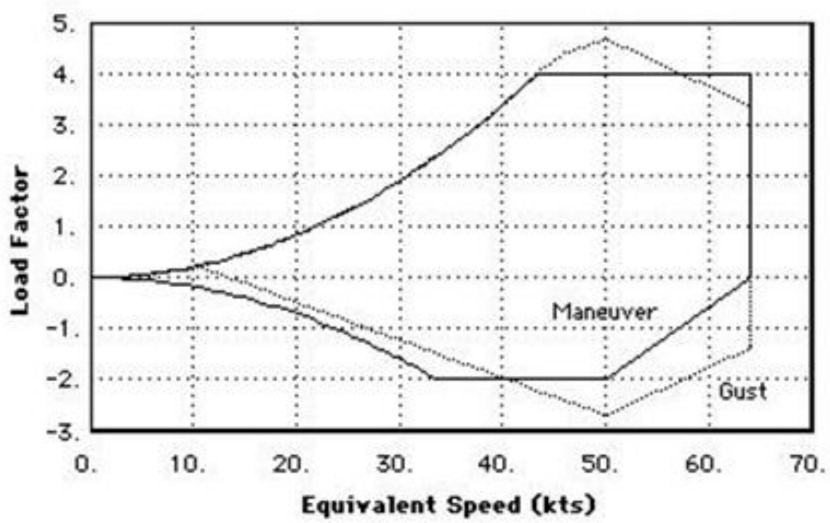


Fig 4.3 V-n Diagram

4.4 MANEUVER DIAGRAM:

This diagram illustrates the variation in load factor with airspeed for maneuvers. At low speeds the maximum load factor is constrained by aircraft maximum C_L. At higher speeds the maneuver load factor may be restricted as specified by FAR Part 25.

The maximum maneuver load factor is usually +2.5 . If the airplane weighs less than 50,000 lbs., however, the load factor must be given by: n= 2.1 + 24,000 / (W+10,000) n need not be greater than 3.8. This is the required maneuver load factor at all speeds up to V_c, unless the maximum achievable load factor is limited by stall.

The negative value of n is -1.0 at speeds up to V_c decreasing linearly to 0 at V_D. Maximum elevator deflection at V_A and pitch rates from V_A to V_D must also be considered.

4.5 GUST DIAGRAM:

Loads associated with vertical gusts must also be evaluated over the range of speeds.

The FAR's describe the calculation of these loads in some detail. Here is a summary of the method for constructing the V-n diagram. Because some of the speeds (e.g. V_B) are determined by the gust loads, the process may be iterative. Be careful to consider the alternative specifications for speeds such as V_B.

The gust load may be computed from the expression given in FAR Part 25. This formula is the result of considering a vertical gust of specified speed and computing the resulting change in lift. The associated incremental load factor is then multiplied by a load alleviation factor that accounts primarily for the aircraft dynamics in a gust.

4.6 MATERIAL SELECTION:

The basic question is how do we go about selecting a material for a given part? This may seem like a very complicated process until we realize than we are often restrained by choices we have already made. For example, if different parts have to interact then material choice becomes limited. When we talk about choosing materials for a component, we take into account many different factors. These factors can be broken down into the following areas.

1) Material Properties
    a) The expected level of performance from the material
2) Material Cost and Availability.
    a) Material must be priced appropriately (not cheap but right)
    b) Material must be available (better to have multiple sources)
3) Processing
    a) Must consider how to make the part,
    b) for example: Casting, Machining, Welding
4) Environment
    a) The effect that the service environment has on the part
    b) The effect the part has on the environment
    c) The effect that processing has on the environment
5) Kinds of Materials.
    a) Iron
    b) Aluminum
    c) Copper
    d) Magnesium
    e) Composites
    f) Ceramics
    g) Glass
    h) Semi-conductors
    i) Structural ceramics (SiN, SiC)
    j) Refractory Composites
    k) Polymers
    l) Rubber
    m) Plastics

UNIT 5
BASELINE DESIGN- STABILITY AND CONTROL, PERFORMANCE AND CONSTRAINY ANALYSIS

5.1 INTRODUCTION:

The final design of horizontal and vertical tails requires calculation of the dynamic stability and response of the airplane in cases like:

(a) Different flight conditions

(b) Different weights and c.g. locations and

(c) Possible variations in configuration.

