Chapter 5

Wing design - selection of wing parameters

5.1 Introduction:

In the context of wing design the following aspects need consideration.

I. Wing area (S): this is calculated from the wing loading and gross weight which have been already been decided i.e.

$$S = W / (W/S)$$

II. Location on the fuselage: high-, low- or mid-wing

III. Aerofoil: thickness ratio and shape

IV. Sweep (Λ): swept forward, swept backward, no sweep, cranked wing, variable sweep (with one pivot or two)

V. Aspect ratio (A): high or low, winglets

VI. Taper ratio (λ): straight taper or variable taper.

VII. Twist (ε): amount and distribution
VIII. Wing incidence or setting (i_w)
IX. High lift devices : type of flaps and slats
X. Ailerons and spoilers
XI. Leading edge strakes
XII. Dihedral angle (Γ)
XIII. Other aspects: variable camber, planform tailoring, area ruling, braced wing, aerodynamic coupling (intentionally adding a coupling lifting surface like canard) .

For details see

Reference .1.11 chapter 4

Reference 1.10 part.	I	ch. 3
	II	ch. 6
	III	ch. 4

Following Ref. 1.11 , chapter 4, we deal with selection of the above parameters in the following order.

i) Airfoil
ii) Aspect ratio
iii) Sweep
iv) Taper ratio
v) Twist
vi) Incidence
vii) Dihedral
viii) Vertical location
ix) Wing tips
x) Other aspects

For ready reference, Fig. 5.1 gives the geometric details of the wing. Appendix 2.1 and section 4.2.1 (Fig. 4.4a and b) give the definitions of various terms.

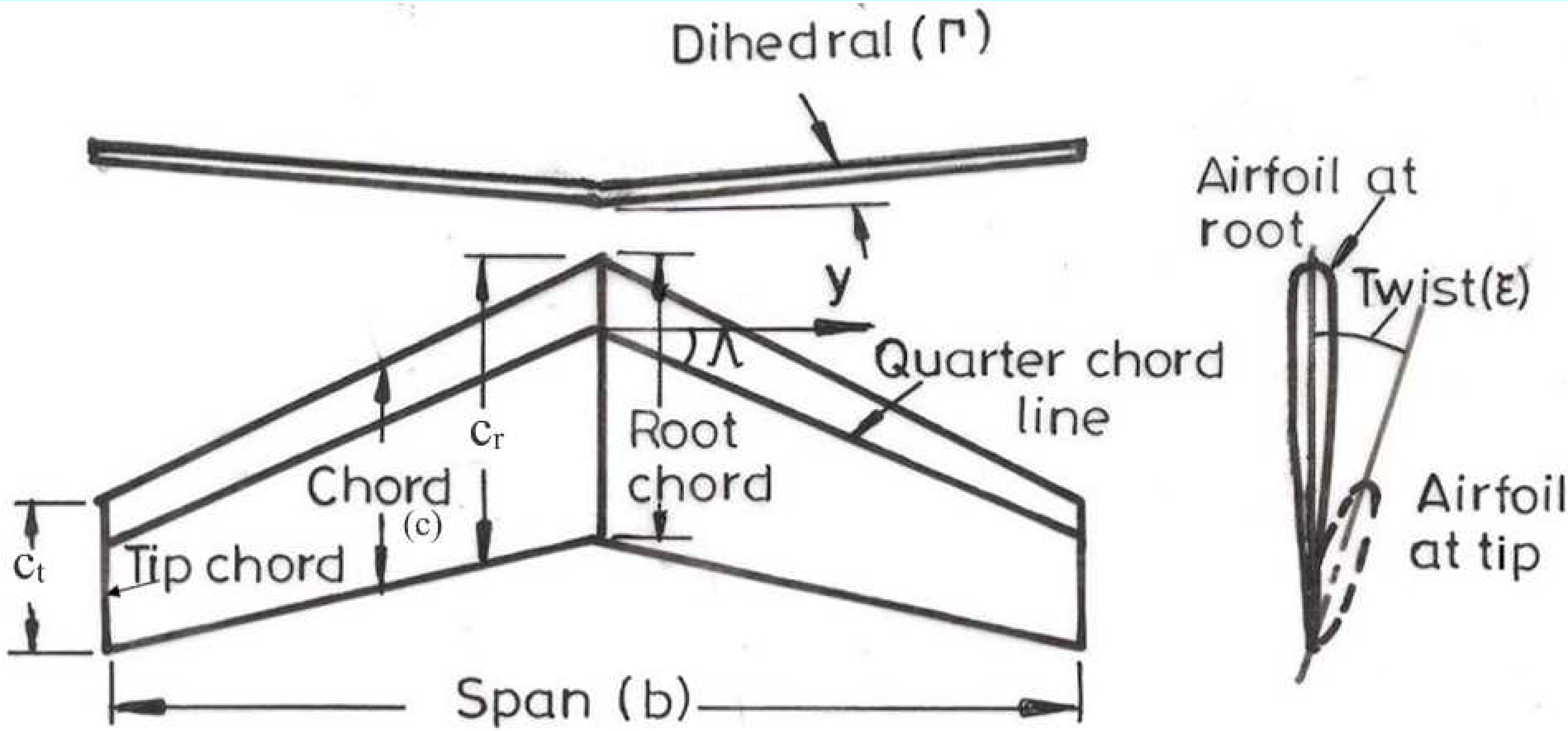

Fig 5.1 Wing Geometry

5.2. Airfoil Selection

5.2.1 Airfoil geometry: Let us recapitulate certain aspects of aerofoil geometry. See also Fig. 5.2.

Based on Ref. 5.1, the camber line or mean line is the basic line for definition of aerofoil shape (Fig.5.2a). The line joining the extremities of the camber line is the chord. The leading and trailing edges are defined as the forward and rearward extremities, respectively, of the mean line. Various camber line shapes have been suggested and they characterize various families of airfoils. The maximum camber as a fraction of the chord length and its location are some of the parameters of the camber line.

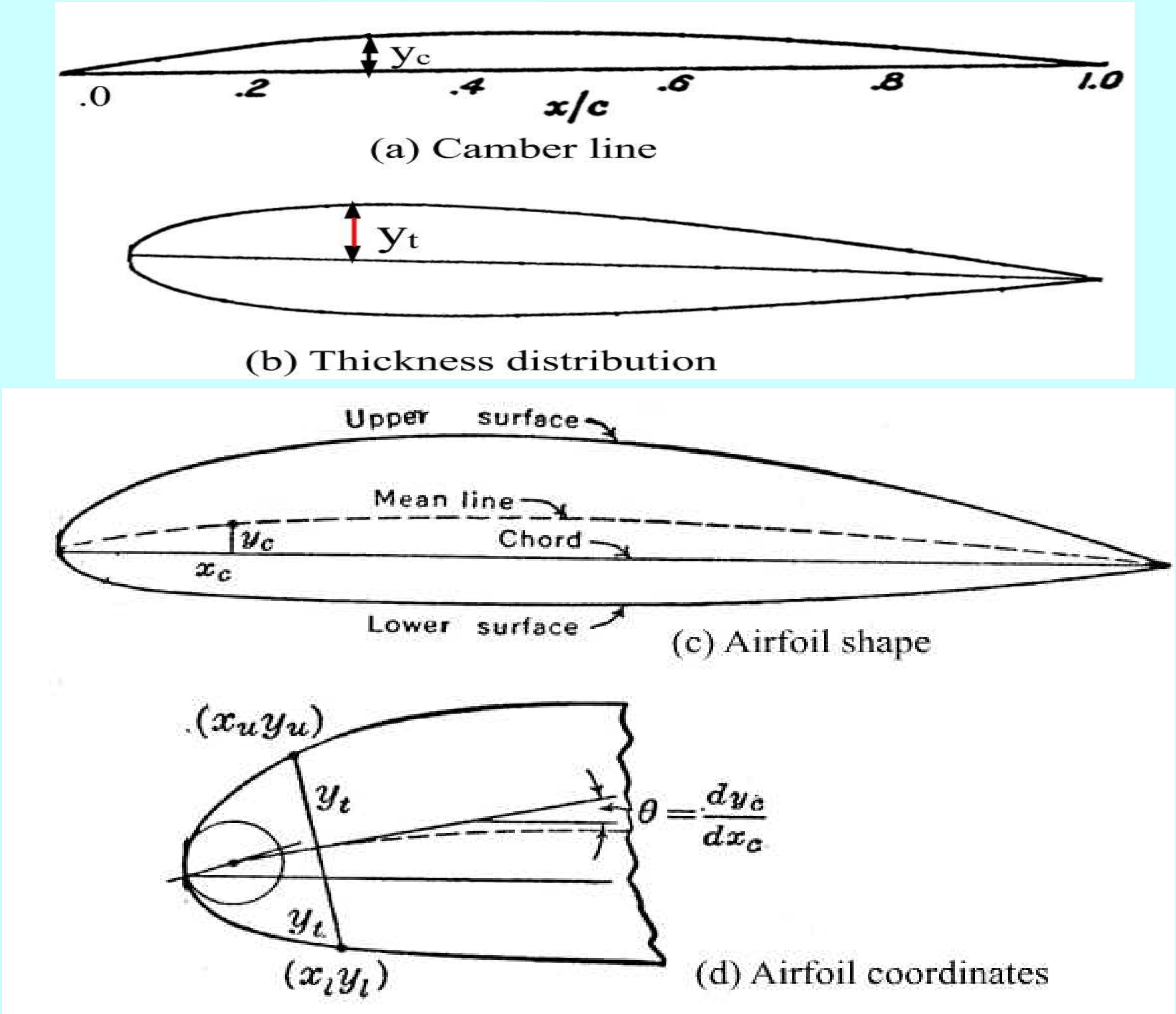

Fig 5.2 Airfoil geometry
(Adapted from Ref. 5.2, chapter 2)

Various thickness distributions have been suggested and they also characterize different families of airfoils (Fig. 5.2b). The maximum ordinate of the thickness distribution and its location are some of the parameters of the thickness distribution.

