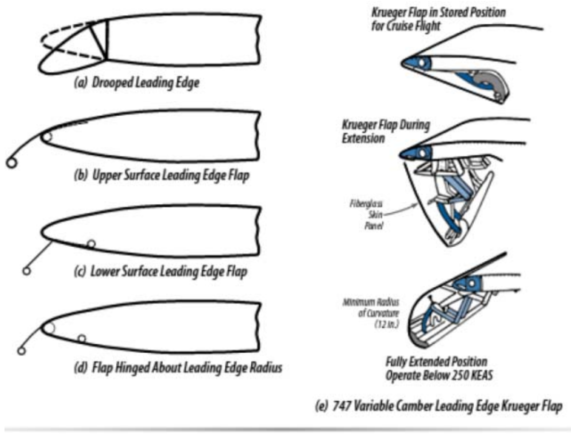


Figure 9.5 Various LE flap devices.

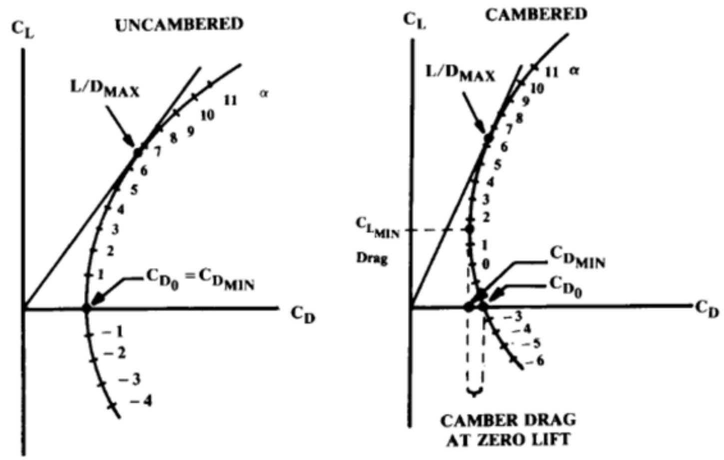


Fig. 12.3 Drag polar.

Aerodinámica Fase Preliminar

Tema 5

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

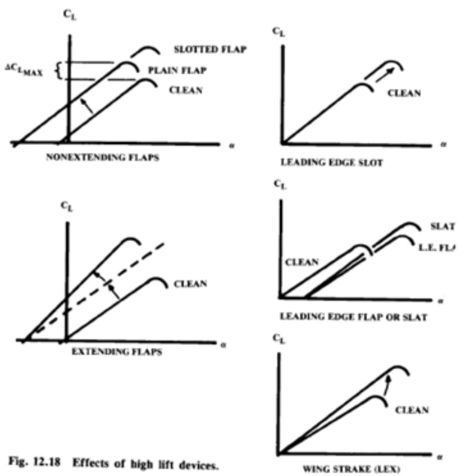


Fig. 12.18 Effects of high lift devices.

Sergio Esteban Roncero
Departamento de Ingeniería Aeroespacial
Y Mecánica de Fluidos

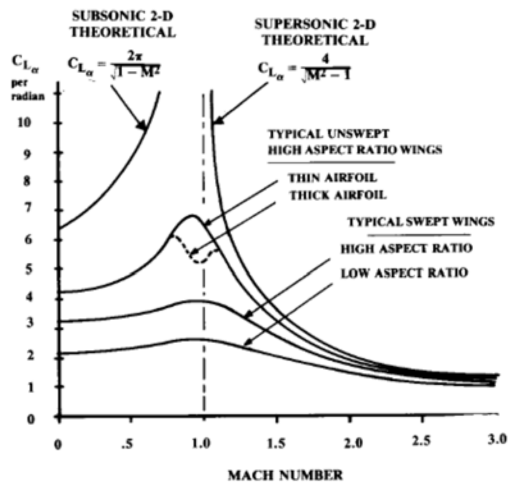


Fig. 12.5 Lift curve slope vs Mach number.

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

Aerodinámica – Estudio Preliminar

¿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, λ
- Interacción con diferentes áreas de Ingeniería:
  - Sustentación requerida: pesos (Estructura)
  - Generación de parámetros aerodinámicos (Estabilidad y Actuaciones/Porpulsión)
  - Geometría: Diseño y sistemas

Aerodinámica - I

- Lo que se espera:
  - Polar parabólica de coeficientes constantes:
    - CD del avión
      - Configuración limpia
        - Vuelo de subida
        - Vuelos de crucero
      - Configuración sucia
        - Despegue y aterrizaje
          - Superficies Hipersustentadoras, tren de aterrizaje…

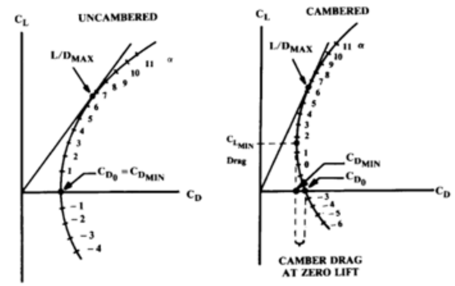


Fig. 12.3 Drag polar.

  - Características aerodinámicas de los perfiles:
    - Estimar C_L
      - C_{Lo}, $C_{L\alpha}$, $C_{Mo,}$ $C_{M\alpha}$
      - Ala, canard, deriva horizontal y vertical, cola en V…
    - Corrección para alas finitas
- Como conseguirlo…

Aerodinámica - II

- CD del avión
  - Configuración limpia: Tren retraído, flaps recogidos
    - Vuelo de subida
    - Vuelos de crucero y alcance
  - Configuración sucia: flaps y tren de aterrizaje desplegados
    - Despegue y aterrizaje
- Como conseguirlo:
  - Modelo de polar parabólica de coeficientes constantes

$$C_D = C_{D_0} + KC_L^2 \Rightarrow K = \frac{1}{\pi A e}$$

  - Component Buildup Method

$$(C_{D_0})_{\text{subsonic}} = \frac{\Sigma(C_{f_c} FF_c Q_c S_{\text{wet}_c})}{S_{\text{ref}}} + C_{D_{\text{misc}}} + C_{D_{\text{L\&P}}}$$

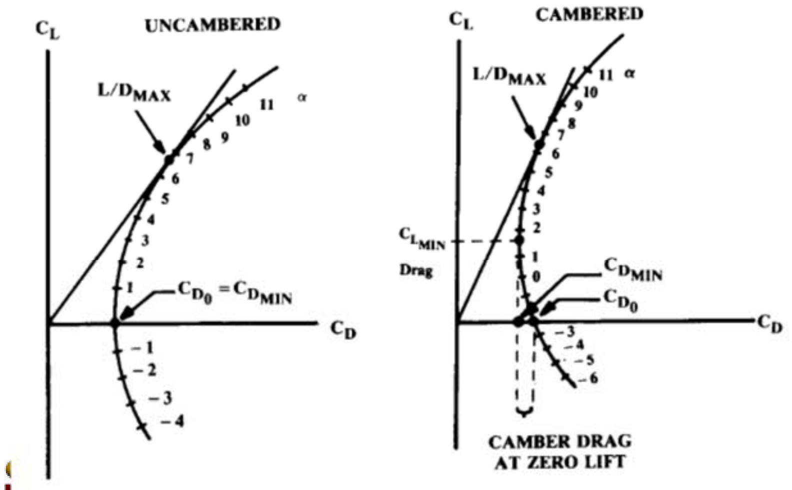


Fig. 12.3 Drag polar

Aerodinámica - III

- Características aerodinámicas de los perfiles:
  - Estimar C_L
    - CLmax:
      - 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_{Lo}, $C_{L\alpha}$, C_{Mo}, $C_{M\alpha}$
    - Ala, canard, deriva horizontal y vertical, cola en V…
  - Corrección para alas finitas

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

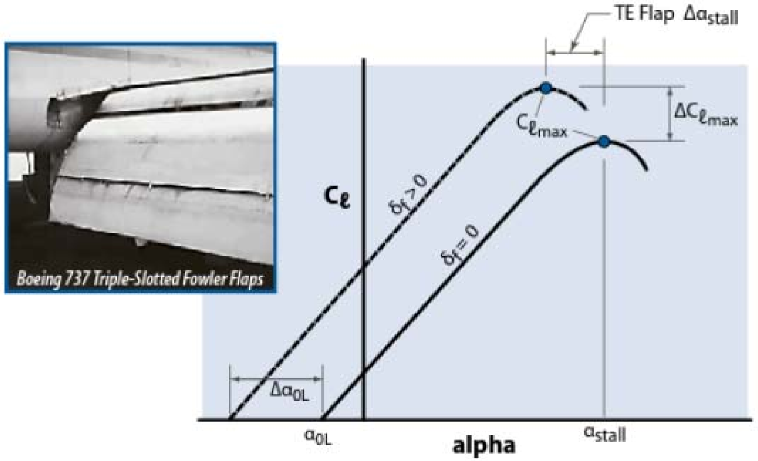


Figure 9.8 Construction of section lift curves for TE flaps.

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

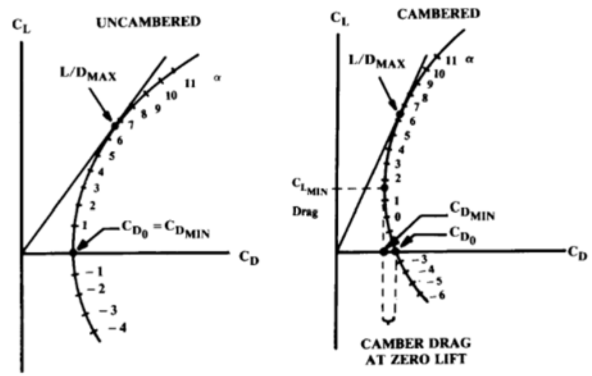


Fig. 12.3 Drag polar.

    - Modelo de Polar
      - Polar parabólica **No compensada**

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

      - Polar parabólica **Compensada**

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

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

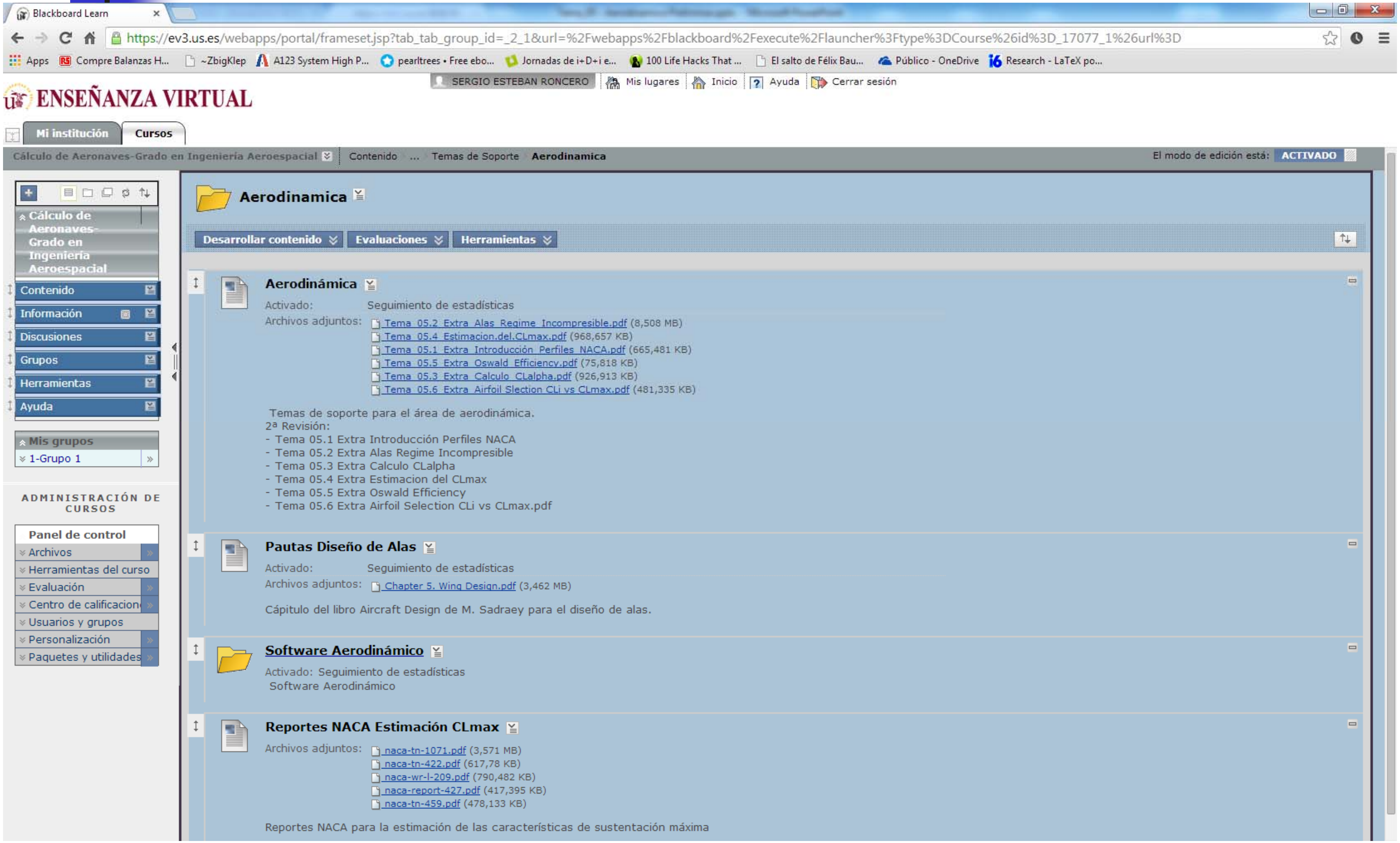
Aerodinámica – IV

- 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 -> W_0/S -> 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 parasitaría
    - Estudio comparativo de diferentes resistencias para diferentes configuraciones de fuselaje
    - Comparación con métodos clásicos

Material Docente - Aerodinámica



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**.

$$C_D = C_D\left(C_L, M\right)$$

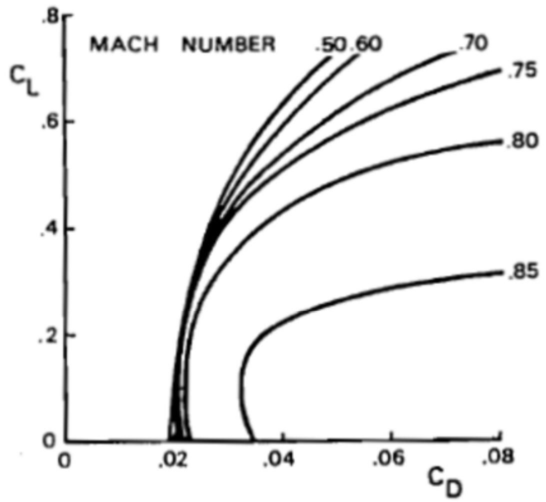
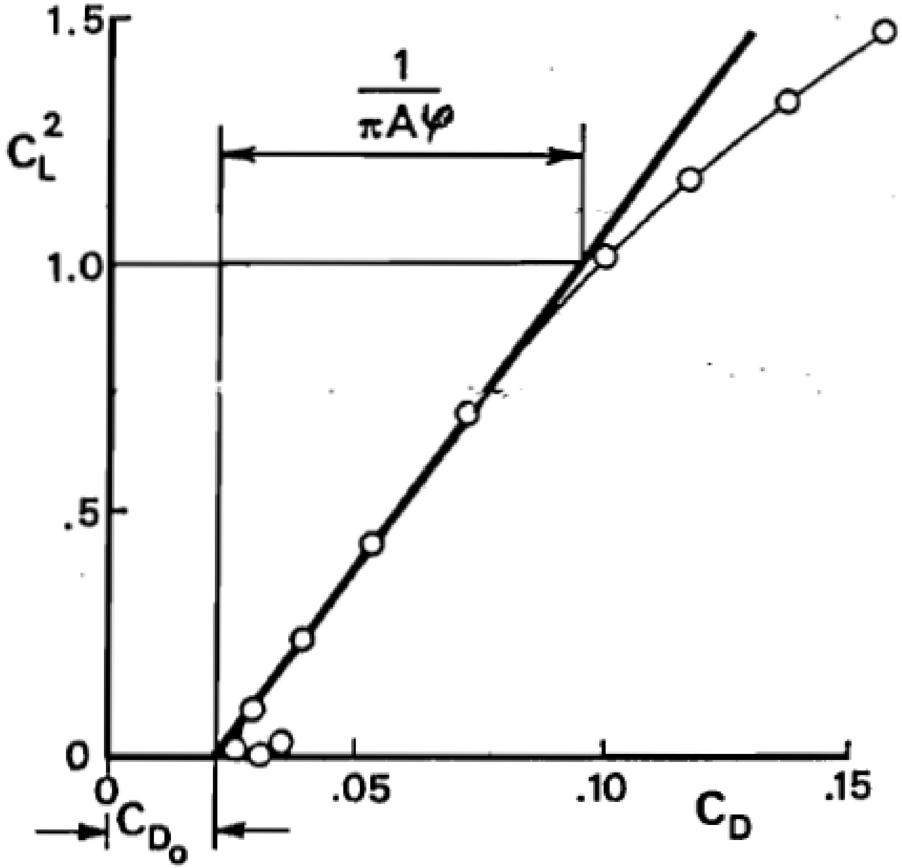


Fig. 5-6. Effect of compressibility on the drag polar

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°
  - Alargamientos mayores a 4.
- **Metodología rápida**:
  - estimación gráfica del C_{Do} y de la eficiencia aerodinámica e (Oswald efficiency) para polares parabólicas simples:

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



- 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 α=0
      - $C_{Do} = C_{Dmin}$

$$C_D = C_{D_0} + K C_L^2 \Leftarrow K = \frac{1}{\pi A e}$$

  - Alas con curvatura
    - Mínima resistencia α>0
      - $C_{Do} \neq C_{Dmin}$

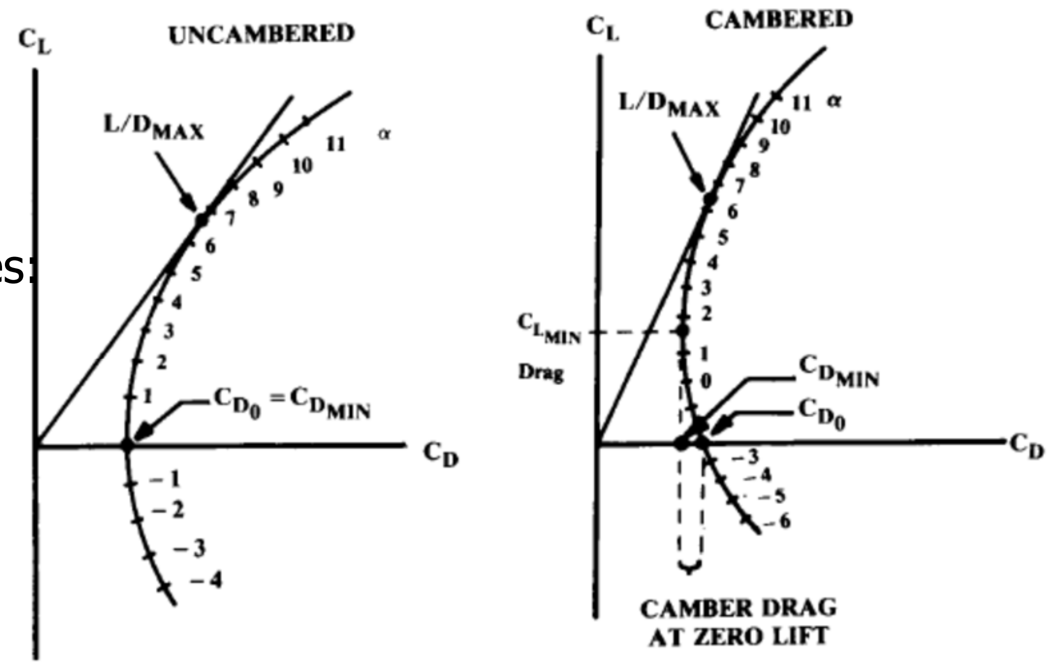


Fig. 12.3 Drag polar.

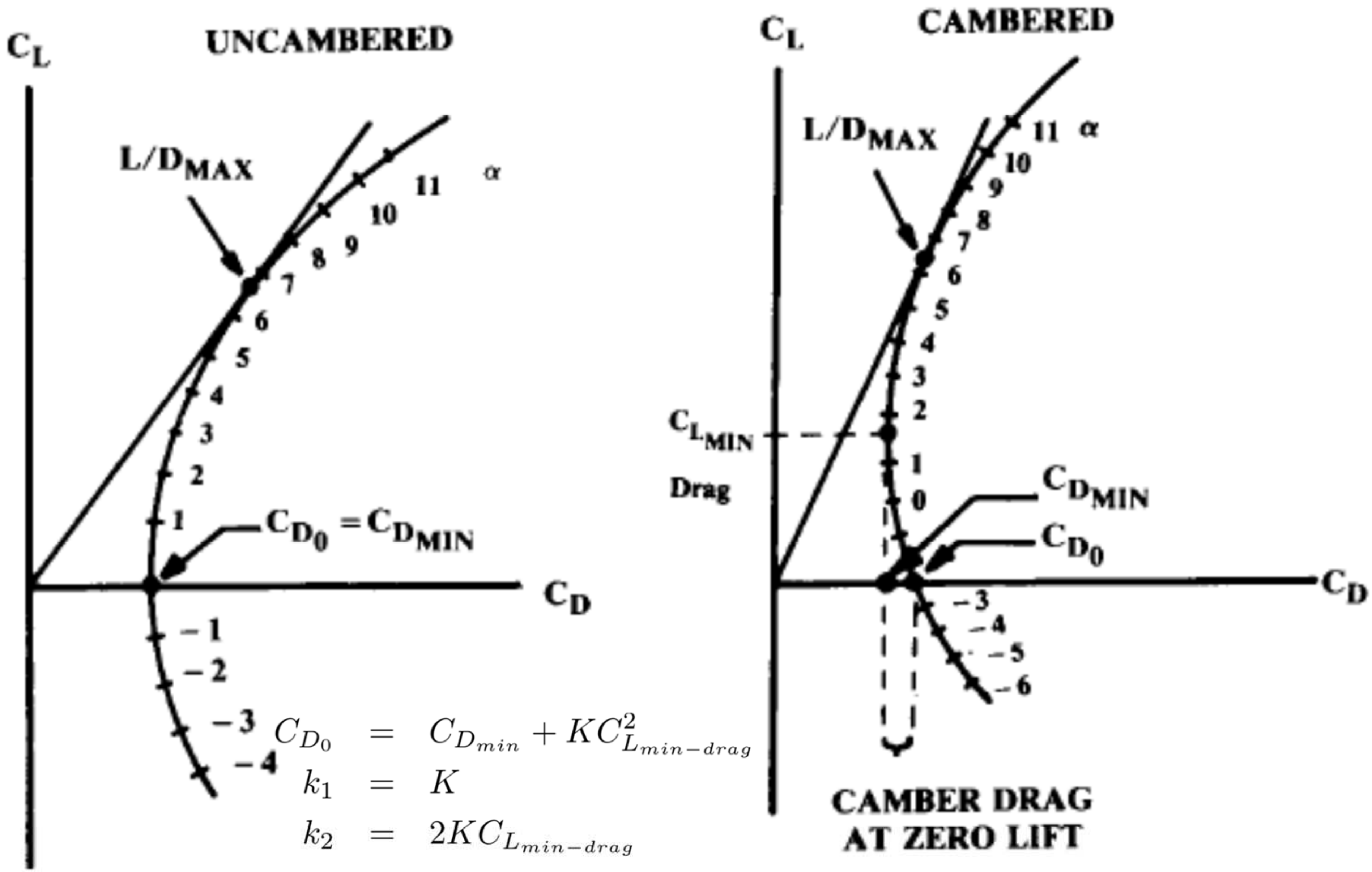
corregido

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

corregido

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

- 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



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

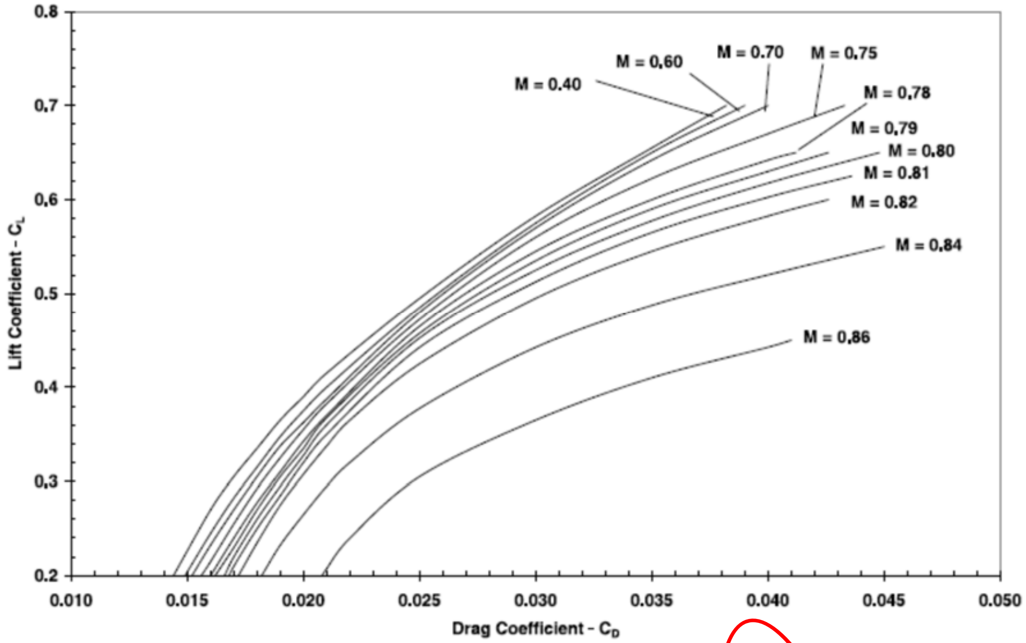
Fig. 12.3 Drag polar.

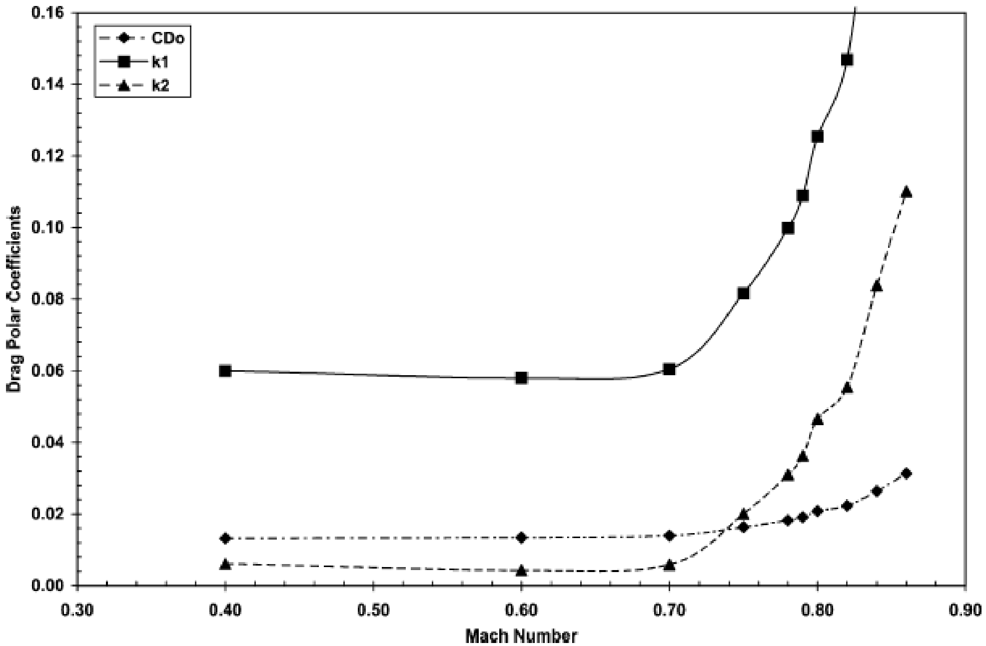
Polar del avión - III

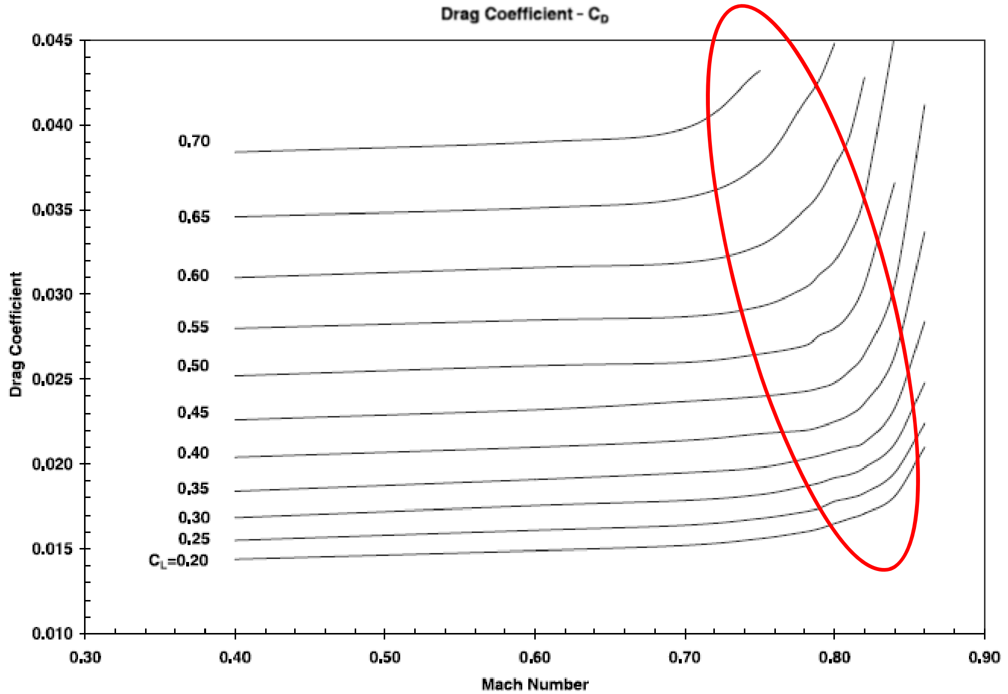
- Aproximación de los coeficientes en función del Mach de divergencia M_{DD}

$$C_D = C_D(C_L, M)$$
$$C_D = C_{D_0} + k_1 C_L^2 - k_2 C_L$$
$$C_{D_0} = C_{D_0}(M),\ k_1 = k_1(M),\ k_2 = k_2(M)$$

Normalización con respecto a los valores propios







Polar del avión - IV

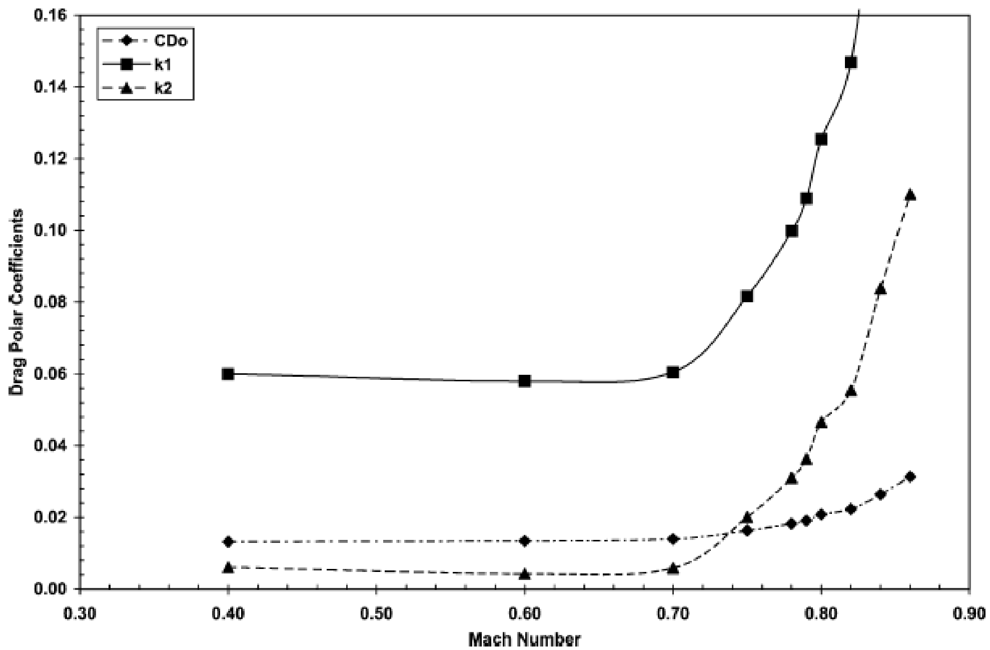
- Para aviones con perfiles de velocidad por debajo M<0.7 se observa que los coeficientes son prácticamente constantes.
- Para aviones con regímenes de vuelo M>0.7, el aumento de valor de los coeficientes es exponencial:
  - Se harán estimaciones normalizando los valores presentes para aviones (Boeing 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

$$C_D = C_D\left(C_L, M\right)$$

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

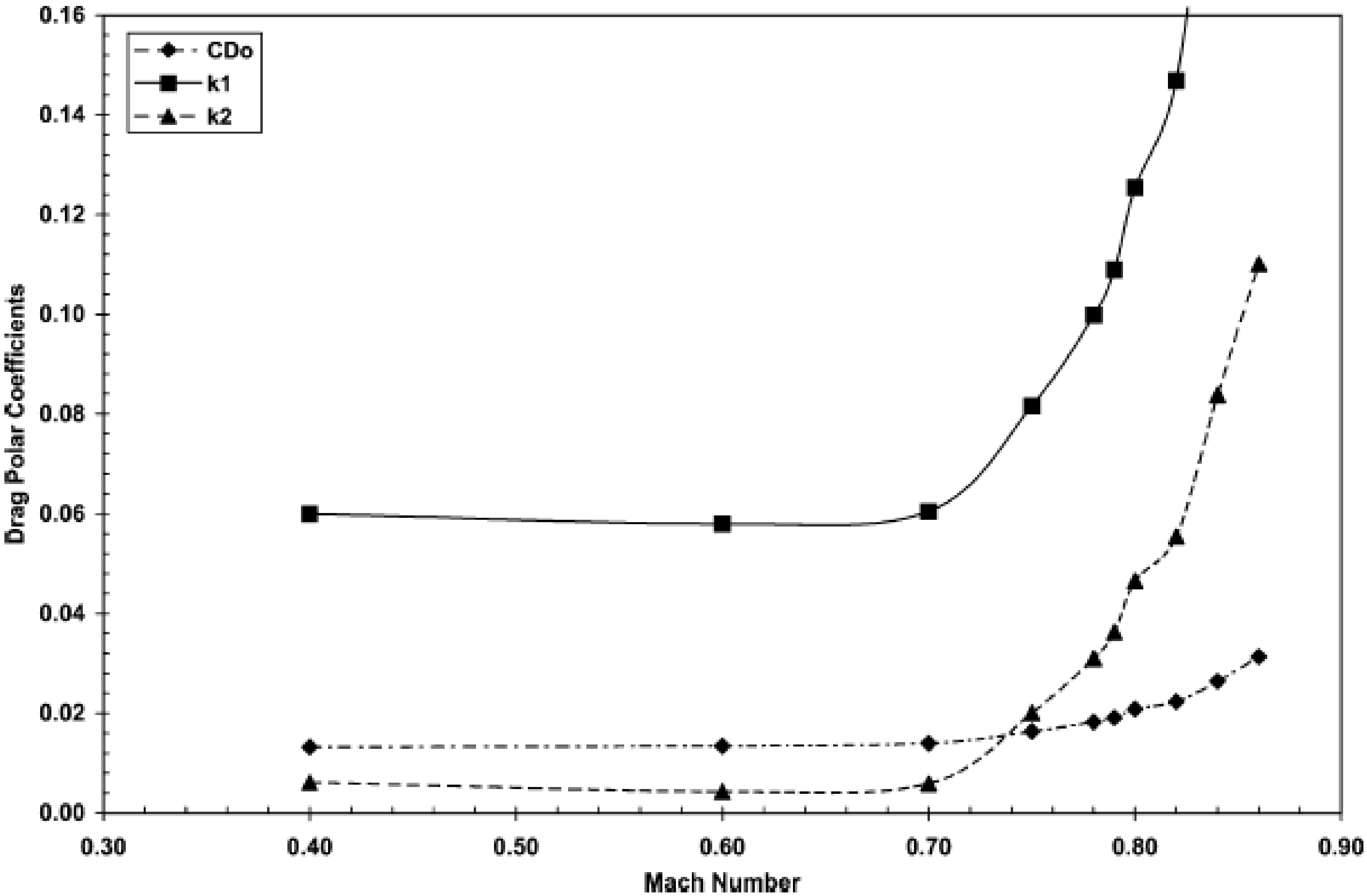
$$C_{D_0} = C_{D_0}\left(M\right),\ k_1 = k_1\left(M\right),\ k_2 = k_2\left(M\right)$$



Polar del avión - V

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

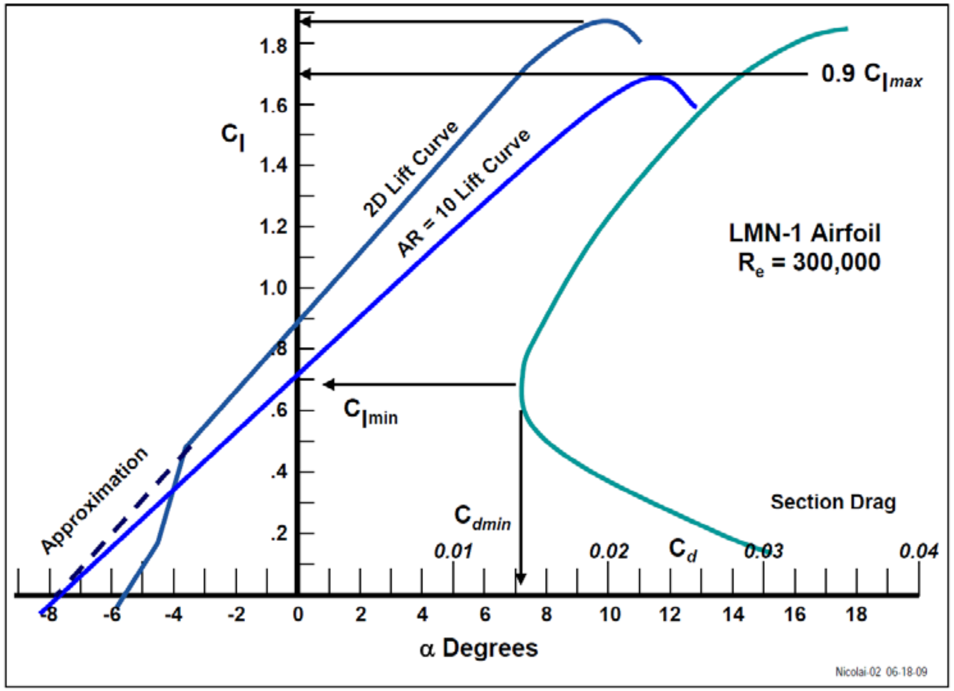
$$C_{D_0} = C_{D_0}(M)\,,\ k_1 = k_1(M)\,,\ k_2 = k_2(M)$$

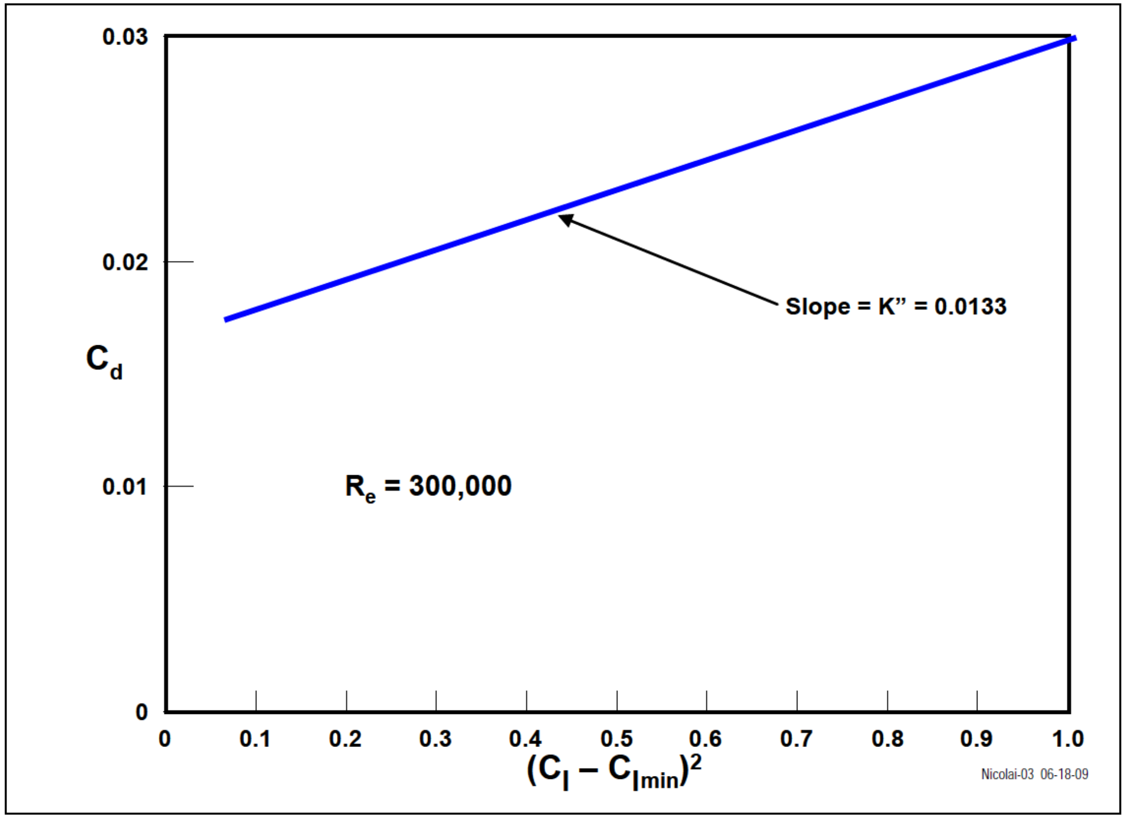


Polar del avión - VI

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

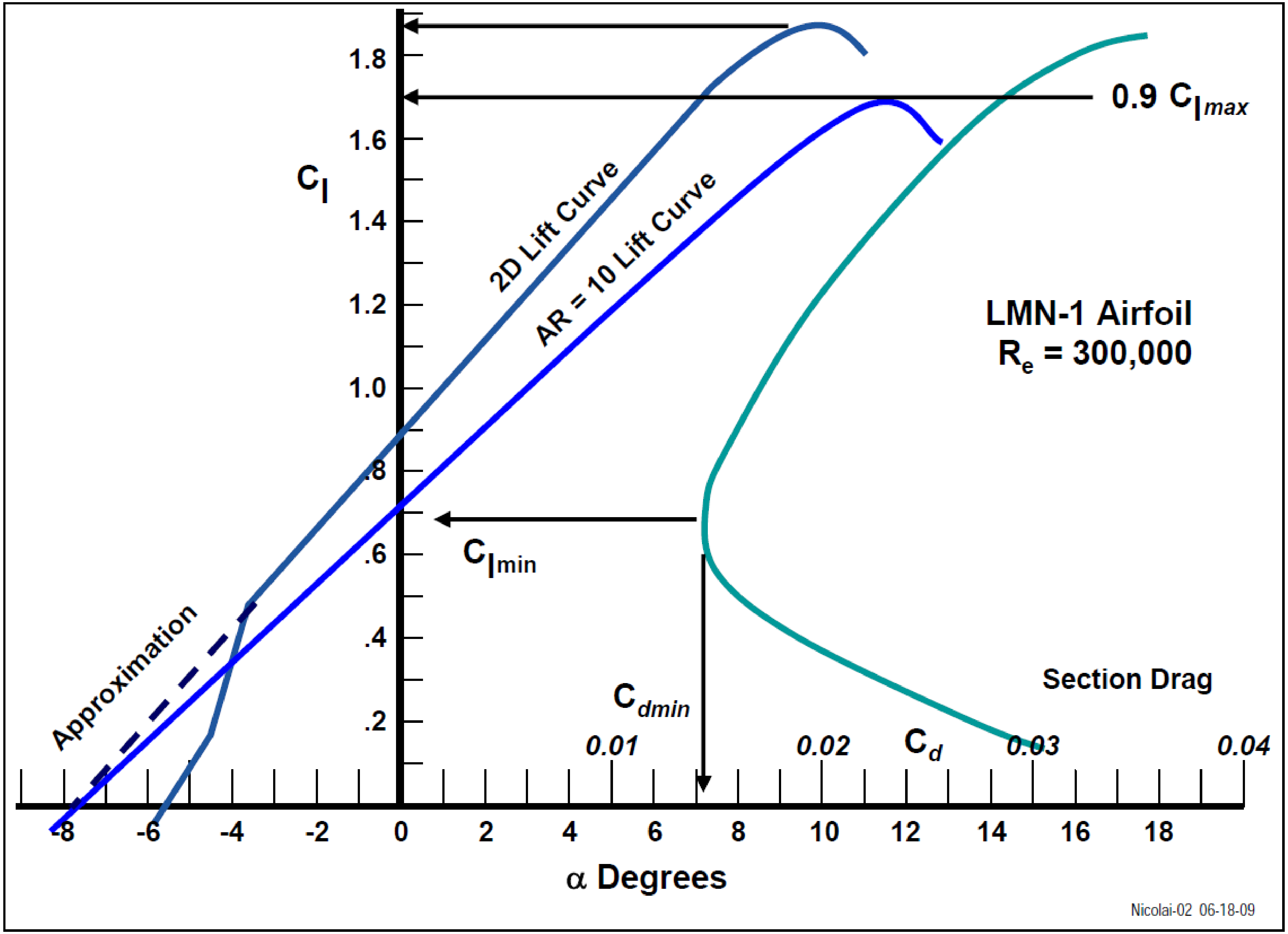
$$C_D = C_{D_{min}} + K'C_L^2 + K''\left(C_L - C_{L_{min-drag}}\right)^2 \qquad K' = \frac{1}{\pi A e}$$



