PROGETTO DELL'ALA

Scelta dell'ala:

- → Configurazione strutturale
  - ala a sbalzo
  - ala controventata
- → Posiz in fusoliera
  - alta
  - media
  - bassa

(considerat Interferenza
Stabilità
visibilità

- → freccia
- → spess % profilo
- → scelta λ e disegno forma in pianta
- → scelta Γ
- → scelta PROFILO ALARE
- → scelta calettamento i_w (C_D in crociera)
- → scelta svergolamento geometrico

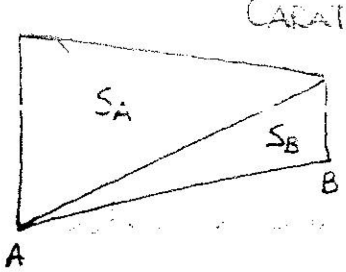
PROFILO

Vedere da Jane's i velivoli simili

VELIVOLI LEGGERI	NACA 23015	
BIMOTORI ELICA	64_3-418 64_2-415	GAW

Scelta profilo radice 18% 15%
estremità 15% 12%

- DETERMINARE LE CARATTERISTICHE AERODINAMICHE DEL PROFILO MEDIO



$$K_A = \frac{2S_A}{S} \qquad K_B = \frac{2S_B}{S}$$

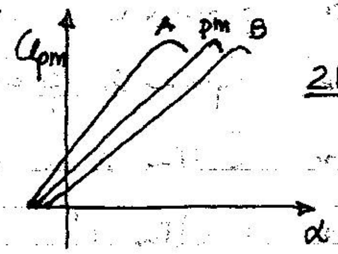
$$\underline{C_\ell = K_A C_{\ell_A} + K_B C_{\ell_B}}$$

α α_A α_B C_{ℓ_A} C_{ℓ_B} $K_A C_{\ell_A}$ $K_B C_{\ell_B}$ C_ℓ V Re_A Re_B

Una volta determinati i Re_A e Re_B si riprendono i C_{ℓ_A} e C_{ℓ_B} agli effettivi Re di volo e si rifà la tabella.

Si fa lo stesso procedimento per il C_d

Quindi



2D

α	$C_{\ell_{pm}}$	$C_{d_{pm}}$
⋮	⋮	⋮

vedi pag W9-W10

ALA ISOLATA

→ Coeff angolare retta di portanza C_{L_α} (o anche a_W)

Per il coeff angolare ci sono varie formule

$$C_{L_\alpha} = f \frac{C_{\ell_{\alpha_{pm}}} / \left(\frac{P}{b}\right)}{1 + \left[57.3\, C_{\ell_{\alpha_{pm}}} \Big/ \left(\pi AR \frac{P}{b}\right)\right]} \qquad C_{\ell_\alpha} \text{ in } [1/°]$$

o anche la stessa con $\left(\frac{P}{b}\right) = 1$

Per angoli di freccia e/o Mach elevati (comprimibile)

$$C_{L_\alpha} = \frac{2\pi AR}{2 + \sqrt{\frac{AR^2 \beta^2}{K^2}\left(1 + \frac{\tan^2\Lambda}{\beta^2}\right) + 4}}$$

con $\beta^2 = 1 - M^2$

$\Lambda = \Lambda_{c/2}$

$K = \dfrac{C_{l\alpha}\,\beta}{2\pi}$ con $C_{l\alpha}$ in rad

(Quindi è il rapp tra il $C_{l\alpha}$ del profilo, ad es. 5.8 e il $C_{l\alpha}$ della lastra piana che è 2π).

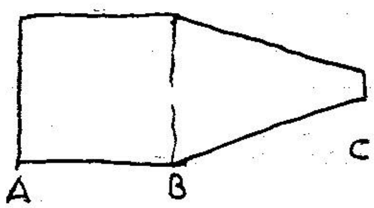
Il $C_{L\alpha}$ si può ricavare anche con MULTHOPP, cioè con il codice che vi do.

→ α_{ZL} dell'ala

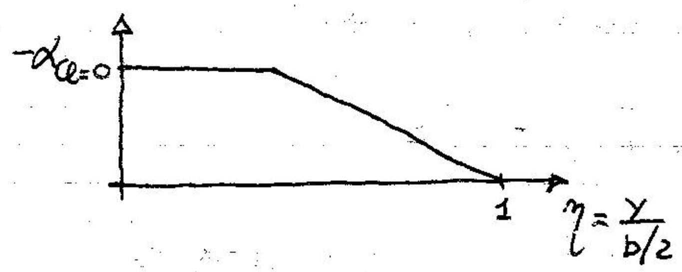
Per determinare l'angolo di portanza nulla dell'ala si deve determinare lo svergolamento aerodinamico equivalente.

Per ogni sezione ~~[illegible]~~ valuto

$$\alpha_{Cl=0} = \alpha_{Zl} - \varepsilon$$

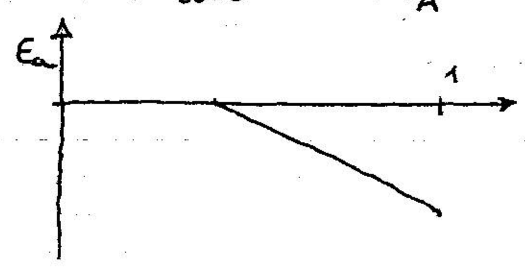


Es :

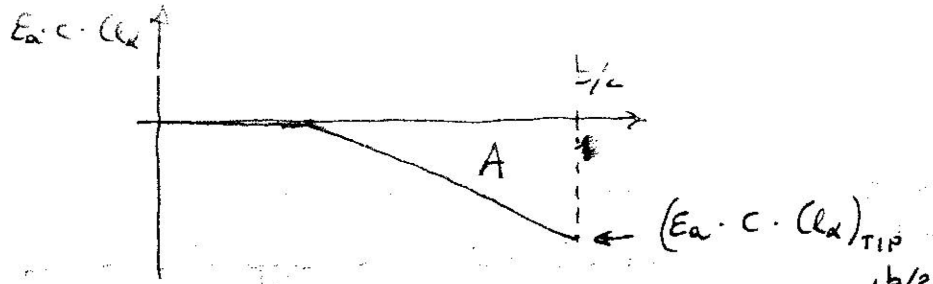


Poi

$$\varepsilon_a = -\left(\alpha_{Cl=0} - \alpha_{Cl=0_A}\right)$$



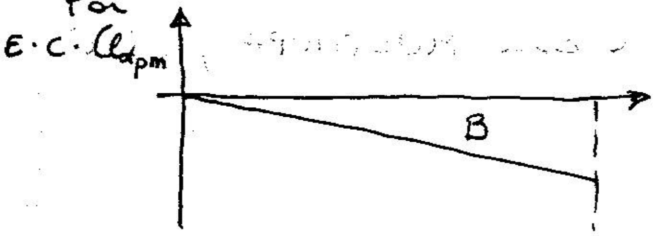
Poi si diagramma $\varepsilon_a \cdot c \cdot C_{l\alpha}$



$$A = \int_0^{b/2} \varepsilon_a(y) \cdot c(y) \cdot Cl_\alpha(y)\, dy$$

Si valuta l'area A e

Poi



Imponendo $A = B = \varepsilon \cdot c_{TIP} \cdot Cl_{\alpha_{pm}} \cdot \frac{b}{2} \cdot \frac{1}{2}$

Da cui

$$\varepsilon = \frac{4A}{c_{TIP}\; Cl_{\alpha_{pm}}\; b}$$

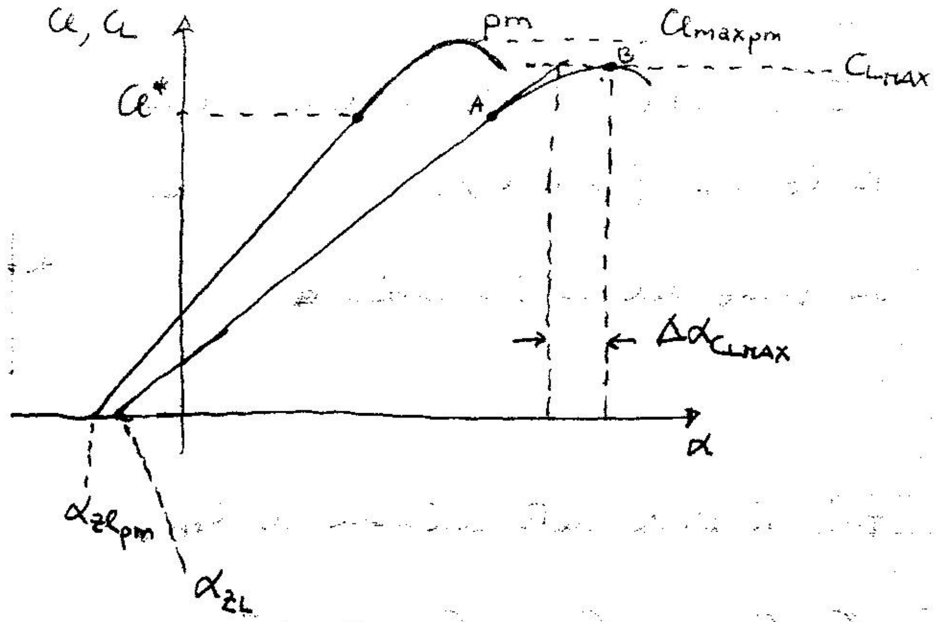
Valutato ε (svergolamento aerod. equival.)

$$\alpha_{ZL} = (\alpha_{CL=0})_{ROOT} + J\varepsilon$$

Dove J si det dal grafico di fig 9 (pag 13)

Comunque α_{ZL} si può valutare anche da Multhopp.
(codice che vi ho dato).

CURVA DI PORTANZA ALA



→ $C_{LMAX} = f(Cl_{max_{pm}})$ pag 20/b fig 4-2-4

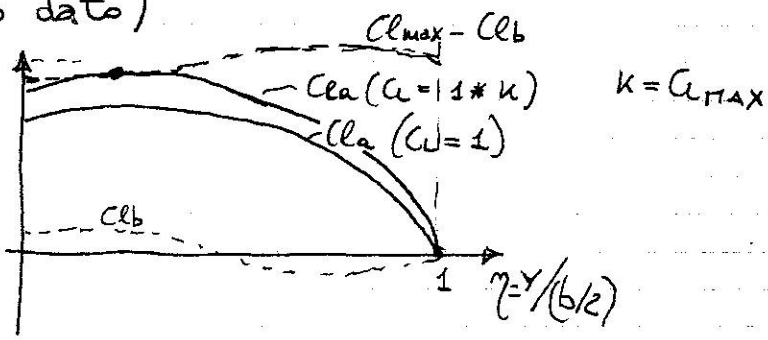
→ $\Delta\alpha_{CLMAX}$ ⇒ pag 20/b fig 4-2-5

Per il tratto non lineare si può usare un polinomio che si ottiene imponendo le condizioni

→ Pos. per A
→ Pos per B
→ Der. prima in A = $C_{L\alpha}$
→ " " B = 0

(Quindi una cubica tipo $y(x) = a_0 + a_1 x + a_2 x^2 + a_3 x^3$)

Il C_{LMAX} si potrebbe trovare anche tramite il sentiero di stallo.
(Programma che vi ho dato)

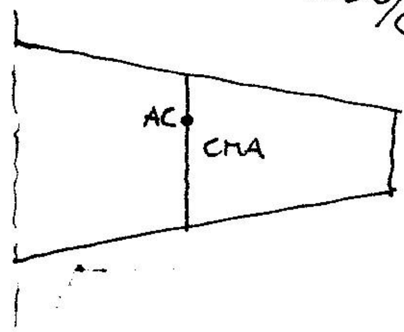


DETERMINARE ANCHE LE CARATTERISTICHE
GEOMETRICHE E AERODINAMICHE

$\bar{c}$ o MAC (Mean Aer. Chord) (vedi pag W20/c - W25)

Posiz AC (in x e y)

