AIRCRAFT PERFORMANCE AND STATIC STABILITY

16BTAR602

16BTAR602 AIRCRAFT PERFORMANCE AND STATIC STABILITY
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OBJECTIVES:

- Ability to analyse the performance of aircraft under various Flight conditions such as take off, cruise, landing, climbing, gliding, turning and other maneuvers
- Ability to analyse the static response of aircraft for both voluntary and involuntary changes in flight conditions

UNIT - I LIFT AND DRAG ON FLIGHT PERFORMANCE

Streamlined and bluff bodies, aerofoil classification, Aerofoil characteristics, Pressure distribution around aerofoil's.. Types of drag, Effects of Reynolds number on skin friction and pressure drag, Drag reduction of airplanes., Induced drag, chord wise and span wise pressure distribution. Aspect ratio, Camber and plan form characteristics ,drag polar.

UNIT - II STEADY FLIGHT:

Steady level flight, Thrust/power, available and required with altitude Estimation of maximum level flight speed, conditions for minimum drag and minimum power required.

UNIT - III GLIDING,CLIMBING ANDTURING PERFORMANCE:

Maximum range, Minimum rate of skin a glide, Shallow angle of climb,Rate of climb ,time to climb and ceilings, Glide hodograph. Bank angle and load factor, Limitations on turn, Pull up and pushover, the v-n diagram.

UNIT - IV SPECIAL PERFORMANCE:

Range and endurance of jet and propeller type of airplanes , estimation of take-off and landing distance .High lift devices, Use of thrust augmentation and reverse thrust.

UNIT - V STATIC LONGITUDINAL STABILITY

Degree of freedom of rigid bodies in space - Static and dynamic stability - Purpose of controls in airplanes -Inherently stable and marginal stable airplanes – Static, Longitudinal stability - Stick fixed stability - Basic equilibrium equation - Stability criterion - Effects of fuselage and nacelle - Influence of CG location - Power effects - Stick fixed neutral point - Stick free stability-Hinge moment coefficient -Stick free neutral points-Symmetric maneuvers - Stick force gradients - Stick force per 'g' -Aerodynamic balancing.

TEXT BOOKS:

S.NO.	AUTHOR(S)	TITLE OF THE BOOK	PUBLISHER	YEAR OF PUBLICATION
1.	John D. Anderson	Aircraft Performance and Design	Tata McGraw-Hill, New Delhi.	2011
2.	Perkins C.D &Hage,	Airplane performance, stability and control	Wiley &Sons,New Delhi	2011
3.	Bandu N. Pamadi	Performance, stability, dynamics, and control of airplanes, second edition	AIAA Education Series,Washington DC	2004

REFERENCE BOOKS:

S.NO.	AUTHOR(S)	TITLE OF THE BOOK	PUBLISHER	YEAR OF PUBLICATION
1.	Perkins, C. D., and Hage, R, E.	Airplane Performance, Stability and Control	Wiley Toppan, Mumbai.	1974
2.	Babister, A.W.	Aircraft Stability and Response	Pergamon Press, Newyork.	1980
3.	McCormick B.W	Aerodynamics, Aeronautics &Flight Mechanics	John Wiley, New York.	2009
4.	Nelson R.C	Flight stability & Automatic Control	Tata McGraw Hill,New Delhi.	2007
5.	Warren F Phillips	Mechanics of flight	John Wiley, New York.	2009

WEB REFERENCE:

- nptel.ac.in/courses/101106042
- https://www.classle.net/category/subject-area/**flight-dynamics-ii**
- www.myopencourses.com › Courses › Aerospace Engineering
- www.nal.res.in/.../Flight%20Mechanics%
- www.myopencourses.com › Courses › Aerospace Engineering

KARPAGAM ACADEMY OF HIGHER EDUCATION

(Deemed to be University Established Under Section 3 of UGC Act 1956)

Faculty of Engineering

DEPARTMENT OF MECHANICAL ENGINEERING(AEROSPACE)

LESSON PLAN

Subject Name : **AIRCRAFT PERFORMANCE AND STATIC STABILITY**
Subject Code : **16BTAR602** **(Credits - 3)**
Name of the Faculty : **ARUN PRAKSASH J**
Designation : **ASSISTANT PROFESSOR**
Year/Semester/Section : **III/VI SEM**
Branch : **B.Tech Aerospace Engineering**

Sl. No.	No. of Periods	Topics to be Covered	Support Materials
1.	**1**	**Introduction and Fundamentals for the Course**	
UNIT – I : LIFT AND DRAG ON FLIGHT PERFORMANCE			
2.	1	Streamlined and bluff bodies	R2
3.	1	Aerofoil classification	R2
4.	1	Aerofoil characteristics	T1,R2
5.	1	Pressure distribution around airfoil's	T1,R2
6.	1	Types of drag.	R2
7.	1	Effects of Reynolds number on skin friction and pressure drag	T1,R2
8.	1	Induced drag and Drag reduction of airplanes	T1,R2
9.	1	Chordwise and span wise pressure distribution. Aspect ratio, Camber and plan form characteristics	T1,R1
10	1	Drag polar	T1,R1
11	1	Tutorial - Summary of Unit I and Objective Type Questions Discussion	
Total No. of Hours Planned for Unit - I			**11**

Sl. No.	No. of Periods	Topics to be Covered	Support Materials
UNIT II : STEADY FLIGHT			
12.	1	Steady level flight	T1,R3
13.	1	Thrust available with altitude	T1,R3

14.	1	Thrust required with altitude	T1,R3
15.	1	Power available with altitude	T1,R3
16.	1	power required with altitude	T1,R3
17.	1	Estimation of maximum level flight speed	T1,R3
18.	1	Conditions for minimum drag	T1,R3
19.	1	Conditions for minimum power required	T1,R3
20.	1	Problems on thrust and power	T1,R3
21.	1	Tutorial - Summary of Unit II and Objective Type Questions Discussion	
Total No. of Hours Planned for Unit - II			**10**

Sl. No.	No. of Periods	Topics to be Covered	Support Materials
UNIT III - GLIDING, CLIMBING AND TURING PERFORMANCE			
22.	1	Maximum range and Minimum rate of skin a glide	T1,T2
23.	1	Shallow angle of climb	T1,T2
24.	1	Rate of climb and time to climb	T1
25.	1	Ceilings	T1,R3
26.	1	Glide hodograph.	T1,R3
27.	1	Bank angle and load factor	T1,R1
28.	1	Turning Performance and Limitations on turn	T1,R1
29.	1	Pull up and push over maneuverer	T1
30.	1	The V-n diagram.	T1,R3
31.	1	Tutorial -Summary of Unit III and Objective Type Questions Discussion	
Total No. of Hours Planned for Unit - III			**10**

Sl. No.	No. of Periods	Topics to be Covered	Support Materials
UNIT – IV SPECIAL PERFORMANCE			
32.	1	Range of jet and propeller type of airplane	T1,R3
33.	1	Problems on Range	R3
34.	1	Endurance of jet and propeller type of airplane	T1,R3
35.	1	Problems on Endurance	R3
36.	1	Estimation of take-off distance.	T1,R3
37.	1	Estimation of landing distance.	T1,R3
38.	1	High lift devices	R3

39.	1	Thrust augmentation	R2,R3
40.	1	Reverse thrust.	R2,R3
41.	1	Tutorial - Summary of Unit IV and Objective Type Questions Discussion	
Total No. of Hours Planned for Unit - IV			**10**

Sl. No.	No. of Periods	Topics to be Covered	Support Materials
UNIT V- STATIC LONGITUDINAL STABILITY			
42.	1	Degree of freedom of rigid bodies in space - Static and dynamic stability	T2 ,R2
43.	1	Purpose of controls in airplanes -Inherently stable and marginal stable airplanes	T2 ,R2
44.	1	Static, Longitudinal stability and Stick fixed stability	T2 ,R2
45.	1	Basic equilibrium equation and Stability criterion , Effects of fuselage and nacelle	T2 ,R2
46.	1	Influence of CG location and Power effects	T2 ,R2
47.	1	Stick fixed neutral point and Stick free stability	T2 ,R2
48.	1	Hinge moment coefficient and Stick free neutral points	T2 ,R2
49.	1	Symmetric maneuvers and Stick force gradients	T2 ,R2
50.	1	Stick force per 'g' and Aerodynamic balancing.	T2 ,R2
51.	1	Tutorial -Summary of Unit V and Objective Type Questions Discussion	
52.	1	**Discussion on University previous year questions**	
Total No. of Hours Planned for Unit - V			**10+1**

TOTAL PERIODS : 52

TEXT BOOKS

T [1] – Aircraft Performance and Design, John D. Anderson, Tata McGraw-Hill, 2011
T [2] – Airplane performance, stability and control by Perkins C.D and Hage, Wiley & Sons,2011

REFERENCES

R [1] - Aerodynamics, Aeronautics &Flight Mechanics by McCormick B.W, John Wiley, 1995
R [2]- Aerodynamics, Clancey L.J, Sterling Book House ,2006
R [2] - Flight stability & Automatic Control by Nelson R.C, Tata McGraw,2007

WEBSITES

W [1] - nptel.ac.in//
W [2] - www.nal.res.in
W [3] - www.myopencourses.com › Courses › Aerospace Engineering

JOURNALS

J [1] - Progress in Aerospace Sciences – Elsevier
J [2] – AIAA Journal- AIAA.
J [3] – Canadian Aeronautics and Space Journal - Canadian Aeronautics and Space Institute.
J [4] – Journal of Guidance, Control, and Dynamics - AIAA
J [5] – Aerospace Science and Technology - Elsevier

UNIT	Total No. of Periods Planned	Lecture Periods	Tutorial Periods
I	11	9+1	1
II	10	9	1
III	10	9	1
IV	10	9	1
V	10+1	9+1	1
TOTAL	52	45+2	5

I. CONTINUOUS INTERNAL ASSESSMENT : 40 Marks

(Internal Assessment Tests: 25, Attendance: 5, Assignment5 ,Seminar: 5)

II. END SEMESTER EXAMINATION : 60 Marks

TOTAL : 100 Marks

FACULTY HOD / MECH DEAN / FOE

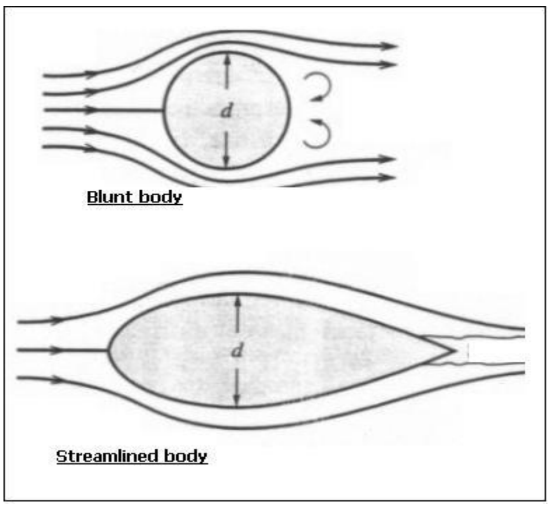
AIRCRAFT PERFORMANCE

UNIT I

LIFT AND DRAG ON FLIGHT PERFORMANCE

STREAMLINED AND BLUFF BODIES

A body moving through a fluid experiences a drag force, which is usually divided into two components: frictional drag and pressure drag. Frictional drag comes from friction between the fluid and the surfaces over which it is flowing. This friction is associated with the development of boundary layers, and it scales with Reynolds number as we have seen above. Pressure drag comes from the eddying motions that are set up in the fluid by the passage of the body. This drag is associated with the formation of a wake, which can be readily seen behind a passing boat, and it is usually less sensitive to Reynolds number than the frictional drag. Formally, both types of drag are due to viscosity (if the body was moving through an inviscid fluid there would be no drag at all), but the distinction is useful because the two types of drag are due to different flow phenomena. Frictional drag is important for attached flows (that is, there is no separation), and it is related to the surface area exposed to the flow. Pressure drag is important for separated flows, and it is related to the cross-sectional area of the body. When the drag is dominated by viscous drag, we say the body is streamlined, and when it is dominated by pressure drag, we say the body is bluff.



AEROFOIL CLASSIFICATION

An airfoil is a stream lined body, or a lifting surface, of simple shape that provides sufficient lift and considerably less drag at small angle of attack. Airfoils are of different shapes are sizes depending on the specifications and configuration of the intended aircraft.

There are three basic types of airfoils.

- Symmetrical Airfoils
- Unsymmetrical Airfoils
- Flat Bottom Airfoils
- Supersonic Airfoils

An airfoil can be either positive camber or a negative camber.

AEROFOIL CHARACTERISTICS

The geometry of the airfoil can be described with the following terms

- LEADING EDGE

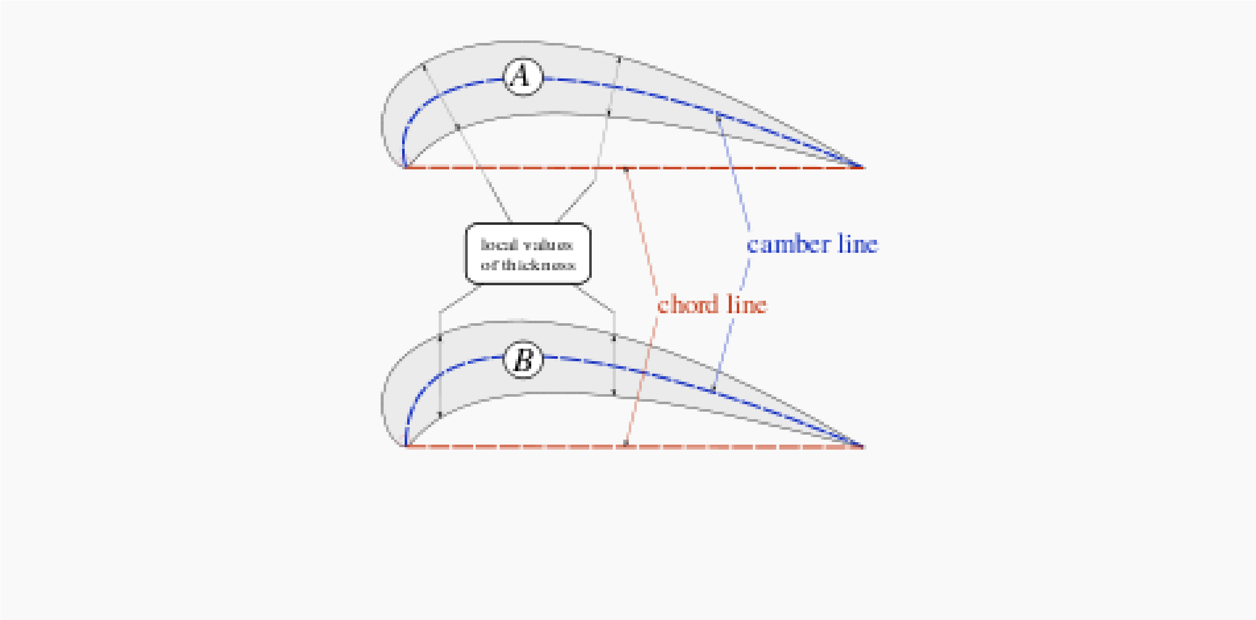
The point at the front of the airfoil that has maximum curvature (minimum radius).

- TRAILING EDGE

The point of maximum curvature at the rear of the airfoil.

- CHORD LINE

The straight line connecting leading and trailing edges. The chord length, or simply chord, c, is the length of the chord line.



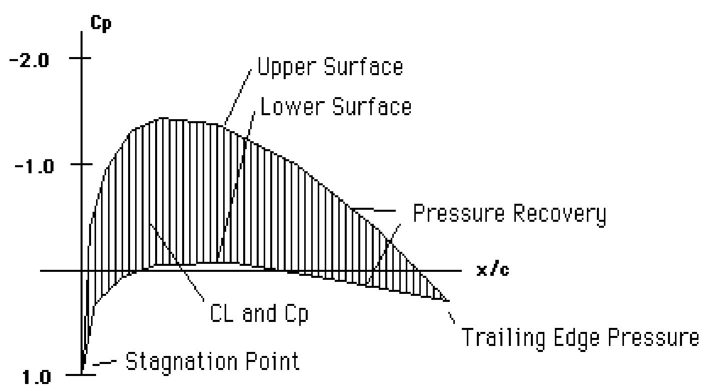
- MEAN CAMBER LINE OR MEAN LINE

The locus of point's midway between the upper and lower surfaces. Its shape depends on the thickness distribution along the chord.

PRESSURE DISTRIBUTION AROUND AEROFOILS

The aerodynamic performance of airfoil sections can be studied most easily by reference to the distribution of pressure over the airfoil. This distribution is usually expressed in terms of the pressure coefficient.

- Cp is the difference between local static pressure and freestream static pressure, non dimensionalized by the free stream dynamic pressure.
- x/c varies from 0 at the leading edge to 1.0 at the trailing edge. C_p is plotted "upside-down" with negative values (suction), higher on the plot. (This is done so that the upper surface of a conventional lifting airfoil corresponds to the upper curve.)
- The C_p starts from about 1.0 at the stagnation point near the leading edge.
- It rises rapidly (pressure decreases) on both the upper and lower surfaces and finally recovers to a small positive value of C_p near the trailing edge.
- Various parts of the pressure distribution are depicted in the figure below and are described in the following sections.



- Upper Surface
  The upper surface pressure is lower (plotted higher on the usual scale) than the lower surface Cp in this case. But it doesn't have to be.

- Lower Surface
  The lower surface sometimes carries a positive pressure, but at many design conditions is actually pulling the wing downward. In this case, some suction (negative Cp -> downward force on lower surface) is present near the mid chord.

- Stagnation Point
  The stagnation point occurs near the leading edge. It is the place at which V = 0. Note that in incompressible flow C_p = 1.0 at this point. In compressible flow it may be somewhat larger.

TYPES OF DRAG

Drag refers to forces acting opposite to the relative motion of any object moving with respect to a surrounding fluid. This can exist between two fluid layers (or surfaces) or a fluid and a solid surface.

There are at two main types of drag: parasite and induceddrag.

- **Parasitic drag:** caused by moving a solid object through a fluid.
- **Induced drag:** occurs as the result of the creation of lift on a three-dimensional lifting body, such as the wing or fuselage of an airplane. Induced drag consists of two primary components, including drag due to the creation of vortices (vortex drag) and the presence of additional viscous drag (lift-induced viscous drag)

There are 3 types of parasite drag: form drag, interference drag, and skin drag.

Form drag

Form drag is the portion of parasite drag generated by the aircraft due to its shape and airflow around it. Examples include engine cowlings, antennas, and aerodynamic shape of other components.

Interference drag

Interference drag comes from the intersection of air streams that creates eddy currents, turbulence, or restricts smooth airflow. For example, the intersection of the wing and the fuselage at the wing root has significant interference drag. It is also highest when two surfaces meet at perpendicular angles.

Skin friction drag

Skin friction drag is the aerodynamic resistance due to the contact of moving air with the surface of the aircraft. No matter how apparently smooth a surface appears, has a rough, ragged surface when viewed under a microscope. The actual speed at which the air molecules move depends upon the shape of the wing, the stickiness of the air through which the wing or airfoil is moving, and the compressibility.

Profile drag

Profile Drag is the sum of Form drag and Skin Friction drag.

Induced drag

In level flight the aerodynamic properties of a wing or rotor produce a required lift, but at the expense of a certain penalty. Induced drag is the name of the penalty. Induced drag is inherent whenever an airfoil is producing lift, and this type of drag is inseparable from the production of lift. It is always present of lift is produced.

Whenever an airfoil is producing lift, the pressure on the lower surface of it is greater than that on the upper surface (Bernoulli's Principle).

In the area of the wing tips, there is a tendency for these pressures to equalize, resulting in a lateral flow outward from the underside to the upper surface. When viewed from the tail, vortices from the wing tips trail behind the airfoils. This also creates a downwash flow behind the wing's trailing edge. In simple terms, this downwash flow, in a sense, is the induced drag that is created when the plane has produced lift.

Wave drag

Wave drag (also called compressibility drag) is drag which is created by the presence of a body moving at high speed through a compressible fluid.

Wave drag is the result of the formation of shockwaves on the body, formed when areas of local supersonic (Mach number greater than 1.0) flow are created.

Apart from these we also have the following drags:

External store drag: An increase in parasite drag due to external fuel tanks, bombs, rockets, etc., carried as payload by the airplane, but mounted externally from the airframe.

Landing gear drag: An increase in parasite drag when the landing gear is deployed.

Protuberance drag: An increase in parasite drag due to aerodynamic blemishes on the external surface, such as antennas, lights, protruding rivets, and rough or misaligned skin panels.

Leakage drag: An increase in parasite drag due to air leaking into and out of holes and gaps in the surface. Air tends to leak in where the external pressure distribution is highest and to leak out where the external pressure distribution is lowest.

Engine cooling drag: An increase in parasite drag due to airflow through the internal cooling passages for reciprocating engines.

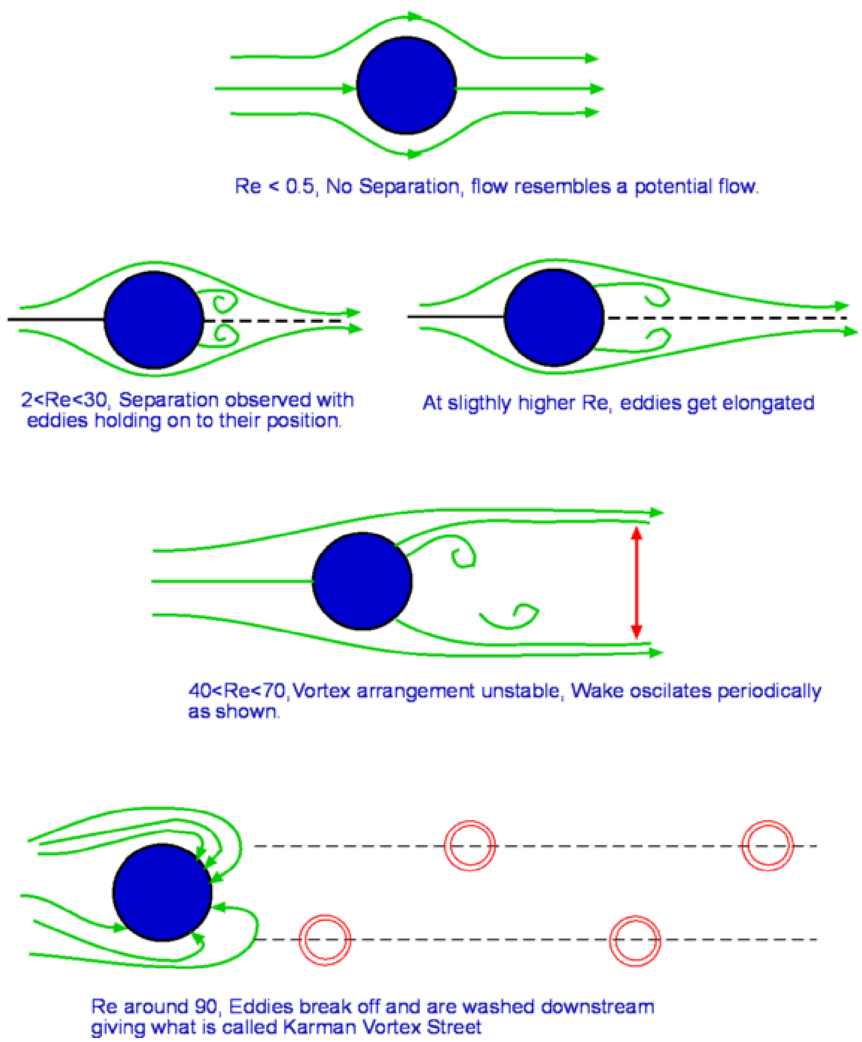
Flap drag: An increase in both parasite drag and induced drag due to the deflection of flaps for high-lift purposes.

Trim drag: The induced drag of the tail caused by the tail lift necessary to balance the pitching moments about the airplane's center of gravity.

EFFECTS OF REYNOLDS NUMBER ON SKIN FRICTION AND PRESSURE DRAG

- The drag force is primarily due to friction drag at low Reynolds numbers ($Re<10$) and to pressure drag at high Reynolds numbers ($Re>5000$).
- Both effects are significant at intermediate Reynolds numbers.

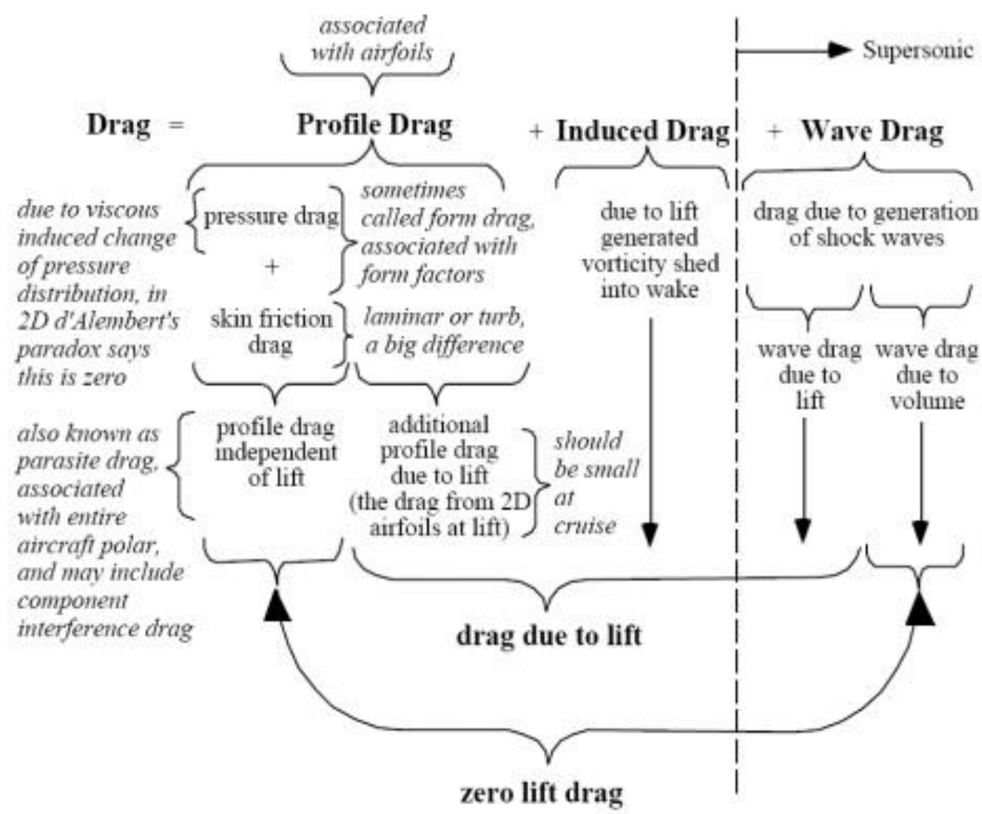
- For blunt bodies such as a circular cylinder or sphere, however, an increase in the surface roughness may increase or decrease the drag coefficient depending on Reynolds number.
- At low Reynolds numbers, most drag is due to friction drag.
- The friction drag is proportional to the surface area.
- The pressure drag is proportional to the frontal area and to the difference between the pressures acting on the front and back of the immersed body.
- The pressure drag is usually dominant for blunt bodies and negligible for streamlined bodies.



Flow past a Circular Cylinder at various Reynolds Numbers

The relative importance of the two kinds of drag is very apparent in the case of flow over a circular cylinder or a sphere. The flow depends strongly upon Reynolds number as is clear from Figure given above When the Reynolds numbers are small (1 and below)the flow behaves like a potential flow. There is no separation. The drag is all due to skin friction. As the Reynolds number is increased this drag decreases. At Reynolds numbers around 2 - 30, there is a separation of theboundary layer, but the wake is of a limited length.The eddies formed seem fixed behind the cylinder. For Reynolds numbers close to 40 -70, there is a periodic oscillation of the wake. For higher Reynolds numbers the eddies break off from the cylinder. As the Reynolds number is increased, the eddies are continuously shed from the cylinder and washed downstream. Two rows of vortices are formed called the Vortex Street. Now the pressure drag contributes to almost 90% of the total drag. The value of C_D reaches a minimum of around 0.9 at a Reynolds number of around 2000.Increasing the Reynolds numbers further results in large angular velocities and a degeneration of vortices into turbulence.

DRAG REDUCTION OF AIRPLANES



Skin friction drag reduction

Two methods are generally considered for skin friction drag reduction. The first one aims at reducing the turbulent skin friction while the second one aims at delaying transition to maintainlarge extent of laminar flow.

Laminar flow control technology

Controlling the laminar flow to reduce the drag

Lift-induced drag reduction

This can be done by increasing the aspect ratio of the wing. Wing aspect ratio is a compromise between aerodynamic and structure characteristics and it is clear that for a given technology, there is not a great possibility to increase aspect ratios. The alternative is to develop wing tip devices acting on the tip vortex which is at the origin of the lift-induced drag.

Wave drag reduction

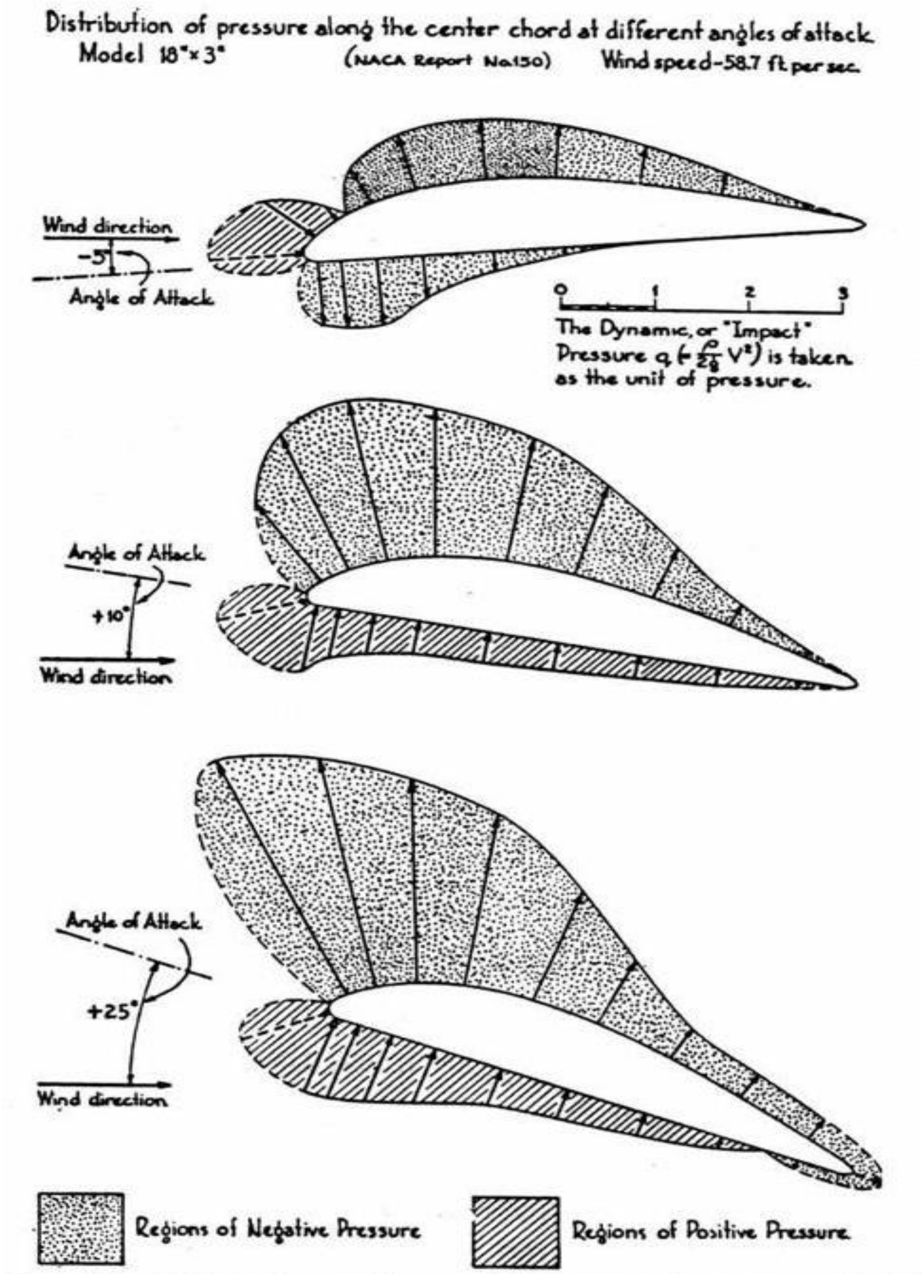
Wave drag reduction includes wing sweep, area ruling and reduced thickness as well as wing twist

Various other methods include

- The turbulent skin friction drag reduction by the use of riblets
- The hybrid laminar flow technology
- The innovative wing-tip devices
- The shock control and trailing edge devices which allow to adapt the wing geometry to flight conditions

CHORDWISE AND SPANWISE PRESSURE DISTRIBUTION

The increment in aerodynamic efficiency of wings in ground effect is accredited to bothspanwise and chordwise contributing factors. The chordwise dominated influence produces an increment in the lifting performance, whereby the interaction with a flat ground or free watersurface reduces the lower surface velocity distribution of the airfoil, resulting in a high staticpressure region.



ASPECT RATIO, CAMBER AND PLAN FORM CHARACTERISTICS

Aspect ratio: Aspect ratio is the relationship between the length and width of a wing. It is one of the primary factors in determining lift/drag characteristics. At a given angle of attack, a higher aspect ratio produces less drag for the same amount of lift. Thus an aspect ratio formula is:-

Aspect Ratio = Total length of the wing / Average width of the wing

Camber: It affects the difference in the velocity of the airflow between the upper and lower surfaces of the wing. If the upper camber increases and the lower camber remains the same, the velocity differential increases. There is, of course, a limit to the amount of camber which can be used. After a certain point, air will no longer flow smoothly over the airfoil. Once this happens, the lifting capacity diminishes. The ideal camber varies with the airplane's performance specification, especially the speed range and the load-carrying requirements.

Wing plan form: This refers to the shape of the airplane wing when viewed from above or below. Each plan form design has its advantages and disadvantages.

DRAG POLAR

For every aerodynamic body, there is a relation between CD and CL that can be expressed as an equation or plotted on a graph. Both the equation and the graph arecalled the drag polar.

(Totaldrag)=(parasitedrag)+(wavedrag)+(induceddrag)

$$C_D = C_{D,e} + C_{D,w} + \frac{C_L^2}{\pi e \mathrm{AR}}$$

The parasite drag coefficient CD,e can be treated as the sum of its value at zero liftCD,e,o and the increment in parasite drag ΔCD,edue to lift. The skin-friction drag (to a lesserextent) and the pressure drag due to flow separation (to a greater extent) changewhen α changes; the sum of these changes createsΔCD,e

$$C_{D,e} = C_{D,e,0} + \Delta C_{D,e} = C_{D,e,0} + k_1 C_L^2$$

Foraflatplateatangleofattack,

$$c_{d,w} = \frac{4\alpha^2}{\sqrt{M_\infty^2 - 1}} = \frac{4}{\sqrt{M_\infty^2 - 1}} \left(\frac{c_l \sqrt{M_\infty^2 - 1}}{4} \right)^2$$
$$= \frac{c_l^2 \sqrt{M_\infty^2 - 1}}{4}$$

Since CD,w is simply the wave drag coefficient due to lift, and since equation shows that CD,w varies asC^2LHence,

$$C_{D,w} = C_{D,w,0} + \Delta C_{D,w} = C_{D,w,0} + k_2 C_L^2$$

Also

$$C_D = C_{D,e,0} + C_{D,w,0} + k_1 C_L^2 + k_2 C_L^2 + \frac{C_L^2}{\pi e \mathrm{AR}}$$

Assume k_3 to be a constant such that

$$k_3 \equiv 1/(\pi e \mathrm{AR})$$

So that the equation becomes

$$C_D = C_{D,e,0} + C_{D,w,0} + (k_1 + k_2 + k_3)C_L^2$$

Sum of the first two terms is equal to the zero lift drag coefficientC_{D0}

$$C_{D,e,0} + C_{D,w,0} \equiv C_{D,0}$$

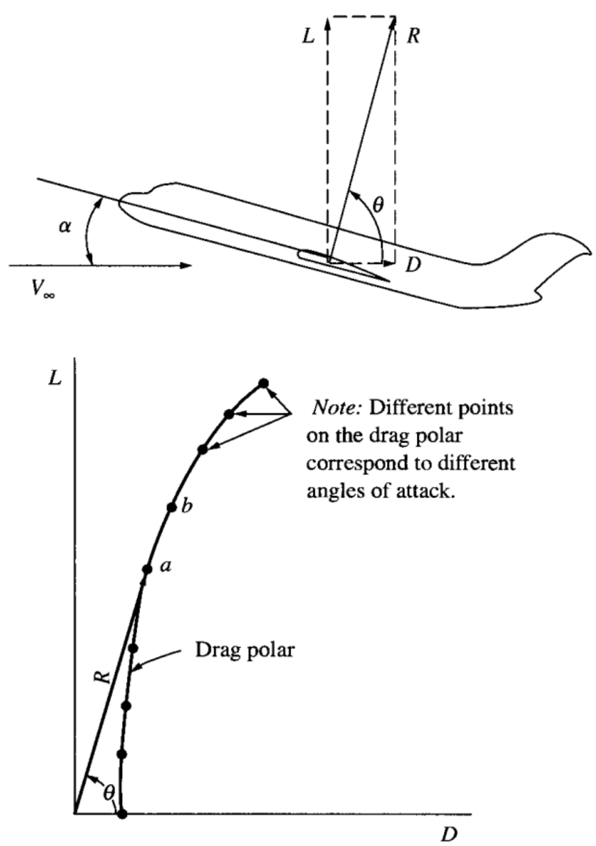
$$k_1 + k_2 + k_3 \equiv K$$

On substituting the above equations we have for the complete airplane

$$\boxed{C_D = C_{D,0} + KC_L^2}$$

This is called the drag polar equation

Construction for the resultant aerodynamic force on a drag polar



UNIT II

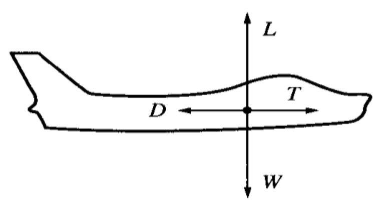
STEADY FLIGHT

STEADY LEVEL FLIGHT

Equations ofmotion forsteady,levelflight

Consider the figure, whichshowsanairplanewithahorizontalflightpath.Thisairplane isinlevelflight; thatis,theclimbangleθandrollangleφ arezero.Moreover,by definition, steadyflightisflightwithnoacceleration.

Thegoverningequations ofmotionforsteady,levelflightareobtained by setting θ,φ, dv_∞/dt and v^2_∞/r_1 equal to zero



Force diagram for steady, level flight.

Thenormalaccelerationiszero by definitionofsteadyflight,i.e.,noacceleration;thisisalsoconsistentwiththeflight pathbeingastraightline,wheretheradiusofcurvatureris infinitelylarge.

$$0 = T\cos\epsilon - D$$

$$0 = L + T\sin\epsilon - W$$

AlthoughtheenginethrustlineisinclinedatangleEtothefree-streamdirection, thisangleisusuallysmall forconventionalairplanesandcanbeneglected.Hence, forthischapterweassumethatthethrust isaligned withtheflightdirection,thatis, ϵ=0.

$$\boxed{T = D}$$

$$\boxed{L = W}$$

Inthesimpleforcebalanceshownin the above figure liftequalsweightandthrustequalsdrag.

THRUST REQUIRED WITH ALTITUDE

Imaginethisairplaneinsteady,levelflightatagivenvelocityandaltitude. Tomaintainthisspeedandaltitude,enough

Thrustmustbegeneratedtoexactlyovercomethedragandtokeeptheairplanegoing-thisisthethrustrequiredtomaintaintheseflightconditions.ThethrustrequiredTRdependson thevelocity,thealtitude,andtheaerodynamicshape,size,andweightof theairplane.

Thethrustrequired issimplyequaltothedrag oftheairplane-itisthethrustrequiredtoovercomethe aerodynamicdrag.

1. GraphicalApproach

Consideragivenairplaneflyingatagivenaltitudeinsteady,levelflight.Forthegivenairplane,weknowthefollowingphysicalcharacteristics: weightW,aspectratioAR,andwingplan formareaS.

We know that $C_D=C_{D,0}+KC_L^2$

where$C_D$andKareknownforthegivenairplane.Tocalculatethethrustrequired curve,proceed as follows:

1.ChooseavalueofV∞.

2.Forthechosen V∞, calculateCLfromtherelation

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

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

3. CalculateC_D

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

4. Calculatedrag,henceT_R,

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

ThisisthevalueofTRcorrespondingtothevelocity choseninstep1. This combination(TR,V∞) isonepointonthethrustrequiredcurve.

5. Repeatsteps1to4foralargenumberofdifferentvaluesofV_∞,thusgenerating enoughpointstoplotthethrustrequiredcurve.

2. AnalyticalApproach

Thethrustrequiredcurvefromananalyticalpointof view is examined here.

Forsteady,levelflightwehave

$$T_R = D = \frac{D}{W} W = \frac{D}{L} W$$

$$\boxed{T_R = \frac{W}{L/D}}$$

The lift to drag ratio can be written as

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

From the drag polar equation, we have

$$D = q_\infty S C_D = q_\infty S (C_{D,0} + K C_L^2)$$

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

From which

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

Substituting in the drag polar equation, we have

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

$$\boxed{D = \frac{1}{2}\rho_\infty V_\infty^2 S C_{D,0} + \frac{2KS}{\rho_\infty V_\infty^2} \left(\frac{W}{S} \right)^2}$$

Now replacing the value of q_∞ as

$$q_\infty = \tfrac{1}{2}\rho_\infty V_\infty^2$$

We know that $D = T_R$

$$T_R = q_\infty S C_{D,0} + \frac{KS}{q_\infty} \left(\frac{W}{S} \right)^2$$

Now multiplying by q_∞ and rearranging

$$q_\infty^2 S C_{D,0} - q_\infty T_R + KS \left(\frac{W}{S} \right)^2 = 0$$

Obtaining the value of q_∞ from the above equation

$$\begin{aligned} q_\infty &= \frac{T_R \pm \sqrt{T_R^2 - 4SC_{D,0}K(W/S)^2}}{2SC_{D,0}} \\ &= \frac{T_R/S \pm \sqrt{(T_R/S)^2 - 4C_{D,0}K(W/S)^2}}{2C_{D,0}} \end{aligned}$$

Replacing q_∞ with

$$q_\infty = \tfrac{1}{2}\rho_\infty V_\infty^2$$

Now the value of V_∞ can be obtained as

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

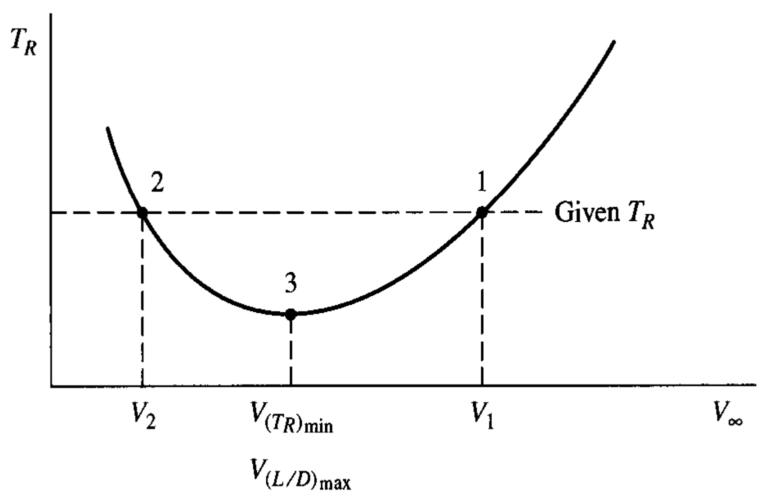
We know that

$$\frac{T_R}{S} = \frac{T_R}{W}\frac{W}{S}$$

On substituting the above value and taking the square root we get the value of V_∞ as

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

Plot of Thrust required (T_R) and Velocity(V) gives the following graph



THRUST REQUIRED

The thrust available, denoted by TA is the thrust provided by the power plant of the airplane.