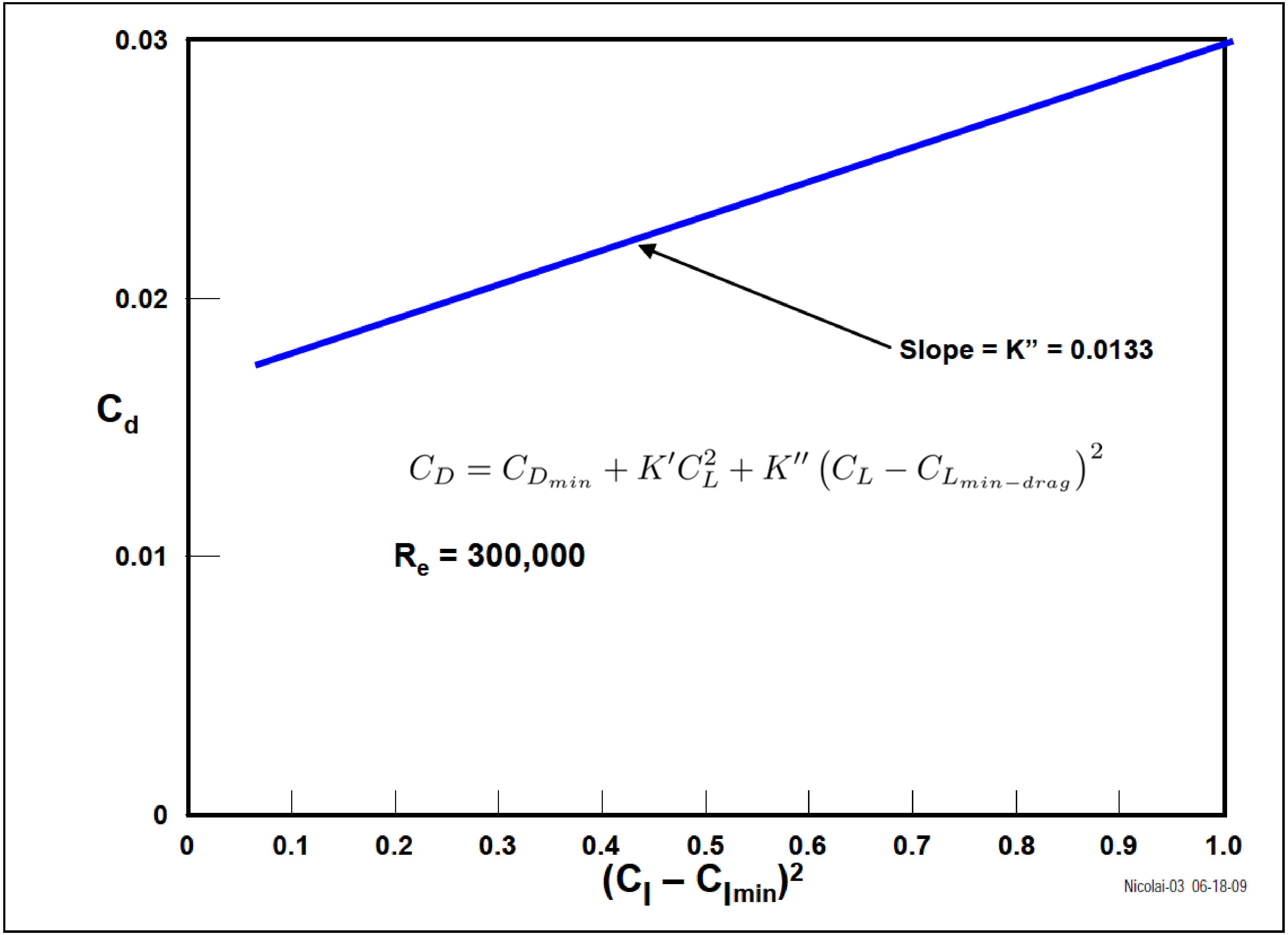


Aproximación Ala

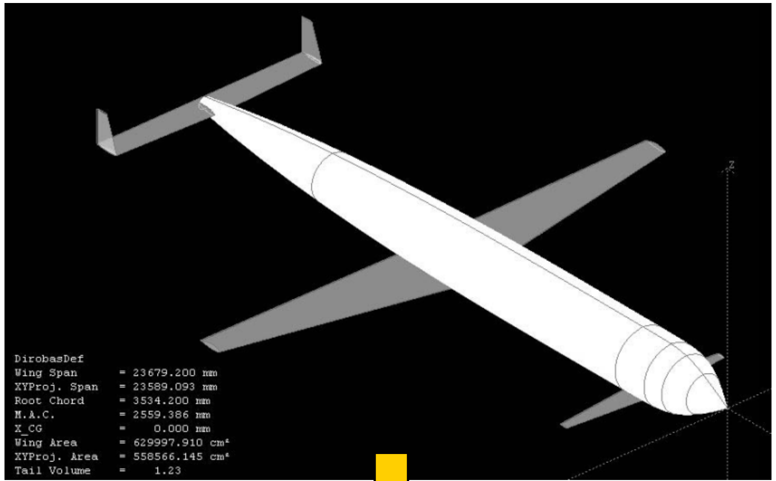
Polar del avión - VII

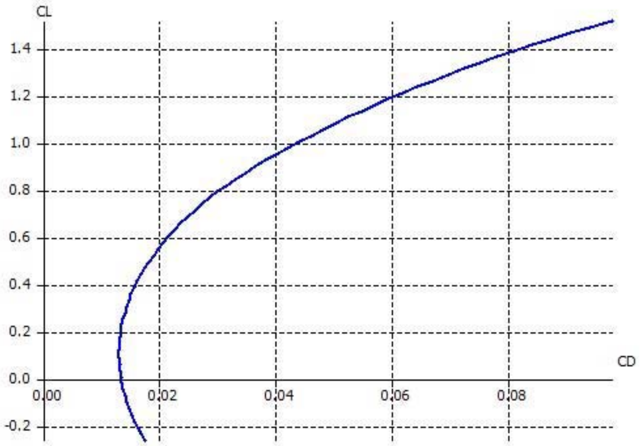


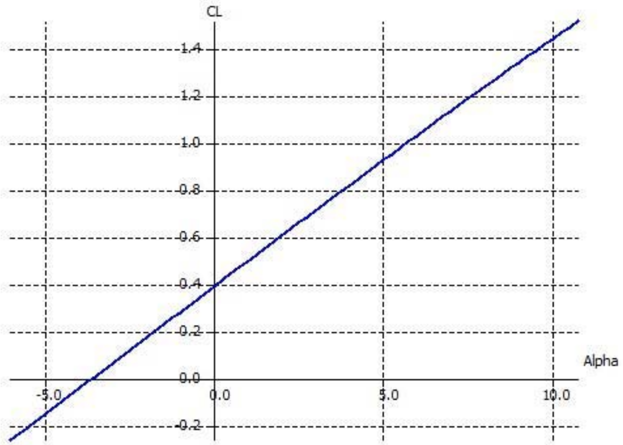
Polar del avión - VIII



Polar del avión - X



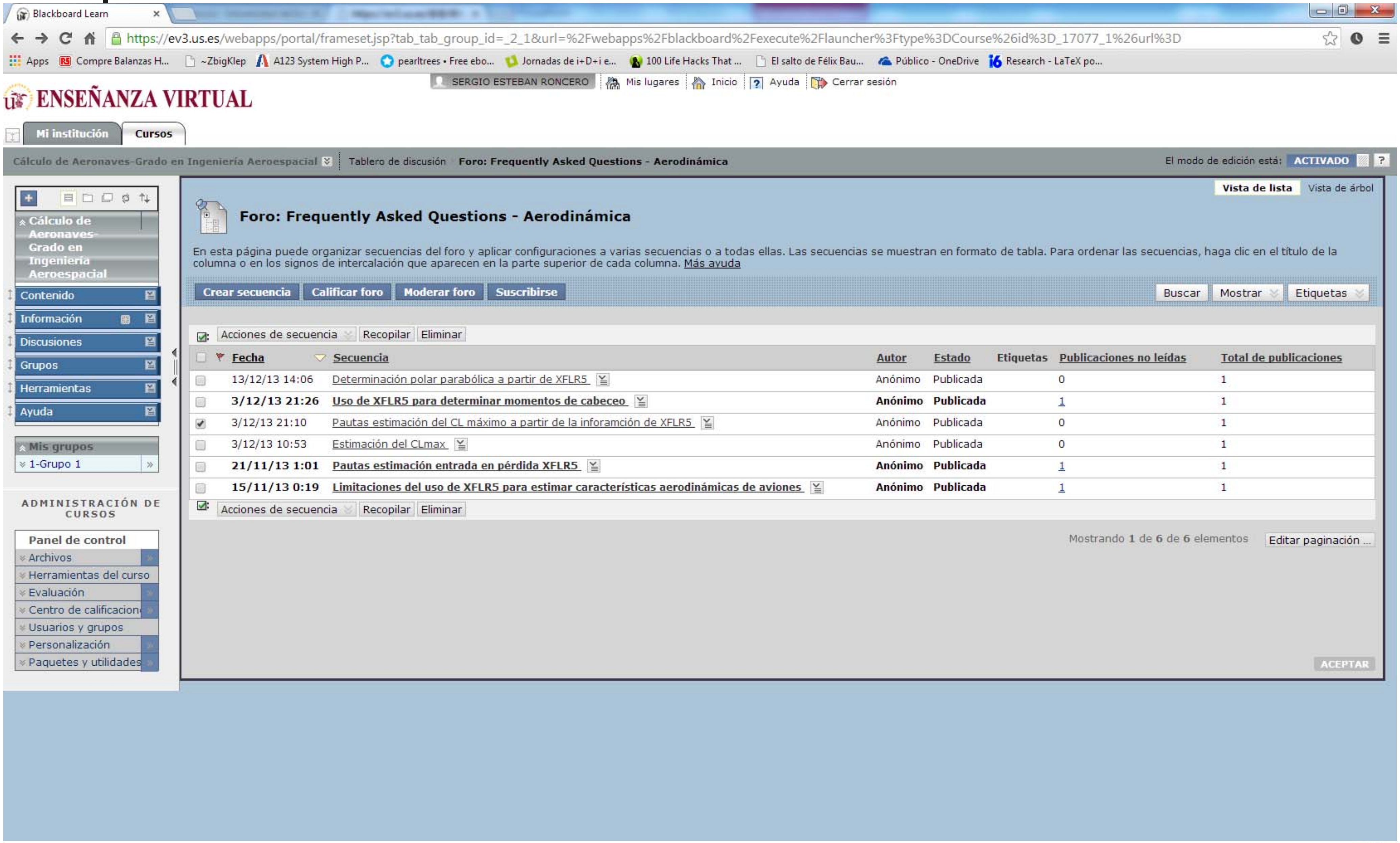




$$C_D = C_{D_0} + k_1 C_L^2 + k_2 C_L$$

Ajuste polinómico

Polar del avión - XI

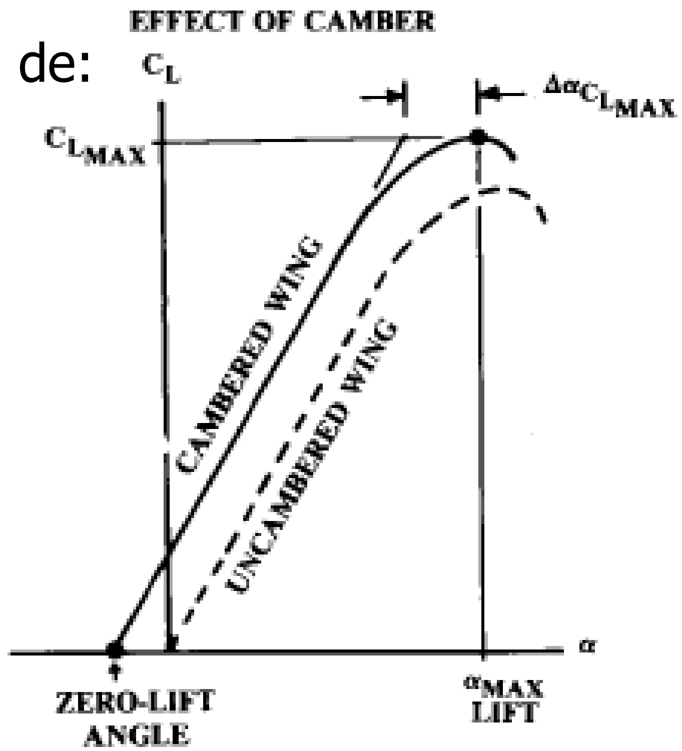


Coeficiente de Sustentación - C_L

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

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

  - La pendiente de la curva de empuje se ve modificado con el alargamiento, siendo el **valor teórico** para toda ala con **alargamiento ∞ = 2π**
  - Dicha pendiente tiene que ser **corregida** para el **alargamiento** de cada ala por lo que deja de ser el teórico 2π.
  - 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 (Λ).
    - Eficiencia aerodinámica del perfil (E).
  - Método diferentes para subsónico y supersónico



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

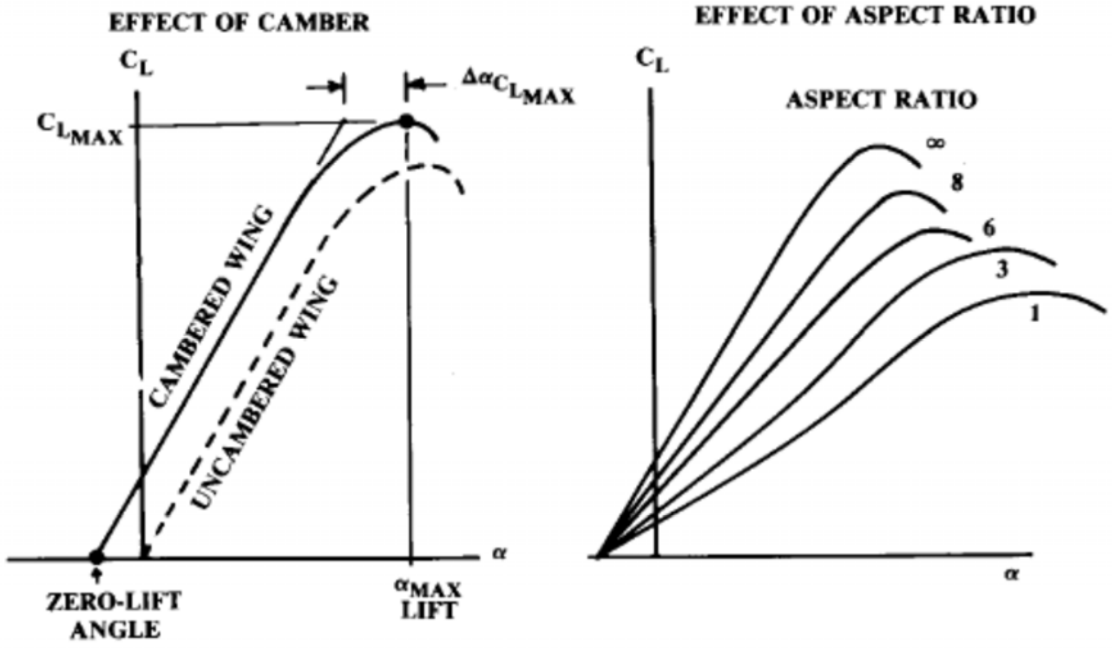


Fig. 12.4 Wing lift curve.

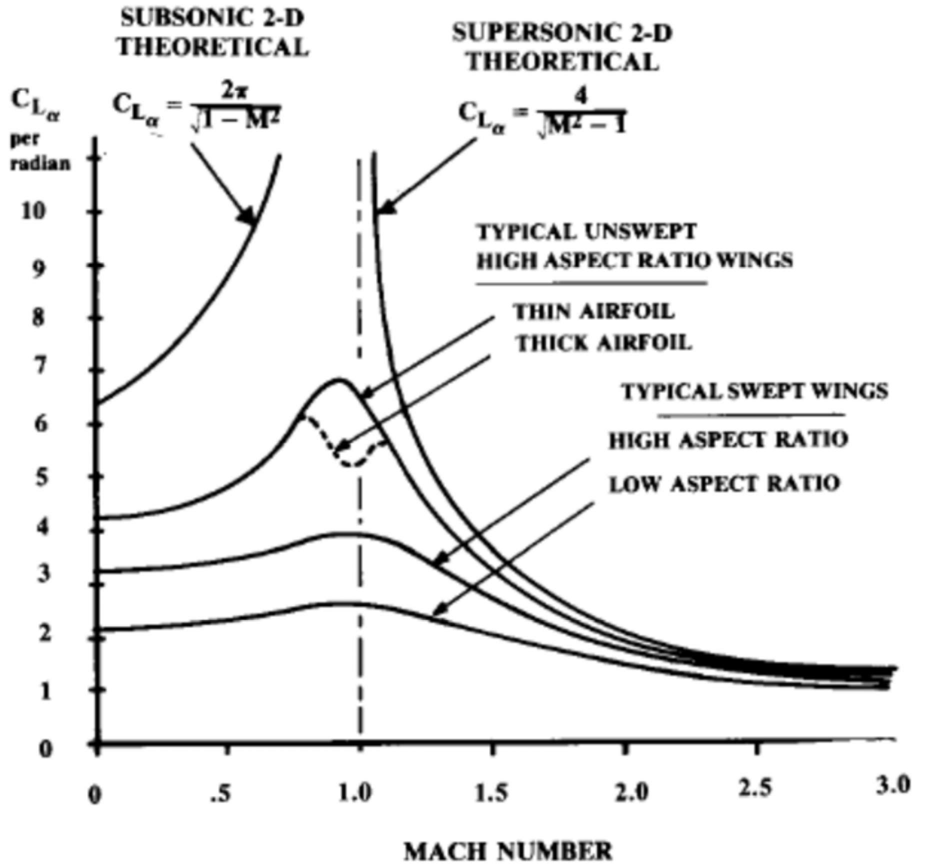


Fig. 12.5 Lift curve slope vs Mach number.

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

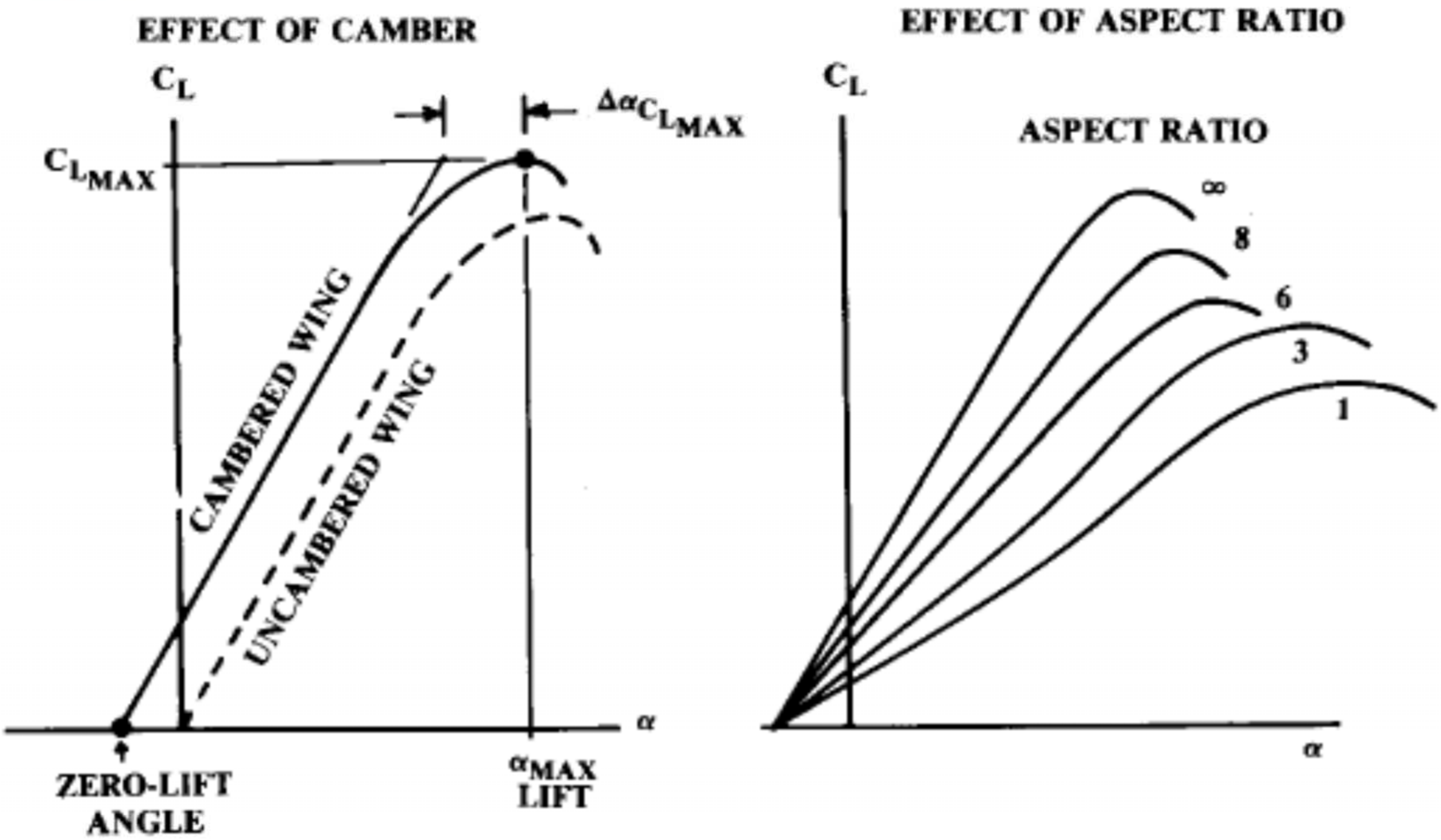


Fig. 12.4 Wing lift curve.

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

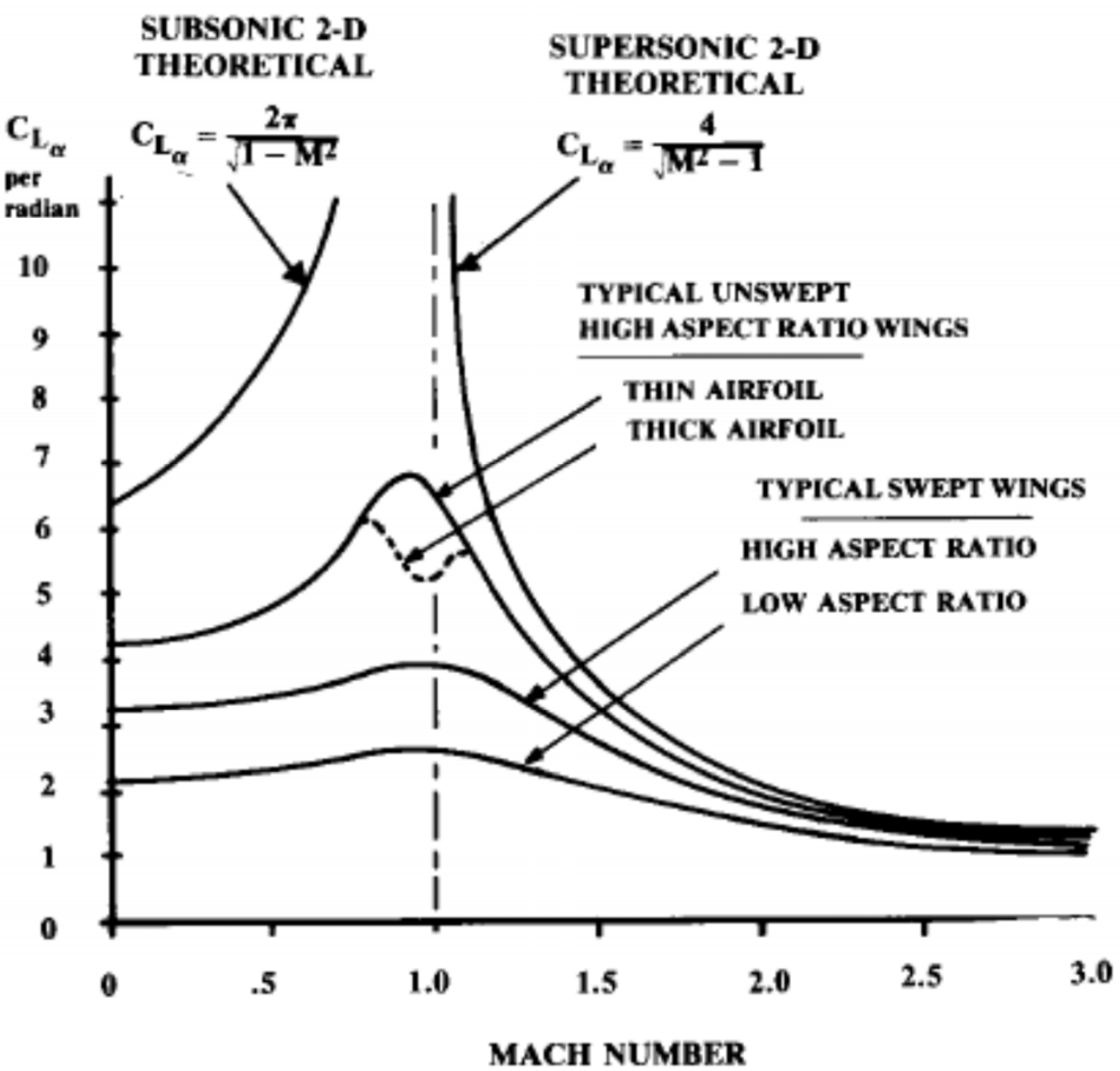


Fig. 12.5 Lift curve slope vs Mach number.

$C_{L\alpha}$ 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 perdida 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 (η)
- 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}!!

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

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

Pendiente sustentación 2D

$$\eta = \frac{C_{l_\alpha}}{2\pi/\beta}$$

Eficiencia aerodinámica del perfil

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

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

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

$C_{L\alpha,w}$, $C_{L\alpha,t}$ 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

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

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

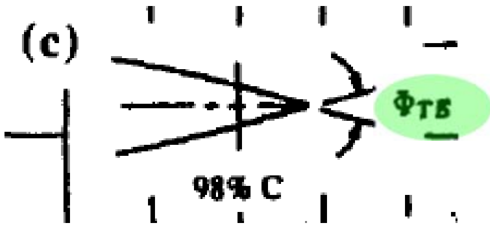
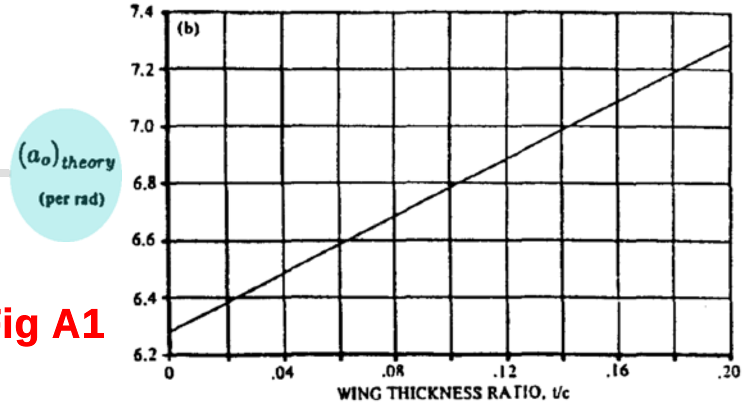
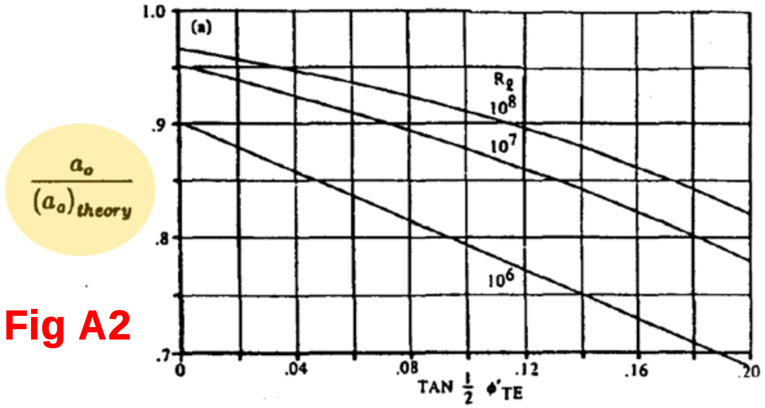


Fig A1



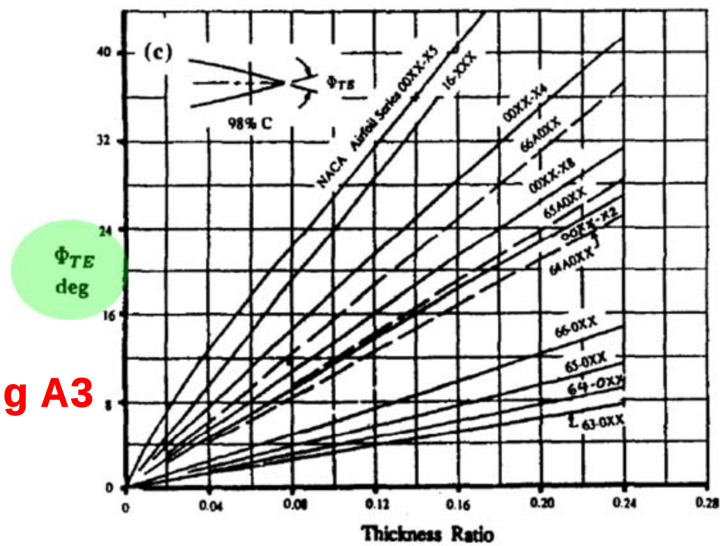
a) Theoretical sectional lift-curve slope

Fig A2



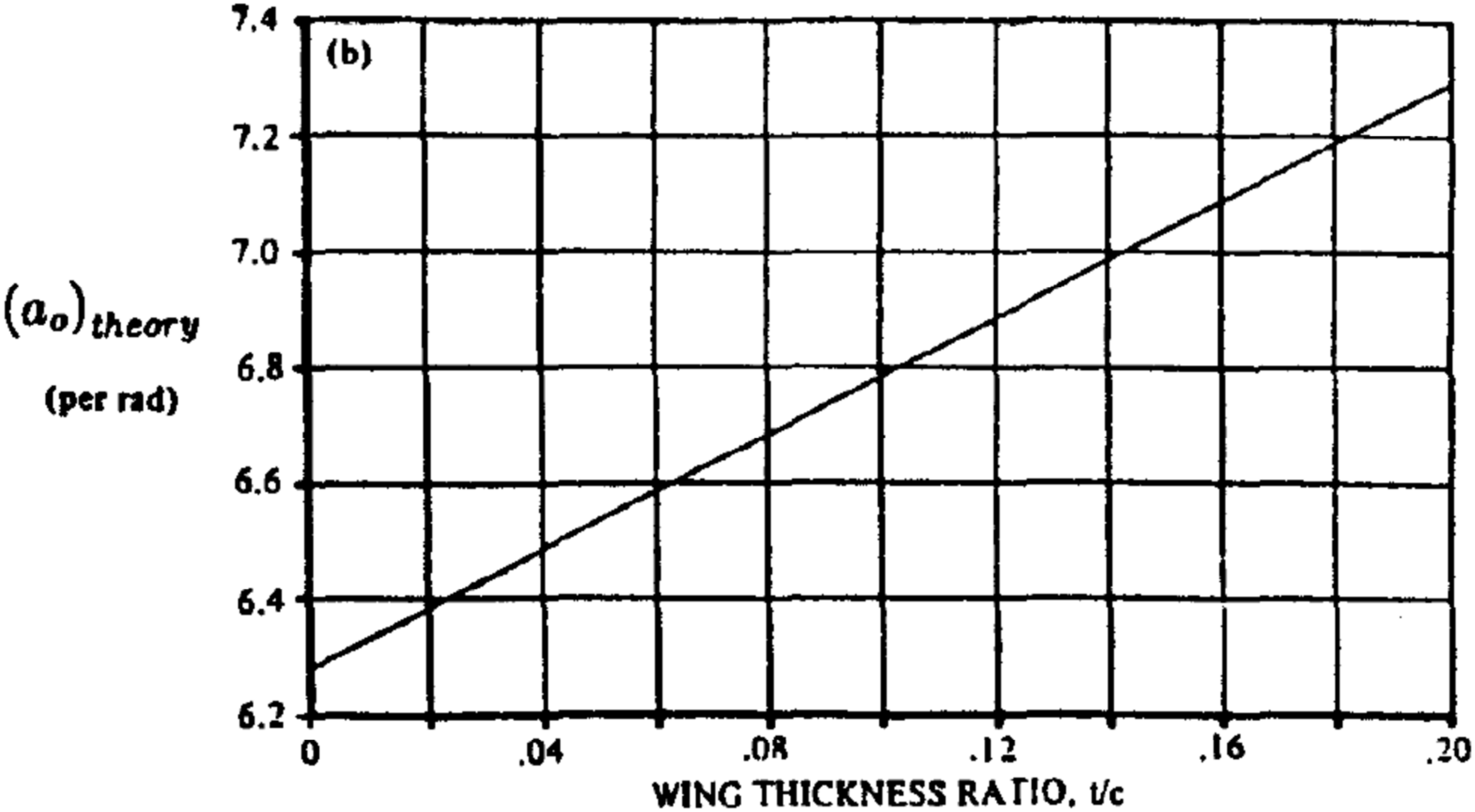
b) Empirical correction factor

Fig A3



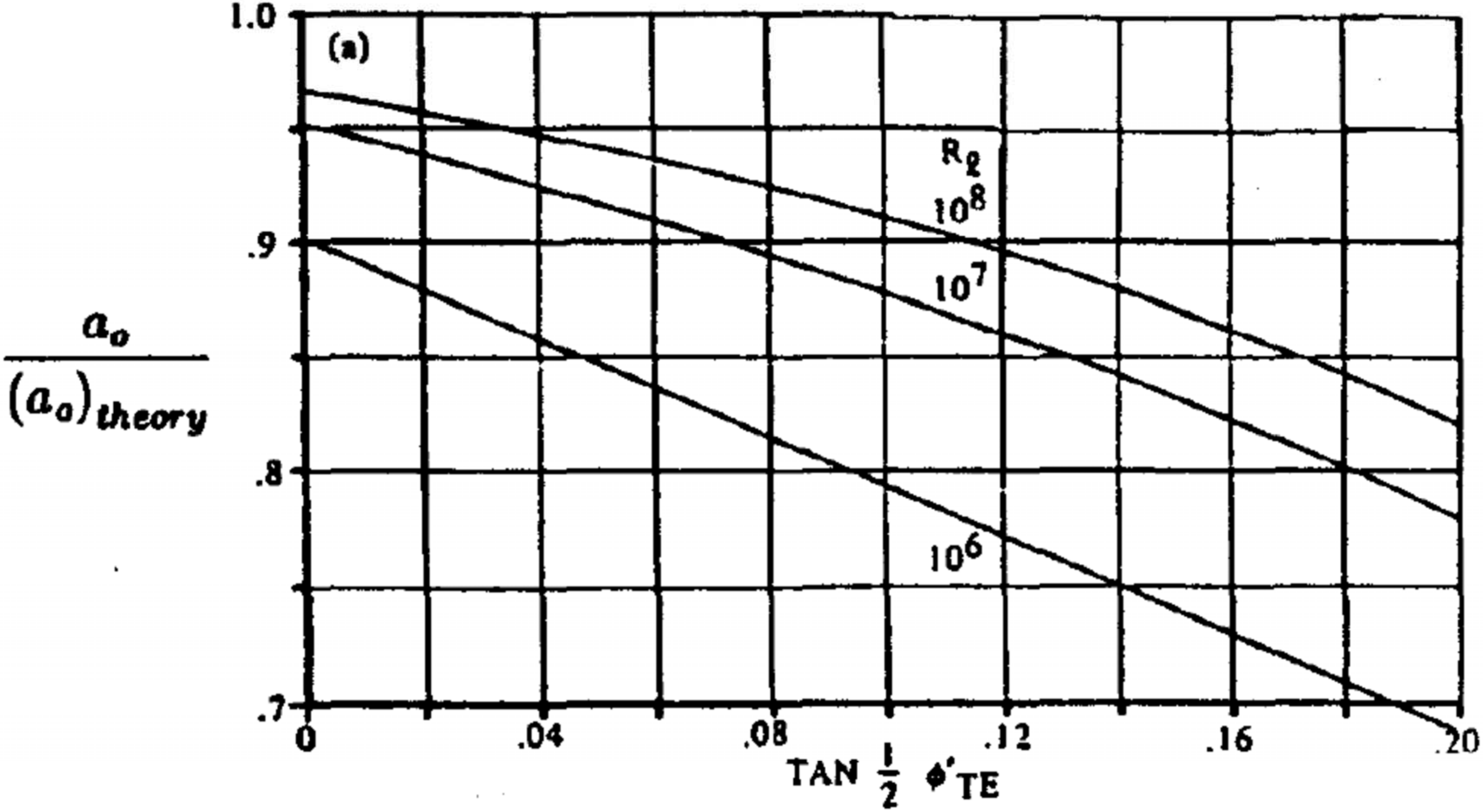
c) Variation of trailing-edge angle ϕ_{TE}

Fig A1



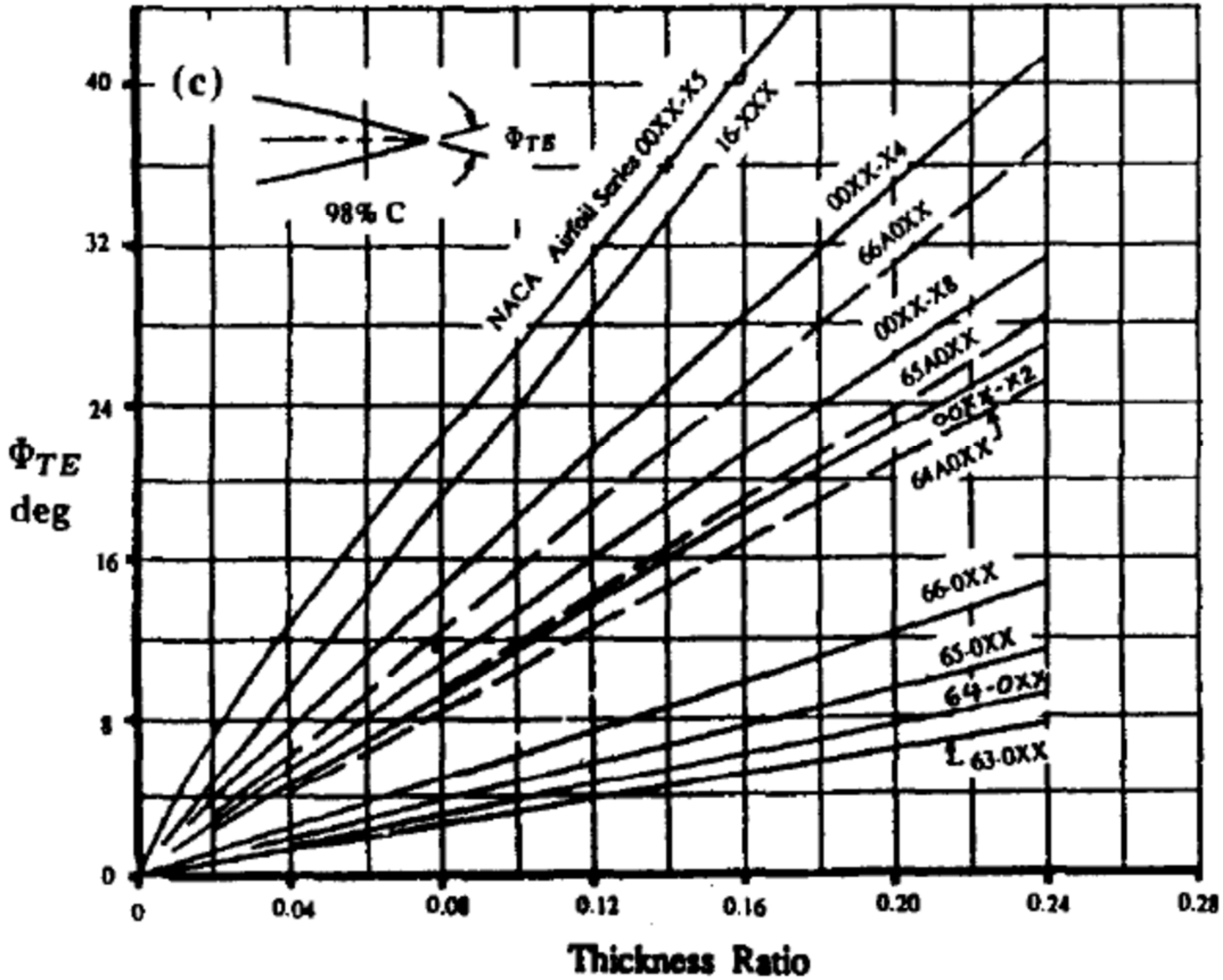
a) Theoretical sectional lift-curve slope

Fig A2



b) Empirical correction factor

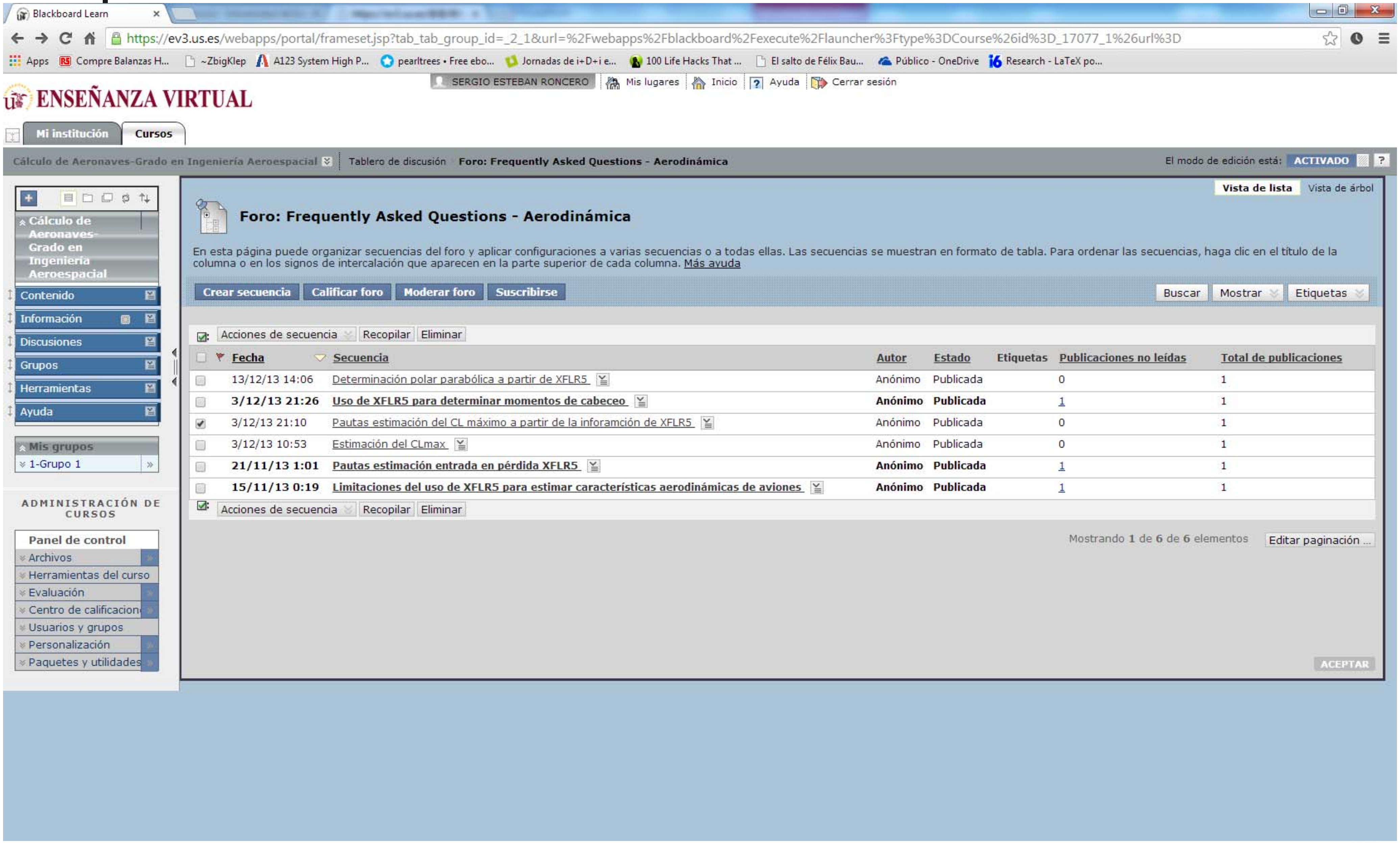
Fig A3



c) Variation of trailing-edge angle ϕ_{TE}

Fig. 3.13 Sectional (two-dimensional) lift-curve slope of wings, continued.[1]

Estimación $C_{L\alpha}$ - XFLR5



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

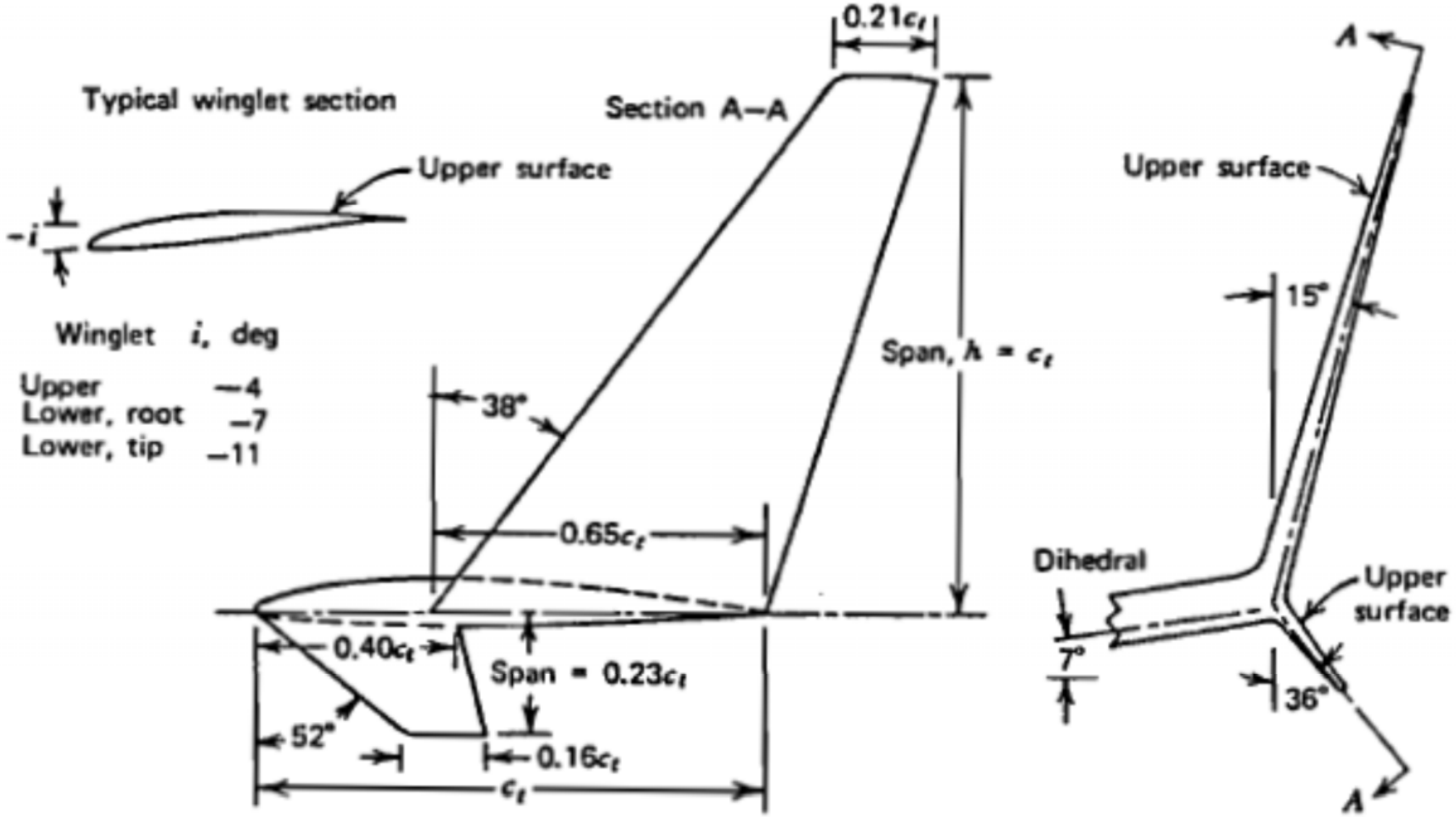
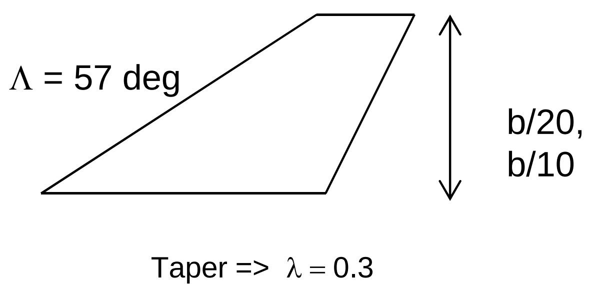


Figure 4.35 Winglet geometry.

Winglets – Design - II

- Perfil (airfoil)
  - Tiene que generar suficiente sustentación mientras mantiene un 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 regimenes de vuelo



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 montado 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

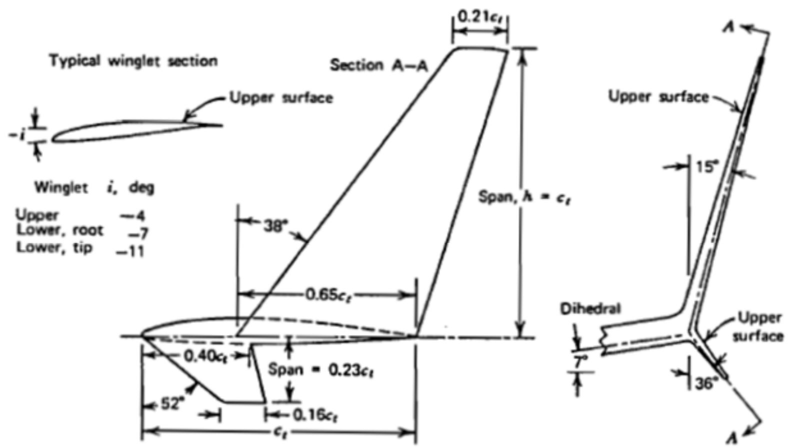


Figure 4.35 Winglet geometry.

Winglets - II

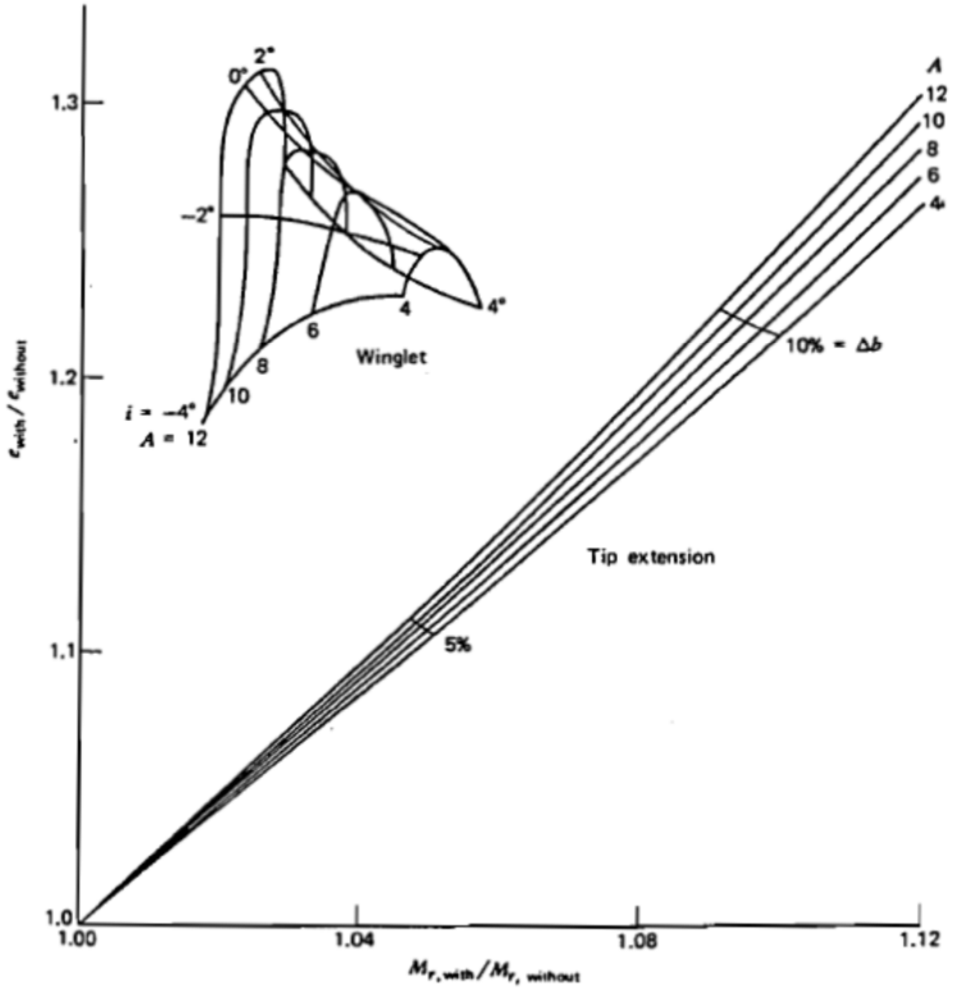


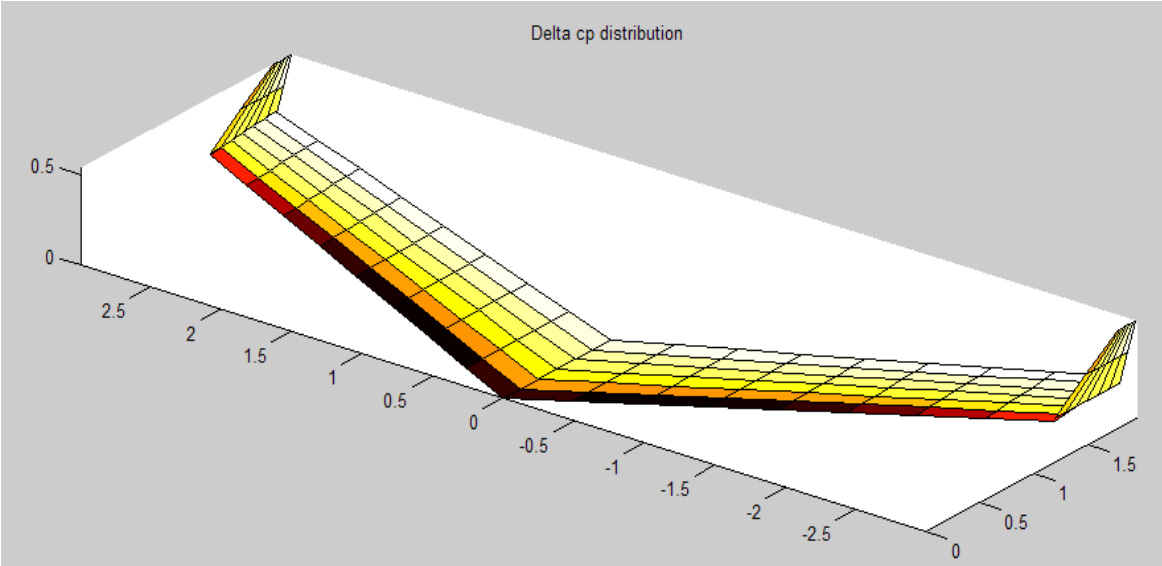
Figure 4.39 Comparison of tip extension and winglet when added to an untwisted wing.

