Subject: Theory of Flight — **الموضوع : نظرية طيران**

Weekly Hours : Theoretical : 2 Units : 4 — **الساعات الأسبوعية : نظري : 2 الوحدات :4**

Tutorial : — **مناقشة :**

Experimental : — **عملي :**

Week	**Contents**	**المحتويات**	**الأسبوع**
1	Standard Atmosphere (ISA)	الغلاف الجوي القياسي	1
2	Aerodynamic forces and moments on aircraft	القوى و العزوم الايروديناميكية على الطائرة	2
3	Lift – Lift coefficient – Lift curves characteristics	الرفع – معامل الرفع – خصائص منحنيات الرفع	3
4	Drag – Drag estimation	الكبح – تخمين قوة الكبح	4
5	Types of drag	أنواع الكبح	5
6	Stalling	الانهواء	6
7	Subsonic wings and sections characteristics	الأجنحة و المقاطع تحت الصوتية – التوصيف و الخصائص	7
8	Boundary layer control	السيطرة على الطبقة المتاخمة	8
9	High-lift and control devices	معدات الرفع العالي و السيطرة	9
10	High speed flow – Mach number	جريان السرع العالية – عدد ماخ	10
11	Wave drag	الكبح الموجي	11
12	Compressibility corrections	تصحيحات الانضغاطية	12
13	Supersonic wings and sections characteristics	الأجنحة و المقاطع فوق الصوتية - الخصائص	13
14	Transonic aircraft and the Area Rule	الطائرة الانتقالية و قاعدة المساحة	14
15	The supercritical airfoils	المقاطع فوق الحرجة	15
16	Introduction to performance computation	مدخل إلى حساب الأداء	16
17	Performance in level flight	أداء الطيران الأفقي	17
18	The wing loading	تحميل الجناح	18
19	Performance curves in terms of thrust	منحنيات الأداء بدلالة الدفع	19
20	Performance curves in terms of power	منحنيات الأداء بدلالة القدرة	20
21	Climb performance	أداء التسلق	21
22	Descent performance	أداء النزول	22
23	Gliding performance	أداء الانحدار بدون قدرة محرك	23
24	Range and Endurance (piston a/c)	المدى و زمن البقاء (للطائرة بمحرك مكبسي)	24
25	Range and Endurance (jet a/c)	المدى و زمن البقاء (للطائرة بمحرك نفاث)	25
26	Take-off	الإقلاع	26
27	Landing	الهبوط	27
28	Acceleration in climb	التعجيل في التسلق	28
29	Turning performance	أداء الاستدارة	29
30	Design performance	أداء التصميم	30

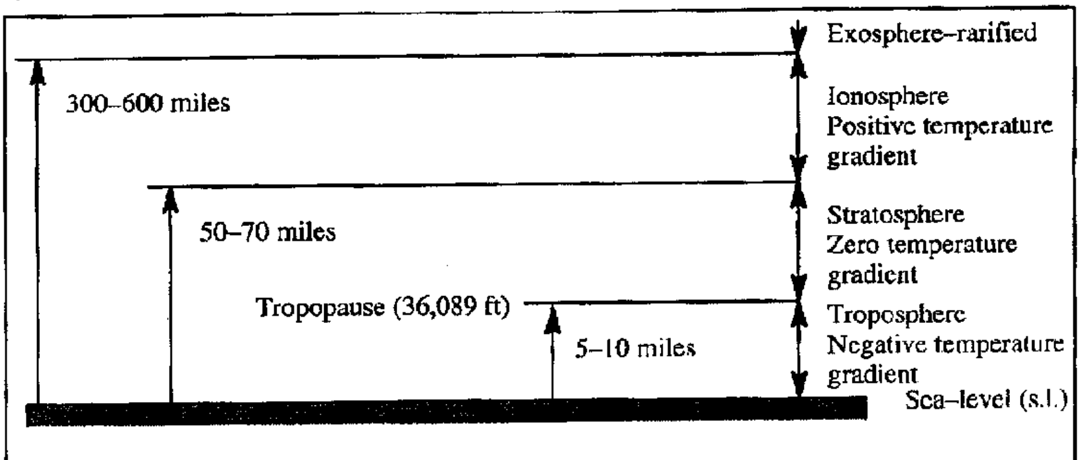


Figure 1.1 The Four Regions of the Atmosphere

According to the standard atmosphere the standard sea–level properties of the atmosphere are as follows:

$$g_0 = 32.17 \text{ ft/sec}^2 = 9.806 \text{ m/sec}^2 \tag{1.6a}$$

$$p_0 = 29.92 \text{ in Hg} = 2,116.2 \text{ lbs/ft}^2 = 1.013\text{x}10^5 \text{ N/m}^2 \tag{1.6b}$$

$$T_0 = 59^0\text{F} = 518.7^0\text{R} = 15^0\text{C} = 288.2^0\text{K} \tag{1.6c}$$

$$\varrho_0 = 0.002377 \text{ slug/ft}^3 = 1.225 \text{ Kg/m}^3 \tag{1.6d}$$

For subsonic airplanes only the troposphere and the stratosphere are important.

1.2.1 TEMPERATURE VARIATION WITH ALTITUDE

In the standard atmosphere it is assumed that below an altitude of 36,089 ft, there is a constant drop of temperature of 0.00356616 deg. F per foot of altitude. This is referred to as the lapse rate of the atmosphere. Therefore, the temperature at any given altitude, h, can be written as:

$$T = T_1 + a(h - h_1) \tag{1.7}$$

where:

a = the lapse rate of the atmosphere $= -0.00356616\ {}^0\text{F/ft}$

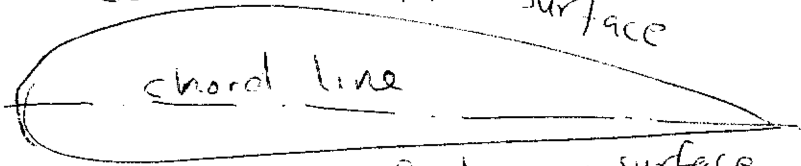
T_1 is the reference temperature at altitude h_1

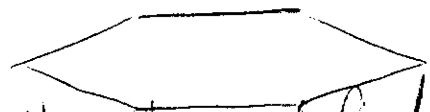
h_1 is the reference altitude

At sea–level, $h_1 = 0$ and $T_1 = T_0$. Above an altitude of 36,089 ft in the stratosphere, the standard temperature is constant and roughly equal to –69.7 deg. F.

