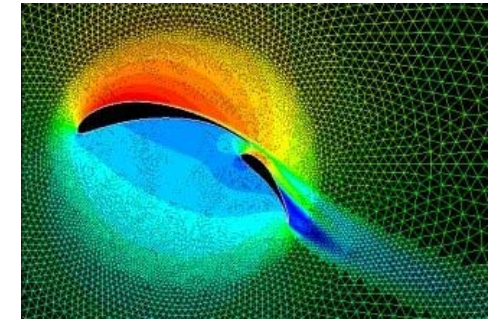


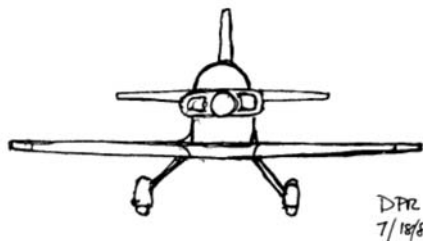
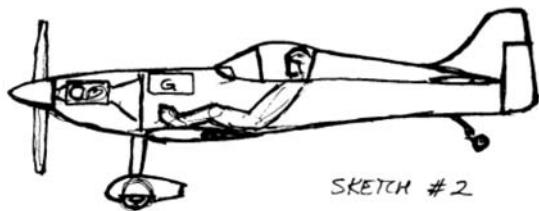
Fig. 4.22 Effect of taper on lift distribution.



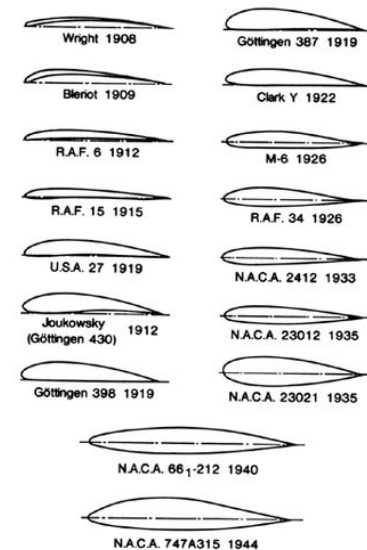
# Ejemplo de Diseño: Single Seat Aerobatic

## Tema 8

Sergio Esteban Roncero  
 Departamento de Ingeniería Aeroespacial  
 Y Mecánica de Fluidos



With a View to Practical Solutions



The historical evolution of airfoil sections, 1908-1944. The last two shapes (N.A.C.A. 66\_1-212 and N.A.C.A. 747A315) are low-drag sections designed to have laminar flow over 60 to 70 percent of chord on both the upper and the lower surface. Note that the laminar flow sections are thickest near the center of their chords.



# Outline

---

- Introducción – RFP.
- Aviones Semejantes.
- Selección geometría.
- Introducción a herramientas (Software).

# RFP

- Avión acrobático con mejores prestaciones que las del "homebuilt" *Great Lakes Biplane*, pero con "handling qualities peiores que el *Pitts Special*.
  - También se compara con el Stevens Acro
- Diseño clásico pero aprovechando las nuevas tecnologías que utilizan compuestos para simplificar y reducir costes de construcción (fibra de vidrio, espuma, ...)
  - Diseño que permita la rápida fabricación en el garaje.
- Utilización de un motor ya preparado para vuelo acrobático: Lycoming 0.320-A2B



# GREAT LAKES 2T-1A2

## DIMENSIONS AND PERFORMANCE

- **Engine:** Lycoming AEIO-360-B1G6.
- **Fuel cap:** 27 gal.
- **Horsepower:** 180. Propeller: Hartzell C/S.
- **Gross Wt:** 1800 lbs.
- **Empty Wt:** 1230 lbs.
- **Payload:** 384 lbs.
- **Wingspan:** 26'8"
- **Height:** 7'8"
- **Overall length:** 21'2"
- **Wing area:** 187.6 sq. ft.
- **Wheel track:** 5'10"
- **Wing load:** 9.6 lbs/sq. ft.
- **Cruise speed:** (75%): 102 kt
- **Cabin width:** 24"
- **Endurance:** 2.2 hs.
- **Best rate of climb:** 1150 fpm
- **Stall speed:** 50 K.
- **Service ceiling:** 17000 ft
- **Takeoff roll:** 475 ft
- **Landing roll:** 400 ft.

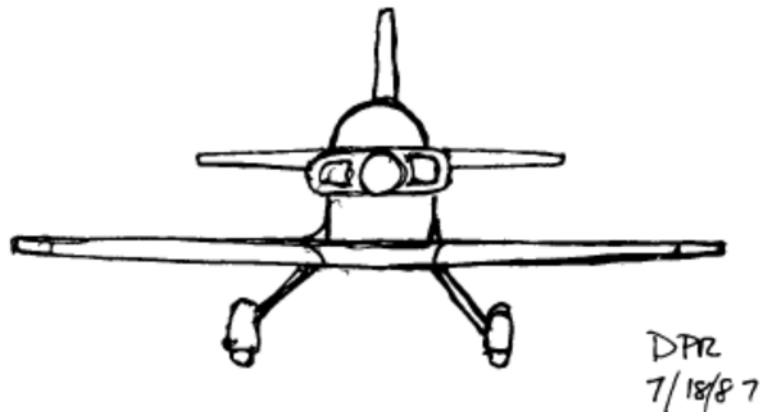
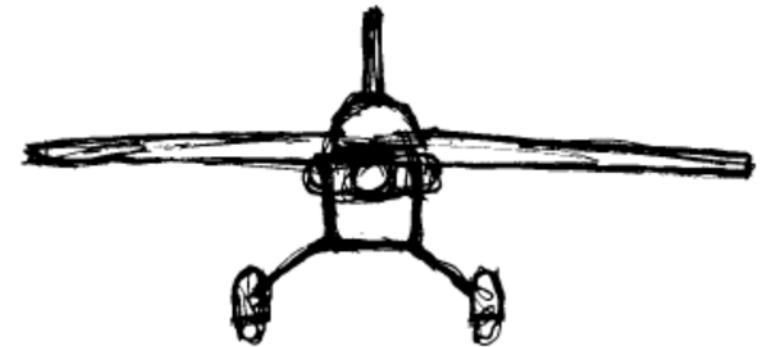
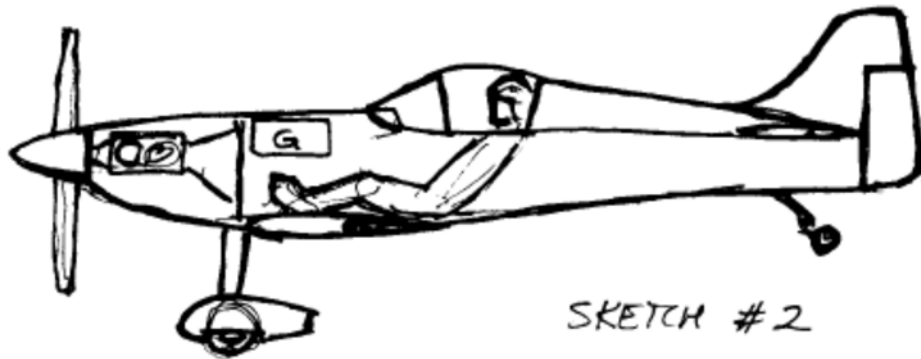


# Metas de Diseño

- Fabricación rápida:
  - Empleo de fibra de vidrio y espuma
- Actuaciones definidas teniendo en cuenta las actuaciones de tanto Pitts S-15 y Great Planes:
  - $V_{\max} \geq 130$  knts
  - $V_{\text{stall}} \leq 50$  knts
  - Takeoff  $\leq 1000$  ft over 50'
  - Rate of Climb S.L.  $\geq 1500$  fpm
  - Range  $\geq 280$  nm (sin reservas)
    - $V_{\text{cruise}} = 115$  knts
  - $n = +9/-6$  g's
  - $W_{\text{crew}} = 220$  (incluyendo paracaídas)
    - Opcional cabina abierta (utilizado para el análisis)
- Cualidades de vuelo:
  - Ligeramente estable para poder maniobrar.
  - Buena respuesta para recuperar del tirabuzón, tanto en vuelo normal como inverso
- Misión:
  - Despeque
  - Subida a 8000 ft
  - Crucero 280 nm con  $V_{\text{cruise}} = 115$  knts
  - Descenso
  - Aterrizaje

# Diseño conceptual - Servilleta

DPR  
7/17/87



# Selección de la Geometría del Ala - 1

- Utilizando la información en el capítulo 4 de Raymer (gráficas y tablas) se determina que los primeros valores para la geometría del ala sean:
  - $AR = 6$
  - $\lambda = 0.4$
  - $\Gamma = 3^\circ$ . ( $0^\circ$  efectivos)
  - Perfil:
    - NACA 63<sub>2</sub>015 (perfil de punta)
    - NACA 63<sub>2</sub>012 (perfil de raíz)
    - t/c más elevado en la punta que en la raíz ayuda a prevenir la entrada en pérdida de la punta del ala
  - Perfil sin torsión para evitar que en vuelo invertido se produzca entrada en pérdida de la punta del ala.
- Geometría de la deriva horizontal:
  - $AR = 4$
  - $\lambda = 0.4$
  - Perfil: NACA 0012

# Selección de la Geometría del Ala - 2

- Geometría de la deriva vertical:
  - $AR = 1.5$
  - $\lambda = 0.4$
  - Perfil: NACA 0012
- Notas
  - La torsión revisa la distribución de la sustentación intentado aproximarla a una distribución elíptica.
    - Torsión geométrica.
    - Torsión aerodinámica.
  - Información sobre perfiles:
    - <http://www.ae.uiuc.edu/m-selig/ads/aircraft.html>



# Selección Alargamiento

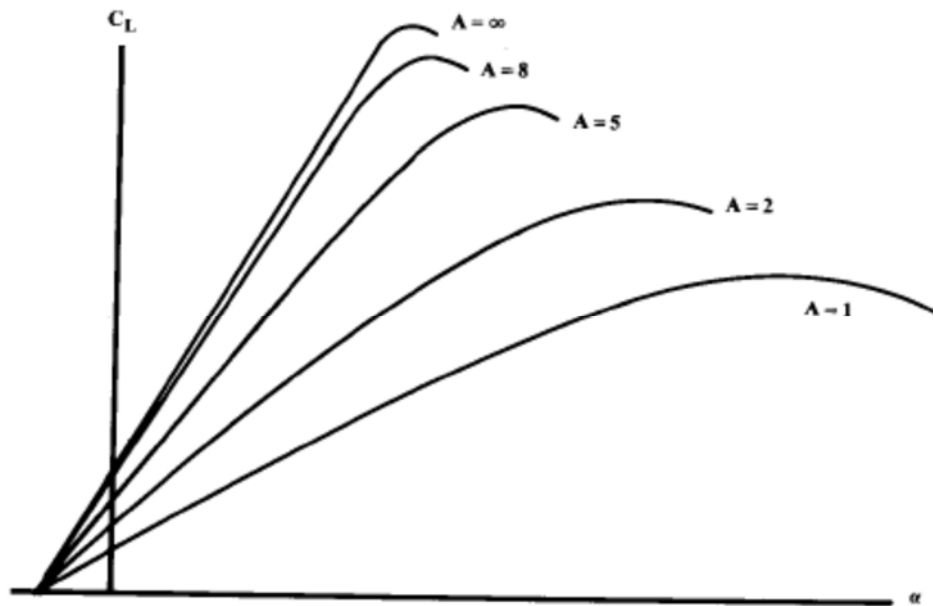


Fig. 4.18 Effect of aspect ratio on lift.

Table 4.1 Aspect ratio

Sailplane equivalent* aspect ratio = 4.464 (best $L/D$ ). <sup>69</sup>		
Propeller aircraft		
	Equivalent aspect ratio	
Homebuilt	6.0	
General aviation—single engine	7.6	
General aviation—twin engine	7.8	
Agricultural aircraft	7.5	
Twin turboprop	9.2	
Flying boat	8.0	
Jet aircraft		
	Equivalent aspect Ratio = $aM_{max}^C$	
	$a$	$C$
Jet trainer	4.737	-0.979
Jet fighter (dogfighter)	5.416	-0.622
Jet fighter (other)	4.110	-0.622
Military cargo/bomber	5.570	-1.075
Jet transport	7.50	0

\*Equivalent aspect ratio = wing span squared/(wing and canard areas)

# Selección Estrechamiento – Distribución Elíptica

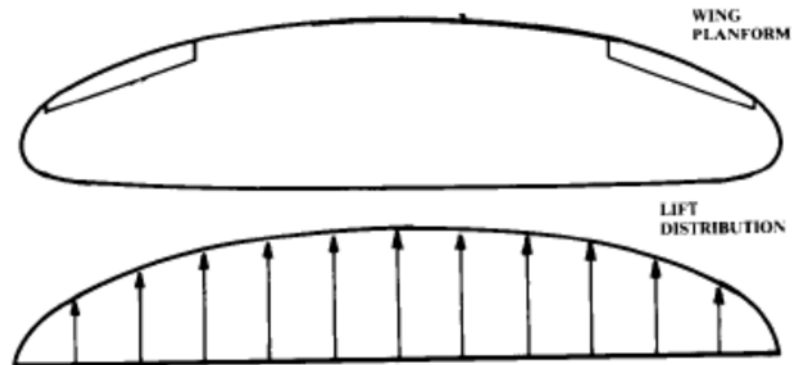


Fig. 4.21 Elliptical wing.

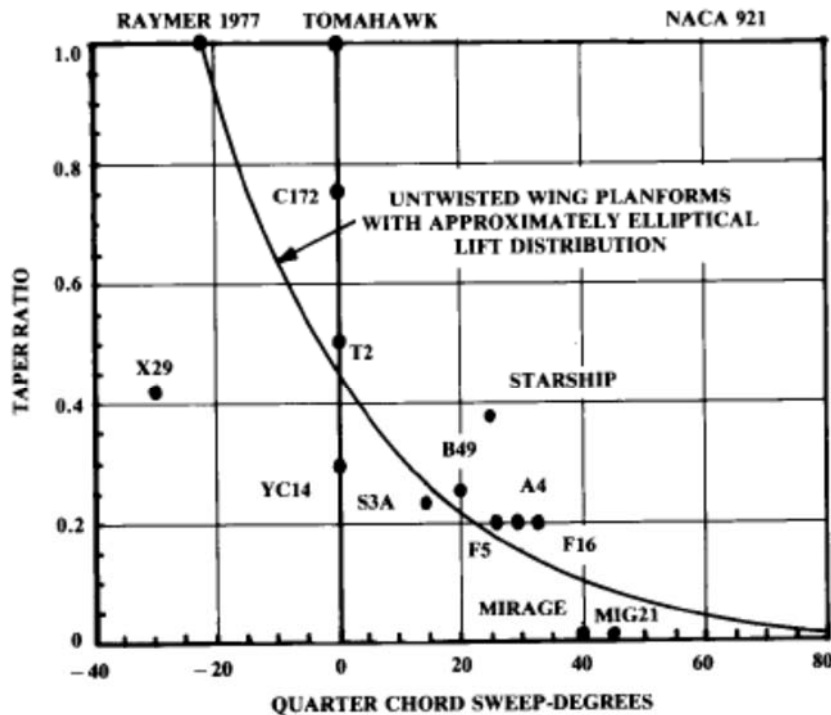


Fig. 4.23 Effect of sweep on desired taper ratio.