The topics dealt with in this chapter are based on the following two observations.

1) For conventional subsonic airplanes, if the airplane has adequate level of static stability, then it would have reasonable dynamic stability.
2) If the areas of the control surfaces are adequate to trim the airplane (i.e. bring the moments about the three airplane axes to zero) in certain critical conditions, then the airplane would have reasonable level of controllability.

5.2 NEUTRAL POINT:

It is known that the c.g. of the airplane moves during the flight. Further, the contribution of wing to Cmα depends sensitively on the location of the c.g. When the c.g. moves aft, xcg increases and the wing contribution becomes less and less negative or more and more positive.

There is a c.g. location at which (Cmα)stick-fixed becomes zero. This location of c.g. is called the stick-fixed neutral point. In this case, the airplane is neutrally stable. The location of the neutral point can be obtained by putting Cmα = 0.

It is noted that the contribution of tail to (Cmα) changes when the stick is fixed or free. It may be recalled, from course on stability analysis, that in stick-free case, the elevator is free to move about its hinge**.**

5.3: PERFORMANCE ESTIMATION:

After carrying out the stability analysis, the major dimensions of the airplane have been arrived at. This will enable preparation of the revised three view drawing. Using this drawing and the flight conditions, a drag polar of the airplane can be estimated.

The performance analysis includes the following:

1) The variation of stalling speed (Vs) at various altitudes
2) Variations with altitude of maximum speed (Vmax) and minimum speed from power output consideration (Vmin) power. The minimum speed of the airplane at an altitude

will be the higher of Vs and (Vmin) power. The maximum speed and minimum speed will decide the flight envelope.

3) Variations with altitude of the maximum rate of climb ((R/C)max) and maximum angle of climb (γmax) ; the flight being treated as steady climb. Variations with altitude of V(R/C)max and Vγmax. To arrive at these quantities choose a set of altitudes and at each of these altitudes, obtain the R/C and γ at different flight velocities. From the plot of (R/C)max vs. h, the values of absolute ceiling and service ceiling can be obtained. At absolute ceiling (R/C)max is zero and at service ceiling (R/C)max is 30 m/min. For multi-engined airplanes, the rate of climb with one engine inoperative must satisfy the airworthiness regulations.
4) To arrive at the cruising speed and altitude, choose a range of altitudes around the cruising altitude mentioned in the specifications. At each of these altitudes obtain the range in constant velocity flights choosing different velocities. The information on appropriate values of specific fuel consumption (SFC) can be obtained from the engine charts. The values of range obtained at different speeds and altitudes be plotted as range vs velocity curves with altitude as parameter. Draw an envelope of this curves.The altitude and velocity at which the range is maximum can be considered as the cruising speed (Vcruise) and cruising altitude (hcruise). These curves also give information about the range of flight speeds and altitudes around Vcruise and hcruise at which near optimum performance is obtained.
5) The maximum rate of turn (max ψ) and the minimum radius of turn (rmin) in steady level turn depend on the thrust available,CLmax and the permissible load factor (nmax). The value of CLmax used here is that without the flaps. For high speed airplanes the value of C Lmax depends also on Mach number. The value of nmax depends on the weight and the type of airplane. Choose a set of altitudes and at each of these altitudes obtain the values of Vψ max and Vrmin. From plots of these quantities obtain variations, with altitude, of rmin, and Vrmin.
6) Take - off run and take - off distance: During take-off an airplane accelerates on the ground. For an airplane with nose wheel type of landing gear, around a speed of 85% of the take-off speed, the pilot pulls the stick back. Then, the airplane attains the angle of attack corresponding to take-off and the airplane leaves the ground. The point at which the main wheels leave the ground is called the unstick point and the distance from the start of take-off point to the unstick point is called the ground run. After the unstick, the airplane goes along a curved path as lift is more than the weight. This phase of take-off is called transition at the end of which the airplane climbs along a straight line. The take-off phase is said to be over when the airplane attains screen height which is generally 15 m above the ground. The horizontal distance from the start of the take off to the where the airplane attains screen height is called take off distance (Fig.3.4). The take off run and the take-off distance can be estimated by writing down equations of motion in different phases.

