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SCHOOL OF MECHANICAL ENGINEERING

DEPARTMENT OF AERONAUTICAL ENGINEERING

UNIT – I –PREREQUSITES TO EVALUATE AIRCRAFT PERFORMANCE - SEAE1501

1. PREREQUISITES TO EVALUATE AIRCRAFT PERFORMANCE

1.1 The role and design mission of an aircraft, specification of performance requirements

- Performance can be used as a measure of the capability of the aircraft in many ways.
- Performance can be defined as a measure of the ability of the aircraft to carry out a specified task.
- In the case of a civil transport aircraft it determines an element of the cost of the operation of the aircraft and hence it contributes to its economic viability as a transport vehicle.
- In military combat operations, time, manoeuvre and radius of action are some of the more critical performance parameters in the overall evaluation of the effectiveness and air superiority of the aircraft.
- Performance can also be regarded as a measure of safety.
- An aircraft has an excess of thrust over drag it can increase its energy by either climbing or accelerating if the drag exceeds the thrust then it will be losing energy as it either decelerates or descends.
- In safe flight the thrust must not be committed to a decrease of energy that would endanger it so that at all critical points in the mission thrust must exceed the drag. This is a consideration of the performance aspect of the airworthiness of the aircraft.
- The design of an aircraft starts from the statement of the flight path related performance that the aircraft is expected to achieve.
- The basic statement of the performance will be concerned with the payload the aircraft will be required to carry and the mission profile it will be required to fly.
- The payload of a civil transport aircraft may be defined in the terms of no. of passengers, tonnage of freight, volume of freight or as combinations of freight and passengers.
- The definition of military aircraft mission payloads may cover a wide range of possibilities including personnel, troops, support equipment and supplies in transport aircraft and internally carried stores, externally carried stores and sensor pods on combat aircraft.

Aircraft performance is a measurement of how well the plane flies. Many factors affect how the plane flies: its weight, the atmospheric conditions, pressure, temperature, humidity, even the runway at takeoff. Aircraft performance will tell you the speeds the plane can hope to achieve in different conditions. Aircraft must be able to operate safely throughout their flight regime in such a way that a safe outcome will result from specified malfunctions (e.g. power unit failure), occurring at any point throughout the flight range. Even though modern aircraft are designed and built according to strict standards which are laid down by national and international authorities due to off design conditions or some unexpected climatic changes may result in poor performance of aircraft. In the present study the performance will be based on flight path of aircraft in atmosphere which depends upon the mission profile of aircraft.Performance of an aircraft changes based upon its role and mission. In broad terms the aircraft role and mission can be classified into Civil Operations, which are commercial flights transporting passengers or cargo from one location to another, or Military Operations, which are concerned with defensive or offensive, flight operations or their associated support operations.

Civil operations are fundamentally, intended to profit making while military operations are more concerned with the need to achieve their objectives than with the cost of doing so; this distinction points to the different performance criteria that will need to be applied to the mission.

1.2 Mission profile

The design requirements stipulate the required performance of the new airplane. The airplane performance is critically dependent on several parameters, especially

(1) Maximum lift coefficient (CLmax);

(2) lift-to-drag ratio L/ D, usually at cruise;

(3) Wing loading W / S; and

(4) Thrust-to-weight ratio T / W.

A typical mission profile of a civil transport aircraft is shown in the above figure.

- The primary mission is to fly a payload from the departure point to the destination. This requires the aircraft to takeoff from the departure point, climb to the cruising height and cruise to the destination, where the aircraft descends and lands.

❖ If the aircraft be unable to land the destination when it arrives, it will have to divert to an alternate airfield and the flight plan will need to include provision for the diversion.

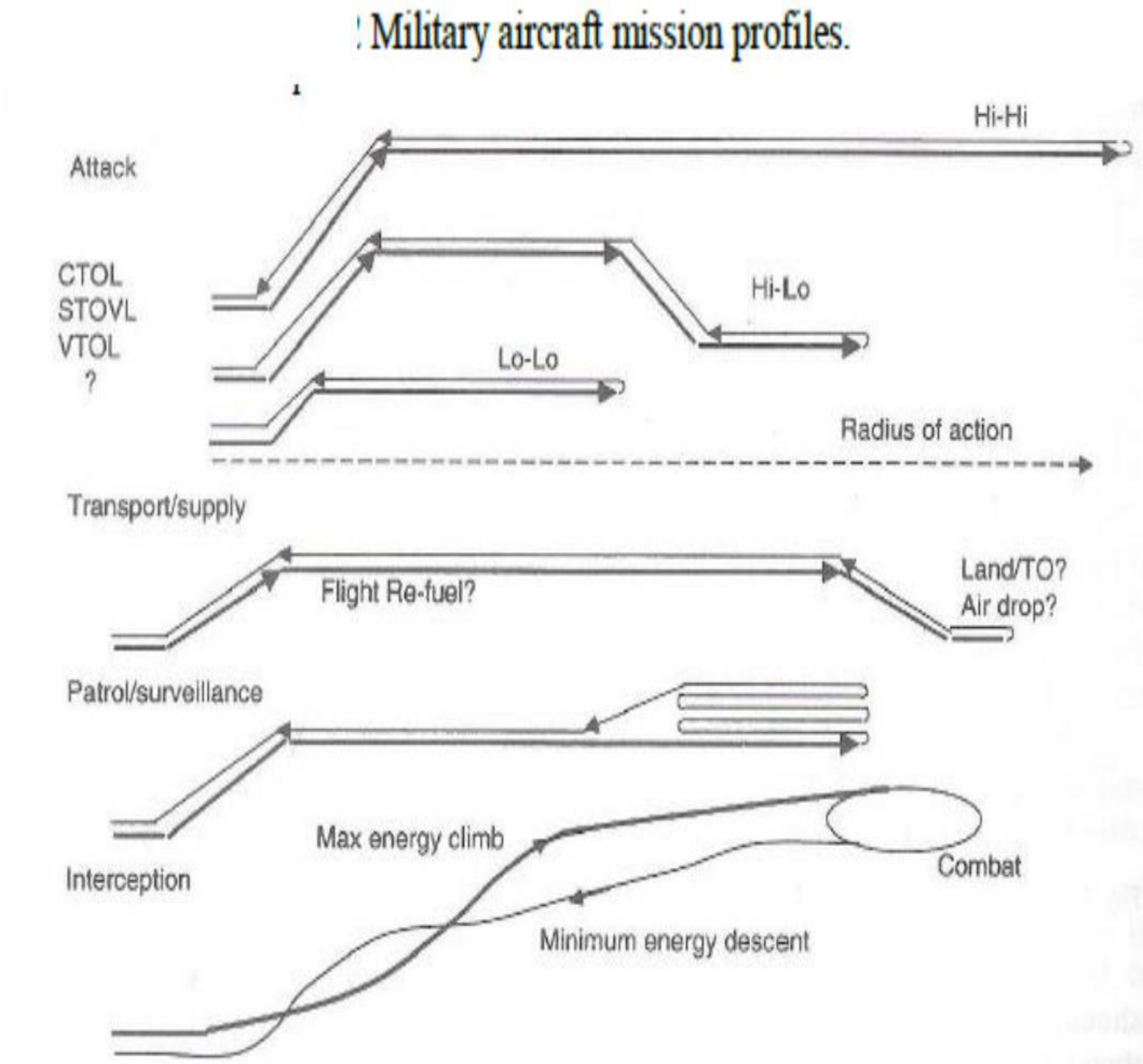

Figure 1.1 Mission profiles

Once the mission profile and the payload of the aircraft have been specified the design process can commence. From the performance standpoint the total design process extends from the initial project design estimations right through to the delivery of the aircraft into service, fig 1.1.

In the final phase the aircraft is prepared for its operational role. The overall procedure can be divided into 3 broad areas:

- Performance estimation
- Performance measurement
- Operational performance

1.3 Performance estimation

Performance estimation involves the prediction of the capabilities of the aircraft from the considerations of its aerodynamic design, power plant and operating environment.

It can be applied to

- ❖ The design of new type of aircraft
- ❖ Modification of existing aircraft type in respect to design changes affecting its aerodynamic characteristics or power plant or
- ❖ To supplement or extend the full scale measured performance of an aircraft type for conditions outside those already established

Performance estimation process begins with the proposal of some performance target. Initially the estimation will centre on individual elements of the flight path. Performance estimation is usually based on the assumption of a simple atmosphere model, the international standard atmosphere (ISA). This is a linearized model of the temperature atmosphere which represents the mean global atmosphere state with respect to seasonal changes and latitude and is used as the basis for aircraft design.

1.4 Performance measurement

A performance measurement is required for the three main purposes: It is required for three main purposes:

- ❖ To verify that the aircraft achieves the estimated design performance targets;
- ❖ To demonstrate that the aircraft can satisfy the safety criteria set down in the airworthiness requirements; and
- ❖ To provide validated performance data for the performance section of the flight manual.

- The performance of the aircraft is measured in development trials and compared with the estimated performance where there is difference in the characteristics of the aircraft and of the power plant can be measured and compared with those used in the models
- As the design of the aircraft is developed and the flight trials show that it is meeting its performance targets, data was measured for submission to the airworthiness authority for the certification of the aircraft.
- As a part of the certification process validated performance data are required for the performance section of the flight manual, known as the flight performance manual or the operating data manual(ODM), which contains the information of the performance of the aircraft needed by the operator for flight planning.

1.5 Operational Performance

The basic requirements for the safe flight are that the space required for the aircraft to manoeuvre should never exceed the space available, and that the aircraft carries sufficient fuel for the flight these fundamental requirements form the basis of performance planning and fuel planning.

1.5.1 Performance planning:

It is a part of the flight plan mode in advance of the flight ensures that at any point in the flight the aircraft has sufficient performance to be able to manoeuvre within the space available. The space required for any given manoeuvre is a function of the weight of the aircraft and the space required increases as the weight increases.

1.5.2 Fuel planning:

It ensures that the aircraft carries sufficient fuel for the mission, taking into account reserves for contingencies, diversions and safety. Since the fuel required for the mission will depend on the takeoff weight of the aircraft the fuel planning must follow the flight planning.

1.5.3 Operational Safety and Economy

Any transport operation carried out by an aircraft must be shown to be safe. The basic requirements for safe flight are that the space required for the aircraft to Manoeuvre should never exceed the space available and the aircraft carries sufficient fuel for the flight; these fundamental requirements form the basis of performance planning and fuel planning. The Standard Atmosphere, Off-standard and design atmospheres. It is important to understand the definition of various altitudes that are usually used to analyze/compare the performance of flying vehicles in standard atmosphere.

The International Standard Atmosphere is only a model for use in the estimation and measurement of aircraft performance. The real atmosphere encountered at any particular time and place will generally not conforms to the ISA model but will have its own temperature at datum pressure and its own temperature-height profile. The datum of the real atmosphere is usually taken to be an arbitrary height – for example, mean sea level at which the pressure will vary with time and the temperature – height profile may range between arctic winter and tropical summer. Any atmosphere that does not confirm to the ISA profile is referred to as an off-standard atmosphere. This is simply because of the fact that the atmospheric properties viz; Pressure, density and Temperature (P, ρ, T) also changes with altitude. Aerodynamic forces are strong function of these atmospheric properties (P, ρ, T). It is a necessity to specify the altitude that will help in postulating gravitational and aerodynamic forces explicitly.

- Absolute altitude (ha) – The altitude as measured from the center of the earth
- Geometric altitude (hg) – The altitude as measured from the mean sea level
- Geo-potential altitude (h) – The geometric altitude corrected for the gravity variation.

From Figure 1, it can be concluded that the absolute altitude is the sum of geometric altitude and mean radius of the earth. Mathematically, this relationship can be numericized

The gravitational force experienced by any aircraft varies with altitude. Also, an air-craft experiences variation in aerodynamic forces with altitude.

$$ha = hg + r \qquad (1)$$

where, r is the mean radius of earth.Acceleration due to gravity and altitude relationship

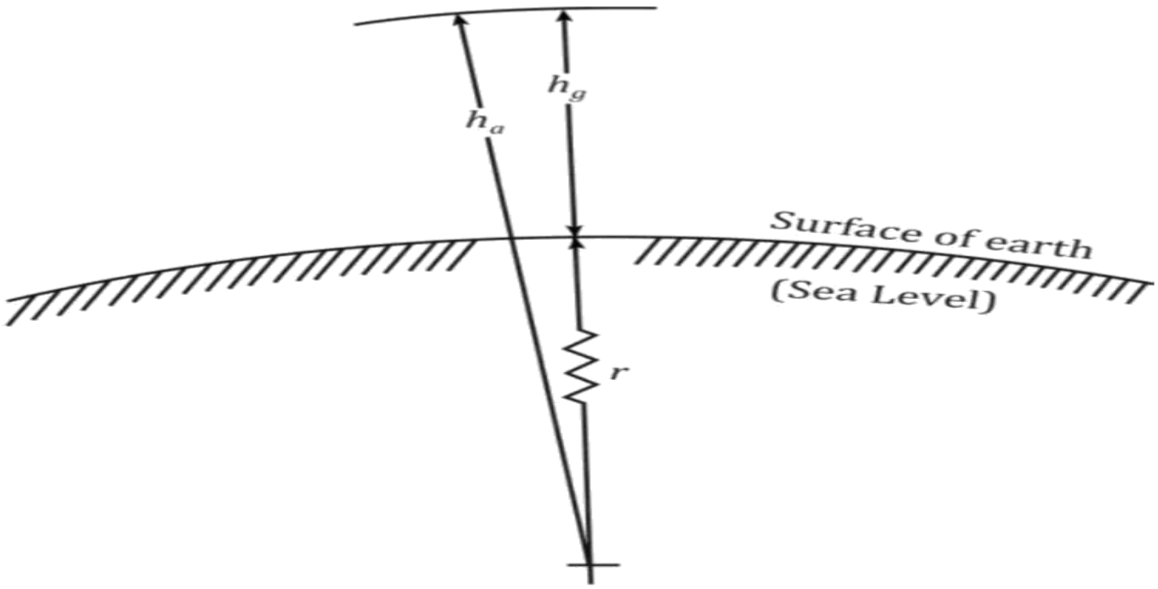

Figure 1.2 Schematic diagram representing geometric altitude and absolute altitude

Now that various concepts about altitudes are familiarized, the variations of acceleration due to gravity with altitude needs to be understood. It can be obtained using Newton's universal law of gravitation, fig 1.2. If gravitational acceleration at the sea level is go and the local gravitational constant is g for a given absolute altitude ha; then the relationship between g and go follow,

1.6 Geo-potential and geometric altitudes.

The hydrostatic equation of an infinitesimal fluid is given by

$$dP = -\rho g dhg \quad (3)$$

where, P - hydostatic pressure (P a), ρ - fluid density kg/m3 and

g - Acceleration (m/s2) due to gravity corresponding to the geometric altitude hg

In order to obtain the hydrostatic pressure (P) at a particular geometric altitude (hg), the above expression has to be integrated. Density and acceleration due to gravity are functions of altitude makes integration a bit more complex/difficult.