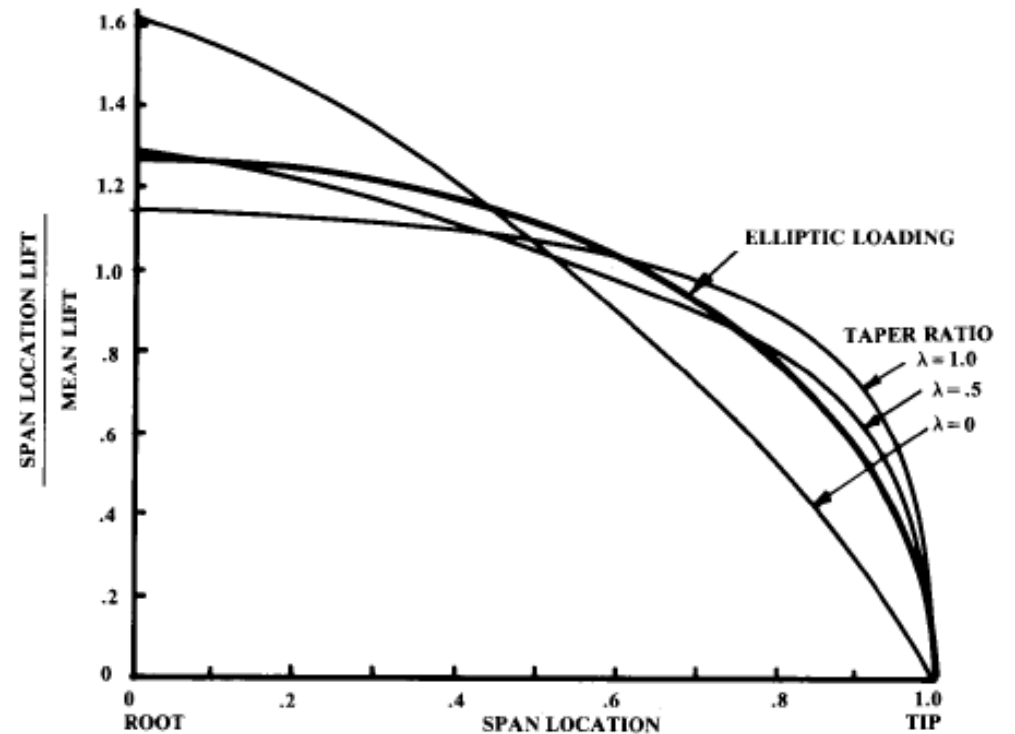


Fig. 4.22 Effect of taper on lift distribution.

- Hp/T Ratio

- Pitts W/Hp = 6.4
- Great Lakes W/Hp = 10
- Seleccionamos inicialmente W/Hp = 8

- De la tabla 5.4 de Raymer calculamos el Hp/W de forma estadística

$$H_p/W = .004 V_{\max}^{.57} = .004 (130 \times 1.151)^{.57} = .069$$

$$\text{or } W/H_p = 14.4$$

- Pero hay que tener en cuenta que estos valores estadísticos son para aviones en crucero, y un avión acrobático tiene una misión de vuelo totalmente diferente, por lo que nos mantendremos con el valor entre los dos aviones usados como referencia (Pitts y Great Lakes)

$$\underline{W/H_p = 8} \quad \text{and } W_0 = 8 \times 150 = 1200 \text{ lb}$$

Table 5.4 hp/W<sub>0</sub> vs V<sub>max</sub> (mph)

hp/W <sub>0</sub> = a V <sub>max</sub> <sup>C</sup>	a	C
Sailplane—powered	0.043	0
Homebuilt—metal/wood	0.005	0.57
Homebuilt—composite	0.004	0.57
General aviation—single engine	0.024	0.22
General aviation—twin engine	0.034	0.32
Agricultural aircraft	0.008	0.50
Twin turboprop	0.012	0.50
Flying boat	0.029	0.23

# Estudio Carga Alar - 1

- Histórico
  - Pitts W/S = 11.7
  - Greast Lakes W/S = 9.6
  - Stevens Acro W/S = 13.0
- Entrada en pérdida.

$$C_{L_{max}} \cong 0.9 \left\{ (C_{l_{max}})_{flapped} \frac{S_{flapped}}{S_{ref}} + (C_{l_{max}})_{unflapped} \frac{S_{unflapped}}{S_{ref}} \right\}$$

STALL: No flaps, so  $C_{L_{max}} \cong .9 C_{l_{max}} \cong 1.2$

eq 5.6)  $W/S \leq \frac{1}{2} (.00238)(50 \times 1.689)^2 (1.2) = 10.2 \text{ lb/ft}^2$

$$W = L = q_{stall} S C_{L_{max}} = \frac{1}{2} \rho V_{stall}^2 S C_{L_{max}}$$

- Despegue

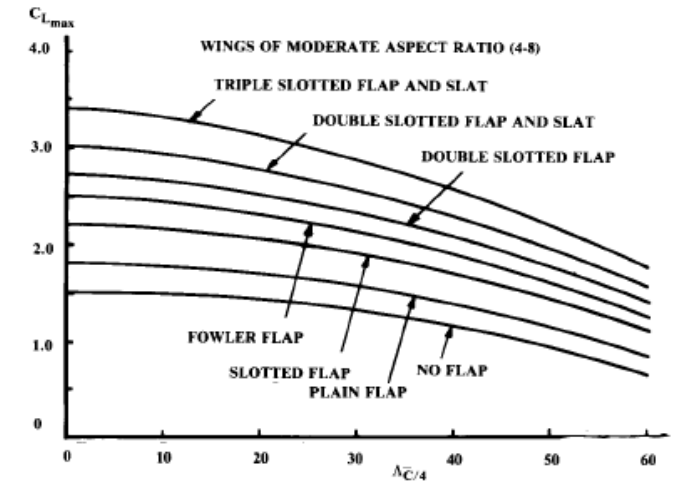
$$W/S = \frac{1}{2} \rho V_{stall}^2 C_{L_{max}}$$

Prop:  $(W/S) = (TOP) \sigma C_{L_{TO}} (hp/W)$

TAKEOFF: 1000 Ft. TAKEOFF  $\Rightarrow$  TAKEOFF PARAMETER = 120

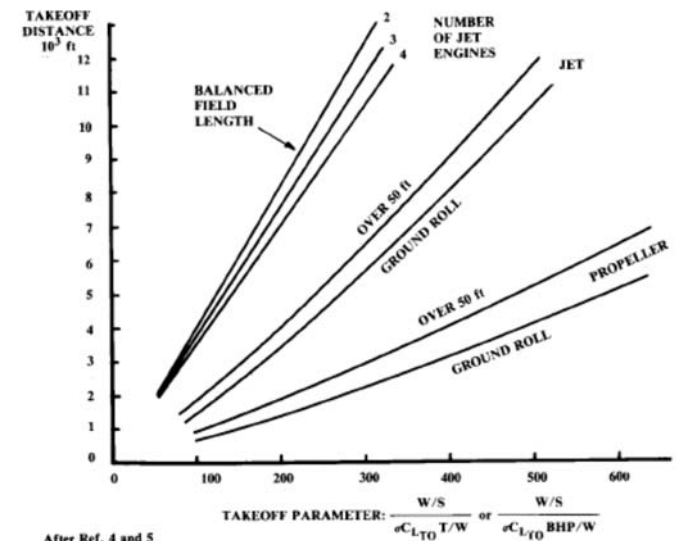
eq 5.8)  $C_{L_{takeoff}} = C_{L_{max}} \left( \frac{V_{stall}}{V_{takeoff}} \right)^2 = 1.2 \left( \frac{1}{1.1} \right)^2 = .99$

T.O.P. = 120 =  $\frac{W/S}{(1)(.99)(1/8)}$  ;  $W/S \leq 14.9$



After Ref. 39

Fig. 5.3 Maximum lift coefficient.



After Ref. 4 and 5

Fig. 5.4 Takeoff distance estimation.

# Estudio Carga Alar - 2

- Subida

- Se asume una velocidad de subida de 70 knots

$$G = (T - D)/W$$

Climb gradient

$$\frac{D}{W} = \frac{T}{W} - G$$

$$\frac{D}{W} = \frac{qSC_{D0} + qS(C_L^2/\pi Ae)}{W} = \frac{qC_{D0}}{W/S} + \frac{W}{S} \frac{1}{q\pi Ae}$$

$$\frac{W}{S} = \frac{[(T/W) - G] \pm \sqrt{[(T/W) - G]^2 - (4C_{D0}/\pi Ae)}}{2/q\pi Ae}$$

$$H_p/W = 1/8 \text{ and } H_p = T V$$

CLIMB: Assume climb speed = 70 kts, so  $q = 16.6 \text{ lb/ft}^2$

$$G = \frac{V_y}{V} = \frac{1500/60}{70 \times 1.689} = .212$$

$$T/W = \frac{550 \times 0.8}{70 \times 1.689} \left(\frac{1}{8}\right) = .465$$

Assume  $e = .8$

From sketch,  $\frac{S_{wet}}{S_{ref}} \cong 3.5$  so  $C_{D0} \cong 3.5 \times .0055 = .02$   
eg 12.23

$$\text{eg 5.30) } W/S \leq \left[ (.465 - .212) + \sqrt{(.465 - .212)^2 - \frac{4(.02)}{\pi \times 6 \times 8}} \right] \sqrt{\frac{2}{16.6 \pi \times 6 \times 8}} = 62$$

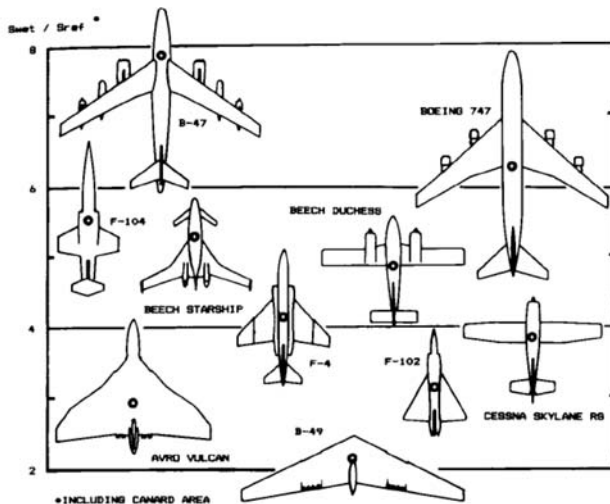


Fig. 3.5 Wetted area ratios.

# Estudio Carga Alar - 3

## ■ Crucero

- Para aviones con hélices, para maximizar el alcance en crucero se debe de seleccionar una carga alar que genere un L/D elevado.
- Para aviones con hélice, la eficiencia de empuje desciende a medida que se aumenta la velocidad, por lo que obtiene el máximo alcance cuando vuela a la velocidad óptima para L/D tal que  $CD_0 = CD_i$

	Cruise	Loiter
Jet Prop	$0.866 \frac{L/D_{max}}{L/D_{max}}$	$\frac{L/D_{max}}{0.866 L/D_{max}}$

$$qSC_{D_0} = qS \frac{C_L^2}{\pi A e}$$

$$\text{Maximum Prop Range: } W/S = q \sqrt{\pi A e C_{D_0}}$$

$$\text{CRUISE: } q_{cruise} = 35 \text{ lb/ft}^2$$

$$W/S = 35 \sqrt{\pi \times 6 \times 0.8 \times 0.02} = 20$$

$$\star \text{ SELECT LOWEST: } W/S = 10.2 ; S = \frac{1200}{10.2} = 118 \text{ FT}^2$$

# Dimensionado inicial - 1

- Fracción del peso en vacío

$$\frac{W_e}{W_0} = .59 W_0^{-1} (6)^{.05} (1/8)^{.1} (10.2)^{-.05} (130)^{-.17} = 1.093 W_0^{-1}$$

- Ésta ecuación es para aviones en crucero, no para aviones acrobáticos. Ajustamos al ecuación utilizando la información del Stevens ACRO

Stevens Acro:  $W_0 = 1300$   $W_e = 950$   $W_e/W_0 = .73$   
 Our equation gives:  $\frac{W_e}{W_0} = 1.093(1300)^{-1} = .533$  (too low!)