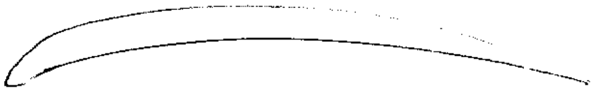
1

Typical airfoil section

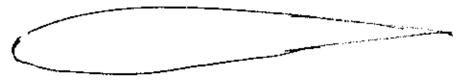




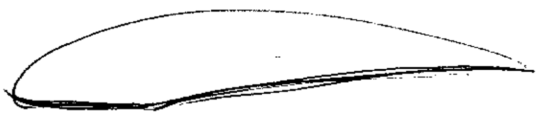
Double wedge modified Airfoil M>1



Early Airfoil

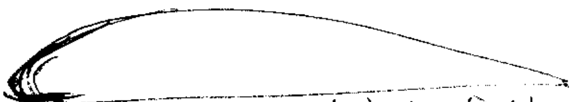


Laminar flow Airfoil
subsonic
M<1

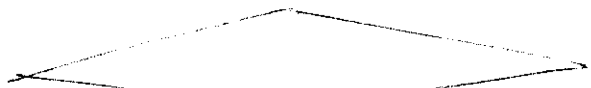


Later Airfoil

Circular Arc Airfoil
supersonic
M>1



Clark 'Y' Airfoil
subsonic
M<1



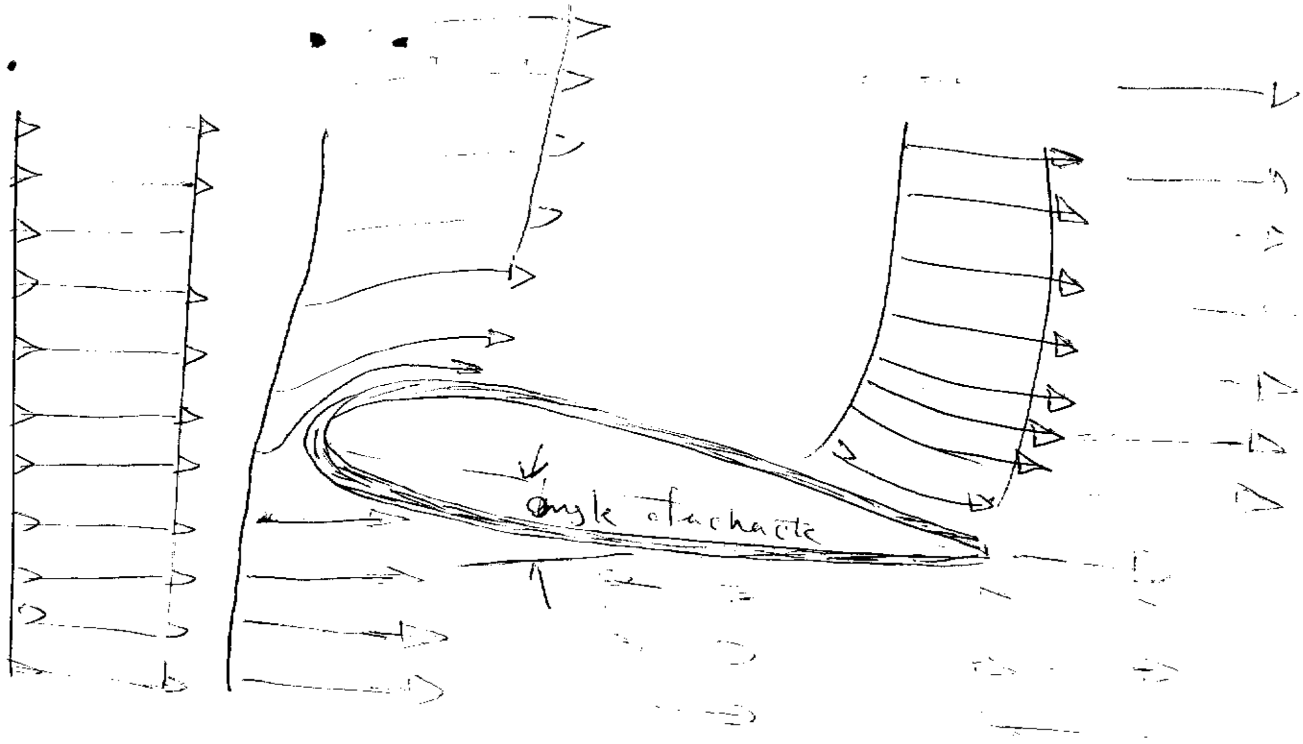
Double Wedge Airfoil
supersonic
M>1

point of reversal of curvature of the path

air to exert more than normal pressure on the leading edge of the airfoil

Momentum influences airflow over an airfoil

cause strong resistance to forward motion



Simulated air flowing around an airfoil

The popular Description of lift which fixates on the shape of the wind

air accelerates over the top of the wing

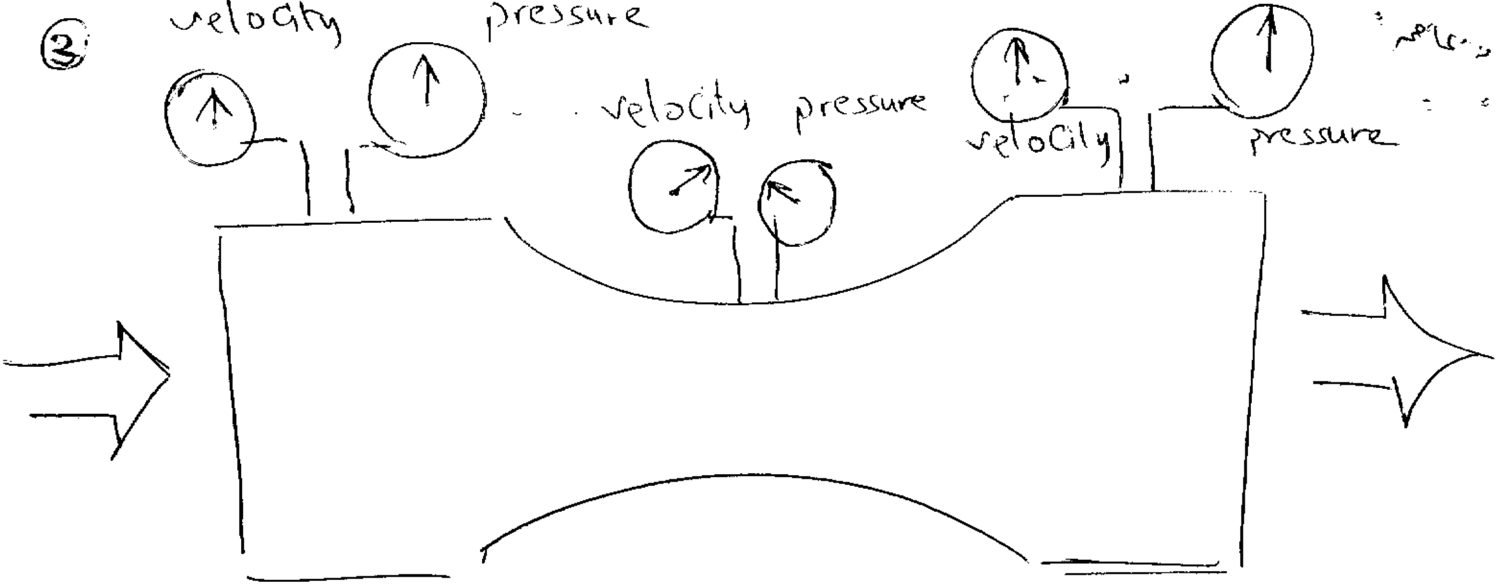
Bernoulli effect which relates the speed of the air to the static pressure