7) Landing Distance: The landing flight begins when the airplane is at the screen height at a velocity called the approach speed. During the approach phase the airplane descends along a flight path of about 3 degrees. Subsequently the flight path becomes horizontal in the phase called _flare‘. In this phase the pilot also tries to touch the ground gently. The point where the main wheels touch the ground is called touchdown point. Subsequent to touch down, the airplane rolls along the ground for about 3 seconds during which the nose wheel touches the ground. This phase is called free roll. After this phase the brakes are applied and the airplane comes to halt. In some airplanes, thrust in the reverse direction is produced by changing the direction of jet exhaust or by reversible pitch propeller. In some airplanes, the drag is increased by speed brakes, spoilers or parachutes. For airplanes which land on the deck of the ship, an arresting gear is employed to reduce the landing distance. The horizontal distance from the start of approach at screen height till the airplane comes to rest is called landing distance.

5.4 BALANCED FIELD LENGTH:

It is the length of the run way required from, consideration of safety in the event of engine failure. If the engine fails soon after the aircraft begins to roll, the pilot can stop the airplane without difficulty. If the engine fails when the airplane is near the unstick point, then he should not have difficulty in completing the take-off. The speed of the airplane, during take-off, at which the distance to stop after an engine failure equals the distance to continue the takeoff on the remaining engine (s) is called a decision speed. The balanced field length is the take-off distance to clear the screen height when one engine fails at the decision speed. Subsection 4.8.1 can be referred to for estimating the balance field length.

Remarks:

1) The landing distance is considerably affected by piloting techniques. To take into account the uncertainty, the landing distance is multiplied by 1.67 to get the FAR/EASA(European Air Safety Agency) landing distance.
2) Take-off and landing distance calculations are repeated assuming different altitudes of air fields and different atmospheric conditions e.g. A + 200 C i.e. the sea level temperature is 350 C instead of 150 C in the standard atmosphere.

5.5 GENERAL REMARKS ON PERFORMANCE ESTIMATION:

5.5.1 OPERATING ENVELOPE:

The maximum speed and minimum speed can be calculated from the level flight analysis. However, the attainment of maximum speed may be limited by other considerations. The operating envelope for an airplane is the range of flight speeds permissible at different altitudes.

5.5.2 ENERGY HEIGHT TECHNIQUE FOR CLIMB PERFORMANCE:

The analysis of a steady climb shows that the velocity corresponding to maximum rate of climb (V(R/C) max) increases with altitude. Consequently, climb with (R/C) max involves acceleration and the rate of climb will actually be lower than that given by the steady climb analysis. This is because a part of the engine output would be used to increase the kinetic energy.

Secondly, the aim of the climb is to start from velocity near Vtake-off and at htake-off and attain a velocity near Vcruise at hcruise.

5.5.3 RANGE PERFORMANCE:

For commercial airplanes the range performance is of paramount importance. Hence, range performance with different amounts of payload and fuel on board the airplane, needs to be worked out. In this context the following three limitations should to be considered.

a) **MAXIMUM PAYLOAD:**
The number of seats and the size of the cargo compartment are limited. Hence maximum payload capacity is limited.

b) **MAXIMUM FUEL:**
The size of the fuel tanks depends on the space in the wing and the fuselage to store the fuel. Hence, there is limit on the maximum amount of fuel that can be carried by the airplane.

c) **MAXIMUM TAKE-OFF WEIGHT:**
The airplane structure is designed for a certain load factor and maximum takeoff weight.

5.6 CASE STUDIES AND DESIGN OF UNIQUE AIRCRAFT CONCEPTS:

5.6.1 DC 1:

Development of the DC-1 can be traced back to the 1931 crash of TWA Flight 599, which was caused by a Fokker F.10 Trimotor airliner suffering a structural failure of one of its wings, probably due to water which had over time seeped between the layers of the wood laminate and dissolved the glue holding the layers together. Following the accident, the Aeronautics Branch of the U.S. Department of Commerce placed stringent restrictions on the use of wooden wings on passenger airliners. Boeing developed an answer, the 247, a twin-engined all-metal monoplane with a retractable undercarriage, but their production capacity was reserved to meet the needs of United Airlines, part of United Aircraft and Transport Corporation which also owned Boeing. TWA needed a similar aircraft to respond to competition from the Boeing 247 and they asked five manufacturers to bid for construction of a three-engined, 12-seat aircraft of all-metal construction, capable of flying 1,080 mi (1,740 km) at 150 mph (242 km/h). The most demanding part of the specification was that the airliner would have to be capable of safely taking off from any airport on TWA's main routes (and in particular Albuquerque, at high altitude and with severe summer temperatures) with one engine non-functioning.