Use a fudge-factor to adjust the equation.

$$\frac{W_e}{W_0} = \left(\frac{.73}{.533}\right) 1.093 W_0^{-1} = 1.495 W_0^{-1}$$

At  $W_0 = 1200$  Lb,  $W_e = 883$  Lb

Table 6.2 Empty weight fraction vs  $W_0$ , A, hp/ $W_0$ ,  $W_0/S$ , and  $V_{max}$  (mph)

	$W_e/W_0 = a + bW_0^{C1} A^{C2} (hp/W_0)^{C3} (W_0/S)^{C4} V_{max}^{C5}$						
	a	b	C1	C2	C3	C4	C5
Sailplane—unpowered	0	0.75	-0.05	0.14	0	-0.30	0.06
Sailplane—powered	0	1.20	-0.04	0.14	0.19	-0.20	0.05
Homebuilt—metal/wood	0	0.69	-0.10	0.05	0.10	-0.05	0.17
Homebuilt—composite	0	0.59	-0.10	0.05	0.10	-0.05	0.17
General aviation—single engine	-0.25	1.14	-0.20	0.08	0.05	-0.05	0.27
General aviation—twin engine	-0.90	1.32	-0.10	0.08	0.05	-0.05	0.20
Agricultural aircraft	0	1.64	-0.14	0.07	0.10	-0.10	0.11
Twin turboprop	0.37	0.08	-0.06	0.08	0.08	-0.05	0.30
Flying boat	0	0.41	-0.01	0.10	0.05	-0.12	0.18

# Dimensionado inicial - 2

- Fracciones de peso por segmentos de vuelo

- Crucero

$$\frac{W_1}{W_0} = .97 \quad \frac{W_2}{W_1} = .985$$

$$\frac{L}{D} = \frac{1}{\frac{qC_{D0}}{W/S} + \frac{W}{S} \frac{1}{q\pi Ae}}$$

$$\text{CRUISE: } W/S = (10.2)(.97)(.985) = 9.7 \quad ; q = 35 \text{ lb/ft}^2$$

$R$  = range

$C$  = specific fuel consumption

$V$  = velocity

$L/D$  = lift-to-drag ratio

$$\frac{W_3}{W_2} = e^{\frac{-(280 \times 6076)(.5/3600)}{11.04 \times 550 \times 0.8}} = .953$$

$$\text{Prop: } \frac{W_i}{W_{i-1}} = \exp\left[\frac{-RC_{bbp}}{550 \eta_p (L/D)}\right]$$

aterrizaje  $\frac{W_4}{W_3} = .995$

$$\frac{W_4}{W_0} = .97 \times .985 \times .953 \times .995 = .906$$

$$\frac{W_f}{W_0} = 1.06 \left(1 - \frac{W_x}{W_0}\right)$$

$$\frac{W_f}{W_0} = 1.06(1 - .906) = .0997$$

$$W_f = 1200 \times .0997 = 120 \text{ lb (total)}$$

$$W_{f \text{ usable}} = 120 / 1.06 = 113 \text{ lb}$$

Table 3.2 Historical mission segment weight fractions

	$(W_i/W_{i-1})$
Warmup and takeoff	0.970
Climb	0.985
Landing	0.995



# Dimensionado inicial - 3

$$W_0 = \frac{W_{\text{crew}} + W_{\text{payload}}}{1 - (W_f/W_0) - (W_e/W_0)}$$

INITIAL SIZING (RUBBER ENGINE)

eg 3.4)  $W_0 = \frac{220}{1 - 0.0977 - \frac{W_e}{W_0}}$  where  $\frac{W_e}{W_0} = 1.495 W_0^{-1}$

INPUT W0drawn, WEdrawn, We/Wo Exponent:1200,883,-.1  
 INPUT CREW + PAYLOAD WEIGHT 220

MISSION SEG TYPES  
 MISSION SEG WT FRACTS .97 .985 .953 .995

\*\*\*\*\* SIZING ITERATIONS \*\*\*\*\*

WOG	WF	WE	WOCALC
1200.0	119.6	883.0	1222.6
1218.1	121.4	895.0	1236.3
1232.7	122.8	904.6	1247.4
1244.5	124.0	912.4	1256.4
1254.0	125.0	918.7	1263.7
1261.7	125.7	923.8	1269.5
1268.0	126.4	927.9	1274.2
1273.0	126.9	931.2	1278.0
1277.0	127.3	933.9	1281.1
1280.3	127.6	936.0	1283.6
1282.9	127.8	937.7	1285.6
1285.0	128.1	939.1	1287.2
1286.8	128.2	940.2	1288.5
1288.1	128.4	941.2	1289.5
1289.2	128.5	941.9	1290.4

BUT THIS HEAVIER W0 WOULD GIVE REDUCED PERFORMANCE WITH A FIXED-SIZE ENGINE!

## FIXED-ENGINE SIZING

- Varied  $\frac{W_4}{W_0}$  until  $W_{0, \text{Calc.}} = W_{0, \text{drawn}} = 1200 \text{ Lb}$

INPUT W0drawn, WEdrawn, We/Wo Exponent:1200,883,-.1  
 INPUT CREW + PAYLOAD WEIGHT 220

MISSION SEG TYPES  
 MISSION SEG WT FRACTS .925

\*\*\*\*\* SIZING ITERATIONS \*\*\*\*\*

WOG	WF	WE	WOCALC
1200.0	95.4	883.0	1198.4
1198.7	95.3	882.2	1197.5
1197.7	95.2	881.5	1196.7

- Occurs when  $\frac{W_4}{W_0} = .925$

- Solve for  $W_3/W_2$  to determine range:

$$\frac{W_3}{W_2} = \frac{W_4/W_0}{\frac{W_1}{W_0} \times \frac{W_2}{W_1} \times \frac{W_4}{W_3}} = \frac{.925}{(.97)(.985)(.995)} = .973$$

eg 6.12)  $\frac{W_3}{W_2} = .973 = e^{-\frac{R(.5/3600)}{11.04 \times 550 \times 0.8}}$  ;  $R = 957300 \text{ ft}$   
 $R = 158 \text{ nm.}$

Range of 158 nm. is less than goal of 280 nm. We will lay out design anyway, and use refined sizing methods and optimization techniques to maximize range and performance.

# Layout Data - 1

- Una vez calculado el peso inicial ( $W_o$ ), el peso en vacio ( $W_e$ ), el peso de combustible ( $W_f$ )
  - $W_o=1200$  lbs,  $W_e=883$ lbs y  $W_f=120$  lbs. (112 utilizable).
  - $S=118$  ft<sup>2</sup>,  $A=6$ ,  $\lambda=0.4$

$$b = \sqrt{AS}$$

$$C_{\text{root}} = \frac{2S}{b(1 + \lambda)}$$

$$C_{\text{tip}} = \lambda C_{\text{root}}$$

$$\bar{C} = \left(\frac{2}{3}\right) C_{\text{root}} \frac{1 + \lambda + \lambda^2}{1 + \lambda}$$

$$\bar{Y} = \left(\frac{b}{6}\right) \left(\frac{1 + 2\lambda}{1 + \lambda}\right)$$

$b = 26.6 \text{ ft} = 319.3 \text{ in}$   
 $C_r = 75.6 \text{ in}$   
 $C_t = 30.4 \text{ in}$   
 $\bar{C} = 56 \text{ in}$   
 $\bar{Y} = 68.4 \text{ in}$

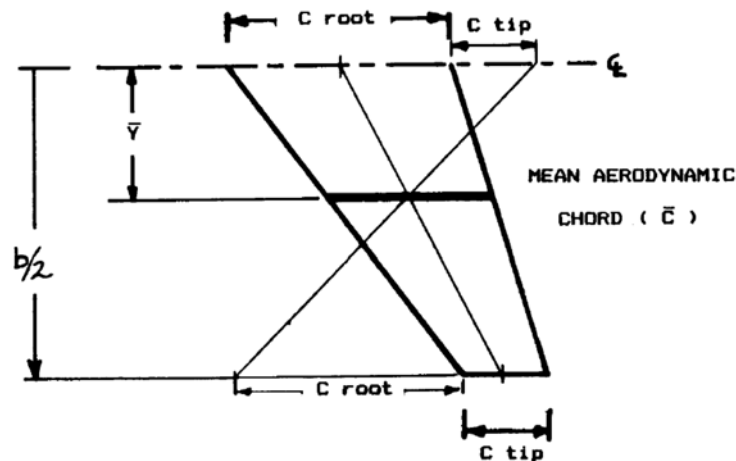
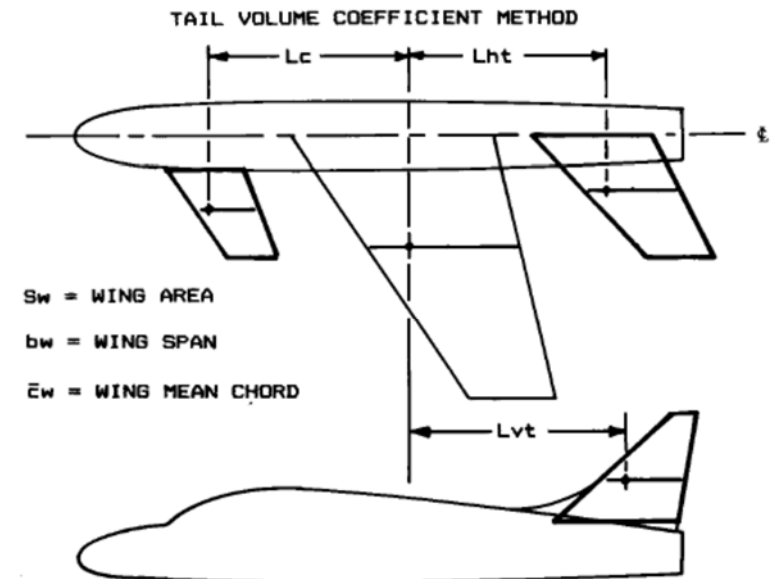


Fig. 7.23 Reference (trapezoidal) wing/tail.



$S_w$  = WING AREA  
 $b_w$  = WING SPAN  
 $\bar{c}_w$  = WING MEAN CHORD

Fig. 6.2 Initial tail sizing.

# Layout Data - 2

- La configuración del fuselaje viene dada por:

$$L \cong 3.5 (1200)^{-0.23} = 18 \text{ ft}$$

- Para aviones con un motor de hélice montado en la parte frontal el brazo de la cola es aproximadamente el 60% del largo del fuselaje.

$$L_{\text{tail arm}} \cong 60\% L = 10.8 \text{ ft}$$

- NOTA: para aviones con los motores en el ala, el brazo de la cola es aproximadamente 50-55% del largo del fuselaje.
- Para los aviones con configuración canard el brazo es aproximadamente 30-50%
- Para la deriva horizontal y vertical, el tail volume coefficient viene dado por:

$$c_{VT} = \frac{L_{VT} S_{VT}}{b_W S_W}$$

$$c_{HT} = \frac{L_{HT} S_{HT}}{C_W S_W}$$

Table 6.4 Tail volume coefficient

	Typical values	
	Horizontal $c_{HT}$	Vertical $c_{VT}$
Sailplane	0.50	0.02
Homebuilt	0.50	0.04
General aviation—single engine	0.70	0.04
General aviation—twin engine	0.80	0.07
Agricultural	0.50	0.04
Twin turboprop	0.90	0.08
Flying boat	0.70	0.06
Jet trainer	0.70	0.06
Jet fighter	0.40	0.07
Military cargo/bomber	1.00	0.08
Jet transport	1.00	0.09

Table 6.3 Fuselage length vs  $W_0$

Length = $aW_0^c$	$a$	$C$
Sailplane—unpowered	0.86	0.48
Sailplane—powered	0.71	0.48
Homebuilt—metal/wood	3.68	0.23
Homebuilt—composite	3.50	0.23
General aviation—single engine	4.37	0.23
General aviation—twin engine	0.86	0.42
Agricultural aircraft	4.04	0.23
Twin turboprop	0.37	0.51
Flying boat	1.05	0.40
Jet trainer	0.79	0.41
Jet fighter	0.93	0.39
Military cargo/bomber	0.23	0.50
Jet transport	0.67	0.43

# Layout Data - 3

Vertical tail volume coefficient

$$C_{VT} = \frac{L_{VT} S_{VT}}{b_W S_W}$$

$$C_{VT} \cong .04 = \frac{10.8 S_{VT}}{26.6 \times 118} ; S_{VT} = 11.6 \text{ ft}^2$$

$$A = 1.5 ; \lambda = 0.4$$

$$b = 4.1 \text{ ft} \quad C_r = 4.0 \text{ ft} \quad C_x = 1.6 \text{ ft}$$

$$b = \sqrt{AS}$$

$$C_{\text{root}} = \frac{2S}{b(1+\lambda)}$$

$$C_{\text{tip}} = \lambda C_{\text{root}}$$

$$\bar{C} = \left(\frac{2}{3}\right) C_{\text{root}} \frac{1+\lambda+\lambda^2}{1+\lambda}$$

$$\bar{Y} = \left(\frac{b}{6}\right) \left(\frac{1+2\lambda}{1+\lambda}\right)$$

Horizontal tail volume coefficient

$$C_{HT} = \frac{L_{HT} S_{HT}}{C_W S_W}$$

$$C_{HT} \cong .5 = \frac{10.8 S_{HT}}{4.7 \times 118} ; S_{HT} = 25.5 \text{ ft}^2$$

$$A = 4.0 ; \lambda = 0.4$$

$$b = 10.1 \text{ ft} \quad C_r = 3.6 \text{ ft} \quad C_x = 1.4 \text{ ft}$$

## ■ NOTA:

- Para la deriva vertical, la distancia a lo largo de la envergadura (Y) dónde se sitúa la mean aerodynamic chord (c) es el doble de la calculada.
- Para una cola móvil, el tail volume coefficient puede ser reducido un 10-15%
- Para una T-tail, el vertical tail volume coefficient puede ser reducido un 5% debido al efecto de end-plate, y el horizontal tail volume coefficient puede reducirse un 5% debido al flujo limpio de la superficie.
- El horizontal tail volume coefficient en las H-tail también puede reducirse en un 5%.
- Para una superficie sustentadora canard el método del tail volume coefficient no es válido, y se distribuye el área necesaria para sustentación, siendo la forma típica 25% (canard) 75% (ala).