In order to simplify this integration, the concept of geo-potential altitude (h) has been introduced. We will consider this concept next.

1.6.1. Geo-potential altitude (h)

2. It is a fictitious altitude corrected for the gravity variation, which is typically used to ease the integration process (Equation 3). In simple terms, it can be called as “gravity adjusted height”. The adjustment uses Earth’s mean sea level as reference.

2. Now we can rewrite the hydrostatic equation, by replacing the geometric altitude with geo-potential altitude (h).

$$dP = -\rho g_o dh \quad (4)$$

Where, go is the acceleration due to gravity at mean sea level.

3. Using the two hydrostatic equations, viz, Equation 3 & Equation 4, we can derive the relationship between geometric and geo-potential altitude
4. Integrating dh from sea level up to any given value of h (different from hg)

1.7.The troposphere:

This is the region closest to the earth’s surface. It is characterized by turbulent conditions of air. The temperature decreases linearly at an approximate rate of 6.5 K / km. The highest point of the troposphere is called tropopause. The height of the tropopause varies from about 9 km at the poles to about 16 km at the equator.

1.8.The stratosphere:

This extends from the tropopause to about 50 km. High velocity winds may be encountered in this region, but they are not gusty. Temperature remains constant up to about 25 km and then increases. The highest point of the stratosphere is called the stratopause.

1.9.The mesosphere:

The mesosphere extends from the stratopause to about 80 km. The temperature decreases to about -900 C in this region. In the mesosphere, the pressure and density of air are very low, but the air still retains its composition as at sea level. The highest point of the mesosphere is called the mesopause.

1.10.The ionosphere or thermosphere:

This region extends from the mesopause to about 1000 km. It is characterized by the presence of ions and free electrons. The temperature increases to about 0° C at 110 km, to about 1000° C at 150 km and peak of about 1780° C at 700 km (Ref.2.1). Some electrical phenomena like the aurora borealis occur in this region.

1.11.The exosphere:

This is the outer fringe of the earth's atmosphere. Very few molecules are found in this region. The region gradually merges into the interplanetary space.

1.12. International Standard Atmosphere (ISA)

Need for ISA and agency prescribing it The properties of earth's atmosphere like pressure, temperature and density vary not only with height above the earth's surface but also with the location on earth, from day to day and even during the day. As mentioned in section 1.9, the performance of an airplane is dependent on the

- ❖ physical properties of the earth's atmosphere.
- ❖ Hence, for the purpose of comparing (a) the performance of different airplanes and

The performance of the same airplane measured in flight tests on different days, a set of values for atmospheric properties have been agreed upon, which represent average conditions prevailing for most of the year, in Europe and North America. Though the agreed values do not represent the actual conditions anywhere at any given time, they are useful as a reference. This set of values called the International Standard Atmosphere (ISA) is prescribed by ICAO (International Civil Aviation Organization). It is defined by the pressure and temperature at mean sea level

1.13. Measurement of air data

The essential requirements in the measurement of aircraft performance are first, the knowledge of the state of the atmosphere in which the aircraft is flying and secondly the relative motion between the aircraft and the air mass. This information is collected by the air data system.

The air data system of an aircraft in fig 1.11 consists of a pitot-static installation to sense the airflow pressures from which height, airspeed and Mach number are derived. An air thermometer from which the air temperature can be determined and in some cases airflow direction detectors (ADD) which sense the local flow directions relative to the aircraft body axes are part of the system

Both the air data computer and the mechanical instruments use the same basic calibration equations to convert the measured data into a suitable form for operational use. The calibration equation will be developed in the subsequent sections. The units used in the display of air data are usually the foot for measurement of height and the knot for airspeed since international regulation requires primary flight instruments to be calibrated in these units.

1.14.1 Indicated Air Speed (IAS):

The speed indicated by the airspeed indicator in the cockpit, which is based on the Pitot - static tube attached to the airplane.

1.14.2 Calibrated Air Speed (CAS):

The indicated airspeed corrects for the position and instrument errors. In standard atmospheric conditions, this is equal to the True Air Speed (TAS).

1.14.3 Equivalent Air Speed (EAS):

- ❖ The calibrated air speed corrected for adiabatic and compressibility effects. The al-titude effects are included in this speed.
- ❖ To explain the three airspeed, let us assume that an aircraft is in cruise at an al- titude of h, where the density is ρ, dynamic pressure is q and the corresponding velocity is v.

❖ Now if we want to simulate the same dynamic pressure at mean sea level (qo), the corresponding velocity is known as equivalent air speed

1.14.4 True Air Speed (TAS)

Pitot - static tube

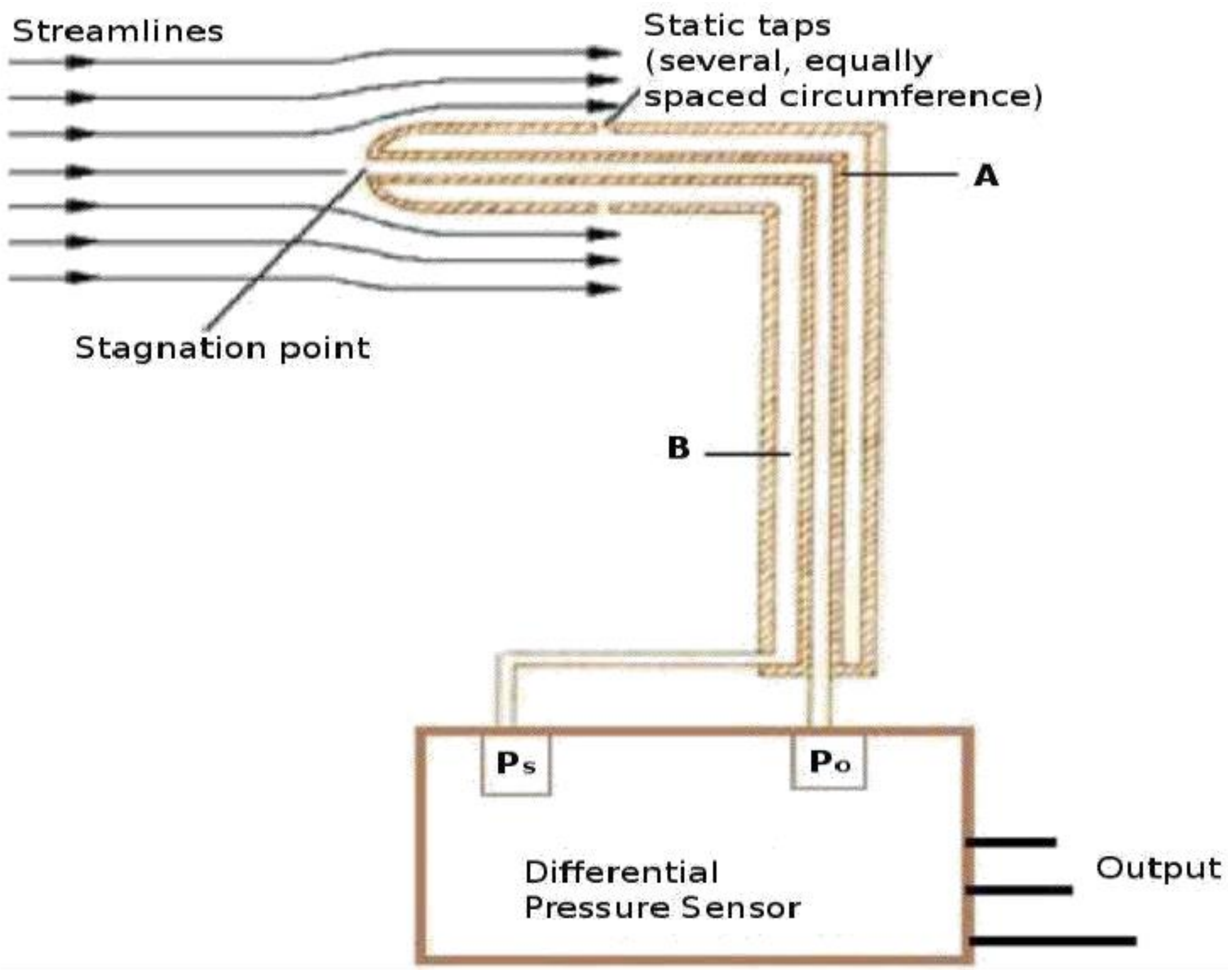

Figure 1.3: Schematic diagram representing the function of Pitot-Static tube

The airspeed of the airplane relative to the undisturbed air. True Air Speed (TAS) & Equivalent Air Speed (EAS) relation Suppose an airplane is flying at an altitude (ha) and experiencing dynamic pressure (Pa). We define equivalent airspeed as that speed with which the aircraft needs to fly at sea level to duplicate the actual dynamic pressure (ha) at a given altitude. so, we can write;

1.A schematic drawing of a pitot-static tube is presented in the Fig 1.3. It consists of two concentric tubes, say A and B.

2.The rear ends of these two tubes are connected to a differential pressure sensor. Typically a pressure sensor is a micro electro-mechanical device that is capable of measuring pressure and provides an equivalent analog output.

3.The front end of the tube A is open to free stream to trap the total pressure (Po) during the flight. Whereas tube B is a closed mouth tube having equally spaced peripheral holes to communicate with the surrounding air. Thus, tube B will be capturing the static pressure (Ps).

4.The output of the pressure sensor is usually in volts (V). In order to use this setup for velocity measurement, one has to calibrate the pressure sensor. Once the calibration chart is available for the specific pressure sensor, we can convert voltage (V) to Pascal (P a).

1.14.5 Air data computers

The measurement of the air data has been considered in terms of individual, mechanical instruments each providing an indication of one air data parameter. In many applications, the air data instruments have been replaced by an air data computer that carries out all the functions of the air data instruments. In addition it applies the known corrections to give the airspeed, altitude and air temperature in their most convenient form either for display to the pilot or to be transmitted to other systems on the aircraft.

1.14.6 The Force System Of The Aircraft And The Equations Of Motion

The equations of motion for performance

• The equations of motion of the aircraft are statements of Newton‘s law, F = ma, in each of three mutually perpendicular axes. The general force F is the sum of the components of a system of forces acting on the aircraft, which results in the inertial force, ma. The system of forces acting on the aircraft can be categorized into four groups; the gravitational forces, Fg, the aerodynamic forces, Fa, and the propulsive forces, Fp, which result in the inertial forces, F1, so that the statement of Newton‘s law becomes.

• There will also be a system of moments acting on the aircraft but, as these do not affect the flight path directly, they do not need to be taken into account in the equations of motion for performance. Each group of forces acts in its own axis system and needs to be resolved into the velocity axis system before the equations of motion for performance.

• Each group of forces acts in its own axis system and needs to be resolved into the velocity axis system before the equations of motion can be developed. The axis systems are described in full in Appendix A and the full equations of motion for aircraft performance are developed in Appendix B. Only a summary of the characteristics of the forces and the equations of motion will be considered here.

• The majority of performance analysis is based on the longitudinal equation of motion in which the term, Fn – D, is known as the excess thrust and provides the increase in potential energy (climb), or the increase in kinetic energy (acceleration).

• The equations of motion stated above are written in terms of aircraft with thrust-producing engines. If the ai4rcraft haspower-pr4oducing engines, which drive propellers to convert the power in to thrust, then the equations must be converted into their power form; this will be considered later in the section on propulsive forces.

• The movement of a given airplane through the atmosphere, insofar as it is responding to the four forces of flight. This movement is governed by a set of equations called the equations of motion.

• The equations of motion for an airplane are simply statements of Newton's second law, namely Consider an aircraft flying a curvilinear path as shown in Fig. below. To fly a curved path, a net force (Fnet), composed of force components parallel and perpendicular to the flight path, must exist. In applying Newton’s 2nd law, acceleration parallel (linear) to the flight path is dV/dt—centripetal acceleration is V2/r, where r is the radius of curvature of the curvilinear path.

Lift, drag, thrust, and weight are the four forces acting on an aircraft. A few points to remember about the figure:

1.Lift and drag, are the components of the aerodynamic force perpendicular and parallel to the relative wind (V∞).
2.The x axis is a body-fixed axis, aligned with the ‘‘reference line’’ of the aircraft.
3.The aircraft’s angle of attack (α) is between the x axis and the relative wind.
4.In general, the thrust vector is not aligned with the aircraft’s reference line. Rather, it is displaced through an angle φT , called the thrust angle.
5.The flight path angle (γ) is the angle between the horizon and the relative wind.
6.The pitch angle (θ) is the angle between the aircraft’s reference line and the horizon. It is sometimes referred to as pitch attitude or the deck angle (on transports, the cargo deck is usually taken as the aircraft’s reference line). Summing forces, parallel and perpendicular to the flight path, yields the following.