Fig 5.1 DC 1

Donald Douglas was initially reluctant to participate in the invitation from TWA. He doubted that there would be a market for 100 aircraft, the number of sales necessary to cover development costs. Nevertheless, he submitted a design consisting of an all-metal, low-wing, twin-engined aircraft seating 12 passengers, a crew of two and a flight attendant. The aircraft exceeded the specifications of TWA even with two engines. It was insulated against noise, heated, and fully capable of both flying and performing a controlled takeoff or landing on one engine.

Don Douglas stated in a 1935 article on the DC-2 that the first DC-1 cost $325,000 to design and build.

5.6.2 DC 2:

In the early 1930s, fears about the safety of wooden aircraft structures (responsible for the crash of a Fokker Trimotor) compelled the American aviation industry to develop all-metal types. The United Aircraft and Transport Corporation had a monopoly on the Boeing 247; rival Transcontinental and Western Air issued a specification for an all-metal trimotor.

The Douglas response was more radical. When it flew on July 1, 1933, the prototype DC-1 had a robust tapered wing, retractable landing gear, and two 690 hp (515 kW) Wright radial engines driving variable-pitch propellers. It seated 12 passengers.

Fig 5.2 DC 2

TWA accepted the basic design and ordered twenty DC-2s having more powerful engines and a bit more length, to carry 14 passengers in a 66-inch-wide cabin. The design impressed American and European airlines and further orders followed. Those for European customers KLM, LOT, Swissair, CLS and LAPE were assembled by Fokker in the Netherlands after that company bought a licence from Douglas.Airspeed Ltd. took a similar licence for DC-2s to be delivered in Britain and assigned the company designation Airspeed AS.23, but although a registration for one aircraft was reserved none were actually delivered. Another licence was taken by the Nakajima Aircraft Company in Japan; unlike Fokker and Airspeed, Nakajima built five aircraft as well as assembling at least one Douglas-built aircraft. A total of 130 civil DC-2s were built with another 62 for the United States military. In 1935 Don Douglas stated in an article that the DC-2 cost about $80,000 per aircraft if mass-produced.

5.6.3 DC 3:

The designation "DC" stands for "Douglas Commercial". The DC-3 was the culmination of a development effort that began after an inquiry from Transcontinental and Western Airlines (TWA) to Donald Douglas. TWA's rival in transcontinental air service, United Airlines, was starting service with the Boeing 247 and Boeing refused to sell any 247s to other airlines until United's order for 60 aircraft had been filled. TWA asked Douglas to design and build an aircraft to allow TWA to compete with United. Douglas' design, the 1933 DC-1, was promising, and led to the DC-2 in 1934. The DC-2 was a success, but there was room for improvement.

Fig 5.3 DC 3

The DC-3 resulted from a marathon telephone call from American Airlines CEO C. R. Smith to Donald Douglas, when Smith persuaded a reluctant Douglas to design a sleeper aircraft based on the DC-2 to replace American's Curtiss Condor II biplanes. (The DC-2's cabin was 66 inches wide, too narrow for side-by-side berths.) Douglas agreed to go ahead with development only after Smith informed him of American's intention to purchase twenty aircraft. The new aircraft was engineered by a team led by chief engineer Arthur E. Raymond over the next two years, and the prototype DST (for Douglas Sleeper Transport) first flew on December 17, 1935 (the 32nd anniversary of the Wright Brothers' flight at Kitty Hawk). Its cabin was 92 inches wide, and a version with 21 seats instead of the 14-16 sleeping berths of the DST was given the designation DC-3. There was no prototype DC-3; the first DC-3 built followed seven DSTs off the production line and was delivered to American Airlines.