Airfoil ordinates:

An aerofoil shape (Fig.5.2c) is obtained by combining the camber line and the thickness distribution in the following manner.

a) Draw the camber line shape and draw lines perpendicular to it at various locations along the chord (Fig.5.2d).

b) Lay off the thickness distribution along the lines drawn perpendicular to the mean line (Fig.5.2d).

c) The coordinates of the upper surface (x_u,y_u) and lower surface (x_l,y_l) are given as (Fig.5.2d):

$$x_u = x - y_t \sin\theta$$
$$y_u = y_c + y_t \cos\theta$$
$$x_l = x + y_t \sin\theta$$
$$y_l = y_c - y_t \cos\theta$$

where y_c and y_t are the ordinates, at location x , of the camber line and the thickness distribution respectively. Tan θ is the slope of camber line at the location x.

d) The leading edge radius is also prescribed for the aerofoil. The center of the leading edge radius is located along the tangent to the mean line at the leading edge (Fig.5.2d).

e) Depending on the thickness distribution, the trailing edge angle may be zero or have a finite value. In some cases, thickness may be non-zero at the trailing edge .

5.2.2. Aerodynamic characteristics of airfoils:

Fig. 5.3 shows typical experimental characteristics of an aerofoil. The three plots are:

Lift coefficient (C_l) vs. angle of attack (α). This curve (Fig.5.3a) has four important features – angle of zero lift (α_{0l}), slope of the lift curve ($dC_l/d\alpha$ or a_0 or $C_{l\,\alpha}$), maximum lift coefficient (C_{lmax}), angle of attack (α_{stall}) corresponding to C_{lmax}.

Drag coefficient (C_d) vs C_l. This curve has the following important features – minimum drag coefficient (C_{dmin}), lift coefficient (C_{lopt}) corresponding to C_{dmin}. In some airfoils, called laminar flow airfoils or low-drag airfoils, the minimum drag coefficient extends over a range of

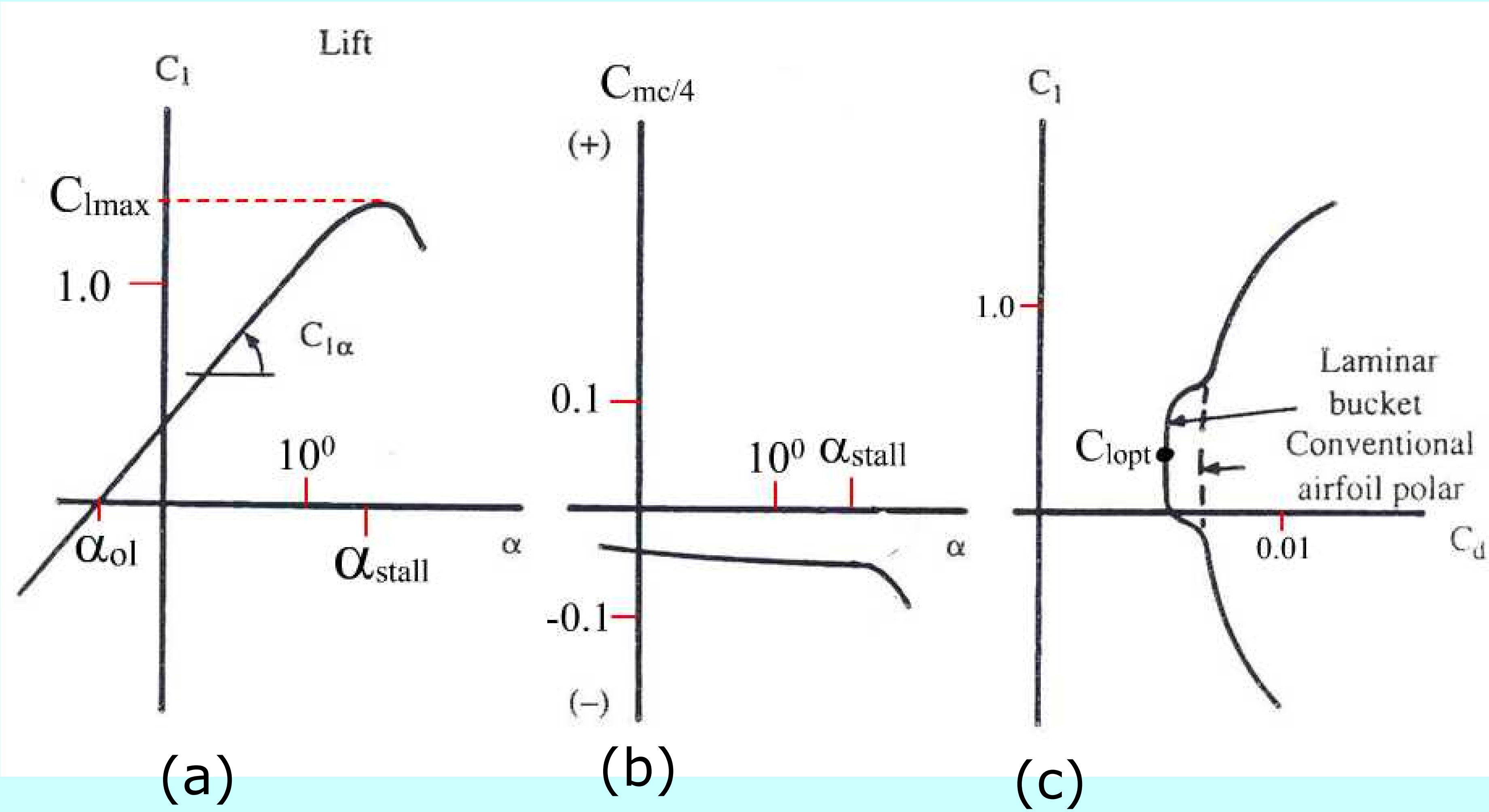

Fig 5.3 Airfoil characteristics

(Adapted from Ref. 1.11 , chapter 4)

(a) C_l vs α (b) C_l vs C_d (c) C_l vs C_m

lift coefficients (Fig. 5.3b). This feature is called drag bucket. The extent of drag bucket and the lift coefficient at the middle are also characteristic features of the airfoil. It may be added that the camber decides the C_{lopt} and thickness ratio decides the extent of the drag bucket (Ref.5.1, chapter 7).

c) Moment coefficient about quarter-chord ($C_{mc/4}$) vs. α or C_l (Fig.5.3c) From this curve, the location of the aerodynamic center (a.c.) and moment about it (C_{mac}) can be worked out. It may be recalled that a.c. is the point on the chord about which, the moment coefficient is independent of C_l.

d) Stall pattern:

Variation of lift coefficient with angle of attack near the stall is an indication of the stall pattern. A gradual pattern as shown in the C_l vs α curve in Fig. 5.3a is a desirable feature. Some airfoils display abrupt decrease in C_l after stall. This is undesirable as pilot does not get adequate warning regarding impending loss of lift. Airfoils with thickness ratio between 6 – 10% generally display abrupt stall. Those with t/c more than 14% display gradual stall. It may be added that stall patterns on the wing and on the airfoil are directly related only for high aspect ratio upswept wings. For low aspect ratio highly swept wings three dimensional effects may dominate.

A large number of airfoils have been designed over the years. Some of them are shown in Fig. 5.4.

Remarks:

i) In 1930s, NACA (National Advisory Committee for Aeronautics) of USA designed and tested a large number of airfoils. These are called NACA airfoils. In 1958, NACA was converted to NASA (National Aeronautics and Space Administration). NASA also has designed and tested some airfoils e.g. GA(W) airfoils, supercritical airfoils etc.

ii) Many other organizations like Gottingen Laboratory of DLR (Germany), RAE (UK) and individuals like Lissaman, Liebeck, Eppler have designed airfoils.

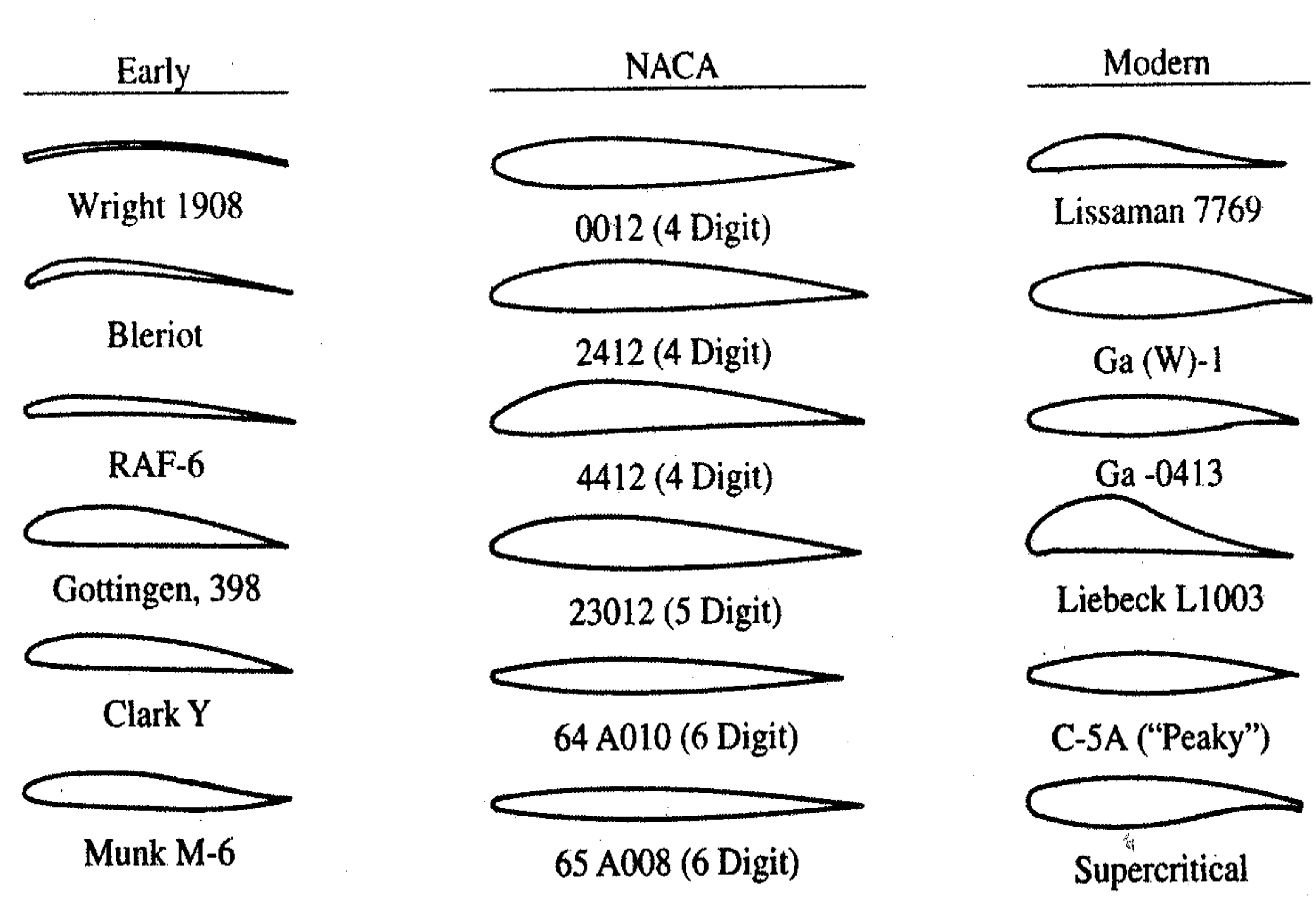

Fig 5.4 Typical Airfoils
(Adapted from Ref. 1.11, chapter 4)

iii) Characteristics of airfoils are available in Refs. 1.3, 5.1, 5.3 and 5.4, Appendix D of Ref.1.11 and NASA reports. Reference 5.1 contains extensive information on NACA airfoils. Reference 4.2 contains information on GA(W) airfoils and Ref.5.5 contains information about the supercritical airfoils.

