

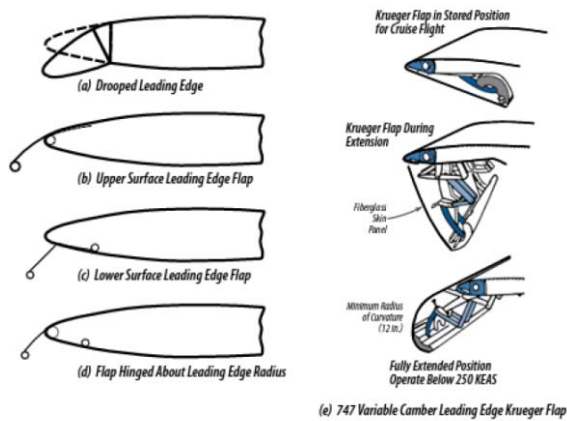


Figure 9.5 Various LE flap devices.

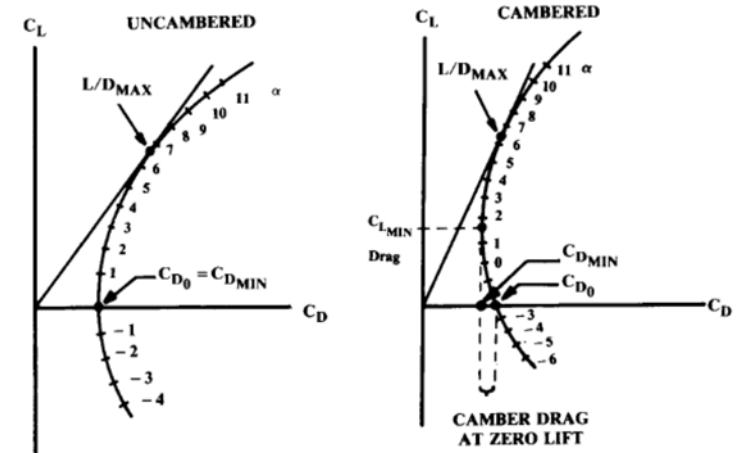


Fig. 12.3 Drag polar.

# Aerodinámica Preliminar

## Tema 5 $C_{L_\alpha} = \frac{2\pi A}{2 + \sqrt{4 + \frac{A^2 \beta^2}{\eta^2} \left(1 + \frac{\tan^2 \Lambda_{max,t}}{\beta^2}\right)}} \left(\frac{S_{exposed}}{S_{ref}}\right) (F)$

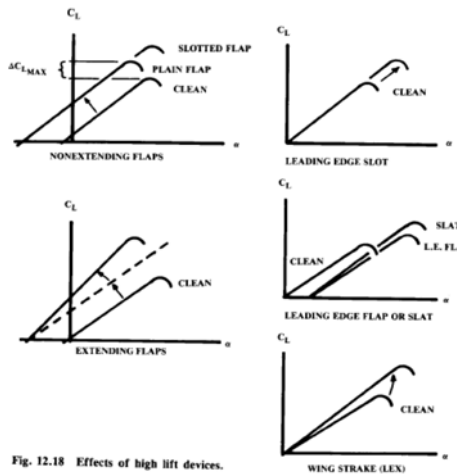


Fig. 12.18 Effects of high lift devices.

**Sergio Esteban Roncero**  
**Departamento de Ingeniería Aeroespacial**  
**Y Mecánica de Fluidos**

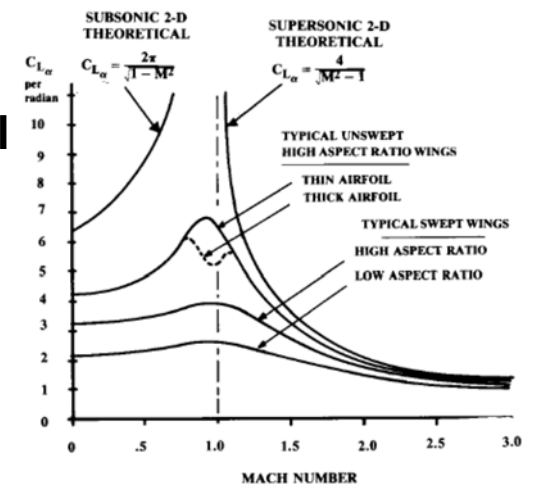
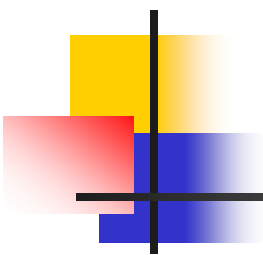


Fig. 12.5 Lift curve slope vs Mach number.



# Outline

---

- Introducción – ¿Dónde estamos?
- ¿Que hay que hacer?
- Polar del Avión.
- Fuerzas y Coeficientes Aerodinámicos.
- Cálculo de los coeficientes de la polar.
- Reducción de la resistencia aerodinámica.
- Bibliografía.

# Introducción – ¿Dónde estamos?

- Se han presentado metodologías para:
  - Definir del dimensionado inicial:
    - *Rubber engine sizing.*
    - *Fixed Engine sizing.*
  - Selección del  $T/W$  y la  $W/S$  en diferentes segmentos de vuelo generando un “boceto” conceptual del avión.
  - Dimensionado de derivas mediante estimación muy burda: *Tail volume coefficient.*
- Dicho “boceto” (creíble) se ha obtenido **asumiendo estimaciones burdas** de las características aerodinámicas, de pesos, y características propulsivas.
- Una vez que el boceto/diseño inicial se ha establecido se pueden estudiar las características del avión para ver si cumplen las especificaciones de la misión
- Para ello necesitamos unas pautas
  - Estudios comparativos para determinar la mejor combinación de  $T/W$  ( $P/W$ ) y  $W/S$  y alargamiento ( $AR$ ), para cumplir los requisitos de actuaciones de los diferentes segmentos con mínimo peso y coste.
  - Estudio de la estabilidad para mejorar los diseños iniciales de las derivas (horizontal y vertical).
- Necesitamos **herramientas más precisas** para determinar valores que hasta ahora nos hemos basado en estimaciones burdas para refinar el diseño:
  - Herramientas Aerodinámicas:
    - Estudio de la Polar del avión

## ¿Que hacer?

- **Selección preliminar** de los **perfiles** para las superficies sustentadoras.
  - Aviones semejantes
- Definir la precisión en los **modelos** de **polares** más exactos.
- Determinación inicial de las **características** iniciales **aerodinámicas**.
- Definición de parámetros adimensionales:
  - Optimización del ala:  $AR$ ,  $e$ ,  $\lambda$
- Interacción con diferentes áreas de Ingeniería:
  - Sustentación requerida: pesos (Estructura)
  - Generación de parámetros aerodinámicos (Estabilidad y Actuaciones/Propulsión)
  - Geometría: Diseño y sistemas

# Aerodinámica - I

## ■ Lo que se espera:

### ■ Polar parabólica de coeficientes constantes:

#### ■ $C_D$ del avión

##### ■ Configuración limpia

- Vuelo de subida

- Vuelos de crucero

##### ■ Configuración sucia

- Despegue y aterrizaje

- Superficies Hipersustentadoras, tren de aterrizaje...

### ■ Características aerodinámicas de los perfiles:

#### ■ Estimar $C_L$

- $C_{L0}$ ,  $C_{L\alpha}$ ,  $C_{M0}$ ,  $C_{M\alpha}$

- Ala, canard, deriva horizontal y vertical, cola en V...

#### ■ Corrección para alas finitas

## ■ Como conseguirlo...

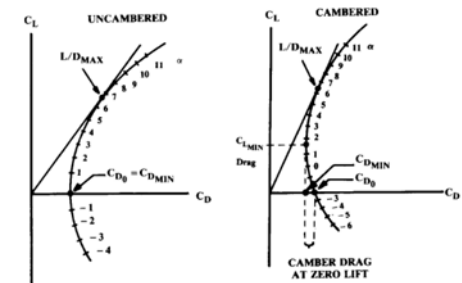


Fig. 12.3 Drag polar.

- Como conseguirlo:

## ■ Component Buildup Method



Cálculo de Aeronaves © 2014 Sergio Esteban Roncero, sesteban@us.es

$$(C_{D0})_{\text{subsonic}} = \frac{\Sigma(C_{fc} FF_c Q_c S_{\text{wet}c})}{S_{\text{ref}}} + C_{D\text{misc}} + C_{D\text{L\&P}}$$

# Aerodinámica - III

## ■ Características aerodinámicas de los perfiles:

### ■ Estimar $C_L$

#### ■ $C_{Lmax}$ :

- configuración limpia – métodos gráficos transparencias de clase: ejemplo clase
- configuración sucia: túnel de viento virtual: **ejemplo practico de clase**
  - **XFLR5 o similares**

#### ■ Métodos gráficos

### ■ $C_{L0}$ , $C_{L\alpha}$ , $C_{M0}$ , $C_{M\alpha}$

- Ala, canard, deriva horizontal y vertical, cola en V...

### ■ Corrección para alas finitas

$$C_{L\alpha} = \frac{2\pi A}{2 + \sqrt{4 + \frac{A^2 \beta^2}{\eta^2} \left(1 + \frac{\tan^2 \Lambda_{max,t}}{\beta^2}\right)}} \left( \frac{S_{exposed}}{S_{ref}} \right) (F)$$

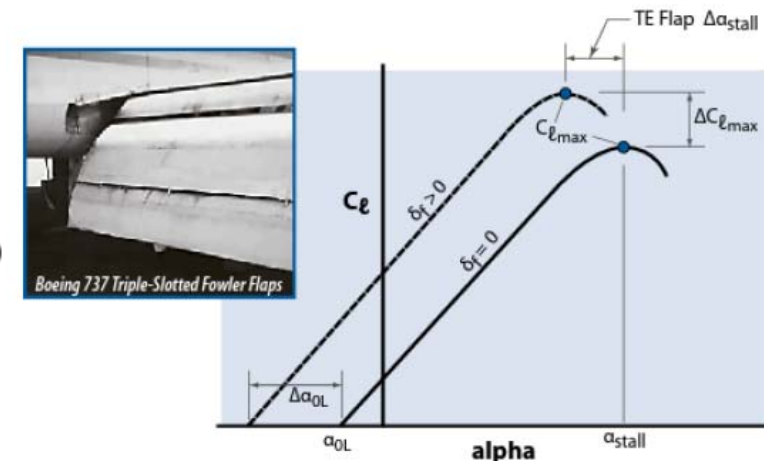


Figure 9.8 Construction of section lift curves for TE flaps.

# Aerodinámica – Estudio Avanzado

- 1º - Conversión de 2D a 3D
  - 1 - Calcular las características del ala en 2D
  - 2 - Calcular la corrección de 2D a 3D: del ala
  - 3 - Calcular como se ven afectados por HLD (High Lift Devices)
- 2º - Calculo de la polar
  - 1 – Calcular la polar simplificada
    - Configuración limpia
    - Configuración sucia
  - 2 – Calcular la polar corregida:
- 3º - Evaluar requisitos asociados a las áreas de trabajo
  - Actuaciones
  - Estructuras
  - Estabilidad

# Aerodinámica – II

- Estudio Aerodinámico más detallado:
  - Selección depurada de los perfiles para superficies sustentadoras.
    - Requisitos estabilidad.
    - Requisitos actuaciones.
  - Estudio comparativo de XFLR5 (usar todo el potencial)
    - Comparación 3 perfiles (2D)
    - Comparación 3 planta alares (3D)
      - Posible comparación de diferentes plantas con diferentes perfiles
    - Elección de la configuración elegida en función de parámetros
      - Eficiencia (E)
      - Coeficiente de Oswald (e)
      - Resistencia ...
    - Comparativa configuración de cola y perfiles (dpto. estabilidad)
    - Cálculo de sustentación máxima
      - Métodos mixtos, XFLR5 + métodos clásicos
    - Estudio del avión por partes
      - Superficies aerodinámicas
    - Estudio del avión al completo
      - Superficies aerodinámicas + fuselaje+...

# Aerodinámica – III

- Estudio Aerodinámico más detallado:
  - Estudio de la polar del avión para las diferentes configuraciones:
    - Depuración de polar (dept de Diseño y sistemas)
      - Configuración limpia:
        - Subida,
        - Crucero,
        - Descenso
      - Configuración sucia:
        - despegue,
        - aterrizaje
    - Modelo de Polar
      - Polar parabólica **No compensada**
      - Polar parabólica **Compensada**

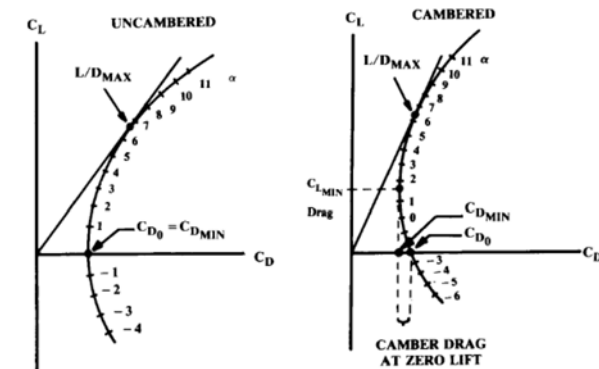


Fig. 12.3 Drag polar.

$$C_D = C_{D_0} + KC_L^2$$

$$\begin{aligned}
 C_D &= C_{D_{min}} + K(C_L - C_{L_{min-drag}})^2 \\
 &= C_{D_{min}} + KC_{L_{min-drag}}^2 + KC_L^2 - 2KC_L C_{L_{min-drag}} \\
 &= C_{D_0} + k_1 C_L^2 - k_2 C_L
 \end{aligned}$$

$$\begin{aligned}
 C_{D_0} &= C_{D_{min}} + KC_{L_{min-drag}}^2 \\
 k_1 &= K \\
 k_2 &= 2KC_L C_{L_{min-drag}}
 \end{aligned}$$

# Aerodinámica – IV

- Estudio Aerodinámico más detallado:
  - Polar parabólica **No compensada**
    - Análisis de las actuaciones
      - Despegue y aterrizaje, subida, crucero, espera, descenso
    - Definir núm Reynolds de análisis (simplificado)
      - No es factible definir polar para todo el rango de velocidades
  - Polar parabólica **Compensada**
    - Mejora actuaciones Crucero.
    - Análisis de Empuje (Potencia) necesaria vs. disponible
  - Evaluar requisitos asociados a las áreas de trabajo
    - Actuaciones:
      - Despegue, crucero:  $C_{L,max}$
    - Estructuras:
      - Estimado un nuevo  $W_0 \rightarrow W_0/S \rightarrow$  cálculo de nueva  $S_w$
    - Estabilidad:
      - Trimado (aumento o disminución de  $S_w$ )
      - Modificación geometría alar o perfil

# Aerodinámica – V

- Estudio Aerodinámico más detallado:
  - Estudio de resistencia de trimado
    - Incluir  $C_{dtrimado}$ 
      - Coordinar con departamento de Estabilidad y Actuaciones
  - Métodos Clásicos
    - Composite Build-Up Methods
  - Métodos modernos
    - XFLR5:
      - XFLR5 no proporciona resistencia parasitaria de cuerpos sin perfil
  - Análisis Mixto
    - Análisis Mixto Composite Build-UP + XFLR5
  - Estimación de resistencia parasitaria
    - Estudio comparativo de diferentes resistencias para diferentes configuraciones de fuselaje
    - Comparación con métodos clásicos

# Material Docente - Aerodinámica

Blackboard Learn

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Apps Compré Balanzas H... ~ZbigKlep A123 System High P... pearltrees • Free ebo... Jornadas de i+D+i e... 100 Life Hacks That ... El salto de Félix Bau... Público - OneDrive 16 Research - LaTeX po...

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**ENSEÑANZA VIRTUAL**

Mi institución Cursos

Cálculo de Aeronaves-Grado en Ingeniería Aeroespacial Contenido ... Temas de Soporte **Aerodinámica** El modo de edición está: **ACTIVADO**

**Aerodinámica**

Desarrollar contenido Evaluaciones Herramientas

**Aerodinámica**

Activado: Seguimiento de estadísticas

Archivos adjuntos:

- Tema\_05.2\_Extra\_Alas\_Regime\_Incompresible.pdf (8,508 MB)
- Tema\_05.4\_Estimacion.del.CLmax.pdf (968,657 KB)
- Tema\_05.1\_Extra\_Introducción\_Perfiles\_NACA.pdf (665,481 KB)
- Tema\_05.5\_Extra\_Oswald\_Efficiency.pdf (75,818 KB)
- Tema\_05.3\_Extra\_Calculo\_CLalpha.pdf (926,913 KB)
- Tema\_05.6\_Extra\_Airfoil\_Selection\_CLI\_vs\_CLmax.pdf (481,335 KB)

Temas de soporte para el área de aerodinámica.  
2ª Revisión:

- Tema 05.1 Extra Introducción Perfiles NACA
- Tema 05.2 Extra Alas Regime Incompresible
- Tema 05.3 Extra Calculo CLalpha
- Tema 05.4 Extra Estimacion del CLmax
- Tema 05.5 Extra Oswald Efficiency
- Tema 05.6 Extra Airfoil Selection CLI vs CLmax.pdf

**Pautas Diseño de Alas**

Activado: Seguimiento de estadísticas

Archivos adjuntos:

- Chapter 5. Wing Design.pdf (3,462 MB)

Cápítulo del libro Aircraft Design de M. Sadraey para el diseño de alas.

**Software Aerodinámico**

Activado: Seguimiento de estadísticas

Software Aerodinámico

**Reportes NACA Estimación CLmax**

Archivos adjuntos:

- naca-tn-1071.pdf (3,571 MB)
- naca-tn-422.pdf (617,78 KB)
- naca-wr-1-209.pdf (790,482 KB)
- naca-report-427.pdf (417,395 KB)
- naca-tn-459.pdf (478,133 KB)

Reportes NACA para la estimación de las características de sustentación máxima

**Panel de control**

- Archivos
- Herramientas del curso
- Evaluación
- Centro de calificación
- Usuarios y grupos
- Personalización
- Paquetes y utilidades

# Polar del avión - I

- La **polar del avión** (relación L/D) es **muy importante** para estimar **correctamente** las **actuaciones** y comprobar que se cumplen las especificaciones iniciales del **RFP**.
- La polar depende de la variación:
  - Número de **Mach**
  - Número de **Reynolds**
  - Configuración del avión: no hay una sola polar sino **varias** según el segmento en el que se encuentre el avión.
    - Conf. **Sucia**: despegue, aterrizaje
    - Conf. **limpia**: subida, crucero
  - La construcción de la polar se construye **contabilizando** las **distintas partes** por separado y sumándolas luego con **factores de corrección**.

$$C_D = C_D(C_L, M)$$

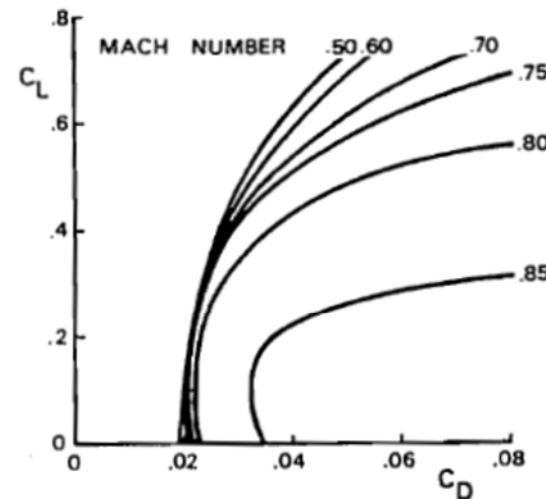


Fig. 5-6. Effect of compressibility on the drag polar

# Cálculo de los coeficientes de la polar

- La mayoría de los métodos se refieren a **geometrías** mas o menos **clásicas**:

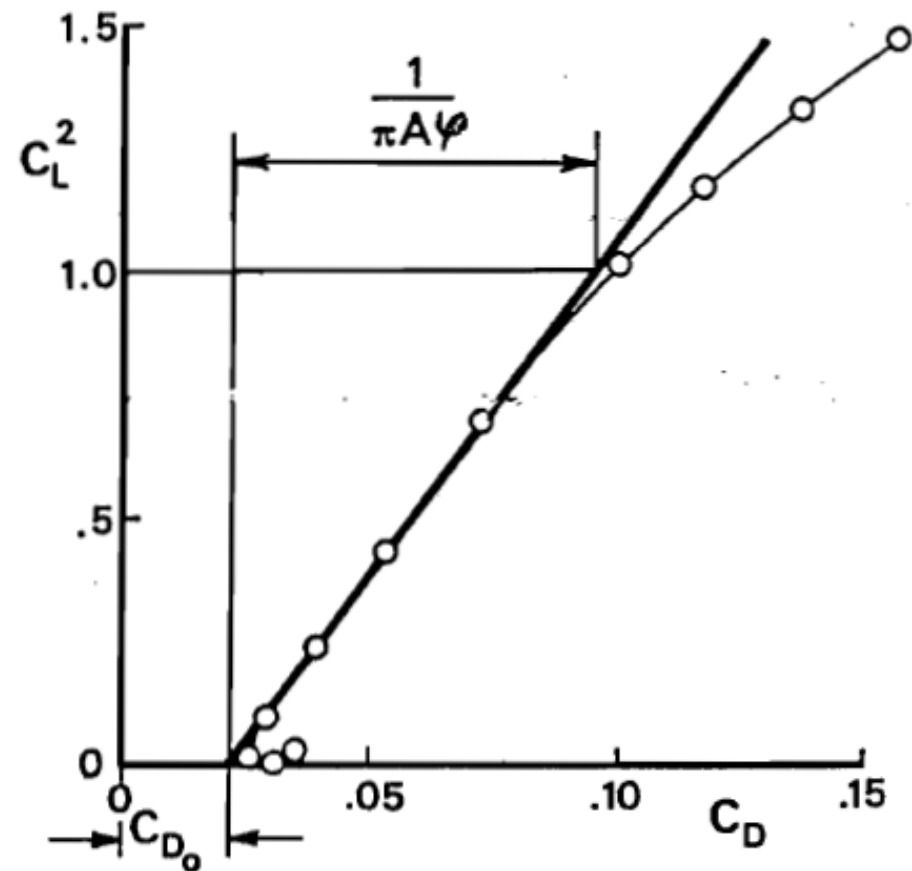
- Alas con flecha inferiores a  $40^\circ$
- Alargamientos mayores a 4.

- **Metodología rápida:**

- estimación gráfica del  $C_{D0}$  y de la eficiencia aerodinámica  $e$  (Oswald efficiency) para polares parabólicas simples:

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

- Para la estimación del parámetro de la eficiencia aerodinámica es necesario el considerar las **contribuciones** de la **estela de torbellinos** y de los perfiles a variación de la pendiente de la curva de empuje (*lift curve slope*) como veremos más adelante.
- También hay que considerar los efectos de **compresibilidad**, principalmente en el ala:
  - Variación de la resistencia de un perfil y del ala en función de  $M$  y  $C_L$ .



# Polar del avión - II

- Cada **segmento** de vuelo define unas características de **polar diferentes** en función de la relación  $L/D$  a las que se vuela.
- En crucero se suele aproximar con una polar parabólica de coeficientes constantes:

- Alas sin curvatura

- Mínima resistencia  $\alpha=0$

$$C_{D0} = C_{Dmin}$$

$$C_D = C_{D0} + KC_L^2 \Leftrightarrow K = \frac{1}{\pi Ae}$$

- Alas con curvatura

- Mínima resistencia  $\alpha>0$

$$C_{D0} \neq C_{Dmin}$$

corregido

$$\begin{aligned} C_D &= C_{Dmin} + K (C_L - C_{Lmin-drag})^2 \\ &= C_{Dmin} + KC_{Lmin-drag}^2 + KC_L^2 - 2KC_L C_{Lmin-drag} \\ &= C_{D0} + k_1 C_L^2 - k_2 C_L \end{aligned}$$

- En la mayoría de los textos se asume que la **sustentación** procede **únicamente** del **ala**, lo que se conoce como **polar no equilibrada**, ya que tanto las derivas como el fuselaje influyen en la sustentación

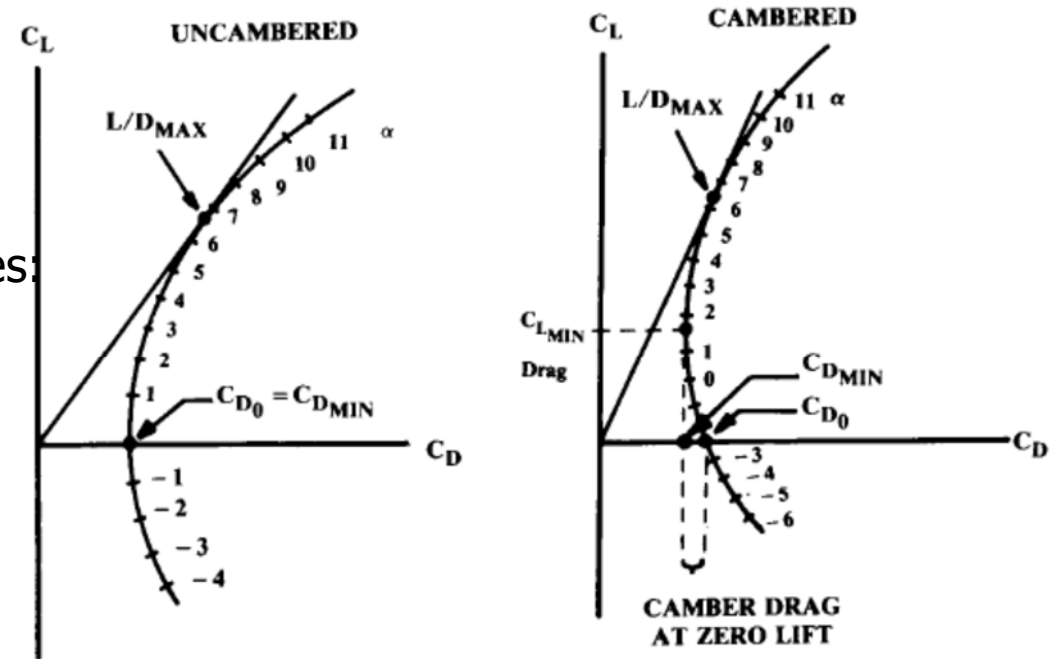
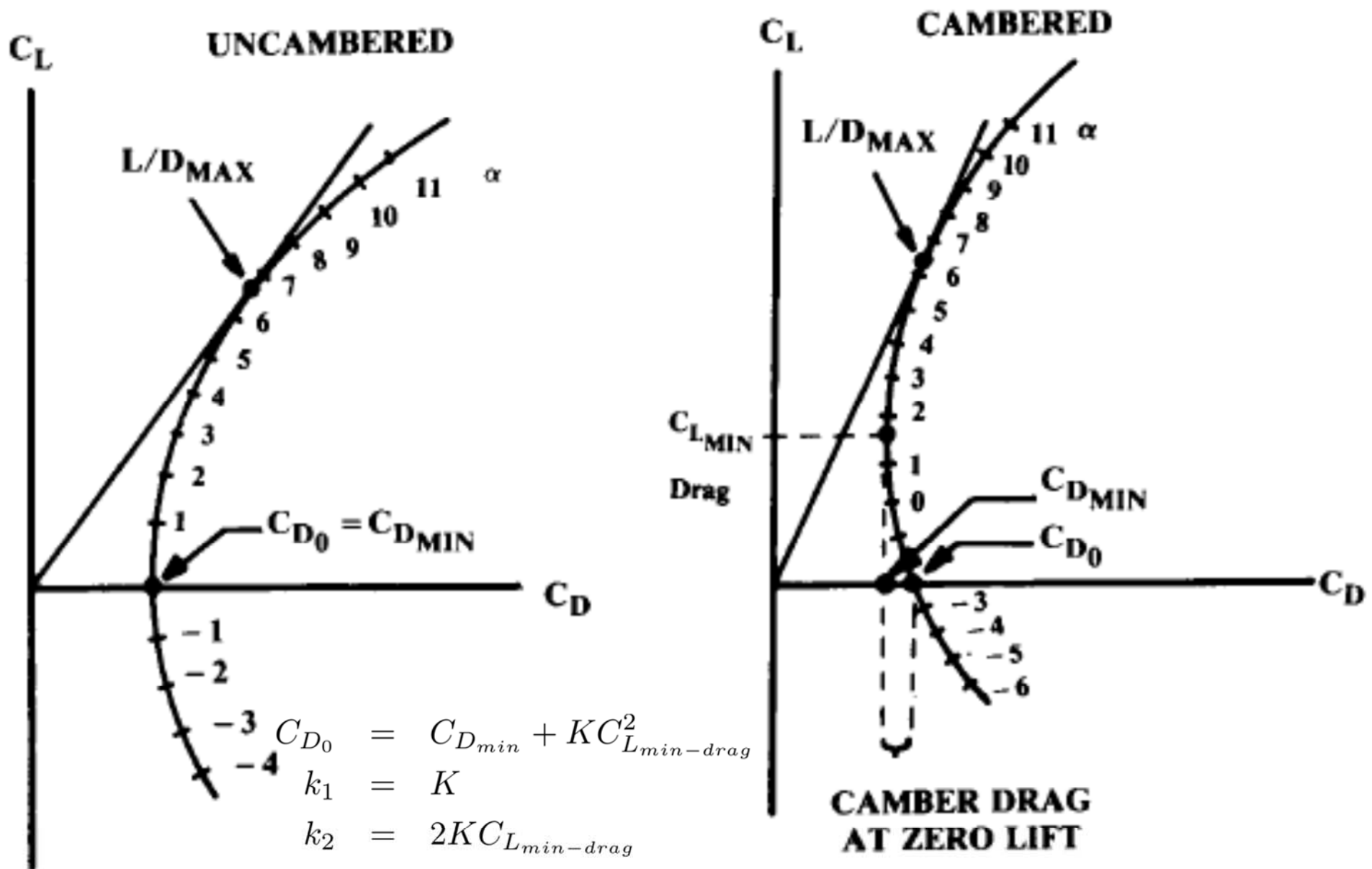


Fig. 12.3 Drag polar.

corregido

$$\begin{aligned} C_{D0} &= C_{Dmin} + KC_{Lmin-drag}^2 \\ k_1 &= K \\ k_2 &= 2KC_L C_{Lmin-drag} \end{aligned}$$



**Fig. 12.3 Drag polar.**

# Polar del avión - III

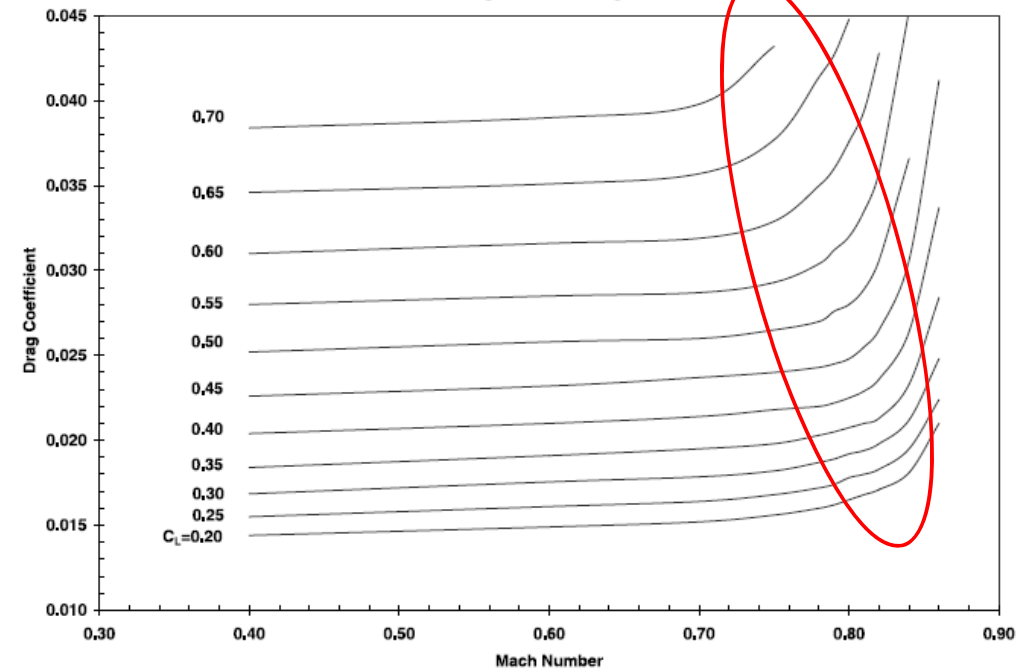
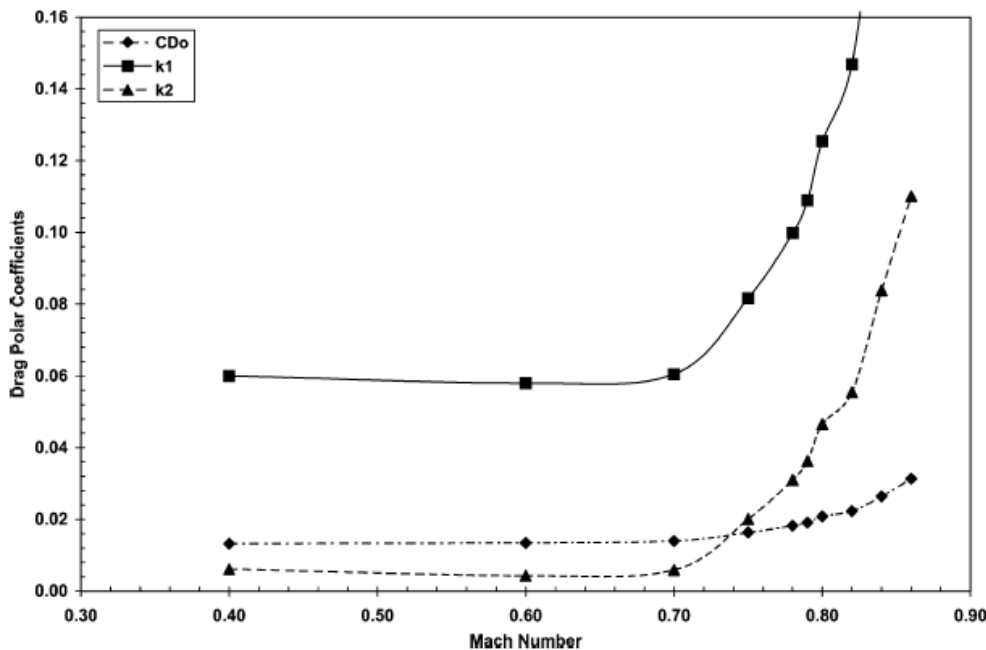
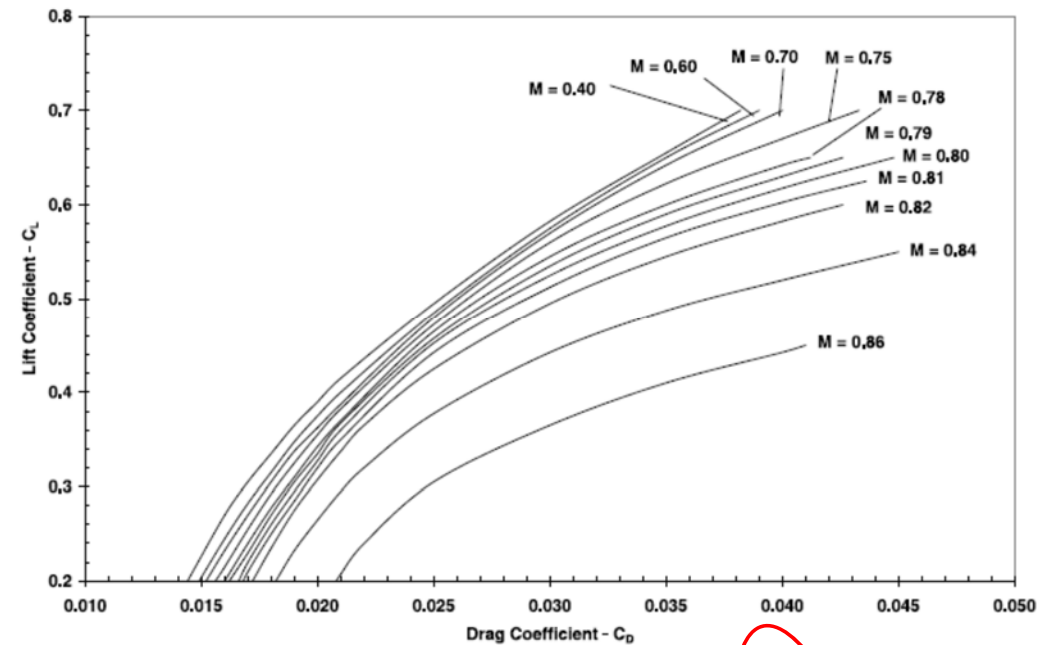
- Aproximación de los coeficientes en función del Mach de divergencia  $M_{DD}$

$$C_D = C_D(C_L, M)$$

$$C_D = C_{D_0} + k_1 C_L^2 - k_2 C_L$$

$$C_{D_0} = C_{D_0}(M), k_1 = k_1(M), k_2 = k_2(M)$$

Normalización con respecto a los valores propios



# Polar del avión - IV

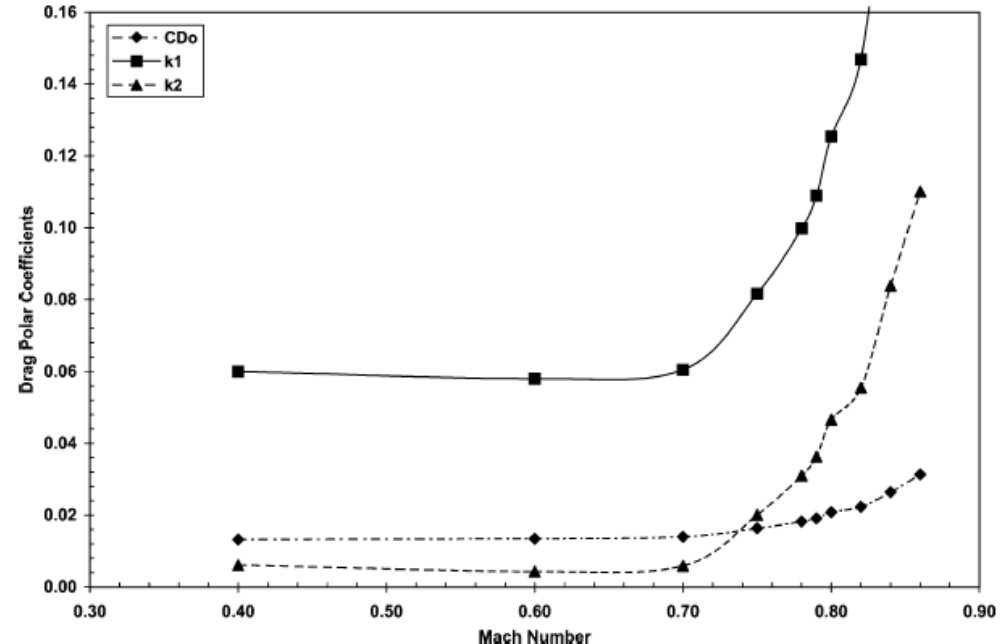
- Para aviones con perfiles de velocidad por debajo  $M < 0.7$  se observa que los coeficientes son prácticamente constantes.
- Para aviones con regímenes de vuelo  $M > 0.7$ , el aumento de valor de los coeficientes es exponencial:
  - Se harán estimaciones normalizando los valores presentes para aviones (Boeing 767)
    - Cavcar A. and Cavcar M., "Approximate solutions of range for constant altitude – constant high subsonic speed flight of transport aircraft"
  - Métodos más elaborados:
    - Roskam, "Methods for estimating drag polar for subsonic airplanes", (1973)

Normalización con respecto a los valores propios

$$C_D = C_D(C_L, M)$$

$$C_D = C_{D_0} + k_1 C_L^2 - k_2 C_L$$

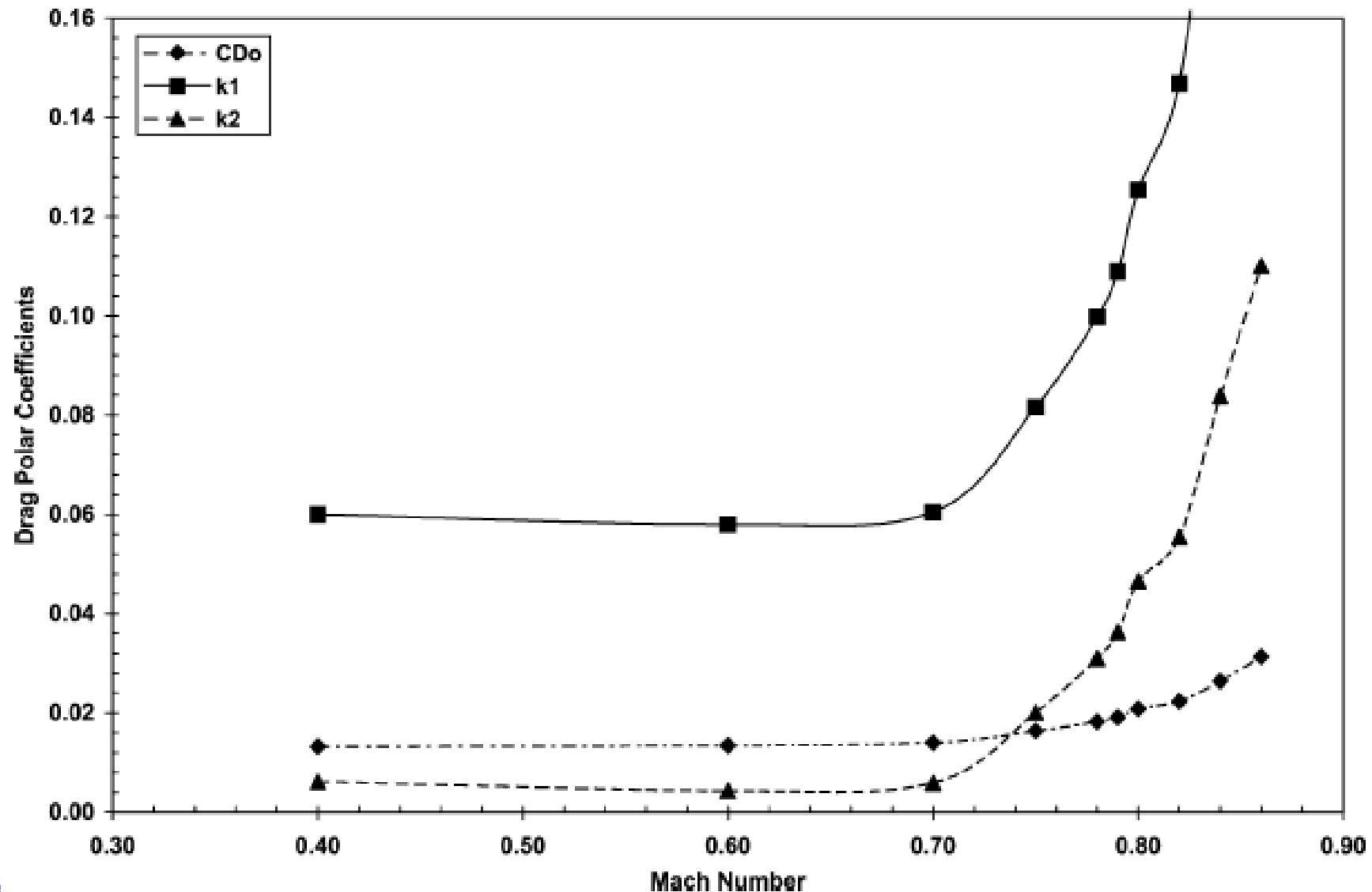
$$C_{D_0} = C_{D_0}(M), k_1 = k_1(M), k_2 = k_2(M)$$



# Polar del avión - V

- Aproximación de los coeficientes en función del Mach de divergencia  $M_{DD}$

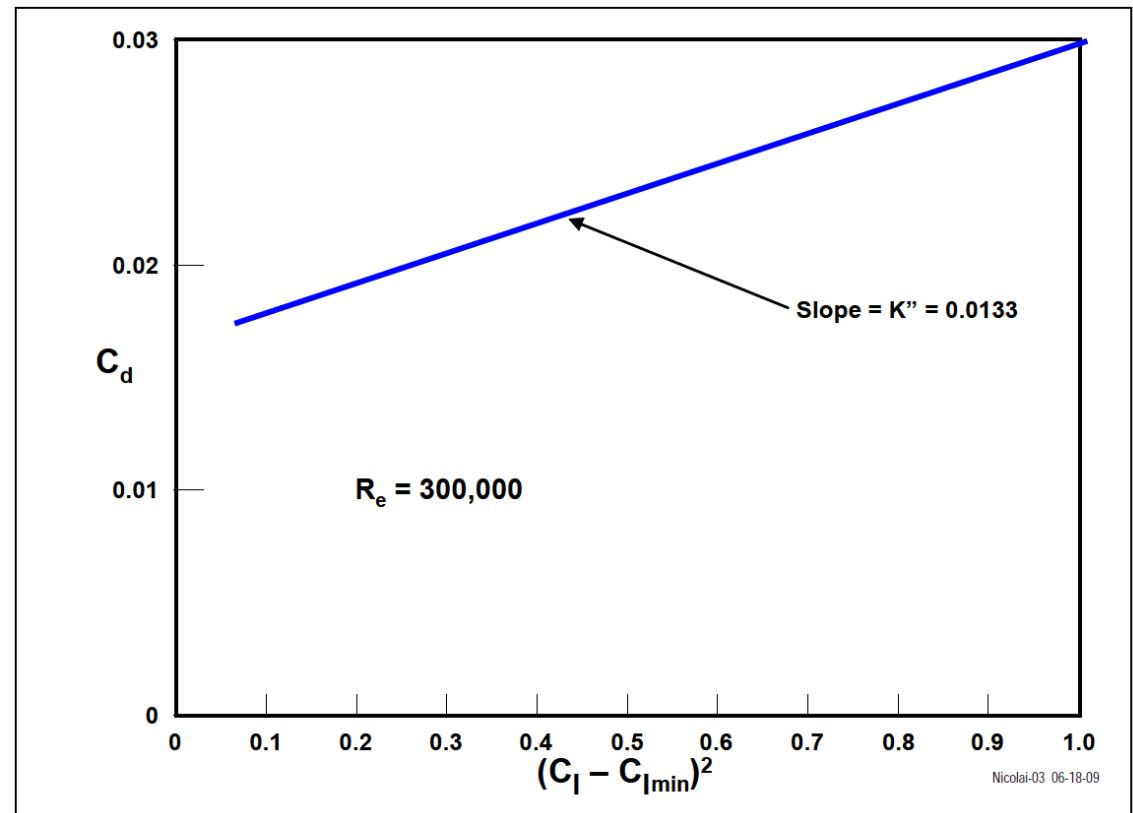
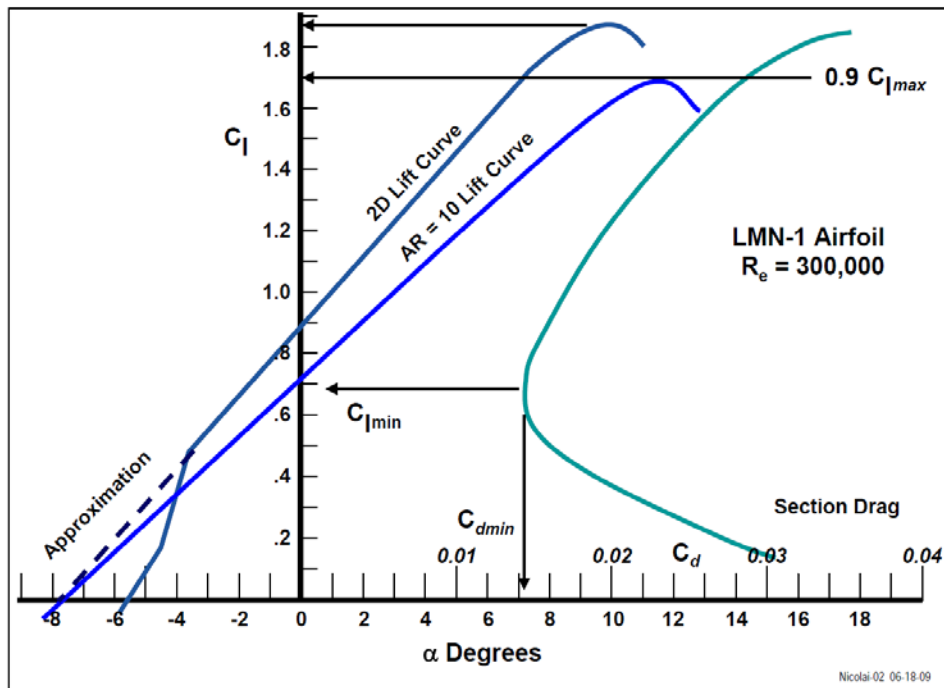
$$C_{D_0} = C_{D_0}(M), k_1 = k_1(M), k_2 = k_2(M)$$



# Polar del avión - VI

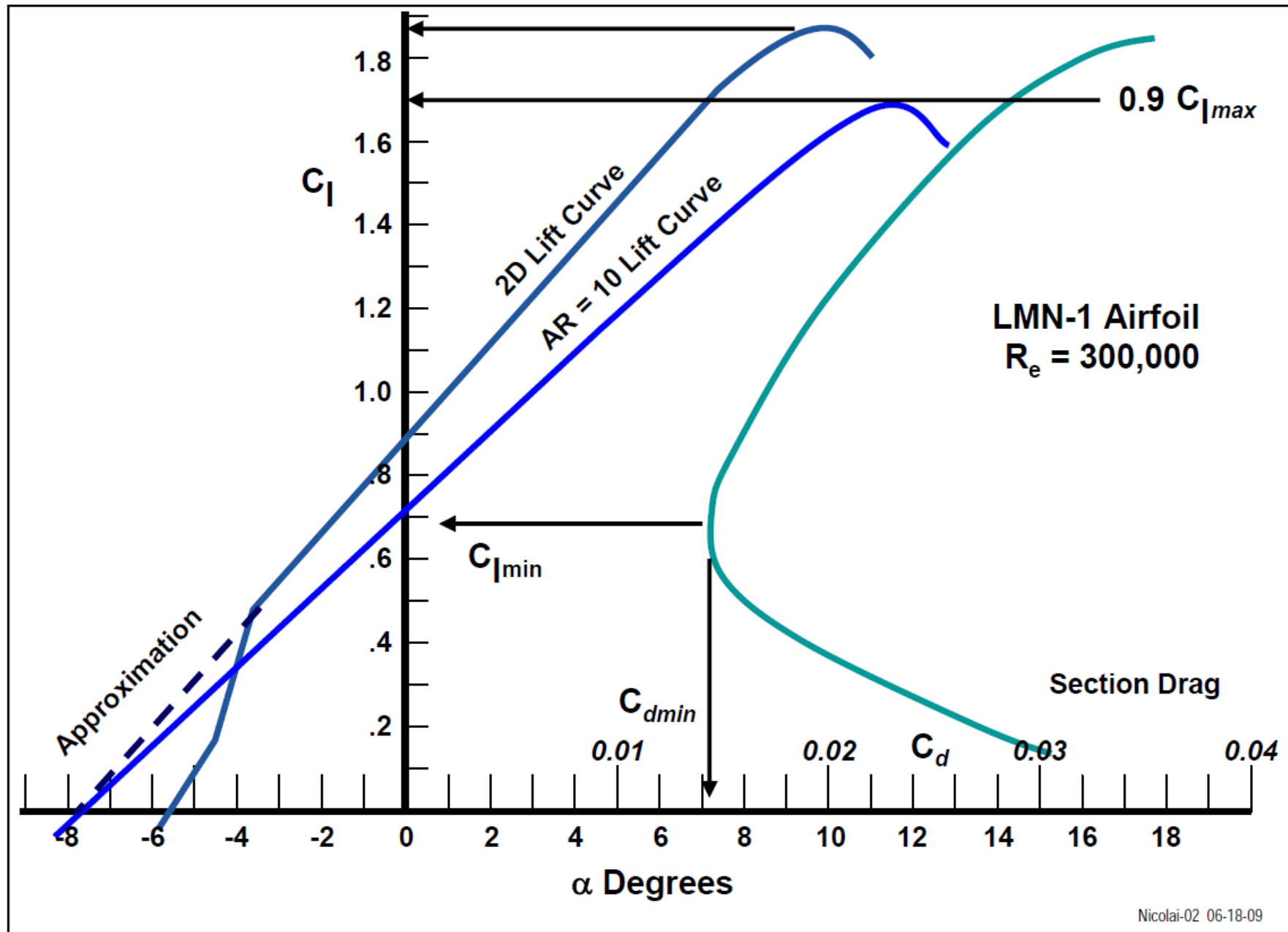
- Aproximación Nicolai (Nicolai, Leland M. and Carichner, Grant, "Fundamentals of Aircraft and Airship Design, Volume I, Aircraft Design", AIAA, Reston, Va, 2010)

$$C_D = C_{D_{min}} + K' C_L^2 + K'' (C_L - C_{L_{min-drag}})^2 \quad K' = \frac{1}{\pi A e}$$



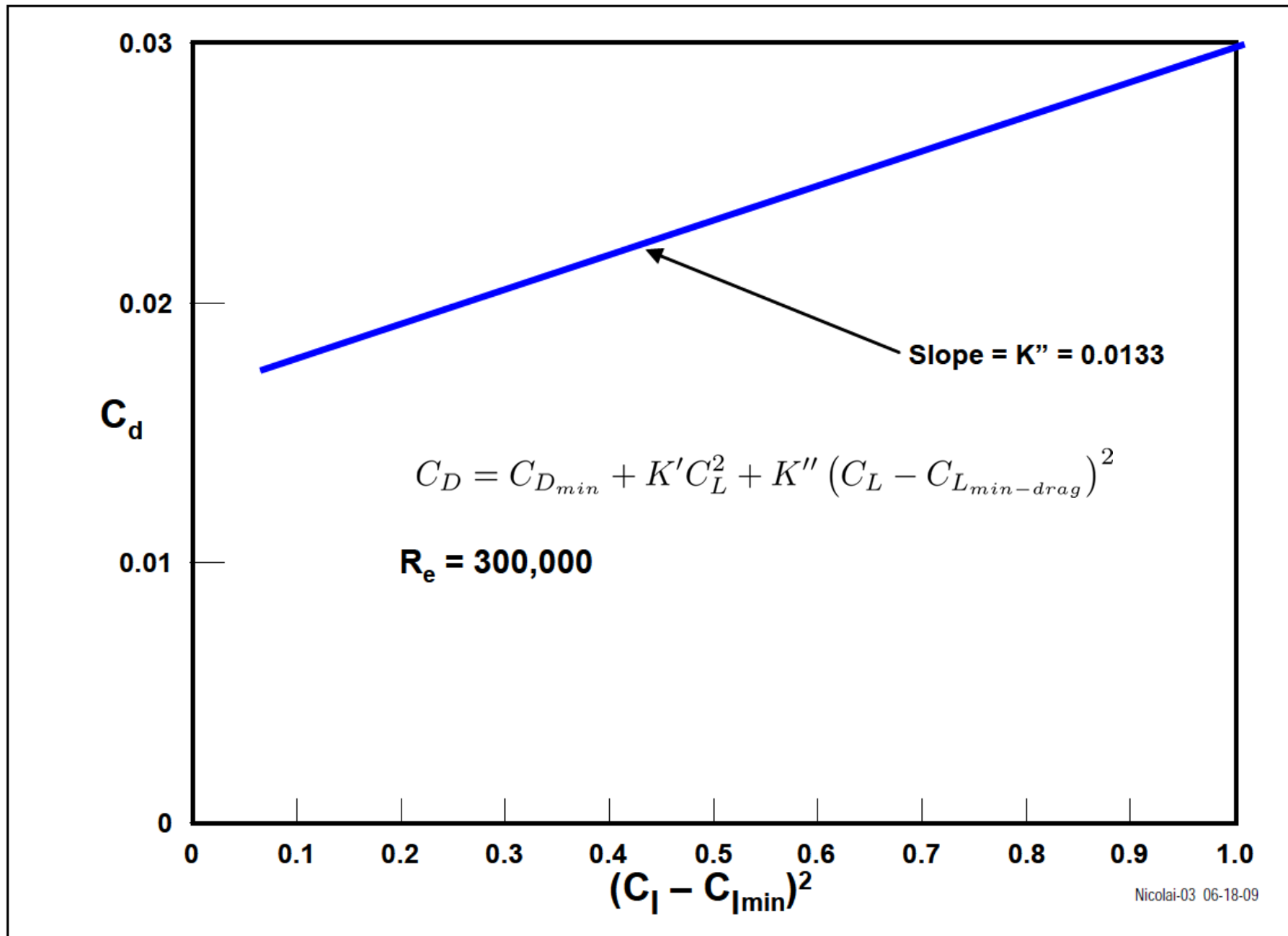
## Aproximación Ala

# Polar del avión - VII

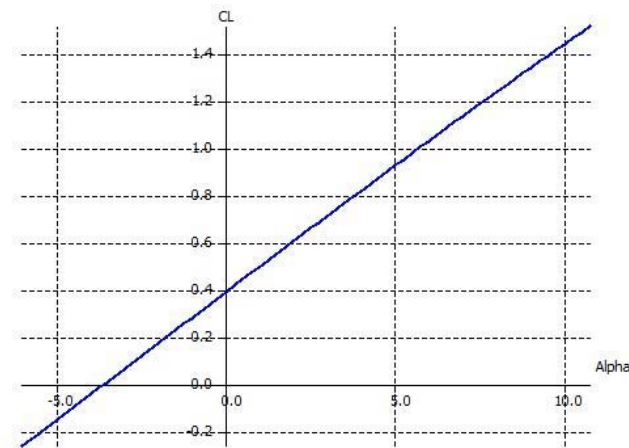
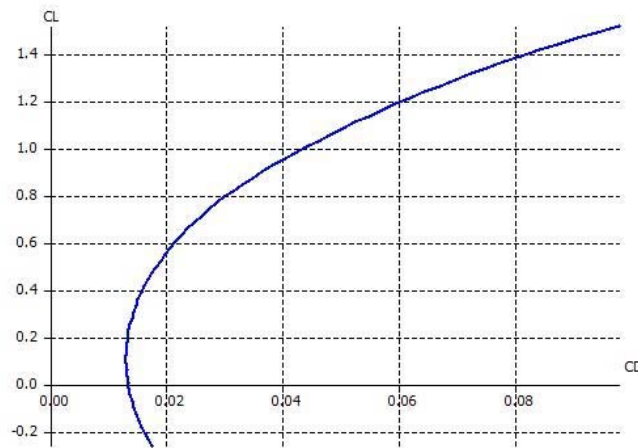
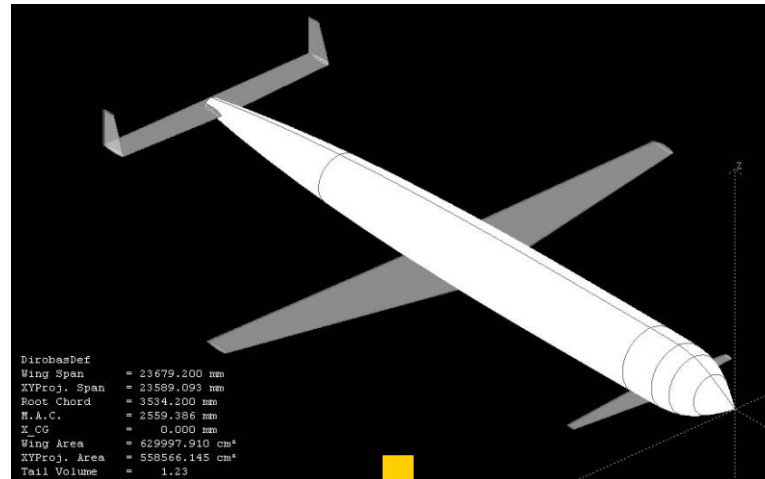


Nicolai-02 06-18-09

# Polar del avión - VIII



# Polar del avión - X



$$C_D = C_{D_0} + k_1 C_L^2 + k_2 C_L$$

Ajuste polinómico

# Polar del avión - XI

Blackboard Learn

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Apps Compre Balanzas H... ~ZbigKlep A123 System High P... pearltrees • Free ebo... Jornadas de i+D+i e... 100 Life Hacks That ... El salto de Félix Bau... Público - OneDrive 16 Research - LaTeX po...