# Layout Data - 4

- Tanque de combustible viene dado por el volumen útil de combustible. Se calculó previamente que  $W_f = 120$  lbs (todo y que útiles para el cálculo del alcance max sea un 6% menos  $W_{\text{useful}} = 112$  lbs)

- El keroseno (gasolina aviación) pesa  $\approx 5.64$  lb/gallon

$$W_f = 120 \text{ lb} = 20 \text{ gallons} = 2.7 \text{ ft}^3$$

- Tamaño de las ruedas:

$$D = 11 \text{ in.}; W = 5.3 \text{ in.}$$

- Diámetro de la hélice:

Two blade:  $d = 22 \sqrt[4]{\text{Hp}}$

$$d = 22 \sqrt[4]{150} = 77 \text{ in } (?)$$

- Hay que comprobar la velocidad en la punta de la hélice

$$(V_{\text{tip}})_{\text{static}} = \pi n d / 60$$

$n$  = rotational rate (rpm) obtained from engine data

$d$  = diameter

$$(V_{\text{tip}})_{\text{helical}} = \sqrt{V_{\text{tip}}^2 + V^2}$$

$$V = 115 \text{ kts} = 194 \text{ ft/sec}$$

$$n = 2700 \text{ rpm} = 45 \text{ rev/sec}$$

$$\sqrt{194^2 + (\pi n d)^2} \leq 850 \text{ ft/sec (Wood Prop.)}$$

$$\text{so } d \leq 5.85 \text{ ft} = 70 \text{ in}$$

Wood propeller  $V_{\text{tip}} < 950$  fps

Metal propeller  $V_{\text{tip}} < 850$  fps

Table 10.5 Fuel densities (lb/gal)

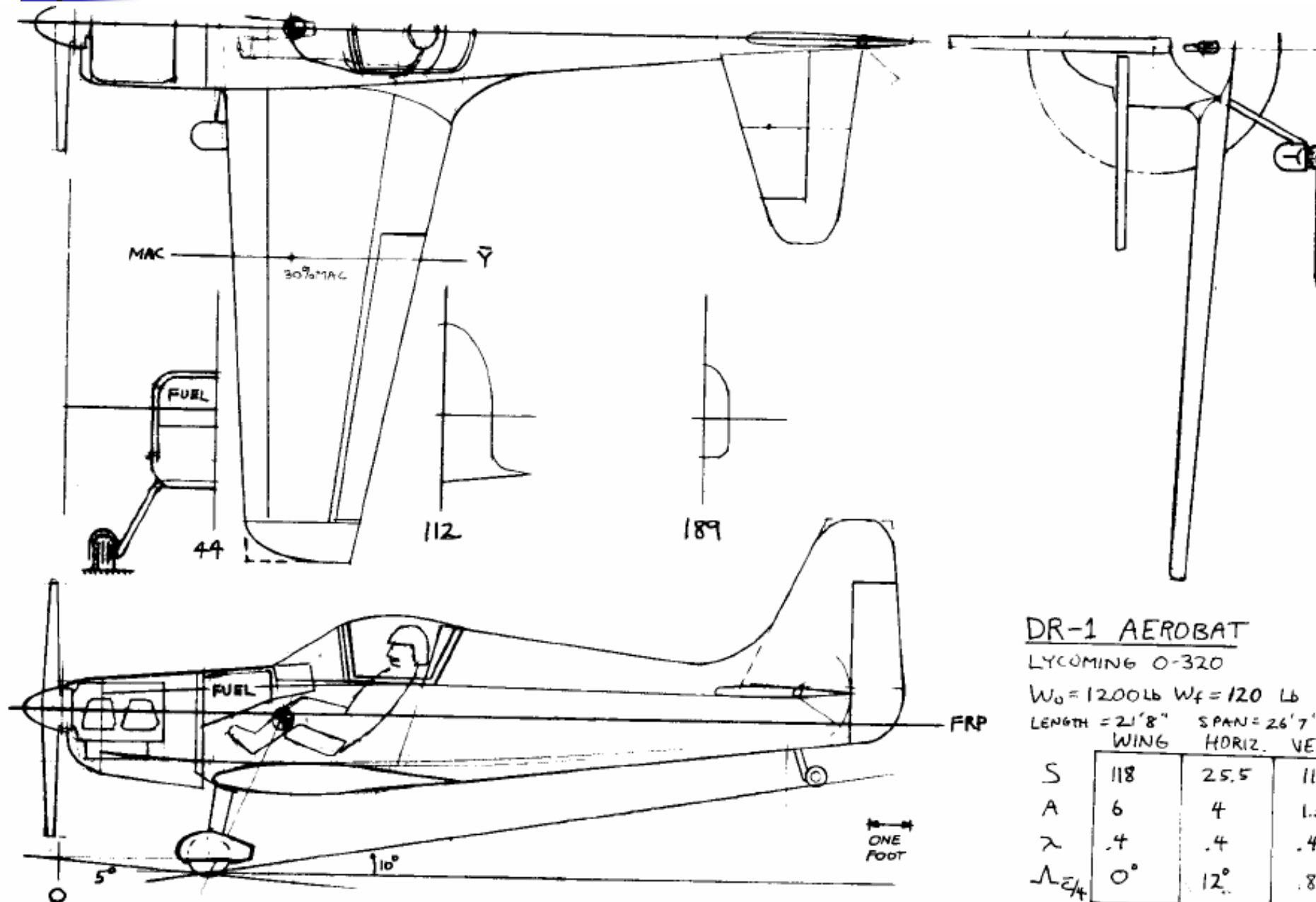
	Average actual density		Mil-spec density
	0°F	100°F	
Aviation gasoline	6.1	5.7	6.0
JP-4	6.7	6.4	6.5
JP-5	7.2	6.8	6.8
JP-8	—	—	6.7

Table 11.1 Statistical tire sizing

Main wheels diameter or width (in.) = $A W_W^B$	Diameter		Width	
	A	B	A	B
	General aviation	1.51	0.349	0.7150
Business twin	2.69	0.251	1.170	0.216
Transport/bomber	1.63	0.315	0.1043	0.480
Jet fighter/trainer	1.59	0.302	0.0980	0.467

$W_W$  = Weight on Wheel

# Layout Data - 5



## DR-1 AEROBAT

LYCOMING O-320

$W_0 = 1200 \text{ lb}$   $W_f = 120 \text{ lb}$

LENGTH = 21'8" SPAN = 26'7"

WING HORIZ. VERT.

S	118	25.5	11.6
A	6	4	1.5
$\lambda$	.4	.4	.4
$\Lambda_{\text{eff}}$	0°	12°	8°

D.P. RAYMER JAN 88

# Coeficiente de Sustentación - $C_L$

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

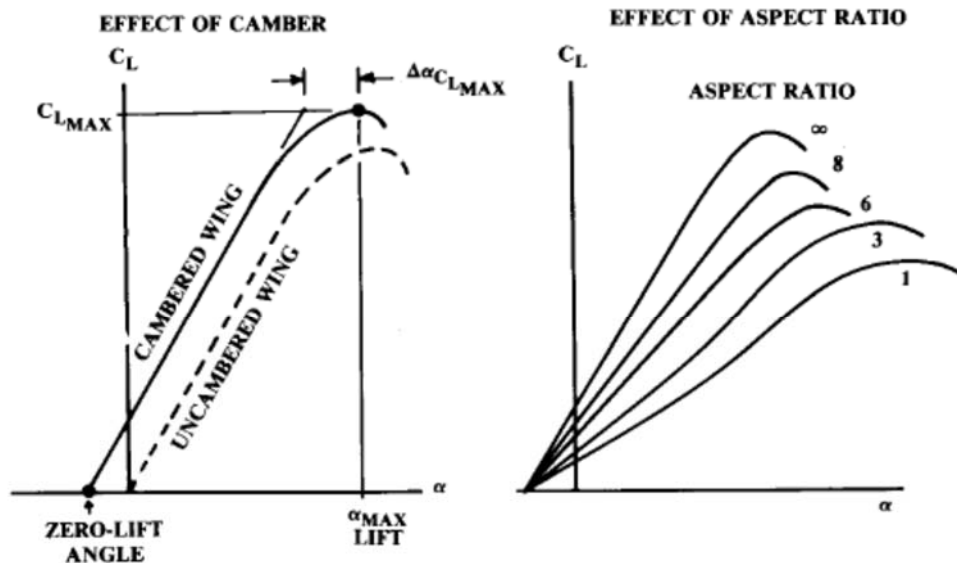


Fig. 12.4 Wing lift curve.

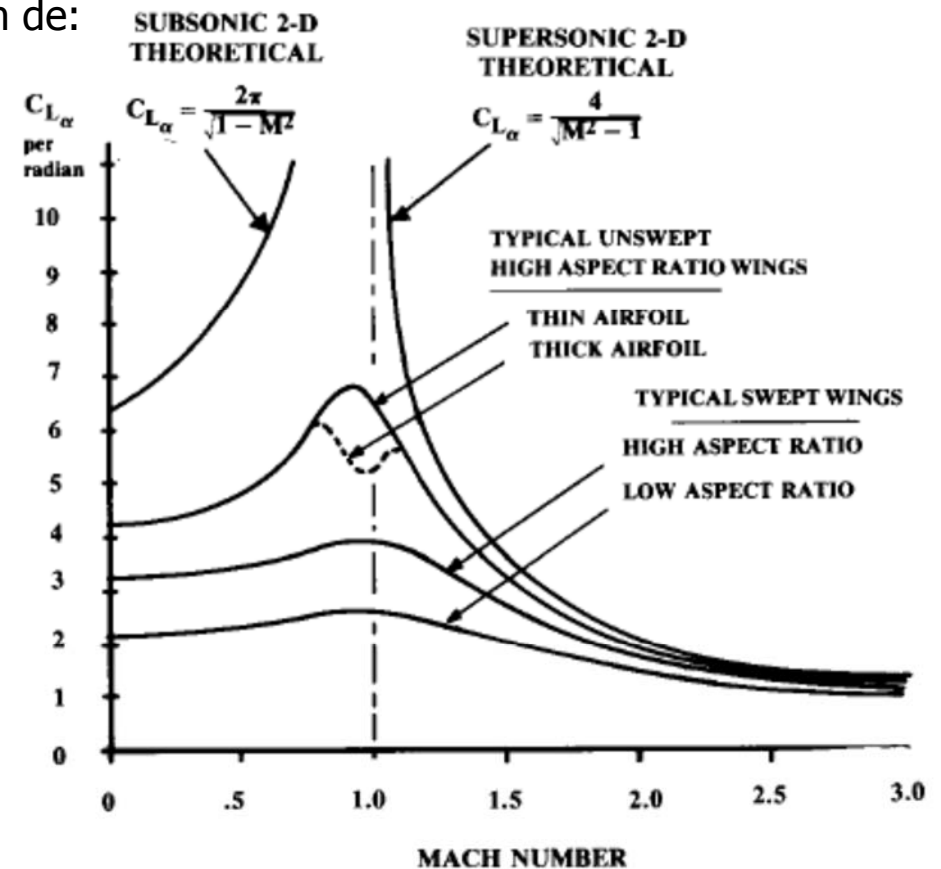


Fig. 12.5 Lift curve slope vs Mach number.

# Importancia del $C_{L\alpha}$

- $C_{L\alpha}$  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 pares y momentos
- $C_{L\alpha}$  disminuye con alargamiento.
  - Al disminuir el alargamiento, la habilidad del aire para escapar por las alas previene la entrada en pérdida incluso a ángulos de ataque elevados.
- $C_{L\alpha}$  disminuye con el aumento de flecha, y los efectos son similares.
- Perfiles actuales tiene un 90-100% eficiencia aerodinámica del perfil ( $\eta$ )
- Influencia con el Mach
- Alargamiento del ala al incluir *winglets* y *endplates*.



# Winglet

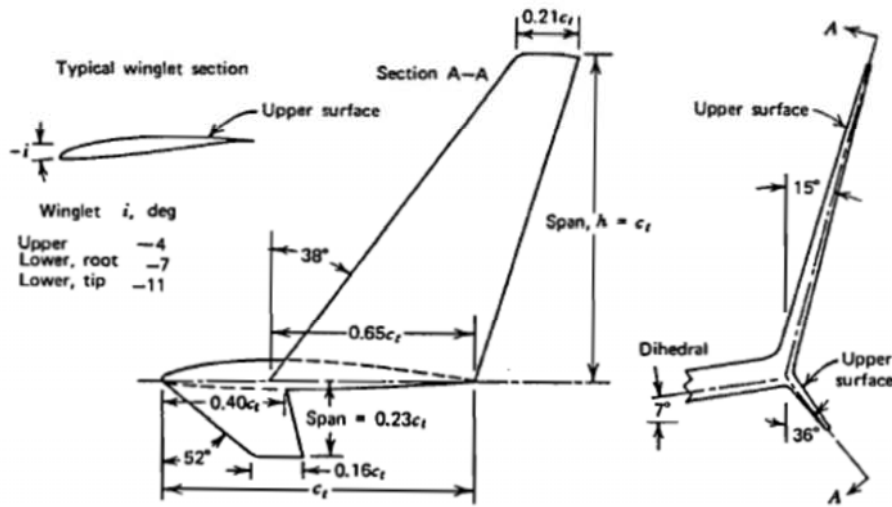


Figure 4.35 Winglet geometry.

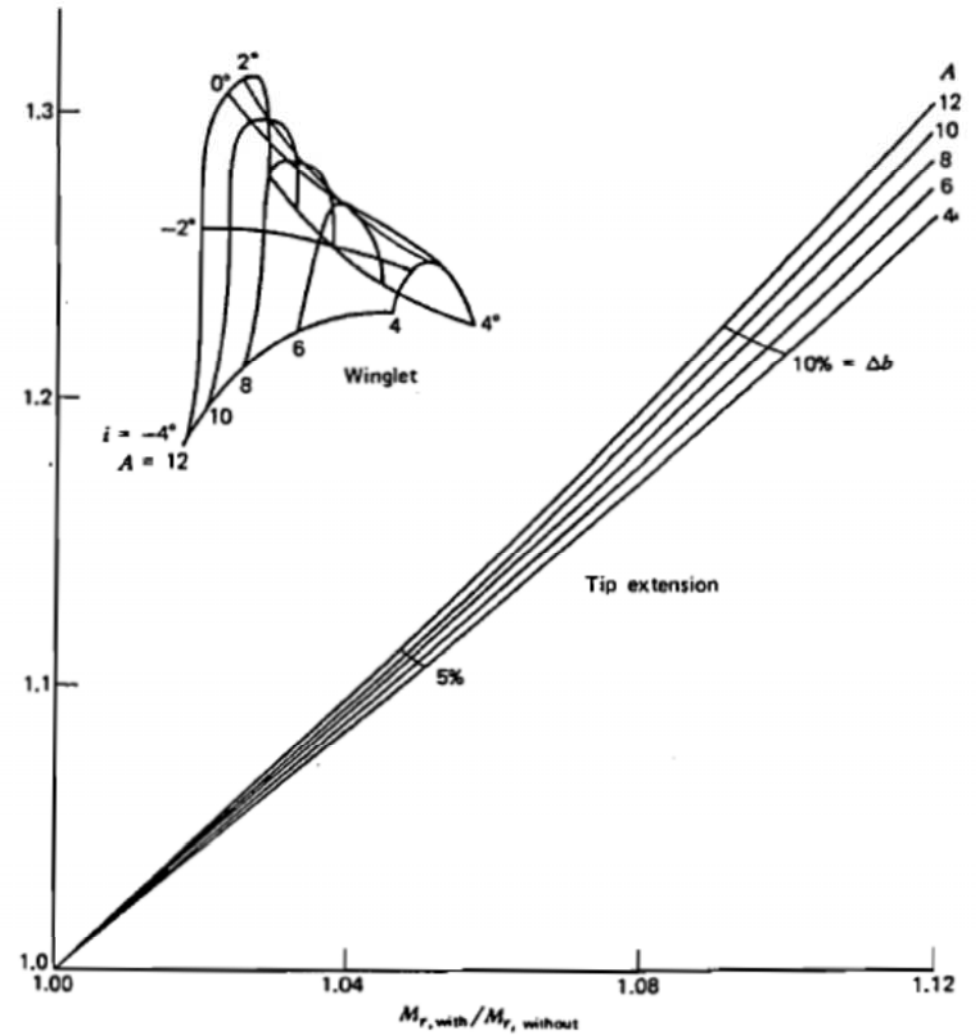


Figure 4.39 Comparison of tip extension and winglet when added to an untwisted wing.