Propeller-Driven Aircraft

An aerodynamic force is generated on a propeller that is translating and rotating through the air. The component of this force in the forward direction is the thrust of the propeller. For propeller/reciprocating engine combination, this propeller thrust is the thrust available TA.

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

Where P ηPR is the propeller efficiency and P is the shaft power from the piston engine. Since power is given by force times velocity thepower available from any flight propulsion device is

$$P_A = T_A V_\infty$$

Combining the equations and solvingforT_A, we get

$$TA=\eta_{pr}P/V_\infty$$

Similarly for turboprop, the power available is given by

$$P_A = \eta P_r P_{es}$$

Thus,

$$T_A = \eta P_r P_{es}/V_\infty$$

Hence,forthegivenpowerratings,theshaftpower*P*forapistonengineandtheequivalentshaftpower*Pes*foraturboprop. The above equations provide the thrust available for each type of power plant.

Jet propelled aircraft

Turbojet and turbofan engines are rated in terms of thrust. Hence, for such power plants, TA is the quantity for the analysis of airplane performance.

Forturbojetengine,at subsonicspeeds,

$$T_A \approx \text{constant with } V_\infty$$

Andforsupersonic speeds,

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

Theeffectofaltitudeon$T_A$isgivenbythe equation

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

Where$(T_A)o$isthethrustavailableatsealevelandρoisthestandardsea-leveldensity.

Unliketheturbojet,thethrustofaturbofanisafunctionofvelocity.Forthehigh-bypass-ratioturbofanscommonly usedforciviltransports,thrustdecreaseswithincreasingvelocity.(Thisisanalogousto thethrustdecreasewithvelocityforpropellers,which makessensebecausethelargefanonahigh-bypass-ratioturbofanisfunctioning muchasapropeller.)

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

Where$(T_A)_{V=0}$isthestaticthrustavailable(thrustatzerovelocity)atstandardsea level,and*A*and*n*arefunctionsof altitude,obtained bycorrelating thedatafora givenengine.Ontheotherhand,foralow-bypass-ratioturbofan,thethrustvariation withvelocityismuchclosertothatofaturbojet,essentiallyconstantatsubsonic speedsandincreasingwithvelocityatsupersonicspeeds.

Thealtitudevariationofthrustforahigh-bypass-ratiociviIturbofaniscorrelated inequation given

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

Where$(T_A)_0$isthethrustavailableatsealevelandρ$_0$isstandardsea-leveldensity.

Foraperformanceanalysisofaturbofan-poweredairplane,thethrustavailable shouldbeobtainedfromtheenginecharacteristicsprovidedbythemanufacturer.

POWER REQUIRED

ConsideraforceFactingonan objectmovingwithvelocityV. BothFandVarevectorsandmayhavedifferentdirections. Atsomeinstant,theobjectislocatedatapositiongivenbythepositionvectorr.Overatimeincrement*dt*,theobjectisdis placedthroughthevectordr;

TheworkdoneontheobjectbytheforceFactingthroughthedisplacementdrisF. dr.Poweristhetimerateofdoing work,or

$$\text{Power} = \frac{d}{dt}(\mathbf{F} \cdot \mathbf{dr}) = \mathbf{F} \cdot \frac{\mathbf{dr}}{dt}$$

Since,

$$\frac{\mathbf{dr}}{dt} = \mathbf{V}$$

Then

$$\text{Power} = \mathbf{F} \cdot \mathbf{V}$$

Consideranairplaneinstraight andlevelflight. ThevelocityoftheairplaneisV$_\infty$.Theconceptof thrustrequired$T_R$wasintroduced,whereT_R= D. Inthissection,weintroducethe analogousconceptofpowerrequired,denotedbyP_R.SincebothTand V$_\infty$ arehorizontal, thedotproductgivesforthepowerrequired

$$\boxed{P_R = T_R V_\infty}$$

Now,

$$P_R = T_R V_\infty = \frac{W}{C_L/C_D} V_\infty$$

Since L=W for steady level flight

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

Solving the above equation gives

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

$$P_R = \frac{W}{C_L/C_D}\sqrt{\frac{2W}{\rho_\infty S C_L}}$$

or

$$\boxed{P_R = \sqrt{\frac{2W^3 C_D^2}{\rho_\infty S C_L^3}}}$$

Minimum power required

Examining the equation we can obtain

$$\boxed{P_R \propto \frac{C_L^{3/2}}{C_D}}$$

Hence minimum power occurs when the airplane is flying such that the the above value is maximum

Thus it can be written that

$$\left(\frac{C_L^{3/2}}{C_D}\right)_{\max} = \frac{1}{4}\left(\frac{3}{K C_{D,0}^{1/3}}\right)^{3/4}$$

Zero lift drag equals one third of the drag due to lift

The velocity at which power required is minimum is given by

$$V_{(C_L^{3/2}/C_D)_{\max}} = \left(\frac{2}{\rho_\infty}\sqrt{\frac{K}{3C_{D,0}}}\frac{W}{S}\right)^{1/2}$$

This is less than that for minimum thrust required

$$V_{(C_L^{3/2}/C_D)_{\max}} = 0.76 V_{(L/D)_{\max}}$$

POWER AVAILABLE

Power available is the power provided by the powerplant of the airplane
It is given by,

$$P_A = T_A V_\infty$$

Propeller driven aircraft

They are driven by reciprocating piston engines or gas turbine engines

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

where η_{pr} is the propeller efficiency and P is the shaft power from the reciprocating engine.

The velocity and altitude effects are as follows

1. power is reasonably constant with

2. for an unsupercharged engine

$$\boxed{\frac{P}{P_0} = \frac{\rho}{\rho_0}}$$

Where P and are the shaft per output and density, respectively, at altitude and P_o and $\rho_o$are the corresponding values at sea level. Considering the temperature effects

$$\boxed{\frac{P}{P_0} = 1.132\frac{\rho}{\rho_0} - 0.132}$$

For a supercharged engine, P is essentially constant up to the critical design altitude of the supercharger. Above this critical altitude, P decreases according to the above equation.

Thus we have,

$$\boxed{P_A = \eta_{pr} P_{es}}$$

WhereP_{es} is the equivalent shaft power.

Thus it can be written that

$$\boxed{\frac{P_A}{P_{A,0}} = \left(\frac{\rho}{\rho_0}\right)^n \qquad n = 0.7}$$

Turbojet and turbofan engines

Turbofan and turbojet are rated in terms of thust. Thus it can be written as

$$\boxed{P_A = T_A V_\infty}$$

For a turbojet engine,

At subsonic speed T_A is essentially constant. So that,

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

The effect of altitude on T_A is given by,

$$\boxed{\frac{P_A}{(P_A)_0} = \frac{\rho}{\rho_0}}$$

The mach number variation of thrust is given by

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

The altitude variation of turbofan thrust is given by

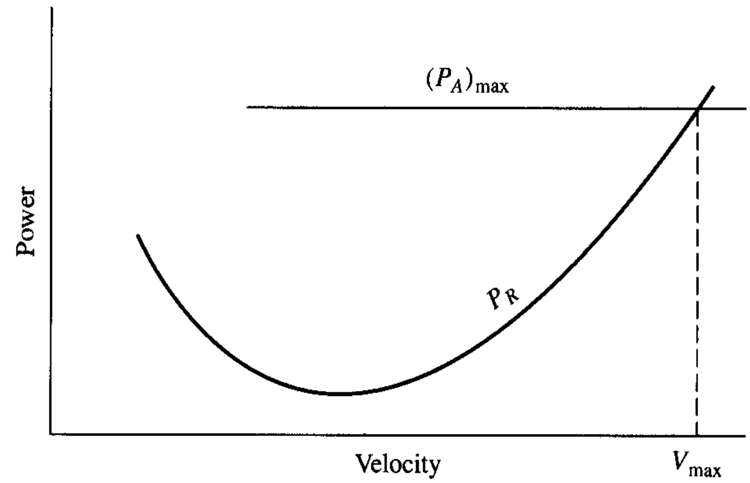
$$\frac{T_A}{(T_A)_0} = \left[\frac{\rho}{\rho_0}\right]^m$$

Hence the variation of P_A with altitude is also the same

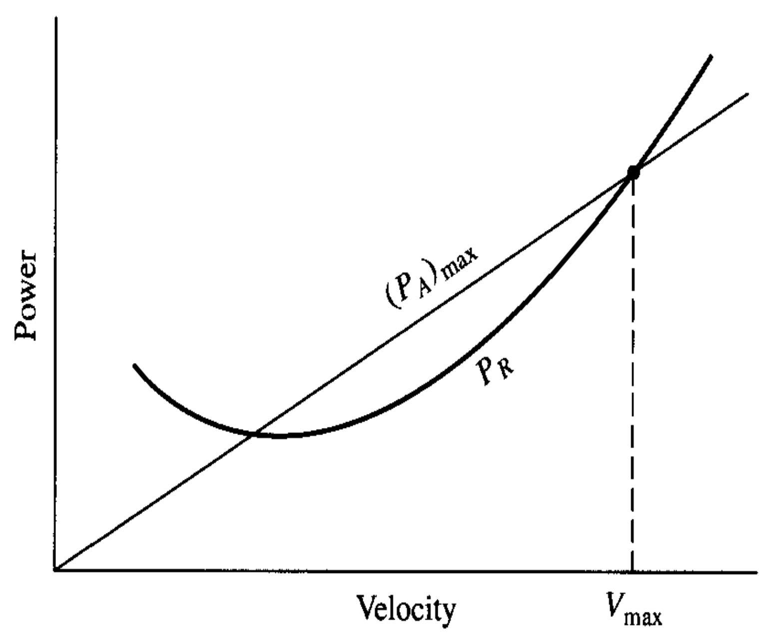
$$\boxed{\frac{P_A}{(P_A)_0} = \left[\frac{\rho}{\rho_0}\right]^m}$$

ESTIMATION OF MAXIMUM LEVEL FLIGHT SPEED

Consider a propeller driven aircraft. The power available P_A is essentially constant with velocity. The intersection of the maximum power available curve and the power required curve defines the maximum velocity for straight and level flight



For a Propeller driven aircraft



For a Turbojet powered airplane

UNIT III

GLIDING, CLIMBING AND TURNING PERFORMANCE

MAXIMUM RANGE

By definition, *range* is the total distance (measured with respect to the ground) traversed by an airplane on one load of fuel. We denote the range by R. We also consider the following weights:

W_0—gross weight of the airplane including *everything*; full fuel load, payload, crew, structure, etc.

W_f—weight of fuel; this is an instantaneous value, and it changes as fuel is consumed during flight.

W_1—weight of the airplane when the fuel tanks are empty.

At any instant during the flight, the weight of the airplane W is

$$W = W_1 + W_f \qquad \text{Eq.(1)}$$

Since W_f is decreasing during the flight, W is also decreasing. Indeed, the time rate of change of weight is, from Eq. (1)

$$\frac{dW}{dt} = \frac{dW_f}{dt} = \dot{W}_f \qquad \text{Eq.(2)}$$

where both dW/dt and $\dot{W}_f$ are negative numbers because fuel is being consumed, and hence both W and W_f are decreasing.

For a propeller-driven/reciprocating engine, the specific fuel consumption is defined by the following equation

$$c \equiv -\frac{\dot{W}_f}{P}$$

where P is the shaft power and the minus sign is necessary because $\dot{W}_f$ is negative and c is always treated as a positive quantity. For a jet-propelled airplane, the thrust specific fuel consumption is defined by Eq.

$$c_t \equiv -\frac{\dot{W}_f}{T}$$

where T is the thrust available.

Now

$$c_t = \frac{cV_\infty}{\eta_{pr}}$$

where η_{pr} is the propeller efficiency.

A general relation for the calculation of range can be obtained as follows. Consider an airplane in steady, level flight. Let s denote horizontal distance covered over the ground. Assuming a stationary atmosphere (no wind), the airplane's velocity V_∞ is

$$V_\infty = \frac{ds}{dt}$$

or

$$ds = V_\infty \, dt$$

$$c_t = -\frac{dW_f/dt}{T}$$

or

$$dt = -\frac{dW_f}{c_t T} \qquad \text{Eq.(3)}$$

Substitute Eq.(3) in ds

$$ds = -\frac{V_\infty}{c_t T} dW_f \qquad \text{Eq.(4)}$$

Since $dW_f = dW$. Equation (4) then becomes

$$ds = -\frac{V_\infty}{c_t T} dW = -\frac{V_\infty}{c_t} \frac{W}{T} \frac{dW}{W}$$

In steady, level flight, $L = W$ and $T = D$.

$$ds = -\frac{V_\infty}{c_t} \frac{L}{D} \frac{dW}{W} \qquad \text{Eq. (5)}$$

The range of the airplane is obtained by integrating Eq. (5) between $s = 0$, where the fuel tanks are full and hence $W = W_0$, and $s = R$, where the fuel tanks are empty and hence $W = W_1$.

$$R = \int_0^R ds = -\int_{W_0}^{W_1} \frac{V_\infty}{c_t} \frac{L}{D} \frac{dW}{W}$$

$$\boxed{R = \int_{W_1}^{W_0} \frac{V_\infty}{c_t} \frac{L}{D} \frac{dW}{W}}$$

Eq.(6)

For a preliminary performance analysis, Eq. (6) is usually simplified. If we assume flight at constant V_∞, c_t, and L/D, Eq. (6) becomes

$$R = \frac{V_\infty}{c_t} \frac{L}{D} \int_{W_1}^{W_0} \frac{dW}{W}$$

or

$$\boxed{R = \frac{V_\infty}{c_t} \frac{L}{D} \ln \frac{W_0}{W_1}} \qquad \text{Eq. (7)}$$

Equation (7) is frequently called the *Breguet range equation*.

Range for Propeller- driven aircraft

Eq. (7) can be written as

$$R = \frac{V_\infty}{c_t} \frac{L}{D} \ln \frac{W_0}{W_1} = \frac{\eta_{pr}}{cV_\infty} V_\infty \frac{L}{D} \ln \frac{W_0}{W_1}$$

or

$$\boxed{R = \frac{\eta_{pr}}{c} \frac{L}{D} \ln \frac{W_0}{W_1}}$$

For maximum range

1. Fly at maximum L/D.
2. Have the highest possible propeller efficiency.
3. Have the lowest possible specific fuel consumption.
4. Have the highest possible ratio of gross weight to empty weight (i.e., carry a lot of fuel).

For Jet-Propelled aircraft

Consider steady level flight

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

or

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

Thus,

$$V_\infty \frac{L}{D} = \sqrt{\frac{2W}{\rho_\infty S C_L}} \frac{C_L}{C_D} = \sqrt{\frac{2W}{\rho_\infty S}} \frac{C_L^{1/2}}{C_D}$$

Thus the product $V_\infty(L/D)$ is maximum when the airplane is flying at a maximum value of $C_L^{1/2}/C_D$.

$$R = \int_{W_1}^{W_0} \frac{1}{c_t}\sqrt{\frac{2W}{\rho_\infty S}} \frac{C_L^{1/2}}{C_D} \frac{dW}{W} \qquad \text{Eq. (8)}$$

Assuming c_t, ρ_∞, S, and $C_L^{1/2}/C_D$ are constant, Eq. (8) can be written as

$$R = \frac{1}{c_t}\sqrt{\frac{2}{\rho_\infty S}} \frac{C_L^{1/2}}{C_D} \int_{W_1}^{W_0} \frac{dW}{W^{1/2}}$$

or

$$\boxed{R = \frac{2}{c_t}\sqrt{\frac{2}{\rho_\infty S}} \frac{C_L^{1/2}}{C_D}\left(W_0^{1/2} - W_1^{1/2}\right)} \qquad \text{Eq.(9)}$$

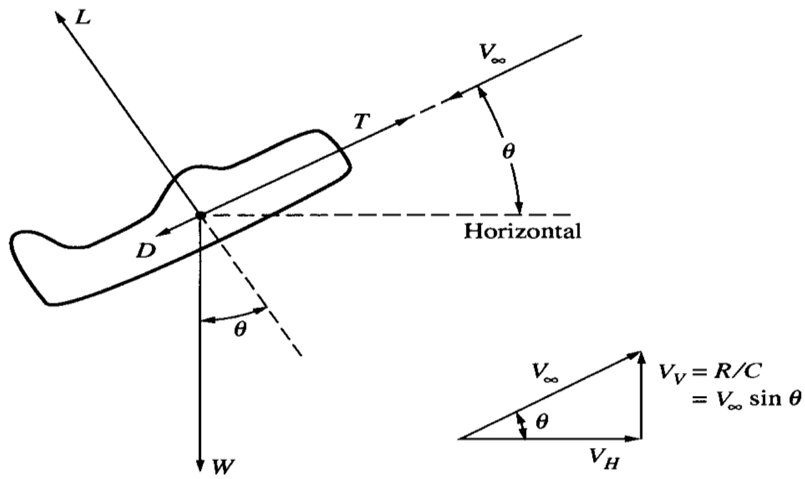
Equation (9) is a simplified range equation for a jet-propelled airplane. From this equation, the flight conditions for maximum range for a jet-propelled airplane are

1. Fly at maximum $C_L^{1/2}/C_D$.
2. Have the lowest possible thrust specific fuel consumption.
3. Fly at high altitude, where ρ_∞ is small.
4. Carry a lot of fuel.

RATE OF CLIMB

Consider a steady unaccelerated climb. The equations of motion for this condition is given by

$$T\cos\epsilon - D - W\sin\theta = 0$$
$$L + T\sin\epsilon - W\cos\theta = 0$$



Force and velocity diagrams for climbing flight.

Here the vertical component gives the rate of climb.

the rate of climb by R/C. From this diagram,

$$\boxed{R/C = V_\infty \sin\theta}$$

On multiplying the equation by V_∞/W we have

$$\boxed{V_\infty \sin\theta = R/C = \frac{TV_\infty - DV_\infty}{W}}$$

Thus power available is the power required to overcome this drag

$$\boxed{TV_\infty - DV_\infty \equiv \text{excess power}}$$

Hence,

$$\boxed{R/C = \frac{\text{excess power}}{W}}$$

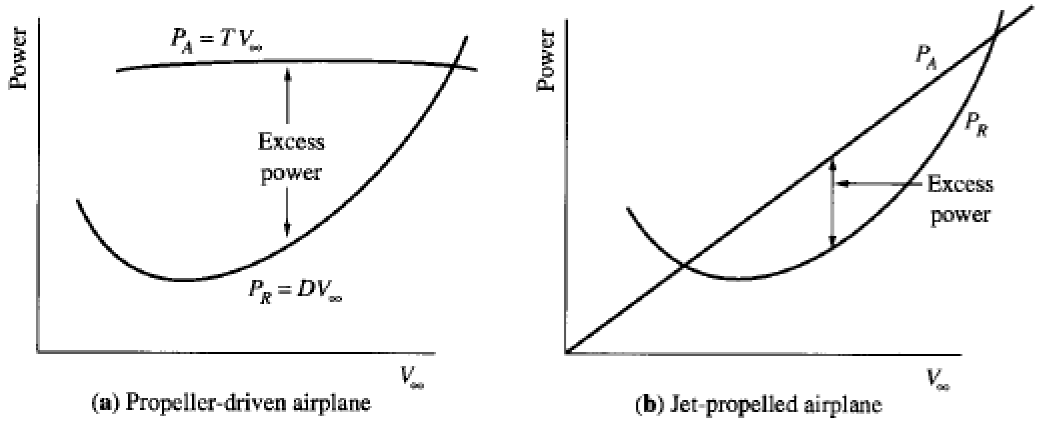
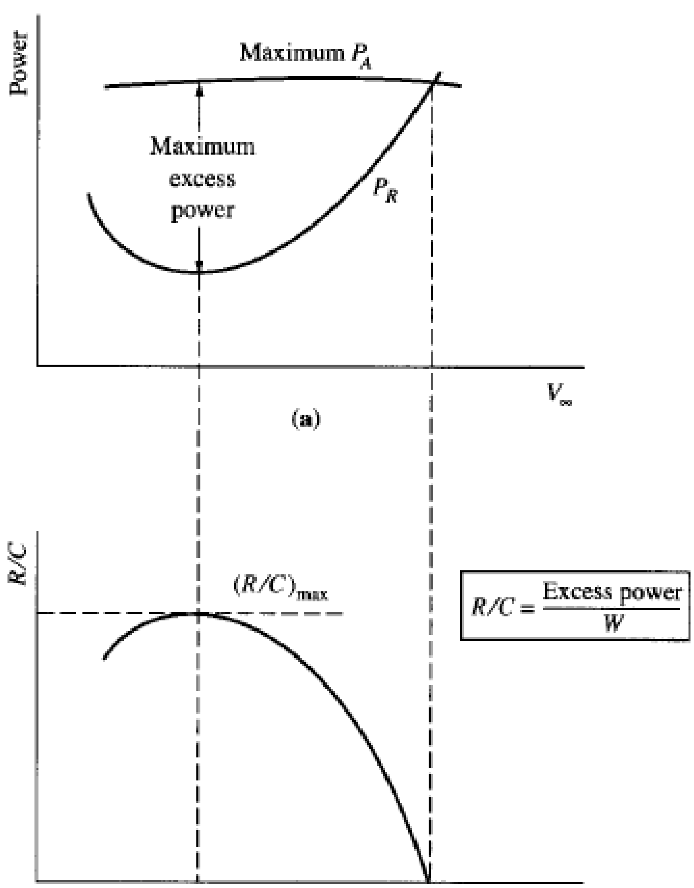


Illustration of excess power for (**a**) propellor-driven airplane and (**b**) jet-propelled airplane.



Variation of rate of climb with velocity at a given altitude.

TIME TO CLIMB

The rate of climb, by definition, is the vertical component of the airplane's velocity, which is simply the time rate of change of altitude dh/dt. Hence,

$$\frac{dh}{dt} = R/C$$

or

$$dt = \frac{dh}{R/C} \qquad \text{Eq.(1)}$$

In Eq. (1) , R/C is a function of altitude, and dt is the small increment in time required to climb the small height dh at a given instantaneous altitude. The time to climb from one altitude h_1 to another h_2 is obtained by integrating Eq. (1). between the two altitudes:

$$t = \int_{h_1}^{h_2} \frac{dh}{R/C} \qquad \text{Eq.(2)}$$

Normally, the performance characteristic labeled *time to climb* is considered from sea level, where $h_1 = 0$. Hence, the time to climb from sea level to any given altitude h_2 is, from Eq. (2)

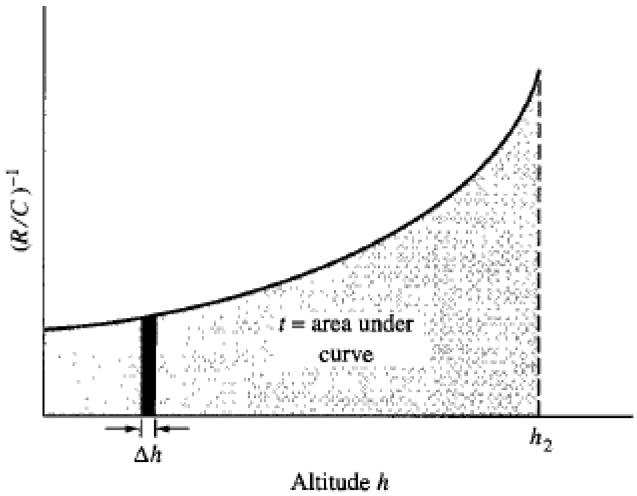
$$\boxed{t = \int_{0}^{h_2} \frac{dh}{R/C}} \qquad \text{Eq. (3)}$$

If in Eq. (3) the maximum rate of climb is used at each altitude, then t becomes the minimum time to climb to altitude h_2.

$$\boxed{t_{\min} = \int_{0}^{h_2} \frac{dh}{(R/C)_{\max}}}$$

Graphical approach

Consider a plot of $(R/C)^{-1}$ versus altitude, as shown in figure. The time to climb to altitude h_2 is simply the area under the curve, shown by the shaded area in figure.

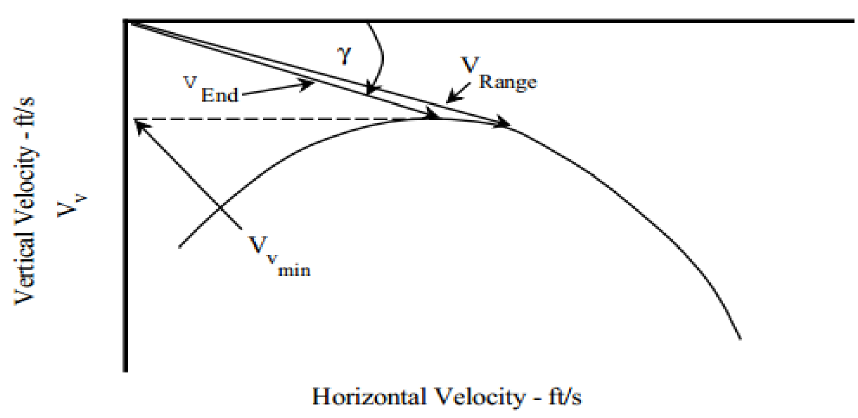


Graphical representation of the time to climb to altitude h_2.

GLIDE HODOGRAPH

A glide hodograph is obtained when horizontal velocity (V_h) is plotted on the x-axis and the rate of sink (V_d) is plotted on the y-axis. Such a diagram gives complete information about glide performance at an altitude especially, γ_{min}, $V_{\gamma\, min}$, $(R/S)_{min}$, $V_{(R/S)min}$ and $\gamma_{(R/S)min}$.

On a hodograph, the radius vector from the origin to any part on the plot has a length proportional to the flight path speed and makes an angle to the horizontal equal to theactual descent angle (γ).



Glide hodograph

1. The vertical axis is: dh/dt = V_v = rate of descent (ROD) = $V_T \sin \gamma$.

2. The horizontal axis is $V_{hor} = V_T \cos \gamma$.

BANK ANGLE AND LOAD FACTOR

We have the equation,

$$\boxed{\omega = \frac{g\sqrt{n^2 - 1}}{V_\infty}}$$

Where n is the load factor.

It can be noted that as the airplane's bankangle (ϕ) is increased, the magnitude of the lift must increase. As L increases, the drag due to lift increases. Hence, to maintain a sustained level turn at a given velocity and at a given bank angle ϕ, the thrust must be increased from its straight and level flight value to compensate for the increase in drag. If this increase in thrust pushes the required thrust beyond the maximum thrust available from the power plant, then the level turn cannot be sustained at the given velocity and bank angle. In this case, to maintain a turn at the given V_∞, ϕ will have to be decreased in order to decrease the drag sufficiently that the thrust required does not exceed the thrust available.

Since the load factor is a function of ϕ,

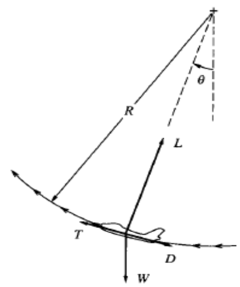
It can be written that

$$n = \frac{1}{\cos \phi}$$

Where ϕ is the bank angle.

PULL UP AND PUSH OVER

Consideranairplaneinitially instraightandlevelflight,whereL=W.Thepilotsuddenlypitchestheairplanetoahigherangleofattacksuchthatthe liftsuddenlyincreases. BecauseL >W,theairplanewillarch upward,assketchedinFigureTheflightpathbecomescurvedin theverticalplane,withaturn radiusRandturn ratedθ/dt.Thisiscalledthepull-upmaneuver.



The equation can be written as

$$\boxed{m\frac{V_\infty^2}{R} = L - W\cos\theta}$$

Since θ is small, we get

$$m\frac{V_\infty^2}{r_2} = L\sin\phi$$

This is the governing equation of motion for a level turn.

$$\cos\phi = \frac{W}{L} = \frac{1}{L/W} \qquad \text{Eq. (1)}$$

In Eq. (1) , the ratio L/W is an important parameter in turning performance; it is defined as the *load factor* n, where

$$\boxed{n \equiv \frac{L}{W}} \qquad \text{Eq. (2)}$$

Hence, Eq. (1) can be written as

$$\boxed{\phi = \text{Arccos}\,\frac{1}{n}} \qquad \text{Eq. (3)}$$

The roll angle ϕ depends *only* on the load factor; if you know the load factor, then you know ϕ, and vice versa.

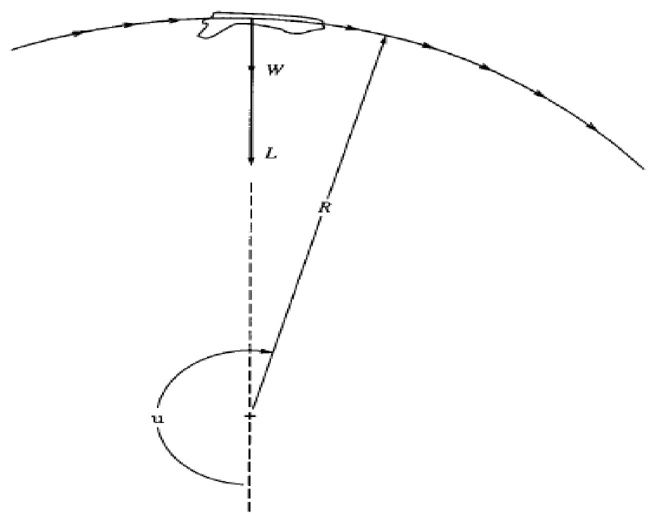
By solving the above equations the turn radius can be expressed as

$$\boxed{R = \frac{V_\infty^2}{g\sqrt{n^2 - 1}}} \qquad \text{Eq. (4)}$$

From Eq. (4) , the turn radius depends only on V_∞ and n. To obtain the smallest possible R, we want

1. The highest possible load factor (i.e., the highest possible L/W).
2. The lowest possible velocity.

PULL DOWN MANEUVER



The pulldown maneuver.

From the above given equation it can be written as

$$R = \frac{mV_\infty^2}{L + W} = \frac{W}{g}\frac{V_\infty^2}{L + W} = \frac{V_\infty^2}{g(L/W + 1)} \qquad \text{Eq. (1)}$$

Since $n = L/W$, Eq. (1) becomes

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

and $\omega = V_\infty / R$ becomes

$$\boxed{\omega = \frac{g(n + 1)}{V_\infty}}$$

THE V-n DIAGRAM

Therearestructurallimitations onthemaximumloadfactorallowedforagiven airplane.Thesestructurallimitationswerenotconsideredinthepreviоussections; letusexamine themnow.

Therearetwo categoriesofstructurallimitationsinairplanedesign:

I. **Limitloadfactor.**Thisistheboundaryassociatedwithpermanentstructuraldeformation ofoneormorepartsof theairplane.Ifnislessthanthelimitload factor,thestructuremaydeflectduring amaneuver,butitwillreturntoits originalstatewhenn = l.Ifnisgreater thanthelimitloadfactor,thentheairplanestructurewillexperienceapermanent deformation,thatis,itwill incur**structuraldamage**.

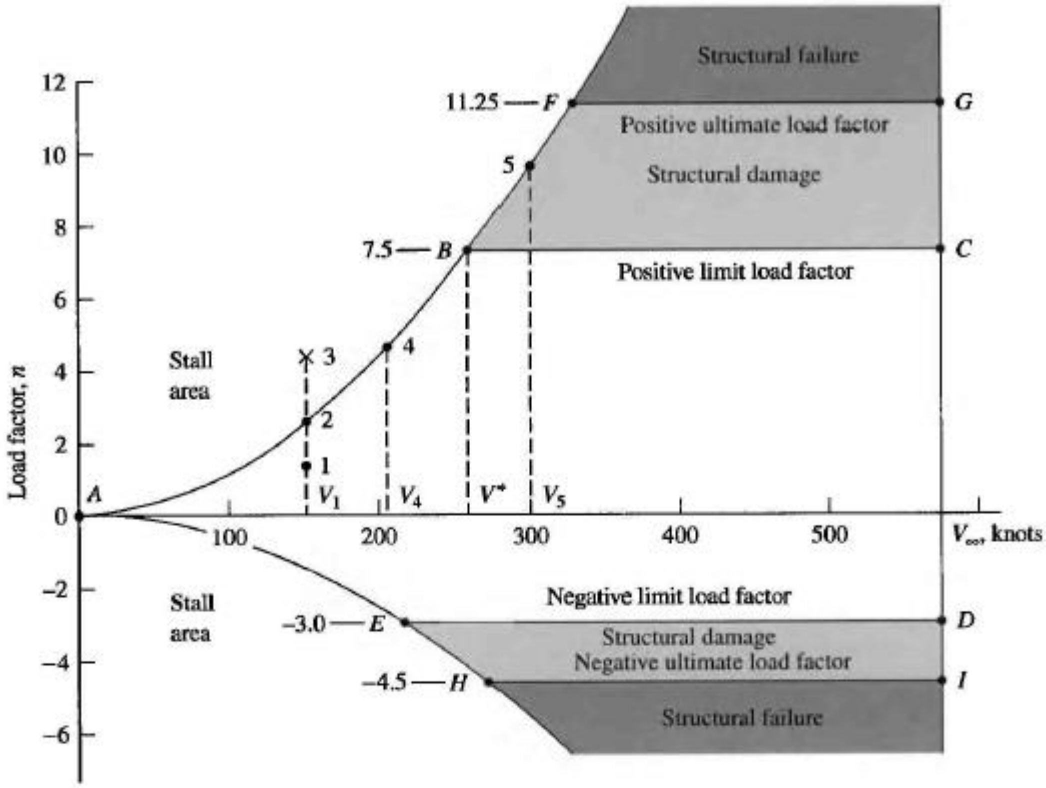
2. **Ultimateloadfactor**.Thisistheboundaryassociatedwithoutrightstructural failure. Ifnisgreater thantheultimateloadfactor,partsoftheairplane will break.

Boththe aerodynamicandstructurallimitationsforagivenairplaneareillustratedintheV-ndiagram,aplot ofloadfactorversusflightvelocity,asgiveninFigure. AV-ndiagramisatypeof"flightenvelope"foragivenairplane;itestablishesthemaneuver boundaries.

The curvebetween pointsAandB in Figurerepresentsthe aerodynamiclimiton loadfactorimposed by$(C_d)_{max}$.The regionabovecurveABintheV-ndiagram isthestallregion.Tounderstand the significance ofcurveABbetter,consideranairplaneflyingatvelocity V_1,whereV_1isshownin Figure.AssumetheairplaneisatanangleofattacksuchthatCL<$(C_d)_{max}$·This flightconditionisrepresented bypoint1 inthe figure.

Nowassume theangleof attack isincreasedtothatfor$(C_d)_{max}$,keeping thevelocityconstant atV_i.TheliftincreasestoitsmaximumvalueforthegivenV_1, andhencethelocalfactorn=L_lWreachesitsmaximumvalueforthegiven*Vi*.Thecorrespondingflightcon ditionisgivenbypoint2inFigure.Iftheangleofattackisincreasedfurther,thewingstallsandtheloadfactord ecreases.Therefore, point3 inFigureisunobtainable inflight.

Point3isinthestallregionof theV-ndiagram.



The V–n diagram for a typical jet trainer aircraft. Free-stream velocity V_∞ is given in knots. 1 knot (kn) = 1.15 mi/h.

At the given velocity Vi. As VOO is increased, say, to a value of V4, then the maximum possible load factor nmax also increases, as given by point 4 in Figure. However, nmax cannot be allowed to increase indefinitely. It is constrained by the structural limit load factor, given by point B in Figure

The horizontal line BC denotes the positive limit load factor in the V-n diagram. The flight velocity corresponding to B is designated as V*. At velocities higher than V*, say, Vs, the airplane must fly at values of CL less than (Cd)max so that the positive limit load factor is not exceeded. If flight at (C d) max is obtained at velocity V5, corresponding to point 5 in Figure, then structural damage or possibly structural failure will occur.

The right-hand side of the V-n diagram, line CD, is a high-speed limit. At flight velocities higher than this limit (to the right of line CD), the dynamic pressure is higher than the design range for the airplane. This will exacerbate the consequences of other undesirable phenomena that may occur in high- speed flight, such as encountering a critical gust and experiencing destructive flutter, aileron reversal, wing or surface divergence, and severe compressibility buffeting. Any one of these phenomena in combination with the high dynamic pressure could cause structural damage or failure. The high-speed limit velocity is a red-line speed for the airplane; it should never be exceeded. By design, it is higher than the level flight maximum cruise velocity Vmax. It may be as high as the terminal dive velocity of the aircraft. The bottom part of the V-n diagram, given by curve AE and the horizontal line ED in Figure, corresponds to negative absolute angles of attack, that is, negative lift, and hence the load factors are negative quantities.

Curve AE defines the stall limit. (If the wing is pitched downward to a large enough negative angle of attack, the flow will separate from the bottom surface of the wing and the negative lift will decrease in magnitude; that is, the wing "stalls.") Line ED gives the negative limit load factor, beyond which structural damage will occur. Line HI gives the negative ultimate load factor beyond which structural failure will occur.

For instantaneous maneuver performance, point B on the V-n diagram in Fig. 6.7 is very important. This point is called the maneuver point. At this point, both CL and n are simultaneously at their highest possible values that can be obtained anywhere throughout the allowable flight envelope of the airplane

The velocity corresponding to point B is called the corner velocity and is designated by V* in Figure The corner velocity can be obtained by solving Equation for velocity yielding

$$V^* = \sqrt{\frac{2n_{\max}}{\rho_\infty (C_L)_{\max}} \frac{W}{S}}$$

At flight velocities less than V*, it is not possible to structurally damage the airplane due to the generation of too much lift. In contrast, at velocities greater than V*, lift can be obtained that can structurally damage the aircraft (point 5 in the figure) and the pilot must make certain to avoid such a case.

UNIT IV

SPECIAL PERFORMANCE

RANGE AND ENDURANCE

Endurance

It is the amount of time an airplane can stay in the air on one load of fuel.

Now consider the equation,

$$\frac{dW_f}{dt} = -c_t T$$

or

$$dt = -\frac{dW_f}{c_t T} \qquad \text{Eq. (1)}$$

Since $T = D$ and $L = W$ in steady, level flight, Eq. (1) can be written as

$$dt = -\frac{dW_f}{c_t D} = -\frac{L}{D}\frac{1}{c_t}\frac{dW_f}{W} \qquad \text{Eq. (2)}$$

Integrating Eq. (2) from $t = 0$, where $W = W_0$, to $t = E$, where $W = W_1$, we have

$$E = -\int_{W_0}^{W_1} \frac{1}{c_t}\frac{L}{D}\frac{dW_f}{W} = \int_{W_1}^{W_0} \frac{1}{c_t}\frac{L}{D}\frac{dW_f}{W} \qquad \text{Eq. (3)}$$

Equation (3) is the general equation for the endurance E of an airplane. If the detailed variations of c_t, L/D, and W are known throughout the flight, Eq. (3) can be numerically integrated to obtain an *exact* result for the endurance.

For preliminary performance analysis, Eq. (3) is usually simplified. If we assume flight at constant c_t and L/D, Eq. (3) becomes

$$E = \frac{1}{c_t}\frac{L}{D}\int_{W_1}^{W_0} \frac{dW_f}{W}$$

or

$$\boxed{E = \frac{1}{c_t}\frac{L}{D}\ln\frac{W_0}{W_1}} \qquad \text{Eq. (4)}$$

Endurance for Propeller-Driven aircraft

The specific fuel consumption for propeller-driven airplanes is given in terms of power rather than thrust.

$$c_t = \frac{cV_\infty}{\eta_{pr}}$$

Substituting this relation into Eq.   , we have

$$E = \int_{W_1}^{W_0} \frac{\eta_{pr}}{cV_\infty} \frac{C_L}{C_D} \frac{dW_f}{W}$$

Substituting the above values in the equation

$$E = \int_{W_1}^{W_0} \frac{\eta_{pr}}{c} \sqrt{\frac{\rho_\infty S C_L}{2W}} \frac{C_L}{C_D} \frac{dW_f}{W}$$

or

$$E = \int_{W_1}^{W_0} \frac{\eta_{pr}}{c} \sqrt{\frac{\rho_\infty S}{2}} \frac{C_L^{3/2}}{C_D} \frac{dW_f}{W^{3/2}} \qquad \text{Eq. (5)}$$

By making the assumptions of constant η_{pr}, c, ρ_∞, and $C_L^{3/2}/C_D$, Eq. (5) becomes

$$\boxed{E = \frac{\eta_{pr}}{c} \sqrt{2\rho_\infty S} \frac{C_L^{3/2}}{C_D} \left(W_1^{-1/2} - W_0^{-1/2} \right)} \qquad \text{Eq. (6)}$$

The maximum endurance corresponds to the following conditions,

1. Fly at maximum $C_L^{3/2}/C_D$.
2. Have the highest possible propeller efficiency.
3. Have the lowest possible specific fuel consumption.
4. Have the highest possible difference between W_0 and W_1 (i.e., carry a lot of fuel).
5. Fly at sea level, where ρ_∞ is the largest value.

Endurance for Jet – Propelled aircraft

Here we have the equation,

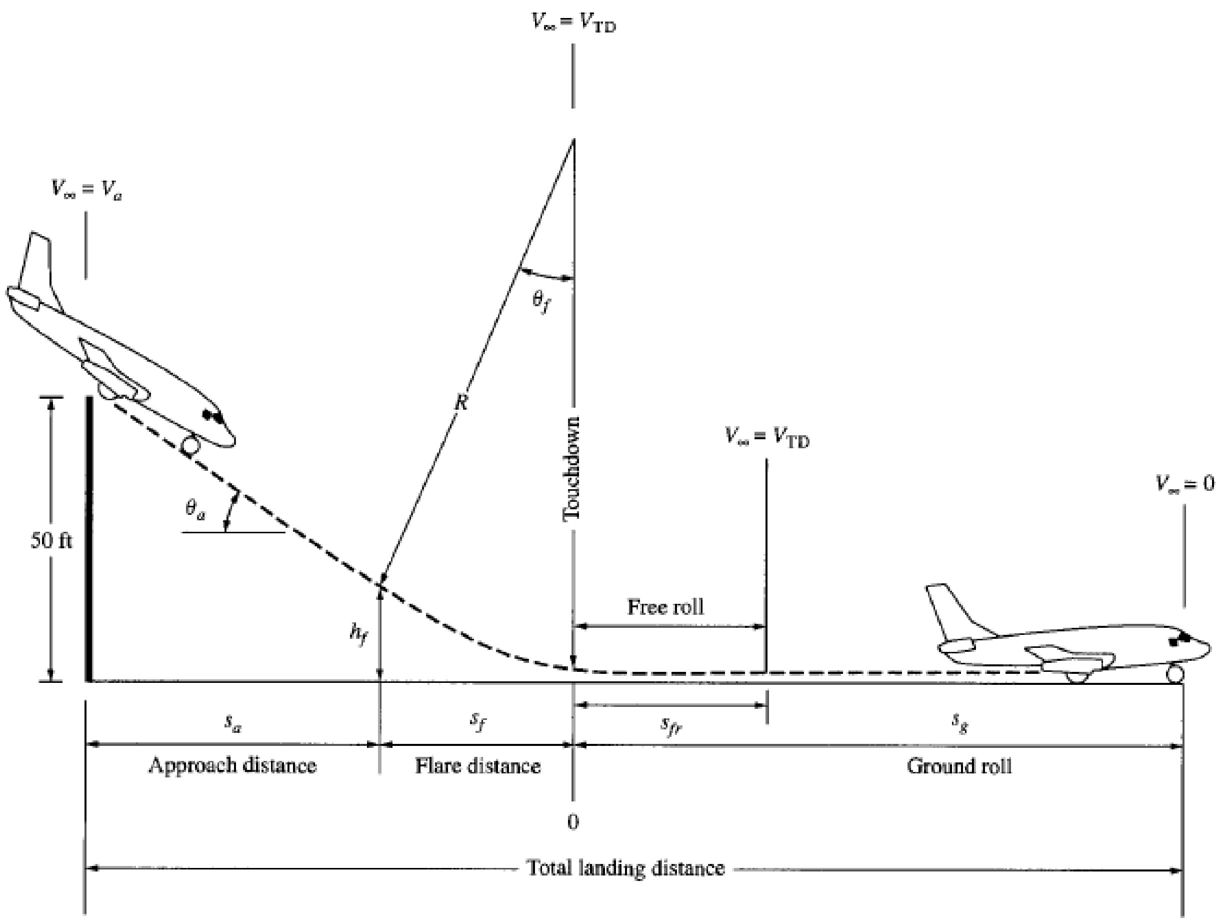
$$\boxed{E = \frac{1}{c_t} \frac{L}{D} \ln \frac{W_0}{W_1}}$$

1. Fly at maximum L/D.
2. Have the lowest possible thrust specific fuel consumption.
3. Have the highest possible ratio of W_0 to W_1 (i.e., carry a lot of fuel).