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Mi institución Cursos

Cálculo de Aeronaves-Grado en Ingeniería Aeroespacial Tablero de discusión Foro: Frequently Asked Questions - Aerodinámica El modo de edición está: **ACTIVADO**

Vista de lista Vista de árbol

## Foro: Frequently Asked Questions - Aerodinámica

En esta página puede organizar secuencias del foro y aplicar configuraciones a varias secuencias o a todas ellas. Las secuencias se muestran en formato de tabla. Para ordenar las secuencias, haga clic en el título de la columna o en los signos de intercalación que aparecen en la parte superior de cada columna. [Más ayuda](#)

Crear secuencia Calificar foro Moderar foro Suscribirse

Acciones de secuencia Recopilar Eliminar

| Fecha          | Secuencia                                                                           | Autor   | Estado    | Etiquetas | Publicaciones no leídas | Total de publicaciones |
|----------------|-------------------------------------------------------------------------------------|---------|-----------|-----------|-------------------------|------------------------|
| 13/12/13 14:06 | Determinación polar parabólica a partir de XFLR5                                    | Anónimo | Publicada | 0         | 1                       |                        |
| 3/12/13 21:26  | Uso de XFLR5 para determinar momentos de cabeceo                                    | Anónimo | Publicada | 1         | 1                       |                        |
| 3/12/13 21:10  | Pautas estimación del CL máximo a partir de la información de XFLR5                 | Anónimo | Publicada | 0         | 1                       |                        |
| 3/12/13 10:53  | Estimación del CLmax                                                                | Anónimo | Publicada | 0         | 1                       |                        |
| 21/11/13 1:01  | Pautas estimación entrada en pérdida XFLR5                                          | Anónimo | Publicada | 1         | 1                       |                        |
| 15/11/13 0:19  | Limitaciones del uso de XFLR5 para estimar características aerodinámicas de aviones | Anónimo | Publicada | 1         | 1                       |                        |

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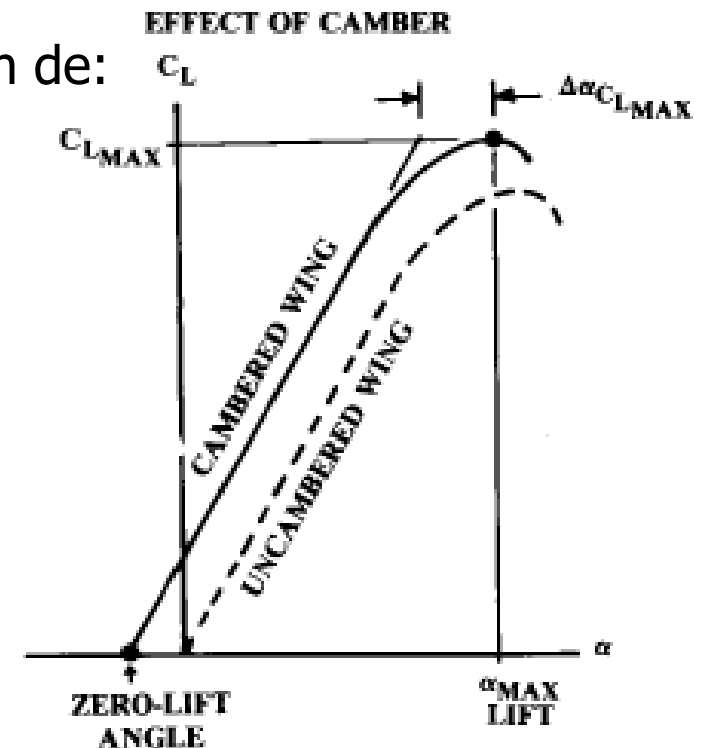
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# Coeficiente de Sustentación - $C_L$

- La sustentación, y por consiguiente el coeficiente de sustentación es función de  $C_{L\alpha}$  y el ángulo de ataque ( $\alpha$ ):

$$C_L = C_{L_0} + C_{L\alpha} \alpha$$

- La pendiente de la curva de empuje se ve modificado con el alargamiento, siendo el **valor teórico** para toda ala con **alargamiento**  $\infty = 2\pi$
- Dicha pendiente tiene que ser **corregida** para el **alargamiento** de cada ala por lo que deja de ser el teórico  $2\pi$ .
- Métodos analíticos para determinar  $C_{L\alpha}$  en función de:
  - Alargamiento (AR).
  - Área expuesta del ala ( $S_{exp}$ ).
  - Mach (M)
  - Factor de sustentación del fuselaje.
  - Flecha ( $\Lambda$ ).
  - Eficiencia aerodinámica del perfil ( $E$ ).
- Método diferentes para subsónico y supersónico



# $C_{L\alpha}$ para alas en 3D - I

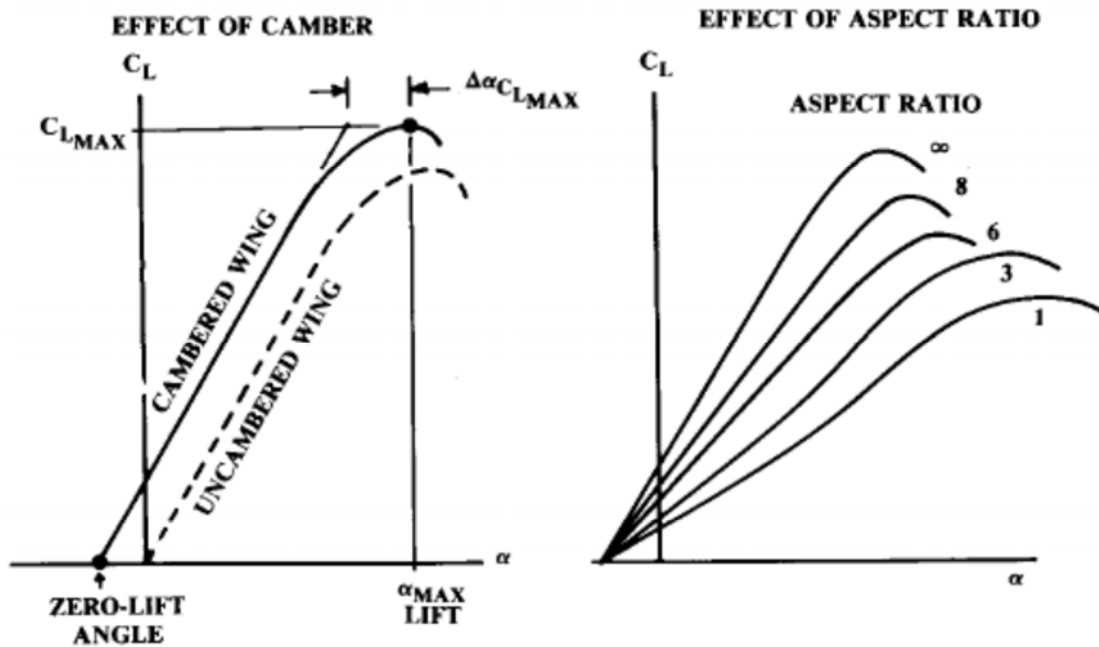


Fig. 12.4 Wing lift curve.

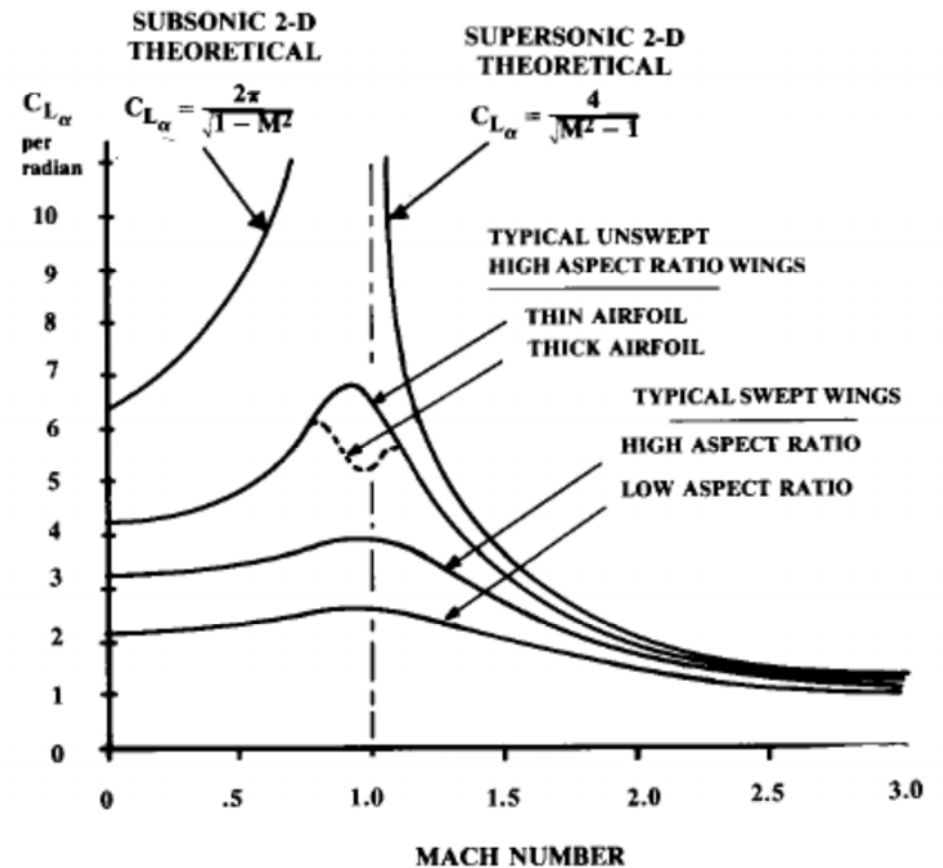


Fig. 12.5 Lift curve slope vs Mach number.

# $C_{L\alpha}$ para alas en 3D - II

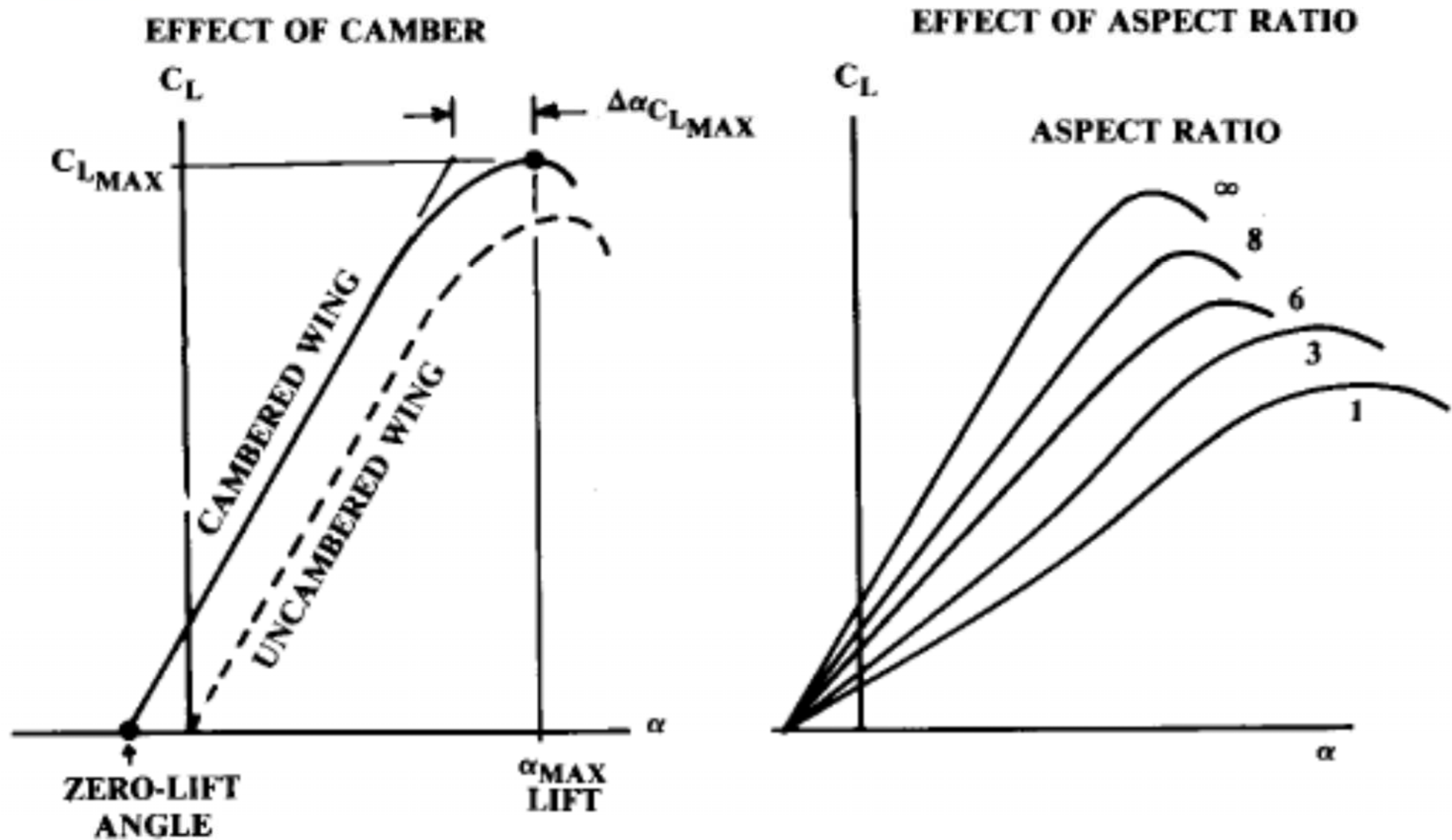


Fig. 12.4 Wing lift curve.

# $C_{L\alpha}$ para alas en 3D - III

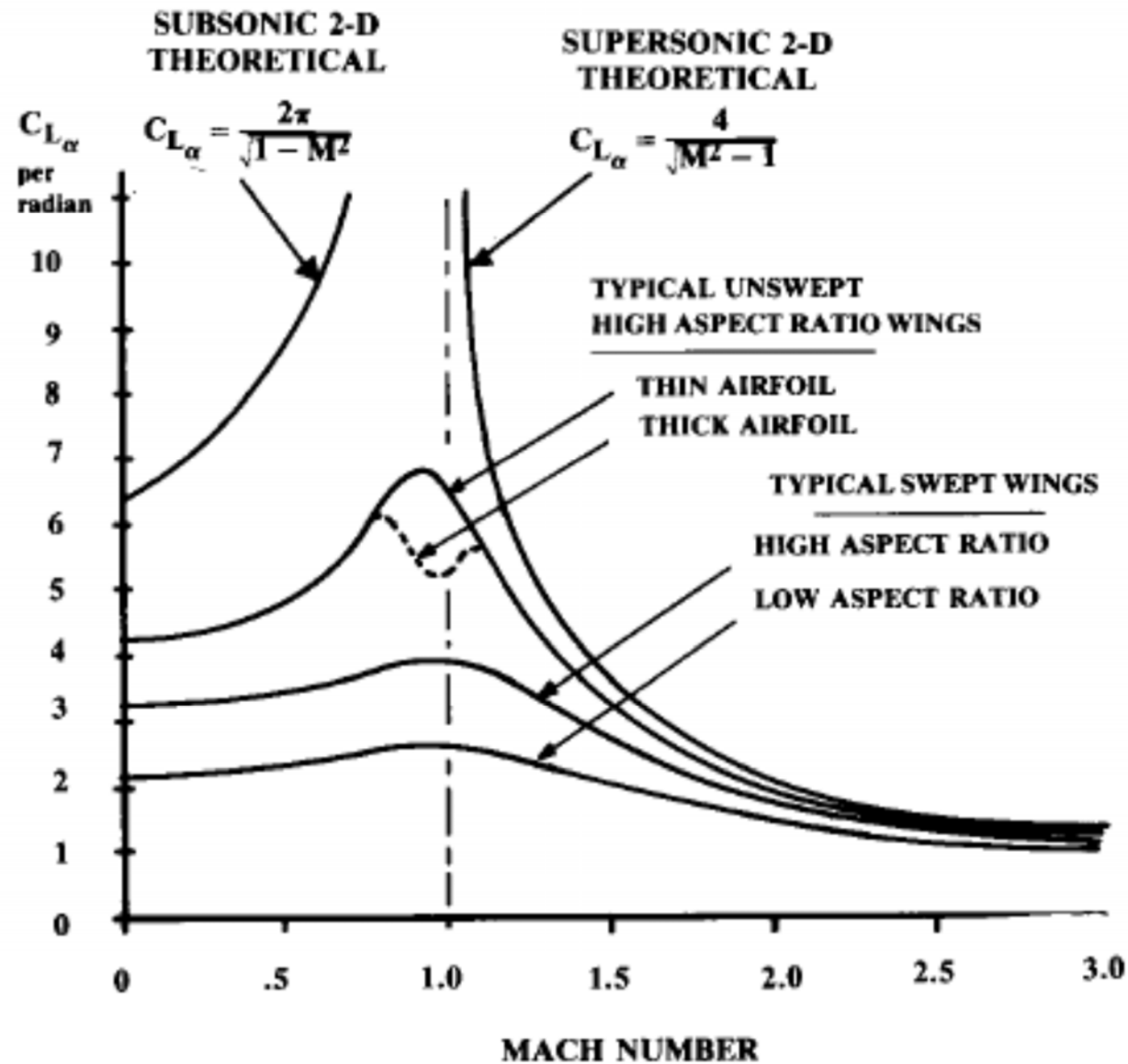


Fig. 12.5 Lift curve slope vs Mach number.

# $C_{L\alpha}$ para alas en 3D - II

- $C_{L\alpha}$  se necesita en el diseño conceptual del avión en tres etapas primordiales:
  1. **Correcta selección** del ángulo de **incidencia** de las alas.
    - En aviones de transporte es primordial que durante crucero el suelo este nivelado.
    - El ángulo de incidencia influye en el ángulo de ataque del fuselaje durante despegue y aterrizaje.
      - Altura Tren de aterrizaje.
      - Envergadura del fuselaje detrás del ala.
  2. Método para obtener la **resistencia** debido a la **sustentación** para aviones con requisitos elevados en las actuaciones.
  3. **Análisis** más detallado de la **estabilidad longitudinal** del avión:
    - Balance fuerzas y momentos
- $C_{L\alpha}$  disminuye con alargamiento.
  - Al disminuir el alargamiento, la habilidad del aire para escapar por las alas previene la entrada en pérdida incluso a ángulos de ataque elevados.
- $C_{L\alpha}$  disminuye con el aumento de flecha, y los efectos son similares.
- Perfiles actuales tiene un 90-100% eficiencia aerodinámica del perfil ( $\eta$ )
- Influencia con el Mach
- Alargamiento del ala al incluir *winglets* y *endplates*.

# Estimación $C_{L\alpha}$ - subsónica

## Corrección 2D → 3D

$h$  es la altura del "endplate"

$b$  es la envergadura del ala

$$\text{endplate} \Rightarrow A_{effective} \cong A \left( 1 + 1.9 \frac{h}{b} \right)$$

$$\text{winglet} \Rightarrow A_{effective} \cong 1.2A$$

Flecha del ala en la cuerda donde el perfil tiene el máximo espesor

$S_{exposed}$  es la parte de área de la  $S_{ref}$  que ve el flujo (que no está cubierta por el fuselaje). ¡¡No confundir con la superficie mojada  $S_{wet}$ !!

$$C_{L\alpha} = \frac{2\pi A}{2 + \sqrt{4 + \frac{A^2 \beta^2}{\eta^2} \left( 1 + \frac{\tan^2 \Lambda_{max,t}}{\beta^2} \right)}} \left( \frac{S_{exposed}}{S_{ref}} \right) (F)$$

Pendiente sustentación 2D  $\rightarrow \eta = \frac{C_{l\alpha}}{2\pi/\beta}$

Eficiencia aerodinámica del perfil  $\rightarrow \beta^2 = 1 - M^2$

Factor de sustentación del fuselaje  $d$  – diámetro del fuselaje  $b$  - envergadura

$$F = 1.07 \left( 1 + \frac{d}{b} \right)^2$$

$$\left( \frac{S_{exposed}}{S_{ref}} \right) (F) < 1 \sim 0.98$$

# $C_{L\alpha,w}$ , $C_{L\alpha,t}$ and $C_{L\alpha,c}$

Corrección 2D → 3D Método alternativo

$$a_w = \frac{2\pi A}{2 + \sqrt{\frac{A^2 \beta^2}{k^2} \left(1 + \frac{\tan^2 \Lambda_{c/2}}{\beta^2}\right) + 4}}$$

$$\beta = \sqrt{1 - M^2}$$

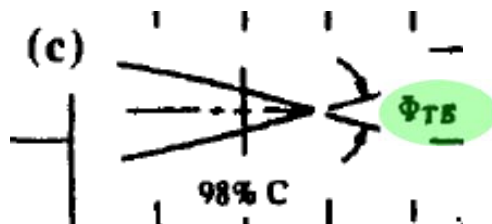
$$k = a_o / 2\pi$$

$\Lambda_{c/2}$  is the midchord sweep.

$a_o$  The sectional (two-dimensional) lift-curve slope  $a_o$

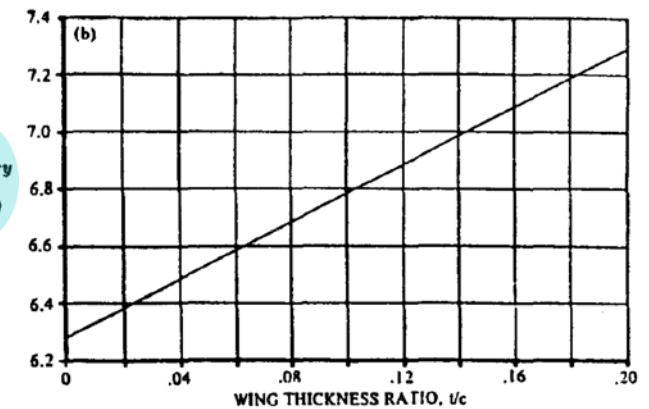
$$a_o = \frac{1.05}{\sqrt{1 - M^2}} \left[ \frac{a_o}{(a_o)_{theory}} \right] (a_o)_{theory}$$

$$\tan \frac{\phi'_{TE}}{2} = \frac{0.5 y_{90} - 0.5 y_{99}}{9}$$



$(a_o)_{theory}$   
(per rad)

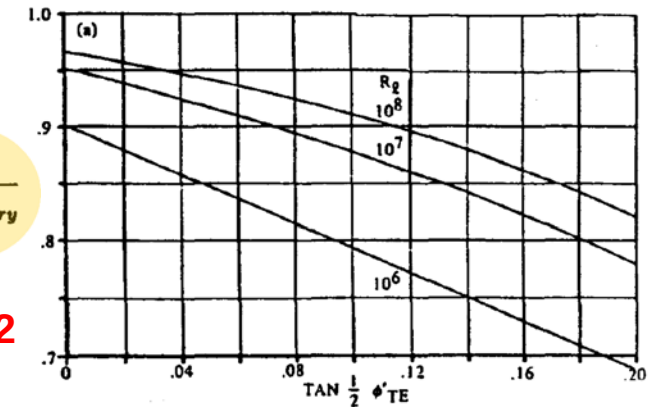
Fig A1



a) Theoretical sectional lift-curve slope

$\frac{a_o}{(a_o)_{theory}}$

Fig A2



b) Empirical correction factor

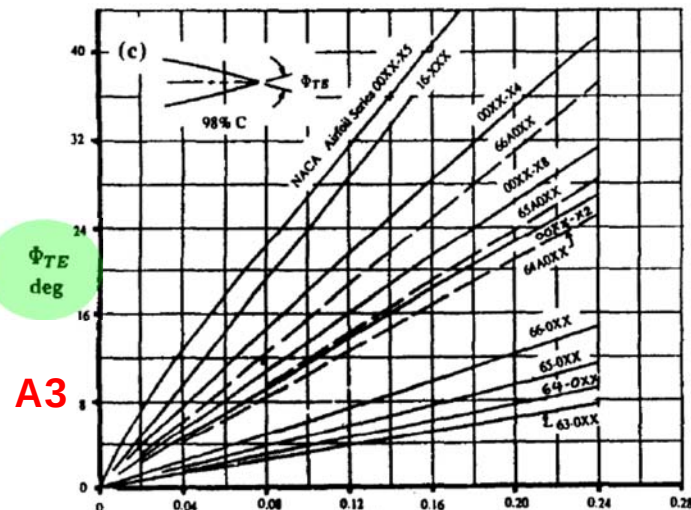
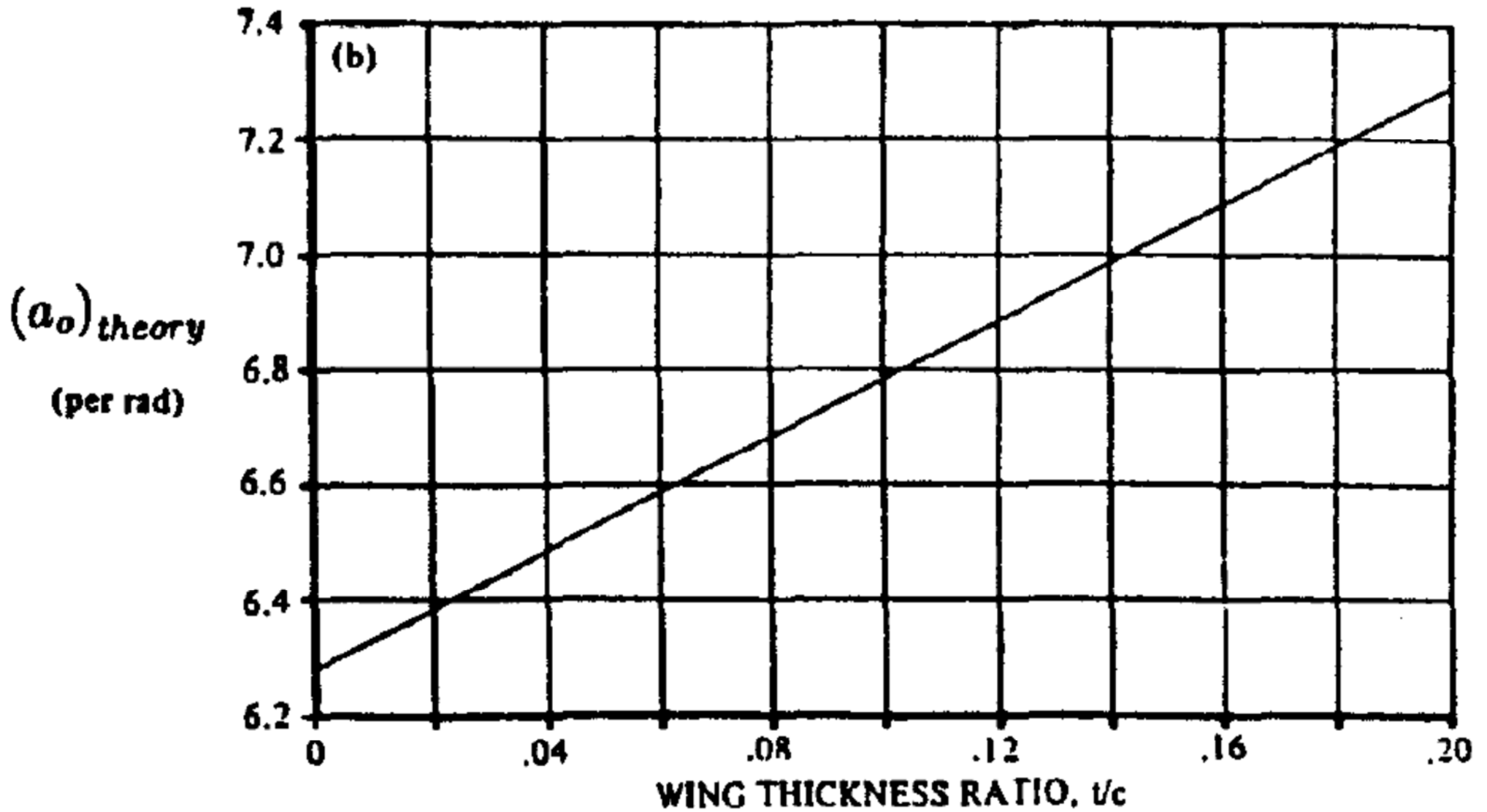


Fig A3

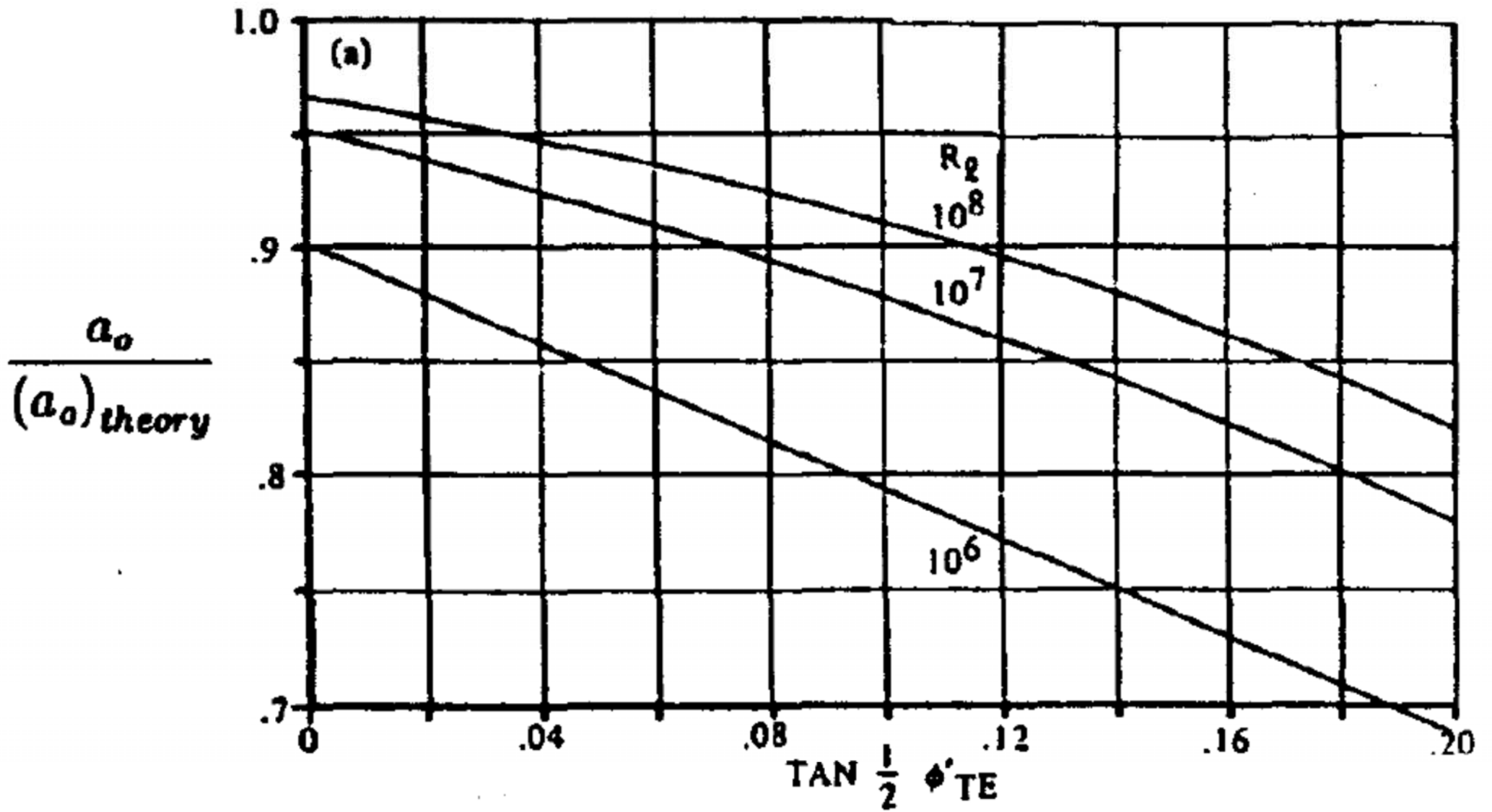
c) Variation of trailing-edge angle  $\phi_{TE}$

# Fig A1



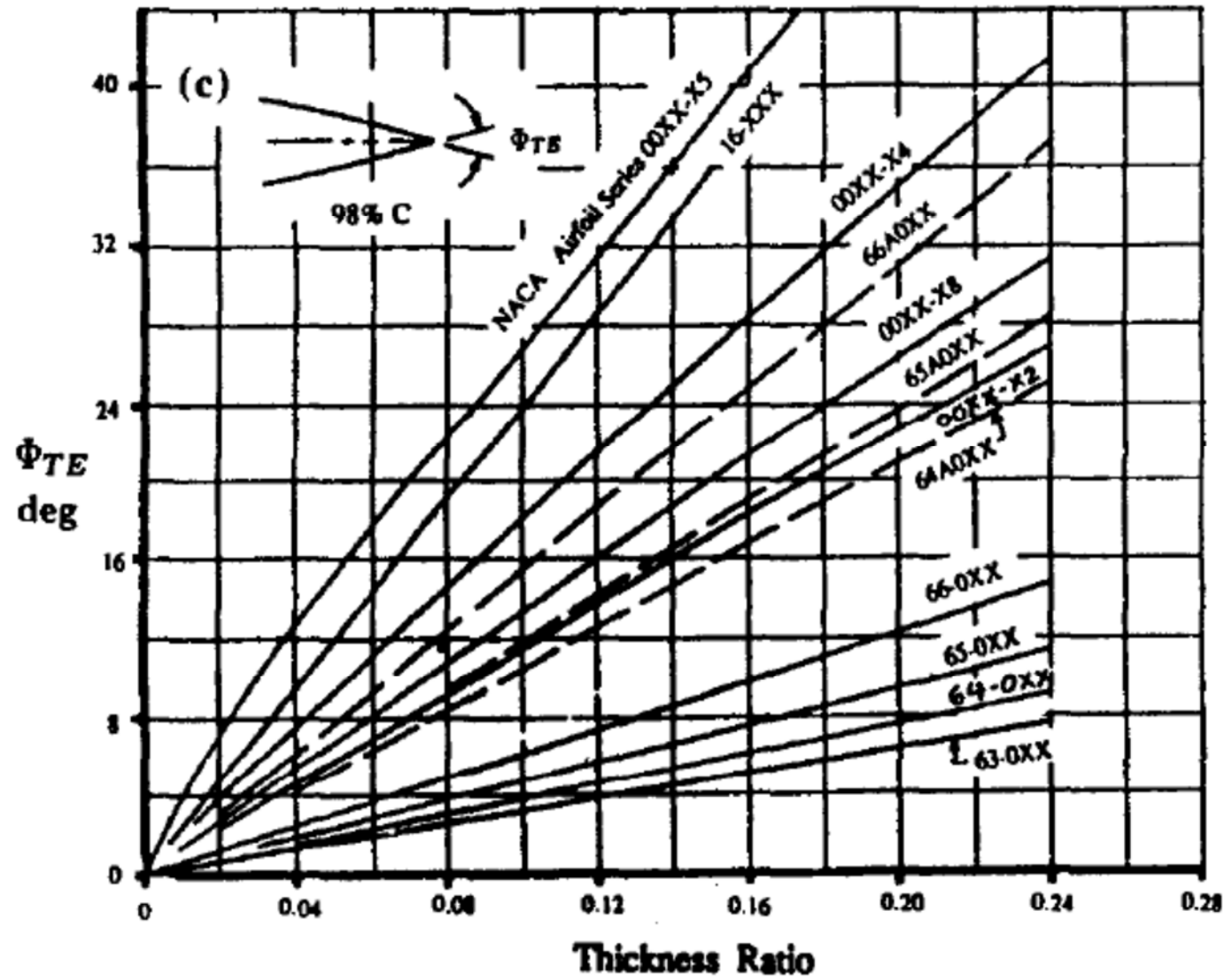
**a) Theoretical sectional lift-curve slope**

# Fig A2



**b) Empirical correction factor**

# Fig A3



c) Variation of trailing-edge angle  $\phi_{TE}$

Fig. 3.13 Sectional (two-dimensional) lift-curve slope of wings, continued.<sup>1</sup>

# Estimación $C_{L\alpha}$ - XFLR5

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En esta página puede organizar secuencias del foro y aplicar configuraciones a varias secuencias o a todas ellas. Las secuencias se muestran en formato de tabla. Para ordenar las secuencias, haga clic en el título de la columna o en los signos de intercalación que aparecen en la parte superior de cada columna. [Más ayuda](#)

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# Winglets - I

- Superficies al final de las alas diseñadas para mejorar la eficiencia de las alas fijas.
  - Reducción de la resistencia hasta un 20%
- También tienen una componente **estética**
- Su principal objetivo es **modificar la estela del ala** de manera que resulte beneficiosa.
- De igual forma, los winglets pueden mejorar
  - **Aumentan el alargamiento efectivo** de un ala con menos área alar:
    - El **aumentar la envergadura reduce la resistencia inducida** pero **aumenta la resistencia parasitaria**, y además requiere **aumentar el refuerzo estructural** y el peso de las alas.
    - El uso de winglets aumenta la sustentación generada en las puntas, reduce la resistencia inducida en puntas causadas por los vórtices:
      - Mejora el **L/D**
      - Aumenta la eficiencia de consumo de combustible **aumentando el alcance**



# Winglets – Design - I

- Aspectos a tener en cuenta a la hora de decidir la geometría de los winglets:
  - Perfil
  - Distribución de cuerda
  - Altura
  - Torsión
  - Flecha
  - Toe Angle

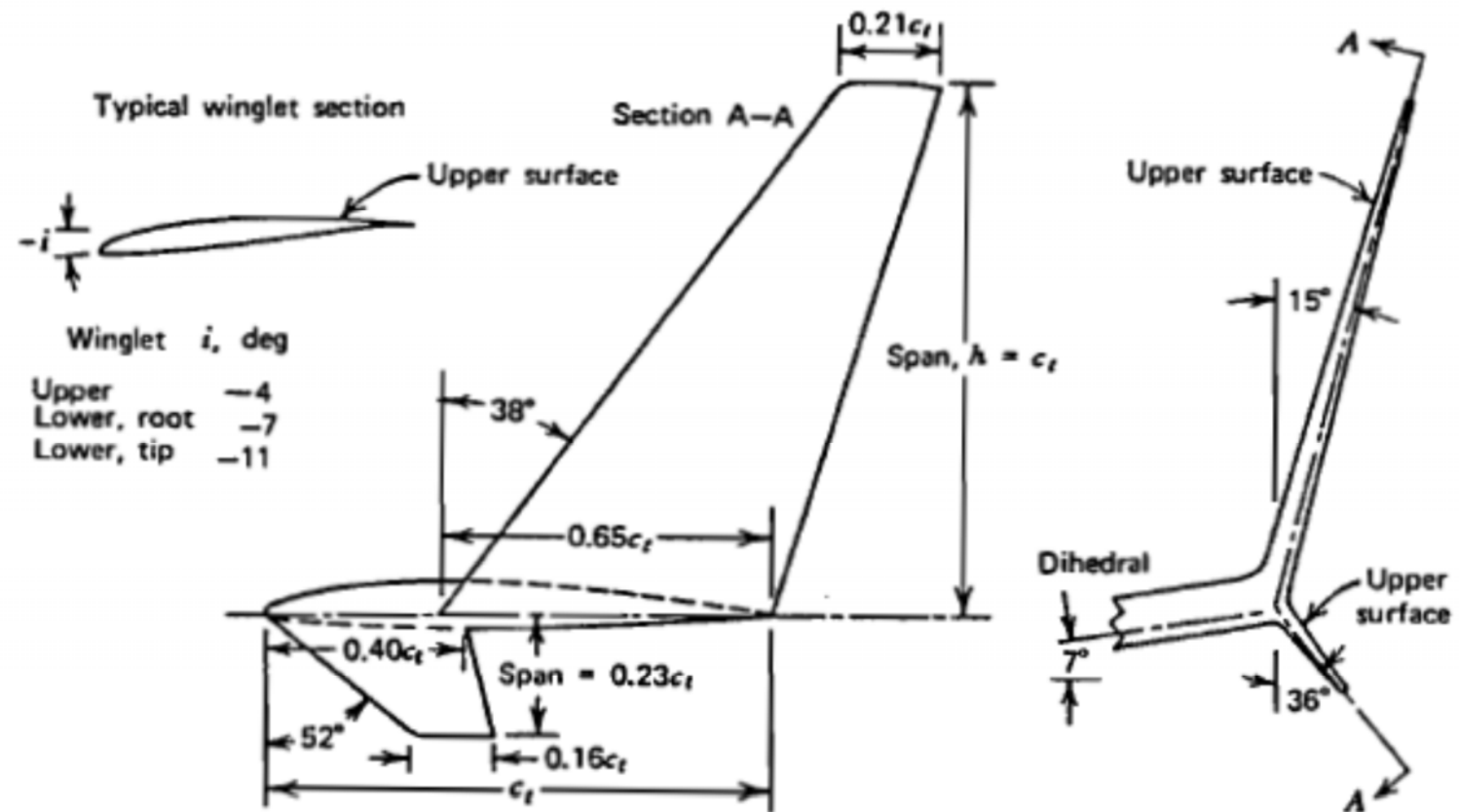
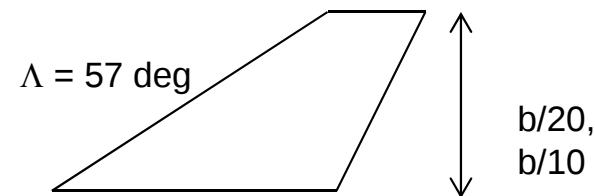


Figure 4.35 Winglet geometry.

# Winglets – Design - II

- Perfil (airfoil)
  - Tiene que generar suficiente sustentación mientras mantiene una resistencia reducida
  - No puede entrar en pérdida antes que el ala a vuelos a baja velocidad.
  - La geometría final viene dictada por las características aerodinámicas del perfil seleccionado.
  - Número de Reynolds bajo ( $1E5$  a  $1E6$ ) definido por cuerdas estrechas
- Distribución de cuerda – dimensionado (chord distribution)
  - Demasiado pequeños el perfil necesitará un coeficiente de sustentación elevado
  - Demasiado elevado
    - Carga alar del winglet elevada y puede causar que la sección exterior del ala entre en pérdida de forma prematura
  - Una distribución de cuerdas que favorece una distribución elíptica ayuda a distribuir la carga alar para un mayor número de regímenes de vuelo



Taper  $\Rightarrow \lambda = 0.3$

# Winglets – Design - III

- Altura (height)
  - Determinada por la **relación óptima** entre la **resistencia inducida** y **resistencia parasitaria**.
- Torsión/Flecha (twist/sweep)
  - Tanto la torsión como flecha tiene similares efectos en el winglet y para decidir cual es la configuración más adecuada hay que tener en cuenta la **carga alar** para cada **configuración**.
- Toe Angle - El ángulo de montaje en la base controla:
  - La **distribución alar** del winglet.
  - Efectos de la distribución de la carga alar en el ala principal.
  - Cada ángulo sólo es optimo para una condición de vuelo por lo que hay que llegar a un compromiso

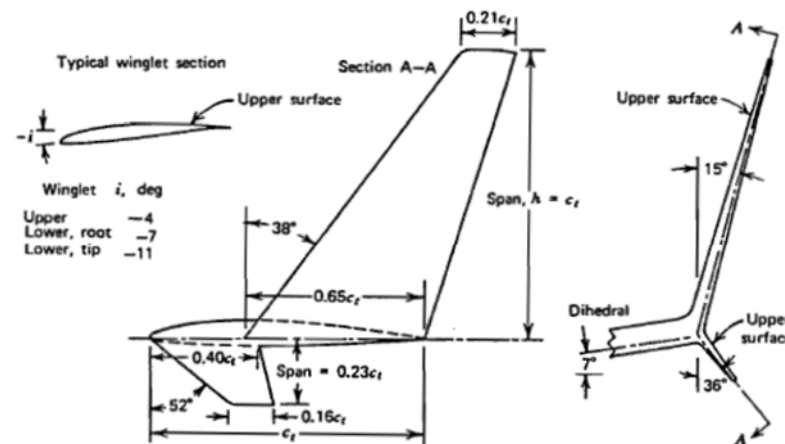
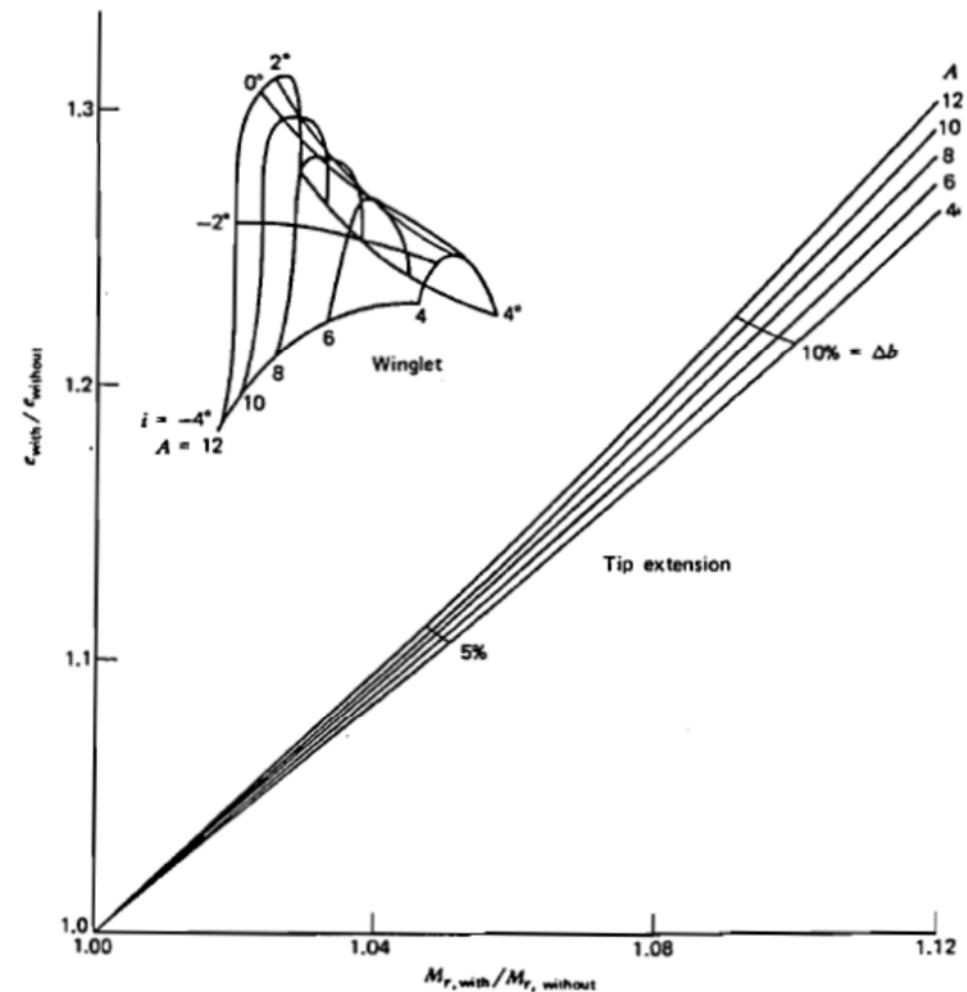


Figure 4.35 Winglet geometry.

# Winglets - II



**Figure 4.39** Comparison of tip extension and winglet when added to an untwisted wing.

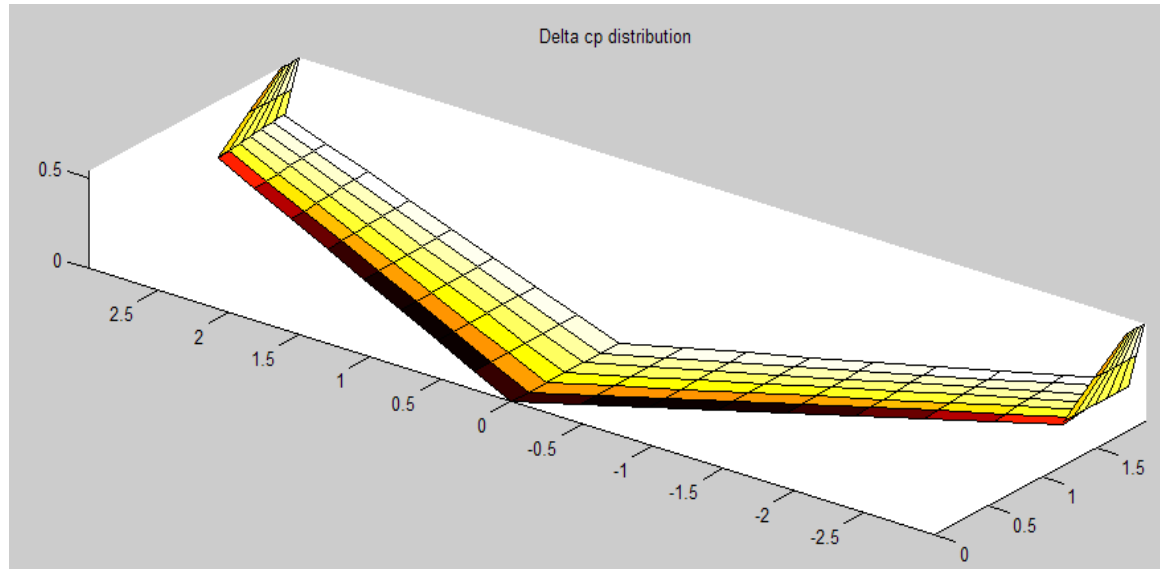
# Winglets – Resultados - I

- Reducciones de hasta el 20%
- Winglets son sólo necesarios para aquellas configuraciones en las que la resistencia inducida sea más elevada de lo normal.
- Muy recomendada para configuraciones canard
- Caso tipo: Boeing BBJ
  - Consumo combustible reducción 4%
  - Alcance incrementado 200 nm (BBJ)
  - Reducción de la marca cústica en un 6.5 % en aeropuertos
  - Reducción del 4% en emisiones NOX (vuelo de 2,000-nmi).



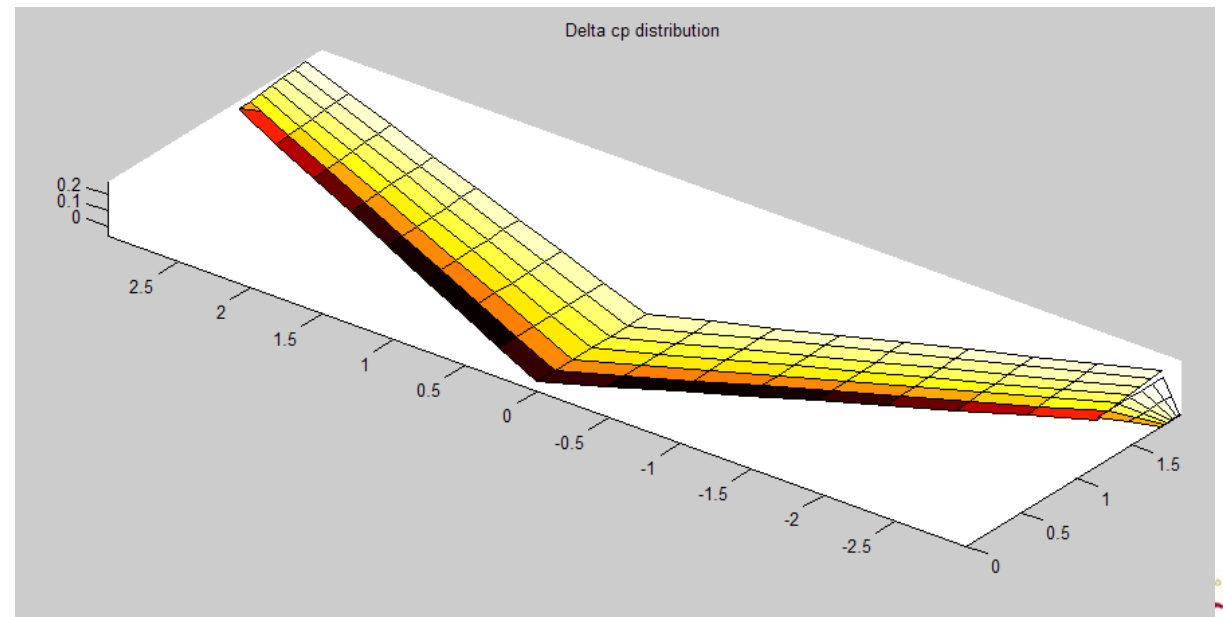
# Winglets – Resultados - II

## Small Version



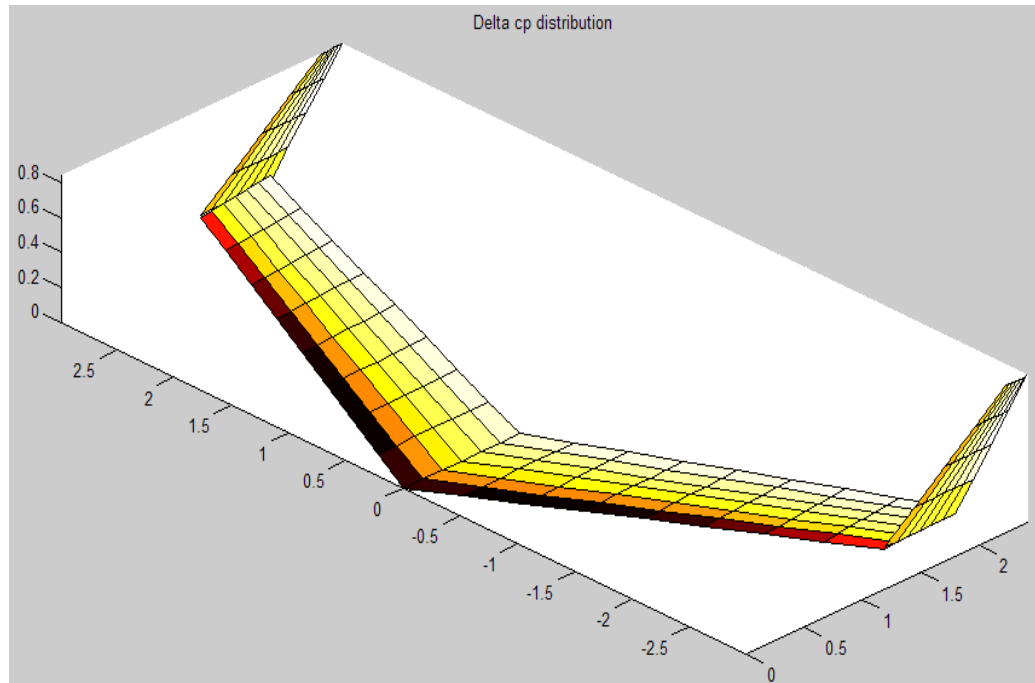
11% drag reduction  
(7% when compared to an extended wing)

8% drag reduction  
(4% when compared to an extended wing)



# Winglets – Resultados - III

## Large Version

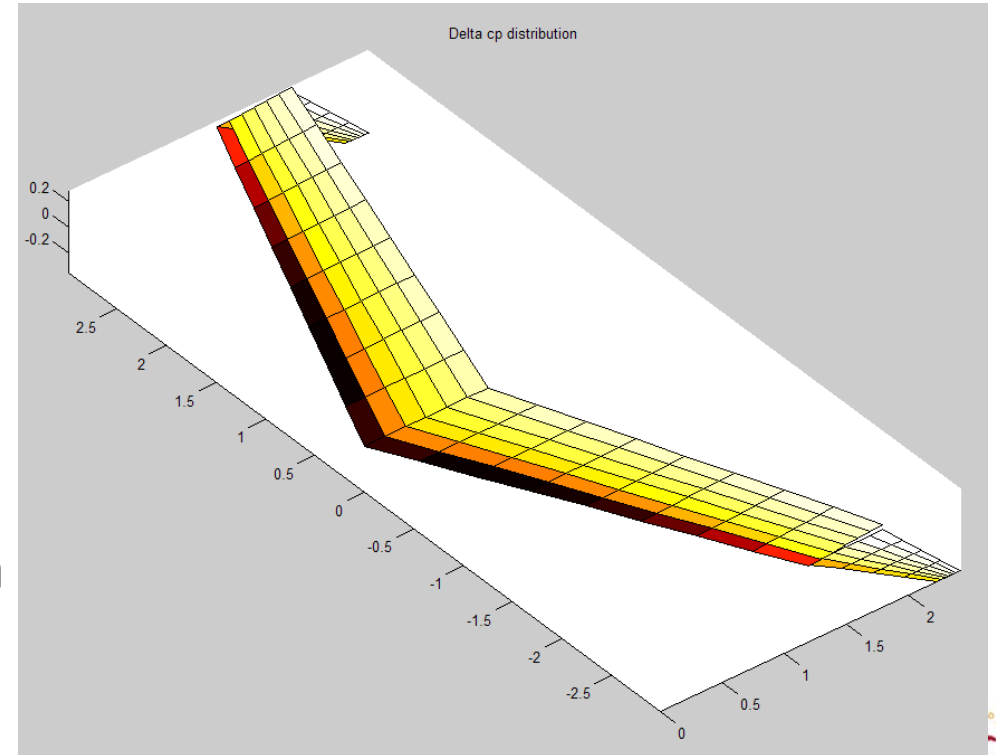


22% drag reduction

(14% when compared to an extended wing)

12% drag reduction

(4% when compared to an extended wing)



# Winglets



B 737-800 blended winglet

B 747-400 tip plus winglet



Raked Wings



KC-135 winglet

MD-11 Extended winglet

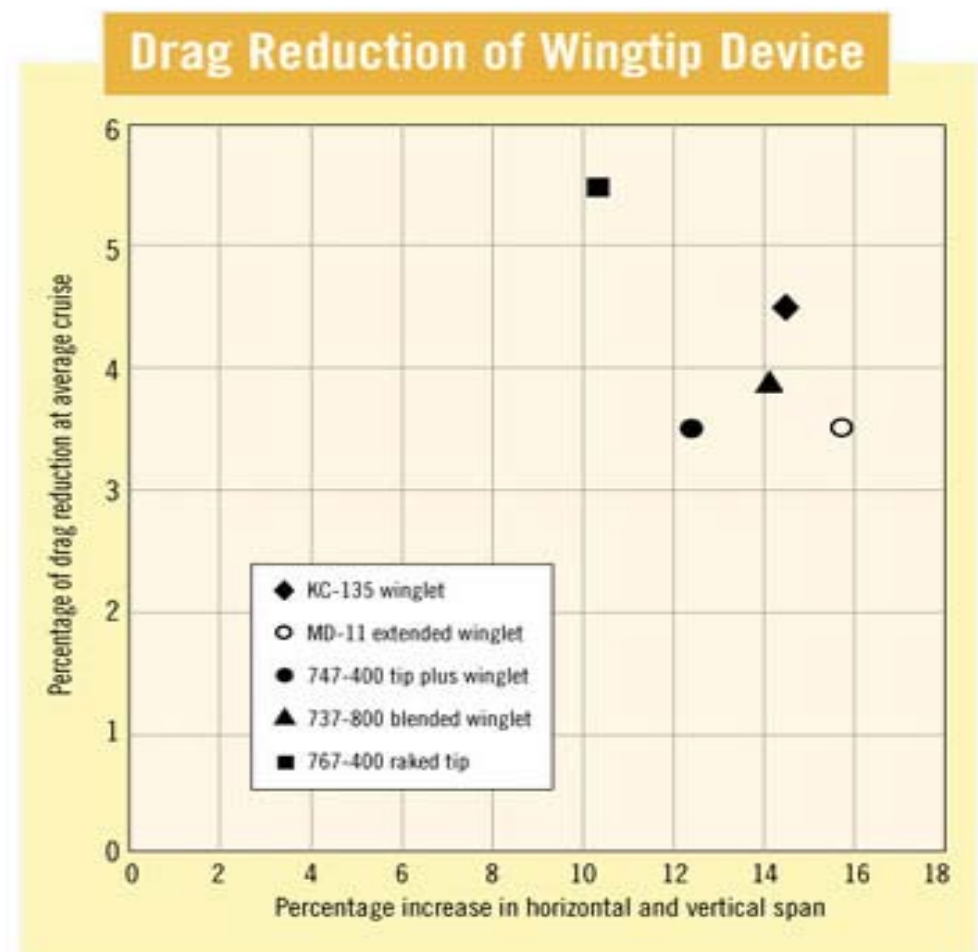
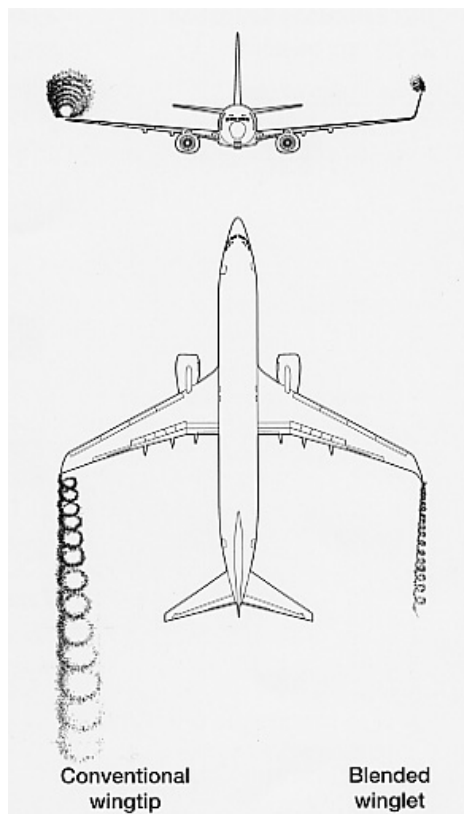


# Reducción Resistencia Inducida

reducción %

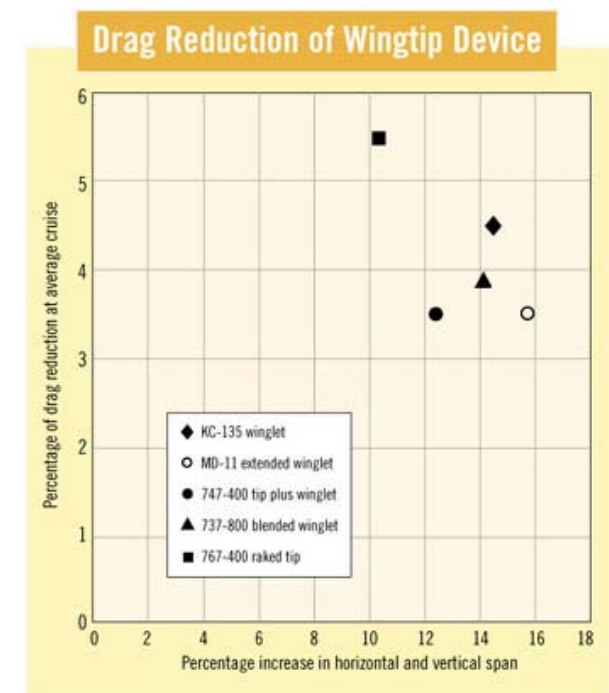
- Raked tips: reducción 5.5 %
- KC-135 winglet: reducción 4.5 %
- Blended winglet: reducción 4 %
- Tip plus winglet: reducción 3.5%
- MD-11 extended winglet: reducción 3.5%

¿Cómo lo aplico?  $\Rightarrow C_D = C_{D_0} + KC_L^2$



# Raked, Wingtips, Extended, Blended ?

- Raked wingtips are a feature on some Boeing airliners, where the tip of the wing has a higher degree of sweep, than the rest of the wing.
- The stated purpose of this additional feature is to improve fuel efficiency, climb performance and to shorten takeoff field length.
- It does this in much the same way that winglets do, by increasing the effective aspect ratio of the wing and interrupting harmful wingtip vortices.
- This decreases the amount of lift-induced drag experienced by the aircraft. In testing by Boeing and NASA, raked wingtips have been shown to reduce drag by as much as 5.5%, as opposed to improvements of 3.5% to 4.5% from conventional winglets. [11]
- An increase in wingspan is generally more effective than a winglet of the same length, but may present difficulties in ground handling.