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 incrementeado 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

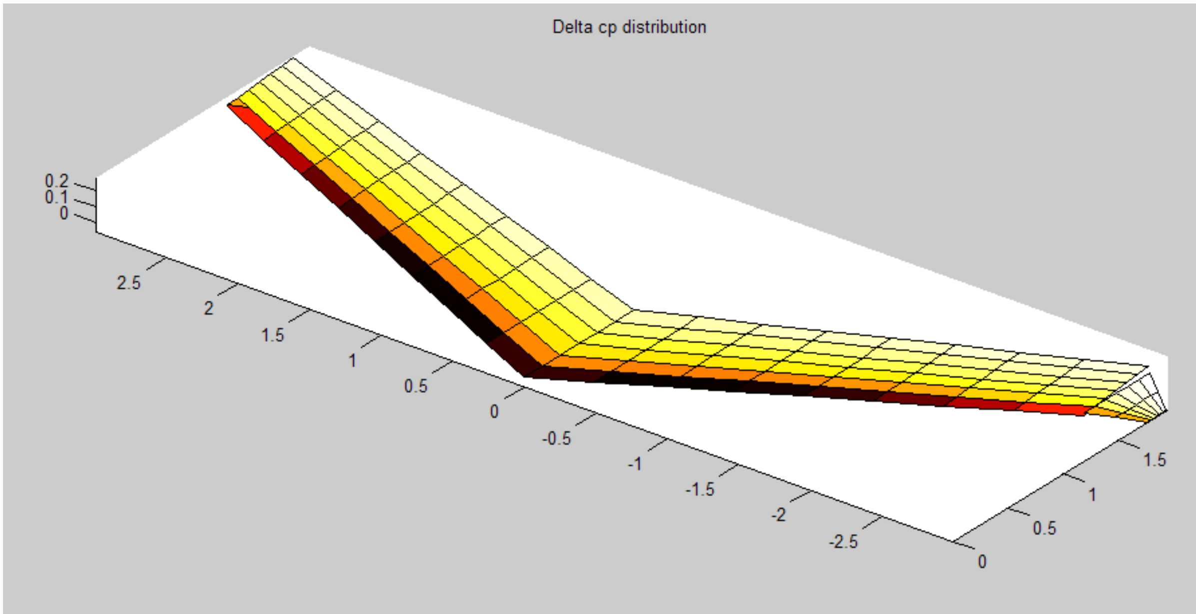


11% drag reduction

(7% when compared to an extended wing)

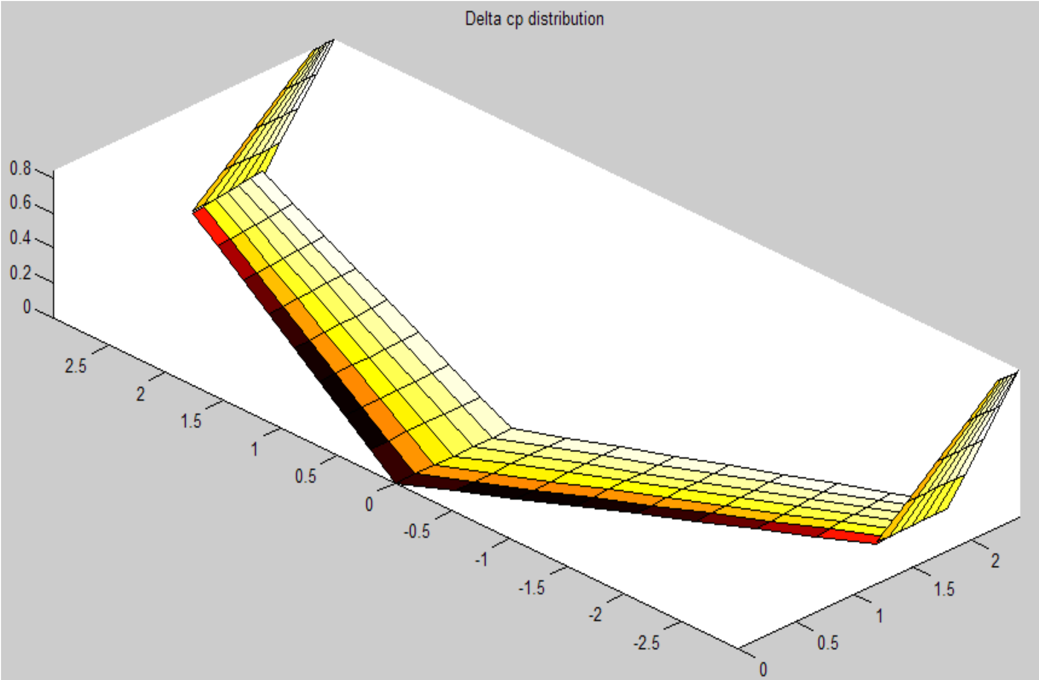
8% drag reduction

(4% when compared to an extended wing)



Winglets – Resultados - III

Large Version

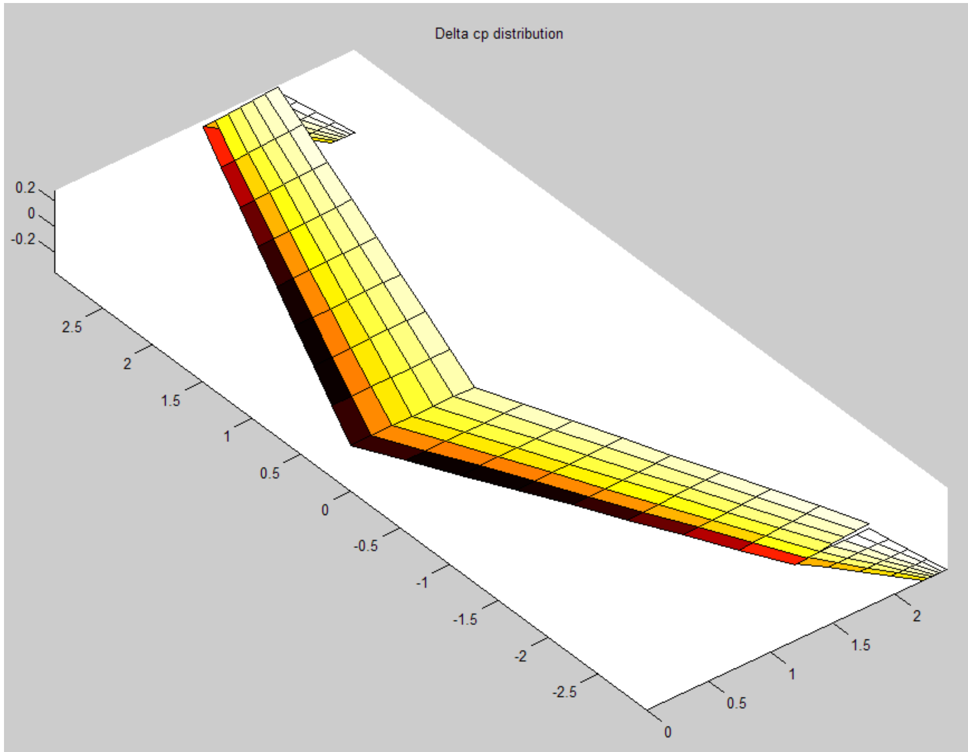


22% drag reduction

(14% when compared to an extended wing)

12% drag reduction

(4% when compared to an extended wing)



Winglets

B 737-800 blended winglet

KC-135 winglet

Raked Wings

B 747-400 tip plus winglet

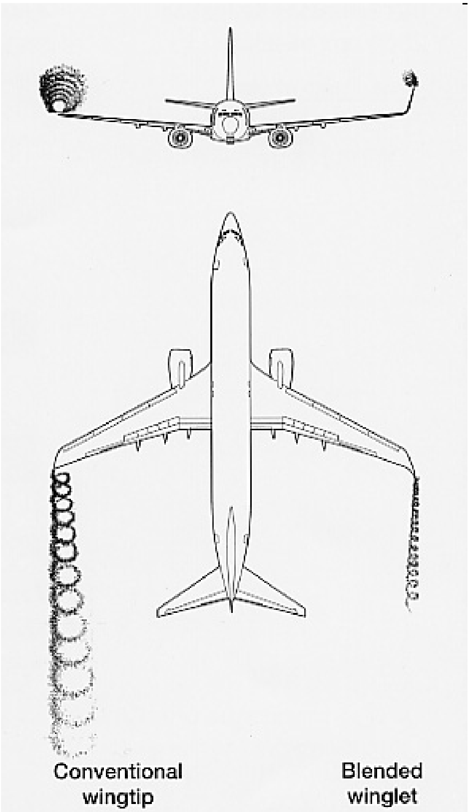
MD-11 Extended winglet

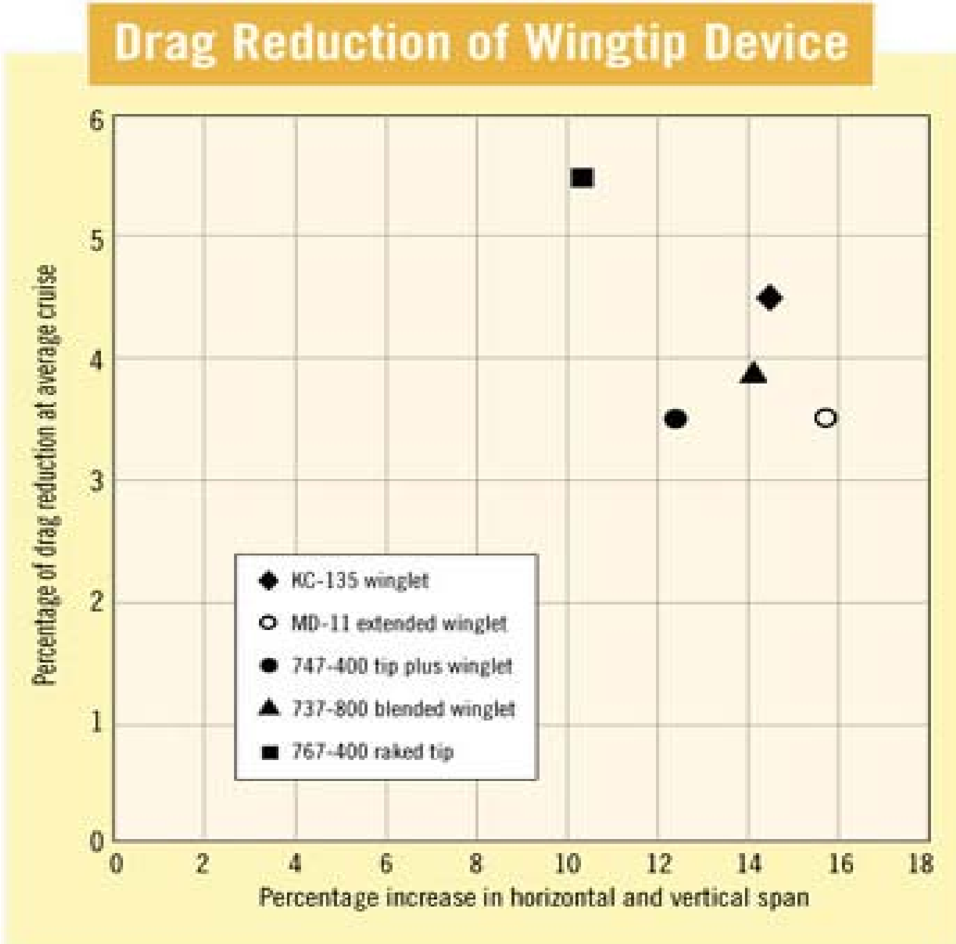
Reducción Resistencia Inducida

- 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%

reducción %

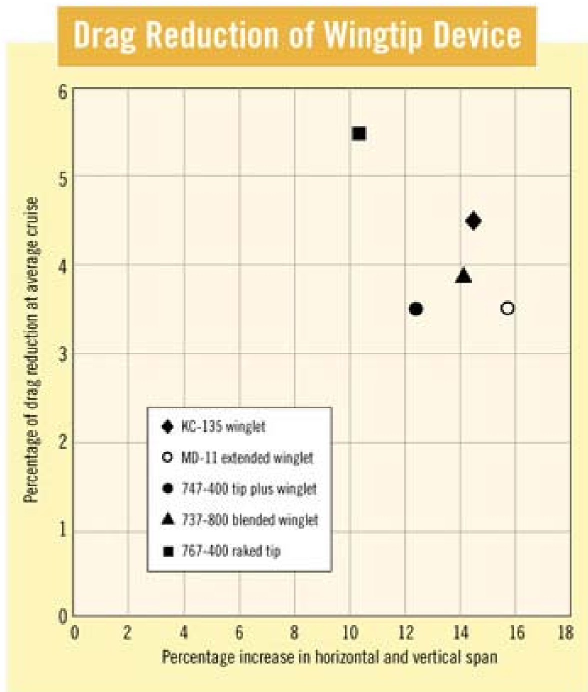
¿Cómo lo aplico? $\Rightarrow$ $C_D = C_{D_0} + KC_L^2$





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

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

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

Cálculo del C_{Lmax} - I

- El $\mathbf{C_{Lmax}}$ **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 $\mathbf{C_{Lmax}}$ 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.

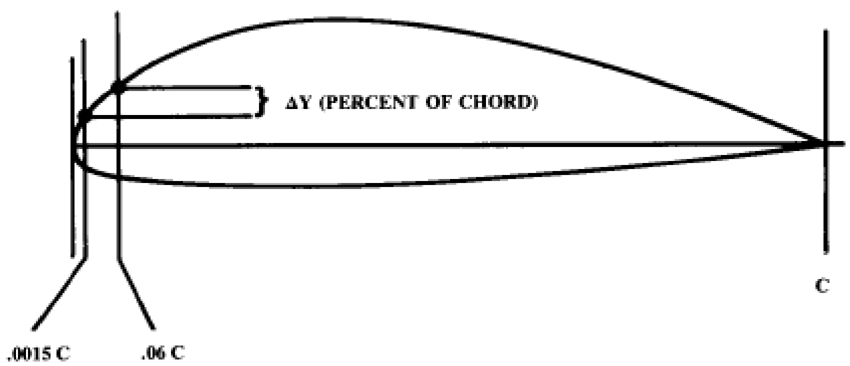
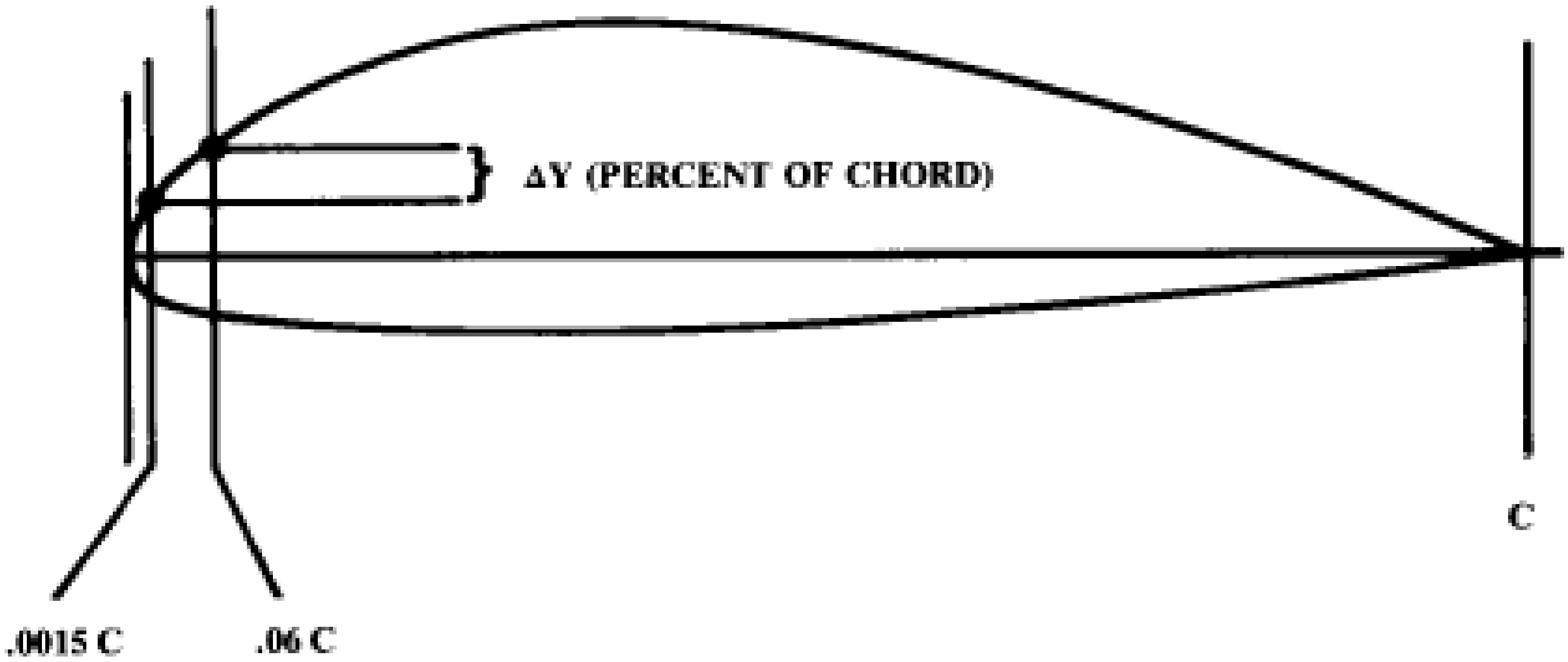


Fig. 12.7 Airfoil leading edge sharpness parameter.

Introducción a los perfiles NACA

Table 12.1 ΔY for common airfoils

Airfoil type	ΔY
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



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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 %.

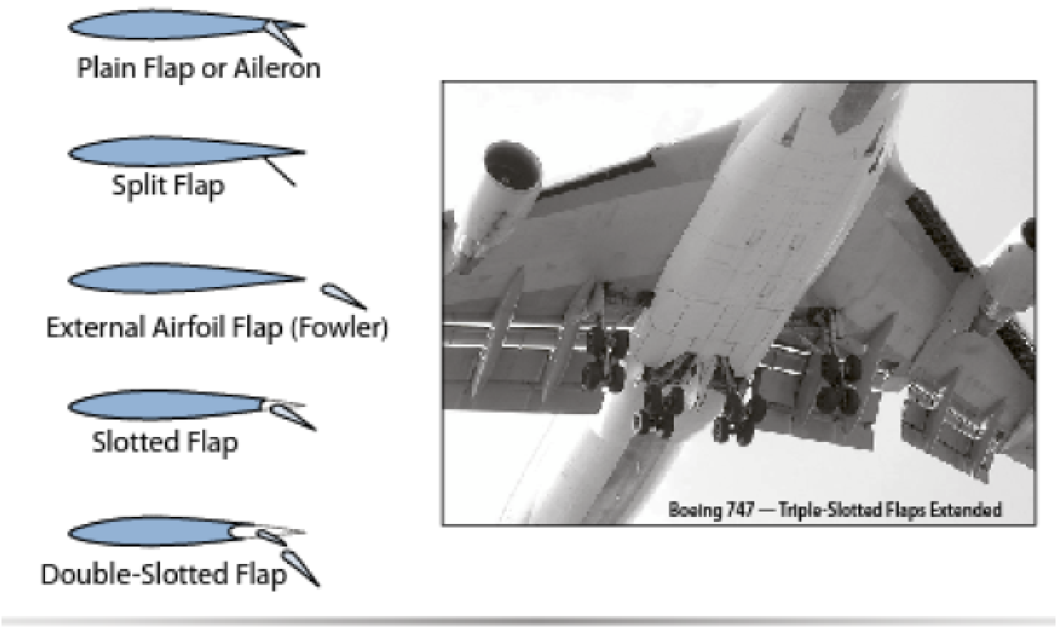


Figure 9.1 Typical TE high-lift devices.

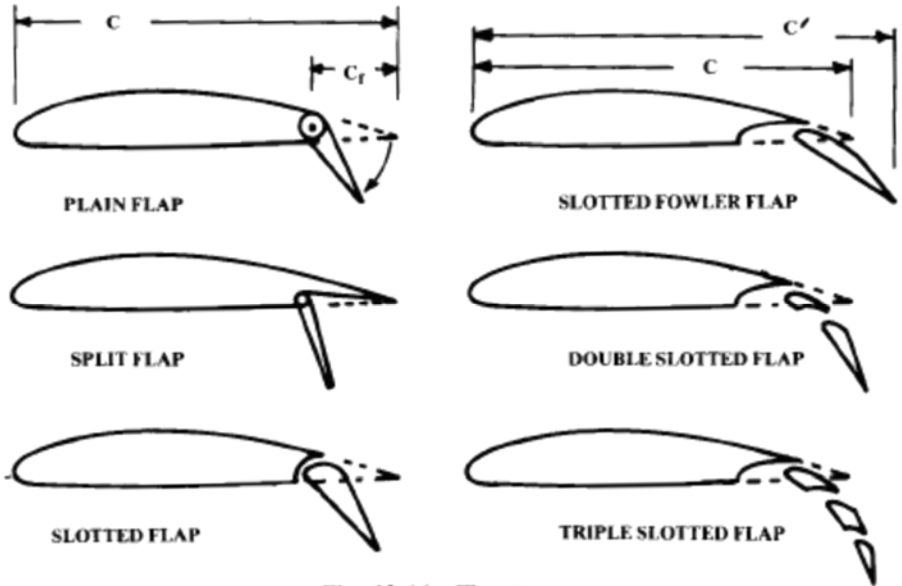


Fig. 12.16 Flap types.

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.

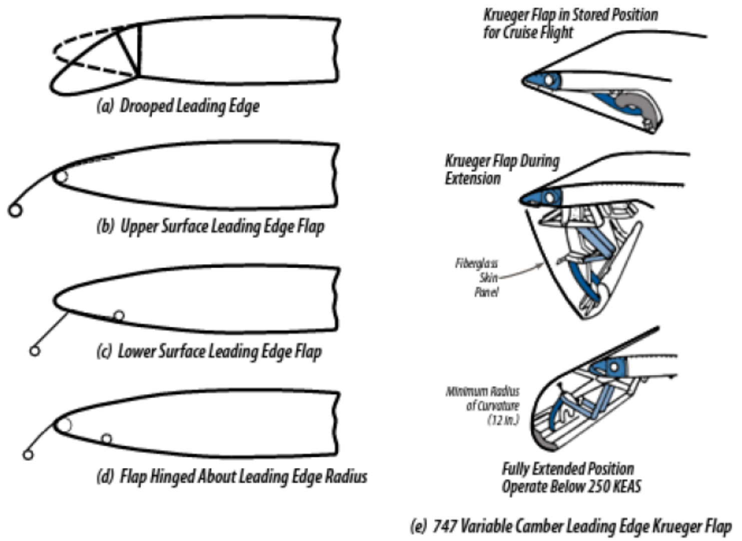


Figure 9.5 Various LE flap devices.

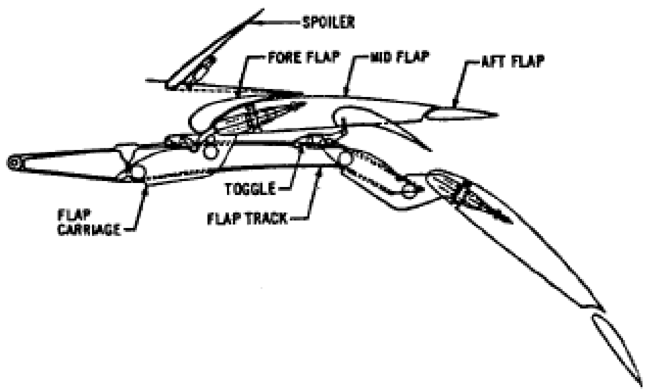


Fig. 7-25h. Triple slotted flap (Boeing 727)

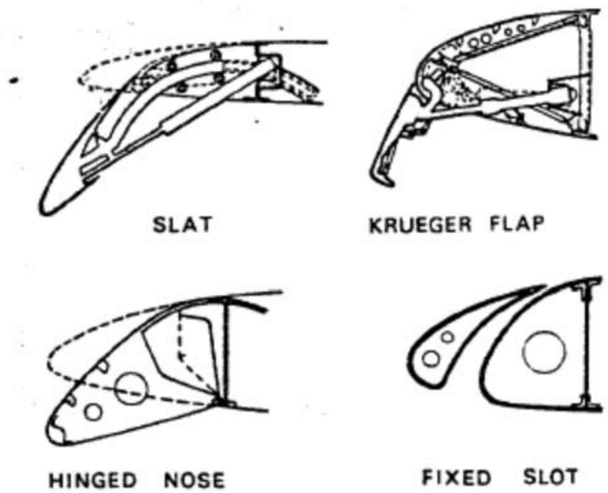


Fig. 7-26. Leading-edge high-lift devices.

Superficies Hipersustentadoras – II - cont

TE HLD Complejos

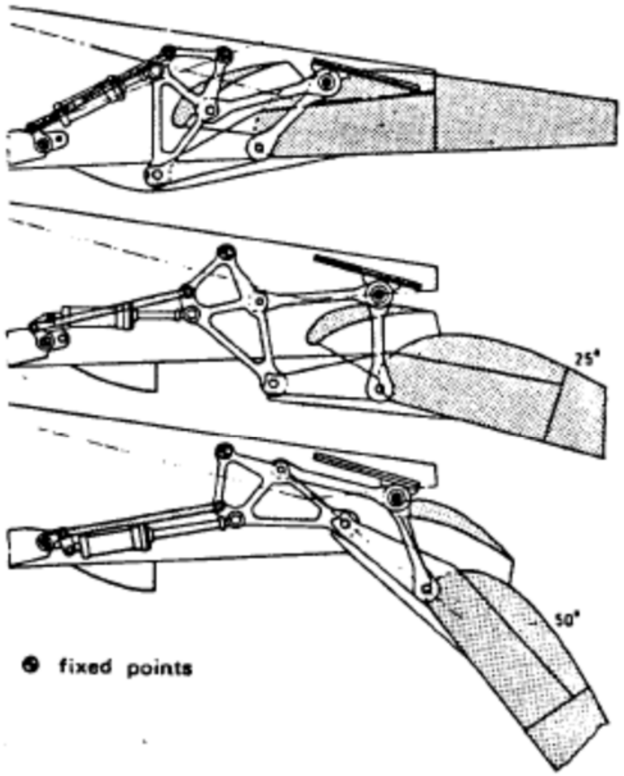


Fig. 7-25f. Double slotted flap with four-bar motion (Douglas DC-8)

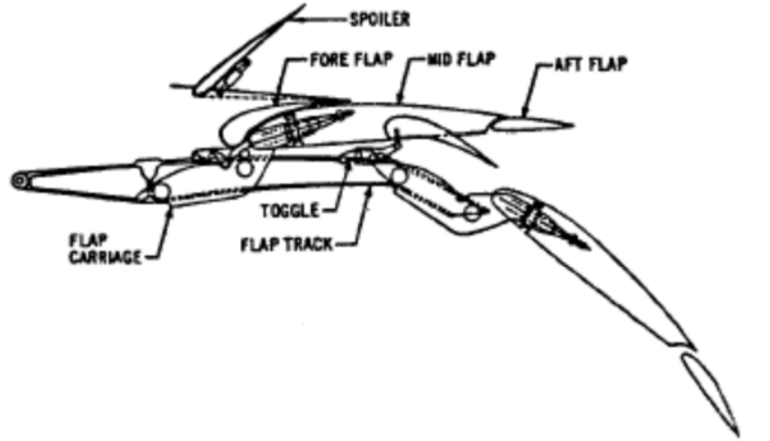


Fig. 7-25h. Triple slotted flap (Boeing 727)

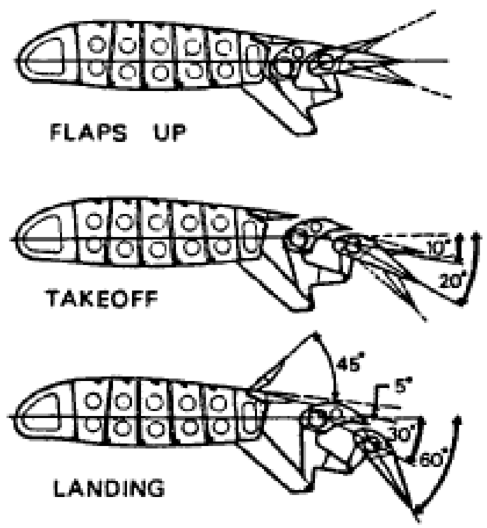


Fig. 7-25g. Double slotted flaps with individual adjustment of flap segments and drooped aileron (GAF N-22 Nomad)

Superficies Hipersustentadoras - III

Efectos HLD

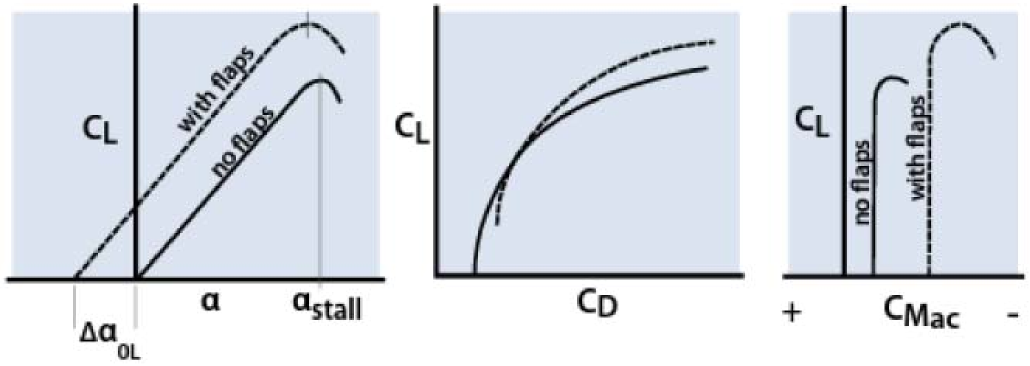
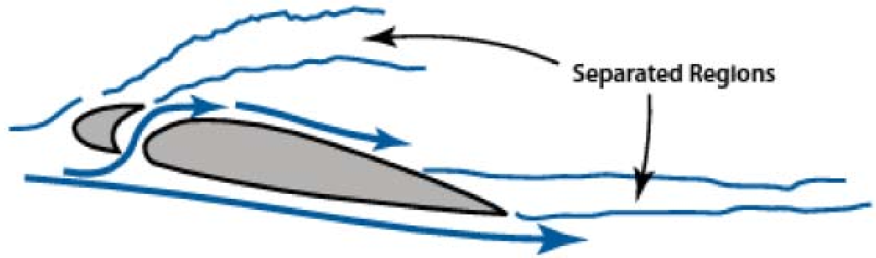


Figure 9.2 Characteristics of TE flaps.



Boeing 737 Triple-Slotted Fowler Flaps

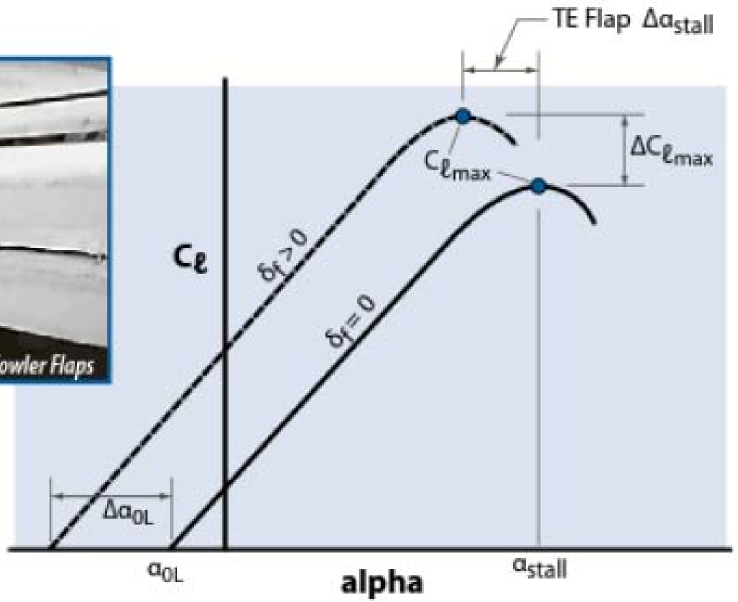


Figure 9.8 Construction of section lift curves for TE flaps.

A300 Leading Edge Slats Extended

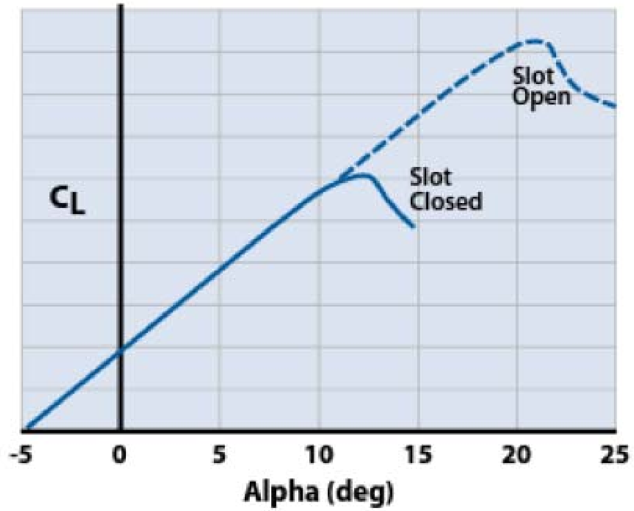


Figure 9.4 Characteristics of slots and slats.

Superficies Hipersustentadoras - IV

Efectos HLD

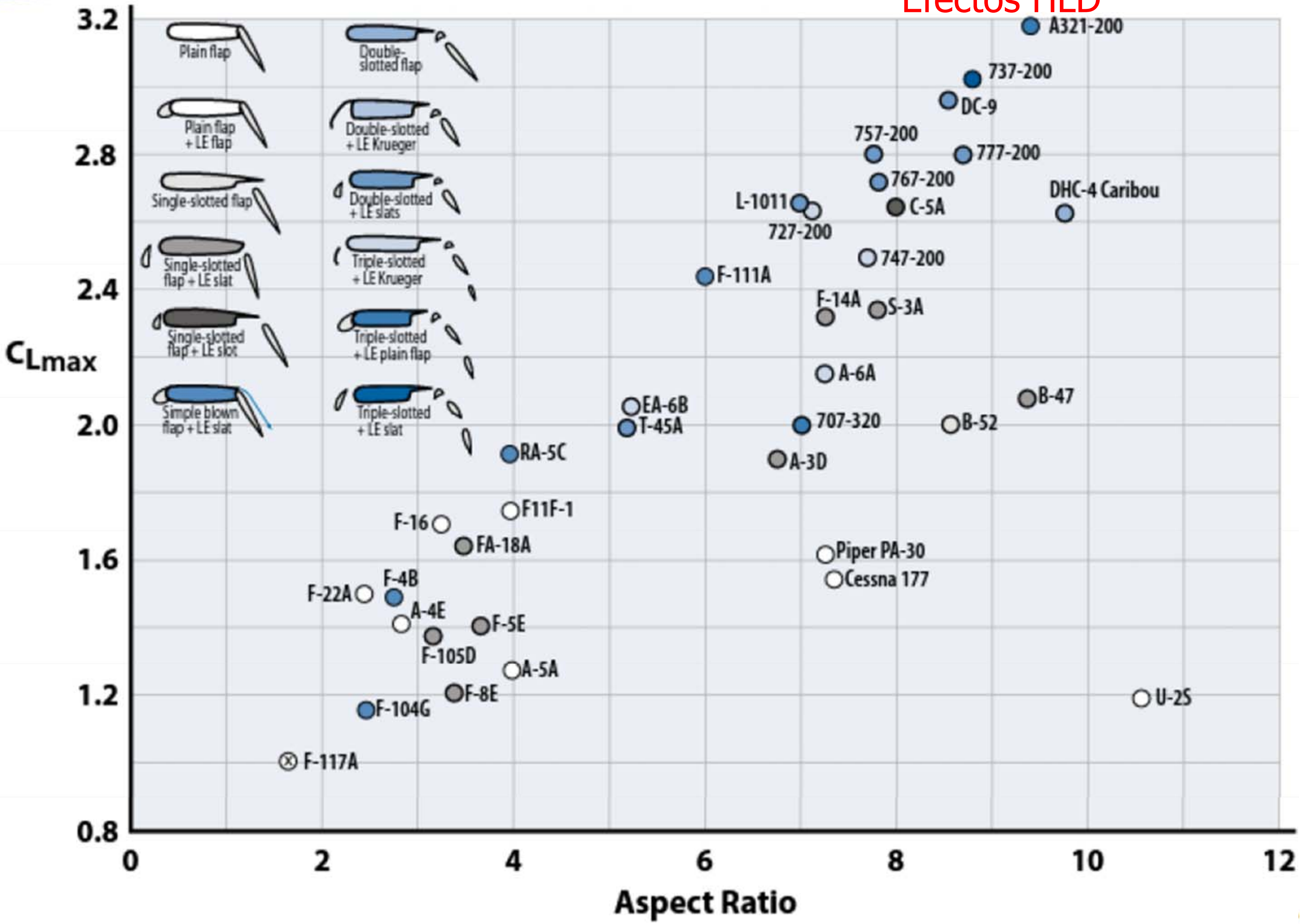


Figure 9.7 Practical low-speed $C_{l_{max}}$ limits for mechanical high-lift systems (data from [7]).

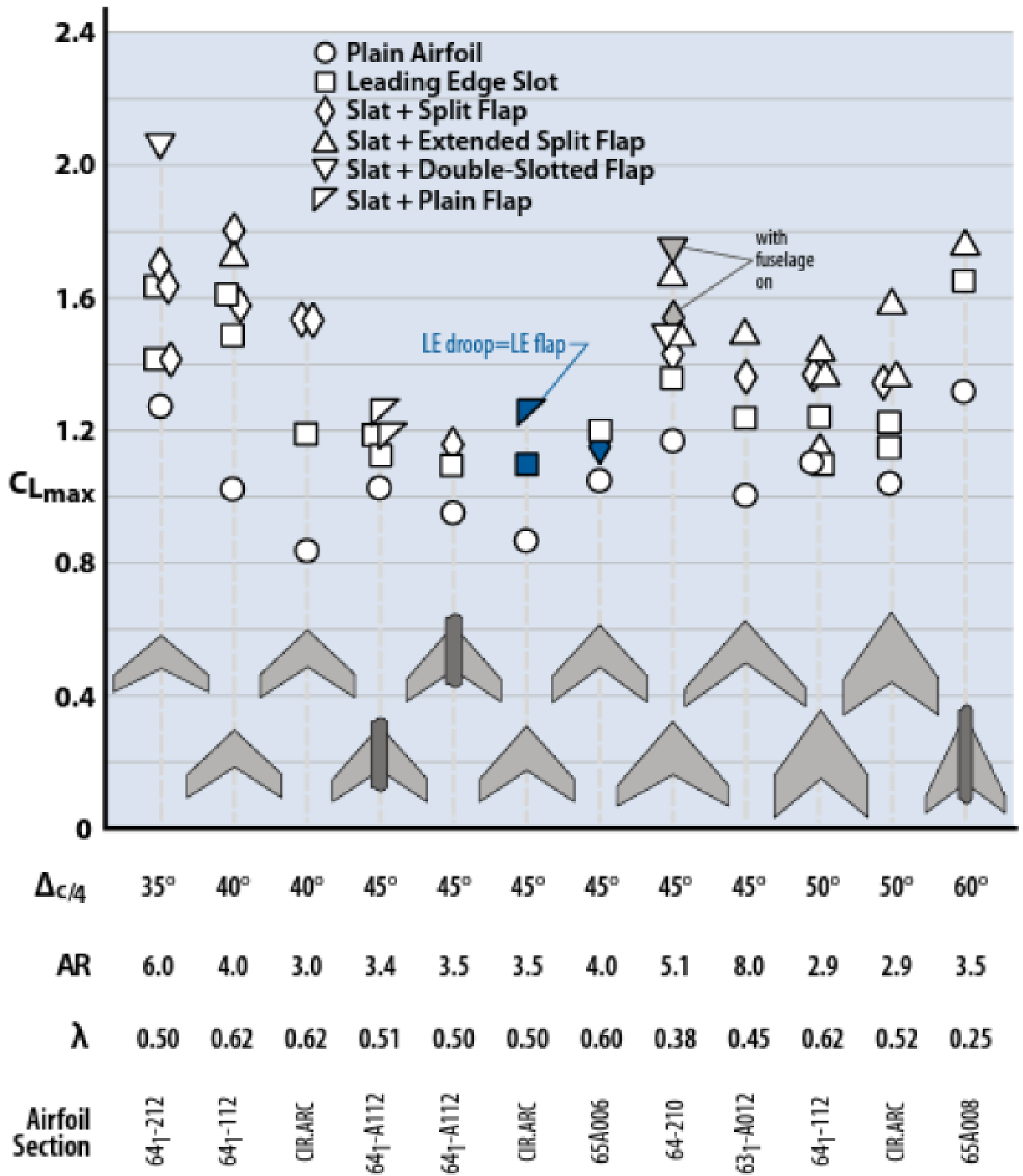
Superficies Hipersustentadoras - V

Efectos HLD

Table 9.3 Typical High-Lift Device Data

$\Delta = 35$ deg, AR = 5.76, $\lambda = 0.54$ Airfoil section: 10% symmetrical		
Arrangement	$C_{L_{max}}$	α_{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
$\Delta = 0$ deg, AR = 4.0, $\lambda = 1.0$ $Re = 10^5$ Airfoil section: NACA 0010		
Arrangement	$C_{L_{max}}$	α_{stall}
Plain wing	0.80	13
30% full-span split flap, $\delta f = 40$ 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])



Efectos HLD

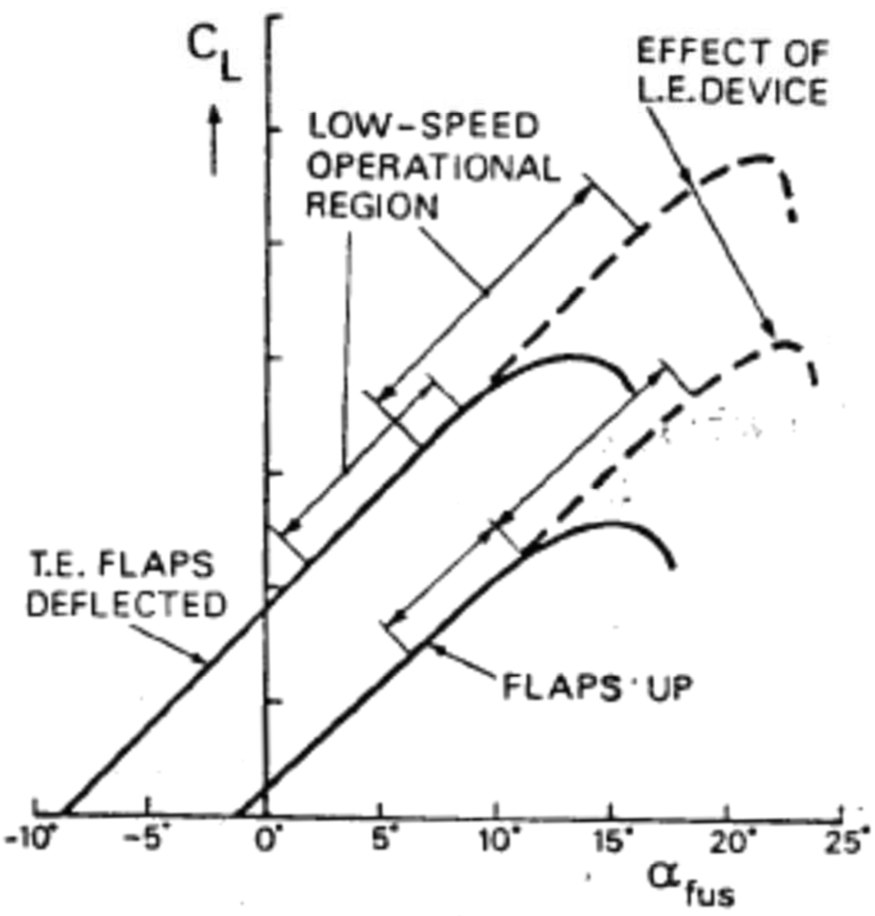


Fig. 7-23. Lift curves with and without high-lift devices.

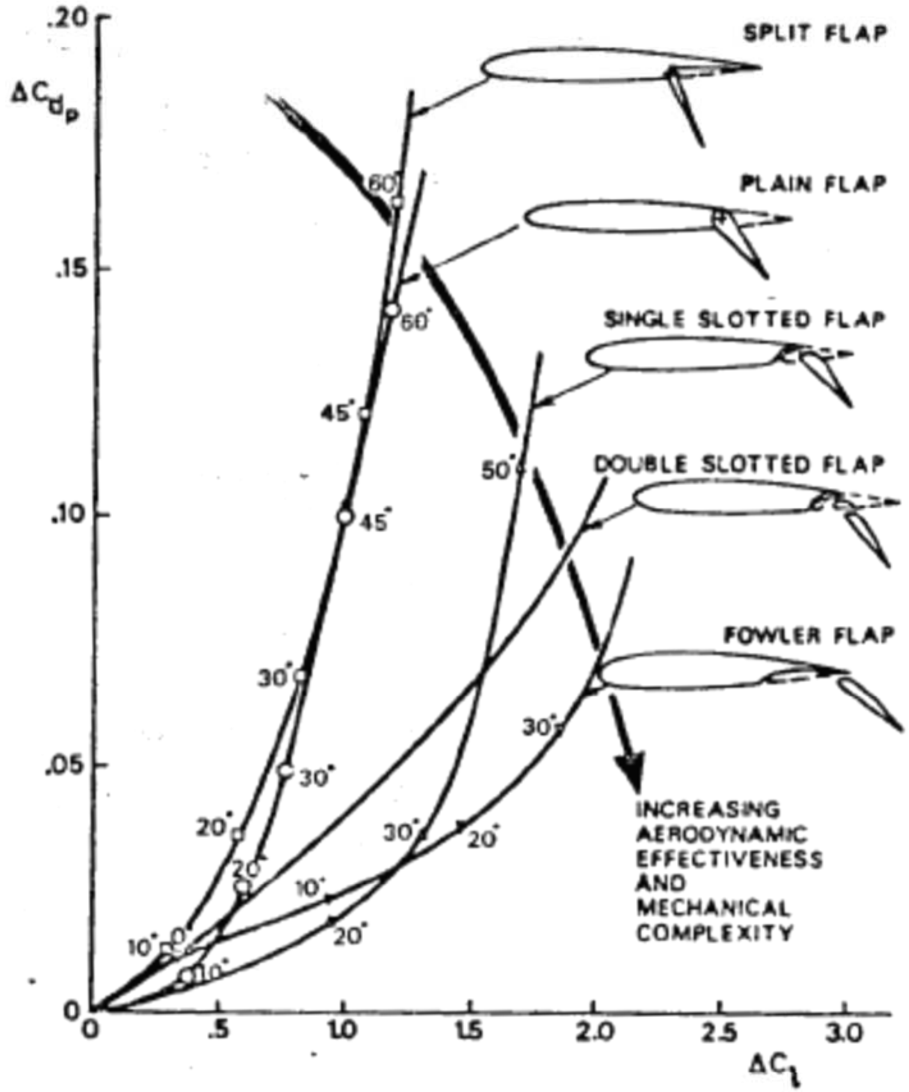
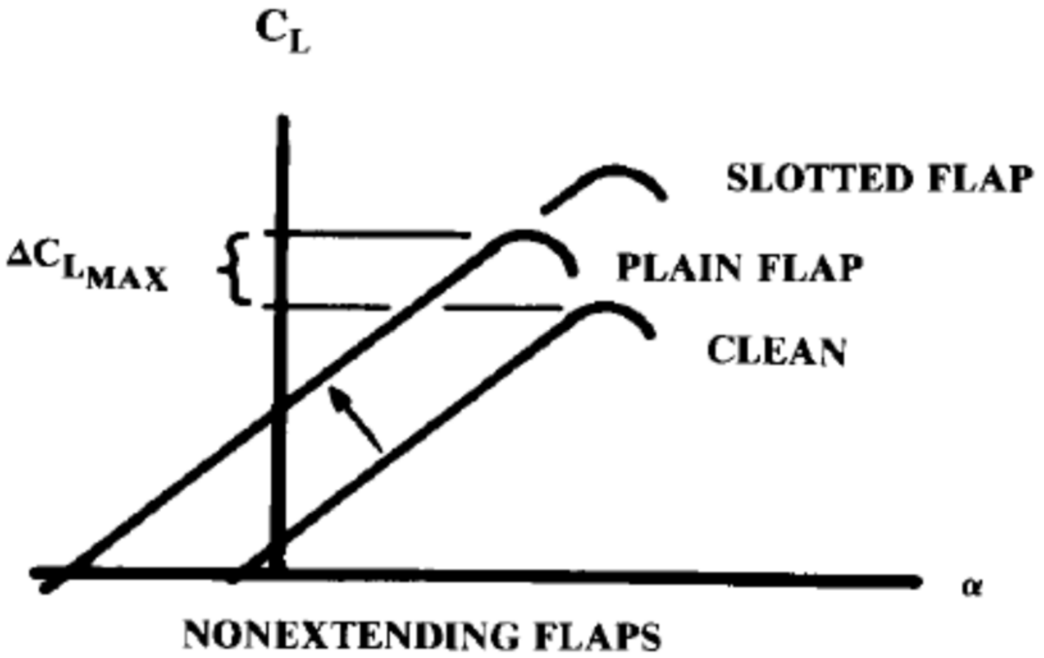
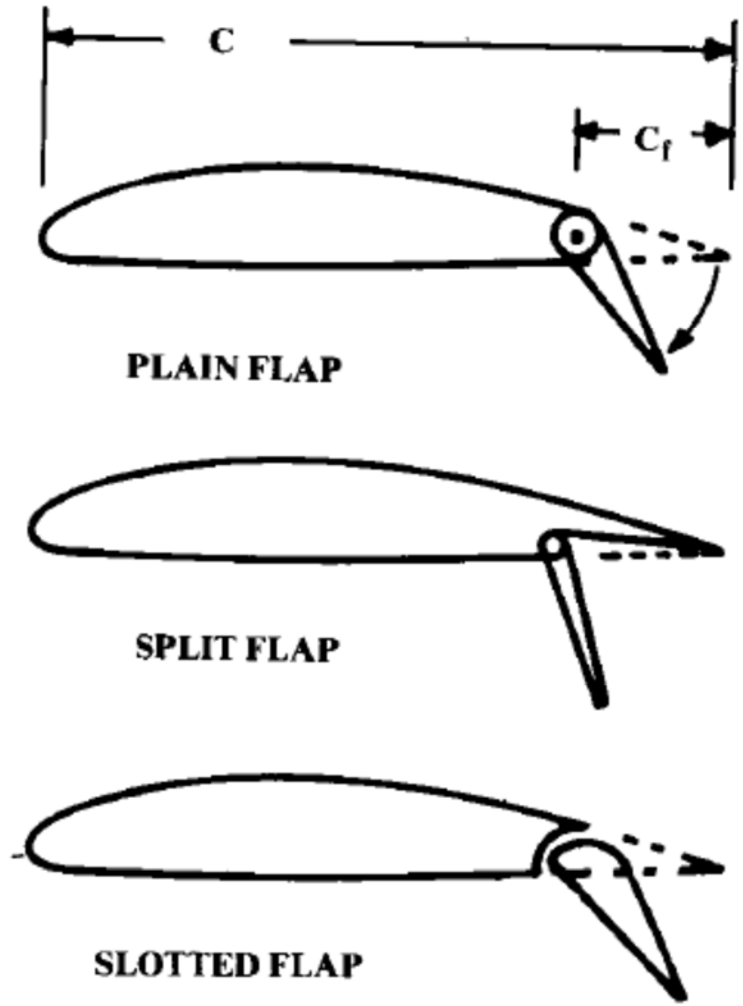


Fig. 7-24. Trends in performance of trailing-edge flaps.