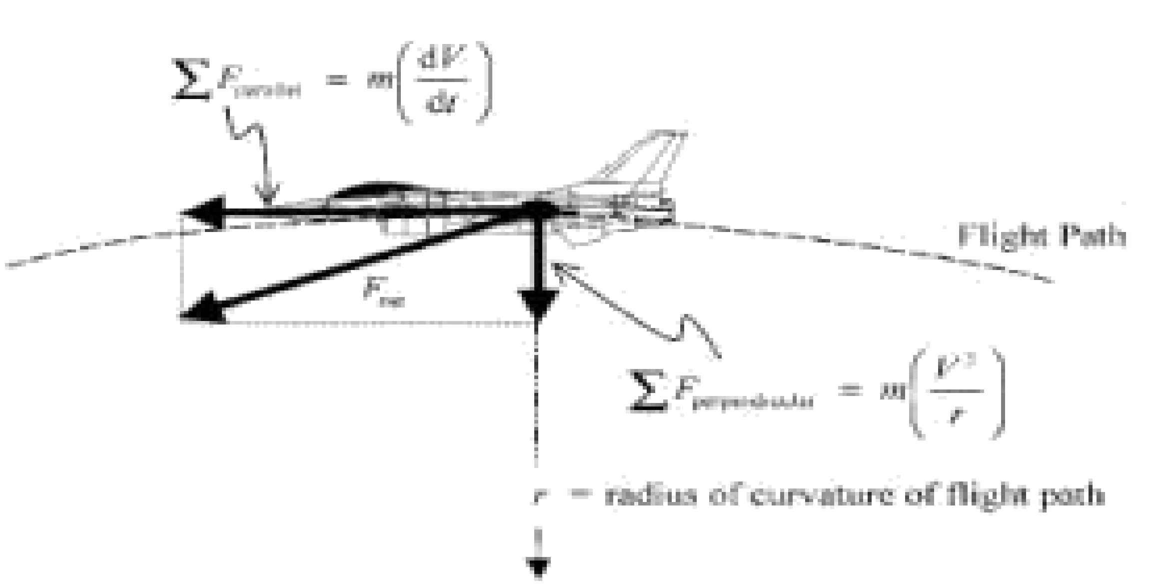

Fig 1.4 Aircraft flying a curvilinear path

1.15 The aircraft force system

•In the development of the equation of motion, the forces acting on the aircraft are represented as simple force terms and appear as constants.

•However, the forces stated in eqn (3.1) and in the equations of motion are not simple forces but depend on the performance variables, aircraft weight, airspeed (or flight Mach number) and the state of the atmosphere. In particular, the aerodynamic forces and the propulsive forces are of great importance to the performance of the aircraft. Their characteristics will define, for example, the airspeeds for best climb rate and gradient and for optimum range or endurance in the cruise part of the flight.

•Each group of forces can be considered in turn to determine how its characteristics very with the flight variables.

1.15.1 The inertial forces, f1

•The inertial forces arise from the mass of the aircraft and its acceleration. The accelerations may be linear accelerations or result from the combination of the forward speed of the aircraft with its rates of pitch and turn. The inertial forces act in the velocity axis system, which is discussed fully in Appendix B.

1.15.2The gravitational forces, fg

•The gravitational force acts downwards in the Earth axis system and is the product of the aircraft mass, m, and the acceleration due to gravity, g. It may be referred to either as weight, W, or ass the product mg; each form of reference has its own applications within the theory and practice of aircraft performance.

1.15.3 The aerodynamic forces, fa

• The aerodynamic forces arise from the relative motion between the aircraft and the air mass in which it is flying; they act in the wind axis system.It will be assumed that the reader is familiar with the concepts of aerodynamics and this treatment will only consider the aerodynamic characteristics of the aircraft that are directly applicable to the study of performance.

• The dynamic pressure of the airflow, q, may be considered in terms of either airspeed or Mach number,Whilst either form may be used when considering the non-dimensional aerodynamic forces, the form involving the mach number is particularly useful when considering operational performance. If the airspeed is considered in terms of the flight Mach number, then the temperature of the atmosphere is implicit in the statement of the Mach number and the atmosphere pressure can be considered independently. Since altitude is related uniquely to the static pressure of the atmosphere, the altitude becomes a basic variable of the aerodynamic forces. Therefore, the forces need to be considered only in terms of their variation with aircraft weight, flight Mach number and altitude a rather than in terms of aircraft weight, airspeed, altitude and temperature.

• The aerodynamic forces that concern performance are the lift, L, the drag, D, and the side force, Y. In the case of an air4craft, the speed of flight is relatively high and the non-dimensional flow variables that characterize the flow are,

1.15.4The Reynolds number,

Typically the flight value of Re is large, 10 to 10 , and the flow can be treated as continue flow. If the aerodynamic characteristics of the aircraft have been determined from experimental sources (e.g. wind tunnels), any Reynolds number effects should have been accounted for before being used in any performance estimation process. It is unlikely that the Reynolds number will influence the analysis of the full scale flight performance of the aircraft significantly, except in extreme cases.

1.16.The Mach number, M = V/a

This may vary from almost zero up to a typical maximum of 2.2 for conventional aircraft; higher Mach number is possible but raises special problems. Since this treatment of performance is concerned mainly with subsonic flight, the supersonic flow characteristics will not be considered in depth. Only in the transonic region, where the Mach numbers Will the effects be considered. In flight up to Mach number of 0.5 the flow can be regarded as incompressible and Mach number effects ignored; for $0.5<M<0.8$ compressibility becomes significant and may lead to small changes in the lift and drag force characteristics. For most subsonic aircraft the critical Mach number occurs typically around M = 0.8; at this Mach number the local flow at points on the aircraft becomes supersonic and shock waves begin to form. This effect starts the change from subsonic to supersonic flow and affects the characteristics of both the lift and drag forces, leading to significant effects on the performance of the aircraft. March number is one of the most important variables of performance and its effect on the aerodynamic forces needs to be considered.

1.16. Lift, drag, side force

The aerodynamic force exerted on a body immersed in airflow is due to the two hands of nature which are in direct contact with the surface of the body; these two hands of nature are the pressure and shear stress distributions acting all over the exposed surface of the body. The pressure and shear stress distributions exerted on the surface of an airfoil due to the airflow over the body are sketched; pressure acts locally perpendicular to the surface, and shear stress acts locally parallel to the surface. The net aerodynamic force on the body is due to the pressure and shear stress distributions integrated over the total exposed surface area.

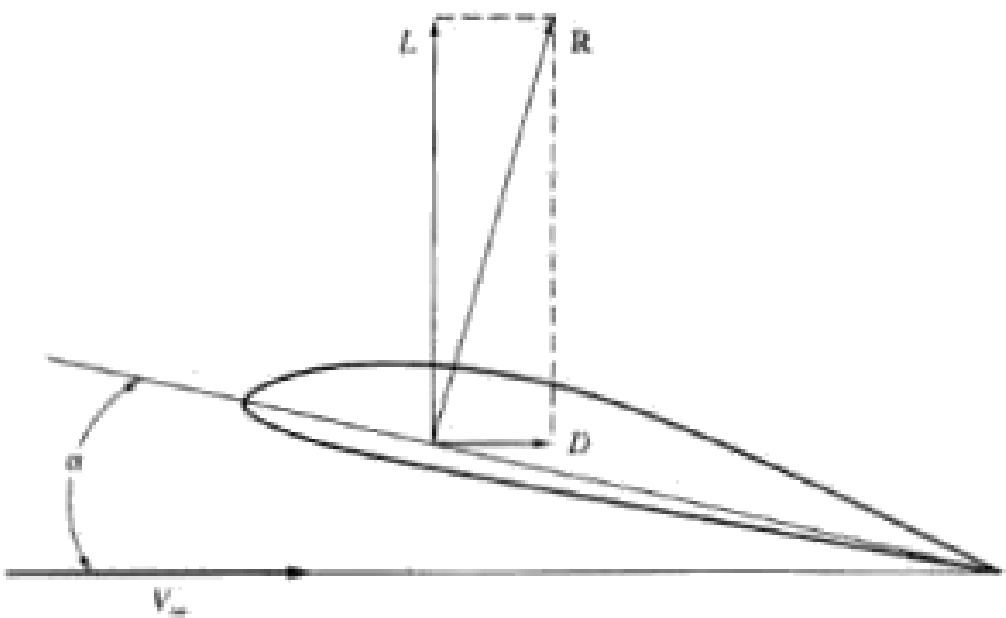

Fig 1.5 Airfoil nomenclature

The component of R perpendicular to the free-stream velocity is the lift and the component of R parallel to the free-stream direction is the D. The force perpendicular to the other two is called the side force. The side force concept is mostly used in connection with turning aircraft.

1.17 Total airplane drag- estimation, drag reduction methods.

Drag is simply force that opposes the motion of an aircraft through the air. However it does have separate components that comprise it. Fig 1.5, shows the air foil nomenclature.

Total Drag produced by an aircraft is the sum of the Profile drag, Induced drag, and Parasite drag. Total drag is primarily a function of airspeed. The airspeed that produces the lowest total drag normally determines the aircraft best-rate-of-climb speed, minimum rate-of-descent speed for autorotation, and maximum endurance speed.

•**Profile Drag** is the drag incurred from frictional resistance of the blades passing through the air. It does not change significantly with angle of attack of the airfoil section, but increases moderately as airspeed increases. Induced Drag is the drag incurred as a result of production of lift. Higher angles of attack which produce more lift also produce increased induced drag.

In rotary-wing aircraft, induced drag decreases with increased aircraft airspeed. The induced drag is the portion of the Total Aerodynamic Force which is oriented in the direction opposing the movement of the airfoil

•**Parasite Drag** is the drag incurred from the non lifting portions of the aircraft. It includes the form drag and skin friction associated with the fuselage, cockpit, engine cowlings, rotor hub, landing gear, and tail boom to mention a few. Parasite drag increases with airspeed.

The graphic illustrates the different forms of drag versus airspeed:

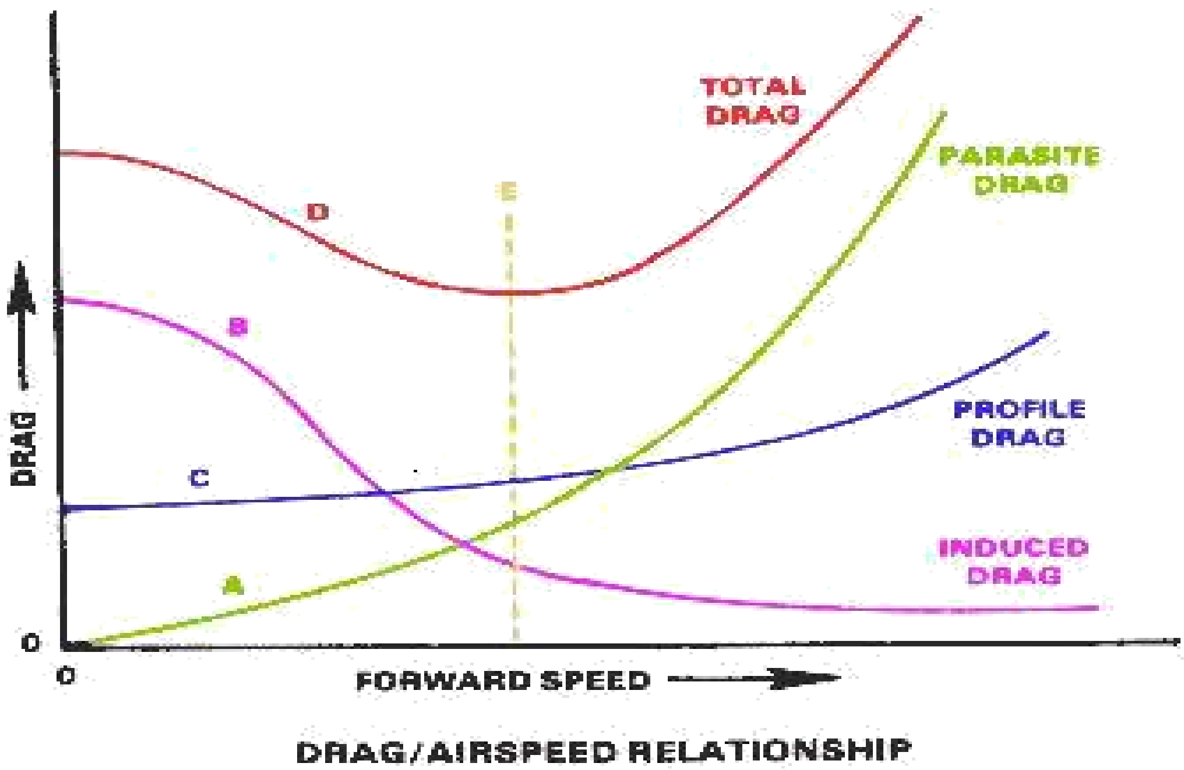

Fig 1.6 Drag to airspeed variation

Curve "A" shows that parasite drag is very low at slow airspeeds and increases with higher airspeeds. Parasite drag goes up at an increasing rate at airspeeds above the midrange, fig 1.6.
Curve "B" shows how induced drag decreases as aircraft airspeed increases. At a hover, or at lower airspeeds, induced drag is highest. It decreases as airspeed increases and the helicopter moves into undisturbed air. Curve "C" shows the profile drag curve. Profile drag remains relatively constant throughout the speed range with some increase at the higher airspeeds.
Curve "D" shows total drag and represents the sum of the other three curves. It identifies the airspeed range, line "E", at which total drag is lowest. That airspeed is the best airspeed for maximum endurance, best rate of climb, and minimum rate of descent in autorotation.

(Total drag) = (parasite drag) + (wave drag) + (induced drag)

• For the subsonic transport, the elements labeled wing, body, empennage, engine installations, interference, leaks, undercarriage, and flaps are the contributors to the zero-lift parasite drag; that is, they stem from friction drag and pressure drag (due to flow separation).

• The element labeled lift-dependent drag (drag due to lift) stems from the increment of parasite drag associated with the change in angle of attack from the zero-lift valve, and the induced drag. Note that most of the drag at cruise is parasite drag, whereas most of the drag at takeoff is lift-dependent drag, which in this case is mostly

• For the supersonic transport, more than two-thirds of the cruise drag is wave drag-a combination of zero-lift wave drag and the lift-dependent drag (which is mainly wave drag due to lift). This dominance of wave drag is the major aerodynamic characteristic of supersonic airplanes. At takeoff, the drag of the supersonic transport is much like that of the subsonic transport, except that the supersonic transport experiences more lift-dependent drag. This is because the low-aspect-ratio delta wing increases the induced drag, and the higher angle of attack required for the delta wing at takeoff (because of the lower lift slope) increases the increment in parasite drag due to lift.