5.2.3 Design Lift Coefficient:

The choice of the airfoil involves the selection of camber, thickness ratio and shape of the airfoil. The camber decides the C_{lopt} of the airfoil and the thickness ratio decides the characteristics like C_{lmax}, C_{dmin}, critical Mach number (M_{cr}), weight of the wing and stall pattern. For a good design the C_{lopt} of the airfoil should be close to the lift coefficient of the aircraft (C_L) in the flight corresponding to the principal segment mission of the airplane. This lift coefficient is called design lift coefficient ($C_{Ldesign}$). In most of the cases this would correspond to the cruise flight condition.

In steady level flight ,

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

Hence, $$C_{L_{design}} = \frac{W}{\frac{1}{2}\rho V^2 S}$$; ρ and V corresponding to mission e.g cruise

Remark:

The camber of the airfoil will be chosen such that C_{lopt} approximately equals $C_{Ldesign}$.

5.2.4 Airfoil thickness ratio (t/c):

As mentioned earlier, the thickness ratio affects C_{dmin}, $(C_l)_{max}$, stall pattern, wing structural weight and M_{cr}. Following Ref. 1.11 chapter 4, the effects of thickness ratio on C_{dmin}, C_{lmax} and M_{cr} are shown in Figs. 5.5, 5.6 and 5.7 respectively.

Remarks:

i) The minimum drag coefficient of the airfoil (C_{dmin}) at low subsonic speeds is dependent on Reynolds number, shape of the airfoil, surface roughness and thickness ratio. The last parameter being most important one. Figure 5.5 shows the variations of C_d for NACA 4 - digit series airfoils with thickness ratio and for C_l values of 0 and 1. In general, C_d

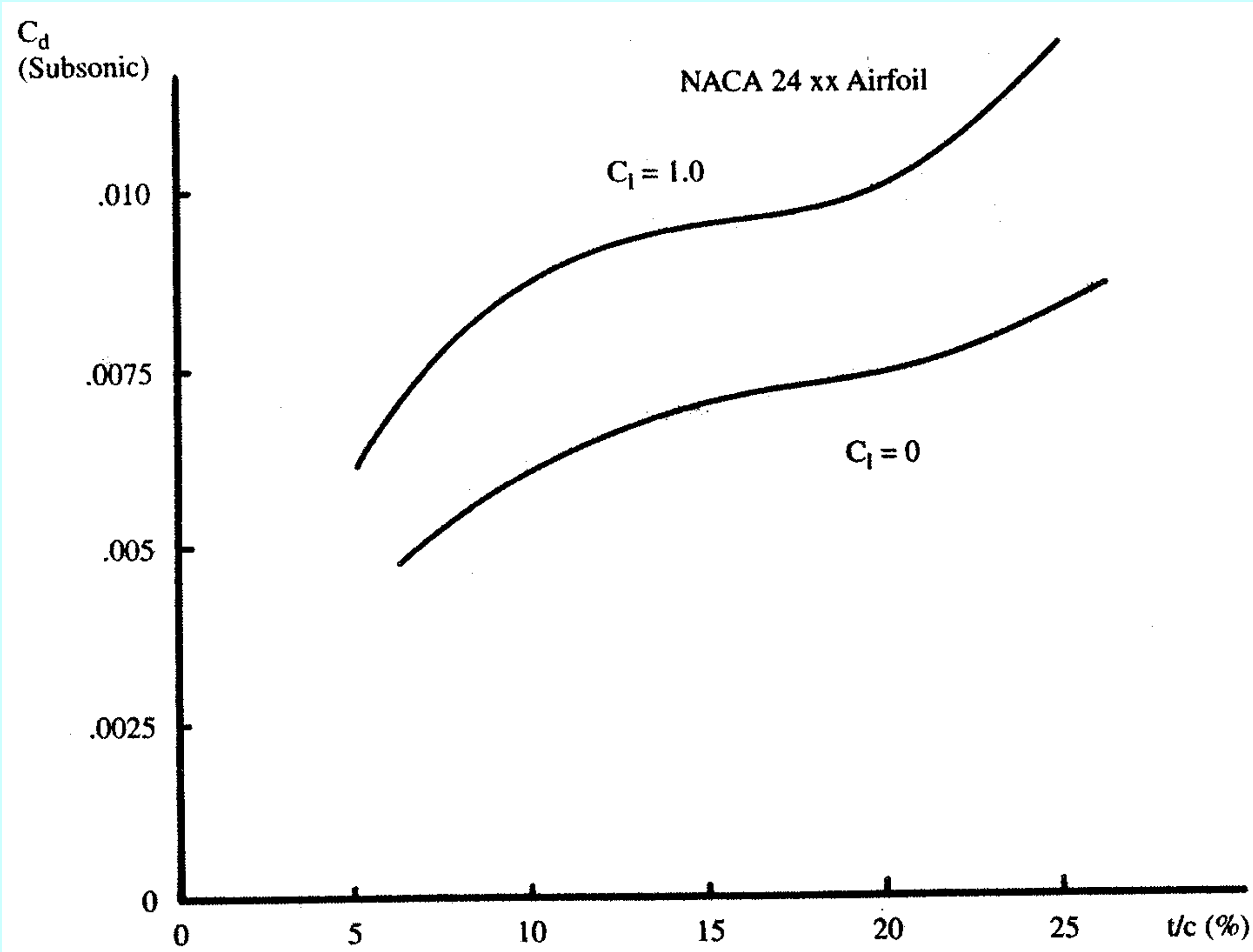

Fig 5.5 Effect of thickness ratio on drag.
(Adapted from Ref. 1.11, chapter 4)

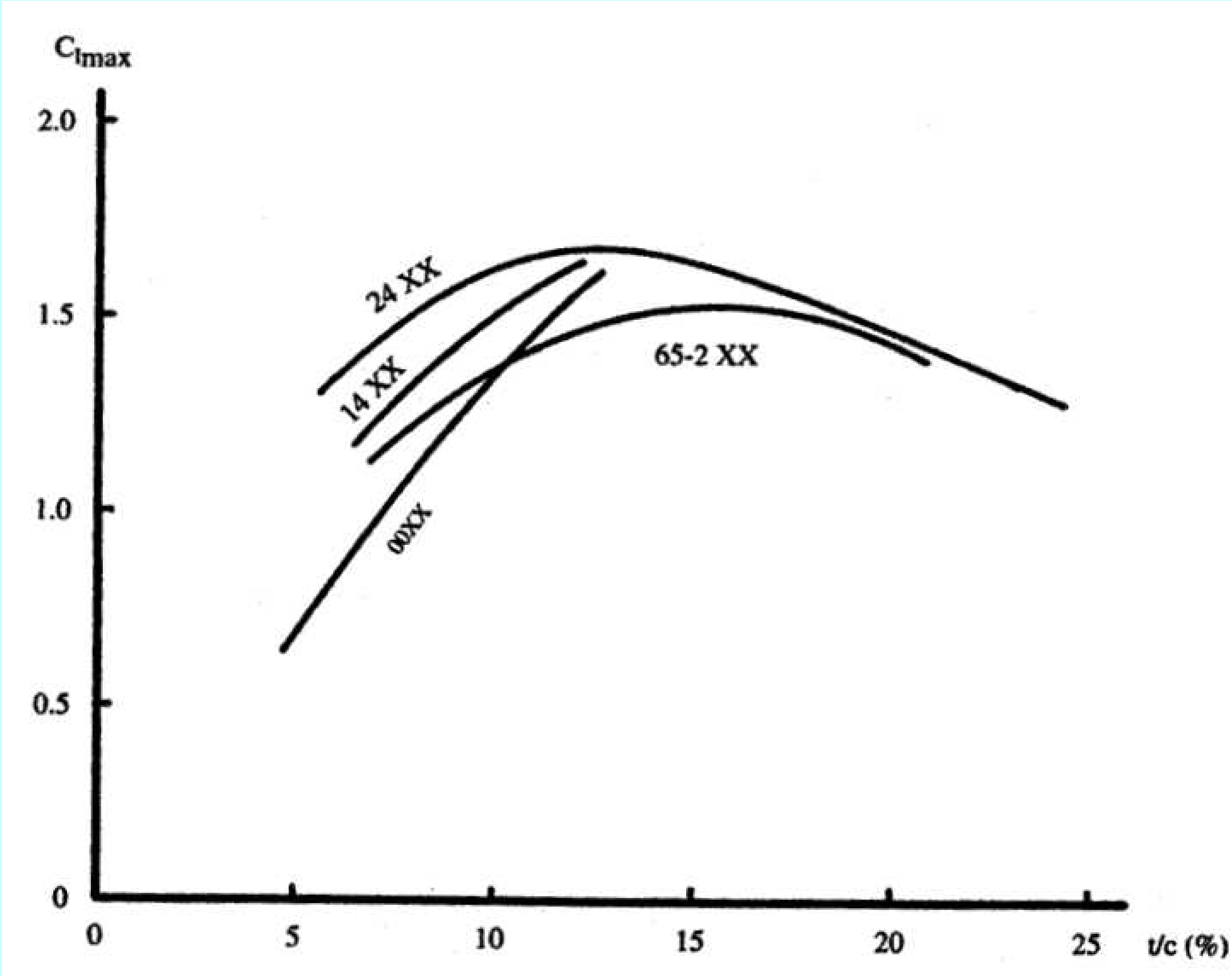

Fig 5.6 Effect of thickness ratio on maximum lift coefficient (Adapted from Ref. 1.11, chapter 4)