∴ a reduced static pressure is produced above the wing, creating lift

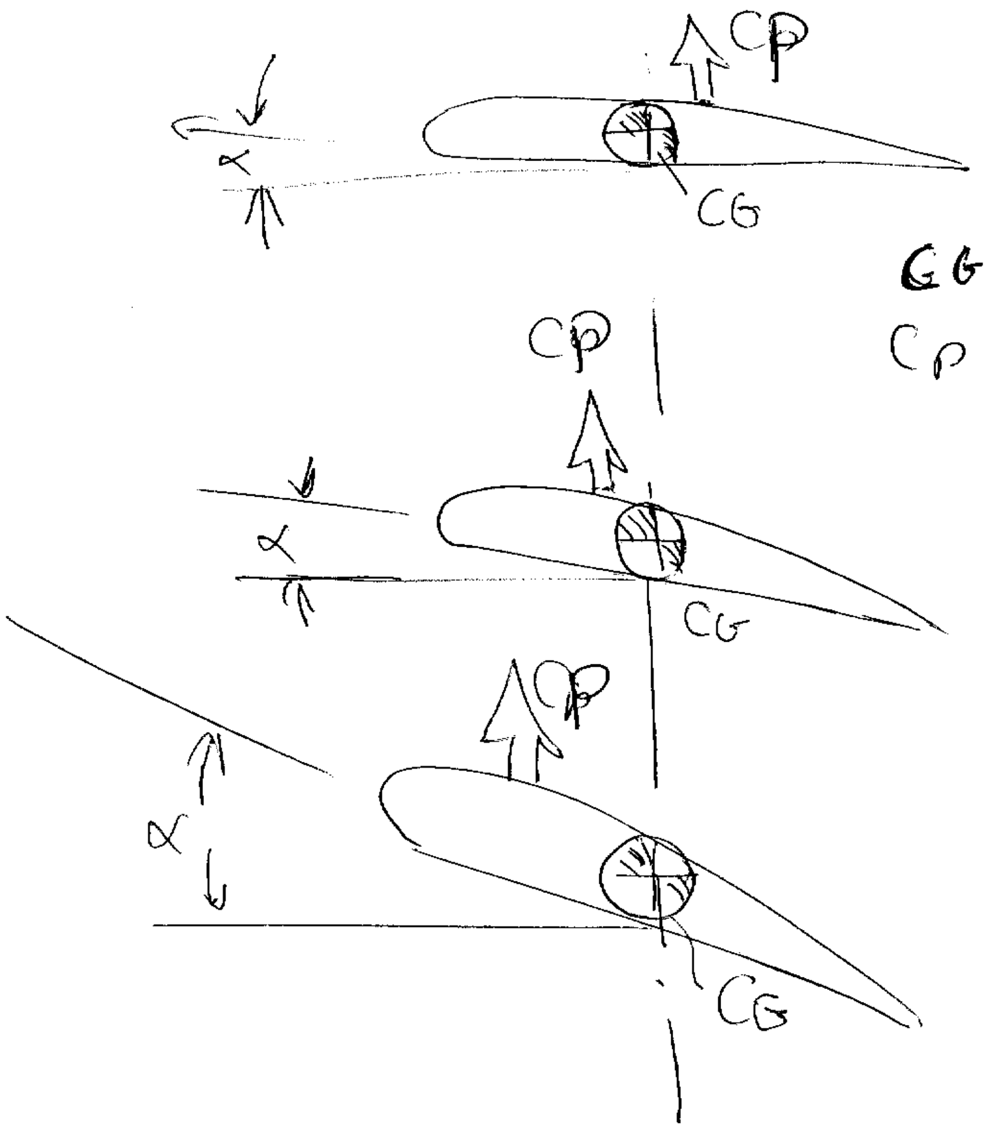
OR "principle of equal transit times"

The air that separates at the leading edge of the wing must rejoin at the trailing edge. Since the wing has a hump on the top, the air going over the tope travels farther. Thus it must go faster to regoin at the trailing edge