In funz del conico basico



Poi si deve determinare il $C_{M_{0W}}$

$$C_{M_{0W}} = C_{M_1} + C_{M_2} = \text{cost}$$

$$C_{M_1} = \frac{2}{S\bar{c}} \int_0^{b/2} Cl_b \, x \, c \, dy$$

$$C_{M_2} = \frac{2}{S\bar{c}} \int_0^{b/2} C_{mac}(y) \, c^2 \, dy$$

C_D dell'ala

$$C_{D_W} = C_{Do_W} + C_{Di_W}$$

$$C_{Do_W} = Cd_{pm}$$

Cioè $\quad C_{Lw} \rightarrow Cl_{pm} \rightarrow Cd_{pm}$

Nel tratto non lineare la corrisp. tra Cl_{pm} e C_{Lw} è data dalla formula:

$$C_{L_W} = Cl_{pm}\left[1 - K \frac{(Cl - Cl^*)_{pm}}{(Cl_{max} - Cl^*)_{pm}}\right]$$

dove $K = \dfrac{C_{L_{MAX}}}{Cl_{max_{pm}}}$

Res. indotta

(Vedi Abbott)

$$C_{Di_W} = \frac{C_{Lw}^2}{\pi AR\, u} + C_L \cdot \varepsilon \frac{Cl_{\alpha_{pm}}}{p/b} v + \varepsilon^2 \left[\frac{Cl_{\alpha_{pm}}}{p/b}\right]^2 w$$

Dove

u, v, w sono in grafici pag 13 e 14

ε è lo sverg. aerod. equivalente

La resistenza indotta si può anche ricavare dal codice che vi ho dato, ad ogni C_L

Riportando C_{Lw}^2 e C_{Di_W} si potrebbe ricavare il fattore di Oswald (solo la parte indotta) dell'ala isolata

Esiste anche una formula più approssimata per il calcolo del C_{Do_W}

(W8)

Da Roskam (vedi pag ~~[illegible]~~ appunti)

$$C_{D_{0w}} = R_{wf}\ R_{LS}\ C_{f_w} \left[1 + L' \left(\frac{t}{c}\right) + 100 \left(\frac{t}{c}\right)^4 \right] \frac{S_{wet}}{S}$$

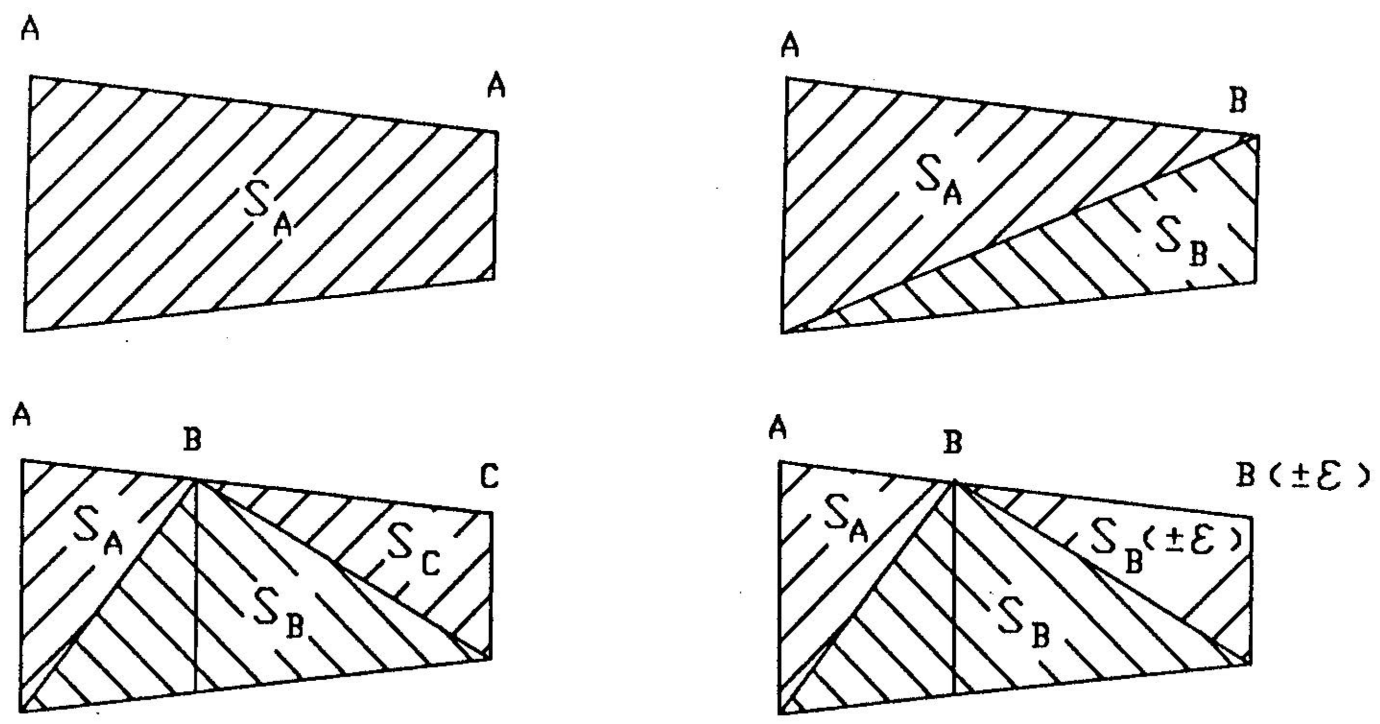
R_{wf} fig 4.1 pag P10

R_{LS} fig 4.2 P10

C_{f_w} coeff attrito lastra piana turbol. fig 4.3 pag P11

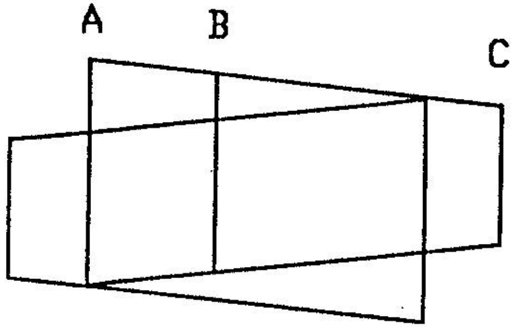
L' fig 4.4. pag P12

t/c spess % profilo



$$K_A = \frac{2S_A}{S}$$

$$K_B = \frac{2S_B}{S}$$ ecc. ecc. $$\sum_{i=1}^{n} K_i = 1$$

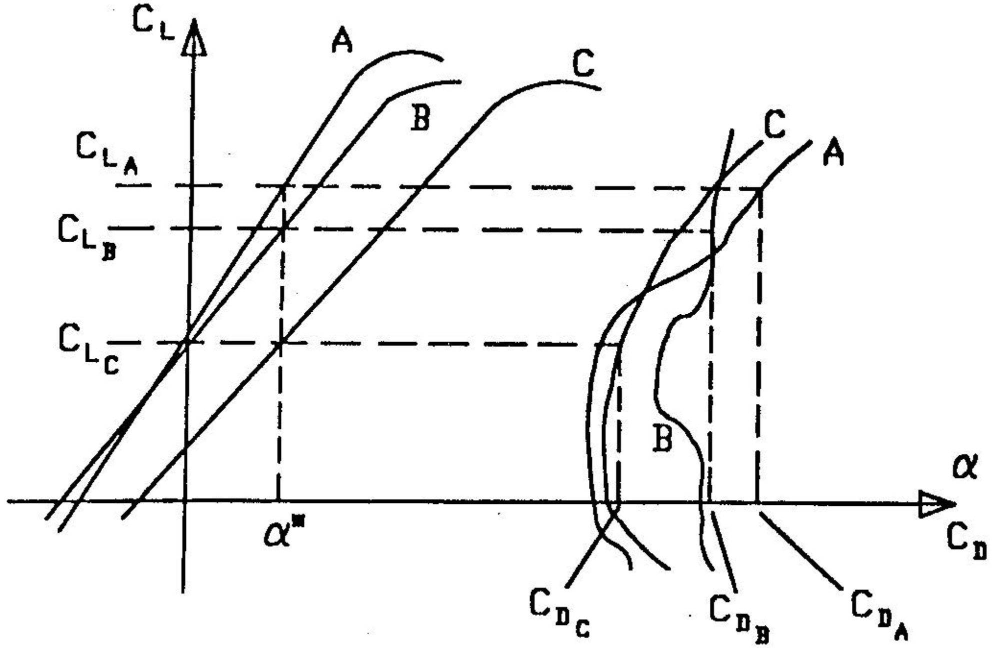


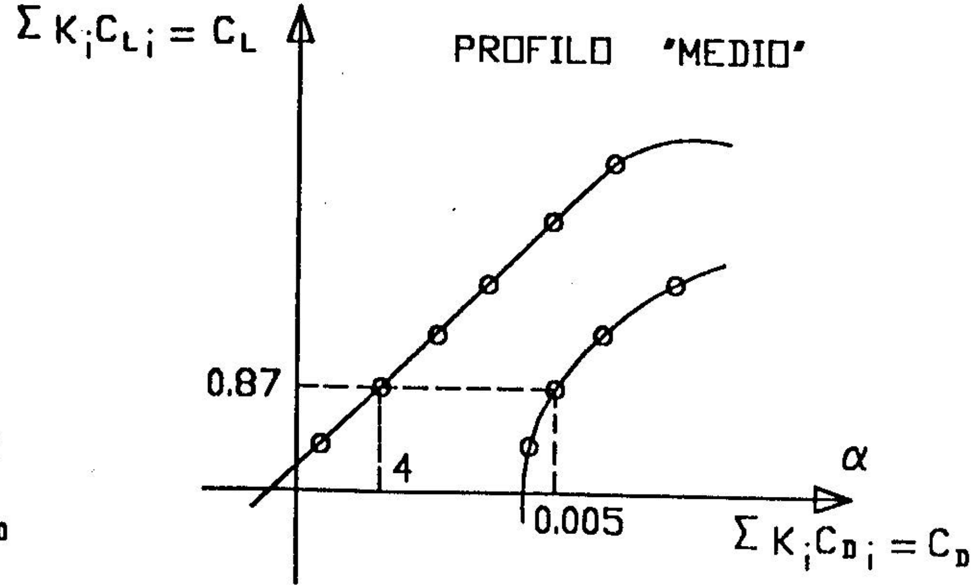
$K_A \cong 0.2$

$K_B \cong 0.65$

$K_C \cong 0.15$

* α	C_{L_A}	C_{L_B}	C_{L_C}	$K_A C_{L_A}$	$K_B C_{L_B}$	$K_C C_{L_C}$	* $\sum K_I C_{L_I}$	C_{D_A}	C_{D_B}	C_{D_C}	$K_A C_{D_A}$	$K_B C_{D_B}$	$K_C C_{D_C}$	* $\sum K_I C_{D_I}$
.	.	.	.	.	.	.	.	.	.	.	.	.	.	.
.	.	.	.	.	.	.	.	.	.	.	.	.	.	.
4	1.05	0.9	0.5	0.21	0.58	0.08	0.87	.006	.005	.004	.001	.005	.001	.005
.	.	.	.	.	.	.	.	.	.	.	.	.	.	.
.	.	.	.	.	.	.	.	.	.	.	.	.	.	.