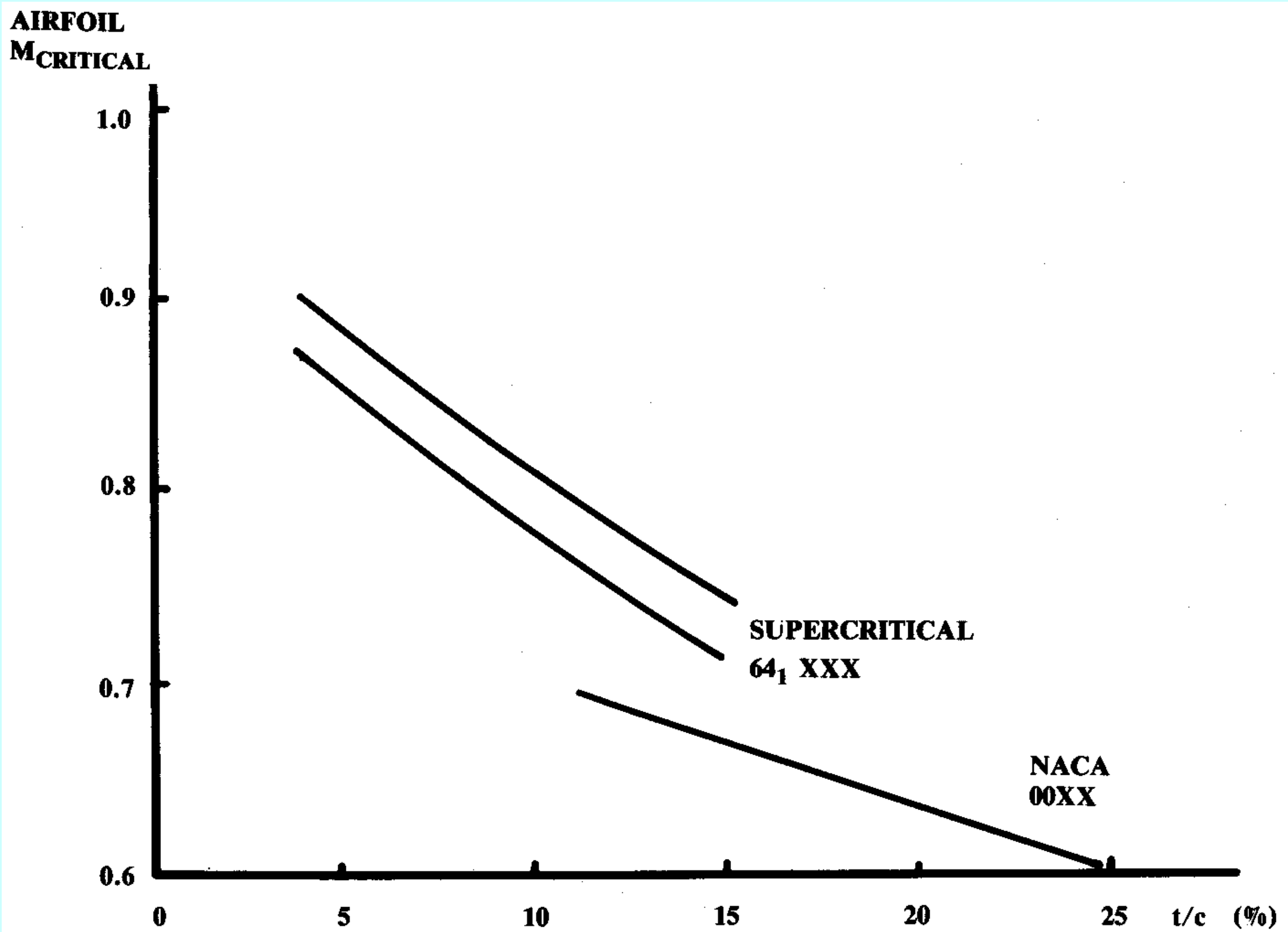

Fig 5.7 Effect of thickness ratio on critical Mach number, C_l=0 (Adapted from Ref. 1.11, chapter 4)

increases with thickness ratio.

ii) The maximum lift coefficient (C_{lmax}) of an airfoil depends on Reynolds number, surface roughness, airfoil shape and thickness ratio. Typical variation of C_{lmax} with thickness ratio is shown in Fig. 5.6. C_{lmax} is high for t/c between 12 to 16%.

iii) The critical Mach number (M_{cr}) is the free stream Mach number for which the maximum Mach number on the airfoil equals unity. The critical Mach number depends on the shape of airfoil, thickness ratio and angle of attack. Figure 5.7 shows the effect of thickness ratio on the critical Mach number. As the thickness ratio increases, the ratio of maximum velocity on the airfoil to the free stream velocity increases and as a consequence, the critical Mach number decreases. The ratio of maximum velocity on the airfoil to the free stream velocity also depends on the shape of the airfoil. In 1970s, special airfoils called supercritical airfoils were developed which had higher critical Mach Number than the earlier airfoils. Figure 5.4 includes a supercritical airfoil along with other airfoils.

d) Structural weight : The wing structure consists of spars (front and rear), stingers and skin. The spars are like 'I' section beams. The flanges of the 'I' section take the bending moment and the web takes the shear. If the wing section is thicker, then the spar flanges will be away from the centroidal axis of the section. This would result in a lighter structure for a given bending moment acting on the wing. Thus a thicker wing will result in lighter wing. According to Ref.1.11, chapter 15, for a cargo/ transport, the wing structural weight is given by :

$$W_{wing} = C S_W^{0.649} A^{0.5} \left(t/c\right)_{root}^{-0.4} \left(1 + \lambda\right)^{0.1} \left(\cos\Lambda\right)^{-1} \qquad (5.1)$$

where S_W = wing area, A = aspect ratio, t/c = thickness ratio, λ = taper ratio and Λ = sweep and C is a constant.

The expressions for weight of the wing for fighter and general aviation airplanes are also available in Ref.1.11, chapter 15.

Remark:

Equation 5.1 shows that if t/c increase from 10% to 15% , it would result in a saving of 17% in wing weight. The actual selection of the thickness ratio will involve trade-off studies. At the preliminary design stage, we take into account the airfoils used on similar airplanes. The thickness ratio can be chosen based on the trends given in Fig 5.8. For low subsonic speed airplanes, GA(W) airfoil can be used. NACA 64A or 65A are used for intermediate Mach numbers (around 0.5). For high subsonic airplanes, supercritical airfoil is a choice and at supersonic speeds, airfoils with 3 to 5 % thickness ratio are used. Concorde used a bi-convex airfoil.

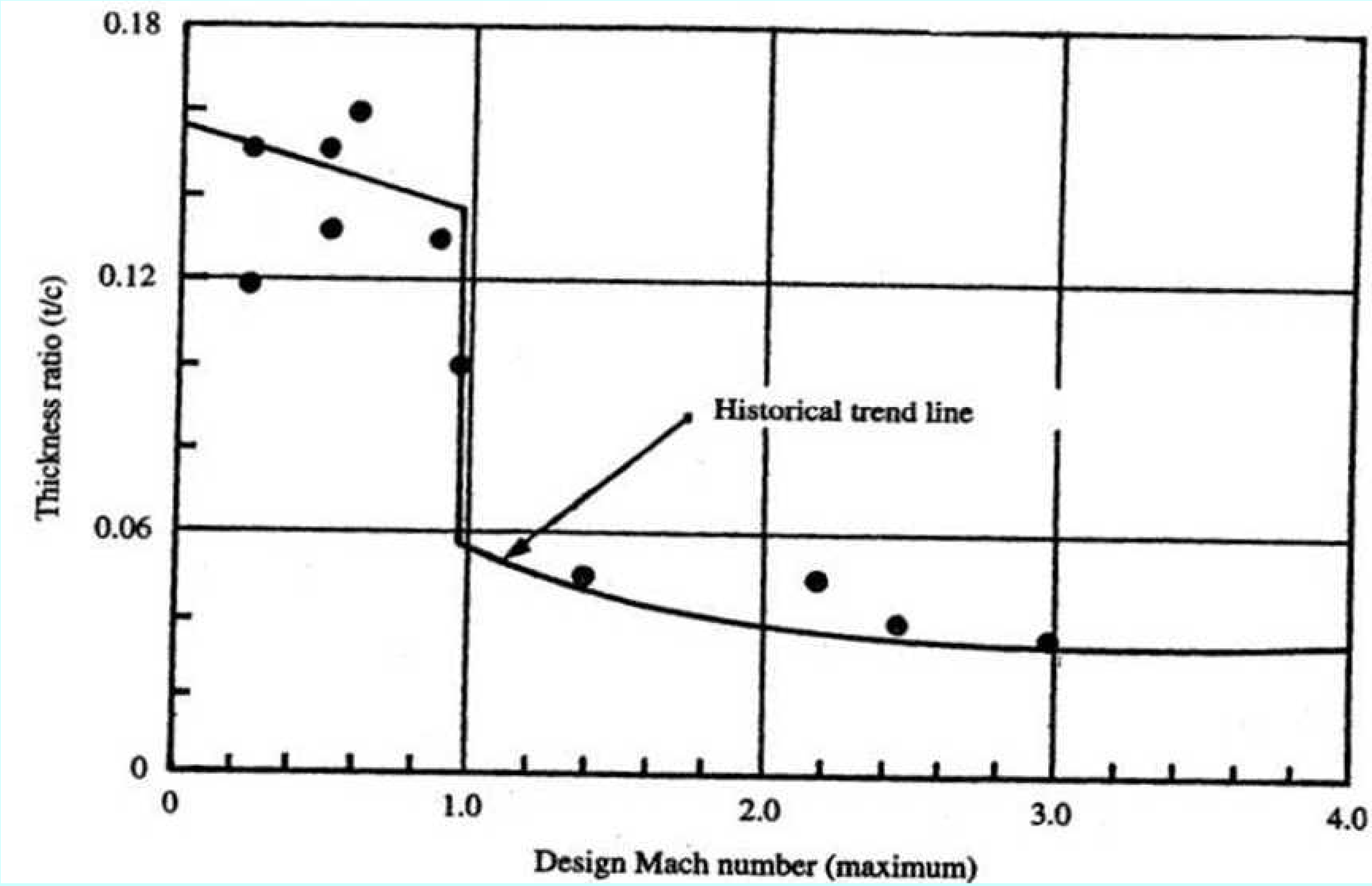

Fig 5.8 Thickness ratio historical trend.
(Adapted from Ref. 1.11, chapter 4)

5.3 Wing geometry:

In this subsection, we deal with selection of aspect ratio (A), sweep (Λ) and taper ratio (λ).