③

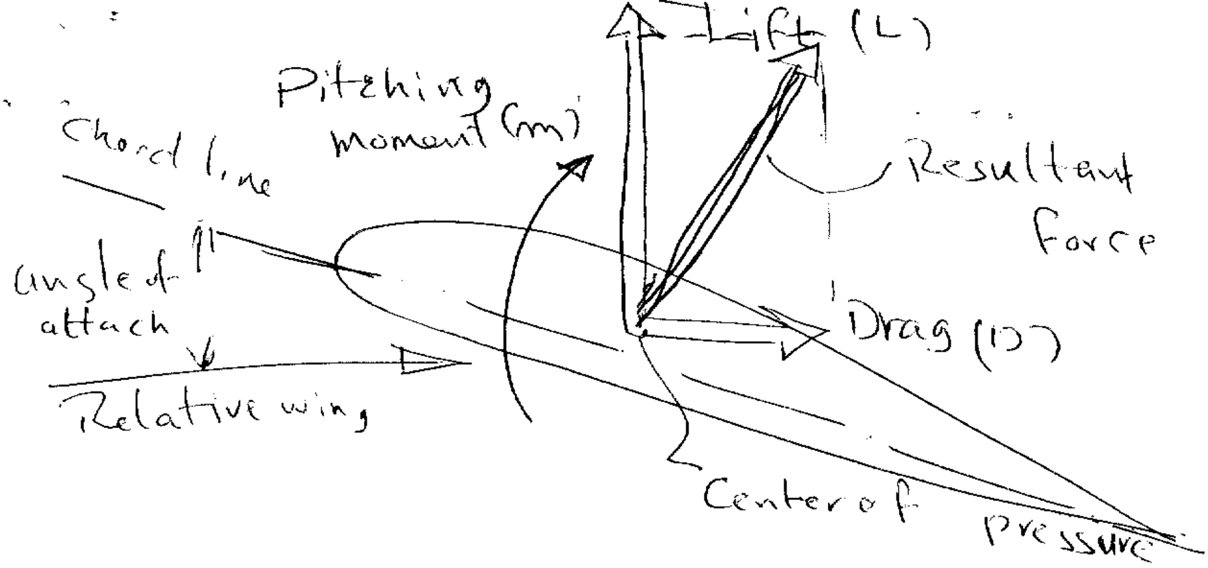


Air pressure decreases in a venturi

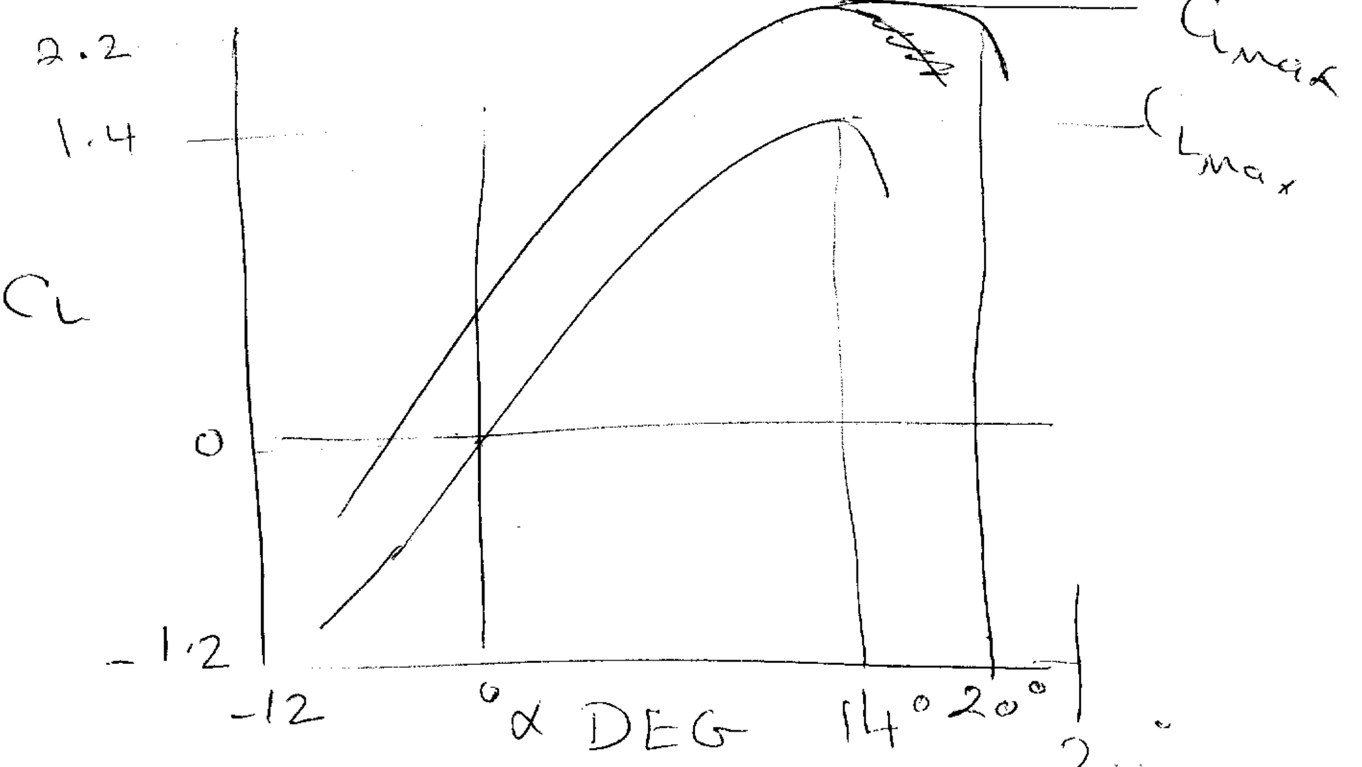


CG center of gravity
Cp Center of pressure

Cp changes with an angle of attach



Force vectors on an airfoil and Moment



Lift Coefficient C_L versuses α for Two airfoils

$$L = \frac{1}{2} \rho V^2 C_L S \quad \text{—} (1) \qquad [N] = \frac{kg}{m^3} \frac{m^2}{sec^2} m^2 = \frac{kg.m}{sec} [N]$$

$$L = \rho V \Gamma \quad \text{—} (2) \qquad [N] \; \frac{kg}{m^2} \; \frac{m}{sec} \; m^2$$

ρ = air density kg/m³

V = air velocity m/sec

C_l = Lift Coefficient

S = wings area m²

Γ = circulation m²

⑤

Force and moment coefficients

C_F : aerodynamic force coefficient
F : force

M = moment

S = area, wing area, b = wing span
V = velocity

ρ = air density, ν = fluid or air kinematic viscosity
K = fluid bulk elasticity
C_L = Lift coefficient
C_D = drag coefficient
non-dimensional quantity $\frac{F}{\rho v^2 b^2}$. By Dimensional theory

$$F = fn(V, b, \rho, \nu, K)$$
$$= C . V^a b^f \rho^c \nu^d K^e$$

In dimensional form

$$\left[\frac{ML}{T^2}\right] = \left[\left(\frac{L}{T}\right)^a L^f \left[\frac{M}{L^3}\right]^c \left\{\frac{L^2}{T}\right]^d \left[\frac{M}{LT^2}\right]^e \quad \text{—} (1)$$

Mass = $1 = c + e$ — (2)
Length $1 = a + f - 3c + 2d - e$ — (3)
Time $-2 = -a - d - 2e$ — (4)

5 unknown 3 equation $\Rightarrow$
Find as a function of d and e
$a = 2 - d - 2e$
$f = 2 - d$
$c = 1 - e$

6

Sub in ①

$$F = V^{2-d-2e}\, b^{2-d}\, \rho^{1-e}\, \nu^{d}\, K^{e} = \rho V^2 b^2 \left(\frac{\nu}{VD}\right)^d \left[\frac{K}{\rho V^2}\right]^{e}$$

Since $a^2 = \frac{\gamma P}{\rho} = \frac{K}{\rho}$ sound speed

Then $\frac{K}{\rho V^2} = \frac{\rho a^2}{\rho V^2} = \left[\frac{a}{V}\right]^2$ ⇐ Mach number compressibility effect

$$F = \rho V^2 b^2 \cdot g\left[\frac{Vb}{\nu}\right] \cdot h[M]$$

$$\therefore \frac{F}{\rho V^2 b^2} = fn\left(\frac{VD}{\nu}; M\right) \quad \text{——— } 6$$

⇐ Rayleigh Equation

$\frac{VD}{\mu/\rho} = \frac{\rho VD}{\mu}$ = Reynold's number viscosity effect

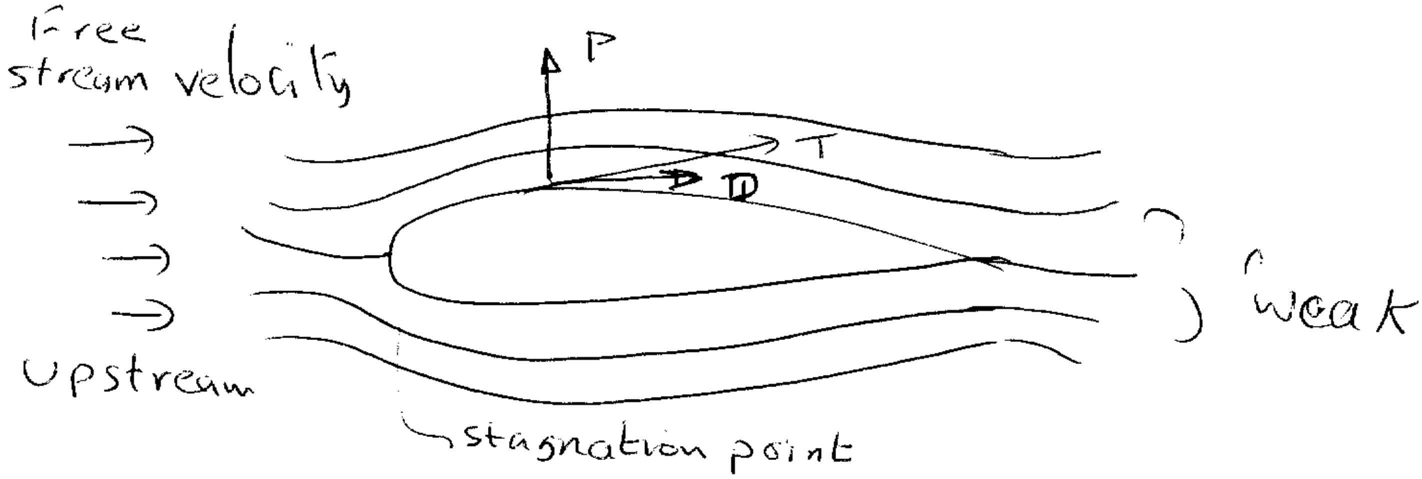
$\therefore$ $C_F = \frac{F}{\frac{1}{2}\rho V^2 S}$, $C_L = \frac{L}{\frac{1}{2}\rho V^2 S}$ and $C_D = \frac{D}{\frac{1}{2}\rho V^2 S}$

$C_M = \frac{M}{\frac{1}{2}\rho V^2 S c}$ ⇐ Pitching moment of the wing area = wing area

c wing chord

Drag Aerodynamic

Drag Force



At relatively lowspeeds on an aerodynamic surface, the ~~drag~~ is ~~the~~ fundamentally the results of horizontal Components of the normal and tangential forces transmitted fromair to the body

principal Components of drag on the airplane

a- pressure drag (form drag)

The pressure or normal forces Can be determined from streamline pattern about the body and the use of Bernoullie's principle, for Ideal flow.

b- Kin friction Drag

Tangential forces are results of the effects of viscosity with boundary layer

c- Induced drag (C_{Di})

due to lift

d- Drag due to Compresibilty (C_{DW})

effect due to the change in specific volume (v) ⇒ density(ρ) per unit change in pressure.