The propulsive forces - the thrust production engines, power producing engines, There are two basic forms of power plant used for aircraft propulsion. The thrust-producing power plant , which produces its propulsive force directly by increasing the momentum of he airflow through the engine, and The power-producing power plant, which produces shat power that is then turned into a propulsive force by a propeller. Each form of power plant has different characteristics and needs to be considered separately.

1.17.1. The thrust-producing power plant

The usual from of thrust-producing engine is the turbojet or turbofan, although rocket could be included in this category. The turbojet or turbofan engine uses atmospheric air as its working fluid and, with the addition of fuel, burns the air to increase its energy. The high-energy air is then expelled through a nozzle with increased momentum to produce the thrust force.

•In simplified terms, the turbojet engine can be considered to produce thrust by increasing the momentum of its internal flow stream. The net propulsive force, Fn, is the difference between the stream force entering the engine and the stream force exiting the engine. The thrusts produced at the exit plane of the nozzle is known as the gross thrust, Fg, and is equal to the rate of change of momentum of the exhaust gas flow,

$$Fg = mVj.$$

The flow into the intake also contributes to the engine thrust. In this case, the momentum of the flow is lost as the air enters the engine.The force due to the intake flow, known as the momentum drag, Dm, is equal to the rate of change of momentum in the intake airflow, Dm = mV. The net propulsive thrust is given by The net thrust of the power plant will be affected by the flight Mach number and altitude. It is not possible to postulate any precise function that will relate thrust to Mach number or altitude for all thrust-producing power plants. However, simple relationships can be developed that will enable the general characteristic of the thrust variation with Mach number and altitude to be deduced. From above equation the net thrust can be expressed as,the power-producing power plant. The power-producing power plant delivers its power through a rotating shaft to a propeller that converts the power into propulsive thrust. The power plant may be either a piston engine or a gas turbine that converts the energy of its gas flow into shaft-power rather than into thrust. In either case the shaft-power output is not greatly affected by airspeed and, to a first-order approximation, the power can be regarded as independent of airspeed. The shaft-power, P, is converted into thrust, T, by the propeller. In the process, losses occur and the thrust-power produced will be less than the shaft-power delivered. The propeller efficiency, n, is the ratio of the thrust-power output to the shaft-power input so that, Thrust is the propulsive force which moves an aircraft through the air. Thrust is used to overcome the drag of an airplane, and to overcome the weight of a rocket. Thrust is generated by the engines of the aircraft through some kind of propulsion system. Thrust is a mechanical force, so the propulsion system must be in physical contact with a working fluid to produce thrust. Thrust is generated most often through the reaction of accelerating a mass of gas. Since thrust is a force, it is a vector quantity having both a magnitude and a direction. The engine does work on the gas and accelerates the gas to the rear of the engine; the thrust is generated in the opposite direction from the accelerated gas. The magnitude of the thrust depends on the amount of gas that is accelerated and on the difference in velocity of the gas through the engine

• To accelerate the gas, we have to expend energy. The energy is generated as heat by the combustion of some fuel. The thrust equation describes how the acceleration of the gas produces a force. The type of propulsion system used on an aircraft may vary from airplane to airplane and each device produces thrust in a slightly different way. The four principal propulsion systems: the propeller, the jet engine, the ramjet, and the rocket. Reciprocating engine is the rotating crankshaft-this is the means by which the engine's power is transmitted to the outside world-a wheel axle in the case of an automobile, or a propeller in the case of an airplane. The power output of the engine is directly proportional to the rpm. Third, the amount of force applied by the burned gas on the face of the piston after combustion will affect the work performed during each power stroke. Hence, the higher the pressure in the cylinder during the power stroke, the larger will be the power output. An average pressure which is indicative of the pressure level in the cylinder is defined as the mean effective pressure Pe· Therefore, we can state that the power output from the engine to the crankshaft, called the shaft brake power P, is given by equation (a) below. The thrust generated by the engine is due to the net resultant of the pressure and shear stress distributions acting on the exposed surface areas, internal and external, at each point at which the gas contacts any part of the device. The calculation of jet engine thrust is carried out infinitely more simply by drawing a control volume around the engine, looking at the time rate of change of momentum of the gas flow through the engine, and using Newton's second law to obtain the thrust. The measurable primary output from a jet engine is thrust, whereas that for a piston engine is power. Therefore, for a turbojet the specific fuel consumption is based on thrust rather than power; to make this clear, it is frequently called the thrust specific fuel consumption. We denote it by ct, and it is defined as ct = weight of fuel burned per unit thrust per unit time. Variation of propulsive power and specific fuel consumption with altitude and flight speed.

Consider the engine mounted on an airplane. As the airplane velocity V∞ is changed, the only variable affected in power Eq. is the pressure of the air entering the engine manifold, due to the stagnation of the airflow in the engine inlet. (Sometimes this is called a ram effect.) In effect, as V∞ increases, this "ram pressure" is it is reflected as an increase in Pe in Eq. which in tum increases P via Eq. For the high-velocity propeller driven fighter airplanes of World War II, this effect had some significance. However today reciprocating engines are used only on low-speed general aviation aircraft and the ram effect can be ignored. Hence Power is reasonably constant with $V\infty$. Similarly, SFC is also constant with $V\infty$ Variation of thrust and specific fuel consumption with altitude and flight speed. Consider the mass flow of air mair· The mass flow of air entering the inlet is ∞ A1 $V\infty$, where A1 is the cross-sectional area of the inlet. As $V\infty$. is increased, Vj stays essentially the same; the value of Vj is much more a function of the internal compression and combustion processes taking place inside the engine than it is of $V\infty$. Hence, the difference Vj - $V\infty$ tends to decrease as $V\infty$. Increases. From thrust eq with $V\infty$. increasing but Vj staying about the same, the value of T is decreased. These two effects tend to cancel in thrust eq, and therefore we might expect the thrust generated by a turbojet to be only a weak function of $V\infty$. Here the thrust for a typical small turbojet is given as a function of flight Mach number for two altitudes, sea level and 40,000 ft, and for three different throttle settings (denoted by different compressor rpm values) at each altitude. Note that, especially at altitude, T is a very weak function of Mach number.There is a strong altitude effect on thrust, as can be seen by examining thrust eq. Again, we note that m air = ∞A1$V\infty$; hence mair is directly proportional to ∞. As the altitude increases, ∞ decreases. In turn, from thrust eq where T is almost directly proportional to m air, thrust also decreases with altitude. Indeed, it is reasonable to express the variation of T with altitude in terms of the density ratio, where p is the density at a given altitude and 0 is sea-level density.

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DEPARTMENT OF AERONAUTICAL ENGINEERING

UNIT – II –ENGINE CHARACTERISTICS– SAEA1501

2 Engine Characteristic

2.1 Introduction

The cruise performance of an aircraft is one of the fundamental building blacks of the overall mission. In the cruising segment of the mission, both height and airspeed are essentially constant and the (airspeed) aircraft is required to cover distance in the most expedient manner. Usually, the majority of the fuel carried in the aircraft will be used during the cruise. The distance that can be flown, on the time that the aircraft can remain aloft, on a given quantity of fuel are important factors in the assessment of the cruise performance

2.2 Maximum and minimum speeds in level flight

By definition, the thrust available curve is the variation of TA with velocity at a given throttle setting and altitude. For the throttle full forward, (TA)max is obtained. The maximum thrust available curve is the variation of (TA)max with velocity at a given altitude. For turbojet and low-bypass-ratio turbofans, we have seen that at subsonic speeds, the thrust is essentially constant with velocity. Hence, for such power plants, the thrust available curve is a horizontal line, as sketched in Figure below.

2.2 Thrust

In steady, level flight, the maximum velocity of the airplane is determined by the highspeed intersection of the thrust required and thrust available curves.

Finding Vmax from the intersection of the thrust required and thrust available curves, as shown in above Figure 2.1, is a graphical technique. An analytical method for the direct solution of Vmax follows from the velocity Eq through thrust required. For steady, level flight, TR = TA.

For flight at Vmax, the thrust available is at its maximum value. Hence,

In Eq., replacing V∞ with Vmax and TR with (TA)max, and taking the plus sign in the quadratic expression because we are interested in the highest velocity, we have

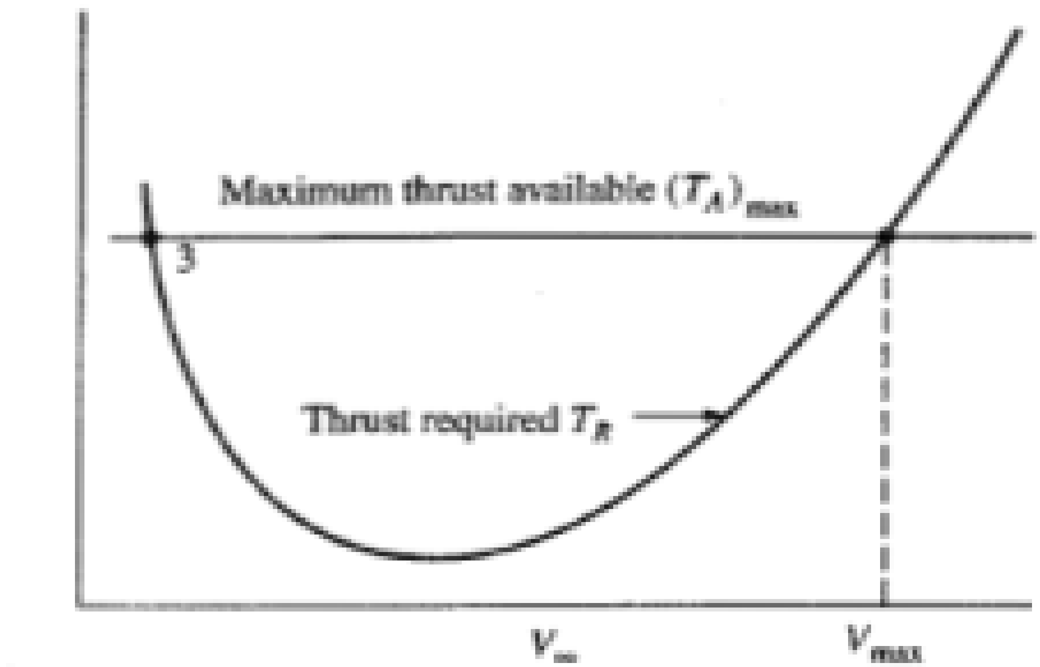

Fig 2.1 Power required vs available curves

The above equation allows the direct calculation of the maximum velocity. Moreover, being an analytic equation, it clearly points out the parameters that influence Vmax· Note in Eq. 2.2 that depends on (1) the maximum thrust-to-weight ratio (TA)max/W, (2) wing loading W/S, the drag polar via CD,o and K, and altitude .

From this equation we see that

1. Vmax increases as (TA) max, W increases.
2. Vmax increases as W / S increases.
3. Vmax decreases as CD.o and/or K increases.

When the aircraft is cruising in trim the equations of motion of a conventional aircraft. The above equations can be reduced to the simple statements Range and endurance with thrust production, and power producing engines

•Range (R) is the horizontal distance covered, with respect to a given point on the ground, with a given amount of fuel. It is measured in km.

•Endurance (E) is the time for which an airplane can remain in air with a given amount of fuel. It is measured in hours.

2.3. Cruising efficiency

It can be measured in terms of either the range or the endurance of the aircraft . The Specific Air Range (SAR) is defined as the horizontal distance flown per unit of fuel consumed and the Specific Endurance (SE) is defined as the time of flight per unit of fuel consumed. The distance travelled, x, in still air is given by the time integral of the true airspeed, V, so that And in cruising flight the true airspeed is usually quoted in knots, or nautical miles per hour (nm/hour). In addition, during cruise, fuel is burned and the fuel mass flow, Qf, will determine the rate of change of mass of the aircraft.The above definition of range is very general and terms like safe range and gross still air range are commonly used. These terms include details of the flight plan and are explained in the subsequent subsections.The fuel mass flow is usually quoted in kg/hour and is negative since the mass of the aircraft decreases with time as fuel is burned.. Cruising efficiency can be measured in terms of either the range or the endurance of the aircraft. The specific Air Range (SAR) is defined as the horizontal distance flown per unit of fuel consumed.Where V=dx/dt is the velocity and Qf =-dm/dt fuel burned during cruise. This implies that the aircraft must be allowed to climb during the cruise to maintain the parameter W/p constant and the method is known as the Cruise-Climb technique. Since the angle of attack is constant throughout the cruise the lift coefficient, and hence lit-drag ratio. L/D, will be constant. The constant Mach number implies flight at constant true airspeed.

2.4. Cruise method

Let the range under cruise method 1 be then from above equation, This expression is the best-known expression for the range of an aircraft and is known as the Breguet Range expression.

2.4.1. Cruise method 1:

•Although it offers the optimum performance in terms of distance flown on a given fuel load there are practical reasons that make its application to flight operations difficult, and further consideration of this cruise method is necessary. As weight decreases during the cruise the variables on the right side of above equation has to compensate. So, by using three variables (Pressure, Mach number and Coefficient of lift) thus three methods can be considered to compensate reduction in weight by varying one variable with other two constants.

2.4.2. Cruise method 2: Constant angle of attack, Constant altitude

In this case, the Mach number, or true airspeed, must be reduced during cruise, so that This implies that the lift coefficient and lift-drag ratio, will be constant during the cruise and that substituting for airspeed in SAR equation then range under cruise method 2 is endurance of the aircraft in cruise method 1 is same as cruise method 2.

2.4.3. Cruise method 3: Constant Mach number, Constant altitude

In this case, the angle of attack must be allowed to decrease as the weight decreases to maintain constant W/CL1. Assuming that the aircraft is cruising at a speed greater than its minimum drag speed, then the decrease in lift coefficient during the cruise will cause the drag coefficient and hence the drag force to decrease. This will require a progressive decrease in the thrust required to maintain the Mach number or true airspeed constant. Since the lift-drag ratio will not be constant, the range will need to be found by integrating the drag over the weight change during cruise.
Now for an aircraft with a parabolic drag polar. Substituting in above equation and integrating gives,

2.5. Methods- comparison of performance

It has been possible to write the range and endurance attained by each method of cruise in the form of a product of a range factor and a range function, and of endurance factor and endurance function, giving

•The range functions are compared for a fuel ratio of 1.5 and shows that the cruise-climb is the optimum method which gives 10% better range than the other methods.