ESTIMATION OF TAKE OFF AND LANDING DISTANCE

Take off performance

The analysis of the landing performance of an airplane is somewhat analogous to that for takeoff, only in reverse. Consider an airplane on a landing approach. The landing distance, as sketched in Figure, begins when the airplane clears an obstacle, the instant it clears the obstacle, denoted by Va, is required to be equal to 1.3 Vstall for commercial airplanes and 1.2Vstall for military airplanes.



At a distance h_f above the ground, the airplane begins the flare, which is the transition from the straight approach path to the horizontal ground roll. The flight path for the flare can be considered a circular arc with radius R, as shown in Figure. The distance measured along the ground from the obstacle to the point of initiation of the flare is the approach distance S_g

Touchdown occurs when the wheels touch the ground. The distance over the ground covered during the flare is the flare distance S1. The velocity at the touch down Vm is 1.15Vstall for commercial airplanes and 1.1 Vstall for military airplanes. After touchdown, the airplane is in free roll for a few seconds before thepilot applies the brakes and/or thrust reverser.

The free-roll distance is short enough that the velocity over this length is assumed constant, equal to Vm. The distance that the airplane rolls on the ground from touchdown to the point where the velocity goesto zero is called the ground roll Sg.

Calculation of approach distance

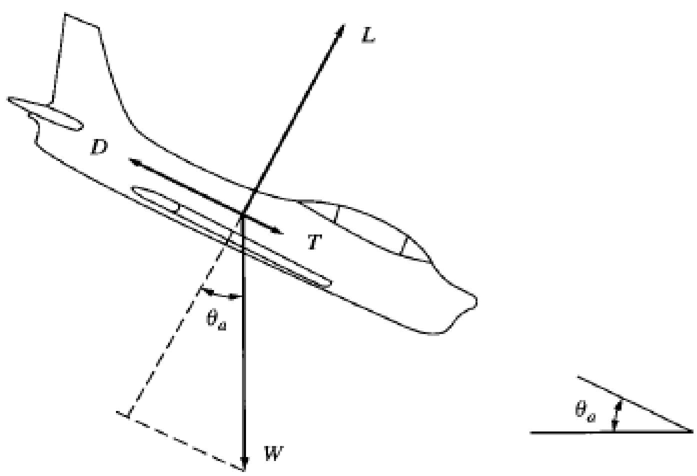
Assuming equilibrium flight conditions,

$$L = W \cos\theta_a \qquad \text{Eq.(1)}$$

$$D = T + W \sin\theta_a \qquad \text{Eq.(2)}$$

From the previous equation

$$\sin\theta_a = \frac{D-T}{W} = \frac{D}{W} - \frac{T}{W} \qquad \text{Eq.(3)}$$



Force diagram for an airplane on the landing approach flight path.

$$\sin\theta_a = \frac{1}{L/D} - \frac{T}{W} \qquad \text{Eq.(4)}$$

The flare height h_f, shown in figure , can be calculated from the construction shown in figure as follows.

$$h_f = R - R\cos\theta_f \qquad \text{Eq.(5)}$$

However, because the circular arc flight path of the flare is tangent to both the approach path and the ground, as shown in figure, $\theta_f = \theta_a$. Hence, Eq. (5), becomes

$$h_f = R(1 - \cos\theta_a) \qquad \text{Eq.(6)}$$

In Eq. (6), R is obtained from Eq. (1) by assuming that V_∞ varies from $V_a = 1.3V_{\text{stall}}$ for commercial aircraft and $1.2V_{\text{stall}}$ for military aircraft to $V_{\text{TD}} = 1.15V_{\text{stall}}$ for commercial aircraft and $1.1V_{\text{stall}}$ for military aircraft, yielding an average velocity during the flare of $V_f = 1.23V_{\text{stall}}$ for commercial airplanes and $1.15V_{\text{stall}}$ for military airplanes. With the load factor n stipulated as $n = 1.2$, Eq. (1) yields

$$R = \frac{V_f^2}{0.2g} \qquad \text{Eq.(7)}$$

Flare distance is given by

$$s_f = R\sin\theta_a$$

Force diagram for the airplane remains the same hence

$$m\frac{dV_\infty}{dt} = -D - \mu_r(W - L)$$

Many jet aircraft are equipped with thrust reversers which typically produce a negative thrust equal in magnitude to 40% or 50% of the maximum forward thrust. Some reciprocating engine/propeller-driven airplanes are equipped with reversible propellers that can produce a negative thrust equal in magnitude to about 40% of the static forward thrust. For turboprops, this increases to about 60%. In such cases, if T_{rev} denotes the absolute magnitude of the reverse thrust, then equation becomes

$$m\frac{dV_\infty}{dt} = -T_{\text{rev}} - D - \mu_r(W - L)$$

An expression for S_g can be obtained from the following,

$$\begin{aligned}
\frac{dV_\infty}{dt} &= -\frac{g}{W}\left[T_{\text{rev}} + \frac{1}{2}\rho_\infty V_\infty^2 SC_D + \mu_r\left(W - \frac{1}{2}\rho_\infty V_\infty^2 SC_L\right)\right] \\
&= -g\left[\frac{T_{\text{rev}}}{W} + \mu_r + \frac{\rho_\infty}{2(W/S)}\left(C_D - \mu_r C_L\right)V_\infty^2\right] \\
&= -g\left\{\frac{T_{\text{rev}}}{W} + \mu_r + \frac{\rho_\infty}{2(W/S)}\left[C_{D,0} + \Delta C_{D,0} + \left(k_1 + \frac{G}{\pi e \text{AR}}\right)C_L^2 - \mu_r C_L\right]V_\infty^2\right\}
\end{aligned}$$

Which can be written as,

$$\frac{dV_\infty}{dt} = -g\left(J_T + J_A V_\infty^2\right)$$

To get the total distance we integrate from 0 to V_{TD}

$$\int_{s_{fr}}^{s_g} ds = -\int_{V_{TD}}^{0} \frac{d(V_\infty^2)}{2g(J_T + J_A V_\infty^2)}$$

or

$$s_g - s_{fr} = \int_0^{V_{TD}} \frac{d(V_\infty^2)}{2g(J_T + J_A V_\infty^2)}$$

If J_T and J_A are assumed as constant then,

$$s_g - s_{fr} = \frac{1}{2gJ_A} \ln\left(1 + \frac{J_A}{J_T} V_{TD}^2\right)$$

The final equation can be written as

$$\boxed{s_g = N V_{TD} + \frac{1}{2gJ_A} \ln\left(1 + \frac{J_A}{J_T} V_{TD}^2\right)}$$

HIGH LIFT DEVICES

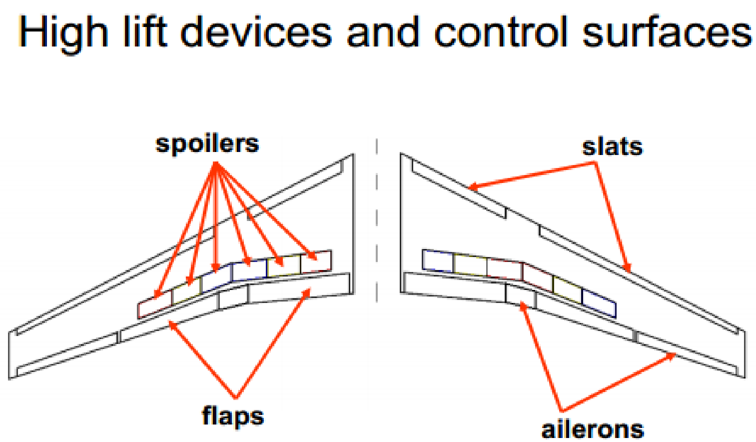
Devices to increase the lift coefficient by geometry changes (camber and/or chord) and/or boundary-layer control (avoid flow separation)

In aircraft design, high-lift devices are moving surfaces or stationary components intended to increase lift during certain flight conditions. They include common devices such as flaps and slats, as well as less common features such as leading edge extensions and blown flaps.

The various types of high lift devices are

Flaps

The most common high-lift device is the flap, a movable portion of the wing that can be lowered into the airflow to produce extra lift. Their purpose is to re-shape the wing section into one that has more camber. Flaps are usually located on the trailing edge of a wing, while leading edge flaps are occasionally used as well. Some flap designs also increase the wing chord when deployed, increasing the wing area to help produce more lift; such complex flap arrangements are found on many modern aircraft.



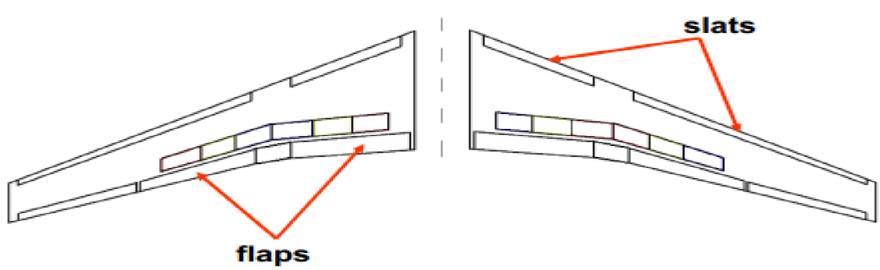
Slats and slots

They can be further divided as

- Leading edge slats
- Leading edge slot

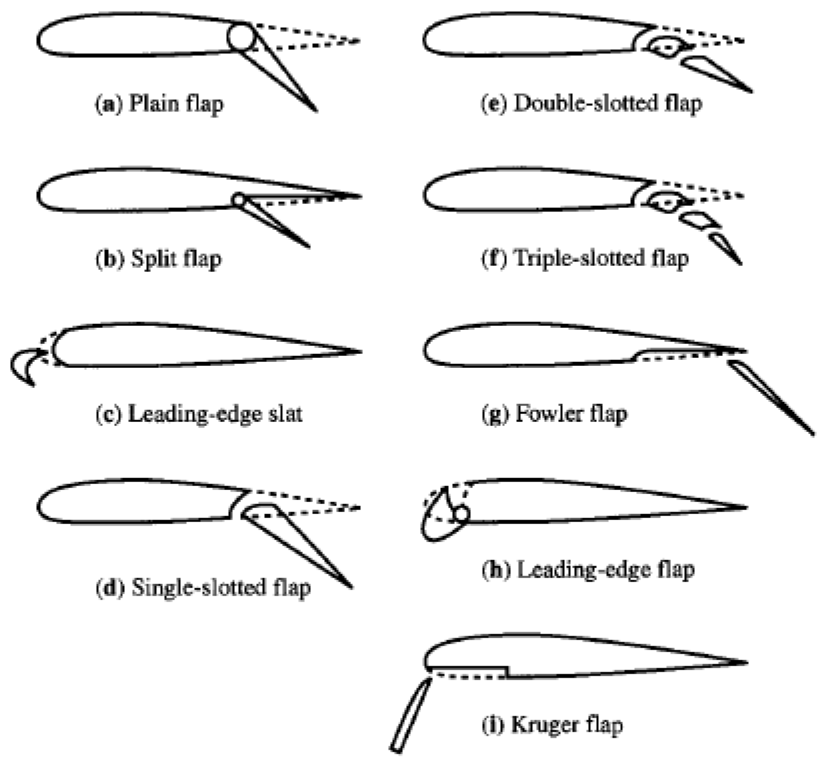
Another common high-lift device is the slat, a small aerofoil shaped device attached just in front of the wing leading edge. The slat re-directs the airflow at the front of the wing, allowing it to flow more smoothly over the upper surface while at a high angle of attack. This allows the wing to be operated effectively at the higher angles required to produce more lift.

A slot is the gap between the slat and the wing. The slat may be fixed in position, or it may be retractable. If it is fixed, then it may appear as a normal part of the leading edge of a wing which has slot. The slat or slot may be either full span, or may occur on only part of the wing (usually outboard), depending on how the lift characteristics need to be modified for good low speed control. Often it is desirable for part of the wing where there are no controls to stall first, allowing aileron control well into the stall.



Leading edge root extensions

Although not as common, another high-lift device is the leading edge root extension (LERX) or leading edge extension (LEX). A LERX typically consist of a small triangular fillet between the wing leading edge root and fuselage. In normal flight the LERX generates little lift. At higher angles of attack, however, it generates a vortex that is positioned to lie on the upper surface of the main wing.



Various types of high-lift devices.

Leading edge devices such as nose flaps, Kruger flaps, and slats reduce the pressure peak near the nose by changing the nose camber. Slots and slats permit a new boundary layer to start on the main wing portion, eliminating the detrimental effect of the initial adverse gradient.

THRUST AUGMENTATION

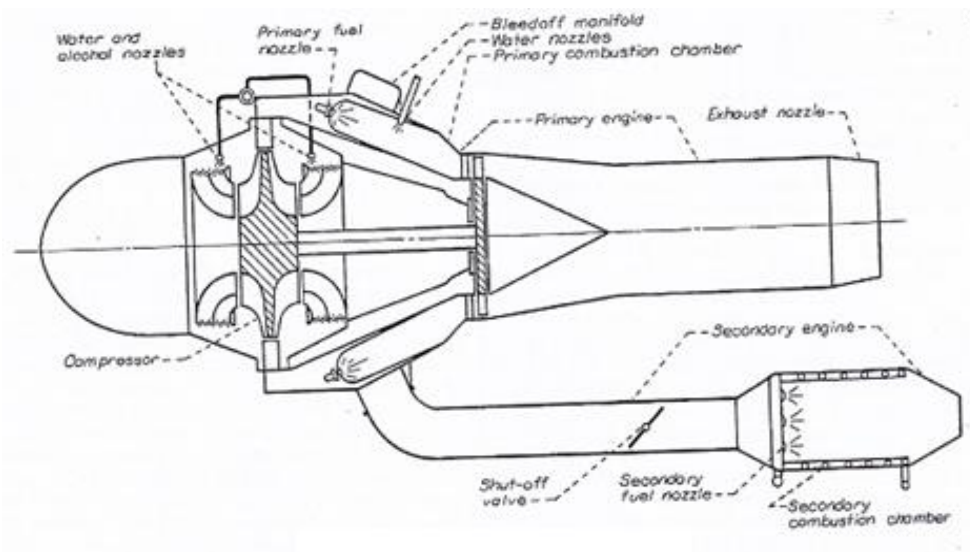
Thrust Augmentation is a method to increase the thrust produced by gas turbine engine, when the outside condition are not suitable for smooth running of the engine.

For example, when it is a very hot day outside the density of the airflow entering the engine is not sufficient enough to produce the required thrust, hence thrust augmentation is used.

There are two common methods:

1. Water, Water/Methanol Injection

Injection a mixture of water and methanol either to compressor inlet or to combustion chamber, water used is de-mineralized water to prevent corrosion and it is used to increase density of air and cool the combustion chamber walls and methanol prevents freezing up of water since methanol has low melting point.



2.After - Burners

The exhaust gases coming into the exhaust of the engine passes through a area of after burner where more fuel is added and combusted which increases the velocity of the air and this increase is proportional to the increase in thrust obtained.

THRUST REVERSAL

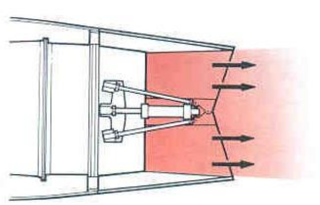
Thrust reversal, also called reverse thrust, is the temporary diversion of an aircraft engine's exhaust so that the exhaust produced is directed forward, rather than aft. This acts against the forward travel of the aircraft, providing deceleration. Thrust reverser systems are featured on many jet aircraft to help slow down just after touch-down, reducing wear on the brakes and enabling shorter landing distances. Such devices affect the aircraft significantly and are considered important for safe operation by airlines. There have been accidents involving thrust reversal systems.

Reverse thrust is also available on many propeller-driven aircraft through reversing the controllable pitch propellers to a negative angle.

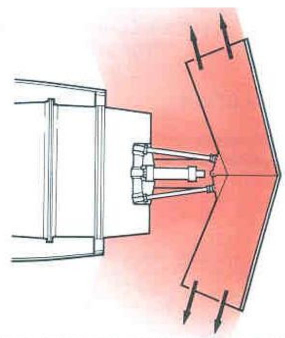
Types of thrust reversal system

1.Target type

The target thrust reverser uses a pair of hydraulically-operated 'bucket' type doors to reverse the hot gas stream. For forward thrust, these doors form the propelling nozzle of the engine. In the original implementation of this system on the Boeing 707 and still common today, two reverser buckets were hinged so when they deployed they blocked the rearward flow of the exhaust and redirected it with a forward component. This type of reverser is visible at the rear of the engine during deployment.



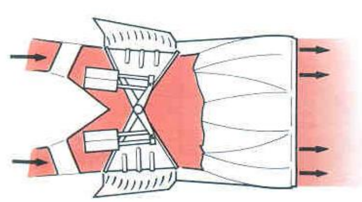
ACTUATOR EXTENDED AND BUCKET DOORS IN FORWARD THRUST POSITION



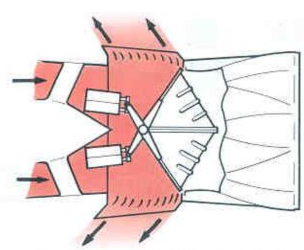
ACTUATOR AND BUCKET DOORS IN REVERSE THRUST POSITION

2. Clam-shell type

The clam-shell door, or cascade, system is pneumatically-operated. When activated, the doors rotate to open the ducts and close the normal exit, causing the thrust to be directed forward. The cascade thrust reverser is commonly used on turbofan engines. On turbojet engines, this system would be less effective than the target system, as the cascade system only makes use of the turbine airflow and does not affect the main engine core, which continues to produce thrust.



CLAMSHELL DOORS IN FORWARD THRUST POSITION

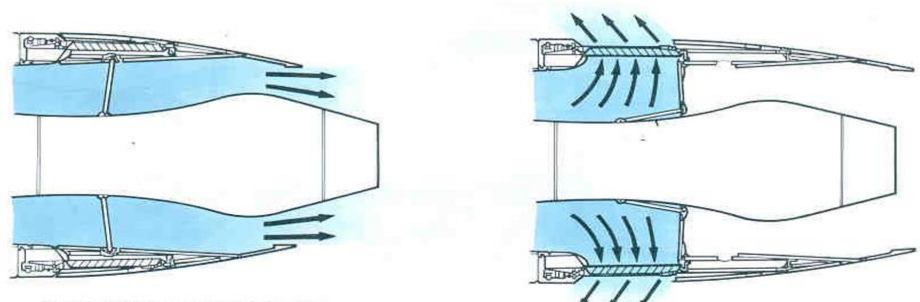


CLAMSHELL DOORS IN REVERSE THRUST POSITION

3. Cold stream type

In addition to the two types used on turbojet and low-bypass turbofan engines, a third type of thrust reverser is found on some high-bypass turbofan engines. Doors in the bypass duct are used to redirect the air that is accelerated by the engine's fan section but does not pass through the combustion chamber (called bypass air) such that it provides reverse thrust. The cold stream reverser system is activated by an air motor. During normal operation, the reverse thrust vanes are blocked. On selection, the system folds the doors to block off the cold stream final nozzle and redirect this airflow to the cascade vanes. This system can redirect both the exhaust flow of the fan and of the core.

The cold stream system is known for structural integrity, reliability, and versatility. During thrust reverser activation, a sleeve mounted around the perimeter of the engine nacelle moves aft to expose cascade vanes which act to redirect the engine fan flow. This thrust reverser system can be heavy and difficult to integrate into nacelles housing large engines.



COLD STREAM REVERSER IN FORWARD THRUST POSITION

COLD STREAM REVERSER IN REVERSE THRUST POSITION

UNIT V

AIRCRAFT PERFORMANCE

TWO MARKS QUESTIONS AND ANSWERS

Chapter I

LIFT AND DRAG ON FLIGHT PERFORMANCE

1. Definitions:

Lift:

Lift is the total force acting on a body immersed in a fluid flow in a direction perpendicular to the flow direction. Lift may also be defined as the component of the aerodynamic force in a perpendicular to the flow direction.

Drag:

Drag is the total force acting on a body immersed in a fluid flow in a direction parallel to the flow direction. Drag may also be defined as the component of the aerodynamic force in a parallel to the flow direction.

Aerodynamic Force:

It is the resultant of the pressure forces acting on a body immersed in the fluid flow. In case of airfoil aerodynamic force passes through a point on the airfoil called centre of pressure.

Centre of pressure:

It is the point on the chord line of the airfoil through which the aerodynamic force passes through. The position of centre of pressure changes with the angle of attack of the airfoil. It moves forward with the increase in angle of attack upto the stalling angle. It moves backward when the angle of attack increases beyond the stalling angle.

2. Explain the geometric characteristics of the airfoil.

Chord line: It is the straight line joining the leading edge and the trailing edge of the airfoil.

Camber line: It the line which passes through the mid points of the upper and lower surfaces of the airfoil.

Zero Lift Line: It is the line parallel to which if the air approaches the airfoil, the lift force produced is zero.

Angle of Attack: It is the angle between the flow direction and the chord line.

Absolute angle of attack: It is the angle between the flow direction and the zero lift line.

Zero lift angle of attack: It is the angle between the chord line and the camberline.

3. Explain Skin Friction Drag?

It is the drag due to the friction between the fluid particle and the surface. Skin friction drag can be reduced by using smooth surface.

4. Definitions:

Pressure Drag:

Pressure drag is the drag force due to the pressure difference between the frontal and rear portion of the body. Pressure drag can be reduced by reducing the frontal area exposed to the flow.

Profile Drag:

Profile drag is the drag of the two dimensional body. It is the sum of the skin frication drag and the pressure drag.

Induced Drag:

Induced drag is the drag due to the tilting of lift vector as a result of wing tip vortices and the trailing edge vortices. Induced drag can be reduced by increasing the aspect ratio of the wing.

Parasite Drag:

Parasite drag is the drag due to the non-lifting surfaces of the airplane such as fuselage, horizontal and vertical tail, empennage etc.

Wave Drag:

Wave drag is the drag due the shock wave formation over the upper and lower surfaces of the airplane. Wave drag can be reduced by avoiding the sudden deflection of the flow.

5. Differentiate between Streamlined and Bluff bodies.

Streamlined body is the one which has low pressure drag and comparatively high skin friction drag.

6. Differentiate between cambered and Symmetrical airfoils.

Cambered airfoil is the one which is not symmetrical about the chord line. Symmetrical airfoil is symmetrical about the chord line. Magnitude of the zero lift angle of attack is zero for the symmetrical airfoil and less than zero for a cambered airfoil.

7. Explain NACA 4 digit series airfoils with examples.

The NACA four-digit wing sections define the profile by

One digit describing maximum camber as percentage of the chord.

One digit describing the distance of maximum camber from the airfoil leading edge in tens of percents of the chord.

Two digits describing maximum thickness of the airfoil as percent of the chord.

For example, the NACA 2412 airfoil has a maximum camber of 2% located 40% (0.4 chords) from the leading edge with a maximum thickness of 12% of the chord. Four-digit series airfoils by default have maximum thickness at 30% of the chord (0.3 chords) from the leading edge.

The NACA 0015 airfoil is symmetrical, the 00 indicating that it has no camber. The 15 indicates that the airfoil has a 15% thickness to chord length ratio: it is 15% as thick as it is long.

8. Explain NACA 5 digit and 6 digit series airfoils with examples.

The NACA five-digit series describes more complex airfoil shapes:

The first digit, when multiplied by 0.15, gives the designed.

Second and third digits, when divided by 2, give *p*, the distance of maximum camber from the leading edge (as per cent of chord).

Fourth and fifth digits give the maximum thickness of the airfoil (as per cent of the chord).

For example, the NACA 12018 airfoil would give an airfoil with maximum thickness of 18% chord, maximum camber located at 10% chord, with a lift coefficient of 0.15

9. Differentiate between AOA and AOI.

Angle of Attack(AOA) is the angle between the airfoil chord line and its direction of motion relative to the air (the resulting Relative Wind).
AOA is one of the primary factors that determines amount of lift and drag produced by an airfoil.
Angle of Incidence (or AOI) is the angle between the blade chord line and the plane of rotation of the rotor system.
It is a mechanical angle rather than an aerodynamic angle

10. List the characteristics of the airfoils.
Characteristics of the airfoil are Chord length, Camber, Maximum thickness, location of maximum thickness etc.

11. Define Stalling angle of attack and Zero lift angle of attack.

Stalling angle is the angle at which the lift coefficient is maximum and beyond which the lift coefficient decreases with the increase in angle of attack.

Zero lift angle of attack is the angle of attack at which the lift coefficient is zero and is denoted as α_0. For a symmetrical airfoil $\alpha_0 = 0$ and for au unsymmetrical airfoil α_0 is negative. α_0 may also be defined as the angle between the chord line and the zero lift line.

12. Explain the phenomenon stalling.

At nominal angles the lift coefficient increases with the increase in angle of attack. The wake thickness behind the airfoil also increases with the angle of attack. The rear stagnation point moves forward over the upper surface. The lift coefficient is maximum at certain angle of attack which is known as the stalling angle beyond which lift coefficient decreases with the increase in angle of attack. This condition is known as stalling.

13. Define the Terms: Coefficient of lift and coefficient of drag.

The coefficient of lift and the drag are the non-dimensional expression of the lift and drag forces acting on the airplanes. The lift and drag coefficients are given by

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

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

14. A monoplane wing of area 36 m^2 has a span of 15 m and chord of 2.4 m is travelling at a speed of 96 km/hr and that the air density is 1.225 kg/m^3. What is the induced drag if the lift coefficient is 1.2?

Solution:

Wing area **$S = 36$ m^2**

Span **$b = 15$ m**

Chord length**, $c = 2.4$ m**

Velocity **$v = 96$ km/hr $= 26.67$ m/s**

Density of freestream air **$\rho = 1.225$ kg/m^3**

Lift coefficient **$C_L = 1.2$**

Aspect ratio **$A/R = b^2/S = 15^2/36 = 6.25$**

Induced Drag$\boldsymbol{D_i = C_{Di}\frac{1}{2}\rho V^2 S = \frac{1}{\pi e A/R}\frac{1}{2}\rho V^2 S}$

Substituting the given values $\boldsymbol{D_i = \frac{1}{\pi (1)6.25} X \frac{1}{2} X 1.225 X 26.67^2 X 36}$

Therefore Di = 802 N **Answer**

15. Define pitching moment coefficient.
Pitching moment about any point is the moment due the lift and drag forces acting on the airfoil.

16. What is centre of pressure?
It is the point on the chord line of the airfoil through which the lift and drag forces act. The position of the centre of pressure changes with the angle of attack.

17. **What is aerodynamic centre?**
It is the fixed point on the chordlineof the airfoil about which the pitching moment coefficient remains constant and does not vary with the angle of attack.

18. Describe the movement of the centre of pressure with the change in angle of attack.
At moderate angles of attack the centre of pressure moves forward with the increase in angle of attack. Beyond the stalling angle the centre of pressure moves backward with the increase in angle of attack.

19. Define the term: Aspect ratio.
It is the ratio between the wing span and the mean aerodynamic chord of the wing. It is also equal to the ratio between the square of the wing span and the wing area.

20. What is the use of winglets?

Winglet refers to a near-vertical extension of the wing tips provided to reduce the strength of wingtip vortices, which trail behind the plane. Wing tip prevents the flow of air from the lower surface of the wing to the upper surface at the tips minimizing the side flow of air over the wings. As a result of this the induced drag acting on the wing decreases.

Unit II
STEADY FLIGHT

21. Explain the condition: steady straight and level flight.
In this condition the aircraft is assumed to fly along a straight line with the uniform velocity without any change in altitude.

22. Differentiate between the equivalent air speed and the true air speed.

Equivalent air speed is the air speed measured by the airspeed indicator which is calibrated as per sea level conditions.

True air speed is the actual speed of the aircraft with respect to the ground. It is the speed calculated taking into account the actual density of the atmospheric air.

23. Define Thrust power.

It is the product of the thrust force and the velocity of the aircraft.

Thrust Power, P = Thrust X Velocity.

24. Define the term 'Power Available'

It is the power produced by the power plant of an aircraft

Power available $P_A = T_A V_\infty$

25. Explain the maximum velocity condition in level flight.

The velocity of the airplane is maximum when the maximum thrust force produced by the engine is equal to the drag force. Beyond the maximum velocity the drag force is more than the thrust force produced by the engine and hence steady level flight is not possible.

26. Show that the $C_L$min.drag $= \sqrt{C_{D0}\, K}$ with reference to $C_D = C_{D0} + KC_L^2$

Given: $C_D = C_{D0} + KC_L^2$

The drag force or C_D is minimum when $\frac{dC_D}{dC_L} = 0$

Applying the above condition we get **$C_L$min.drag** $= \sqrt{C_{D0}\, K}$

27. Describe the condition for minimum drag and minimum power required during level flight.

Conditions for Minimum drag

Profile drag = Induced drag

$C_{DO} = KC_L^2$

Conditions for MinimumPower

3 X Profile drag = Induced drag

$3C_{DO} = KC_L^2$

28. How does the power required for steady straight and level flight varies with the altitude?

Referring the equation Power Required $P_r = \frac{W^{3/2}}{\sqrt{\frac{1}{2}\rho_0\, S}} \frac{C_D}{C_L^{3/2}} \frac{1}{\sqrt{\sigma}}$

Where σ is the density ratio of the flight altitude which decreases as the height increases. Therefore the power required for straight and level flight increases as the altitude of the aircraft increases.

29. Explain how minimum thrust required for steady straight and level flight, remains constant with altitude.

For steady straight and level flight, Thrust required = Drag and hence minimum thrust required is equal to minimum drag force ($T_{min} = D_{min}$).

The minimum drag force is given by $T_{min} = D_{min} = 2W\sqrt{C_{Do}K}$

Since T_{min} is independent of density of the atmospheric air, it is also independent of the altitude.

30. Describe the effect of altitude on minimum drag of the airplane in steady straight and level flight.

The minimum drag of the airplane in steady straight and level flight is given by $D_{min} = 2W\sqrt{C_{Do}K}$ = constant. Minimum drag is independent of the altitude.

31. Why minimum drag speed condition is important in regards to a stable level flight speed of an aircraft?

When the speed of the airplane is less than the minimum drag speed, the airplane is in the unstable condition.
When the speed of the airplane is more than the minimum drag speed, the airplane is in the stable condition.

32. Define Thrust grading.

The ratio of change of thrust to torque with radius is called thrust grading

33. What are the main aspects of requirements to be considered in airplane design?

1) Range
2) Takeoff distance
3) Stalling velocity

4) Endurance
5) Maximum velocity

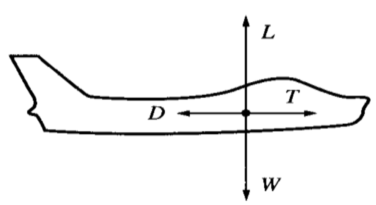
34. Define weight fraction
Weight fraction is defined as the airplane weight at end of the segment divided by the weight of the airplane at the beginning of the segment.

35. Define cruise weight fraction.
Cruise weight fraction is defined as the airplane weight at end of the cruise divided by the weight of the airplane at the beginning of the cruise.

36. Write down the selection criteria for airfoil?
- Maximum lift coefficient CL max
- Lowest minimum drag coefficient CDmin
- Highest lift to drag ratio (Cl/ Cd)max
- Highest lift curve slope.

37. What are the various air loads on an aircraft?
- Maneuver
- Gust
- Control Deflection
- Component Interaction
- Buffet

38. Write the equation of motions for a Aircraft at steady level flight.



Force diagram for steady, level flight.

$$0 = T \cos \epsilon - D$$

$$0 = L + T \sin \epsilon - W$$

39. Write the formula for thrust required.

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

---

40. Write the formula for thrust available.

Thrust Available $T_A = \frac{pr\,P}{V\infty}$

41. Write the formula for power required and available.

Power required $P_R = T_R V\infty$

Power available $P_A = T_A V\infty$

42. Why jet propulsion is insufficient at low speeds?

Jet engine throws back small mass of air at a very high velocity. Hence energy wasted is more at slow speeds.

43. Why propeller propulsion is insufficient at high speeds?

Propeller throws back a large mass of air at a comparably lower velocity. Hence at low speeds energy wasted is lesser.

44. Define wash in

When the tip chord incidence angle is higher than that of root chord the configuration is called wash in.

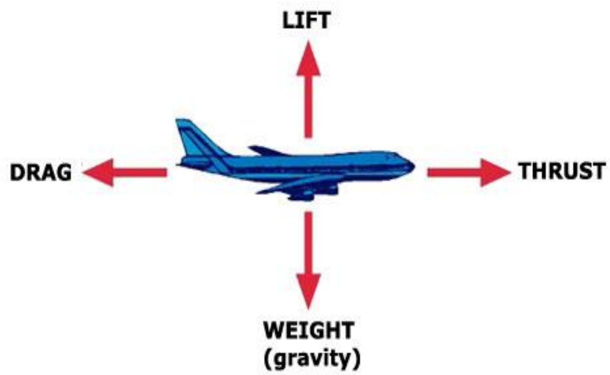
45. Define mean aerodynamic chord

The mean aerodynamic chord is defined as the chord length that when multiplied by the wing area, the dynamic pressure and the moment coefficient about the aerodynamic Centre yields the value of the aerodynamic moment about the airplane's aerodynamic Centre.

Unit III

GLIDING, CLIMBING AND TURING PERFORMANCE

46. Show the major forces acting on the airplane during steady glide of an airplane.



47. Define range and endurance of an aircraft.

Range of an aircraft is defined as the total distance travelled by an aircraft on one load of fuel. It is denoted by R.

Range $R=\frac{V\infty}{ct}\frac{L}{D}\ln\frac{Wo}{W1}$

Endurance of an aircraft is defined as the amount of time that an aircraft can stay in the air on one load of fuel. It is denoted by E.

Endurance $E=\frac{1}{ct}\frac{L}{D}\ln\frac{Wo}{W1}$

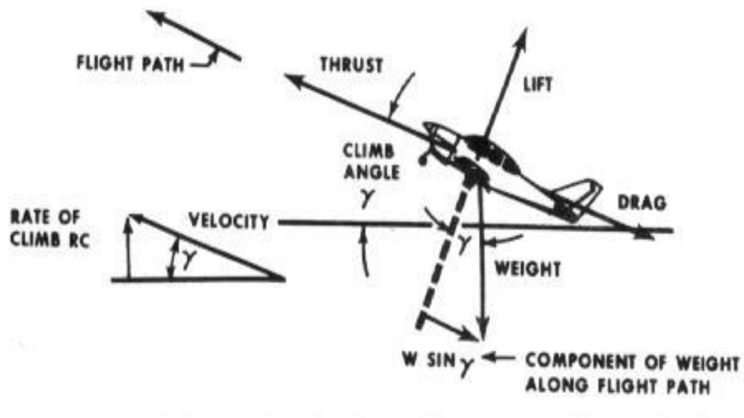
48. Describe the conditions for maximum range of an airplane.

Range of an aircraft is given by

Range $R=\frac{V\infty}{ct}\frac{L}{D}\ln\frac{Wo}{W1}$

Maximum range (distance traveled) is obtained when the aircraft is flown at the most aerodynamically efficient condition (maximum C_L/C_D).

49. Show the major forces acting on the airplane during steady climbing flight.



50. Define rate of climb.

Rate of climb is defined as the rate at which the aircraft improves its altitude.

51. Which is the main parameter affecting rate of climb?

The main parameter affecting rate of climb is Excess power.

Rate of climb = Excess power / Weight

52. Define absolute ceiling

Absolute ceiling of the aircraft is defined as the maximum height to which the aircraft can reach. At this altitude Power required (P_R) = Power available (P_A)

53. Define Service ceiling

Service ceiling of the aircraft is defined as the altitude at which the rate of climb is 100 units per unit time.

54. Write the formula for thrust required and available.

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

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

55. Describe the conditions for maximum rate of climb of an airplane.

Maximum Rate of climb will be achieved when the excess power is greatest and the aircraft is lightest.

ROC = (Power available - Power required)/W or excess power/W, so the maximum rate of climb will be achieved when the excess power is greatest and the aircraft is lightest.

56. What do you mean by load factor ?

The load factor is defined as the ratio of the lift of an aircraft to its weight and represents a global measure of the stress to which the structure of the aircraft is subjected.

$$n = \frac{L}{W}$$

Where:

n = Load factor
L = Lift
W = Weight

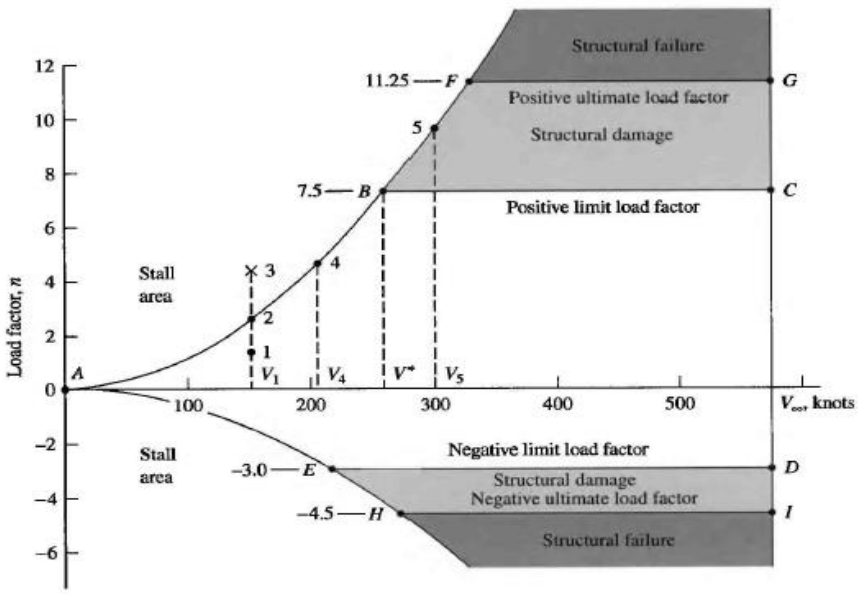
57. Write the expression for minimum radius of turn.

$$\boxed{R = \frac{V_\infty^2}{g\sqrt{n^2 - 1}}}$$

1. The highest possible load factor (i.e., the highest possible L/W).
2. The lowest possible velocity.

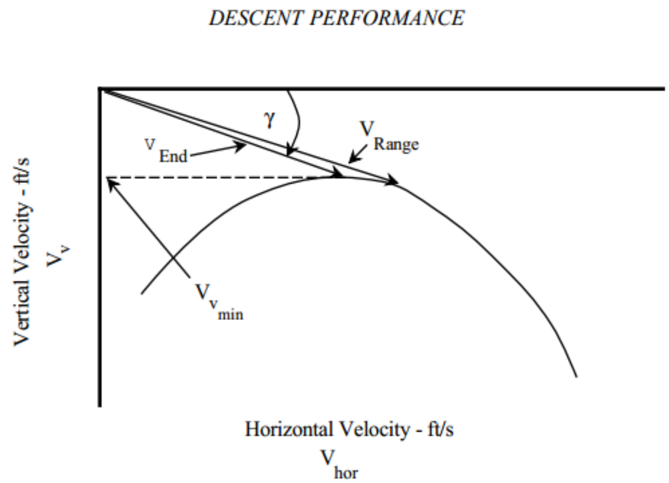
58. What is V-n diagram?

Therearestructurallimitations onthemaximumloadfactorallowedforagiven airplane.Boththe aerodynamicandstructurallimitationsforagivenairplaneareillustratedintheV-ndiagram,aplot ofloadfactorversusflightvelocity. AV-ndiagramisatypeof"flightenvelope"foragivenairplane;itestablishesthemaneuver boundaries.
Draw the v-n diagram.



59. What is hodograph?

On a hodograph, the radius vector from the origin to any part on the plot has a length proportional to the flight path speed and makes an angle to the horizontal equal to the actual descent angle (γ).



60. What are the various inertial loads experienced by an aircraft?

 Acceleration
 Rotation
 Dynamic
 Vibration
 Flutter

61. What are the various loads experienced due to landing gear by an aircraft?

 Vertical load factor
 Spin up
 Spring back
 Crabbed
 One wheel arrested
Braking

62. What are the important properties required for the material selection of an aircraft?

 Yield strength
 Ultimate strength
 Stiffness
 Density
 Fracture toughness

63. Define weight fraction

Weight fraction is defined as the airplane weight at end of the segment divided by the weight of the airplane at the beginning of the segment.

64. Define cruise weight fraction.
Cruise weight fraction is defined as the airplane weight at end of the cruise divided by the weight of the airplane at the beginning of the cruise.

65. Define aspect ratio
Aspect ratio is defines as ratio of square of wing span to the wing area.

66. Define taper ratio
Taper ratio is defined as ratio between tipchord to root chord

67. What are the advantages of Mid- Wing configuration?
- The aircraft structure is heavier.
- The mid wing is more attractive compared with two other configurations

68. What are the disadvantages of Mid- Wing configuration?
- Worst structure
- The mid wing is more expensive compared with high and low wing configurations
- The strut is usually not used to reinforce the wing structure

69. What are the advantages of Low- Wing configuration?
- The aircraft take off performance is better; compared with a high wing configuration; due to the ground effect.
- The pilot has a better higher-than-horizon view, since he/she is above the wing.
- The retraction system inside the wing is an option along with inside the fuselage

70. Write down the selection criteria for airfoil?
- Maximum lift coefficient CL max
- Lowest minimum drag coefficient CDmin
- Highest lift to drag ratio (Cl/ Cd)max
- Highest lift curve slope.

Unit IV

SPECIAL PERFORMANCE

71. Define the term Endurance of airplanes during level flight.

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

Endurance can be written as:

$$E = \int_{t_1}^{t_2} dt = -\int_{W_1}^{W_2} \frac{dW}{F} = \int_{W_2}^{W_1} \frac{dW}{F}.$$

Define fuel weight in aircraft?

This is the weight of the fuel in the fuel tanks. Since fuel is consumed during the course of the fuel Wfuelis a variable decreasing with time during the flight. 44) Explain empty weight of an aircraft? This is the weight of everything else the structure, engine, electronic components,.Landing gear, fixed equipment and anything else that is not crew, payload and fuel.