Superficies Hipersustentadoras - VII

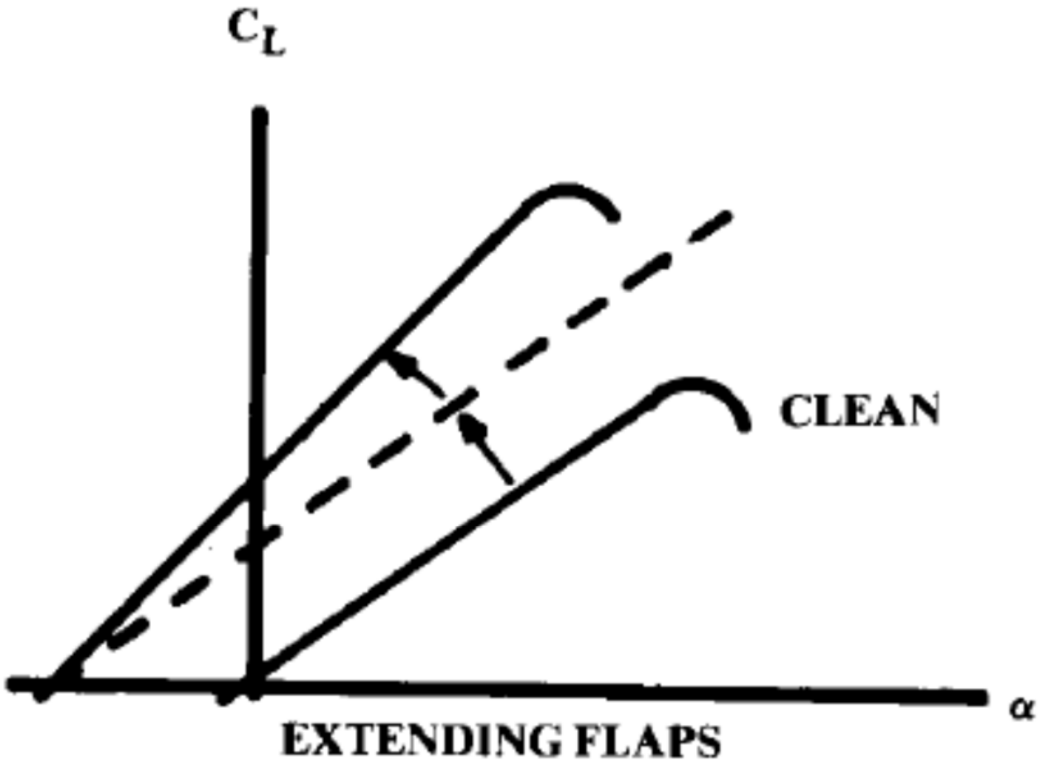
Efectos HLD

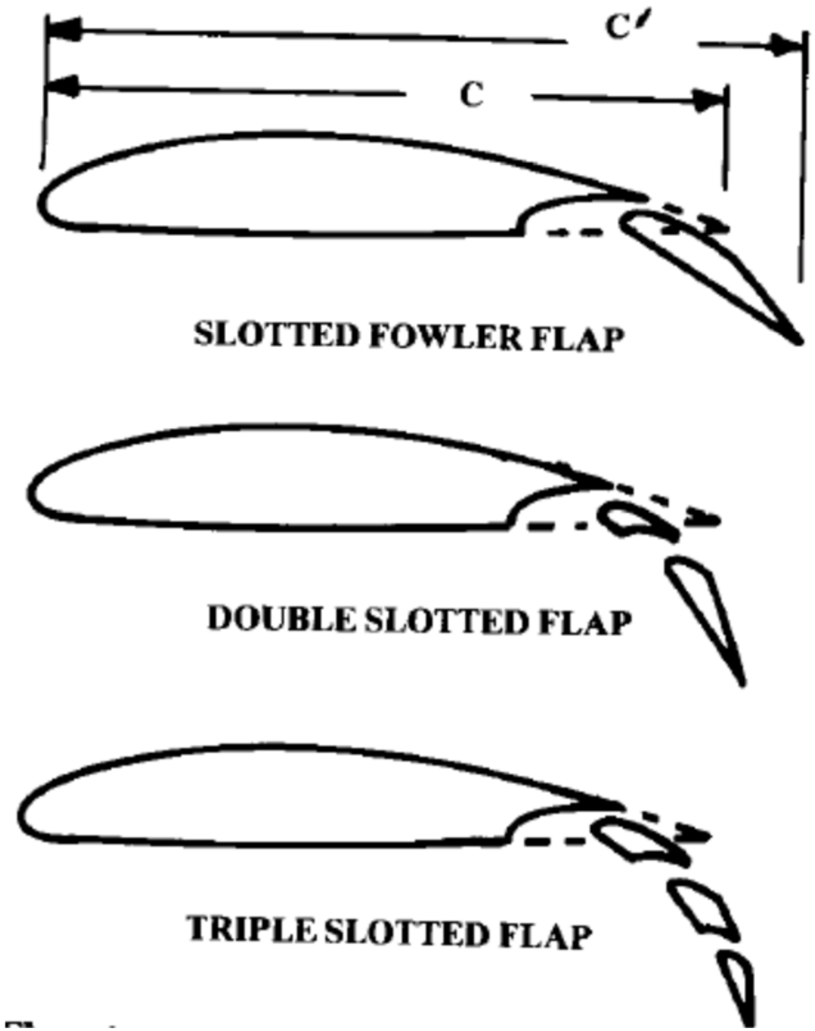




Superficies Hipersustentadoras - VIII

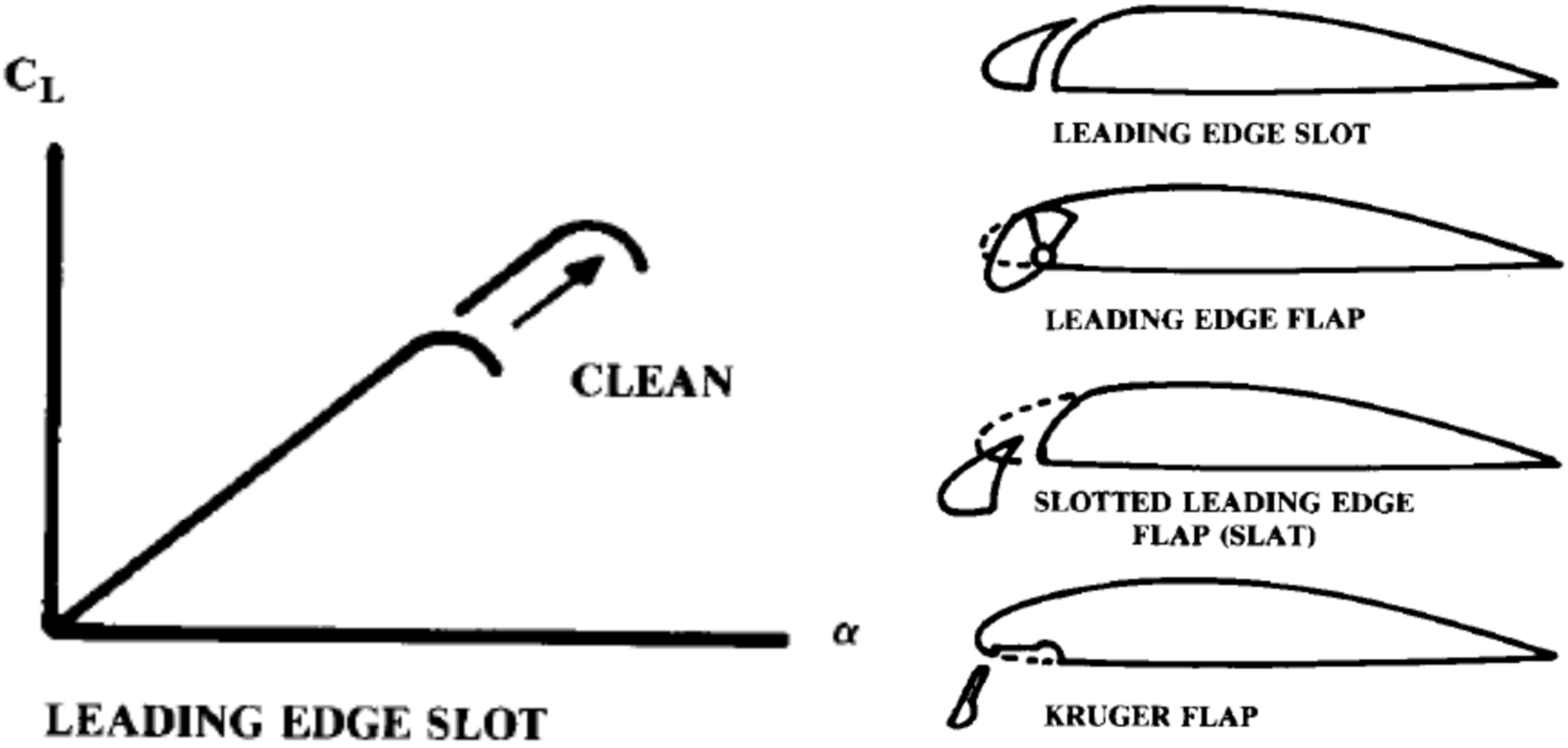
Efectos HLD





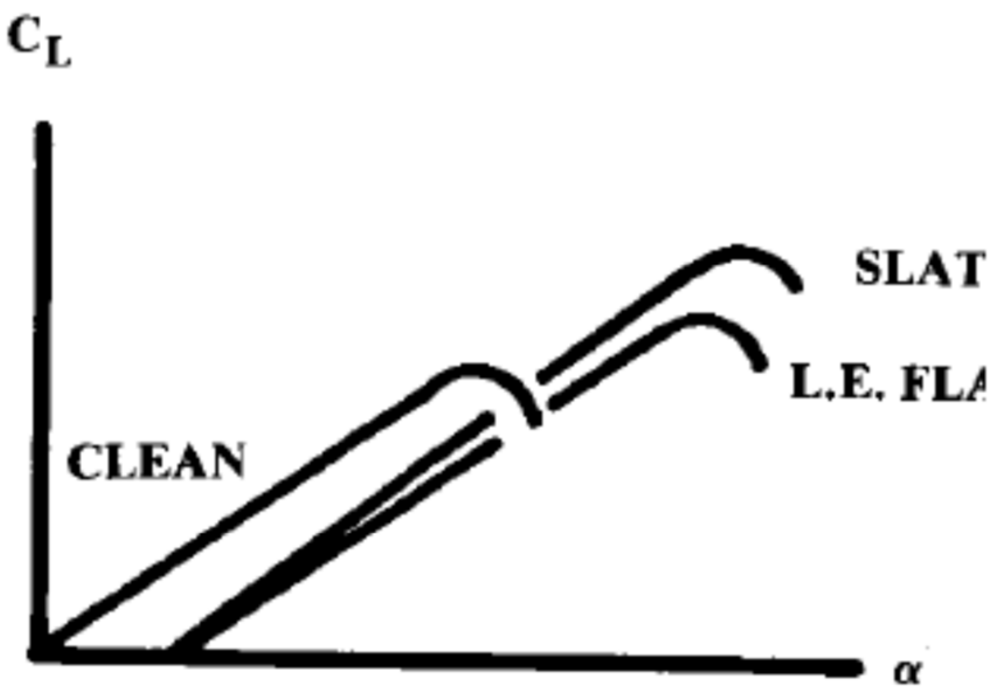
Superficies Hipersustentadoras - IX

Efectos HLD

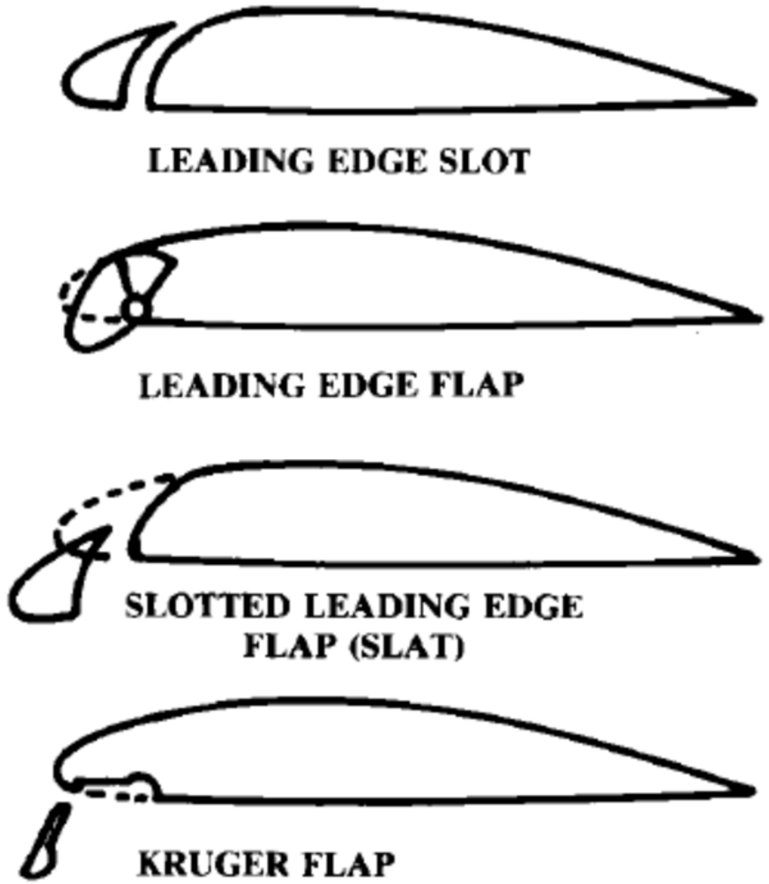


Superficies Hipersustentadoras - X

Efectos HLD

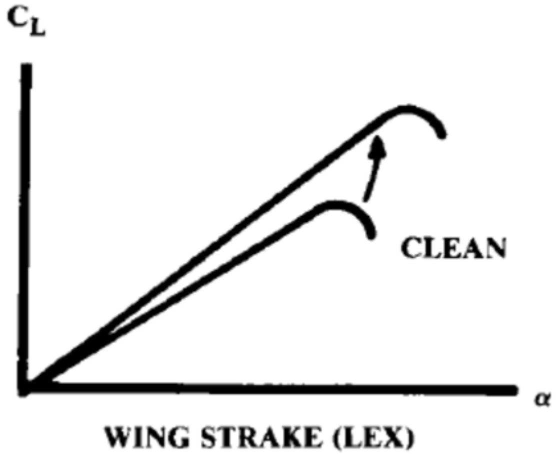


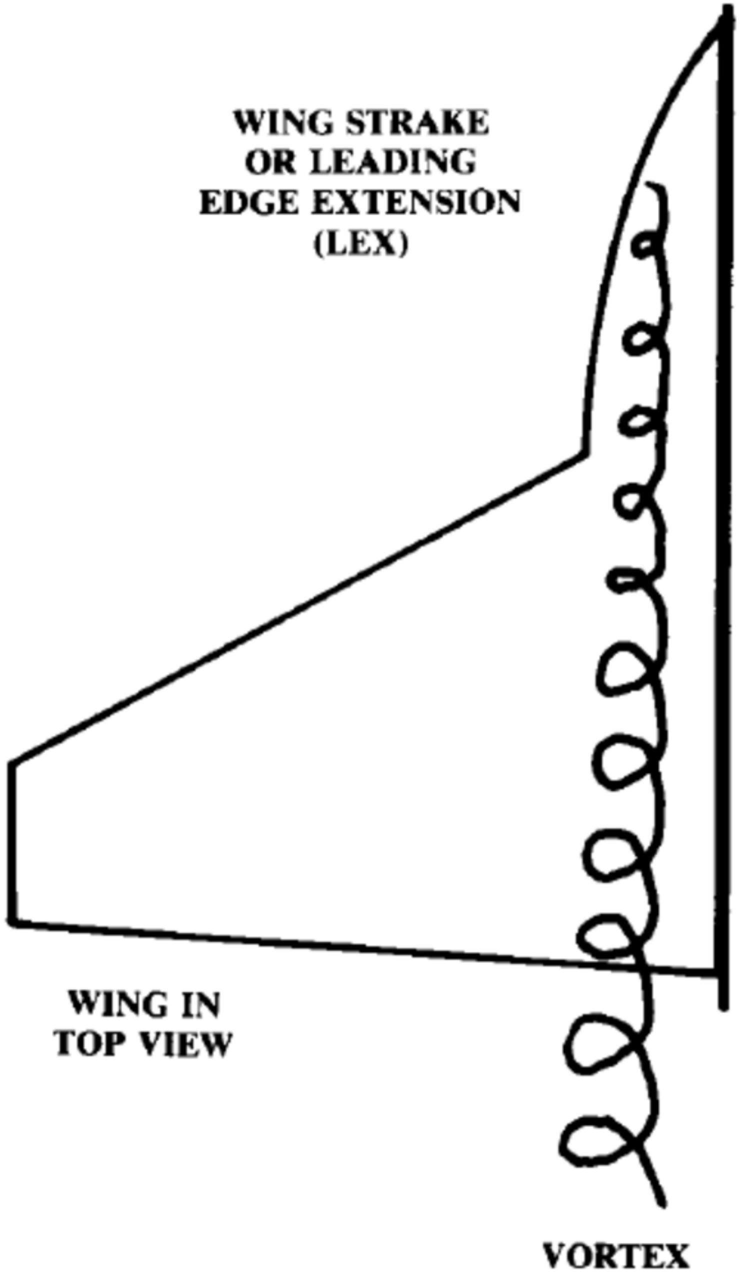
LEADING EDGE FLAP OR SLAT



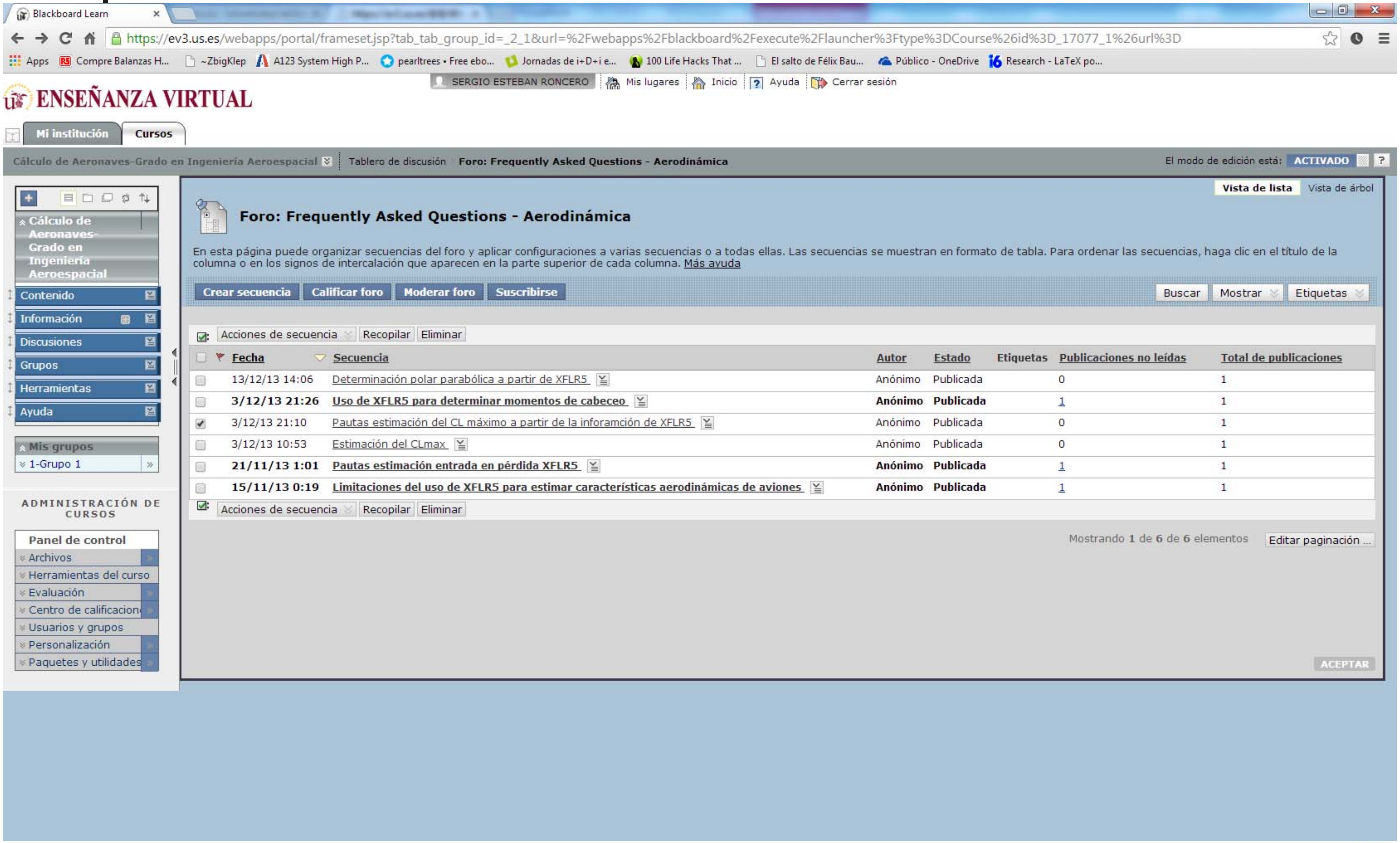
Superficies Hipersustentadoras - XI

Efectos HLD





Estimación C_{Lmax} - XFLR5



Cálculo C_{Lmax} - I

- En una primera iteración se pueden emplear métodos empíricos :
  - Es necesario primero determinar C_L vs. α para el ala básica.
    - Después corregir para diferentes superficies hipersustentadoras.
  - Necesario determinar $\Delta\alpha_{0L}$, $\Delta \mathbf{C}_{\mathbf{lmax}}$ y $\Delta\alpha_{\mathbf{STALL}}$
  - Primer paso es obtener α_{0L}, $\mathbf{C}_{\mathbf{lmax}}$ **y** $\alpha_{\mathbf{STALL}}$ 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

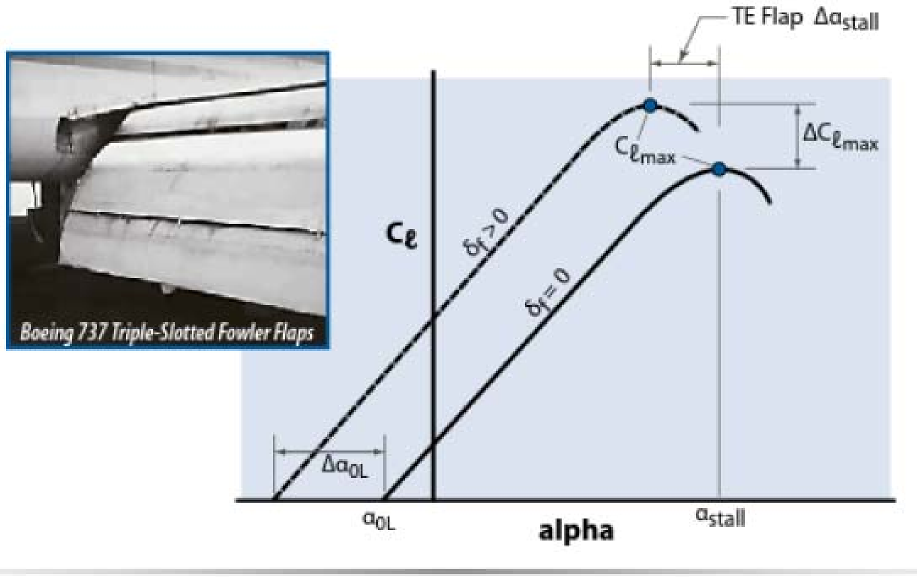


Figure 9.8 Construction of section lift curves for TE flaps.

Cálculo C_{Lmax} - I

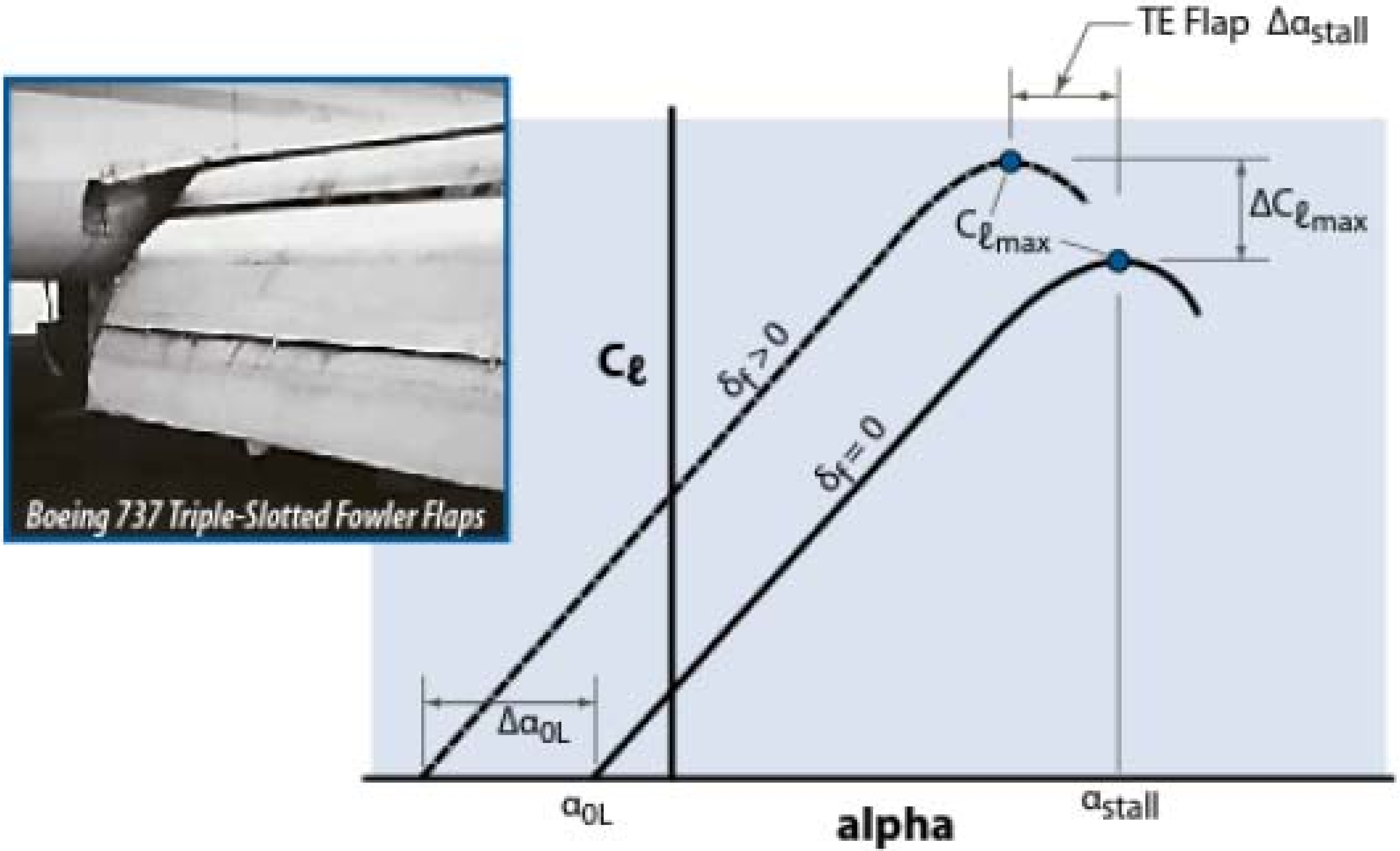


Figure 9.8 Construction of section lift curves for TE flaps.

Cálculo C_{Lmax} - Datos Experimentales - I

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

Airfoil	α_{0L} (deg)	Cm_0	C_{ℓ_α} (per deg)	a.c.	$\alpha_{C_{\ell max}}$ (per deg)	$C_{\ell max}$	α^{*a} (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_1-012	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_2-015	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_3-018	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	α_{0L} (deg)	Cm_0	C_{ℓ_α} (per deg)	a.c.	$\alpha_{C_{\ell max}}$ (per deg)	$C_{\ell max}$	α^{*a} (deg)
63_4-021	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 $_{a=3}$	−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_1-012	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_2-015	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_3-018	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_4-021	+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_1-012	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)

NACA Report 824 - Summary of airfoil data

Cálculo C_{Lmax} - Datos Experimentales - II

Airfoil	α_{0L} (deg)	Cm_0	C_{ℓ_α} (per deg)	a.c.	$\alpha_{c_{\ell max}}$ (per deg)	$C_{\ell_{max}}$	α^{*a} (deg)
-212 $_{a=6}$	−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_2-015	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 $_{a=5}$	−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 $_{a=5}$	−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 $_{a=8}$	−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_3-018	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 $_{a=5}$	−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 $_{a=5}$	−4.2	−0.078	0.104	0.265	20.0	1.51	5.3
65_4-021	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 $_{a=5}$	−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_1-012	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_2-015	0	0.005	0.105	0.265	15.5	1.35	12.0
-215	−1.3	−0.031	0.106	Z60	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 $_{a=6}$	−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	α_{0L} (deg)	Cm_0	C_{ℓ_α} (per deg)	a.c.	$\alpha_{c_{\ell max}}$ (per deg)	$C_{\ell_{max}}$	α^{*a} (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
64_1A212	−2.0	−0.040	0.100	0.252	14.0	1.54	11.0
64_2A215	−2.0	−0.040	0.095	0.252	15.0	1.50	12.0

${}^a\alpha^*$ = angle of attack at which lift curve ceases to be linear (incipient stall).
Note: C_{M0} is about the aerodynamic center (a.c.).

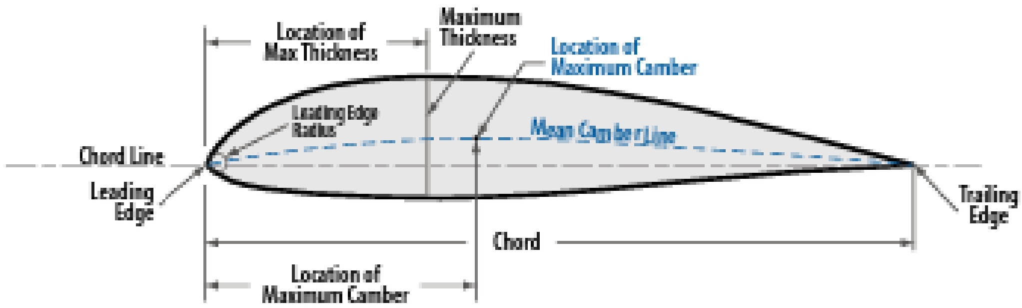


Figure F.1 Airfoil terminology.

NACA Report 824 - Summary of airfoil data

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	Biconvex
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 III/IV	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 64A011
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	150/152/172/180/182206	NACA 2412
Cessna	208 Caravan	NACA 23017
Cessna	310/L-27/U-3	NACA 2412
Cessna	T-37	NACA 23018
Cessna	Citation I/II/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 64-210.68
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 64A010

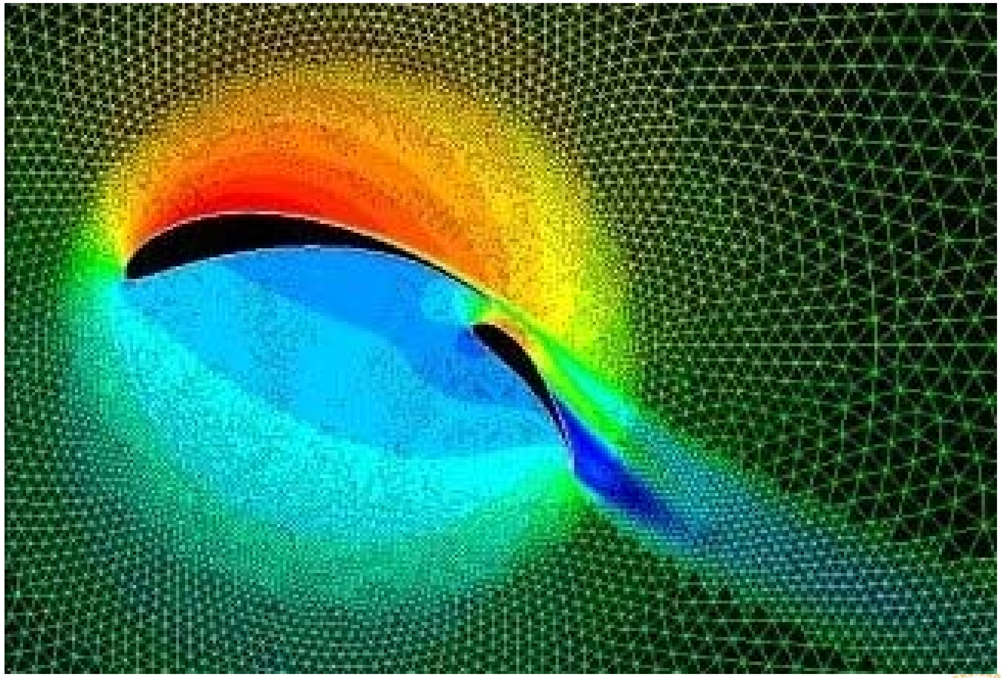
Company	Aircraft	Airfoil (Root)
Gulfstream	GII/GIII/GIV	NACA 0012 mod
Hawker	Hurricane	Clark YH (19%)
Hawker	Typhoon	NACA 2219
Hughes	H-1 (Long Wing)	NACA 23016.5
Hughes	H-1 (short wing)	NACA 2418
Hughes	H-4 Hercules	NACA 63(420)-321
Learjet	23–60	NACA 64A109
Lesher	(Univ of Mich)Teal	NACA 63A615
Lockheed	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	Polecat	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	VariEze	NACA 4414
Rutan	Pond Racer	Roncz
Rutan	Raptor	Roncz RQW17B
Rutan	Global Flyer	Roncz
Rutan	Space Ship One	Rutan
Rutan	White Knight	Hatfield
Ryan	AQM-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.

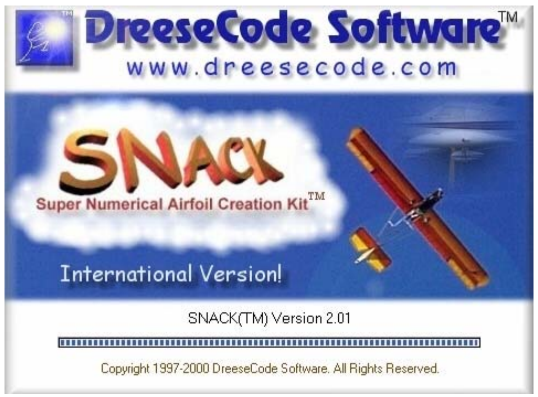
- The Incomplete Guide to Airfoil Usage
  - http://www.ae.uiuc.edu/m-selig/ads/aircraft.html
  - David Lednicer
    Analytical Methods, Inc.
    2133 152nd Ave NE
    Redmond, WA 98052
    dave@amiwest.com



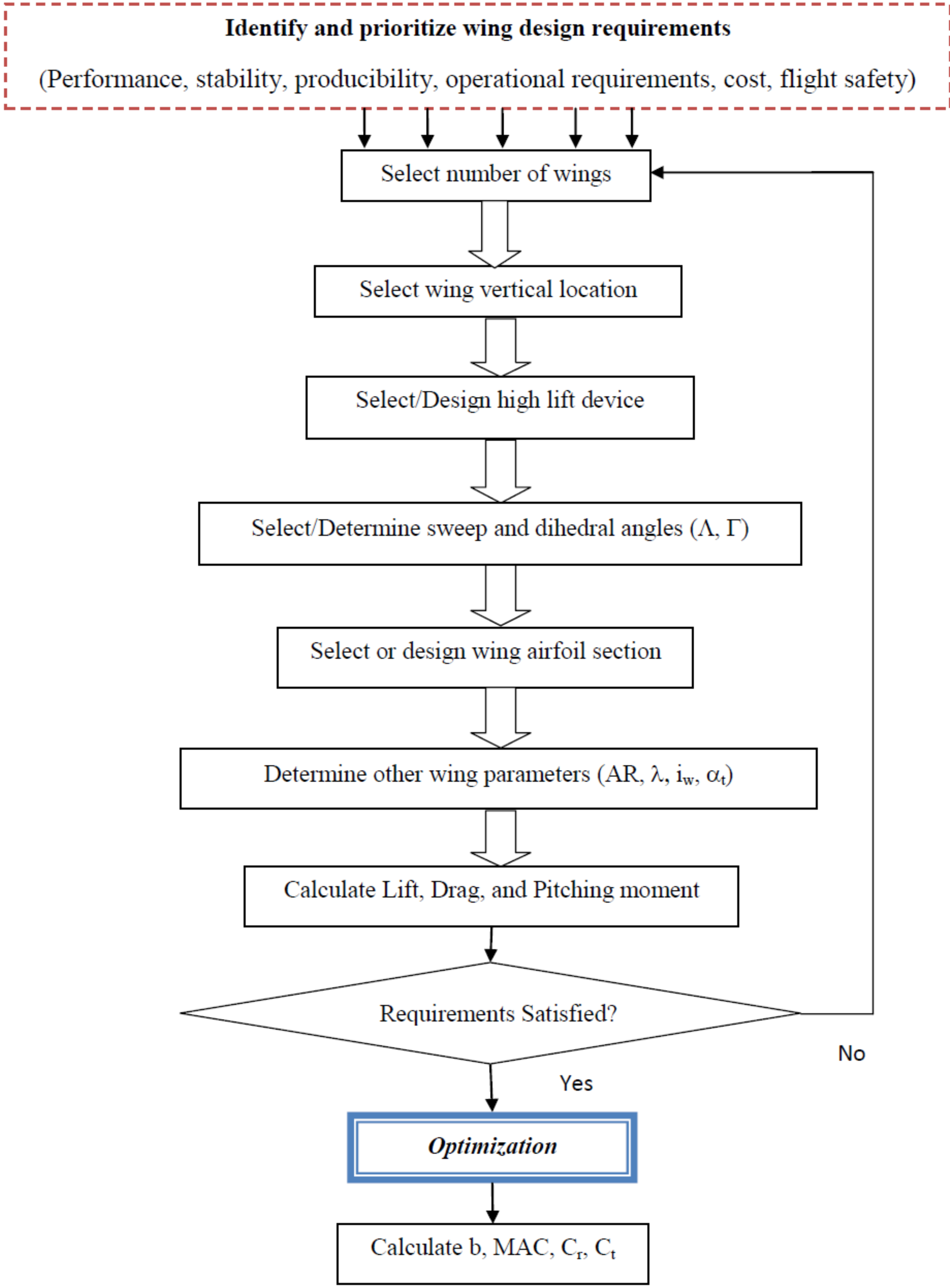
NACA Report 824 - Summary of airfoil data

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 Cp 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 & Pohlhauson**.
    - 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)
Select number of wings
Select wing vertical location
Select/Design high lift device
Select/Determine sweep and dihedral angles (Λ, Γ)
Select or design wing airfoil section
Determine other wing parameters (AR, λ, i_w, α_t)
Calculate Lift, Drag, and Pitching moment
Requirements Satisfied?
No
Yes
Optimization
Calculate b, MAC, C_r, C_t

Coeficiente de sustentación vs. Ángulo de ataque

The maximum lift coefficient ($C_{l\max}$)

The stall angle (α_s)

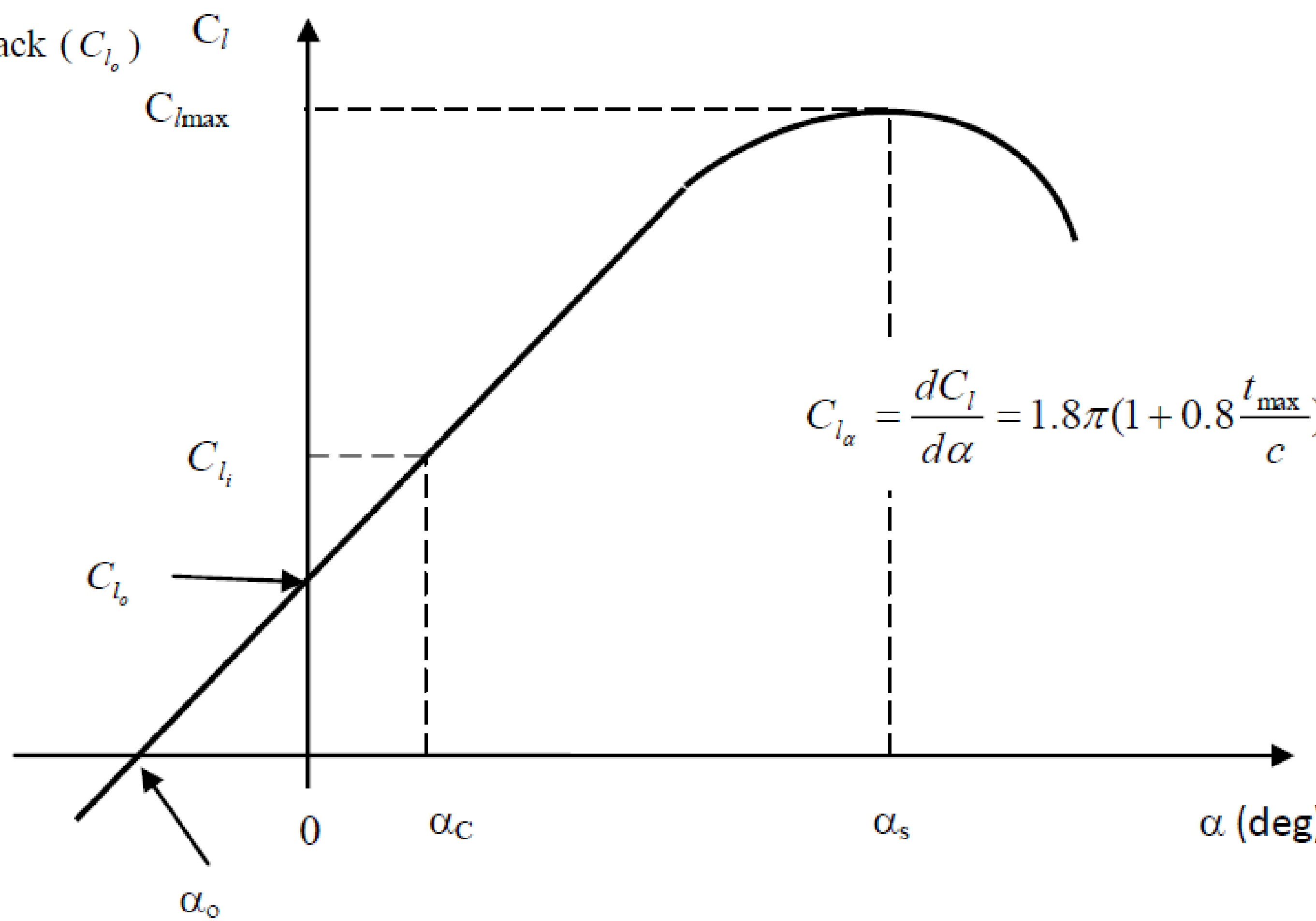
The ideal lift coefficient (C_{l_i})

The angle of attack corresponding to ideal lift coefficient (α_{Cli})

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

The zero lift angle of attack (α_o)

The lift curve slope (C_{l_α})



$$C_{l_\alpha} = \frac{dC_l}{d\alpha} = 1.8\pi\left(1 + 0.8\frac{t_{\max}}{c}\right)$$

Variaciones de Pitch Moment

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

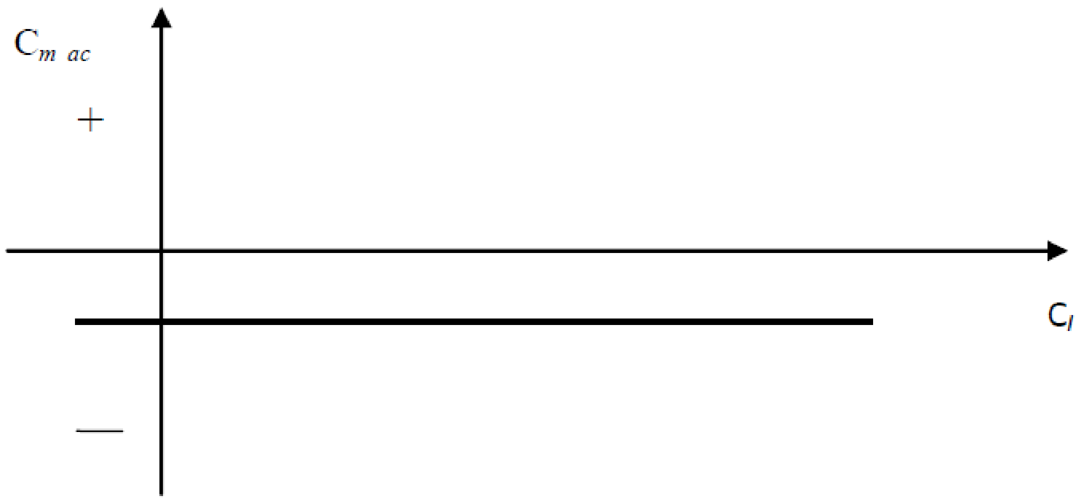
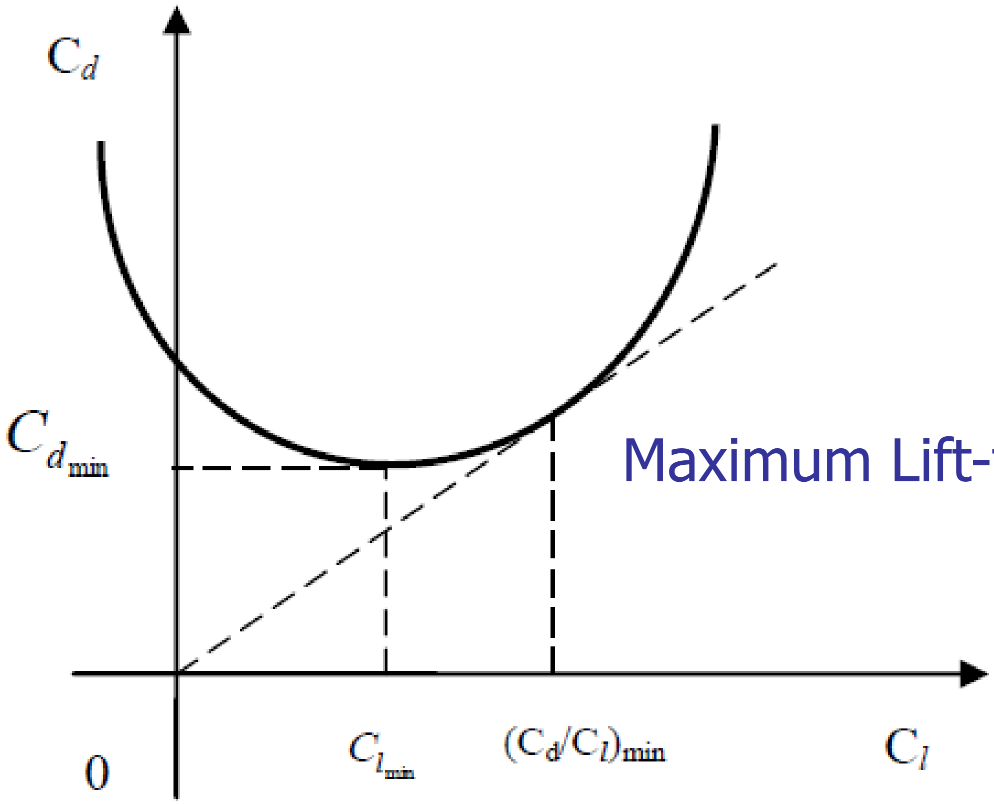


Figure 5.14. The variations of pitching moment coefficient versus lift coefficient

Variaciones de Resistencia

Minimum drag coefficient



Maximum Lift-to-Drag

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 gentile, not sharp).
8. The airfoil must be structurally reinforceable. The airfoil should not that much 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 places 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:

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

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, ρ is the air density at cruising altitude, and S is the wing planform area.

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

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

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

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.

$$C_{L_{C_w}} = \frac{C_{L_C}}{0.95}$$ 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

$$C_{l_i} = \frac{C_{L_{C_w}}}{0.9}$$ 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, ρ_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

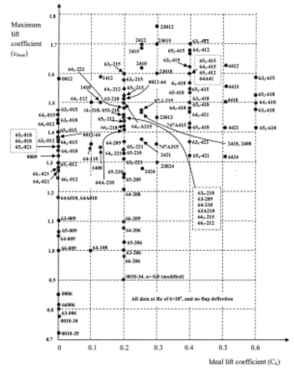
9. Determine the high lift device (HLD) contribution to the wing maximum lift coefficient $\Delta C_{l_{HLD}}$

10. Calculate the wing airfoil "*net*" maximum lift coefficient ($C_{l_{\max}}$)

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

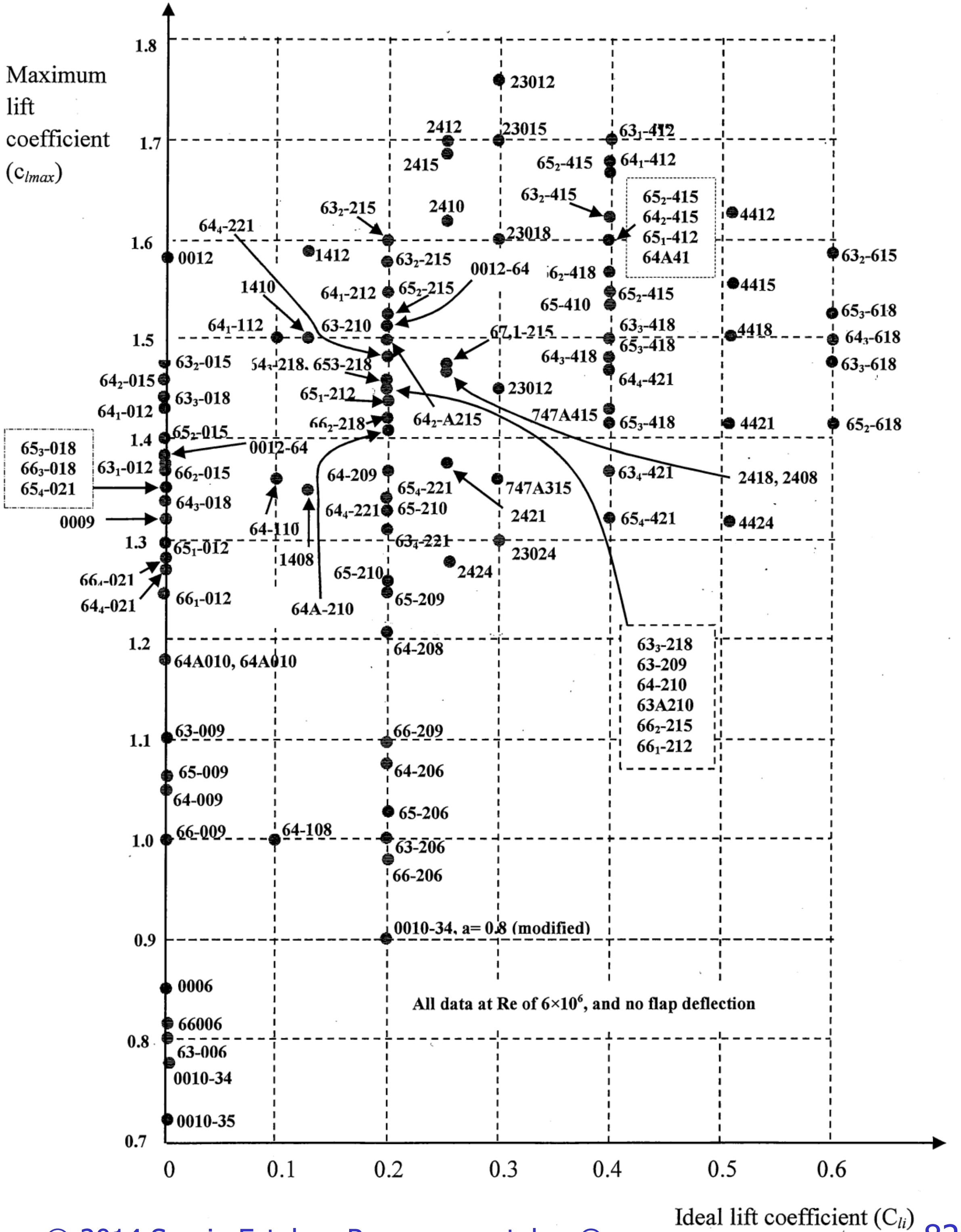
11. Identify airfoil section alternatives that deliver the desired C_{li} (step 4) and C_{lmax} (step 10).