L'ALA ISOLATA

Determinare dapprima la pendenza della curva di portanza del profilo medio ($C_{L\alpha}$) e l'incidenza di portanza nulla ($\alpha_{C_L=0}$), applicare, quindi, la seguente formula semiempirica:

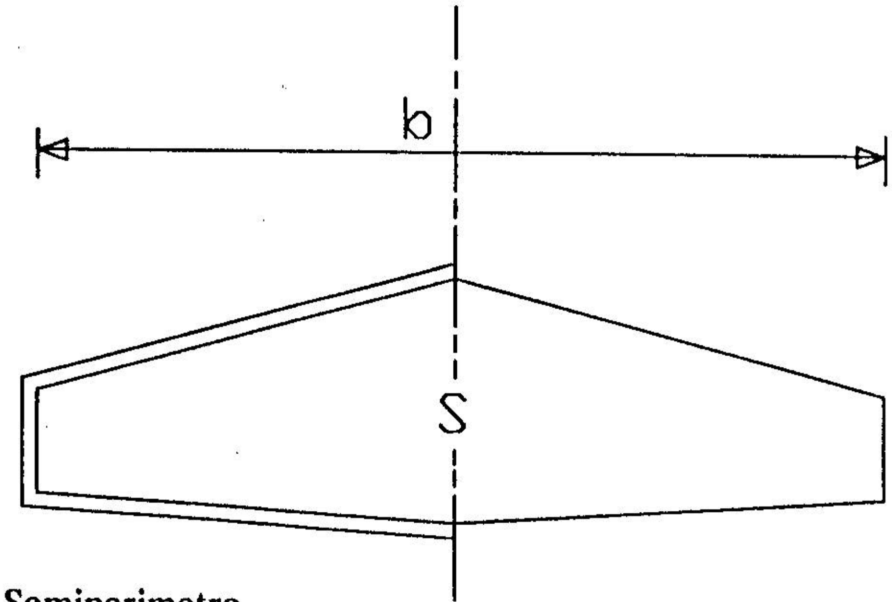
$$\left(\frac{\partial C_L}{\partial \alpha}\right)_{ala} = f \frac{\left(\frac{\partial C_L}{\partial \alpha}\right)_{\substack{prof\\medio}} \Big/ \left(\frac{p}{b}\right)}{1 + \left[57{,}3 \left(\frac{\partial C_L}{\partial \alpha}\right)_{\substack{prof\\medio}} \Big/ \left(\pi \lambda \frac{p}{b}\right)\right]}$$

f = *coefficiente correttivo*

p = *semiperimetro alare*

b = *apertura alare*

λ = *allungamento geometrico* $= \frac{b^2}{S}$

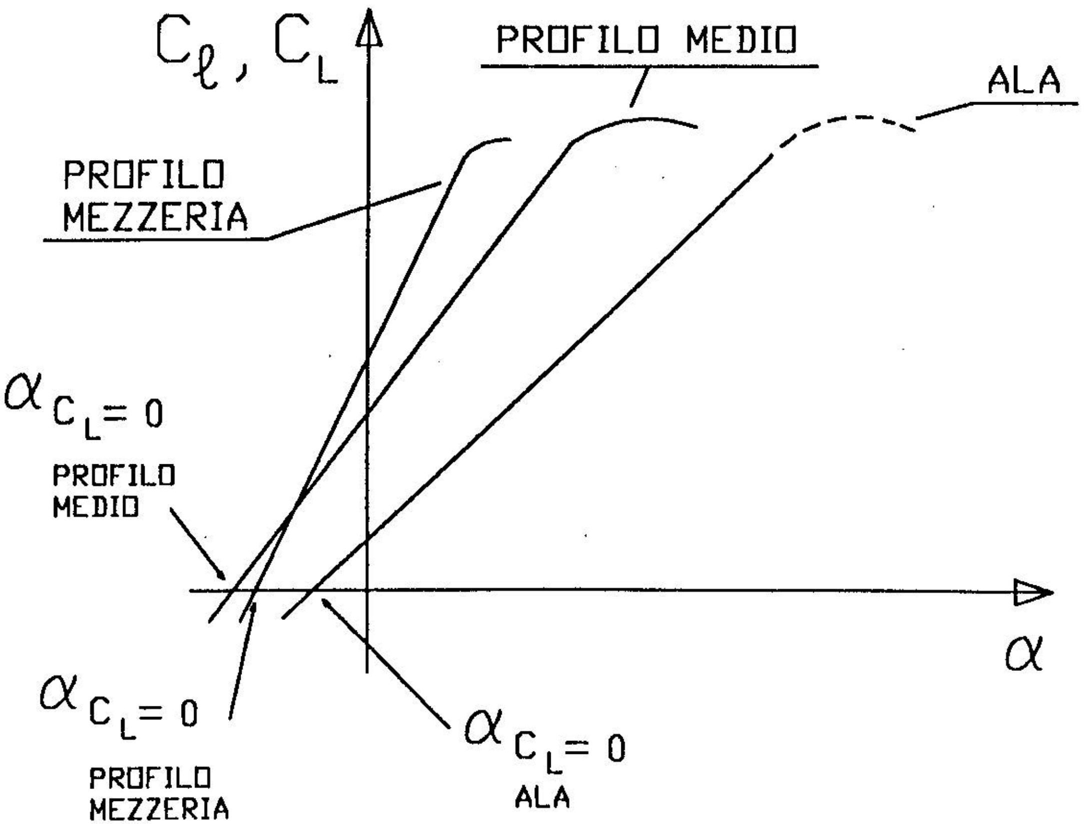


Semiperimetro

W11

$$\left(\alpha_{C_L=0}\right)_{\text{ala}} = \left(\alpha_{C_L=0}\right)_{\substack{\text{profilo in} \\ \text{mezzeria alare}}} + J\,\varepsilon$$

j = fattore correttivo
ε = svergolamento aerodinamico equivalente



W13

where the effective section lift-curve slope a_e is taken as the average for

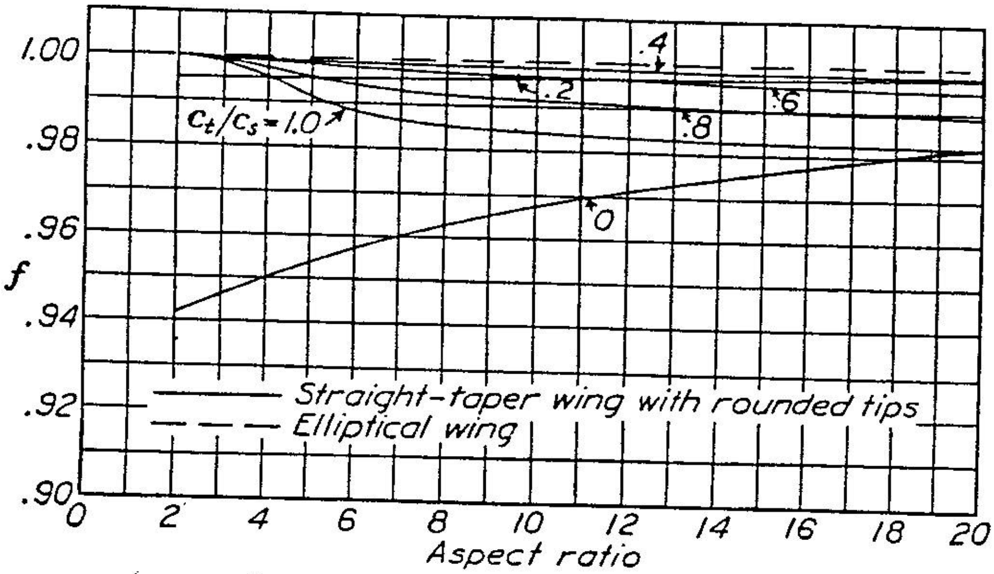


FIG. 8. Chart for determining lift-curve slope.

$$a = f\frac{a_e}{1 + (57.3a_e/\pi A)}$$

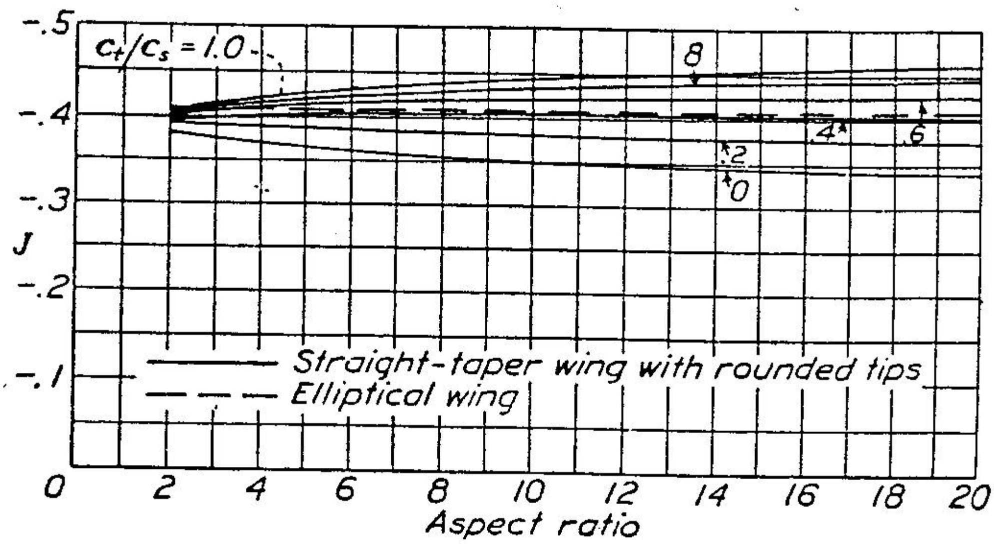
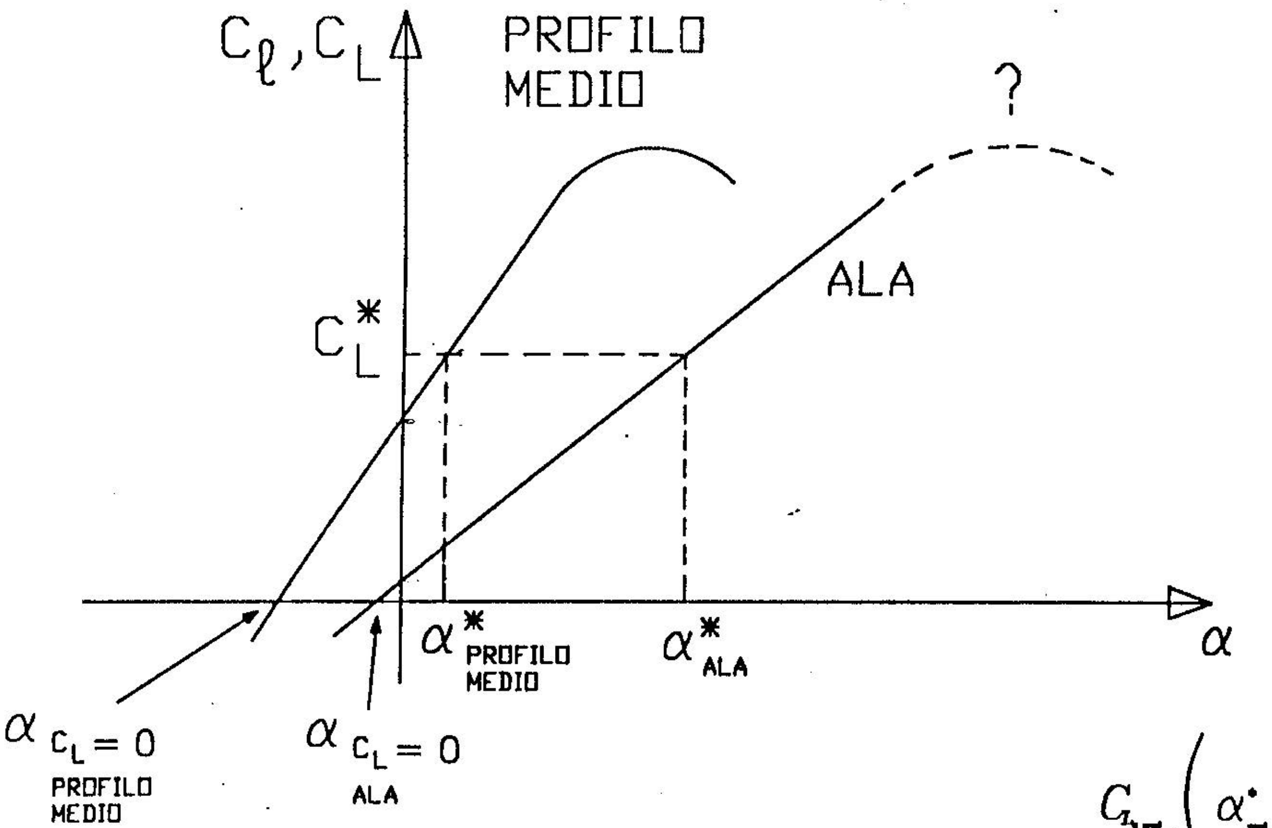


FIG. 9. Chart for determining angle of attack.

$$\alpha_s = \frac{C_L}{a} + \alpha_{l_{0_s}} + J\epsilon \qquad \alpha_{s(L=0)} = \alpha_{l_{0_s}} + J\epsilon$$