5.3.1 Choice of aspect ratio (A):

Aspect ratio affects the slope of the lift curve of wing ($C_{L\alpha}$), induced drag (C_{Di}) , structural weight of the wing and the wing span.

a) **Effect of aspect ratio on slope of the lift curve**: The slope of lift curve of an elliptic wing in a low subsonic flow is given as:

$$C_{L_\alpha} \approx C_{l_\alpha} \frac{A}{A+2} \qquad (5.2)$$

For other types of wing, the $C_{L\alpha}$ would in general be slightly lower than that for elliptic wing. However Eq.(5.2) shows that $C_{L\alpha}$ decreases as aspect ratio decreases. (See Ref.1.11, chapter 12, for effect of planform and Mach number on $C_{L\alpha}$).

b) Effect of aspect ratio on induced drag: The induced drag coefficient of a subsonic airplane is given by:

$$C_D = \frac{{C_L}^2}{\pi A}(1+\delta) \qquad (5.3)$$

where δ depends on wing geometry i.e. aspect ratio, taper ratio and sweep.

c) **Effect of aspect ratio on structural weight**: Equation (5.1) shows that the wing weight increases as square root of the aspect ratio. The reason for this is that the span increases as the aspect ratio increases ($A = b^2/S$). An increase in the span would increase the bending moment at the wing root. This would require higher moment of inertia of the spar and hence higher weight.

d) **Effect of aspect ratio on span**: For a chosen wing area, the aspect ratio decides the span of the wing $\{b=(AxS)^{1/2}\}$. In turn the span decides the hanger space needed for the airplane. Hence for personal airplanes, a moderate aspect ratio of 6 to 7 is generally chosen.

Ride in turbulent weather is poor for a high aspect ratio wing. Hence agricultural and other airplanes, which fly in proximity of ground, are subjected to air turbulence and have moderate aspect ratio of 6 to 7.

The final choice of the aspect ratio would depend on the trade-off studies. For preliminary design purposes Tables 5.1 and 5.2 based on Ref. 1.11, chapter 4, can be used as guidelines. Table 5.1 is for airplanes with engine propeller combination. As regards the guidelines for the jet airplane, the aspect ratio (A) is expressed as :

$$A = a\,M^{c}_{max}, \qquad (5.2a)$$

M_{max} = Maximum flight Mach number.

The vales of a and c for different airplanes are given in Table 5.2

Type of airplane	Aspect ratio
Homebuilt	6.0
General aviation-single engine	7.6
General aviation-twin engine	7.8
Agricultural aircraft	7.5
Twin turboprop	9.2
Flying boat	8.0

Table 5.1 Guidelines for aspect ratio of propeller airplanes
(Adapted from Ref.1.11, chapter 4)

Type of airplane	*a*	*C*
Jet trainer	4.737	-0.979
Jet fighter (dogfighter)	5.416	-0.622
Jet fighter (other)	4.110	-0.622
Military cargo/bomber	5.570	-1.075
Jet transport	7.500	0

Table 5.2 Guidelines for aspect ratio of jet airplanes
(Adapted from Ref.1.11, chapter 4)

5.3.2 Choice of sweep (Λ):

The wing sweep affects slope of the lift curve ($C_{L\alpha}$), induced drag coefficient (C_{Di}), critical Mach number (M_{cr}), wing weight and tip stalling.

a) Effect of sweep on slope of lift curve: According to Ref. 1.11, chapter 12, $C_{L\alpha}$ of a wing is given by

$$C_{L_\alpha} = \frac{2\pi A}{2 + \sqrt{4 + \frac{A^2\beta^2}{\eta^2}\left(1 + \frac{\tan^2 \Lambda_{\max t}}{\beta^2}\right)}} \qquad (5.4)$$

$\beta^2 = 1 - M^2$, $\eta = C_{l\alpha} / (2\pi/\beta)$,

Λ_{maxt} = sweep of the line of maximum thickness,

$C_{l\alpha}$ is the slope of lift curve of the airfoil used on wing at chosen flight Mach number. In the absence of this information, η can be taken as 1.

From Eq.(5.4) it is seen that $C_{L\alpha}$ decreases as sweep increases. It can be shown that $C_{L\alpha}$ of a wing of aspect ratio 8, operating at Mach number 0.8, would decrease by about 25% when sweep increases from 0^0 to 35^0.

b) **Effect of sweep on induced drag**: Based on experimental data on swept wing, Ref. 5.4 chapter 7 mentions that induced drag of a swept wing is inversely proportional to cosine of $(\Lambda\text{-}5^0)$ i.e.

$$C_{D_i} \propto \frac{1}{\cos\left(\Lambda - 5^0\right)}, \Lambda\ 75$$

c) **Effect of sweep on critical Mach number (M_{cr}) or drag divergence Mach number (M_{DD}):** The critical Mach number in connection with the airfoil was defined as the free stream Mach number at which the maximum Mach number on the airfoil is unity. This quantity can be obtained theoretically by calculating the pressure distribution on the airfoil, but cannot be determined experimentally.

However when the critical Mach number is exceeded, the drag coefficient starts to increase. Making use of this behavior we define the term 'Drag divergence Mach number (M_{DD})' as the Mach number at which the drag coefficient shows an increase of 0.002 over the subsonic drag value (Fig.5.9).

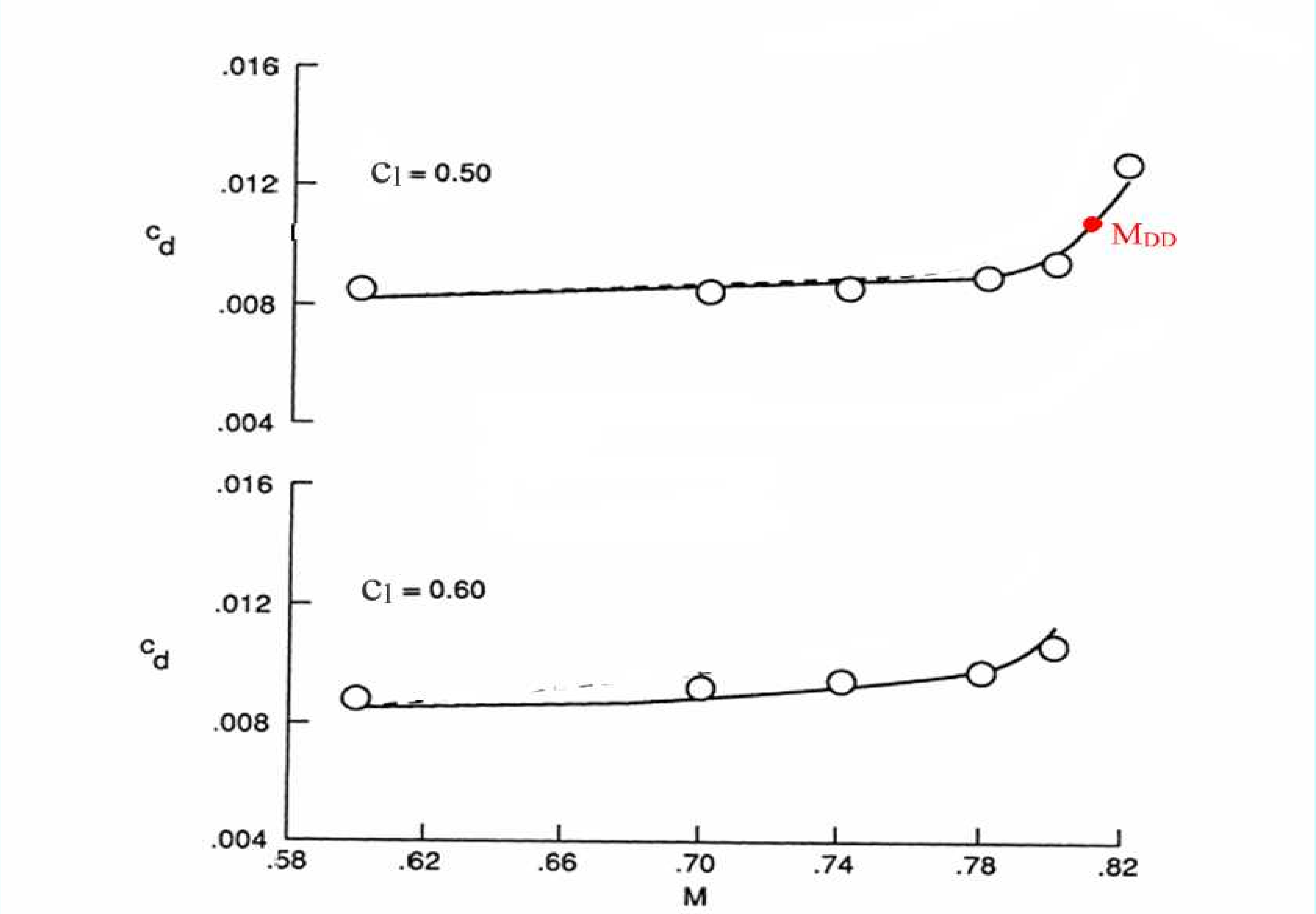

Fig 5.9 Drag divergence Mach number of a supercritical (Based on Data in Ref.5.5)

Some authors (Ref 5.6) define M_{DD} as the Mach number at which the slope of the C_d vs. M curve has a value of 0.1 i.e. $(dC_d / dM) = 0.1$.

For a swept wing the change in drag divergence Mach number due to sweep angle Λ, is given by the following equation (Ref. 5.6 , Chapter 15):

$$\frac{1-(M_{DD})_{\Lambda}}{1-(M_{DD})_{\Lambda=0}} = 1-\frac{\Lambda}{90} \qquad (5.5)$$

where $(M_{DD})_{\Lambda=0}$ and $(M_{DD})_{\Lambda}$ are the drag divergence Mach numbers of the unswept and the swept wings respectively, Λ is quarter-chord sweep in degrees.