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

Endplate:  $A_{\text{effective}} = A(1 + 1.9 h/b)$

Winglet:  $A_{\text{effective}} \cong 1.2A$

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

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

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

Flecha máxima en la cuerda

$$F = 1.07(1 + d/b)^2$$

Factor de sustentación del fuselaje  
d – diámetro del fuselaje

Eficiencia aerodinámica del perfil

# 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.
  - Máximo empuje (configuración limpia)
    - Solo valida para configuraciones moderadas de alargamiento

$$C_{L_{\max}} = 0.9C_{l_{\max}} \cos\Lambda_{0.25c}$$

- Corrección para alargamiento bajo y elevado
- Máximo empuje con superficies hipersustentadoras
  - Incremento de sustentación.

# Aerodynamics - 1

- Determinación de las características aerodinámicas teniendo en cuenta las áreas mojadas y expuestas del dibujo conceptual.
- Obtención del máximo coeficiente de sustentación, teniendo en cuenta la penalización de la flecha:

$$C_{L_{max}} = 0.9 C_{l_{max}} \cos \Lambda_{0.25c} \quad C_{L_{max}} \approx 0.9 C_{l_{max}} = (1.35)(0.9) = 1.2$$

- Pendiente de la curva de sustentación:

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

$$\beta^2 = 1 - M^2 \quad \eta = \frac{C_{l_{\alpha}}}{2\pi/\beta} \quad F = 1.07(1 + d/b)^2$$

$$C_{L_{\alpha}} = \frac{2\pi \times 6 \times \frac{S_{exposed}}{S_{ref}} F}{2 + \sqrt{4 + \frac{6^2}{.95^2} \left(1 + \frac{\tan^2 0}{1}\right)}} = 4.37 \times .837 \times 1.33$$

$$C_{L_{\alpha}} = 4.85 \text{ per radian} = .085 \text{ per degree}$$

# Coeficiente de Resistencia - $C_D$

- La resistencia esta compuesta por:
  - Fricción
  - Flat plate friction
  - Profile
  - Roughness
  - Excrescences
  - Interference
  - Inducido
    - Tren de aterrizaje.
    - Gondolas
    - Protuberancias.
  - 3-D efectos
  - Efectos de compresibilidad
  - Inducido
- 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.

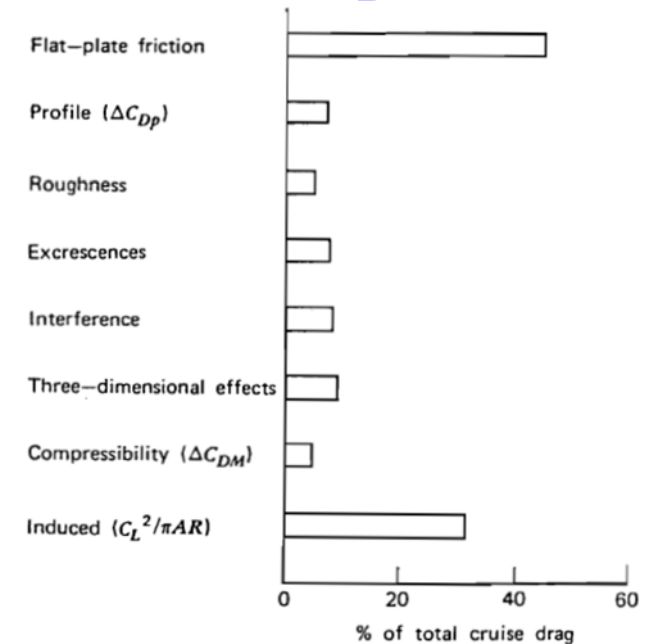


Figure 4.34 Typical drag buildup for jet transport.

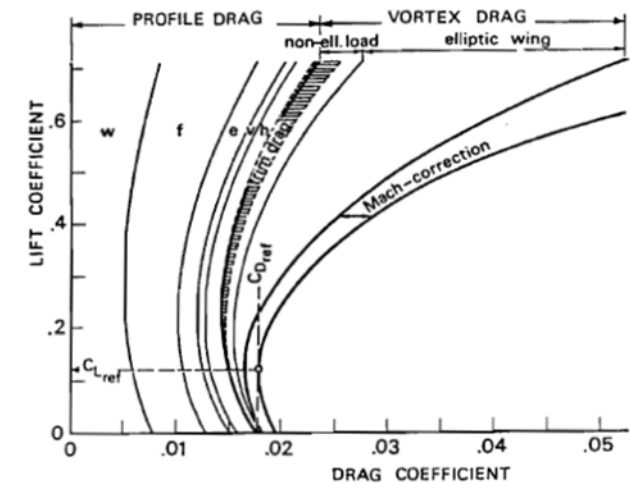


Fig. 11-2. Drag buildup by analysis (w = wing; f = fuselage; e = engine installation; v = vertical tailplane; h = horizontal tailplane)

# Coeficiente de Resistencia - $C_D$

## ■ Hipótesis simplificada

- Polar parabólica de coeficientes constantes
- Solo influye los términos de resistencia de fricción y la resistencia inducida

$$\text{Unicambered: } C_D = C_{D_0} + KC_L^2$$

$$\text{Cambered: } C_D = C_{D_{\min}} + K(C_L - C_{L_{\min \text{ drag}}})^2$$

- Donde 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

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

$$\text{Straight-Wing Aircraft: } e = 1.78(1 - 0.045A^{0.68}) - 0.64$$

$$\text{Swept-Wing Aircraft: } e = 4.61(1 - 0.045A^{0.68})(\cos\Lambda_{LE})^{0.15} - 3.1$$

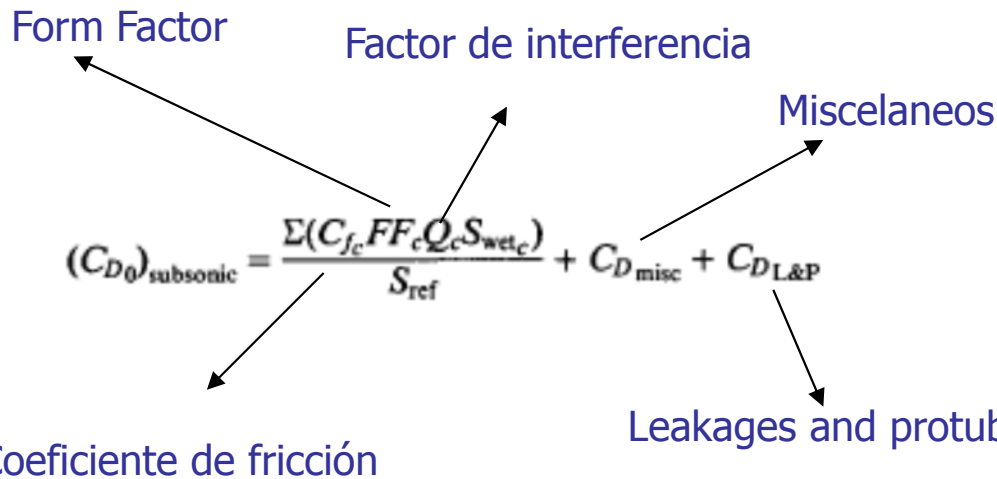
$$(\Lambda_{LE} > 30 \text{ deg})$$

# Coeficiente de Resistencia - $C_{D0}$

- La resistencia de fricción (*parasite drag*) se conoce como la resistencia en la que la sustentación es el cero.
- Para alas sin curvatura, equivale al valor mínimo de la resistencia.
- Métodos para estimar  $C_{D0}$ :
  - Método de la equivalencia de la superficie de fricción

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

- Método del coeficiente de placa plana (*Flat-plate*) de superficie de fricción mediante



**Table 12.3 Equivalent skin friction coefficients**

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

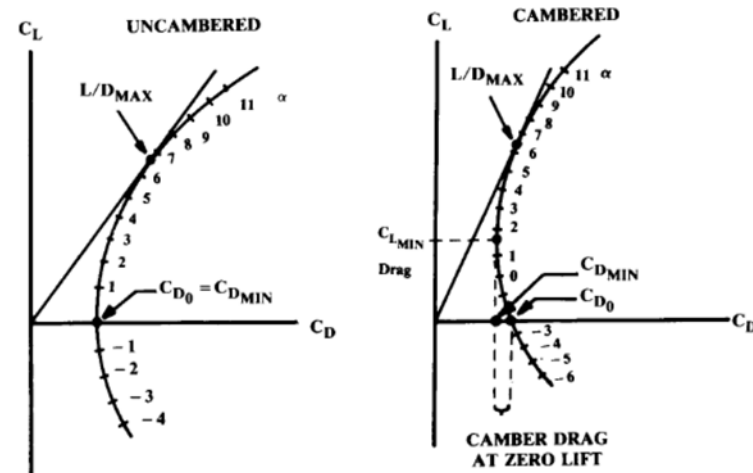
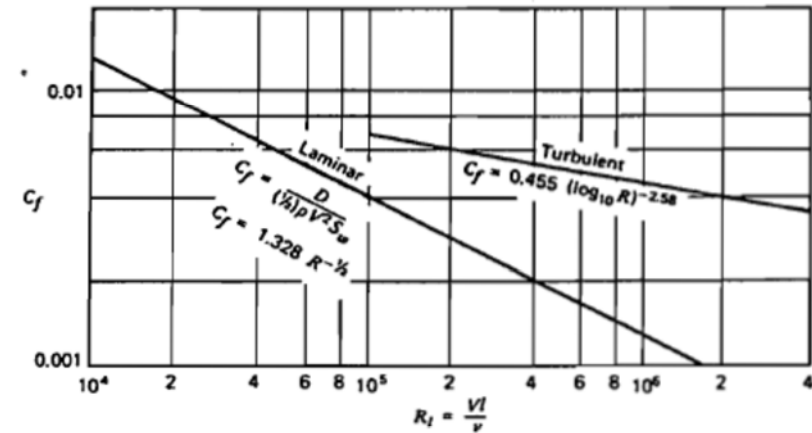


Fig. 12.3 Drag polar.

# Método del coeficiente de placa plana - 1

- Calcula el coeficiente de resistencia de fricción para cada parte del avión utilizando una resistencia de fricción basada en el método de placa plana, y luego los suma todos
- El coeficiente de resistencia ( $C_f$ ) de fricción de placa plana depende de:
  - Número de Reynolds.
  - Mach.
  - Superficie
- Puede ser turbulento o laminar



Laminar:  $C_f = 1.328/\sqrt{R}$   $R = \rho V \ell / \mu$

Turbulent:  $C_f = \frac{0.455}{(\log_{10} R)^{2.58} (1 + 0.144 M^2)^{0.65}}$

- Característica de la superficie afectan al número de Reynolds

Subsonic:  $R_{\text{cutoff}} = 38.21(\ell/k)^{1.053}$

Transonic or Supersonic:  $R_{\text{cutoff}} = 44.62(\ell/k)^{1.053} M^{1.16}$

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

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

**Table 12.4** Skin roughness value ( $k$ )

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



# Método del coeficiente de placa plana - 2

- Form Factor (FF)

Fuselage and Smooth Canopy:

$$FF = \left(1 + \frac{60}{f^3} + \frac{f}{400}\right)$$

Nacelle and Smooth External Store:

$$FF = 1 + (0.35/f)$$

- Factor de interferencia (Q)

- Góndola Q=1.5
- Ala sin carenado Q=1.25
- Colas Q=1.02..

- Resistencia Miscelanea

- Tanques de combustible externos

- Resistencia Leakage and protuberances

- Antenas, puertas, bordes, carenado de superficies de control, defectos de construcción...

Wing, Tail, Strut, and Pylon:

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

$$f = \frac{\ell}{d} = \frac{\ell}{\sqrt{(4/\pi) A_{\max}}}$$

Ubicación del máximo grosor del perfil con respecto a la cuerda

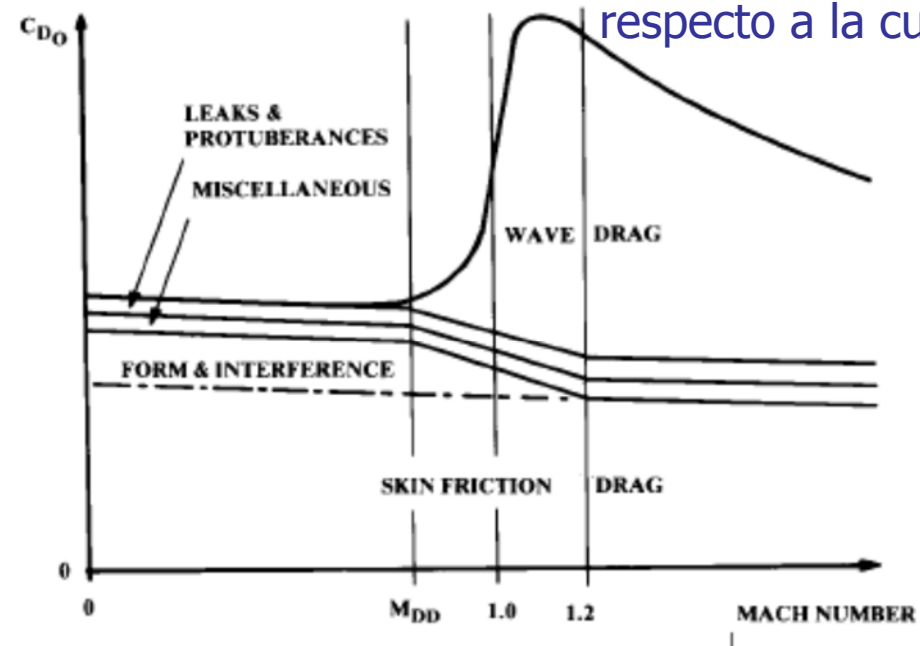


Fig. 12.30 Complete parasite drag vs Mach number.