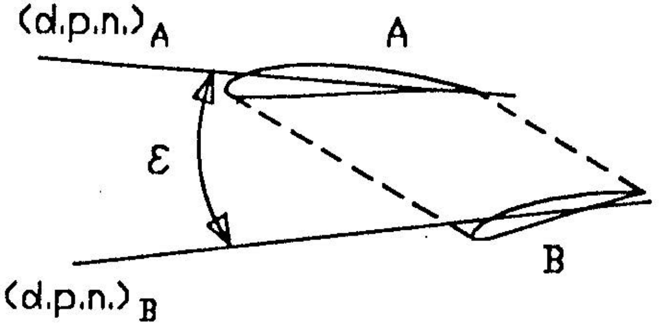


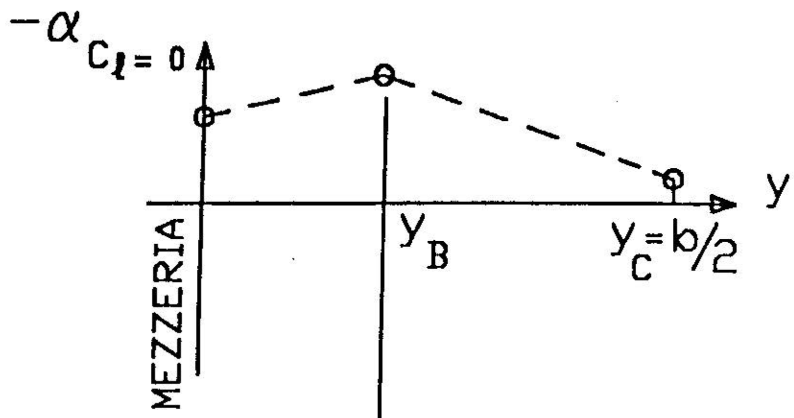
$$C_{L_\alpha,\,\text{p.m.}}\left(\alpha^*_{\text{p.m.}} - \alpha_{0,\,\text{p.m.}}\right) = C_L^* =$$

$$= C_{L_\alpha}\left(\alpha^*_{a} - \alpha_{0,\,a}\right) =$$

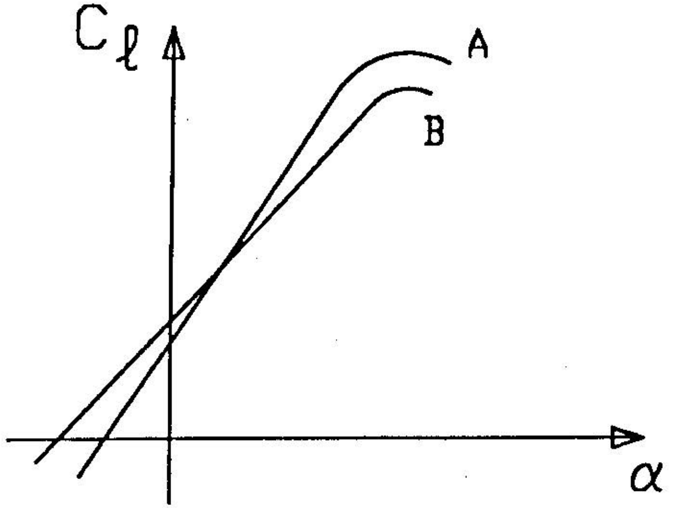
$$\alpha_{a} = \left(\alpha^*_{\text{p.m.}} - \alpha_{0,\,\text{p.m.}}\right)\frac{C_{L_\alpha,\,\text{p.m.}}}{C_{L_\alpha}} + \alpha_{0,\,a}$$

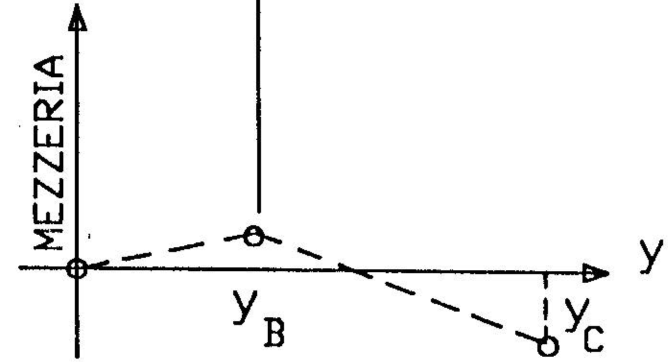
ε = SVERGOLAMENTO AERODINAMICO



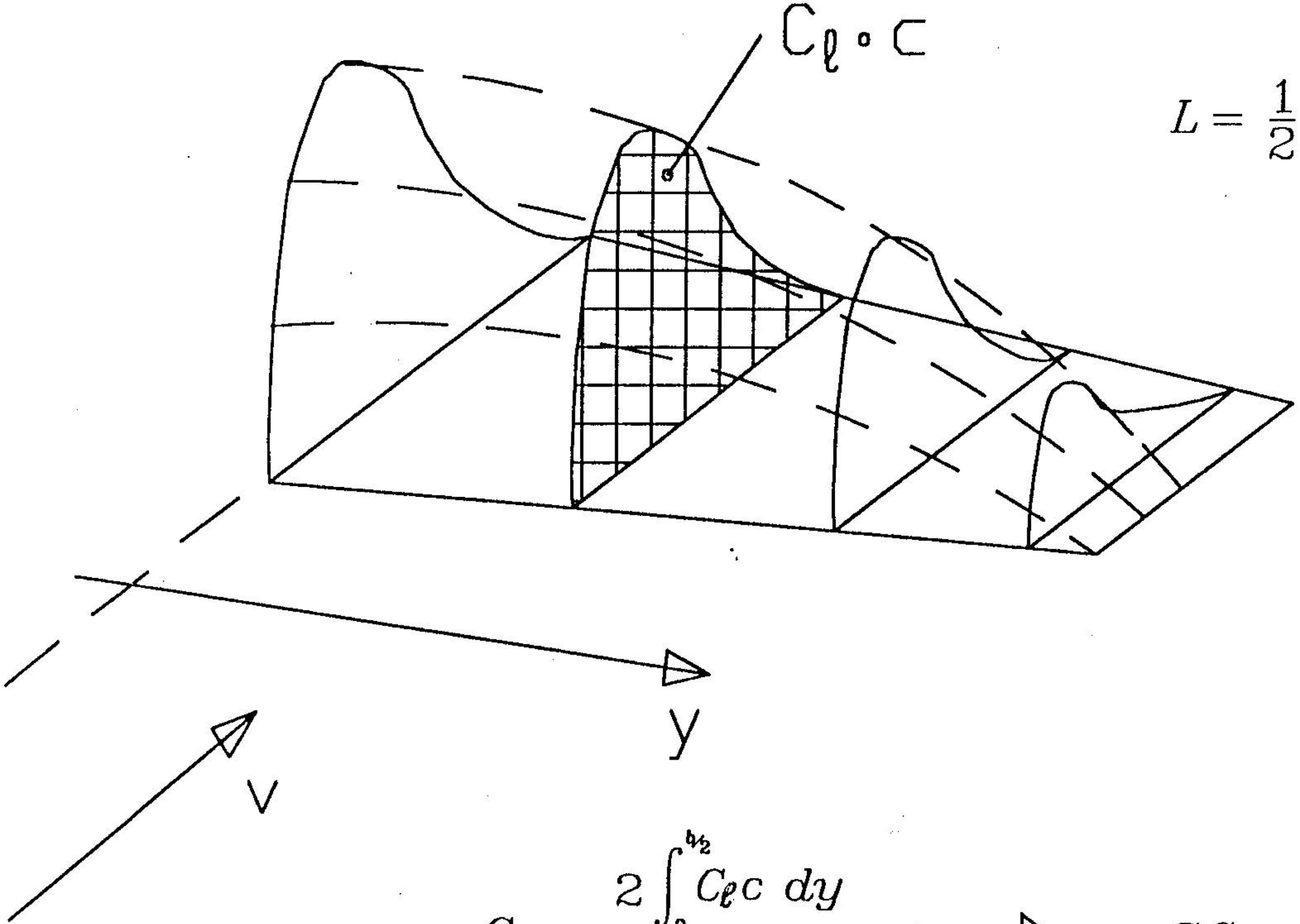


$$-\left(\alpha_{C_\ell=0} - \alpha_{C_\ell=0_A}\right) = \varepsilon$$





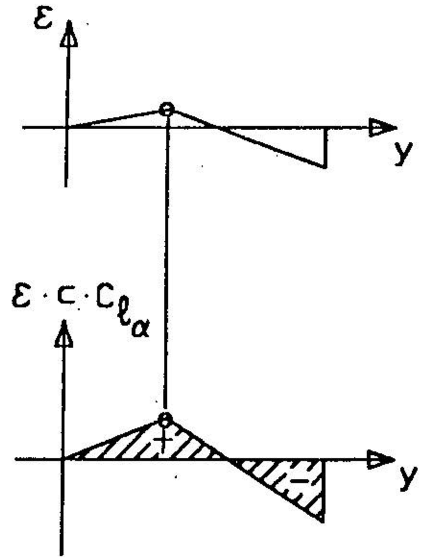
W15



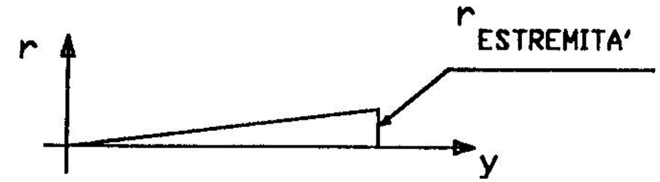
$$L = \frac{1}{2}\rho v^2 S C_L$$

$$C_L = \frac{2\int_0^{b/2} C_\ell c \, dy}{S} \Longrightarrow S C_L = 2\int_0^{b/2} C_\ell c \, dy$$