- 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

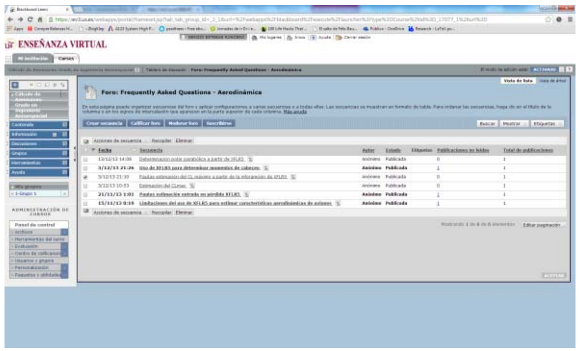


3. Abbott I. H. and Von Donehoff A. F., **Theory of Wing Sections**, Dover, 1959
4. Anderson J. D., **Fundamentals of Aerodynamics**, McGraw-Hill, Fifth edition, 2010

C_{li} and C_{lmax}



Enseñanza Virtual
FAQ Aerodynamics



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 (Mcr) and drag-divergent Mach number (Mdd).
- This allow the aircraft fly closer to Mach one before the drag rise is encountered.
- Thinner airfoil will have a higher Mcr than a thicker airfoil

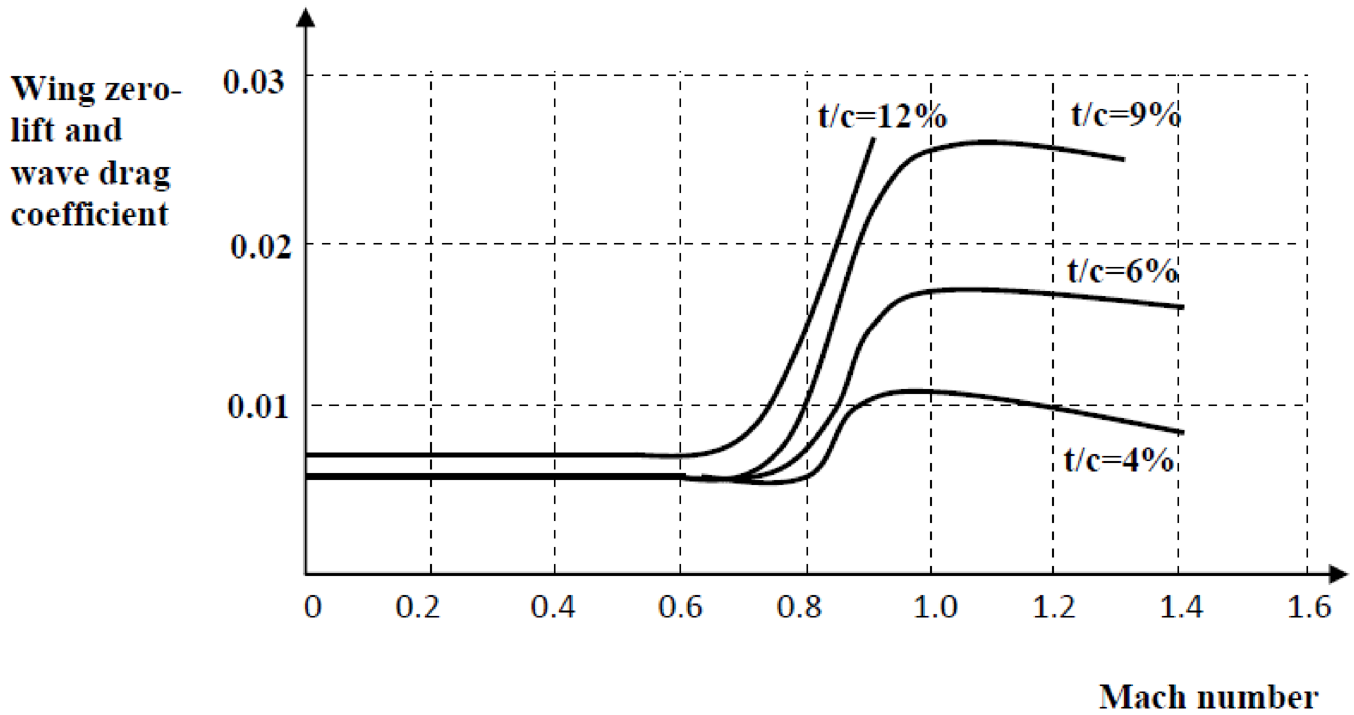


Figure 5.24. Variation of wing zero-lift and wave drag coefficient versus Mach number for various airfoil thickness ratio.

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%					
α_s	15					
α_o	10					
$(C_l/C_d)_{max}$	10%					
$C_{l\alpha}$	5%					
Stall quality	20%					
Summation	100%	64	76	93	68	68

NACA	C_dmin	Cmo	alpha stall	alpha ZL	Cl/Cd max	C_L max	CL0	Stall qual	Claplha
4415	0,00559	-0,1049	19,4	-4,1	173	1,9806	0,4882	moderate-	0,1175
4418	0,00585	-0,1036	19,2	-4,1	171,3488	1,9308	0,4892	moderate-	0,1185
4412	0,00524	-0,1055	19,3	-4,1	167,6032	2,022	0,4846	moderate	0,1108
4421	0,00616	-0,1017	19,2	-4,1	163,664	1,895	0,4857	docile-mod	0,1118
63_3 215	0,0042	-0,1228	21	-4,56	167,93	1,9981	0,5446	moderate	0,1103
63_3 618	0,00438	-0,121	20,6	-4,5	179,0716	1,9032	0,5489	docile	0,1124
64_3 618	0,00431	-0,1218	22,4	-4,5	187,4848	1,9062	0,5468	docile +	0,1099
65_3 618	0,00386	-0,1245	23,6	-4,7	195,4118	1,8796	0,5489	docile ++	0,1135
12	0,00509	0	20,4	0	126,6477	1,8816	0	Sharp	0,1132

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:

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

$$C_{L_{C_w}} = \frac{C_{L_C}}{0.95} = \frac{0.176}{0.95} = 0.185$$

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

Maximum lift coefficient:

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

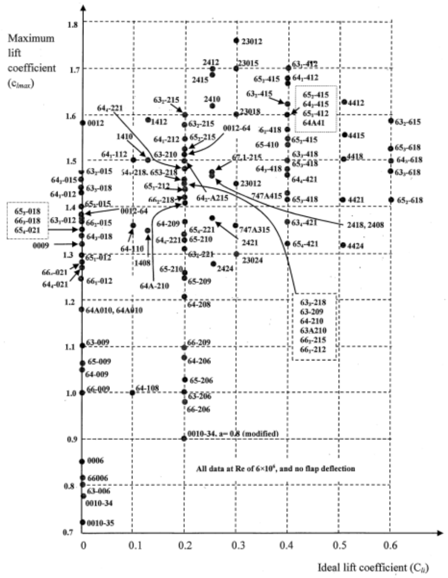
$$C_{L_{\max_w}} = \frac{C_{L_{\max}}}{0.95} = \frac{1.909}{0.95} = 2.01$$

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

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

Pasos para la selección de perfil - VII

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



63_3-218, 64-210, 66_1-212, 66_2-215, 65_3-218

No	NACA	C_{dmin}	C_{mo}	α_s (deg) Flap up	α_o (deg) $\delta_f = 60^o$	$(C_l/C_d)_{max}$	Stall quality
1	63_3-218	0.005	-0.028	12	-12	100	Docile
2	64-210	0.004	-0.040	12	-13	75	Moderate
3	66_1-212	0.0032	-0.030	12	-13	86	Sharp
4	66_2-215	0.0035	-0.028	14	-13.5	86	Sharp
5	65_3-218	0.0045	-0.028	16	-13	111	Moderate

Pasos para la selección de perfil - VIII

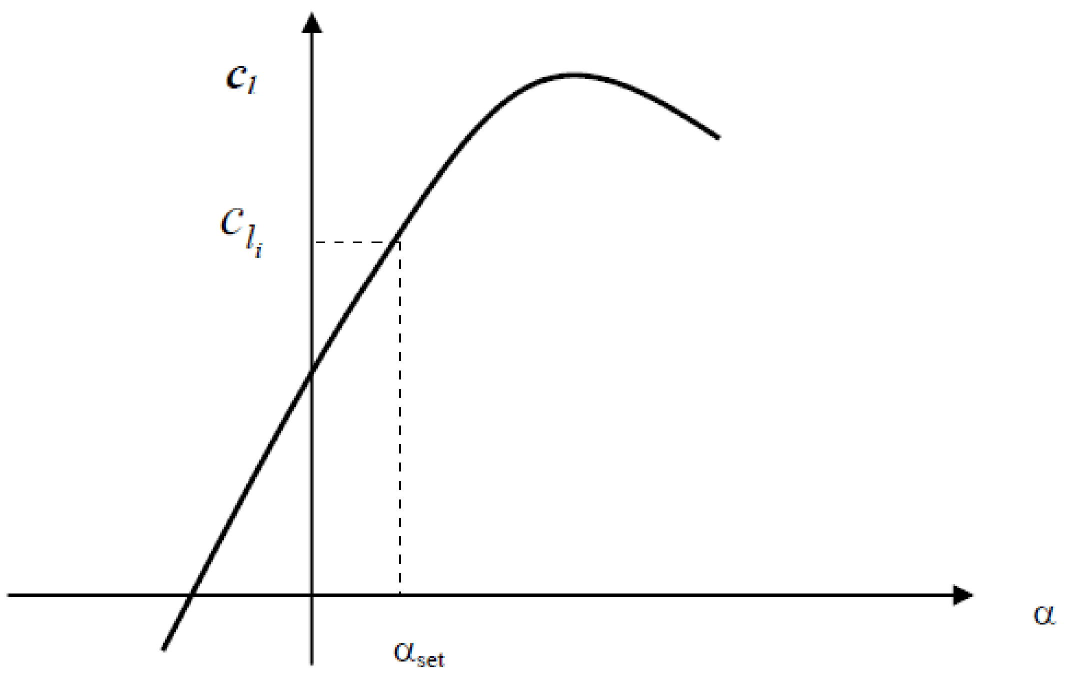
1- The NACA airfoil section 66_1-212 yields the highest maximum speed, since it has the lowest C_{dmin} (0.0032).
2- The NACA airfoil section 65_3-218 yields the lowest stall speed, since it has the highest stall angle (16 degrees).
3- The NACA airfoil section 65_3-218 yields the highest endurance, since it has the highest $(C_l/C_d)_{max}$ (111).
4- The NACA 63_3-218 yields the safest flight, due to its docile stall quality.
5- The NACA airfoil sections 63_3-218, 66_2-215, and 65_3-218 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 Cdmin 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

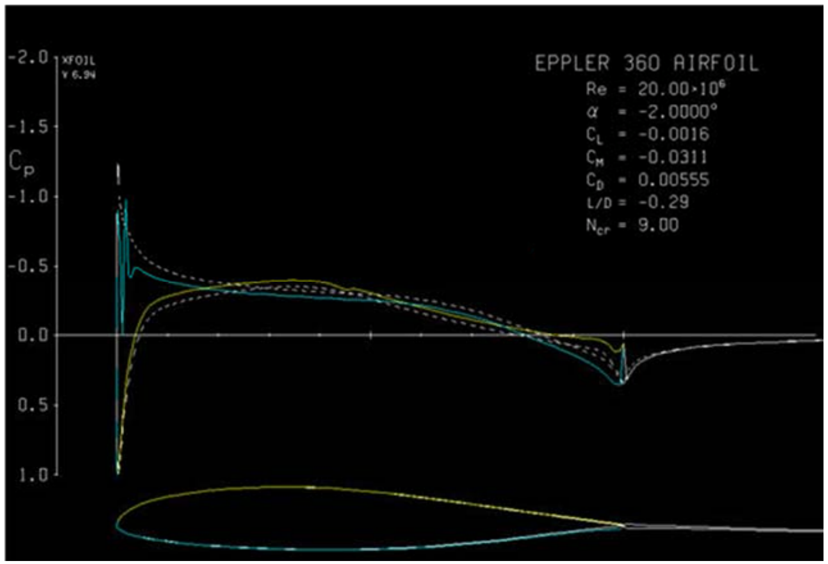
Wing Incidence

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).

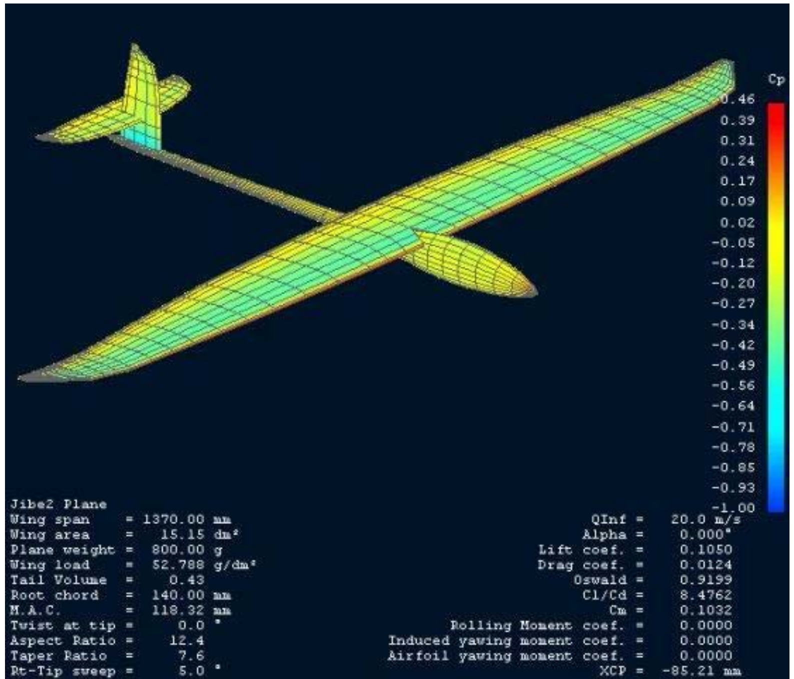


Herramientas – XFOIL & XFLR5

- 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



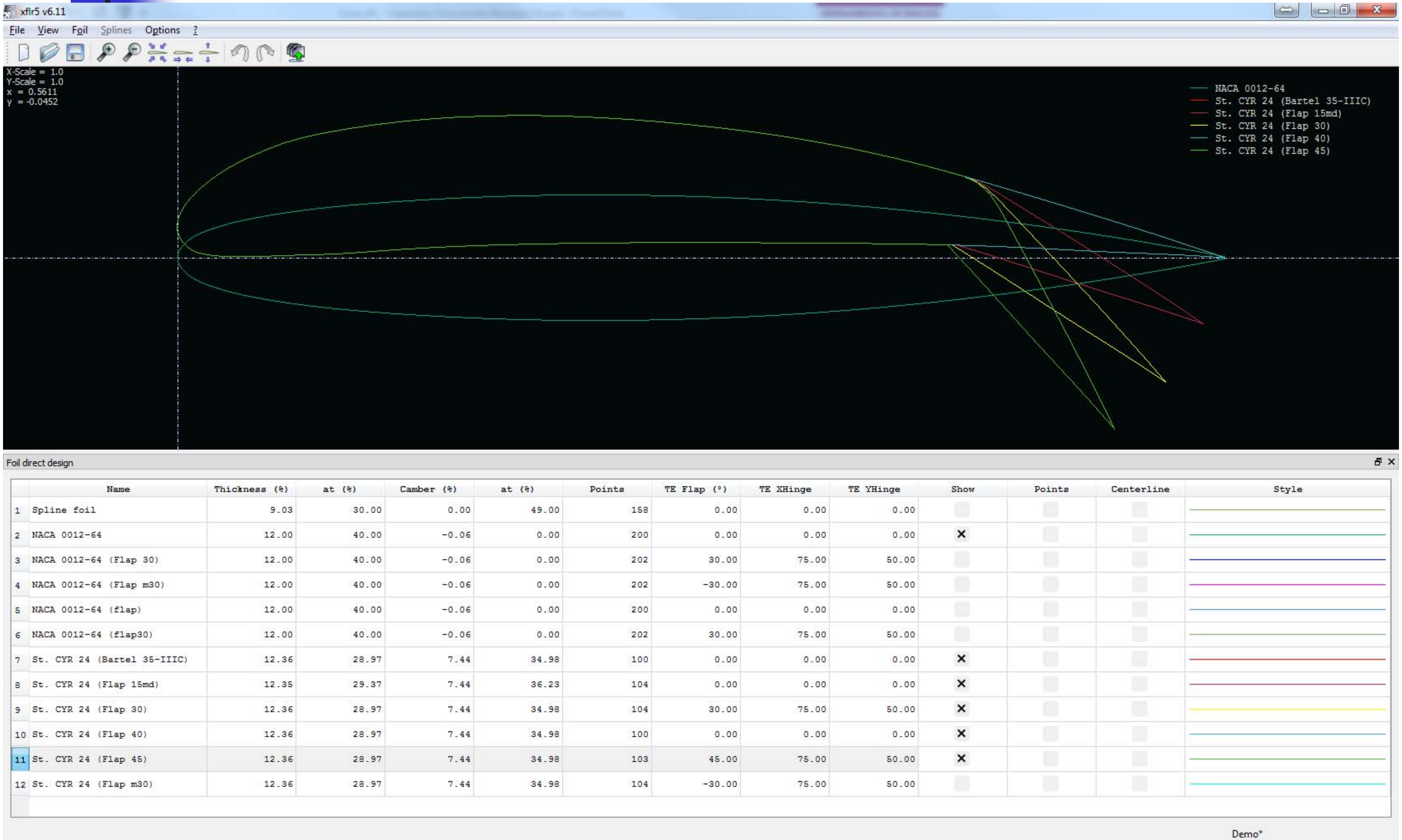
DEMO



Herramientas – XFOIL & XFLR5 - 2

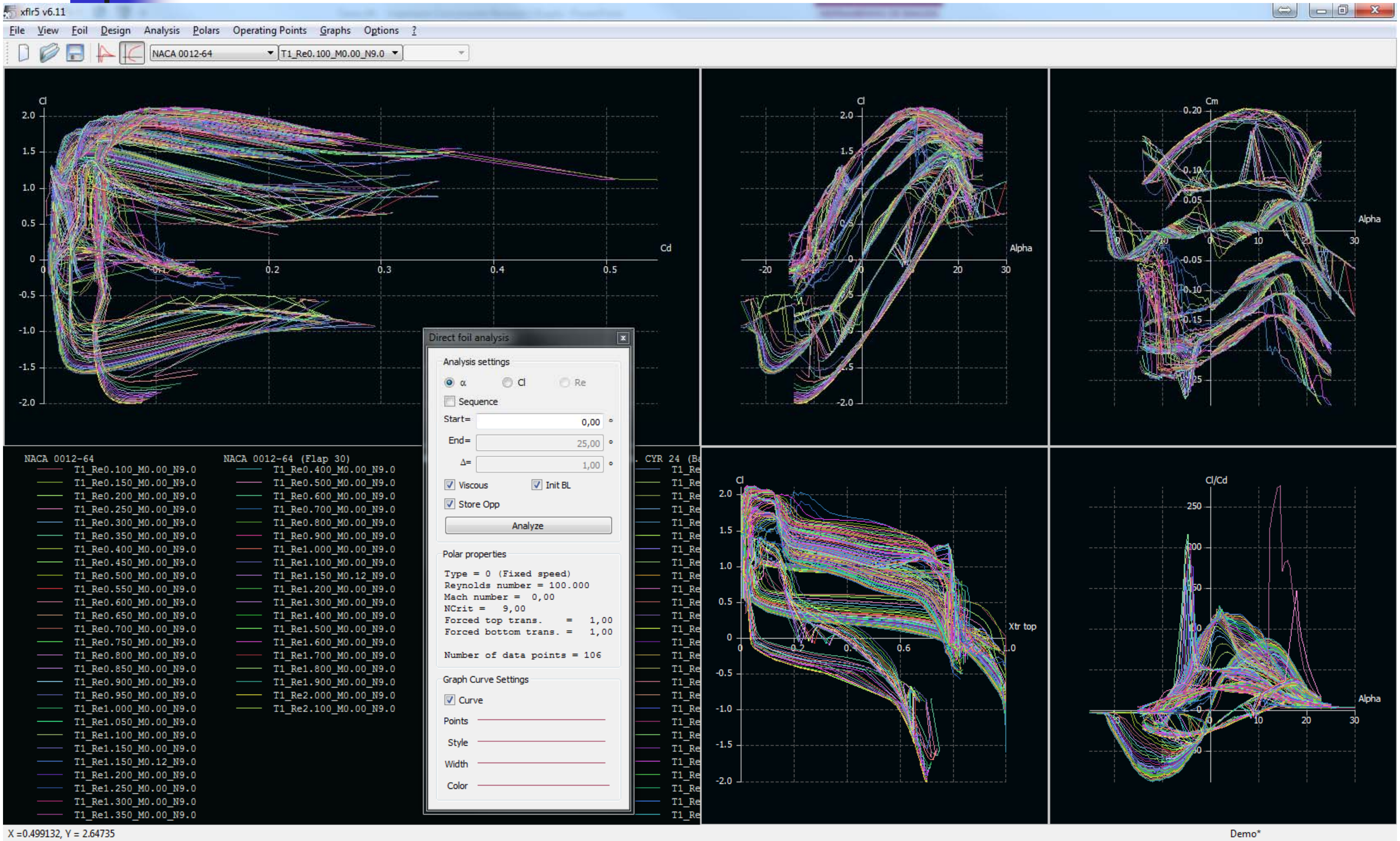
- Empleo de herramientas
  - Estudios 2D:
    - Selección del Perfil (Direct Foil Design)
    - Refinado del perfil
    - Modificación del perfil
    - Estudio 2D (XFOIL Direct Analysis)
  - Estudio 3D
    - Definición geometría y perfiles: ala, HTP y VTP
    - Definición de condiciones de vuelo
      - LLT: Lifting Line Theory
      - VLM1: viscoso y no viscoso
    - 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 Clmax y Polar
    - VLM1: determinación de polar (no resultados buenos)

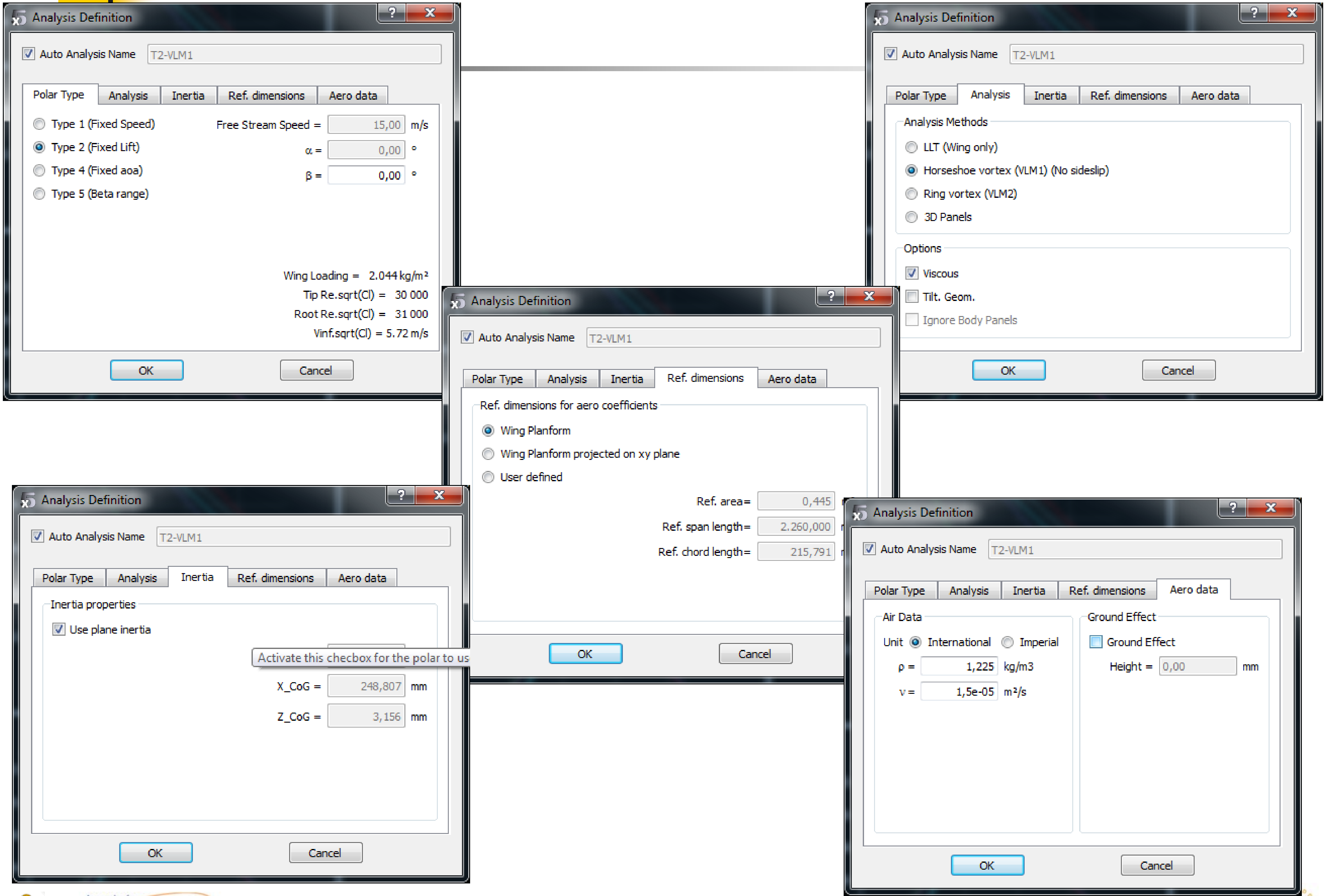
XFLR5 – Estudio 2D

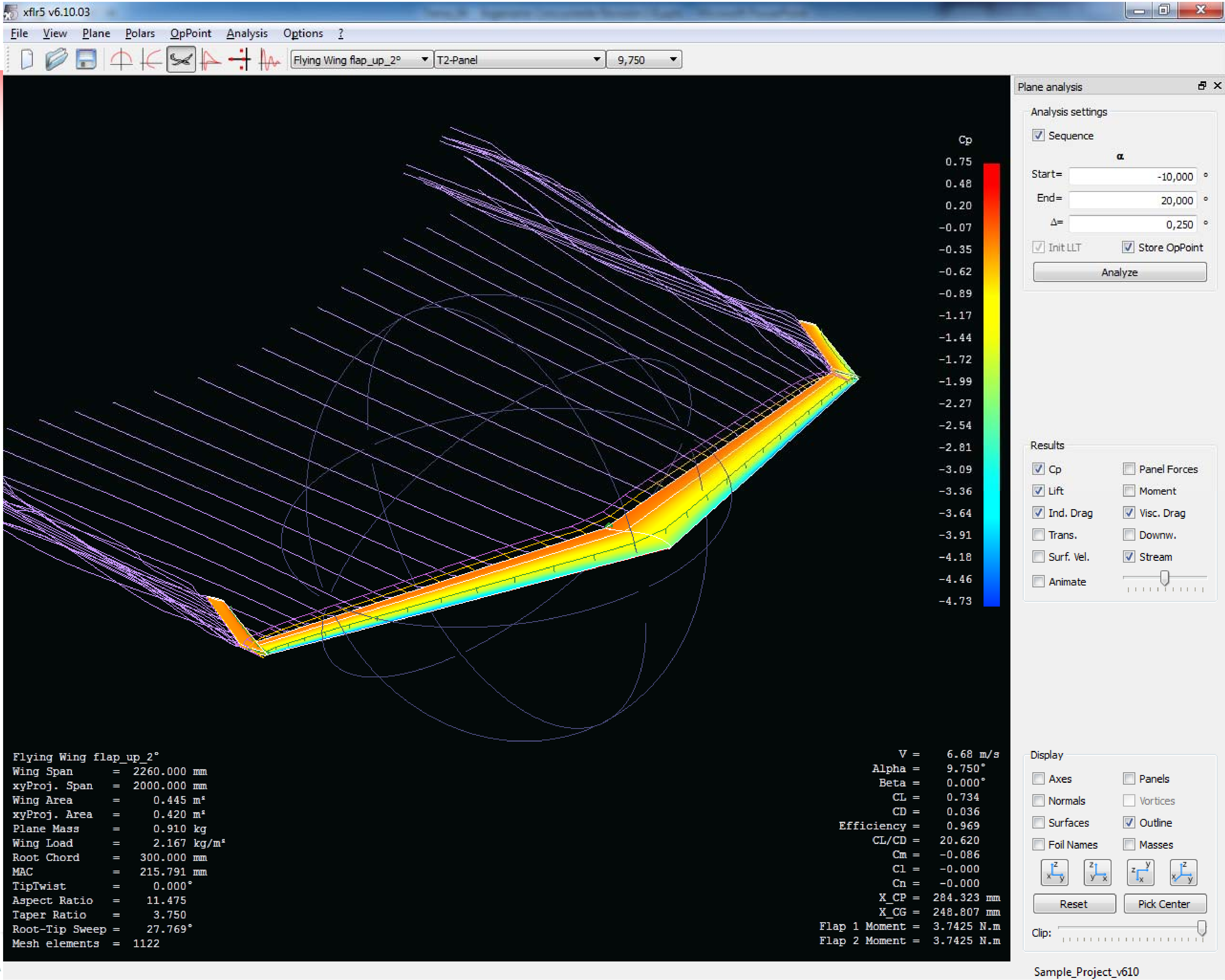


	Name	Thickness (%)	at (%)	Camber (%)	at (%)	Points	TE Flap (°)	TE XHinge	TE YHinge	Show	Points	Centerline	Style
1	Spline foil	9.03	30.00	0.00	49.00	158	0.00	0.00	0.00				
2	NACA 0012-64	12.00	40.00	-0.06	0.00	200	0.00	0.00	0.00	X			
3	NACA 0012-64 (Flap 30)	12.00	40.00	-0.06	0.00	202	30.00	75.00	50.00				
4	NACA 0012-64 (Flap m30)	12.00	40.00	-0.06	0.00	202	-30.00	75.00	50.00				
5	NACA 0012-64 (flap)	12.00	40.00	-0.06	0.00	200	0.00	0.00	0.00				
6	NACA 0012-64 (flap30)	12.00	40.00	-0.06	0.00	202	30.00	75.00	50.00				
7	St. CYR 24 (Bartel 35-IIIC)	12.36	28.97	7.44	34.98	100	0.00	0.00	0.00	X			
8	St. CYR 24 (Flap 15md)	12.35	29.37	7.44	36.23	104	0.00	0.00	0.00	X			
9	St. CYR 24 (Flap 30)	12.36	28.97	7.44	34.98	104	30.00	75.00	50.00	X			
10	St. CYR 24 (Flap 40)	12.36	28.97	7.44	34.98	100	0.00	0.00	0.00	X			
11	St. CYR 24 (Flap 45)	12.36	28.97	7.44	34.98	103	45.00	75.00	50.00	X			
12	St. CYR 24 (Flap m30)	12.36	28.97	7.44	34.98	104	-30.00	75.00	50.00				

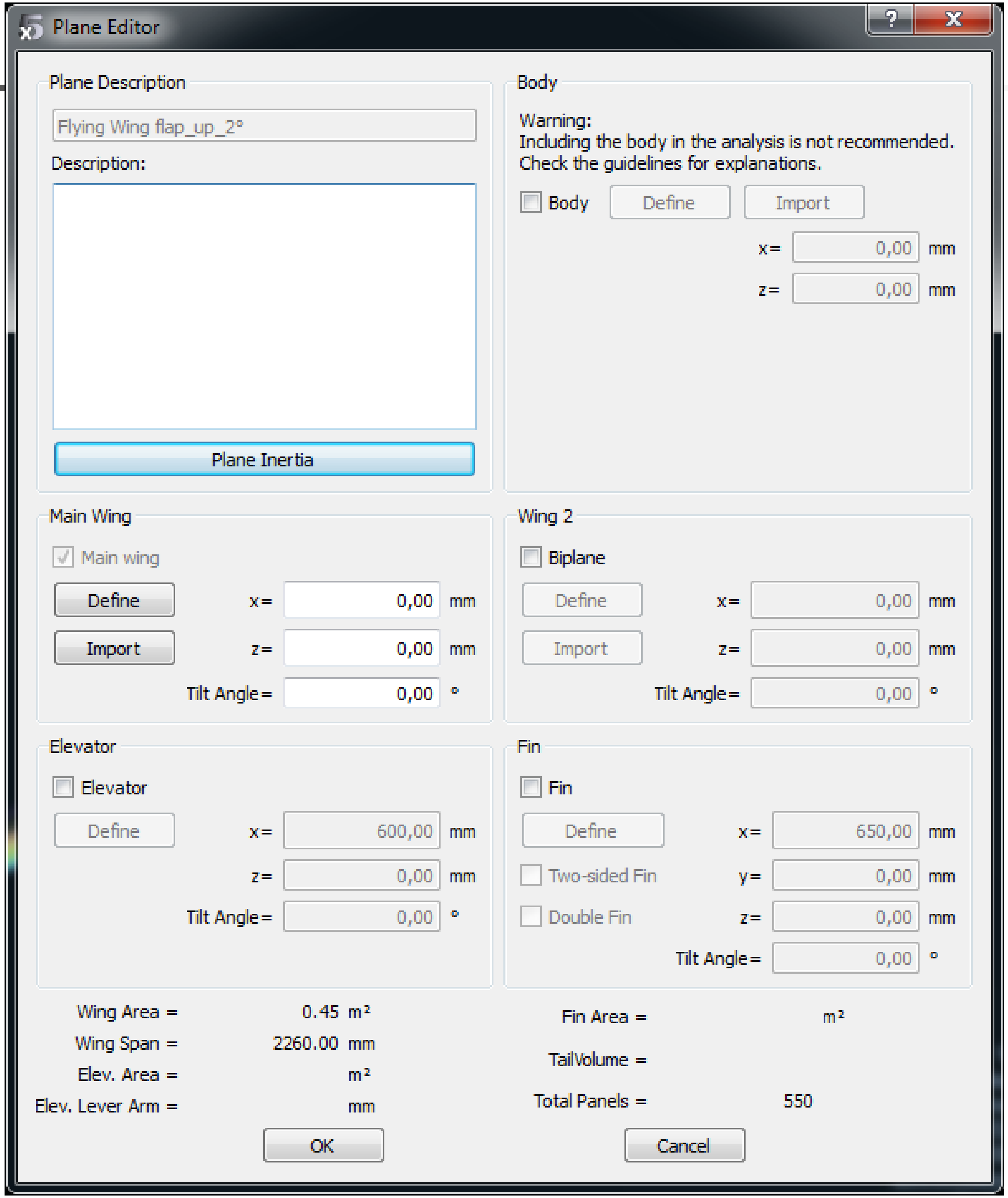
XFLR5 – Estudio 2D

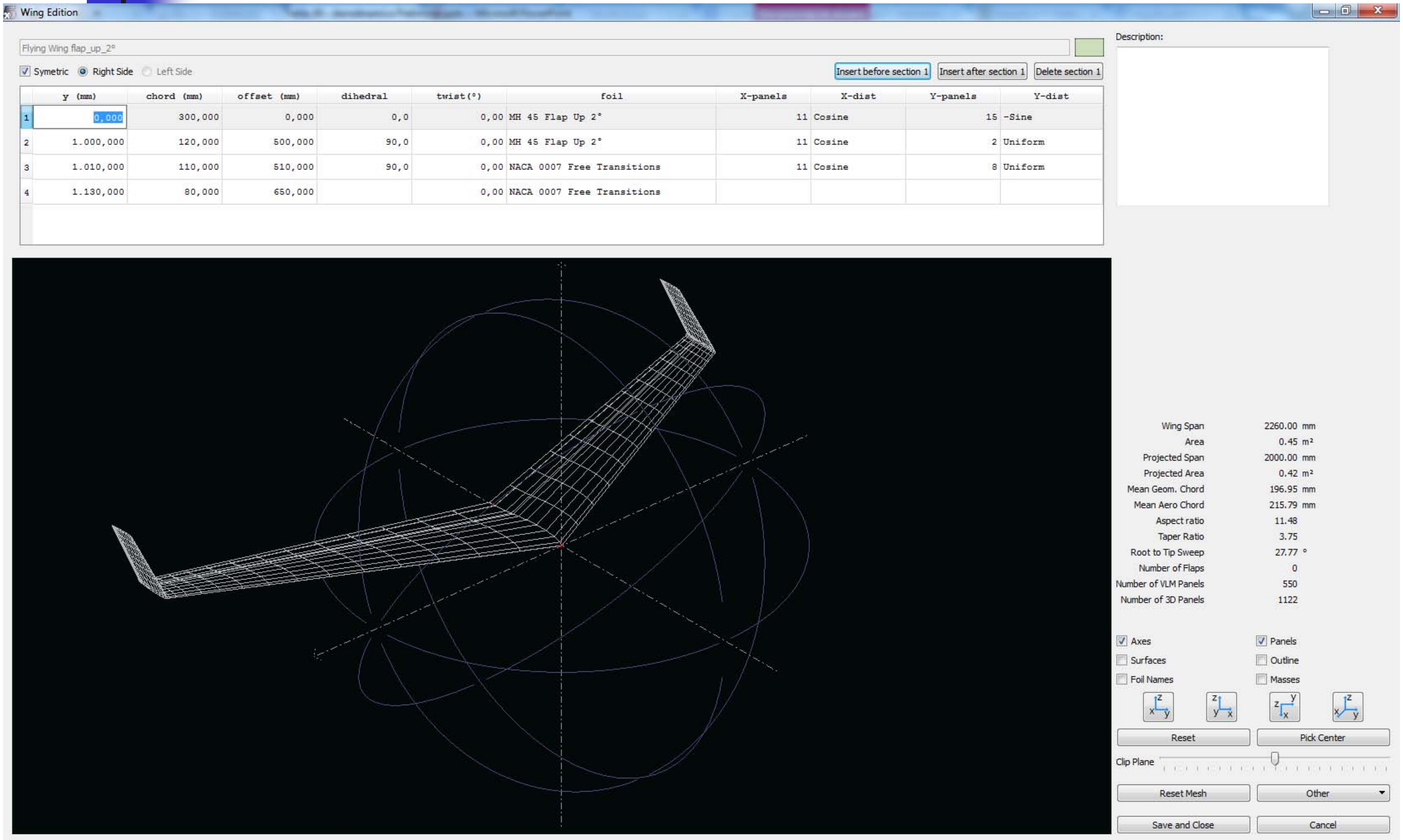



Analysis Definition
Auto Analysis Name
T2-VLM1
Polar Type
Analysis
Inertia
Ref. dimensions
Aero data
Type 1 (Fixed Speed)
Type 2 (Fixed Lift)
Type 4 (Fixed aoa)
Type 5 (Beta range)
Free Stream Speed = 15,00 m/s
α = 0,00 °
β = 0,00 °
Wing Loading = 2.044 kg/m²
Tip Re.sqrt(Cl) = 30 000
Root Re.sqrt(Cl) = 31 000
Vinf.sqrt(Cl) = 5.72 m/s
OK
Cancel
Analysis Methods
LLT (Wing only)
Horseshoe vortex (VLM1) (No sideslip)
Ring vortex (VLM2)
3D Panels
Options
Viscous
Tilt. Geom.
Ignore Body Panels
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
Inertia properties
Use plane inertia
Activate this checbox for the polar to us
X_CoG = 248,807 mm
Z_CoG = 3,156 mm
Air Data
Unit International Imperial
ρ = 1,225 kg/m3
ν = 1,5e-05 m²/s
Ground Effect
Ground Effect
Height = 0,00 mm

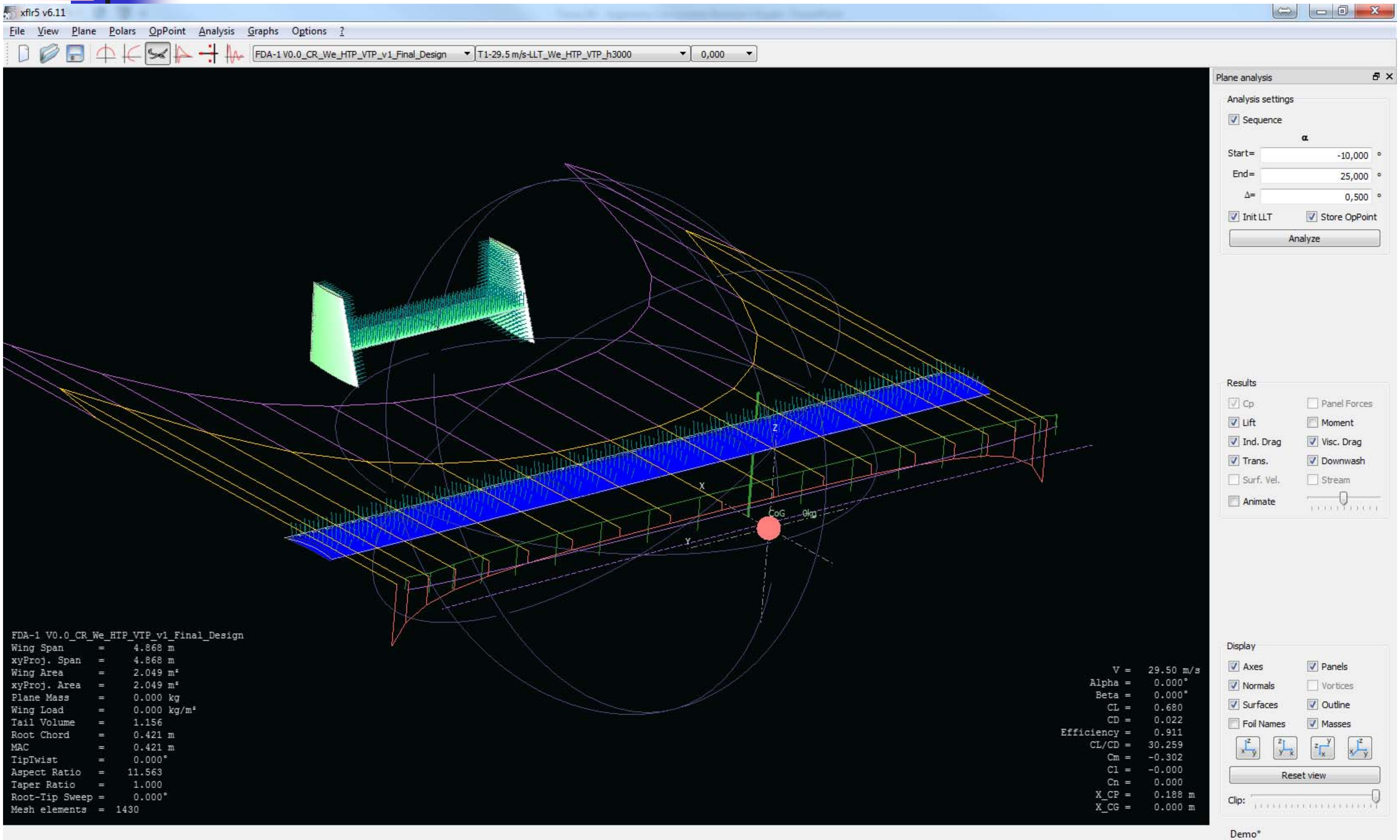


xflr5 v6.10.03
File View Plane Polars OpPoint Analysis Options ?
Flying Wing flap_up_2°
T2-Panel
9,750
Plane analysis
Analysis settings
Sequence
α
Start= -10,000 °
End= 20,000 °
Δ= 0,250 °
Init LLT
Store OpPoint
Analyze
Results
Cp
Panel Forces
Lift
Moment
Ind. Drag
Visc. Drag
Trans.
Downw.
Surf. Vel.
Stream
Animate
Display
Axes
Panels
Normals
Vortices
Surfaces
Outline
Foil Names
Masses
Reset
Pick Center
Clip:
Sample_Project_v610
Cp
0.75
0.48
0.20
-0.07
-0.35
-0.62
-0.89
-1.17
-1.44
-1.72
-1.99
-2.27
-2.54
-2.81
-3.09
-3.36
-3.64
-3.91
-4.18
-4.46
-4.73
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

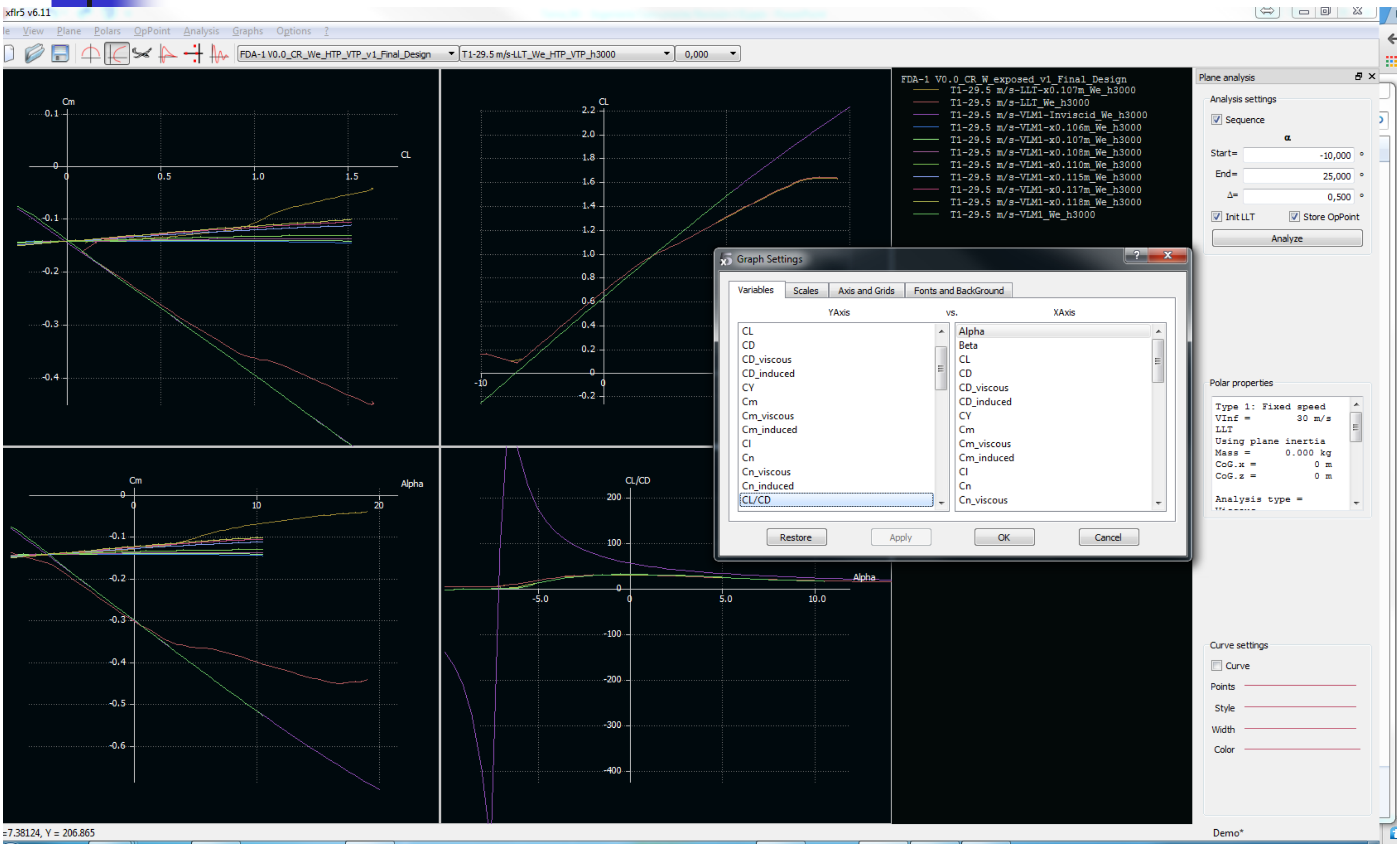

Plane Editor
Plane Description
Flying Wing flap_up_2°
Description:
Plane Inertia
Body
Warning:
Including the body in the analysis is not recommended.
Check the guidelines for explanations.
Body
Define
Import
x= 0,00 mm
z= 0,00 mm
Main Wing
Main wing
Define
Import
x= 0,00 mm
z= 0,00 mm
Tilt Angle= 0,00 °
Wing 2
Biplane
Define
Import
x= 0,00 mm
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
Two-sided Fin
Double Fin
x= 650,00 mm
y= 0,00 mm
z= 0,00 mm
Tilt Angle= 0,00 °
Wing Area = 0.45 m²
Wing Span = 2260.00 mm
Elev. Area = m²
Elev. Lever Arm = mm
Fin Area = m²
TailVolume =
Total Panels = 550
OK
Cancel


Wing Edition
Flying Wing flap_up_2°
Description:
Symetric
Right Side
Left Side
Insert before section 1
Insert after section 1
Delete section 1
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 | | | |
Wing Span 2260.00 mm
Area 0.45 m²
Projected Span 2000.00 mm
Projected Area 0.42 m²
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
Reset
Pick Center
Clip Plane
Reset Mesh
Other
Save and Close
Cancel

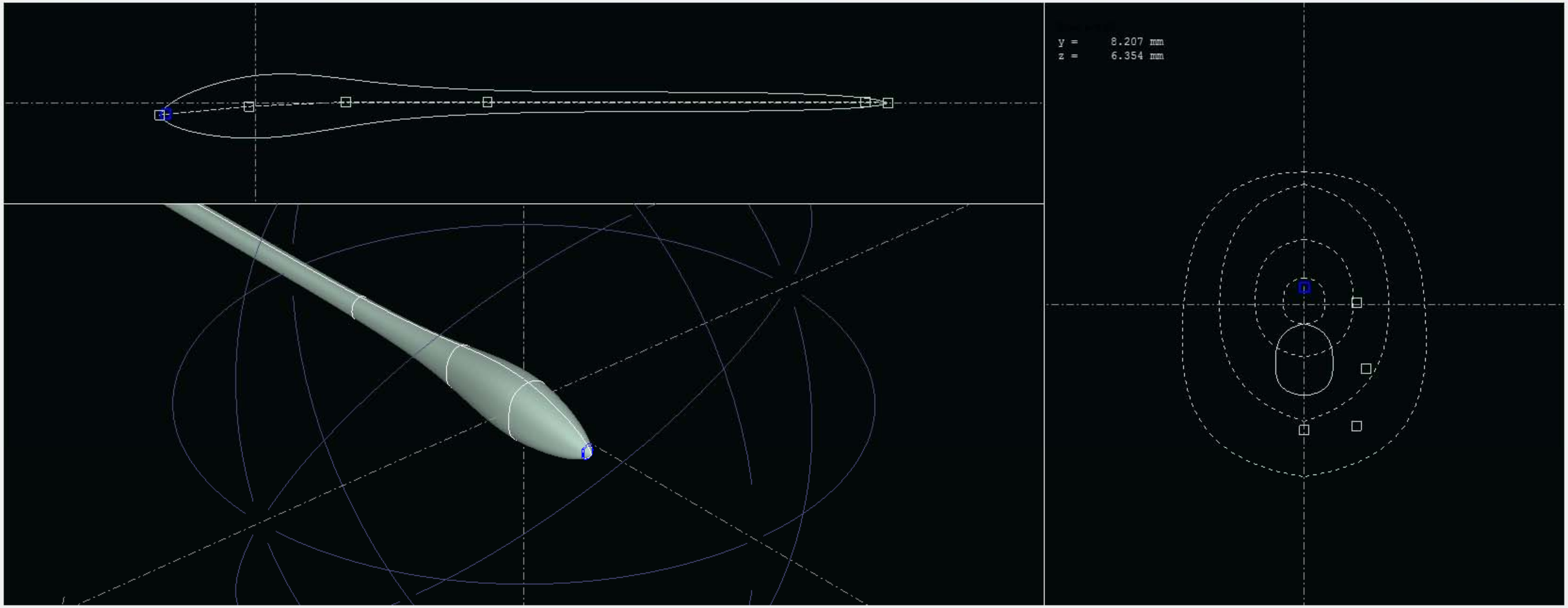
XFLR5 – Estudio 3D



XFLR5 – Estudio 3D



Body Edition



Flying Wing flap_up_2°_body

Description:

Flat Panels BSplines

	x	Hoop
Degree	3	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

```
xflr5 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 α_{0L}, C_{lmax} y α_{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: δ_f

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

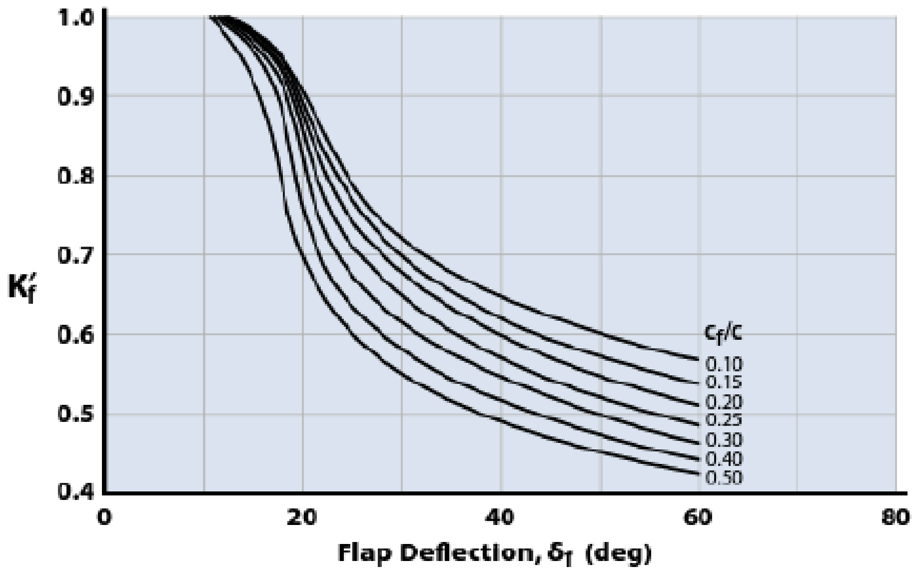


Figure 9.9 Nonlinear correction for plain TE flaps (adapted [10]).

- 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_ℓ for a change in δ_f, Fig. 9.10

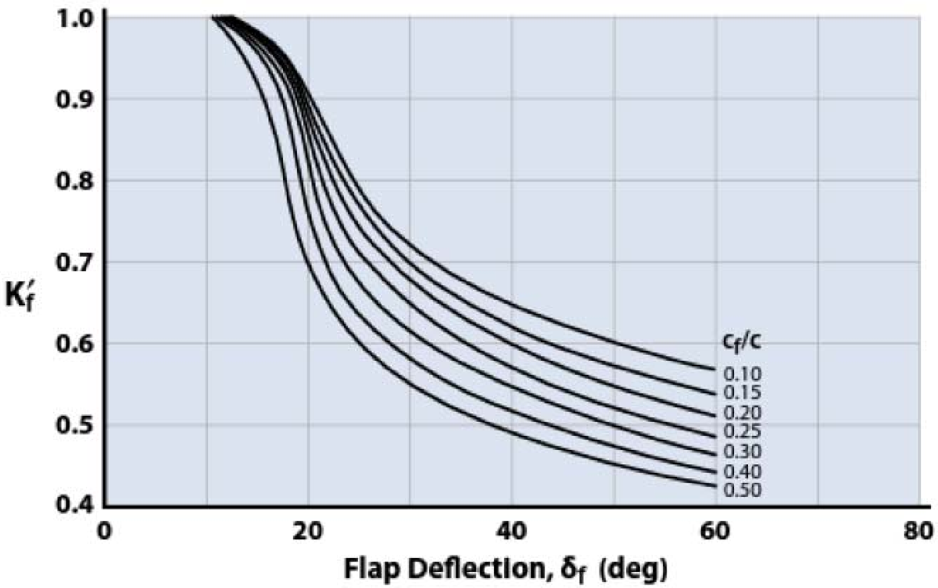


Figure 9.9 Nonlinear correction for plain TE flaps (adapted [10]).

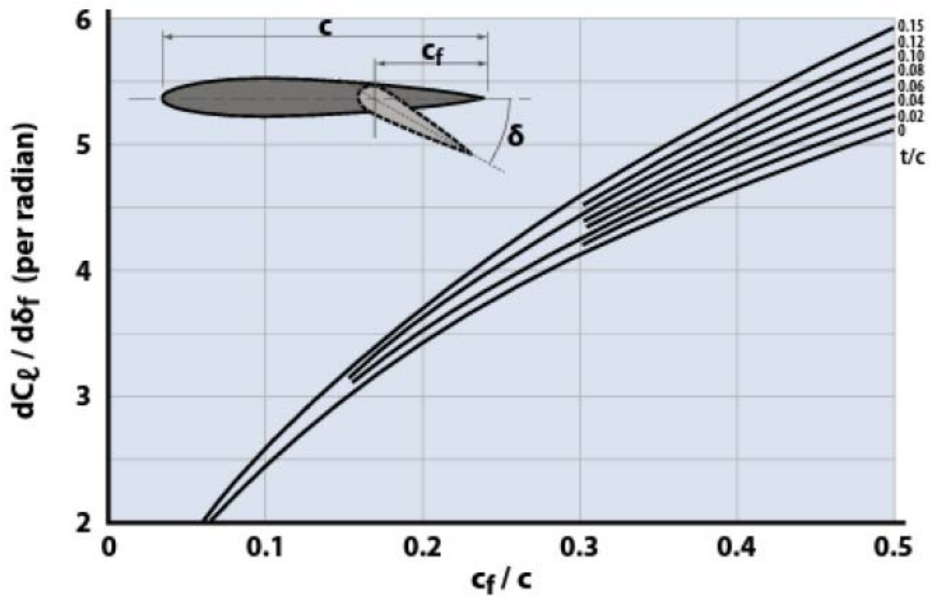


Figure 9.10 Variation of $dC_\ell/d\delta_f$ with flap chord ratio (adapted [10]).

Plain TE Flap - I

Section Lift Coefficient

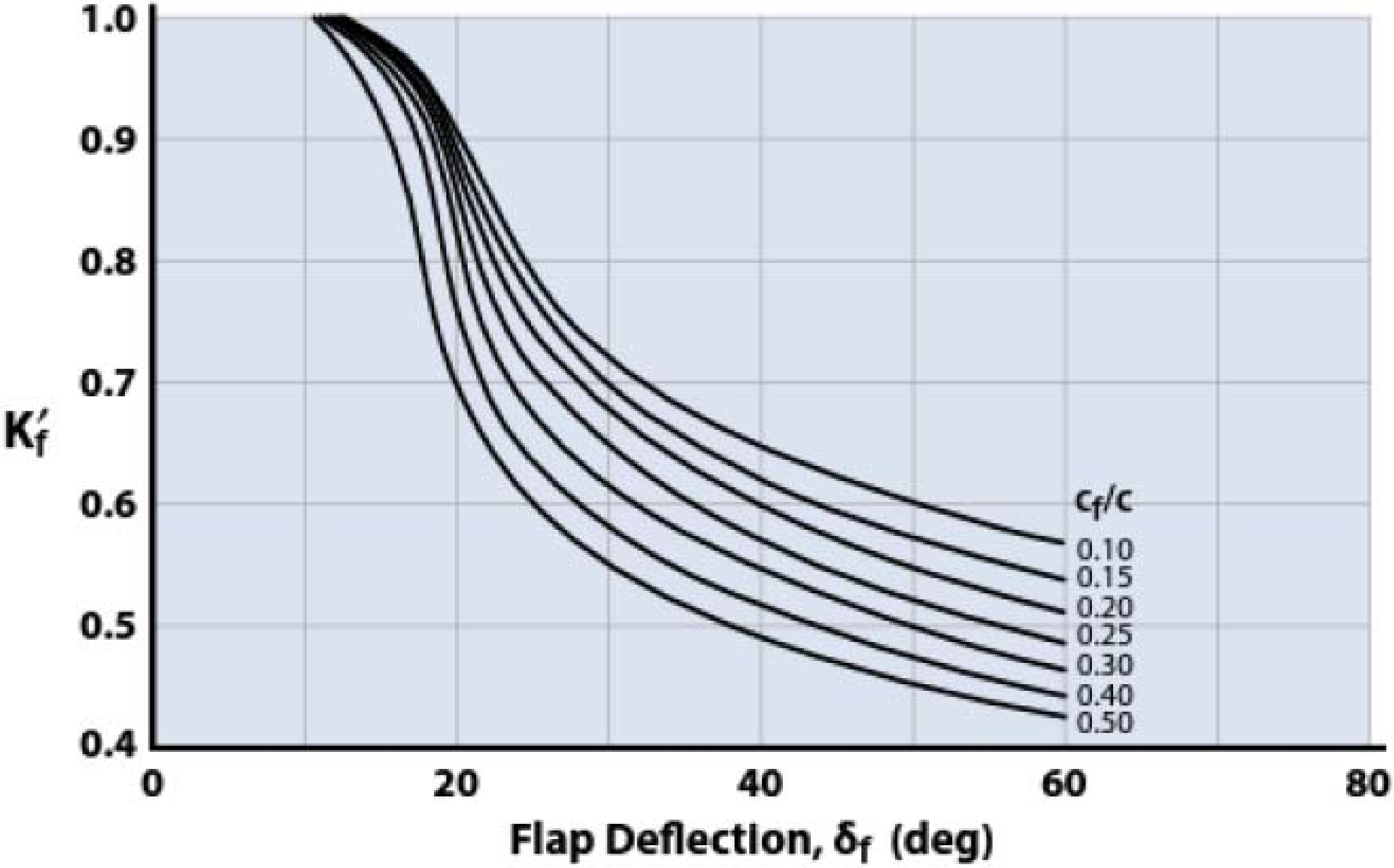


Figure 9.9 Nonlinear correction for plain TE flaps (adapted [10]).

Plain TE Flap - II

Section Lift Coefficient

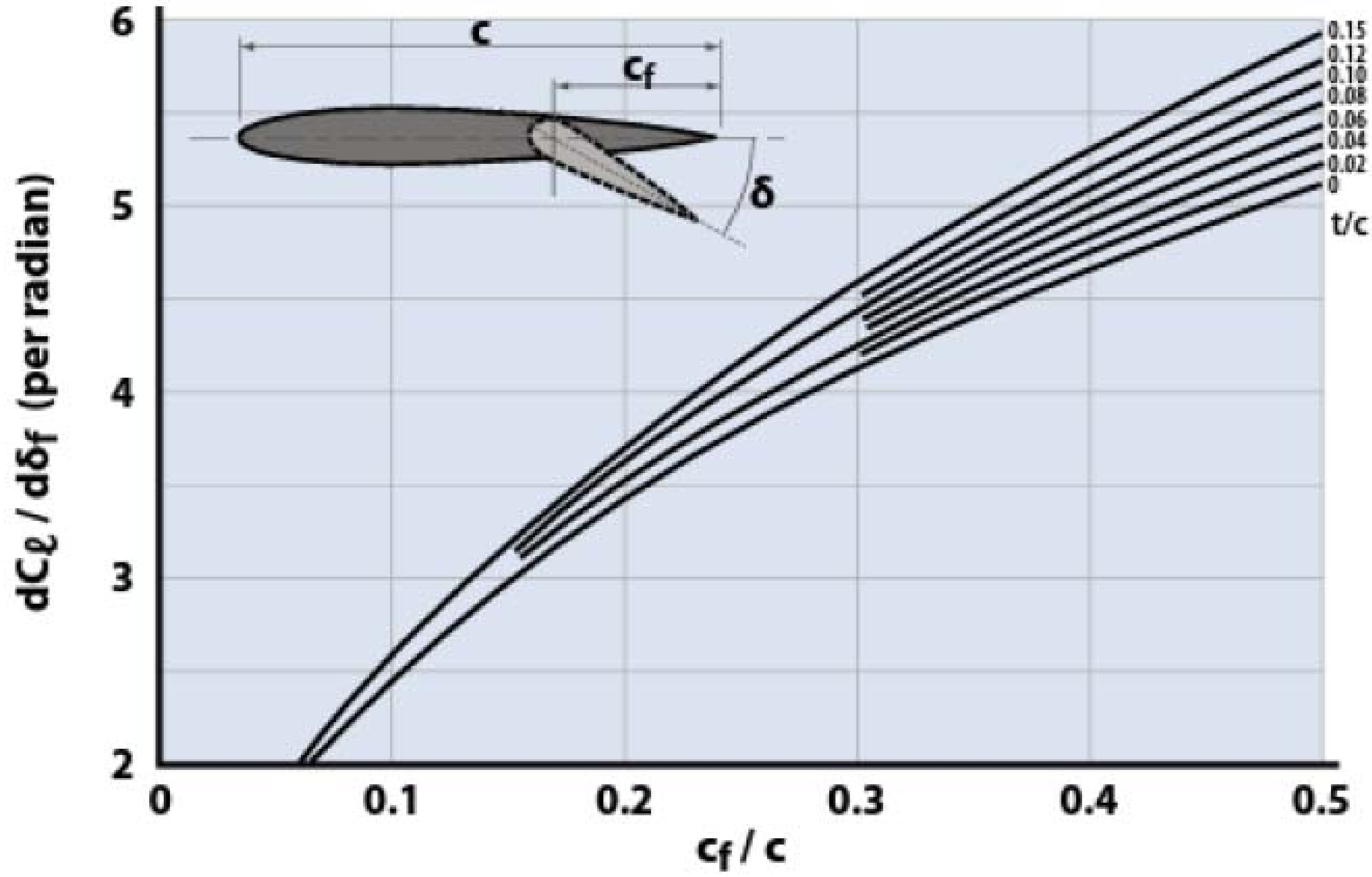


Figure 9.10 Variation of $dC_\ell/d\delta_f$ with flap chord ratio (adapted [10]).

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

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

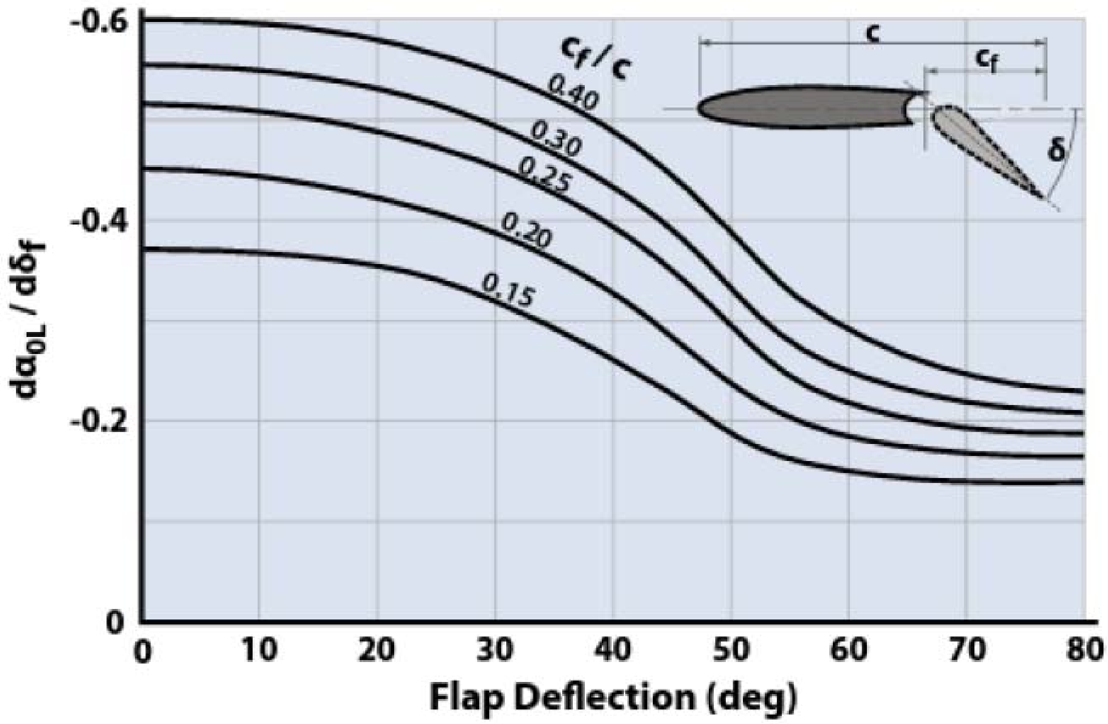


Figure 9.11 Section lift effectiveness parameter for single-slotted flaps (adapted [10]).

Single-Slotted TE Flap - I

Section Lift Coefficient

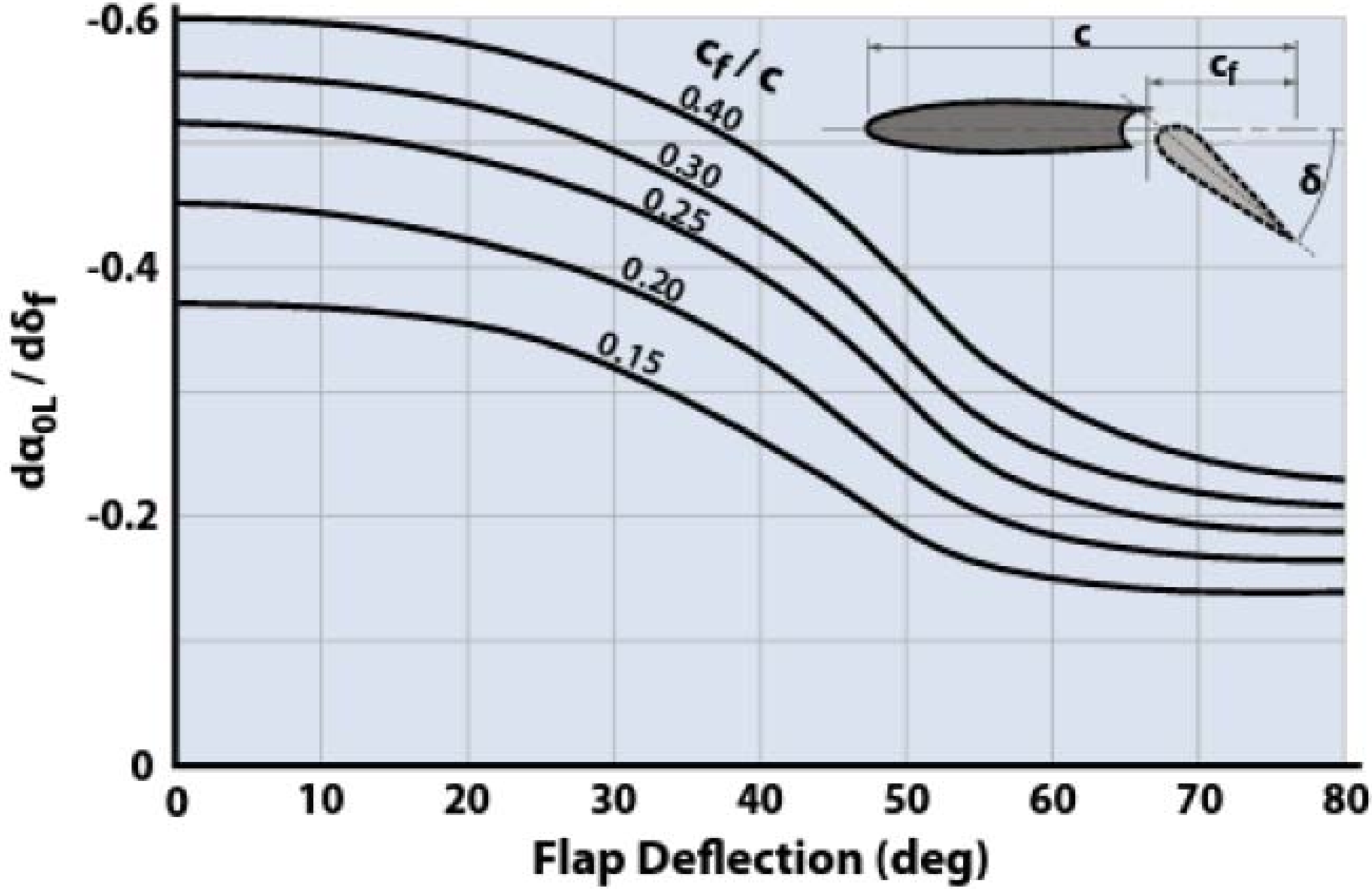


Figure 9.11 Section lift effectiveness parameter for single-slotted flaps (adapted [10]).

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

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

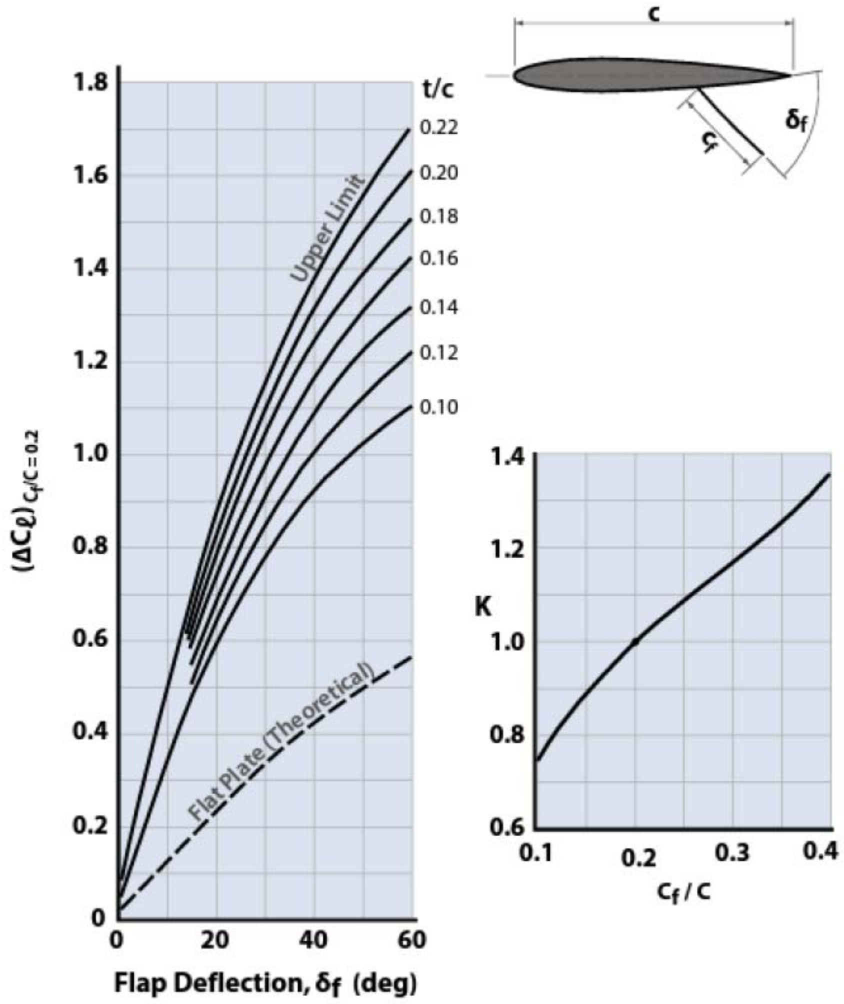


Figure 9.12 Empirical constants for split flap analysis (adapted [10]).

Split TE Flap - I

Section Lift Coefficient

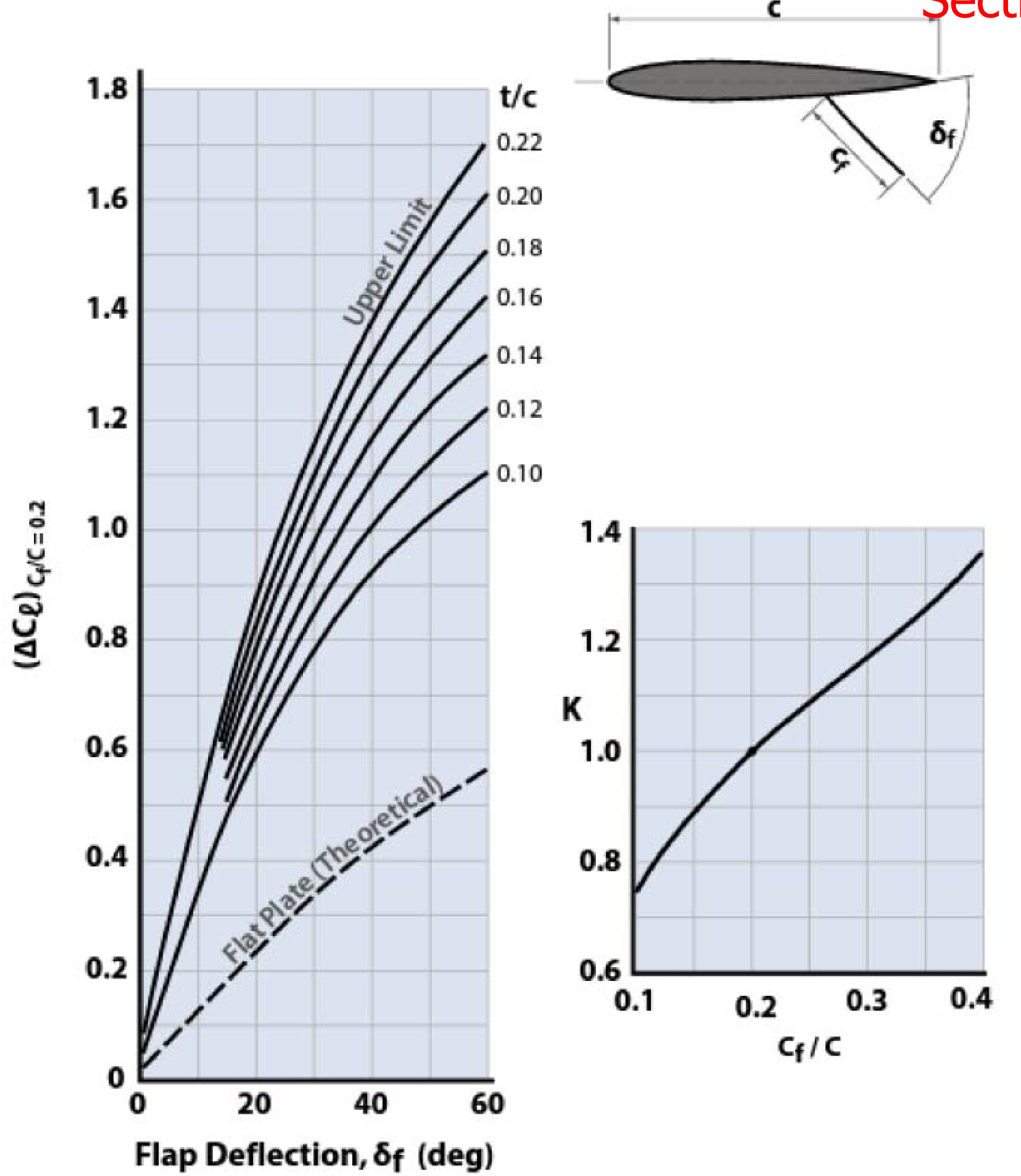


Figure 9.12 Empirical constants for split flap analysis (adapted [10]).

Cálculo C_{Lmax} – Métodos Empíricos - V

Section Lift Coefficient

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

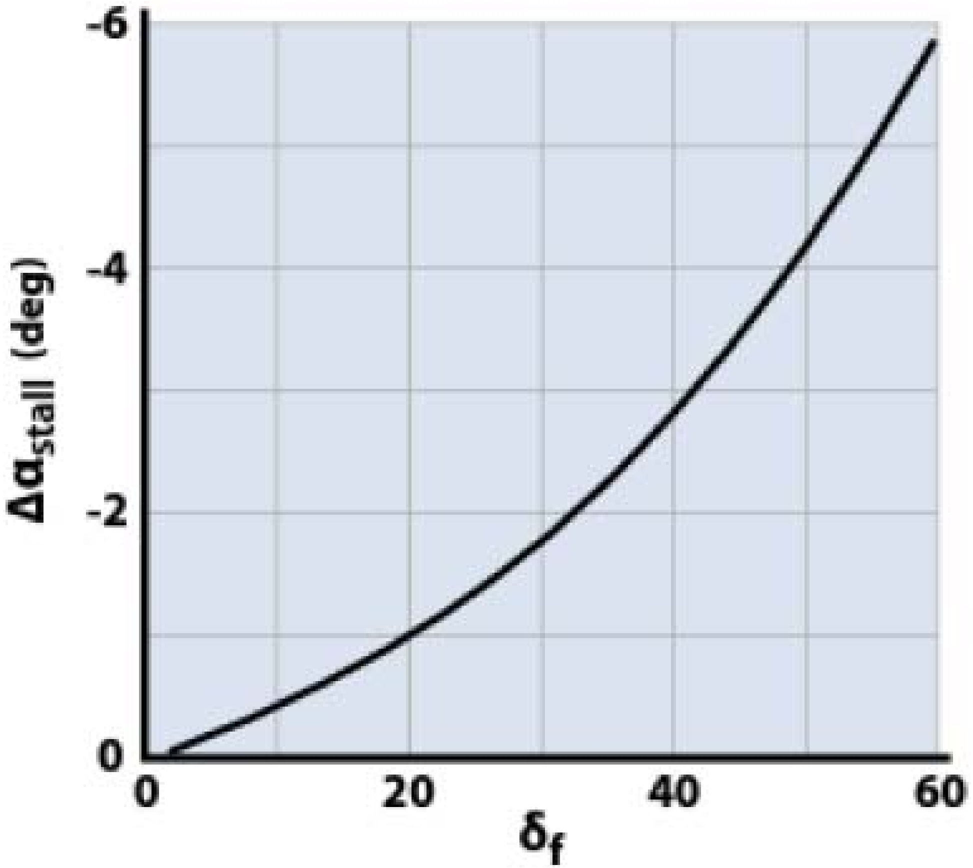


Figure 9.13 Decrease in stall angle with flap deflection (data from [2]).

Cálculo C_{Lmax} – básico - I

- Correcciones 2D a 3D
  - Cálculo de las características del ala básica (sin HLD)
    - Corrección C_{L_α} 2D -> 3D

corregido

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

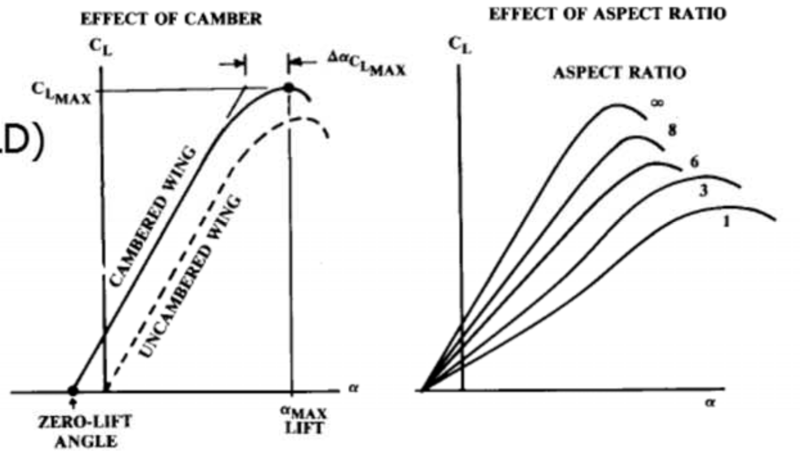


Fig. 12.4 Wing lift curve.

- Cálculo de características básicas
  - Alas con bajo alargamiento (AR)
    - Depende de la forma del ala

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

$\Rightarrow$

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

  - Alas con alto alargamiento (AR)
    - Depende del perfil seleccionado

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

$\Rightarrow$

$$\left(\frac{C_{L_{max}}}{C_{l_{max}}}\right)$$
$$C_{l_{max}}$$
$$C_{l_\alpha}$$
$$\alpha_{0L}$$
$$\Delta\alpha_{C_{L_{max}}}$$

Cálculo C_{Lmax} - II

Section Lift Coefficient

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

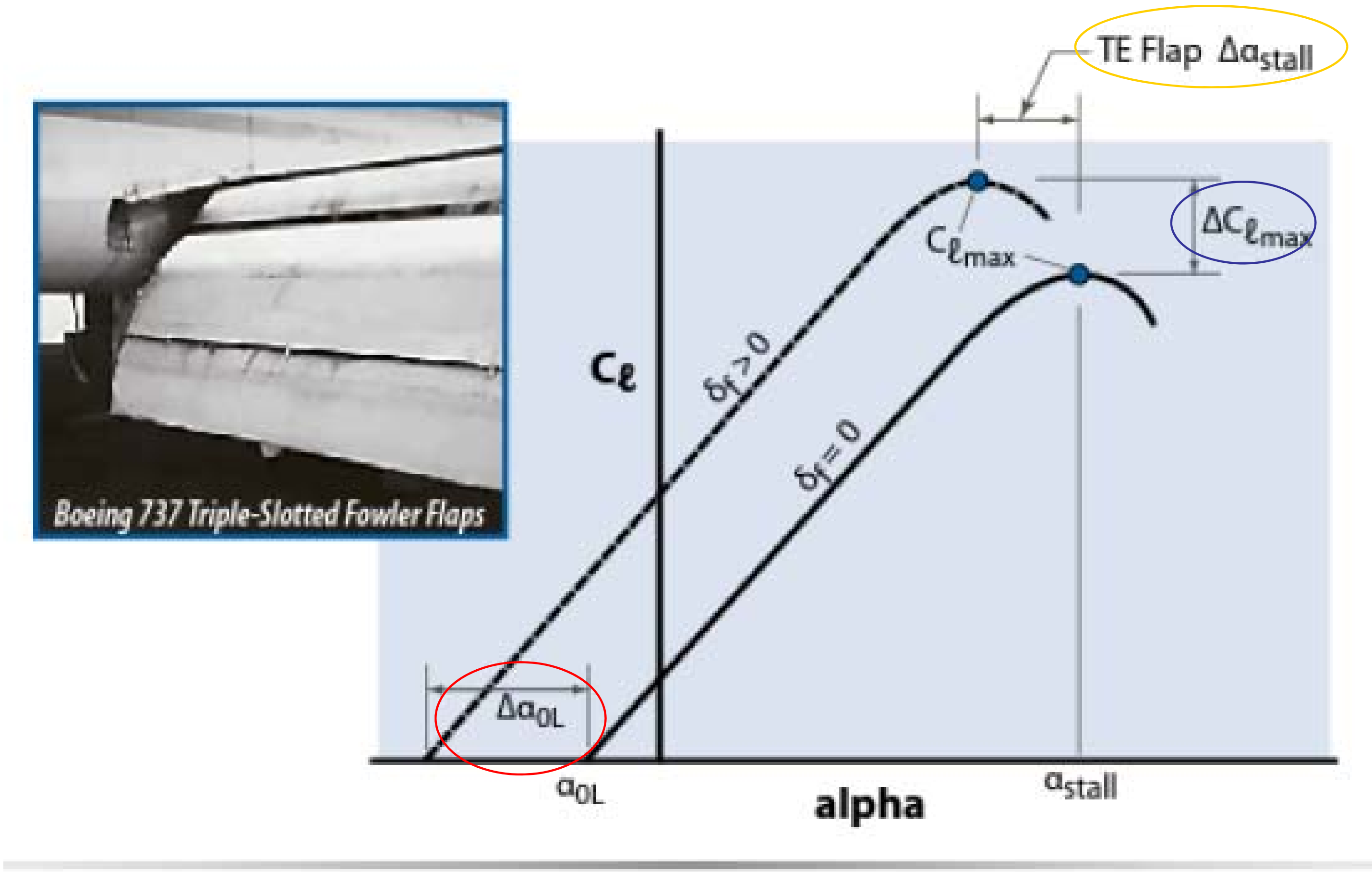


Figure 9.8 Construction of section lift curves for TE flaps.

Cálculo C_{Lmax} - III

Wing Lift Coefficient

- Para determinar C_{Lmax} se distingue entre :
  - Alas con alto alargamiento (AR) → $A > \dfrac{4}{(C_1+1)\cos\Delta_{LE}}$
    - Depende del perfil seleccionado
  - Alas con bajo alargamiento (AR) → $A < \dfrac{4}{(C_1+1)\cos\Delta_{LE}}$
    - Depende de la forma del ala

Δ_{LE} = leading edge sweep

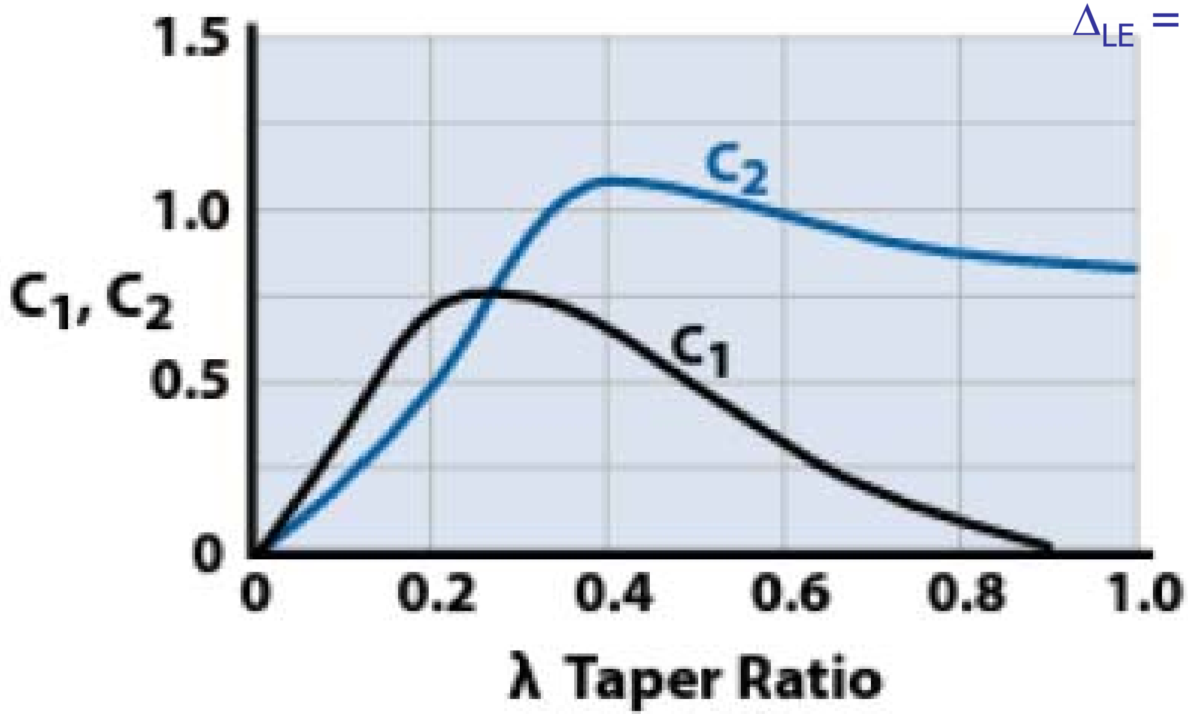


Figure 9.14 Taper ratio correction factors (adapted [10]).

Cálculo C_{Lmax} – High Aspect Ratio - I

Wing Lift Coefficient

- Obtener C_{Lmax} y α_{STALL} para el ala básica

Ángulo de ataque para sustentación nula

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

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

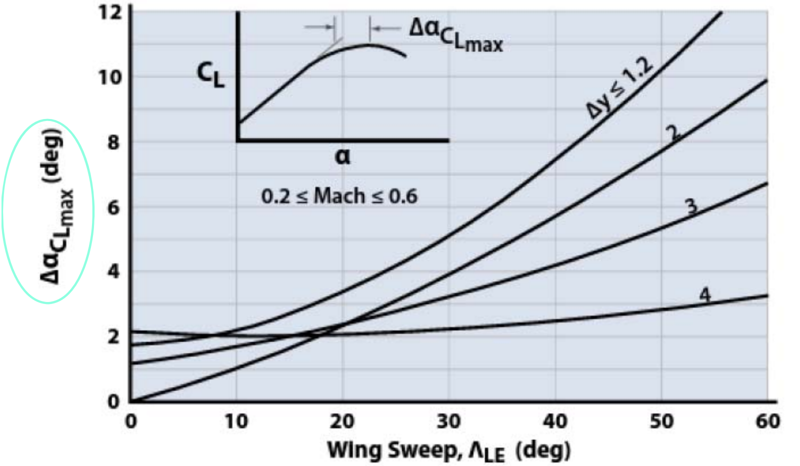


Figure 9.16 Angle-of-attack increment for subsonic maximum lift of high-AR wings (adapted [10]).

$(C_{L_{max}}/C_{\ell_{max}})$

C_{L_α}

α_{0L}

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

$C_{\ell_{max}}$

Airfoil maximum lift coefficient at M=0.2

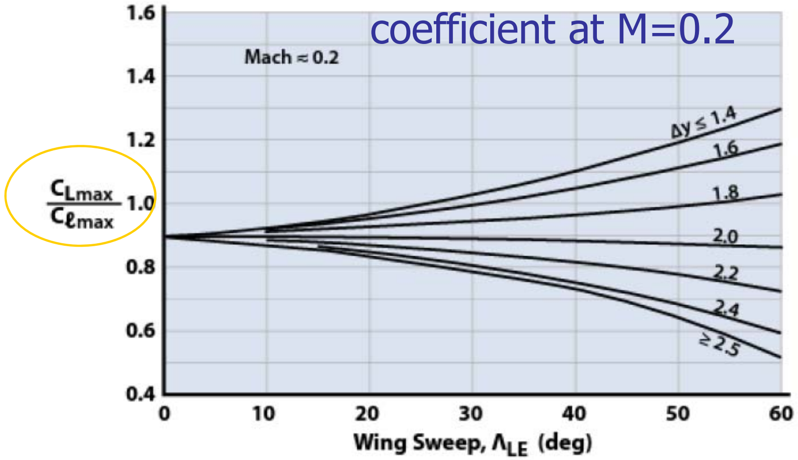
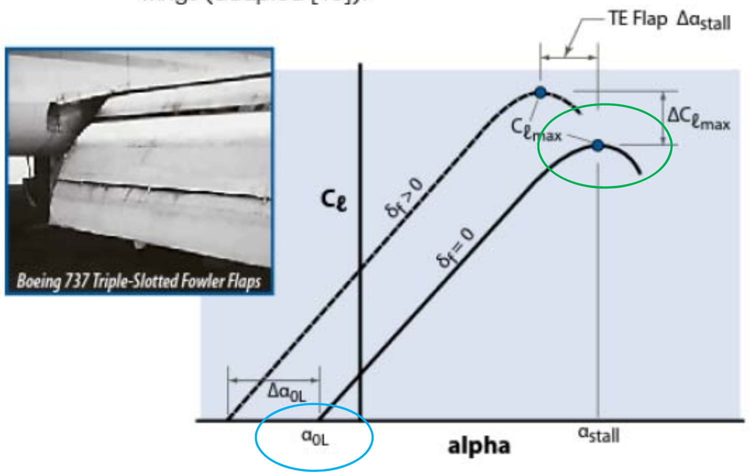


Figure 9.15 Subsonic maximum lift of high-AR wings (adapted [10]).



corregido

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

Figure 9.8 Construction of section lift curves for TE flaps.

Cálculo C_{Lmax} – High Aspect Ratio - I

Wing Lift Coefficient

- Obtener C_{Lmax} y α_{STALL} para el ala básica

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

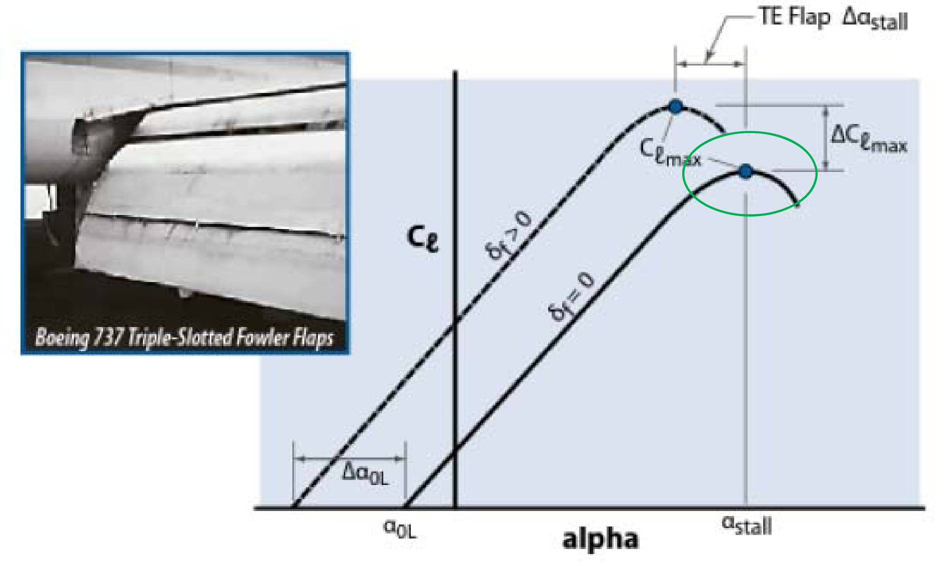


Figure 9.8 Construction of section lift curves for TE flaps.

Airfoil maximum lift coefficient at M=0.2

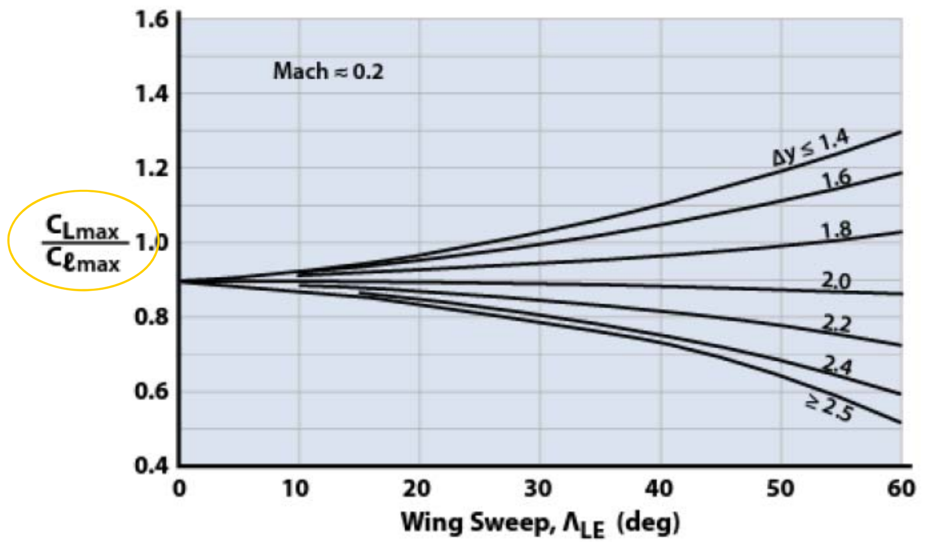


Figure 9.15 Subsonic maximum lift of high-AR wings (adapted [10]).

Cálculo C_{Lmax} – High Aspect Ratio - I

Wing Lift Coefficient

- Obtener C_{Lmax} y α_{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

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

Corrección de los efectos no Lineales del flujo de vórtices

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

Propiedades lineales del ala

Ángulo de ataque para sustentación nula

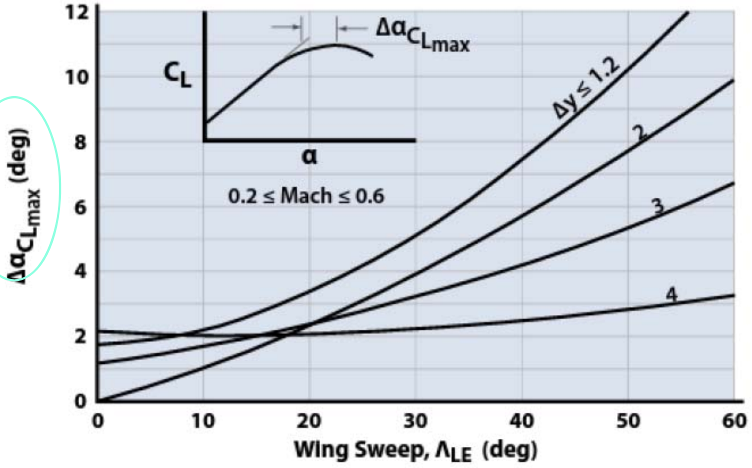


Figure 9.16 Angle-of-attack increment for subsonic maximum lift of high-AR wings (adapted [10]).

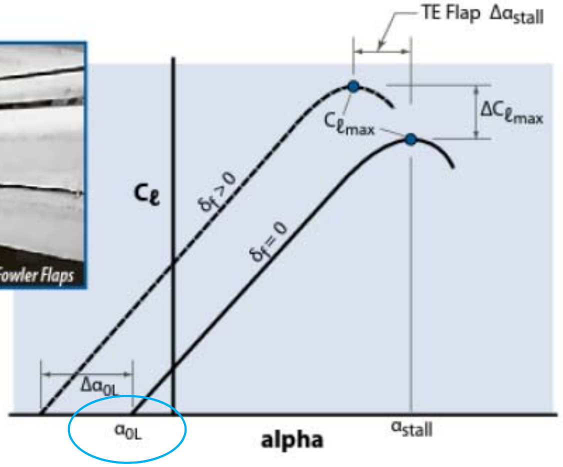
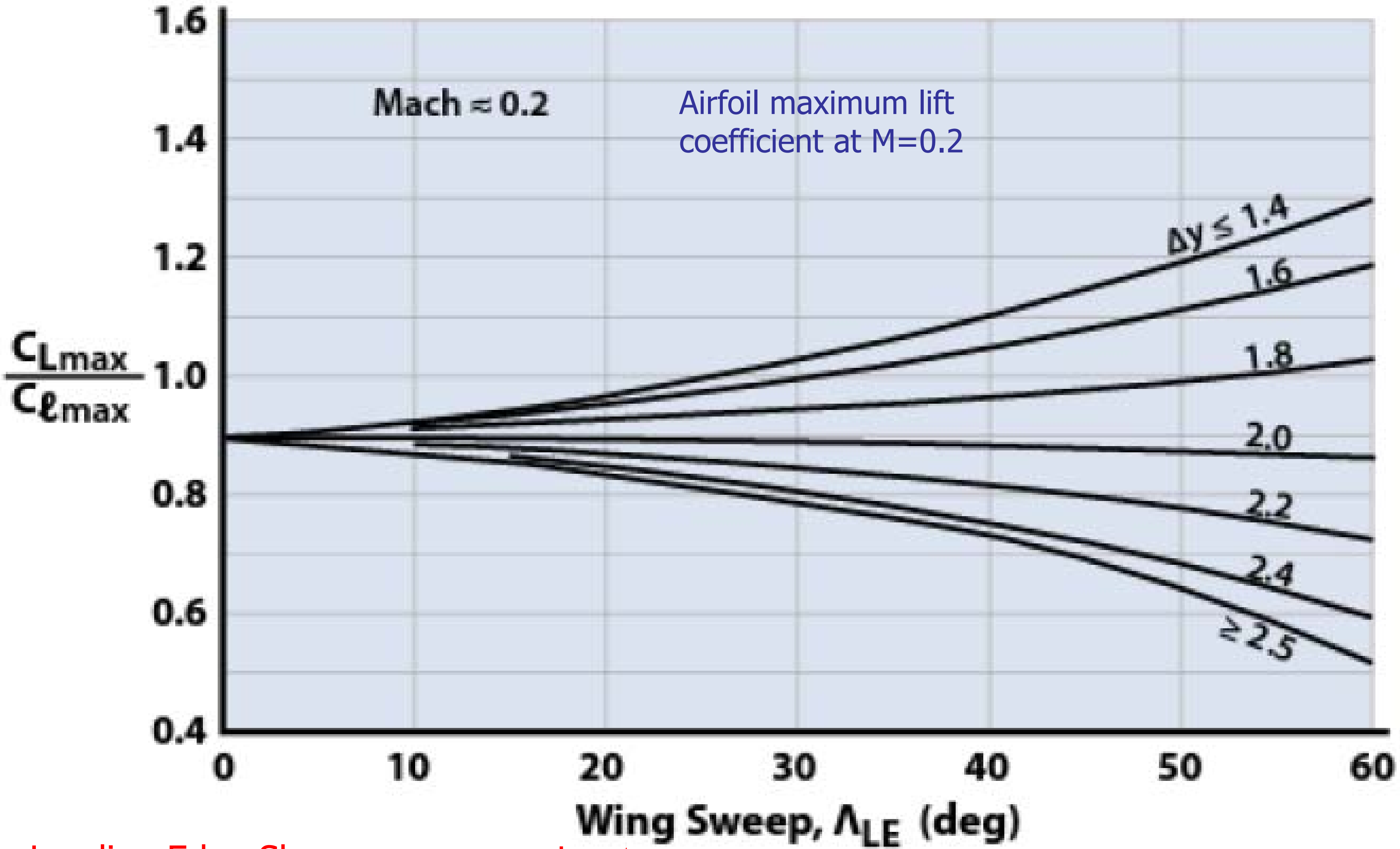


Figure 9.8 Construction of section lift curves for TE flaps.

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

Cálculo C_{Lmax} – High Aspect Ratio - II

Wing Lift Coefficient

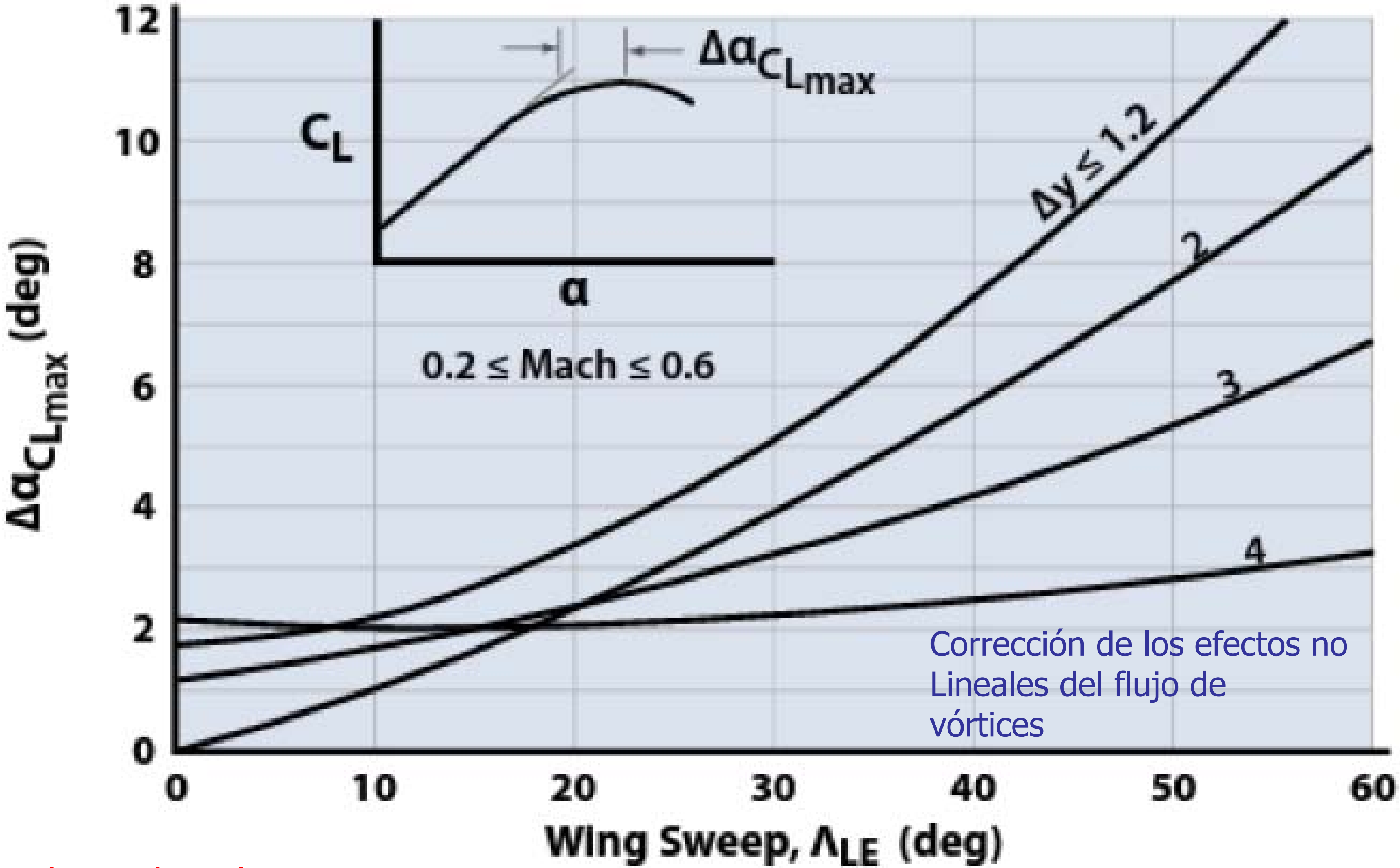


Leading Edge Sharpness parameter: Δy

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: Δy

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

Leading Edge Sharpness parameter : Δy

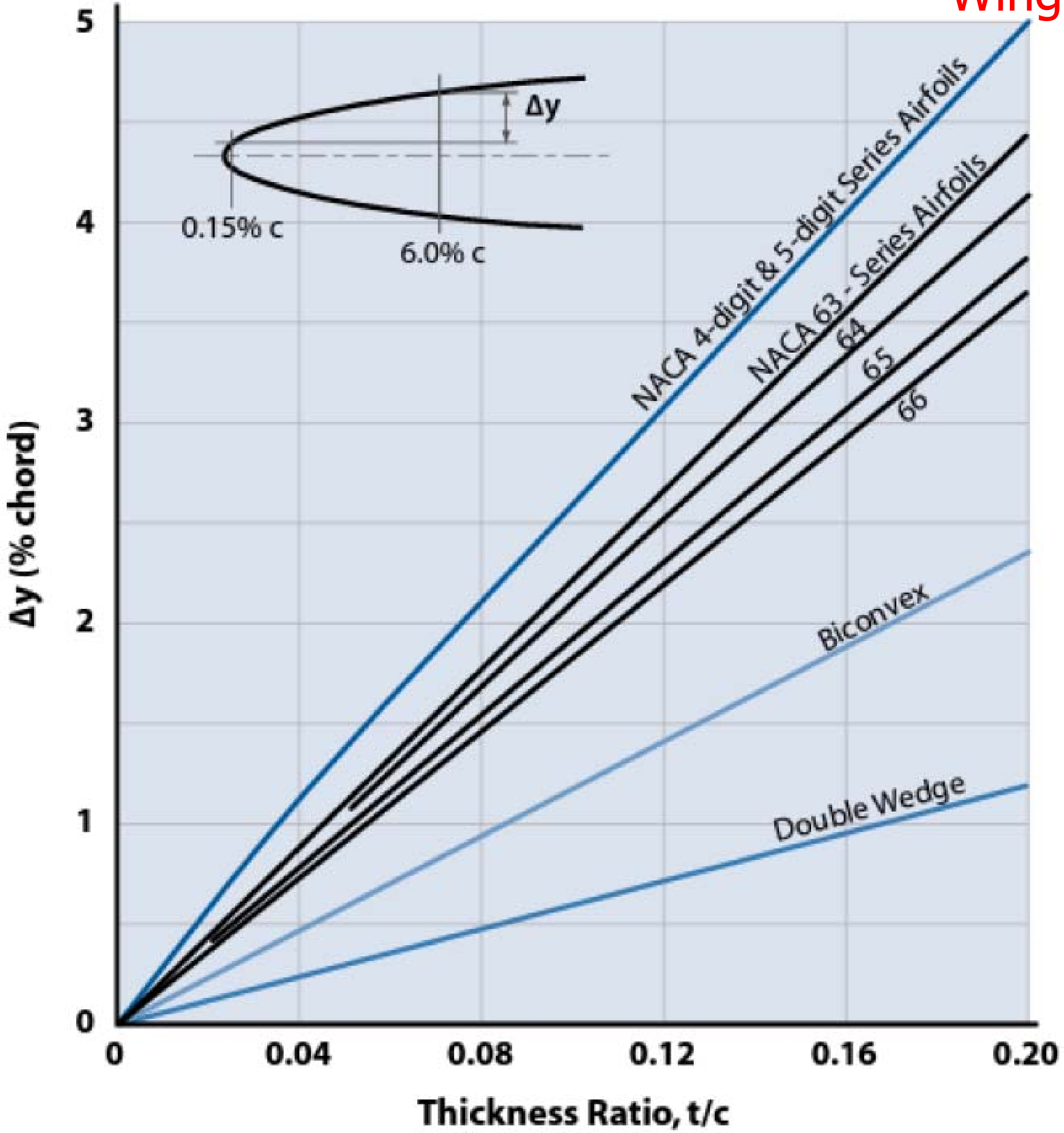
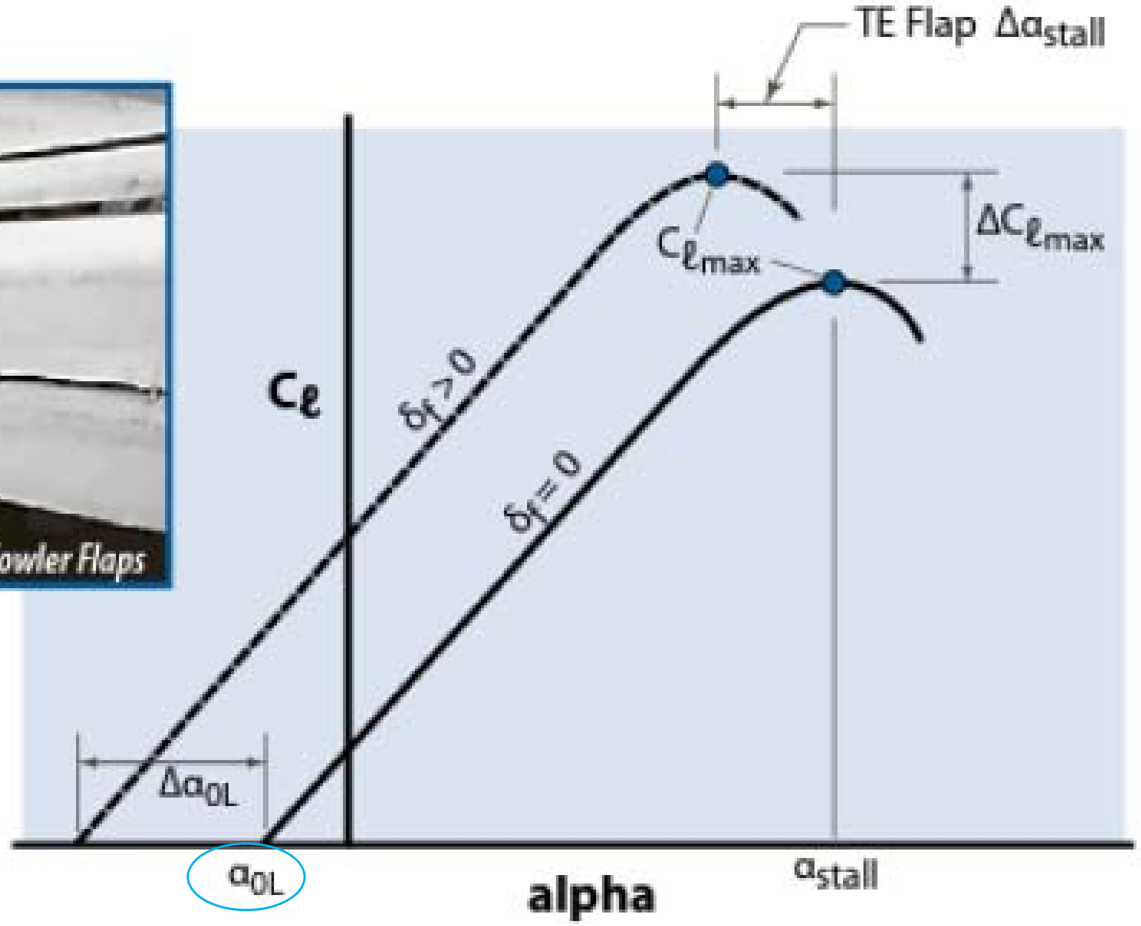