# Otras influencias del $c_L$

- Hay que considerar otros aspectos de la generación de sustentación
  - Efectos no lineales de la sustentación
    - En alas con una **flecha elevada**, o **alargamiento**, el aire se escapa alrededor del borde de entrada de la flecha o en las puntas generando **vórtices** bastante fuertes que crean **sustentación adicional** que varía proporcionalmente con el cuadrado del ángulo de ataque.
    - Muy **difícil de estimar**.
  - Máxima sustentación (configuración limpia)
    - Sólo válida para configuraciones moderadas de alargamiento

$$C_{L_{max}} = 0.9C_{l_{max}} \cos \Lambda_{0,25c}$$

- Corrección para **alargamiento bajo y elevado**
- Máxima sustentación con superficies hipersustentadoras
  - **Incremento de sustentación**.

# Cálculo del $C_{Lmax}$ - I

- El  $C_{Lmax}$  **determinará** por lo general el **área del ala**.
  - Esto a su vez tendrá una gran **influencia** en la **resistencia del segmento de crucero**.
  - Lo que a su vez tendrá una gran **influencia** en el **peso de despegue** necesario para poder completar la misión de diseño.
- El coeficiente de sustentación máximo es uno de los **parámetros más críticos** para la **determinación del peso del avión**, pero **paradójicamente**:
  - La **estimación** del  $C_{Lmax}$  es uno de los **métodos menos fiables** de todos los que se emplean en los cálculos para el diseño conceptual de aeronaves.
  - **Ni siquiera** ensayos en **túneles de viento** pueden predecir de forma precisa las características de los valores de coeficiente de sustentación máximo.
- La **forma del borde de ataque** es uno de los **parámetros determinantes** ya que un ala con estrechamiento relativamente bajo, y un borde de ataque relativamente afilado tendrá un aumento en la sustentación debido a la generación de vórtices en el borde de entrada.

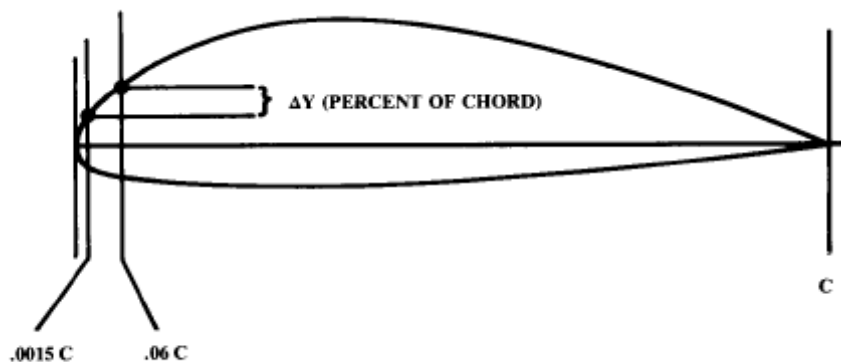
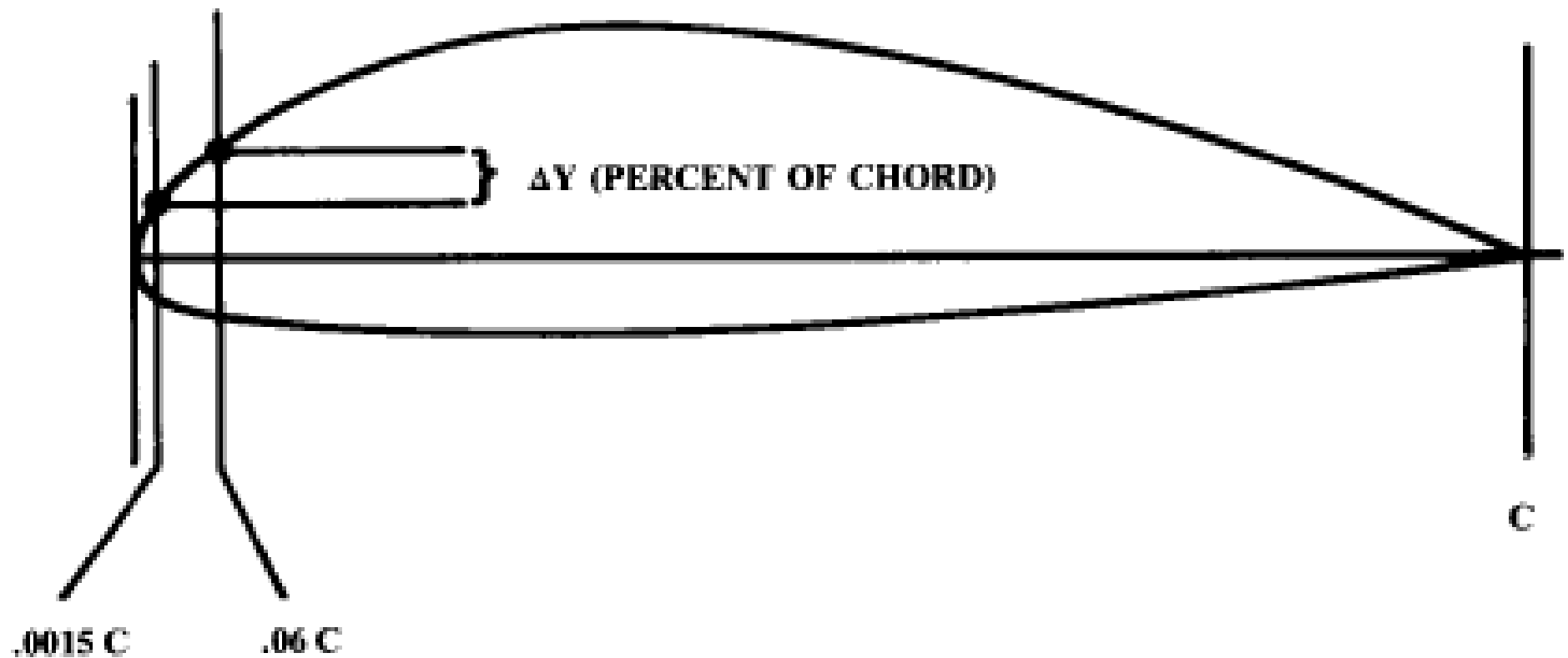


Fig. 12.7 Airfoil leading edge sharpness parameter.

## Introducción a los perfiles NACA

Table 12.1  $\Delta Y$  for common airfoils

| Airfoil type   | $\Delta Y$ |
|----------------|------------|
| NACA 4 digit   | 26 t/c     |
| NACA 5 digit   | 26 t/c     |
| NACA 64 series | 21.3 t/c   |
| NACA 65 series | 19.3 t/c   |
| Biconvex       | 11.8 t/c   |



**Fig. 12.7 Airfoil leading edge sharpness parameter.**

**Table 12.1  $\Delta Y$  for common airfoils**

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| NACA 64 series | 21.3 t/c   |
| NACA 65 series | 19.3 t/c   |
| Biconvex       | 11.8 t/c   |

# Superficies Hipersustentadoras (HLD) - I

- Siempre hay incompatibilidades en el diseño de las alas cuando se consideran todos los diferentes segmentos:
  - Idealmente durante crucero la curvatura del ala tendría que ser lo más pequeña posible para poder operar a una carga alar ( $W/S$ ) elevada.
  - Mientras que para despegue y aterrizajes ala tiene que generar mucha sustentación lo que implica elevadas curvaturas. ( $W/S$ ) reducida
- Para llegar a un compromiso entre los diferentes segmentos se utilizan superficies hipersustentadoras que aumentan la cuerda efectiva.
- El aumento en sustentación es del orden del 60-80 %.

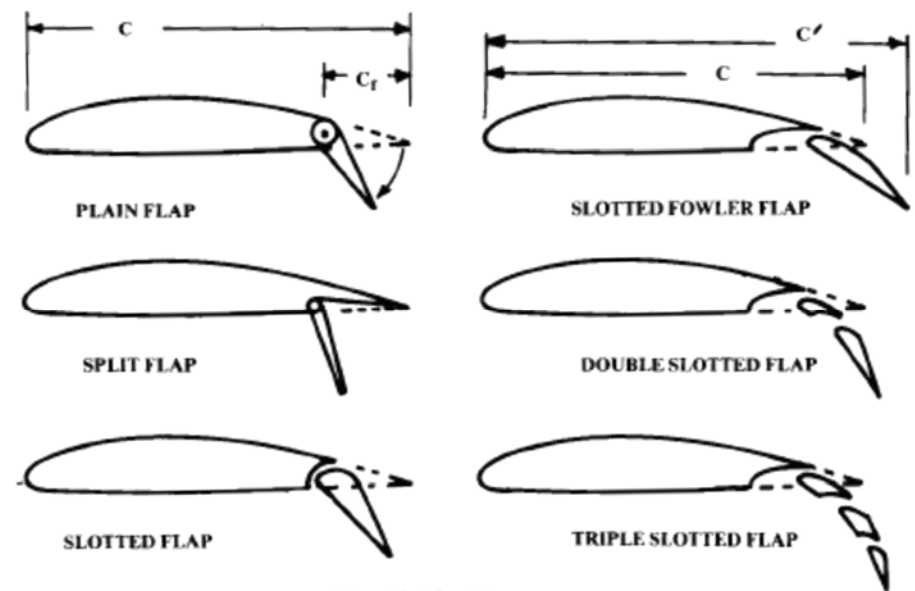
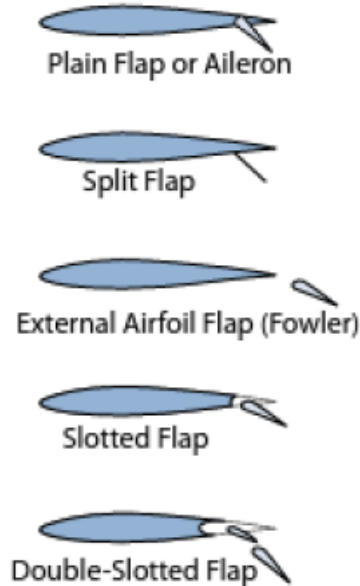


Fig. 12.16 Flap types.

Figure 9.1 Typical TE high-lift devices.

# Superficies Hipersustentadoras - II

- Superficies hipersustentadoras:
  - Prevenir que el **avión** alcance **valores no aceptables de velocidad** durante **despegue, acercamiento y aterrizaje**.
- ¿Como funcionan? – Aunque no todas al mismo tiempo
  - **Aumentan** la **curvatura** del perfil.
  - **Control** de la **capa límite** mejorando la distribución de presiones, re-energizando o eliminado las capas límite de baja energía.
  - **Aumento efectivo** del **área total del ala**:
    - Extendiendo la cuerda del perfil.
- Aleta hipersustentadora – *Flaps & Slats*
  - Flaps de borde de salida - *Flaps*:
    - **Aumentan** la **curvatura** y mejoran el flujo en el borde de salida.
    - Tienden a **promover entrada** en **perdida** del borde de ataque
  - Flaps de borde de entrada - *Slats*:
    - **Posponen** o eliminan la **entrada** en **perdida** del borde de ataque
    - **No aporta beneficio** a la curvatura.

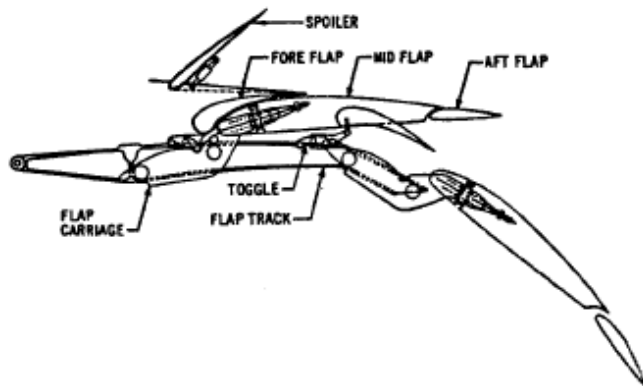


Fig. 7-25h. Triple slotted flap (Boeing 727)

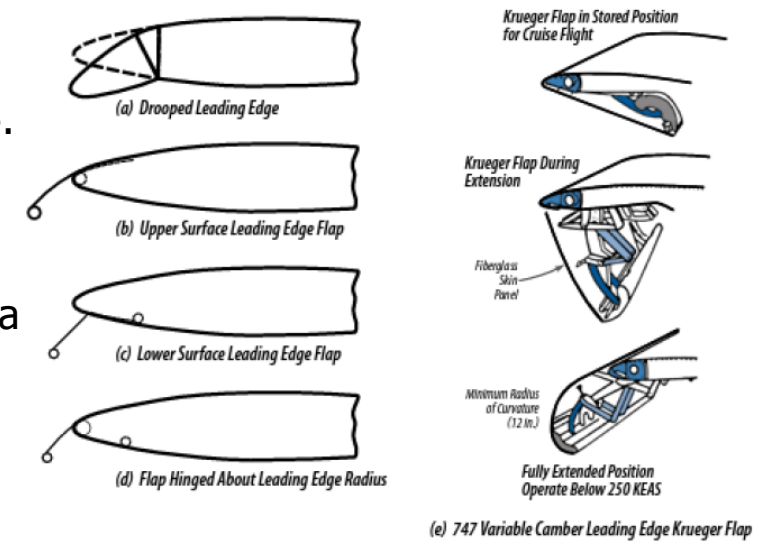


Figure 9.5 Various LE flap devices.

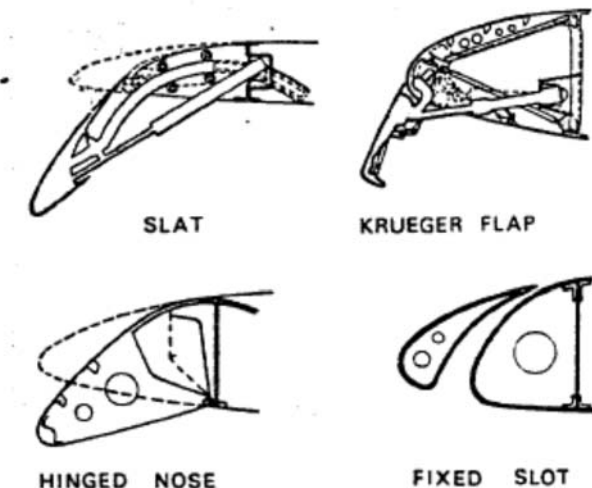


Fig. 7-26. Leading-edge high-lift devices.

# Superficies Hipersustentadoras – II - cont

## TE HLD Complejos

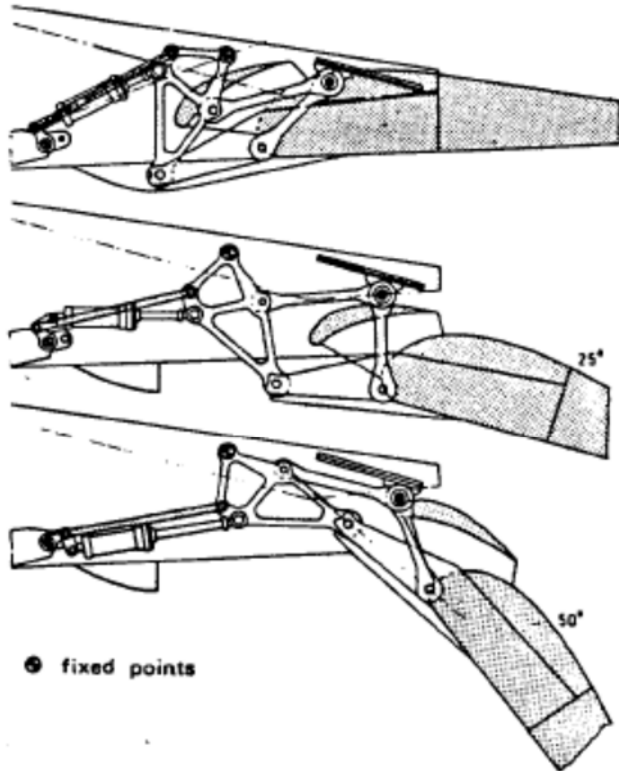


Fig. 7-25f. Double slotted flap with four-bar motion (Douglas DC-8)

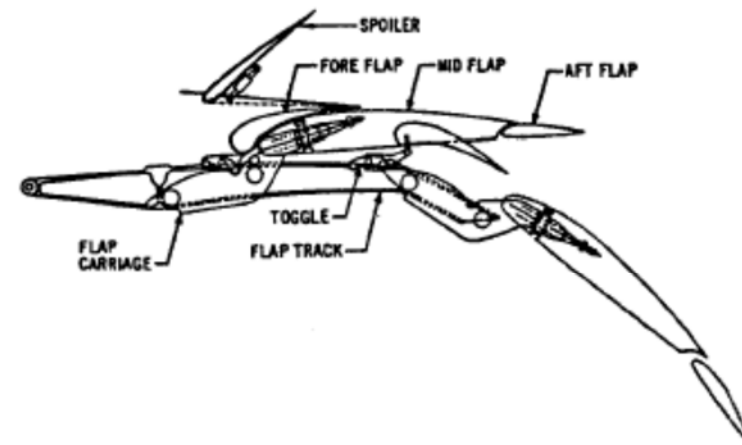


Fig. 7-25h. Triple slotted flap (Boeing 727)

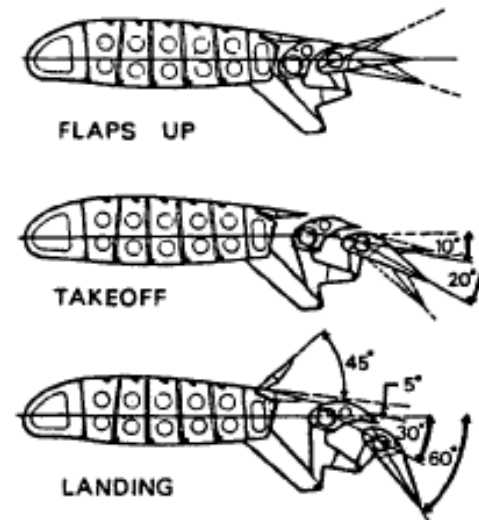


Fig. 7-25g. Double slotted flaps with individual adjustment of flap segments and drooped aileron (GAF N-22 Nomad)

# Superficies Hipersustentadoras - III

## Efectos HLD

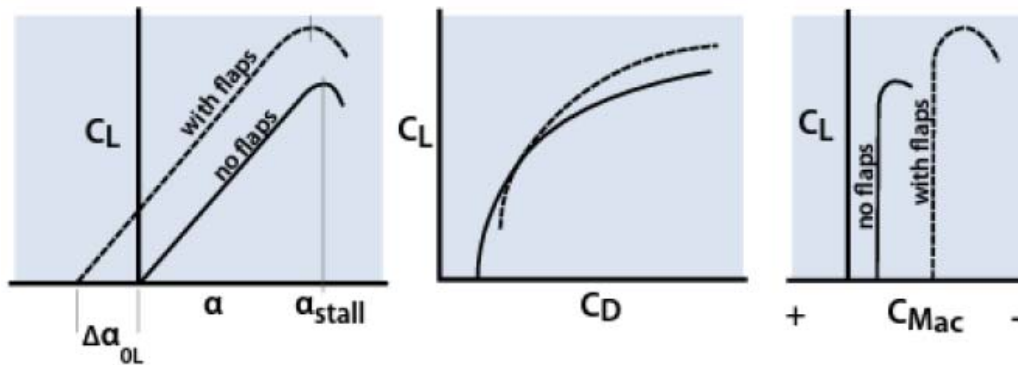


Figure 9.2 Characteristics of TE flaps.

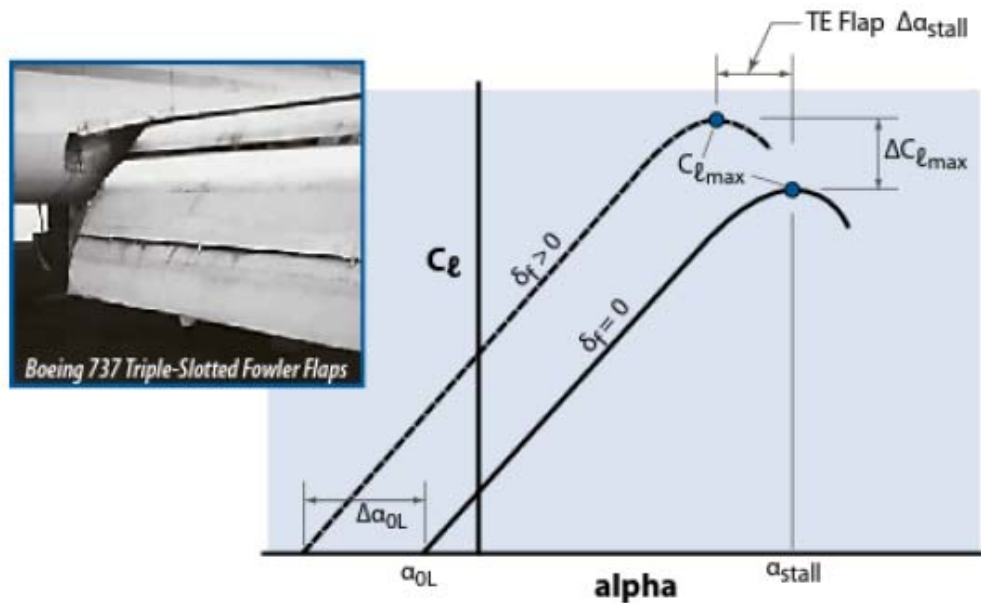
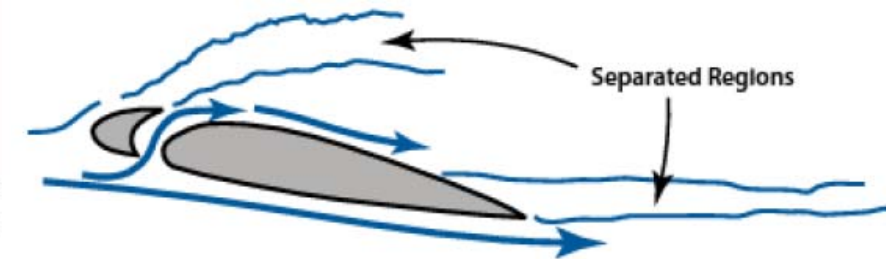


Figure 9.8 Construction of section lift curves for TE flaps.



A300 Leading Edge Slats Extended

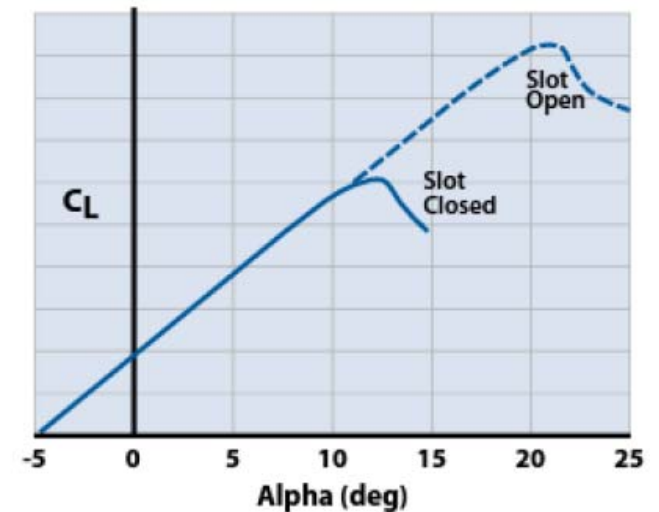
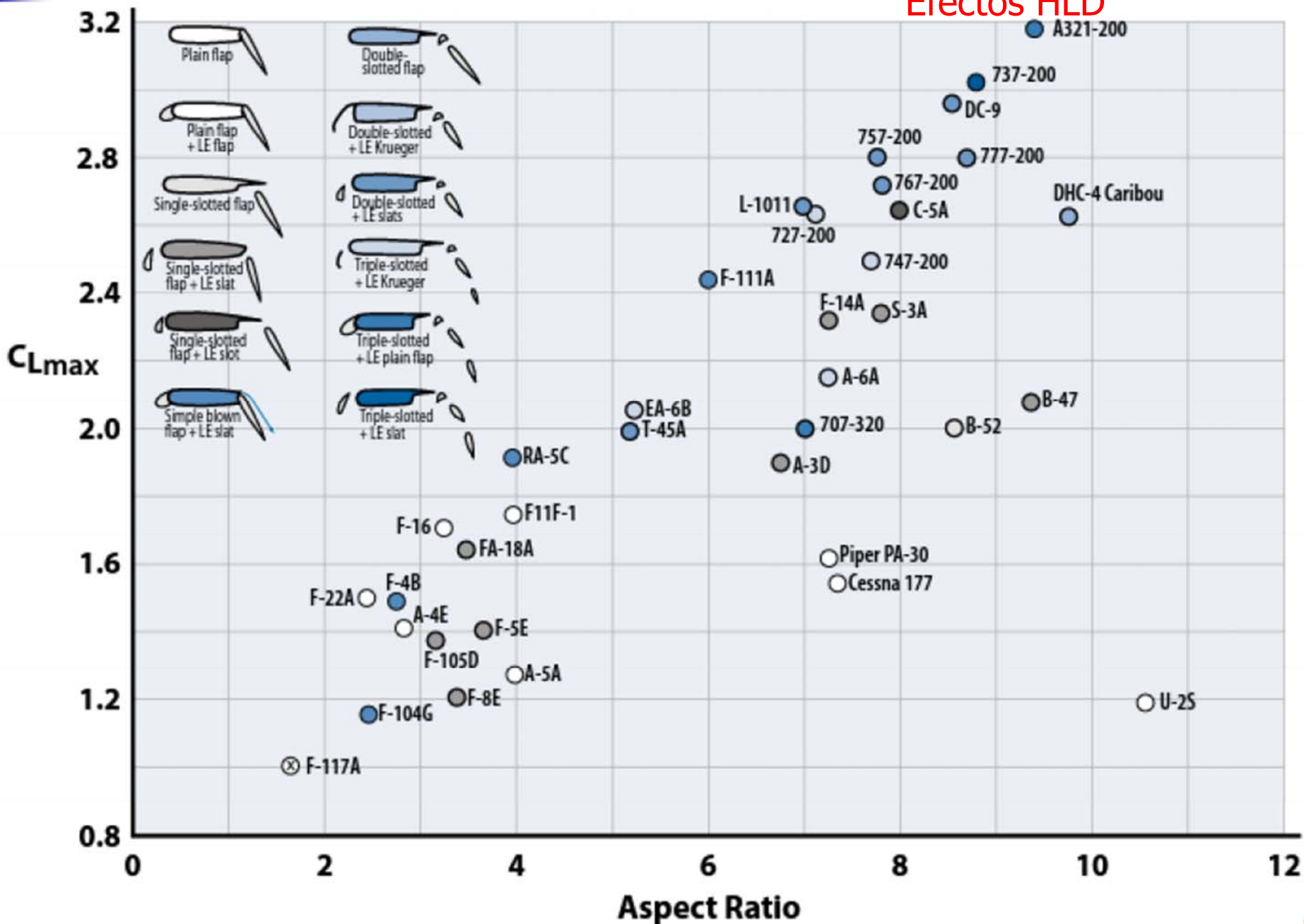


Figure 9.4 Characteristics of slots and slats.

# Superficies Hipersustentadoras - IV

Efectos HLD

Figure 9.7 Practical low-speed  $C_{Lmax}$  limits for mechanical high-lift systems (data from [7]).



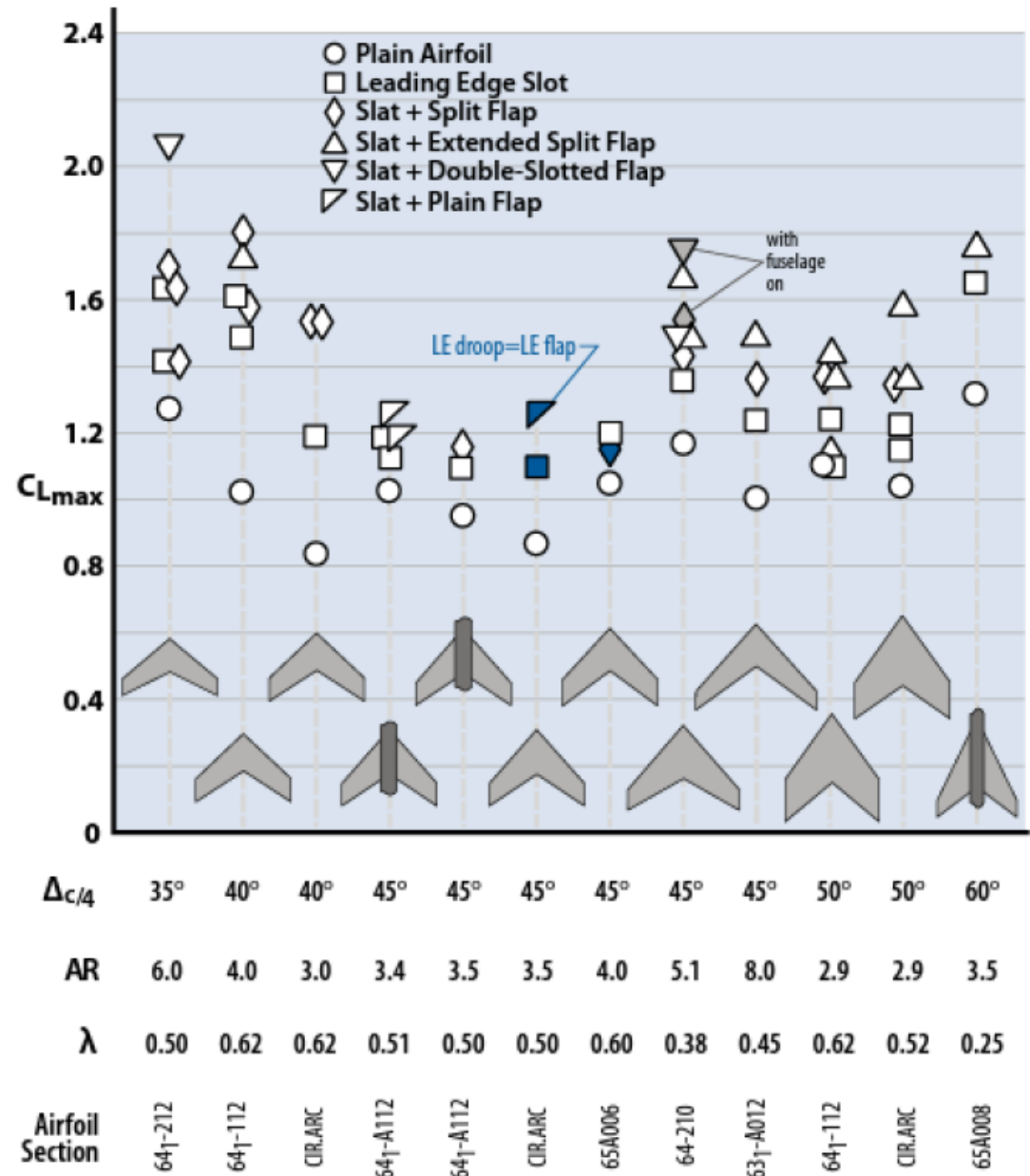
# Superficies Hipersustentadoras - V

Efectos HLD

**Table 9.3** Typical High-Lift Device Data

| $\Delta = 35 \text{ deg}, AR = 5.76, \lambda = 0.54$  |               |                  |
|-------------------------------------------------------|---------------|------------------|
| Airfoil section: 10% symmetrical                      |               |                  |
| Arrangement                                           | $C_{L_{max}}$ | $\alpha_{stall}$ |
| Plain wing                                            | 0.90          | 16               |
| 20% full-span split flap, $\delta f = 60$             | 1.45          | 10.6             |
| 20% full-span slat                                    | 1.38          | 23.6             |
| 20% full-span LE flap                                 | 1.49          | 26.5             |
| 20% full-span split flap + 20% full-span LE flap      | 2.01          | 19.7             |
| $\Delta = 0 \text{ deg}, AR = 4.0, \lambda = 1.0$     |               |                  |
| $Re = 10^5$                                           |               |                  |
| Airfoil section: NACA 0010                            |               |                  |
| Arrangement                                           | $C_{L_{max}}$ | $\alpha_{stall}$ |
| Plain wing                                            | 0.80          | 13               |
| 30% full-span split flap, $\delta f = 40 \text{ deg}$ | 1.52          | 10               |
| 20% full-span slat                                    | 1.36          | 24               |

**Table 9.2** Summary of Maximum Lift Coefficient Obtained with Various Types of High-lift Devices (data from [5,8,9])



# Superficies Hipersustentadoras - VI

## Efectos HLD

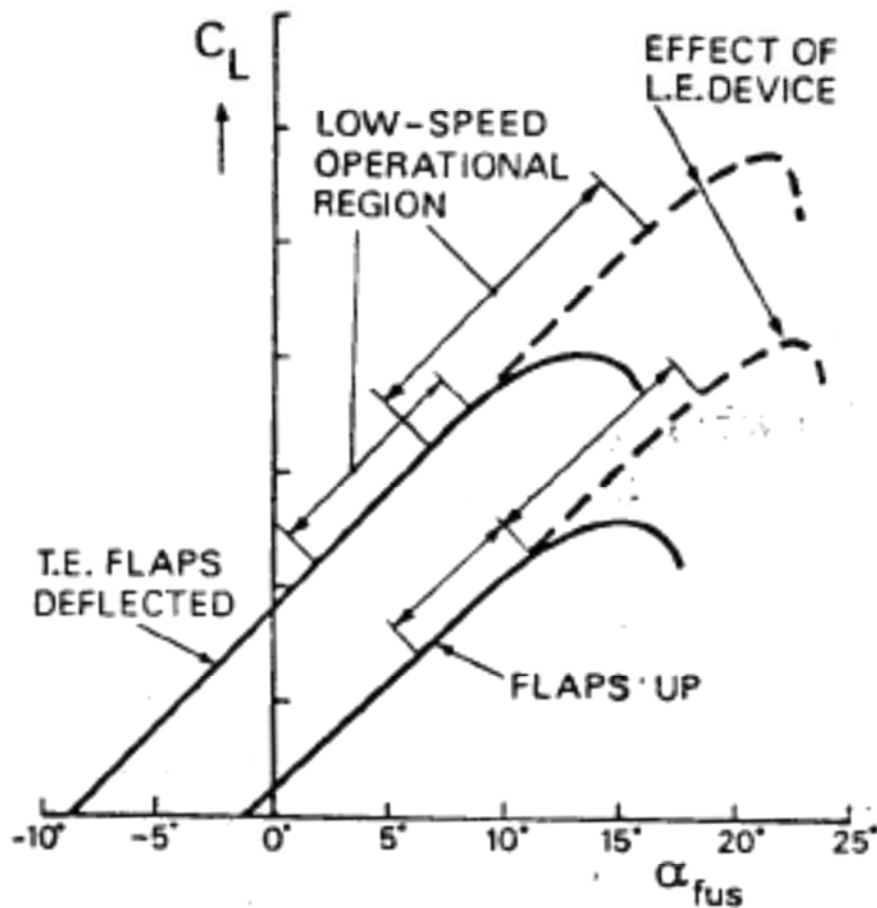


Fig. 7-23. Lift curves with and without high-lift devices.

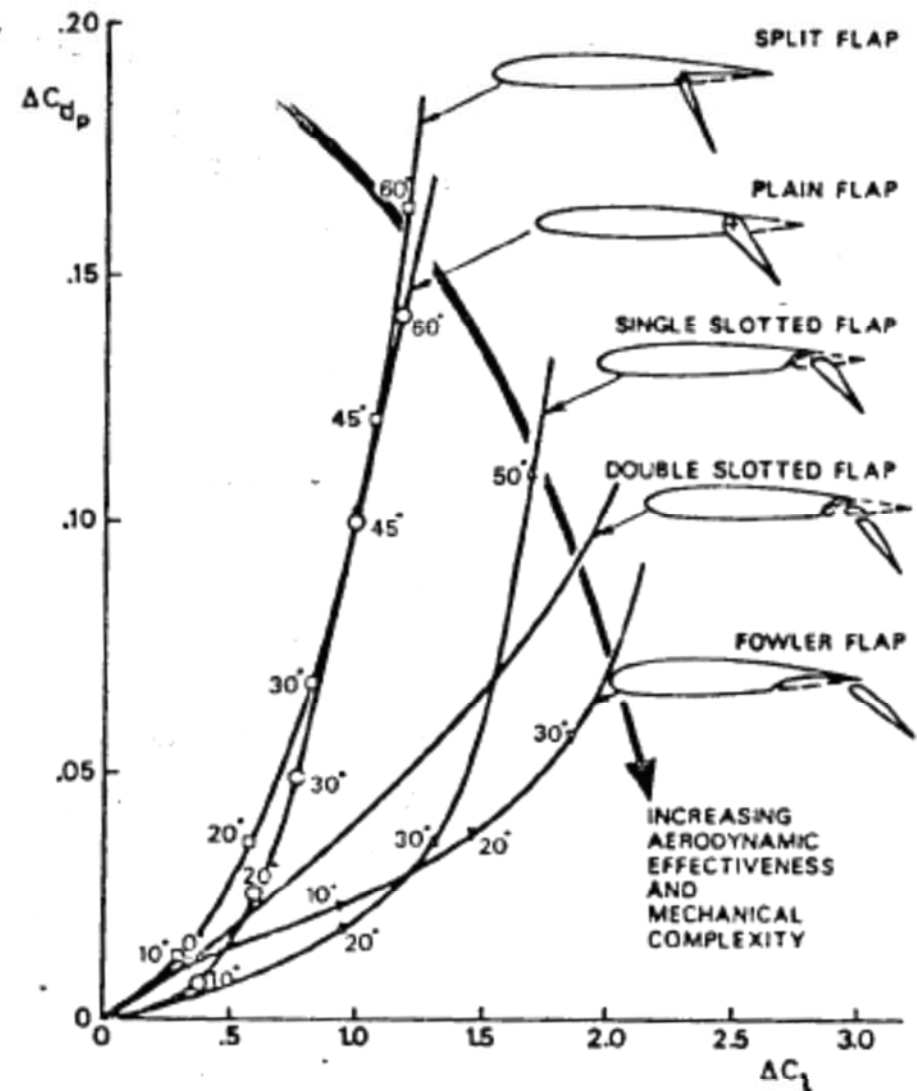
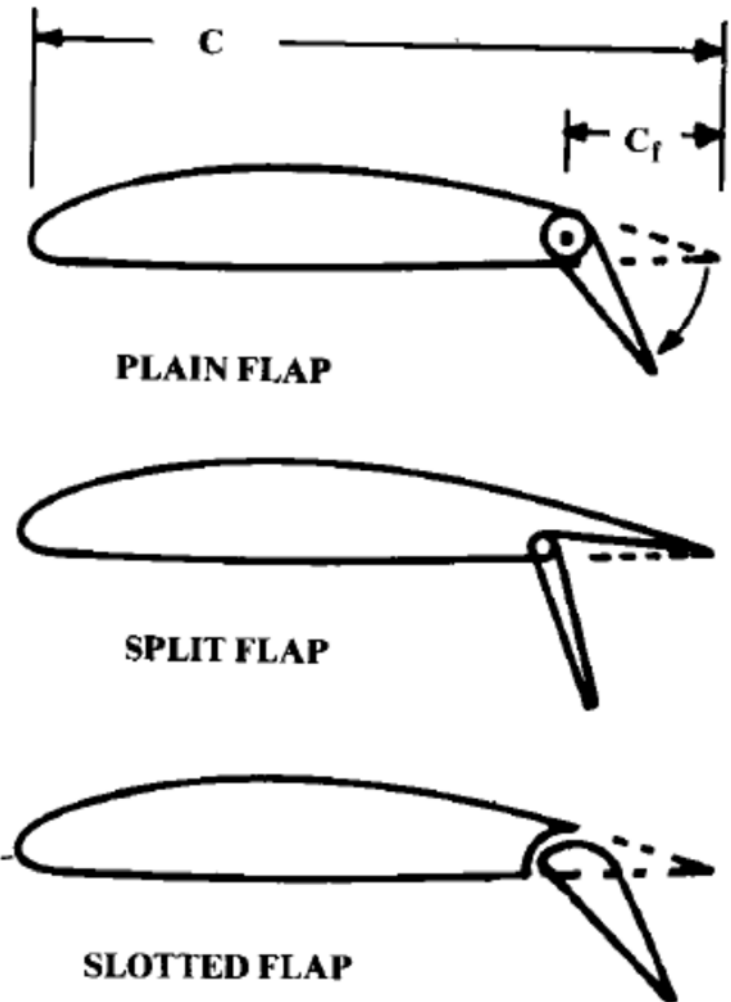
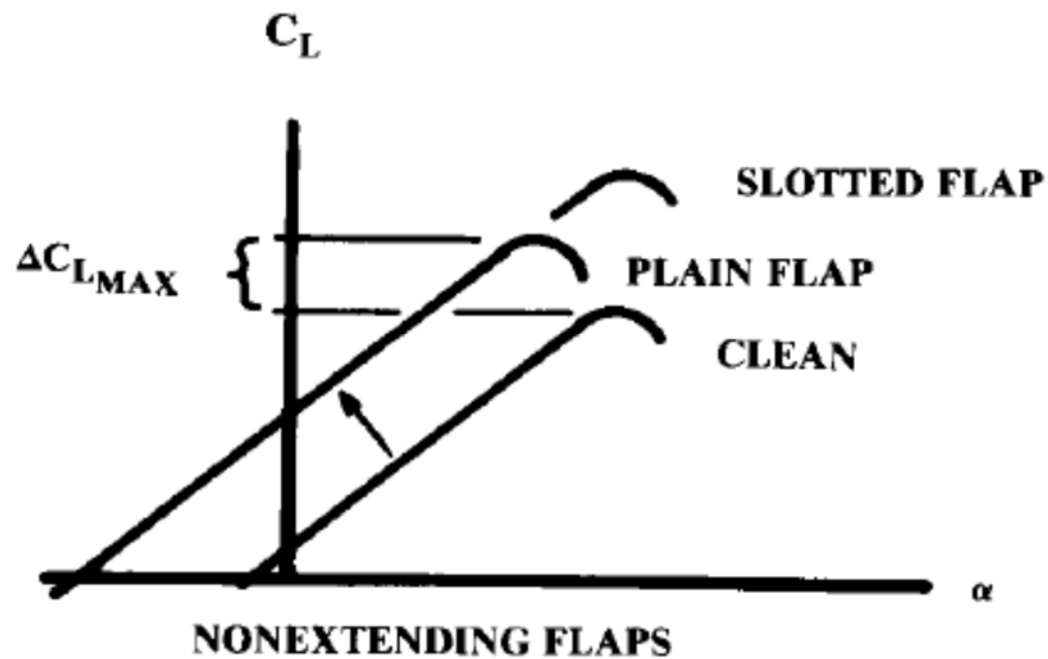


Fig. 7-24. Trends in performance of trailing-edge flaps.

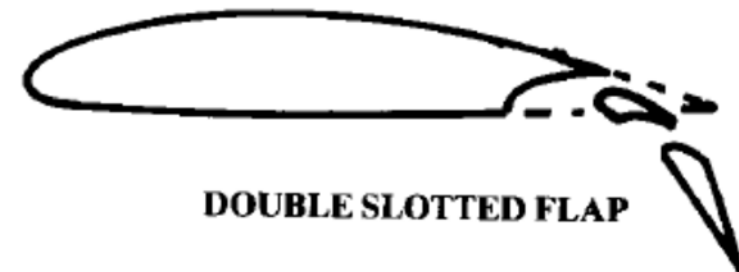
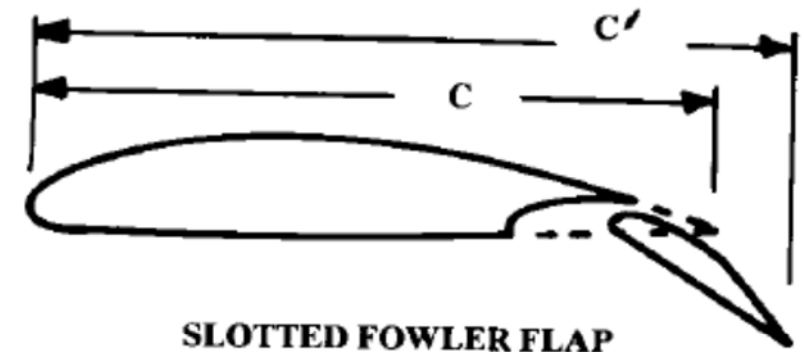
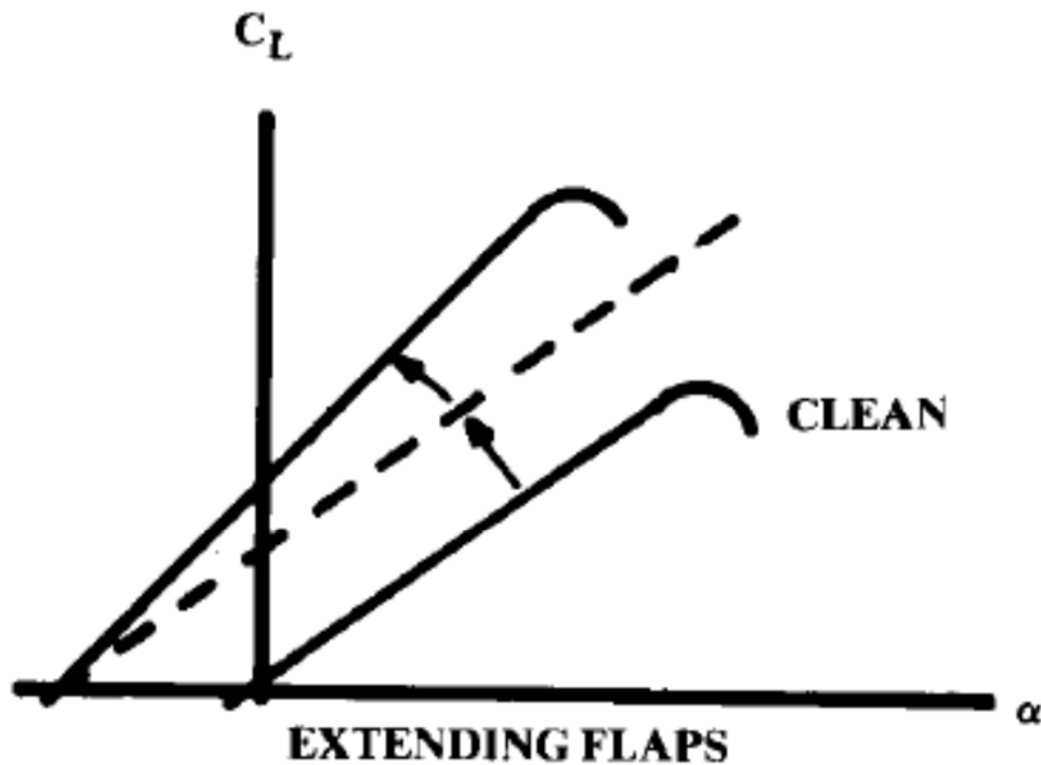
# Superficies Hipersustentadoras - VII

Efectos HLD



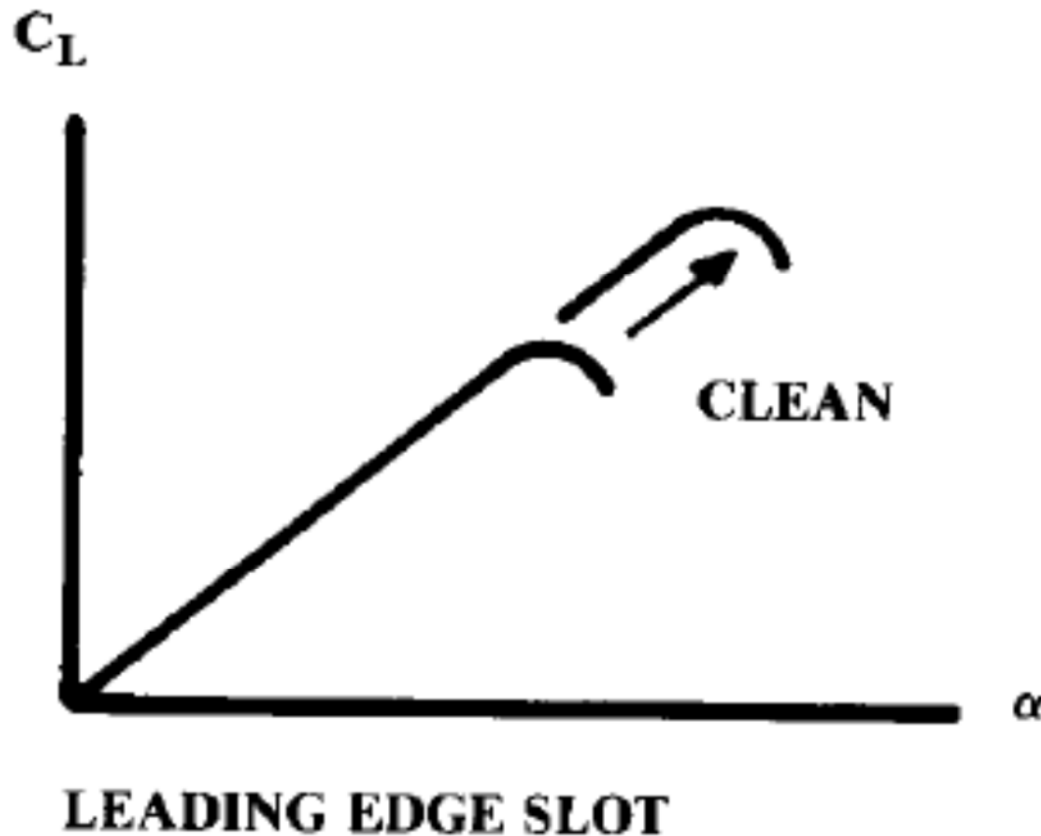
# Superficies Hipersustentadoras - VIII

Efectos HLD

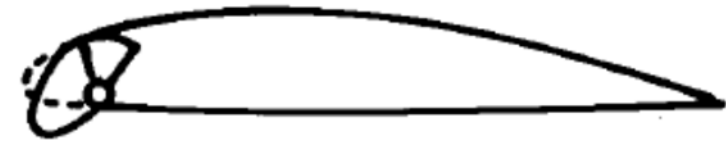


# Superficies Hipersustentadoras - IX

Efectos HLD



LEADING EDGE SLOT



LEADING EDGE FLAP



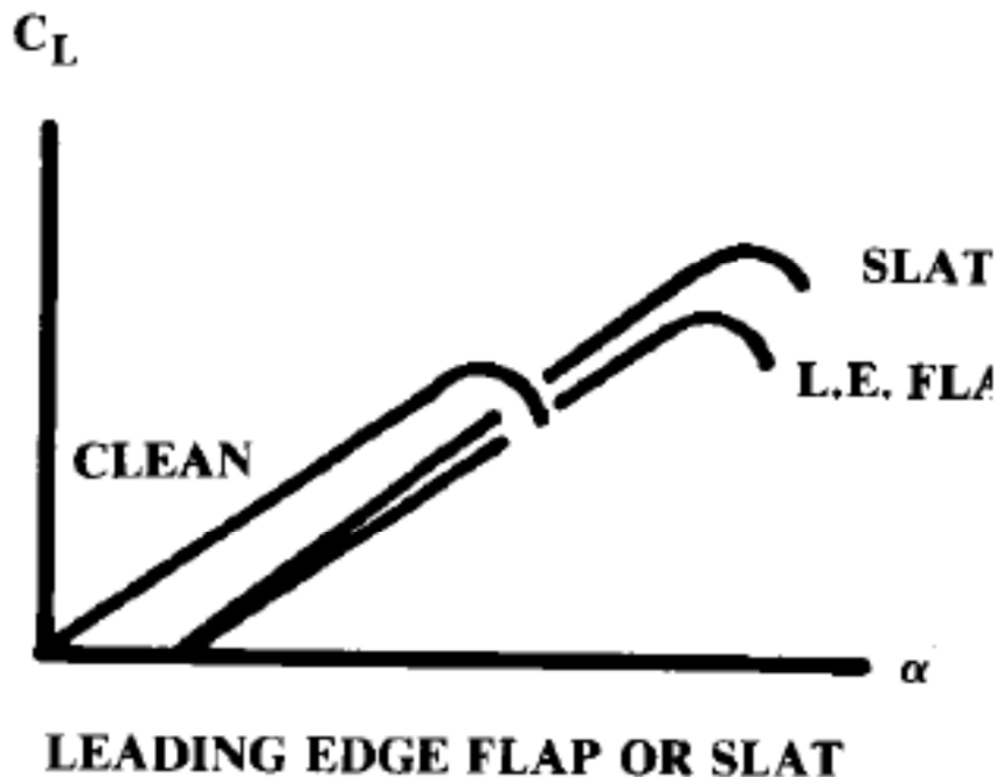
SLOTTED LEADING EDGE  
FLAP (SLAT)



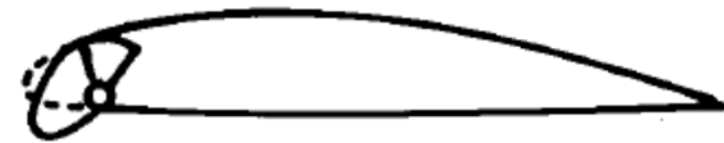
KRUGER FLAP

# Superficies Hipersustentadoras - X

Efectos HLD



LEADING EDGE SLOT



LEADING EDGE FLAP



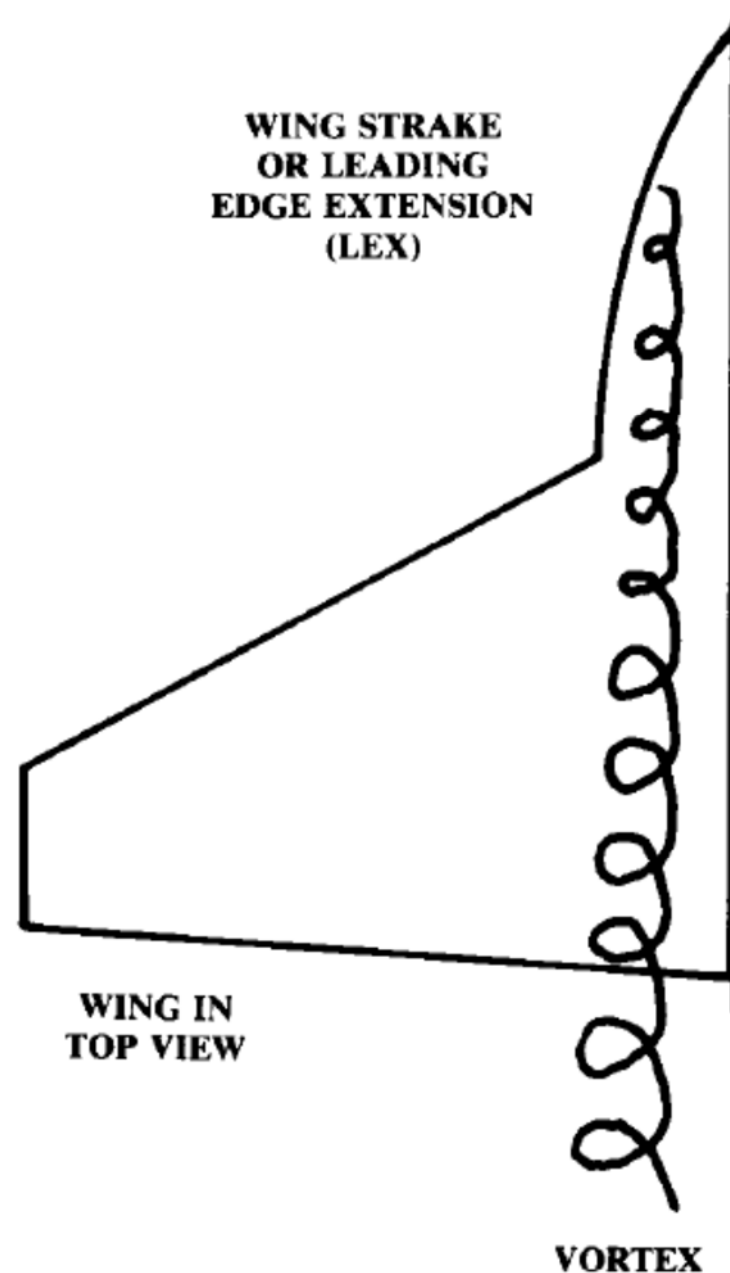
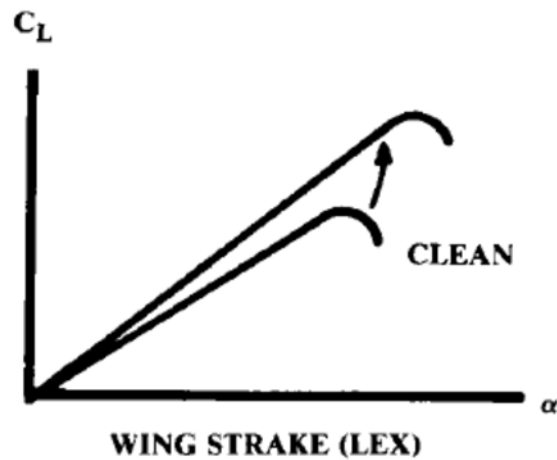
SLOTTED LEADING EDGE  
FLAP (SLAT)



KRUGER FLAP

# Superficies Hipersustentadoras - XI

Efectos HLD



Dryden Flight Research Center EC89-0096-149 Photographed 1990  
F-18 HARV flow-visualization smoke marks vortex flows  
along the leading edge extension. NASA photo



Historic Photographs / Historische Fotografien

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# Estimación $C_{Lmax}$ - XFLR5

The screenshot shows a Blackboard Learn web interface. At the top, there's a navigation bar with the user's name 'SERGIO ESTEBAN RONCERO' and links for 'Mis lugares', 'Inicio', 'Ayuda', and 'Cerrar sesión'. Below this is a sidebar with 'Mi institución' and 'Cursos'. The main content area is titled 'Cálculo de Aeronaves-Grado en Ingeniería Aeroespacial' and 'Tablero de discusión'. The forum is titled 'Foro: Frequently Asked Questions - Aerodinámica'. It contains a table of forum posts with columns for 'Fecha', 'Secuencia', 'Autor', 'Estado', 'Etiquetas', 'Publicaciones no leídas', and 'Total de publicaciones'. The table lists six posts, all by 'Anónimo' and in 'Publicada' state. The last post is selected. At the bottom right, there's an 'ACEPTAR' button.