ملاحظة: ⑧

a and b. is parasite drag (C_{Do})

at low speed

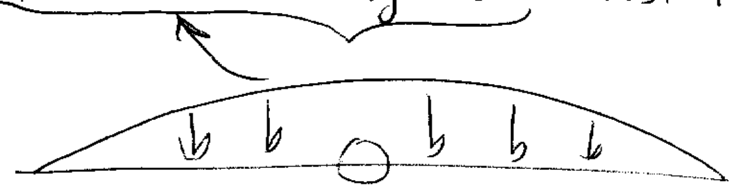
$$C_D = C_{Do} + C_{Di}$$

$$C_{Di} = \frac{C_L^2}{\pi A e}$$

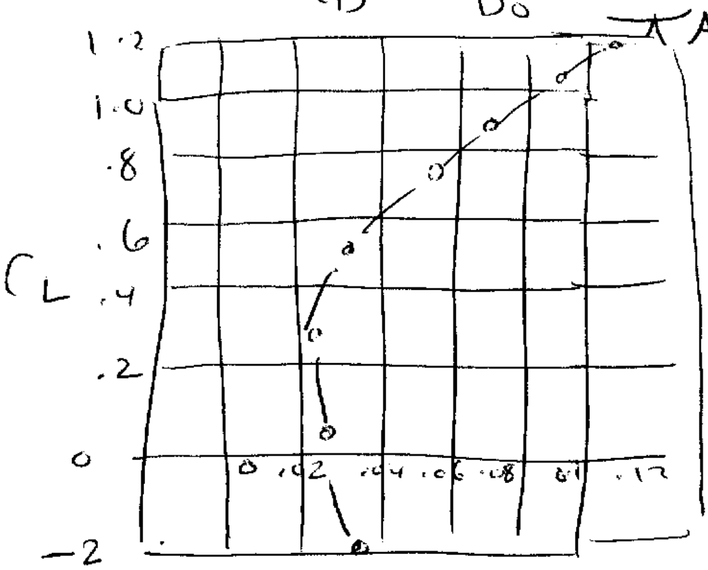
$$A = \text{Aspect ratio } \frac{b^2}{S}$$

e airplane efficiency factor (or Oswald's) effeciency factor ⟹ for variation of parasite drag D_o with angle of attack ~~and~~ (α+δ) in the induced drag term

for elliptical wing load distribution $e = 1$

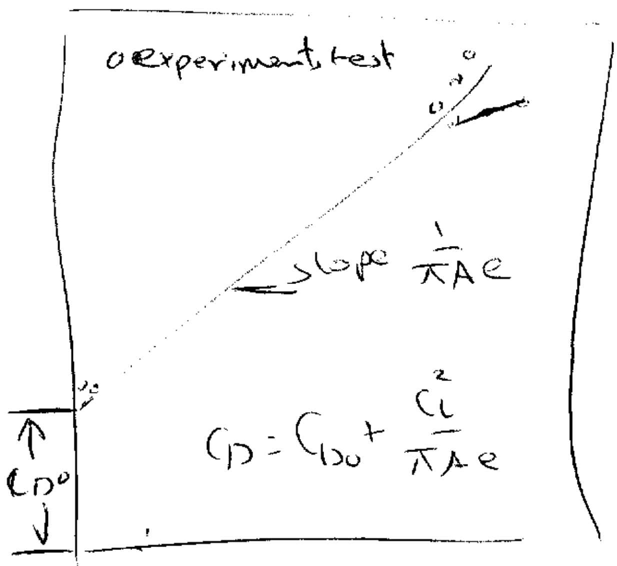


$$C_D = C_{Do} + \frac{C_L^2}{\pi A e}$$



C_D

Typical polar



C_L^2

C_{Do} = min drag

Propulsion

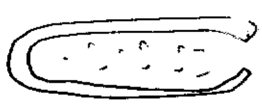
1- Power-plant effciency

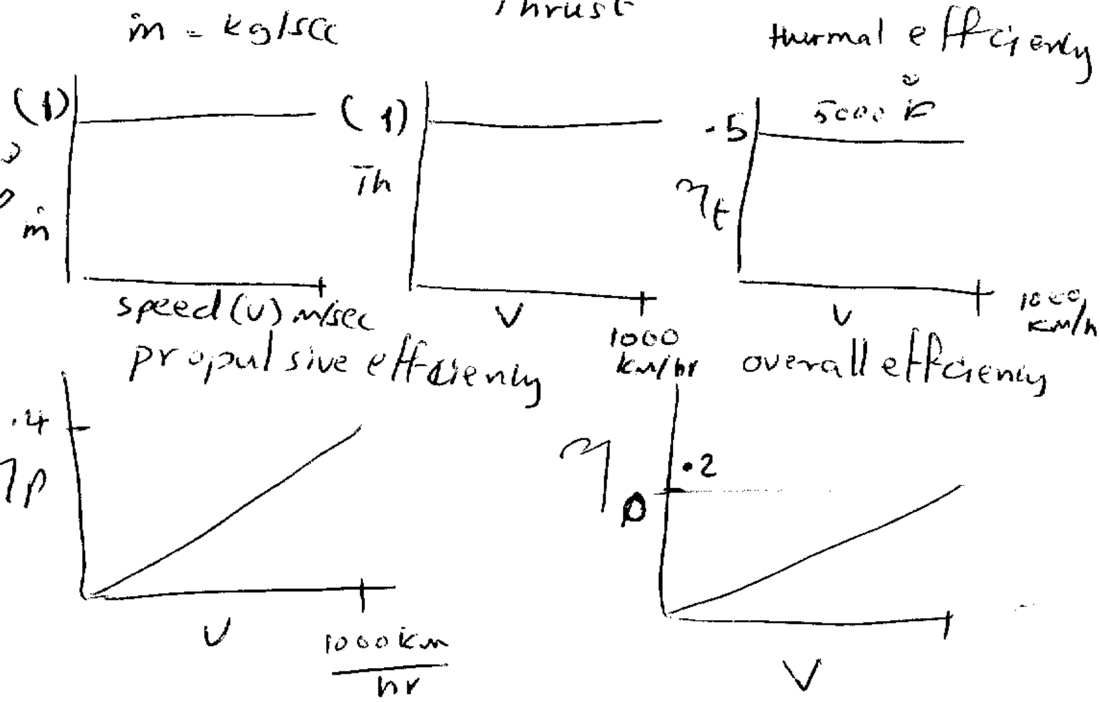
over all effciency (η_o) = $\dfrac{\text{usefull work done on airplane}}{\text{Thermal energy for fuel and oxgen (oxdizer)}}$

2- thermal effciency (η_t) = $\dfrac{\text{Mechanical energy produce in the system}}{\text{Thermal energy for fuel and oxdizer}}$