Figure 9.17 Variation of LE sharpness parameter with airfoil thickness ratio (adapted [10]).

Wing Lift Coefficient

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

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 α_{STALL} para el ala básica

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

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

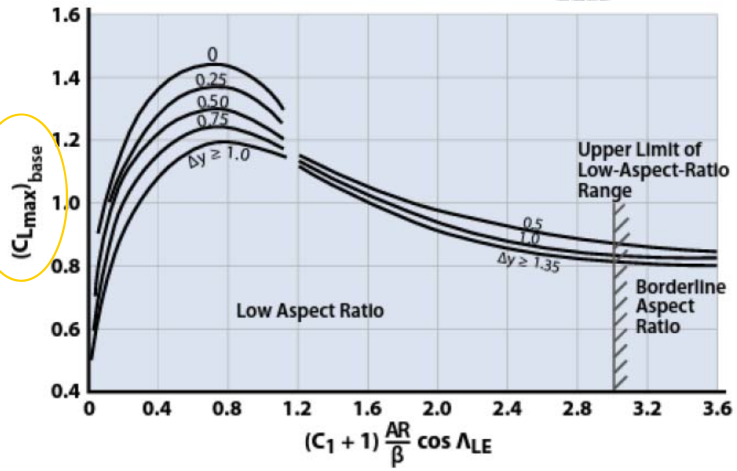


Figure 9.18 Subsonic maximum lift of low-AR wings (adapted [10]).

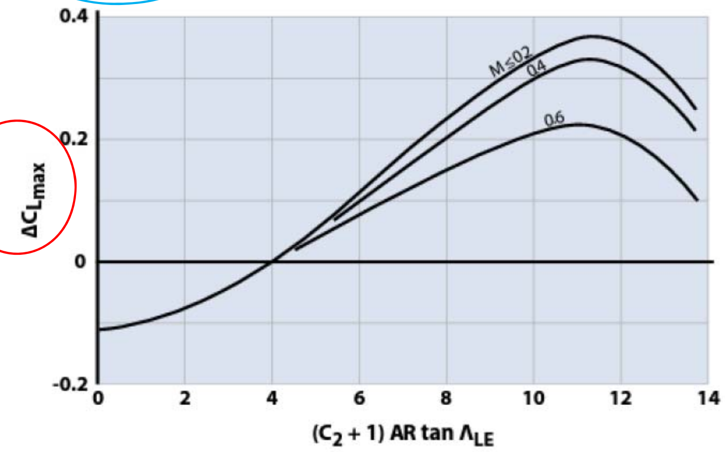


Figure 9.19 Subsonic maximum-lift increment for low-AR wings (adapted [10]).

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

$\Delta C_{L_{max}}$

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

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

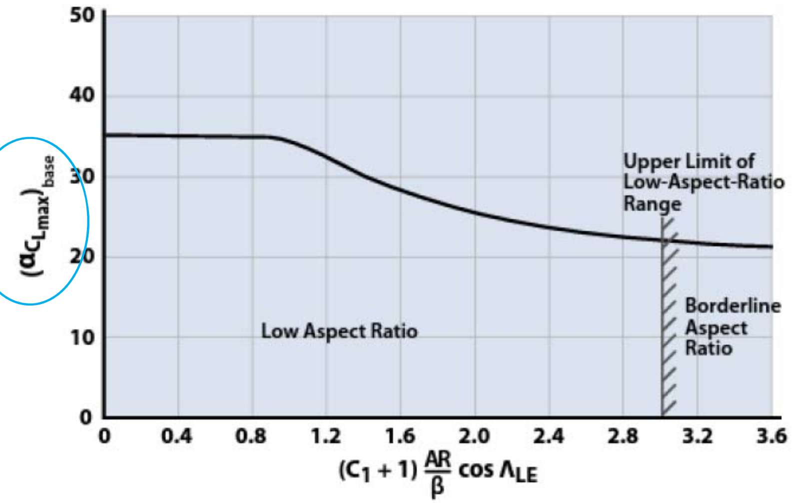


Figure 9.20 Angle-of-attack for subsonic maximum lift of low-AR wings.

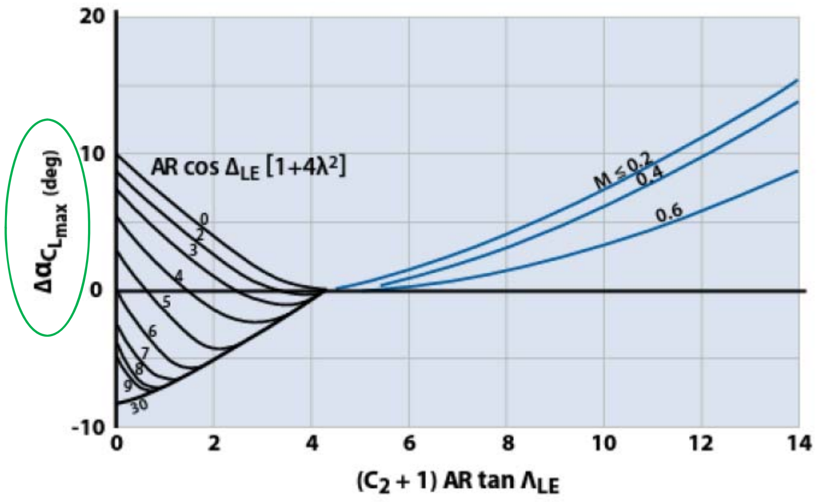


Figure 9.21 Angle-of-attack increment for subsonic maximum lift of low-AR wings.

Cálculo C_{Lmax} – Low Aspect Ratio - I

Wing Lift Coefficient

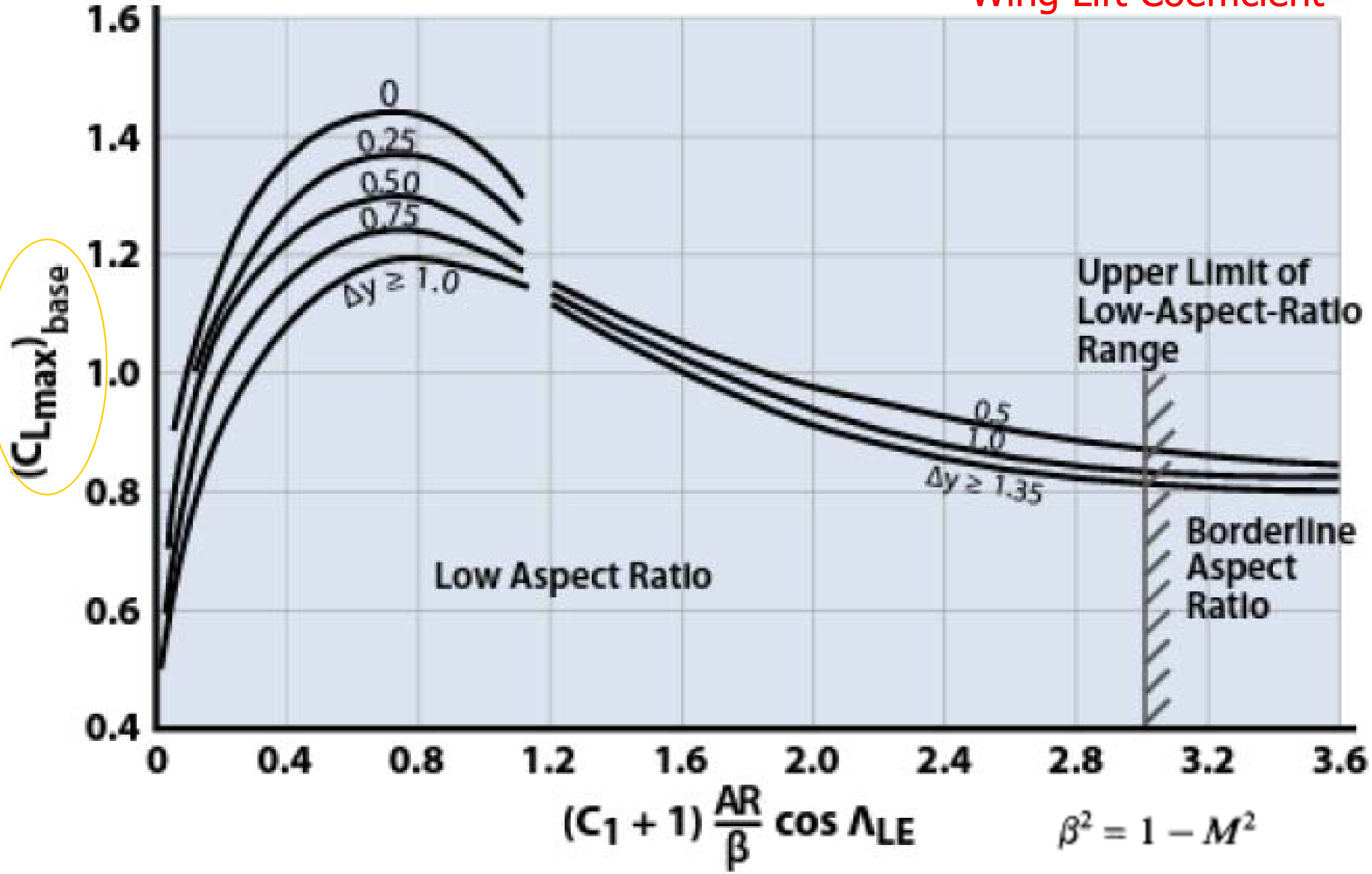


Figure 9.18 Subsonic maximum lift of low-AR wings (adapted [10]).

Cálculo C_{Lmax} – Low Aspect Ratio - II

Wing Lift Coefficient

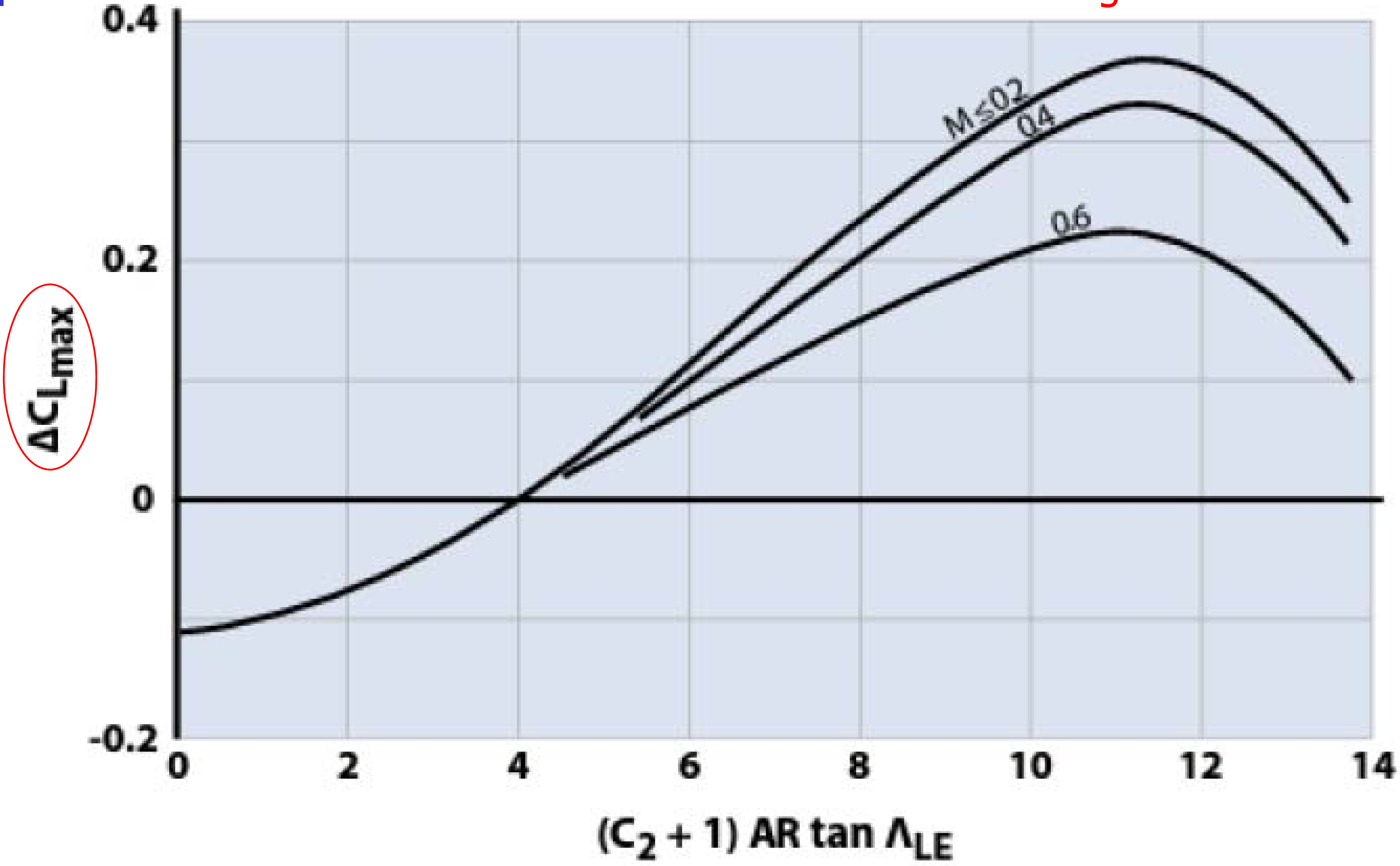


Figure 9.19 Subsonic maximum-lift increment for low-AR wings (adapted [10]).

Cálculo C_{Lmax} – Low Aspect Ratio - III

Wing Lift Coefficient

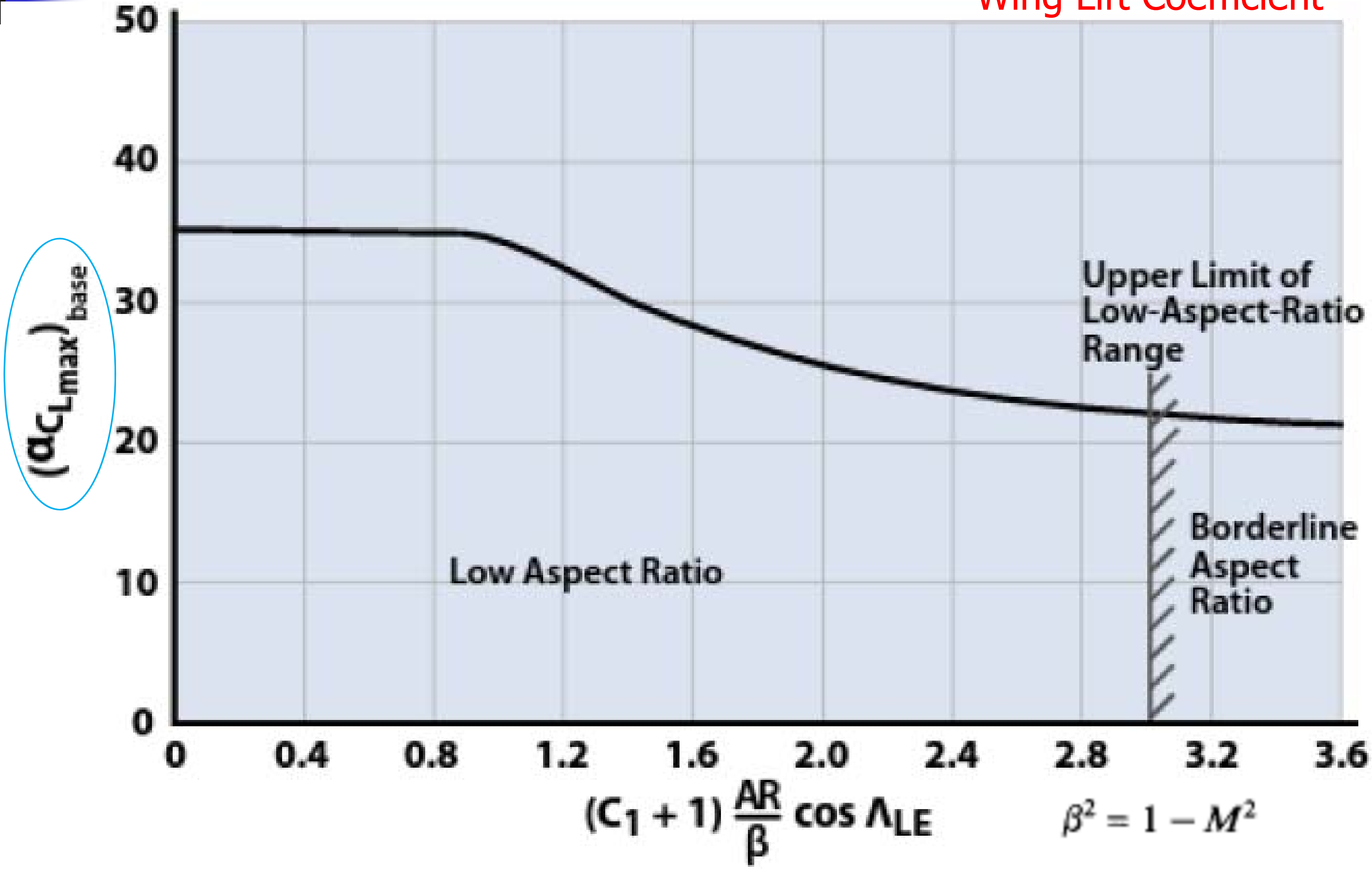


Figure 9.20 Angle-of-attack for subsonic maximum lift of low-AR wings.

Cálculo C_{Lmax} – Low Aspect Ratio - IV

Wing Lift Coefficient

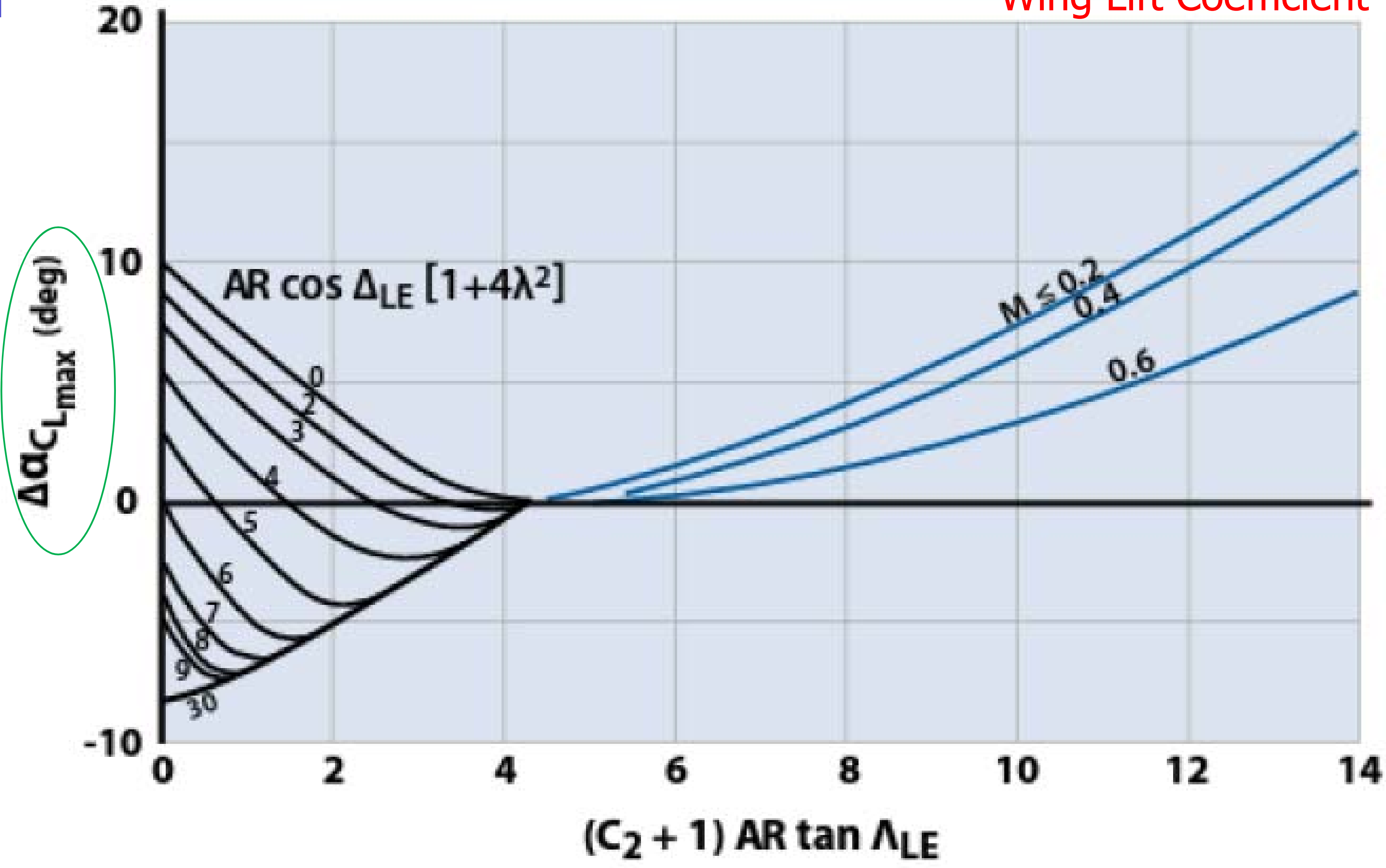


Figure 9.21 Angle-of-attack increment for subsonic maximum lift of low-AR wings.

Cálculo C_{Lmax} - IV

Wing Lift Coefficient with TE HLD

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

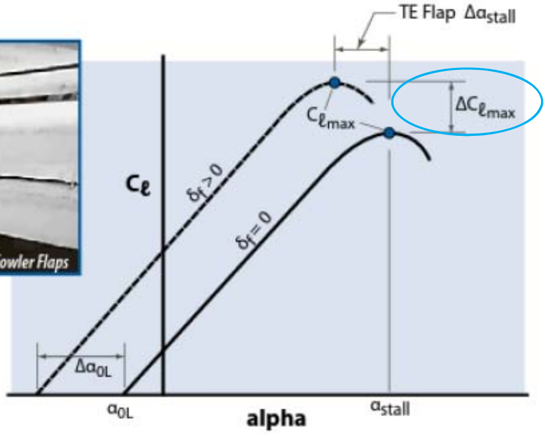


Figure 9.8 Construction of section lift curves for TE flaps.

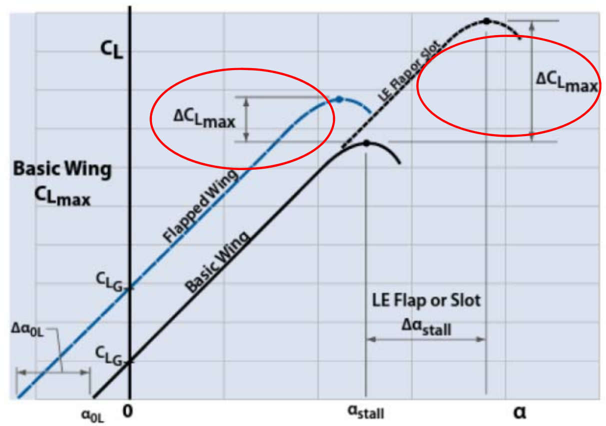


Figure 9.22 Construction of wing lift curves for mechanical high-lift devices.

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

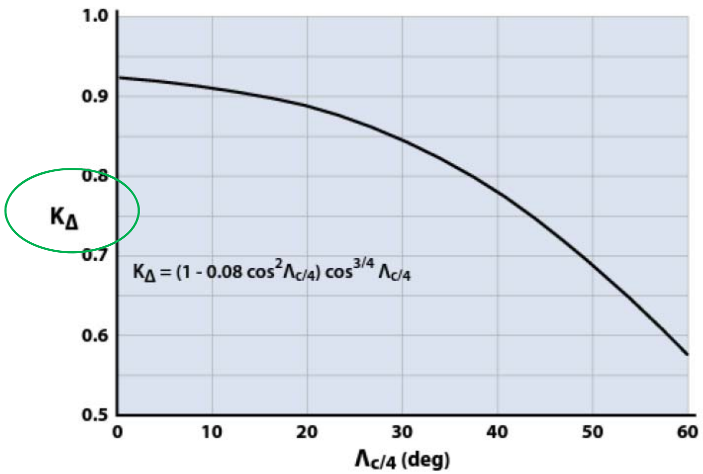


Figure 9.23 Planform correction factors for TE flaps (adapted [10]).

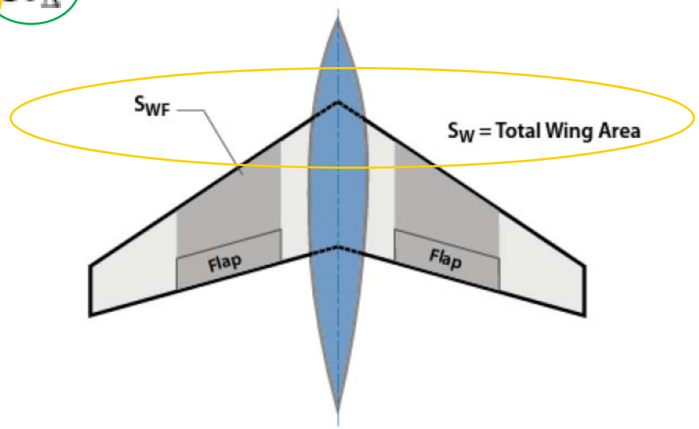


Figure 9.24 Schematic showing flapped wing area.

Cálculo C_{Lmax} - V

Wing Lift Coefficient with TE HLD

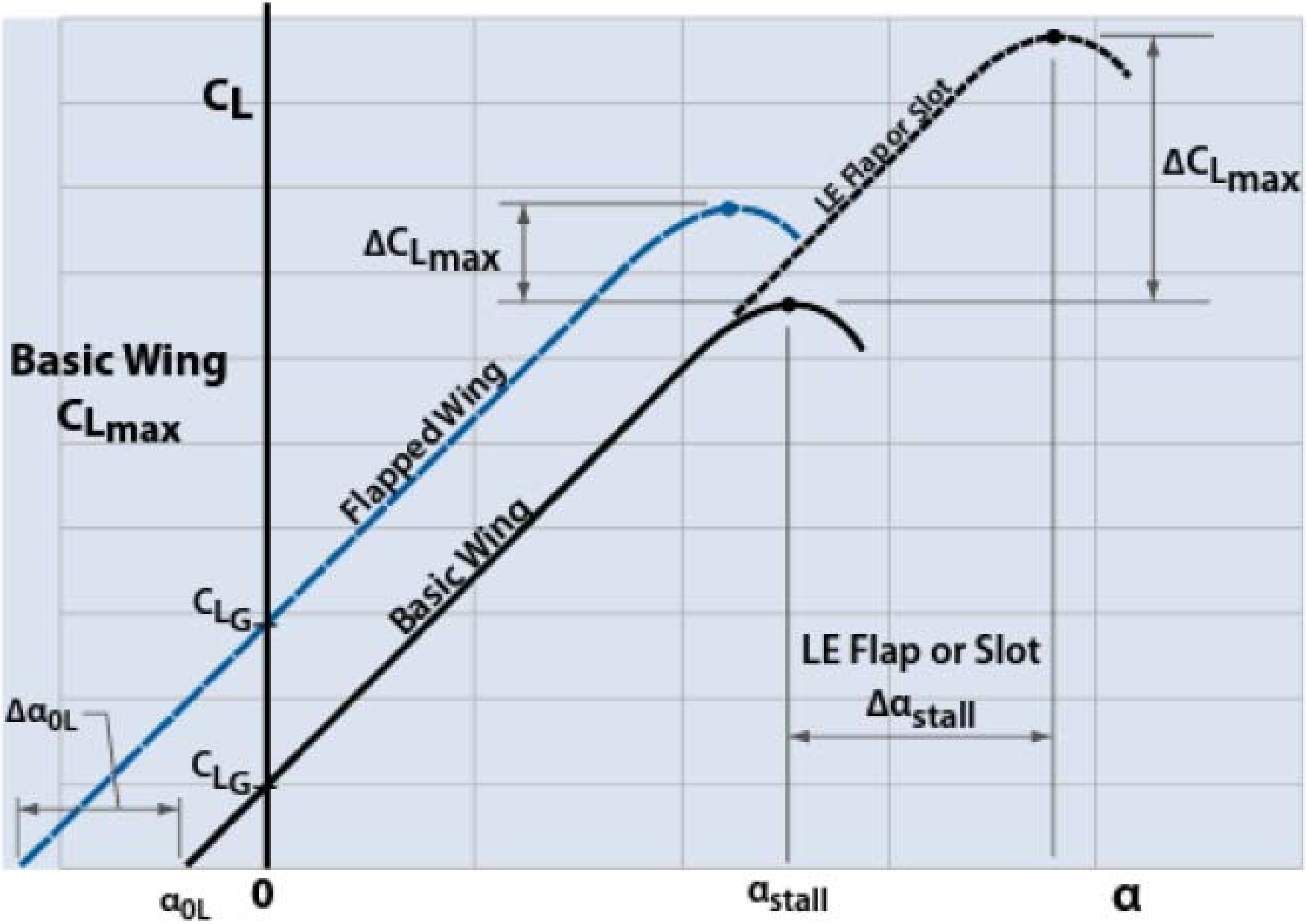


Figure 9.22 Construction of wing lift curves for mechanical high-lift devices.

Cálculo C_{Lmax} - VI

Wing Lift Coefficient with TE HLD

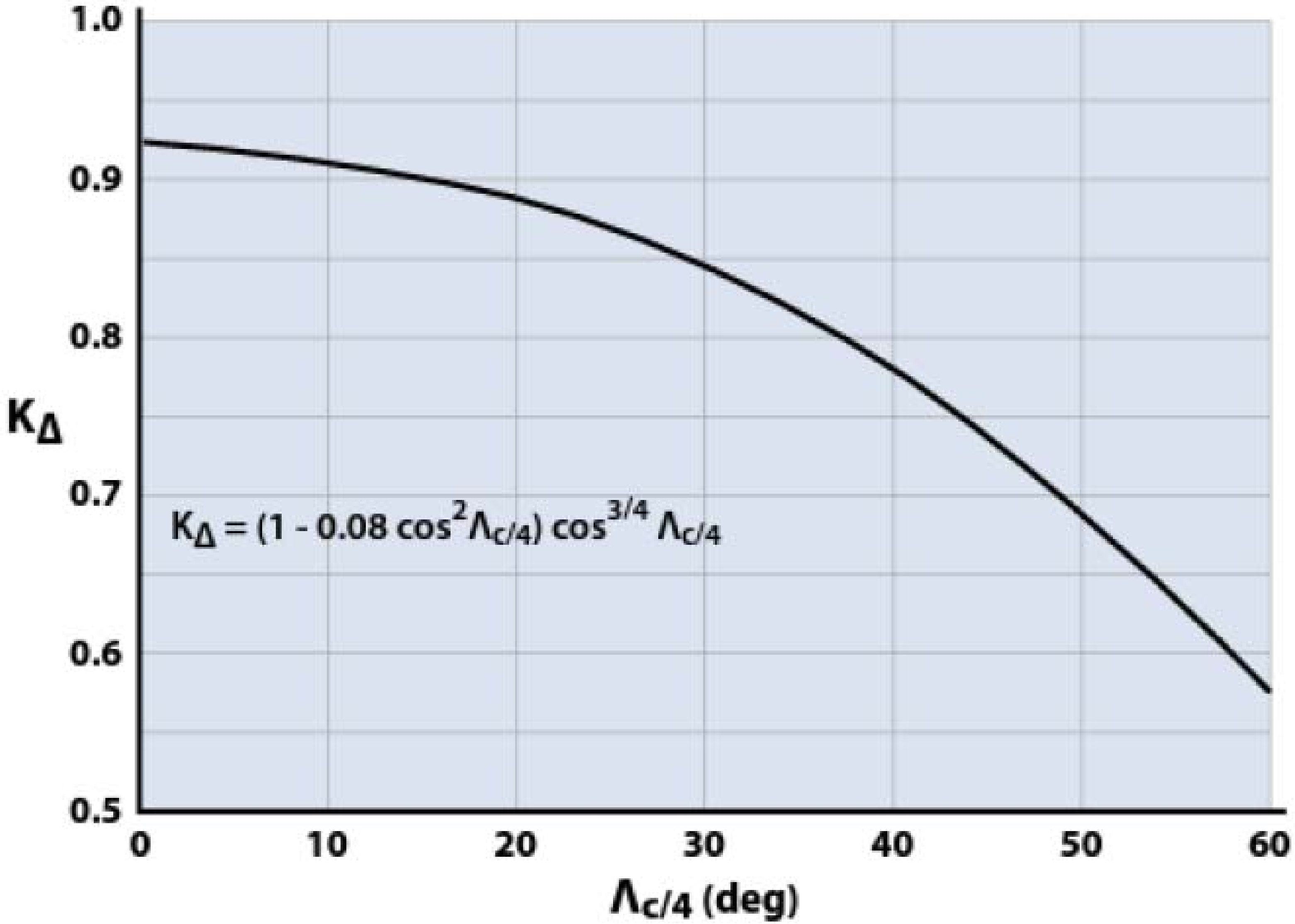


Figure 9.23 Planform correction factors for TE flaps (adapted [10]).

Cálculo C_{Lmax} - VII

Wing Lift Coefficient with TE HLD

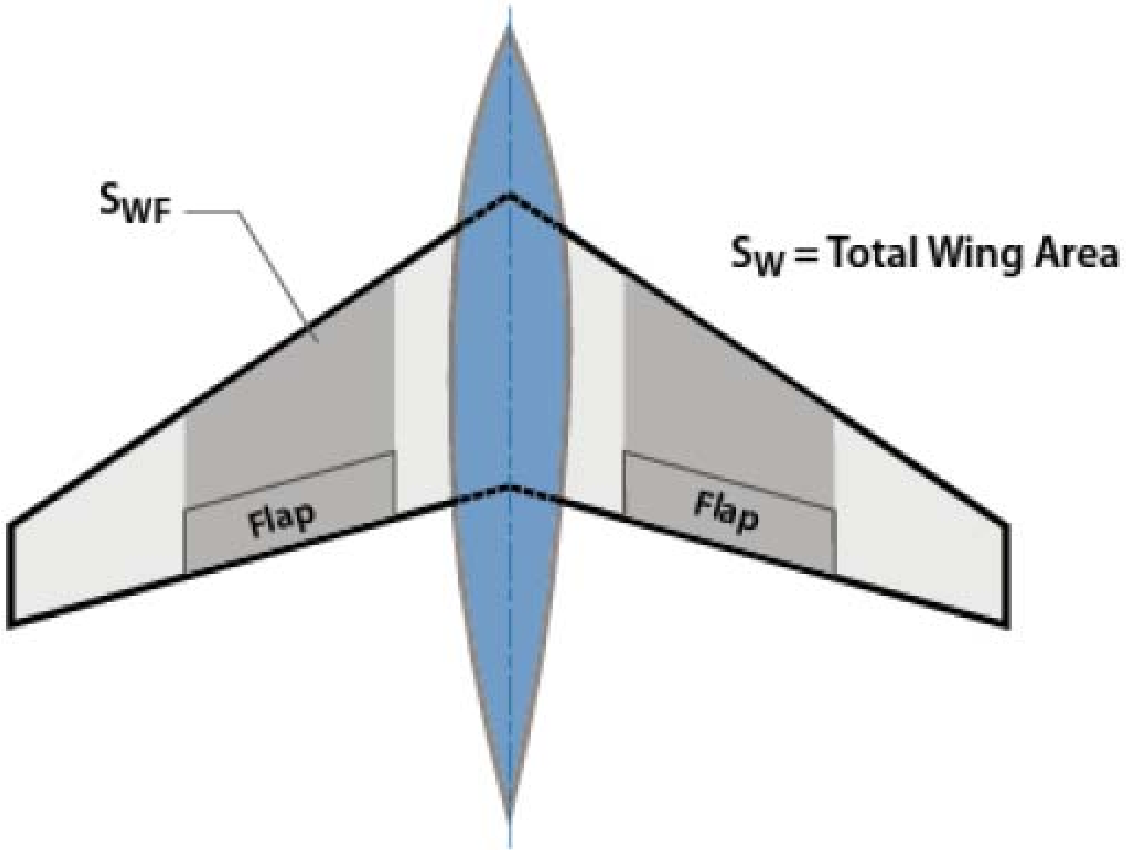


Figure 9.24 Schematic showing flapped wing area.

Cálculo C_{Lmax} - VIII

- Incremento de la sustentación

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

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

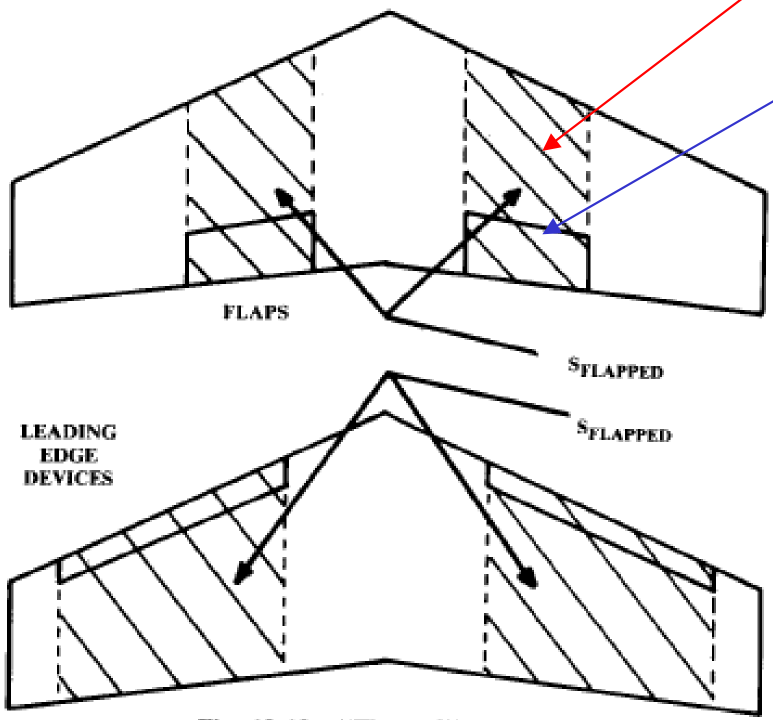


Fig. 12.19 "Flapped" wing area.

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 (ΔC_{Dp})
  - Aspereza
  - Protuberancias
  - Interferencias
    - Tren de aterrizaje.
    - Góndolas
    - Protuberancias.
  - Efectos 3-D
  - Efectos de compresibilidad (ΔC_{DM})
  - Inducido ($C_L^2/(\pi ARe)$)
- 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.

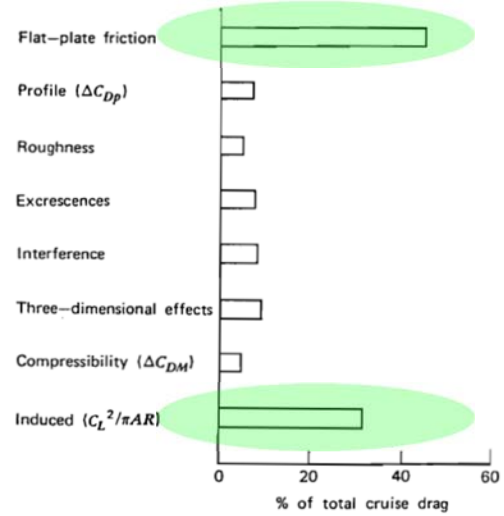


Figure 4.34 Typical drag buildup for jet transport.

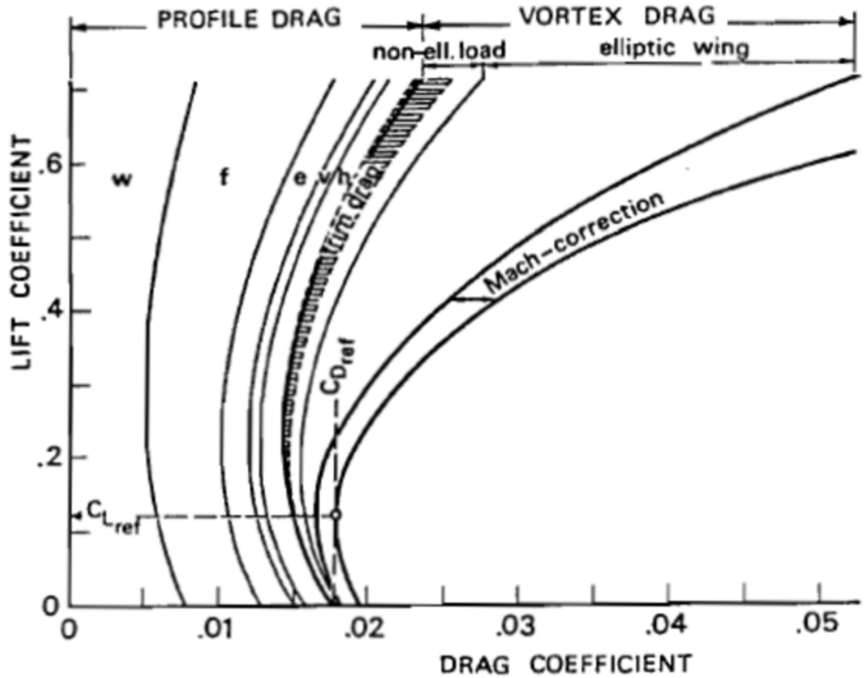


Fig. 11-2. Drag buildup by analysis (w = wing; f = fuselage; e = engine installation; v = vertical tailplane; h = horizontal tailplane)

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 α=0
      - $C_{Do} = C_{Dmin}$

$$C_D = C_{D_0} + KC_L^2 \Leftarrow K = \frac{1}{\pi A e}$$

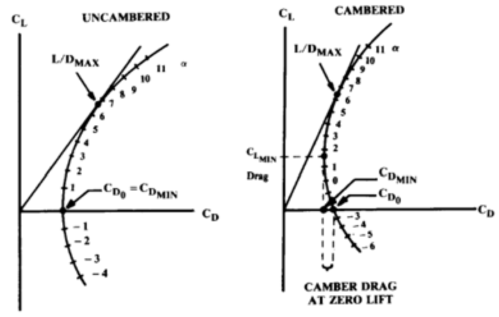


Fig. 12.3 Drag polar.

  - Alas con curvatura
    - Mínima resistencia α>0
      - $C_{Do} \neq C_{Dmin}$

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

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

- 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

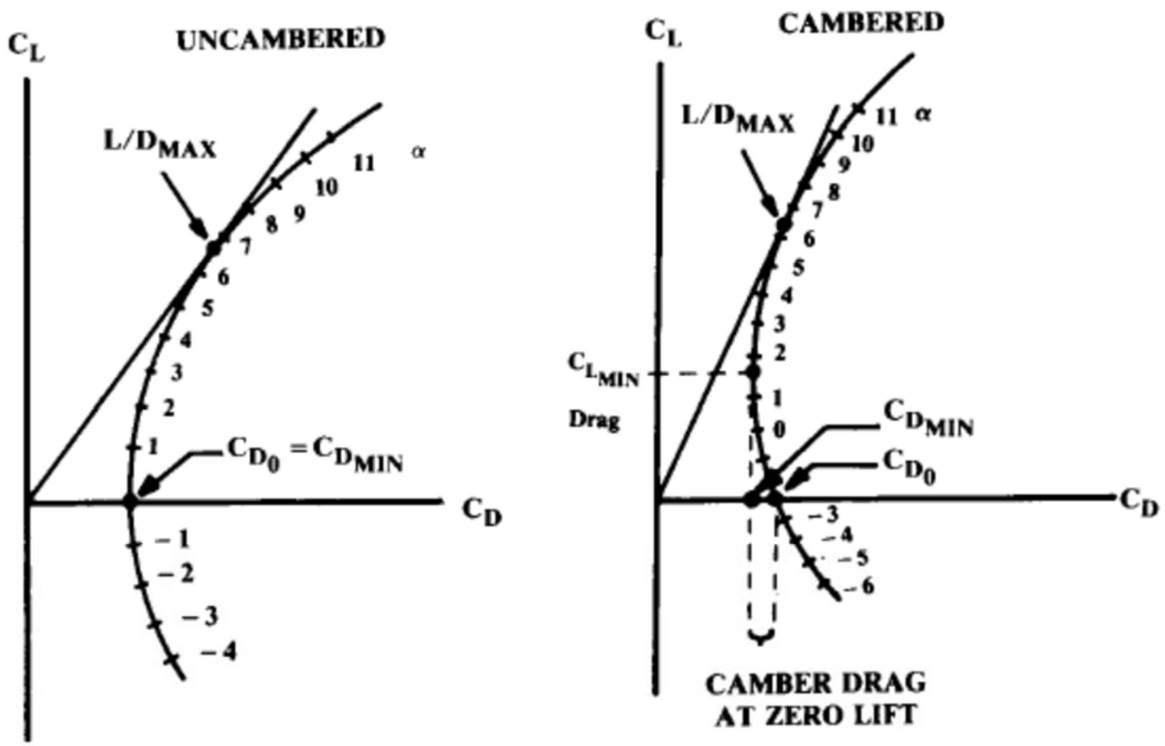
Coeficiente de Resistencia - C_D - III

- 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

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

$$\text{Straight-wing aircraft: } e = 1.78\left(1 - 0.045A^{0.68}\right) - 0.64$$

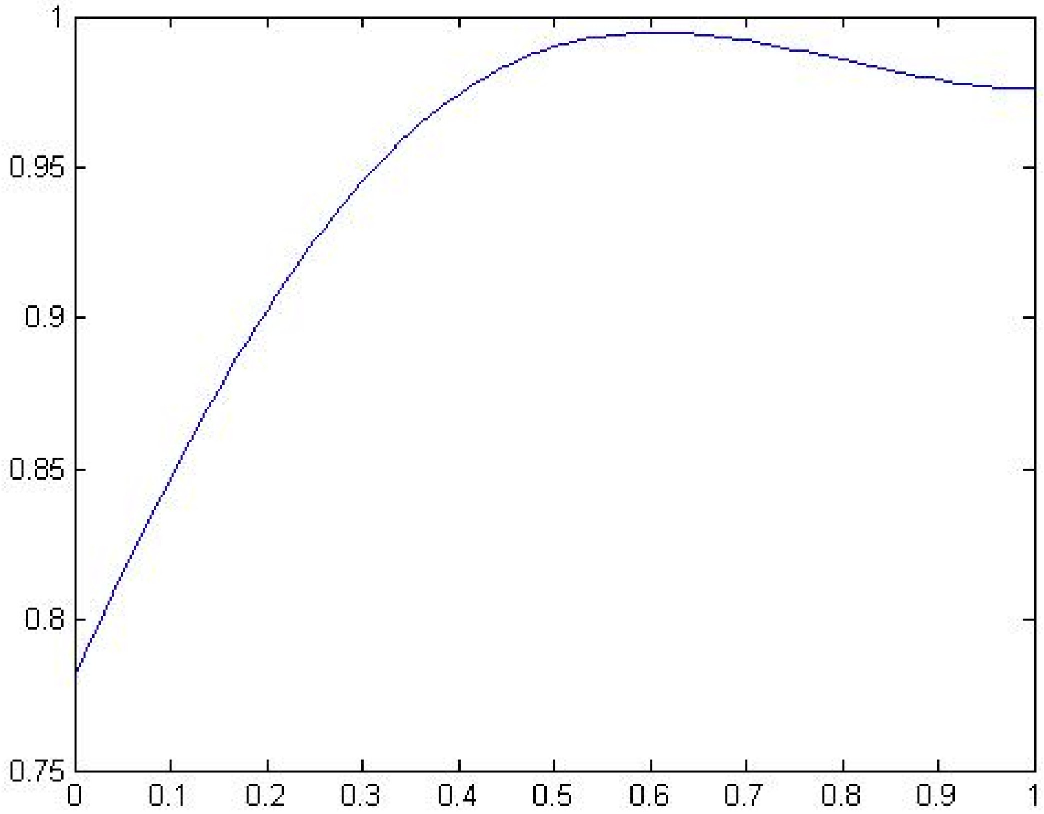
$$\text{Swept-wing aircraft: } e = 4.61\left(1 - 0.045A^{0.68}\right)(\cos\Lambda_{LE})^{0.15} - 3.1$$
$$(\Lambda_{LE} > 30°)$$



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

$$C_{D_0} = C_{f_e}\frac{S_{wet}}{S_{ref}}$$

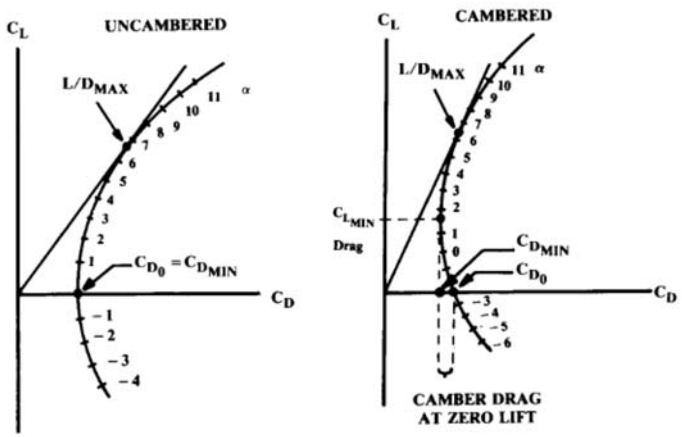


Fig. 12.3 Drag polar.

Table 12.3 Equivalent skin friction coefficients

$C_{D_0} = C_{fe}\frac{S_{wet}}{S_{ref}}$	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_{Do} – 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 está unidos.
  - **C_{Dmisc}** 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)

Form Factor

Factor de interferencia

Miscelaneos

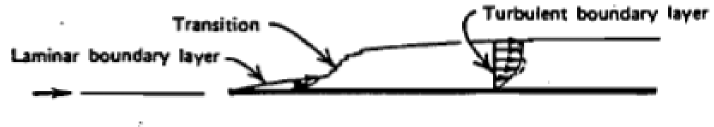
$$(C_{D_0})_{subsonic} = \frac{\Sigma\left(C_{f_c} \cdot FF_c \cdot Q_c \cdot S_{wet_c}\right)}{S_{ref}} + C_{D_{misc}} + C_{D_{L\&P}}$$

Leakages and protuberances

Coeficiente de fricción

Estimación C_{Do} – 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**.



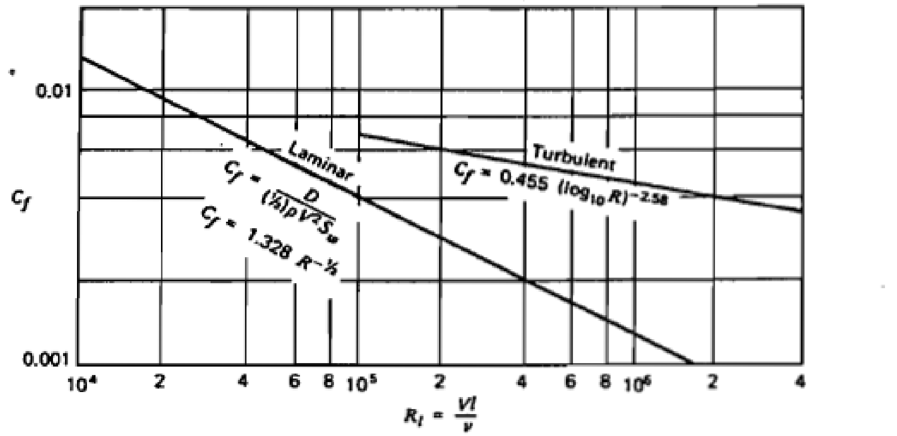


Table 4.3 Typical Total Skin Friction Coefficient Values for Different Airplane Configurations

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	3.33×10^{-5}
Smooth paint	2.08×10^{-5}
Production sheet metal	1.33×10^{-5}
Polished sheet metal	0.50×10^{-5}
Smooth molded composite	0.17×10^{-5}

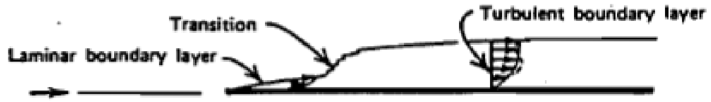
Estimación C_{Do} – IV

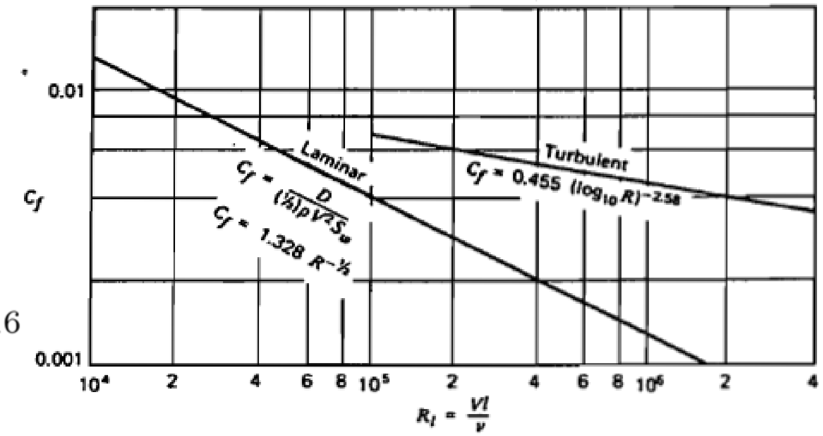
- El Flat Plate skin friction coefficient:
  - Laminar