72. What is SFC?

SFC for thrust engines (e.g. turbojets, turbofans, ramjets, rocket engines, etc.) is the mass of fuel needed to provide the net thrust for a given period e.g. lb/(h·lbf) (pounds of fuel per hour-pound of thrust)

73. Define critical engine failure speed.

It is the decision speed nominated by the pilot which satisfies all safety rules, and above which the takeoff will continue even if an engine fails. The speed will vary among aircraft types and varies according to factors such as aircraft weight, runway length, wing flap setting, engine thrust used and runway surface contamination.

74. Describe the condition for maximum endurance of a jet engine airplane.

The maximum endurance corresponds to the following conditions,

- Highest propeller efficiency
- Lowest possible specific fuel consumption
- Flight at sea level
- Maximum $C_L^{3/2}/C_D$ value

75. What is "SAR" ?

Specific Air Range is the derivative of range with respect to mass.

$$\text{SAR} = \frac{\partial X}{\partial m}$$

76. State the relation between wing loading and landing speed.

Given that the stall always occurs at the same critical angle of attack, by increasing the load factor such critical angle and the stall , will be reached with the airspeed remaining well above the normal stall speed, that is:

L =nW

Where

L=Lift

n=Load factor

W= Weight of the aircraft

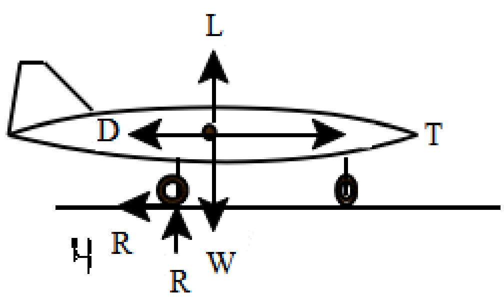
77. Why longer runway is required for an aircraft when airport is located at high altitude?

This is because of the reduction in density of air with altitude. When density decreases lift also reduces. So longer runway is used to produce the required lift.

78. State the purpose of spoilers.

A spoiler (sometimes called a lift dumper) is a device intended to reduce lift in an aircraft. Spoilers are plates on the top surface of a wing which can be extended upward into the airflow and spoil it. By doing so, the spoiler creates a carefully controlled stall over the portion of the wing behind it, greatly reducing the lift of that wing section. Spoilers differ from airbrakes in that airbrakes are designed to increase drag making little change to lift, while spoilers greatly reduce lift making only a moderate increase in drag.

79. Sketch the forces acting on an airplane during ground run.



Where,

T= Thrust, L=Lift, D= Drag, W= Weight, R=Reaction Force, R= Coefficient of friction

80. What are high lift devices?

In aircraft design, high-lift devices are moving surfaces or stationary components intended to increase lift during certain flight conditions. They include common devices such as flaps and slats, as well as less common features such as leading edge extensions and blown flaps.

81. Explain the roll of high lift devices for short take off and landing.

A larger wing will provide more lift and reduce takeoff and landing distance, but will increase drag during cruising flight and thereby lead to lower than optimum fuel economy. High-lift devices are used to smooth out the differences between the two goals, allowing the use of an efficient cruising wing, and adding lift for takeoff and landing.

82. Name two thrust augmentation devices used.

- After burner
- Water injection system

83. Explain the concept of reverse thrust.

Thrust reversal, also called reverse thrust, is the temporary diversion of an aircraft engine's exhaust so that the exhaust produced is directed forward, rather than aft. This acts against the forward travel of the aircraft, providing deceleration.

84. On a hot day, a given airplane requires a longer ground roll for take off (true/false). Justify your answer.

True this is because on a hotter day the density of air will be less consequently the effective thrust produced will also be low. So the aircraft should run for longer distance before it achieves the sufficient lift for takeoff.

85. What is ground effect?

In fixed-wing aircraft, ground effect is the increased lift and decreased drag that an aircraft's wings generate when they are close to a fixed surface.[1] When landing, ground effect can give the pilot the feeling that the aircraft is "floating". When taking off, ground effect may temporarily reduce the stall speed.

86. Define Range of an aircraft.

Range of an aircraft is defined as the total distance travelled by an aircraft on one load of fuel. It is denoted by R.

87. Write the formula for range and endurance.

Range $R=\frac{V\infty}{ct}\frac{L}{D}\ln\frac{Wo}{W1}$ **Endurance** $E=\frac{1}{ct}\frac{L}{D}\ln\frac{Wo}{W1}$

88. Write down the conditions for unaccelerated, steady level flight?

Thrust is equal to Drag,T=D and Lift is equal to weightL=W. Climb angle and roll angle is equal to zero

89. Define bye-pass ratio.
Bye-pass ratio is defined as the ratio of mass flow passing through the fan, via bye pass duct to the mass flow passing through the core itself.

90. Write the different types of power plants.
1. Reciprocating Engine
2. Turbojet
3. Turbofan
4. Turboprop

91. Define thrust and propulsive efficiency.
Thrust is defined as the forward motion of an aircraft and is denoted by T, propulsive efficiency is defined as the ratio of useful power available to the total power generated.

92. What are the important components in turboprop engine?
The important component are diffuser,propeller,reduction gear, high pressure compressor,low pressure compressor, combustionchamber, high speed turbine, low speed turbine and nozzle.

93. Write down the formulas to calculate landing distance?
Landing distance = Sa+Sj+Sg
Where,
Sa= 50-hf
Tanθa
Sj= R sinθs
Sg= JN×(2/ρα×W/S × 1/Clmax)^0.5+ J2× W/S
g× ραClmaxμr

94. Write down the phases of airplane design?
1) Conceptual design
2) Preliminary design
3) Detailed design

95. Define crew weight.
The crew comprises the people necessary to operate the airplane in flight. E.g. Pilot

96. Define payload in aircraft.
The payload is what the airplane is intended to transport e.g. Passenger, baggage, freight etc. If the airplane is intended for military purpose then the payload includes bombs, rockets and other disposable ordnance.

97. Which are segments in landing performance?
The segments in landing performance are ground roll, approach distance and flare distance.

98. Define landing distance with formula.
Landing distance is defined as the sum of approach distance, flare distance and ground roll

99. Write down the advantages of swept wing?
Improving the wing aerodynamic feature
Adjusting the aircraft center of gravity
Improving longitudinal and directional stability
Increasing pilot view.

Unit V
PROPELLERS

100. What is the use of propeller in an airplane?
Aircraft propellers or airscrews convert rotary motion from piston engines, turboprops or electric motors to provide propulsive force. The propeller is usually attached to the crankshaft of a piston engine, either directly or through a reduction unit.

101. What is an actuator disk?

Actuator disc is an imaginary replacement to a propeller. It is assumed that an actuator disc has

- Infinite number of blades on it
- Produce uniform change in velocity of fluid passing through the disc.

102. Name the types of propellers.
- Fixed pitch propeller
- Variable pitch propeller
- Reverse pitch propeller
- Contra rotating propellers
- Counter rotating propellers

103. List the assumptions made in the Froud momentum theory.

- The flow is inviscid and steady (ideal flow), therefore the propeller does not experience energy losses due to frictional drag.
- Also the rotor is thought of as an actuator disk with an infinite number of blades, each with an infinite aspect ratio.
- The propeller can produce thrust without causing rotation in the slipstream.

- Here the rotor is assumed as an infinitely thin disc, which induces a constant velocity along the axis of rotation.

104. Why do we need "Geometrical twist" on propeller?

If the blades had the same geometric pitch throughout their lengths, at cruise speed the portions near the hub could have negative angles of attack while the propeller tips would be stalled. Propeller blades are twisted to change the blade angle in proportion to the differences in speed of rotation along the length of the propeller and thereby keep thrust more nearly equalized along this length.

105. What do you mean by pitch of the propeller?

The pitch in inches is the distance which the propeller would screw through the air in one revolution if there were no slippage.
The pitch of a propeller may be designated in inches.

106. Define activity factor of a propeller.

Activity factor is a measure of how much power any given propeller can absorb. It is roughly equal to solidity times a constant for any particular prop.

107. What are the merits and demerits of fine and coarse pitch propellers?

PROPELLER	ADVANTAGES	DISADVANTAGES
Cruise – Coarse pitch	Good cruise speed Poor climb performance	Long take-off roll
Climb – Fine pitch	Short take-off roll Good climb performance	Low cruise speed

108. Why is the tip speed an important factor in propeller design?

Maximum helical tip velocity is an important parameter for propeller selection. In the absence of specific data from the prop manufacturer, it is safe to assume that (a) the maximum prop efficiency will be about 87% (for any metal prop a non-governmental agency can afford), and (b) that the prop efficiency begins to decrease dramatically when the prop is operated at a helical tip velocity in excess of 0.85 Mach. That occurs because the local air velocity over the surface of the prop (near the point of maximum airfoil thickness) will reach Mach 1, and create a shock wave, separating the flow and dissipating prop energy.

109. **Define Solidity**.

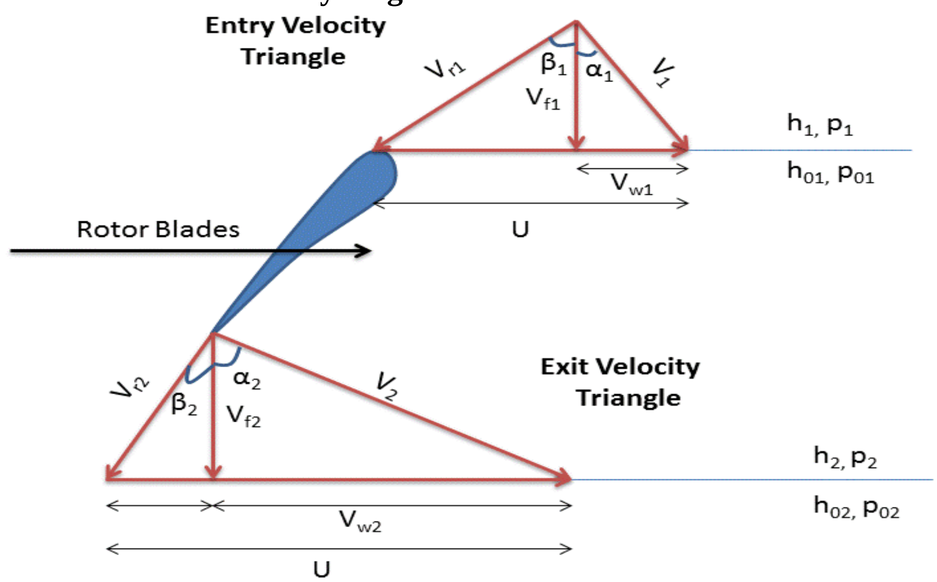
The solidity of a propeller is defined as the ratio of the area of all the bade elements to the area of the complete annulus of the actuator disc of same outer diameter as the propeller.it is seen that minimum solidity of propeller yields maximum efficiency.

110. Differentiate between fixed pitch and variable pitch propellers.

Fixed pitch and ground adjustable propellers are designed for best efficiency at one rotation and forward speed. They are designed for a given airplane and engine combination. A propeller may be used that provides the maximum propeller efficiency for either takeoff, climb, cruise, or high speed flight. Any change in these conditions results in lowering the efficiency of both the propeller and the engine. Since the efficiency of any machine is the ratio of the useful power output to the actual power input, propeller efficiency is the ratio of thrust horsepower to brake horsepower.

A constant speed propeller, however, automatically keeps the blade angle adjusted for maximum efficiency for most conditions encountered in flight. During takeoff, when maximum power and thrust are required, the constant speed propeller is at a low propeller blade angle or pitch. The low blade angle keeps the angle of attack small and efficient with respect to the relative wind. At the same time, it allows the propeller to handle a smaller mass of air per revolution.

111. Draw the velocity diagram for the flow over a blade cross section.



112. What are the blade- coefficients used to describe the performance of the propeller?

- Efficiency
- Torque coefficients
- Power coefficients
- Thrust coefficients

113. Define pitch angle

Pitch angle is defined angle between the plane of symmetry and the plane containing the chord of the airfoil. It is the angle at which each blade is set.

114. What are the factors affecting the actual efficiency of the propeller?

a) Thrust is not uniform over the disc due to losses at root and tip of blades.
b) There is loss of energy due to the rotation of the slip stream of real fluid.
c) Losses due to skin friction drag as the fluid is a real one.

115. What is meant by advance ratio in a propeller blade?

Advance ratio is the dimensionless quantity given by $J = V / ND$
Where,
V= relative velocity of forward motion
N= speed of propeller in rps
D= Diameter of propeller.

116. Define geometrical pitch.

Geometrical pitch is defined as theoretical forward distance which should be advanced by a point on the blade if the blade can move into air as into a solid without any slip.

117. Define Blade angle.

The angle at which the chord of the blade element set to the plane of rotation is called blade angle.

118. Define Solidity.

The solidity of a propeller is defined as the ratio of the area of all the bade elements to the area of the complete annulus of the actuator disc of same outer diameter as the propeller.it is seen that minimum solidity of propeller yields maximum efficiency.

119. Define effective pitch

It is the average of all pitch values at all points on the blades.

120. Define efficiency of propeller?

Efficiency of propeller is defined as the ratio of the power output to power input.

121. What do you mean by breaking propeller?

In these propellers, the angle β is reduced to a negative value so that the power supplied to the to the propeller will result in a negative thrust or an anti-directional torque.

122. What factors make the efficiency of a propeller?

It is dependent on
Forward velocity
Thrust of propeller
Rotational Speed
Torque exerted by engine.

123. How solidity can be increased?

Either increase the chord of the airfoil blade or increase number of blades.

124. Why propeller propulsion is insufficient at high speeds?

Propeller throws back a large mass of air at a comparably lower velocity. Hence at low speeds energy wasted is lesser.

125. Define aspect ratio

Aspect ratio is defines as ratio of square of wing span to the wing area.

126. Define taper ratio

Taper ratio is defined as ratio between tipchord to root chord

127. Define twist angle.

When the tip chord incidence angle is smaller than that of root chord the configuration is called twist angle αt.

KARPAGAM UNIVERSITY

Coimbatore 21.

Faculty of Engineering

B.E – Aeronautical Engineering –V Semester

END SEMESTER EXAMINATION ANSWER KEY

Subject Code: 12BEAR502 **Maximum Marks: 100**

Subject Name: Aircraft Performance **Time: 3 Hrs**

PART A (15 x 2 = 30 Marks)

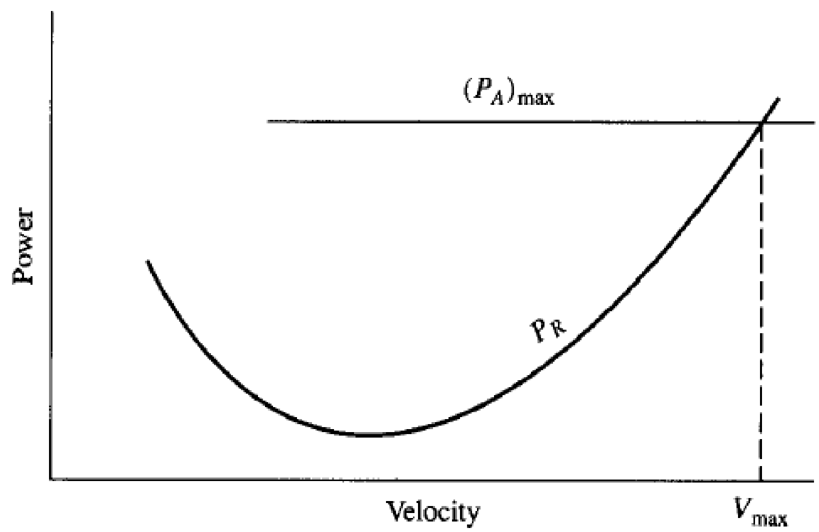
1. Define Lift.
   Lift is the total force acting on a body immersed in a fluid flow in a direction perpendicular to the flow direction. Lift may also be defined as the component of the aerodynamic force in a perpendicular to the flow direction.

2. What do you mean by Skin Friction Drag?
   It is the drag due to the friction between the fluid particle and the surface. Skin friction drag can be reduced by using smooth surface.

3. Differentiate between Streamlined and Bluff bodies.
   Streamlined body is the one which has low pressure drag and comparatively high skin friction drag.
4. Write the expression for Coefficient of drag.
   The coefficient of drag is the non-dimensional expression of the drag forces acting on the airplanes. The drag coefficient is given by
   $$C_D = \frac{D}{\frac{1}{2}\rho V^2 S}$$

5. Define the term Power Available.
   It is the power produced by the power plant of an aircraft
   Power available PA=TAV∞

6. Draw the power required vs Velocity Curve during level flight of an airplane at the same altitude.



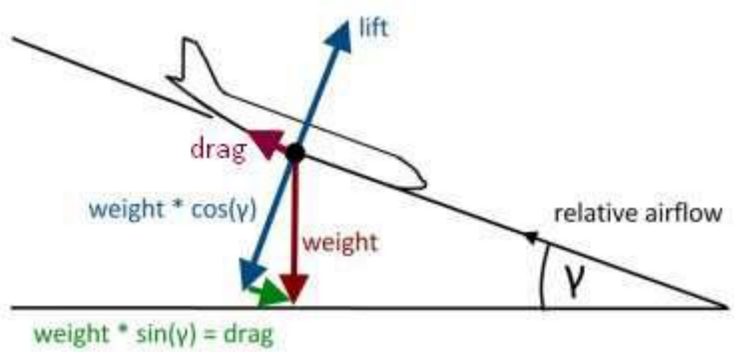
7. How does minimum thrust required for steady straight and level flight remain constant with altitude?

For steady straight and level flight, Thrust required = Drag and hence minimum thrust required is equal to minimum drag force ($T_{min} = D_{min}$).

The minimum drag force is given by $T_{min} = D_{min} = 2W\sqrt{C_{Do}K}$

Since T_{min} is independent of density of the atmospheric air, it is also independent of the altitude.

8. Show the major forces acting on the airplane during steady glide of an airplane.



9. Describe the conditions for minimum sinking speed of a gliding flight

   The minimum sink speed is the airspeed speed at which the glider is descending as slowly as possible through the air. In still air flying at minimum sink speed is how you achieve maximum flight time.

10. What is a shallow angle of climb?
When angle of attack becomes greater during climb as the bank angle increases it is called as shallow angle of climb.

11. What factors dictate the minimum turn radius?
1. Large CL
2. Low altitude (high density)
3. High load factor n (n max)
4. Low wing loading (W/S)

12. Define the term Range of airplanes during level flight.
Range of an aircraft is defined as the total distance travelled by an aircraft on one load of fuel. It is denoted by R.

Range $R = \frac{V\infty}{ct} \frac{L}{D} \ln \frac{Wo}{W1}$

13. Describe the condition for maximum endurance of a piston engine airplane.
The maximum endurance corresponds to the following conditions,

- Highest propeller efficiency
- Lowest possible specific fuel consumption
- Flight at sea level
- Maximum $C_L^{3/2}/C_D$ value

14. Name the types of propellers.

- Fixed pitch propeller
- Variable pitch propeller
- Reverse pitch propeller
- Contra rotating propellers
- Counter rotating propellers

15. What is an actuator disk?
Actuator disc is an imaginary replacement to a propeller. It is assumed that an actuator disc has

- Infinite number of blades on it
- Produce uniform change in velocity of fluid passing through the disc.

PART – B (5 X 14 = 70 Marks)

20 Derive the expression for drag polar and explain it with a neat plot.

For every aerodynamic body, thereis a relation between CD and CL that can be expressed as an equation or plotted on a graph. Both the equation and the graph are called the drag polar.

(Totaldrag)=(parasitedrag)+(wavedrag)+(induceddrag)

$$C_D = C_{D,e} + C_{D,w} + \frac{C_L^2}{\pi e \text{AR}}$$

The parasite drag coefficient CD,e can be treated as the sum of its value at zero lift CD,e,o and the increment in parasite drag ΔCD,e due to lift. The skin-friction drag (to a lesser extent) and the pressure drag due to flow separation (to a greater extent) change when α changes; the sum of these changes creates ΔCD,e

$$C_{D,e} = C_{D,e,0} + \Delta C_{D,e} = C_{D,e,0} + k_1 C_L^2$$

For a flat plate at angle of attack,

$$c_{d,w} = \frac{4\alpha^2}{\sqrt{M_\infty^2 - 1}} = \frac{4}{\sqrt{M_\infty^2 - 1}} \left(\frac{c_l \sqrt{M_\infty^2 - 1}}{4} \right)^2$$
$$= \frac{c_l^2 \sqrt{M_\infty^2 - 1}}{4}$$

Since CD,w is simply the wave drag coefficient due to lift, and since equation shows that $C_{D,w}$ varies asC_L^2 Hence,

$$C_{D,w} = C_{D,w,0} + \Delta C_{D,w} = C_{D,w,0} + k_2 C_L^2$$

Also

$$C_D = C_{D,e,0} + C_{D,w,0} + k_1 C_L^2 + k_2 C_L^2 + \frac{C_L^2}{\pi e \text{AR}}$$

Assume k_3 to be a constant such that

$$k_3 \equiv 1/(\pi e \text{AR})$$

So that the equation becomes

$$C_D = C_{D,e,0} + C_{D,w,0} + (k_1 + k_2 + k_3)C_L^2$$

Sum of the first two terms is equal to the zero lift drag coefficientC_{D0}

$$C_{D,e,0} + C_{D,w,0} \equiv C_{D,0}$$

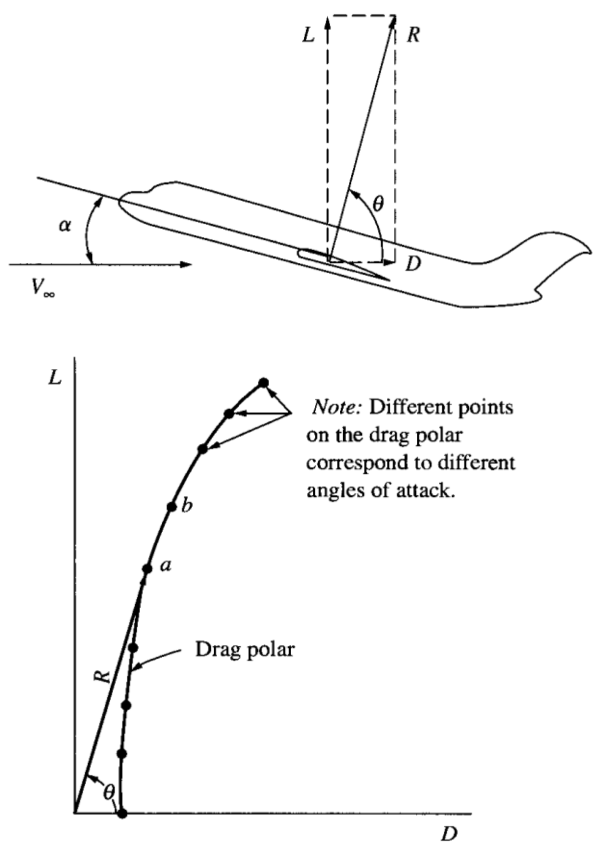
$$k_1 + k_2 + k_3 \equiv K$$

On substituting the above equations we have for the complete airplane

$$\boxed{C_D = C_{D,0} + KC_L^2}$$

This is called the drag polar equation

Construction for the resultant aerodynamic force on a drag polar



17. a) i) Explain the conditions for thrust or power required for an aircraft.

POWER REQUIRED

Consider a force F acting on an object moving with velocity V. Both F and V are vectors and may have different directions. At some instant, the object is located at a position given by the position vector r. Over a time increment dt, the object is displaced through the vector dr,
The work done on the object by the force F acting through the displacement dr is F.dr. Power is the time rate of doing work, or

$$\text{Power} = \frac{d}{dt}(\mathbf{F} \cdot \mathbf{dr}) = \mathbf{F} \cdot \frac{\mathbf{dr}}{dt}$$

Since,

$$\frac{\mathbf{dr}}{dt} = \mathbf{V}$$

Then

$$\text{Power} = \mathbf{F} \cdot \mathbf{V}$$

Consider an airplane in straight and level flight. The velocity of the airplane is $V\infty$.The concept of thrust required TR was introduced, where TR = D. In this section, we introduce the analogous concept of power required, denoted by PR. Since both T and $V\infty$ are horizontal, the dot product gives for the power required

$$\boxed{P_R = T_R V_\infty}$$

Now,

$$P_R = T_R V_\infty = \frac{W}{C_L/C_D} V_\infty$$

Since L=W for steady level
flight

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

Solving the above equation
gives

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

$$P_R = \frac{W}{C_L/C_D}\sqrt{\frac{2W}{\rho_\infty S C_L}}$$

or

$$\boxed{P_R = \sqrt{\frac{2W^3 C_D^2}{\rho_\infty S C_L^3}}}$$

ii) Explain the conditions for power available for an aircraft.

POWER AVAILABLE

Power available is the power provided by the powerplant of the airplane
It is given by,

$$P_A = T_A V_\infty$$

Propeller driven aircraft

They are driven by reciprocating piston engines or gas turbine engines

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

where η_{pr} is the propeller efficiency and P is the shaft power from the reciprocating engine.

The velocity and altitude effects are as follows

1. power is reasonably constant with

2. for an unsupercharged engine

$$\boxed{\frac{P}{P_0} = \frac{\rho}{\rho_0}}$$

Where P and are the shaft per output and density, respectively, at altitude and P_o and $\rho_o$are the corresponding values at sea level. Considering the temperature effects

$$\boxed{\frac{P}{P_0} = 1.132\frac{\rho}{\rho_0} - 0.132}$$

For a supercharged engine, P is essentially constant up to the critical design altitude of the supercharger. Above this critical altitude, P decreases according to the above equation.

Thus we have,

$$\boxed{P_A = \eta_{pr} P_{es}}$$

Where P_{es} is the equivalent shaft power.

Thus it can be written that

$$\boxed{\frac{P_A}{P_{A,0}} = \left(\frac{\rho}{\rho_0}\right)^n \qquad n = 0.7}$$

Turbojet and turbofan engines

Turbofan and turbojet are rated in terms of thust. Thus it can be written as

$$\boxed{P_A = T_A V_\infty}$$

For a turbojet engine,

At subsonic speed T_A is essentially constant. So that,

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

The effect of altitude on T_A is given by,

$$\boxed{\frac{P_A}{(P_A)_0} = \frac{\rho}{\rho_0}}$$

The mach number variation of thrust is given by

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

The altitude variation of turbofan thrust is given by

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

Hence the variation of P_A with altitude is also the same

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

(OR)

b) Using the analytical and graphical approach derive the expressions for the thrust required.

THRUST REQUIRED

Imagine this airplane in steady, level flight at a given velocity and altitude. To maintain this speed and altitude, enoughThrust must be generated to exactly overcome the drag and to keep the airplane going- this is the thrust required to maintain these flight conditions. The thrust required TR depends on the velocity, the altitude, and the aerodynamic shape, size, and weight of the airplane.

The thrust required is simply equal to the drag of the airplane-it is the thrust required to overcome the aerodynamic drag.

1. Graphical Approach

Consider a given airplane flying at a given altitude in steady, level flight. For the given airplane, we know the following physical characteristics: weight W, aspect ratio AR, and wing plan form area S.

We know that $C_D = C_{D,0} + KC_L^2$

where CD and K are known for the given airplane. To calculate the thrust required curve, proceed as follows:

1. Choose a value of V_∞•
2. For the chosen V_∞, calculate CL from the relation

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

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

3. Calculate CD

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

4. Calculate drag, hence TR,

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

This is the value of TR corresponding to the velocity chosen in step 1. This combination (T_R, V_∞) is one point on the thrust required curve.

5. Repeat steps 1 to 4 for a large number of different values of V_∞, thus generating enough points to plot the thrust required curve.

2. Analytical Approach

The thrust required curve from an analytical point of view is examined here.
For steady, level flight we have

$$T_R = D = \frac{D}{W}W = \frac{D}{L}W$$

$$\boxed{T_R = \frac{W}{L/D}}$$

The lift to drag ratio can be written as

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

From the drag polar equation, we have

$$D = q_\infty S C_D = q_\infty S(C_{D,0} + KC_L^2)$$

$$L = W = q_\infty S C_L = \tfrac{1}{2}\rho_\infty V_\infty^2 S C_L$$

From which

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

Substituting in the drag polar equation, we have

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

$$\boxed{D=\frac{1}{2}\rho_\infty V_\infty^2 S C_{D,0}+\frac{2KS}{\rho_\infty V_\infty^2}\left(\frac{W}{S}\right)^2}$$

Now replacing the value of q∞ as

$$q_\infty=\tfrac{1}{2}\rho_\infty V_\infty^2$$

We know that D=TR

$$T_R=q_\infty S C_{D,0}+\frac{KS}{q_\infty}\left(\frac{W}{S}\right)^2$$

Now multiplying by q∞ and rearranging

$$q_\infty^2 S C_{D,0}-q_\infty T_R+KS\left(\frac{W}{S}\right)^2=0$$

Obtaining the value of q∞ from the above equation

$$q_\infty=\frac{T_R\pm\sqrt{T_R^2-4SC_{D,0}K(W/S)^2}}{2SC_{D,0}}$$

$$=\frac{T_R/S\pm\sqrt{(T_R/S)^2-4C_{D,0}K(W/S)^2}}{2C_{D,0}}$$

Replacing q∞ with

$$q_\infty=\tfrac{1}{2}\rho_\infty V_\infty^2$$

Now the value of V∞ can be obtained as

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

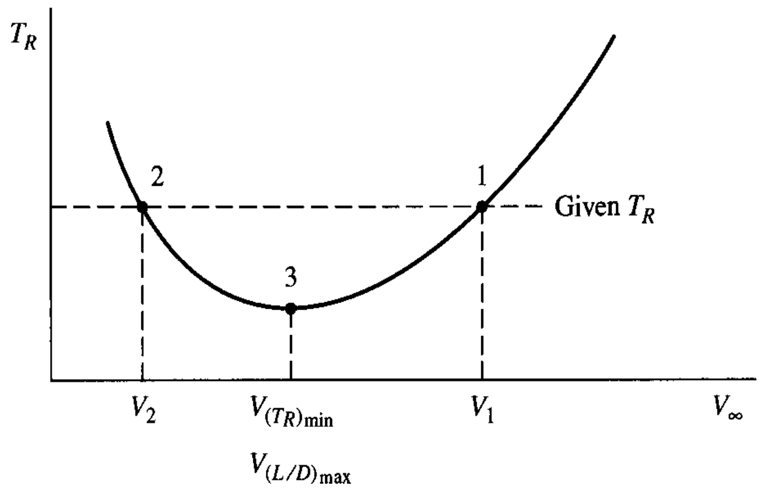
We know that

$$\frac{T_R}{S}=\frac{T_R}{W}\frac{W}{S}$$

On substituting the above value and taking the square root we get the value of V∞ as

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

Plot of Thrust required (TR) and Velocity(V) gives the following graph

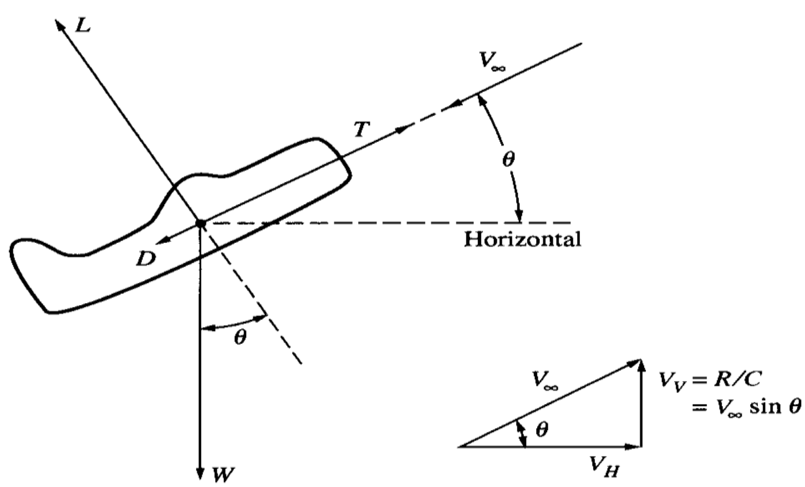


18. **a)** Explain and derive an expression for rate of climb.

RATE OF CLIMB

Consider a steady unaccelerated climb. The equations of motion for this condition is given by

$$T\cos\epsilon - D - W\sin\theta = 0$$
$$L + T\sin\epsilon - W\cos\theta = 0$$



Force and velocity diagrams for climbing flight.

Here the vertical component gives the rate of climb.

the rate of climb by R/C. From this diagram,

$$\boxed{R/C = V_\infty \sin\theta}$$

On multiplying the equation by V_∞/W we have

$$\boxed{V_\infty \sin\theta = R/C = \frac{TV_\infty - DV_\infty}{W}}$$

Thus power available is the power required to overcome this drag

$$\boxed{TV_\infty - DV_\infty \equiv \text{excess power}}$$

Hence,

$$\boxed{R/C = \frac{\text{excess power}}{W}}$$

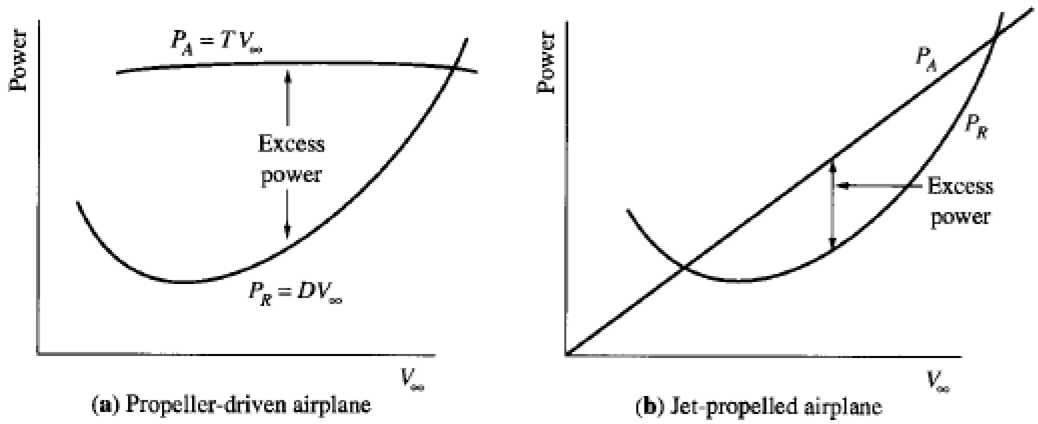
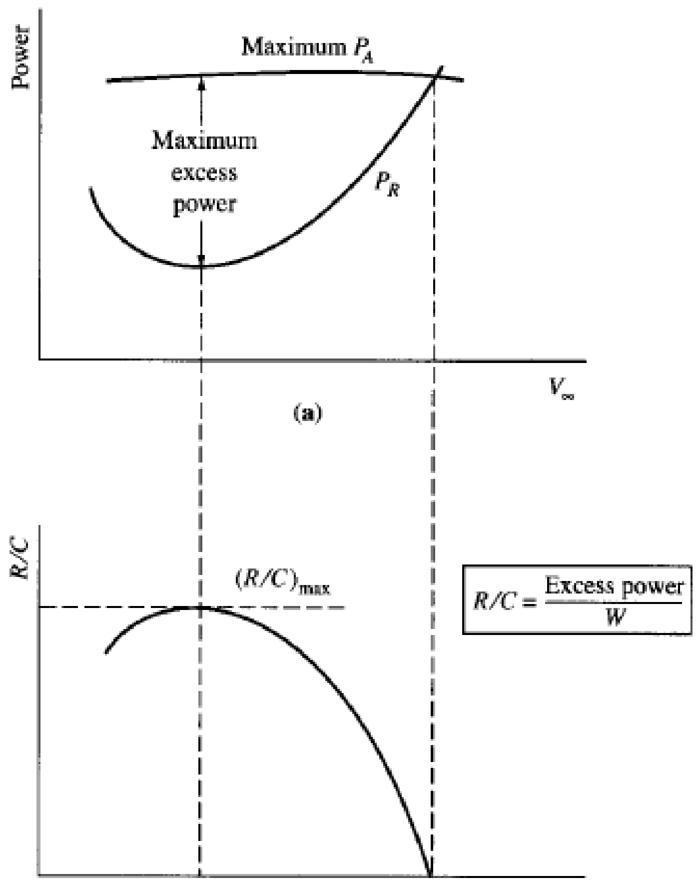


Illustration of excess power for (a) propellor-driven airplane and (b) jet-propelled airplane.



Variation of rate of climb with velocity at a given altitude.

(OR)

b) Explain briefly about

(i) Service ceiling and absolute ceiling
(ii) Time to climb and Rate of climb

Service Ceiling:

The service ceiling is the maximum usable altitude of an aircraft. Specifically, it is the density altitude at which flying in a clean configuration, at the best rate of climbairspeedfor that altitude and with all engines operating and producing maximum continuous power, will produce a given rate of climb (a typical value might be 100 feet per minute climb or 30 metres per minute, or on the order of 500 feet per minute climb for jet aircraft). Margin to stall at service ceiling is 1.5 g.

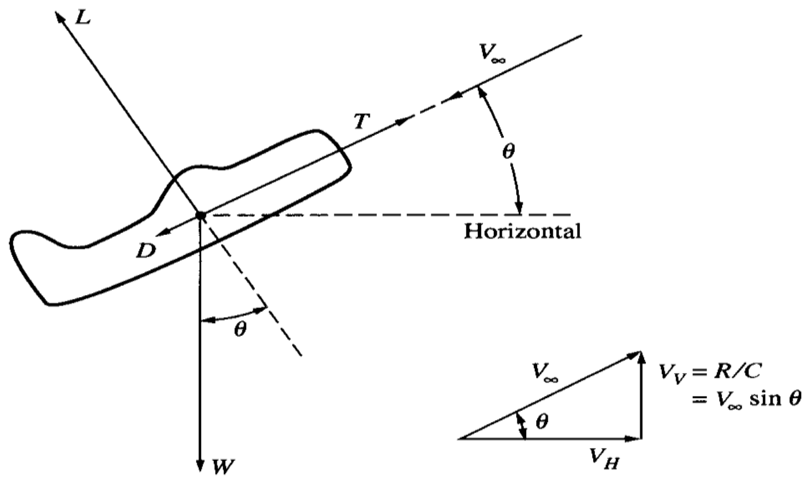
Absolute ceiling

The **absolute ceiling**, also known as coffin corner, is the highest altitude at which an aircraft can sustain level flight, which means the altitude at which the thrust of the engines at full power is equal to the total drag at minimum drag speed. In other words, it is the altitude where maximum thrust available equals minimum thrust required, so the altitude where the maximum sustained (with no decreasing airspeed) rate of climb is zero. Most commercial jetliners have a service (or certificated) ceiling of about 42,000 feet (12,802 m)[*citation needed*] and some business jets about 51,000 feet (15,545 m). While these aircraft's absolute ceiling is much higher than standard operational purposes, it is impossible to reach (because of the vertical speed asymptotically approaching zero) without afterburners or other devices temporarily increasing thrust. Flight at the absolute ceiling is also not economically advantageous due to the low indicated airspeed which can be sustained: although the true airspeed (TAS) at an altitude is typically greater than indicated airspeed (IAS), the difference is not enough to compensate for the fact that IAS at which minimum drag is achieved is usually very low, so a flight at an absolute ceiling altitude results in a low TAS as well, and hence in a high fuel burn rate per distance traveled. The absolute ceiling varies with the air temperature and, overall, the aircraft weight (usually calculated at MTOW).

RATE OF CLIMB

Consider a steady unaccelerated climb. The equations of motion for this condition is given by

$$T\cos\epsilon - D - W\sin\theta = 0$$
$$L + T\sin\epsilon - W\cos\theta = 0$$



Force and velocity diagrams for climbing flight.

Here the vertical component gives the rate of climb.

the rate of climb by R/C. From this diagram,

$$\boxed{R/C = V_\infty \sin\theta}$$

On multiplying the equation by V_∞/W we have

$$\boxed{V_\infty \sin\theta = R/C = \frac{T V_\infty - D V_\infty}{W}}$$

Thus power available is the power required to overcome this drag

$$\boxed{T V_\infty - D V_\infty \equiv \text{excess power}}$$

Hence,

$$\boxed{R/C = \frac{\text{excess power}}{W}}$$

TIME TO CLIMB

The rate of climb, by definition, is the vertical component of the airplane's velocity, which is simply the time rate of change of altitude dh/dt. Hence,

$$\frac{dh}{dt} = R/C$$

or

$$dt = \frac{dh}{R/C} \qquad \text{Eq.(1)}$$

In Eq. (1) , R/C is a function of altitude, and dt is the small increment in time required to climb the small height dh at a given instantaneous altitude. The time to climb from one altitude h_1 to another h_2 is obtained by integrating Eq. (1). between the two altitudes:

$$t = \int_{h_1}^{h_2} \frac{dh}{R/C} \qquad \text{Eq.(2)}$$

Normally, the performance characteristic labeled *time to climb* is considered from sea level, where $h_1 = 0$. Hence, the time to climb from sea level to any given altitude h_2 is, from Eq. (2)

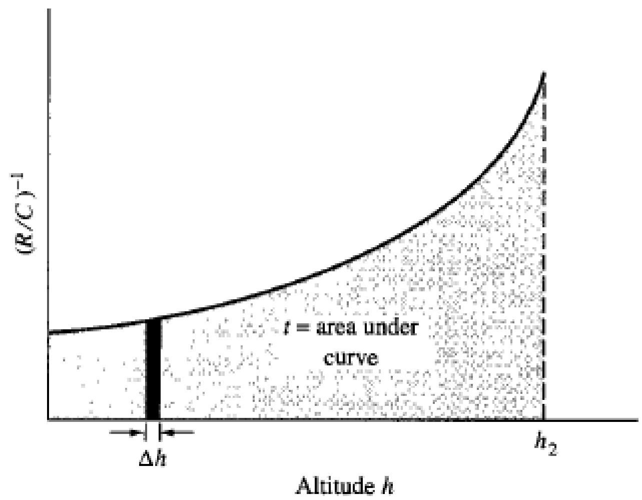
$$\boxed{t = \int_{0}^{h_2} \frac{dh}{R/C}} \qquad \text{Eq. (3)}$$

If in Eq. (3) the maximum rate of climb is used at each altitude, then t becomes the minimum time to climb to altitude h_2.

$$\boxed{t_{\min} = \int_{0}^{h_2} \frac{dh}{(R/C)_{\max}}}$$

Graphical approach

Consider a plot of $(R/C)^{-1}$ versus altitude, as shown in figure. The time to climb to altitude h_2 is simply the area under the curve, shown by the shaded area in figure.



Graphical representation of the time to climb to altitude h_2.

19. a) Derive the Breguet range equation.

By definition, *range* is the total distance (measured with respect to the ground) traversed by an airplane on one load of fuel. We denote the range by R. We also consider the following weights:

W_0—gross weight of the airplane including *everything*; full fuel load, payload, crew, structure, etc.

W_f—weight of fuel; this is an instantaneous value, and it changes as fuel is consumed during flight.

W_1—weight of the airplane when the fuel tanks are empty.