•The comparison between the cruise methods for endurance shows less disparity but favours the constant angle of attack methods, fig 2.2.

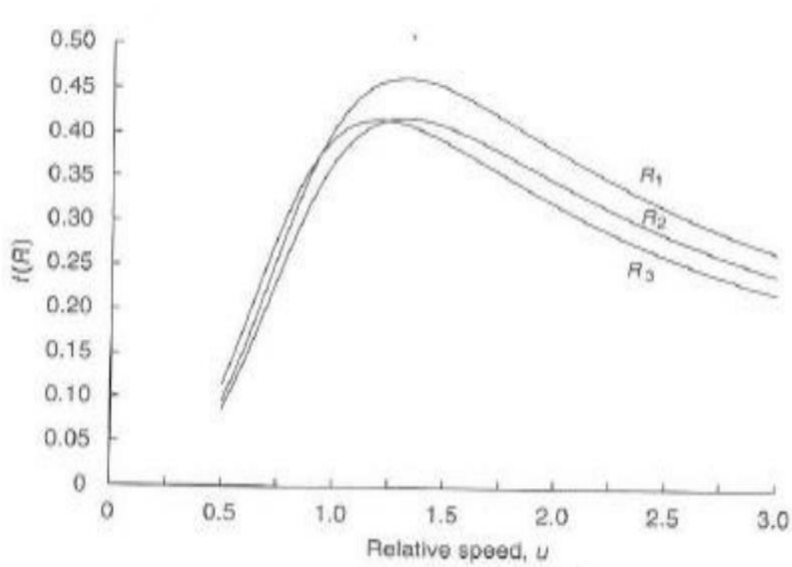

Fig 2.2 Cruise conditions

2.6. The effect of alternative fuel flow laws

•The range and endurance functions developed above have used the simple fuel flow law, above equation which assumes that the fuel flow is proportional only to net thrust. It was accepted that this fuel flow law is idealized, probably not full describing the characteristics of the engine, and that a more realistic law should be applied. If an alternative fuel flow law is considered, for example above equation then it may produce different optimum operating airspeeds. Above equation states a fuel flow law of the form,where n may vary from 0.2 for a turbojet to about 0.6 for hybass ratio turbofan. If SAR and SE equations are written in terms of Mach number and the alternative fuel flow substituted, they give, When these expressions are differentiated, they give the airspeeds for maximum SAR and SE to be, effect of weight, altitude and temperature on cruise performance

•For a given final weight of the aircraft, the increase in initial weight implies an increase in the fuel available for the cruise; it is this that extends the range not the weight of the aircraft itself.
•An increase in air temperature increases the range since the TAS (True Air Speed) is increased and the aircraft flies further in given time, during which it burns the same quantity of fuel. If the

specific fuel consumption is a function of air temperature, then the effect of the temperature may be lessened and the range may even decrease as temperature increases.

•Increasing altitude will produce an increase in the range as the ambient relative pressure decreases. In the troposphere, the effect will be reduced by the accompanying decrease of temperature with altitude and further affected by any dependency of the SFC on air temperature.

2.7. Cruise performance with mixed power-plants.

•Turboprop is the engine which converts the turbine shaft power into thrust by using
propeller and the exhaust gases convert into thrust using nozzle.

•By using cruise-climb range expressions for thrust and power producing engines cruise performance of mixed power plants in terms of Range can be obtained as

•Where π is the proportion of the thrust derived from the shaft power in the overall thrust of the power plant and CP and CT are the specific fuel consumptions based on the shaft power and net thrust respectively.

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UNIT – III –EVALUATION OF UN - ACCELERATED FLIGHT PERFORMANCE– SAEA1501

3.Evaluation of Un - Accelerated Flight Performance

3.1 Climb and descent techniques

•Climb is a phase in the overall mission of the aircraft in which the aircraft increases its height to the required cruising level from takeoff and in a descent phase the aircraft decreases its height from the end of the cruise to the landing. Climb performance is important from both economic and flight safety point of view. The descent is less critical than the climb economically since the aircraft will be operating at low thrust and hence low fuel flow.

•In these phases of flight, the difference between the propulsive thrust and the airframe drag is used to change the potential and kinetic energy of aircraft. If the thrust exceeds drag the aircraft will climb and if drag exceeds thrust then aircraft will descend.

•The fuel required to climb to a given height can be minimized by the use of the correct climb technique. A climb or a descent will usually be performed with reference to an indication either of airspeed or of Mach number.

•The constant EAS climb implies, as the aircraft climbs, the ambient air density will be decreasing and the true airspeed will be increasing thus the aircraft will be accelerating throughout climb.

• A constant Mach number results in a constant TAS and there is no acceleration. Steady state climb and descent can be measured in terms of rate or gradient. The rate of climb is the vertical, dH/dt

Where the height, H, is geopotential height.

3.2 Climb and descent performance analysis

• The power plant of the aircraft may be thrust producing or power producing. Thrust-producing engineers, turbojets and turbofans, produce thrust that is relatively constant with small change of airspeed in subsonic flight. Power-producing engines, piston engines or turbo-shaft engineers, produce shaft power, which is relatively constant with change of airspeed, and which needs to be converted into propulsive thrust by a propeller. The differing characteristics of these power plants lead to different criteria for optimum climb performance and each need to be considered separately.

3.3. Safety Considerations

•The safety of the aircraft depends on its ability to climb above obstructions at all points on the flight path. Sufficient excess thrust must be available to ensure that the aircraft can meet certain minimum gradients of climb in any of the safety critical segments of the flight. Several safety-related considerations will affect the choice of the flight path in the descent. Among them are the attitude of the aircraft, the rate of change of cabin pressure and the need for the engines to supply power for airframe services.

3.4 Performance analysis- Aircraft with Thrust producing engines

•The equations of motion of an aircraft with thrust-producing engines in a climb or descent can obtained using equation 1.1. In a straight, wings level, climb in which the flight path gradient is constant the equations of motion can be written as

•If the aircraft has a normal takeoff thrust-to-weight ratio of about 0.3 then the rates of climb will be low enough to assume that the acceleration associated with the rate of climb is negligible. The climb can then be assumed to be made either at constant airspeed or constant Mach number. Also the gradient of climb and descent will be low enough to allow the assumption that 2=1 and the equation can be simplified as

•The excess thrust [FN-D] provides the gradient of climb, If the thrust is constant the best gradient of climb will be obtained by flying at the minimum drag speed. Using eq 3.1 in 3.6 leads to an expression for the best rate of climb,

•The above equation indicates that the airspeed for the best rate of climb occurs when the excess thrust power [FN V], over drag power, [DV] is a maximum. Since the ideal thrust power increases linearly with true airspeed, the best rate of climb is predicted to be at airspeed greater than the minimum drag speed.

• The climb and descent performance can best be analysed in a general manner by considering a dimensionless form of the performance equation

3.5 Maximum Climb Gradient

•By differentiating eq 3.8 the relative airspeeds for best climb or descent performance can be found. From eq 3.8 the gradient of climb is given by

•For maximum gradient 2 = 0, which occurs when u=1 if is constant, and confirms that the steepest climb occurs at the minimum drag speed. In the special case of gliding flight, in which = 0, flight at the minimum drag speed will give the shallowest glide angle, which will give the greatest range of glide; this speed is used when cruising between thermals. The minimum glide angle will be

3.6 Performance analysis- Aircraft with Power producing engines

The generalized performance equation for climb and descent is given from equations of motion for aircraft with power-producing engines as

3.6.1 Climb performance in aircraft operations

•One of the usual ways of determining the best climbing performance is by partial climbs. This technique consists of flying a number of short climb segments through a datum height. Each of the climbs is flown at a different airspeed around the predicted speed for best climb performance. The measured rates of climb are corrected for change of aircraft weight due to the consumption of fuel during the series of climbs. So that the performance relates to a common standard aircraft weight. The measured climb rate and climb gradient can then be plotted against airspeed to give the actual airspeeds for best performance under the weight, altitude and temperature conditions of the test. From these measured data the airspeeds at which optimum climb performance is achieved can be found. An alternative method is to measure the maximum excess thrust or power by level accelerations. In this technique, the aircraft is flown as slowly as possible in level flight; maximum thrust or power is selected and the airspeed recorded in a level acceleration to maximum airspeed. From the acceleration, the excess thrust or power can bededuced and thus so can that speeds for best climb performance.

•The level acceleration method is best suited to aircraft with thrust-producing engines and the partial climb method is best suited to aircraft with power producing engines since the best climb speeds tend to be towards the lower end of their speed range.

3.6.2 Minimum fuel climbs

•The fuel consumed in the climb can be expressed in terms of the specific climb, SC, as

•If the fuel flow is measured during the flight trails to determine the optimum climb speed, then the specific climb function can be formed to give the airspeed for a minimum fuel climb.

3.6.3 Descent performance in aircraft operations

•If the propulsive thrust is less than the airframe drag then the aircraft will decelerate or descend. The aircraft has wide range of descent flight paths like shallow descent, steep descent which occurs by changing thrust and drag. In the special case of gliding flight, in which there is no propulsive thrust, the descent will be determined based on lift-drag ratio.

•A descent can be produced by flying at airspeed less than the minimum drag speed; the aircraft will not have a stable flight path. The stability of flight path occurs when the flight path gradient can be controlled by the use of elevator control only. If the aircraft is flying at airspeed greater than the minimum drag speed then the flight path gradient of descent can be increased (steepened) by increasing airspeed. This is achieved by nose down pitch change with no adjustment in engine thrust setting. The optimization of the descent is not straightforward as the optimization of the climb.

•In transport operations it would be undesirable to make a very steep, high airspeed descent since this would entail a steep nose-down attitude that could be uncomfortable, if not dangerous to persons in the cabin. In addition the rate of increase of cabin pressure during descent must be kept to a reasonably low value to prevent discomfort due to the re-pressurization of the passenger's ear passages. The rate of change of cabin pressure should not exceed the equivalent of 300 ft/min at sea level. This implies that, if the cabin pressurized to the equivalent of 8000 ft pressure height, the descent to sea level should not take less than 24 min regardless of the pressure height from which the aircraft descends. An exception to this general rule is the emergency descent following the loss of cabin pressure. In this case, the aircraft must descend to a safe altitude as quickly as possible and the highest rate of descent must be used.

•There are several phases of a typical descent flight path from cruising altitude down to landing. Each phase of the descent has different criteria that govern the manner in which the aircraft is flown. Figure above shows the phases of atypical descent.

3.8 Effect of wind on climb and descent performance

•Wind is the relative velocity between the general air mass and the ground. Wind can be assumed to have only a horizontal component of velocity but due to ground profile vertical velocity components are also to be considered where sloping or undulating ground occurs. So, in the present case the aircraft will be taken to be operating over ground level so that only horizontal component will be considered.

•The performance relative to the ground will be affected by the wind which is known as perceived performance seen by the observer, from either the aircraft or ground and which affects the ability of the aircraft to clear ground-based obstructions.

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UNIT – IV –ACCELERATED AND MANOEUVERING FLIGHT PERFORMANCE– SAEA1501

4. Accelerated motion of aircraft - equations of motion- the manoeuvre envelope

•An aircraft can be said to be in manoeuvreing flight when its flight path is in a continuous change of state and in which there is an inertial force due to acceleration. Longitudinal manoeuvres- the pull-up, push over manoeuvres.

•Consider an airplane initially in straight and level flight, where L = W. The pilot suddenly pitches the airplane to a higher angle of attack such that the lift suddenly increases. Because L > W, the airplane will arch upward, as sketched in Fig. below. The flight path becomes curved in the vertical plane, with a turn radius R and turn rate dθ/dt. This is called the pull-up manoeuvre as shown in Fig 4.1.

4.1 Manoeuvre boundaries, Manoeuvre performance of military aircraft, transport aircraft.

•The effect of manoeuvres on the flight path performance of civil transport aircraft is
generally not very significant. They spend only very low proportion of their time in turning flight and, since the maximum bank angle is typically 200 the turns are generally low rate. The only case in which the effect of turning would be significant to a transport aircraft in terms of its effect on flight safety is a turn made with marginal excess thrust available.

•Military transport aircraft may be required to Manoeuvre more aggressively than civil aircraft and sustained turning performance may be significant in the operational profile of the mission. Requires higher rate of pitch and fuel consumption during Manoeuvres. The design criteria for such aircraft should be sustaining high g Manoeuvres with high rate of turns. Complex Manoeuvres combing turning with pull-ups and acceleration will also need to be considered to establish the limits of aircraft in air – air combat situations.