The DC-3 and DST popularized air travel in the United States. Eastbound transcontinental flights could cross the U.S. in about 15 hours with three refueling stops; westbound trips against the wind took 17½ hours. A few years earlier such a trip entailed short hops in slower and shorter-range aircraft during the day, coupled with train travel overnight

A variety of radial engines were available for the DC-3. Early-production civilian aircraft used Wright R-1820 Cyclone 9s, but later aircraft (and most military versions) used the Pratt & Whitney R-1830 Twin Wasp which gave better high-altitude and single engine performance. Three DC-3S Super DC-3s with Pratt & Whitney R-2000 Twin Wasps were built in the late 1940s.

5.6.4 B 47:

The Boeing B-47 Stratojet (company Model 450) was a long range, six-engine, jet-powered strategic bomber designed to fly at high subsonic speed and at high altitude to avoid enemy interceptor aircraft. The B-47's primary mission was to drop nuclear bombs on the Soviet Union. With its engines carried in nacelles under the swept wing, the B-47 was a major innovation in post-World War II combat jet design, and contributed to the development of modern jet airliners.

Fig 5.4 B 47

The B-47 entered service with the United States Air Force's Strategic Air Command (SAC) in 1951. It never saw combat as a bomber, but was a mainstay of SAC's bomber strength during the late 1950s and early 1960s, and remained in use as a bomber until 1965. It was also adapted to a number of other missions, including photographic reconnaissance, electronic intelligence and weather reconnaissance, remaining in service as a reconnaissance platform until 1969 and as a testbed until 1977.

5.6.5 B 707:

The Boeing 707 is a mid-size, long-range, narrow-body four-engine jet airliner built by Boeing Commercial Airplanes from 1958 to 1979. Its name is commonly pronounced as "Seven Oh Seven". Versions of the aircraft have a capacity from 140 to 189 passengers and a range of 2,500 to 5,750 nautical miles (4,630 to 10,650 km).

Fig 5.5 B707

Developed as Boeing's first jet airliner, the 707 is a swept-wing design with podded engines. Although it was not the first jetliner in service, the 707 was the first to be commercially successful. Dominating passenger air transport in the 1960s and remaining common through the 1970s, the 707 is generally credited with ushering in the Jet Age. It established Boeing as one of the largest manufacturers of passenger aircraft, and led to the later series of airliners with "7x7" designations. The later 727, 737, and 757 share elements of the 707's fuselage design.

The 707 was developed from the Boeing 367-80, a prototype jet first flown in 1954. A larger fuselage cross-section and other modifications resulted in the initial production 707-120, powered by Pratt & Whitney JT3C turbojet engines, which first flew on December 20, 1957. Pan American World Airways began regular 707 service on October 26, 1958. Later derivatives included the shortened long-range 707-138 and the stretched 707-320, both of which entered service in 1959. A smaller short-range variant, the 720, was introduced in 1960. The 707-420, a version of the stretched 707 with Rolls-Royce Conway turbofans, debuted in 1960, while Pratt & Whitney JT3D turbofans debuted on the 707-120B and 707-320B models in 1961 and 1962, respectively.

The 707 has been used on domestic, transcontinental and transatlantic flights, and for cargo and military applications. A convertible passenger-freighter model, the 707-320C, entered service in 1963, and passenger 707s have been modified to freighter configurations. Military derivatives include the E-3 Sentry airborne reconnaissance aircraft and the C-137 Stratoliner VIP transports. Boeing produced and delivered 1,011 airliners including the smaller 720 series; over 800 military versions were also produced. There were 10 Boeing 707s in commercial service in July 2013.

5.6.6 GENERAL DYNAMICS F 16:

The General Dynamics (now Lockheed Martin) F-16 Fighting Falcon is a single-engine multirole fighter aircraft originally developed by General Dynamics for the United States Air Force (USAF). Designed as an air superiority day fighter, it evolved into a successful all-weather multirole aircraft. Over 4,500 aircraft have been built since production was approved in 1976. Although no longer being purchased by the U.S. Air Force, improved versions are still being built for export customers. In 1993, General Dynamics sold its aircraft manufacturing business to the Lockheed Corporation, which in turn became part of Lockheed Martin after a 1995 merger with Martin Marietta.