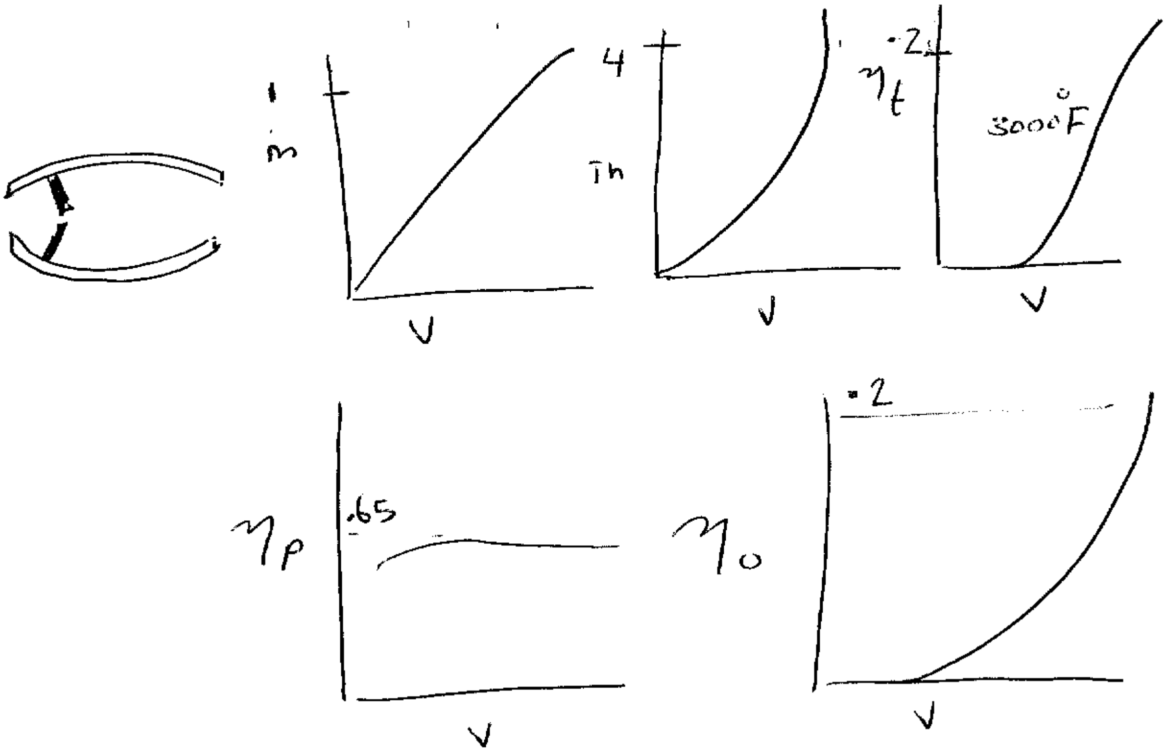
3- propulsive effcieny (η_p) = $\dfrac{\text{usefull work don on airplane}}{\text{Mechanical energy produce in the system}}$