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| <input type="checkbox"/>            | 3/12/13 21:26  | <a href="#">Uso de XFLR5 para determinar momentos de cabeceo</a>                                    | Anónimo | Publicada | 1         | 1                       |                        |
| <input checked="" type="checkbox"/> | 3/12/13 21:10  | <a href="#">Pautas estimación del CL máximo a partir de la inforamción de XFLR5</a>                 | Anónimo | Publicada | 0         | 1                       |                        |
| <input type="checkbox"/>            | 3/12/13 10:53  | <a href="#">Estimación del CLmax</a>                                                                | Anónimo | Publicada | 0         | 1                       |                        |
| <input type="checkbox"/>            | 21/11/13 1:01  | <a href="#">Pautas estimación entrada en pérdida XFLR5</a>                                          | Anónimo | Publicada | 1         | 1                       |                        |
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ACEPTAR

# Cálculo $C_{Lmax}$ - I

- En una primera iteración se pueden emplear métodos empíricos :
  - Es necesario primero determinar  $C_L$  vs.  $\alpha$  para el ala básica.
    - Después corregir para diferentes superficies hipersustentadoras.
  - Necesario determinar  $\Delta\alpha_{0L}$ ,  $\Delta C_{lmax}$  y  $\Delta\alpha_{STALL}$
  - Primer paso es obtener  $\alpha_{0L}$ ,  $C_{lmax}$  y  $\alpha_{STALL}$  del perfil:
    - Datos experimentales
      - [NACA Report 824 - Summary of airfoil data.](#)
      - Theory of Wing Sections, by Abbott.
    - Datos empíricos
  - Una vez calculado para el perfil básico extender para HLD

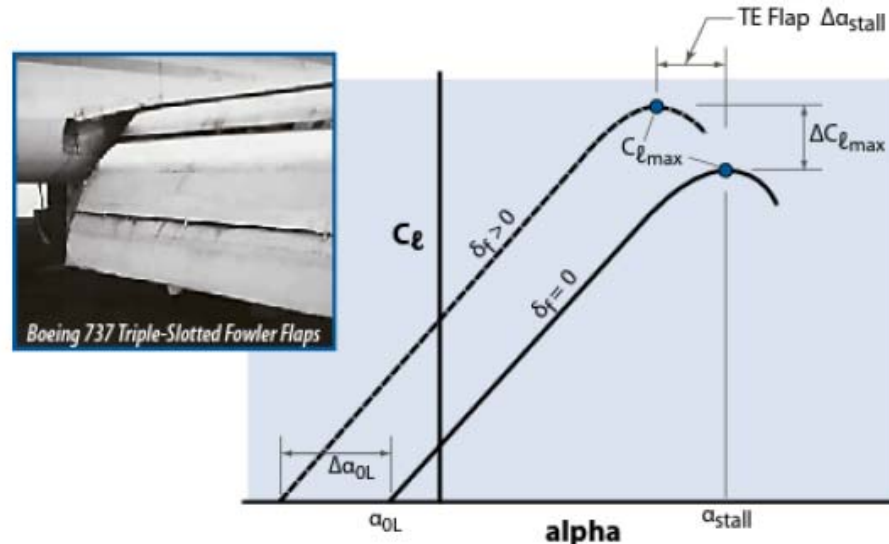


Figure 9.8 Construction of section lift curves for TE flaps.

# Cálculo $C_{Lmax}$ - I

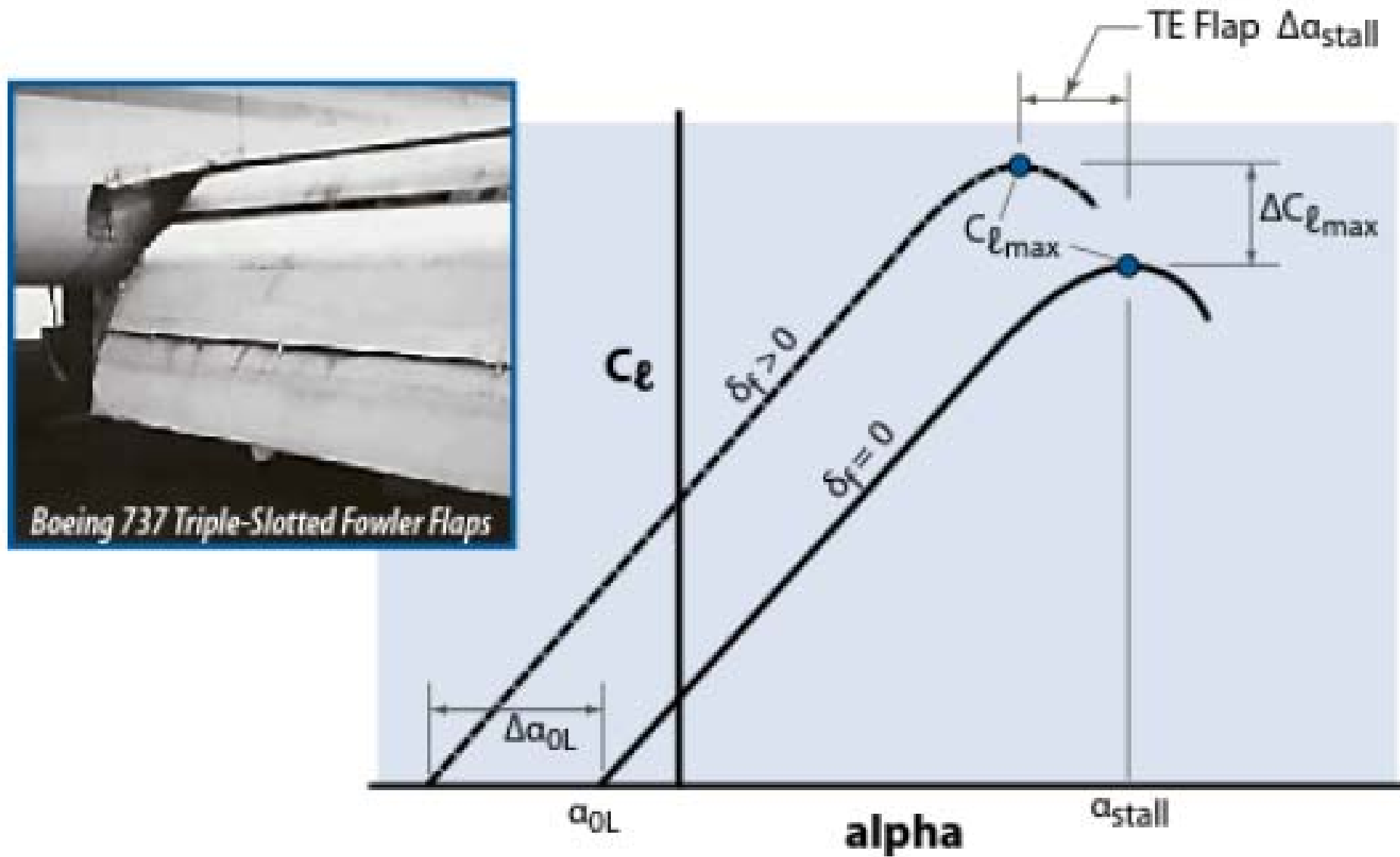


Figure 9.8 Construction of section lift curves for TE flaps.

# Cálculo $C_{Lmax}$ - Datos Experimentales - I

**Table F.1** Low-Speed Data on Airfoil Sections,  $Re = 9 \times 10^6$  [1]

| Airfoil | $\alpha_{0L}$<br>(deg) | $C_{m0}$ | $C_{\ell_{\alpha}}$<br>(per deg) | a.c.  | $\alpha_{C_{\ell_{max}}}$<br>(per deg) | $C_{\ell_{max}}$ | $\alpha^{*a}$<br>(deg) |
|---------|------------------------|----------|----------------------------------|-------|----------------------------------------|------------------|------------------------|
| 0006    | 0                      | 0        | 0.108                            | 0.250 | 9.0                                    | 0.92             | 9.0                    |
| 0009    | 0                      | 0        | 0.109                            | 0.250 | 13.4                                   | 1.32             | 11.4                   |
| 1408    | -0.8                   | -0.023   | 0.109                            | 0.250 | 14.0                                   | 1.35             | 10.0                   |
| 1410    | -1.0                   | -0.020   | 0.108                            | 0.247 | 14.3                                   | 1.50             | 11.0                   |
| 1412    | -1.1                   | -0.025   | 0.108                            | 0.252 | 15.2                                   | 1.58             | 12.0                   |
| 2412    | -2.0                   | -0.047   | 0.105                            | 0.247 | 16.8                                   | 1.68             | 9.5                    |
| 2415    | -2.0                   | -0.049   | 0.106                            | 0.246 | 16.4                                   | 1.63             | 10.0                   |
| 2418    | -2.3                   | -0.050   | 0.103                            | 0.241 | 14.0                                   | 1.47             | 10.0                   |
| 2421    | -1.8                   | -0.040   | 0.103                            | 0.241 | 16.0                                   | 1.47             | 8.0                    |
| 2424    | -1.8                   | -0.040   | 0.098                            | 0.231 | 16.0                                   | 1.29             | 8.4                    |
| 4412    | -3.8                   | -0.093   | 0.105                            | 0.247 | 14.0                                   | 1.67             | 7.5                    |
| 4415    | -4.3                   | -0.093   | 0.105                            | 0.245 | 15.0                                   | 1.64             | 8.0                    |
| 4418    | -3.8                   | -0.088   | 0.105                            | 0.242 | 14.0                                   | 1.53             | 7.2                    |
| 4421    | -3.8                   | -0.085   | 0.103                            | 0.238 | 16.0                                   | 1.47             | 6.0                    |
| 4424    | -3.8                   | -0.082   | 0.100                            | 0.239 | 16.0                                   | 1.38             | 4.8                    |
| 23012   | -1.4                   | -0.014   | 0.107                            | 0.247 | 18.0                                   | 1.79             | 12.0                   |
| 23015   | -1.0                   | -0.007   | 0.107                            | 0.243 | 18.0                                   | 1.72             | 10.0                   |
| 23018   | -1.2                   | -0.005   | 0.104                            | 0.243 | 16.0                                   | 1.60             | 11.8                   |
| 23021   | -1.2                   | 0        | 0.103                            | 0.238 | 15.0                                   | 1.50             | 10.3                   |
| 23024   | -0.8                   | 0        | 0.097                            | 0.231 | 15.0                                   | 1.40             | 9.7                    |
| 63-006  | 0                      | 0.005    | 0.112                            | 0.258 | 10.0                                   | 0.87             | 7.7                    |
| -009    | 0                      | 0        | 0.111                            | 0.258 | 11.0                                   | 1.15             | 10.7                   |
| 63-206  | -1.9                   | -0.037   | 0.112                            | 0.254 | 10.5                                   | 1.06             | 6.0                    |
| -209    | -1.4                   | -0.032   | 0.110                            | 0.262 | 12.0                                   | 1.4              | 10.8                   |
| -210    | -1.2                   | -0.035   | 0.113                            | 0.261 | 14.5                                   | 1.56             | 9.6                    |
| 63-012  | 0                      | 0        | 0.116                            | 0.265 | 14.0                                   | 1.45             | 12.8                   |
| -212    | -2.0                   | -0.035   | 0.114                            | 0.263 | 14.5                                   | 1.63             | 11.4                   |
| -412    | -2.8                   | -0.075   | 0.117                            | 0.271 | 15.                                    | 0                | 1.77                   |
| 63-015  | 0                      | 0        | 0.117                            | 0.271 | 14.5                                   | 1.47             | 11.0                   |
| -215    | -1.0                   | -0.030   | 0.116                            | 0.267 | 15.0                                   | 1.60             | 8.8                    |
| -415    | -2.8                   | -0.069   | 0.118                            | 0.262 | 15.0                                   | 1.68             | 10.0                   |
| -615    | -3.6                   | -0.108   | 0.117                            | 0.266 | 15.0                                   | 1.67             | 8.6                    |
| 63-018  | 0                      | 0        | 0.118                            | 0.271 | 15.5                                   | 1.54             | 11.2                   |
| -218    | -1.4                   | -0.033   | 0.118                            | 0.271 | 14.5                                   | 1.85             | 8.0                    |
| -418    | -2.7                   | -0.064   | 0.118                            | 0.272 | 16.0                                   | 1.57             | 7.0                    |
| -618    | -3.8                   | -0.097   | 0.118                            | 0.267 | 16.0                                   | 1.59             | 4.2                    |

| Airfoil                 | $\alpha_{0L}$<br>(deg) | $C_{m0}$ | $C_{\ell_{\alpha}}$<br>(per deg) | a.c.  | $\alpha_{C_{\ell_{max}}}$<br>(per deg) | $C_{\ell_{max}}$ | $\alpha^{*a}$<br>(deg) |
|-------------------------|------------------------|----------|----------------------------------|-------|----------------------------------------|------------------|------------------------|
| 63-021                  | 0                      | 0        | 0.118                            | 0.273 | 17.0                                   | 1.38             | 9.0                    |
| -221                    | -1.5                   | -0.035   | 0.118                            | 0.269 | 15.0                                   | 1.44             | 9.2                    |
| -421                    | -2.8                   | -0.062   | 0.120                            | 0.275 | 16.0                                   | 1.48             | 6.7                    |
| 63.4-420                | -2.2                   | -0.059   | 0.109                            | 0.265 | 14.0                                   | 1.42             | 7.6                    |
| 63.4-420 <sub>0-3</sub> | -2.4                   | -0.037   | 0.111                            | 0.265 | 16.0                                   | 1.35             | 6.0                    |
| 63(420)-422             | -3.2                   | -0.065   | 0.112                            | 0.271 | 14.0                                   | 1.36             | 6.0                    |
| 63(420)-517             | -3.0                   | -0.084   | 0.108                            | 0.264 | 15.0                                   | 1.60             | 8.0                    |
| 64-006                  | 0                      | 0        | 0.109                            | 0.256 | 9.0                                    | 0.8              | 7.2                    |
| -009                    | 0                      | 0        | 0.110                            | 0.262 | 11.0                                   | 1.17             | 10.0                   |
| 64-108                  | 0                      | -0.015   | 0.110                            | 0.255 | 10.0                                   | 1.1              | 10.0                   |
| -110                    | -1.0                   | -0.020   | 0.110                            | 0.261 | 13.0                                   | 1.4              | 10.0                   |
| 64-206                  | -1.0                   | -0.040   | 0.110                            | 0.253 | 12.0                                   | 1.03             | 8.0                    |
| -208                    | -1.2                   | -0.039   | 0.113                            | 0.257 | 10.5                                   | 1.23             | 8.8                    |
| -209                    | -1.5                   | -0.040   | 0.107                            | 0.261 | 13.0                                   | 1.40             | 8.9                    |
| -210                    | -1.6                   | -0.040   | 0.110                            | 0.258 | 14.0                                   | 1.45             | 10.8                   |
| 64-012                  | 0                      | 0        | 0.111                            | 0.262 | 14.5                                   | 1.45             | 11.0                   |
| -112                    | -0.8                   | -0.017   | 0.113                            | 0.267 | 14.0                                   | 1.50             | 12.2                   |
| 212                     | -1.3                   | -0.027   | 0.113                            | 0.262 | 15.0                                   | 1.55             | 11.0                   |
| -412                    | -2.6                   | -0.065   | 0.112                            | 0.267 | 15.0                                   | 1.67             | 8.0                    |
| 64-015                  | 0                      | 0        | 0.112                            | 0.267 | 15.0                                   | 1.48             | 13.0                   |
| -215                    | -1.6                   | -0.030   | 0.112                            | 0.265 | 15.0                                   | 1.57             | 10.0                   |
| -415                    | -2.8                   | -0.070   | 0.115                            | 0.264 | 15.0                                   | 1.65             | 8.0                    |
| 64-018                  | 0                      | 0.004    | 0.111                            | 0.266 | 17.0                                   | 1.50             | 12.0                   |
| -218                    | -1.3                   | -0.027   | 0.115                            | 0.271 | 16.0                                   | 1.53             | 10.0                   |
| -418                    | -2.9                   | -0.065   | 0.116                            | 0.273 | 14.0                                   | 1.57             | 8.0                    |
| -618                    | -3.8                   | -0.095   | 0.116                            | 0.273 | 16.0                                   | 1.58             | 5.6                    |
| 64-021                  | +0.005                 | -0.029   | 0.110                            | 0.274 | 14.0                                   | 1.30             | 10.3                   |
| -221                    | -1.2                   | -0.029   | 0.117                            | 0.271 | 13.0                                   | 1.32             | 6.8                    |
| -421                    | -2.8                   | -0.068   | 0.120                            | 0.276 | 13.0                                   | 1.42             | 6.4                    |
| 65-006                  | 0                      | 0        | 0.105                            | 0.258 | 12.0                                   | 0.92             | 7.6                    |
| -009                    | 0                      | 0        | 0.107                            | 0.264 | 11.0                                   | 1.08             | 9.8                    |
| 65-206                  | -1.6                   | -0.031   | 0.105                            | 0.257 | 12.0                                   | 1.03             | 6.0                    |
| -209                    | -1.2                   | -0.031   | 0.106                            | 0.259 | 12.0                                   | 1.30             | 10.0                   |
| -210                    | -1.6                   | -0.034   | 0.108                            | 0.262 | 13.0                                   | 1.40             | 9.6                    |
| 65-410                  | -2.5                   | -0.067   | 0.112                            | 0.262 | 14.0                                   | 1.52             | 8.0                    |
| 65-012                  | 0                      | 0        | 0.110                            | 0.261 | 14.0                                   | 1.36             | 10.0                   |
| -212                    | -1.0                   | -0.032   | 0.108                            | 0.261 | 14.0                                   | 1.47             | 9.4                    |

(continued)

# Cálculo $C_{Lmax}$ - Datos Experimentales - II

| Airfoil                    | $\alpha_{0L}$<br>(deg) | $C_{m0}$ | $C_{L\alpha}$<br>(per deg) | a.c.  | $\alpha_{C_{Lmax}}$<br>(per deg) | $C_{Lmax}$ | $\alpha^{*a}$<br>(deg) |
|----------------------------|------------------------|----------|----------------------------|-------|----------------------------------|------------|------------------------|
| -212 <sub>0-6</sub>        | -1.4                   | -0.033   | 0.108                      | 0.269 | 14.0                             | 1.50       | 9.6                    |
| -412                       | -3.0                   | -0.070   | 0.111                      | 0.265 | 15.5                             | 1.66       | 10.5                   |
| 65-015                     | 0                      | 0        | 0.111                      | 0.257 | 15.0                             | 1.42       | 11.2                   |
| -215                       | -1.2                   | -0.032   | 0.112                      | 0.269 | 15.5                             | 1.53       | 10.0                   |
| -415                       | -2.6                   | -0.060   | 0.111                      | 0.268 | 16.0                             | 1.61       | 8.7                    |
| -415 <sub>0-5</sub>        | -2.6                   | -0.051   | 0.111                      | 0.264 | 20.0                             | 1.60       | 7.0                    |
| 65(215)-114                | -0.7                   | -0.019   | 0.112                      | 0.265 | 15.0                             | 1.44       | 10.5                   |
| 65(216)-415 <sub>0-5</sub> | -3.0                   | -0.057   | 0.106                      | 0.267 | 18.0                             | 1.60       | 6.0                    |
| 65-3-018                   | 0                      | 0        | 0.100                      | 0.262 | 17.0                             | 1.44       | 10.0                   |
| -418 <sub>0-8</sub>        | -3.0                   | -0.081   | 0.112                      | 0.266 | 20.0                             | 1.58       | 4.4                    |
| -618                       | -4.0                   | -0.100   | 0.110                      | 0.273 | 20.0                             | 1.60       | 4.9                    |
| 65-018                     | 0                      | 0        | 0.100                      | 0.267 | 16.0                             | 1.37       | 10.0                   |
| -218                       | -1.2                   | -0.030   | 0.100                      | 0.263 | 18.0                             | 1.48       | 8.8                    |
| -418                       | -2.4                   | -0.059   | 0.110                      | 0.265 | 18.0                             | 1.54       | 4.9                    |
| -418 <sub>0-5</sub>        | -2.8                   | -0.055   | 0.115                      | 0.267 | 18.0                             | 1.50       | 6.0                    |
| -618                       | -4.0                   | -0.102   | 0.113                      | 0.276 | 18.0                             | 1.64       | 5.2                    |
| -618 <sub>0-5</sub>        | -4.2                   | -0.078   | 0.104                      | 0.265 | 20.0                             | 1.51       | 5.3                    |
| 65-021                     | 0                      | 0        | 0.112                      | 0.267 | 18.5                             | 1.40       | 7.4                    |
| -221                       | -1.3                   | -0.029   | 0.115                      | 0.274 | 20.5                             | 1.46       | 6.0                    |
| -421                       | -2.8                   | -0.066   | 0.116                      | 0.272 | 22.0                             | 1.56       | 5.0                    |
| -421 <sub>0-5</sub>        | -2.8                   | -0.052   | 0.116                      | 0.272 | 20.0                             | 1.43       | 5.6                    |
| 65(421)-420                | -2.4                   | -0.061   | 0.116                      | 0.276 | 20.0                             | 1.52       | 4.7                    |
| 66-006                     | 0                      | 0        | 0.100                      | 0.252 | 9.0                              | 0.80       | 6.5                    |
| -009                       | 0                      | 0        | 0.103                      | 0.259 | 10.0                             | 1.05       | 10.0                   |
| 66-206                     | -1.6                   | -0.038   | 0.108                      | 0.257 | 10.5                             | 1.00       | 7.0                    |
| -209                       | -1.0                   | -0.034   | 0.107                      | 0.257 | 11.0                             | 1.17       | 9.0                    |
| -210                       | -1.3                   | -0.035   | 0.110                      | 0.261 | 11.0                             | 1.27       | 10.0                   |
| 66-012                     | 0                      | 0        | 0.106                      | 0.258 | 14.0                             | 1.25       | 11.2                   |
| -212                       | -1.2                   | -0.032   | 0.102                      | 0.259 | 15.0                             | 1.46       | 11.6                   |
| 66-015                     | 0                      | 0.005    | 0.105                      | 0.265 | 15.5                             | 1.35       | 12.0                   |
| -215                       | -1.3                   | -0.031   | 0.106                      | 0.260 | 16.0                             | 1.50       | 11.4                   |
| -415                       | -2.6                   | -0.069   | 0.106                      | 0.260 | 17.0                             | 1.60       | 10.0                   |
| 66(215)-016                | 0                      | 0        | 0.105                      | 0.260 | 14.0                             | 1.33       | 10.0                   |
| -216                       | -2.0                   | -0.044   | 0.114                      | 0.262 | 16.0                             | 1.55       | 8.8                    |
| -216 <sub>0-6</sub>        | -1.2                   | -0.030   | 0.100                      | 0.257 | 16.0                             | 1.46       | 7.0                    |
| -416                       | -2.6                   | -0.068   | 0.100                      | 0.265 | 18.0                             | 1.60       | 4.0                    |
| 63A010                     | 0                      | 0.005    | 0.105                      | 0.254 | 13.0                             | 1.20       | 10.0                   |

| Airfoil | $\alpha_{0L}$<br>(deg) | $C_{m0}$ | $C_{L\alpha}$<br>(per deg) | a.c.  | $\alpha_{C_{Lmax}}$<br>(per deg) | $C_{Lmax}$ | $\alpha^{*a}$<br>(deg) |
|---------|------------------------|----------|----------------------------|-------|----------------------------------|------------|------------------------|
| 63A210  | -1.5                   | -0.040   | 0.103                      | 0.257 | 14.0                             | 1.43       | 10.0                   |
| 64A010  | 0                      | 0        | 0.110                      | 0.253 | 12.0                             | 1.23       | 10.0                   |
| 64A210  | -1.5                   | -0.040   | 0.105                      | 0.251 | 13.0                             | 1.44       | 10.0                   |
| 64A410  | -3.0                   | -0.080   | 0.100                      | 0.254 | 15.0                             | 1.61       | 10.0                   |
| 64A212  | -2.0                   | -0.040   | 0.100                      | 0.252 | 14.0                             | 1.54       | 11.0                   |
| 64A215  | -2.0                   | -0.040   | 0.095                      | 0.252 | 15.0                             | 1.50       | 12.0                   |

$\alpha^*$  = angle of attack at which lift curve ceases to be linear (incipient stall).  
Note:  $C_{m0}$  is about the aerodynamic center (a.c.).

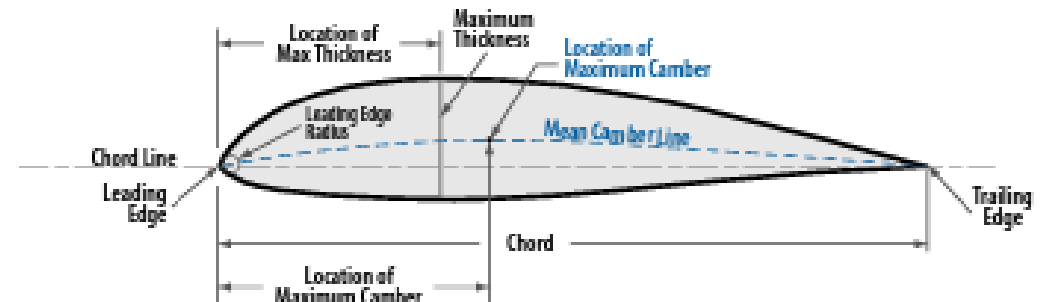


Figure F.1 Airfoil terminology.

**NACA Report 824 - Summary of airfoil data**

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# Companies Datos Experimentales - III

| Company        | Aircraft           | Airfoil (Root)  |
|----------------|--------------------|-----------------|
| AAI            | RQ-2 Pioneer       | NACA 4415       |
| AAI            | Shadow 200/400     | NACA 4415       |
| AerMacchi      | MB. 339            | NACA 64A114     |
| AerMacchi      | MC. 202/205        | NACA 23018      |
| AerMacchi      | MC. 72             | Blaconex        |
| AerMacchi      | SF-260             | NACA 64-212     |
| Aero Commander | Shrike 500/600/720 | NACA 23012      |
| Aero           | L-139/39/59        | NACA 64A012     |
| AeroVironment  | Gossamer Condor    | Lissaman 7769   |
| AeroVironment  | Helios             | Selig S6078     |
| AeroVironment  | Pathfinder         | Liebeck LA2573A |
| Airbus         | A300B              | 15%             |
| Avro           | Vulcan             | NACA 0010 mod   |
| Avro           | CF-105 Arrow       | NACA 0003.5 mod |
| Bede           | BD-4/6/7           | NACA 64-415     |
| Bede           | BD-5               | NACA 64-212     |
| Beech          | Bonanza/Lightning  | NACA 23016.5    |
| Beech          | King Air           | NACA 23016.5    |
| Beech          | Queen Air          | NACA 23018      |

| Company           | Aircraft        | Airfoil (Root)     |
|-------------------|-----------------|--------------------|
| Schleicher        | ASW 24          | Delft DU89-158     |
| Schleicher        | ASW 28          | Delft DU99-146     |
| Schleicher        | Ka-6 Rhonsegler | NACA 63-618        |
| Slingsby          | T.4 Falcon      | Goettingen 535 mod |
| Slingsby          | T.41 Skylark 2  | NACA 63-620        |
| Slingsby          | T.67 Firefly    | NACA 23015         |
| Stinson           | Reliant         | Clark Y            |
| Stoddard-Hamilton | Glasair         | NASA GA(W)-2       |
| Stoddard-Hamilton | Glastar         | NASA GA(W)-2       |
| Supermarine       | Spitfire        | NACA 2213          |
| Swearingen        | Queen Air 800   | NACA 23018         |
| Swearingen        | Merlin II/V     | NACA 65A215        |

| Company          | Aircraft                | Airfoil (Root)            |
|------------------|-------------------------|---------------------------|
| Bell             | P-39 Aircobra           | NACA 0015                 |
| Bell             | P-59 Aircomet           | NACA 66-014               |
| Bell             | P-63 Kingcobra          | NACA 66-116               |
| Bell             | X-5                     | NACA 64AD11               |
| Bellanca         | Citabria                | NACA 4412                 |
| Boeing           | B-17                    | NACA 0018                 |
| Boeing           | B-29                    | Boeing 117 (22%)          |
| Boeing           | 707/727/737/747/757     | Boeing airfoils (12%-15%) |
| Boeing           | Condor                  | Liebeck LD-17A            |
| Boeing           | C-17                    | DLBA 142                  |
| Boeing           | F-15 Eagle              | NACA 64A006.6             |
| Boeing           | F-18                    | NACA 65A005 mod           |
| Breguet          | 941                     | NACA 63A416               |
| BAE              | AV-8 Harrier            | Hawker 10%                |
| Cessna           | 150/152/172/180/182/206 | NACA 2412                 |
| Cessna           | 208 Caravan             | NACA 23017                |
| Cessna           | 310/L-27/U-3            | NACA 2412                 |
| Cessna           | T-37                    | NACA 23018                |
| Cessna           | Citation III/V          | NACA 23014                |
| Convair          | B-58 Hustler            | NACA 0003.46              |
| Convair          | F-102                   | Delta Dagger              |
| Convair          | F-106 Delta Dart        | NACA 0004-65 mod          |
| DeHavilland      | DH-106 Comet            | NACA 63A116 mod           |
| Douglas          | A-4 Skyhawk             | NACA 0008-1.1-25          |
| Douglas          | DC-3                    | NACA 2215                 |
| Douglas          | DC-4                    | NACA 23016                |
| Douglas          | DC-6                    | NACA 23016                |
| Douglas          | DC-8/DC-9/DC-10         | DSMA                      |
| Douglas          | D-558-II Skyrocket      | NACA 63-010               |
| Fairchild        | A-10 Thunderbolt II     | NACA6716                  |
| Ford             | Trimotor                | Goettingen 386            |
| General Atomics  | RQ-1 Predator           | A                         |
| General Atomics  | RQ-9                    | Predator B                |
| General Dynamics | F-111                   | NACA 64-210.6B            |
| Grumman          | SA-16/HU-16 Albatross   | NACA 23017                |
| Grumman          | E-2 Hawkeye             | NACA 63A216               |
| Grumman          | A-6 Intruder            | NACA 64A009 mod           |
| Grumman          | F-14 Tomcat             | NACA 64A209.65 mod        |
| Grumman          | F9F Cougar              | NACA 64AD10               |

| Company           | Aircraft                 | Airfoil (Root)           |
|-------------------|--------------------------|--------------------------|
| Gulfstream        | GI/GII/GIV               | NACA 0012 mod            |
| Hawker            | Hurricane                | Clark YH (19%)           |
| Hawker            | Typhoon                  | NACA 2219                |
| Hughes            | H-1 (Long Wing)          | NACA 23016.5             |
| Hughes            | H-1 (short wing)         | NACA 2418                |
| Hughes            | H-4 Hercules             | NACA 63(420)-321         |
| Learjet           | 23-60                    | NACA 64A109              |
| Leshner           | (Univ of Mich) Teal      | NACA 63A615              |
| Lockheed          | P-80 Shooting Star       | NACA 65-213              |
| Lockheed          | P-38 Lightning           | NACA 23016               |
| Lockheed          | F-94                     | NACA 65-213              |
| Lockheed          | P-3 Orion/Electra        | NACA 0014-1.1            |
| Lockheed          | U-2A/R/S                 | NACA 64A409              |
| Lockheed          | C-130                    | NACA 64A318              |
| Lockheed          | L-1011 Tristar           | Lockheed airfoil (12.4%) |
| Lockheed          | F-117 Nighthawk          | 3 flats upper, 2 lower   |
| Lockheed Martin   | F-16                     | NACA 64A204              |
| Lockheed Martin   | Falcon                   | LM airfoil               |
| McDonnell Douglas | F-4 Phantom              | NACA 0006.4-64 mod       |
| McDonnell         | F-101 Voodoo             | NACA 65A007 mod          |
| Messerschmitt     | Bf 109                   | NACA 2R1 14.2            |
| Messerschmitt     | Bf 110/161/162           | NACA 2R1 18.5            |
| Messerschmitt     | Me 209                   | NACA 2R1 16              |
| Messerschmitt     | Me 263                   | Me 1.8 25 14-1.1-30      |
| Mooney            | (All)                    | NACA 63-215              |
| North American    | P-51B/C/D Mustang        | NACA 45-100              |
| North American    | B-25 Mitchell            | NACA 23017               |
| North American    | F-86F                    | NACA 0009-64 mod         |
| North American    | F-100C/D                 | NACA 64A007              |
| North American    | T-39 Sabreliner (-40/60) | NACA 64A212              |
| Northrop          | B-2 Spirit               | Modified supercritical   |
| Northrop          | F-5 Tiger                | NACA 65A004.8            |
| Northrop          | F-20 Tigershark          | NACA 65A004.8            |
| Northrop          | T-38 Talon               | NACA 65A004.8            |
| Northrop          | Tacit Blue               | Clark Y mod              |
| Northrop Grumman  | RQ-4A Global Hawk        | NASA LRN 1015            |
| Pilatus           | PC-12                    | NASA LS(1)-0417          |
| Pilatus           | PC-6 Turbo Porter        | NACA 64-514              |

| Company       | Aircraft            | Airfoil (Root)            |
|---------------|---------------------|---------------------------|
| Pilatus       | PC-7 Turbo Trainer  | NACA 64A415               |
| Pilatus       | PC-8B Twin Porter   | NACA 64-514               |
| Pilatus       | PC-9                | PIL15M825                 |
| Piper         | J-3 Cub             | USA 35B                   |
| Piper         | L-14                | USA 35B                   |
| Piper         | PA-22 Tripacer      | USA 35B                   |
| Piper         | PA-24 Comanche      | NACA64A215                |
| Piper         | PA-28 Cherokee      | NACA 65-415               |
| Piper         | PA-31 Cheyenne      | NACA 63A415               |
| Piper         | PA-32 Saratoga      | NACA 65-415               |
| Piper         | PA-34 Seneca        | NACA 65-415               |
| Piper         | PA-38 Tomahawk      | NASA GA(W)-1              |
| Piper         | PA-40 Arapaho       | NACA 64A215               |
| Piper         | PA-46 Malibu        | NACA 23015                |
| Piper         | PA-48 Enforcer      | NACA 45-100               |
| Pitts         | S-1C/D              | NACA M-6                  |
| Pitts         | S-1E                | Symmetrical               |
| Republic      | F-84F Thunderstreak | NACA 64A010               |
| Republic      | F-105 Thunderchief  | NACA 65A005.5             |
| Republic      | P-47 Thunderbolt    | Seversky S-3              |
| Rutan         | Variviggen          | Roncz R1145MS             |
| Rutan         | Proteus             | NACA 4414                 |
| Rutan         | Varizee             | NACA 4414                 |
| Rutan         | Pond Racer          | Roncz                     |
| Rutan         | Raptor              | Roncz RQW17B              |
| Rutan         | Global Flyer        | Roncz                     |
| Rutan         | Space Ship One      | Rutan                     |
| Rutan         | White Knight        | Hatfield                  |
| Ryan          | AGM-34 Firebee      | NACA 10% hybrid           |
| Ryan          | XV-5A               | NACA 0012-64              |
| Ryan          | Supersonic Firebee  | Symmetrical 3%            |
| Ryan          | Spirit of St. Louis | Clark Y                   |
| Ryan          | Navion              | NACA 4415R                |
| Ryan          | RQ-4 Global Hawk    | NASA LRN 1015             |
| Schempp-Hirth | Nimbus II           | Wortmann FX 67-K-170      |
| Schempp-Hirth | Nimbus 3            | Goettingen 681            |
| Schleicher    | ASW 20C             | Wortmann FX 62-131 (14.4) |
| Schleicher    | ASW 22              | Wortmann FX S-02-196      |
| Schleicher    | ASW 23              | Wortmann FX 61-168        |

# Ejemplo de Base de Datos

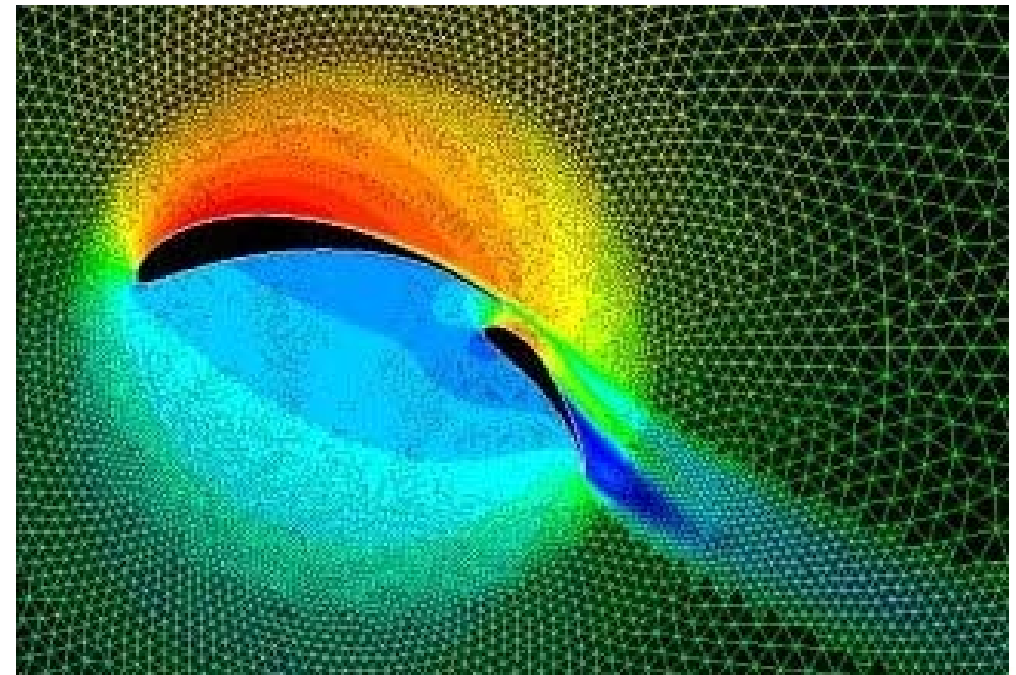
- UIUC Airfoil Data Site

- <http://www.ae.uiuc.edu/m-selig/ads.html>
- Michael Selig  
Department of Aerospace Engineering  
University of Illinois at Urbana-Champaign, Urbana, Illinois 61801
- Software y bases de datos sobre información de perfiles.



- The Incomplete Guide to Airfoil Usage

- <http://www.ae.uiuc.edu/m-selig/ads/aircraft.html>
- David Lednicer  
Analytical Methods, Inc.  
2133 152nd Ave NE  
Redmond, WA 98052  
dave@amiwest.com



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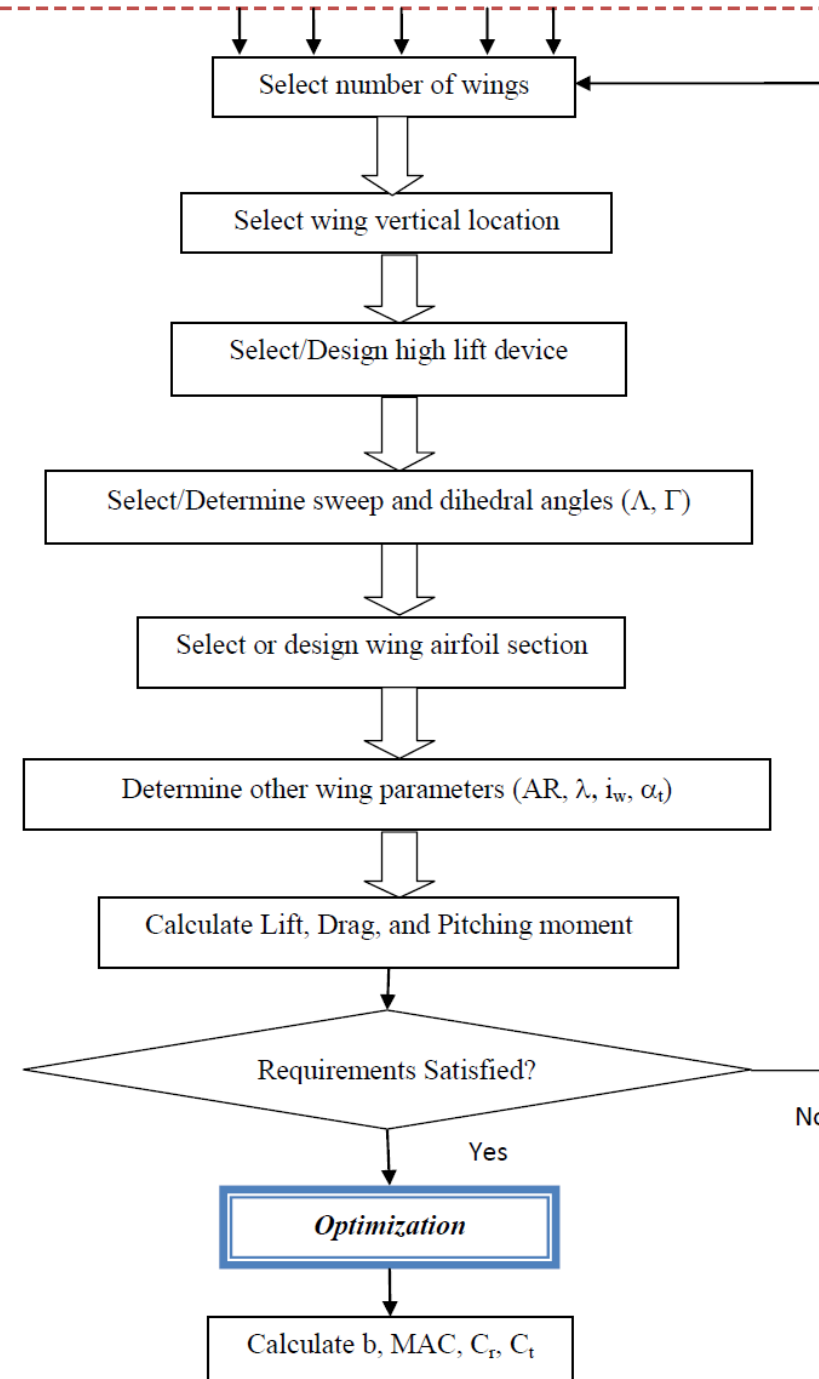
# Ejemplo de Programa

- Ejemplo:
  - SNACK. El cual parte del paquete DesignFOIL (\$179)
- Propiedades SNACK
  - The **pressure distribution** is obtained using a proprietary **panel method** developed over the last five years.
    - Implement the panel method via the linearly varying strength vortex across each panel.
    - After enforcing the boundary conditions of tangential velocity and the trailing edge Kutta condition, the final velocity distribution (and therefore the  $C_p$  distribution) is obtained by solving the matrix of equations.
  - The **boundary layer calculations** are based on **integral method theory**.
    - Integral methods involve backing out the **boundary layer shape** from the given **pressure distribution**.
    - The **laminar flow portion** is based on the approximation method developed by **von Karman & Pohlhausen**.
    - The **turbulent flow** is modeled on methods similar to that of the laminar flow; the approximation method attributed to **Buri**.
    - The **drag coefficient** is obtained using the **Squire-Young method** based on **momentum boundary layer thickness**.



# Wing Design

Identify and prioritize wing design requirements  
(Performance, stability, producibility, operational requirements, cost, flight safety)



# Coeficiente de sustentación vs. Ángulo de ataque

The maximum lift coefficient ( $C_{l_{max}}$ )

The stall angle ( $\alpha_s$ )

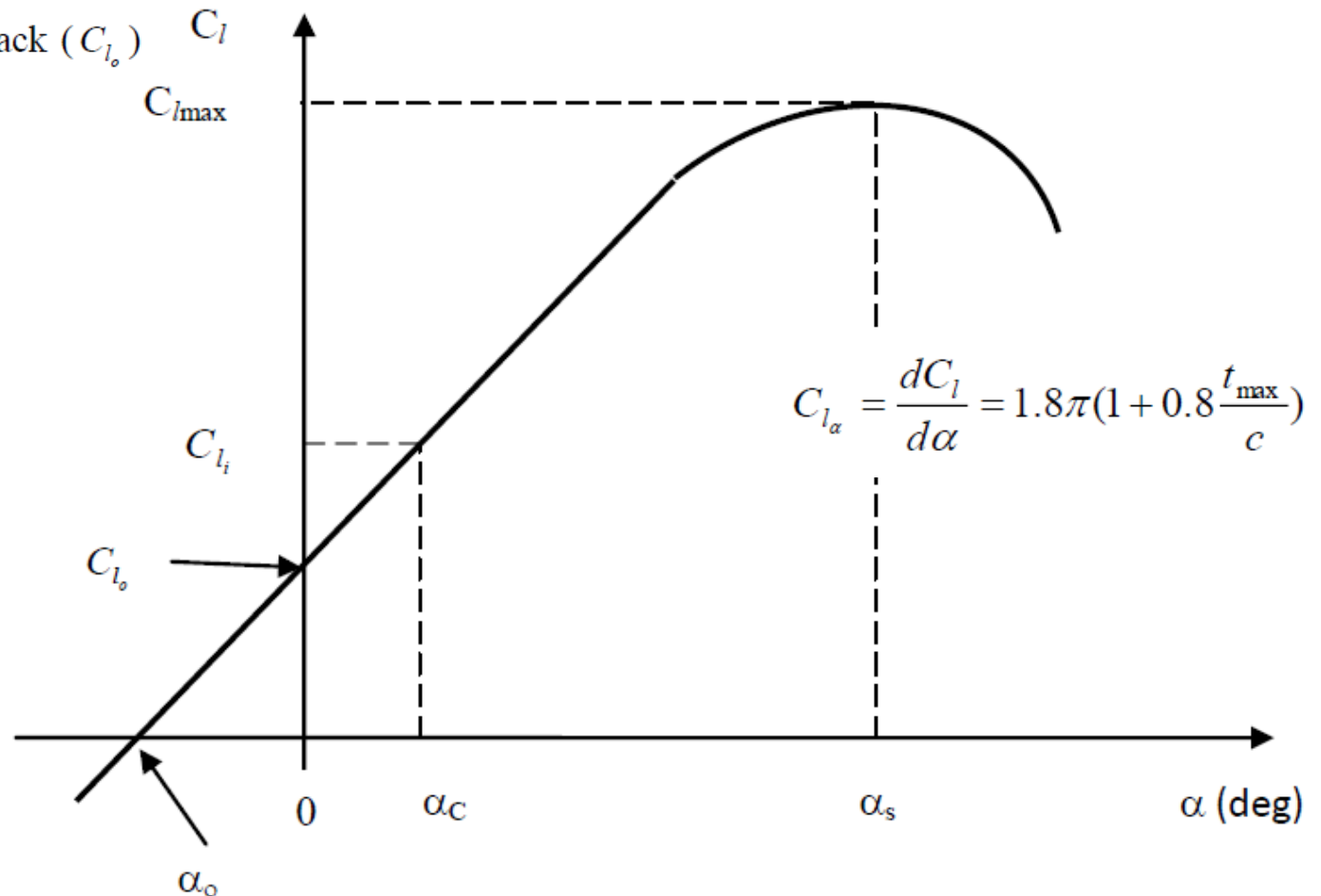
The ideal lift coefficient ( $C_{l_i}$ )

The angle of attack corresponding to ideal lift coefficient ( $\alpha_{cli}$ )

The lift coefficient at zero angle of attack ( $C_{l_o}$ )

The zero lift angle of attack ( $\alpha_o$ )

The lift curve slope ( $C_{l_\alpha}$ )



# Variaciones de Pitch Moment

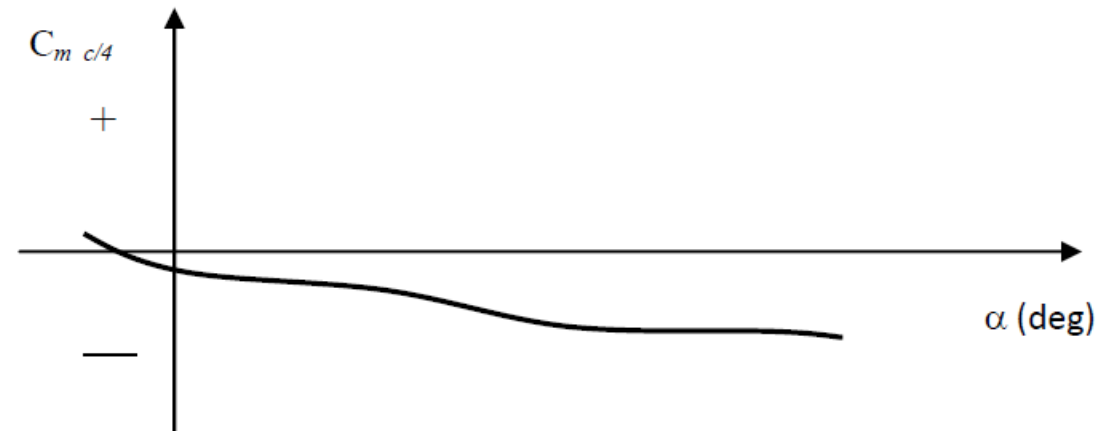


Figure 5.13. The variations of pitching moment coefficient versus angle of attack

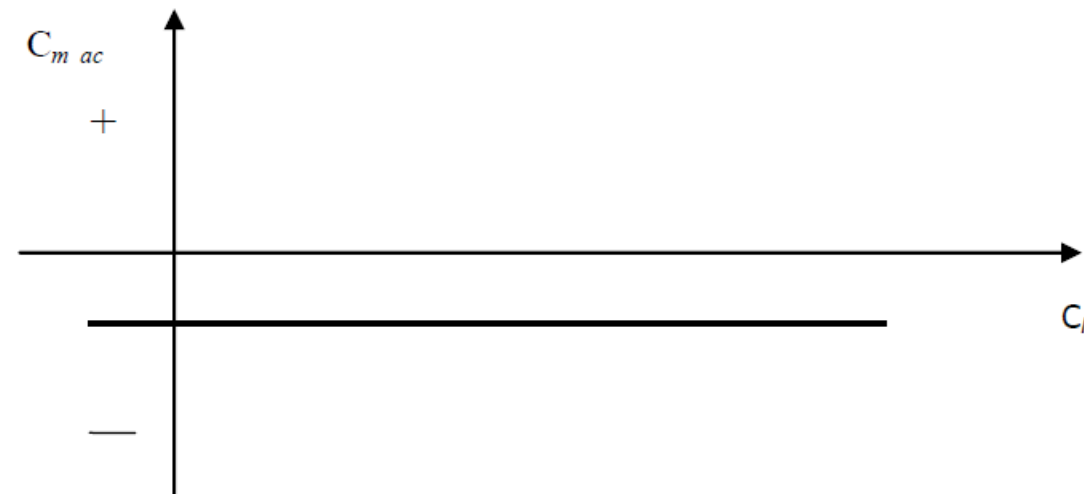
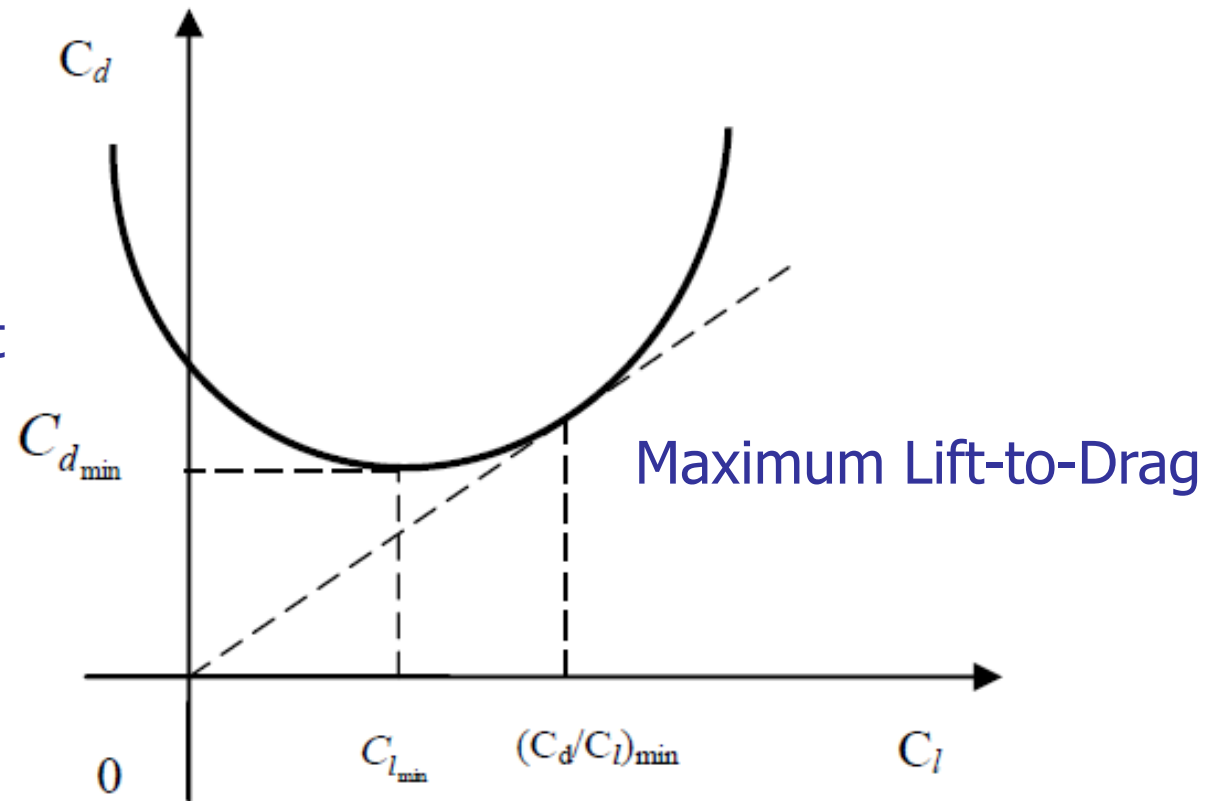


Figure 5.14. The variations of pitching moment coefficient versus lift coefficient

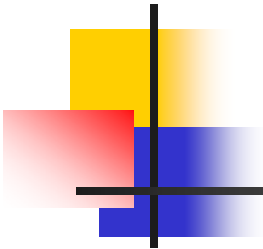
# Variaciones de Resistencia

Minimum drag coefficient



# Criterios de Selección de los Perfiles - I

1. The airfoil with the highest maximum lift coefficient ( $C_{l_{\max}}$ ).
2. The airfoil with the proper ideal or design lift coefficient ( $C_{l_d}$  or  $C_{l_i}$ ).
3. The airfoil with the lowest minimum drag coefficient ( $C_{d_{\min}}$ ).
4. The airfoil with the highest lift-to-drag ratio ( $(C_l/C_d)_{\max}$ ).
5. The airfoil with the highest lift curve slope ( $C_{l_{\alpha_{\max}}}$ ).
6. The airfoil with the lowest (closest to zero; negative or positive) pitching moment coefficient ( $C_m$ ).
7. The proper stall quality in the stall region (the variation must be gentle, not sharp).
8. The airfoil must be structurally reinforceable. The airfoil should not be so thin that spars cannot be placed inside.
9. The airfoil must be such that the cross section is manufacturable.
10. The cost requirements must be considered.
11. Other design requirements must be considered. For instance, if the fuel tank has been designated to be placed inside the wing inboard section, the airfoil must allow the sufficient space for this purpose.