At any instant during the flight, the weight of the airplane W is

$$W = W_1 + W_f$$

While time decreases the rate of change of weight also decreases

$$V_\infty = \frac{ds}{dt}$$

$$\frac{dW}{dt} = \frac{dW_f}{dt} = \dot{W}_f \qquad ds = V_\infty\, dt$$

A general relation for the calculation of range can be obtained as follows. Consider an airplane in steady, level flight. Let s denote horizontal distance covered over the ground. Assuming a stationary atmosphere (no wind), the airplane's velocity V_∞ is

$$c_t = -\frac{dW_f/dt}{T}$$

$$dt = -\frac{dW_f}{c_t T}$$

$$ds = -\frac{V_\infty}{c_t T}\, dW_f$$

$dW_f = dW$. Equation then becomes

$$ds = -\frac{V_\infty}{c_t T}\, dW = -\frac{V_\infty}{c_t}\frac{W}{T}\frac{dW}{W}$$

In steady, level flight, $L = W$ and $T = D$.

$$ds = -\frac{V_\infty}{c_t T}\, dW = -\frac{V_\infty}{c_t}\frac{W}{T}\frac{dW}{W}$$

The range of the airplane is obtained by integrating Eq. (5.150) between $s = 0$, where the fuel tanks are full and hence $W = W_0$, and $s = R$, where the fuel tanks are empty and hence $W = W_1$.

$$R = \int_0^R ds = -\int_{W_0}^{W_1} \frac{V_\infty}{c_t} \frac{L}{D} \frac{dW}{W}$$

$$\boxed{R = \int_{W_1}^{W_0} \frac{V_\infty}{c_t} \frac{L}{D} \frac{dW}{W}}$$

OR

(OR)

b) Explain briefly what thrust reversal is and discuss any one method with a neat sketch.

THRUST REVERSAL

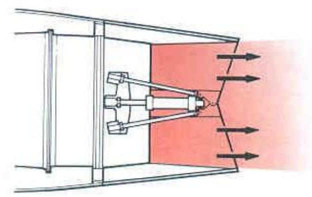
Thrust reversal, also called reverse thrust, is the temporary diversion of an aircraft engine's exhaust so that the exhaust produced is directed forward, rather than aft. This acts against the forward travel of the aircraft, providing deceleration. Thrust reverser systems are featured on many jet aircraft to help slow down just after touch-down, reducing wear on the brakes and enabling shorter landing distances. Such devices affect the aircraft significantly and are considered important for safe operation by airlines. There have been accidents involving thrust reversal systems.

Reverse thrust is also available on many propeller-driven aircraft through reversing the controllable pitch propellers to a negative angle.

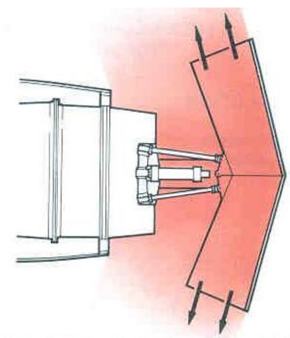
Types of thrust reversal system

1.Target type

The target thrust reverser uses a pair of hydraulically-operated 'bucket' type doors to reverse the hot gas stream. For forward thrust, these doors form the propelling nozzle of the engine. In the original implementation of this system on the Boeing 707 and still common today, two reverser buckets were hinged so when they deployed they blocked the rearward flow of the exhaust and redirected it with a forward component. This type of reverser is visible at the rear of the engine during deployment.



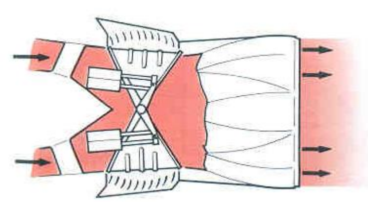
ACTUATOR EXTENDED AND BUCKET DOORS IN FORWARD THRUST POSITION



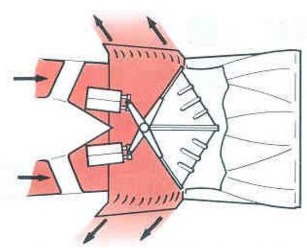
ACTUATOR AND BUCKET DOORS IN REVERSE THRUST POSITION

2. Clam-shell type

The clam-shell door, or cascade, system is pneumatically-operated. When activated, the doors rotate to open the ducts and close the normal exit, causing the thrust to be directed forward. The cascade thrust reverser is commonly used on turbofan engines. On turbojet engines, this system would be less effective than the target system, as the cascade system only makes use of the turbine airflow and does not affect the main engine core, which continues to produce thrust.



CLAMSHELL DOORS IN FORWARD THRUST POSITION

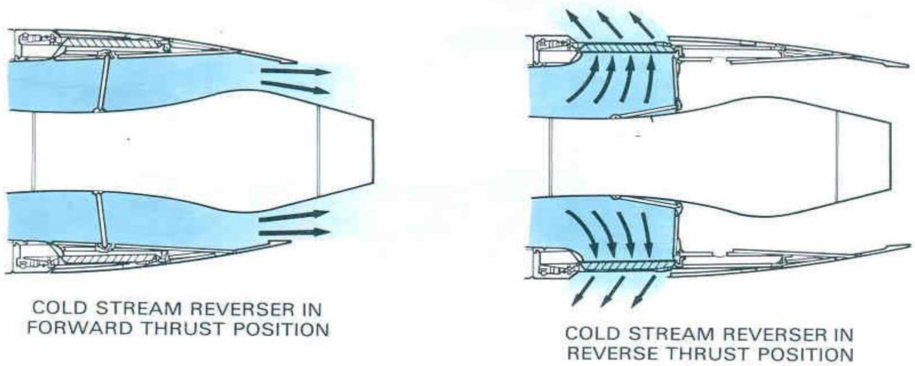


CLAMSHELL DOORS IN REVERSE THRUST POSITION

3. Cold stream type

In addition to the two types used on turbojet and low-bypass turbofan engines, a third type of thrust reverser is found on some high-bypass turbofan engines. Doors in the bypass duct are used to redirect the air that is accelerated by the engine's fan section but does not pass through the combustion chamber (called bypass air) such that it provides reverse thrust. The cold stream reverser system is activated by an air motor. During normal operation, the reverse thrust vanes are blocked. On selection, the system folds the doors to block off the cold stream final nozzle and redirect this airflow to the cascade vanes. This system can redirect both the exhaust flow of the fan and of the core.

The cold stream system is known for structural integrity, reliability, and versatility. During thrust reverser activation, a sleeve mounted around the perimeter of the engine nacelle moves aft to expose cascade vanes which act to redirect the engine fan flow. This thrust reverser system can be heavy and difficult to integrate into nacelles housing large engines.



20. a) Derive the Froude momentum theory.

Momentum theory

Mathematical model of an ideal propeller or helicopter rotor can be described by The Momentum theory or Disk actuator theory by W.J.M.Rankine, Alfred George Greenhill and R.E. Froude.
In fluid dynamics, the momentum theory describes a mathematical model of an ideal actuator disk, such as a propeller or helicopter rotor. The rotor is modeled as an infinitely thin disc, inducing a constant velocity along the axis of rotation. The basic state of a helicopter is hovering. This disc creates a flow around the rotor. Under certain mathematical premises of the fluid, there can be extracted a mathematical connection between power, radius of the rotor, torque and induced velocity. Friction is not included.

Assumptions made in the Froude momentum theory

1. Infinitely thin disc of area A which offers no resistance to air passing through it.
2. Purely 1-D analysis
3. Thrust loading and velocity are uniform over the disk.
4. Far upstream and far downstream the pressure is free stream static pressure.
5. Viscous effects are not considered (no drag, no momentum diffusion)

For a stationary rotor, such as a helicopter in hover, the power required to produce a given thrust is:

$$P = \sqrt{\frac{T^3}{2\rho A}}$$

Where:

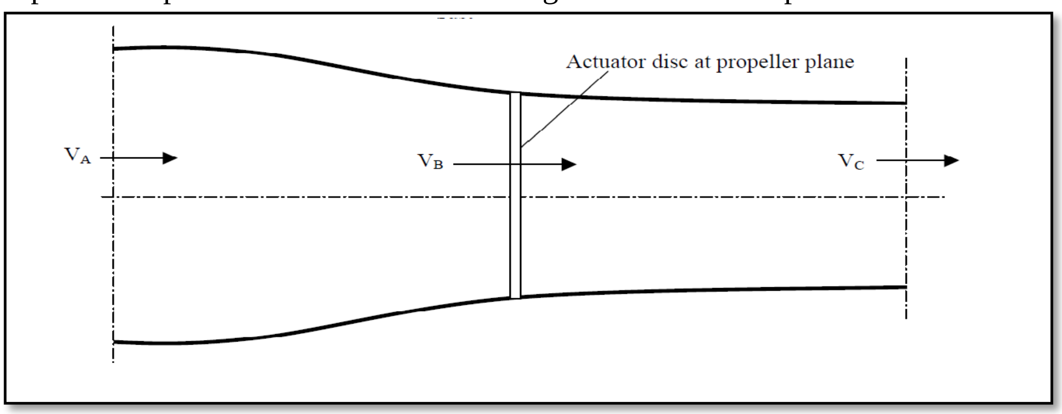
- T is the thrust

- ρ is the density of air (or other medium)
- A is the area of the rotor disc

A device which converts the translational energy of the fluid into rotational energy of the axis or vice versa is called a Rankine disk actuator.

It was originally intended to provide an analytical means for evaluating ship propellers. Momentum Theory is also well known as Disk Actuator Theory. Momentum Theory assumes that

• The flow is inviscid and steady (ideal flow), therefore the propeller does not experience energy losses due to frictional drag.
• Also the rotor is thought of as an actuator disk with an infinite number of blades, each with an infinite aspect ratio.
• The propeller can produce thrust without causing rotation in the slipstream.



Rankine disk actuator

Here the rotor is assumed as an infinitely thin disc, which induces a constant velocity along the axis of rotation.
From the basic thrust equation, we know that the amount of thrust depends on the mass flow rate through the propeller and the velocity change through the propulsion system. In the above figure the flow is proceeding from left to right. Let us denote the subscripts "A and C" for the stations assumed to be far upstream and downstream of the propeller respectively and the location of the actuator disc by the subscript "B". The thrust (T) is equal to the mass flow rate (m) times the difference in velocity (V).

$$T = m(V_C - V_A)$$

There is no pressure-area term because the pressure at the C is equal to the pressure at A.

- **The power P_D absorbed by the propeller is given by:**

$$P_D = \frac{1}{2}\dot{m}(V_C^2 - V_A^2)$$

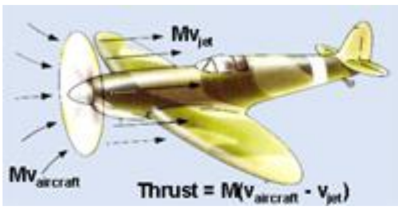
- **Momentum theory thrust is given by**,

$T = \frac{\pi}{4}D^2(v+\frac{\Delta v}{2})\rho\Delta v$

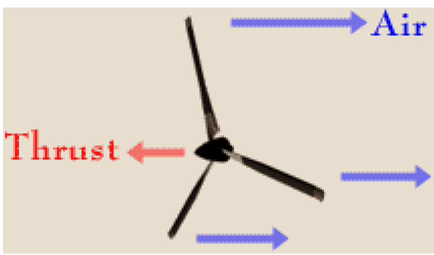
(OR)

b) i) With a help of a neat sketch explain about the geometry of the propeller blade.

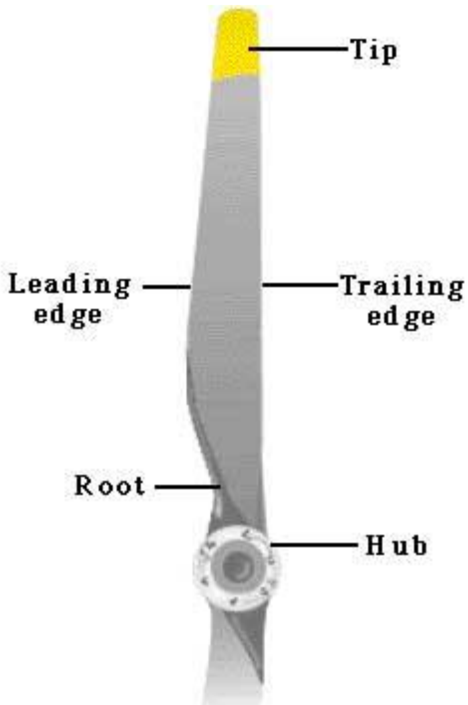
Thrust is the force that move the aircraft through the air.Thrust is generated by the propulsion system of the aircraft. There are different types of propulsion systems develop thrust in different ways, although it usually generated through some application of Newton's Third Law. Propeller is one of the propulsion system. The purpose of the propeller is to move the aircraft through the air. The propeller consist of two or more blades connected together by a hub. The hub serves to attach the blades to the engine shaft. .



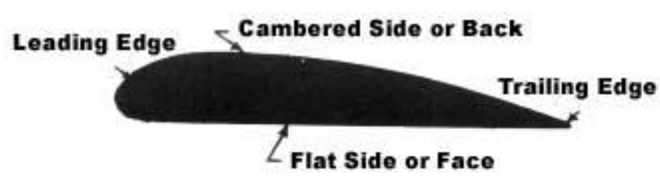
The blades are made in the shape of an airfoil like wing of an aircraft. When the engine rotates the propeller blades, the blades produce lift. This lift is called **thrust** and moves the aircraft forward. most aircraft have propellers that pull the aircraft through the air. These are called **tractor** propellers. Some aircraft have propellers that **push** the aircraft. These are called **pusher** propellers.



Leading Edge of the airfoil is the cutting edge that slices into the air. As the leading edge cuts the air, air flows over the blade face and the cambe side.

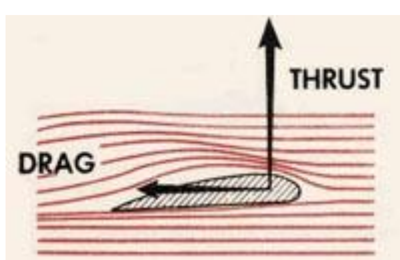


Blade Face is the surface of the propeller blade that corresponds to the lower surface of an airfoil or flat side, we called Blade Face.



Cross section of a propeller blade.

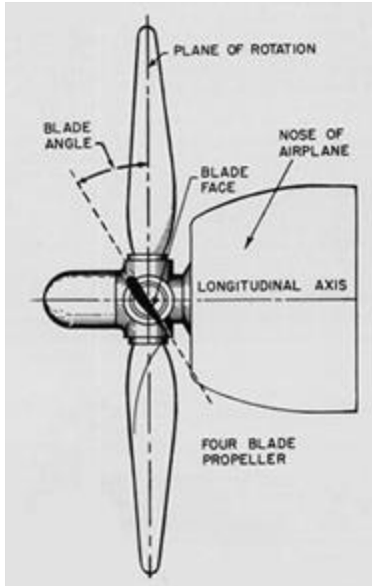
Blade Back / Thrust Face is the curved surface of the airfoil.



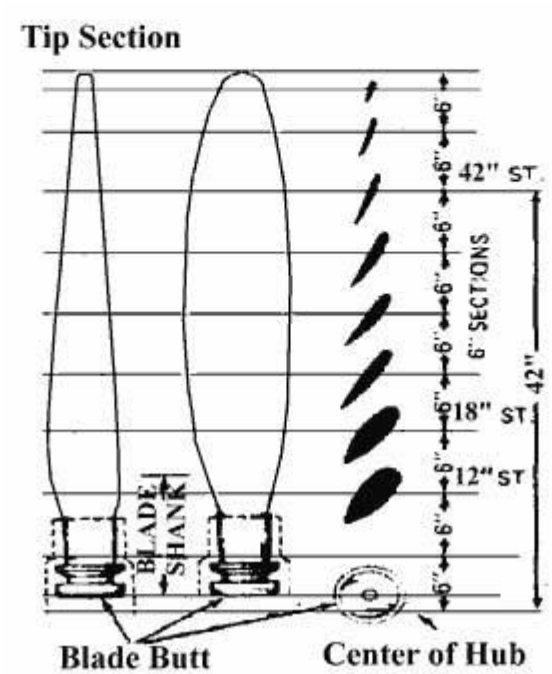
Blade Shank (Root) is the section of the blade nearest the hub.

Blade Tip is the outer end of the blade fartest from the hub.

Plane of Rotation is an imaginary plane perpendicular to the shaft. It is the plane that contains the circle in which the blades rotate.



Blade Angle is formed between the face of an element and the plane of rotation. The blade angle throughout the length of the blade is not the same. The reason for placing the blade element sections at different angles is because the various sections of the blade travel at different speed. Each element must be designed as part of the blade to operate at its own best angle of attack to create thrust when revolving at its best design speed

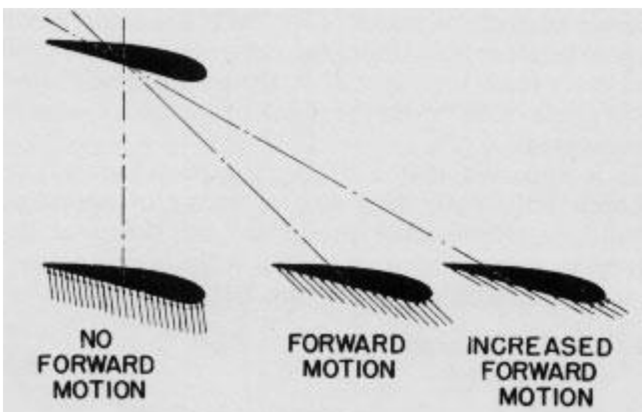


Blade Element are the airfoil sections joined side by side to form the blade airfoil. These elements are placed at different angles in rotation of the plane of rotation.

The reason for placing the blade element sections at different angles is because the various sections of the blade travel at different speeds. The inner part of the blade section travels slower than the outer part near the tip of the blade. If all the elements along a blade is at the same blade angle, the relative wind will not strike the elements at the same angle of attack. This is because of the different in velocity of the blade element due to distance from the center of rotation.

The blade has a small twist (due to different angle in each section) in it for a very important reason. When the propeller is spinning round, each section of the blade travel at different speed, The twist in the propeller blade means that each section advance forward at the same rate so stopping the propeller from bending.

Thrust is produced by the propeller attached to the engine driveshaft. While the propeller is rotating in flight, each section of the blade has a motion that combines the forward motion of the aircraft with circular movement of the propeller. The slower the speed, the steeper the angle of attack must be to generate lift. Therefore, the shape of the propeller's airfoil (cross section) must chang from the center to the tips. The changing shape of the airfoil (cross section) across the blade results in the twisting shape of the propeller.

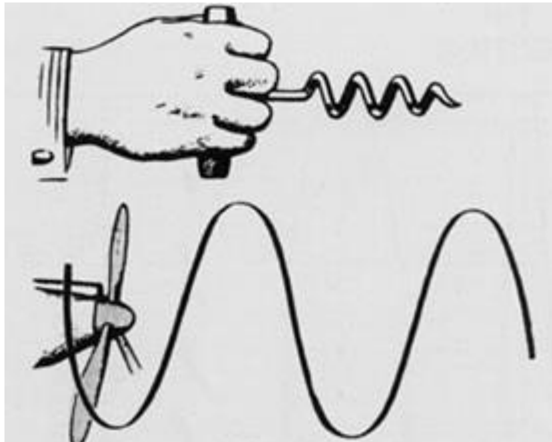


Relative wind with respect to propeller blade

Relative Wind is the air that strikes and pass over the airfoil as the airfoil is driven through the air.

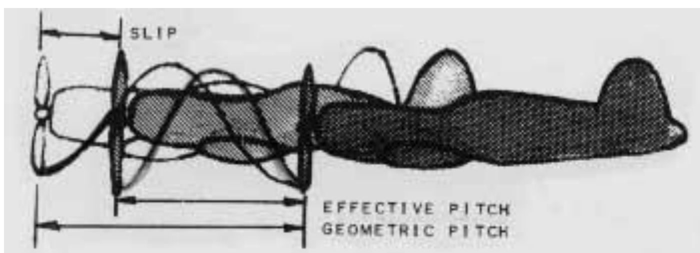
Angle of Attack is the angle between the chord of the element and the relative wind. The best efficiency of the propeller is obtained at an angle of attack around 2 to 4 degrees.

Blade Path is the path of the direction of the blade element moves.


Propeller Path

Pitch refers to the distance a spiral threaded object moves forward in one revolution. As a wood screw moves forward when turned in wood, same with the propeller move forward when turn in the air.

Geometric Pitch is the theoritical distance a propeller would advance in one revolution.



Geometric and Effective Pitch

Effective Pitch is the actual distance a propeller advances in one revolution in the air. The effective pitch is always shorter than geometric pitch due to the air is a fluid and always slip.

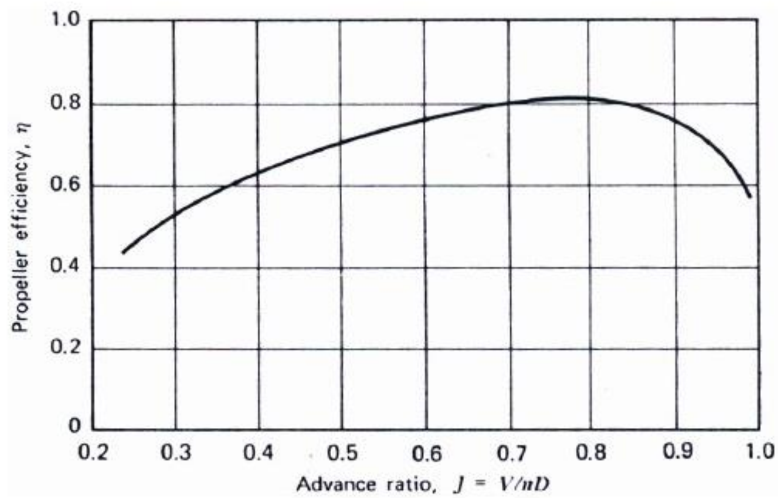
ii) Explain briefly the uses of propeller charts.

.

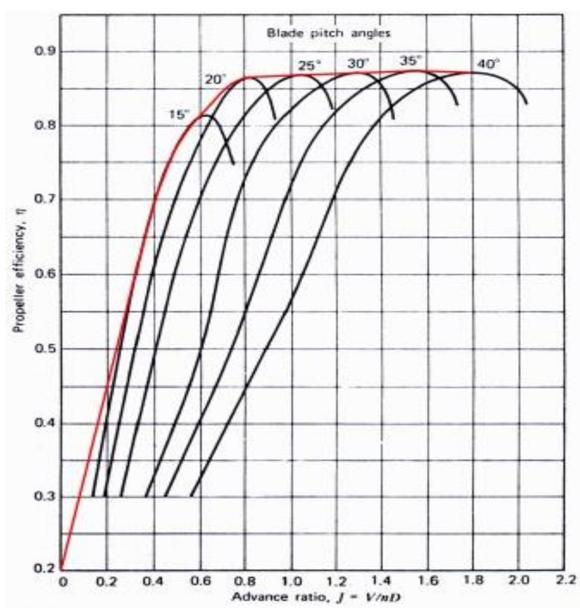
For a variable pitch propeller, the device called “propeller governor” changes thepropeller pitch to a higher blade angle, as the forward velocity of the aircraftincreases. Therefore, maximum efficiency is obtained for a wide range of forwardvelocities from take-off to cruise. In case of fixed pitch propellers, they aredesigned to provide optimum efficiency for only one flight phase, either climb orcruise, thus take-off performance is poor with the fixed pitch propellers.

The propeller is a twisted airfoil that converts the rotating power of the engine into thrust, which propels the airplane through the air. Sections of the propeller near the center are moving at a slower rate of speed than those near the tip, which is why the blades are twisted.

For a propeller driven aircraft, thrust is produced by a propeller converting theshaft torque into propulsive force, and depends on the propeller efficiency.However, propeller efficiency depends on the propeller angle of attack,consequently on the advance ratio given bywhere V is the forward velocity of the aircraft, n is the rotational speed and D is the diameter of the propeller. Thus, for a constant RPM, propeller efficiency depends on the forward velocity of the aircraft as shown in Figure.



Efficiency versus advance ratio for a fixed pitch propeller



Efficiency of a variable pitch propeller

If the propeller is a fixed pitch propeller, for a constant RPM, there is only oneforward velocity where the efficiency reaches to a maximum. Consider thedrawing given in Figure given, where the forward velocity, blade angle, angle ofattack, and rotational velocity relations are shown. If the blade angle is fixed,hence the propeller is fixed pitch; angle of attack will decrease as the forwardvelocity of aircraft increases. Although this will result in an efficiency increaseinitially, further velocity increase will bring the angle of attack to zero, and thepropeller will not be able to generate thrust. In order to avoid this, variable pitch or constant speed propellers are used.

Aircraft Performance – 11BEAR502

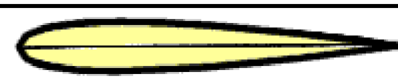
End Semester ExaminationNovember 2013

Answer Key

Part A

1. Streamlining, in aerodynamics, the contouring of an object, such as an aircraft body, to reduce its drag, or resistance to motion through a stream of air.

2.

	Cambered Airfoil	Symmetrical Airfoil
1	The natural shape of aCamberedairfoil turns the flow even at 0 degree angle of attack producinglift.	In a symmetric airfoil, the air flows evenly across the top and bottom, so no lift is generated at 0 degrees angle of attack.
2	Aircraft can fly with the wings level and generate lift.	Aircraft has to pitch to generate lift.
3		

3. Lift-induced drag is a drag force that occurs whenever a moving object redirects the airflow coming at it. This drag force occurs in airplanes due to wings or a lifting body redirecting air to cause lift.

4. The Flight Condition where L=W and D=T is called as Level Flight.

5. Power required is Given by $P_{req} = T_{req} V$

Thrust required is given by $T_{req} = D$

$D = \frac{1}{2}\rho v^2 S C_d$ Since drag force is related to density and density varies with altitude, the power

required for level flight varies with altitude.

6. Starting speed is given by $V_s = (2W / \rho S C_{Lmax})^{1/2}$

Where W = Weight

ρ = Density

C_L = Maximum Coefficient of Lift

S = Wing Planform

7. The absolute ceiling is the highest altitude at which an airplane can sustain level flight, which means the altitude at which the thrust of the engines at full power is equal to the total drag at minimum drag speed.

8. Load Factor is the ratio of lift to the weight. It is denoted by n.

$$n = L/W$$

9. The rate of climb (R/C) is an aircraft's vertical speed i.e the rate of change in altitude. This is expressed in metre per second.

10. A process by which the thrust produced by a jet-propulsion engine may be increased temporarily over its normal value by some secondary means (as the burning of additional fuel in the tail pipe, or the injection of water into the engine inlet and the combustion chambers) which increases the mass flow, the velocity, or both is called as Thrust Augmentation

11. A vertical take-off and landing (VTOL) aircraft is one that can hover, take off, and land vertically.

A short takeoff and landing (STOL) aircraft is an aircraft with short runway requirements for

takeoff and landing.

12. Condition for maximum range of propeller driven aircraft:

- Fly at maximum L/D
- Have highest possible propeller efficiency
- Have lowest possible specific fuel consumption
- Have the highest possible ratio between gross weight and empty weight.

13. The advance ratio is the ratio between the distance the propeller moves forward through the fluid during one revolution and the diameter of the propeller.

14. Factors affecting the efficiency of a propeller are

- o Speed of the propeller rotation
- o Blade Pitch
- o Pitch Angle
- o Location

15. Blade pitch or simply pitch refers to turning the angle of attack of the blades of a propeller or

helicopter rotor into or out of the wind to control the production or absorption of power.

Part B

16. a

Drag

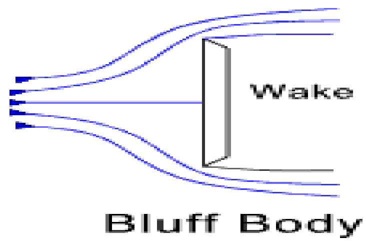
Aerodynamic drag is the fluid drag force that acts on any moving solid body in the direction of the fluid freestream flow.From the body's perspective (near-field approach), the drag comes from forces due to pressure distributions over the body surface.The pressure distribution over the body surface exerts normal forces which, summed and projected into the freestream direction, represent the drag force due to pressure.

Drag Reduction Methods:

Streamlining the Surface:

Streamlining, in aerodynamics, the contouring of an object, such as an aircraft body, to reduce its drag, or resistance to motion through a stream of air.

A moving body causes the air to flow around it in definite patterns, the components of which are called streamlines. Smooth, regular airflow patterns around an object are called laminar flow; they denote a minimum of disturbance of the air by the object's motion through it. Turbulent flow occurs when air is disturbed and separates from the surface of the moving body, with the consequent formation of a zone of swirling eddies in the body's wake. This eddy formation represents a reduction in the downstream pressure on the moving object and is a principal source of drag. Streamlining, then, is the contouring of an aircraft or other body in such a way that its turbulent wake is reduced to a minimum. The mechanics of airflow patterns lead to two principles for subsonic streamlining: (1) the forward part of the object should be well rounded, and (2) the body should gradually curve back from the midsection to a tapering rear section. An efficiently streamlined body thus takes on the look of a horizontally inclined teardrop shape.



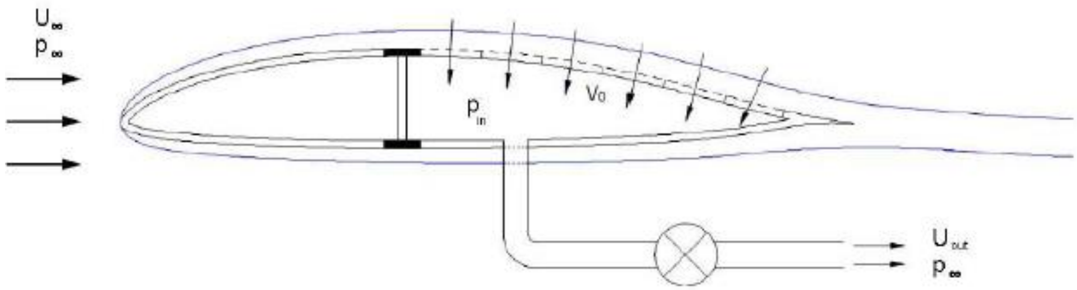
An aircraft or other body that is traveling at supersonic speeds requires a different streamlined form from that of a subsonic aircraft because it is moving faster than the speed at which the pressure impulses it creates are propagated in air. Because the pressure waves can no longer be transmitted ahead of an aircraft moving at supersonic speed, they pile up in front of it, creating a compression, or shock, wave. Further shock waves are created at the midsection and tail of the supersonic aircraft. The strength of these shock waves is dependent on the magnitude of the change in the air's direction, which in turn is dependent on the sharpness or angle of the forward tip and other surfaces of the aircraft's body. Supersonic aircraft thus have sharply pointed noses and tails and straight, narrow bodies to minimize the intensity of the shock waves.

Boundary layer suction:

Boundary layer suction is technique in which an air pump is used to extract the boundary layer at the wing or the inlet of an aircraft. Improving the air flow can reduce drag. Improvements in fuel efficiency have been estimated as high as 30%.The air molecules at the surface of a wing are effectively stationary. If the flow is smooth, known as laminar flow, the velocity of the air increases steadily as measurements are taken further away from the surface. However the smooth flow is often disturbed by the boundary layer breaking away from the surface and creating a low pressure region immediately behind the airfoil. This low pressure region results in increased overall drag. Attempts have been made over the years to delay the onset of this flow separation by careful design and smooth surfaces.

There are two reasons to apply boundarylayer suction; one is to postpone separation, the other reason is to postpone transition.To postpone separation, a part of the turbulent boundary layer is sucked away, which willprevent the growth of the boundary layer and keeping it attached, therefore preventing
Separation. In doing so, it is possible to fly with higher angles of attack and lowervelocities.
To postpone transition of the boundary layer from a laminar one to a turbulent one andavoid it to separate, laminar boundary layer suction can be used. A portion of the laminarboundary layer will be removed which stabilizes the boundary layer. This is because thegrowth of instabilities in

the laminar boundary layer will decrease, i.e. the Tollmien-Schlichting waves will be damped. Because this results in larger areas of laminar flow,the profile drag will be reduced.

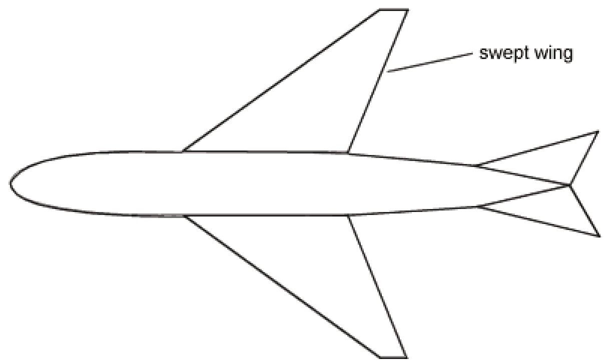


The reason that a laminar boundary layer gives lower profile drag compared to a turbulentone is twofold; the friction drag as well as the pressure drag is reduced.

Swept back wings:

A swept wing is a wing planform favored for high subsonic and supersonic speeds, and is found on almost all jet aircraft in one form or another, as well as some high speed propeller aircraft. Compared with straight wings common to slower aircraft, they have a "swept" wing root to wingtip direction angled beyond (usually aftward) the spanwise axis. This has the effect of delaying the drag rise caused by fluid compressibility near the speed of sound, increasing performance.The characteristic sweep angle is normally measured by drawing a line from root to tip, 25% of the way back from the leading edge, and comparing that to the longitudinal axis of the aircraft. Typical sweep angles vary from 0 for a straight-wing aircraft, to 45 degrees or more for fighters and other high-speed designs.

As an aircraft enters the transonic speeds just below the speed of sound, an effect known as wave drag starts to appear. As airflow accelerates around curved surfaces, near the speed of sound this can cause the airflow to reach supersonic speeds. When this occurs, an oblique shock wave is generated at the point where the flow slows back to subsonic speed. Since this occurs on curved areas, these shock waves are normally associated with the upper surfaces of the wing, the cockpit canopy, and the nose cone of the aircraft, areas with the highest local curvature.



Shock waves require energy to form. This energy is taken out of the aircraft, which has to supply extra thrust to make up for this energy loss. Thus the shocks are seen as a form of drag.

Airflow at supersonic speeds generates lift through the formation of shock waves, as opposed to the patterns of airflow over and under the wing. These shock waves, as in the transonic case, generate large amounts of drag. One of these shock waves is created by the leading edge of the wing, but contributes little to the lift. In order to minimize the strength of this shock it needs to remain "attached" to the front of the wing, which demands a very sharp leading edge. To better shape the shocks that will contribute to lift, the rest of an ideal supersonic airfoil is roughly diamond-shaped in cross-section. For low-speed lift these same airfoils are very inefficient, leading to poor handling and very high landing speeds.To avoid the need for a dedicated supersonic wing is to use a highly swept subsonic design. Airflow behind the shock waves of a moving body are reduced to subsonic speeds. This effect is used within the intakes of engines meant to operate in the supersonic, as jet engines are generally incapable of ingesting supersonic air directly.

16.b

Given:

α	$= 6^o$
Span Efficiency Factor	= 0.95
a_o	= 0.105/degree
$\alpha_{L=0}$	$= -2.2^o$
C_d	= 0.0076

Solution

$$a_o = dC_L/d\alpha$$

$C_L = \frac{l}{\frac{1}{2}\rho v_\infty s}$

$0.105 = \frac{dcl\,[cl-0]}{6+2.2}$

$dC_L = 0.861$ $\longrightarrow$ Coefficient of Lift

$\frac{CL}{CD} = \frac{0.861}{0.0076} = 113.289$

17. a Thrust Required

Thrust required is dependent on Velocity, Altitude, Aerodynamic shape and Size. It is simply equal to the Drag on the Airplane.

Derivation

For a Steady level flight we have $T_R = D = \frac{D}{W}W = \frac{D}{L}W$

or $T_R = \frac{W}{L/D}$

The lift to drag ratio for an airplane is given as $\frac{L}{D} = \frac{\frac{1}{2}\rho_\infty V_\infty^2 S C_L}{\frac{1}{2}\rho_\infty V_\infty^2 S C_D} = \frac{C_L}{C_D}$

The Drag is a function of altitude, density and weight and it can be expressed as $D = f(h, V_\infty, W)$

From the drag polar equation we have, $D = q_\infty S C_D = q_\infty S(C_{D,0} + K C_L^2)$

This can also be written as $L = W = q_\infty S C_L = \frac{1}{2}\rho_\infty V_\infty^2 S C_L$

The Coefficient of lift is given by $C_L = \frac{2W}{\rho_\infty V_\infty^2 S}$

Substituting the value of C_L we get , $D = \frac{1}{2}\rho_\infty V_\infty^2 S\left[C_{D,0} + 4K\left(\frac{W}{\rho_\infty V_\infty^2 S}\right)^2\right]$

Which can also be expressed as $D = \frac{1}{2}\rho_\infty V_\infty^2 S C_{D,0} + \frac{2KS}{\rho_\infty V_\infty^2}\left(\frac{W}{S}\right)^2$

Expressing this equation in terms of dynamic pressure we have $T_R = q_\infty S C_{D,0} + \frac{KS}{q_\infty}\left(\frac{W}{S}\right)^2$

Rearranging this equation we get $q_\infty^2 S C_{D,0} - q_\infty T_R + KS\left(\frac{W}{S}\right)^2 = 0$

Since this equation is in the form of a quadratic equation we get two roots as

$$q_\infty = \frac{T_R \pm \sqrt{T_R^2 - 4SC_{D,0}K(W/S)^2}}{2SC_{D,0}}$$
$$= \frac{T_R/S \pm \sqrt{(T_R/S)^2 - 4C_{D,0}K(W/S)^2}}{2C_{D,0}}$$

Substituting the value of q_∞ we get $V_\infty^2 = \frac{T_R/S \pm \sqrt{(T_R/S)^2 - 4C_{D,0}K(W/S)^2}}{\rho_\infty C_{D,0}}$

From this we have the expression for velocity as

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

17. b

Given

Wing Loading = 2400 N/m^2

Drag Polar **= 0.0016 + 0.055C^2_L**

Solution :

(i) **Maximum L/D ratio =**
C_{D0} = 0.016 + 0.055 C^2_L

$$\frac{L}{D}max = \sqrt{\frac{1}{4 \text{ x} C_{D0} \text{x } K}}$$

$$\sqrt{\frac{1}{4 \text{ x} 0.016 \text{x } 0.055}}$$

$$\frac{L}{D}max = 16.85$$

ii) Minimum Drag Speed

$$V_{md} = \left[\frac{K}{Cdo}\right]^{1/4}\sqrt{\frac{2w}{\rho s}}$$

$$V_{md} = \left[\frac{0.055}{0.016}\right]^{1/4}\sqrt{\frac{2 \text{ x } 2400}{1.225}}$$

= 1.361 x 62.596

V_{md} = 85.1943 m/s

iii) L/D Ratio at a speed of 100 m/sec:

$$\frac{L}{D} = \left[\frac{\rho_\infty V_\infty C_{D0}}{2w/s} + \frac{2k}{\rho_\infty {V_\infty}^2}\frac{W}{S}\right]^{-1}$$

$$\frac{L}{D} = \left[\frac{1.225 \text{ x } 100 \text{ x } 0.016}{2 \text{ x}2400} + \frac{2 \text{ x } 0.055}{1.225 \text{ x}100^2} \text{ x } 2400\right]^{-1}$$

= 45.64

18 a .

Given

Wing Loading = W/S = 300 kg/m^2

Planform Area = 35m^2

$C_{D1o} =$ 0.018

K = 0.055

Maximum Rate of Climb

$$V_{\max\frac{R}{C}} = \frac{1}{2}\rho V^2 S\, C_{D10} \quad + \frac{KW^2}{\frac{1}{2}\,\rho r^2 S}\left[1 - \left(\frac{Vc}{V}\right)^2\right] + \frac{WV_c}{V}$$

$$= \frac{1}{2}\text{x}0.735 \text{ x}2000^2\text{x}35 \text{ x } 0.018 \quad + \frac{0.055 \text{ x}10500^2}{\frac{1}{2}\ 0.735 \text{ x}2000^2 35}\left[1 - \left(\frac{Vc}{V}\right)^2\right] + \frac{10500V_c}{V}$$

100000 = 926100 + (0.11 – 0.11x $\left(\frac{Vc}{V}\right)^2$) x+ 10500$\frac{V_c}{V}$

$$73899 = -\ 0.11\ \left(\left(\frac{Vc}{V}\right)^2\right) + 10500\frac{V_c}{V}$$

$$\frac{V_c}{V} = 9.546 \text{ x } 10^4,\ 0.7037$$

$$\frac{V_c}{V} = 0.7037$$

$$\frac{V_c}{V} = \sin\theta = 0.7037$$

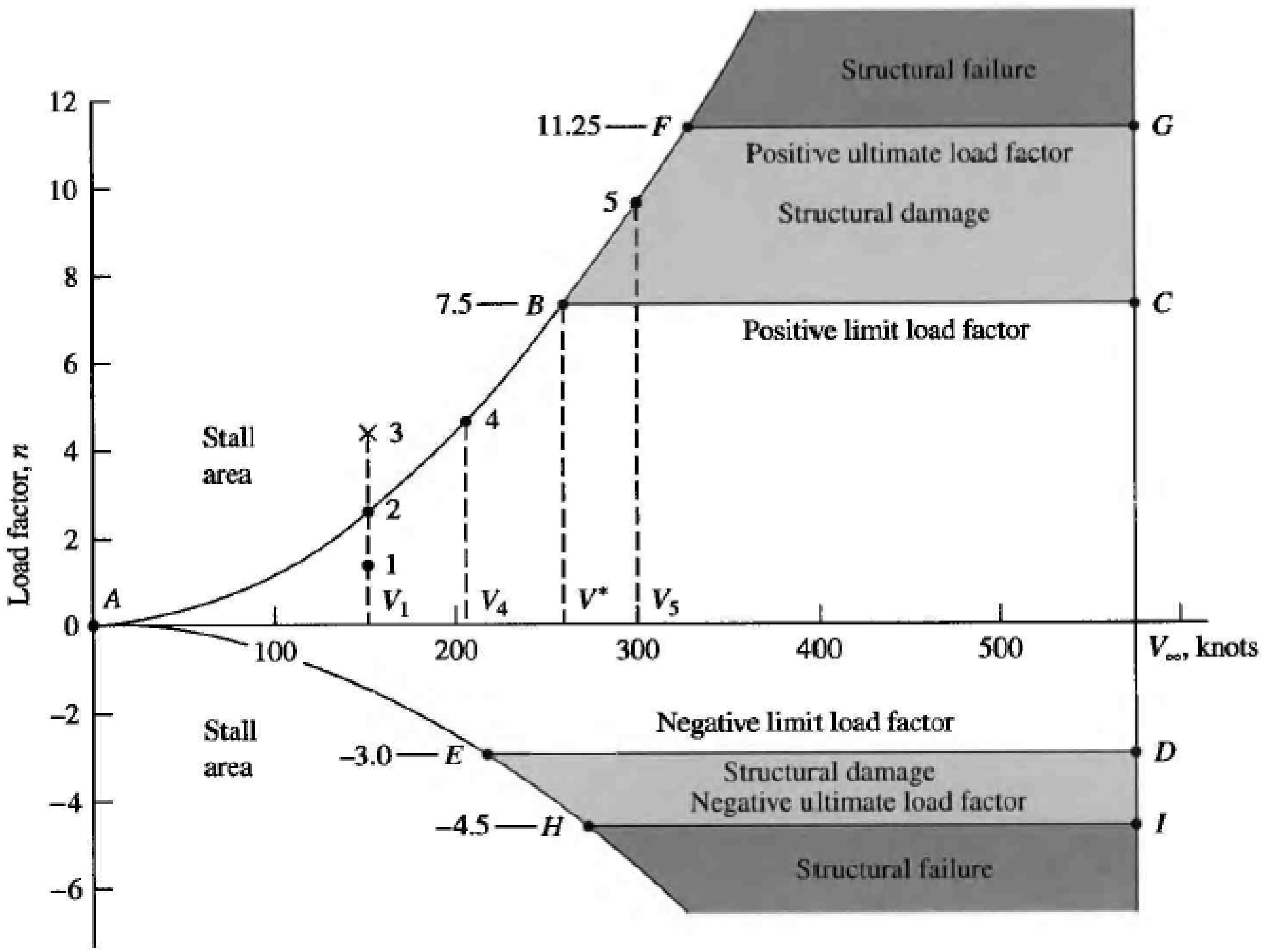
$$\theta = 44^0 6681'$$

$$V_C = 1407.4 \text{ m/s}$$

18.b

V-n Diagram

Flight regime of any aircraft includes all permissible combinations of speeds, altitudes, weights, centers of gravity, and configurations. This regime is shaped by aerodynamics, propulsion, structure, and dynamics of aircraft. The borders of this flight regime are called flight envelope or maneuvering envelope. The safety of human onboard is guaranteed by aircraft designer and manufacturer. Pilots are always trained and warned through flight instruction manual not to fly out of flight envelope, since the aircraft is not stable, or not controllable or not structurally strong enough outside the boundaries of flight envelope. A mishap or crash is expected, if an aircraft is flown outside flight envelope.