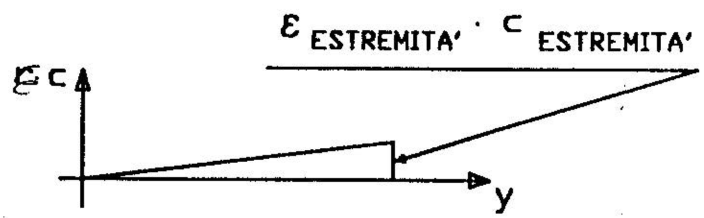
W17



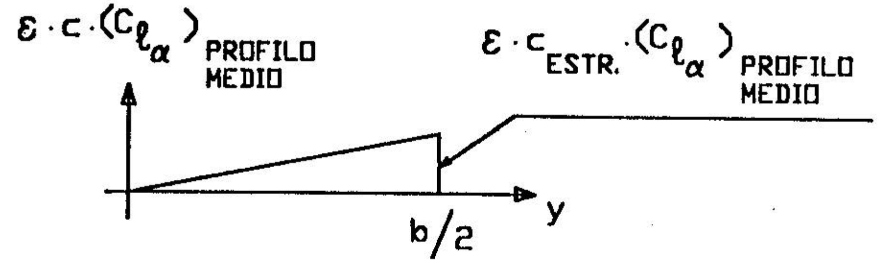
Integrale = A



SVERGOLAMENTO LINEARMENTE DISTRIBUITO



SVERGOLAMENTO UNIFORMEMENTE DISTRIBUITO

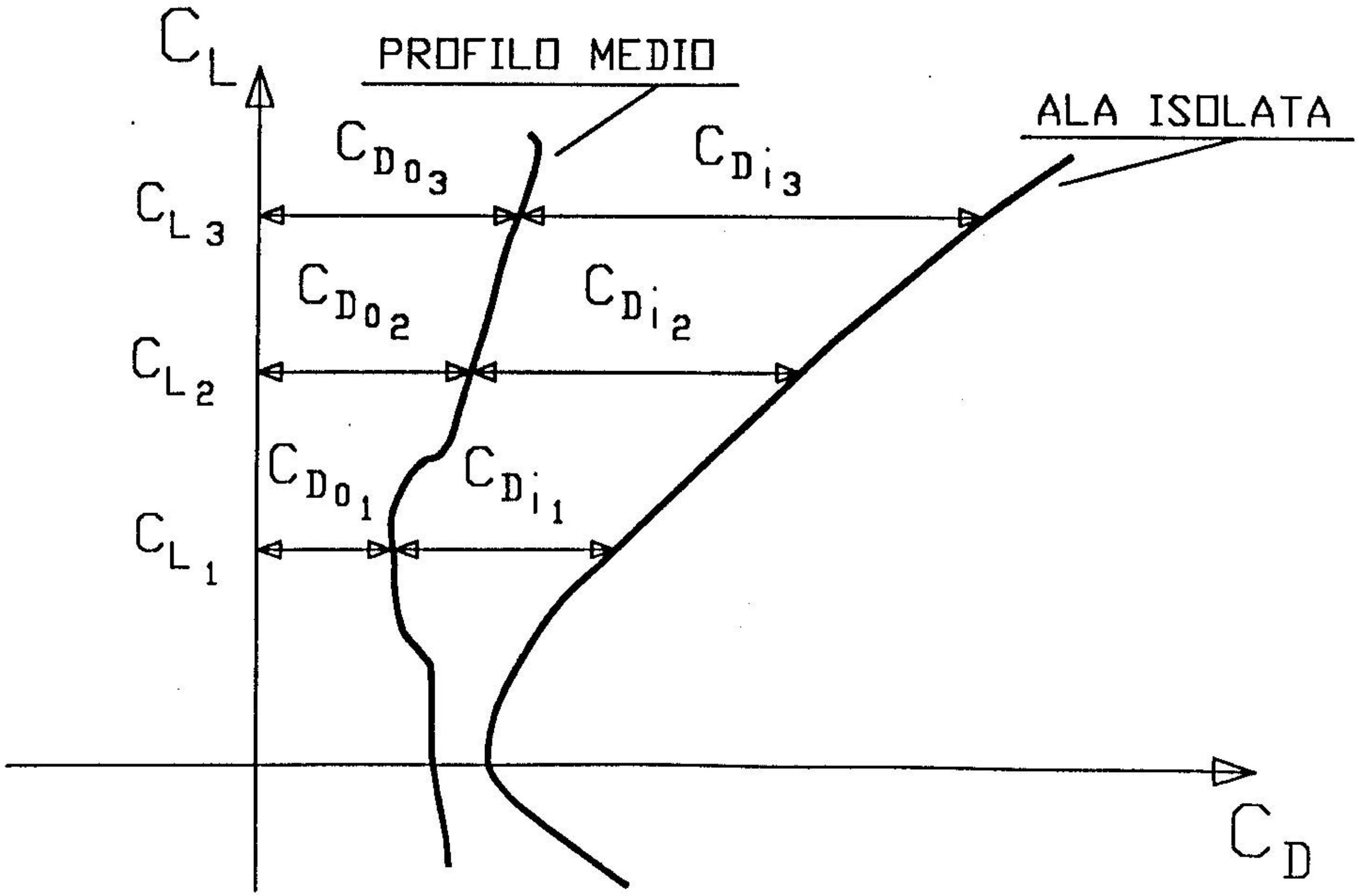


SVERGOLAMENTO AERODINAMICO EQUIVALENTE

$$A = \varepsilon\, C_{-} \left(C_{L}\right)_{pm} \frac{b}{2}\,\frac{1}{2}$$

$$\varepsilon = \frac{4\,A}{C_{-}\left(C_{L}\right)_{pm} b}$$

$$C_{D_i} = \frac{C_L^{\,2}}{\pi \lambda u} + C_L \varepsilon \, \frac{\left(C_{L_0}\right)}{p/b} v + \varepsilon^2 \left[\frac{\left(C_{L_0}\right)}{p/b} \right]^2 w$$



W18

RES.
INDOTTA
ALA

W19

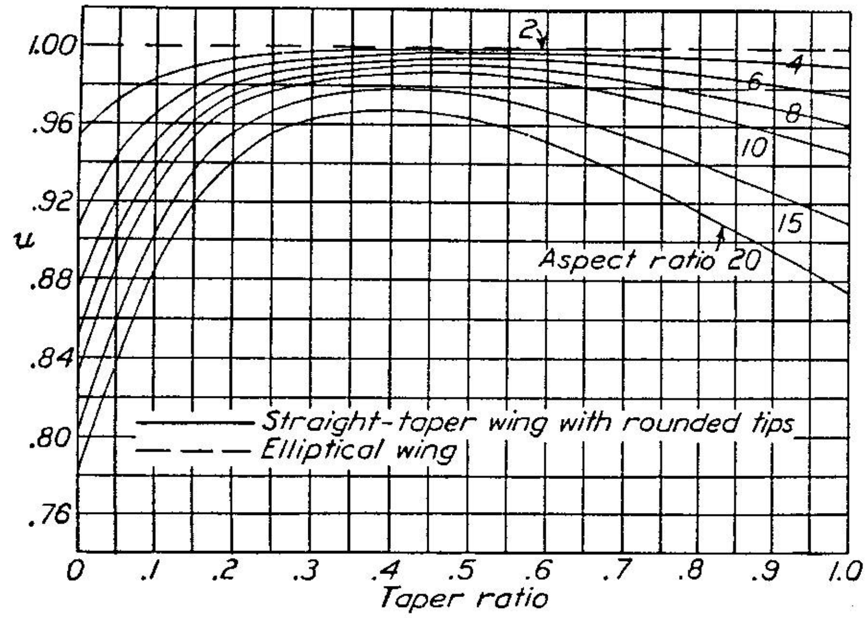


FIG. 10. Chart for determining induced-drag factor u.

$$\boxed{C_{D_i} = \frac{C_L{}^2}{\pi A u} + C_L \epsilon a_e v + (\epsilon a_e)^2 w}$$

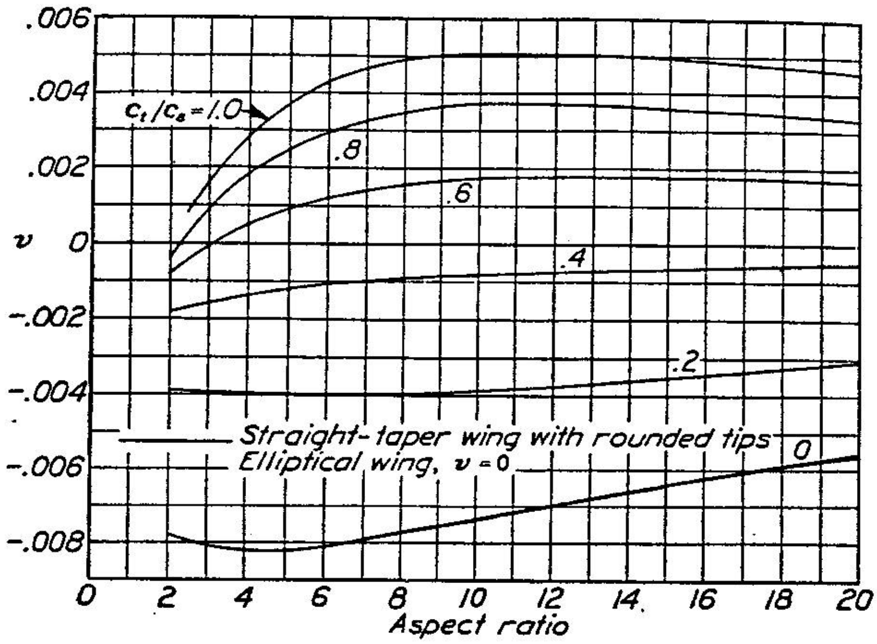


FIG. 11. Chart for determining induced-drag factor v.

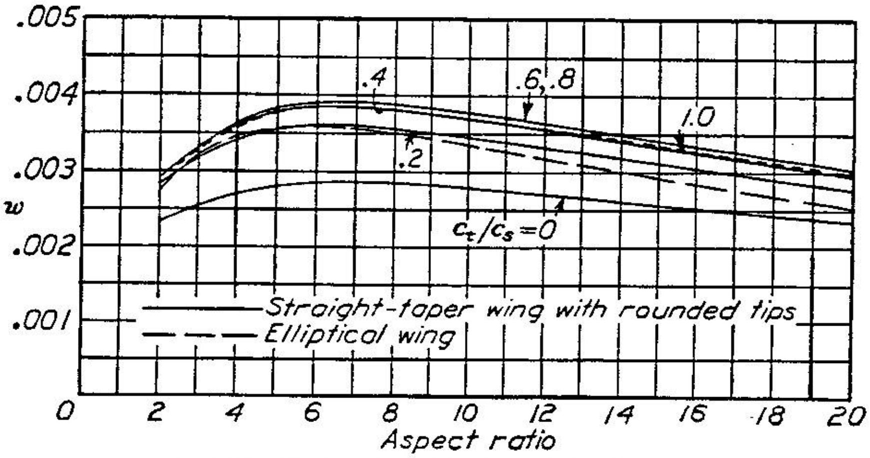


FIG. 12. Chart for determining induced-drag factor w.

W 20/a

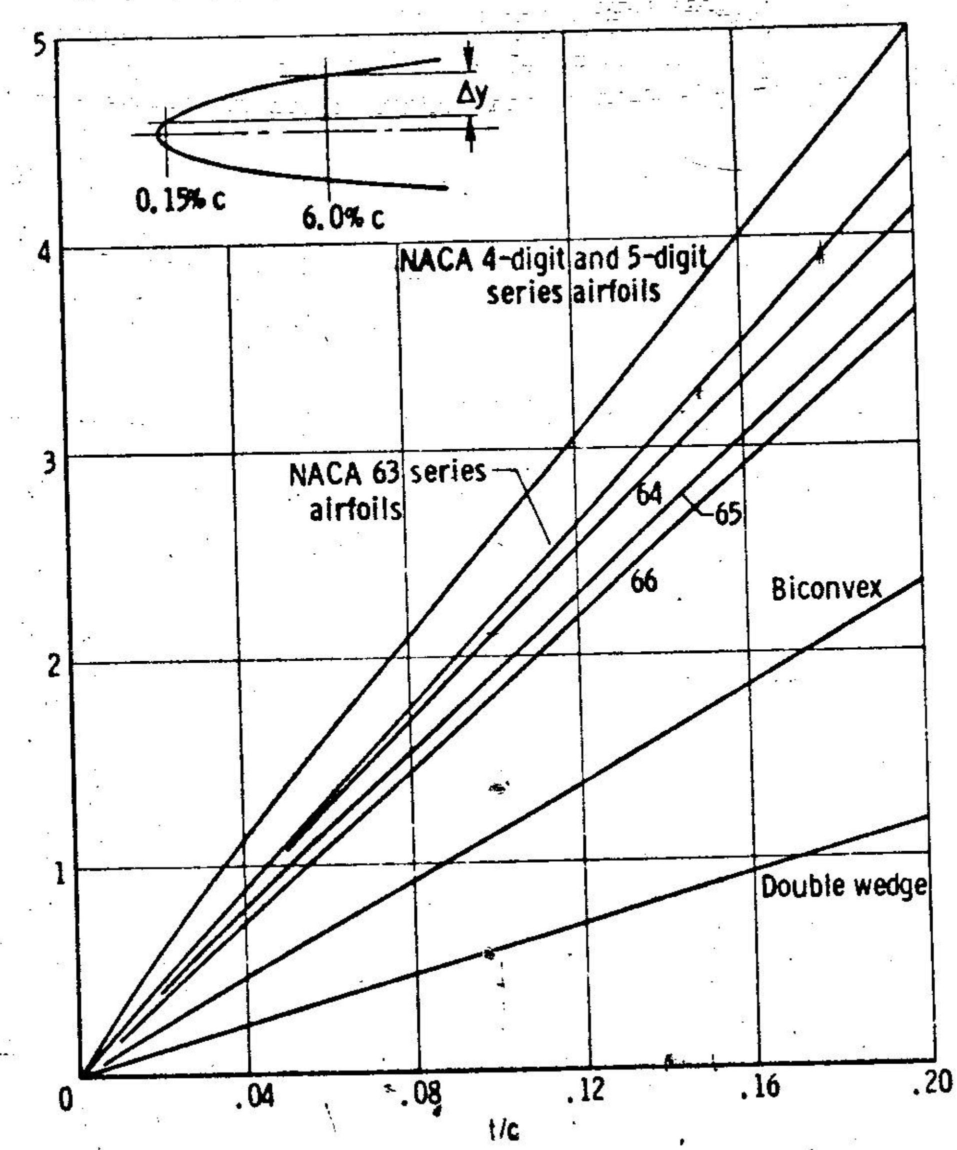


Figure 4. 1-2. Variation of leading-edge sharpness parameter with airfoil thickness ratio (ref. 1).