# Criterios de Selección de los Perfiles - II

As a guidance; the typical values for the airfoil maximum thickness-to-chord ratio of majority of aircraft are about 6% to 18%.

- 1- For a low speed aircraft with a high lift requirement (such as cargo aircraft), the typical wing  $(t/c)_{\max}$  is about 15% - 18%.
- 2- For a high speed aircraft with a low lift requirement (such as high subsonic passenger aircraft), the typical wing  $(t/c)_{\max}$  is about 9% - 12%.
- 3- For the supersonic aircraft, the typical wing  $(t/c)_{\max}$  is about 3% - 9%.

## Introducción a los perfiles NACA

# Pasos para la selección de perfil - I

1. Determine the average aircraft weight ( $W_{avg}$ ) in cruising flight:

$$W_{avg} = \frac{1}{2}(W_i + W_f)$$

$W_i$  is the initial aircraft weight at the beginning of cruise  
 $W_f$  is the final aircraft weight at the end of cruise.

2. Calculate the aircraft ideal cruise lift coefficient ( $C_{L_c}$ ). In a cruising flight, the aircraft weight is equal to the lift force (equation 5.1), so:

$$C_{L_c} = \frac{2W_{ave}}{\rho V_c^2 S}$$

where  $V_c$  is the aircraft cruise speed,  $\rho$  is the air density at cruising altitude, and  $S$  is the wing planform area.

$$L = W \Rightarrow \frac{1}{2} \rho V^2 S C_L = mg \quad (5.1)$$

$$D = T \Rightarrow \frac{1}{2} \rho V^2 S C_D = n T_{max} \quad (\text{jet engine}) \quad (5.2)$$

$$D = T \Rightarrow \frac{1}{2} \rho V^2 S C_D = \frac{n \eta_P P_{max}}{V_c} \quad (\text{prop-driven engine}) \quad (5.3)$$

The variable “n” ranges between 0.6 to 0.9.

only a partial engine throttle is used in a cruising flight and maximum engine power or engine thrust is not employed.

For the airfoil initial design, it is suggested to use 0.75.

# Pasos para la selección de perfil - II

## 3. Calculate the wing cruise lift coefficient ( $C_{L_{c_w}}$ ).

- The wing is solely responsible for the generation of the lift.
- Other aircraft components also contribute to the total lift; negatively, or positively; (20%)
- Thus the relation between aircraft cruise lift coefficient and wing cruise lift coefficient is a function of aircraft configuration.

$$C_{L_{c_w}} = \frac{C_{L_c}}{0.95} \quad \text{Estimación preliminar}$$

## 4. Calculate the wing airfoil ideal lift coefficient ( $C_{l_i}$ ).

- The wing is a three-dimensional body, while an airfoil is a two-dimensional section.
- If the wing chord is constant, with no sweep angle, no dihedral, and the wing span is assumed to be infinity; theoretically; the wing lift coefficient would be the same as wing airfoil lift coefficient.
- However, at this moment, the wing has not been designed yet, we have to resort to an approximate relationship. In reality, the span is limited, and in most cases, wing has sweep angle, and non-constant chord, so the wing lift coefficient will be slightly less than airfoil lift

$$C_{l_i} = \frac{C_{L_{c_w}}}{0.9} \quad \text{Estimación preliminar}$$

# Pasos para la selección de perfil - III

5. Calculate the aircraft maximum lift coefficient ( $C_{L_{max}}$ ):

$$C_{L_{max}} = \frac{2W_{TO}}{\rho_o V_s^2 S}$$

where  $V_s$  is the aircraft stall speed,  $\rho_o$  is the air density at sea level, and  $W_{TO}$  is the aircraft maximum take-off weight.

6. Calculate the wing maximum lift coefficient ( $C_{L_{max_w}}$ ). With the same logic that was described in step 3, the following relationship is recommended.

$$C_{L_{max_w}} = \frac{C_{L_{max}}}{0.95}$$

7. Calculate the wing airfoil gross maximum lift coefficient ( $C_{l_{max_{gross}}}$ ).

$$C_{l_{max_{gross}}} = \frac{C_{L_{max_w}}}{0.9}$$

where the wing airfoil “gross” maximum lift coefficient is the airfoil maximum lift coefficient in which the effect of high lift device (e.g. flap) is included.

# Pasos para la selección de perfil - IV

8. Select/Design the high lift device (type, geometry, and maximum deflection).

Se va a explicar a posterior

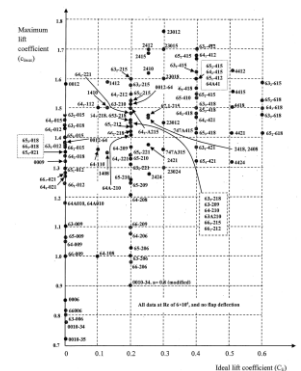
9. Determine the high lift device (HLD) contribution to the wing maximum lift coefficient  $\Delta C_{l_{HLD}}$

10. Calculate the wing airfoil “net” maximum lift coefficient ( $C_{l_{max}}$ )

$$C_{l_{max}} = C_{l_{max, gross}} - \Delta C_{l_{HLD}}$$

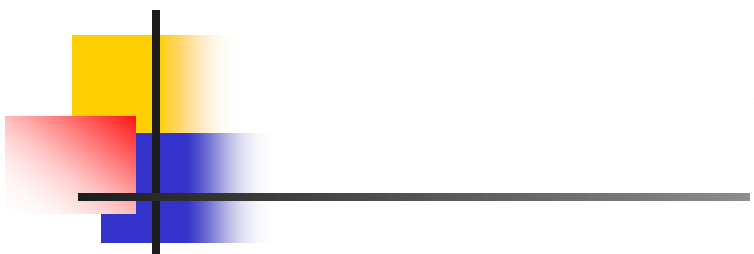
11. Identify airfoil section alternatives that deliver the desired  $C_{li}$  (step 4) and  $C_{lmax}$  (step 10).

- The horizontal axis represents the airfoil ideal lift coefficient while the vertical axis the airfoil maximum lift coefficient.
- Every black circle represents one NACA airfoil section
- Bibliografía adicional



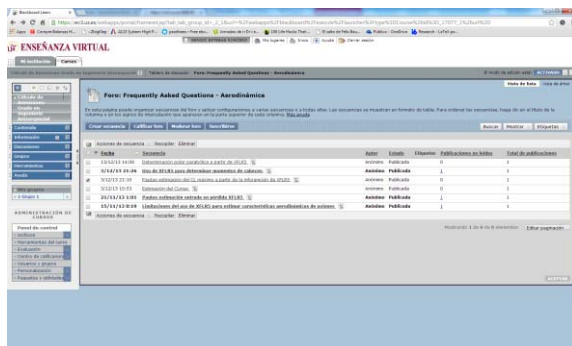
3. Abbott I. H. and Von Doenhoff A. F., **Theory of Wing Sections**, Dover, 1959

4. Anderson J. D., **Fundamentals of Aerodynamics**, McGraw-Hill, Fifth edition, 2010

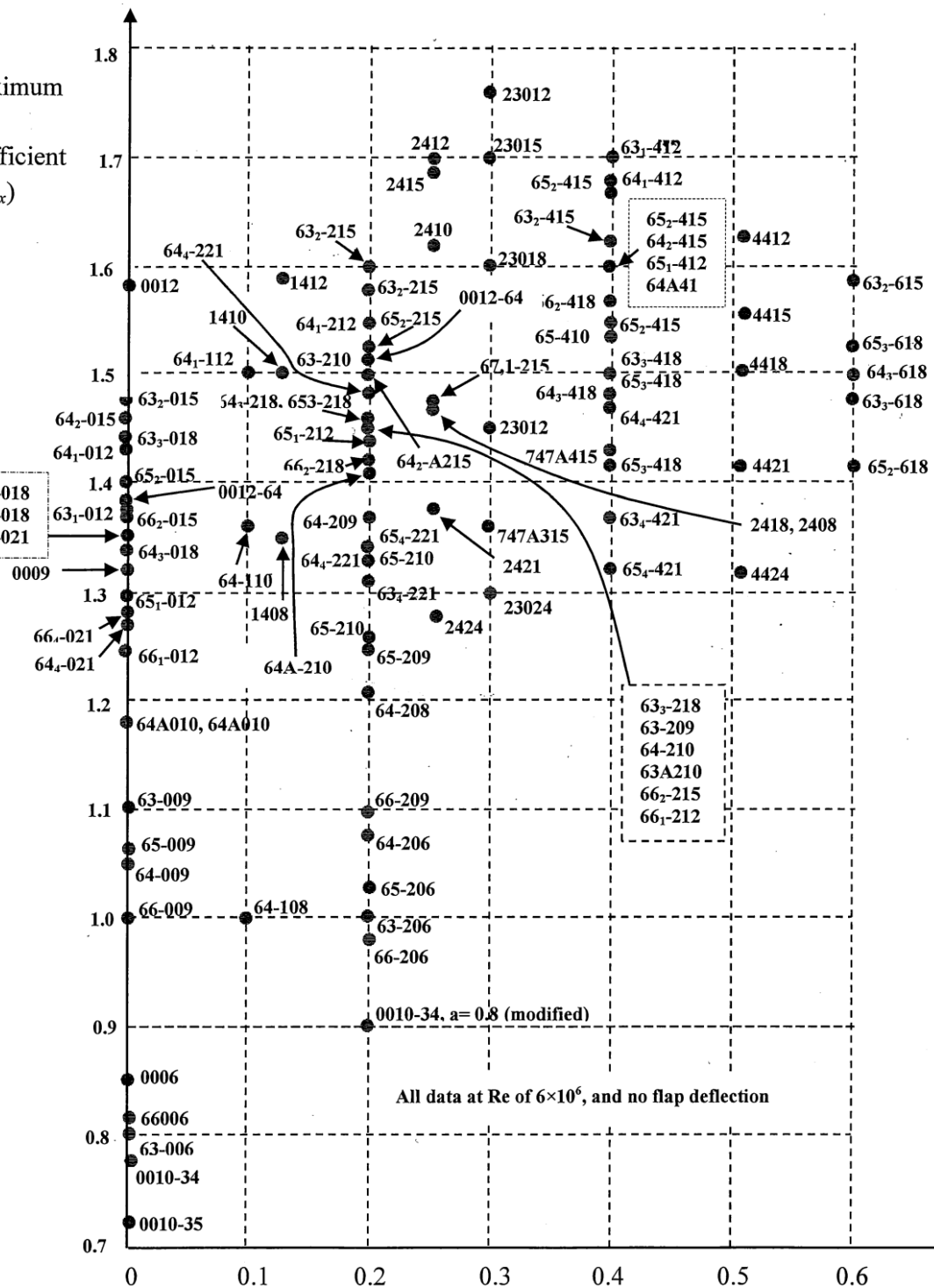


$C_{li}$  and  $C_{lmax}$

## Enseñanza Virtual FAQ Aerodynamics



Maximum  
lift  
coefficient  
( $C_{lmax}$ )



# Pasos para la selección de perfil - V

12. If the wing is designed for a high subsonic passenger aircraft, select the thinnest airfoil (the lowest  $(t/c)_{\max}$ ).

- Reduce the critical Mach number ( $M_{cr}$ ) and drag-divergent Mach number ( $M_{dd}$ ).
- This allow the aircraft fly closer to Mach one before the drag rise is encountered.
- Thinner airfoil will have a higher  $M_{cr}$  than a thicker airfoil

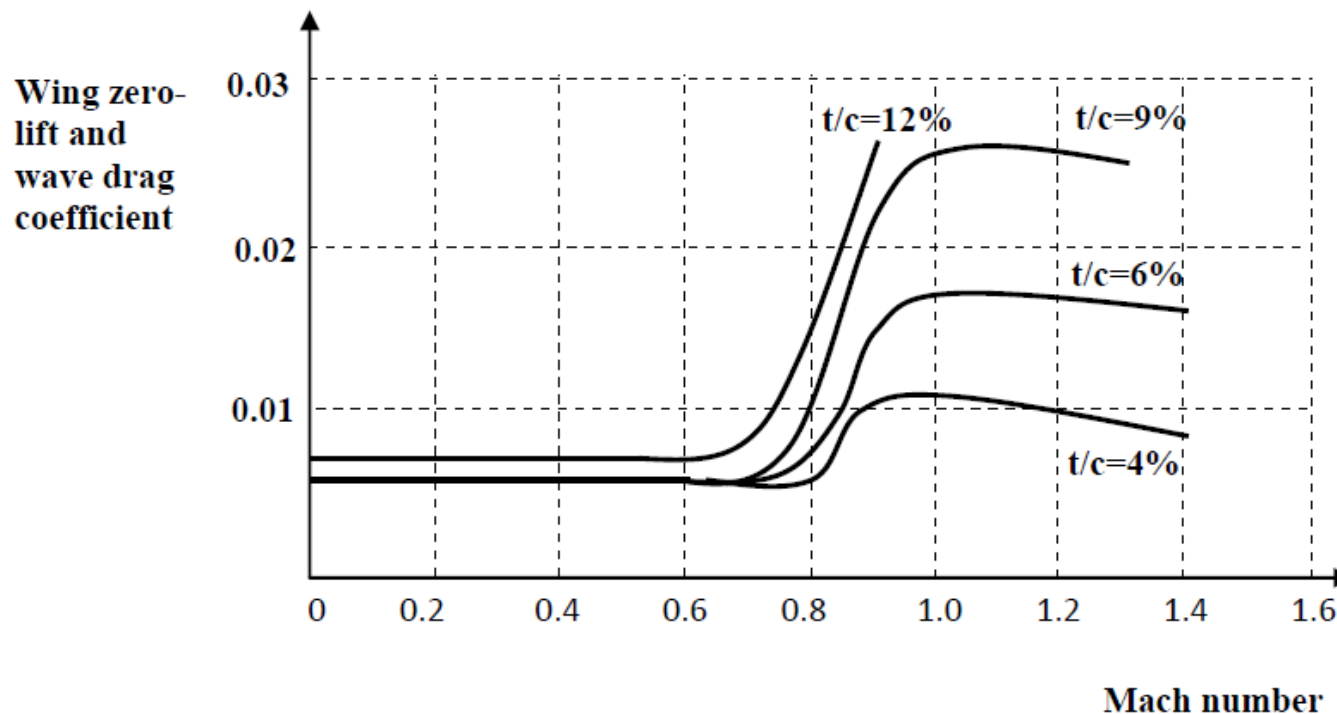


Figure 5.24. Variation of wing zero-lift and wave drag coefficient versus Mach number for various airfoil thickness ratio.

# Pasos para la selección de perfil - VI

Tabla comparativa entre perfiles

| Design objectives | Weight | Airfoil 1 | Airfoil 2 | Airfoil 3 | Airfoil 4 | Airfoil 5 |
|-------------------|--------|-----------|-----------|-----------|-----------|-----------|
| $C_{dmin}$        | 25%    |           |           |           |           |           |
| $C_{mo}$          | 15%    |           |           |           |           |           |
| $\alpha_s$        | 15     |           |           |           |           |           |
| $\alpha_o$        | 10     |           |           |           |           |           |
| $(C_l/C_d)_{max}$ | 10%    |           |           |           |           |           |
| $C_{l\alpha}$     | 5%     |           |           |           |           |           |
| Stall quality     | 20%    |           |           |           |           |           |
| Summation         | 100%   | 64        | 76        | 93        | 68        | 68        |

# Pasos para la selección de perfil - VII

- Ejemplo

Select a NACA airfoil section for the wing for a jet non-maneuverable GA aircraft with the following characteristics:

$$m_{TO} = 4000 \text{ kg}, S = 30 \text{ m}^2, V_c = 250 \text{ knot (at 3000 m)}, V_s = 65 \text{ knot (sea level)}$$

The high lift device (split flap) will provide  $\Delta C_L = 0.8$  when deflected.

Ideal lift coefficient:

$$C_{L_c} = \frac{2W_{ave}}{\rho V_c^2 S} = \frac{2 \times 4000 \times 9.81}{0.9 \times (250 \times 0.514)^2 \times 30} = 0.176$$

$$C_{L_{cw}} = \frac{C_{L_c}}{0.95} = \frac{0.176}{0.95} = 0.185$$

$$C_{l_i} = \frac{C_{L_{cw}}}{0.9} = \frac{0.185}{0.9} = 0.205 \cong 0.2$$

Maximum lift coefficient:

$$C_{L_{max}} = \frac{2W_{TO}}{\rho_o V_s^2 S} = \frac{2 \times 4000 \times 9.81}{1.225 \times (65 \times 0.514)^2 \times 30} = 1.909$$

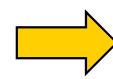
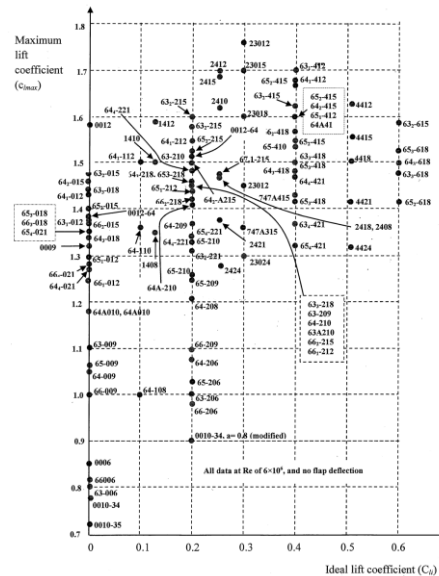
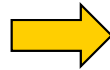
$$C_{L_{max_w}} = \frac{C_{L_{max}}}{0.95} = \frac{1.909}{0.95} = 2.01$$

$$C_{l_{max_{gross}}} = \frac{C_{L_{max_w}}}{0.9} = \frac{2.01}{0.9} = 2.233$$

$$C_{l_{max}} = C_{l_{max_{gross}}} - \Delta C_{l_{max_{HLD}}} = 2.233 - 0.8 = 1.433$$

# Pasos para la selección de perfil - VII

$$C_{li} = 0.2, C_{lmax} = 1.43$$



**63<sub>3</sub>-218, 64-210, 66<sub>1</sub>-212, 66<sub>2</sub>-215, 65<sub>3</sub>-218**

| No | NACA                 | $C_{dmin}$ | $C_{mo}$ | $\alpha_s$ (deg)<br>Flap up | $\alpha_o$ (deg)<br>$\delta_f = 60^\circ$ | $(C_l/C_d)_{max}$ | Stall quality |
|----|----------------------|------------|----------|-----------------------------|-------------------------------------------|-------------------|---------------|
| 1  | 63 <sub>3</sub> -218 | 0.005      | -0.028   | 12                          | -12                                       | 100               | Docile        |
| 2  | 64-210               | 0.004      | -0.040   | 12                          | -13                                       | 75                | Moderate      |
| 3  | 66 <sub>1</sub> -212 | 0.0032     | -0.030   | 12                          | -13                                       | 86                | Sharp         |
| 4  | 66 <sub>2</sub> -215 | 0.0035     | -0.028   | 14                          | -13.5                                     | 86                | Sharp         |
| 5  | 65 <sub>3</sub> -218 | 0.0045     | -0.028   | 16                          | -13                                       | 111               | Moderate      |

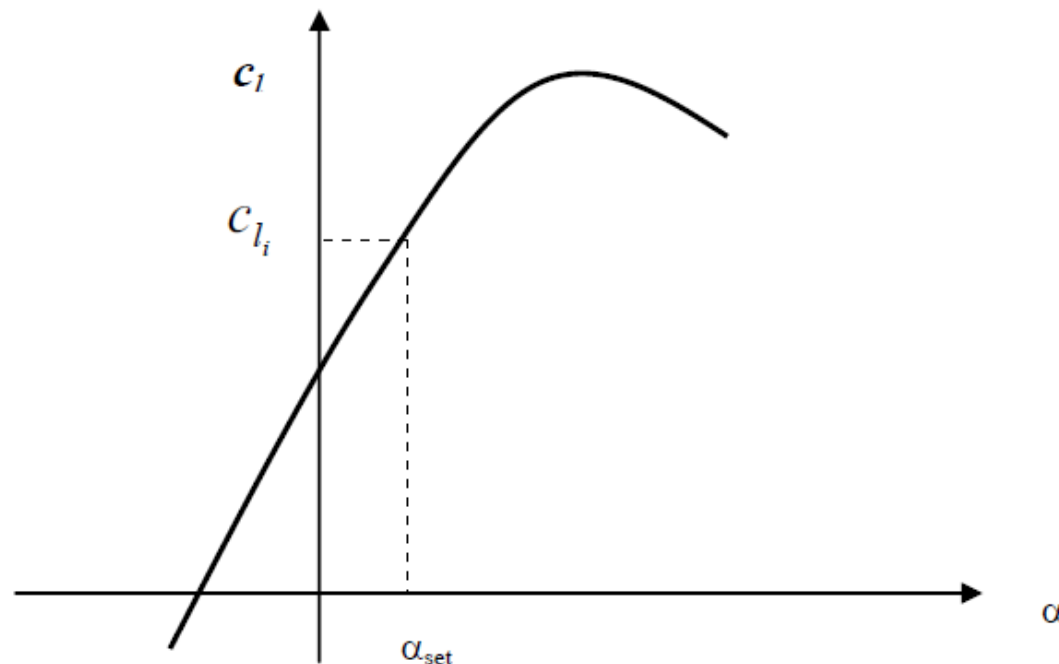
# Pasos para la selección de perfil - VIII

- 1- The NACA airfoil section 66<sub>1</sub>-212 yields the highest maximum speed, since it has the lowest  $C_{dmin}$  (0.0032).
  - 2- The NACA airfoil section 65<sub>3</sub>-218 yields the lowest stall speed, since it has the highest stall angle (16 degrees).
  - 3- The NACA airfoil section 65<sub>3</sub>-218 yields the highest endurance, since it has the highest  $(C_l/C_d)_{max}$  (111).
  - 4- The NACA 63<sub>3</sub>-218 yields the safest flight, due to its docile stall quality.
  - 5- The NACA airfoil sections 63<sub>3</sub>-218, 66<sub>2</sub>-215, and 65<sub>3</sub>-218 deliver the lowest control problem in flight, due to the lowest  $C_{mo}$  (-0.028).
- Since the aircraft is a non-maneuverable GA aircraft, the stall quality cannot be sharp;
  - NACA airfoil sections 661-212 and 662-215 are not acceptable.
  - If the safety is the highest requirement, the best airfoil is NACA 632-218.
  - If the low cost is the most important requirement, NACA 64-210 with the lowest  $C_{dmin}$  is the best.
  - If the aircraft performance (stall speed, endurance or maximum speed) are of greatest important design requirement, the NACA airfoil section 653-218, 653-218, or 661-212 are the best respectively.
  - This may be performed by using a comparison table incorporating the weighted design requirements.

# Pasos para la selección de perfil - IX

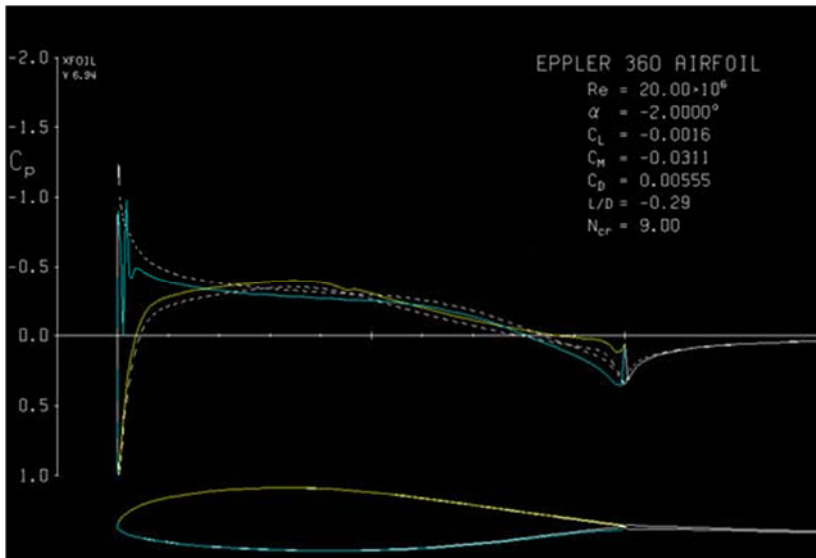
## Wing Incidence

1. The wing must be able to generate the desired lift coefficient during cruising flight.
2. The wing must produce minimum drag during cruising flight.
3. The wing setting angle must be such that the wing angle of attack could be safely varied (in fact increased) during take-off operation.
4. The wing setting angle must be such that the fuselage generates minimum drag during cruising flight (i.e. the fuselage angle of attack must be zero in cruise).

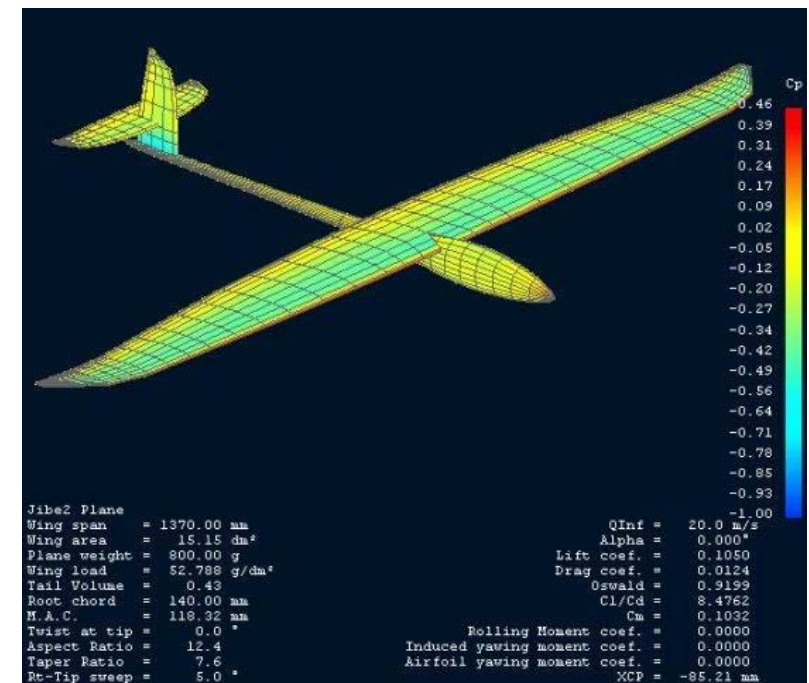


# Herramientas – XFOIL & XFLR5

- El XFOIL, predecesor del XFLR5, software libre y permite el análisis y diseño de perfiles alares subsónicos.
  - Creado por Mark Drela como una herramienta de diseño para el proyecto Daedalus en el MIT (Massachusetts Institute of Technology) allá por la década de los 80
  - Programado en FORTRAN (última versión data de 2001): muy utilizado
- XFLR5 is an analysis tool for airfoils, wings and planes operating at low Reynolds Numbers
  - El XFLR5 sucesor natural del XFOIL: programado en C++
  - <http://www.xflr5.com/xflr5.htm>



DEMO



# Herramientas – XFOIL & XFLR5 - 2

- Empleo de herramientas
  - Direct Foild Design: Estudios 2D
  - Wing and Plane Design
    - Polar Type
      - Fixed Speed, Fixed Lift, Fixed aoa, Fixed beta Range
    - Analysis
      - LLT, VLM1 (no sideslip), VLM2, 3DPanels
    - Inertia
    - Ref. Dimensions
      - Wing planform, projected wing planform, user defined
    - Aero Data
      - Densidad, viscosidad, Ground Effect
  - Analizar los distintos métodos de análisis
  - Recomendaciones iniciales
    - LLT: determinación de  $C_{lmax}$
    - VLM1: determinación de polar

**Analysis Definition**

☒ Auto Analysis Name T2-VLM1

Polar Type Analysis Inertia Ref. dimensions Aero data

☐ Type 1 (Fixed Speed) Free Stream Speed = 15,00 m/s  
☒ Type 2 (Fixed Lift)  $\alpha$  = 0,00 °  
☐ Type 4 (Fixed aoa)  $\beta$  = 0,00 °  
☐ Type 5 (Beta range)

Wing Loading = 2.044 kg/m<sup>2</sup>  
 Tip Re.sqrt(Cl) = 30 000  
 Root Re.sqrt(Cl) = 31 000  
 Vinf.sqrt(Cl) = 5.72 m/s

OK Cancel

**Analysis Definition**

☒ Auto Analysis Name T2-VLM1

Polar Type Analysis Inertia Ref. dimensions Aero data

Analysis Methods

☐ LLT (Wing only)  
☒ Horseshoe vortex (VLM1) (No sideslip)  
☐ Ring vortex (VLM2)  
☐ 3D Panels

Options

☒ Viscous  
☐ Tilt. Geom.  
☐ Ignore Body Panels

OK Cancel

**Analysis Definition**

☒ Auto Analysis Name T2-VLM1

Polar Type Analysis Inertia Ref. dimensions Aero data

Ref. dimensions for aero coefficients

☒ Wing Planform  
☐ Wing Planform projected on xy plane  
☐ User defined

Ref. area = 0,445  
 Ref. span length = 2.260,000  
 Ref. chord length = 215,791

OK Cancel

**Analysis Definition**

☒ Auto Analysis Name T2-VLM1

Polar Type Analysis Inertia Ref. dimensions Aero data

Inertia properties

☒ Use plane inertia

X\_CoG = 248,807 mm  
 Z\_CoG = 3,156 mm

OK Cancel

**Analysis Definition**

☒ Auto Analysis Name T2-VLM1

Polar Type Analysis Inertia Ref. dimensions Aero data

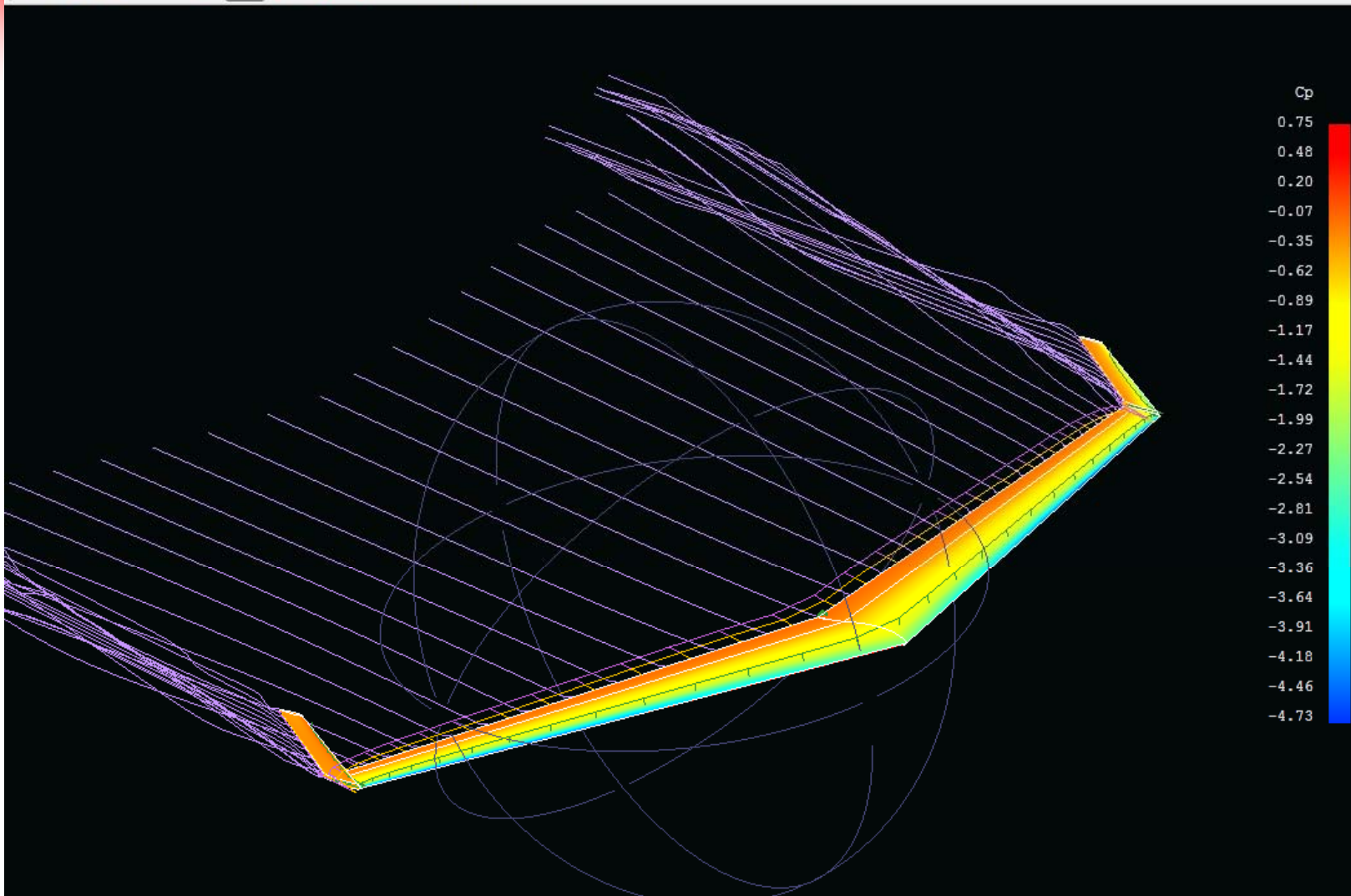
Air Data

Unit ☒ International ☐ Imperial  
 $\rho$  = 1,225 kg/m<sup>3</sup>  
 $\nu$  = 1,5e-05 m<sup>2</sup>/s

Ground Effect

☐ Ground Effect  
 Height = 0,00 mm

OK Cancel



Flying Wing flap\_up\_2°  
 Wing Span = 2260.000 mm  
 xyProj. Span = 2000.000 mm  
 Wing Area = 0.445 m²  
 xyProj. Area = 0.420 m²  
 Plane Mass = 0.910 kg  
 Wing Load = 2.167 kg/m²  
 Root Chord = 300.000 mm  
 MAC = 215.791 mm  
 TipTwist = 0.000°  
 Aspect Ratio = 11.475  
 Taper Ratio = 3.750  
 Root-Tip Sweep = 27.769°  
 Mesh elements = 1122

V = 6.68 m/s  
 Alpha = 9.750°  
 Beta = 0.000°  
 CL = 0.734  
 CD = 0.036  
 Efficiency = 0.969  
 CL/CD = 20.620  
 Cm = -0.086  
 Cl = -0.000  
 Cn = -0.000  
 X\_CP = 284.323 mm  
 X\_CG = 248.807 mm  
 Flap 1 Moment = 3.7425 N.m  
 Flap 2 Moment = 3.7425 N.m

Plane analysis

Analysis settings

☒ Sequence

$\alpha$

Start= -10,000 °

End= 20,000 °

$\Delta$ = 0,250 °

☒ Init LLT ☒ Store OpPoint

Analyze

Results

☒ Cp ☐ Panel Forces

☒ Lift ☐ Moment

☒ Ind. Drag ☒ Visc. Drag

☐ Trans. ☐ Downw.

☐ Surf. Vel. ☒ Stream

☐ Animate

Display

☐ Axes ☐ Panels

☐ Normals ☐ Vortices

☐ Surfaces ☒ Outline

☐ Foil Names ☐ Masses

Reset Pick Center

Clip: [Slider]

**Plane Editor**

**Plane Description**

Flying Wing flap\_up\_2°

Description:

Plane Inertia

**Body**

Warning:  
Including the body in the analysis is not recommended.  
Check the guidelines for explanations.

☐ Body    Define    Import

x = 0,00 mm

z = 0,00 mm

**Main Wing**

☒ Main wing

Define    x = 0,00 mm

Import    z = 0,00 mm

Tilt Angle = 0,00 °

**Wing 2**

☐ Biplane

Define    x = 0,00 mm

Import    z = 0,00 mm

Tilt Angle = 0,00 °

**Elevator**

☐ Elevator

Define    x = 600,00 mm

z = 0,00 mm

Tilt Angle = 0,00 °

**Fin**

☐ Fin

Define    x = 650,00 mm

☐ Two-sided Fin    y = 0,00 mm

☐ Double Fin    z = 0,00 mm

Tilt Angle = 0,00 °

Wing Area = 0.45 m<sup>2</sup>

Wing Span = 2260.00 mm

Elev. Area = m<sup>2</sup>

Elev. Lever Arm = mm

Fin Area = m<sup>2</sup>

TailVolume =

Total Panels = 550

OK    Cancel

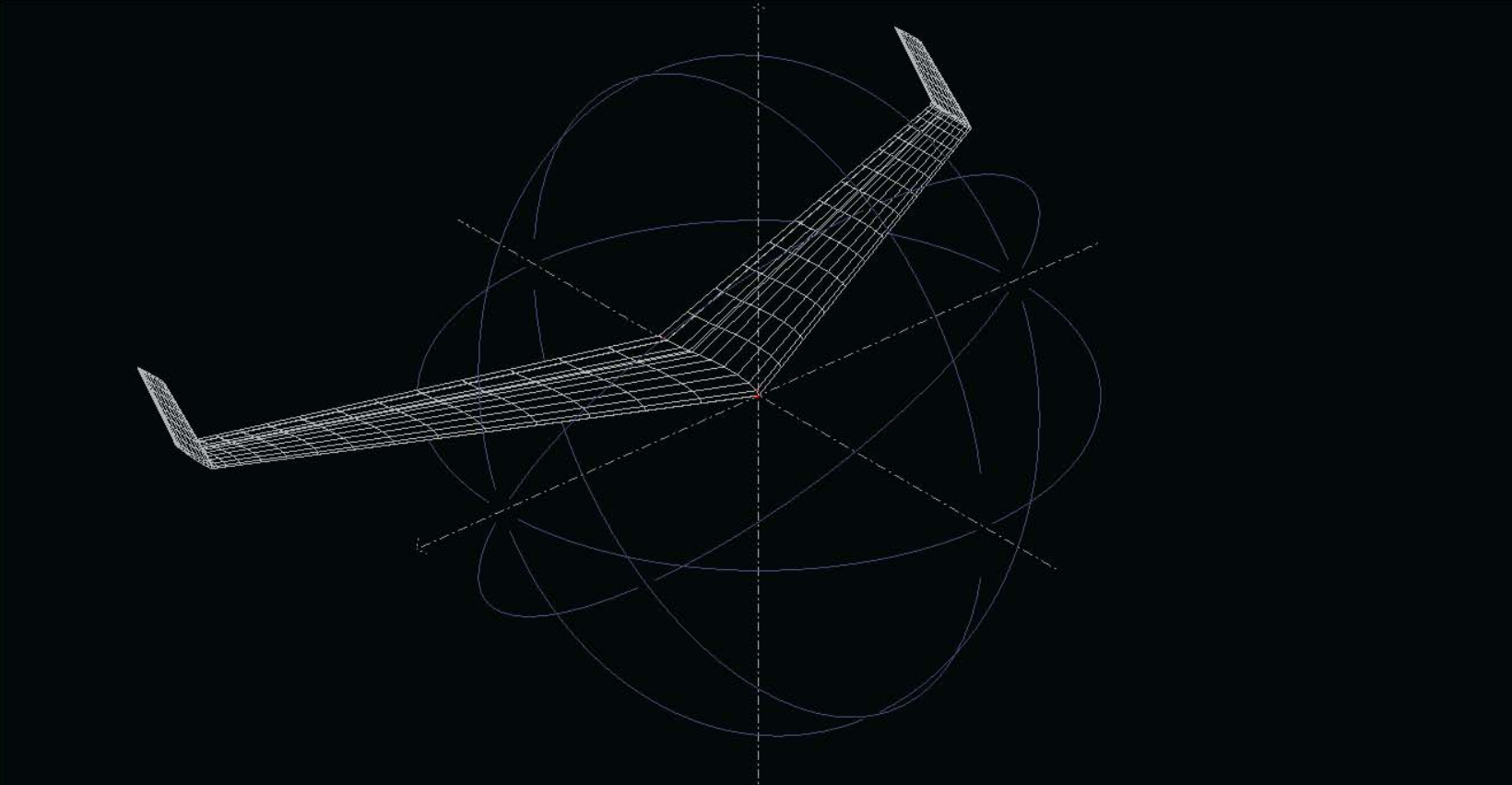
Wing Edition

Flying Wing flap\_up\_2°

☒ Symetric
 ☒ Right Side
 ☐ Left Side

|   | y (mm)    | chord (mm) | offset (mm) | dihedral | twist(°) | foil                       | X-panels | X-dist | Y-panels | Y-dist  |
|---|-----------|------------|-------------|----------|----------|----------------------------|----------|--------|----------|---------|
| 1 | 0,000     | 300,000    | 0,000       | 0,0      | 0,00     | MH 45 Flap Up 2°           | 11       | Cosine | 15       | -Sine   |
| 2 | 1.000,000 | 120,000    | 500,000     | 90,0     | 0,00     | MH 45 Flap Up 2°           | 11       | Cosine | 2        | Uniform |
| 3 | 1.010,000 | 110,000    | 510,000     | 90,0     | 0,00     | NACA 0007 Free Transitions | 11       | Cosine | 8        | Uniform |
| 4 | 1.130,000 | 80,000     | 650,000     |          | 0,00     | NACA 0007 Free Transitions |          |        |          |         |

Description:



Wing Span 2260.00 mm

Area 0.45 m<sup>2</sup>

Projected Span 2000.00 mm

Projected Area 0.42 m<sup>2</sup>

Mean Geom. Chord 196.95 mm

Mean Aero Chord 215.79 mm

Aspect ratio 11.48

Taper Ratio 3.75

Root to Tip Sweep 27.77 °

Number of Flaps 0

Number of VLM Panels 550

Number of 3D Panels 1122

☒ Axes
 ☒ Panels

☐ Surfaces
 ☐ Outline

☐ Foil Names
 ☐ Masses



Clip Plane

Body Edition

y = 8.207 mm  
z = 6.354 mm

Flying Wing flap\_up\_2°\_body

Description:

☐ Flat Panels
 ☒ BSplines

Degree 3
 Hoop 3

Panels 19
11

Panel bunch

Frame Positions

|   | x (mm)   | NPanels |
|---|----------|---------|
| 1 | -100.000 | 1       |
| 2 | -93.600  | 1       |
| 3 | -6.700   | 1       |
| 4 | 94.300   | 1       |
| 5 | 242.000  | 1       |
| 6 | 636.000  | 1       |
| 7 | 660.000  | 1       |

Current Frame Definition

|   | y (mm) | z (mm)  | NPanels |
|---|--------|---------|---------|
| 1 | 0.000  | 3.500   | 1       |
| 2 | 11.000 | 0.300   | 1       |
| 3 | 13.000 | -13.600 | 1       |
| 4 | 11.000 | -25.700 | 1       |
| 5 | 0.000  | -26.600 | 1       |

Clip Plane

☒ Axes
 ☐ Panels
 ☒ Light

☒ Surfaces
 ☒ Outline
 ☐ Masses

Reset Scales
Pick Center

Undo
Redo
Other

Save and Close
Cancel

case2\_V24.6\_VLM1\_novisc.txt: Bloc de notas

Archivo Edición Formato Ver Ayuda

xf1r5 v6.10.03

wing name : FDA-1 V0.0  
wing polar name : T1-24.6 m/s-VLM1-Inviscid  
Freestream speed : 24.600 m/s

| alpha   | CL        | CDi      | CDv      | CD       | CY       | Cl        | Cm         | Cn        | Cni       | Qinf    | XCP     |
|---------|-----------|----------|----------|----------|----------|-----------|------------|-----------|-----------|---------|---------|
| -10.000 | -0.061914 | 0.001503 | 0.000000 | 0.001503 | 0.000000 | 0.000000  | 0.563211   | -0.000000 | -0.000000 | 24.6000 | 4.5481  |
| -9.500  | -0.016841 | 0.001897 | 0.000000 | 0.001897 | 0.000000 | 0.000000  | 0.345125   | -0.000000 | -0.000000 | 24.6000 | 10.3535 |
| -9.000  | 0.028268  | 0.002430 | 0.000000 | 0.002430 | 0.000000 | 0.000000  | 0.126511   | -0.000000 | -0.000000 | 24.6000 | -2.3754 |
| -8.500  | 0.073402  | 0.003103 | 0.000000 | 0.003103 | 0.000000 | 0.000000  | -0.092564  | -0.000000 | -0.000000 | 24.6000 | 0.5459  |
| -8.000  | 0.118550  | 0.003917 | 0.000000 | 0.003917 | 0.000000 | 0.000000  | -0.312035  | -0.000000 | -0.000000 | 24.6000 | 1.2422  |
| -7.500  | 0.163702  | 0.004871 | 0.000000 | 0.004871 | 0.000000 | 0.000000  | -0.531833  | -0.000000 | -0.000000 | 24.6000 | 1.5543  |
| -7.000  | 0.208847  | 0.005966 | 0.000000 | 0.005966 | 0.000000 | 0.000000  | -0.751892  | -0.000000 | -0.000000 | 24.6000 | 1.7313  |
| -6.500  | 0.253974  | 0.007201 | 0.000000 | 0.007201 | 0.000000 | 0.000000  | -0.972146  | -0.000000 | -0.000000 | 24.6000 | 1.8454  |
| -6.000  | 0.299072  | 0.008576 | 0.000000 | 0.008576 | 0.000000 | 0.000000  | -1.192526  | -0.000000 | -0.000000 | 24.6000 | 1.9251  |
| -5.500  | 0.344130  | 0.010092 | 0.000000 | 0.010092 | 0.000000 | 0.000000  | -1.412967  | -0.000000 | -0.000000 | 24.6000 | 1.9840  |
| -5.000  | 0.389138  | 0.011747 | 0.000000 | 0.011747 | 0.000000 | 0.000000  | -1.633400  | -0.000000 | -0.000000 | 24.6000 | 2.0293  |
| -4.500  | 0.434084  | 0.013541 | 0.000000 | 0.013541 | 0.000000 | 0.000000  | -1.853758  | -0.000000 | -0.000000 | 24.6000 | 2.0654  |
| -4.000  | 0.478959  | 0.015474 | 0.000000 | 0.015474 | 0.000000 | 0.000000  | -2.073975  | -0.000000 | -0.000000 | 24.6000 | 2.0949  |
| -3.500  | 0.523751  | 0.017544 | 0.000000 | 0.017544 | 0.000000 | 0.000000  | -2.293983  | -0.000000 | -0.000000 | 24.6000 | 2.1195  |
| -3.000  | 0.568449  | 0.019752 | 0.000000 | 0.019752 | 0.000000 | 0.000000  | -2.513716  | -0.000000 | -0.000000 | 24.6000 | 2.1404  |
| -2.500  | 0.613044  | 0.022096 | 0.000000 | 0.022096 | 0.000000 | 0.000000  | -2.733105  | -0.000000 | -0.000000 | 24.6000 | 2.1584  |
| -2.000  | 0.657524  | 0.024574 | 0.000000 | 0.024574 | 0.000000 | 0.000000  | -2.952086  | -0.000000 | -0.000000 | 24.6000 | 2.1743  |
| -1.500  | 0.701879  | 0.027187 | 0.000000 | 0.027187 | 0.000000 | 0.000000  | -3.170590  | -0.000000 | -0.000000 | 24.6000 | 2.1883  |
| -1.000  | 0.746098  | 0.029932 | 0.000000 | 0.029932 | 0.000000 | 0.000000  | -3.388551  | -0.000000 | -0.000000 | 24.6000 | 2.2009  |
| -0.500  | 0.790171  | 0.032809 | 0.000000 | 0.032809 | 0.000000 | 0.000000  | -3.605904  | -0.000000 | -0.000000 | 24.6000 | 2.2124  |
| 0.000   | 0.834088  | 0.035816 | 0.000000 | 0.035816 | 0.000000 | 0.000000  | -3.822580  | -0.000000 | -0.000000 | 24.6000 | 2.2229  |
| 0.500   | 0.877839  | 0.038951 | 0.000000 | 0.038951 | 0.000000 | 0.000000  | -4.038516  | -0.000000 | -0.000000 | 24.6000 | 2.2326  |
| 1.000   | 0.921412  | 0.042213 | 0.000000 | 0.042213 | 0.000000 | -0.000000 | -4.253645  | -0.000000 | -0.000000 | 24.6000 | 2.2417  |
| 1.500   | 0.964799  | 0.045600 | 0.000000 | 0.045600 | 0.000000 | -0.000000 | -4.467901  | -0.000000 | -0.000000 | 24.6000 | 2.2502  |
| 2.000   | 1.007990  | 0.049111 | 0.000000 | 0.049111 | 0.000000 | -0.000000 | -4.681219  | -0.000000 | -0.000000 | 24.6000 | 2.2582  |
| 2.500   | 1.050973  | 0.052742 | 0.000000 | 0.052742 | 0.000000 | -0.000000 | -4.893534  | -0.000000 | -0.000000 | 24.6000 | 2.2659  |
| 3.000   | 1.093740  | 0.056493 | 0.000000 | 0.056493 | 0.000000 | 0.000000  | -5.104781  | -0.000000 | -0.000000 | 24.6000 | 2.2733  |
| 3.500   | 1.136282  | 0.060362 | 0.000000 | 0.060362 | 0.000000 | 0.000000  | -5.314897  | -0.000000 | -0.000000 | 24.6000 | 2.2804  |
| 4.000   | 1.178587  | 0.064345 | 0.000000 | 0.064345 | 0.000000 | 0.000000  | -5.523817  | -0.000000 | -0.000000 | 24.6000 | 2.2872  |
| 4.500   | 1.220648  | 0.068441 | 0.000000 | 0.068441 | 0.000000 | 0.000000  | -5.731477  | -0.000000 | -0.000000 | 24.6000 | 2.2939  |
| 5.000   | 1.262454  | 0.072647 | 0.000000 | 0.072647 | 0.000000 | 0.000000  | -5.937814  | -0.000000 | -0.000000 | 24.6000 | 2.3005  |
| 5.500   | 1.303996  | 0.076961 | 0.000000 | 0.076961 | 0.000000 | 0.000000  | -6.142766  | -0.000000 | -0.000000 | 24.6000 | 2.3069  |
| 6.000   | 1.345266  | 0.081380 | 0.000000 | 0.081380 | 0.000000 | 0.000000  | -6.346270  | -0.000000 | -0.000000 | 24.6000 | 2.3132  |
| 6.500   | 1.386255  | 0.085902 | 0.000000 | 0.085902 | 0.000000 | 0.000000  | -6.548263  | -0.000000 | -0.000000 | 24.6000 | 2.3195  |
| 7.000   | 1.426953  | 0.090524 | 0.000000 | 0.090524 | 0.000000 | 0.000000  | -6.748685  | -0.000000 | -0.000000 | 24.6000 | 2.3257  |
| 7.500   | 1.467353  | 0.095242 | 0.000000 | 0.095242 | 0.000000 | 0.000000  | -6.947474  | -0.000000 | -0.000000 | 24.6000 | 2.3319  |
| 8.000   | 1.507445  | 0.100056 | 0.000000 | 0.100056 | 0.000000 | 0.000000  | -7.144571  | -0.000000 | -0.000000 | 24.6000 | 2.3380  |
| 8.500   | 1.547221  | 0.104960 | 0.000000 | 0.104960 | 0.000000 | 0.000000  | -7.339913  | -0.000000 | -0.000000 | 24.6000 | 2.3442  |
| 9.000   | 1.586674  | 0.109952 | 0.000000 | 0.109952 | 0.000000 | 0.000000  | -7.533444  | -0.000000 | -0.000000 | 24.6000 | 2.3503  |
| 9.500   | 1.625795  | 0.115030 | 0.000000 | 0.115030 | 0.000000 | 0.000000  | -7.725102  | -0.000000 | -0.000000 | 24.6000 | 2.3565  |
| 10.000  | 1.664577  | 0.120190 | 0.000000 | 0.120190 | 0.000000 | 0.000000  | -7.914830  | -0.000000 | -0.000000 | 24.6000 | 2.3626  |
| 10.500  | 1.703011  | 0.125428 | 0.000000 | 0.125428 | 0.000000 | 0.000000  | -8.102571  | -0.000000 | -0.000000 | 24.6000 | 2.3689  |
| 11.000  | 1.741091  | 0.130742 | 0.000000 | 0.130742 | 0.000000 | 0.000000  | -8.288266  | -0.000000 | -0.000000 | 24.6000 | 2.3751  |
| 11.500  | 1.778809  | 0.136127 | 0.000000 | 0.136127 | 0.000000 | 0.000000  | -8.471860  | -0.000000 | -0.000000 | 24.6000 | 2.3814  |
| 12.000  | 1.816158  | 0.141582 | 0.000000 | 0.141582 | 0.000000 | 0.000000  | -8.653296  | -0.000000 | -0.000000 | 24.6000 | 2.3878  |
| 12.500  | 1.853132  | 0.147101 | 0.000000 | 0.147101 | 0.000000 | 0.000000  | -8.832520  | -0.000000 | -0.000000 | 24.6000 | 2.3942  |
| 13.000  | 1.889722  | 0.152682 | 0.000000 | 0.152682 | 0.000000 | 0.000000  | -9.009476  | -0.000000 | -0.000000 | 24.6000 | 2.4007  |
| 13.500  | 1.925924  | 0.158320 | 0.000000 | 0.158320 | 0.000000 | 0.000000  | -9.184110  | -0.000000 | -0.000000 | 24.6000 | 2.4072  |
| 14.000  | 1.961730  | 0.164013 | 0.000000 | 0.164013 | 0.000000 | 0.000000  | -9.356371  | -0.000000 | -0.000000 | 24.6000 | 2.4139  |
| 14.500  | 1.997135  | 0.169757 | 0.000000 | 0.169757 | 0.000000 | 0.000000  | -9.526204  | -0.000000 | -0.000000 | 24.6000 | 2.4206  |
| 15.000  | 2.032132  | 0.175547 | 0.000000 | 0.175547 | 0.000000 | 0.000000  | -9.693558  | -0.000000 | -0.000000 | 24.6000 | 2.4273  |
| 15.500  | 2.066717  | 0.181380 | 0.000000 | 0.181380 | 0.000000 | 0.000000  | -9.858383  | -0.000000 | -0.000000 | 24.6000 | 2.4342  |
| 16.000  | 2.100883  | 0.187252 | 0.000000 | 0.187252 | 0.000000 | 0.000000  | -10.020628 | -0.000000 | -0.000000 | 24.6000 | 2.4412  |
| 16.500  | 2.134624  | 0.193160 | 0.000000 | 0.193160 | 0.000000 | 0.000000  | -10.180244 | -0.000000 | -0.000000 | 24.6000 | 2.4482  |
| 17.000  | 2.167937  | 0.199099 | 0.000000 | 0.199099 | 0.000000 | 0.000000  | -10.337181 | -0.000000 | -0.000000 | 24.6000 | 2.4553  |
| 17.500  | 2.200817  | 0.205065 | 0.000000 | 0.205065 | 0.000000 | 0.000000  | -10.491393 | -0.000000 | -0.000000 | 24.6000 | 2.4626  |
| 18.000  | 2.233257  | 0.211054 | 0.000000 | 0.211054 | 0.000000 | 0.000000  | -10.642833 | -0.000000 | -0.000000 | 24.6000 | 2.4699  |
| 18.500  | 2.265255  | 0.217063 | 0.000000 | 0.217063 | 0.000000 | 0.000000  | -10.791453 | -0.000000 | -0.000000 | 24.6000 | 2.4773  |
| 19.000  | 2.296806  | 0.223087 | 0.000000 | 0.223087 | 0.000000 | 0.000000  | -10.937209 | -0.000000 | -0.000000 | 24.6000 | 2.4848  |
| 19.500  | 2.327906  | 0.229123 | 0.000000 | 0.229123 | 0.000000 | 0.000000  | -11.080057 | -0.000000 | -0.000000 | 24.6000 | 2.4924  |
| 20.000  | 2.358551  | 0.235165 | 0.000000 | 0.235165 | 0.000000 | 0.000000  | -11.219953 | -0.000000 | -0.000000 | 24.6000 | 2.5002  |

# Cálculo $C_{Lmax}$ – Métodos Empíricos - I

- Una vez elegidos  $\alpha_{0L}$ ,  $C_{lmax}$  y  $\alpha_{STALL}$  del perfil se requiere elegir:
  - Configuración del Flap de Borde de Salida (TE):
    - Plain
    - Single-slotted flap
    - Fowler flap
    - Split flap
  - La relación ente flap y la cuerda: flap-to-chord ratio  $c_f/c$
  - Deflexión del flap:  $\delta_f$

Nicolai, L.M. Carichner, G.E. Fundamentals of Aircraft and Airship Design: Vol 1, 2010

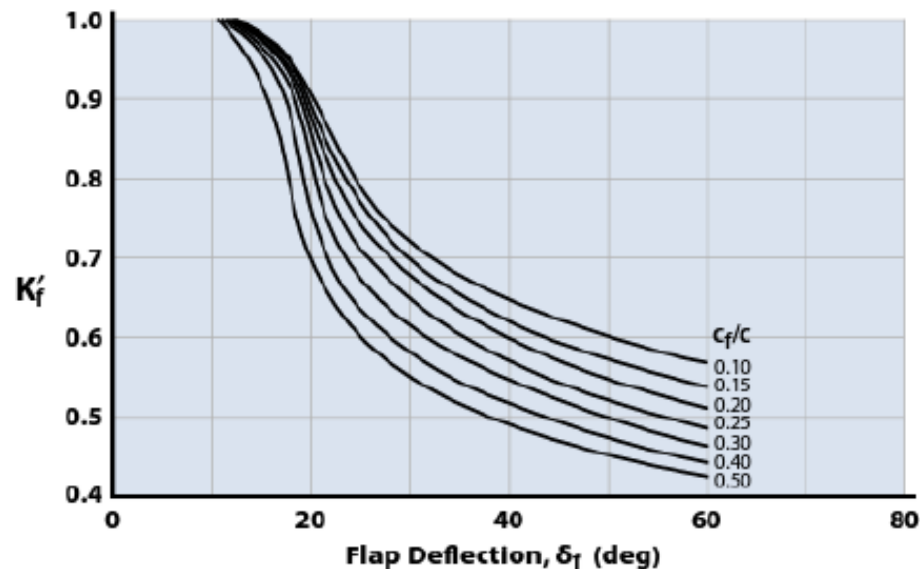


Figure 9.9 Nonlinear correction for plain TE flaps (adapted [10]).