Fig 5.6 F 16

The Fighting Falcon has key features including a frameless bubble canopy for better visibility, side-mounted control stick to ease control while maneuvering, a seat reclined 30 degrees to reduce the effect of g-forces on the pilot, and the first use of a relaxed static stability/fly-by-wire flight control system helps to make it a nimble aircraft. The F-16 has an internal M61 Vulcan cannon and 11 locations for mounting weapons and other mission equipment. The F-16's official name is "Fighting Falcon", but "Viper" is commonly used by its pilots, due to a perceived resemblance to a viper snake as well as the Battlestar Galactica Colonial Viper starfighter.

In addition to active duty U.S. Air Force, Air Force Reserve Command, and Air National Guard units, the aircraft is also used by the USAF aerial demonstration team, the U.S. Air Force Thunderbirds, and as an adversary/aggressor aircraft by the United States Navy. The F-16 has also been procured to serve in the air forces of 25 other nations.

5.6.7 SR 71 BLACK BIRD:

The Lockheed SR-71 "Blackbird" is a long-range, Mach 3+ strategic reconnaissance aircraft that was operated by the United States Air Force. It was developed as a black project from the Lockheed A-12 reconnaissance aircraft in the 1960s by Lockheed and its Skunk Works division. Renowned American aerospace engineer Clarence "Kelly" Johnson was responsible for many of the design's innovative concepts. During reconnaissance missions, the SR-71 operated at high speeds and altitudes to allow it to outrace threats. If a surface-to-air missile launch was detected, the standard evasive action was simply to accelerate and outfly the missile. The SR-71 was designed to have basic stealth characteristics and served as a precursor to future stealth aircraft.

Fig 5.7 SR 71 BLACK BIRD

The SR-71 served with the U.S. Air Force from 1964 to 1998. A total of 32 aircraft were built; 12 were lost in accidents, but none lost to enemy action. The SR-71 has been given several nicknames, including Blackbird and Habu. Since 1976, it has held the world record for the fastest air-breathing manned aircraft, a record previously held by the YF-12.

5.6.8 NORTHROP GRUMMAN B2 STEALTH BOMBER:

The Northrop (later Northrop Grumman) B-2 Spirit, also known as the Stealth Bomber, is an American heavy strategic bomber, featuring low observable stealth technology designed for penetrating dense anti-aircraft defenses; it is able to deploy both conventional and thermonuclear weapons. The bomber has a crew of two and can drop up to eighty 500 lb (230 kg)-class (Mk 82) JDAM Global Positioning System-guided bombs, or sixteen 2,400 lb (1,100 kg) B83 nuclear bombs. The B-2 is the only known aircraft that can carry large air-to-surface standoff weapons in a stealth configuration.

Fig 5.8 NORTHROP GRUMMAN B2 STEALTH BOMBER

Development originally started under the "Advanced Technology Bomber" (ATB) project during the Carter administration, and its performance was one of his reasons for the cancellation of the supersonic B-1A bomber. ATB continued during the Reagan administration, but worries about delays in its introduction led to the reinstatement of the B-1 program as well. Program costs rose throughout development. Designed and manufactured by Northrop Grumman, the cost of each aircraft averaged US$737 million (in 1997 dollars). Total procurement costs averaged $929 million per aircraft, which includes spare parts, equipment, retrofitting, and software support. The total program cost including development, engineering and testing, averaged $2.1 billion per aircraft in 1997.

Because of its considerable capital and operating costs, the project was controversial in the U.S. Congress and among the Joint Chiefs of Staff. The winding-down of the Cold War in the latter portion of the 1980s dramatically reduced the need for the aircraft, which was designed with the intention of penetrating Soviet airspace and attacking high-value targets. During the late 1980s and 1990s, Congress slashed plans to purchase 132 bombers to 21. In 2008, a B-2 was destroyed in a crash shortly after takeoff, though the crew ejected safely. A total of 20 B-2s remain in service with the United States Air Force, which plans to operate the B-2 until 2058.

The B-2 is capable of all-altitude attack missions up to 50,000 feet (15,000 m), with a range of more than 6,000 nautical miles (11,000 km) on internal fuel and over 10,000 nautical miles (19,000 km) with one mid air refueling. Though originally designed primarily as a nuclear bomber, it was first used in combat dropping conventional ordnance in the Kosovo War in 1999 and saw further service in Iraq and Afghanistan.