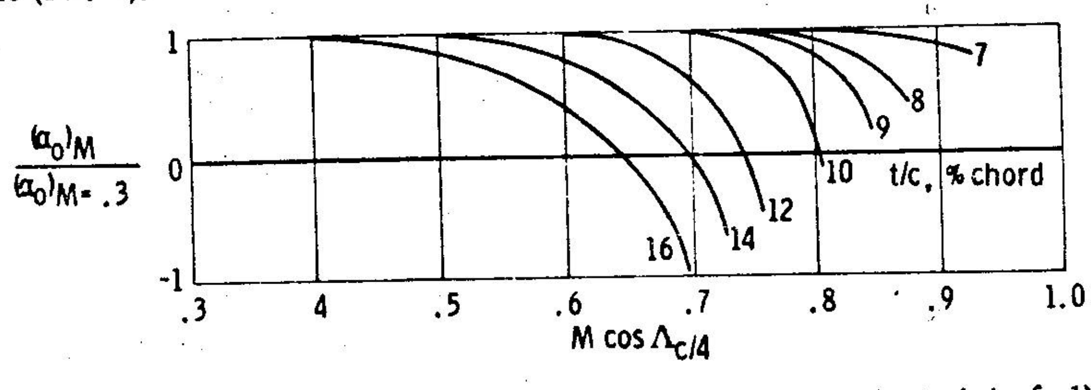


Figure 4. 1-3. Mach number correction for zero-lift angle of attack (ref. 1).

W20/b

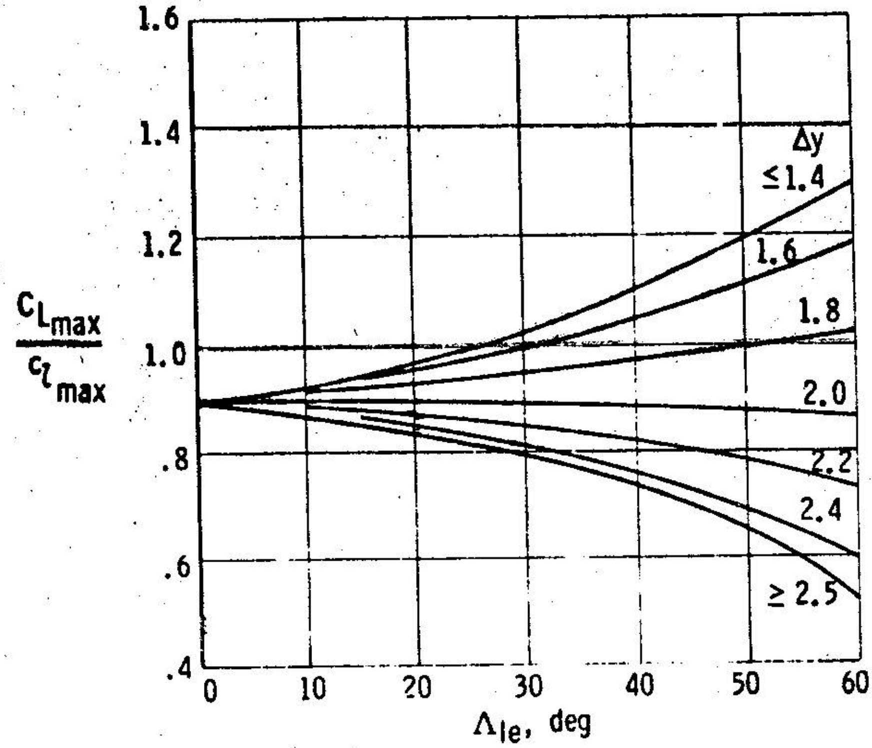


Figure 4.2-4. Subsonic maximum lift of high-aspect ratio, untwisted, constant airfoil section wings (ref. 1). $M \approx 0.2$.

3.3

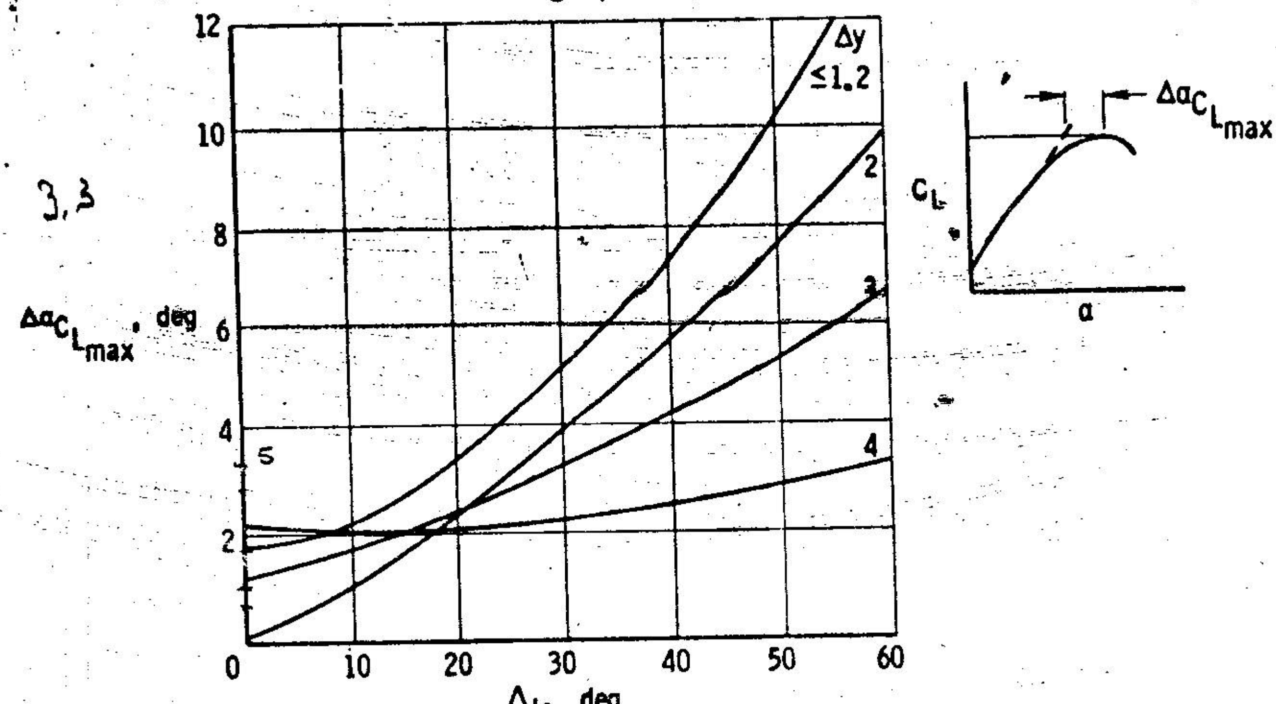


Figure 4.2-5. Angle-of-attack increment for subsonic maximum lift of high-aspect-ratio, untwisted, constant airfoil section wings (ref. 1). $M \approx 0.2$ to 0.6.

Lateral

$$(Y_v)' = Y_v$$
$$(Y_p)' = Y_p \cos \varepsilon - Y_r \sin \varepsilon$$
$$(Y_r)' = Y_r \cos \varepsilon + Y_p \sin \varepsilon$$
$$(L_v)' = L_v \cos \varepsilon - N_v \sin \varepsilon$$
$$(L_p)' = L_p \cos^2 \varepsilon - (L_r + N_p) \sin \varepsilon \cos \varepsilon + N_r \sin^2 \varepsilon$$
$$(L_r)' = L_r \cos^2 \varepsilon - (N_r - L_p) \sin \varepsilon \cos \varepsilon - N_p \sin^2 \varepsilon$$
$$(N_v)' = N_v \cos \varepsilon + L_v \sin \varepsilon$$
$$(N_p)' = N_p \cos^2 \varepsilon - (N_r - L_p) \sin \varepsilon \cos \varepsilon - L_r \sin^2 \varepsilon$$
$$(N_r)' = N_r \cos^2 \varepsilon + (L_r + N_p) \sin \varepsilon \cos \varepsilon + L_p \sin^2 \varepsilon$$

MEAN AERODYNAMIC CHORD, MEAN AERODYNAMIC CENTER, AND C_{m0_w}

G. K. DIMOCK

APPENDIX C

C.1 BASIC DEFINITIONS

In the normal flight range, the resultant aerodynamic forces acting on any lifting surface can be represented as a lift and drag acting at the mean aerodynamic center $(\bar{x}, \bar{y}, \bar{z})$, together with a pitching couple C_{mo_w} which is independent of angle of attack (see Fig. 2.6).

The pitching moment of a wing is nondimensionalized by the use of the mean aerodynamic chord $\bar{c}$.

Both the m.a. center and the m.a. chord lie in the plane of symmetry of the wing. However, in determining them it is convenient to work with the half-wing.

These quantities are defined by (Fig. C.1)

$$\bar{c} = \frac{2}{S} \int_0^{b/2} c^2 \, dy \tag{C.1,1}$$

$$\bar{x} = \frac{2}{C_L S} \int_0^{b/2} C_{l_a} c x \, dy \tag{C.1,2}$$

$$\bar{y} = \frac{2}{C_L S} \int_0^{b/2} C_{l_a} c y \, dy = \eta_{cp} \frac{b}{2} \tag{C.1,3}$$

$$\bar{z} = \frac{2}{C_L S} \int_0^{b/2} C_{l_a} c z \, dy \tag{C.1,4}$$

where b = wing span
c = local chord
C_L = total lift coefficient
C_{l_a} = local additional lift coefficient

W20/C

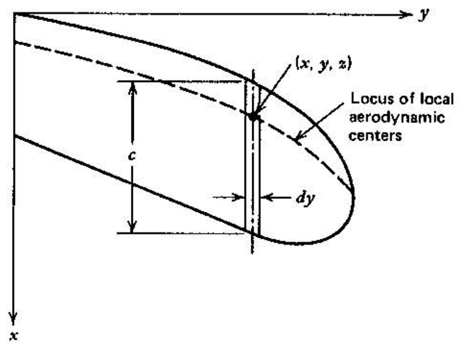


FIG. C.1 Local aerodynamic center coordinates.

S = wing area

y = spanwise coordinate of local aerodynamic center measured from axis of symmetry

x = chordwise coordinate of local aerodynamic center measured aft of wing apex

z = vertical coordinate of local aerodynamic center measured from xy plane

η_{cp} = lateral position of the center of pressure of the additional load on the half-wing as a fraction of the semispan

The coordinates of the m.a. center depend on the additional load distribution; hence the position of the true m.a. center will vary with wing incidence if the form of the additional loading varies with incidence (as it may with swept wings). For a wing that has no aerodynamic twist, the m.a. center of the half-wing is also the center of pressure of the half-wing. If there is a basic loading (i.e., at zero overall lift, due to wing twist), then $(\bar{x}, \bar{y}, \bar{z})$ is the center of pressure of the additional loading.

The height and spanwise position of the local aerodynamic centers may be assumed known, and hence $\bar{y}$ and $\bar{z}$ for the half-wing can be calculated once the additional spanwise loading distribution is known. However, in order to calculate $\bar{x}$, the fore-and-aft position of each local aerodynamic center must be known first. If all the local aerodynamic centers are assumed to lie on the nth-chord line (assumed to be straight), then

$$\bar{x} = nc_r + \bar{y} \tan \Lambda_n \qquad \text{(C.1,5)}$$

where c_r = wing root chord

Λ_n = sweepback of nth-chord line, degrees

Ideal two-dimensional flow theory gives $n = \frac{1}{4}$ for subsonic speeds and $n = \frac{1}{2}$ for supersonic speeds.

The m.a. chord is located relative to the wing by the following procedure:

1. In Eq. C.1,2 replace C_{l_α} by C_L, and for x use the coordinates of the $\frac{1}{4}$-chord line.
2. The value of $\bar{x}$ so obtained (the mean quarter-chord point) is the $\frac{1}{4}$-point of the m.a. chord.

The above procedure and the definition of $\bar{c}$ (Eq. C.1,1) are used for *all* wings.

C.2 COMPARISON OF M.A. CHORD AND M.A. CENTER FOR BASIC PLANFORMS AND LOADING DISTRIBUTIONS

In Table C.1 taken from ref. C.1, values of m.a. chord and $\bar{y}$ are given for some basic planforms and loading distributions.