# Cálculo $C_{Lmax}$ – Métodos Empíricos - II

## Section Lift Coefficient

- Configuración del Flap de Borde de Salida (TE): Plain TE Flap
  - Cálculo del cambio en  $\Delta\alpha_{0L}$  para una deflexión de flap

$$\Delta\alpha_{0L} = -\frac{dC_\ell}{d\delta_f} \frac{1}{C_{\ell\alpha}} \delta_f K'_f$$

$C_{\ell\alpha}$  = section lift curve slope (per radian)

$K'_f$  = correction for nonlinear effects, Fig. 9.9

$dC_\ell/d\delta_f$  = change in  $C_\ell$  for a change in  $\delta_f$ , Fig. 9.10

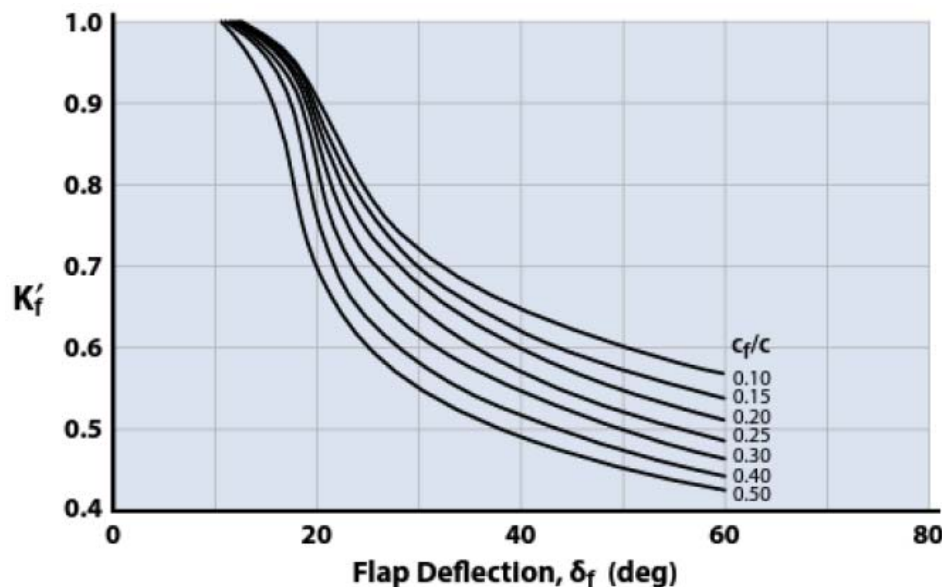


Figure 9.9 Nonlinear correction for plain TE flaps (adapted [10]).

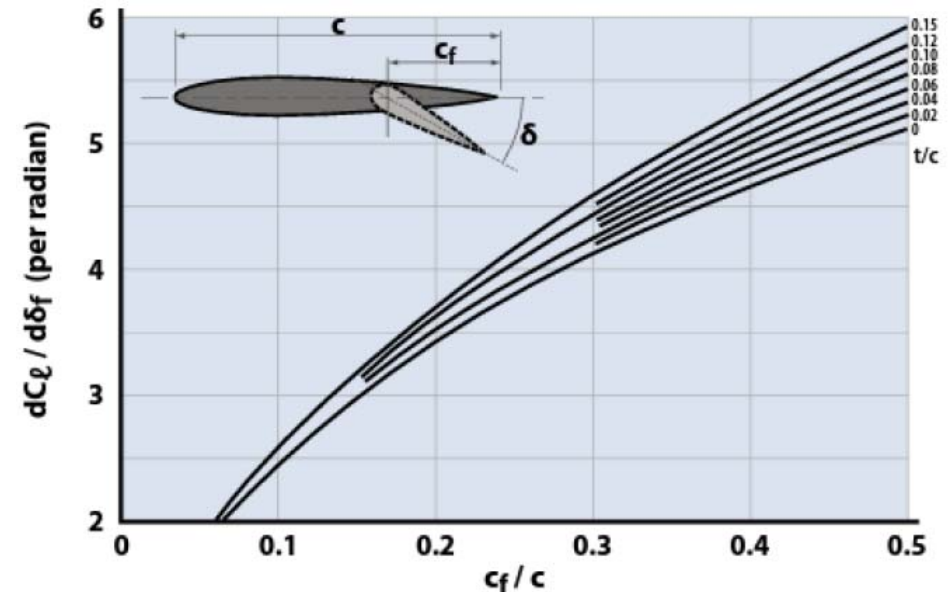
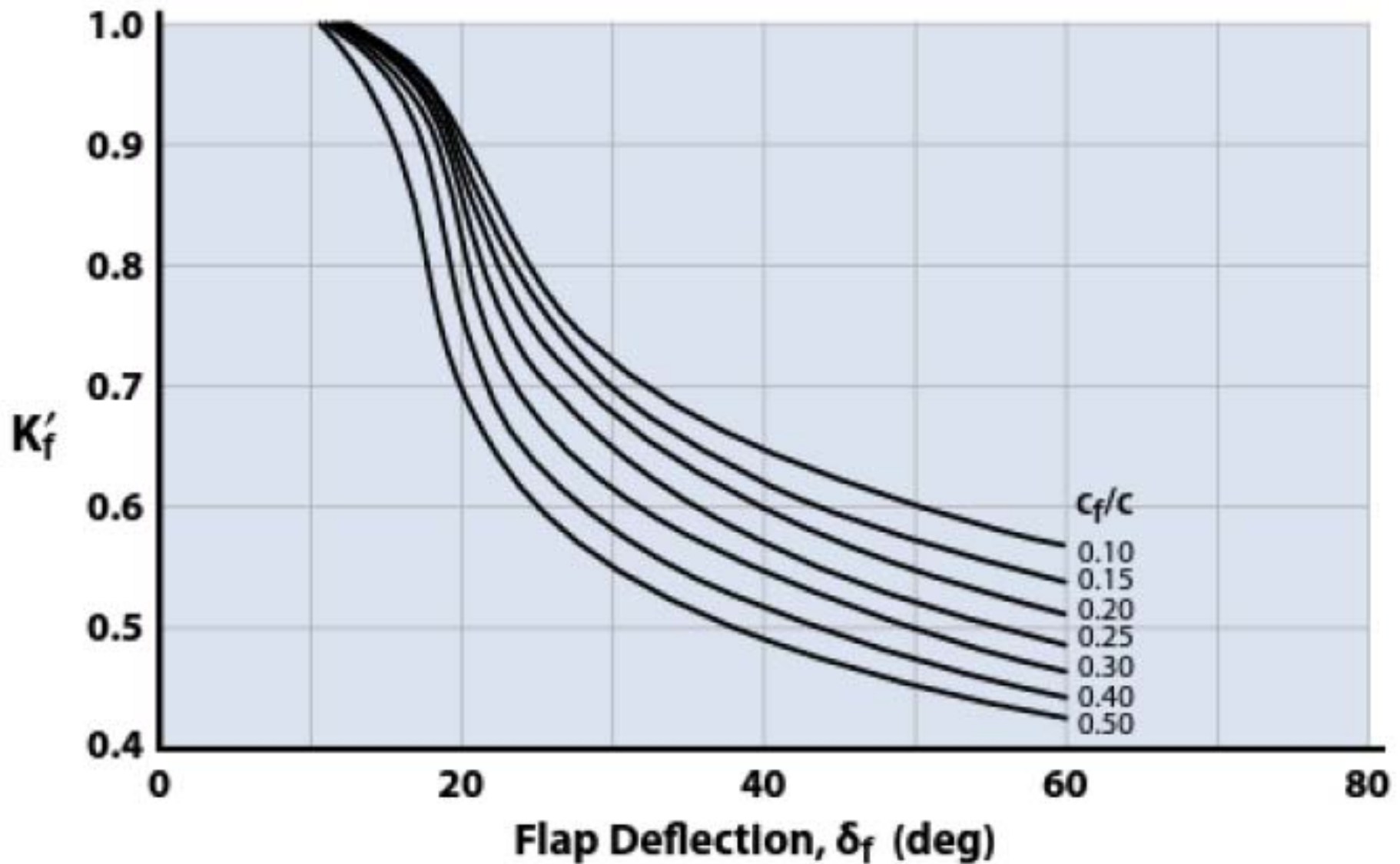


Figure 9.10 Variation of  $dC_\ell/d\delta_f$  with flap chord ratio (adapted [10]).

# Plain TE Flap - I

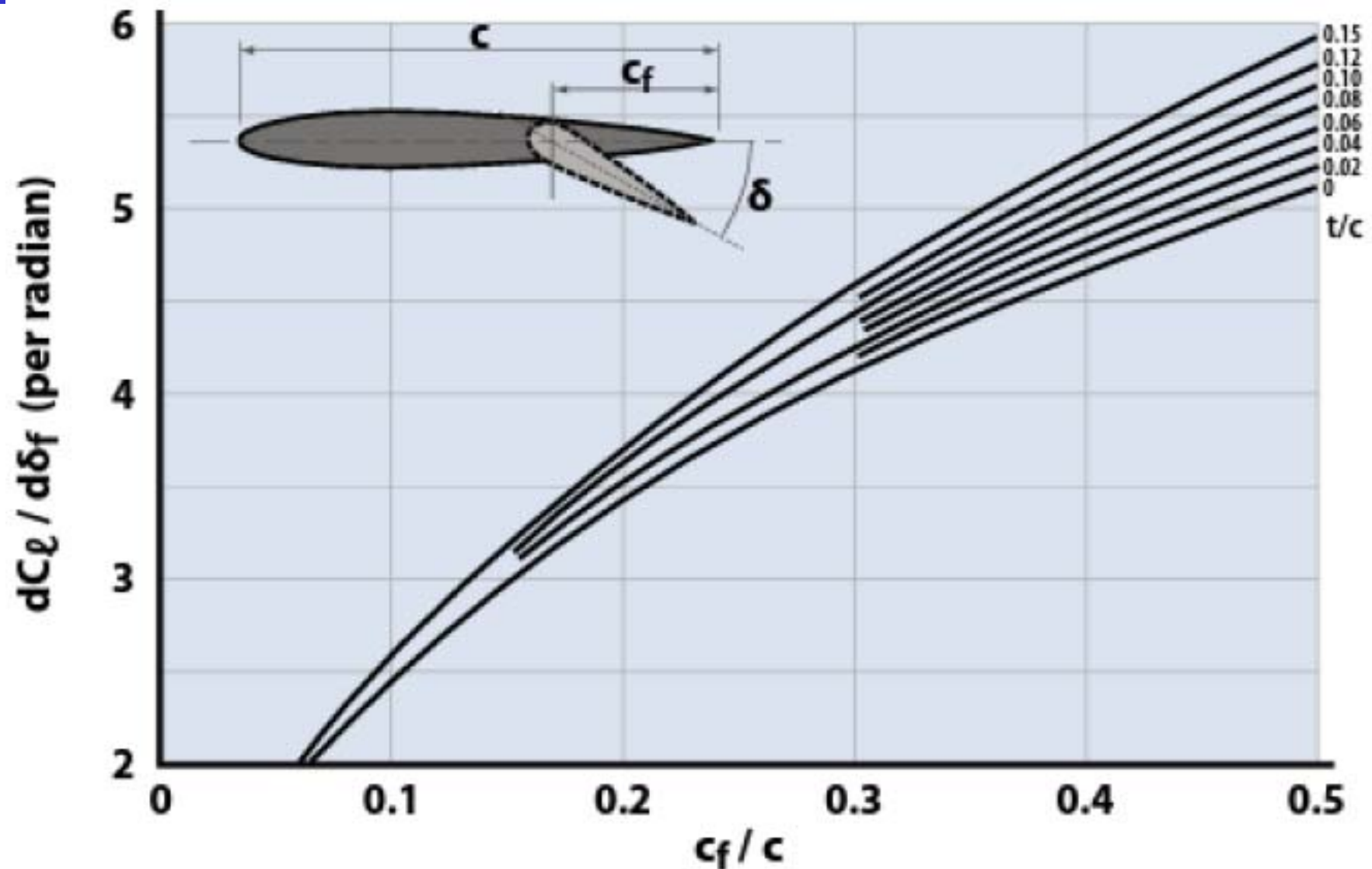
Section Lift Coefficient



**Figure 9.9** Nonlinear correction for plain TE flaps (adapted [10]).

# Plain TE Flap - II

Section Lift Coefficient



**Figure 9.10** Variation of  $dC_l / d\delta_f$  with flap chord ratio (adapted [10]).

# Cálculo $C_{Lmax}$ – Métodos Empíricos - III

## Section Lift Coefficient

- Configuración del Flap de Borde de Salida (TE): Single-Slotted Flap

$$\Delta\alpha_{0L} = \frac{d\alpha}{d\delta_f} \delta_f$$

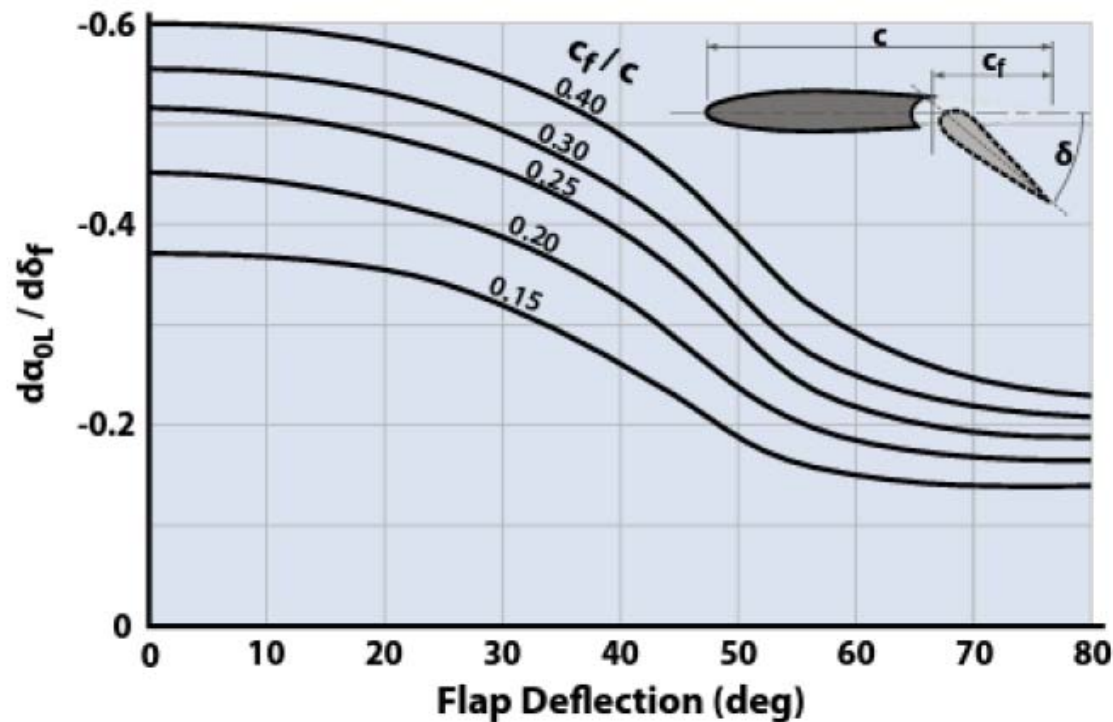
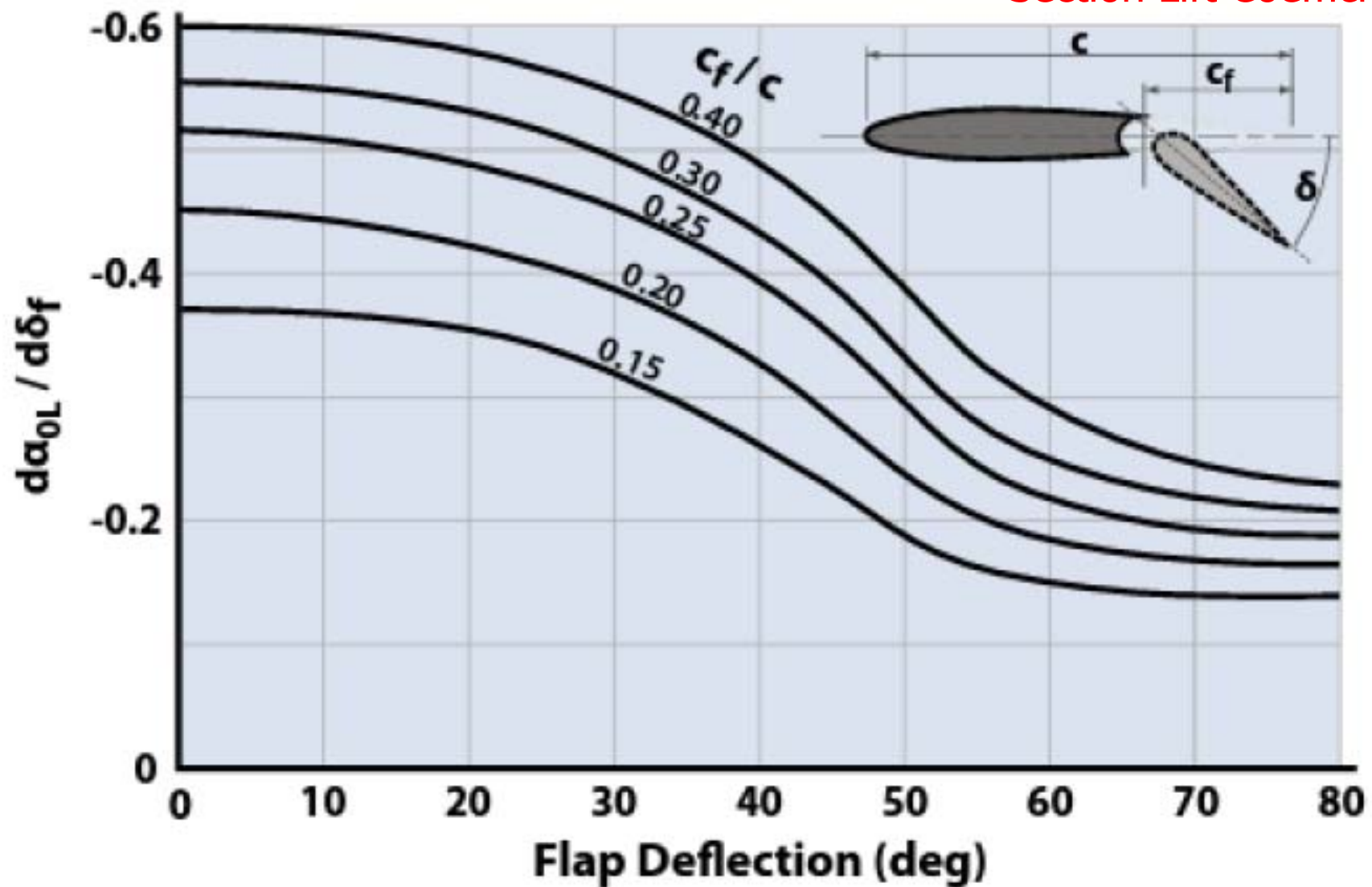


Figure 9.11 Section lift effectiveness parameter for single-slotted flaps (adapted [10]).

# Single-Slotted TE Flap - I

Section Lift Coefficient



**Figure 9.11** Section lift effectiveness parameter for single-slotted flaps (adapted [10]).

# Cálculo $C_{Lmax}$ – Métodos Empíricos - IV

Section Lift Coefficient

- Configuración del Flap de Borde de Salida (TE): Fowler Flap
  - Mismo método que para single-slotted flap
- Configuración del Flap de Borde de Salida (TE): Split Flap

$$\Delta\alpha_{0L} = -\frac{k}{C_{l\alpha}} (\Delta C_l)_{c_f/c=0.2}$$

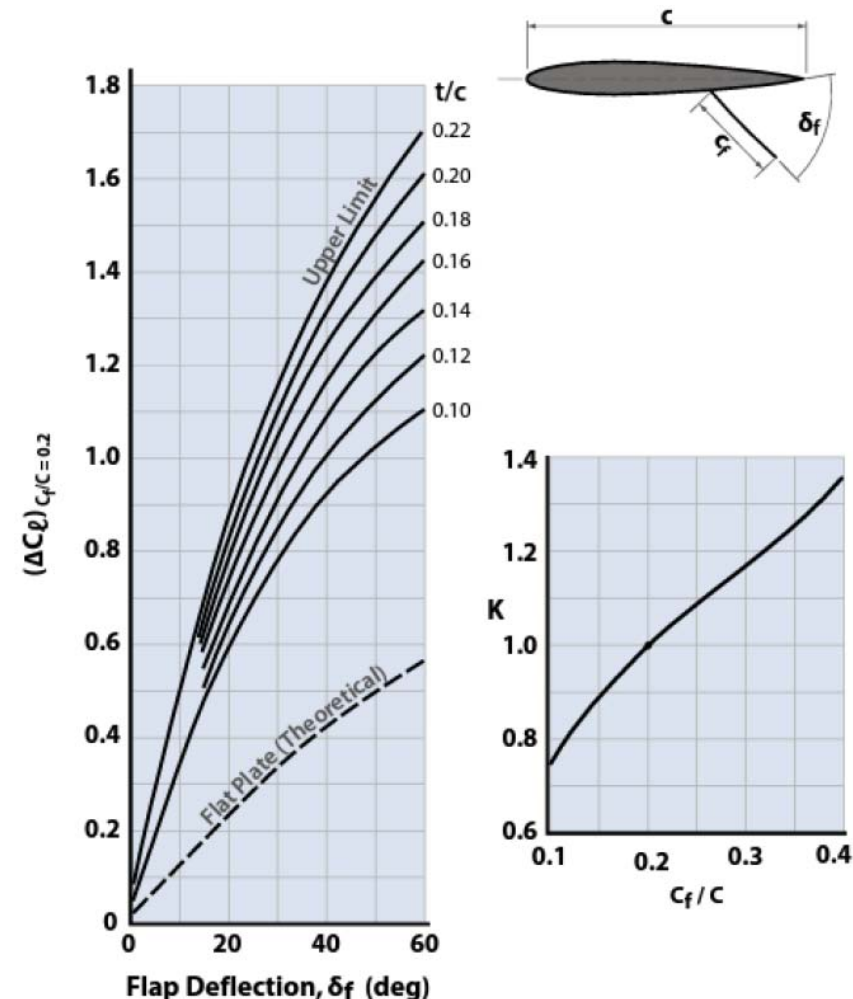


Figure 9.12 Empirical constants for split flap analysis (adapted [10]).

# Split TE Flap - I

Section Lift Coefficient

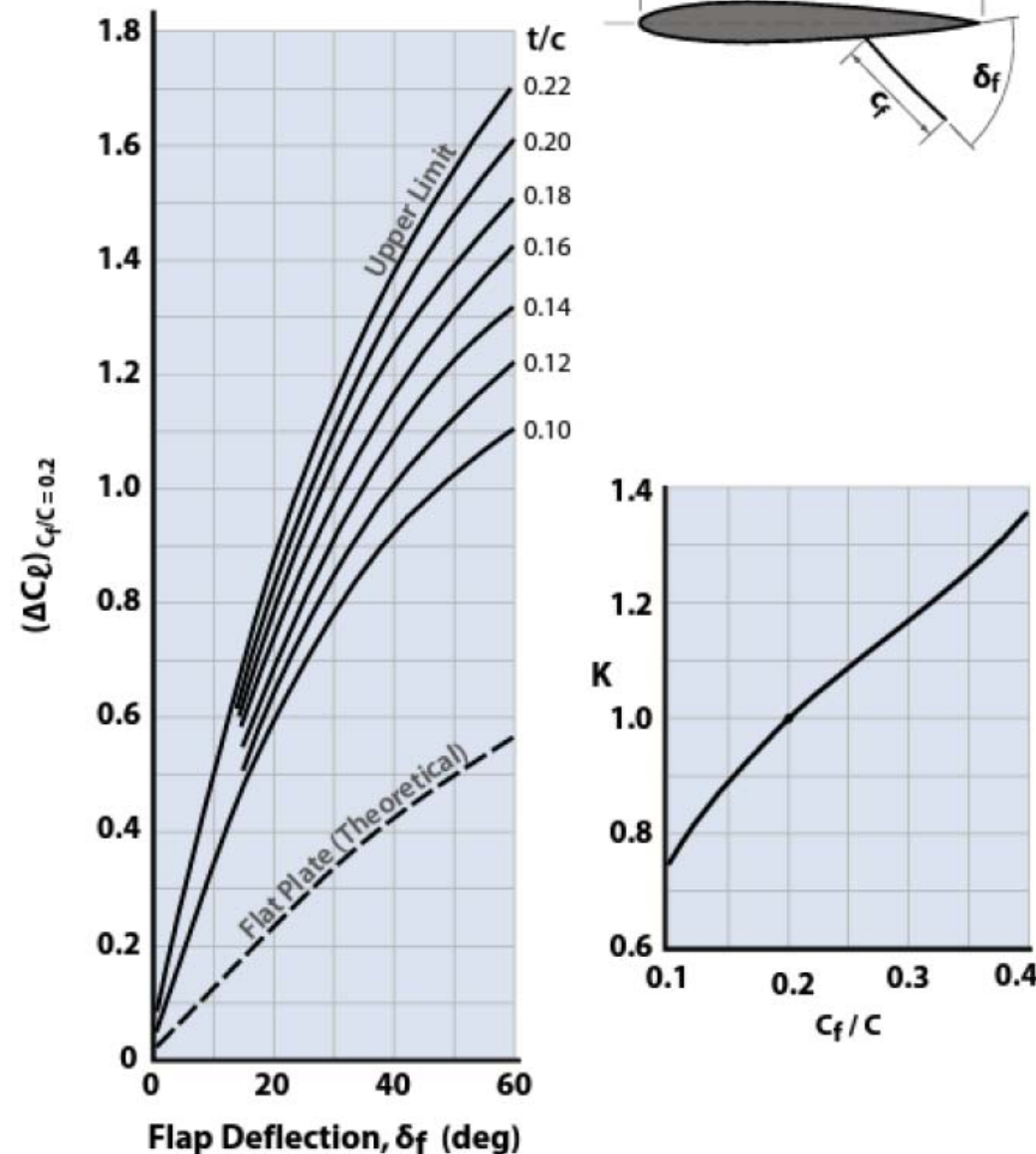
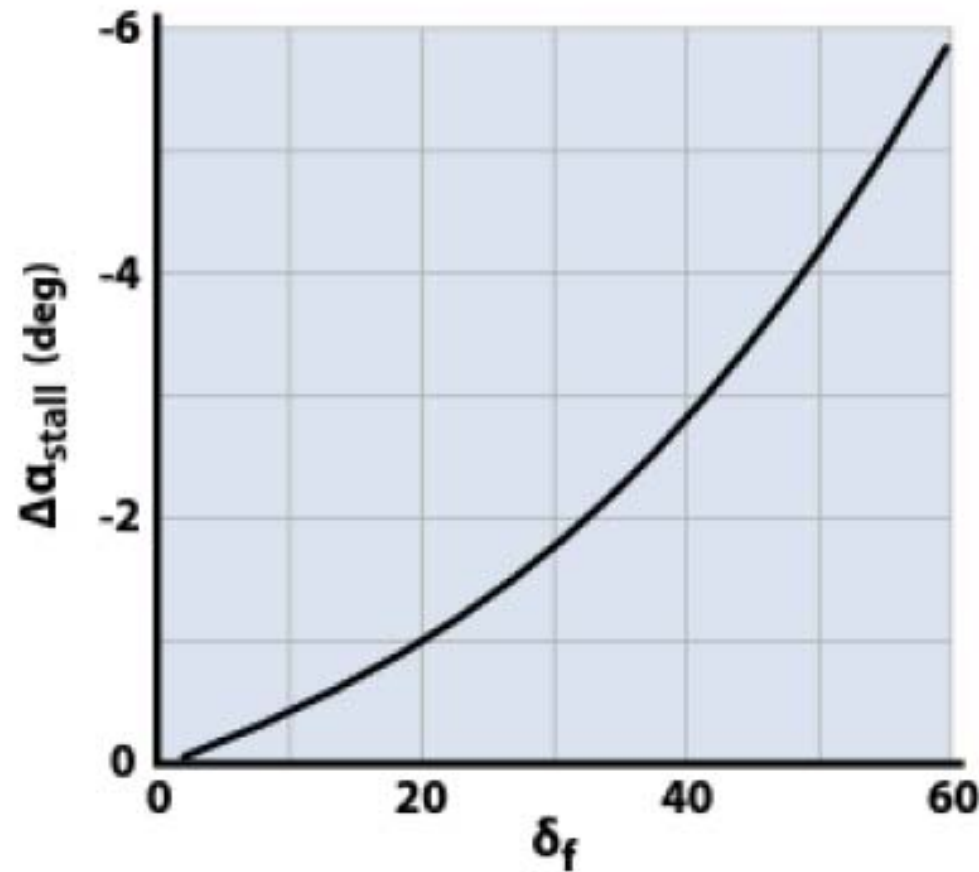


Figure 9.12 Empirical constants for split flap analysis (adapted [10]).

# Cálculo $C_{Lmax}$ – Métodos Empíricos - V

Section Lift Coefficient

- El uso de TE flaps empeora la separación, por lo que  $\alpha_{STALL}$  disminuye:  $\Delta\alpha_{STALL}$



**Figure 9.13** Decrease in stall angle with flap deflection (data from [2]).

# Cálculo $C_{L_{max}}$ – básico - I

## Correcciones 2D a 3D

### ■ Cálculo de las características del ala básica (sin HLD)

- Corrección  $C_{L\alpha}$  2D -> 3D

corregido

$$C_{L\alpha} = \frac{2\pi A}{2 + \sqrt{4 + \frac{A^2 \beta^2}{\eta^2} \left(1 + \frac{\tan^2 \Lambda_{max,t}}{\beta^2}\right)}} \left( \frac{S_{exposed}}{S_{ref}} \right) (F)$$

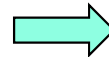
## ■ Cálculo de características básicas

### ■ Alas con bajo alargamiento (AR)

- Depende de la forma del ala

$$C_{L_{max}} = (C_{L_{max}})_{base} + \Delta C_{L_{max}}$$

$$\alpha_{stall} = (\alpha_{C_{L_{max}}})_{base} + \Delta \alpha_{C_{L_{max}}}$$



### ■ Alas con alto alargamiento (AR)

- Depende del perfil seleccionado

$$C_{L_{max}} = \frac{C_{L_{max}}}{C_{l_{max}}} C_{l_{max}}$$

$$\alpha_{stall} = \frac{C_{L_{max}}}{C_{l_{\alpha}}} + \alpha_{0L} + \Delta \alpha_{C_{L_{max}}}$$

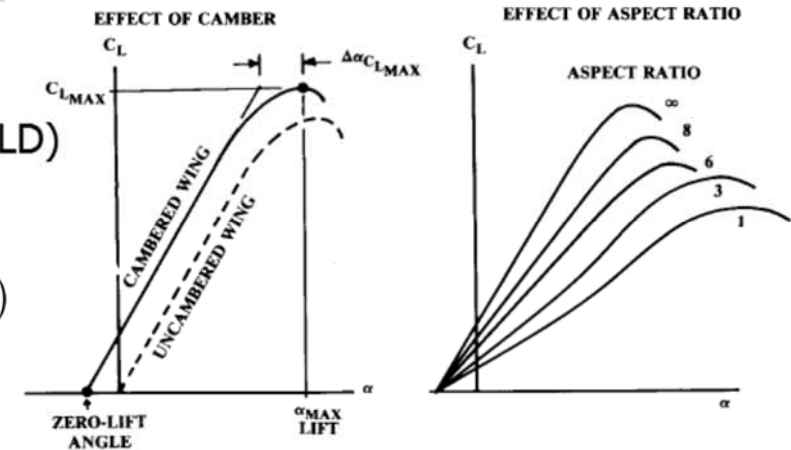
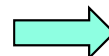


Fig. 12.4 Wing lift curve.

$$(C_{L_{max}})_{base}$$

$$\Delta C_{L_{max}}$$

$$(\alpha_{C_{L_{max}}})_{base}$$

$$\Delta \alpha_{C_{L_{max}}}$$

$$\left( \frac{C_{L_{max}}}{C_{l_{max}}} \right)$$

$$C_{l_{max}}$$

$$C_{l_{\alpha}}$$

$$\alpha_{0L}$$

$$\Delta \alpha_{C_{L_{max}}}$$

# Cálculo $C_{Lmax}$ - II

## Section Lift Coefficient

- Construir la curva  $C_L$  vs.  $\alpha$  utilizando:
  - El uso de TE flaps empeora la separación, por lo que  $\alpha_{STALL}$  disminuye
  - $\Delta\alpha_{0L}$  y  $\Delta\alpha_{STALL}$  por lo que se puede determinar  $\Delta C_{Lmax}$

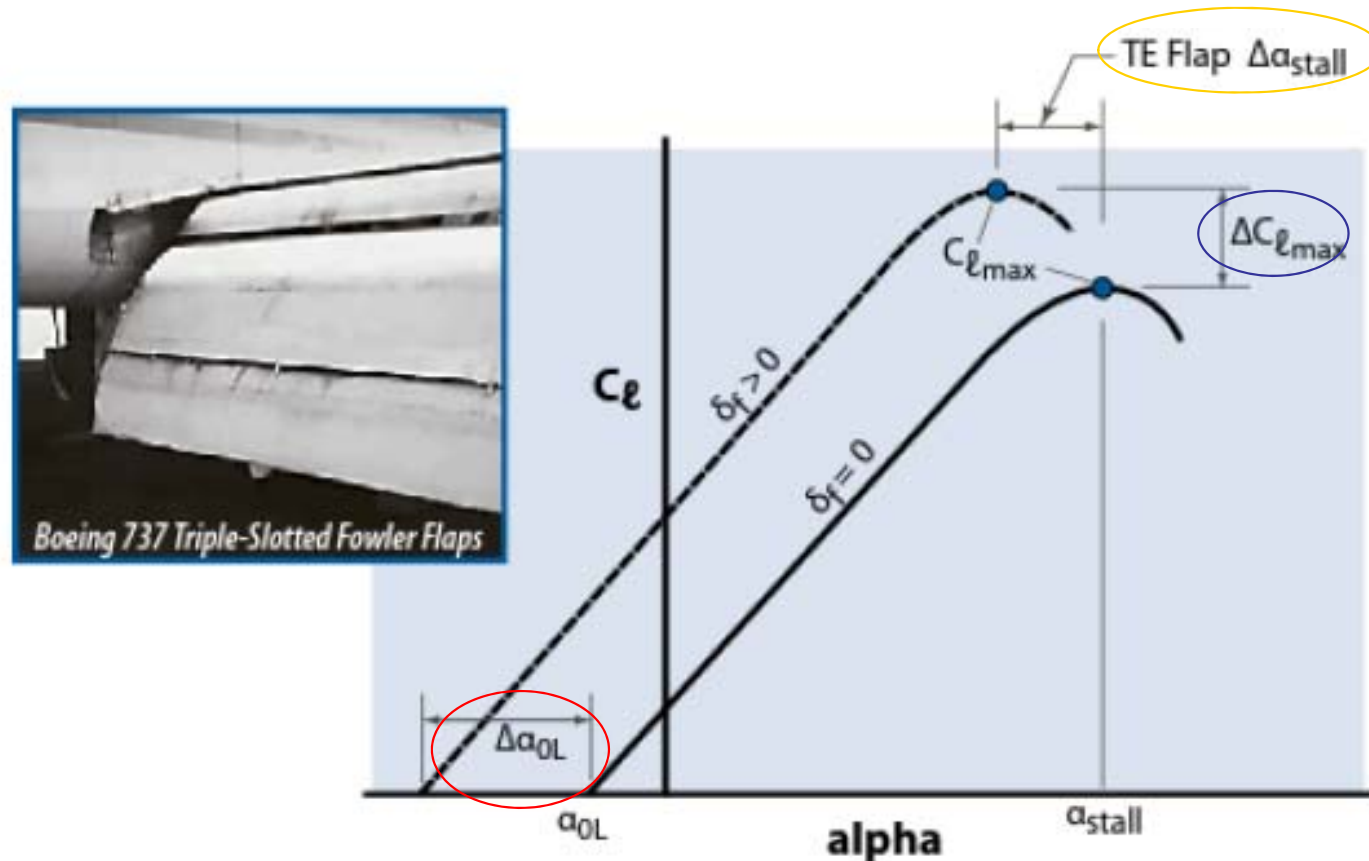


Figure 9.8 Construction of section lift curves for TE flaps.

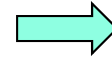
# Cálculo $C_{Lmax}$ - III

Wing Lift Coefficient

- Para determinar  $C_{Lmax}$  se distingue entre :

- Alas con alto alargamiento (AR)

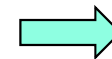
- Depende del perfil seleccionado



$$A > \frac{4}{(C_1 + 1) \cos \Delta_{LE}}$$

- Alas con bajo alargamiento (AR)

- Depende de la forma del ala



$$A < \frac{4}{(C_1 + 1) \cos \Delta_{LE}}$$

$\Delta_{LE}$  = leading edge sweep

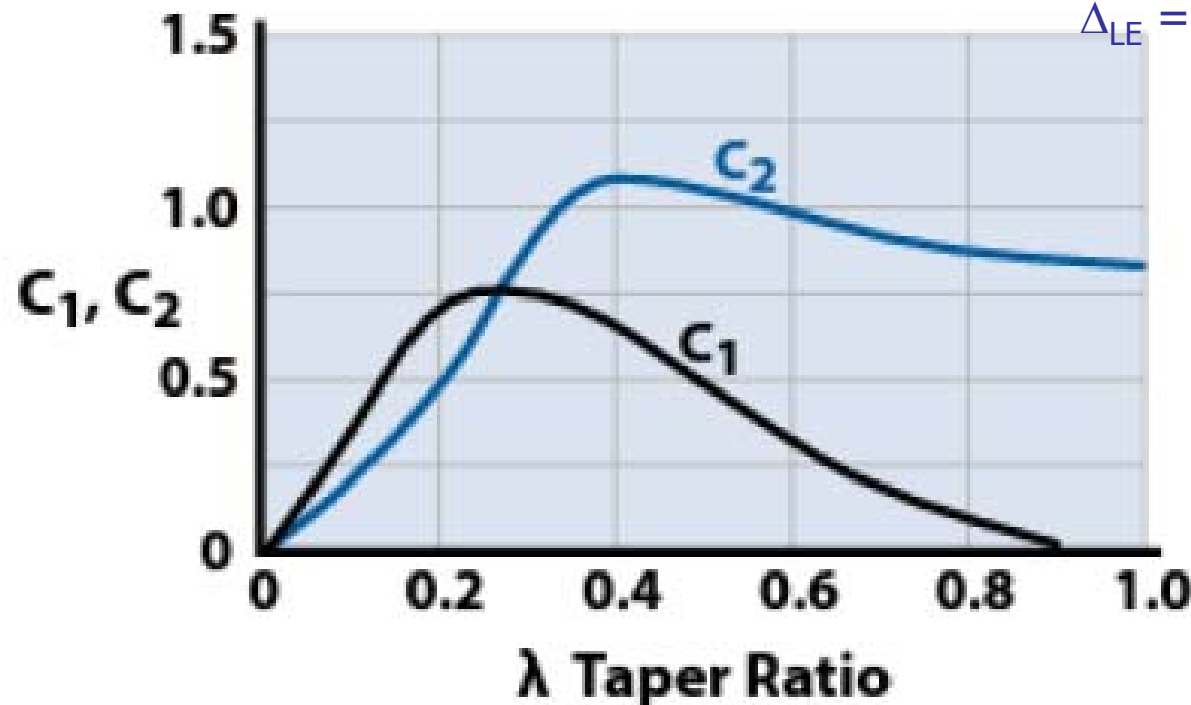


Figure 9.14 Taper ratio correction factors (adapted [10]).

# Cálculo $C_{L_{max}}$ – High Aspect Ratio - I

Wing Lift Coefficient

- Obtener  $C_{L_{max}}$  y  $\alpha_{STALL}$  para el ala básica      Ángulo de ataque para sustentación nula

$$C_{L_{max}} = \frac{C_{L_{max}}}{C_{l_{max}}} C_{l_{max}}$$

$$\alpha_{stall} = \frac{C_{L_{max}}}{C_{L_{\alpha}}} + \alpha_{0L} + \Delta\alpha_{C_{L_{max}}}$$

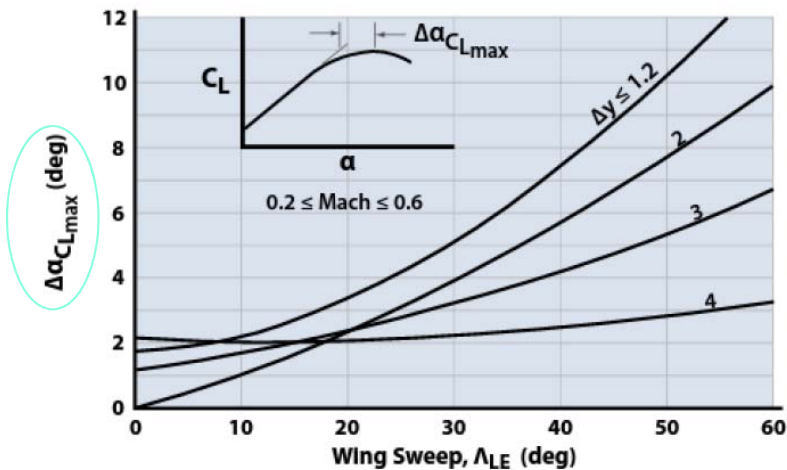


Figure 9.16 Angle-of-attack increment for subsonic maximum lift of high-AR wings (adapted [10]).

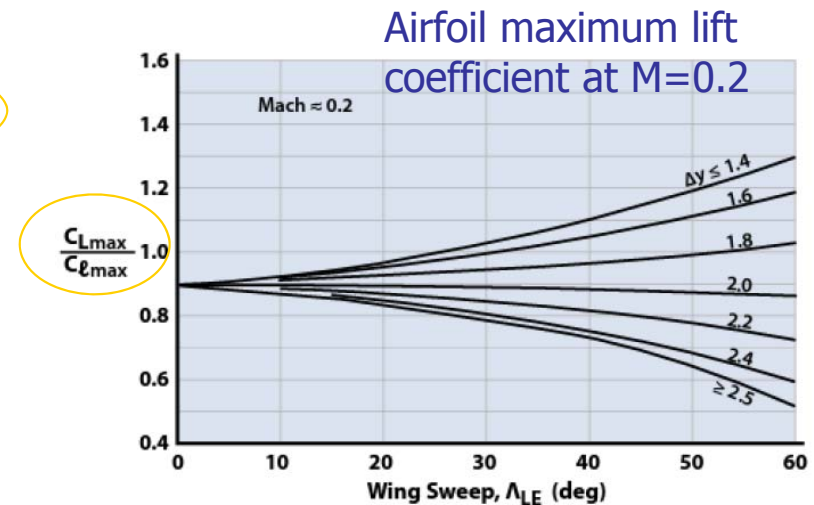


Figure 9.15 Subsonic maximum lift of high-AR wings (adapted [10]).

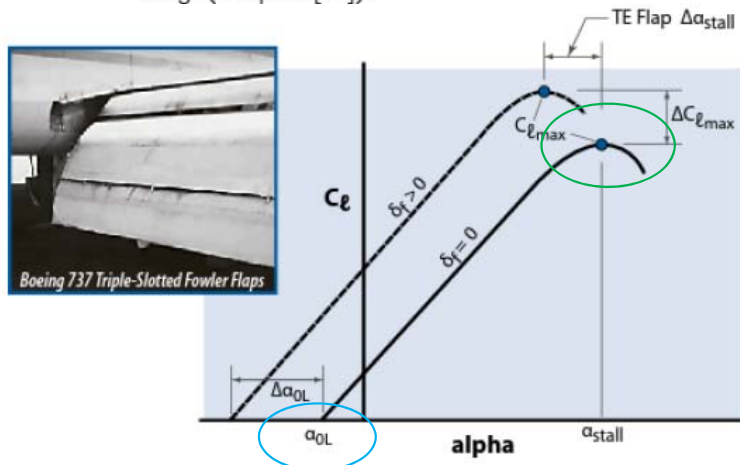


Figure 9.8 Construction of section lift curves for TE flaps.

corregido

$$C_{L_{\alpha}} = \frac{2\pi A}{2 + \sqrt{4 + \frac{A^2 \beta^2}{\eta^2} \left(1 + \frac{\tan^2 \Lambda_{max,t}}{\beta^2}\right)}} \left(\frac{S_{exposed}}{S_{ref}}\right) (F)$$

# Cálculo $C_{L_{max}}$ – High Aspect Ratio - I

Wing Lift Coefficient

- Obtener  $C_{L_{max}}$  y  $\alpha_{STALL}$  para el ala básica

$$C_{L_{max}} = \frac{C_{L_{max}}}{C_{l_{max}}} C_{l_{max}}$$

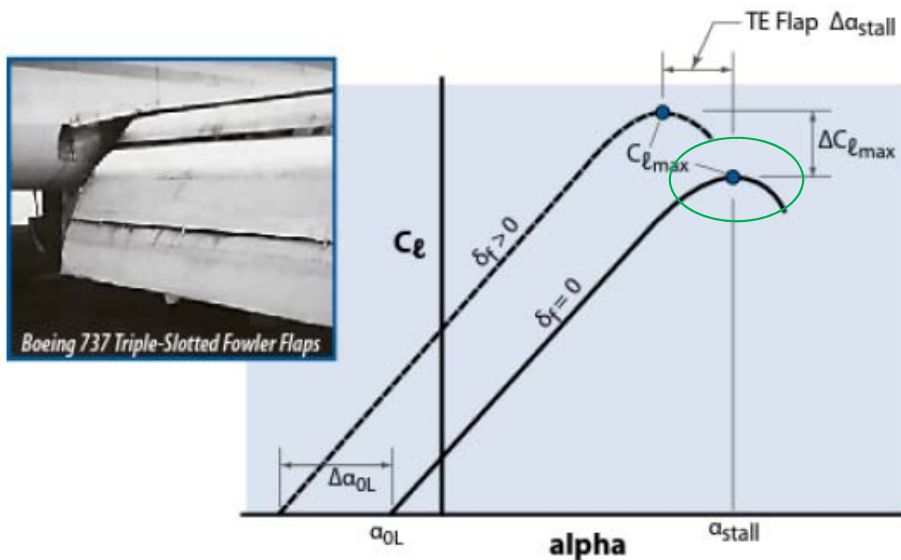


Figure 9.8 Construction of section lift curves for TE flaps.

Airfoil maximum lift coefficient at  $M=0.2$

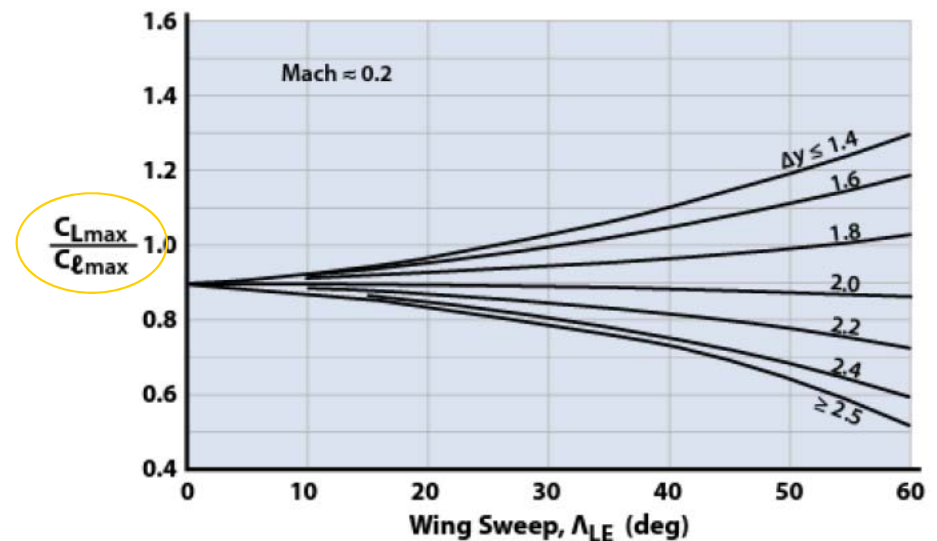


Figure 9.15 Subsonic maximum lift of high-AR wings (adapted [10]).

# Cálculo $C_{L_{max}}$ – High Aspect Ratio - I

Wing Lift Coefficient

- Obtener  $C_{L_{max}}$  y  $\alpha_{STALL}$  para el ala básica
  - Igual de importante es el ángulo al que se obtiene la máxima sustentación:
    - Los dos primeros términos representan el ángulo de ataque si la pendiente de la curva tuviera propiedades lineales hasta llegar a la zona de entrada en pérdida

$$C_{L_{max}} = \frac{C_{L_{max}}}{C_{l_{max}}} C_{l_{max}}$$

$$\alpha_{stall} = \frac{C_{L_{max}}}{C_{L_{\alpha}}} + \alpha_{0L} + \Delta\alpha_{C_{L_{max}}}$$

Corrección de los efectos no lineales del flujo de vórtices

Ángulo de ataque para sustentación nula

Propiedades lineales del ala

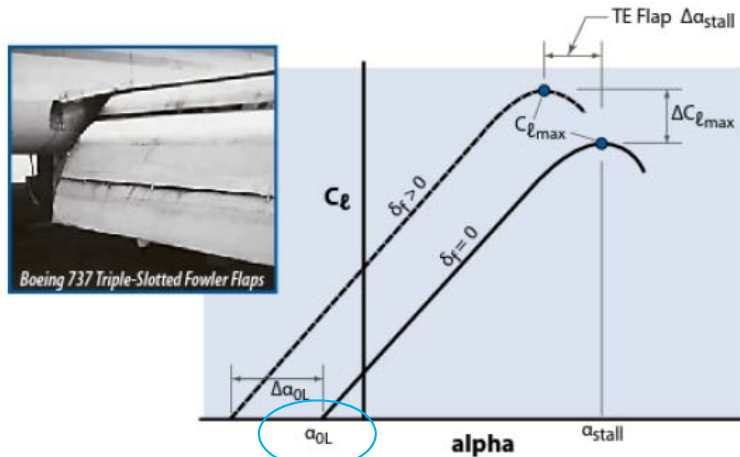


Figure 9.8 Construction of section lift curves for TE flaps.

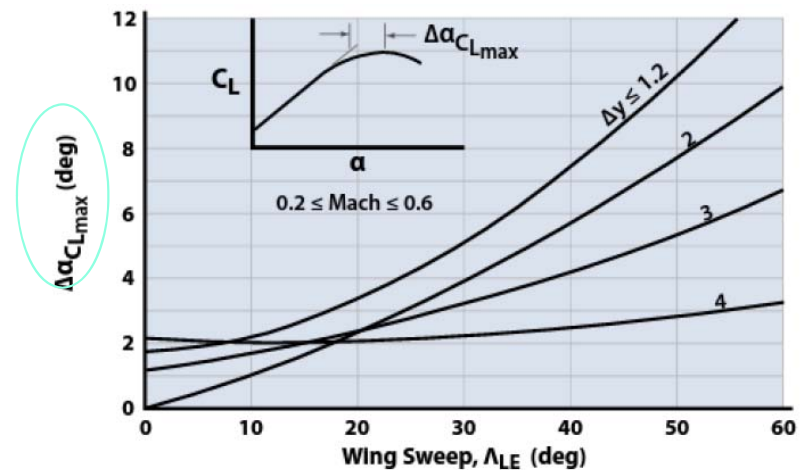
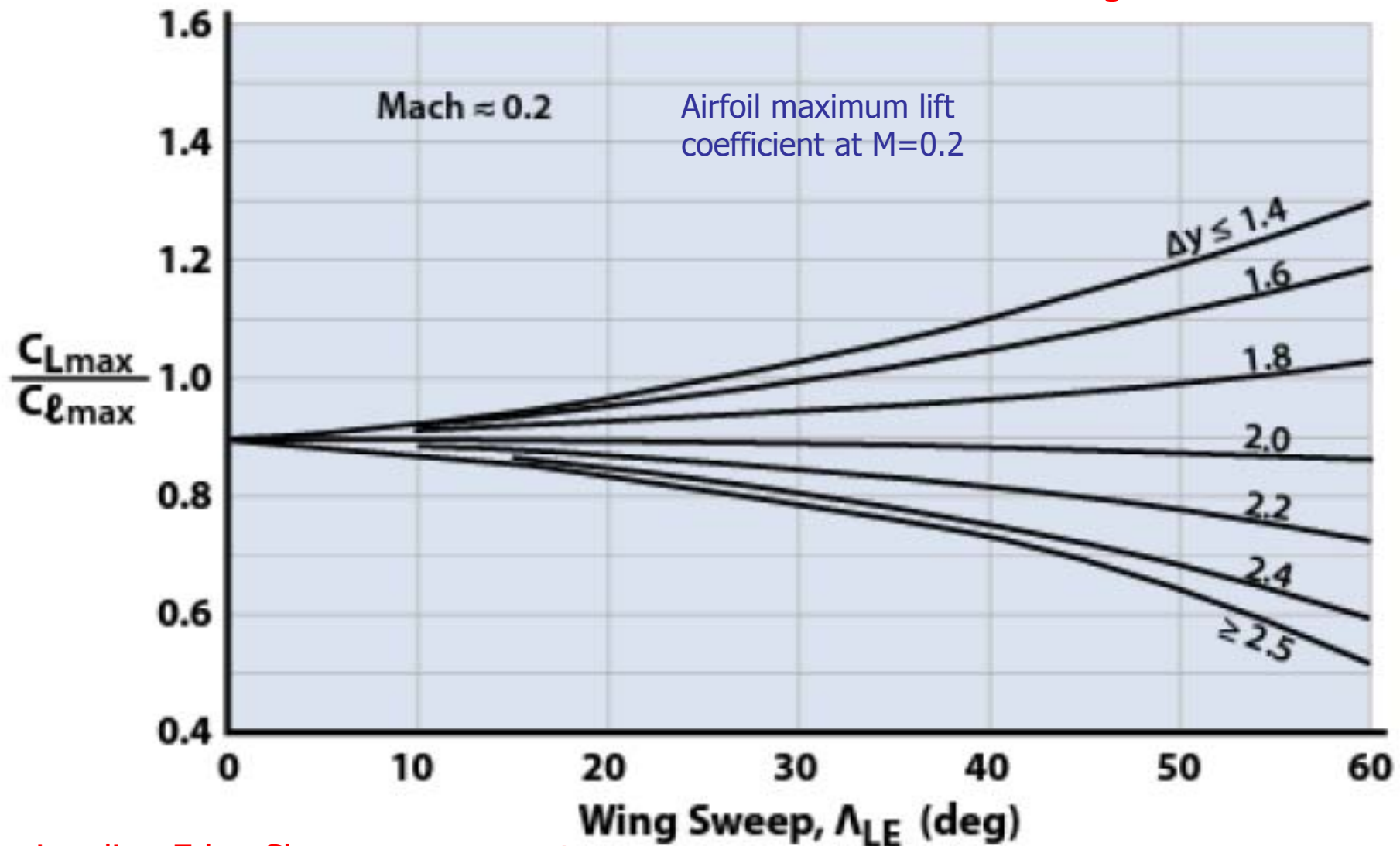


Figure 9.16 Angle-of-attack increment for subsonic maximum lift of high-AR wings (adapted [10]).

$$C_{L_{\alpha}} = \frac{2\pi A}{2 + \sqrt{4 + \frac{A^2 \beta^2}{\eta^2} \left(1 + \frac{\tan^2 \Lambda_{max,t}}{\beta^2}\right)}} \left(\frac{S_{exposed}}{S_{ref}}\right) (F)$$

# Cálculo $C_{Lmax}$ – High Aspect Ratio - II

Wing Lift Coefficient

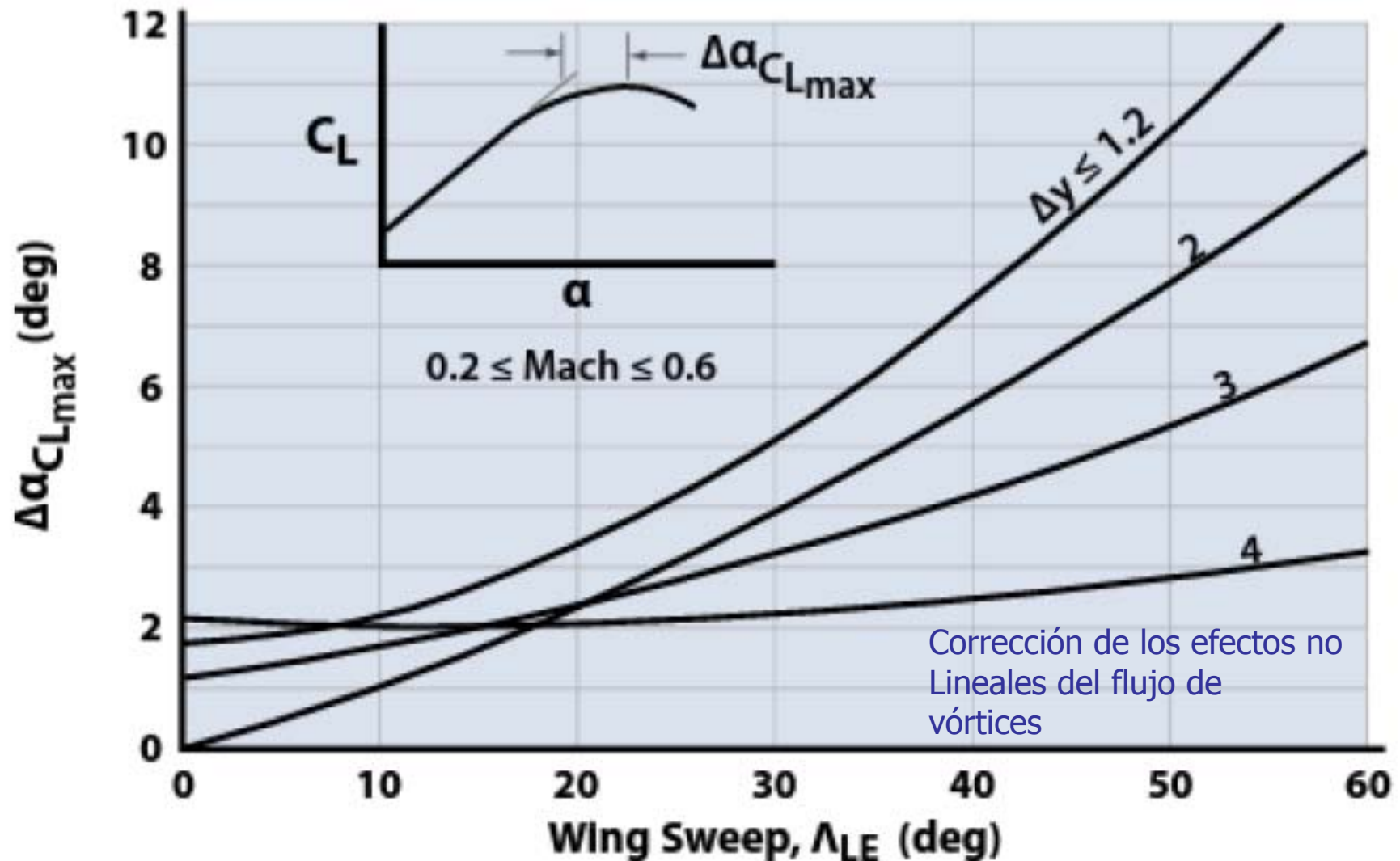


Leading Edge Sharpness parameter:  $\Delta y$

Figure 9.15 Subsonic maximum lift of high-AR wings (adapted [10]).