As an illustrations consider a wing employing a supercritical airfoil with M_{DD} of 0.78. Ignoring the effects of aspect ratio on M_{DD}, the value of M_{DD} would be 0.78 for a wing with $\Lambda=0$. If the wing has a sweep of 30^0, then its M_{DD} from Eq.(5.5) would be 0.853.

Remark:

As regards the effect of sweep on critical Mach number is concerned a sweep back or sweep forward has the same effect. However from structural point of view a swept forward wing has lower flutter speed and is seldom use. See also Remark on HFB Hamsa in section 5.7.

d) **Effect of sweep on wing weight:** Equation 5.1 shows that the weight of the wing is proportional to $(1/\cos\ \Lambda)$. Thus the weight of the wing increases as sweep increases.

Remarks:

i) The final choice of sweep will be done after trade-off studies. Following can be given as guidelines. Low subsonic airplanes have unswept wings. For high speed airplanes, the angle of sweep can be chosen based on Fig. 5.10 taken from Ref. 1.11 chapter 4.

ii) **Wing with cranked trailing edge:** Instead of having a trapezoidal wing planform, the wings of high subsonic airplanes have an unswept trailing edge up to about 30% of semi-span in the inboard region (Fig. 4.4b). These wings have the following favorable effects.

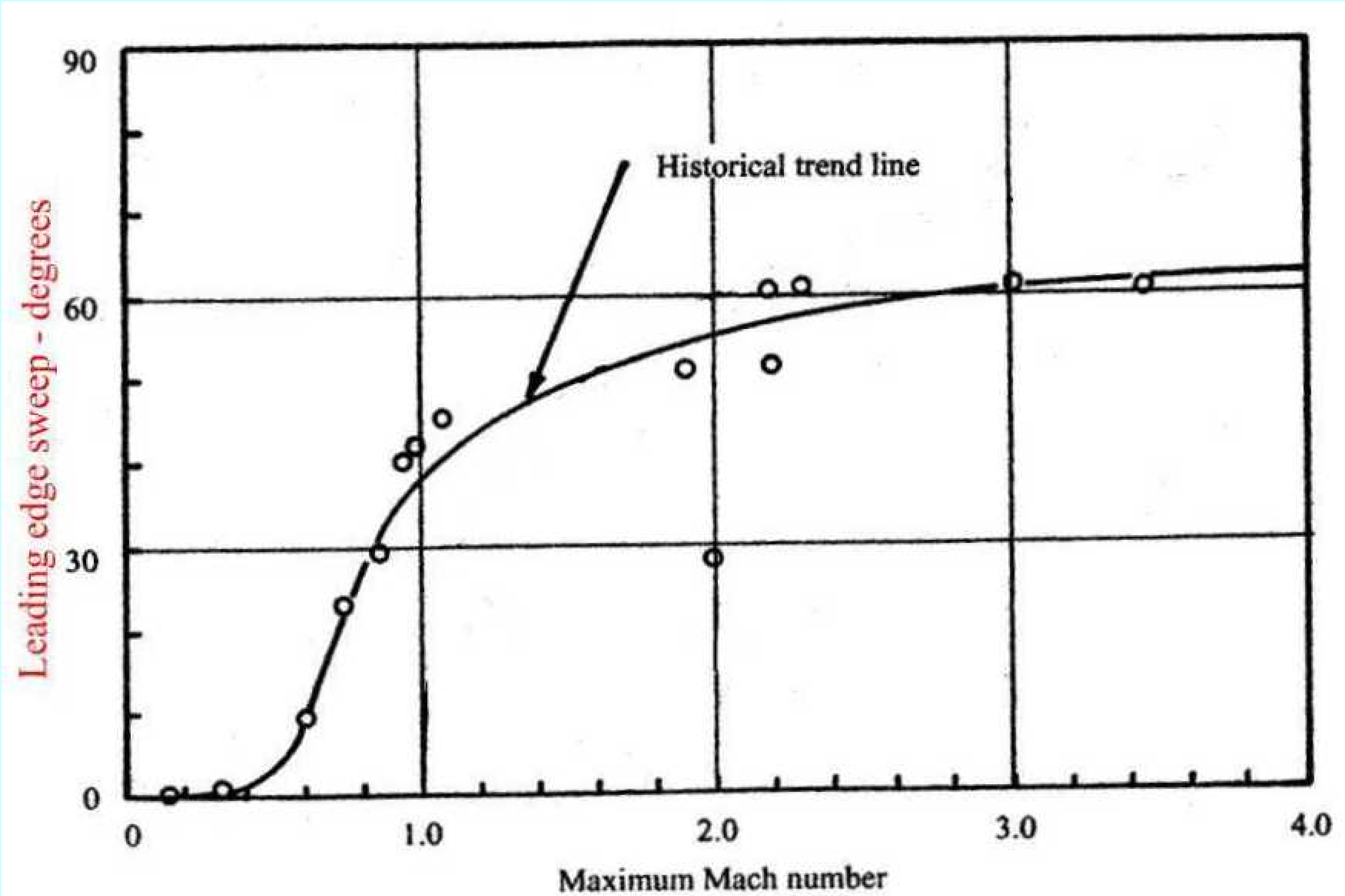

Fig 5.10 Guidelines for selection of wing sweep
(Adapted from Ref. 1.11, chapter 4)

a) Higher thickness at the root and

b) Span-wise center of pressure is brought slightly inboard which reduces the bending moment at the root as compared to the trapezoidal wing.

These two effects tend to reduce the weight of wing structure. The thicker inboard section also provides room for accommodating the backup structure for the landing gear.

5.3.3 Choice of taper ratio (λ):

The taper ratio influences the following quantities.

a) Induced drag

b) Structural weight

c) Ease of fabrication

a) **Effect of taper ratio on induced drag:** It is known that an elliptic wing has the lowest induced drag (δ = 0 in Eq.5.3). However this planform shape is difficult to fabricate.

A rectangular wing is easy to fabricate but has about 7% higher C_{Di} as compared to the elliptic wing (δ = 0.07). It is also heavier structurally (Eq.5.1).

An unswept wing, with λ between 0.3 to 0.5,has a slightly positive value of δ. Further in a tapered wing, the span loading is concentrated in the inboard portions of the wing and the airfoil at the root is thicker than near the tip. These factors help in reducing the wing weight (Eq.5.1). Tip stalling (discussed in section 5.4) is also not a problem when the taper ratio is between 0.3 and 0.5.

From these considerations, a taper ratio between 0.3 and 0.5 is common for low speed airplanes. For swept wings, a taper ratio of 0.2 is commonly used. This would necessitate measures for avoiding tip-stalling (section 5.4).Figure 5.11 shows the values of λ for swept wings as recommended by Ref. 1.11, chapter 4.

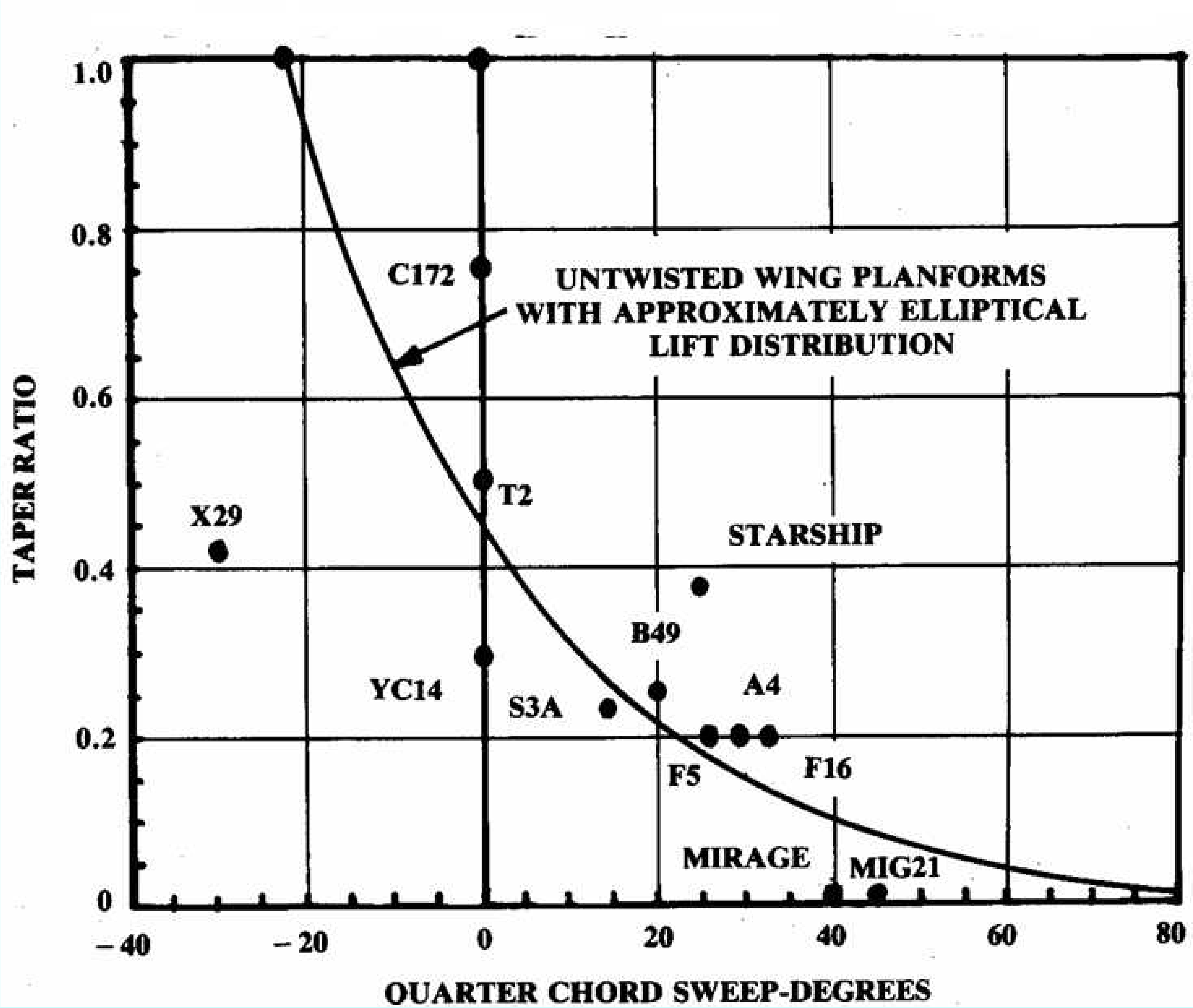

Fig 5.11 Guidelines for taper ratio of swept wings (Adapted from Ref. 1.11, chapter 4)

5.4 Twist

It is given to prevent tip stalling which is explained below.