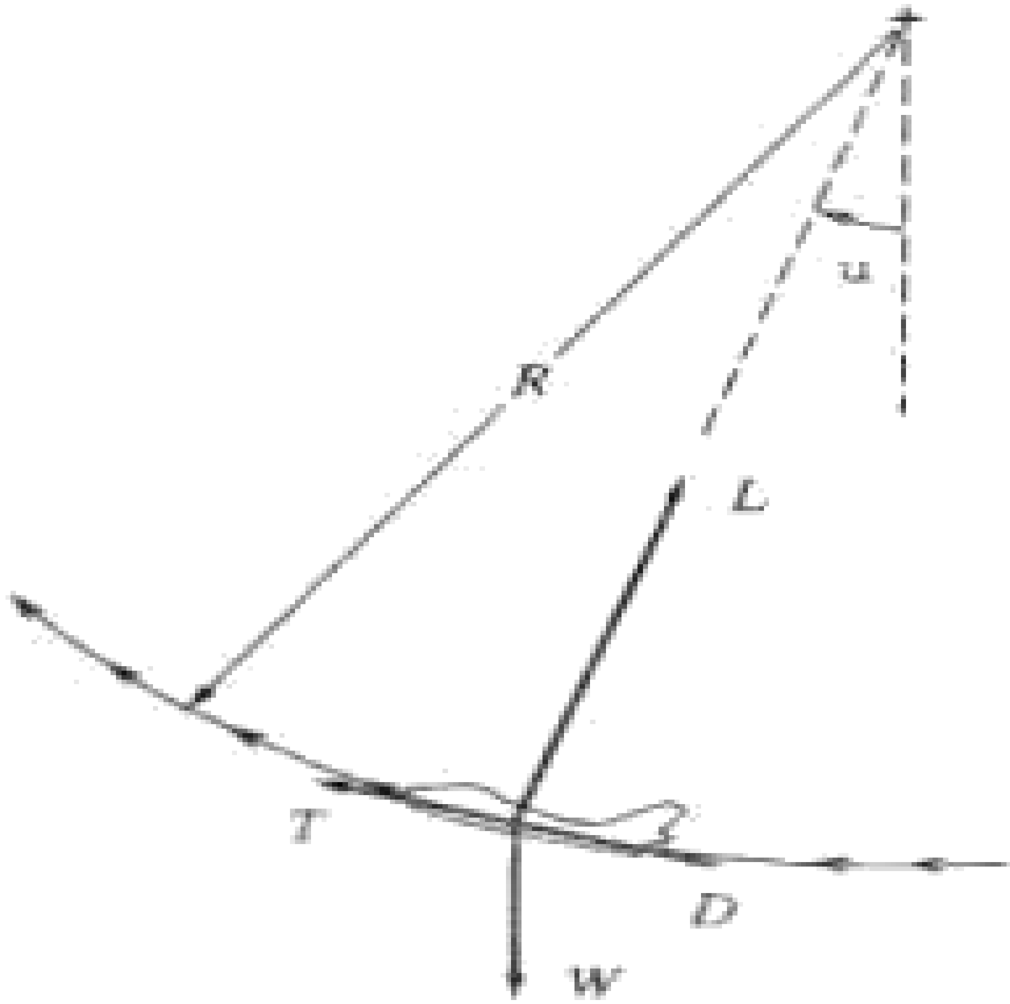

Fig 4.1 Pull up Manoeuvre

4.2 The Manoeuvre envelope

•The manoeuvre performance of the aircraft will be limited by the structural strength of the airframe; there are two basic reasons for this.

•The pressure loading produced by the dynamic pressure of the airflow increases with the square of the airspeed and with it the air loads on structural com opponents of the aircraft. This is particularly obvious in the case of the deflection of devices such as flaps or landing gear into the airstream. The design maximum dynamic equivalent airspeed, EAS.

•The normal acceleration associated with manoeuvreing flight produces structural loads in the airframe. The maximum allowable load factor in a manoeuvre is determined by the load bearing capability of the airframe structure.

•These limitations do not always apply universally. Since the configuration of the airframe may be changed for certain parts or the flight, for example by the lowering of the landing gear or the deflection of flaps, the structural loading and strength limits may be affected. Therefore, different maximum airspeed and load factor limitations may exist for each configuration. The design manoeuvre envelope, or n-V diagram, describes the design limitations on airspeed and load factor.

•The structural strength of the airframe must be capable of sustaining the structural loading generated by flight manoeuvres and gusts at a given air4raft weight5 and by the dynamic pressure of the airflow at the maximum permissible airspeed. The load factor limits and maximum airspeeds depend on the role of the aircraft.

•A military combat aircraft requires a high level of manoeuvreability and, therefore, the capability of sustaining a high load factor. In addition, it will be called upon to perform the manoeuvres over a wide range of airspeeds. These requirements lead to the need for a strong airframe to withstand the combined aerodynamic and manoeuvre loading; this implies that the airframe will be relatively heavily constructed.

•A large, subsonic, transport aircraft need only low manoeuvreability and can meet the entire manoeuvre and gust loading requirements with a relatively low load factor.

• The airspeed is likely to be limited by a sub sonic Mach number, which may enable a relatively low maximum EAS to be scheduled. These considerations lead to a much lower structural strength requirement and, consequently, a lighter airframe construction.

• Fig above shows the main elements of a typical manoeuvre envelope. Although the manoeuvre envelopes for civil and military aircraft differ in detail, the definitions of the boundaries and the principal airspeeds are similar enough to be generalized.

4.3 Transport aircraft manoeuvre performance

The effect of manoeuvres on the flight path performance of civil transport aircraft its generally not very significant. They spend only a very low proportion of their time in tur4ning flight and since the maximum bank angle used in the turn is typically 20*, the turns are generally of very low rate. The effect of such turns on the overall performance is minimal. En-route turns, which are generally associated with heading changes, are usually made through angles of less than 900. If the aircraft is stacked in the hold during the descent to landing it is required to fly an oval flight path, with a 180* turn at each turn, known as the holding pattern. The holding pattern turn is flown at Rate1, or 3*/s, and takes one minute. This is probably the most sustained turn that the aircraft will be required to carry out in normal operations and will be flown at a speed commensurate with flight safety and air traffic control considerations. This speed should be losing to the maximum endurance speed if possible.

•The rate and radius of the turns made by aircraft with differing airspeeds can be used to advantage in control of air traffic. The smaller regional‘aircraft, often turbo-props, can fly at lower speeds than the big jets and can use bank angles up to 30* compared with the normal maximum of 20* in the case of the big jets. This enables the regional aircraft to turn at a higher rate and with a smaller radius. These differences in performance can be used to aid the separation and flow of traffic in airport terminal areas.

•The only case in which the effect of turning would be significant to a transport aircraft, in terms of its effect on flight safety, is a turn made with marginal excess thrust available. For example, during the after-take-off climb with one engine inoperative, the additional drag due to the turn may decrease the already small gradient of climb. Similarly, the sustained pull-up is not a manoeuvre associated with civil transport operations. The most significant pitching manoeuvres, other than the transition at take-off and the flare on landing, will be transient manoeuvres between the limb and cruising states or between cruising and descending flight.

4.4 Military aircraft manoeuvre performance

•Military transport aircraft may be required to manoeuvre more aggressively than civil aircraft and sustained turning performance may be significant in the operational profile of the mission. Although pure pull-ups are still unlikely to be more than transient manoeuvres in path, they may extend, in the case of tactical transport aircraft, to pop-up‘ manoeuvres from low-level flight that would demand a higher rate of pitch than the simple gradient changing manoeuvre.

•The increased fuel burned due to the turning manoeuvres may need to be accounted for in the analysis of the fuel requirement for the mission.

•The combat aircraft (or the aerobatic aircraft) spends a significant proportion of its mission time in manoeuvreing flight and the design criterion for the aircraft is that it will be capable of sustaining high-g manoeuvres with high rates op turn. In the design phase of the development of the aircraft, estimates will be needed of the thrust required to achieve the design target manoeuvres combining turning with pull-ups and acceleration will also need to be considered to establish the limits of the aircraft in air-to-air combat situations. The performance in these manoeuvres will be part of the design specification of the aircraft. However, in service combat manoeuvres will be developed from combinations of the three basic manoeuvres to take advantage of the performance equalities peculiar to the aircraft. It is unlikely that such manoeuvres could be foreseen at the design stage, but their development in service through experience of the aircraft, is a necessary phase of the maintenance of the aircraft air superiority.

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UNIT – V –FLIGHT TESTING METHODS TO EVALUATE PERFORMANCE– SAEA1501

5. TAKE-OFF AND LANDING- SAFETY REQUIREMENTS - FLIGHT PLANNING

5.1 Take-off performance:

Consider an airplane standing motionless at the end of a runway. This is denoted by location O. The pilot releases the brakes and pushes the throttle to maximum takeoff power, and the airplane accelerates down the runway. At some distance from its starting point, the airplane lifts into the air. How much distance does the airplane cover along the runway before it lifts into the air? This is the central question in the analysis of takeoff performance. Called the ground roll (or sometimes the ground run) and denoted by sg in Fig. 5.1, it is a major focus of this section. However, this is not the whole consideration. The total takeoff distance also includes the extra distance covered over the ground after the airplane is airborne but before it clears an obstacle of a specified height. This is denoted by sa in Fig. 5.1. The height of the obstacle is generally specified to be 50 ft for military aircraft and 35 ft for commercial aircraft. The sum of s g and sa is the total takeoff distance for the airplane. The ground rolls g is further divided into intermediate segments, as shown in Fig.5.2. These segments are defined by various velocities, as follows:

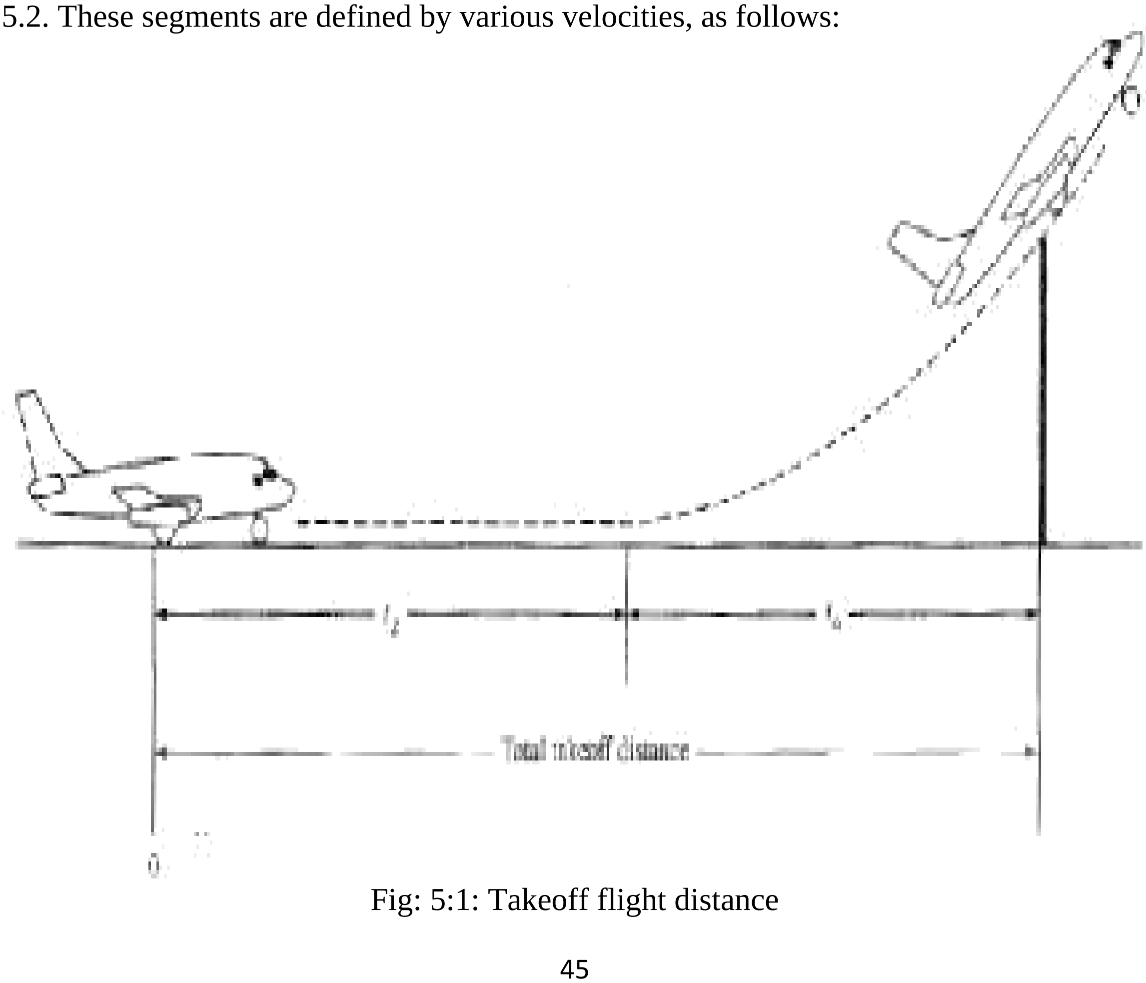


Fig: 5:1: Takeoff flight distance

•As the airplane accelerates from zero velocity, at some point it will reach the stalling velocity Vstall, as noted in Fig above. The airplane continues to accelerate until it reaches the minimum control speed on the ground, denoted by Vmcg in Fig. 5.2. This is the rr1inimum velocity at which enough aerodynamic force can be generated on the vertical fin with rudder deflection while the airplane is still rolling along the ground to produce a yawing moment sufficient to counteract t.'1at produced when there is an engine failure for a multiengine aircraft

•If the airplane were in the air (without the landing gear in contact with the ground), the minimum speed required for yaw control in case of engine failure is slightly greater than Vmcg· This velocity is called the minimum control speed in the denoted by Vmca in Fig. 5.2 For the ground roll shown in Fig. 5.2, Vmca is essentially a reference speed-the airplane is still on the ground when this speed is reached.

•The airplane continues to accelerate until it reaches the decision speed, denoted by V1 in Fig. This is the speed at which the pilot can successfully continue the takeoff even though an engine failure (in a multiengine aircraft) would occur at that point. This speed must be equal to or larger than V mcg in order to maintain control of the airplane. A more descriptive name for V1 is the critical engine failure speed. If an engine fails before V1 is achieved, the takeoff must be stopped. If an engine fails after V1 is reached, the takeoff can still be achieved.

•The airplane continues to accelerate until the takeoff rotational speed, denoted by VR in Fig. above , is achieved. At this velocity, the m1tiates elevator deflection a rotation of the airplane in order to increase the of attack, hence to increase Clearly, the maximum angle of attack achieved during rotation should not exceed the stalling angle of attack. Actually, all that is needed is an angle of attack high enough to produce a lift at the given larger than the so that the airplane will lift off the ground. However, even this angle of attack may not be achievable because the tail may drag the ground. (Ground clearance for the tail after rotation is an important design feature for the airplane, imposed by takeoff considerations.)

•If the rotation of the airplane is limited by ground clearance for the tail, the airplane must continue to accelerate while rolling along the ground after rotation is achieved, until a higher speed is reached where indeed the lift becomes larger than the weight. This speed is called the minimum unstick speed, denoted by Vmu in Fig. 5.2. For the definition of Vmu, it is assumed that the angle of attack achieved during rotation is the maximum allowable by the tail clearance.