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

  - Turbulento

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





- 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

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

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

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

Table 12.4 Skin roughness value (k)

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

- 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

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

Estimación C_{Do} – 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

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

Flecha en el máximo espesor

Fuselage and smooth canopy

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

Nacelle and smooth external store

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

⇨ l = longitud característica
d = diámetro
A_{max} = max cross sectional area

- 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_{Do} – 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 ~ Q=1.5
    - Si la góndola esta separada a menos de 1 diámetro de la góndola ~ Q =1.3
    - Si la góndola esta separada mas de 1 diámetro de la góndola ~ 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 ~1.4
  - Fuselaje ≈ 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_{Do} – VII

- Resistencias Miscelaneas:
  - Corrección para fuselajes con upsweep

Drag-Area

Máxima sección de área Del fuselaje

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

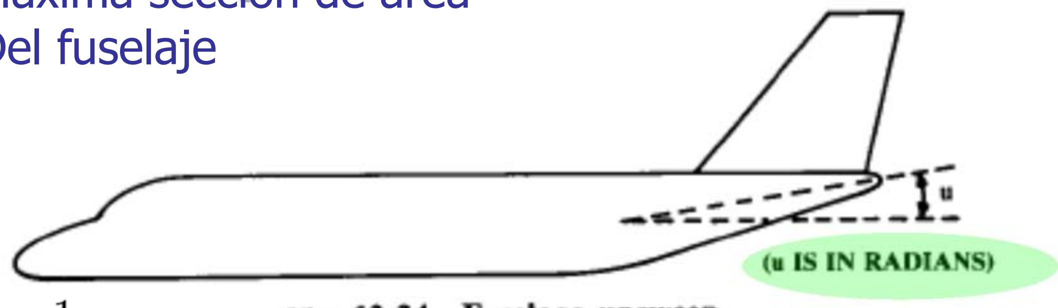


Fig. 12.24 Fuselage upsweep.

$$\frac{D}{q_{upsweep}} = 3.83u^{2.5}A_{max} \Rightarrow C_{D_0} = \frac{D}{q_{upsweep}} \cdot \frac{1}{S_{ref}}$$

[(D/q)/Frontal area] ⇨ Multiplicado por $S_{frontal}/S_{ref}$ En ft^2!!!!!

$$\frac{D/q}{\text{Frontal Area}} \cdot \frac{S_{frontal}}{S_{ref}} = C_{D_0}$$

  - Tren de aterrizaje

Table 12.5 Landing gear component drags

	D/q Frontal area (Ft^2)
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

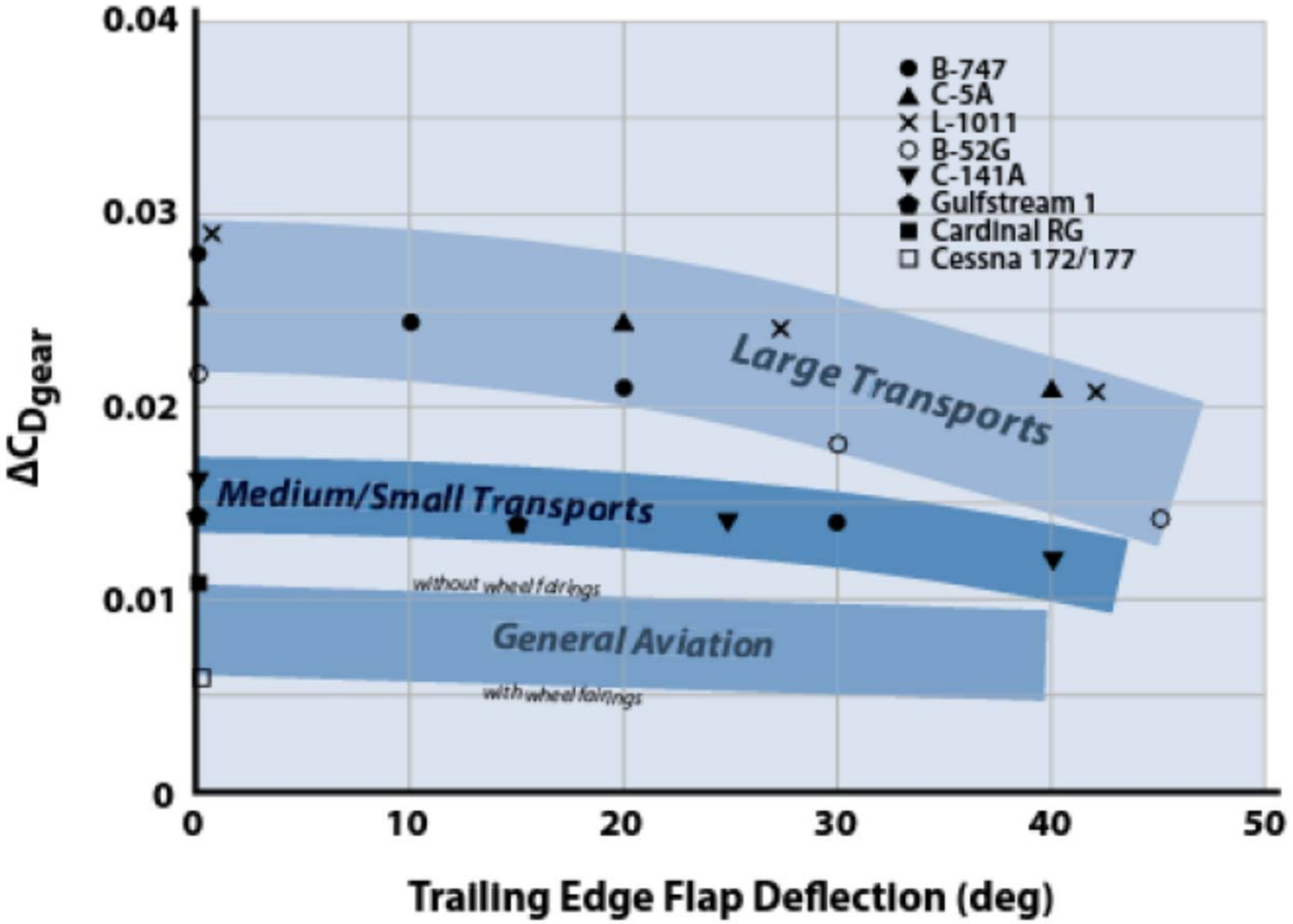


Figure 10.5 Drag of landing gear.

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^2)	$\Delta C_{D_{gear}}$	Landing Gear Configuration[a]
Fighters			
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 pogos
Large transports			
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[b]	One-wheel NLG, two one-wheel MLG
Cessna 177	174	0.006[b]	One-wheel NLG, two one-wheel MLG
Cardinal RG	174	0.011	One-wheel NLG, two one-wheel MLG

[a]Abbreviations: NLG, nose landing gear; MLG, main landing gear.
[b]Fixed landing gear with wheel fairings.

Estimación C_{Do} – VII - cont

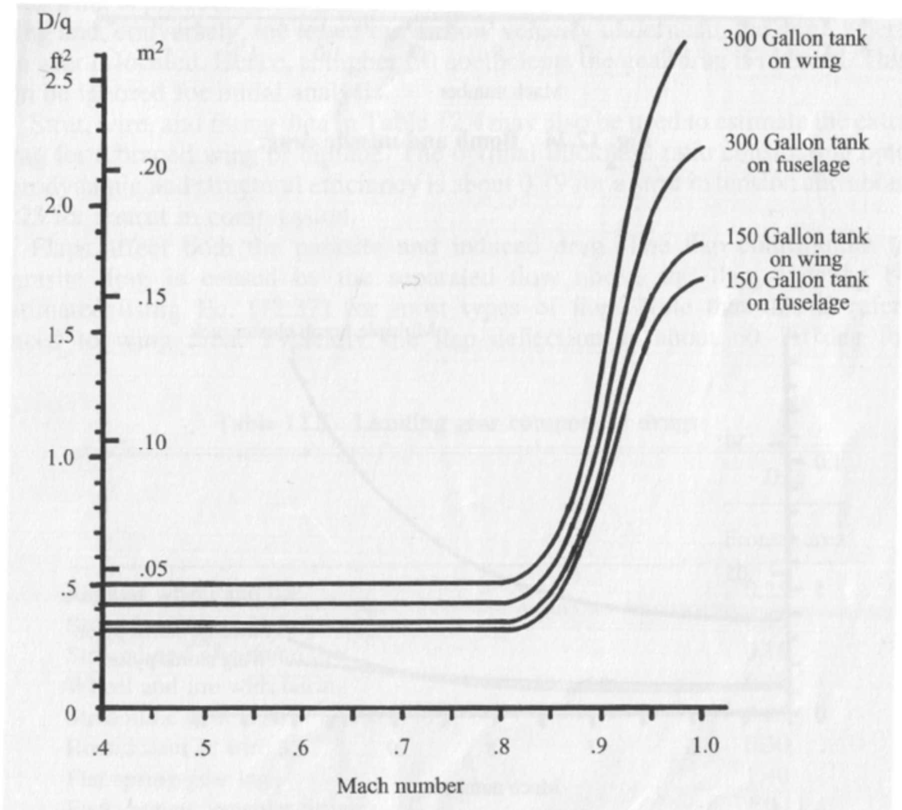


Fig. 12.23 External stores (fuel tanks) drag.

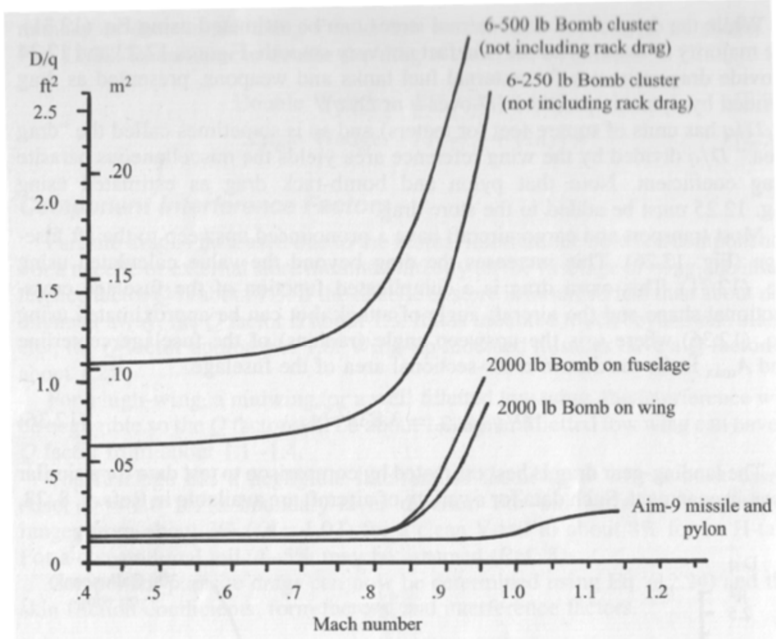


Fig. 12.24 Bomb and missile drag.

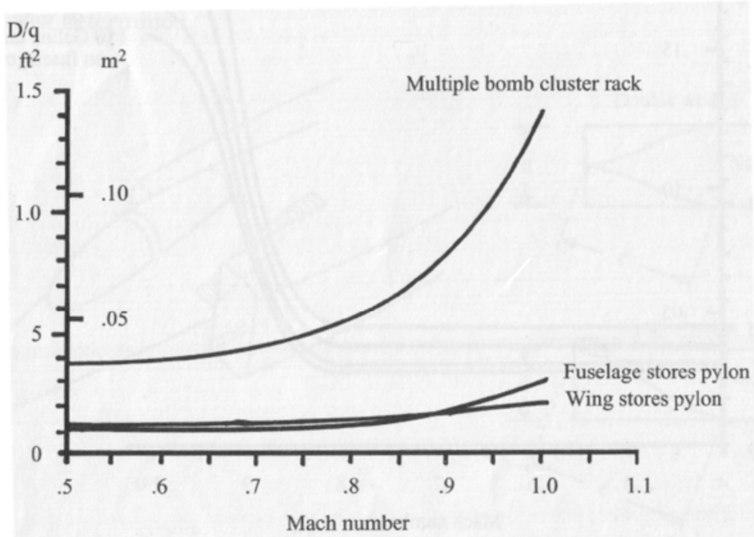


Fig. 12.25 Pylon and bomb rack drag.

Estimación C_{Do} – 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.

$\%C_{Do}$ ⇨ $$(C_{D_0})_{subsonic} = \frac{\Sigma\left(C_{f_c} \cdot FF_c \cdot Q_c \cdot S_{wet_c}\right)}{S_{ref}} + C_{D_{misc}} + C_{D_{L\&P}}$$

Estimación C_{Do} – IX

Drag to Flap deflection

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

$$\Delta C_{D_{0_{flap}}} = F_{flap}\left(\frac{C_f}{C}\right)\left(\frac{S_{flapped}}{S_{ref}}\right)(\delta_{flap} - 10)$$

$$\delta_{flap} \Rightarrow \text{in degrees}$$

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

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

$$C_f = \text{chord length of flap}$$

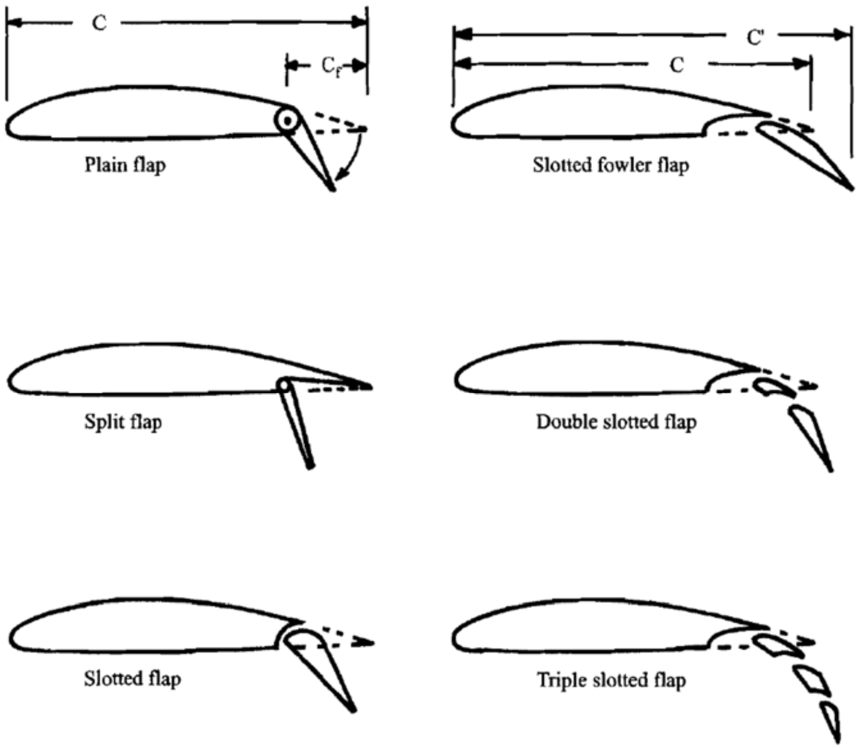


Fig. 12.17 Flap types.

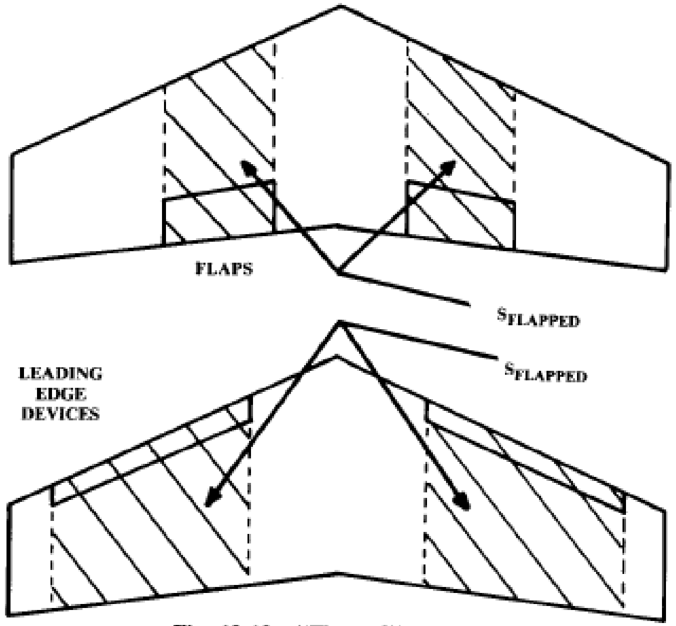


Fig. 12.19 "Flapped" wing area.

Estimación C_{Do} – X

Drag to Flap deflection

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

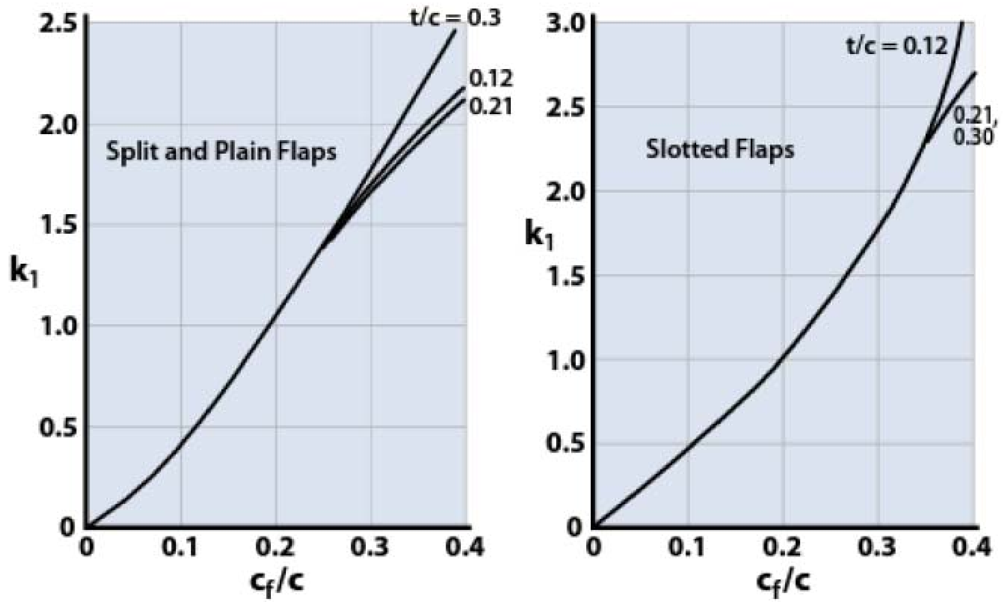


Figure 9.26 Factor k_1 to calculate drag increment due to flaps (data from [13]).

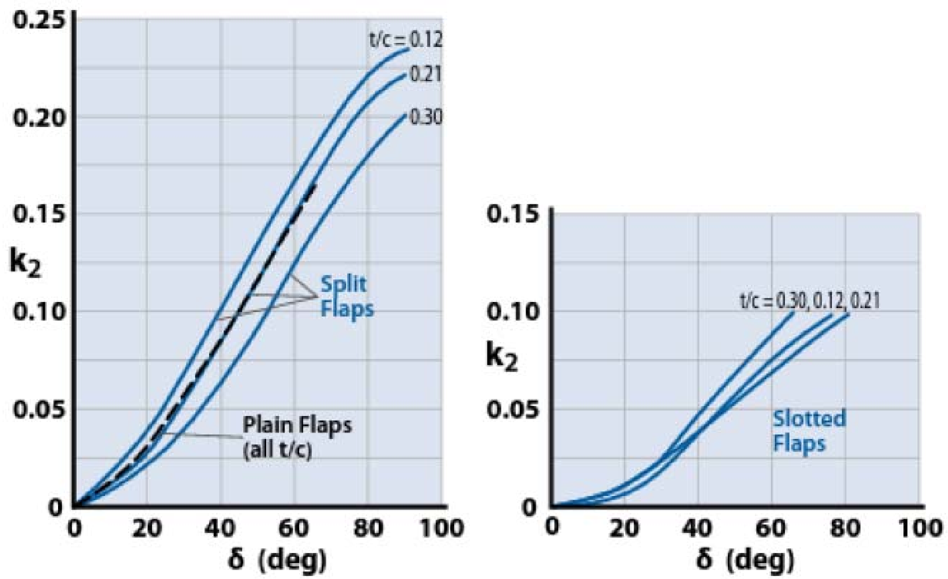


Figure 9.27 Factor k_2 to calculate drag increment due to flaps.

Estimación C_{Do} – XI

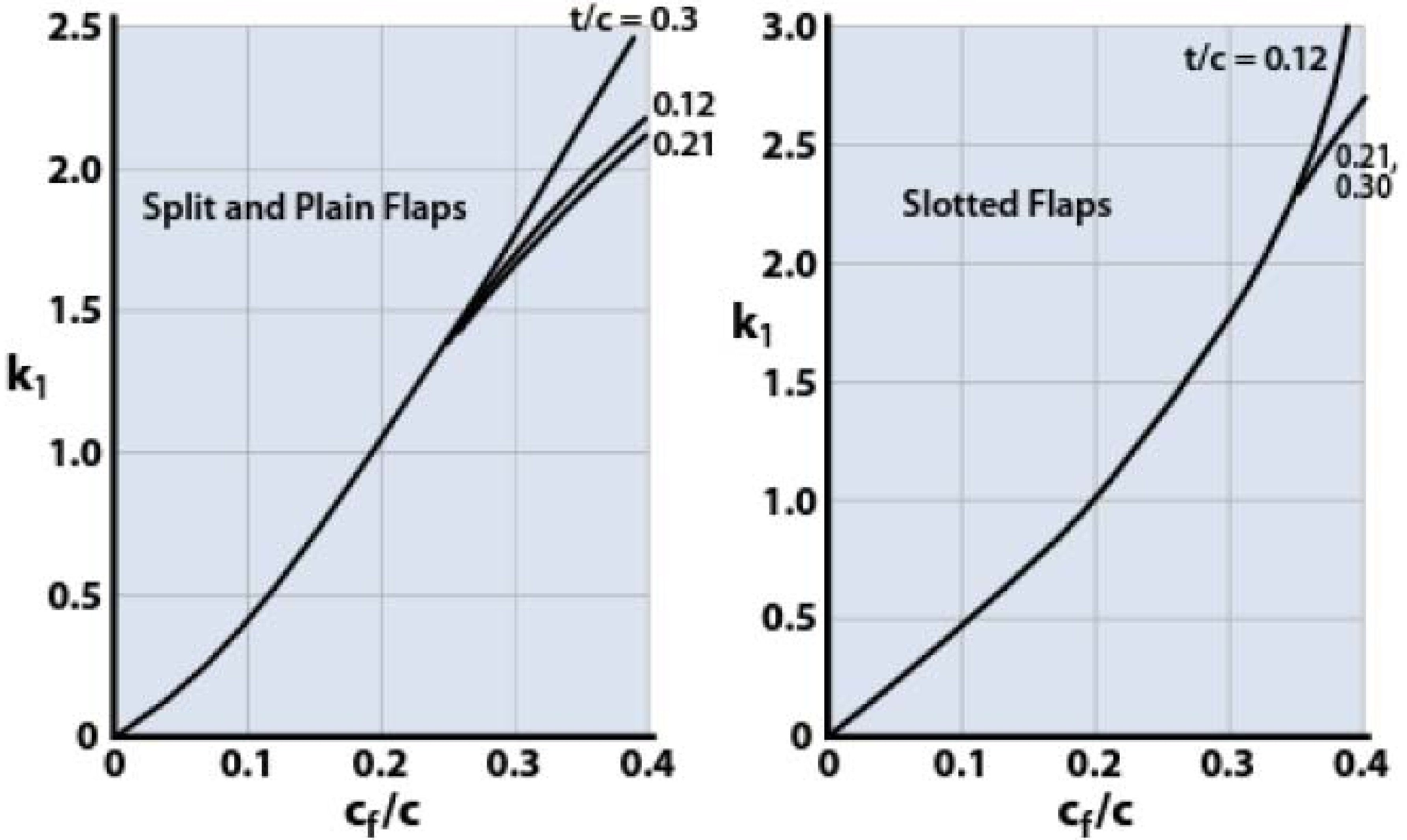


Figure 9.26 Factor k_1 to calculate drag increment due to flaps (data from [13]).

Estimación C_{Do} – XII

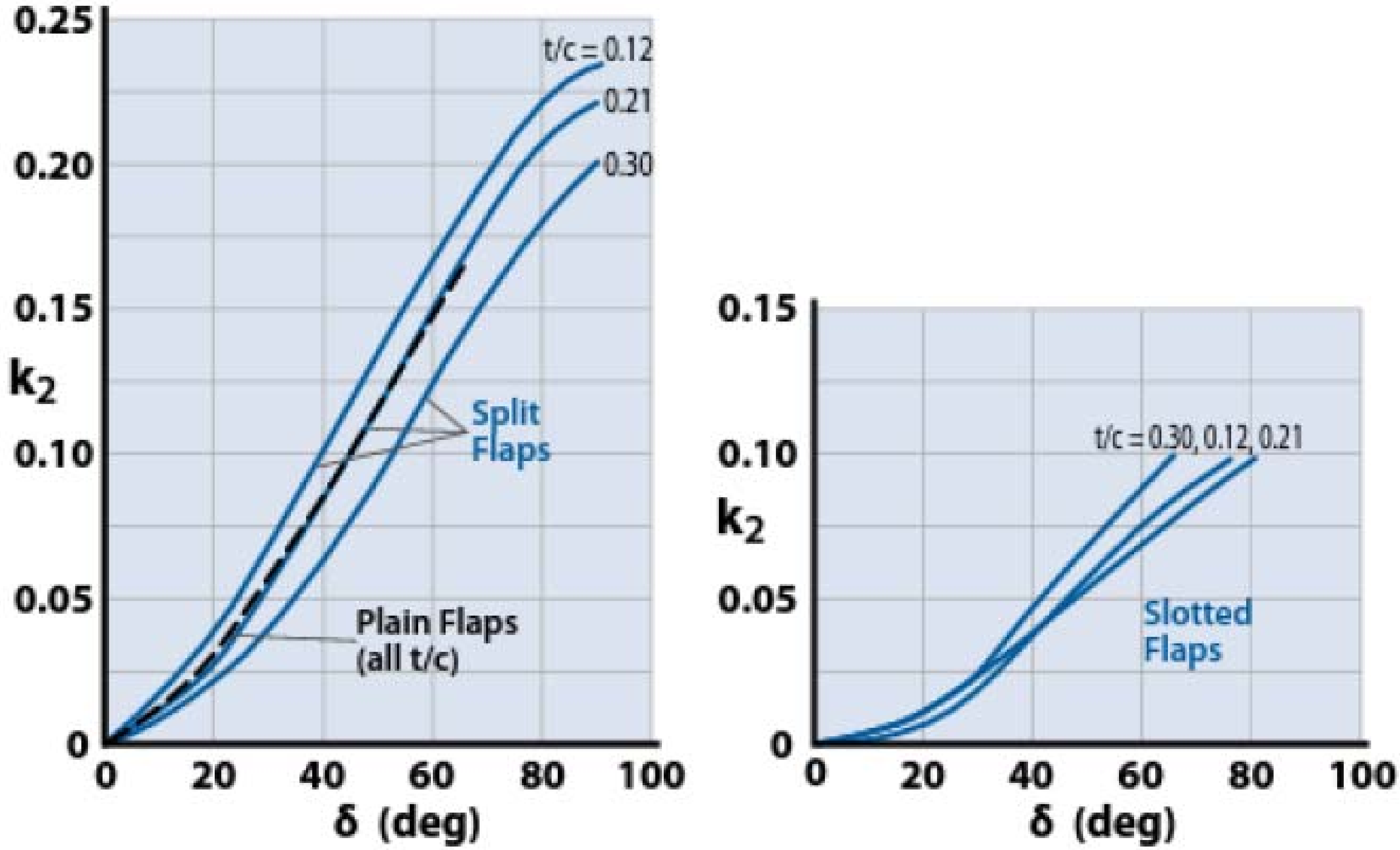


Figure 9.27 Factor k_2 to calculate drag increment due to flaps.

Estimación C_{Do} – XIII

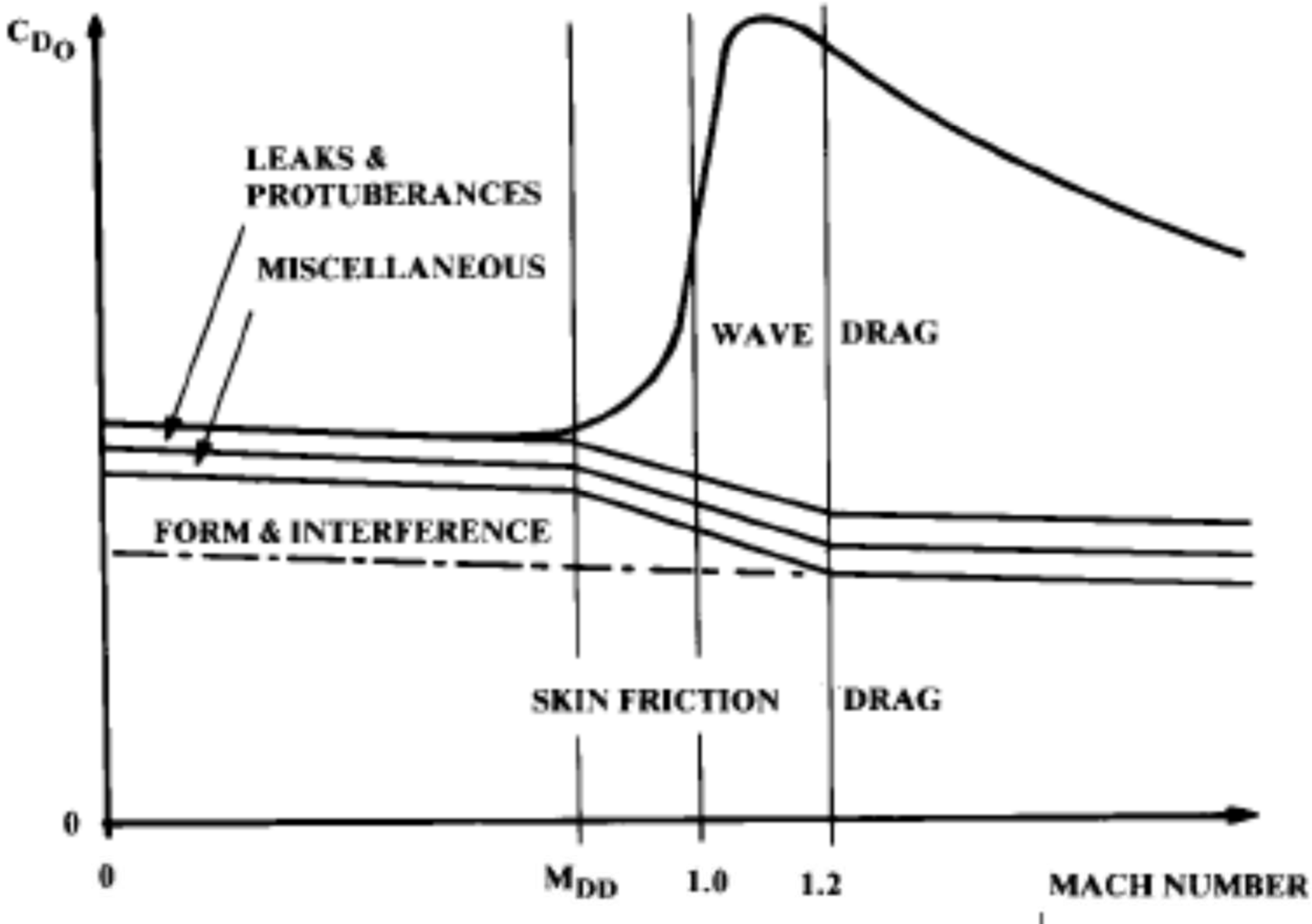


Fig. 12.30 Complete parasite drag vs Mach number.

Eficiencia Aerodinámica - I

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

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

$$k = \frac{1}{\pi ARe}$$

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

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

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

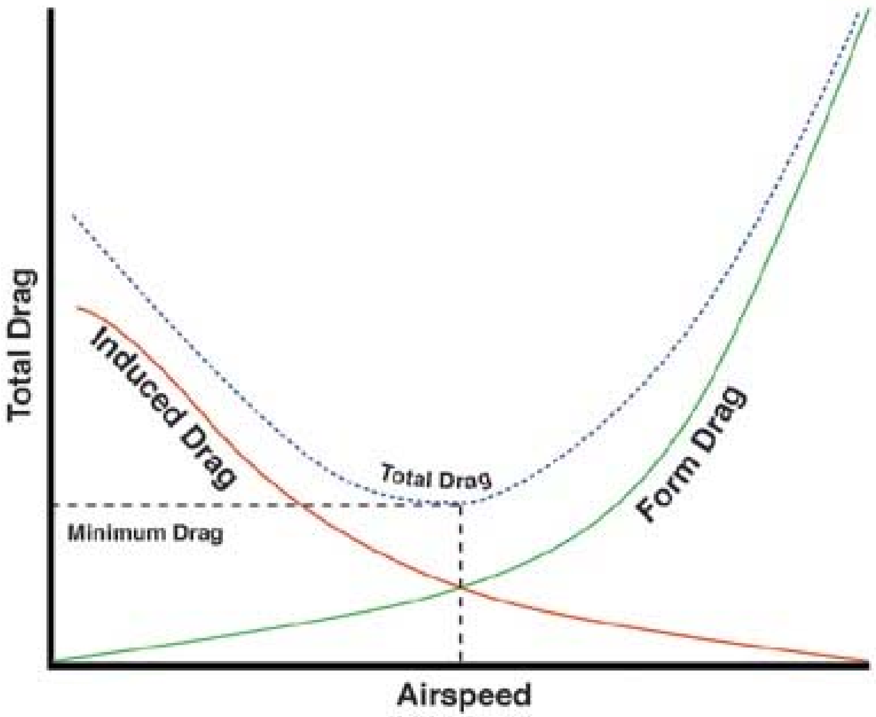
$$E = \frac{C_L}{C_D} = \frac{C_L}{C_{D_0} + kC_L^2} \Longrightarrow E_{opt} = \frac{C_{L_{opt}}}{C_{D_0} + kC_{L^2_{opt}}} = \frac{1}{2\sqrt{kC_{D_0}}}$$

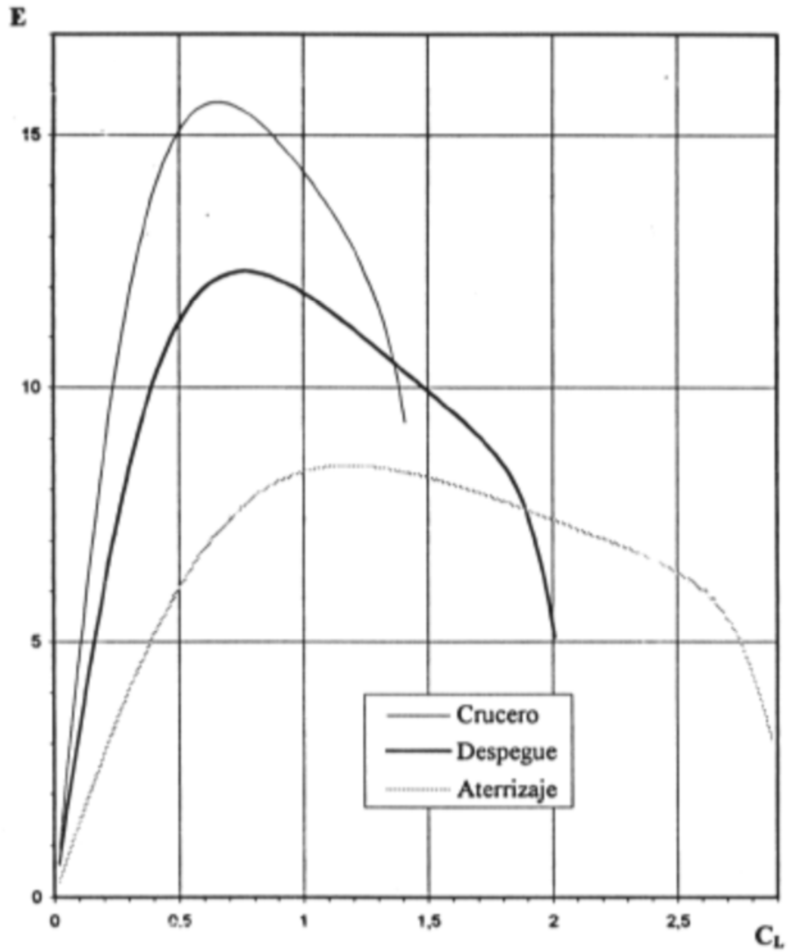
$$\begin{aligned} C_D &= C_{D_{min}} + K\left(C_L - C_{L_{min\ drag}}\right)^2 \\ &= C_{D_{min}} + KC^2_{L_{min\ drag}} + KC_L^2 - 2KC_{L_{min\ drag}}C_L \\ &= C_{D_o} + k_1C_L^2 - k_2C_L \end{aligned}$$

¿¿¿¿$C_{L_{opt}}$????

Eficiencia Aerodinámica - II

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





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 extensión):
- Reducen la resistencia inducida, lo que aumenta la eficiencia aerodinámica y un momento flector

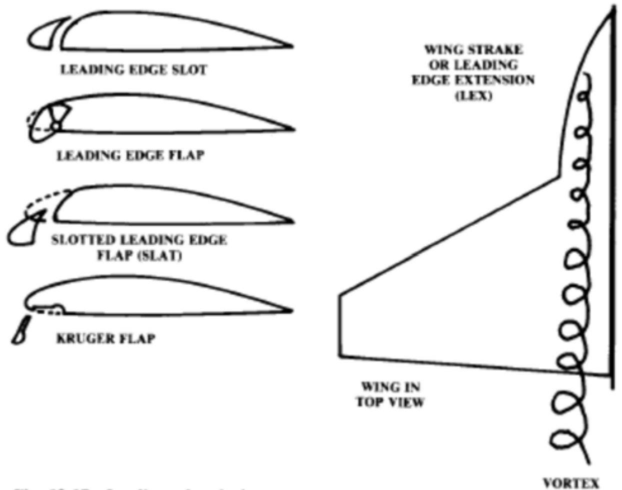


Fig. 12.17 Leading edge devices.

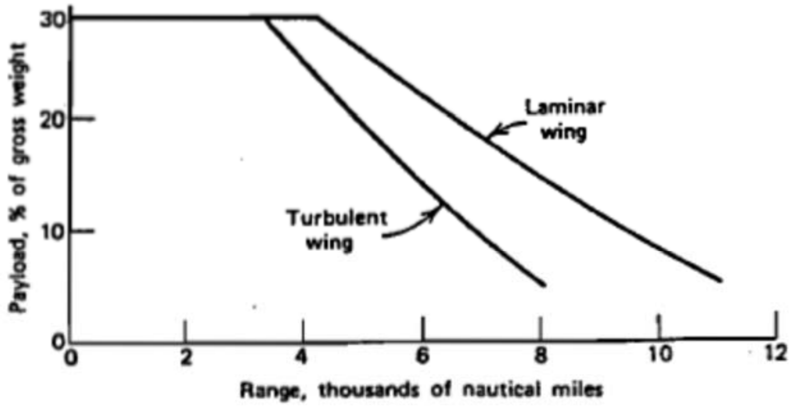


Figure 4.50 Performance gains from laminar flow control. (R. E. Kosin, "Laminar Flow Control by Suction as Applied to X-21A Airplane". AIAA *Journal of Aircraft*, 1965.

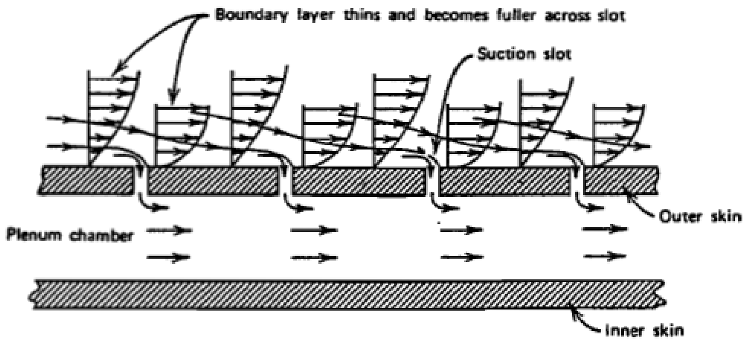


Figure 4.49 Laminar flow control by suction through thin slots transverse to the flow.

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