In the general case the additional loading distribution and the spanwise center-of-pressure position can be obtained by methods such as those of ref. C.2,3 and 4. For a trapezoidal wing with the local aerodynamic centers on the nth-cord line, the chordwise location of the aerodynamic center from the leading edge of the m.a. chord expressed as a fraction of the m.a. chord h_{n_w} is given by

$$h_{n_w} = n + \frac{3(1+\lambda)^2}{8(1+\lambda+\lambda^2)}\left[\eta_{cp} - \frac{1+2\lambda}{3(1+\lambda)}\right] A \tan \Lambda_n \qquad \text{(C.2,1)}$$

TABLE C.1

Planform	*Additional Loading Distribution*	*M.A.C.,* $\bar{c}$	$\bar{y}$
Constant taper and sweep (trapezoidal)	Any	$\frac{2c_r}{3}\frac{1+\lambda+\lambda^2}{1+\lambda}$	$\eta_{cp}\cdot\frac{b}{2}$
Constant taper and sweep (trapezoidal)	Proportional to wing chord (uniform C_{l_α})	$\frac{2c_r}{3}\frac{1+\lambda+\lambda^2}{1+\lambda}$	$\frac{b}{2}\cdot\frac{1+2\lambda}{3(1+\lambda)}$
Constant taper and sweep (trapezoidal)	Elliptic	$\frac{2c_r}{3}\frac{1+\lambda+\lambda^2}{1+\lambda}$	$\frac{b}{2}\cdot\frac{4}{3\pi}$
Elliptic (with straight sweep of line of local a.c.)	Any	$\frac{c_r}{3}\cdot\frac{8}{\pi}$	$\eta_{cp}\cdot\frac{b}{2}$
Elliptic (with straight sweep of line of local a.c.)	Elliptic (uniform C_{l_α})	$\frac{c_r}{3}\cdot\frac{8}{\pi}$	$\frac{b}{2}\cdot\frac{4}{3\pi}$
Any (with straight sweep of line of local a.c.)	Elliptic	$\frac{2}{S}\int_0^{b/2} c^2\,dy$	$\frac{b}{2}\cdot\frac{4}{3\pi}$

W21

where A = aspect ratio, b^2/S

λ = taper ratio, c_t/c_r

c_t = wing-tip chord

The length of the chord through the centroid of area of a trapezoidal half-wing is equal to $\bar{c}$. For the same wing with uniform spanwise lift distribution (i.e., C_{l_a} = const) and local aerodynamic centers on the nth-chord line, the m.a. center also lies on the chord through the centroid of area. The chord through the centroid of area of a wing having an elliptic planform is not the same as $\bar{c}$, but the m.a. center for elliptic loading and the centroid of area both lie on the same chord (see ref. C.1).

C.3 M.A. CHORD AND M.A. CENTER FOR SWEPT AND TAPERED WINGS

The ratio $\bar{c}/c_r$ is plotted against λ in Fig. C.2 for straight tapered wings with streamwise tips. The spanwise position of the m.a. center of the half-wing (or the center of pressure of the additional load) for uniform spanwise loading is also given in Fig. C.2. These functions are given in Table C.1.

The m.a. chord is located by means of the distance x of the leading edge of the m.a. chord aft of the wing apex, given in Fig. C.3:

$$x = \frac{b}{2}\cdot\frac{1}{3}\frac{1+2\lambda}{1+\lambda}\tan\Lambda_0$$

$$= \frac{1+2\lambda}{12}c_r A\tan\Lambda_0 \qquad \text{(C.3,1)}$$

where Λ_0 = sweepback of wing leading edge, degrees.

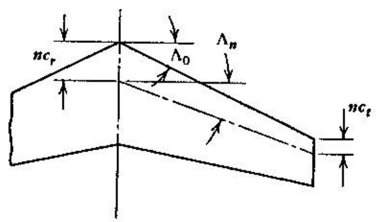
The sweepback of the leading edge is related to the sweep of the nth-chord line Λ_n by the relation

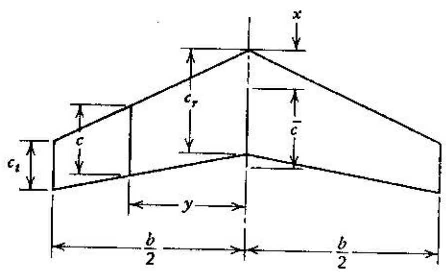
$$A\tan\Lambda_0 = A\tan\Lambda_n + 4n\frac{1-\lambda}{1+\lambda} \qquad \text{(C.3,2)}$$

Using Eq. C.3,2 and the expression for $\bar{c}/c_r$, x can be obtained in terms of $\bar{c}$ and Λ_n from

$$\frac{x}{\bar{c}} = \frac{(1+2\lambda)(1+\lambda)}{8(1+\lambda+\lambda^2)}\left[A\tan\Lambda_n + 4n\frac{1-\lambda}{1+\lambda}\right] \qquad \text{(C.3,3)}$$

The fractional distance of the m.a. center aft of the leading edge of the m.a. chord, h_{n_w}, is given for swept and tapered wings at low speeds and small incidences in Fig. C.4. The dotted lines show the aerodynamic-center position for wings with unswept trailing edges. The curves have been obtained from theoretical and experimental data (ref. C.5). The curves apply only within the linear range of the curve of wing lift against pitching moment, provided that the flow is subsonic over the entire wing. The probable error of h_{n_w} given by the curves is within 3%.





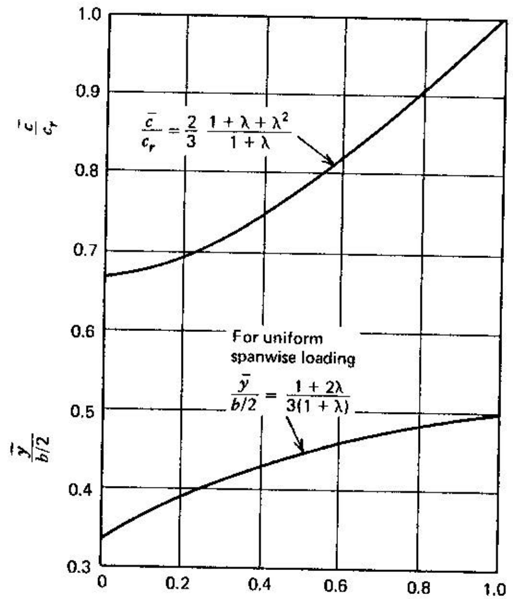


FIG. C.2 Mean aerodynamic chord for straight tapered wings; and spanwise position of mean aerodynamic center for uniform spanwise loading (i.e., constant C_{L_a}).
(Reproduced from "Notes on the Mean Aerodynamic Chord and the Mean Aerodynamic Center of a Wing" by A. H. Yates, *J. Roy. Aero. Soc.*, June 1952.)

W22

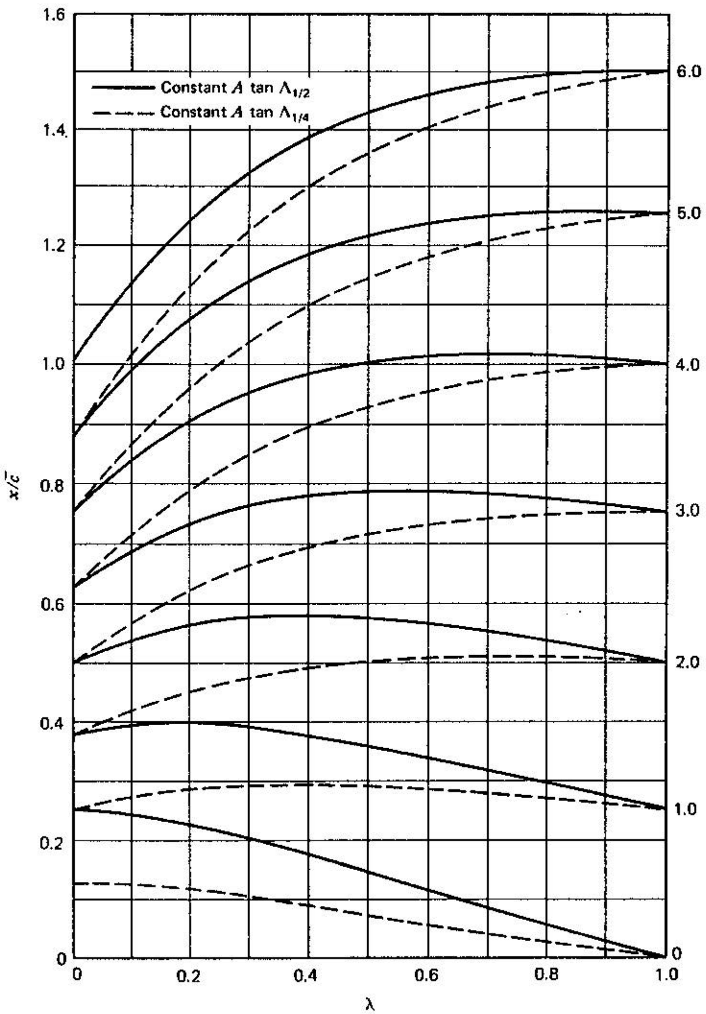


FIG. C.3 Distance of the leading edge of the mean aerodynamic chord aft of the wing apex for swept and tapered wings.
(Reproduced from Royal Aeronautical Society Data Sheet Wings 00. 01. 01.)

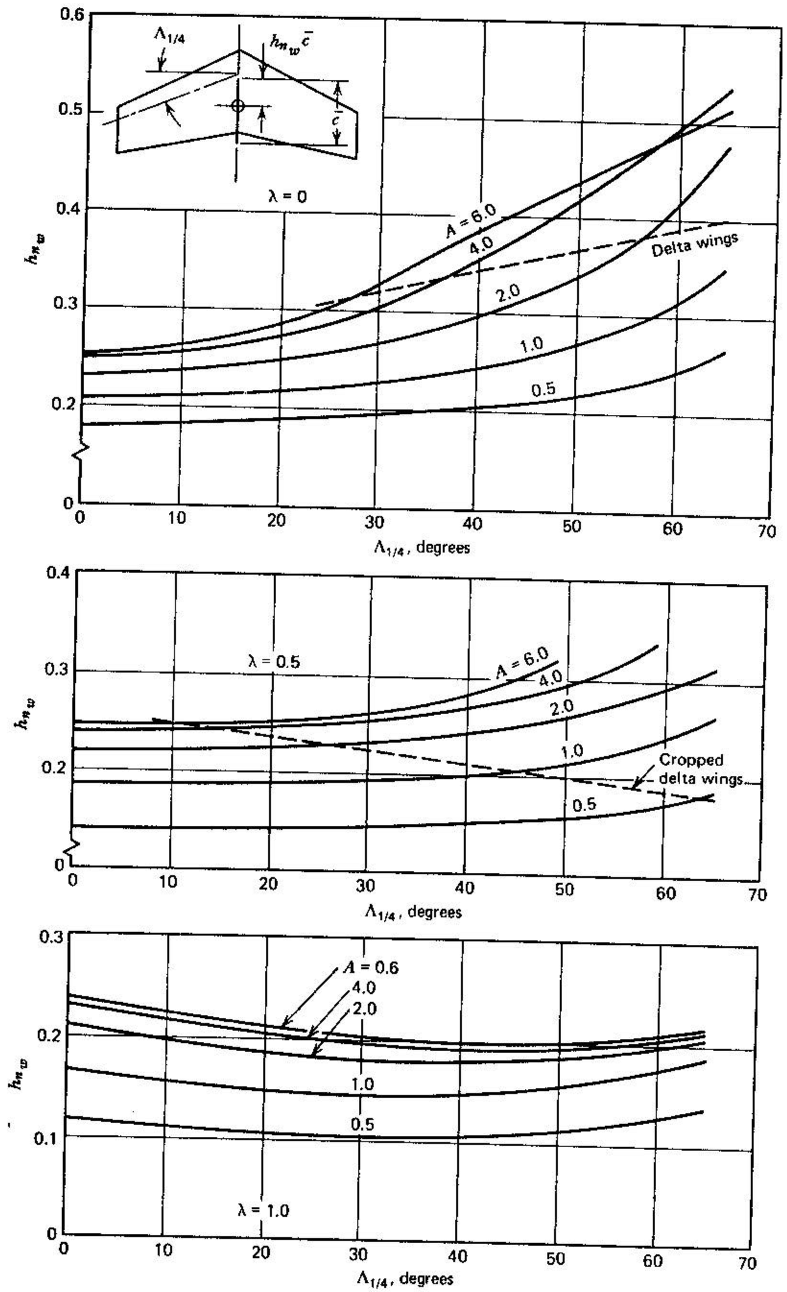


FIG. C.4 Chordwise position of the mean aerodynamic center of swept and tapered wings at low speeds expressed as a fraction of the mean aerodynamic chord.
(Reproduced from Royal Aeronautical Data Sheet Wings 08. 01. 01.)