$\therefore \eta_o = \eta_t \eta_p$

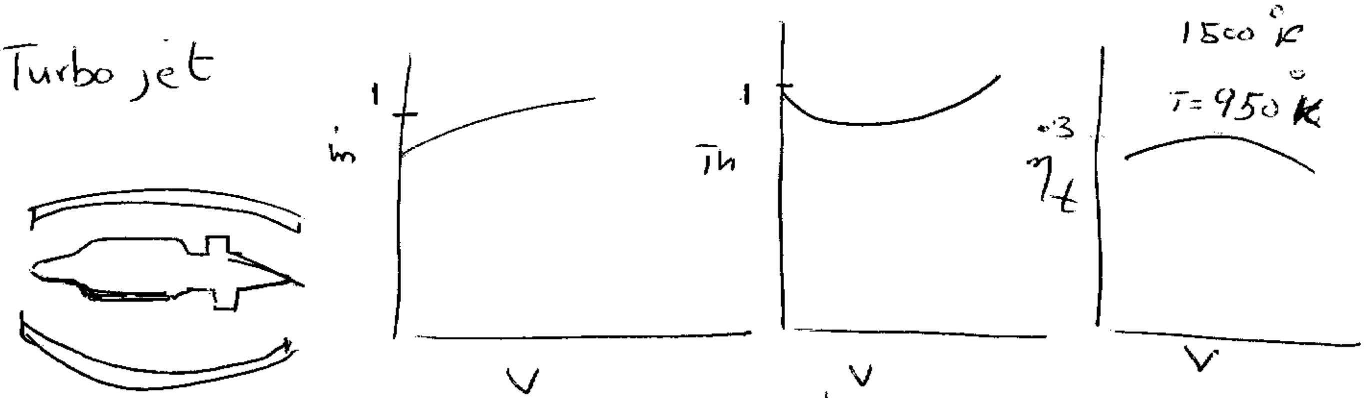
rocket 

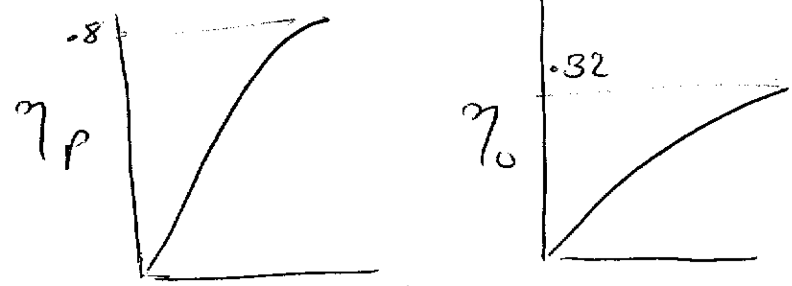


Ram jet

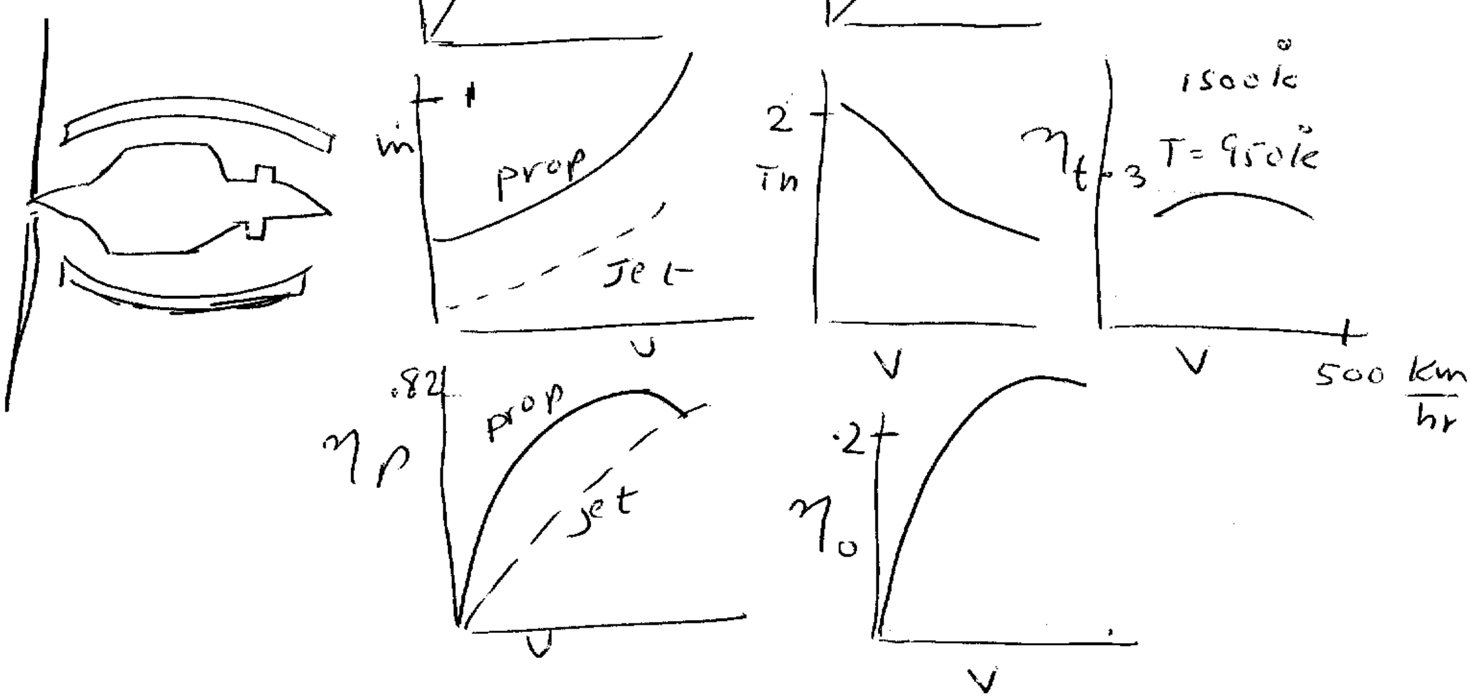


Turbo jet

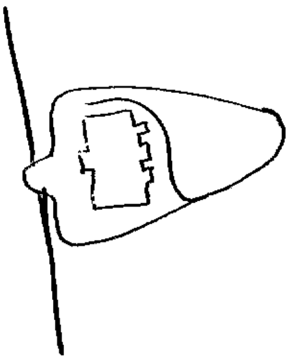


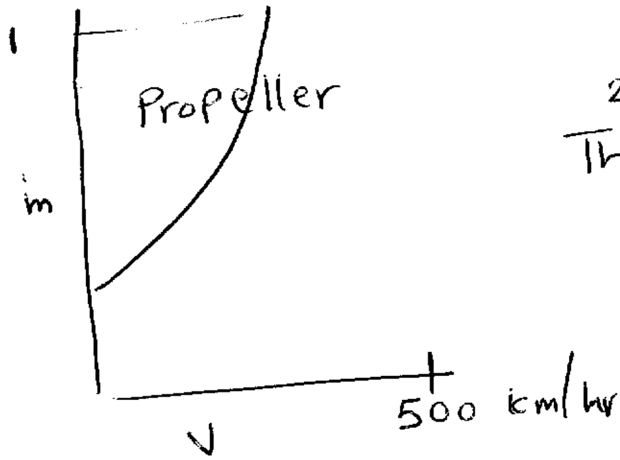


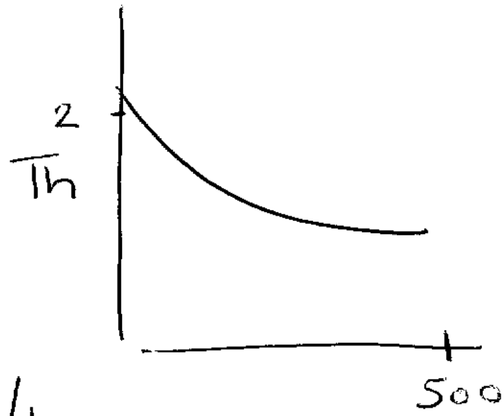
Turboprodler

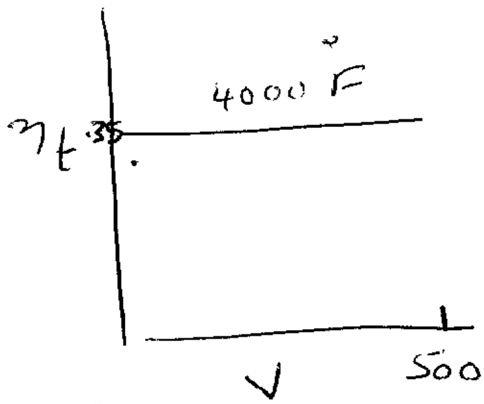


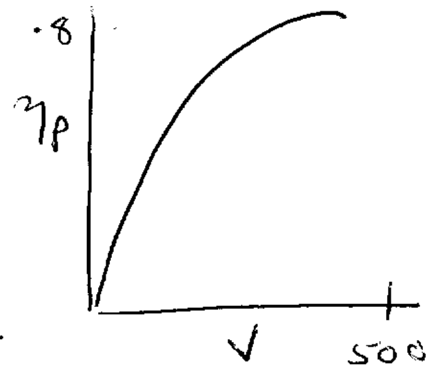
Reciprocating engine

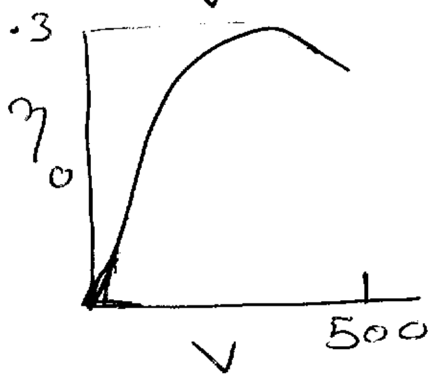












د. علام

16.333: Lecture #1

Equilibrium States

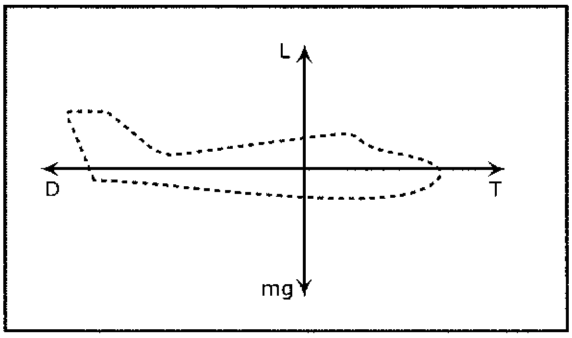
Aircraft performance

Introduction to basic terms

د. علام

Aircraft Performance

- Accelerated horizontal flight - balance of forces
  - Engine thrust T
  - Lift L ($\perp$ to V)
  - Drag D ($\|$ to V)
  - Weight W

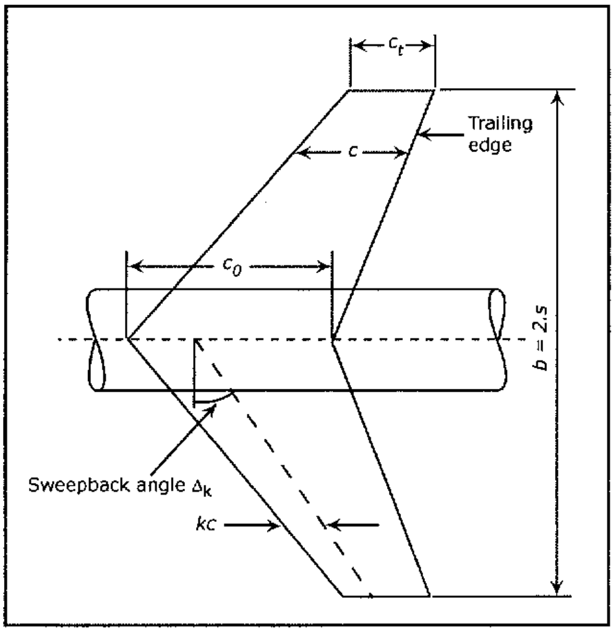


$$T - D = m\frac{dV}{dt} = 0 \text{ for steady flight}$$

and

$$L - W = 0$$

- Define $L = \frac{1}{2}\rho V^2 S C_L$ where
  - ρ – air density (standard tables)
  - S – gross wing area $= \bar{c} \times b$,
  - $\bar{c}$ = mean chord
  - b = wing span
  - $\mathcal{R}$ – wing aspect ratio $= b/\bar{c}$