# Aerodynamics - 2

- Cálculo de la fricción parasitaria (asumiendo que todo el flujo es turbulento)

Use  $V=100 \text{ kts} = 169 \text{ ft/s}$ ,  $h = \text{sea level}$  to determine friction  $C_f$   
 so  $M=.15$ ;  $\mu = 0.37 \times 10^{-6}$

- Fuselaje:

$$l = 22 \text{ ft} \quad l/d = 6.38$$

$$R = \rho V l / \mu$$

$$\text{Reynolds \# : } R = 23,916,000$$

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

$$R_{\text{cutoff}} = 84,158,000 \quad (\text{using smooth paint})$$

$$\text{Turbulent: } C_f = \frac{0.455}{(\log_{10} R)^{2.58} (1 + 0.144 M^2)^{0.65}}$$

$$C_f = \frac{.455}{(\log_{10} 23,916,000)^{2.58} (1 + .144(.15)^2)^{.65}} = .0026$$

$$FF = \left( 1 + \frac{60}{f^3} + \frac{f}{400} \right)$$

$$FF = 1 + \frac{60}{6.38^3} + \frac{6.38}{400} = 1.25$$

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

$$S_{\text{wet}} = 164 \text{ ft}^2$$

$$C_{D0 \text{ fuselage}} = .0026 \times 1.25 \times 164 / 118 = .0046$$

Table 12.4 Skin roughness value (k)

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

# Aerodynamics - 3

- Wing

$$l = \bar{c} = 4.67 \text{ ft} \quad \text{average } t/c = 13.5\%$$

$$R = \rho V l / \mu \quad R = 5 \times 10^6$$

$$R_{\text{cutoff}} = 38.21(l/k)^{1.053} \quad R_{\text{cutoff}} = 16.46 \times 10^6$$

$$\text{Turbulent: } C_f = \frac{0.455}{(\log_{10} R)^{2.58} (1 + 0.144 M^2)^{0.65}} \quad C_f = .0034$$

$$FF = \left[ 1 + \frac{0.6}{(x/c)_m} \left( \frac{t}{c} \right) + 100 \left( \frac{t}{c} \right)^4 \right] \left[ 1.34 M^{0.18} (\cos \Lambda_m)^{0.28} \right] \quad FF = \left[ 1 + \frac{6}{3} (.135) + 100 (.135)^4 \right] \left[ 1.34 (.15)^{.18} \right] = 1.24$$

$$(C_{D0})_{\text{subsonic}} = \frac{\Sigma(C_{fc} FF_c Q_c S_{wet,c})}{S_{\text{ref}}} + C_{D_{\text{misc}}} + C_{D_{L\&P}} \quad S_{\text{wet}} = 202.3 \text{ ft}^2 \quad Q \cong 1 \text{ using fillets}$$

$$C_{D0_{\text{wing}}} = .0034 \times 1.24 \times 202.3 / 118 = .0071$$

# Aerodynamics - 4

- Deriva vertical y horizontal se analizan simultáneamente ya que tienen cuerdas medias ( $c$ ) y  $t/c$  similares:

$$l = \bar{c}_{average} = 2.8 \quad t/c = 12\%$$

$$R = \rho V l / \mu \quad R_{cutoff} = 38.21(l/k)^{1.053}$$

$$R = 3 \times 10^6 \quad R_{cutoff} = 9.6 \times 10^6$$

$$\text{Turbulent: } C_f = \frac{0.455}{(\log_{10} R)^{2.58} (1 + 0.144 M^2)^{0.65}} \quad C_f = .0037$$

$$FF = \left[ 1 + \frac{0.6}{(x/c)_m} \left( \frac{t}{c} \right) + 100 \left( \frac{t}{c} \right)^4 \right] \left[ 1.34 M^{0.18} (\cos \Lambda_m)^{0.28} \right] \quad FF = \left[ 1 + \frac{.6}{.3} (.12) + 100 (.12)^4 \right] \left[ 1.34 (.15)^{.18} \right] = 1.20$$

$$(C_{D0})_{subsonic} = \frac{\Sigma(C_{fc} FF_c Q_c S_{wet_c})}{S_{ref}} + C_{D_{misc}} + C_{D_{L\&P}} \quad C_{D0_{tails}} = .0037 \times 1.2 \times 60.6 / 118 = .0023 \quad (+10\% \text{ for gaps} \Rightarrow .0025)$$

# Aerodynamics - 5

- Gear Drag

Tire frontal area =  $1.03 \text{ ft}^2$ ;  $D/q = 1.03 \times .13 = .134 \text{ ft}^2$   
 Strut frontal area =  $0.67 \text{ ft}^2$ ;  $D/q = 0.67 \times .05 = .033 \text{ ft}^2$

Add 20% for interference, so  $C_{D_{gear}} = 1.2(.134 + .033) / 118 = .0020$

- Cockpit Drag

Frontal Area =  $1.8 \text{ ft}^2$ ;  $C_{D_{open cockpit}} = 1.8 \times .5 / 118 = .0076$

- Total parasite drag:

- $C_{D_0} = C_{D_{fuselage}} + C_{D_{wing}} + C_{D_{tails}} + C_{D_{gear}} + C_{D_{cockpit}} = 0.0238$

(sum plus 5% for leaks & protuberances)

$C_{D_0} = 1.05(.0238) = .0250 \text{ (Aero)}$

Table 12.5 Landing gear component drags

	$D/q$ Frontal area (Ft <sup>2</sup> )
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

# Aerodynamics - 6

- Cooling Drag

$$(D/q)_{\text{cooling}} = (4.9 \times 10^{-7}) \frac{\text{bhp} \cdot T^2}{\sigma V}, \text{ ft}^2$$

$$D/q_{\text{cooling}} = (4.9 \times 10^{-7}) \frac{150 (519)^2}{V} = \frac{19.8}{V} \text{ ft}^2$$

so  $C_{D0_{\text{cooling}}} \approx .0024$  at  $V=115$  kts (small, so ignore change at other speeds)

- Misc Engine Drag:

- Incluye la refrigeración del aceite, la toma de aire, los tubos de escape, y el resto de partes.
- Para tomas de refrigeración y el resto de partes bien diseñados se puede estimar su resistencia parasitaria con:

$$(D/q)_{\text{cooling}} = (4.9 \times 10^{-7}) \frac{\text{bhp} \cdot T^2}{\sigma V}, \text{ ft}^2$$

$T$  = air temperature, deg Rankine  
 $V$  = velocity in ft/s

$$(D/q)_{\text{misc}} = (2 \times 10^{-4}) \text{ bhp}, \text{ ft}^2$$

- Mientras que para un avión ligero, este valor estimado se puede hasta triplicar

$$\text{eg 13.19) } D/q_{\text{misc}} = (2 \times 10^{-4}) 150 = .03 \text{ ft}^2$$

$$C_{D0_{\text{misc}}} = .03/118 = .0003$$

TOTAL PARASITE AND ENGINE DRAG:

$$\underline{\underline{C_{D0} = .0250 + .0024 + .0003 = .0277}}$$

# Aerodynamics - 7

- 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

$$\text{Uncambered: } C_D = C_{D_0} + KC_L^2$$

$$\text{Cambered: } C_D = C_{D_{\min}} + K(C_L - C_{L_{\min \text{ drag}}})^2$$

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

$$\text{Straight-Wing Aircraft: } e = 1.78(1 - 0.045A^{0.68}) - 0.64$$

$$\text{Swept-Wing Aircraft: } e = 4.61(1 - 0.045A^{0.68})(\cos\Lambda_{LE})^{0.15} - 3.1$$

$$(\Lambda_{LE} > 30 \text{ deg})$$

$$e = 1.78 \left[ 1 - 0.045(6)^{0.68} \right] - 0.64 = 0.87$$

$$K = \frac{1}{\pi(6)(0.87)} = 0.061$$

# Propulsion - 1

- Diseño a medida:
  - Hélice de madera, 2 palas, paso fijo,  $D=70$  in.

Advance Ratio:  $J = V/nD$

Power Coefficient:  $c_P = \frac{P}{\rho n^3 D^5} = \frac{550 \text{ bhp}}{\rho n^3 D^5}$

Thrust Coefficient:  $c_T = T/\rho n^2 D^4$

Speed-Power Coefficient:  $c_S = V \sqrt[5]{\rho/Pn^2}$

Activity Factor:

$$AF_{\text{per blade}} = \frac{10^5}{D^3} \int_{0.15R}^R cr^3 dr = \frac{10^5 c_{\text{root}}}{16D} [0.25 - (1 - \lambda)0.2]$$

Propeller Efficiency:  $\eta_p = \frac{TV}{P} = \frac{TV}{550 \text{ bhp}} = J \frac{c_T}{c_P}$

Thrust:  $T = \frac{550 \text{ bhp } \eta_p}{V} = \frac{c_T}{c_P} \frac{550 \text{ bhp}}{nD}$

$T$  = thrust (lb)

$V$  = velocity (ft/s)

$P$  = power (ft-lb/s)

bhp = brake horsepower

$n$  = rotation speed (rev/s)

$D$  = propeller diameter (ft)

$c$  = propeller airfoil chord (ft)



# Propulsion - 2

- On Design

$$J = \frac{V}{nD} = \frac{115 \times 1.689}{(2700/60)(70/12)} = .74$$

$$C_p = \frac{530(150)}{\rho n^3 D^5} = 0.06$$

$$\eta_p = .84 \quad \theta_{.75} = 20^\circ$$

$$\eta_p = 0.9 \times 1.03 \times (.84) = .78 \quad (\text{on-design at 115 kts})$$

• But wooden propeller reduces  $\eta_p$  by 10% , while 2-bladed is about 3% better than 3-bladed data provided by figure 13.9

- Off Design: ajustar la información en la condición de vuelo de diseño

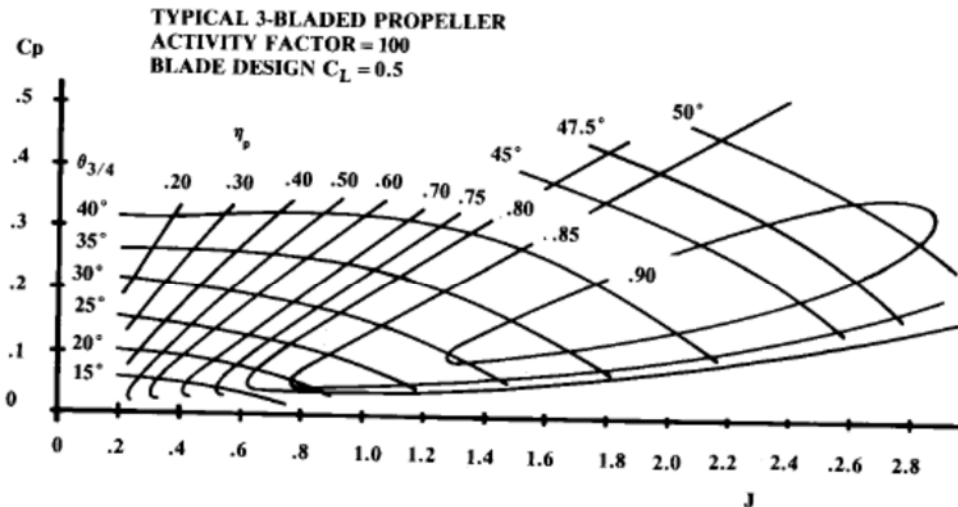


Fig. 13.9 Forward flight thrust and efficiency. (after Ref. 50)

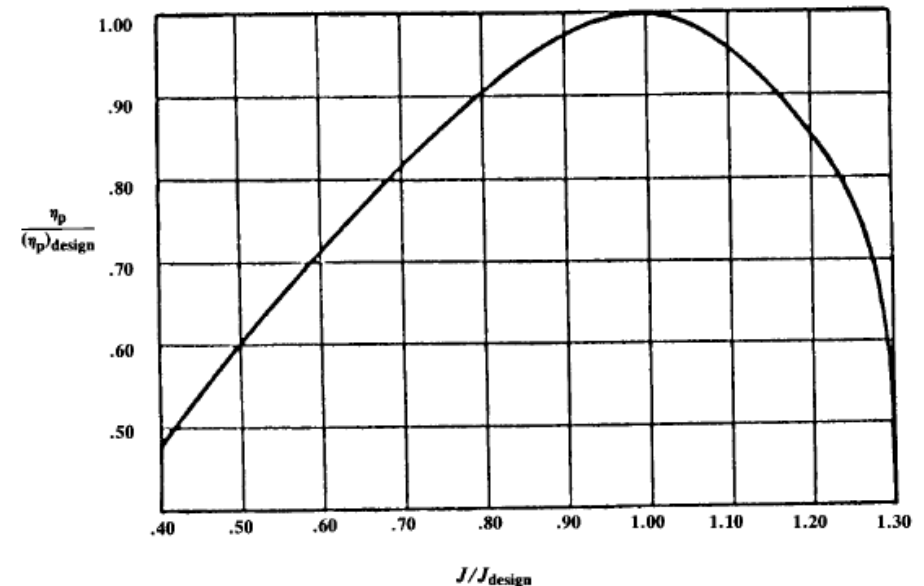


Fig. 13.10 Fixed-pitch propeller adjustment.

# Propulsion - 3

- Static Thrust:

$$C_T/C_P = 2.5$$

$$T = \frac{550 \text{ bhp } \eta_p}{V} = \frac{C_T}{C_P} \frac{550 \text{ bhp}}{nD} \quad T_{\text{Static}} = 2.5 \frac{550 (150)}{nD} = 786 \text{ Lb} \quad (\text{assuming 3-bladed})$$

• 2-bladed propeller has 5% less static thrust, thus:

$$T_{\text{Static}} = .95 \times 786 = 747 \text{ Lb}$$

- But this is for a variable pitch propeller, which changes to a flat blade angle at static conditions.
- Instead, assume the static thrust equals the highest forward thrust value found above (see plot)
- (Crude assumption - better to use fixed-pitch data!)