W23

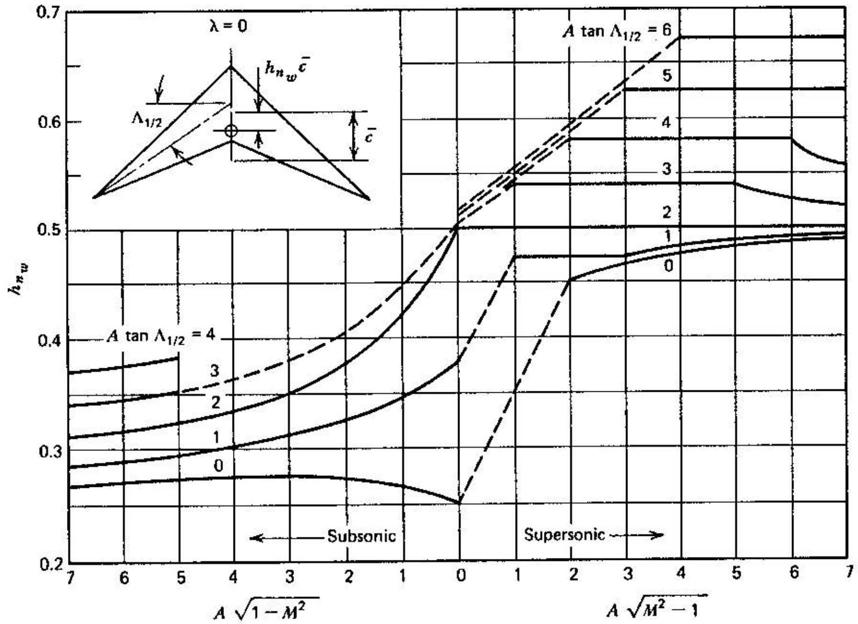


FIG. C.5 Chordwise position of the mean aerodynamic center of swept and tapered wings at high speeds expressed as a fraction of the mean aerodynamic chord. (Reproduced from Royal Aeronautical Society Data Sheet Wings S. 08. 01. 02.)

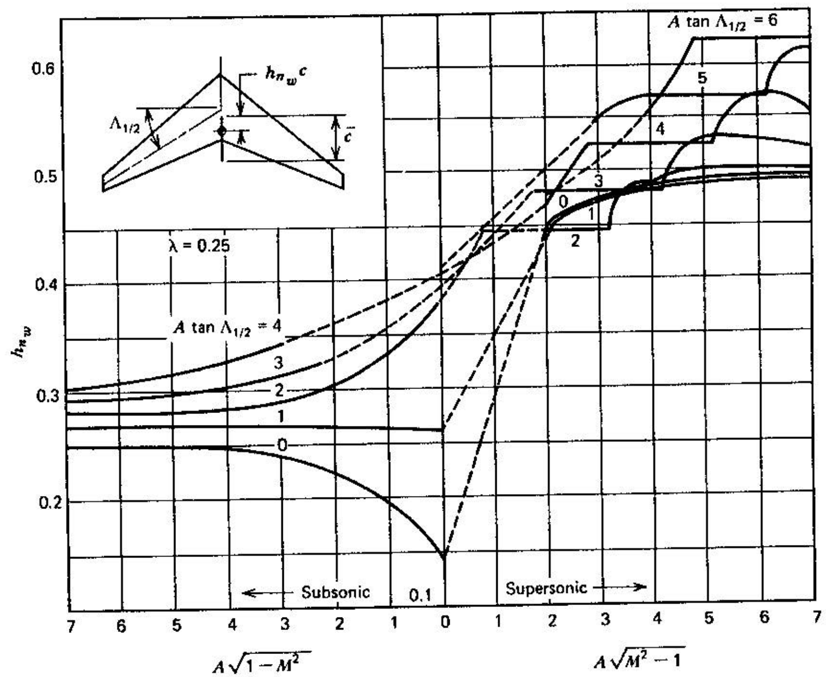


FIG. C.5 (continued)

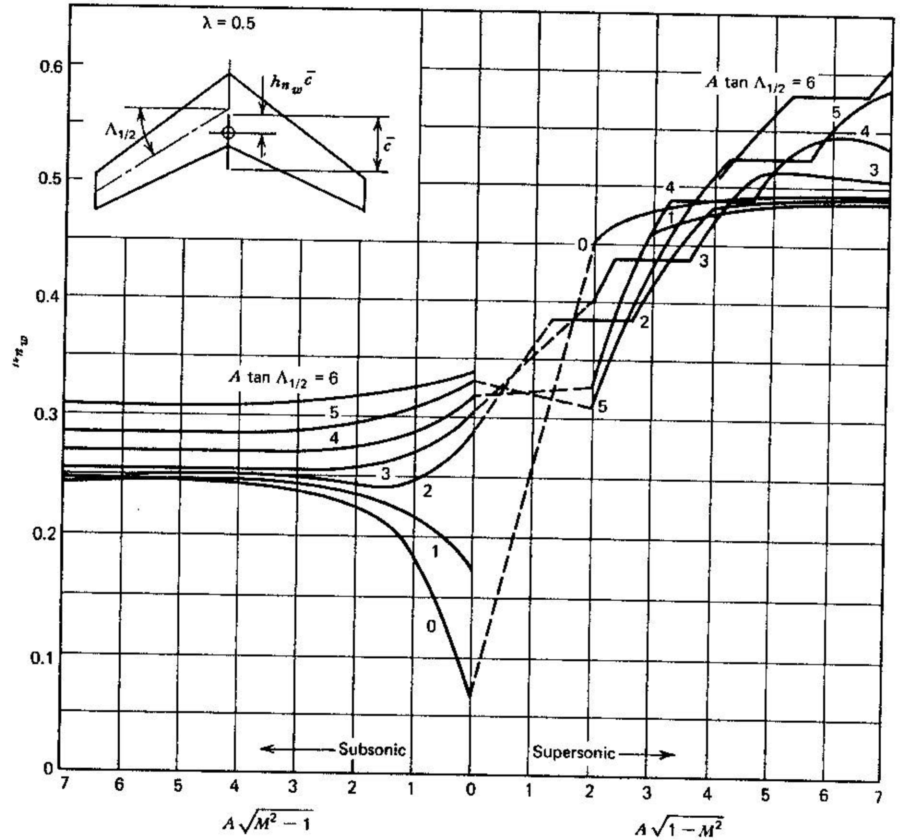


FIG. C.5 (continued)

For higher Mach numbers and in particular for supersonic speeds, h_{n_w} may be obtained from the theoretical data given in Fig. C.5. The curves for subsonic flow were obtained by applying the similarity laws of linearized perturbation theory to Weissinger's method, whereas those for supersonic flow were based on linearized supersonic lifting-surface theory. Slender-wing theory was used to give h_{n_w} for $A\sqrt{M^2-1} = 0$. In regions where the theoretical solutions were inadequate, the curves are shown dotted.

The theories are based on inviscid flow and apply to flat plates at small incidence. For small values of $A\sqrt{M^2-1}$ their practical validity is doubtful. The kinks shown in the curves are not likely to occur in practice. The curves are expected to be of acceptable accuracy where the variation of aerodynamic-center position with Mach number is smooth (ref. C.6).

W24

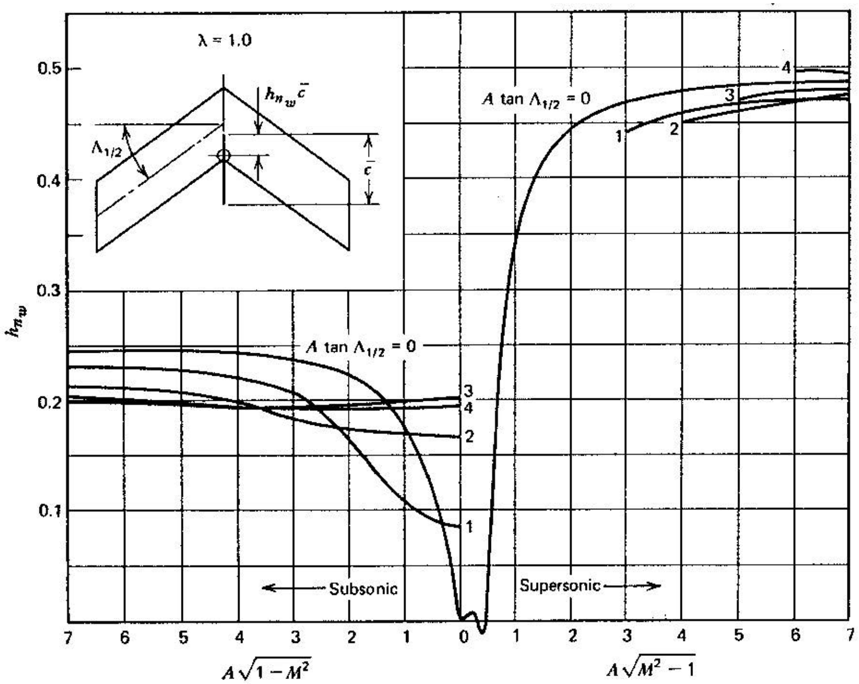


FIG. C.5 (concluded)

C.4 $C_{m_{0_w}}$

The total load on each section of a wing has three parts as illustrated by Fig. C.6*a*. The resultant of the local additional lift l_a, is the lift L_a acting through the m.a. center (Fig. C.6*b*).

The resultant of the distribution of the local basic lift l_b is a pitching couple whenever the line of aerodynamic centers is not straight and perpendicular to x. This couple is given by

$$M_1 = 2\int_0^{b/2} (x - \bar{x}) l_b \, dy = 2\int_0^{b/2} x l_b \, dy$$

since the resultant of $l_b = 0$.

Then

$$C_{m_1} = \frac{2}{qS\bar{c}}\int_0^{b/2} x C_{l_b} qc \, dy$$

$$= \frac{2}{S\bar{c}}\int_0^{b/2} C_{l_b} xc \, dy \tag{C.4,1}$$

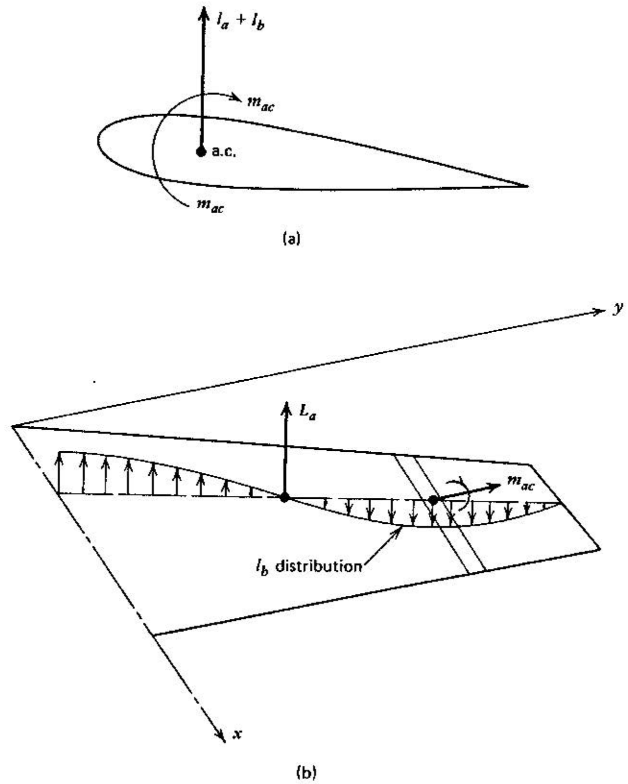


FIG. C.6 (*a*) Section total load. (*b*) Wing loads.

The resultant of the m_{ac} distribution is given by

$$C_{m_2} = \frac{2}{S\bar{c}}\int_0^{b/2} C_{m_{ac}} c^2 \, dy \tag{C.4,2}$$

The total pitching-moment coefficient about the m.a. center is then

$$C_{m_{0_w}} = C_{m_1} + C_{m_2} = \text{const} \tag{C.4,3}$$

If $C_{m_{ac}}$ is constant across the span, and equals C_{m_2}, then Eq. C.4,2 also becomes the defining equation for $\bar{c}$.

W25