  - $Q = \frac{1}{2}\rho V^2$ dynamic pressure
  - V = speed relative to the air

– C_L lift coefficient – for low Mach number, $C_L = C_{L_\alpha}(\alpha - \alpha_0)$

◇ α angle of incidence of wind to the wing

◇ α_0 is the angle associated with zero lift

- Back to the performance:

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

which implies that $V = \sqrt{\frac{2mg}{\rho S C_L}}$ so that

$$V \propto C_L^{-1/2}$$

and we can relate the effect of speed to wing lift

- A key number is stall speed, which is the lowest speed that an aircraft can fly steadily

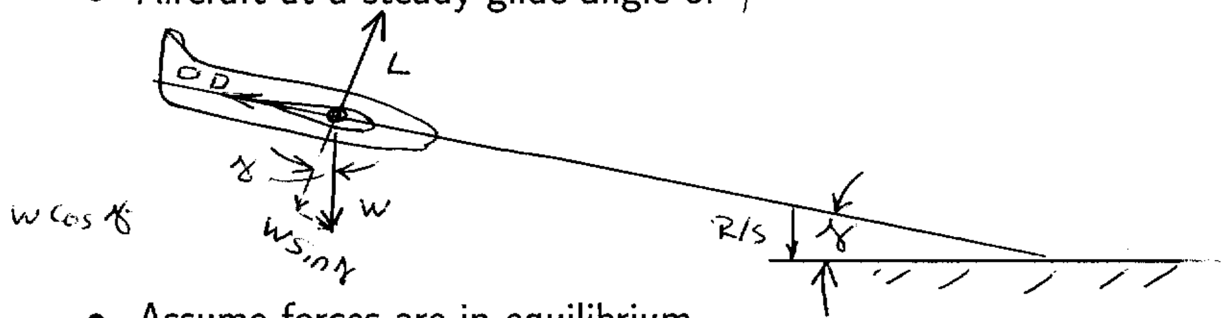
$$V_s = \sqrt{\frac{2mg}{\rho S C_{L_{\max}}}}$$

where typically get $C_{L_{\max}}$ at $\alpha_{\max} = 10°$

دسلام

Steady Gliding Flight

- Aircraft at a steady glide angle of γ



- Assume forces are in equilibrium

$$L - mg\cos\gamma = 0 \quad (1)$$
$$D + mg\sin\gamma = 0 \quad (2)$$

Gives that

$$\tan\gamma = \frac{D}{L} \equiv \frac{C_D}{C_L}$$

$\Rightarrow$ Minimum gliding angle obtained when C_D/C_L is a minimum

– High L/D gives a low gliding angle

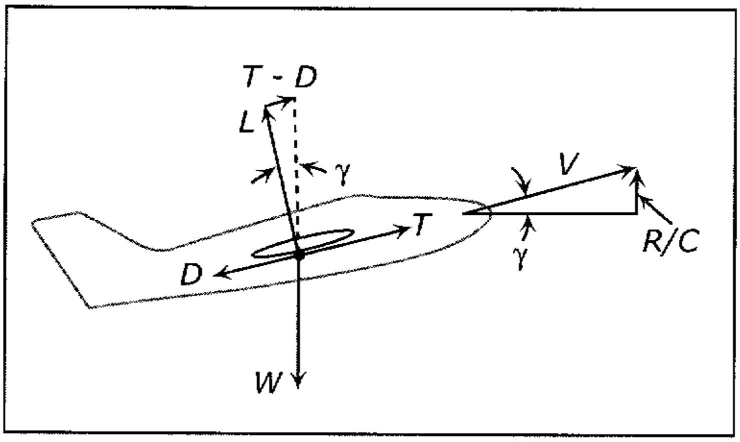
- Note: typically

$$C_D = C_{D_{\min}} + \frac{C_L^2}{\pi \mathcal{R} e}$$

where

– $C_{D_{\min}}$ is the zero lift (friction/parasitic) drag
– C_L^2 gives the **lift induced drag**
– e is Oswald's **efficiency factor** $\approx 0.7 - 0.85$

Steady Climb



- Equations:

$$T - D - W\sin\gamma = 0 \quad (5)$$
$$L - W\cos\gamma = 0 \quad (6)$$

$\Rrightarrow$ which gives

$$T - D - \frac{L}{\sin\gamma}\cos\gamma = 0$$

so that

$$\tan\gamma = \frac{T - D}{L}$$

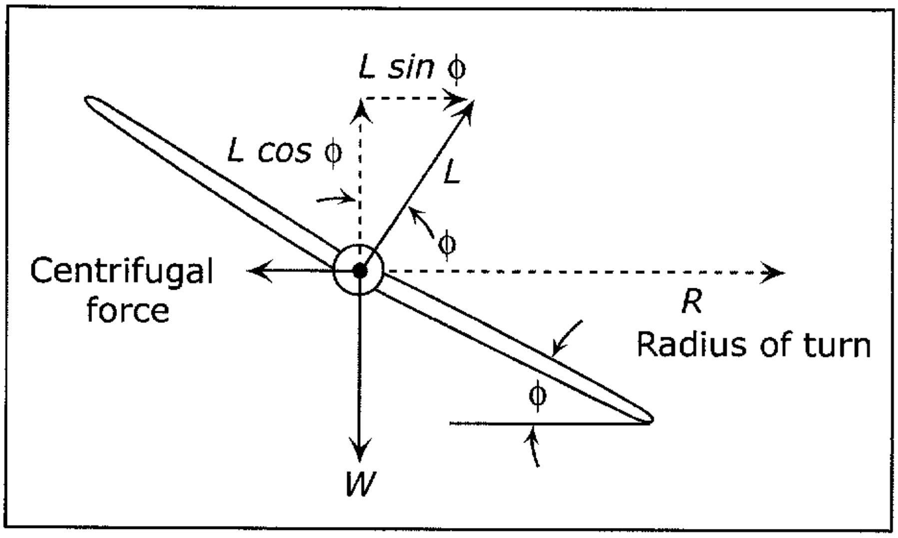
- Consistent with 1–3 if $T = 0$ since then γ as defined above is negative

- Note that for small γ, $\tan\gamma \approx \gamma \approx \sin\gamma$

$$R/C = V\sin\gamma \approx V\gamma \approx \frac{(T - D)V}{L}$$

so that the rate of climb is approximately equal to the excess power available (above that needed to maintain level flight)

Steady Turn



- Equations:

$$L\sin\phi = \text{centrifugal force} \quad (7)$$

$$= \frac{mV^2}{R} \quad (8)$$

$$L\cos\phi = W = mg \quad (9)$$

$$\Rightarrow \tan\phi = \frac{V^2}{Rg} \overset{V=R\omega}{=} \frac{V\omega}{g} \quad (10)$$

- Note: obtain $R_{\min}$ at $C_{L_{\max}}$

$$R_{\min}(\frac{1}{2}\rho V^2 S C_{L_{\max}})\sin\phi = \frac{WV^2}{g}$$

$$\Rightarrow R_{\min} = \frac{W/S}{1/2\rho g C_{L_{\max}}\sin\phi_{\max}}$$

where W/S is the wing loading and $\phi_{\max} < 30°$