V-n Diagram

It is a diagram drawn between velocity and load factor. It is also called as the flight regime or envelope.

This envelope demonstrates the variations of airspeed versus load factor (V – n). In another word, it depicts the aircraft limit load factor as a function of airspeed. One of the primary reasons that this diagram is highly important is that, the maximum load factor; that is extracted from this graph; is a reference number in aircraft structural design. If the maximum load factor is under-calculated, the aircraft cannot withstand flight load safely. For this reason, it is recommended to structural engineers to recalculate the V-n diagram on their own as a safety factor.

The load to the aircraft on the ground is naturally produced by the gravity (i.e. 1 times g). But,there are other sources of load to the aircraft during flight; one of which is the acceleration load.This load is usually normalized through load factor (i.e. "n" times g). In another word, aircraftload is expressed as a multiple of the standard acceleration due to gravity (g = 9.81 m/sec2 =32.17 ft/sec2). Recall that we defined the load factor as the ratio between lift and weight.

$$n = \frac{L}{W}$$

The Curve between the points A and B represents the aerodynamic limit on the load factor imposed between C_{Lmax} **.**The region above the curve AB in V-n diagram is the stall region. As V is increased to the value of V_4then the maximum possible load factor n_{max}also increases .the point B is the Structural limit factor. The horizontal line BC denotes the positive limit load factor in the V-n diagram. The flight velocity corresponding to this velocity is V*.At velocities greater than V* the airplane must fly at values less than $C_{Lmax.}$. If not Structural damage or possible structural failure will occur. If the aircraft flies beyond this speed then it will experience a critical gust and destructive flutter. This is due to the high dynamic pressure that is created than the design range of the Aircraft. The corresponding velocity at point B is called as Corner velocity.

19 a

High Lift Devices:

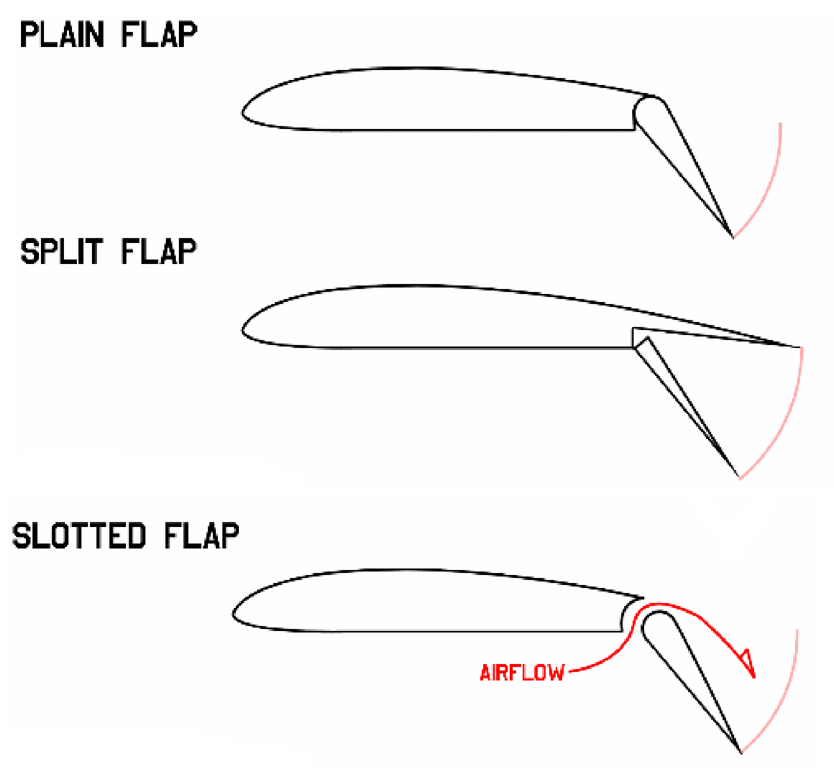
In aircraft design, high-lift devices are moving surfaces or stationary components intended to increase lift during certain flight conditions. They include common devices such as flaps and slats, as well as less common features such as leading edge extensions and blown flaps.Aircraft designs include compromises intended to maximize performance for a particular role. One of the most fundamental of these is the size of the wing; a larger wing will provide more lift and reduce takeoff and landing distance, but will increase drag during cruising flight and thereby lead to lower than optimum fuel economy. High-lift devices are used to smooth out the differences between the two goals, allowing the use of an efficient cruising wing, and adding lift for takeoff and landing.

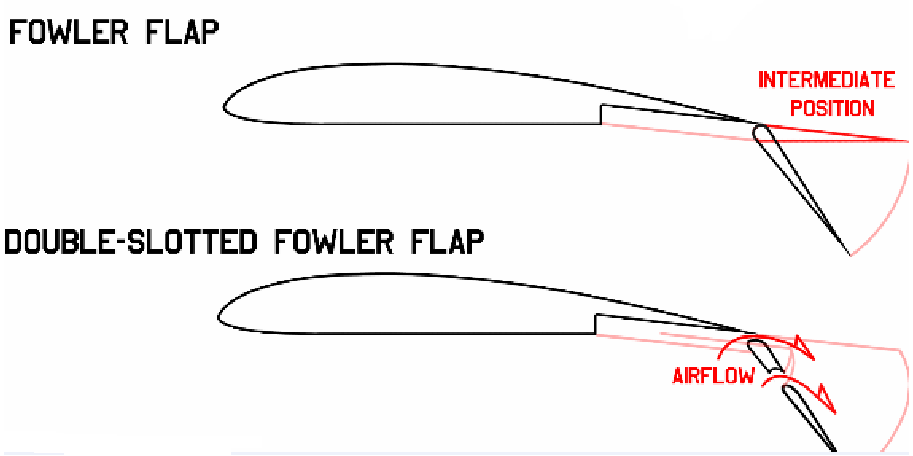
Flaps

Flaps are devices used to improve the lift characteristics of a wing and are mounted on the trailing edges of the wings of a fixed-wing aircraft to reduce the speed at which the aircraft can be safely flown and to increase the angle of descent for landing. They shorten takeoff and landing distances. Flaps do this by lowering the stall speed and increasing the drag.

Extending flaps increases the camber or curvature of the wing, raising the maximum lift coefficientor the lift a wing can generate. This allows the aircraft to generate as much lift but at a lower speed, reducing the stalling speed of the aircraft, or the minimum speed at which the aircraft will maintain flight. Extending flaps increases drag which can be beneficial during approach and landing because it slows the aircraft. On some aircraft, a useful side effect of flap deployment is a decrease in aircraft pitch angle which improves the pilot's view of the runway over the nose of the aircraft during landing. However the flaps may also cause pitch-up, depending on the type of flap and the location of the wing.

Types of Flaps





Plain flap: the rear portion of airfoil rotates downwards on a simple hinge mounted at the front of the flap.

Split flap: the rear portion of the lower surface of the airfoil hinges downwards from the leading edge of the flap, while the upper surface stays immobile.

Slotted flap: a gap between the flap and the wing forces high pressure air from below the wing over the flap helping the airflow remain attached to the flap, increasing lift compared to a split flap.

Fowler flap: split flap that slides backward flat, before hinging downward, thereby increasing first chord, then camber.

Leading-edge slats

Slats are aerodynamic surfaces on the leading edge of the wings of fixed-wing aircraft which, when deployed, allow the wing to operate at a higher angle of attack. A higher coefficient of lift is produced as a result of angle of attack and speed, so by deploying slats an aircraft can fly at slower speeds, or take off and land in shorter distances. They are usually used while landing or performing maneuvers which take the aircraft close to the stall, but are usually retracted in normal flight to minimize drag.

Types of Leading-edge slats

Automatic – the slat lies flush with the wing leading edge until reduced aerodynamic forces allow it

to extend by way of aerodynamics when needed. Sometimes referred to as Handley-Page slats.

Fixed – the slat is permanently extended. This is sometimes used on specialist low-speed aircraft
(these are referred to as slots) or when simplicity takes precedence over speed.
Powered – the slat extension can be controlled by the pilot. This is commonly used on airliners.

Operation of Leading-edge slats

The chord of the slat is typically only a few percent of the wing chord. The slats may extend over the outer third of the wing, or they may cover the entire leading edge.In reality, the slat does not give the air in the slot high velocity (it actually reduces its velocity) and also it cannot be called high-energy air since all the air outside the actual boundary layers has the same total heat. The actual effects of the slat are:

The slat effect: The velocities at the leading edge of the downstream element (main airfoil) are reduced due to the circulation of the upstream element (slat) thus reducing the pressure peaks of the downstream element.

The circulation effect: The circulation of the downstream element increases the circulation of the upstream element thus improving its aerodynamic performance.

The dumping effect: The discharge velocity at the trailing edge of the slat is increased due to the circulation of the main airfoil thus alleviating separation problems or increasing lift.

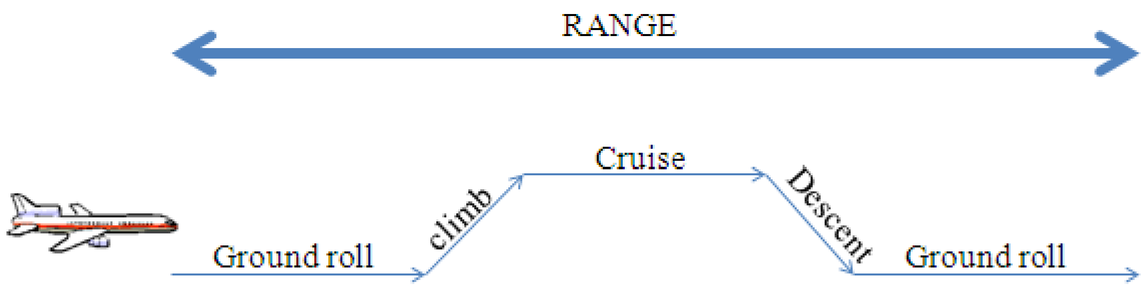
Off the surface pressure recovery: The deceleration of the slat wake occurs in an efficient manner, out of contact with a wall.

Fresh boundary layer effect: Each new element starts out with a fresh boundary layer at its leading edge. Thin boundary layers can withstand stronger adverse gradients than thick ones.

19 .b

Breguet Range Equation:

Range : It is defined as the total distance covered by an aircraft on one load of fuel



Assume ,

$$W = W_1 + W_f$$

W_0 – gross weight of the aeroplane including fuel load, payload, crew and Structure.
W_f– Weight of the fuel
W_1– Weight of airplane when fuel tanks are empty

Consider an aircraft in steady, level flight, with weight W

During the flight, Since is decreasing during the flight , W is also decreasing , then the rate of change of weight w.r.t Time(t)is

$$\frac{dW}{dt} = \frac{dW_f}{dt} = \dot{W}_f$$

Range is intimately connected with engine performance through the Specific Fuel Consumption (S.F.C) denoted by 'C' The S.F.C is defined as mass of flow used by an engine per unit energy delivered
The S.F.C for a propeller driven airplane or reciprocating engine is defined by

$$c \equiv -\frac{\dot{W}_f}{P}$$

Where,
P – Shaft Power.
Negative sign is necessary, because is negative and C is always positive

TSFC is defined as a unit of measurement of fuel efficiency for turbojet engines.

For a jet propelled airplane, the thrust specific fuel consumption is defined by

$$c_t \equiv -\frac{\dot{W}_f}{T}$$

Where,

T – thrust available

The relation between S.F.C and T.S.F.C is

$$c_t = \frac{cV_\infty}{\eta_{pr}}$$

Where, η_{pr} propeller efficiency

Now consider an airplane which is in steady level flight. Let 'S' denote horizontal distance covered over the ground. Assume that there is no wind, the airplane velocity

Then

$$c_t = -\frac{dW_f/dt}{T}$$

We Know that

$$V_\infty = \frac{ds}{dt}$$

$$dt = -\frac{dW_f}{c_t T}$$

$$ds = -\frac{V_\infty}{c_t T}\,dW_f$$

$$ds = -\frac{V_\infty}{c_t T}\,dW = -\frac{V_\infty}{c_t}\frac{W}{T}\frac{dW}{W}$$

In a steady flight $L = W$ and $T = D$

So the equation becomes $ds = -\frac{V_\infty}{c_t}\frac{L}{D}\frac{dW}{W}$

By integrating equation between S= 0 and S=R

$$R = \int_0^R ds = -\int_{W_0}^{W_1} \frac{V_\infty}{c_t}\frac{L}{D}\frac{dW}{W}$$

$$R = \int_{W_1}^{W_0} \frac{V_\infty}{c_t}\frac{L}{D}\frac{dW}{W}$$

This equation is called as the Breguet Range Equation

20 a)

Given

Weight = 446 kg

Propeller diameter = 2.16 m

Head Wind = 19.3 km/hr = $19.3 = \frac{1000}{3600}$ m/sec = 5.36 m/sec

Altitude = 578 m

L/D = 5

η_{prop} = 82%

V_j= 41.52 m/s

Power of the engine = $P_A = TV_\infty$

$T = m\,(V_j - V_\infty)$

$m = \rho_\infty A V_\infty = 1.15 \times \pi/4 \times 2.16^2 \times 5.36$

m = 22.58 kg/sec

T = 22.58 x (41.52-5.36)

T = 816.67 N

P = 816.67 x 5.36 = 4377.34 Nm/s

20 b)

Blade Element Theory

Blade element theory (BET) is a mathematical process originally designed by William Froude, David W. Taylor and Stefan Drzewiecki to determine the behavior of propellers.It involves breaking a blade down into several small parts then determining the forces on each of these small blade elements. These forces are then integrated along the entire blade and over one rotor revolution in order to obtain the forces and moments produced by the entire propeller or rotor. One of the key difficulties lies in modeling the induced velocity on the rotor disk. Because of this the blade element theory is often combined with the momentum theory to provide additional relationships necessary to describe the induced velocity on the rotor disk.

Assumptions:

• The blade is composed of aerodynamically independent, narrow strips orelements.

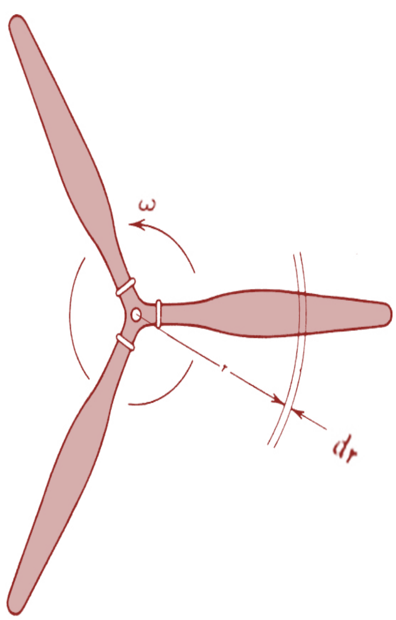
• A differential blade element of chord C and width dr, located at a radius rfrom the rotor axis is considered as an airfoil section.

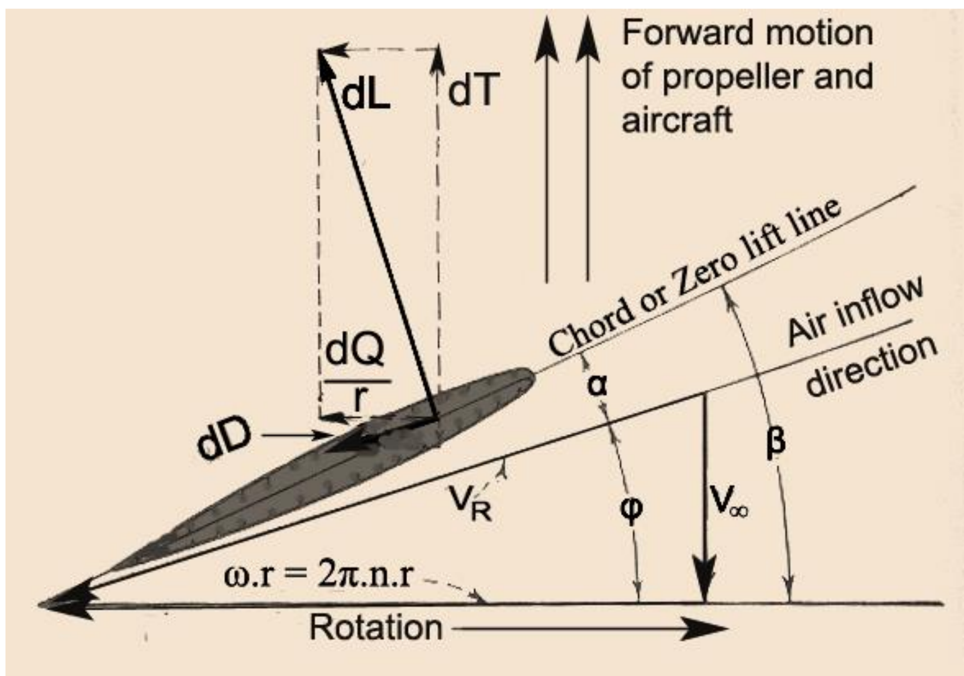
Theory :

The blade elements are assumed to be made up of airfoil shapes of known lift, Cland drag, Cdcharacteristics.

•In practice a large number of different airfoils are used to make up one propeller blade.

•Each of these elements shall have its own lift, $C_l$and drag, C_d coefficient characteristics.





The thrust, dTcreated by an element of elemental radial length dris created with contributions from the airfoil with lift, dLand drag, dD

Using the blade elemental lift and drag characteristics the working capacity of the blade element may be found as :

Thrust produced

$dT = dL.\cos\varphi - dD.\sin\varphi$

$= \frac{1}{2} .\rho .V_R^2 .c.dr. (C_l \cos\varphi - C_d \sin\varphi)$

Torque to be supplied

$dQ = (dL.\sin\varphi + dD.\cos\varphi). r$

$= \frac{1}{2} .\rho.V_R^2 .c.dr.(Cl.\sin\varphi + Cd.\cos\varphi)$

Substituting for Resultantinflow velocityIncident and aligned to the blade element

$V_R = V_\infty / \sin\varphi$,

and for Incoming flow Dynamic head based on forward velocity of the element

$q = \frac{1}{2} \rho V_\infty^2$

The elemental thrust is: $dT = \frac{q.c.dr}{\sin^2\phi}(C_l \cos\phi - C_d \sin\phi)$

The elemental torque is: $dQ = \frac{q.c.r.dr}{\sin^2\phi}\left(C_l \sin\phi + C_d \cos\phi\right)$

Propeller thrustand torqueare now computed by integrating from the root to the tip of the blade and for number of blades, B

$$T = q.B.\int_0^R \frac{c.dr}{\sin^2\phi}(C_l \cos\phi - C_d \sin\phi)$$

$$Q = q.B\int_0^R \frac{c.r.dr}{\sin^2\phi}(C_l \sin\phi + C_d \cos\phi)$$

Thus, the net thrust and the torque are seen to be directly proportional to the number of blades, B and the chord, c.

Chapter 3

Static Longitudinal Stability and Control

The most critical aspects of static longitudinal stability relate to control forces required for changing trim or performing maneuvers. Our textbook [1] treats primarily the situation when the controls are fixed. This is, of course, and idealization, even for the case of powered, irreversible controls, as the position of the control surfaces can he held fixed only to the extent of the maximum available control forces. The opposite limit – that of free control surfaces – also is an idealization, limited by the assumptions of zero friction in the control positioning mechanisms. But, just as the control fixed limit is useful in determining control *position* gradients, the control free limit is useful in determining control *force* gradients. And these latter are among the most important vehicle properties in determining handling qualities.

3.1 Control Fixed Stability

Even for the controls-fixed case, our text is a bit careless with nomenclature and equations, so we review the most important results for this case here. We have seen that for the analysis of longitudinal stability, terms involving products of the drag coefficient and either vertical displacements of the vehicle center-of-gravity or sines of the angle of attack can be neglected. Then, with the axial locations as specified in Fig. 3.1 the pitching moment about the vehicle c.g. can be written

$$\mathbf{C}_{m\,\mathrm{cg}} = \mathbf{C}_{m0_w} + \mathbf{C}_{Lw}\left(\frac{x_{\mathrm{cg}}}{\bar{c}} - \frac{x_{\mathrm{ac}}}{\bar{c}}\right) - \eta\frac{S_t}{S}\mathbf{C}_{Lt}\left[\frac{\ell_t}{\bar{c}} - \left(\frac{x_{\mathrm{cg}}}{\bar{c}} - \frac{x_{\mathrm{ac}}}{\bar{c}}\right)\right] + \mathbf{C}_{mf} \tag{3.1}$$

where we assume that $\mathbf{C}_{m0_t} = 0$, since the tail is usually symmetrical. Note that, as is the usual convention when analyzing *static* longitudinal stability and control, the positive direction of the x-axis is taken to be *aft*;[1] thus, e.g., the second term on the right-hand side of Eq. (3.1) contributes to a positive (nose-up) pitching moment for positive lift when the c.g. is aft of the wing aerodynamic center.

[1] Also, the origin of the x-axis is taken, by convention, to be at the leading edge of the mean aerodynamic chord of the wing, and distances are normalized by the length of the wing mean aerodynamic chord. Thus, for example, we might specify the location of the vehicle center-of-gravity as being at *30 per cent m.a.c.*

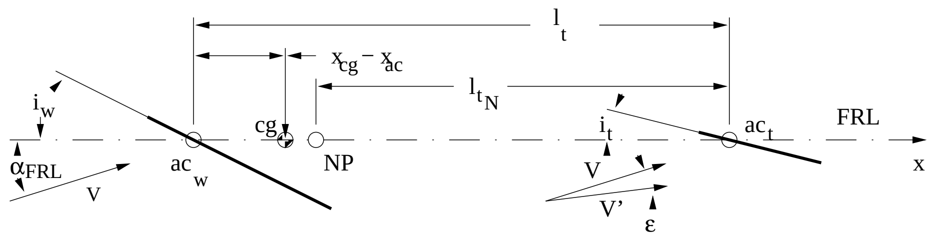


Figure 3.1: Geometry of wing and tail with respect to vehicle c.g., basic neutral point, and wing aerodynamic center. Note that positive direction of the x-axis is aft.

Grouping the terms involving the c.g. location, this equation can be written

$$\mathbf{C}_{m\text{cg}} = \mathbf{C}_{m0_w} + \left(\frac{x_{\text{cg}}}{\bar{c}} - \frac{x_{\text{ac}}}{\bar{c}}\right)\left[\mathbf{C}_{Lw} + \eta\frac{S_t}{S}\mathbf{C}_{Lt}\right] - \eta V_H \mathbf{C}_{Lt} + \mathbf{C}_{mf} \tag{3.2}$$

where $V_H = \frac{\ell_t S_t}{\bar{c}S}$ is the *tail volume parameter*. Note that this definition is based on the distance between the aerodynamic centers of the wing and tail, and is therefore independent of the vehicle c.g. location. Note that the total *vehicle* lift coefficient is

$$\mathbf{C}_L = \frac{L_w + L_t}{QS} = \mathbf{C}_{Lw} + \eta\frac{S_t}{S}\mathbf{C}_{Lt} \tag{3.3}$$

where $\eta = Q_t/Q$ is the tail efficiency factor, and this total vehicle lift coefficient is exactly the quantity appearing in the square brackets in Eq. (3.2). Now, we can introduce the dependence of the lift coefficients on angle of attack as

$$\begin{aligned}
\mathbf{C}_{Lw} &= \mathbf{C}_{L\alpha_w}\left(\alpha_{FRL} + i_w - \alpha_{0_w}\right)\\
\mathbf{C}_{Lt} &= \mathbf{C}_{L\alpha_t}\left(\alpha_{FRL} + i_t - \left[\varepsilon_0 + \frac{\mathrm{d}\varepsilon}{\mathrm{d}\alpha}\alpha_{FRL}\right]\right)
\end{aligned} \tag{3.4}$$

Note that, consistent with the usual use of symmetric sections for the horizontal tail, we have assumed $\alpha_{0_t} = 0$. Introducing these expressions into Eq. (3.3), the latter can be expressed as

$$\mathbf{C}_L = \mathbf{C}_{L\alpha_w}\left(i_w - \alpha_{0_w}\right) + \eta\frac{S_t}{S}\mathbf{C}_{L\alpha_t}\left(i_t - \varepsilon_0\right) + \left(\mathbf{C}_{L\alpha_w} + \eta\frac{S_t}{S}\left[1 - \frac{\mathrm{d}\varepsilon}{\mathrm{d}\alpha}\right]\mathbf{C}_{L\alpha_t}\right)\alpha_{FRL} \tag{3.5}$$

This equation has the form

$$\mathbf{C}_L = \mathbf{C}_{L0} + \mathbf{C}_{L\alpha}\alpha_{FRL} \tag{3.6}$$

where the *vehicle* lift curve slope is

$$\mathbf{C}_{L\alpha} = \mathbf{C}_{L\alpha_w} + \eta\frac{S_t}{S}\left(1 - \frac{\mathrm{d}\varepsilon}{\mathrm{d}\alpha}\right)\mathbf{C}_{L\alpha_t} \tag{3.7}$$

and

$$\mathbf{C}_{L0} = \mathbf{C}_{L\alpha_w}\left(i_w - \alpha_{0_w}\right) + \eta\frac{S_t}{S}\mathbf{C}_{L\alpha_t}\left(i_t - \varepsilon_0\right) \tag{3.8}$$

is the vehicle lift coefficient at zero (fuselage reference line) angle of attack. Finally, if we define the vehicle angle of attack relative to the angle of attack for zero *vehicle* lift, i.e.,

$$\alpha \equiv \alpha_{FRL} - \alpha_0 \tag{3.9}$$

where

$$\alpha_0 = -\frac{\mathbf{C}_{L0}}{\mathbf{C}_{L\alpha}} \tag{3.10}$$

then

$$\mathbf{C}_L = \mathbf{C}_{L\alpha}\alpha \tag{3.11}$$

where $\mathbf{C}_{L\alpha}$ is the vehicle lift curve slope, given by Eq. (3.7).

Introducing the angle of attack into Eq. (3.2), the expression for the vehicle pitching moment coefficient becomes

$$\begin{aligned}\mathbf{C}_{m\text{cg}} =& \mathbf{C}_{m0_w} + \left(\frac{x_{\text{cg}}}{\bar{c}} - \frac{x_{\text{ac}}}{\bar{c}}\right)\left[\mathbf{C}_{L\alpha_w}\left(i_w - \alpha_{0_w}\right) + \eta\frac{S_t}{S}\mathbf{C}_{L\alpha_t}\left(i_t - \varepsilon_0\right)\right] - \eta V_H \mathbf{C}_{L\alpha_t}\left(i_t - \varepsilon_0\right) + \\ &\left\{\left(\frac{x_{\text{cg}}}{\bar{c}} - \frac{x_{\text{ac}}}{\bar{c}}\right)\left[\mathbf{C}_{L\alpha_w} + \eta\frac{S_t}{S}\left(1 - \frac{\mathrm{d}\varepsilon}{\mathrm{d}\alpha}\right)\mathbf{C}_{L\alpha_t}\right] - \eta V_H\left(1 - \frac{\mathrm{d}\varepsilon}{\mathrm{d}\alpha}\right)\mathbf{C}_{L\alpha_t} + \mathbf{C}_{m\alpha_f}\right\}\alpha_{FRL}\end{aligned} \tag{3.12}$$

This can be expressed in terms of the angle of attack from zero vehicle lift as

$$\begin{aligned}\mathbf{C}_{m\text{cg}} = \mathbf{C}_{m0_w} &+ \left(\frac{x_{\text{cg}}}{\bar{c}} - \frac{x_{\text{ac}}}{\bar{c}}\right)\left[\mathbf{C}_{L\alpha_w}\left(i_w - \alpha_{0_w}\right) + \eta\frac{S_t}{S}\mathbf{C}_{L\alpha_t}\left(i_t - \varepsilon_0\right)\right] - \eta V_H \mathbf{C}_{L\alpha_t}\left(i_t - \varepsilon_0\right) \\ &+ \mathbf{C}_{m\alpha}\alpha_0 + \left\{\left(\frac{x_{\text{cg}}}{\bar{c}} - \frac{x_{\text{ac}}}{\bar{c}}\right)\mathbf{C}_{L\alpha} - \eta V_H \mathbf{C}_{L\alpha_t}\left(1 - \frac{\mathrm{d}\varepsilon}{\mathrm{d}\alpha}\right) + \mathbf{C}_{m\alpha_f}\right\}\alpha\end{aligned} \tag{3.13}$$

This equation has the form

$$\mathbf{C}_m = \mathbf{C}_{m0} + \mathbf{C}_{m\alpha}\alpha \tag{3.14}$$

with the *vehicle* pitching moment coefficient at zero lift

$$\mathbf{C}_{m0} = \mathbf{C}_{m0_w} + \left(\frac{x_{\text{cg}}}{\bar{c}} - \frac{x_{\text{ac}}}{\bar{c}}\right)\left[\mathbf{C}_{L\alpha_w}\left(i_w - \alpha_{0_w}\right) + \eta\frac{S_t}{S}\mathbf{C}_{L\alpha_t}\left(i_t - \varepsilon_0\right)\right] - \eta V_H \mathbf{C}_{L\alpha_t}\left(i_t - \varepsilon_0\right) + \mathbf{C}_{m\alpha}\alpha_0 \tag{3.15}$$

and the vehicle pitch stiffness

$$\mathbf{C}_{m\alpha} = \left(\frac{x_{\text{cg}}}{\bar{c}} - \frac{x_{\text{ac}}}{\bar{c}}\right)\mathbf{C}_{L\alpha} - \eta V_H \mathbf{C}_{L\alpha_t}\left(1 - \frac{\mathrm{d}\varepsilon}{\mathrm{d}\alpha}\right) + \mathbf{C}_{m\alpha_f} \tag{3.16}$$

Note that Eq. (3.15) can be simplified (using Eq. (3.16)) to

$$\mathbf{C}_{m0} = \mathbf{C}_{m0_w} - \eta V_H \mathbf{C}_{L\alpha_t}\left[i_t - \varepsilon_0 + \left(1 - \frac{\mathrm{d}\varepsilon}{\mathrm{d}\alpha}\right)\alpha_0\right] + \mathbf{C}_{m\alpha_f}\alpha_0 \tag{3.17}$$

Note that Eq. (3.17) correctly shows that the pitching moment at zero net vehicle lift is independent of the c.g. location, as it must be (since at zero lift the resultant aerodynamic force must sum to a pure couple).

The basic (or control-fixed) *neutral point* is defined as the c.g. location for which the vehicle is neutrally stable in pitch – i.e., the c.g. location for which the pitch stiffness goes to zero. From Eq. (3.16) the neutral point is seen to be located at

$$\frac{x_{NP}}{\bar{c}} = \frac{x_{\text{ac}}}{\bar{c}} + \eta V_H \frac{\mathbf{C}_{L\alpha_t}}{\mathbf{C}_{L\alpha}}\left(1 - \frac{\mathrm{d}\varepsilon}{\mathrm{d}\alpha}\right) - \frac{\mathbf{C}_{m\alpha_f}}{\mathbf{C}_{L\alpha}} \tag{3.18}$$

Note that Eq. (3.16) for the pitch stiffness can be expressed as

$$\mathbf{C}_{m\alpha} = \left\{ \frac{x_{\text{cg}}}{\bar{c}} - \left[\frac{x_{\text{ac}}}{\bar{c}} + \eta V_H \frac{\mathbf{C}_{L\alpha_t}}{\mathbf{C}_{L\alpha}} \left(1 - \frac{\mathrm{d}\varepsilon}{\mathrm{d}\alpha}\right) - \frac{\mathbf{C}_{m\alpha_f}}{\mathbf{C}_{L\alpha}} \right] \right\} \mathbf{C}_{L\alpha} \tag{3.19}$$

where the quantity in square brackets is exactly the location of the basic neutral point, as shown in Eq. (3.18). Thus, we can write

$$\mathbf{C}_{m\alpha} = \left\{ \frac{x_{\text{cg}}}{\bar{c}} - \frac{x_{NP}}{\bar{c}} \right\} \mathbf{C}_{L\alpha} \tag{3.20}$$

or, alternatively,

$$\frac{\partial \mathbf{C}_m}{\partial \mathbf{C}_L} = -\left(\frac{x_{NP}}{\bar{c}} - \frac{x_{\text{cg}}}{\bar{c}} \right) \tag{3.21}$$

Thus, the pitch stiffness, measured with respect to changes in vehicle lift coefficient, is proportional to the distance between the c.g. and the basic neutral point. The quantity in parentheses on the right-hand side of Eq. (3.21), i.e., the distance between the vehicle c.g. and the basic neutral point, expressed as a percentage of the wing mean aerodynamic chord, is called the vehicle *static margin*.[2]

3.2 Static Longitudinal Control

The elevator is the aerodynamic control for pitch angle of the vehicle, and its effect is described in terms of the *elevator effectiveness*

$$a_e = \frac{\partial \mathbf{C}_{Lt}}{\partial \delta_e} \tag{3.22}$$

where $\mathbf{C}_{Lt}$ is the lift coefficient of the horizontal tail and δ_e is the elevator deflection, considered positive trailing edge down. The horizontal tail lift coefficient is then given by

$$\mathbf{C}_{Lt} = \frac{\partial \mathbf{C}_{Lt}}{\partial \alpha_t} (\alpha + i_t - \varepsilon) + a_e \delta_e \tag{3.23}$$

and the change in *vehicle* lift coefficient due to elevator deflection is

$$\mathbf{C}_{L\delta_e} = \eta \frac{S_t}{S} a_e \tag{3.24}$$

while the change in vehicle pitching moment due to elevator deflection is

$$\begin{aligned} \mathbf{C}_{m\delta_e} &= -\eta \frac{S_t}{S} a_e \left[\frac{\ell_t}{\bar{c}} + \frac{x_{\text{ac}} - x_{\text{cg}}}{\bar{c}} \right] \\ &= -\mathbf{C}_{L\delta_e} \left[\frac{\ell_t}{\bar{c}} + \frac{x_{\text{ac}} - x_{\text{cg}}}{\bar{c}} \right] \end{aligned} \tag{3.25}$$

The geometry of the moment arm of the tail lift relative to the vehicle c.g. (which justifies the second term in Eq. (3.25)) is shown in Fig. 3.1.

The vehicle is in equilibrium (i.e., is trimmed) at a given lift coefficient $\mathbf{C}_{L\text{trim}}$ when

$$\begin{aligned} \mathbf{C}_{L\alpha}\alpha + \mathbf{C}_{L\delta_e}\delta_e &= \mathbf{C}_{L\text{trim}} \\ \mathbf{C}_{m\alpha}\alpha + \mathbf{C}_{m\delta_e}\delta_e &= -\mathbf{C}_{m0} \end{aligned} \tag{3.26}$$

[2]Again, it is worth emphasizing that the location of the basic neutral point, and other special c.g. locations to be introduced later, are usually described as fractional distances along the wing mean aerodynamic chord; e.g. we might say that the basic neutral point is located at 40 per cent m.a.c.

These two equations can be solved for the unknown angle of attack and elevator deflection to give

$$\begin{aligned}\alpha_{\text{trim}} &= \frac{-\mathbf{C}_{L\delta_e}\mathbf{C}_{m0} - \mathbf{C}_{m\delta_e}\mathbf{C}_{L\text{trim}}}{\Delta} \\ \delta_{\text{trim}} &= \frac{\mathbf{C}_{L\alpha}\mathbf{C}_{m0} + \mathbf{C}_{m\alpha}\mathbf{C}_{L\text{trim}}}{\Delta}\end{aligned} \tag{3.27}$$

where

$$\Delta = -\mathbf{C}_{L\alpha}\mathbf{C}_{m\delta_e} + \mathbf{C}_{m\alpha}\mathbf{C}_{L\delta_e} \tag{3.28}$$

Note that the parameter

$$\begin{aligned}\Delta &= -\mathbf{C}_{L\alpha}\mathbf{C}_{m\delta_e} + \mathbf{C}_{m\alpha}\mathbf{C}_{L\delta_e} \\ &= -\mathbf{C}_{L\alpha}\left[-\mathbf{C}_{L\delta_e}\left(\frac{\ell_t}{\bar{c}} + \frac{x_{\text{ac}} - x_{\text{cg}}}{\bar{c}}\right)\right] + \mathbf{C}_{L\alpha}\left(\frac{x_{\text{cg}} - x_{NP}}{\bar{c}}\right)\mathbf{C}_{L\delta_e} \\ &= \mathbf{C}_{L\alpha}\mathbf{C}_{L\delta_e}\left(\frac{\ell_t}{\bar{c}} + \frac{x_{\text{ac}} - x_{NP}}{\bar{c}}\right) = \mathbf{C}_{L\alpha}\mathbf{C}_{L\delta_e}\frac{\ell_{t_N}}{\bar{c}}\end{aligned} \tag{3.29}$$

where

$$\ell_{t_N} = \ell_t + x_{\text{ac}} - x_{NP} \tag{3.30}$$

is the distance from the basic neutral point to the tail aerodynamic center. Thus, the parameter Δ is independent of the vehicle c.g. location, and is seen to be positive for conventional (aft tail) configurations, and negative for canard (forward tail) configurations.

An important derivative related to handling qualities is the control position gradient for trim, which can be seen from the second of Eqs. (3.27) to be given by

$$\left.\frac{\mathrm{d}\delta_e}{\mathrm{d}\mathbf{C}_L}\right)_{\text{trim}} = \frac{\mathbf{C}_{m\alpha}}{\Delta} \tag{3.31}$$

It is seen from Eq. (3.31) that the control position gradient, which measures the sensitivity of trimmed lift coefficient to control position, is negative for stable, aft tail configurations, and is proportional to the static margin (since Δ is independent of c.g. location and $\mathbf{C}_{m\alpha}$ is directly proportional to the static margin). In fact, using Eq. 3.29, we can see that

$$\left.\frac{\mathrm{d}\delta_e}{\mathrm{d}\mathbf{C}_L}\right)_{\text{trim}} = \frac{-1}{\mathbf{C}_{L\delta_e}}\frac{x_{NP} - x_{\text{c.g.}}}{\ell_{t_N}} \tag{3.32}$$

Thus, the control position gradient is seen to be determined by the static margin, normalized by ℓ_{t_N}, scaled by the effectiveness of the control deflection at generating lift $\mathbf{C}_{L\delta_e}$.

These results can be used in flight tests to determine the location of the basic neutral point. For each of several different c.g. positions the value of lift coefficient $\mathbf{C}_L$ is determined as a function of control position (as indicated by the data points in Fig. 3.2 (a).) For each c.g. location the value of the control position gradient is estimated by the best straight-line fit through these data, and is then plotted as a function of c.g. location. A best-fit straight line to these data, illustrated in Fig. 3.2 (b), is then extrapolated to zero control position gradient, which corresponds to the basic neutral point.

3.2.1 Longitudinal Maneuvers – the Pull-up

Another important criterion for vehicle handling qualities is the sensitivity of vehicle normal acceleration to control input. This can be analyzed by considering the vehicle in a steady pull-up. This

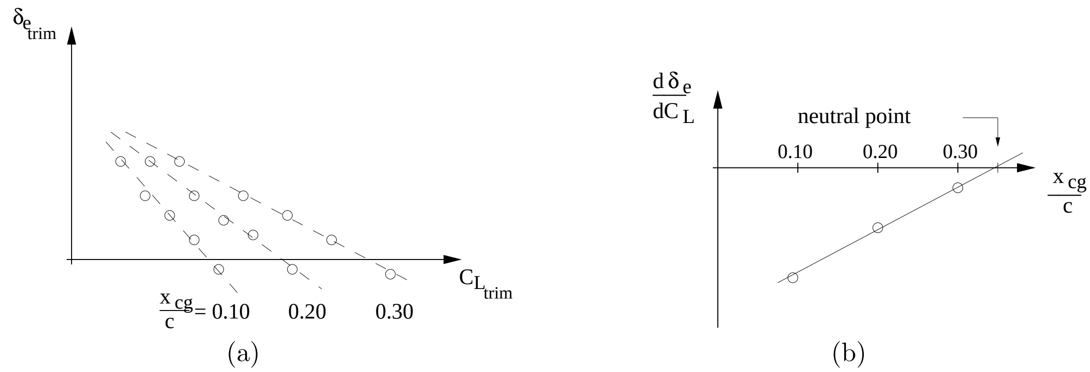


Figure 3.2: Schematic of procedure to estimate the location of the basic neutral point using control position gradient, measured in flight-test.

is a longitudinal maneuver in which the vehicle follows a curved flight path of constant radius R at constant angle of attack, as sketched in Fig. 3.3. For this maneuver, the pitch rate q is constant, and is given by

$$q = \frac{V}{R} \tag{3.33}$$

We define the dimensionless pitch rate

$$\hat{q} = \frac{q}{\frac{2V}{\bar{c}}} = \frac{\bar{c}q}{2V} \tag{3.34}$$

and will need to estimate the additional stability derivatives

$$\mathbf{C}_{Lq} \equiv \frac{\partial \mathbf{C}_L}{\partial \hat{q}} \tag{3.35}$$

and

$$\mathbf{C}_{mq} \equiv \frac{\partial \mathbf{C}_m}{\partial \hat{q}} \tag{3.36}$$

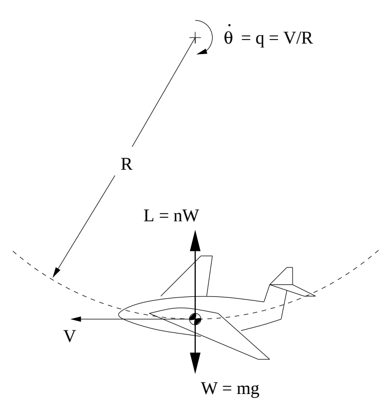


Figure 3.3: Schematic of flight path and forces acting on vehicle in a steady pull-up.

These derivatives characterize the sensitivity of vehicle lift and pitching moment to pitch rate. For vehicles with tails (either aft or canard), the largest contribution to these derivatives comes from the increment in tail lift due to the change in angle of attack of the tail arising from the rotation rate. This change in angle of attack is approximately[3]

$$\Delta\alpha_t = \frac{\ell_t}{V} q = \frac{2\ell_t}{\bar{c}} \hat{q} \tag{3.37}$$

and the resulting change in vehicle lift coefficient is

$$\Delta\mathbf{C}_L = \eta \frac{S_t}{S} \frac{\partial \mathbf{C}_{Lt}}{\partial \alpha_t} \Delta\alpha_t = 2\eta V_H \frac{\partial \mathbf{C}_{Lt}}{\partial \alpha_t} \hat{q} \tag{3.38}$$

so

$$\mathbf{C}_{Lq} = 2\eta V_H \frac{\partial \mathbf{C}_{Lt}}{\partial \alpha_t} \tag{3.39}$$

This increment in tail lift acts through the moment arm ℓ_t, so the corresponding estimate for the tail contribution to pitch damping is

$$\mathbf{C}_{mq} = -\frac{\ell_t}{\bar{c}} \mathbf{C}_{Lq} = -2\eta \frac{\ell_t}{\bar{c}} V_H \frac{\partial \mathbf{C}_{Lt}}{\partial \alpha_t} \tag{3.40}$$

The fuselage and wing (especially if the wing is swept) also contribute to the vehicle pitch damping, but it is difficult to develop simple formulas of general applicability, so these contributions will be neglected here. It should be noted that the tail contribution to pitch damping is sometimes multiplied by the factor 1.1 to account, at least approximately, for the contributions of other components. Finally, note that the derivative $\mathbf{C}_{Lq}$ will be positive for aft tail configurations (and negative for canard configurations), but the pitch damping $\mathbf{C}_{mq}$ will be always be negative, regardless of whether the tail is ahead or behind the vehicle center of gravity.