•However, for increased safety, the angle of attack after rotation is slightly less than the maximum allowable by tail clearance, and the airplane continues to accelerate to a slightly higher velocity, called the liftoff speed, denoted by VLO in Fig. 5.2. This is the point at which the airplane actually lifts off the ground. The total distance covered along the ground to this point is the ground roll. The relative values of the various velocities discussed above, and noted on Fig. 5.2, are all sandwiche,d between the value of Vsta11 and that for VLO, where usually VLO ~ 1.1 Vstall ·

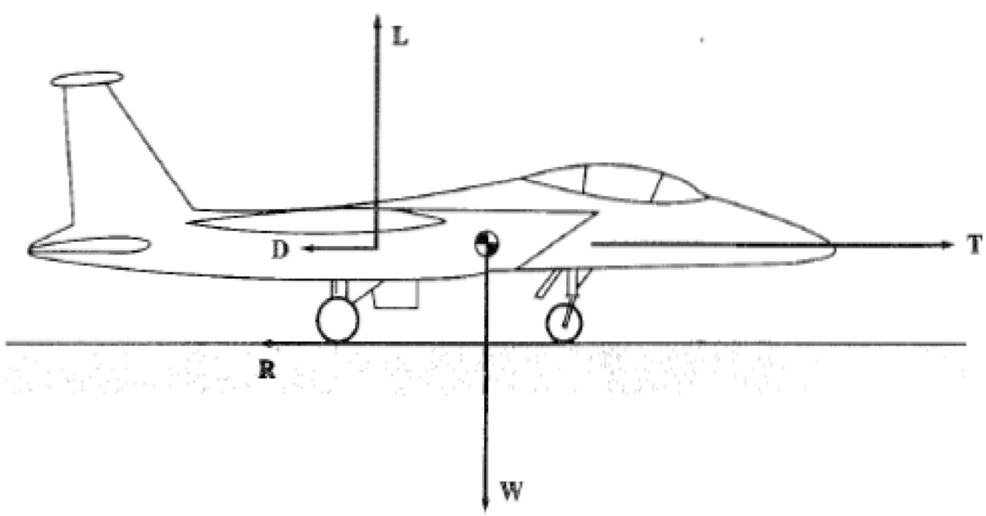
5.2 Related to the above discussion is the concept of balanced field length, defined as follows.

•The decision speed V1 was defined earlier as the minimum velocity at which the pilot can successfully continue the takeoff even though an engine failure would occur at that point. What does it mean that the pilot "can successfully continue the takeoff" in such an event? The answer is that when the airplane reaches V1, if an engine fails at that point,· then the additional distance required to clear the obstacle at the end of takeoff is exactly the same distance as required to bring the airplane to a stop on the ground. If we let A be the distance traveled by the airplane along the ground from the original starting point (point O in Fig. 5.2) to the point where V1 is reached, and we let B be the additional distance traveled with an engine failure (the same distance to clear an obstacle or to brake to a stop), then the balanced field length is by definition the total distance A + B.

The effect on the take-off distances of the flight variables:

•The take-off distances will be affected by the weight of the aircraft, the state of the atmosphere and the airfield conditions. Equations above enable the approximate effect of variation of the flight variables to be discussed in fig 5.2-5.3

Aircraft weight: If above equation is expressed for a take-off on a level runway in still air then the approximation for the ground run distance is.an increase in aircraft weight can be seen to have two direct effects on the ground run
distance.



Forces acting on an airplane during takeoff and landing.

Fig.5.2 Take-off speeds and distances.

First, the ground run distance is directly proportional to aircraft weight, so that the ground run will increase in proportion to the weight increase. Secondly, the ground run distance is direct fly proportional to the square of the light-off speed. Vtof. Now, the lift-off speed is proportional to the stalling speed, Vsl. Which, in turn, is proportional to the square root of the weight of the aircraft? Therefore, the take-off ground run distance will be increased, again in proportion to the aircraft weight.

•In addition to the direct effects, the increased weight will increase the runway friction force acting on the aircraft, but the effect of this on the ground run will be relatively small compared with the direct effects of weight. Summing the individual effects of a weight increase, it can be expected that increasing the aircraft weight by 10% will increase the take-off ground run distance by at least 20%. The airborne distance will be similarly affected

• First, from above equation it can be seen that the airborne distance is directly proportional to the aircraft weight. Secondly, both the lift-off speed and the take-off safety sped will be increased by the increase in aircraft weight. This will produce a proportional increment in the kinetic energy related term since the airborne distance is a function of the square of the airspeeds, which are proportional to the square root of the weight. However, as the potential energy and kinetic energy terms in the expression for the airborne distance are roughly equal, the effect of weight increase of 10% on the kinetic energy term is to increase the airborne distance by about 5%.

•Also, the increase the airframe drags and reduces the excess thrust available. The magnitude of this effect will depend on the excess thrust and, on the lift,-dependent part of the drag characteristic of the airframe. For a transport aircraft, it will probably equate to a reduction in the excess thrust of the order of 2% for a weight increase of10%. By summing these effects, it can be expected that increasing the weight of the aircraft by 10% will increase the airborne distance by between 15% and 20%.

5.3 Atmosphere state effects:

The state of the atmosphere has two basic effects on the take-off distances. The take-off is performed with reference to indicated airspeed (IAS) displayed by the airspeed indicator. In the absence of pitot-static errors, the IAS will be the same as the equivalent airspeed, EAS (since the take-off is made at low speed and flow attitude and the scale-altitude correction will be negligible). However, the take-off distances are functions of true airspeed (TAS), since they are determined by the motion of the aircraft in Earth axes. Since the ground run distance is proportional to the TAS2 it will be proportional to the inverse of the relative density. I/o, or 0/8. This implies that hot or high (low relative pressure) conditions will increase the take-off ground run since the TAS will be increased for a given EAS determined by the aircraft weight. The airborne distance will also be increased but, because only the kinetic energy term is affected, the effect will be about halved.

•The output of the power plant is roughly proportional to the relative density since the engine output is dependent point‘s air mass flow. Therefore, the net thrust will decrease in hot or high conditions increasing the take-off distances; the magnitude of the effect will depend on the characteristics of the power plant.

5.4 Head wind:

The effect of the headwind, Vw, is to change the datum speed of the take-off; the aircraft now only needs to be accelerated to a ground speed of Vl of – Vw. From above equation, it can be seen that the effect of a headwind equal to 10% of the lift-off speed is to decrease the ground run by almost 20%. Conversely, the same tailwind would increase the ground run by a little over 20% The effect of the headwind on the airborne distance is approximately half as severe as it is on the ground run distance since it only affects the kinetic energy term.

5.5 Runway conditions:

The effects of a runway slope (uphill) and the runway friction coefficient on the ground run distance can each be accounted for by considering them as equivalent to a decrease in the take-off thrust-to-weight ratio. There is, of course, no effect on the airborne distance.
It will be seen later, that account is taken of the effect of the flight variables on the take-off performance data in their presentation in the aircraft performance manual. The effects of the weight, wind and runway slopes are each accounted for by using the corrections developed above to factor the datum performance measured at a known atmosphere state.

5.6 Landing performance:

In the landing phase of the flight, the aircraft is on a descending flight path towards the runway. As it approaches the runway, the airspeed and the rate of descent are reduced in the flare so that a touchdown is achieved at a low rate of descent. After touchdown, the nose is lowered onto the runway and the aircraft brought to a halt. During the landing, consideration is given to the need to ensure that the aircraft can be controlled safely and that the distances required for the manoeuvres do not exceed those available. The landing manoeuvre is described in Fig. 5.3.

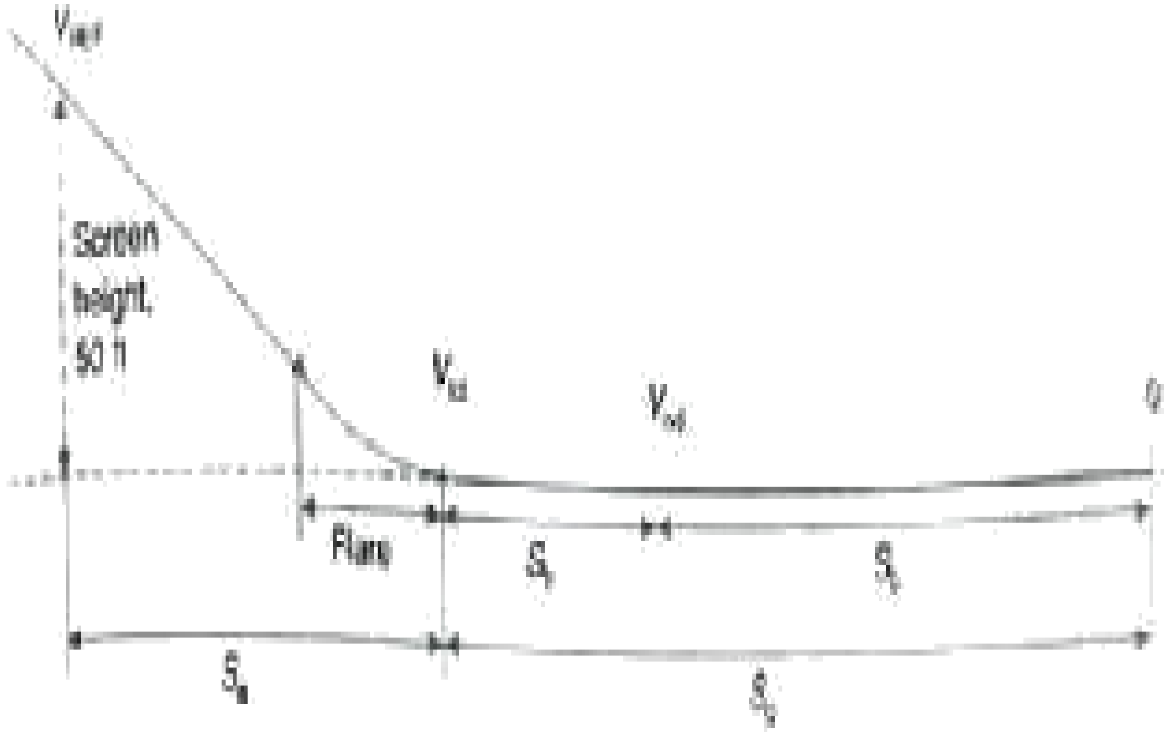


Fig.5.3 landing speeds and distances.

The effect on the landing distances of the flight variables: The landing distances will be affected by the weight of the aircraft, the state of the atmosphere and the airfield conditions, in a broadly similar manner to the take-off distances.

5.7 Aircraft weight:

In the approach phase of the landing the thrust produced by the engines will be less than the maximum thrust available. Therefore, the gradient of the approach flight path can be controlled by the engine thrust and will not be affected by the weight of the air craft. In the flare, the thrust is reduced to idle and the aircraft decelerates as the angle of attack is steadily increased towards the touchdown angle of attack to arrest the rate of descent. Since the angle of attack on the approach and at touchdown is usually determined by the stalling speed of the aircraft in the landing configuration, the lift – drag ratio during the flare will be practically independent of the aircraft weight. This implies that, unlike the airborne take-off distance, the only significant effect of the weight on the airborne landing distance will be due to the kinetic energy term in the flare. Because the approach is made at about 1.3Vso, the kinetic energy loss during the flare will be about twice as much as the potential energy loss. Therefore, a weight increase of 10% would be expected to increase the distance in the flare by about7%. The effect of the weight on the landing group under distance is similar to the take-off case; it increases the ground speed at touchdown through its effect on the stalling speed. The direct effect of the increased weight on the kinetic energy will increase the ground run distance by about 20% for a weight increase of 10%. This affects both the free roll distance and the braked ground run.

5.8 Atmosphere state effects:

The state of the atmosphere will affect the landing distances through its effect on the TAS. There will be no significant effect through the engine thrust since the thrust is either being controlled to maintain the approach gradient or will be at idle. As in the take-off case, the ground run distances will be increased in proportion to the inverse of their density and the flare.

5.9 Headwind:

The effect of the headwind on the ground run distance is to change the datum speed of the landing since the aircraft now only needs to be declared from an effect touch-down speed. The effect of a headwind equal to 10% of the touch-down speed would be to reduce the landing ground run by about20%.The effect of the head window the airborne distance is only felt in the flare since the approach gradient is relative to the ground and is controlled by engine thrust or power. A headwind of 10% of the tough down speed will reduce the kinetic energy loss during the flare by 20% and the distance in the flare by about 14

5.10 Runway conditions:

The effects of a runway slope (uphill) and the runway friction coefficient on the ground run distance can each be accounted for by considering them as equivalent to an increase in the braking force. There is, of course, no effect on the airborne distance

5.11. The space available:

As in the case of take-off the space available for landing is limited by the dimensions of the air5frield and the approach to the runway in the landing direction, see Fig 5.4. The approach path is protected by an obstacle limitation surface in a similar manner to the take-off net flight path. The definition of the obstacle limitation surface depends on the classification of the runway. Figure 5.5shows the limitations for a large airfield with a precision approach category; full definitions for all runway classifications can be found in ICAO Annex 14. As in the case of the take-off obstacle free zone, it may not be possible to comply fully with the requirements, and the flight planning will need to consider any known obstructions near the airfield.