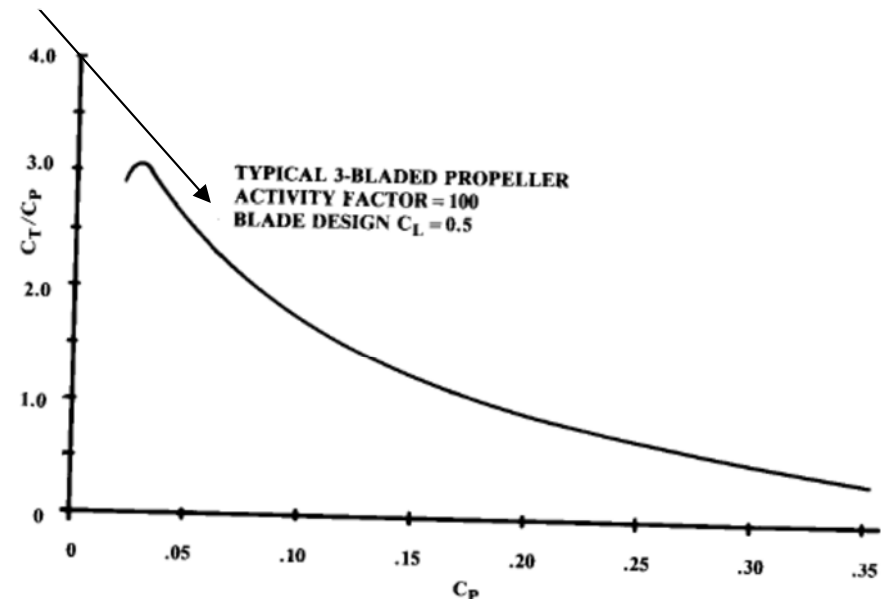
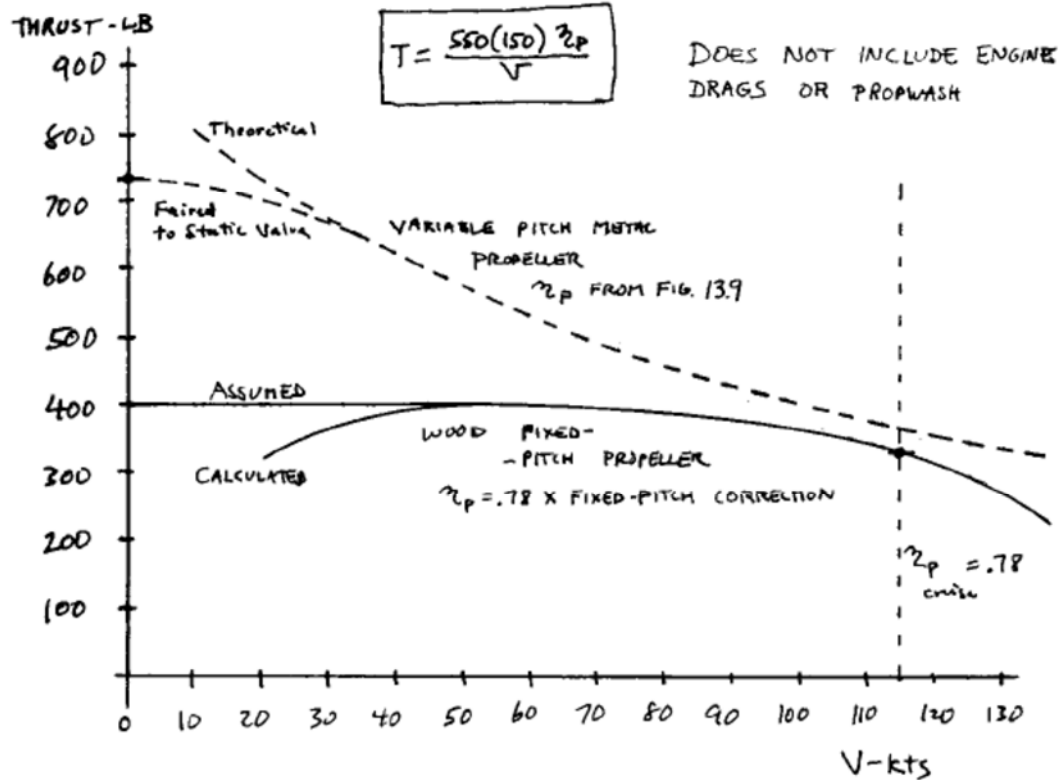


Fig. 13.8 Static propeller thrust. (after Ref. 50)

# Propulsion - 4

## THRUST v.s. VELOCITY



$$\eta_{\text{effective}} = \eta_p \left[ 1 - \frac{1.558}{D^2} \frac{\rho}{\rho_0} \Sigma(C_{f_c} S_{\text{wet}})_{\text{washed}} \right]$$

CORRECTION FOR PROPWASH DRAG EFFECT:  
 Washed Area  $\approx 265 \text{ ft}^2$

$$\eta_{\text{effective}} = \eta_p \left[ 1 - \frac{1.558}{(70/12)^2} (.004)(265) \right] = .95 \eta_p$$

Thus; Thrust<sub>actual</sub> = .95 x Thrust<sub>calculated above</sub>

# Weights - 1

- Estimación de pesos para aviación general

$$W_{\text{wing}} = 0.036 S_w^{0.758} W_{\text{fw}}^{0.0035} \left( \frac{A}{\cos^2 \Lambda} \right)^{0.6} q^{0.006} \lambda^{0.04} \left( \frac{100 t/c}{\cos \Lambda} \right)^{-0.3} (N_z W_{\text{dg}})^{0.49}$$

$$W_{\text{wing}} = .036 (118)^{.758} \left( \frac{6}{\cos^2 0} \right)^{.6} (45)^{.006} (.4)^{.04} \left( \frac{100(.135)}{\cos 0} \right)^{-.3} (9 \times 1200)^{.49} = 168 \text{ Lb}$$

$$W_{\text{horizontal tail}} = 0.016 (N_z W_{\text{dg}})^{0.414} q^{0.168} S_{\text{ht}}^{0.896} \left( \frac{100 t/c}{\cos \Lambda} \right)^{-0.12} \times \left( \frac{A}{\cos^2 \Lambda_{\text{ht}}} \right)^{0.043} \lambda_h^{-0.02}$$

$$W_{\text{ht}} = .016 (9 \times 1200)^{.414} (45)^{.168} (25.5)^{.896} \left( \frac{100(.12)}{\cos 10} \right)^{-.12} \left( \frac{4}{\cos^2 10} \right)^{.043} (.4)^{-.02} = 20 \text{ Lb}$$

$$W_{\text{vertical tail}} = 0.073 \left( 1 + 0.2 \frac{H_t}{H_v} \right) (N_z W_{\text{dg}})^{0.376} q^{0.122} S_{\text{vt}}^{0.873} \left( \frac{100 t/c}{\cos \Lambda_{\text{vt}}} \right)^{-0.49} \times \left( \frac{A}{\cos^2 \Lambda_{\text{vt}}} \right)^{0.357} \lambda_{\text{vt}}^{0.039}$$

$$W_{\text{vt}} = .073 (1 + 2(0)) (9 \times 1200)^{.376} (45)^{.122} (11.6)^{.873} \left( \frac{100(.12)}{\cos 15} \right)^{-.49} \left( \frac{4}{\cos^2 15} \right)^{.357} \times (.4)^{.039} = 15 \text{ Lb}$$

# Weights - 2

$$W_{\text{fuselage}} = 0.052 S_f^{1.086} (N_z W_{dg})^{0.177} L_t^{-0.051} (L/D)^{-0.072} q^{0.241} + W_{\text{press}}$$

$$W_{\text{fus}} = 0.052 (164)^{1.086} (9 \times 1200)^{0.177} (140)^{-0.051} \left(\frac{190}{25}\right)^{-0.072} (45)^{0.241} = 115 \text{ Lb}$$

$$W_{\text{main landing gear}} = 0.095 (N_l W_l)^{0.768} (L_m/12)^{0.409}$$

$$W_{\text{main gear}} = 0.095 (3 \times 1200)^{0.768} (25/12)^{0.409} = 69 \text{ Lb}$$

$$W_{\text{installed engine (total)}} = 2.575 W_{\text{en}}^{0.922} N_{\text{en}}$$

$$W_{\text{engine installed}} = 2.575 (272)^{0.922} = 452 \text{ Lb} \text{ - This seems}$$

too high; table 15.2 gives  $W_{\text{inst. eng}} = 1.3(272) = 380 \text{ Lb}$

$$W_{\text{fuel system}} = 2.49 V_i^{0.726} \left(\frac{1}{1 + V_i/V_l}\right)^{0.363} N_i^{0.242} N_{\text{en}}^{0.157}$$

$$W_{\text{fuel system}} = 2.49 (20)^{0.726} \left(\frac{1}{1+0/1}\right)^{0.363} = 22 \text{ Lb}$$

# Weights - 3

**Table 15.2 Approximate empty weight buildup**

Item	Fighters	Transports and bombers	General aviation	Multiplier <sup>a</sup>	Approximate location
Wing	9.0	10.0	2.5	$S_{\text{exposed planform ft}^2}$	40% MAC
Horizontal tail	4.0	5.5	2.0	$S_{\text{exposed planform ft}^2}$	40% MAC
Vertical tail	5.3	5.5	2.0	$S_{\text{exposed planform ft}^2}$	40% MAC
Fuselage	4.8	5.0	1.4	$S_{\text{wetted area ft}^2}$	40-50% length
Landing gear <sup>b</sup>	.033	.043	.057	TOGW (lb)	—
	.045 Navy				
Installed engine	1.3	1.3	1.4	Engine weight (lb)	—
“All-else empty”	.17	.17	.10	TOGW (lb)	40-50% length

<sup>a</sup>Results are in pounds.

<sup>b</sup>15% to nose gear; 85% to main gear.

# Weights - 4

$$W_{\text{flight controls}} = 0.053 L^{1.536} B_w^{0.371} (N_z W_{dg} \times 10^{-4})^{0.80}$$

$$W_{\text{flight controls}} = 0.053 \left(\frac{190}{12}\right)^{1.536} \left(\frac{26.6}{12}\right)^{0.371} (9 \times 1200 \times 10^{-4})^{0.8} = 5 \text{ Lb}$$

$$W_{\text{electrical}} = 12.57 (W_{\text{fuel system}} + W_{\text{avionics}})^{0.51}$$

$$W_{\text{electrical}} = 12.57 (22 + W_{av})^{0.51} = 73 \text{ Lb}$$

↳ from next equation

$$W_{\text{avionics}} = 2.117 W_{\text{uav}}^{0.933}$$

$$W_{av} = 2.117 W_{\text{uav}}^{0.933} = 2.117 (5)^{0.933} = 9.5 \text{ Lb}$$

↳ From data in Ref. 11

$$W_{\text{furnishings}} \cong 20 \text{ Lb (From data in Ref. 11)}$$

# Weights - 5

## WEIGHTS BY OTHER METHODS

### CESSNA METHODS (Ref. 11)

$$W_{\text{wing}} = .047 W_0^{.397} S^{.36} h^{.397} A^{1.712} = 225 \text{ Lb}$$

$$W_{\text{ht}} = .055 W_0^{.887} S_h^{.101} A_h^{.138} x_{\text{root}}^{-.223} = 60 \text{ Lb}$$

$$W_{\text{vt}} = .108 W_0^{.567} S_v^{.125} A_v^{.482} x_{\text{root}}^{-.747} (\cos \Lambda_{z/a})^{-.882} = 17.7 \text{ Lb}$$

### Ref 10 METHOD

$$W_{\text{fus}} = 200 \left[ \left( \frac{W_0 D}{10^5} \right)^{.286} \left( \frac{L}{10} \right)^{.857} \left( \frac{W+D}{10} \right) \left( \frac{V_e}{100} \right)^{.338} \right]^{1.1} = 114 \text{ Lb}$$

w/o nacelle

$$W_{\text{nacelle}} = 2.5 \sqrt{H_p} = 31 \text{ Lb}$$

### COMPARISON TO ACTUAL DATA (Ref. 11)

$$W_{\text{electrical}} \cong 40 \text{ Lb}$$

$$W_{\text{gear}} = \left( \frac{W_{\text{gear}}}{W_0} \right) W_0 \cong .054 (1200) = 64 \text{ Lb}$$

↳ Average of C-180 and L-19A values



# Weights - 6

## WEIGHTS ADJUSTMENTS & BALANCE

As discussed in Chapter 3, fiberglass homebuilts have reduced weight due to design differences, not due to composite construction. Nonetheless, we will use the factors in table 15.4 to estimate the weight savings for each component.

COMPONENT	FUDGE FACTOR	ADJUSTED WEIGHT: Ch15/Other-Methods	SELECTED WEIGHT	DISTANCE TO DATUM*
FUSELAGE	.90	104 / 128	130 Lb	115 in
WING	.85	143 / 175	160	70
HOR. TAIL	.83	17 / 45	40	210
VERT. TAIL	.83	12 / 13	15	225
ENGINE	-	452 / 380	380	16
GEAR	.95	66 / 57	60	45
FUEL SYS.	-	22	22	50
FL. CONTROLS	-	5	5	80
ELECTRICAL	-	73 / 40	40	40
AVIONICS	-	9.5	10	60
FURNISHINGS	-	20	20	100

$$\Sigma W_e = 882 \text{ Lb @ } 59.5 \text{ in}$$

PILOT & CHUTE	220	85
FUEL (Available, if $W_0 = 1200 \text{ Lb}$ )	98	50

$$\Sigma W_0 = 1200 \text{ Lb @ } 63.3 \text{ in}$$

MOST-AFT C.G IS NO-FUEL:  $W_{e+PILOT} = 1102 \text{ Lb @ } 64.5 \text{ in}$

\* Measured from back of spinner - see drawing

Table 15.4 Weights estimation "fudge factors"

Category	Weight group	Fudge factor (multiplier)
Advanced composites	Wing	0.85
	Tails	0.83
	Fuselage/nacelle	0.90
	Landing gear	0.95
	Air induction system	0.85
Braced wing	Wing	0.82
Wood fuselage	Fuselage	1.60
Steel tube fuselage	Fuselage	1.80
Flying boat hull	Fuselage	1.25

# Definiciones Geométricas - 1

$$\bar{c} = \frac{2}{S} \int_0^{b/2} c^2 dy$$

$$\bar{x} = \frac{2}{C_L S} \int_0^{b/2} C_{l_a} c x dy$$

$$\bar{y} = \frac{2}{C_L S} \int_0^{b/2} C_{l_a} c y dy = \eta_{cp} \frac{b}{2}$$

$$\bar{z} = \frac{2}{C_L S} \int_0^{b/2} C_{l_a} c z dy$$

where  $b$  = wing span

$c$  = local chord

$C_L$  = total lift coefficient

$C_{l_a}$  = local additional lift coefficient, proportional to  $C_L$

$C_{l_b}$  = local basic lift coefficient, independent of  $C_L$

$C_l = C_{l_b} + C_{l_a}$  = total local lift coefficient

$m_{ac}$  = pitching moment, per unit span, about aerodynamic center (Fig. C.4)

$S$  = wing area

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

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

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

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

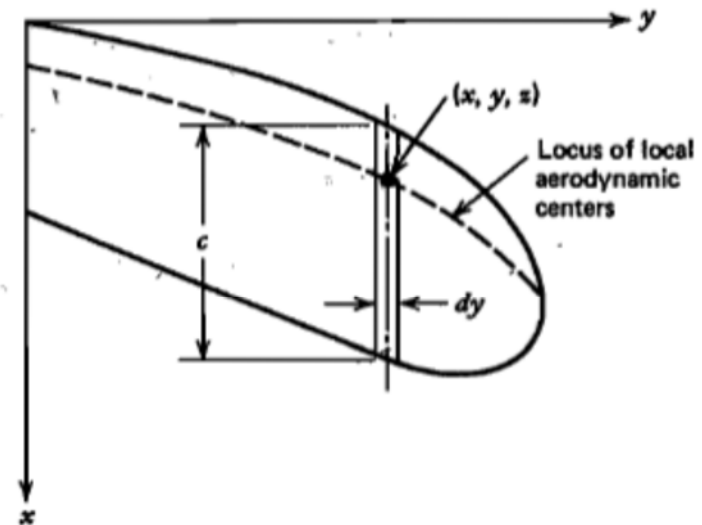


Figure C.1 Local aerodynamic center coordinates.