We analyze the motion at the point on the trajectory when the velocity vector is horizontal, so the balance of forces acting at the vehicle c.g. is

$$L - W = m\frac{V^2}{R} = mVq = \frac{2mV^2}{\bar{c}} \hat{q} \tag{3.41}$$

This equation can be written as

$$QS\left\{\mathbf{C}_{L\alpha}(\alpha + \Delta\alpha) + \mathbf{C}_{L\delta_e}(\delta_e + \Delta\delta_e) + \mathbf{C}_{Lq}\hat{q}\right\} - W = \frac{2mV^2}{\bar{c}} \hat{q} \tag{3.42}$$

where α and δ_e are the angle of attack and elevator deflection for trim in the unaccelerated case, and $\Delta\alpha$ and $\Delta\delta_e$ correspond to the increments in these angles due to the maneuver. If we introduce the *weight coefficient*

$$\mathbf{C}_W \equiv \frac{W/S}{Q} \tag{3.43}$$

the dimensionless form of this equation can be written

$$\left\{\mathbf{C}_{L\alpha}(\alpha + \Delta\alpha) + \mathbf{C}_{L\delta_e}(\delta_e + \Delta\delta_e) + \mathbf{C}_{Lq}\hat{q}\right\} - \mathbf{C}_W = 2\mu\hat{q} \tag{3.44}$$

where

$$\mu \equiv \frac{2m}{\rho S \bar{c}} \tag{3.45}$$

[3]Here, and in the equations through Eq. (3.40), the distance ℓ_t should represent the distance from the vehicle center-of-gravity to the aerodynamic center of the tail. The distance ℓ_t is a good approximation so long as the c.g. is near the wing aerodynamic center, which is usually the case.

is the vehicle *relative mass parameter*, which depends on ρ, the local fluid (air) density. As a result of this dependence on air density, the relative mass parameter is a function of flight altitude.

Subtracting the equilibrium values for the unaccelerated case

$$\mathbf{C}_{L\alpha}\alpha + \mathbf{C}_{L\delta_e}\delta_e - \mathbf{C}_W = 0 \tag{3.46}$$

from Eq. (3.44) gives

$$\mathbf{C}_{L\alpha}\Delta\alpha + \mathbf{C}_{L\delta_e}\Delta\delta_e = \left(2\mu - \mathbf{C}_{Lq}\right)\hat{q} \tag{3.47}$$

Finally, if we introduce the *normal acceleration parameter* n such that $L = nW$, then the force balance of Eq. (3.41) can be written in the dimensionless form

$$(n-1)\mathbf{C}_W = 2\mu\hat{q} \tag{3.48}$$

which provides a direct relation between the normal acceleration and the pitch rate, so that the lift equilibrium equation can be written

$$\mathbf{C}_{L\alpha}\Delta\alpha + \mathbf{C}_{L\delta_e}\Delta\delta_e = (n-1)\mathbf{C}_W\left(1 - \frac{\mathbf{C}_{Lq}}{2\mu}\right) \tag{3.49}$$

The pitching moment must also remain zero for equilibrium (since $\dot{q} = 0$), so

$$\mathbf{C}_{m\alpha}\Delta\alpha + \mathbf{C}_{m\delta_e}\Delta\delta_e + \mathbf{C}_{mq}\hat{q} = 0 \tag{3.50}$$

or

$$\mathbf{C}_{m\alpha}\Delta\alpha + \mathbf{C}_{m\delta_e}\Delta\delta_e = -\mathbf{C}_{mq}\frac{(n-1)\mathbf{C}_W}{2\mu} \tag{3.51}$$

Equations (3.49) and (3.51) provide two equations that can be solved for the unknowns $\Delta\alpha$ and $\Delta\delta_e$ to give

$$\begin{aligned}\Delta\alpha &= \frac{-(n-1)\mathbf{C}_W}{\Delta}\left[\left(1 - \frac{\mathbf{C}_{Lq}}{2\mu}\right)\mathbf{C}_{m\delta_e} + \frac{\mathbf{C}_{mq}}{2\mu}\mathbf{C}_{L\delta_e}\right]\\ \Delta\delta_e &= \frac{(n-1)\mathbf{C}_W}{\Delta}\left[\left(1 - \frac{\mathbf{C}_{Lq}}{2\mu}\right)\mathbf{C}_{m\alpha} + \frac{\mathbf{C}_{mq}}{2\mu}\mathbf{C}_{L\alpha}\right]\end{aligned} \tag{3.52}$$

where

$$\Delta = -\mathbf{C}_{L\alpha}\mathbf{C}_{m\delta_e} + \mathbf{C}_{m\alpha}\mathbf{C}_{L\delta_e} \tag{3.53}$$

is the same parameter as earlier (in Eq. (3.28)).

The control position derivative for normal acceleration is therefore given by

$$\frac{\mathrm{d}\delta_e}{\mathrm{d}n} = \frac{\mathbf{C}_W}{\Delta}\left[\left(1 - \frac{\mathbf{C}_{Lq}}{2\mu}\right)\mathbf{C}_{m\alpha} + \frac{\mathbf{C}_{mq}}{2\mu}\mathbf{C}_{L\alpha}\right] \tag{3.54}$$

Using Eq. (3.20) to express the pitch stiffness in terms of the c.g. location, we have

$$\frac{\mathrm{d}\delta_e}{\mathrm{d}n} = \frac{\mathbf{C}_W}{\Delta}\left[\left(1 - \frac{\mathbf{C}_{Lq}}{2\mu}\right)\left(\frac{x_{\mathrm{cg}}}{\bar{c}} - \frac{x_{NP}}{\bar{c}}\right) + \frac{\mathbf{C}_{mq}}{2\mu}\right]\mathbf{C}_{L\alpha} \tag{3.55}$$

The c.g. location for which this derivative vanishes is called the *basic maneuver point*, and its location, relative to the basic neutral point, is seen to be given by

$$\frac{x_{NP}}{\bar{c}} - \frac{x_{MP}}{\bar{c}} = \frac{\frac{\mathbf{C}_{mq}}{2\mu}}{1 - \frac{\mathbf{C}_{Lq}}{2\mu}} \approx \frac{\mathbf{C}_{mq}}{2\mu} \tag{3.56}$$

Since for all configurations the pitch damping $\mathbf{C}_{mq} < 0$, the maneuver point is aft of the neutral point. Also, since the vehicle relative mass parameter μ increases with altitude, the maneuver point approaches the neutral point with increasing altitude. If Eq. (3.56) is used to eliminate the variable x_{NP} from Eq. (3.55), we have

$$\frac{\mathrm{d}\delta_e}{\mathrm{d}n} = -\frac{\mathbf{C}_W \mathbf{C}_{L\alpha}}{\Delta} \left(1 - \frac{\mathbf{C}_{Lq}}{2\mu}\right) \left(\frac{x_{MP}}{\bar{c}} - \frac{x_{\mathrm{cg}}}{\bar{c}}\right) \tag{3.57}$$

where

$$\left(\frac{x_{MP}}{\bar{c}} - \frac{x_{\mathrm{cg}}}{\bar{c}}\right) \tag{3.58}$$

is called the *maneuver margin*.

3.3 Control Surface Hinge Moments

Just as the control *position* gradient is related to the pitch stiffness of the vehicle when the controls are fixed, the control *force* gradients are related to the pitch stiffness of the vehicle when the controls are allowed to float free.

3.3.1 Control Surface Hinge Moments

Since elevator deflection corresponds to rotation about a hinge line, the forces required to cause a specific control deflection are related to the aerodynamic moments about the hinge line. A free control will float, in the static case, to the position at which the elevator hinge moment is zero:

$$H_e = 0.$$

The elevator hinge moment is usually expressed in terms of the *hinge moment coefficient*

$$\mathbf{C}_{he} = \frac{H_e}{Q S_e \bar{c}_e} \tag{3.59}$$

where the reference area S_e and moment arm $\bar{c}_e$ correspond to the planform area and mean chord of the control surface aft of the hinge line. Note that the elevator hinge moment coefficient is defined relative to Q, not Q_t. While it would seem to make more sense to use Q_t, hinge moments are sufficiently difficult to predict that they are almost always determined from experiments in which the tail efficiency factor is effectively included in the definition of $\mathbf{C}_{he}$ (rather than explicitly isolated in a separate factor).

Assuming that the hinge moment is a linear function of angle of attack, control deflection, etc., we write

$$\mathbf{C}_{he} = \mathbf{C}_{he_0} + \mathbf{C}_{h\alpha}\alpha + \mathbf{C}_{h\delta_e}\delta_e + \mathbf{C}_{h\delta_t}\delta_t \tag{3.60}$$

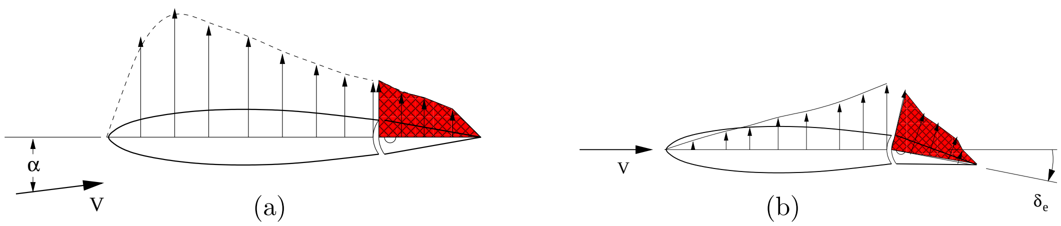


Figure 3.4: Schematic illustration of aerodynamic forces responsible for (a) floating and (b) restoring tendencies of trailing edge control surfaces. Floating (or restoring) tendency represents moment about hinge line of (shaded) lift distribution acting on control surface per unit angle of attack (or control deflection).

In this equation, α is the angle of attack (from angle for zero vehicle lift), δ_e is the elevator deflection, and δ_t is the deflection of the *control tab* (to be described in greater detail later).

The derivative $\mathbf{C}_{h\alpha}$ characterizes the hinge moment created by changes in angle of attack; it is called the *floating tendency*, as the hinge moment generated by an increase in angle of attack generally causes the control surface to float upward. The derivative $\mathbf{C}_{h\delta_e}$ characterizes the hinge moment created by a deflection of the control (considered positive trailing edge down); it is called the *restoring tendency*, as the nose-down hinge moment generated by a positive control deflection tends to restore the control to its original position. The floating tendency in Eq. (3.60) is referred to the vehicle angle of attack, and so it is related to the derivative based on tail angle of attack α_t by

$$\mathbf{C}_{h\alpha} = \left(1 - \frac{\mathrm{d}\epsilon}{\mathrm{d}\alpha}\right)\mathbf{C}_{h\alpha_t} \tag{3.61}$$

which accounts for the effects of wing induced downwash at the tail. The aerodynamic forces responsible for generating the hinge moments reflected in the floating and restoring tendencies are sketched in Fig. 3.4. Only the shaded portion of the lift distribution in these figures acts on the control surface and contributes to the hinge moment.

The angle at which the free elevator floats is determined by the fact that the hinge moment (and, therefore, the hinge moment coefficient) must be zero

$$\mathbf{C}_{h_e} = 0 = \mathbf{C}_{h_{e_0}} + \mathbf{C}_{h\alpha}\alpha + \mathbf{C}_{h\delta_e}\delta_{e\,\mathrm{free}} + \mathbf{C}_{h\delta_t}\delta_t$$

or

$$\delta_{e\,\mathrm{free}} = -\frac{1}{\mathbf{C}_{h\delta_e}}\left(\mathbf{C}_{h_{e_0}} + \mathbf{C}_{h\alpha}\alpha + \mathbf{C}_{h\delta_t}\delta_t\right) \tag{3.62}$$

The corresponding lift and moment coefficients are

$$\begin{aligned}\mathbf{C}_{L\,\mathrm{free}} &= \mathbf{C}_{L\alpha}\alpha + \mathbf{C}_{L\delta_e}\delta_{e\,\mathrm{free}}\\ \mathbf{C}_{m\,\mathrm{free}} &= \mathbf{C}_{m0} + \mathbf{C}_{m\alpha}\alpha + \mathbf{C}_{m\delta_e}\delta_{e\,\mathrm{free}}\end{aligned} \tag{3.63}$$

which, upon substituting from Eq. (3.62), can be written

$$\begin{aligned}\mathbf{C}_{L\,\mathrm{free}} &= \mathbf{C}_{L\alpha}\left(1 - \frac{\mathbf{C}_{L\delta_e}\mathbf{C}_{h\alpha}}{\mathbf{C}_{L\alpha}\mathbf{C}_{h\delta_e}}\right)\alpha - \frac{\mathbf{C}_{L\delta_e}}{\mathbf{C}_{h\delta_e}}\left(\mathbf{C}_{h_{e_0}} + \mathbf{C}_{h\delta_t}\delta_t\right)\\ \mathbf{C}_{m\,\mathrm{free}} &= \mathbf{C}_{m\alpha}\left(1 - \frac{\mathbf{C}_{m\delta_e}\mathbf{C}_{h\alpha}}{\mathbf{C}_{m\alpha}\mathbf{C}_{h\delta_e}}\right)\alpha + \mathbf{C}_{m0} - \frac{\mathbf{C}_{m\delta_e}}{\mathbf{C}_{h\delta_e}}\left(\mathbf{C}_{h_{e_0}} + \mathbf{C}_{h\delta_t}\delta_t\right)\end{aligned} \tag{3.64}$$

Thus, if we denote the control free lift curve slope and pitch stiffness using primes, we see from the above equations that

$$\begin{aligned}
\mathbf{C}_{L_\alpha}' &= \mathbf{C}_{L\alpha}\left(1 - \frac{\mathbf{C}_{L\delta_e}\mathbf{C}_{h\alpha}}{\mathbf{C}_{L\alpha}\mathbf{C}_{h\delta_e}}\right) \\
\mathbf{C}_{m_\alpha}' &= \mathbf{C}_{m\alpha}\left(1 - \frac{\mathbf{C}_{m\delta_e}\mathbf{C}_{h\alpha}}{\mathbf{C}_{m\alpha}\mathbf{C}_{h\delta_e}}\right)
\end{aligned} \tag{3.65}$$

Inspection of these equations shows that the lift curve slope is always reduced by freeing the controls, and the pitch stiffness of a stable configuration is reduced in magnitude by freeing the controls for an aft tail configuration, and increased in magnitude for a forward tail (canard) configuration (in all cases assuming that the floating and restoring tendencies both are negative).

3.3.2 Control free Neutral Point

The c.g. location at which the control free pitch stiffness vanishes is called the *control free neutral point*. The location of the control free neutral point x_{NP}' can be determined by expressing the pitch stiffness in the second of Eqs. (3.65)

$$\mathbf{C}_{m_\alpha}' = \mathbf{C}_{m\alpha} - \frac{\mathbf{C}_{m\delta_e}\mathbf{C}_{h\alpha}}{\mathbf{C}_{h\delta_e}}$$

as

$$\begin{aligned}
\mathbf{C}_{m_\alpha}' &= \left(\frac{x_{\mathrm{cg}}}{\bar{c}} - \frac{x_{NP}}{\bar{c}}\right)\mathbf{C}_{L\alpha} + \frac{\mathbf{C}_{h\alpha}\mathbf{C}_{L\delta_e}}{\mathbf{C}_{h\delta_e}}\left(\frac{\ell_t}{\bar{c}} + \frac{x_{\mathrm{ac}}}{\bar{c}} - \frac{x_{\mathrm{cg}}}{\bar{c}}\right) \\
&= \left(\frac{x_{\mathrm{cg}}}{\bar{c}} - \frac{x_{NP}}{\bar{c}}\right)\mathbf{C}_{L\alpha} + \frac{\mathbf{C}_{h\alpha}}{\mathbf{C}_{h\delta_e}}\eta\frac{S_t}{S}a_e\left(\frac{\ell_t + x_{\mathrm{ac}} - x_{NP}}{\bar{c}} + \frac{x_{NP} - x_{\mathrm{cg}}}{\bar{c}}\right) \\
&= \left(\frac{x_{\mathrm{cg}}}{\bar{c}} - \frac{x_{NP}}{\bar{c}}\right)\left[\mathbf{C}_{L\alpha} - \frac{\mathbf{C}_{L\delta_e}\mathbf{C}_{h\alpha}}{\mathbf{C}_{h\delta_e}}\right] + \eta V_{H_N}\frac{\mathbf{C}_{h\alpha}a_e}{\mathbf{C}_{h\delta_e}}
\end{aligned} \tag{3.66}$$

where $a_e = \partial\mathbf{C}_{Lt}/\partial\delta_e$ is the elevator effectiveness and

$$V_{H_N} = \left(\frac{\ell_t}{\bar{c}} + \frac{x_{\mathrm{ac}}}{\bar{c}} - \frac{x_{NP}}{\bar{c}}\right)\frac{S_t}{S} \tag{3.67}$$

is the tail volume ratio based on ℓ_{t_N}, the distance between the tail aerodynamic center and the basic neutral point, as defined in Eq. (3.30). The quantity in square brackets in the final version of Eq. (3.66) is seen to be simply the control free vehicle lift curve slope $\mathbf{C}_{L_\alpha}'$, so we have

$$\mathbf{C}_{m_\alpha}' = \left(\frac{x_{\mathrm{cg}}}{\bar{c}} - \frac{x_{NP}}{\bar{c}}\right)\mathbf{C}_{L_\alpha}' + \eta V_{H_N}\frac{\mathbf{C}_{h\alpha}a_e}{\mathbf{C}_{h\delta_e}} \tag{3.68}$$

Setting the control free pitch stiffness $\mathbf{C}_{m_\alpha}'$ to zero gives the distance between the control free and basic neutral points as

$$\frac{x_{NP}}{\bar{c}} - \frac{x_{NP}'}{\bar{c}} = \eta V_{H_N}\frac{a_e}{\mathbf{C}_{L_\alpha}'}\frac{\mathbf{C}_{h\alpha}}{\mathbf{C}_{h\delta_e}} \tag{3.69}$$

Finally, if Eq. (3.69) is substituted back into Eq. (3.68) to eliminate the variable x_{NP}, we have

$$\mathbf{C}_{m_\alpha}' = -\left(\frac{x_{NP}'}{\bar{c}} - \frac{x_{\mathrm{cg}}}{\bar{c}}\right)\mathbf{C}_{L_\alpha}' \tag{3.70}$$

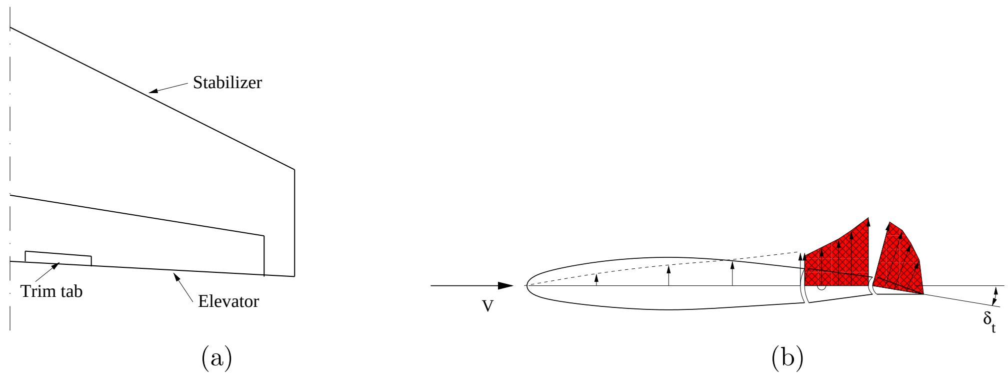


Figure 3.5: (a) Typical location of trim tab on horizontal control (elevator), and (b) schematic illustration of aerodynamic forces responsible for hinge moment due to trim tab deflection.

showing that the control free pitch stiffness is directly proportional to the *control free static margin*

$$\left(\frac{x'_{NP}}{\bar{c}} - \frac{x_{\mathrm{cg}}}{\bar{c}}\right)$$

3.3.3 Trim Tabs

Trim tabs can be used by the pilot to trim the vehicle at zero control force for any desired speed. Trim tabs are small control surfaces mounted at the trailing edges of primary control surfaces. A linkage is provided that allows the pilot to set the angle of the trim tab, relative to the primary control surface, in a way that is independent of the deflection of the primary control surface. Deflection of the trim tab creates a hinge moment that causes the elevator to float at the angle desired for trim. The geometry of a typical trim tab arrangement is shown in Fig. 3.5.

Zero control force corresponds to zero hinge moment, or

$$\mathbf{C}_{he} = 0 = \mathbf{C}_{he_0} + \mathbf{C}_{h\alpha}\alpha + \mathbf{C}_{h\delta_e}\delta_e + \mathbf{C}_{h\delta_t}\delta_t$$

and the trim tab deflection that achieves this for arbitrary angle of attack and control deflection is

$$\delta_t = -\frac{1}{\mathbf{C}_{h\delta_t}}\left(\mathbf{C}_{he_0} + \mathbf{C}_{h\alpha}\alpha + \mathbf{C}_{h\delta_e}\delta_e\right) \tag{3.71}$$

so the tab setting required for zero control force at trim is

$$\delta_{t_{\mathrm{trim}}} = -\frac{1}{\mathbf{C}_{h\delta_t}}\left(\mathbf{C}_{he_0} + \mathbf{C}_{h\alpha}\alpha_{\mathrm{trim}} + \mathbf{C}_{h\delta_e}\delta_{e\mathrm{trim}}\right) \tag{3.72}$$

The values of α_{trim} and $\delta_{e\mathrm{trim}}$ are given by Eqs. (3.27)

$$\begin{aligned} \alpha_{\mathrm{trim}} &= \frac{-\mathbf{C}_{L\delta_e}\mathbf{C}_{m0} - \mathbf{C}_{m\delta_e}\mathbf{C}_{L\mathrm{trim}}}{\Delta} \\ \delta_{e\mathrm{trim}} &= \frac{\mathbf{C}_{L\alpha}\mathbf{C}_{m0} + \mathbf{C}_{m\alpha}\mathbf{C}_{L\mathrm{trim}}}{\Delta} \end{aligned} \tag{3.73}$$

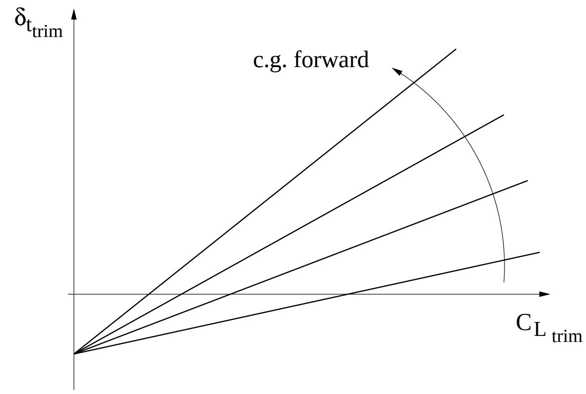


Figure 3.6: Variation in trim tab setting as function of velocity for stable, aft tail vehicle.

Substituting these values into Eq. (3.72) gives the required trim tab setting as

$$\delta_{t_{\rm trim}} = -\frac{1}{\mathbf{C}_{h\delta_t}}\left(\mathbf{C}_{he_0} + \frac{\mathbf{C}_{m0}}{\Delta}\left(-\mathbf{C}_{h\alpha}\mathbf{C}_{L\delta_e} + \mathbf{C}_{h\delta_e}\mathbf{C}_{L\alpha}\right) + \frac{1}{\Delta}\left(-\mathbf{C}_{h\alpha}\mathbf{C}_{m\delta_e} + \mathbf{C}_{h\delta_e}\mathbf{C}_{m\alpha}\right)\mathbf{C}_{L\rm trim}\right) \tag{3.74}$$

Note that the coefficient of $\mathbf{C}_{L\rm trim}$ in this equation – which gives the sensitivity of the trim tab setting to the trim lift coefficient – can be written as

$$\frac{d\delta_t}{d\mathbf{C}_L} = -\frac{\mathbf{C}_{h\delta_e}}{\mathbf{C}_{h\delta_t}\Delta}\left(\mathbf{C}_{m\alpha} - \frac{\mathbf{C}_{h\alpha}\mathbf{C}_{m\delta_e}}{\mathbf{C}_{h\delta_e}}\right) = -\frac{\mathbf{C}_{h\delta_e}}{\mathbf{C}_{h\delta_t}\Delta}{\mathbf{C}_{m\alpha}}' = -\frac{\mathbf{C}_{h\delta_e}}{\mathbf{C}_{h\delta_t}\Delta}\left(\frac{x'_{NP}}{\bar{c}} - \frac{x_{\rm cg}}{\bar{c}}\right){\mathbf{C}_{L\alpha}}' \tag{3.75}$$

and Eq. (3.74) can be written

$$\delta_{t_{\rm trim}} = -\frac{1}{\mathbf{C}_{h\delta_t}}\left[\mathbf{C}_{he_0} + \frac{\mathbf{C}_{m0}}{\Delta}\left(-\mathbf{C}_{h\alpha}\mathbf{C}_{L\delta_e} + \mathbf{C}_{h\delta_e}\mathbf{C}_{L\alpha}\right) + \frac{\mathbf{C}_{h\delta_e}}{\Delta}{\mathbf{C}_{L\alpha}}'\left(\frac{x'_{NP}}{\bar{c}} - \frac{x_{\rm cg}}{\bar{c}}\right)\mathbf{C}_{L\rm trim}\right] \tag{3.76}$$

Thus, the tab setting for trim is a linear function of trimmed lift coefficient whose slope is proportional to the control free static margin. This variation is shown schematically for a conventional (aft tail) configuration in Fig. 3.6.

3.3.4 Control Force for Trim

As mentioned earlier, the most important aspects of stability relating to handling qualities of the vehicle are related to control *forces*. For longitudinal control, the control force F is related to the elevator hinge moment H_e through a gearing constant G, so that

$$F = GH_e \tag{3.77}$$

This equation defines a positive control force as a *pull*, corresponding to the force required to balance a positive (nose up) elevator hinge moment.[4] The units of the gearing constant G are inverse length, which can be interpreted as a mechanical advantage corresponding to radians of control deflection per unit distance (foot) of control yoke displacement.

[4] It is important to be careful when reading other books; positive control force is sometimes defined as a *push*, in which case there is a minus sign inserted on the right hand side of Eq. (3.77) and subsequently throughout the analysis.

Expressing the hinge moment in terms of the corresponding dimensionless coefficient, we have

$$F = GS_e\bar{c}_eQ\mathbf{C}_{he} = GS_e\bar{c}_eQ\left(\mathbf{C}_{he_0} + \mathbf{C}_{h\alpha}\alpha + \mathbf{C}_{h\delta_e}\delta_e + \mathbf{C}_{h\delta_t}\delta_t\right) \tag{3.78}$$

Since this equation is linear in tab deflection, the control force required for a tab setting other than the trim value is

$$F = GS_e\bar{c}_eQ\mathbf{C}_{h\delta_t}\left(\delta_t - \delta_{t_{\text{trim}}}\right) \tag{3.79}$$

and, substituting the tab setting required for trim from Eq. (3.76), we have

$$F = GS_e\bar{c}_eQ\left[\mathbf{C}_{h\delta_t}\delta_t + \mathbf{C}_{he_0} + \frac{\mathbf{C}_{m0}}{\Delta}\left(-\mathbf{C}_{h\alpha}\mathbf{C}_{L\delta_e} + \mathbf{C}_{h\delta_e}\mathbf{C}_{L\alpha}\right) + \frac{\mathbf{C}_{h\delta_e}}{\Delta}\mathbf{C}_{L}{}'_{\alpha}\left(\frac{x_{\text{cg}} - x'_{NP}}{\bar{c}}\right)\mathbf{C}_{L\text{trim}}\right] \tag{3.80}$$

Finally, substituting

$$\mathbf{C}_{L\text{trim}} = \frac{W/S}{Q} \tag{3.81}$$

for level flight with $L = W$, we have

$$\begin{aligned} F =& GS_e\bar{c}_e(W/S)\frac{\mathbf{C}_{h\delta_e}\mathbf{C}_{L}{}'_{\alpha}}{\Delta}\left(\frac{x_{\text{cg}} - x'_{NP}}{\bar{c}}\right) + \\ & GS_e\bar{c}_e\left[\mathbf{C}_{h\delta_t}\delta_t + \mathbf{C}_{he_0} + \frac{\mathbf{C}_{m0}}{\Delta}\left(-\mathbf{C}_{h\alpha}\mathbf{C}_{L\delta_e} + \mathbf{C}_{h\delta_e}\mathbf{C}_{L\alpha}\right)\right]\frac{1}{2}\rho V^2 \end{aligned} \tag{3.82}$$

The dependence of control force on velocity described by this equation is sketched in Fig. 3.7. Note from the equation that:

1. The control force $F \propto S_e\bar{c}_e$, i.e, is proportional to the *cube* of the size of the vehicle; control forces grow rapidly with aircraft size, and large aircraft require powered (or power-assisted) control systems.
2. The location of the c.g. (i.e., the control free static margin) affects only the constant term in the equation.
3. The vehicle weight enters only in the ratio W/S.
4. The effect of trim tab deflection δ_t is to change the coefficient of the V^2 term, and hence controls the intercept of the curve with the velocity axis.

If we denote the velocity at which the control force is zero as V_{trim}, then Eq. (3.82) gives

$$\begin{aligned} & GS_e\bar{c}_e\left(\mathbf{C}_{h\delta_t}\delta_t + \mathbf{C}_{he_0} + \frac{\mathbf{C}_{m0}}{\Delta}\left(-\mathbf{C}_{h\alpha}\mathbf{C}_{L\delta_e} + \mathbf{C}_{h\delta_e}\mathbf{C}_{L\alpha}\right)\right)\frac{1}{2}\rho V_{\text{trim}}^2 = \\ & -GS_e\bar{c}_e(W/S)\frac{\mathbf{C}_{h\delta_e}\mathbf{C}_{L}{}'_{\alpha}}{\Delta}\left(\frac{x_{\text{cg}} - x'_{NP}}{\bar{c}}\right) \end{aligned} \tag{3.83}$$

so

$$F = GS_e\bar{c}_e(W/S)\frac{\mathbf{C}_{h\delta_e}\mathbf{C}_{L}{}'_{\alpha}}{\Delta}\left(\frac{x_{\text{cg}} - x'_{NP}}{\bar{c}}\right)\left[1 - (V/V_{\text{trim}})^2\right] \tag{3.84}$$

and

$$\left.\frac{\mathrm{d}F}{\mathrm{d}V}\right)_{V_{\text{trim}}} = -\frac{2}{V_{\text{trim}}}GS_e\bar{c}_e(W/S)\frac{\mathbf{C}_{h\delta_e}\mathbf{C}_{L}{}'_{\alpha}}{\Delta}\left(\frac{x_{\text{cg}} - x'_{NP}}{\bar{c}}\right) \tag{3.85}$$

These last two equations, which also can be interpreted in terms of Fig. 3.7, show that:

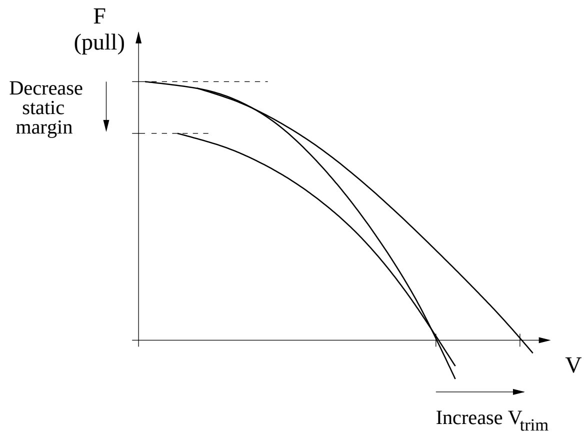


Figure 3.7: Typical variation in control force as function of vehicle velocity for stable configuration.

1. For a given control free static margin (or c.g. position) the control force gradient decreases with increasing flight velocity; and
2. At a given trim velocity, the control force gradient decreases as the c.g. is moved aft toward the control free neutral point (i.e., as the static margin is reduced).

3.3.5 Control-force for Maneuver

Perhaps the single most important stability property of an aircraft, in terms of handling properties, describes the control force required to perform a maneuver. This force must not be too small to avoid over-stressing the airframe, nor too large to avoid making the pilot work too hard.

We will again consider the steady pull-up. The change in control force required to effect the maneuver is

$$\Delta F = G S_e \bar{c}_e Q \Delta \mathbf{C}_{he} \tag{3.86}$$

where

$$\Delta \mathbf{C}_{he} = \mathbf{C}_{h\alpha} \Delta\alpha + \mathbf{C}_{h\delta_e} \Delta\delta_e + \mathbf{C}_{hq} \hat{q} \tag{3.87}$$

where $\hat{q}$ is the dimensionless pitch rate, as defined in Section 3.2.1. It was also seen in that section that the dimensionless pitch rate for a pull-up could be related directly to the excess load factor $(n-1)$, so, using Eq. (3.48), we have

$$\Delta \mathbf{C}_{he} = \mathbf{C}_{h\alpha} \Delta\alpha + \mathbf{C}_{h\delta_e} \Delta\delta_e + \frac{(n-1)\mathbf{C}_W}{2\mu} \mathbf{C}_{hq} \tag{3.88}$$

The derivative $\mathbf{C}_{hq}$ arises from the change in hinge moment due to the change in tail angle of attack arising from the pitch rate. Thus

$$\Delta \mathbf{C}_{he} = \mathbf{C}_{h\alpha_t} \Delta\alpha_t = \mathbf{C}_{h\alpha_t} \frac{2\ell_t}{\bar{c}} \hat{q} \tag{3.89}$$

and

$$\mathbf{C}_{hq} \equiv \frac{\partial \mathbf{C}_{he}}{\partial \hat{q}} = 2\frac{\ell_t}{\bar{c}} \mathbf{C}_{h\alpha_t} \tag{3.90}$$

Now, we can use the solution for $\Delta\delta_e$ from Eq. (3.52)

$$\Delta\delta_e = \frac{(n-1)\mathbf{C}_W}{\Delta}\left[\left(1-\frac{\mathbf{C}_{Lq}}{2\mu}\right)\mathbf{C}_{m\alpha}+\frac{\mathbf{C}_{mq}}{2\mu}\mathbf{C}_{L\alpha}\right] \tag{3.91}$$

along with the lift coefficient equation, Eq. (3.49), which can be written

$$\Delta\alpha = \frac{1}{\mathbf{C}_{L\alpha}}\left[(n-1)\mathbf{C}_W\left(1-\frac{\mathbf{C}_{Lq}}{2\mu}\right)-\mathbf{C}_{L\delta_e}\Delta\delta_e\right] \tag{3.92}$$

in the hinge moment equation to give

$$\Delta\mathbf{C}_{he} = \mathbf{C}_{h\alpha}\frac{n-1}{\mathbf{C}_{L\alpha}}\left[\left(1-\frac{\mathbf{C}_{Lq}}{2\mu}\right)\mathbf{C}_W-\mathbf{C}_{L\delta_e}\frac{\Delta\delta_e}{n-1}\right]+\mathbf{C}_{h\delta_e}\Delta\delta_e+\frac{(n-1)\mathbf{C}_W}{2\mu}\mathbf{C}_{hq} \tag{3.93}$$

which can be rearranged into the form

$$\frac{\Delta\mathbf{C}_{he}}{n-1} = \frac{\mathbf{C}_W}{\mathbf{C}_{L\alpha}}\left[\left(1-\frac{\mathbf{C}_{Lq}}{2\mu}\right)\mathbf{C}_{h\alpha}+\frac{\mathbf{C}_{hq}}{2\mu}\mathbf{C}_{L\alpha}\right]+\frac{\Delta\delta_e}{n-1}\mathbf{C}_{h\delta_e}\frac{\mathbf{C}_{L\alpha}'}{\mathbf{C}_{L\alpha}} \tag{3.94}$$

Finally, using Eq. (3.57) for $\Delta\delta_e/(n-1)$, the equation for the hinge moment increment can be written

$$\frac{\Delta\mathbf{C}_{he}}{n-1} = \frac{\mathbf{C}_W\mathbf{C}_{L\alpha}'\mathbf{C}_{h\delta_e}}{\Delta}\left(1-\frac{\mathbf{C}_{Lq}}{2\mu}\right)\left[\frac{x_{\text{cg}}-x_{MP}}{\bar{c}}+\frac{\Delta}{\mathbf{C}_{L\alpha}'\mathbf{C}_{h\delta_e}}\left(\frac{\mathbf{C}_{h\alpha}}{\mathbf{C}_{L\alpha}}+\frac{\mathbf{C}_{hq}}{2\mu-\mathbf{C}_{Lq}}\right)\right] \tag{3.95}$$

The *control free maneuver point* is defined as the c.g. location for which the control force gradient (per g) (or, equivalently, the hinge moment coefficient gradient) vanishes. This is seen from Eq. (3.95) to give

$$\frac{x_{MP}-x_{MP}'}{\bar{c}} = \frac{\Delta}{\mathbf{C}_{L\alpha}'\mathbf{C}_{h\delta_e}}\left(\frac{\mathbf{C}_{h\alpha}}{\mathbf{C}_{L\alpha}}+\frac{\mathbf{C}_{hq}}{2\mu-\mathbf{C}_{Lq}}\right) \tag{3.96}$$

Note that this quantity is positive for aft tail configurations, and negative for forward tail (canard) configurations. Substitution of this expression back into Eq. (3.95) then gives

$$\frac{\Delta\mathbf{C}_{he}}{n-1} = \frac{\mathbf{C}_W\mathbf{C}_{L\alpha}'\mathbf{C}_{h\delta_e}}{\Delta}\left(1-\frac{\mathbf{C}_{Lq}}{2\mu}\right)\left(\frac{x_{\text{cg}}-x_{MP}'}{\bar{c}}\right) \tag{3.97}$$

Finally, the control force gradient (per g) is

$$\begin{aligned}\frac{\partial F}{\partial n} &= \frac{\Delta F}{n-1} = GS_e\bar{c}_eQ\frac{\Delta\mathbf{C}_{he}}{n-1}\\ &= GS_e\bar{c}_eQ\frac{\mathbf{C}_W\mathbf{C}_{L\alpha}'\mathbf{C}_{h\delta_e}}{\Delta}\left(1-\frac{\mathbf{C}_{Lq}}{2\mu}\right)\left(\frac{x_{\text{cg}}-x_{MP}'}{\bar{c}}\right)\end{aligned} \tag{3.98}$$

or, since $Q\mathbf{C}_W = W/S$,

$$\frac{\partial F}{\partial n} = GS_e\bar{c}_e(W/S)\frac{\mathbf{C}_{L\alpha}'\mathbf{C}_{h\delta_e}}{\Delta}\left(1-\frac{\mathbf{C}_{Lq}}{2\mu}\right)\left(\frac{x_{\text{cg}}-x_{MP}'}{\bar{c}}\right) \tag{3.99}$$

The distance $\frac{x_{MP}'-x_{\text{cg}}}{\bar{c}}$, seen from the above equation to be directly related to the sensitivity of normal acceleration of the vehicle to control force, is called the *control free maneuver margin*.

Note that the control force gradient (per g) is

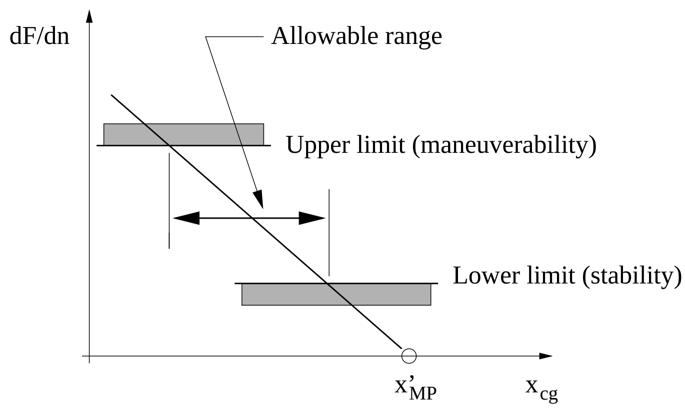


Figure 3.8: Allowable c.g. travel as imposed by limits on control force gradient (per g).

1. Directly proportional to the vehicle wing loading W/S;
2. Directly proportional to the *cube* of the linear size of the vehicle;
3. Directly proportional to the (control free) maneuver margin $(x'_{MP} - x_{\text{cg}})/\bar{c}$; and
4. Independent of airspeed.

The control force gradient should be neither too small nor too large. If the gradient is too small, the vehicle will be overly sensitive to small control inputs and it will be too easy for the pilot to over stress the airframe. At the same time, the control forces required for normal maneuvers must not be larger than the pilot can supply (or so large that the pilot becomes unduly tired performing normal maneuvers). The lower and upper limits on control force gradient (per g) determine allowable rearward and forward limits on c.g. travel, as sketched in Fig. 3.8. The values of these limits will depend on the vehicle mission; in general the limits will be higher for transport aircraft, and lower for vehicles which require greater maneuverability (such as military fighters or aerobatic aircraft).

3.4 Forward and Aft Limits of C.G. Position

The various control position and force gradients impose limits on the acceptable range of travel of the vehicle center of gravity. These include (for most vehicles):

- **Rearward limits:**
  1. The vehicle must be statically stable; i.e., the c.g. must be ahead of the basic and control free neutral points.
  2. The sensitivity of vehicle velocity to control position must not be too small; i.e., the c.g. must be sufficiently far ahead of the basic neutral point.
  3. The sensitivity of vehicle normal acceleration to control force must not be too small; i.e., the c.g. must be sufficiently far ahead of the control free neutral point.
- **Forward limits:**

1. The vehicle must be trimmable at $\mathbf{C}_{L\max}$; i.e., the c.g. must not be so far forward that there is insufficient elevator power to trim the vehicle at maximum lift coefficient.
2. The sensitivity of vehicle normal acceleration to control force must not be too high; i.e., the c.g. must not be so far forward that excessive control force is required to perform maneuvers for which the vehicle is intended.