# Cálculo $C_{Lmax}$ – High Aspect Ratio - III

Wing Lift Coefficient



Leading Edge Sharpness parameter:  $\Delta y$

**Figure 9.16** Angle-of-attack increment for subsonic maximum lift of high-AR wings (adapted [10]).

# Leading Edge Sharpness parameter : $\Delta y$

Wing Lift Coefficient

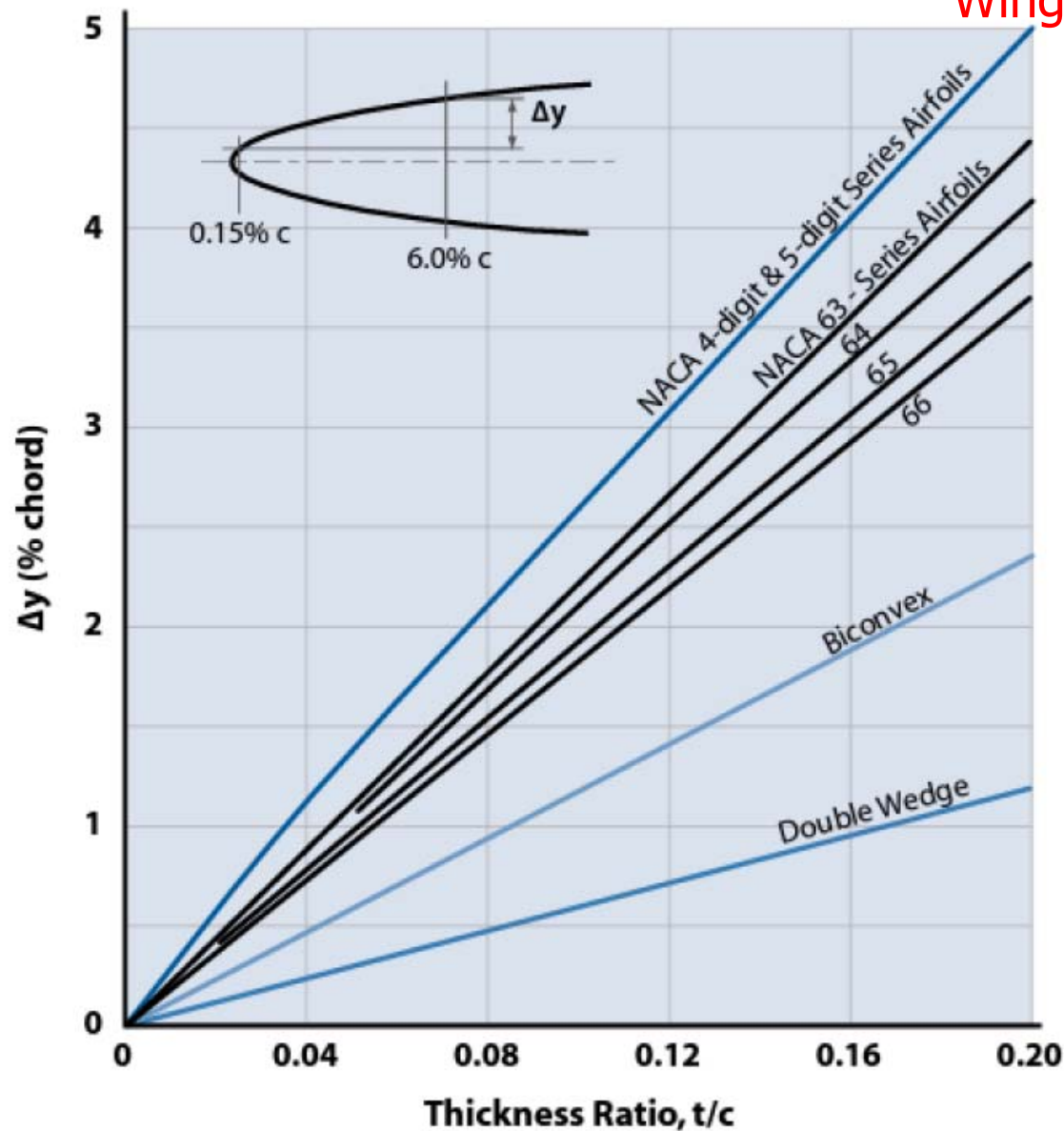
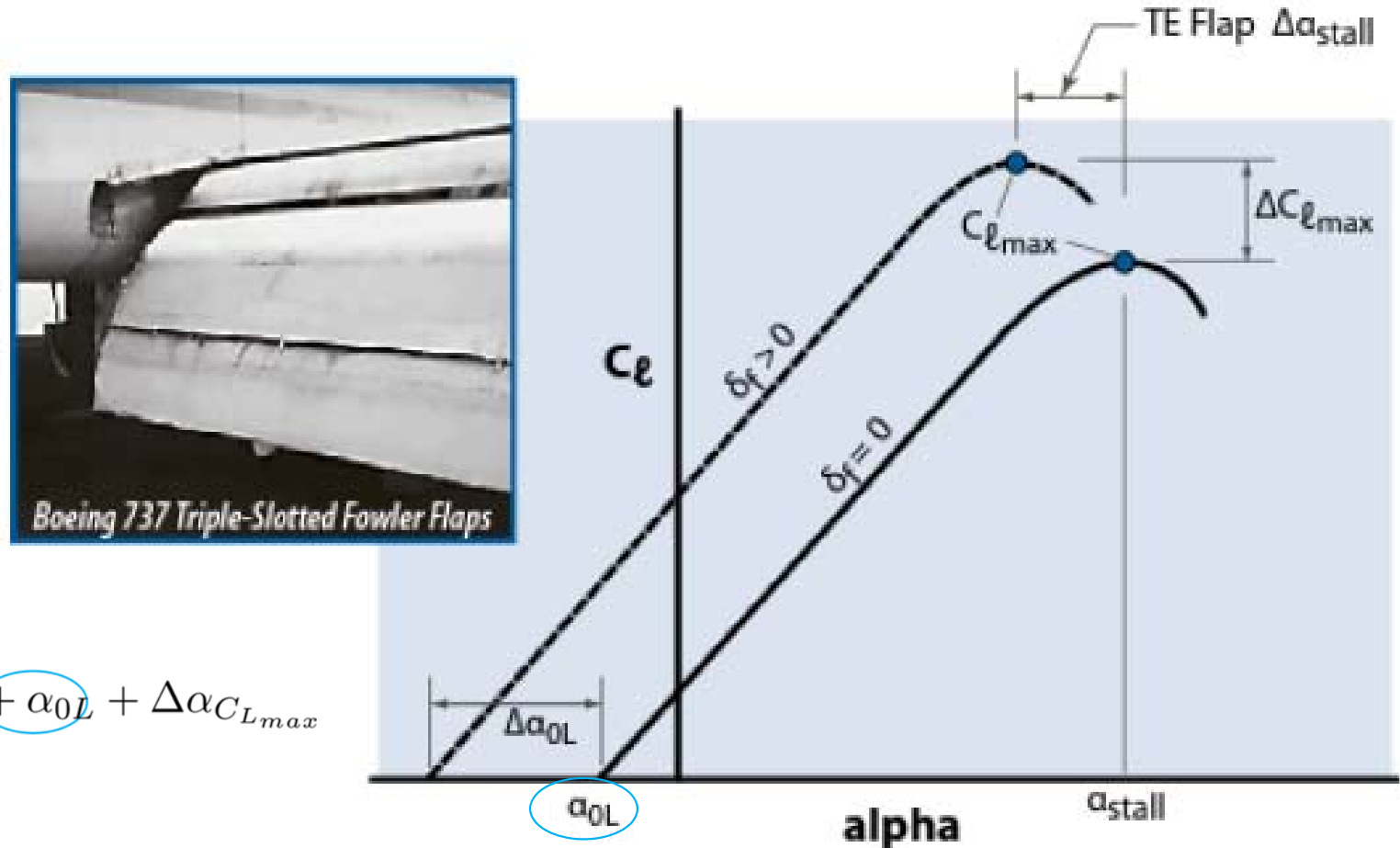


Figure 9.17 Variation of LE sharpness parameter with airfoil thickness ratio (adapted [10]).

# Cálculo $C_{L_{max}}$ – High Aspect Ratio - IV

Wing Lift Coefficient

$$C_{L_{max}} = \frac{C_{L_{max}}}{C_{l_{max}}} C_{l_{max}} \quad \text{Unflapped section maximum lift coefficient}$$



$$\alpha_{stall} = \frac{C_{L_{max}}}{C_{L_{\alpha}}} + \alpha_{0L} + \Delta\alpha_{C_{L_{max}}}$$

Figure 9.8 Construction of section lift curves for TE flaps.

# Cálculo $C_{Lmax}$ – Low Aspect Ratio - I

Wing Lift Coefficient

- Obtener  $C_{Lmax}$  y  $\alpha_{STALL}$  para el ala básica

$$C_{Lmax} = (C_{Lmax})_{base} + \Delta C_{Lmax}$$

$$\alpha_{stall} = (\alpha_{C_{Lmax}})_{base} + \Delta \alpha_{C_{Lmax}}$$

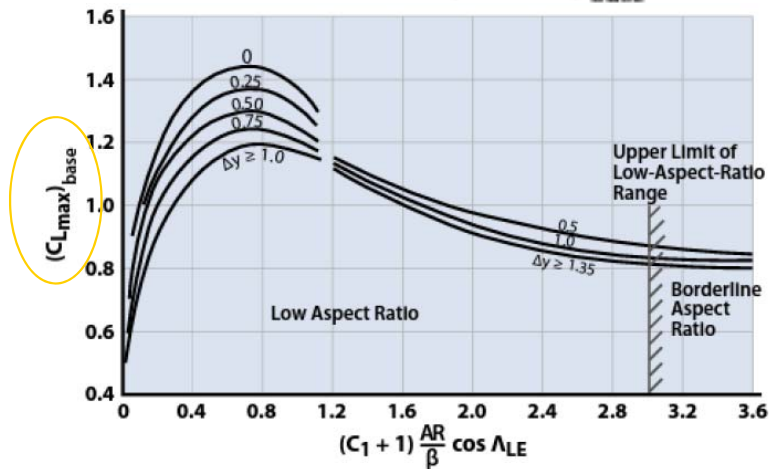


Figure 9.18 Subsonic maximum lift of low-AR wings (adapted [10]).

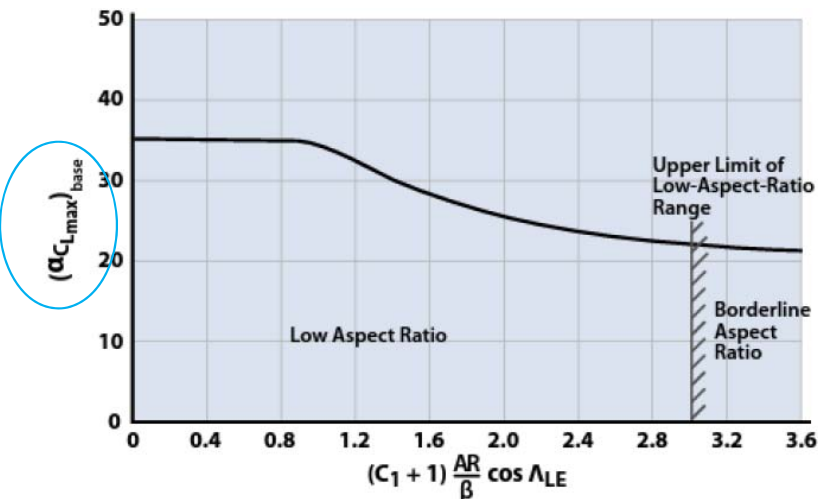


Figure 9.20 Angle-of-attack for subsonic maximum lift of low-AR wings.

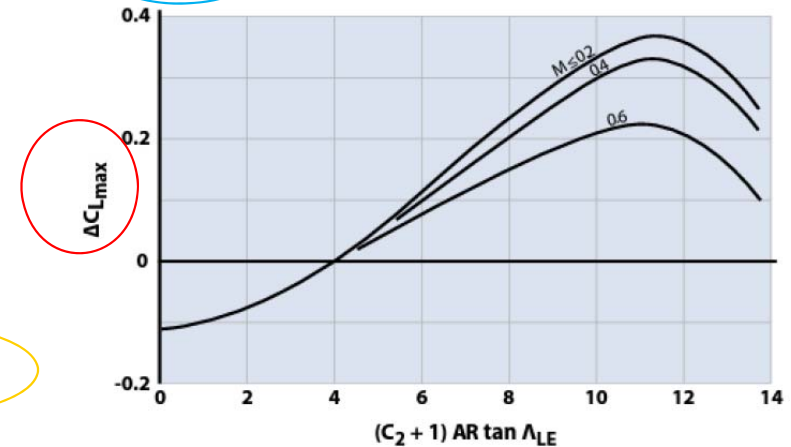


Figure 9.19 Subsonic maximum-lift increment for low-AR wings (adapted [10]).

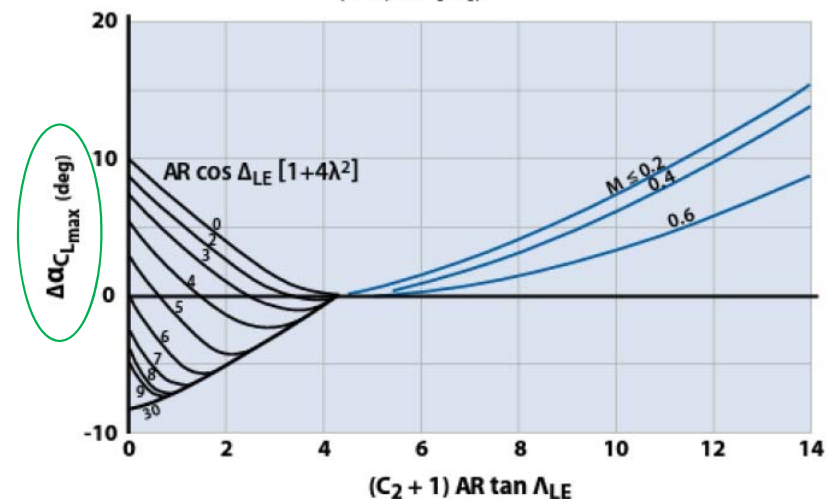


Figure 9.21 Angle-of-attack increment for subsonic maximum lift of low-AR wings.

# Cálculo $C_{Lmax}$ – Low Aspect Ratio - I

Wing Lift Coefficient

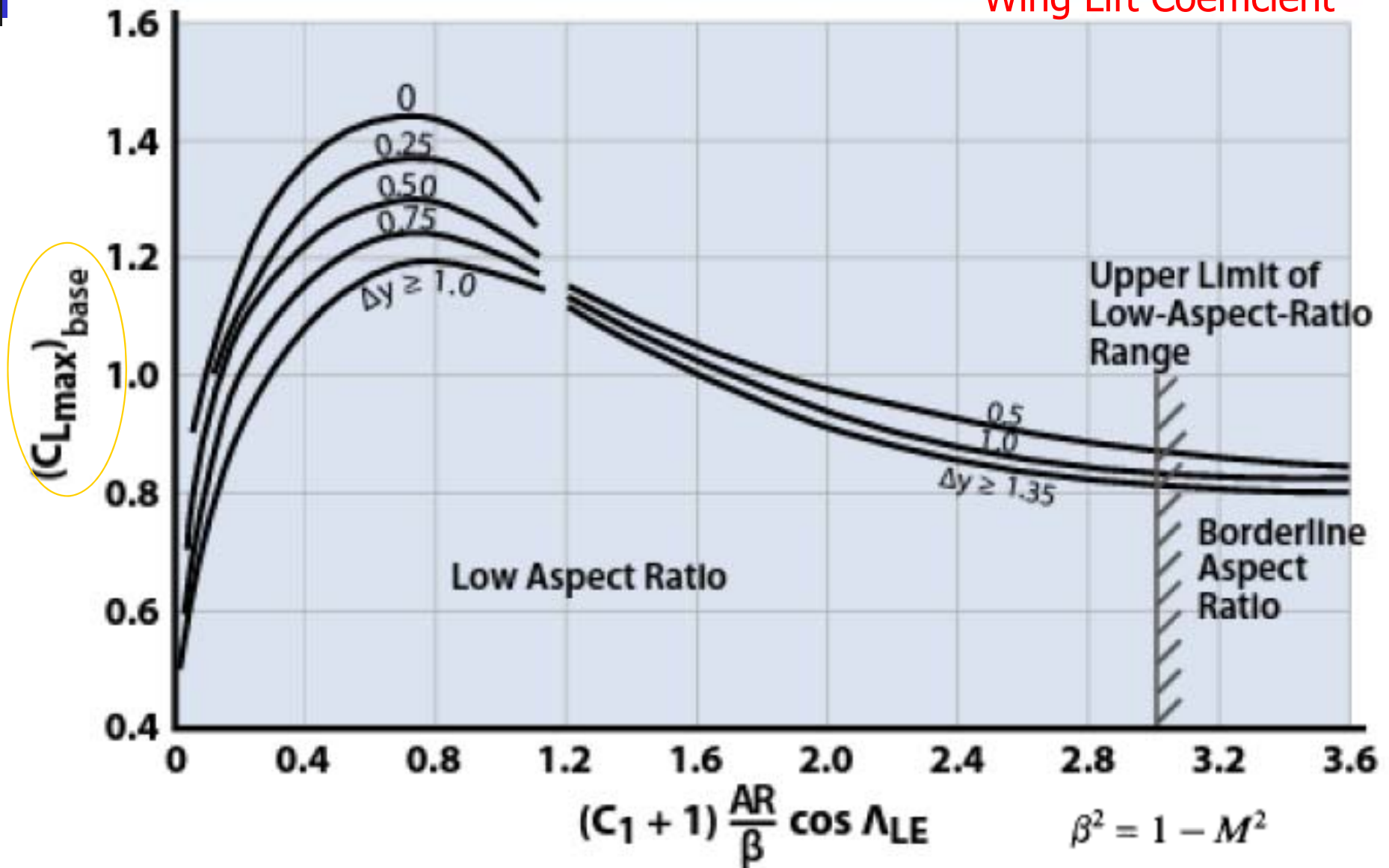


Figure 9.18 Subsonic maximum lift of low-AR wings (adapted [10]).

# Cálculo $C_{Lmax}$ – Low Aspect Ratio - II

Wing Lift Coefficient

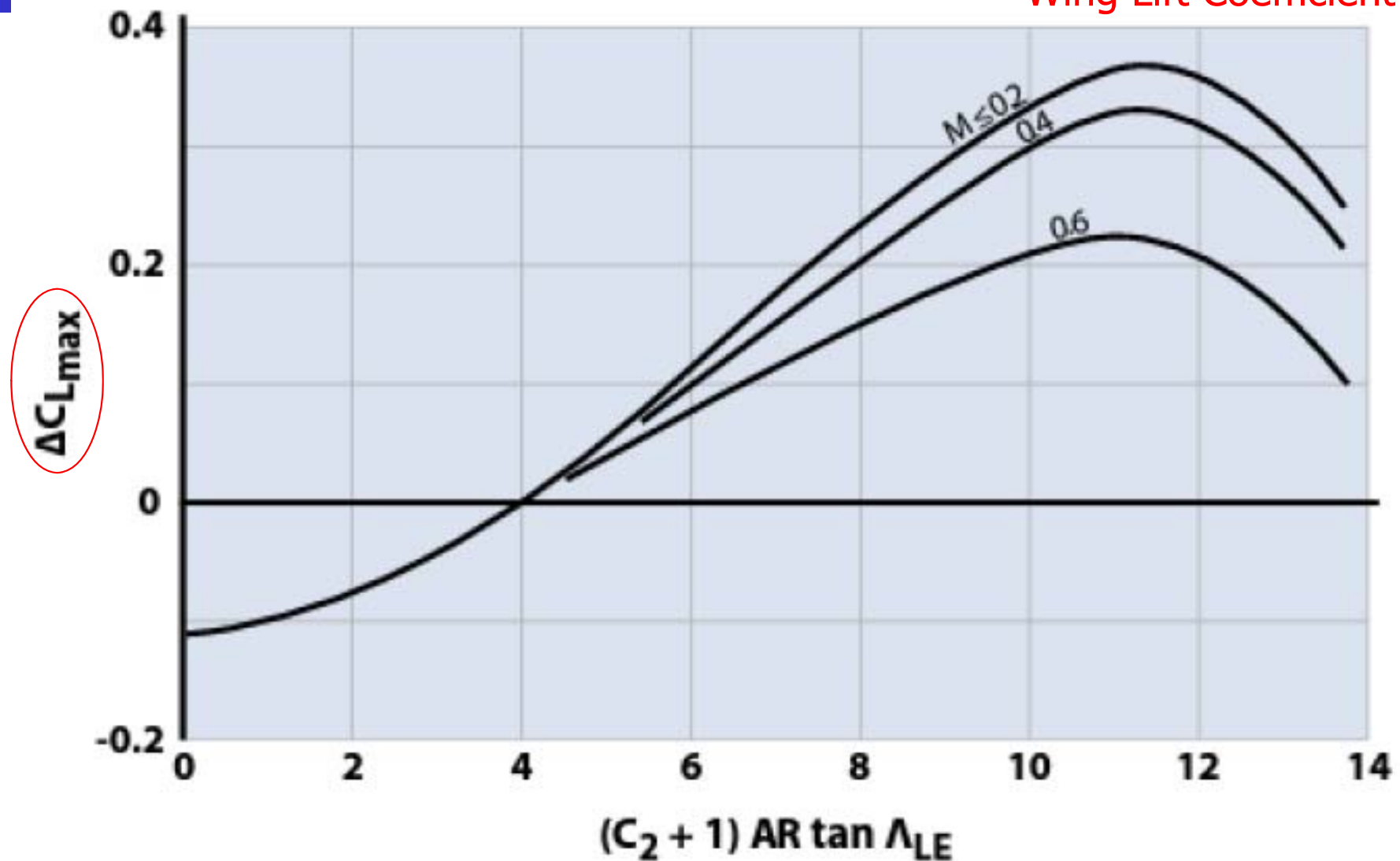
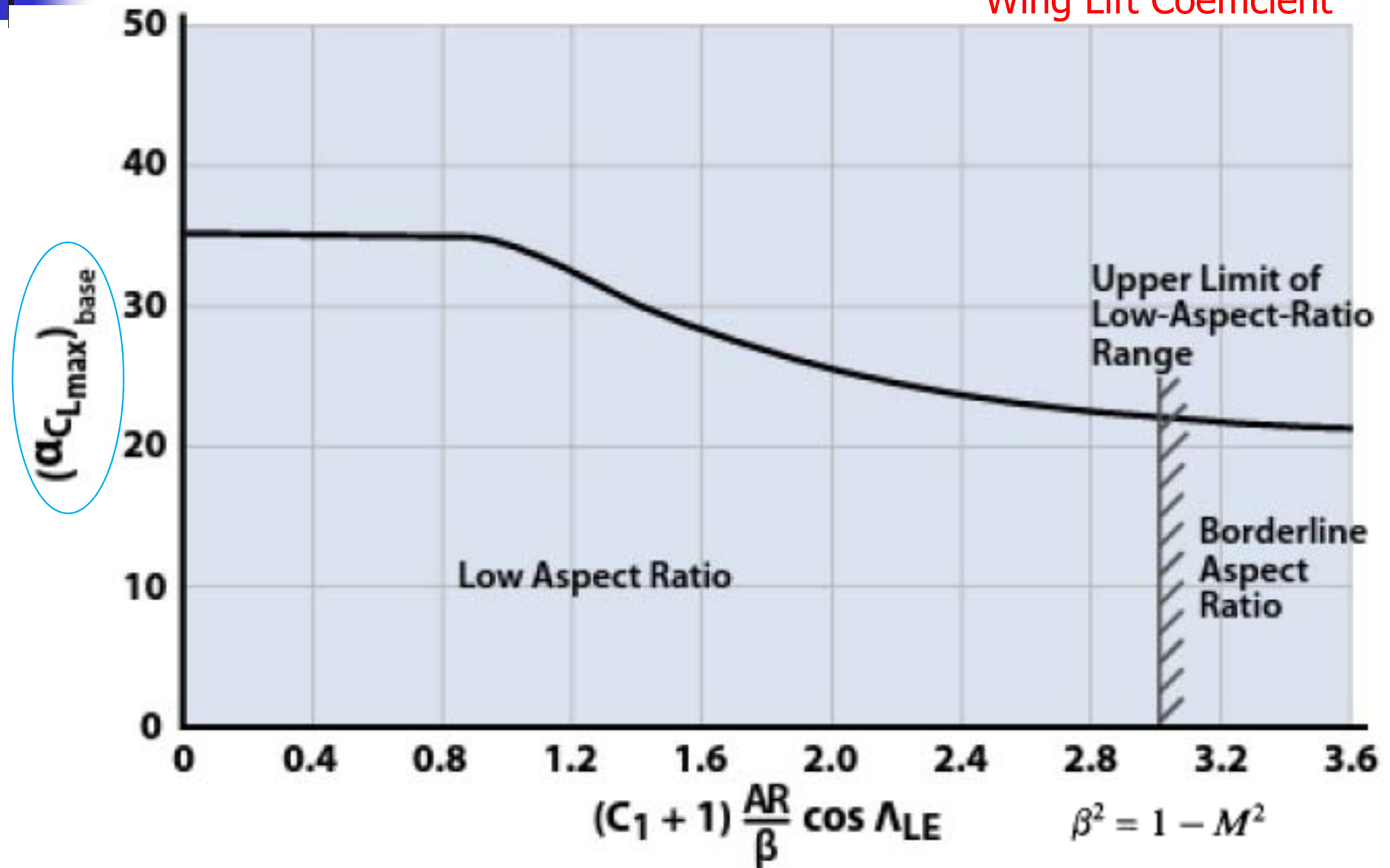


Figure 9.19 Subsonic maximum-lift increment for low-AR wings (adapted [10]).

# Cálculo $C_{Lmax}$ – Low Aspect Ratio - III

Wing Lift Coefficient



**Figure 9.20** Angle-of-attack for subsonic maximum lift of low-AR wings.

# Cálculo $C_{Lmax}$ – Low Aspect Ratio - IV

Wing Lift Coefficient

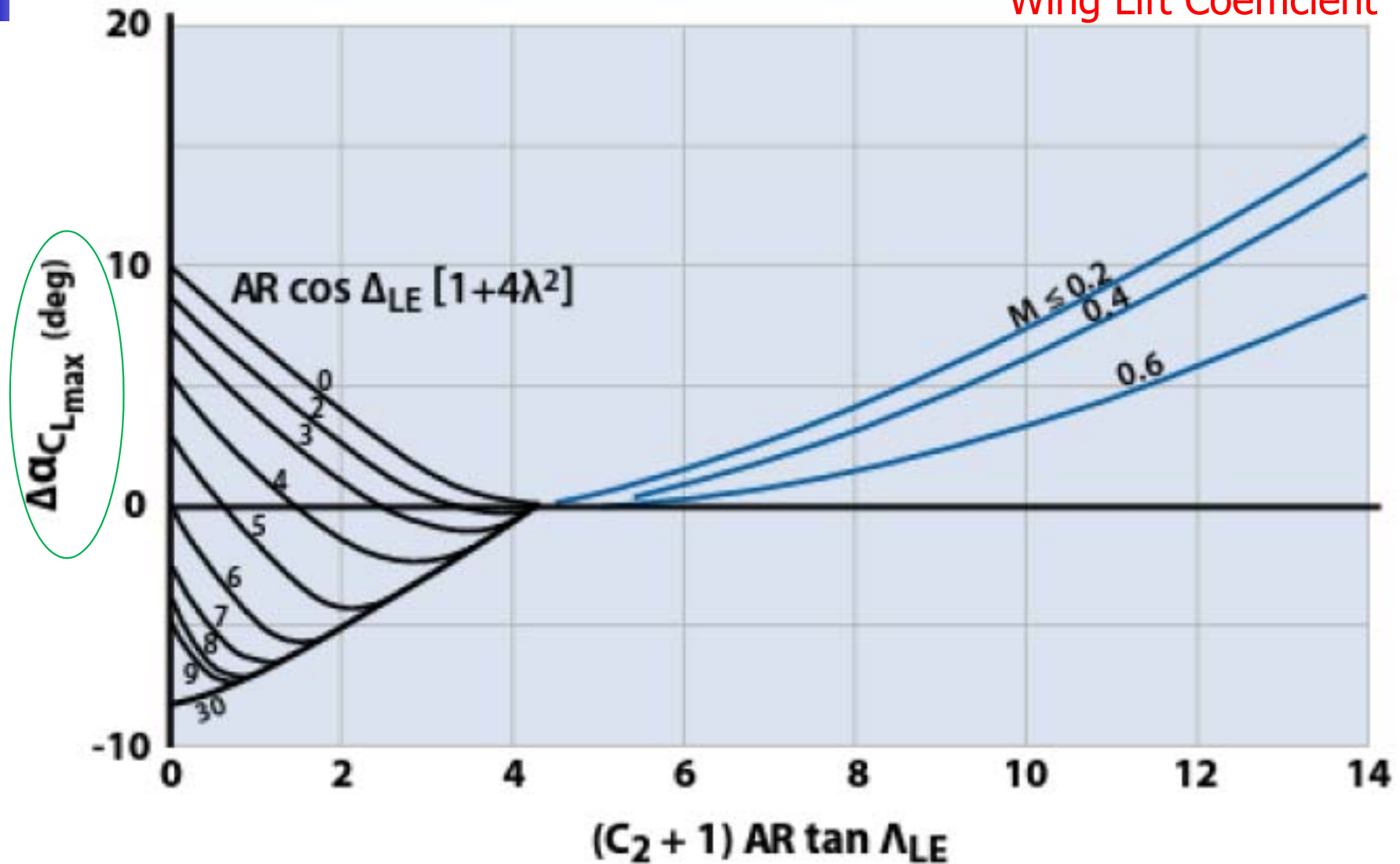


Figure 9.21 Angle-of-attack increment for subsonic maximum lift of low-AR wings.

# Cálculo $C_{Lmax}$ - IV

Wing Lift Coefficient with TE HLD

- Construir la curva  $C_L$  vs.  $\alpha$  para el ala finita teniendo en cuenta geometría:
  - Superficie con flap
  - Corrección por flecha

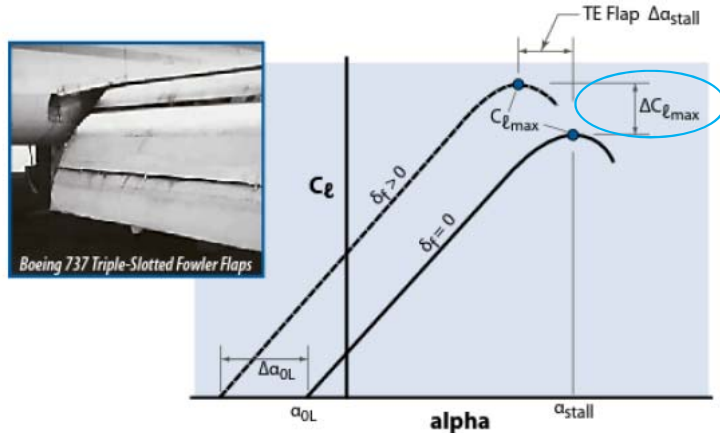


Figure 9.8 Construction of section lift curves for TE flaps.

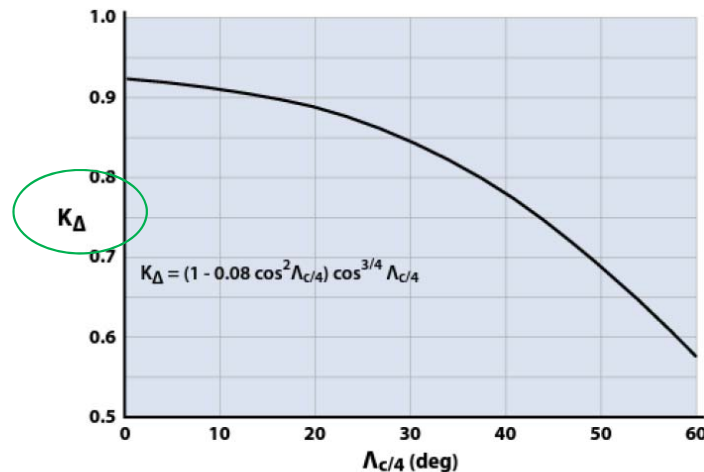


Figure 9.23 Planform correction factors for TE flaps (adapted [10]).

$$\Delta C_{Lmax} = \Delta C_{lmax} \frac{S_{WF}}{S_W} K_\Delta$$

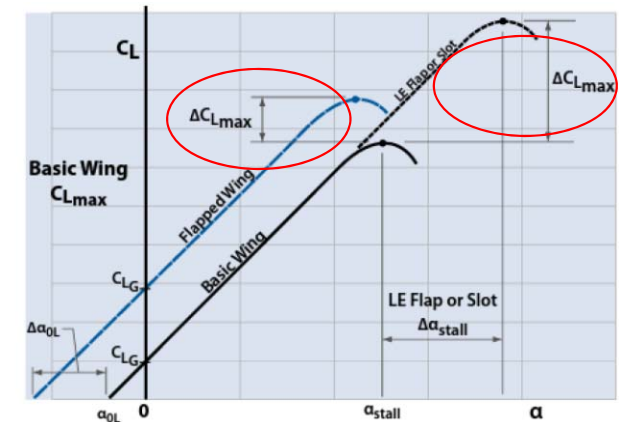


Figure 9.22 Construction of wing lift curves for mechanical high-lift devices.

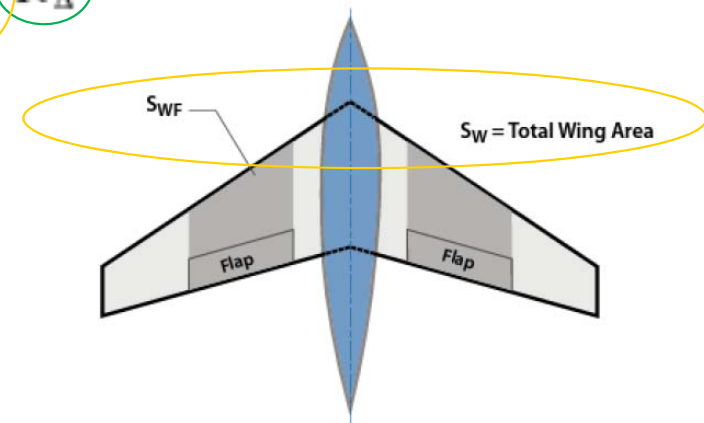


Figure 9.24 Schematic showing flapped wing area.

# Cálculo $C_{Lmax}$ - V

Wing Lift Coefficient with TE HLD

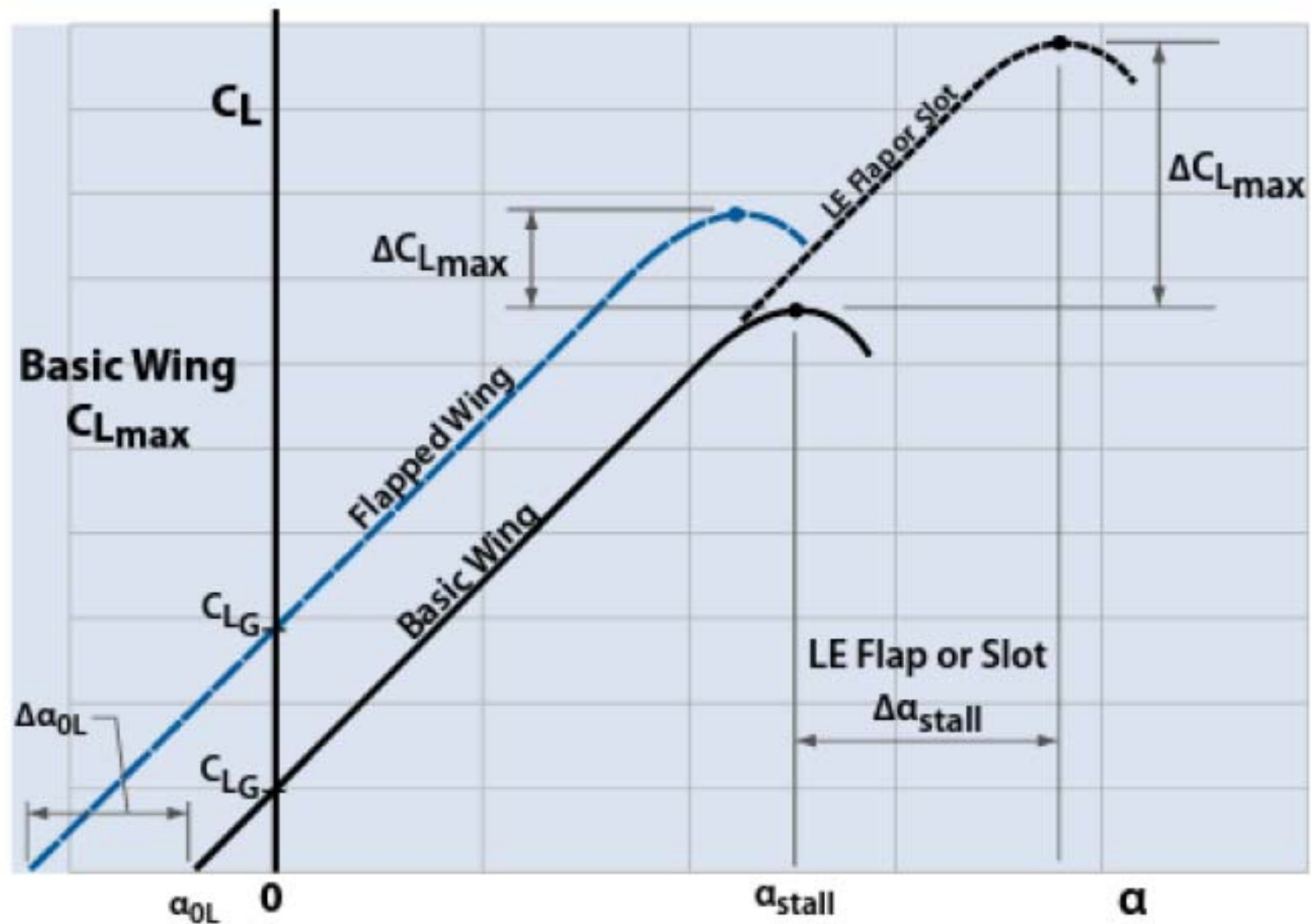


Figure 9.22 Construction of wing lift curves for mechanical high-lift devices.

# Cálculo $C_{Lmax}$ - VI

Wing Lift Coefficient with TE HLD

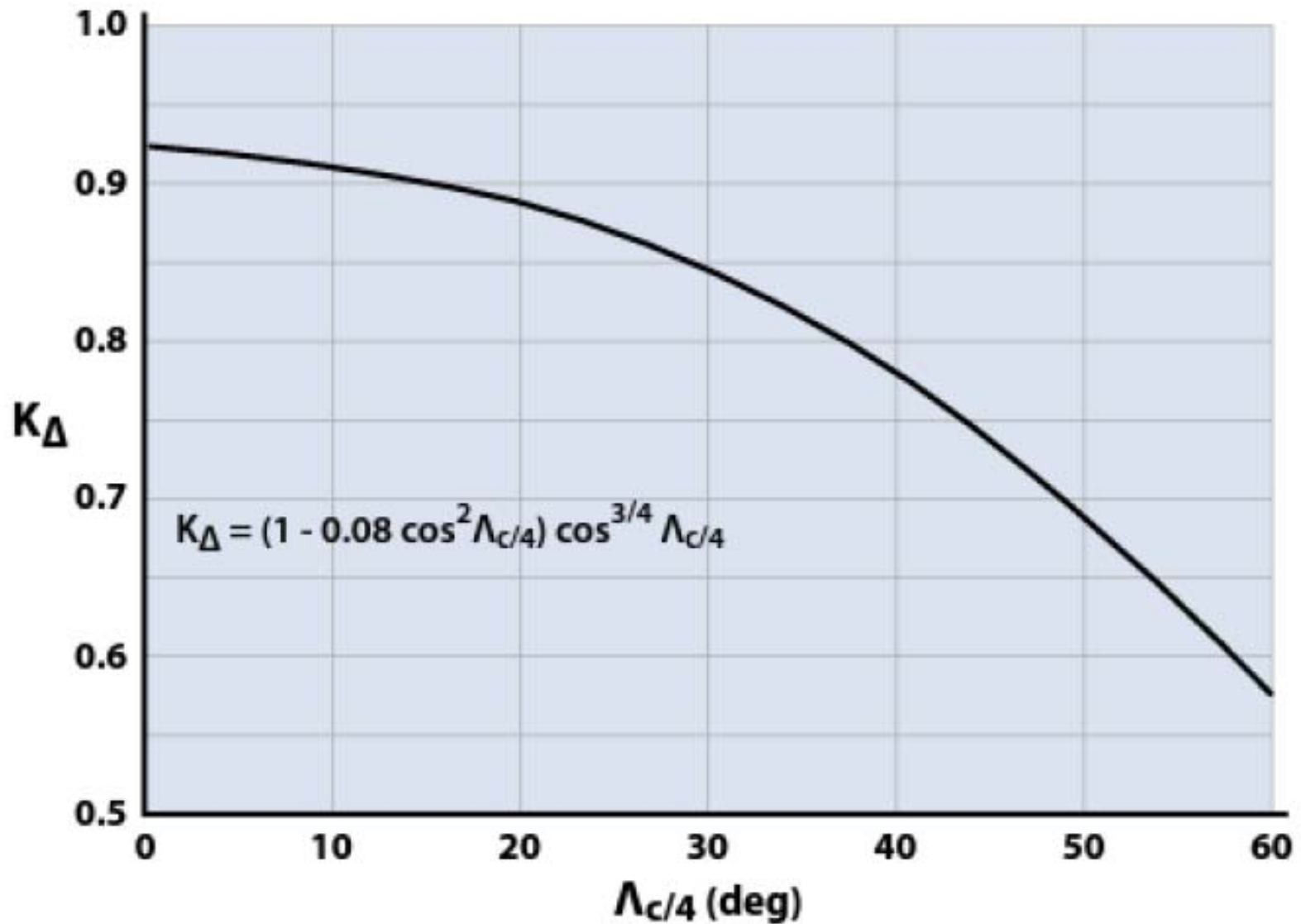
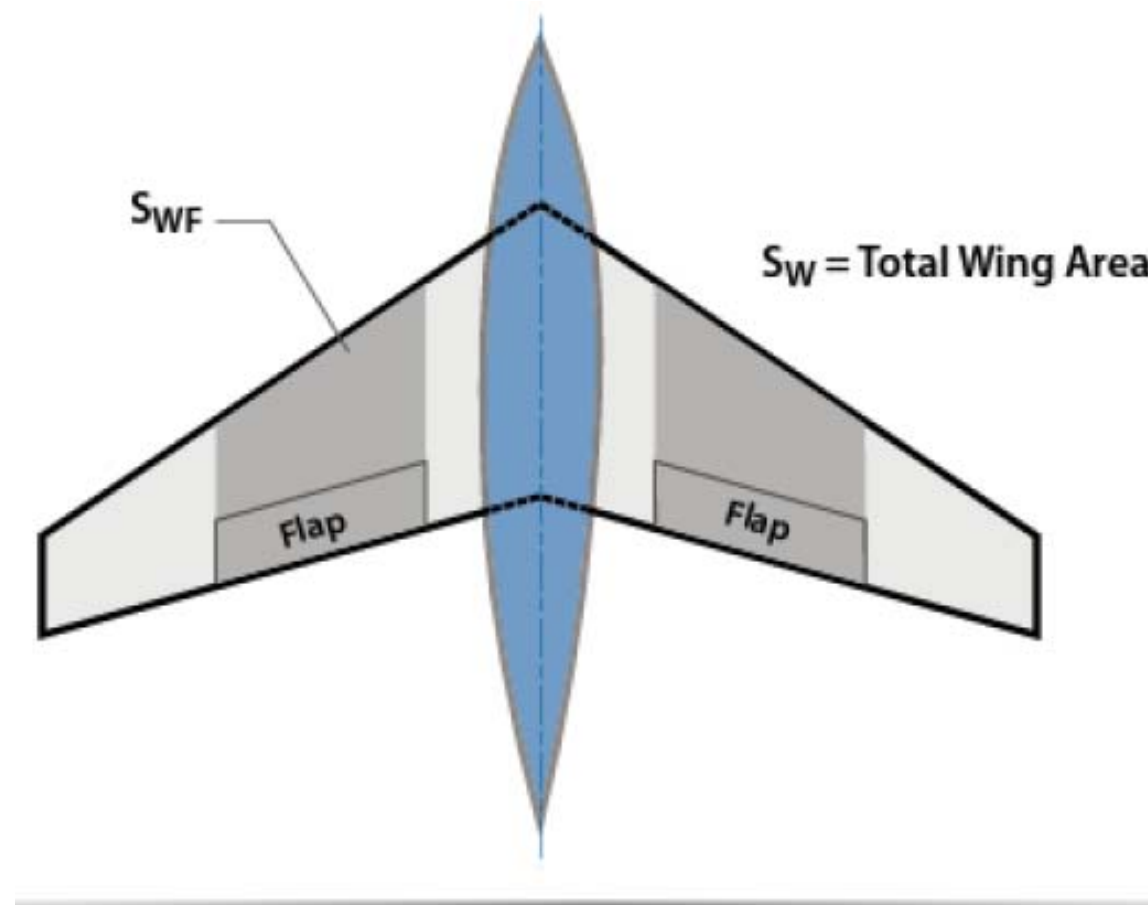


Figure 9.23 Planform correction factors for TE flaps (adapted [10]).

# Cálculo $C_{Lmax}$ - VII

Wing Lift Coefficient with TE HLD



**Figure 9.24** Schematic showing flapped wing area.

# Cálculo $C_{Lmax}$ - VIII

## ■ Incremento de la sustentación

Ángulo de la línea de rotación de la superficie hipersustentadora (High Lift Device)

$$\Delta C_{L_{max}} = \Delta C_{l_{max}} \left( \frac{S_{flapped}}{S_{ref}} \right) \cos(\Lambda_{H.L.})$$

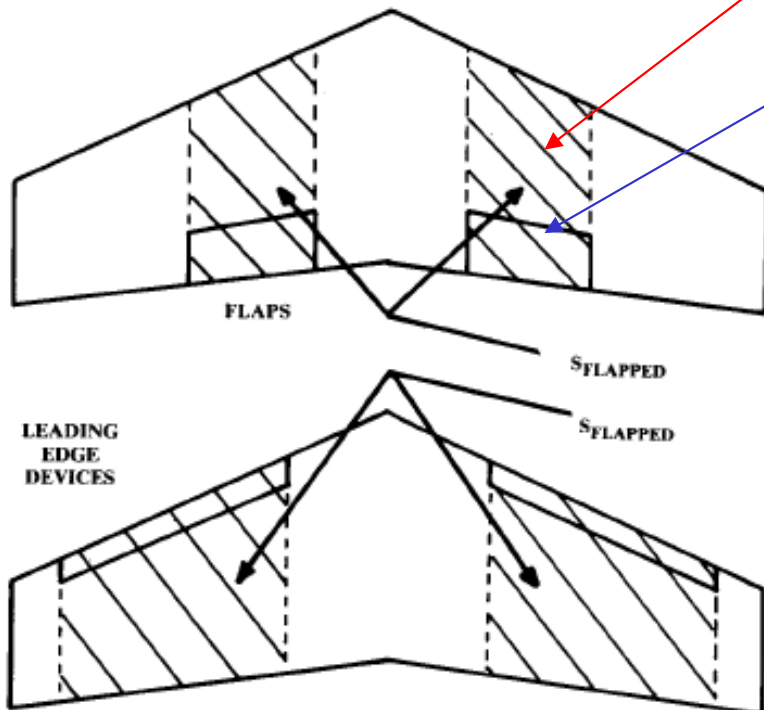


Fig. 12.19 "Flapped" wing area.

Table 12.2 Approximate lift contributions of high-lift devices

| High-lift device     | $\Delta C_{l_{max}}$ |
|----------------------|----------------------|
| Flaps                |                      |
| Plain and split      | 0.9                  |
| Slotted              | 1.3                  |
| Fowler               | 1.3 c'/c             |
| Double slotted       | 1.6 c'/c             |
| Triple slotted       | 1.9 c'/c             |
| Leading edge devices |                      |
| Fixed slot           | 0.2                  |
| Leading edge flap    | 0.3                  |
| Kruger flap          | 0.3                  |
| Slat                 | 0.4 c'/c             |

# Coeficiente de Resistencia - $C_D$ - I

- La resistencia esta compuesta por:
  - Fricción de placa plana
  - Profile ( $\Delta C_{Dp}$ )
  - Aspereza
  - Protuberancias
  - Interferencias
    - Tren de aterrizaje.
    - Góndolas
    - Protuberancias.
  - Efectos 3-D
  - Efectos de compresibilidad ( $\Delta C_{DM}$ )
  - Inducido ( $C_L^2/(\pi A Re)$ )
- Las dos componentes mas importantes de la resistencia aerodinámica son la **fricción** y la **inducida**.
- Se suele simplificar la obtención de la resistencia utilizando tan solo estimaciones para la resistencia de fricción y la inducida.

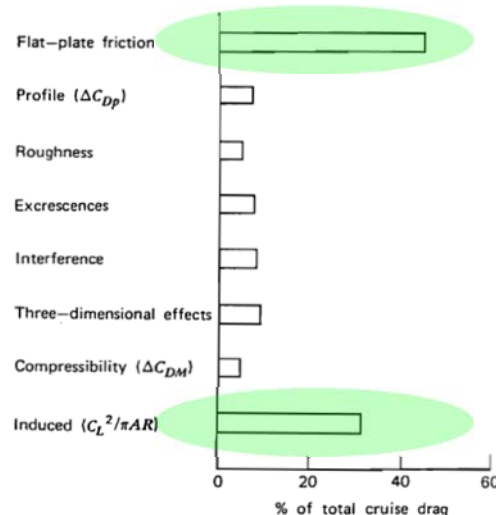


Figure 4.34 Typical drag buildup for jet transport.

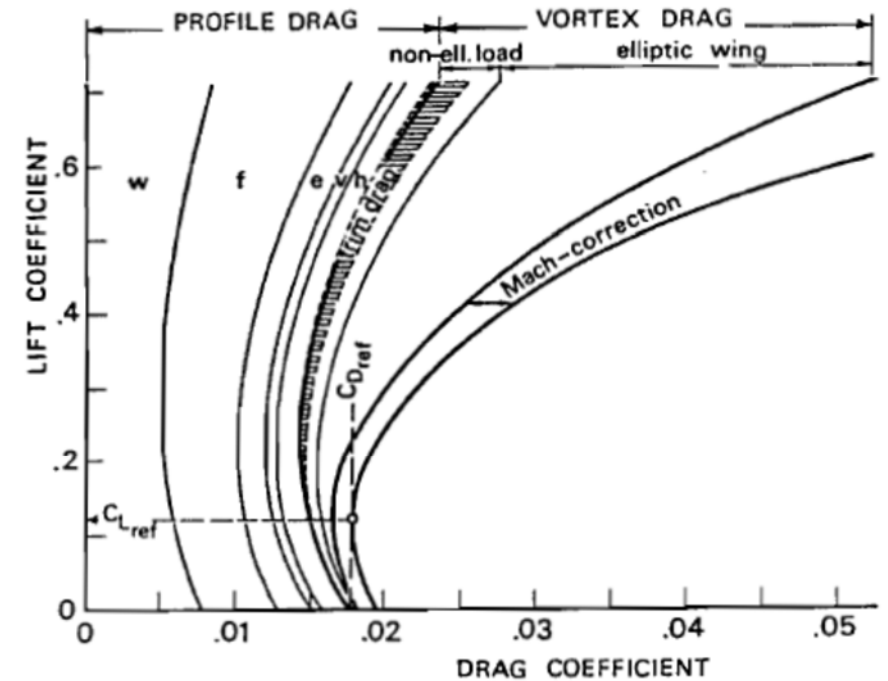


Fig. 11-2. Drag buildup by analysis  
(w = wing; f = fuselage; e = engine installation; v = vertical tailplane; h = horizontal tailplane)

# Coeficiente de Resistencia - $C_D$ - II

- Hipótesis simplificada:
  - Polar parabólica de coeficientes constantes.
- Cada segmento de vuelo define unas características de polar diferentes en función de la relación  $L/D$  a las que se vuela.
- En crucero se suele aproximar con una polar parabólica de coeficientes constantes:

- Alas sin curvatura
  - Mínima resistencia  $\alpha=0$ 
    - $C_{D0} = C_{Dmin}$

$$C_D = C_{D0} + KC_L^2 \Leftarrow K = \frac{1}{\pi Ae}$$

- Alas con curvatura
  - Mínima resistencia  $\alpha>0$ 
    - $C_{D0} \neq C_{Dmin}$

$$\begin{aligned} C_D &= C_{Dmin} + K (C_L - C_{Lmin-drag})^2 \\ &= C_{Dmin} + KC_{Lmin-drag}^2 + KC_L^2 - 2KC_L C_{Lmin-drag} \\ &= C_{D0} + k_1 C_L^2 - k_2 C_L \end{aligned}$$

$$\begin{aligned} C_{D0} &= C_{Dmin} + KC_{Lmin-drag}^2 \\ k_1 &= K \\ k_2 &= 2KC_L C_{Lmin-drag} \end{aligned}$$

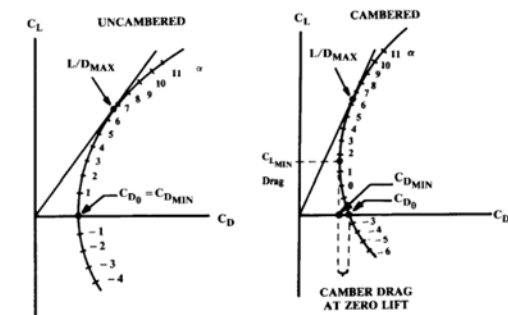


Fig. 12.3 Drag polar.

- En la **mayoría** de los **textos** se asume que la **sustentación** procede **únicamente** del **ala**, lo que se conoce como **polar no equilibrada**, ya que tanto las derivas como el fuselaje influyen en la sustentación

# Coeficiente de Resistencia - $C_D$ - III

- La eficiencia aerodinámica o “Oswald Efficiency”, lo que hace es reducir efectivamente el alargamiento del ala lo que incrementa la resistencia inducida por la sustentación

$$k = \frac{1}{\pi A e}$$



Straight-wing aircraft:  $e = 1.78 (1 - 0.045 A^{0.68}) - 0.64$   
Swept-wing aircraft:  $e = 4.61 (1 - 0.045 A^{0.68}) (\cos \Lambda_{LE})^{0.15} - 3.1$   
( $\Lambda_{LE} > 30^\circ$ )

$A$  – alargamiento del ala

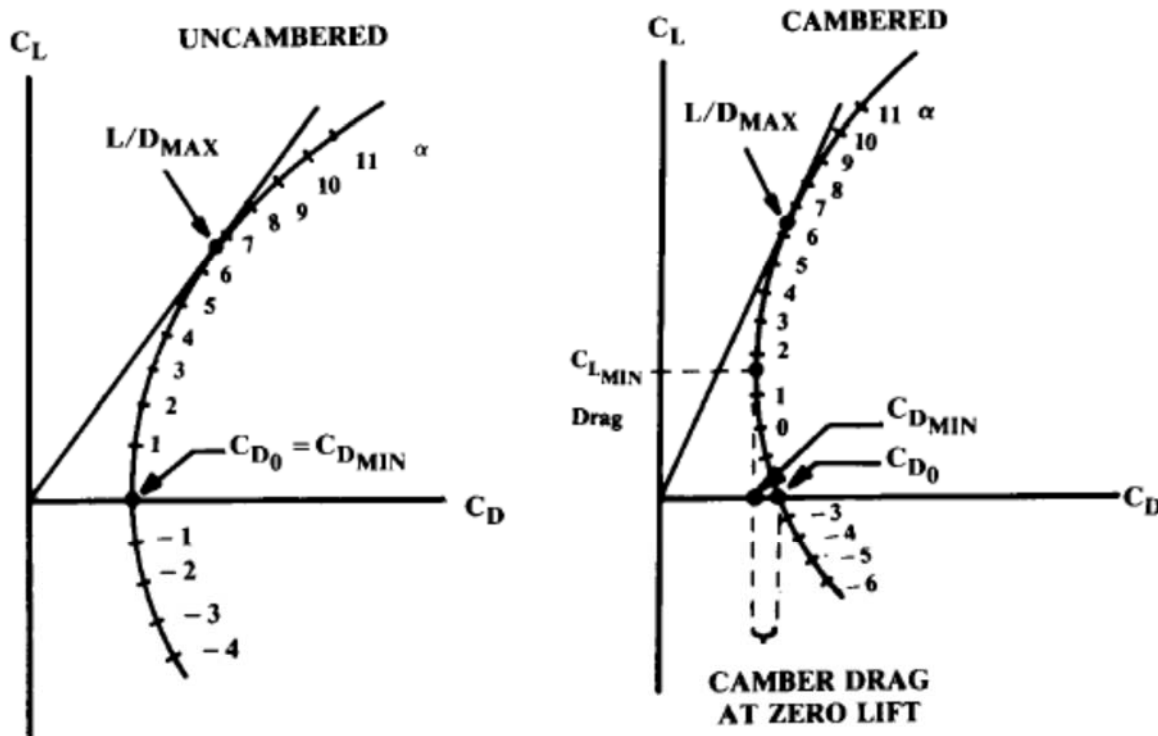
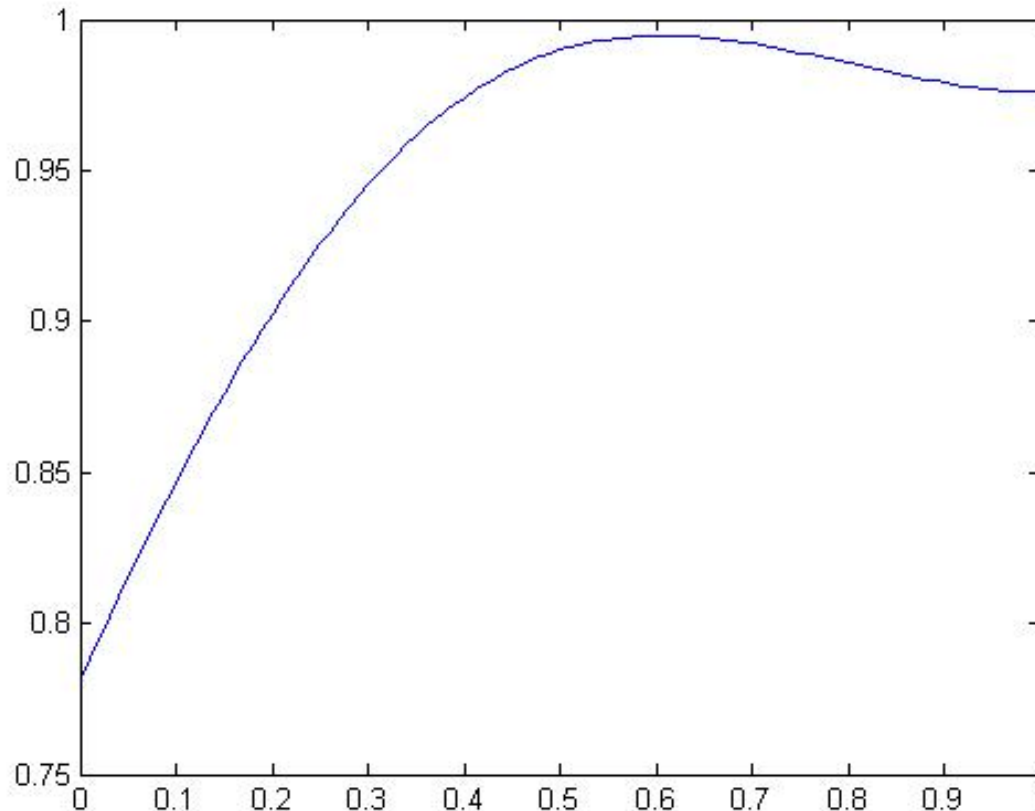


Fig. 12.3 Drag polar.