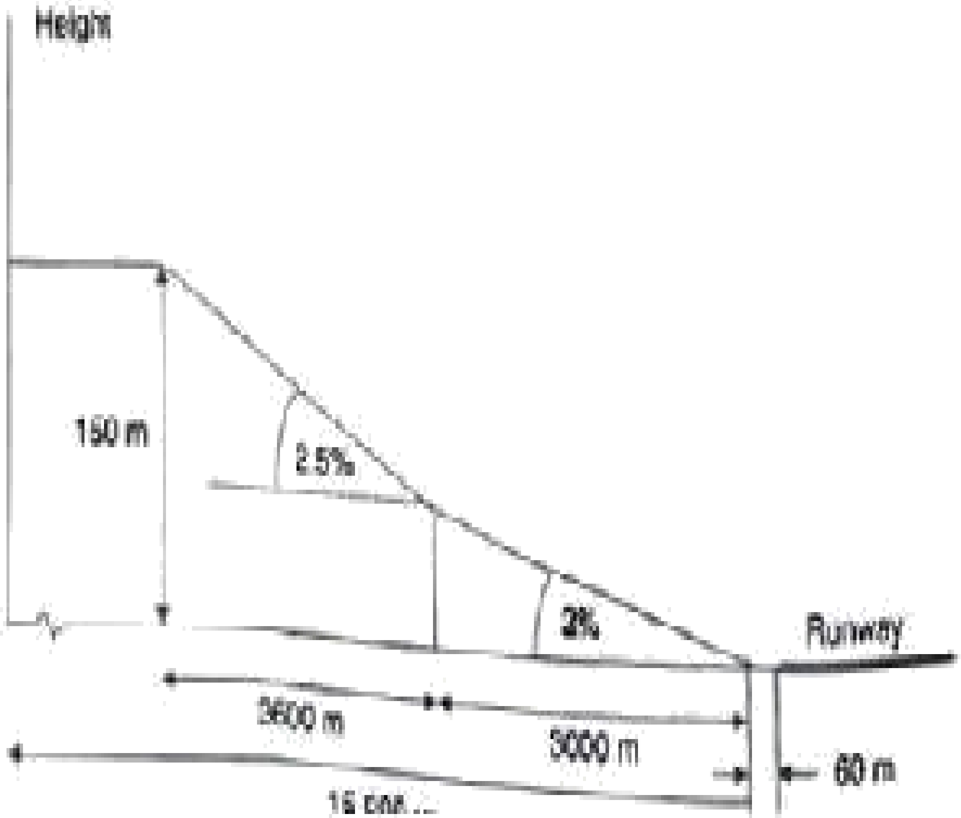


Figure:5.4 Landing approach obstacle limitation surface.

The first section of the obstacle limitation surface starts from a point 60m from the threshold of the runway and edxt5ends rfo4r a distance of 3000m along the appr4oadh path at a gradient of 2%.

The second section extends a further 3600m at a gradient of 2.5%. The horizontal section extends a further 8400m with a base height of 150m above runway surface. The total length of the approach obstacle limitation su4rface is 15000m. There are of course, lateral dimensions associated with the obstacle limitation surface to enable the air45craft to make turning and positioning manoeuvres on the approach. As these do not affect the performance considerations addressed here, they need not be considered in detail. In the event of a missed approach or a baulked landing, in which the aircraft abandons the landing and climbs away from the airfield, there needs to be an obstacle free area in the direction of the climb-out. The baulked-landing obstacle clearance surface extends from 1800m beyond the threshold of the landing runway, or the end of the runway whichever is the lesser, at a gradient of 3.33%, to a distance of 4000m. The landing distance available (LDA) is defined as the length of runway that is declared available and suitable for the ground run of an aircraft landing.

5.12. The space required

The landing distance required (LDR) is the gross distance required o land on a legal, smooth, hard-strafed runway from a specified screen height at the runway threshold and to come to a complete stop, multiplied by a suitable safety factor. The landing flight path assumes that6 the aircraft approaches the runway in a steady descent to a screen height of 50ft at the runway threshold. The gradient of the descent in the approach is usually 5% (or 3*), but may be steeper in some restricted airfield oper4ations. The airspeed at the screen height should be not less than the greater of V mcl of 1.3Vs in the land in configuration. Between the screen height and touchdown, the airspeed is reducedto a safe touchdown speed in the flare. This is done by progressively increasing the angle of attack, so that the touchdown is achieved at an acceptable vertical speed avoiding excessive vertical acceleration or any tendency to bounce, in the flare, no changes to the configuration, addition of thrust or depression of the nose should be required. After touchdown, brakes and other means of retardation, for which a satisfactory means of operation has been approved, can be used to decelerate the aircraft to a halt. If any retarding system depends on the operation of any engine, for example a thus reverser, then it may be necessary to assess the landing distances with that engine inoperative to determine the most critical Landing distance required. To account for wet runway conditions, the dry runway distances are factored by a function of the coefficient of friction of the runway under wet and dry conditions.

The maximum landing weight based on the landing distance required is determined by the balance between the landing distance available and the greatest of the landing distances required. This takes into account the factors for the runway surface, slope and condition, the wind and an overall safety factor to account for the statistical variation in the parameters affecting the landing performance.

5.13 The discontinued landing:

There are two cases to be considered in which the landing has to be abandoned. These are,

•The discontinued approach in which the aircraft terminates the approach before the thrust is reduced in the flare and continues the flight to a point from which a new approach can be made, and the baulked landing in which the aircraft is required to go-around after the thrust has been reduced in the flare.

•The discontinued approach occurs, for example, when the aircraft has reached the minimum decision height on the approach to landing and the run way is not insight. At that point the approach must be discontinued and a climb initiated; it is assumed for the purpose of the requirements that the aircraft is flying with one engine in operative. In a go-around from a discontinued approach, the aircraft must be capable of climbing away, with the critical engine inoperative and in the approach configuration, to a safe height from which it can make another approach or a diversion.

•The steady gradient of climb must not be less than 2.1% for two-engine aircraft, 2.4% for three-engine aircraft and 2.7% for four-engine aircraft; the different gradients reflect the effect of statistical variability in the thrust of the operating engines (For Category II operations the gradient is 2.5% for all types.) The climb gradient must be demonstrated with the critical engine inoperative and the operating engines at take- off thrust, at maximum landing weight and with the landing gear retracted. The configuration for the discontinued approach must be such that the stalling speed is not greater than 1.1Vs for the all-engines operating landing configuration.

In the case of the baulked landing the aircraft must be capable of climbing at a gradient that will maintain clearance from all obstructions with all engines operating in the landing configuration

and with the landing gear down. When the go-around is initiated, take-off thrust is selected and the aircraft is rotated into the climb. With all engines operating, the aircraft must be capable of achieving as gradient of climb of not less than 3.2% in the landing configuration. The engine thrust or power used to calculate the gradient of climb is that which is available 8 seconds after-take-off power is selected from the flight idle condition. The airspeed used for the climb is 1.2Vs, but must not be less than V mcl or more than the greater of V mcl and 1.3Vs.
The gradients of climb in the discontinued landing will determine the maximum weight at which the aircraft can safely terminate the approach taking into account the WAT conditions. In most cases the limit of compliance for the discontinued approach and the baulked landing can be combined into a single WAT chart for the aircraft, this will be discussed further in Chapter 10. The maximum landing weight for the aircraft will be the lowest of the weights determined by consideration of the climb gradient in the discontinued approach and the baulked landing can be combined into a single WAT chart for the aircraft. The maximum landing weight for the aircraft will be the lowest of the weights determined by consideration of the climb gradient in the discontinued land in the space available and the maximum design landing weight.

5.14 The trip fuels

•The flight from the point of departure to the intended destination is made up of the basic elements of the flight path, take-off, climb, cruise, descent and landing in addition, the aircraft will need to taxi from the ramp to the runway for take-off and from the runway to the ramp after landing. The fuel required for each element of the intended flight is derived from flight measured data. This is reduced to a form from which the effects of weight, altitude and temperature can be interpolated separately.The data are presented in graphical or tabular form so that the fuel required for each element of the flight can be found for the particular WAT state and operational parameters.For example, the cruising segment will need a statement of the distance between the end of the climb and the beginning of the descent and the climbing segment will need the height increase between take-off and cruise altitude. In some case, it may be sufficient to state an allowance of fuel for a manoeuvre or fuel burned per unit time.

For example, in the case of smaller aircraft it is not unusual to quote a fixed quantity of fuel for the take-off and landing and for the taxiing fuel in units per minute. This is acceptable if the effect of variation of WAT on the fuel.Since the fuel consumed during the trip depends on the weight of the aircraft, a starting weight�to is needed before the trip fuel can be calculated.The initial weight of the aircraft will depend on the quantity of fuel that will be burned during the trip. Hence, the only weight hat can be determined is the weight at the end of the diversion at which time only the reserve fuel remains.Therefore, the calculation process is reversed, so that the starting weight‘ for the fuel calculation is the weight at the end of the flight, rather than that at the beginning. The starting weight for the calculation of the trip fuel is, therefore, the aircraft prepared for service (APS) weight of the aircraft plus its payload and the fuel remaining at the end of the trip.The fuel required for landing descent, cruise, climb and take-off can be calculated in their reverse order and summed to give the trip fuel.This process estimates the minimum quantity of fuel required for the trip based on assumptions that include the route to be flown, cruise altitude, atmosphere state and forecast winds. In practice, however, the ideal flight plan is rarely achieved and the actual trip will differ from the trip assumed for flight planning; the reasons for this include the following.The flight planning will be completed sometime before the flight and will use forecast whether (temperature profiles and winds), request the best cruise altitude and route and will be based on a specified departure time. The flight plan may not be confirmed until just before take-off, or even after- take-off, and may require changes to the route and cruise altitude to coordinate the flight with other traffic. Any such changes will usually increase the time and distance of the flight and extra fuel will be required. Delays in departure times may result in different air temperatures at take-off and en-route from those used for the flight plan. For example, if an early morning departure is delayed until mid-day, the increase in air temperature may increase the trip fuel required. Consequently, it could mean that the take-off WAT limit would be exceed and the payload would then have to be reduced to comply with the scheduled performance requirements. On arrival at the destination, the aircraft may have to hold to await its turn to land. Flying in the hold will require additional fuel.

To account for the additional fuel requirement caused by environmental effects (temperature and wind), the trip fuel can be increased by a percentage of the calculated en-route fuel. In addition contingency ‘allowances can be added a account for holding and unscheduled manoeuvres or route changes. The percentage increase of the en- route fuel may be based on a requirement set by the regulatory authorities or decided by the operator. The contingency fuel allowances are usually determined by the operator and based on knowledge and experience of the route.

5.15.The diversion fuel:

The fuel for the diversion is calculated in the same manner as the trip fuel, but usually it assumes the same contingency fuel as the trip.

5.16. Reserves:

Minimum reserves are usually set by the regulatory authorities although the operator may increase them at his or her discretion. An aircraft should not need to use fuel from its minimum reserves except in an emergency reserve are additional to the minimum reserv

5.17 Tankering:

•In addition to the fuel required for the mission, the opportunity may be taken to carry extra fuel if it is economically advantageous to do so, this is known as tankering. Fuel can be tinkered if the price at eh departure point is sufficiently below the price at the destination to make it worth using any available surplus weight to carry the extra fuel. In this way, the cost of fuel uplift a for the next flight can be reduced and the overall economy of the operation improved. Tinkered fuel cannot be considered as fuel reserve for fuel planning purposes. The fuel planning progression is shown in Fig 5.4. The payload weight is added to the aircraft prepared for service weight to give the zero-fuel weight; this is the basic weight of the aircraft for that mission. The landing weight at the alternate airfield is the sum of the zero-fuel weight and the fuel reserves that have not been used, together with any tinkered fuel being carried. The landing weight at the destination is the landing weight at the alternate plus the fuel for the diversion and, possibly, a part of the percentage en-route reserve if it has not been required The greatest landing weight as the destination would occur if no reserves had been use; this case is shown in Fig. 5.4. The trip fuel added to the landing weight at the destination gives the take-off weight, which must not exceed the maximum scheduled take-off weight determined by the performance planning.

•Taxing fuel may be carried to permit the aircraft to taxi to the take-off point and take-off as its limiting weight; this is added to the take-off weight to give the ramp weight. The fuel plan usually contains a simple self-check by adding the total fuel weight to the zero-fuel weight to give the ramp weight directly.Comparing this with the ramp weight from the detailed plan helps to eliminate any arithmetic errors that may occur in the process. Although the performance plan takes precedence over the fuel plan in flight planning, the fuel plan may need to be completed first in order to find the fuel required to transport the aircraft and payload to the intended destination. If the aircraft cannot meet any of the performance requirements, then its weight must be reduced until it is able to do so. The APS weight is fixed and cannot be reduced.

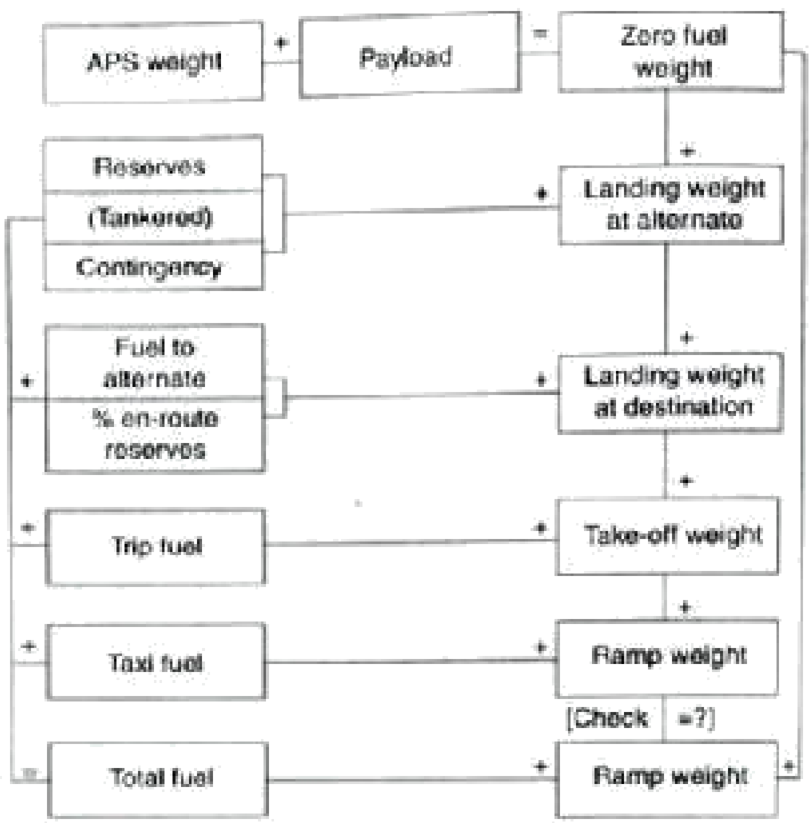


Figure: 5.5. Fuel planning