UNIT I

	Questions	opt1	opt2	opt3	opt4	opt5	opt6	Answer
1	The Function of Aileron is used to obtain __________	longitudinal control	lateral control	directional control	Rolling control			longitudinal control
2	Rudder is fixed on the_________	horizontal stabilizer	vertical stabilizer	nose	cockpit			vertical stabilizer
3	Trim tabs are used under___________	neutral condition	balanced condition	unbalanced condition	motion			unbalanced condition
4	Servo tabs are similar to	trim tabs	spoilers	spring tabs	flaps			trim tabs
5	The forces and moments acting on an airplane is zero, then the airplane is in	danger	equilibrium	unbalanced condition	un equlibirium			equilibrium
6	Fuselage shape is	cylindrical	circle	square	cone			cylindrical
7	In propeller the power is converted into conventional rotational to produce	drag	thrust	lift	weight			thrust
8	Which in the following is a secondary control surface	aileron	rudder	trim tab	all the given			trim tab
9	Yawing movement is performed by	slats	flaps	aileron	rudder			rudder
10	Spoilers are used to decrease	wind turbulance	drag	moment	lift			lift
11	Hunter aircraft is a example of	single engine	multi engine	twin engine	glider			single engine
12	The body of the aircraft where the wings and tail are attached is called as	power plant	cabin	horizontal stabilizer	fuselage			fuselage
13	The aircraft power plant is used to produce	lift	drag	thrust	weight			thrust
14	The major component used for taxing in aircraft is	fuselage	landing gear	control stick	pitot static tube			landing gear
15	Which in the following landing gear does not exist.	single wheel	conventional	tricycle	tandem			single wheel
16	Conventional landing gear consist of	one wheel	two wheel	three wheel	four wheel			three wheel
17	Which of the landing gear has a nose wheel	tandem	tricycle	random	Bicycle			tricycle
18	The vehicle heavier than air and powered by an engine is called	Airship	Free ballon	Airplane	Capture ballon			Airplane
19	Rotating blade located on the front of the airplane is called	rudder	wing	engine	propeller			propeller
20	Which of the following comes under the classification of rotorcraft.	Land plane	Gyro plane	Sea plane	Glider			Gyro plane
21	To provide lift and thrust in helicopter the section involved is	Tail rotor	pylon	tail gear	main rotor			main rotor
22	Cargo airplane comes under the classification of	civil airplane	military airplane	agriculture airplane	cargo			civil airplane
23	Ambulance airplane comes under the category of	civil airplane	military airplane	agriculture airplane	rescue			rescue
24	Hunter aircraft has how many engines	1	2	3	4			1
25	Turbo jet engine is used in	Air bus	Boeing 747	F-16	F-15			F-16
26	Braced sesquiplane configuration is designed in which aircraft.	DC-3	AN-22	AN-2	ANN-2			AN-2
27	Delta wing has a shape of	rectangle	square	pentagon	Triangle			Triangle
28	Ground spoilers are used for	lateral control	longitudnal control	directional control	braking action			braking action
29	The movable outer surface of the airplane are	ribs	control surface	spars	control column			control surface
30	Amphibian plane comes under the classification of	rotorcraft	ornithopter	aeroplane	gyroplane			aeroplane
31	Aerodyne' type of aircraft can be	with or without engine	with engine only	without engine only	with one engine only			with or without engine
32	Gyrohorizon provides positive and direct indications of	altitude	velocity	pressure	attitude			attitude
33	The height of the airplane above the earth is indicated by the	radar altimeter	gyro horizon	vertical speed indicator	turn and bank indicator			radar altimeter
34	Instrument that measure pressure in relative high pressure fluid systems are called as	diaphragm	bellows	bourdon tube	Pitot tube			bourdon tube
35	Pressure gauges designed to provide readings of comparatively low pressure are usually called	diaphragm	static tube	bourdon tube	Dynamic tube			diaphragm
36	The height of the aircraft can be determined by	altimeter	variometer	air speed indicator	gyro horizon			altimeter
37	The speed of the aircraft is usually determined by the instrument	altimeter	air speed indicator	turn and bank indicator	vertical speed indicator			air speed indicator
38	The direction in which an aircraft is headed can be indicated by the	pitot static tube	electrical circuits	magnetic compass	altimeter			magnetic compass
39	The acceleration loads on the aircraft structure can be measured by the	variometer	accelerometer	vertical speed indicator	magnetic compass			accelerometer
40	The power for the operation of the landing gear retraction and extension can be given by	stable system	unstable system	pressure system	pneumatic system			pneumatic system
41	Fin is located on the	horizontal stabilizer	vertical stabilizer	wing	cockpit			vertical stabilizer
42	The body which protects the passengers in the event of a crash is called	black box	fuselage	wing tip	nose			fuselage
43	The aircraft power plant is usually enclosed in housing called a	nozzles	control engine	nacclle	cockpit			nacclle
44	The landing gear used in WRIGHT FLYER is	conventional gear	tricycle	skids	Fixed landing gear			skids
45	Conventional landing gear has two forward wheels and a third small wheel at the	fuselage	nose	mid fuselage	tail			tail
46	The tricycle landing gear has two main wheels and a	tail wheel	mid wheel	low wheel	nosewheel			nosewheel
47	The other name of wing is	airfoil	ribs	main plane	spars			main plane
48	The purpose of the main plane is to generate	lift	decrease fuel ratio	increase angle of attack	manage relative wind			lift
49	The movable sections in the horizontal stabilizer of airplane is	rudder	aileron	propeller	elevator			elevator
50	The tail plane is also known as	vertical stabilizer	horizontal stabilizer	aileron	flaps			horizontal stabilizer
51	The vertical stabilizer for an airplane is the airfoil section forward of the	elevator	trim tab	rudder	aileron			rudder
52	The body which is mounted on the trailing edge of the wing near the wing tip	aileron	flaps	slats	rudder			aileron

53	Rudder are usually balanced statically and	in weight	aerodynamically	vectorly	in shape			aerodynamically
54	The body normally attached to hinges on the rear spar of the horizontal stabilizer	aileron	rudder	elevator	flaps			elevator
55	Undercarriage is the other name of	low fuselage	mid fuselage	landing gear	engine gear			landing gear
56	Which of the following does not come under primary control surface	aileron	elevator	rudder	flaps			flaps
57	The yawing movement takes place in the	normal axis	lateral axis	longitudnal axis	vertical axis			vertical axis
58	Airship comes under the classification of	lighter than air	heavier than air	glider	Free ballon			lighter than air

UNIT II

59	_____ is a wing platform with a wing root to wingtip direction angled beyond the span wise axis, generally used to delay the drag rise caused by fluid compressibility	delta wing	Straight wing	expansion	Swept wings			Swept wings
60	______as a means of reducing wave drag were first used on jet fighter aircraft	delta wing	Straight wing	eliptical wing	eliptical wing			Swept wings
61	The four-engine propeller-driven ______ aircraft has swept wings.	A-10	A380	Swept wings	TU-95			TU-95
62	The ______ is that free-stream Mach number at which sonic flow is first encountered on the airfoil.	subcritical	critical Mach number	pushpak	supercritical			critical Mach number
63	The_______gradient induced by the shock tends to separate the boundary layer on the top surface, causing a large pressure drag	adverse pressure	reverse pressure	Drag divergence	high pressure			adverse pressure
64	________is the Mach number at which the aerodynamic drag on an airfoil or airframe begins to increase rapidly as the Mach number continues to increase	critical	drag divergence	favourable pressure	supercritical			drag divergence
65	can cause the drag coefficient to rise to more than ten times its low speed value.	subcritical	supercritical	subcritical	drag divergence			drag divergence
66	The value of the drag divergence Mach number is typically greater than	0.3	1	critical	2			0.6
67	A turboprop engine is a turbine that drives a propeller through a	landing gear	wheel gear	reduction gear	turbine gear			reduction gear
68	Turbofans were developed to combine some of the best features of the turbojet and the	turboprop	turbofan	ramjet	scramjet			turboprop
69	The propeller mounted on the front of the engine translates the rotating force of the engine into	lift	thrust	weight	power			thrust
70	A propeller is a rotating airfoil which produces thrust through	propulsion force	normal force	drag force	aerodynamic force			aerodynamic force
71	The powerplant usually includes both engine and the	blades	fuel	propeller	cockpit			propeller
72	The propeller may also be mounted on the rear of the engine as in a	puller type aircraft	pusher-type aircraft	rotating -type aircraft	All of the given			pusher-type aircraft
73	The reciprocating engine is also known as	propeller engine	combustion engine	external combustion engine	internal combustion engine			internal combustion engine
74	The propeller has the general shape of the	nose	engine	wing	fuselage			wing
75	The wing has only forward motion but propeller has forward and	backward motion	side motion	running motion	rotary motion			rotary motion
76	The dissipation of Energy to reduce the disturbance is called ______.	Negative damping	Aileron power	Positive damping	All of the given			Positive damping
77	For high value of lift co-efficient, to have a longitudinal static stability, C.G of an airplane should be _________ to a.c of wing	Below	Above	On	Any where			Below
78	Upwash before the wing is due to ______.	Bound vortex	Tailing vortex	Friction	Pressure distribution			Bound vortex
79	Tail contribution to longitudinal stability is greatly affected by __________ on the wing.	Upwash	Downwash	Wing interference	Both b & c			Downwash
80	Rotation of an airplane about longitudinal axis is called ______.	Rolling	Pitching	Yawing	All of the given			Rolling
81	To have some static longitudinal stability C.G should always located _______ the/of Neutral point.	After	Ahead	On	Any where			Ahead
82	Aileron is used to control __________ stability of an airplane.	longitudinal	lateral	directional	All of the given			lateral
83	If the airplanes wing axis is inclined downward to the horizontal plane, then it is called	Anhedral	Dihedral	Wing twist	All of the given			Dihedral
84	Frise ailerons are used to minimize ______.	Longitudinal stability	lateral stability	adverse yaw	rolling			adverse yaw
85	Which control surfaces provide directional and pitch control?	Elerons	Ruddervators	Tailerons	wing			Ruddervators
86	What control surface movements will make an aircraft to yaw left	aileron	elevator	rudder	flap			rudder
87	An aircraft in longitudinally stability will tend to its level flight about which axis?	Adverse yaw	Roll	Yaw	Pitch			Pitch
88	What is the collective term for the fin and rudder and other surfaces aft of the centre of gravity that helps directional stability?	Fuselage surfaces	Empennage	Effective keel surface	Horizontal stabilizer			Effective keel surface
89	Four forces of flight are______.	Lift, gravity, T & D	W, gravity, T & D	L,W, gravity& D	L,W,acceleration &D			Lift, gravity, T & D
90	The most dangerous Pilot Induced Oscillation occurs during ______.	landing	Take-off	climbing	All of the given			landing
91	Which is the advantage of high aspect ratio wing?	increased Dinduced	decreased Dinduced	decreased Dskin friction	Dinduced remains constant			decreased Dinduced
92	Sweepback of the wings will ______ lateral stability	increase	decrease	not affect	make constant			increase
93	Lateral stability may be increased with ______.	increased lateral dihedral	increased lateral anhedral	increased longitudinal dihedral	decreased lateral dihedral			increased lateral dihedral
94	Directional stability is about ______.	normal axis	longitudinal axis	lateral axis	Horizontal axis			normal axis
95	The fin makes stability about ______.	Lateral axis	Normal axis	Longitudinal axis	All of the given			All of the given
96	Disturbance for lateral degrees of freedom is ________.	Angle of attack	Side slip	Both a & b	Location of C.G			Side slip
97	To have a longitudinal static stability for a propeller driven engine, propeller shout be placed ________ C.G.	On the	Ahead of	after	Anywhere from			after
98	If C.G of an aircraft is ahead of No, then the aircraft is_________.	Statically stable	Statically unstable	Neutrally stable	All of the given			Statically stable
99	The ability of the system not to return its original state after some disturbance is called _________.	stable	unstable	neutral	zero			unstable
100	Yawing motion of an airplane is controlled by ________.	Aileron	Rudder	Elevator	flap			Rudder
101	Rudder is used to control ___________ stability of an airplane.	longitudinal	lateral	directional	All of the given			directional
102	In One engine inoperative condition, the design of _________ on multi-engine a/c is critical.	aileron	elevator	compressor	rudder			rudder
103	The movement of C of G in flight is due to ________.	movement of passengers	movement of C of P	consumption of fuels and oils	All of the given			consumption of fuels and oils
104	The decrement in L of left wing results in _______.	left roll	spin	auto rotation	right roll			right roll
105	Non-oscillatory divergent motion occurs at _______.	small directional &large lateral stability	small directional & small lateral stability	large directional&	None of the given			large directional&
106	Long period with poor damping in stick fixed longitudinal motion is known as _______.	weather cocking	phugoid mode	dutch roll	hinge moment			phugoid mode
107	The "wing setting angle" is commonly known as _______.	angle of incidence	angle of attack	dihedral angle	anhedral angle			angle of incidence
108	In a bank and turn _______.	extra L is not required	extra T is required	extra lift is required	Lift is independent one			extra lift is

UNIT III

109	__________theory is frequently linear theory	linearised flow theory	Small-perturbation	cartesian	Compressible Flow theory			Small-perturbation
110	The ______ where the assumption of small perturbations allowed a linearized solution	Small-perturbation	Compressible Flow theory	Small-perturbation	linearised flow theory			acoustic theory
111	___________for a symmetrical airfoil in supersonic flow is predicted at the mid-chord point	Aerodynamic center	center of pressure	acoustic theory	coefficient of drag			center of pressure
112	____________ is a design technique used to reduce an aircraft's drag at transonic	Whitcomb area rule	area rule	coefficient of pressure	thick aerofoil			Whitcomb area rule
113	_______ is a design technique used to reduce an aircraft's drag at transonic and supersonic speeds, particularly between Mach 0.75 and 1.2	subsonic area rule	subcritical	thumbrule	transonic area rule			transonic area rule
114	To reduce the number of these shock waves, an aerodynamic shape should change in cross sectional area as smoothly as possible. This leads to a perfect aerodynamic shape known as the	shark lets	Sears-Haack body	Supercritical	winglet			Sears-Haack body

115	The Mach number in the test section of blow down tunnel is determined by pressure and temperature in the	settling chamber	diaphragm	Wing tip vortices	filter			plenum
116	Test times are limited in ____________ wind tunnels	suction	blowdown	plenum	indraft tunnels			blowdown
117	A ______ is often employed downstream of the test section to shock down the supersonic flow to subsonic before entering the low pressure chamber.	second throat	first throat	subsonic	settling chamber			second throat
118	A closed configuration with both high pressure and low pressure chambers is shown in the figure, but there are other configurations of blowdwon tunnels. Some blowdown tunnels, called	suction	blowdown	third throat	subsonic			indraft tunnels
119	High subsonic wind tunnels opertated at	(1.2<M<5)	(0.4 < M < 0.75)	indraft tunnels	(0.75 < M < 1.2)			(0.4 < M < 0.75)
120	transonic wind tunnels opertated at	(1.2<M<5)	(0.4 < M < 0.75)	M=1	(0.75 < M < 1.2)			(0.75 < M < 1.2)
121	A supersonic wind tunnel is a wind tunnel that produces supersonic speeds	(1.2<M<5)	(0.4 < M < 0.75)	M=1	M=1			(1.2<M<5)
122	__________ have short test times (usually less than one second), relatively high Reynolds number, and low power requirements	Ludwieg tube	shock tubes	(0.75 < M < 1.2)	bourdan tube			Ludwieg tube
123	Stagnation temperatures of at pressures of several hundred atmospheres provide test Mach numbers from 6 to 15 for run durations on the order of 1 minute	3500° F	3500°C	density tube	2500° F			3500° F
124	allow the study of fluid flow at temperatures and pressures that would be difficult to obtain in wind tunnels	pitot tube	Shock tubes	1500° F	bourdan tube			Shock tubes
125	Aerodynamics of a spinning cricket ball is related to	Bernoulli's principle	Magnus effect	density tube	Newton's second law			Magnus effect
126	Velocity potential is valid for	Viscous flow	Real flow	Kutta condition	Irrotational flow			Irrotational flow
127	Streamlined body is one for which	Pressure drag is more than skin friction drag	Induced drag is more than profile drag	Rotational flow	All of the above			Skin friction drag is more than ressure drag
128	Stalling in an incompressible flow is due to	sudden expansion	flow separation	Skin friction drag is more than ressure drag	Isentropic expansion			flow separation
129	Lifting flow over circular cylinder is obtained by the combination of	Uniform flow + source + vortex	Uniform flow + sink + vortex	Adiabatic compression	Uniform flow + doublet + vortex			Uniform flow + doublet + vortex
130	NACA 0014 implies the airfoil is	Symmetric	Positively cambered	Source + Sink + Uniform flow	Cusped			Symmetric
131	Kutta-Joukowski theorem gives the dependence of lift per unit span on	Total pressure	Temperature	Negatively cambered	All of the above			Circulation
132	Aerodynamic center of an airfoil is the point about which	Pitching moment is zero	Pitching moment is constant	Circulation	Pitching moment is negative			Pitching moment is zero
133	Sound propagation is	Isothermal process	Isentropic process	Pitching moment is positive	Isochoric process			Isentropic process
134	When the Mach number ahead of a normal shock is Infinity the Mach number behind the normal shock is	Infinity	High supersonic	Isobaric process	Low subsonic			Low subsonic
135	The lowest value of shock angle for oblique shocks is	Zero	12.5 deg	Zero	115 deg			Mach angle
136	The maximum possible turning angle through Prandtl_Meyer expansion is	130.5 deg	180 deg	Mach angle	115 deg			130.5 deg
137	When the Mach number ahead of a normal shock is infinity the ratio of static density before and after the normal shock is	infinity	Finite	145 deg	Zero			Finite
138	For supersonic flow of Mach number = 2 flowing over a compression corner of turning angle nearly equal to zero the shock angle is	Zero	45 deg	Inversely proportional to the square of the pressure	12.5 deg			30 deg
139	Transonic area rule is applied to the following component of airplane	Wings	Tail	30 deg	Power plants			Fuselage
140	Prandtl's relation for a normal shock is an equation consisting of	Characteristic Mach numbers	Static pressures	Fuselage	density			Characteristic Mach numbers
141	The maximum possible value of Characteristic Mach number is	Infinity	10.58	Total pressures	1			2.45
142	Mach angle is the lowest possible value of	Flow turning angle	shock angle	2.45	Characteristic Mach numbers			shock angle
143	Prandtl-Glauert rule gives the relation between	Viscous and inviscid flow	line integral and surface integral	Angle of incidence	incompressible and compressible flow characteristics			incompressible and compressible flow characteristics
144	Small perturbation theory is applicable for	circular cylinders	slender bodies	Streamlines and equipotential lines	incompressible and compressible flow characteristics			slender bodies
145	In free vortex flow the tangential velocity is	directly proportional to radial distance	inversely proportional to radial distance	spherical bodies	Zero			inversely proportional to radial distance
146	Incompressible inviscid flow can be represented by	second order polynomial	Fourier series	independent of radial distance	line integral and surface integral			Laplace equation
147	Curl of velocity vector is	Acceleration	normal component of velocity	Laplace equation	momentum			Vorticity
148	Viscosity of gases	increases with temperature	decreases with increasing temperature	Vorticity	constant with increasing temperature			increases with temperature
149	Stream function is related to	Volume flow rate	Circulation	is independent of temperature	momentum			Volume flow rate
150	Sweep back results in	less directional stability	less longitudinal stability	Angular velocity	higher directional stability			higher directional stability
151	Supercritical airfoils are characterized by	sharp leading edge	highly cambered upper surface	stronger longitudinal stability	conical upper surface			flattened upper surface
152	A compressible fluid when brought to rest generates greater pressure than an	orthotropic fluid	incompressible fluid	flattened upper surface	isotropic fluid			incompressible fluid
153	Air at lower density is more compressible than air at higher density and therefore compressibility error increases with	increase in altitude	decrease in altitude	barotropic fluid	increase in pressure			increase in altitude
154	is lighter than air and it is present only in the lower layers of the atmosphere.	moisture	liquid	decrease in pressure	Water vapour			Water vapour
155	The speed of sound is directly proportional to the	absolute viscosity	Kinematic viscosity	ice	absolute pressure			square root of the absolute temperature
156	is the ratio of absolute viscosity to density	absolute viscosity	Kinematic viscosity	square root of the absolute temperature.	square root of the absolute temperature.			Kinematic viscosity
157	is a point on the aerofoil chord line through which the resultant aerodynamic force acts.	Centre of pressure	coefficient of pressure	absolute pressure	coefficient of momentum			Centre of pressure
158	is that fixed point on the aerofoil around which the coefficient of pitching moment is a constant	coefficient of momentum	Aerodynamic centre	Aerodynamic centre	coefficient of pressure			Aerodynamic centre
159	The sum of the static and dynamic pressure is called total head pressure and it remains ___________.	constant	decrease	Centre of pressure	greater			constant
160	_________ would form only when the wing is producing lift and would disappear when the wing is not producing lift	Drag	wash in	increase	washout			Wing tip vortices
161	Zero lift drag___________ because of the forward movement of the transition and separation points with increase in lift.	constant	decrease	Wing tip vortices	greater			increases
162	is a layer of retarded air in contact with the surface of the wing	Boundary layer	displacement	increase	compressibility			Boundary layer
163	_____ drag is caused due to the effect of the boundary layer and it increases with increase in speed.	induced	Skin friction	viscosity	interference			Skin friction
164	________ method of preventing wing tips stalling on swept back wings	wing lets	sharklets	pressure	canard			Boundary layer fences
165	Boundary layer control also reduces____________drag	induced	skin friction	Boundary layer fences	interference			skin friction

UNIT IV

166	On a swept wing aircraft if both wing tip sections lose lift simultaneously the aircraft will	roll	pitch nose up	pitch nose down	Yaw			pitch nose up
167	Lift on a delta wing aircraft	increases with an increased angle of incidence (angle of attack)	decreases with an increase in angle of incidence (angle of attack)	does not change with a change in angle of incidence (angle of attack)	increases with an increased angle of incidence upto Stall			increases with an increased angle of incidence (angle of attack)
168	On a straight wing aircraft, stall commences at the	root on a low thickness ratio wing	tip on a high thickness ratio wing	tip on a low thickness ratio wing	root on a high thickness ratio wing			root on a high thickness ratio wing
169	For the same angle of attack, the lift on a delta wing	is greater than the lift on a high aspect ratio wing	is lower than the lift on a high aspect ratio wing	is the same as the lift on a high aspect ratio wing	is greater than the lift on a low aspect ratio wing			is lower than the lift on a high aspect ratio wing
170	The ISA	is taken from the equator	is taken from 45 degrees latitude	is taken from 30 degrees latitude	is taken from 60 degrees latitude			is taken from 45 degrees latitude
171	At higher altitudes as altitude increases, pressure	decreases at constant rate	increases exponentially	remains constant	decreases exponentially			decreases exponentially
172	When the pressure is half of that at sea level, what is the altitude?	12,000 ft	8,000 ft	10,000 ft	18,000 ft			18,000 ft
173	During a turn, the stalling angle	increases with AOA	decreases	remains the same	increases with an increased angle of incidence upto Stall			remains the same
174	The C of G moves in flight. The most likely cause of this is	movement of passengers	movement of the centre of pressure	consumption of fuel and oils	altitude			consumption of fuel and oils
175	The C of P is the point where	all the forces on an aircraft act	the three axis of rotation meet	the lift can be said to act	CG Point			the lift can be said to act
176	The three axis of an aircraft act through the	C of G	C of P	stagnation point	Chord line			C of G
177	Pressure decreases	proportionally with a decreases in temperature	inversely proportional to temperature	Pressure and temperature are not related	proportionally with a increase in temperature			proportionally with a decreases in temperature
178	As air gets colder, the service ceiling of an aircraft	reduces	increases	remains the same	becomes zero			increases
179	When the weight of an aircraft increases, the minimum drag speed	decreases	increases	increases upto stall	remains the same			increases
180	An aircraft will have	less gliding distance if it has more payload	more gliding distance if it has more payload	the same gliding distance if it has more payload	more gliding distance if it has less payload			the same gliding distance if it has more payload
181	When an aircraft experiences induced drag	air flows under the wing span-wise towards the tip and on top of the wing spanwise towards the root	air flows under the wing span-wise towards the root and on top of the wing span-wise towards the tip	air flows under the wing span-wise towards the tip	air flows on top of the wing spanwise towards the root			air flows under the wing span-wise towards the tip and on top of the wing spanwise towards the root
182	At stall, the wingtip stagnation point	moves toward the lower surface of the wing	moves toward the upper surface of the wing	moves toward the lower wing tip	moves toward the upper wing tip			moves toward the lower surface of the wing
183	The rigging angle of incidence of an elevator is	the angle between the mean chord line and the horizontal in the rigging position	the angle between the bottom surface of the elevator and the horizontal in the rigging position	the angle between the bottom surface of the elevator and the longitudinal datum	the angle between the bottom surface of the elevator and the lateral datum			the angle between the mean chord line and the horizontal in the rigging position
184	What is the lapse rate with regard to temperature?	0.98°C per 1000 ft	1.98°F per 1000 ft	4°C per 1000 ft	1.98°C per 1000 ft			1.98°C per 1000 ft
185	What happens to load factor as you decrease turn radius?	It increases	It decreases	It remains constant	load factor is not related to turn radius			It increases
186	If you steepen the angle of a banked turn without increasing airspeed or angle of attack, what will the aircraft do?	It will remain at the same height	It will sideslip with attendant loss of height	It will stall	It will decent			It will sideslip with attendant loss of height
187	An aircraft wing tends to stall first at	the tip due to a higher ratio thickness/chord	the tip due to a lower ratio thickness/chord	the root due to a higher ratio thickness/chord	the root due to a lower ratio thickness/chord			the root due to a higher ratio thickness/chord
188	Dihedral wings combat instability in	pitch	yaw	roll	sideslip			sideslip
189	To stop aircraft decreasing in height during a sideslip, the pilot can	advance the throttle	pull back on the control column	adjust the rudder position	adjust the elevator position			advance the throttle
190	What control surface movements will make an aircraft fitted with ruddervators yaw to the left?	Left ruddervator lowered, right ruddervator raised	Right ruddervator lowered, left ruddervator raised	Both ruddervators raised	Both ruddervators lowered			Left ruddervator lowered, right ruddervator raised
191	When a leading edge slat opens, there is a gap between the slat and the wing. This is	to allow it to retract back into the wing	to allow air through to re-energize the boundary layer on top of the wing	to keep the area of the wing the same	to change the area of the wing			to allow air through to re-energize the boundary layer on top of the wing
192	Which of the following is true?	Lift acts at right angles to the wing chord line and weight acts vertically down	Lift acts at right angles to the relative airflow and weight acts vertically down	Lift acts at right angles to the relative airflow and weight acts at right angles to the aircraft centre line	Lift acts at right angles to the chord line and weight acts at right angles to the aircraft centre line			Lift acts at right angles to the relative airflow and weight acts vertically down
193	If the wing tips stall before the root on a swept wing aircraft, the aircraft will	roll	pitch nose up	pitch nose down	Yaw			pitch nose up
194	Standard sea level temperature is	0 degrees Celsius	15 degrees Celsius	20 degrees Celsius	22 degrees Celsius			15 degrees Celsius
195	As altitude increases, pressure	decreases at constant rate	increases exponentially	decreases exponentially	Remains constant			decreases exponentially
196	Lapse rate usually refers to	Pressure	Density	Temperature	altitude			Temperature
197	The vertical fin of a single engined aircraft is	parallel with both the longitudinal axis and vertical axis	parallel with the longitudinal axis but not the vertical axis	parallel with the vertical axis but not the longitudinal axis	Perpendicular with both the longitudinal axis and vertical axis			parallel with the vertical axis but not the longitudinal axis
198	Aircraft flying in the transonic range most often utilize	sweptback wings	advanced supercritical airfoils	high wings	delta wings			sweptback wings
199	Which type of flap changes the area of the wing?	Fowler	Split	Slotted	plain			Fowler
200	Forward swept wings tend to stall at the root first so the aircraft retains lateral control, so why are they never used on passenger aircraft?	Because the wing tips wash in at high wing loads	Because the wing tips wash out at high wing loads	Because at high loads their angle of incidence increases	Because at high loads their angle of incidence decreases			Because at high loads their angle of incidence increases

201	What happens to air flowing at the speed of sound when it enters a converging duct?	Velocity decreases, pressure and density increase	Velocity increases, pressure and density decreases	Velocity, pressure and density remains constant	Velocity, pressure and density increase			Velocity decreases, pressure and density increase
202	As the angle of attack of an airfoil increases the centre of pressure	moves forward	moves aft	remains stationary	moves towards CG			moves forward
203	An aircraft, which is longitudinally stable, will tend to return to level flight after a movement about which axis?	Pitch	Roll	Yaw	all three axis			Pitch
204	Vapour trails from the wingtips of an aircraft in flight are caused by	low pressure above the wing and high pressure below the wing causing vortices	high pressure above the wing and low pressure below the wing causing vortices	low pressure above the wing and high pressure below the wing causing a temperature rise	low pressure above the wing and low pressure below the wing causing a temperature rise			low pressure above the wing and high pressure below the wing causing vortices
205	Vortex generators on the wing are most effective at	high speed	low speed	low angles of attack	high angles of attack			high angles of attack
206	The chord line of a wing is a line that runs from	the centre of the leading edge of the wing to the trailing edge	half way between the upper and lower surface of the wing	one wing tip to the other wing tip	camber line			the centre of the leading edge of the wing to the trailing edge
207	The angle of incidence of a wing is an angle formed by lines	parallel to the chord line and longitudinal axis	parallel to the chord line and the lateral axis	parallel to the chord line and the vertical axis	perpendicular to the chord line and the lateral axis			parallel to the chord line and longitudinal axis
208	The centre of pressure of an aerofoil is located	30 - 40% of the chord line back from the leading edge	30 - 40% of the chord line forward of the leading edge	50% of the chord line back from the leading edge	10% of the chord line back from the leading edge			30 - 40% of the chord line back from the leading edge
209	Compressibility effect is	drag associated with the form of an aircraft	drag associated with the friction of the air over the surface of the aircraft	the increase in total drag of an aerofoil in transonic flight due to the formation of shock waves	the increase in total drag of an aerofoil insubsonic flight due to the formation of shock waves			the increase in total drag of an aerofoil in transonic flight due to the formation of shock waves
210	Lateral control of an aircraft at high angle of attack can be maximised by using	fences	vortex generators	wing slots	flaps			vortex generators
211	Stall strips are always	on the trailing edge of a wing	on the leading edge of a wing	fitted forward of the ailerons	fitted aft of the ailerons			on the leading edge of a wing
212	Due to the interference of the airflow on a high wing aircraft between the fuselage and the wings, the lateral stability of the aircraft in a gusty wind situation will cause	the upper wing to increase its lift	the upper wing to decrease its lift	the lower wing to decrease its lift	the lower wing to increase its lift			the upper wing to decrease its lift
213	Slats	reduce the stall speed	reduce the tendency of the aircraft to Yaw	decrease the aerofoil drag at high speeds	decrease lift			reduce the stall speed
214	A high aspect ratio wing will give	high profile and low induced drag	low profile and high induced drag	low profile and low induced drag	high profile and high induced drag			high profile and low induced drag
215	Aerofoil efficiency is defined by	lift over drag	drag over lift	lift over weight	drag over weight			lift over drag
216	An aircraft banks into a turn. No change is made to the airspeed or angle of attack. What will happen?	The aircraft enters a sideslip and begins to lose altitude	The aircraft turns with no loss of height	The aircraft yaws and slows down	The aircraft begins to gain altitude			The aircraft enters a sideslip and begins to lose altitude
217	The relationship between induced drag and airspeed is, induced drag is	directly proportional to the square of the speed	inversely proportional to the square of the speed	directly proportional to speed	inversely proportional to speed			inversely proportional to the square of the speed
218	What is Boundary Layer?	Separated layer of air forming a boundary at the leading edge	Turbulent air moving from the leading edge to trailing edge	low energy air that sticks to the wing surface and gradually gets faster until it joins the free stream flow of air	Separated layer of air forming a boundary at the trailing edge			low energy air that sticks to the wing surface and gradually gets faster until it joins the free stream flow of air
219	The normal axis of an aircraft passes through	the centre of gravity	a point at the centre of the wings	at the centre of pressure	Chord line			the centre of gravity
220	On a high winged aircraft, what effect will the fuselage have on the up-going wing?	The up-going wing will have a decrease in angle of attack and therefore a decrease in lift	The down-going will have a decrease in angle of attack and therefore a decrease in lift	The up-going wing will have an increase in angle of attack and therefore a	The up-going wing will have an decrease in angle of attack and therefore a			The up-going wing will have a decrease in angle of attack and therefore a decrease in lift
221	What is the collective term for the fin and rudder and other surfaces aft of the centre of gravity that helps directional stability?	Effective keel surface	Empennage	Fuselage surfaces	ruddervators			Effective keel surface
222	Temperature above 36,000 feet will	decrease exponentially	remain constant	increase exponentially	Increses at 1 degree for 1000 feet			remain constant
223	A decrease in incidence toward the wing tip may be provided to	prevent adverse yaw in a turn	prevent span-wise flow in manoeuvres	retain lateral control effectiveness at high angles of attack	prevent yaw in a turn			retain lateral control effectiveness at high angles of attack
224	The angle of attack which gives the best L/D ratio	decreases with a decrease in density	in unaffected by density changes	increases with a decrease in density	decreases with a increase in density			in unaffected by density changes
225	For a given aerofoil producing lift, where P = pressure and V = velocity:	P1 is greater than P2, and V1 is greater than V2	P1 is less than P2 and V1 is greater than V2	P1 is greater than P2, and V1 is less than V2	P1 ,P2, and V1 , V2 remain constant			P1 is greater than P2, and V1 is less than V2
	UNIT V							
226	Low wing loading	increases stalling speed, landing speed and landing run	increases lift, stalling speed and manoeuvrability	decreases stalling speed, landing speed and landing run	decreases lift, stalling speed and manoeuvrability			decreases stalling speed, landing speed and landing run
227	Due to the change in downwash on an un-tapered wing (i.e. one of constant chord length) it will	not provide any damping effect when rolling	tend to stall first at the root	not suffer adverse yaw effects when turning	provide damping effect when rolling			tend to stall first at the root
228	True stalling speed of an aircraft increases with altitude	because reduced temperature causes compressibility effect	because air density is reduced	because humidity is increased and this increases drag	because increased temperature causes compressibility effect			because air density is reduced
229	As a general rule, if the aerodynamic angle of incidence (angle of attack) of an aerofoil is slightly increased, the centre of pressure will	never move	move towards the root	move towards the tip	move forward towards the leading edge			move forward towards the leading edge
230	The "wing setting angle" is commonly known as	angle of incidence	angle of attack	angle of dihedral	angle of andedral			angle of incidence

231	On a very humid day, an aircraft taking off would require	a shorter take off run	a longer take off run	humidity does not affect the take off run	high air intake			a longer take off run
232	An aircraft is flying at 350 MPH, into a head wind of 75 MPH, what will its ground speed be?	175 mph	350 mph	200 mph	275 mph			275 mph
233	When does the angle of incidence change?	When the aircraft attitude changes	When the aircraft is descending	It never changes	When the aircraft is ascending			It never changes
234	As the angle of attack decreases, what happens to the centre of pressure?	It moves forward	It moves rearwards	Centre of pressure is not affected by angle of attack decrease	increases			It moves rearwards
235	A decrease in pressure over the upper surface of a wing or aerofoil is responsible for	approximately 2/3 (two thirds) of the lift obtained	approximately 1/3 (one third) of the lift obtained	approximately ½ (one half) of the lift obtained	approximately twice of the lift obtained			approximately 2/3 (two thirds) of the lift obtained
236	Which of the four forces act on an aircraft?	Lift, gravity, thrust and drag	Weight, gravity, thrust and drag	Lift, weight, gravity and drag	Lift, weight, gravity and thrust			Lift, gravity, thrust and drag
237	Which of the following types of drag increases as the aircraft gains altitude?	Parasite drag	Induced drag	Interference drag	wave drag			Induced drag
238	The layer of air over the surface of an aerofoil which is slower moving, in relation to the rest of the airflow, is known as	camber layer	boundary layer	chord layer	skin layer			boundary layer
239	What is a controlling factor of turbulence and skin friction?	Aspect ratio	Fineness ratio	Counter-sunk rivets used on engine	Counter-sunk rivets used on skin exterior			Counter-sunk rivets used on skin exterior
240	Changes in aircraft weight	will not affect total drag since it is dependant only upon speed	will not affect total lift since it is dependant only upon speed	will only affect total drag if the lift is kept constant	cause corresponding changes in total drag due to the associated lift change			cause corresponding changes in total drag due to the associated lift change
241	The aircraft stalling speed will	increase with an increase in weight	be unaffected by aircraft weight changes since it is dependant upon the angle of attack	increase with an decrease in weight	decrease with an increase in weight			increase with an increase in weight
242	In a bank and turn	extra lift is not required	extra lift is not required if thrust is increased	extra thrust is not required	extra lift is required			extra lift is required
243	To achieve the maximum distance in a glide, the recommended air speed is	as close to the stall as practical	as high as possible with VNE	the speed where the L/D ratio is maximum	the speed where the L/D ratio is minimum			the speed where the L/D ratio is maximum
244	If the C of G is aft of the Centre of Pressure	changes in lift produce a pitching moment which acts to increase the change in lift	when the aircraft sideslips, the C of G causes the nose to turn into the sideslip thus applying a restoring moment	when the aircraft yaws the aerodynamic forces acting forward of the Centre of Pressure	when the aircraft rolls the aerodynamic forces acting forward of the Centre of Pressure			changes in lift produce a pitching moment which acts to increase the change in lift
245	Porpoising is an oscillatory motion in the	pitch plane	roll plane	yaw plane	all three planes			pitch plane
246	Due to the interference effects of the fuselage, when a high wing aeroplane sideslips	the accompanying rolling due to keel surface area is destabilizing	the accompanying lift changes on the wings produces a stabilizing effect	the accompanying rolling due to the fin is destabilizing	the accompanying drag changes on the wings produces a stabilizing effect			the accompanying lift changes on the wings produces a stabilizing effect
247	The power required in a horizontal turn	is greater than that for level flight at the same airspeed	must be the same as that for level flight at the same airspeed	is less than that for level flight at the same airspeed	is less than that for level flight at the same altitude			is greater than that for level flight at the same airspeed
248	A wing mounted stall sensing device is located	usually on the under surface	always at the wing tip	always on the top surface	always on empennage			usually on the under surface
249	For an aircraft in a glide	thrust, drag, lift and weight act on the aircraft	weight, lift and drag act on the aircraft	weight and drag only act on the aircraft	weight, lift and thrust act on the aircraft			weight, lift and drag act on the aircraft
250	The upper part of the wing in comparison to the lower	develops more drag	develops the same lift	develops less lift	develops more lift			develops more lift
251	What effect would a forward CG have on an aircraft on landing?	Increase stalling speed	No effect on landing	Reduce stalling speed	Reduce ground speed			Increase stalling speed
252	An aspect ratio of 8 would mean	span 64, mean chord 8	mean chord 64 , span 8	span squared 64 ,chord 8	span squared 4 ,chord 8			span 64, mean chord 8
253	If an aircraft in level flight loses engine power it will	pitch nose up	pitch nose down	not change pitch without drag increasing	not change pitch without drag decreasing			pitch nose down
254	The lift /drag ratio at stall	increases	decreases	remains constant	Remains constant upto stalling point			decreases
255	On a straight unswept wing, stall occurs at	the thick portion, at the wing root	the thick portion, at the wing tip	the thin portion, at the wing tip	the thick portion, at the wing tip			the thick portion, at the wing root
256	During a climb from a dive	the thrust required is greater than required for level flight	the thrust required is lower than for level flight	the thrust required is the same as for level flight	the thrust required is equal to thrust available			the thrust required is lower than for level flight
257	When power is off, the aircraft will pitch	nose down	nose up	trim level	remains constant			nose down
258	Angle of attack on a down going wing in a roll	is zero	decreases	is unaffected	increases			increases
259	For any given speed, a decrease in aircraft weight, the induced drag will	increase	decrease	remain the same	be zero			decrease
260	The amount of lift generated by a wing is	greatest at the root	greatest at the tip	constant along the span	constant along the chord			greatest at the root
261	Induced Drag is	greatest towards the tip	greatest towards towards the root	decreased from tip to root	increased from tip to root			greatest towards the tip
262	Induced Drag is	equal to profile drag at stalling angle	equal to profile drag	greater than profile drag	less than profile drag			equal to profile drag
263	For a given IAS an increase in altitude will result in	no change in the value of induced drag	an increase in induced drag	an increase in profile drag	an increase in skin friction drag			an increase in induced drag
264	Stall inducers may be fitted to a wing	at the tip to cause the root to stall first	at the root to cause the tip to stall first	at the tip to cause the tip to stall first	at the root to cause the root to stall first			at the root to cause the root to stall first
265	With increasing altitude pressure decreases and	temperature decreases at the same rate as pressure reduces	temperature decreases but at a lower rate than pressure reduces	temperature remains constant to 8000ft	temperature remains constant			temperature decreases but at a lower rate than pressure reduces

266	Krueger Flap	increases the leading edge camber	allows high pressure air from beneath the wing to flow to the top	energizes the air flowing over the ailerons	increases the trailing edge camber			increases the leading edge camber
267	Which conditions will give the shortest take off distance?	Hot Humid day at high elevation	rainy day at sea level	Hot summer day at sea level	Cold winter day at sea level			Cold winter day at sea level
268	The optimum angle of attack of an aerofoil is the angle at which	the aerofoil produces maximum lift	the aerofoil produces zero lift	the highest lift/drag ratio is produced	the lowest lift/drag ratio is produced			the highest lift/drag ratio is produced
269	A high aspect ratio wing has a	increased induced drag	decreased induced drag	decreased skin friction drag	increased skin friction drag			decreased induced drag
270	Minimum total drag of an aircraft occurs	at the stalling speed	when profile drag equals induced drag	when induced drag is least	when wave drag is least			when profile drag equals induced drag
271	If the weight of an aircraft is increased, the induced drag at a given speed	will increase	will decrease	will remain the same	will remain the same upto 8000 ft			will increase
272	The transition point on a wing is the point where	the flow separates from the wing surface	the boundary layer flow changes from laminar to turbulent	the flow divides to pass above and below the wing	the boundary layer flow changes from turbulent to laminar			the boundary layer flow changes from laminar to turbulent
273	The boundary layer of a body in a moving air stream is	a thin layer of air over the surface where the air is stationary	a layer of separated flow where the air is turbulent	a layer of air over the surface where the airspeed is changing from free stream speed to zero speed	a layer of separated flow where the air is laminar			a layer of air over the surface where the airspeed is changing from free stream speed to zero speed
274	A laminar boundary layer will produce	more skin friction drag than a turbulent one	less skin friction drag than a turbulent one	less pressure drag than a turbulent one	more pressure drag than a turbulent one			less skin friction drag than a turbulent one
275	Longitudinal stability is given by	the fin	the wing dihedral	the horizontal tailplane	the vertical tailplane			the horizontal tailplane
276	Lateral stability is given by	the ailerons	the wing dihedral	the horizontal tailplane	the fin			the wing dihedral
277	Stability about the lateral axis is given by	wing dihedral	the fin	the ailerons	the horizontal tailplane			the horizontal tailplane
278	Sweepback of the wings will	increase lateral stability	decrease lateral stability	decrease longitudinal stability	increase longitudinal stability			increase lateral stability
279	On an aircraft in an un-powered steady speed descent	the lift equals the weight	the weight equals the drag	the weight equals the resultant of the lift and drag				the weight equals the resultant of the lift and drag
280	When an aircraft rolls to enter a turn and power is not increased	the lift equals the weight	the lift is greater than the weight	the lift is less than the weight	the lift equals the drag			the lift is less than the weight
281	The boundary layer is	thickest at the leading edge	thickest at the trailing edge	constant thickness from leading to trailing edges	thickest at the lower trailing edge			thickest at the trailing edge
282	The amount of thrust produced by a jet engine or a propeller can be calculated using	Newton's 1st law	Newton's 2nd law	Newton's 3rd law	all the given			Newton's 2nd law
283	An engine which produces an efflux of high speed will be	more efficient	less efficient	speed of efflux has no affect on the engine efficiency	less efficient in case of propeller engines			less efficient
284	Directional stability may be increased with	pitch dampers	horn balance	alierons	yaw dampers			yaw dampers
285	Lateral stability may be increased with	increased lateral dihedral	increased lateral anhedral	increased longitudinal dihedral	increased longitudinal anhedral			increased lateral dihedral