Tip stalling: It is a phenomenon in which the stalling on the wing begins in the region near the wing tips. This is because the distribution of local lift coefficient (C_l) is not uniform along the span and as the angle of attack of the wing increases, the stalling will begin at a location where the local lift coefficient exceeds the value of maximum lift coefficient (C_{lmax}) there.

To appreciate this phenomenon let us consider an unswept tapered wing. The lift distribution on such a wing has a maximum at the root and goes to zero at the tip. This distribution is also known as Γ distribution.

Further, the local lift (ΔL) can be equated to $(1/2)\rho V_\infty^2\, c\, C_l\, \Delta y$, where c is the local chord and C_l is the local lift coefficient over an element (Δy) of span. Thus Γ distribution is proportional to the product cC_l.

The local lift coefficient (C_l) is proportional to Γ/c and is not uniform along the span.The Γ distribution along the span can be approximately obtained by Schrenk's method. According to this method, cC_l distribution is roughly midway between chord distribution of the actual wing and that of an elliptic wing of the same area. Figure 5.12 shows typical distributions of cC_l and c.

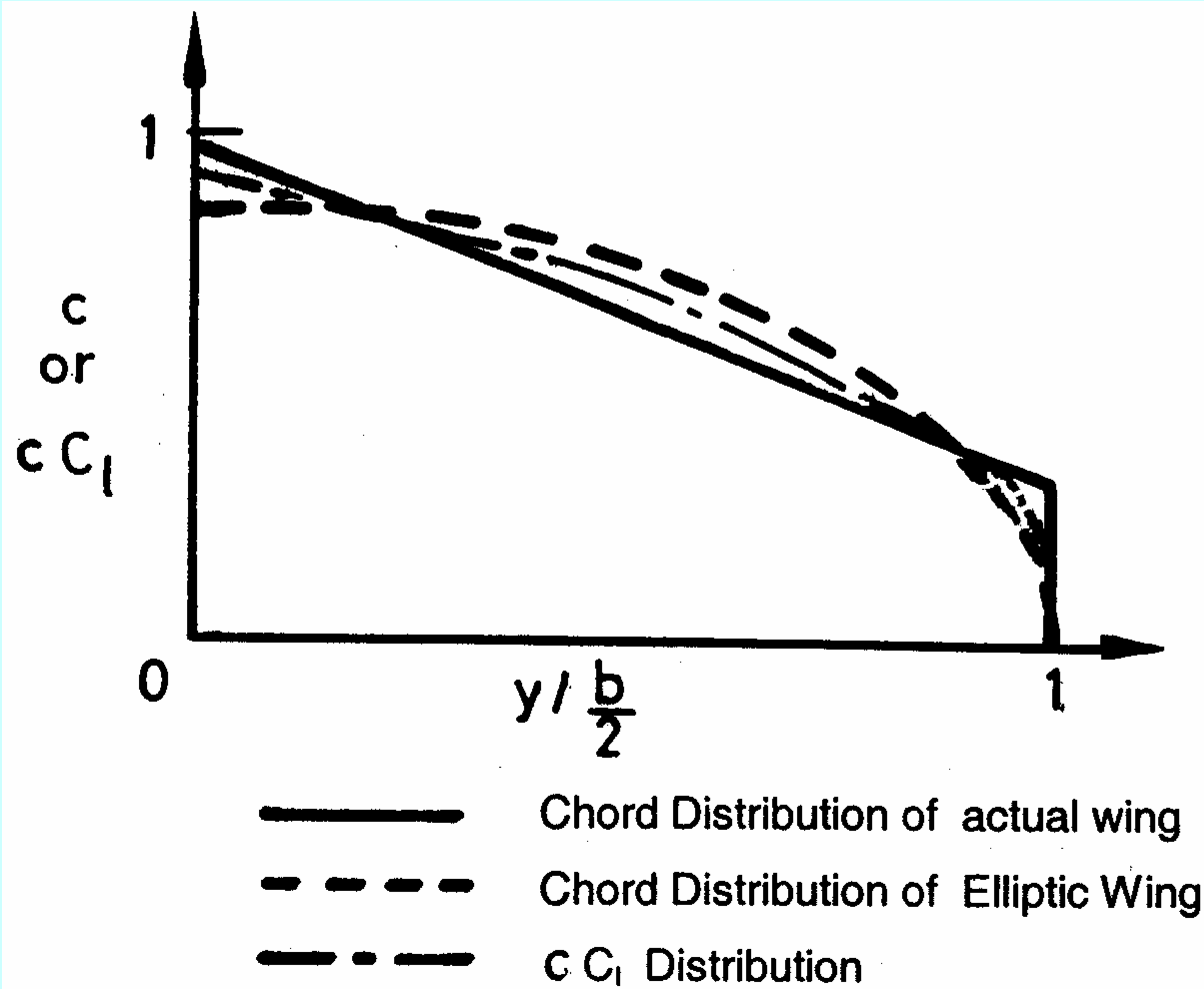

Fig 5.12 Schrenk's Method

From these distributions, the variation of C_l along the span can be calculated (Fig. 5.13). It can be shown that for a wing with taper ratio λ, the local maximum of C_l will occur at a span-wise location where

$$y/(b/2) \approx 1-\lambda. \tag{5.6}$$

It is known that the maximum lift coefficient (C_{lmax}) of an airfoil depends on the airfoil shape, surface roughness and Reynolds number. For simplicity, we can assume that C_{lmax} is approximately constant along the span. Then from the distribution of C_l in Fig. 5.13, we observe that as the angle of attack of the wing increases, the stalling will begin at

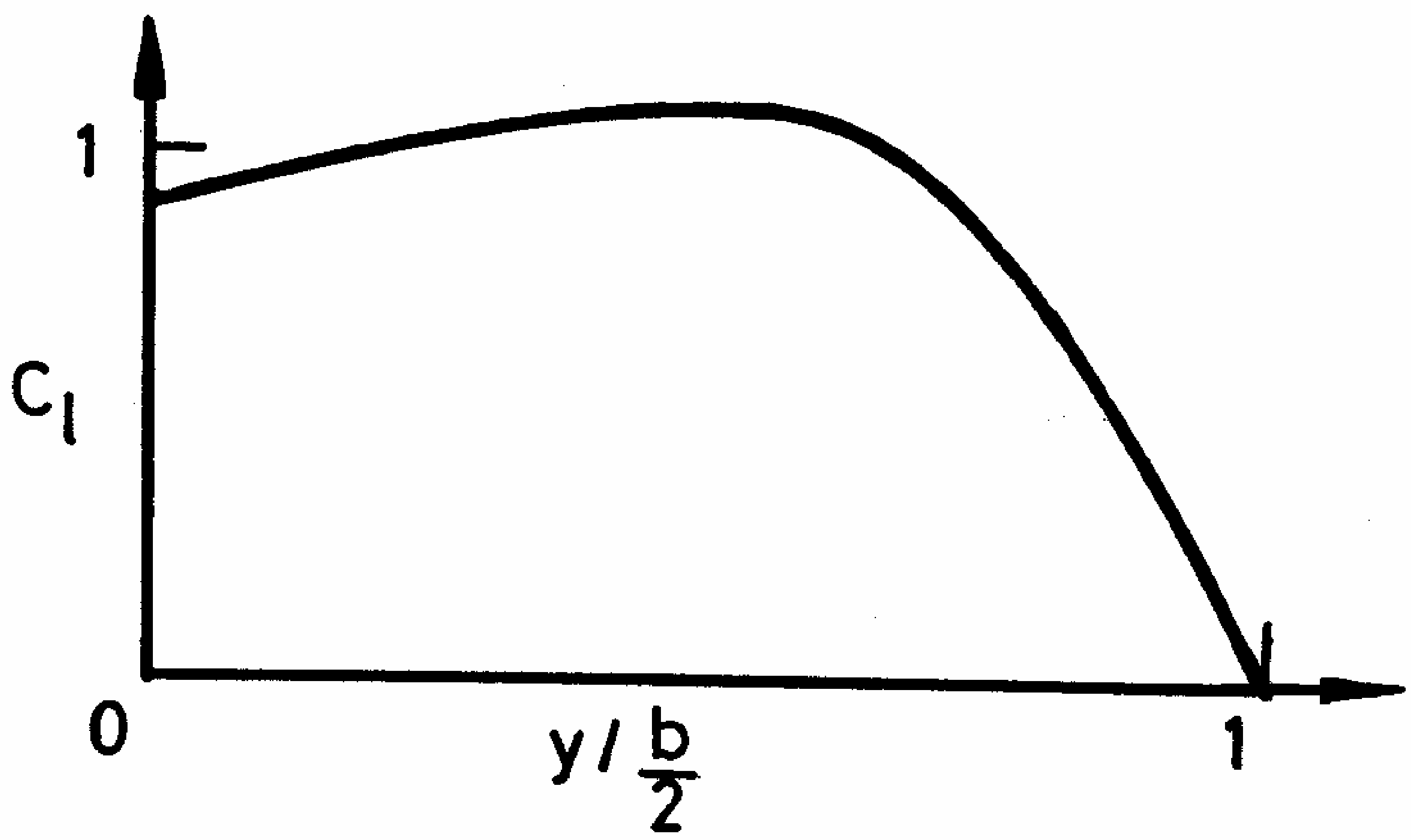

Fig 5.13 Typical Distribution of C_l

the span-wise location where local C_l equals local C_{lmax}.

Subsequently, stalling will progress along the wing span and finally the wing will stall (i.e. C_L of wing will reach a maximum and then decrease).

The beginning of stall near the tip is undesirable as ailerons are located in tip region. Stalling there would reduce aileron effectiveness. For a wing of a taper ratio 0.3, the stall is likely to begin around y/(b/2) of 0.7.

Remark:

In the case of swept wings, there is a cross flow along the span and the tendency for the tip stall is enhanced.

Tip stalling can be prevented if the sections in the tip region have angles of attack lower than those at the root. In this case, the wing acquires a twist. The difference between the angle of attack of the airfoil at the root and that near the tip is called twist and denoted by ε (Fig.5.1). Twist is negative when airfoil near the tip is at an angle of attack lower than that at the root. This is also called wash-out. Sometimes airfoils with higher C_{lmax} are used near the tip. Thus airfoils at the root and near the tip may have a different values of angle of zero lift (α_{0l}). This leads to two different kinds of twists – geometric twist and aerodynamic twist.