# Oswald Efficiency - e



Pendiente de la sustentación

$$e = \frac{1.1 C_{L\alpha}}{R C_{L\alpha} + (1 - R) \pi A}$$

Aspect Ratio

$$R = a_1 \lambda_1^3 + a_2 \lambda_1^2 + a_3 \lambda_1 + a_4$$

$$a_1 = 0.0004, a_2 = -0.0080, a_3 = 0.0501, a_4 = 0.8642$$

estrechamiento

$$\lambda_1 = \frac{A \lambda}{\cos \Lambda_{LE}}$$

Flecha del borde de ataque

# Estimación $C_{D0}$ – I

- La resistencia de fricción (*parasite drag*) se conoce como la **resistencia** en la que la **sustentación es cero**.
- Para alas **sin curvatura**, equivale al valor **mínimo** de la **resistencia**
- Vamos a presentar dos métodos para estimar la resistencia parasitaria ( $C_{D0}$ ):

- **Equivalent Skin-Friction Method.**
- **Component Buildup Method.**

## ■ Equivalent Skin-Friction Method

- Esta basado en el principio que un **avión correctamente** diseñado en **crucero subsónico** tendrá **solo** resistencia **parasitaria**:

- asociada a la **fricción** de la **superficie**.
- una **pequeña componente** de resistencia debida a la **presión de separación**.

$$C_{D0} = C_{fe} \frac{S_{wet}}{S_{ref}}$$

- Este lleva al concepto de un **coeficiente de fricción equivalente** ( $C_{fe}$ ) que incluye **ambos componentes** (fricción de superficie + resistencia debida a la separación)

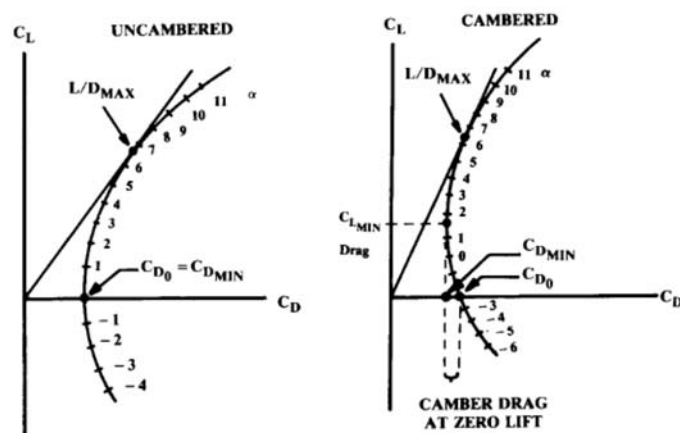


Fig. 12.3 Drag polar.

Table 12.3 Equivalent skin friction coefficients

| $C_{D0} = C_{fe} \frac{S_{wet}}{S_{ref}}$ | $C_{fe}$ -subsonic |
|-------------------------------------------|--------------------|
| Bomber and civil transport                | 0.0030             |
| Military cargo (high upsweep fuselage)    | 0.0035             |
| Air Force fighter                         | 0.0035             |
| Navy fighter                              | 0.0040             |
| Clean supersonic cruise aircraft          | 0.0025             |
| Light aircraft – single engine            | 0.0055             |
| Light aircraft – twin engine              | 0.0045             |
| Prop seaplane                             | 0.0065             |
| Jet seaplane                              | 0.0040             |

# Estimación $C_{D0}$ – II

- El **Component Buildup Method** estima la resistencia parasitaria subsónica de cada uno de los componentes de un avión utilizando el cálculo de:
  - Coeficiente de resistencia de fricción de la placa plana ( $C_{fc}$ )
  - Componente “form factor” (FF) que estima la resistencia de presión debido a la separación viscosa.
  - Se incluye un factor de interferencia “Q” que tiene en cuenta los efectos de cómo los diferentes elementos del avión interfieren entre ellos cuando están unidos.
  - $C_{D_{misc}}$  se refiere a la resistencia parasitaria asociada a flaps, trenes de aterrizaje, porciones del fuselaje que con flecha hacia arriba.
  - $C_{L\&P}$  se refiere a **Leakeges** (perdidas aerodinámicas) y Protuberances (protuberancias)

$$(C_{D0})_{subsonic} = \frac{\Sigma (C_{fc} \cdot FF_c \cdot Q_c \cdot S_{wet_c})}{S_{ref}} + C_{D_{misc}} + C_{D_{L\&P}}$$

Diagram illustrating the Component Buildup Method formula with annotations:

- Form Factor** points to  $FF_c$
- Factor de interferencia** points to  $Q_c$
- Miscelaneos** points to  $C_{D_{misc}}$
- Coeficiente de fricción** points to  $C_{fc}$
- Leakages and protuberances** points to  $C_{D_{L\&P}}$

# Estimación $C_{D0}$ – III

- El coeficiente de resistencia ( $C_f$ ) de fricción de placa plana depende de:
  - Número de Reynolds.
  - Mach.
  - Aspereza de la superficie
- Puede ser **turbulento** o **laminar** todo y que la mayoría de los aviones tiene **flujo turbulento** virtualmente para toda la superficie mojada
- Típicamente un avión correctamente diseñado puede tener
  - flujo laminar** de un **10-20 %** sobre las alas y las derivas (vertical y horizontal)
  - Flujo **turbulento** en el **100 %** del **fuselaje**
- Aviones modernos diseñados con compuestos pueden tener:
  - flujo laminar** de un **50 %** sobre las alas y las derivas (vertical y horizontal)
  - Flujo laminar **20-35%** del **fuselaje**.

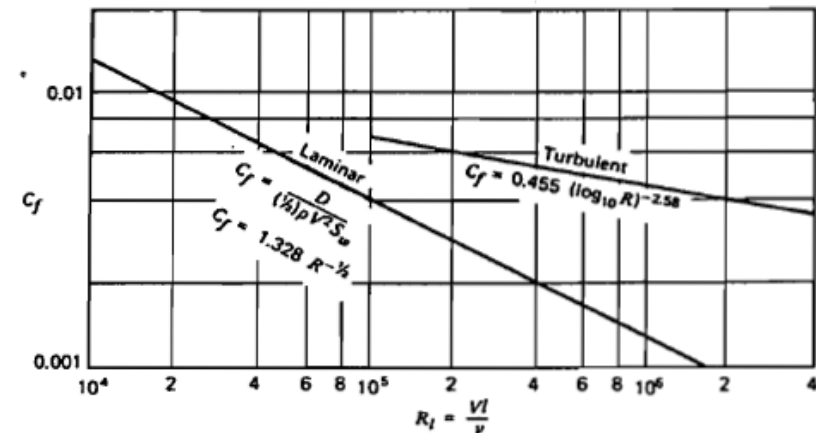
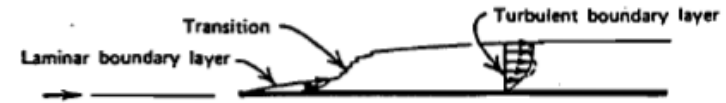


Table 4.3 Typical Total Skin Friction Coefficient Values for Different Airplane Configurations

| Airplane Configuration             | $C_F$ Range at Low Mach Numbers |
|------------------------------------|---------------------------------|
| Propeller driven, fixed gear       | 0.008–0.010                     |
| Propeller driven, retractable gear | 0.0045–0.007                    |
| Jet propelled, engines pod-mounted | 0.0035–0.0045                   |
| Jet propelled, engines internal    | 0.0030–0.0035                   |

Table 12.4 Skin roughness value ( $k$ )

| Surface                      | $k$ (ft)              |
|------------------------------|-----------------------|
| Camouflage paint on aluminum | $3.33 \times 10^{-5}$ |
| Smooth paint                 | $2.08 \times 10^{-5}$ |
| Production sheet metal       | $1.33 \times 10^{-5}$ |
| Polished sheet metal         | $0.50 \times 10^{-5}$ |
| Smooth molded composite      | $0.17 \times 10^{-5}$ |

# Estimación $C_{D0}$ – IV

- El Flat Plate skin friction coefficient:

- Laminar

$$\text{laminar} \Rightarrow C_f = \frac{1.328}{\sqrt{Re}} \quad Re = \frac{\rho V l}{\mu}$$

- Turbulento

$$\text{turbulent} \Rightarrow C_f = \frac{0.455}{(\log_{10} Re)^{2.58} (1 + 0.144 M^2)^{0.65}}$$

- Característica de la superficie afectan al número de Reynolds.
- Es necesario calcular  $R_{cutoff}$  para superficies que no sean suaves:

$$\text{subsonic} \Rightarrow R_{cutoff} = 38.21 (l/k)^{1.053}$$

$$\text{transonic or supersonic} \Rightarrow R_{cutoff} = 44.62 (l/k)^{1.053} M^{1.16}$$

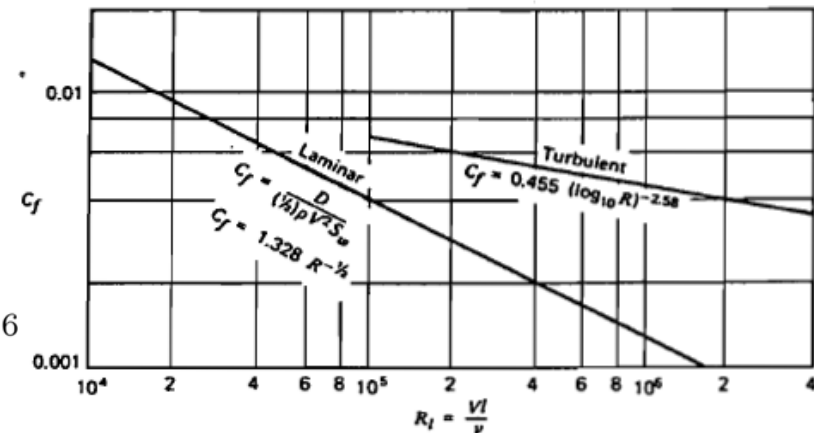
- $k$  es el skin roughness coefficient
- $l$  es distancia en feet

- El **mas bajo** de los dos **números de Reynolds** (actual y cutoff) deberá **ser empleado** para el cálculo del  $C_f$  turbulento.

$$\text{turbulent} \Rightarrow C_f = \frac{0.455}{(\log_{10} Re)^{2.58} (1 + 0.144 M^2)^{0.65}}$$

- Cuando se obtienen los  $C_f$  para **laminar** y **turbulento** se tiene que hacer una **media ponderada** en función de los porcentajes de flujo laminar que se cree que se obtendrán

$$C_f = 15\% C_{f, \text{lam}} + 75\% C_{f, \text{turb}}$$



**Table 4.3** Typical Total Skin Friction Coefficient Values for Different Airplane Configurations

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| Production sheet metal       | $1.33 \times 10^{-5}$ |
| Polished sheet metal         | $0.50 \times 10^{-5}$ |
| Smooth molded composite      | $0.17 \times 10^{-5}$ |

# Estimación $C_{D0} - V$

## ■ Form Factor (FF)

Wing, tail, strut, and pylon

$$\left(\frac{x}{c}\right)_m$$



Ubicación del máximo grosor del perfil con respecto a la cuerda

$$FF = \left[ 1 + \frac{0.6}{(x/c)_m} \left(\frac{t}{c}\right) + 100 \left(\frac{t}{c}\right)^4 \right] \left[ 1.34 M^{0.18} (\cos \Lambda_m)^{0.28} \right]$$

Fuselage and smooth canopy

Flecha en el máximo espesor

$$FF = \left[ 1 + \frac{60}{f^3} + \frac{f}{400} \right] \Rightarrow f = \frac{l}{d} = \frac{l}{\sqrt{\left(\frac{4}{\pi}\right) A_{max}}}$$

Nacelle and smooth external store

$l$  = longitud característica



$d$  = diámetro

$A_{max}$  = max cross sectional area

$$FF = 1 + \frac{0.35}{f} \Rightarrow f = \frac{l}{d} = \frac{l}{\sqrt{\left(\frac{4}{\pi}\right) A_{max}}}$$

## ■ Correcciones:

- Una deriva horizontal con una **línea de bisela** tiene un **10 % adicional FF** debido al **espacio** que produce la **línea de bisela**.
- Un **fuselaje rectangular** tiene incremento de entre un **30-40 %** en el **FF**.

# Estimación $C_{D0}$ – VI

- El factor de interferencia (Q) sirve para calcular la resistencia parasitaria debida a la mutua interferencia entre los diferentes componentes
  - Góndola en el fuselaje o la ala  $\sim Q=1.5$ 
    - Si la góndola esta separada a menos de 1 diámetro de la góndola  $\sim Q = 1.3$
    - Si la góndola esta separada mas de 1 diámetro de la góndola  $\sim Q = 1.0$
  - Alas:
    - Ala con misiles en las puntas  $Q=1.25$
    - Ala alta, media, o baja sin esquinas (con carenados adecuados)  $Q=1.0$
    - Alas con esquinas (sin carenas adecuadas)  $Q=1.1 \sim 1.4$
  - Fuselaje  $\approx 1.00$  para la mayoría de casos
  - Colas
    - $Q=1.03$  para colas en V.
    - $Q=1.08$  para colas en H.
    - $Q=1.04-1.05$  para colas convencionales.

# Estimación $C_{D0}$ – VII

## Resistencias Miscelaneas:

- Corrección para fuselajes con upsweep

Drag-Area

$$\frac{D}{q_{upsweep}} = 3.83u^{2.5} A_{max}$$

Máxima sección de área  
Del fuselaje



Fig. 12.24 Fuselage upsweep.

$$\frac{D}{q_{upsweep}} = 3.83u^{2.5} A_{max} \Rightarrow C_{D0} = \frac{D}{q_{upsweep}} \cdot \frac{1}{S_{ref}}$$

$[(D/q)/\text{Frontal area}] \Rightarrow$  Multiplicado por  $S_{\text{frontal}}/S_{\text{ref}}$

En  $ft^2!!!!$

$$\frac{D/q}{\text{Frontal Area}} \cdot \frac{S_{\text{frontal}}}{S_{\text{ref}}} = C_{D0}$$

- Tren de aterrizaje

Table 12.5 Landing gear component drags

|                                        | $\frac{D/q}{\text{Frontal area (Ft}^2\text{)}}$ |
|----------------------------------------|-------------------------------------------------|
| Regular wheel and tire                 | 0.25                                            |
| Second wheel and tire in tandem        | 0.15                                            |
| Streamlined wheel and tire             | 0.18                                            |
| Wheel and tire with fairing            | 0.13                                            |
| Streamline strut ( $1/6 < t/c < 1/3$ ) | 0.05                                            |
| Round strut or wire                    | 0.30                                            |
| Flat spring gear leg                   | 1.40                                            |
| Fork, bogey, irregular fitting         | 1.0–1.4                                         |

Para tener en cuenta interferencias mutuas entre los diferentes miembros

# Estimación $C_{L_{max}}$ Despegue - III

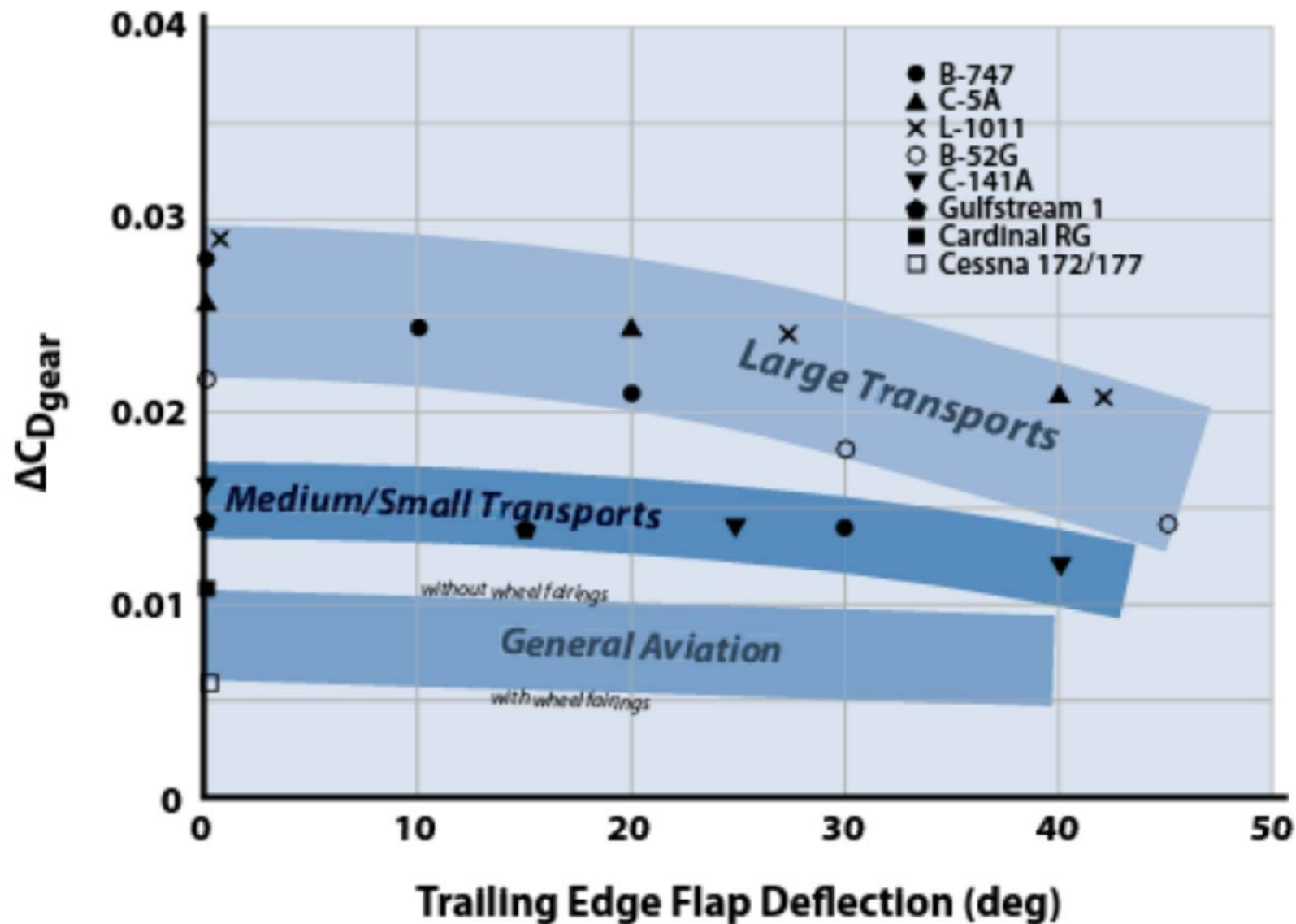


Figure 10.5 Drag of landing gear.

# Estimación $C_{L_{max}}$ Despegue - IV

$$C_{D_{gear}} = \Delta C_{D_{gear}} \cdot \frac{S_{ref-Tabla}}{S_{ref}}$$

En  $ft^2!!!!$

**Table 10.4** Landing Gear Drag Coefficients

| Aircraft                | Reference Area (ft <sup>2</sup> ) | $\Delta C_{D_{gear}}$ | Landing Gear Configuration <sup>a</sup>                    |
|-------------------------|-----------------------------------|-----------------------|------------------------------------------------------------|
| <b>Fighters</b>         |                                   |                       |                                                            |
| A-7                     | 375                               | 0.028                 | Two-wheel NLG, two one-wheel MLG                           |
| F-104                   | 196                               | 0.035                 | One-wheel NLG, two one-wheel MLG                           |
| F-16A1B                 | 300                               | 0.0325                | One-wheel NLG, two one-wheel MLG                           |
| F-22                    | 840                               | 0.014                 | One-wheel NLG, two one-wheel MLG                           |
| U-2S                    | 1000                              | 0.0045                | One dual-wheel MLG, large tail wheel, and two wingtip pogo |
| <b>Large transports</b> |                                   |                       |                                                            |
| L-1011                  | 3456                              | 0.028–0.0205          | Two-wheel NLG, two four-wheel trucks MLG                   |
| C-5A                    | 6200                              | 0.0257–0.021          | Four-wheel NLG, four four-wheel trucks MLG                 |
| B-747                   | 5500                              | 0.028–0.014           | Two-wheel NLG, four four-wheel trucks MLG                  |
| B-52G                   | 4000                              | 0.024–0.0155          | Quadricycle with wingtip gear, four dual-wheel MLG         |

# Estimación $C_{L_{max}}$ Despegue - V

| Medium transports |      |                    |                                          |
|-------------------|------|--------------------|------------------------------------------|
| P-3               | 1300 | 0.020              | Two-wheel NLG, two two-wheel MLG         |
| L-1049 Connie     | 1650 | 0.024              | Two-wheel NLG, two two-wheel MLG         |
| B 727             | 1650 | 0.017              | Two-wheel NLG, two two-wheel MLG         |
| DC-8              | 2771 | 0.012              | Two-wheel NLG, two four-wheel trucks MLG |
| C-141A            | 3228 | 0.0165–0.012       | Two-wheel NLG, two four-wheel trucks MLG |
| Small transports  |      |                    |                                          |
| S-3A              | 598  | 0.023              | Two-wheel NLG, two one-wheel MLG         |
| Gulfstream I      | 615  | 0.015              | Two-wheel NLG, two one-wheel MLG         |
| Fokker F-27       | 754  | 0.024              | One-wheel NLG, two dual-wheel MLG        |
| General aviation  |      |                    |                                          |
| Cessna 172        | 226  | 0.006 <sup>b</sup> | One-wheel NLG, two one-wheel MLG         |
| Cessna 177        | 174  | 0.006 <sup>b</sup> | One-wheel NLG, two one-wheel MLG         |
| Cardinal RG       | 174  | 0.011              | One-wheel NLG, two one-wheel MLG         |

<sup>a</sup>Abbreviations: NLG, nose landing gear; MLG, main landing gear.

<sup>b</sup>Fixed landing gear with wheel fairings.

# Estimación $C_{D0}$ – VII - cont

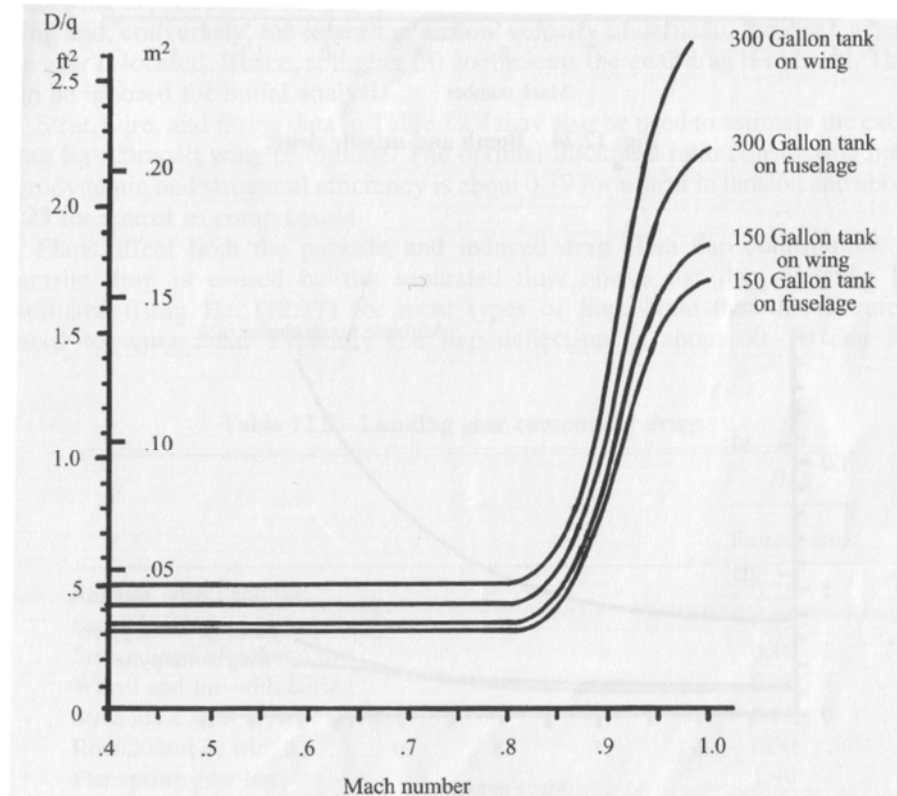


Fig. 12.23 External stores (fuel tanks) drag.

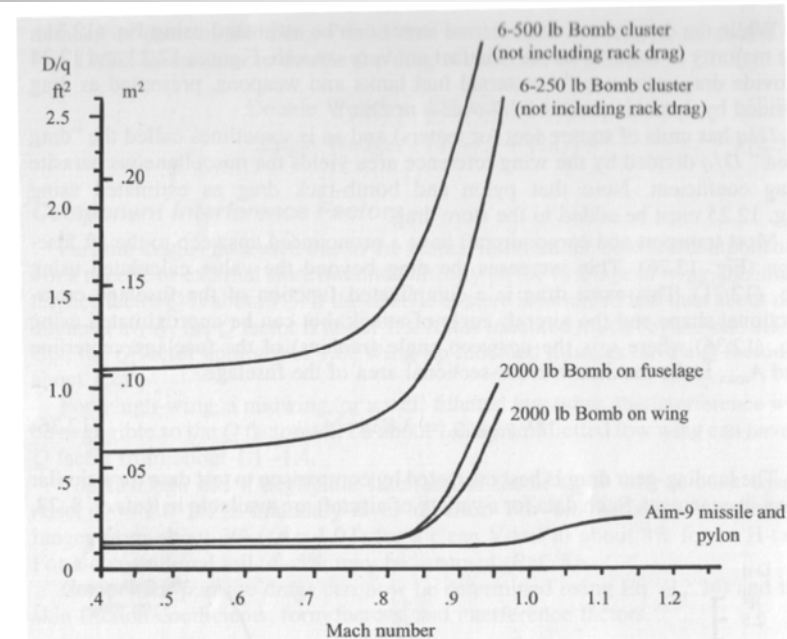


Fig. 12.24 Bomb and missile drag.

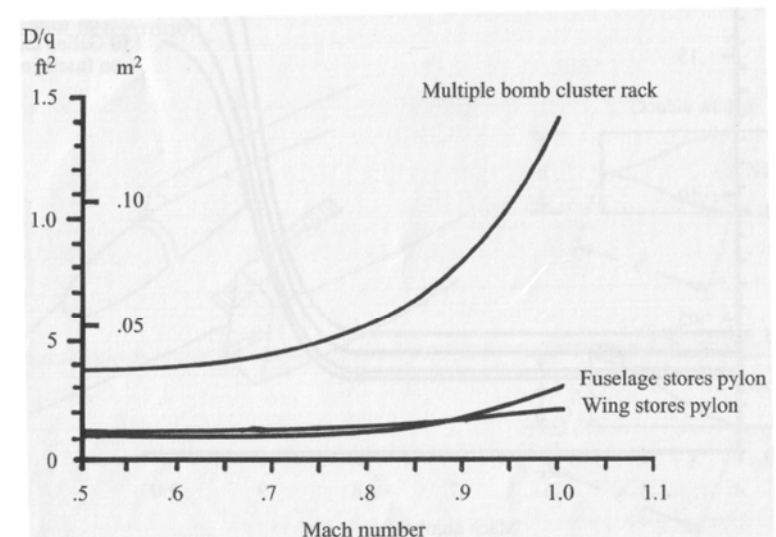


Fig. 12.25 Pylon and bomb rack drag.

# Estimación $C_{D0}$ – VIII

- Resistencia Leakage and protuberances
  - Antenas, puertas, bordes, carenado de superficies de control, defectos de construcción...
  - Dicha resistencia es debido a la tendencia del avión a “inhalar” a través de los orificios y espacios en las zonas de alta presión y “exhalar” aire en las zonas de baja presión.
  - Muy difícil de estimar y se suele aproximar con:
    - Incremento del 2-5% de la resistencia parasitaria en aviones jet y bombarderos.
    - Incremento del 5-10% de la resistencia parasitaria para aviones de pistón.
    - Incremento del 5-10% de la resistencia parasitaria para aviones de combate.

$$\%C_{D0} \Rightarrow (C_{D0})_{subsonic} = \frac{\Sigma (C_{fc} \cdot FF_c \cdot Q_c \cdot S_{wet_c})}{S_{ref}} + C_{D_{misc}} + C_{D_{L\&P}}$$

# Estimación $C_{D0}$ – IX

Drag to Flap deflection

## ■ Resistencia Flaps

- La resistencia asociada a los flaps afecta tanto a la resistencia parásita como a la inducida.

$$\Delta C_{D0_{flap}} = F_{flap} \left( \frac{C_f}{C} \right) \left( \frac{S_{flapped}}{S_{ref}} \right) (\delta_{flap} - 10)$$

$\delta_{flap} \Rightarrow$  in degrees

$F_{flap} = 0.0144 \Rightarrow$  plain flaps

$F_{flap} = 0.0074 \Rightarrow$  slotted flaps

$C_f =$  chord length of flap

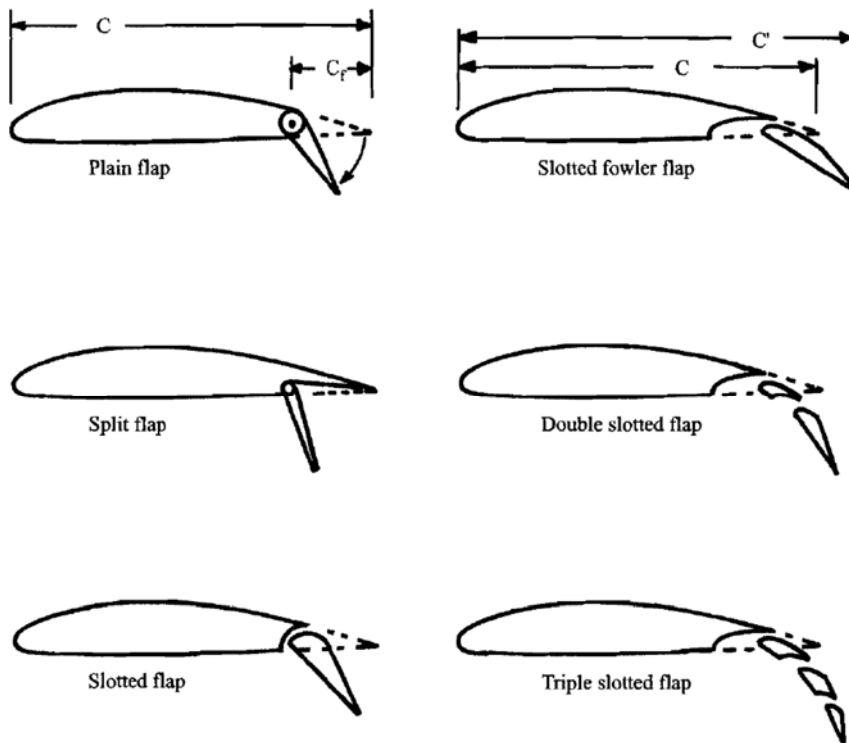


Fig. 12.17 Flap types.

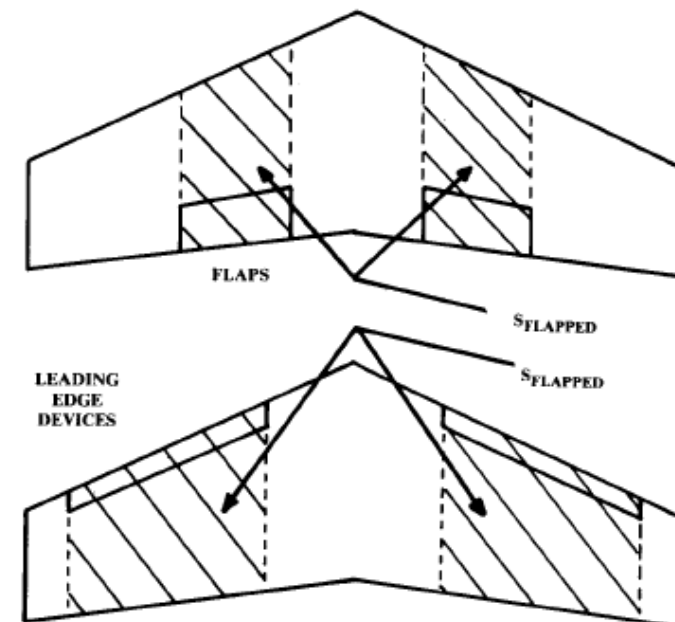
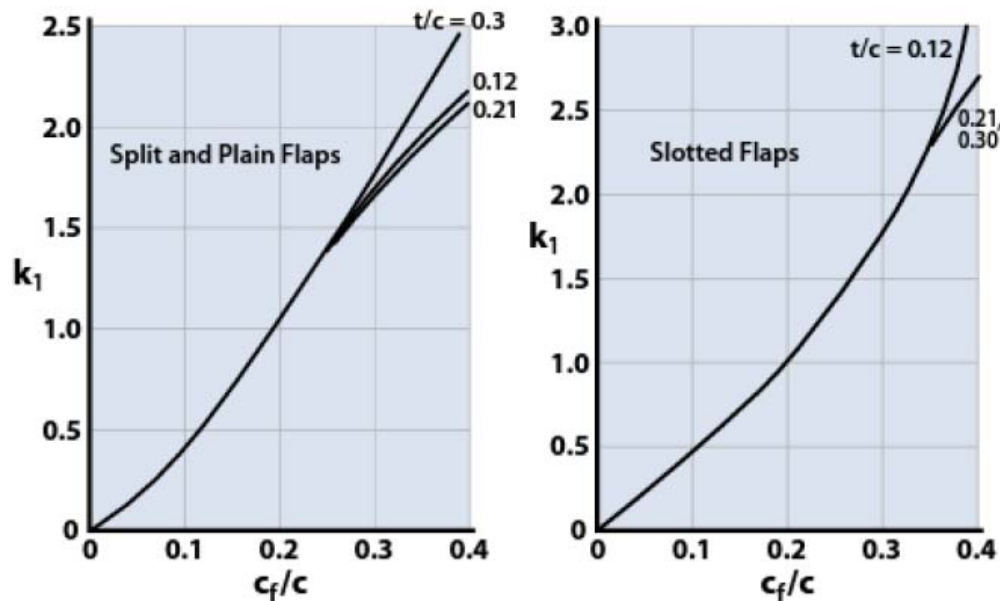


Fig. 12.19 "Flapped" wing area.

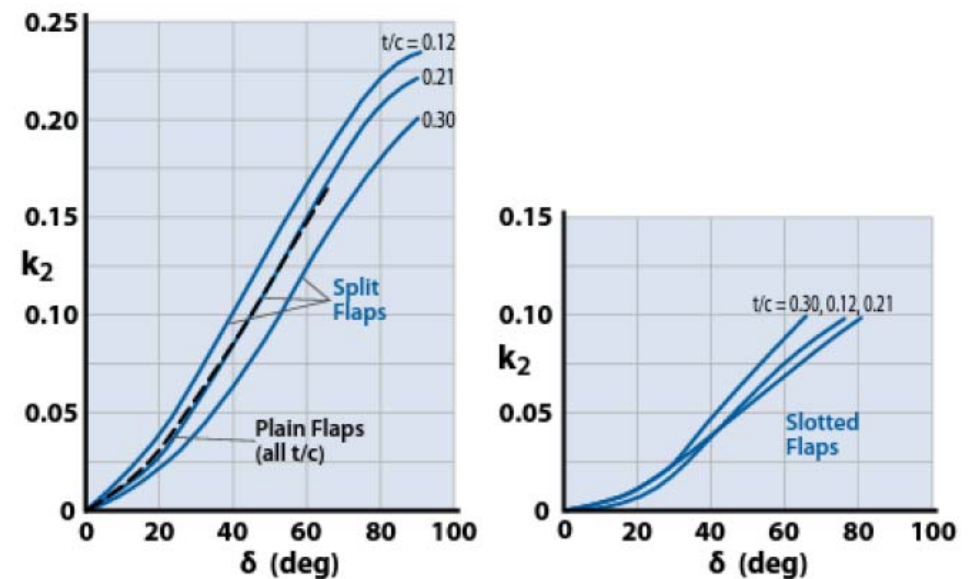
# Estimación $C_{D0} - X$

Drag to Flap deflection

$$\Delta C_{D_{flap}} = k_1 k_2 \frac{S_{WF}}{S_W}$$

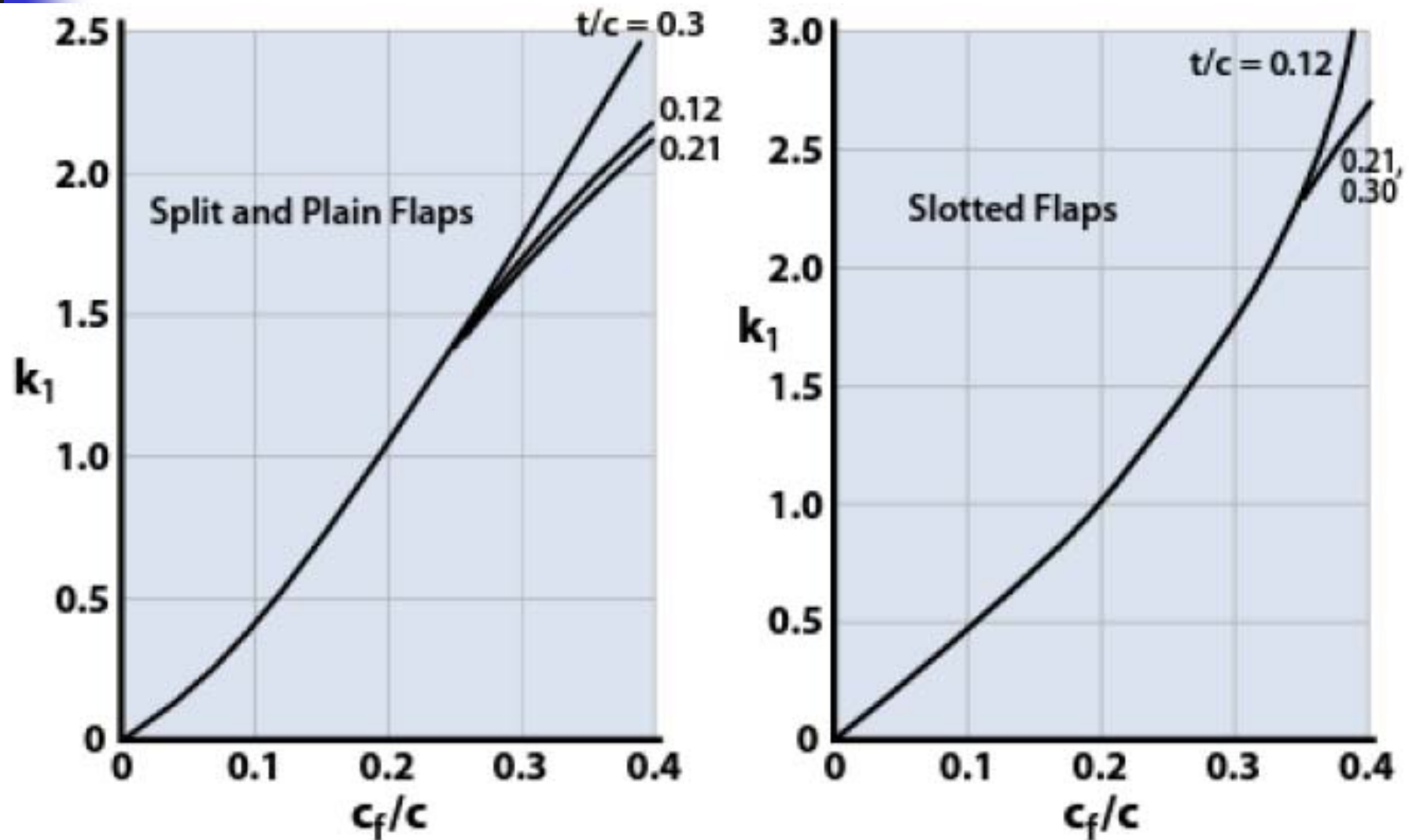


**Figure 9.26** Factor  $k_1$  to calculate drag increment due to flaps (data from [13]).



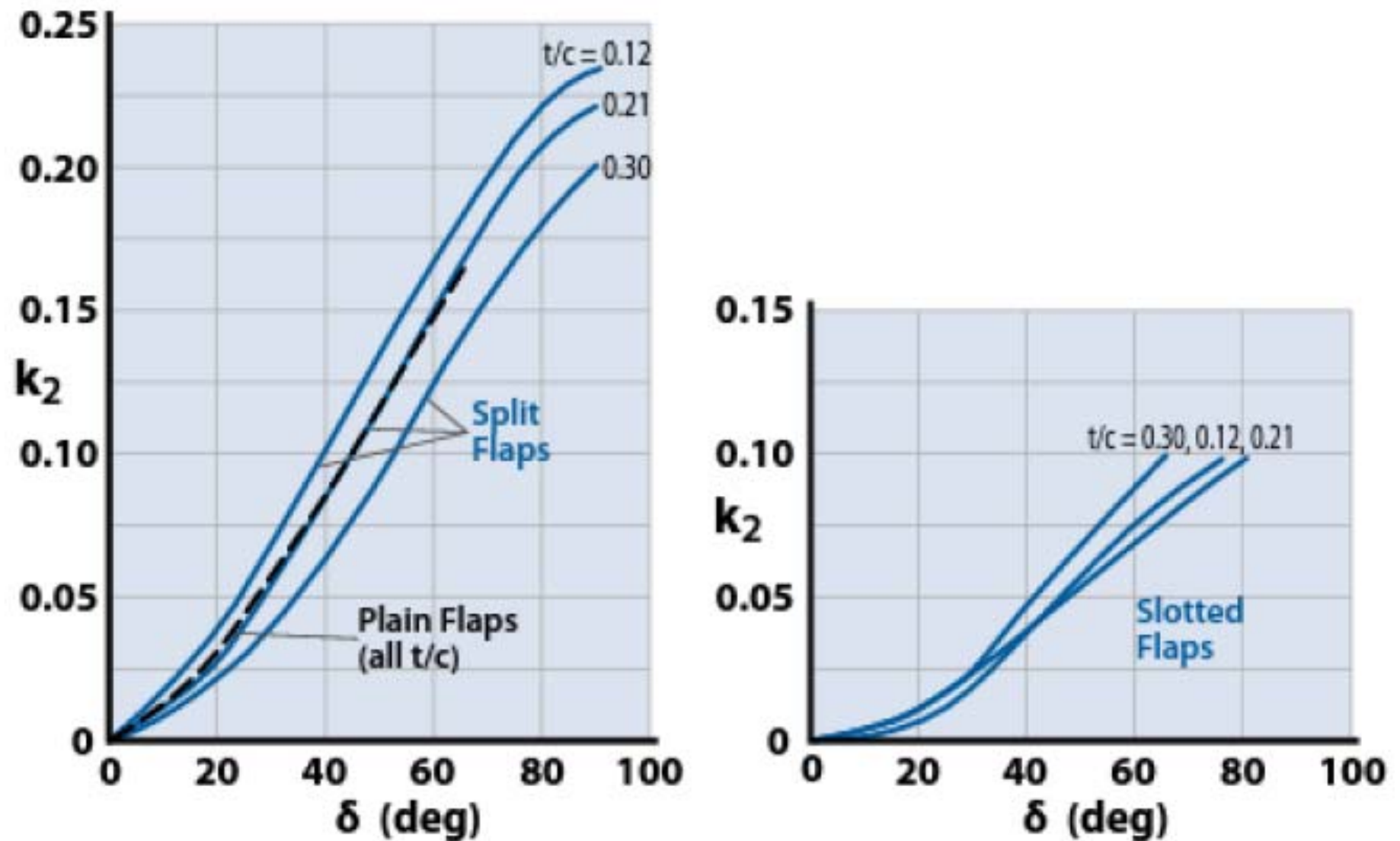
**Figure 9.27** Factor  $k_2$  to calculate drag increment due to flaps.

# Estimación $C_{D0}$ – XI



**Figure 9.26** Factor  $k_1$  to calculate drag increment due to flaps (data from [13]).

# Estimación $C_{D0}$ – XII



**Figure 9.27** Factor  $k_2$  to calculate drag increment due to flaps.

# Estimación $C_{D0}$ – XIII

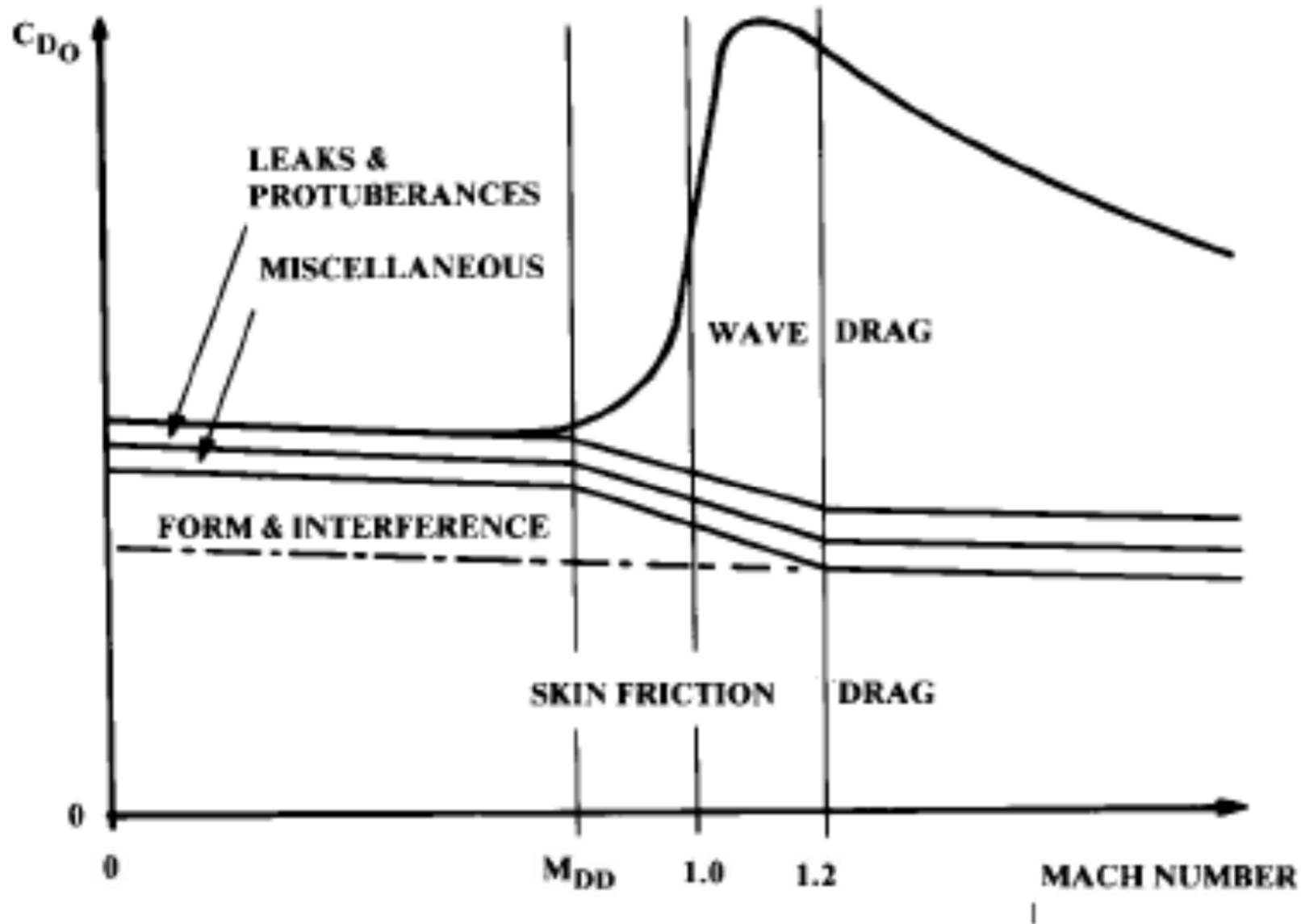


Fig. 12.30 Complete parasite drag vs Mach number.

# Eficiencia Aerodinámica - I

$$E = \frac{C_L}{C_D}$$

$$C_D = C_{D_0} + kC_L^2$$

$$k = \frac{1}{\pi A Re}$$

$$E = \frac{C_L}{C_{D_0} + kC_L^2}$$

$$\begin{aligned} \frac{dE}{dC_L} &= \frac{dE}{dC_L} \left( \frac{C_L}{C_{D_0} + kC_L^2} \right) \\ &= \frac{1}{C_{D_0} + kC_L^2} - \frac{2kC_L}{\sqrt{C_{D_0} + kC_L^2}} \end{aligned}$$

$$\frac{1}{C_{D_0} + kC_L^2} - \frac{2kC_L}{\sqrt{C_{D_0} + kC_L^2}} \implies C_{L_{opt}} = \sqrt{\frac{C_{D_0}}{k}}$$

$$E = \frac{C_L}{C_D} = \frac{C_L}{C_{D_0} + kC_L^2} \implies E_{opt} = \frac{C_{L_{opt}}}{C_{D_0} + kC_{L_{opt}}^2} = \frac{1}{2\sqrt{kC_{D_0}}}$$

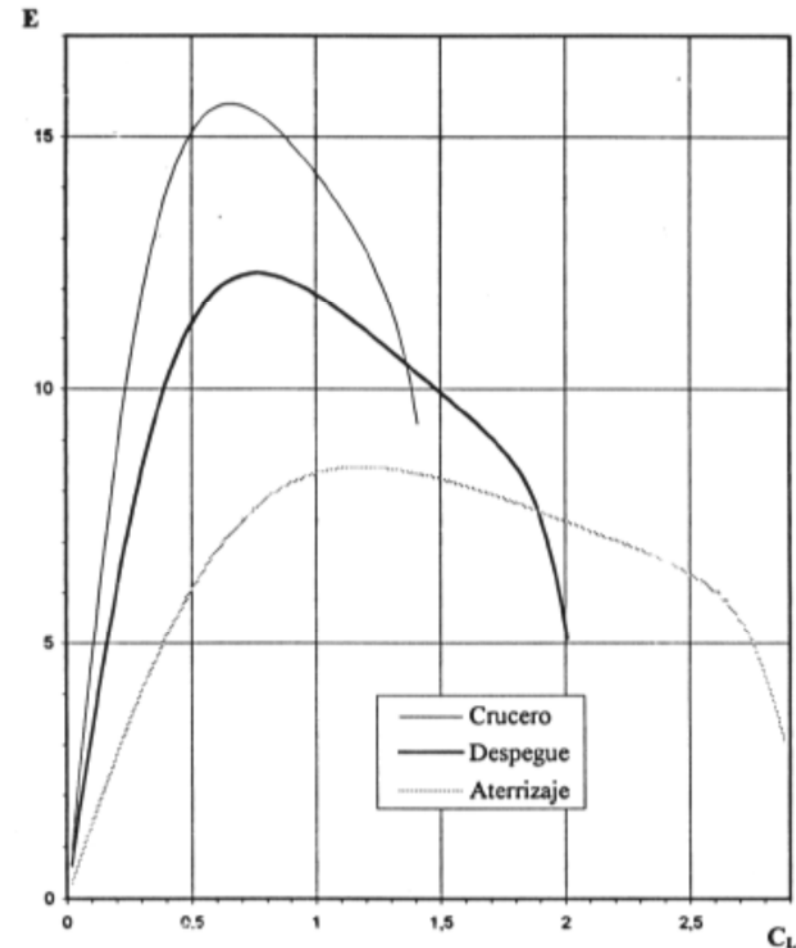
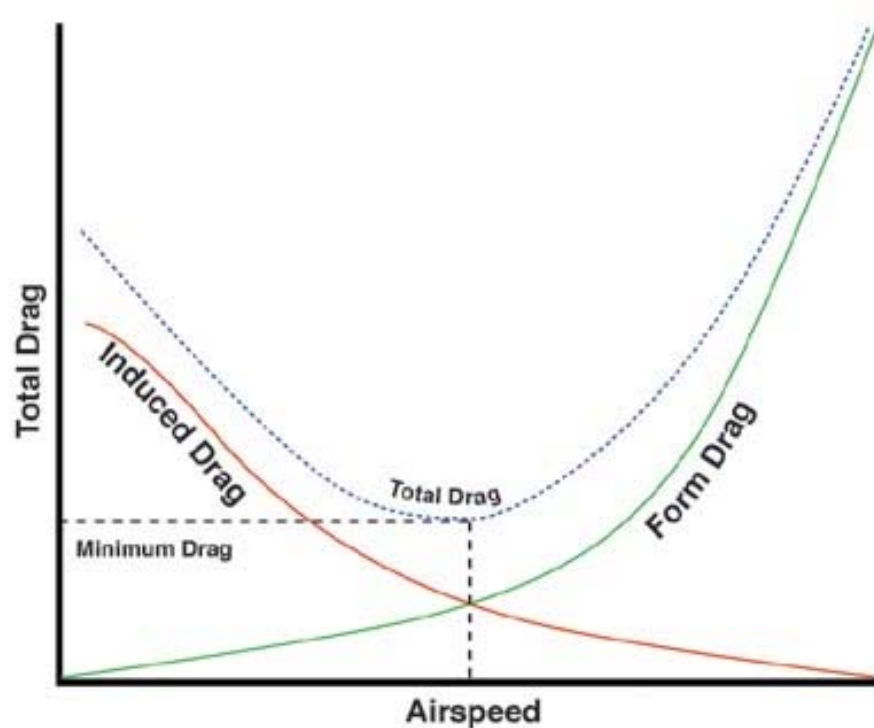
$$\begin{aligned} C_D &= C_{D_{min}} + K(C_L - C_{L_{min \ drag}})^2 \\ &= C_{D_{min}} + KC_{L_{min \ drag}}^2 + KC_L^2 - 2KC_{L_{min \ drag}}C_L \\ &= C_{D_0} + k_1C_L^2 - k_2C_L \end{aligned}$$



¿¿¿¿¿ $C_{L_{opt}}$ ????

# Eficiencia Aerodinámica - II

$$E = \frac{C_L}{C_D} = \frac{C_L}{C_{D_0} + kC_L^2} \Rightarrow E_{opt} = \frac{C_{L_{opt}}}{C_{D_0} + kC_{L_{opt}}^2} = \frac{1}{2\sqrt{kC_{D_0}}}$$



# Reducción de la Resistencia

- Las dos componentes más importantes de la resistencia son la resistencia de fricción y la inducida, por ello es muy importante el esfuerzo que se hace para disminuirlas.
- Control del flujo laminar:
  - Efecto en las actuaciones
- Aletas de borde marginal (LEX – leading edge extension):
- Reducen la resistencia inducida, lo que aumenta la eficiencia aerodinámica y un momento flector

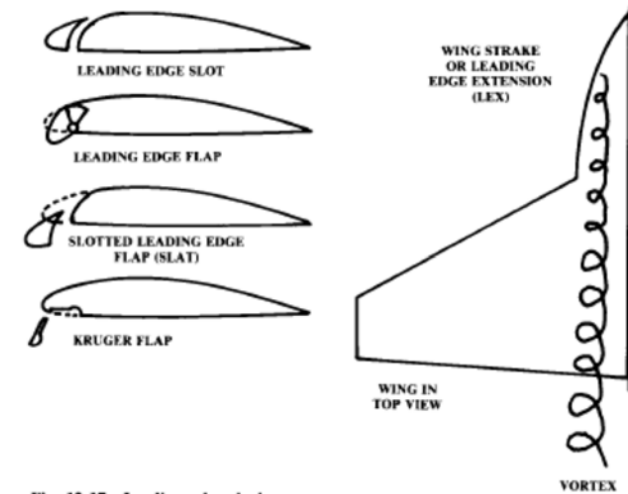


Fig. 12.17 Leading edge devices.

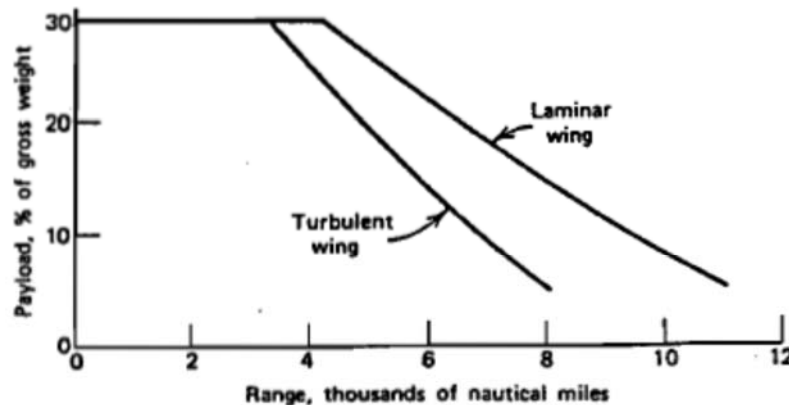


Figure 4.50 Performance gains from laminar flow control. (R. E. Kosin, "Laminar Flow Control by Suction as Applied to X-21A Airplane", AIAA Journal of Aircraft, 1965.

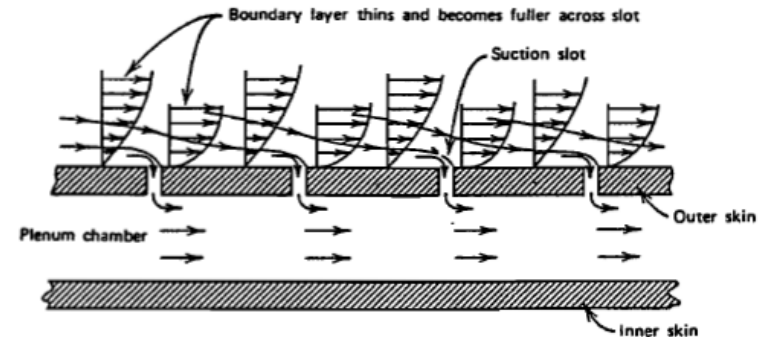


Figure 4.49 Laminar flow control by suction through thin slots transverse to the flow.

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