Geometric twist is the angle between the chords of the airfoils at the root and near the tip. Aerodynamic twist is the angle between the zero lift lines at the root and that near the tip.

To completely eliminate the occurrence of tip stalling, may require complex variation of the angle of twist. However for ease of fabrication, linear twist is given in which the angle of twist varies linearly along the span.

Remarks:

i) Actual value of twist can be obtained by calculating C_l distribution on untwisted wing and then varying the twist such that tip-stalling is avoided. A value of 3^0 is suggested, in Ref. 1.11, chapter 4 , as an initial estimate.

ii) Early swept wing airplanes had the following features to avoid tip stalling . (a) Vortex generators, (b) Fences on top surface.

5.5 Wing incidence

The mean aerodynamic chord is the reference line on the wing. Fuselage reference line (FRL) is the reference line for the entire airplane. The angle between fuselage reference line and the wing reference line is called wing incidence and denoted by i_w. The wing incidence is given for the following reason.

For the economy in fuel consumption, the drag should be minimum during the cruise. The fuselage has a minimum drag when its angle of attack is zero. However, during cruise, the wing should produce sufficient lift to support the weight of the airplane. Keeping these factors in view, the wing is mounted on the fuselage in such a manner that it produces required amount of lift in cruise while the fuselage is at zero angle of attack.

During the preliminary design phase, i_w can be obtained as follows.

a) Obtain $C_{Ldesign}$ corresponding to cruise or any other design condition i.e.

$$C_{Ldesign} = \frac{W}{\frac{1}{2}\rho V^2 S}$$

where ρ and V correspond to the design flight conditions.

b) Obtain $C_{L\alpha}$ for the wing .

c) Obtain zero lift angle (α_{0L}) for wing. This depends on α_{0l} of the airfoil used on the wing and the wing twist (See for example Ref. 5.2, chapter 2).

d) Calculate i_w from the following equation:

$$C_{L_{design}} = C_{L_\alpha}\left(i_w - \alpha_{OL}\right) \tag{5.7}$$

Remark:

The final choice of i_w may be arrived at from wind tunnel tests on the airplane model. Reference 1.11 chapter 4 gives following guidelines for preliminary design purposes.

Airplane type	Wing incidence angle
General aviation/ home built	2^0
Transport	1^0
Military	0^0

Table 5.2 Suggested wing incidence angle
(Adapted from Ref. 1.11, chapter 4)

5.6 Dihedral

Figure 5.1 shows the dihedral angle Γ. Its value is decided after the lateral dynamic stability calculations have been carried out for the airplane. Reference 1.11, chapter 4 gives following guidelines for preliminary design purposes.

Airplane Type	Γ°		
	Wing Location		
	Low	Mid	High
Unswept (civil)	5 to 7	2 to 4	0 to 2
Subsonic swept	3 to 7	-2 to 2	-5 to –2
Supersonic swept	0 to 5	-5 to 0	-5 to 0

Table 5.3 Suggested dihedral angle
(Adapted from Ref. 1.11, chapter 4)

5.7 Wing vertical location

There are three choices for the location of the wing on the fuselage namely high-, mid- and low- wing. Figure 5.14 shows three military airplanes with these locations for the wing.

Following Ref. 1.11, chapter 4, the advantages and disadvantages of the three configurations are given below.

High Wing configuration:

Advantages:

i) Allows placing fuselage closer to ground, thus allowing loading and unloading without special ground handling equipment.

ii) Jet engines & propeller have sufficient ground clearance without excessive landing gear length

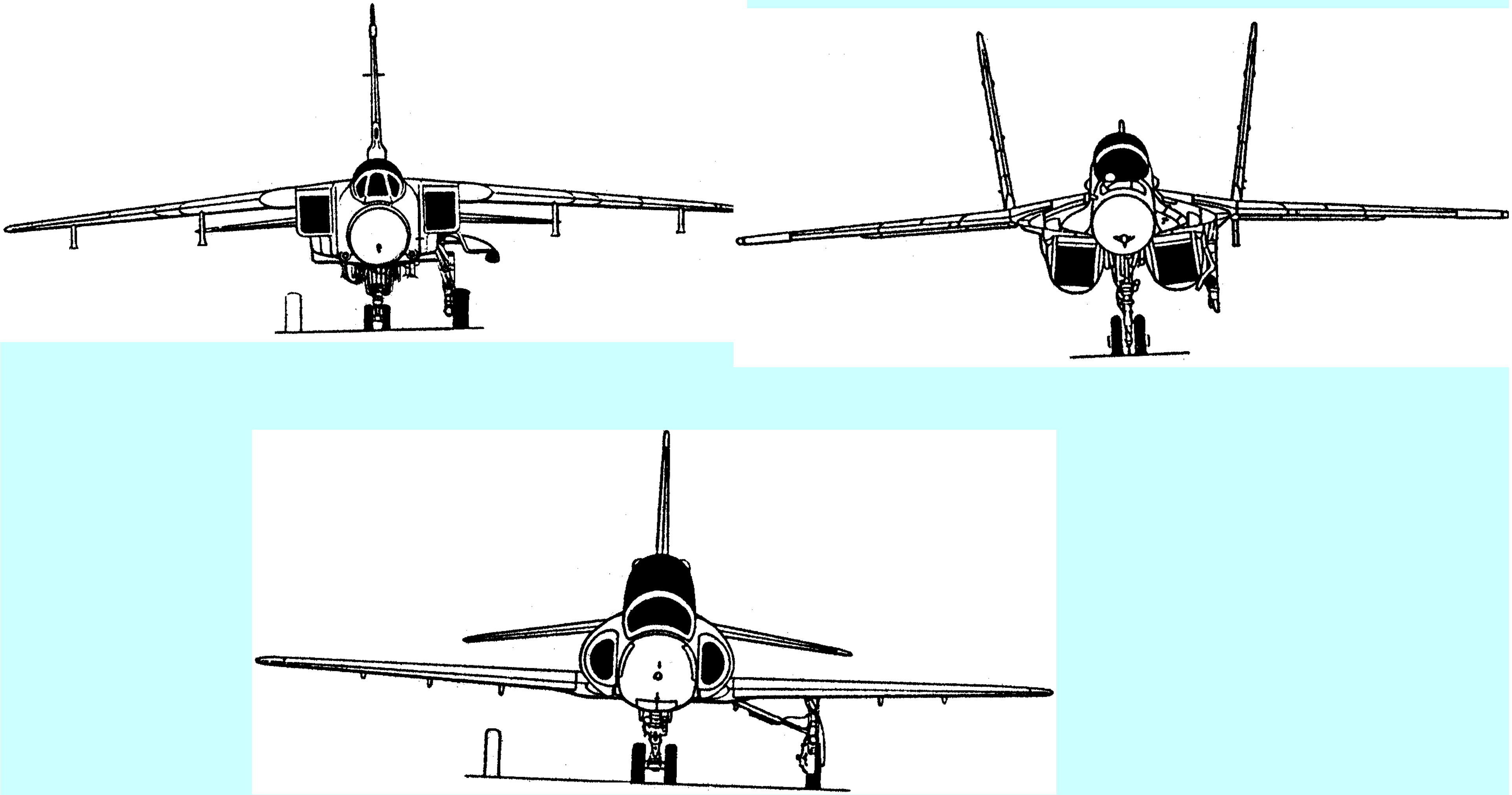

Fig.5.14 High-wing, mid-wing and low-win configurations
(Adapted from Ref. 1.2, p 247, 116 and 211)

leading to lower landing gear weight.

iii) For low speed airplanes, weight saving can be effected by strut braced wing.

iv) For short take off and landing (STOL) airplanes with high wing configuration have following specific advantages. (a) Large wing flaps can be used (b) Engines are away from the ground and hence ingestion of debris rising from unprepared runways is avoided (c) Prevents floating of wing due to ground effect which may occur for low wing configuration.

Disadvantages:

i. Fuselage generally houses the landing gear in special pods leading to higher weight and drag.

ii. Pilot's visibility may be blocked in a turn.

Mid Wing configuration

Advantages:

i. Lower drag.

ii. Advantages of ground clearance as in the high wing configuration.

iii. No blockage of visibility. Hence used on some military airplanes.

Disadvantages:

i. Wing root structure passing through the fuselage is not possible, which leads to higher weight. However in HFB Hansa airplane, a swept forward mid-wing is located behind the passenger cabin and has carry through structure.

Low Wing configuration

Advantages:

i. Landing gear can be located in the wing thereby avoiding pods on the fuselage and hence lower drag. However to provide adequate ground clearance, the fuselage has to be at a higher level as compared to the high wing configuration.

ii. Wing structure can be through the fuselage.

Disadvantages:

i. Low ground clearance.

ii. A low-wing configuration has unstable contribution to the directional stability. Hence a larger vertical tail area is needed.

5.8 Other aspects

See Ref. 1.11, chapter 4 for shapes of wing tips.

See Ref. 4.2 for information on winglets.

See websites for information on a) wing strakes , b) double delta wings, c) behavior of delta wings at high angles of attack etc.

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EXERCISES

5.1 Distinguish between gross area, exposed area and reference area of a wing.

5.2 Consider two untwisted wings ‘T’ and ‘R’. Wing ‘T’ has a taper ratio of 0.3 and wing ‘R’ is rectangular. Both wings have same span ‘b’ and planform area ‘S’. Both have the same airfoil NASA G-A(W)-1 with 17% thickness ratio along the span. Answer the Following:

a. What is the ratio of the maximum thickness at root for these two wings ?

b. Express the chord as function of the spanwise distance ‘y’ for the wing ‘T’.

c. Find the ratio of the internal volumes of the two wings. Assume that the wing cross sectional area at any spanwise station is proportional to the product of local chord and thickness.

[Answer : (a)1.538; (b) $c = c_r\{1 - 0.7(2y/b)\}$; (c) 1.0965]

5.3 The weight of a wing is given by the following expression. Justify the indices of various terms involving S, A, t/c, λ and Λ.

$$W_{wing} = 0.0051(W_{dg}N_Z)^{0.557}\, S_w^{0.649} A^{0.5} (t/c)^{-0.4}\, (1+\lambda)^{0.1}_{caw}$$

$$x\ (\cos\Lambda)^{-1.0}\, S^{0.1}$$