# Definiciones Geométricas - 2

- Para flujos ideales en 2D se asume que  $n=0.25$  para subsónico, y  $n=0.5$  para supersónico.

$$\bar{x} = nc_r + \bar{y} \tan \Lambda_n$$

where  $c_r$  = wing root chord

$\Lambda_n$  = sweepback of  $n$ th-chord line, degrees

$$\bar{x} = \frac{2}{C_L S} \int_0^{b/2} C_{L\alpha} c x dy$$

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

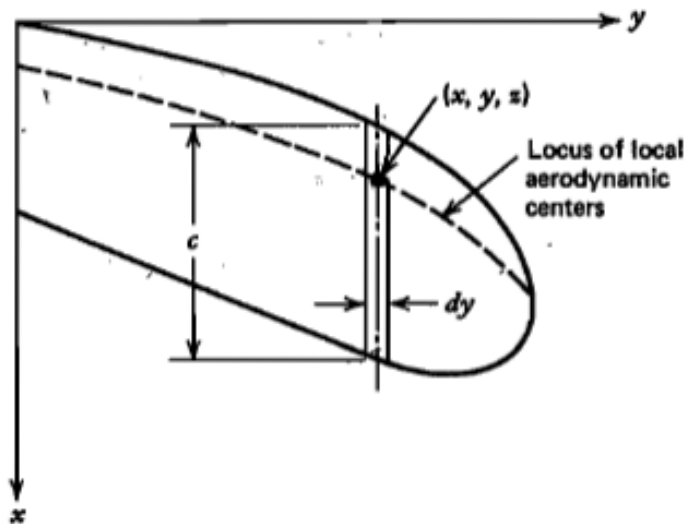


Figure C.1 Local aerodynamic center coordinates.

# Definiciones Geométricas - 3

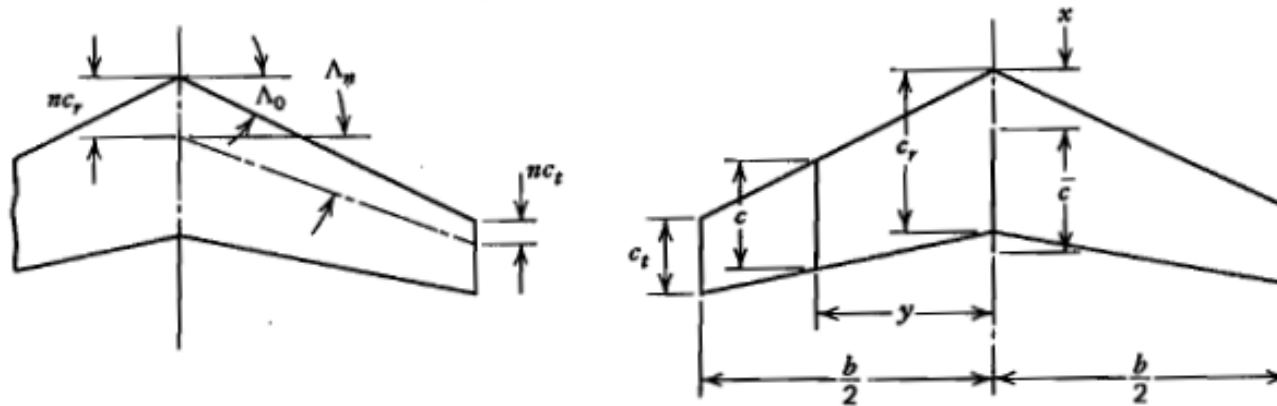
- Comparación del m.a. chord y m.a. center para diferentes alas y distribuciones de carga sustentadora.

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

Table C.1

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

# Definiciones Geométricas - 4



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

$$= \frac{1+2\lambda}{12} c_r A \tan \Lambda_0$$

$\Lambda_0$  = sweepback of wing leading edge, degrees.

$$A \tan \Lambda_0 = A \tan \Lambda_n + 4n \frac{1-\lambda}{1+\lambda}$$

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

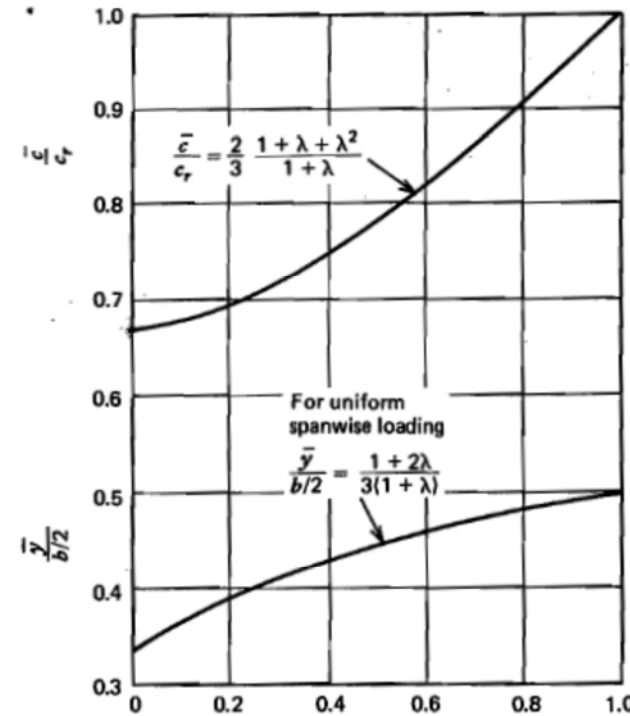


Figure C.2 Mean aerodynamic chord for straight tapered wings; and spanwise position of mean aerodynamic center for uniform spanwise loading (i.e., constant  $C_{L\alpha}$ ). (From "Notes on the Mean Aerodynamic Chord and the Mean Aerodynamic Center of a Wing" by A. H. Yates, *J. Roy. Aero. Soc.*, June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

# Stability and Control - 1

- Partiendo de la información generada previamente tanto de las dimensiones de la cuerda media (mean aerodynamic chord) como de la posición más atrasada del centro de gravedad (condición sin combustible  $X_{cg}=64.5$  in.)
  - calculamos el valor normalizado del centro de gravedad

$$\bar{c} = 56 \text{ in}$$

MOST-AFT C.G. IS AT 64.5 in

$$\bar{x}_{cg} = \frac{64.5}{56} = 1.15$$

- Determinación de las características del ala:
  - El centro aerodinámico está ubicado a 62 in.

$$\bar{x}_{acw} = \frac{62}{56} = 1.107$$

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

$$C_{L\alpha} = 4.85 \text{ per radian}$$

$$C_{m_w} = 0 \text{ (since airfoil is symmetric)}$$

# Stability and Control - 2

- Determinación de las características del fuselaje:
  - $W_f$  = máxima anchura del fuselaje o góndola.
  - $L_f$  = Largo del fuselaje

$$C_{m\alpha_{\text{fuselage}}} = \frac{K_f W_f^2 L_f}{c S_w}, \text{ per deg}$$

$$C_{m\alpha_{\text{fus}}} = \frac{(0.008)(35)(234)}{(56)(118 \times 12^2)} = .002 \text{ per deg.}$$
$$= .12 \text{ per rad.}$$

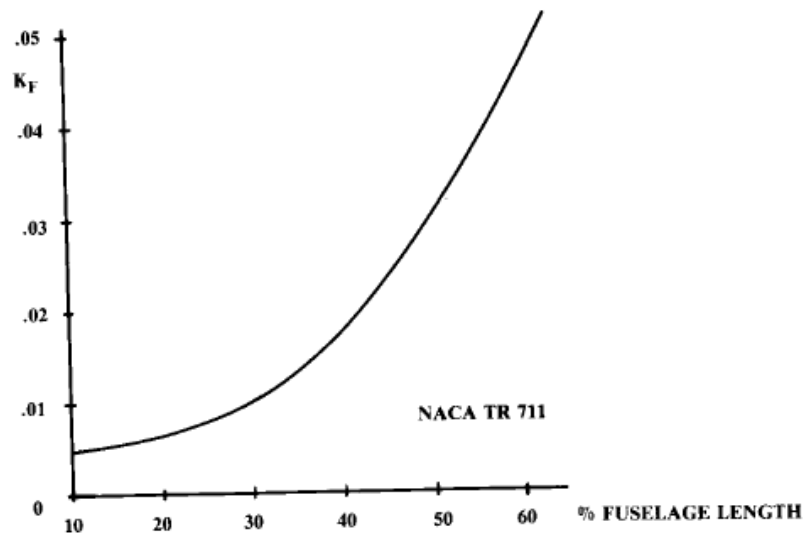


Fig. 16.14 Position of  $\frac{1}{4}$  root chord.

# Stability and Control - 3

- Determinación de las características de la cola:
  - Centro aerodinámico:

AERO. CENTER AT 206 in so  $\bar{X}_{ach} = \frac{206}{56} = 3.68$

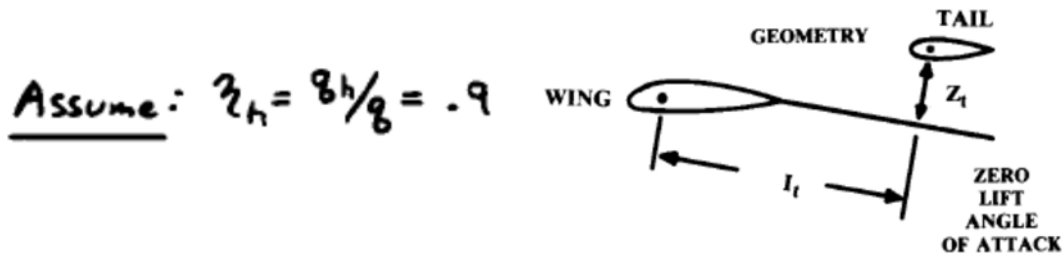
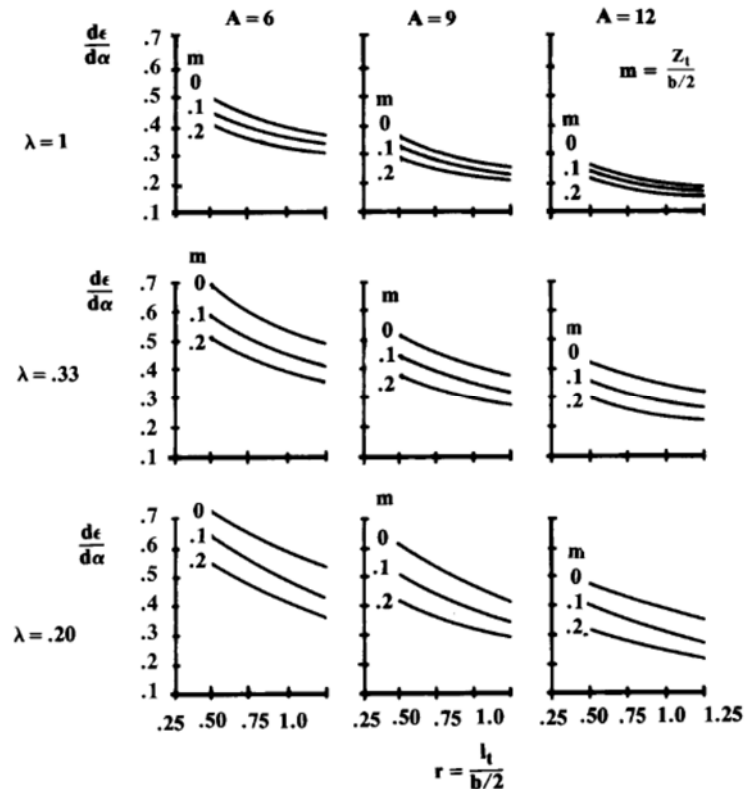
$$C_{L\alpha} = \frac{2\pi A}{2 + \sqrt{4 + \frac{A^2 \beta^2}{\eta^2} \left(1 + \frac{\tan^2 \Lambda_{max f}}{\beta^2}\right)}} \left(\frac{S_{exposed}}{S_{ref}}\right) (F)$$

$C_{L\alpha_h} = 3.77$  per radian

- Downwash:

$$\left. \begin{aligned} r &= \frac{144}{319.3/2} = .9 \\ m &= \frac{28}{319.3/2} = .2 \end{aligned} \right\} \frac{d\epsilon}{d\alpha} = .38$$

eg 16.20)  $\frac{d\alpha_h}{d\alpha} = 1 - .38 = .62$





# Estimación del Downwash - 1

- El flujo de las alas tiene una gran influencia en el flujo del resto de los elementos aerodinámicos del avión.
- Aguas arriba del ala el aire (en vuelo subsónico) es elevado debido a las presiones sobre el ala:
  - Empuja el flujo en la parte inferior del fuselaje.
  - Gira el flujo previo a alcanzar la hélice o la entrada de flujo de la góndola.
- Aguas abajo del ala, el flujo tiene una dirección descendente teórica equivalente al ángulo de ataque del ala justo en el borde de salida, todo y que ese valor desciende hasta aproximadamente la mitad del ángulo de ataque aguas abajo del ala.
- El efecto del *downwash* es la reducción efectiva del ángulo de ataque de la cola, contribuyendo de esa manera al momento de cabeceo del fuselaje.
- El chorro de los motores afecta en gran medida al downwash

$$\text{Upwash: } \frac{\partial \alpha_u}{\partial \alpha} = 1 + \frac{\partial \epsilon_u}{\partial \alpha}$$

$$\text{Downwash: } \frac{\partial \alpha_h}{\partial \alpha} = 1 - \frac{\partial \epsilon}{\partial \alpha}$$

$$\alpha_h = (\alpha + i_w) \left( 1 - \frac{\partial \epsilon}{\partial \alpha} \right) + (i_h - i_w)$$

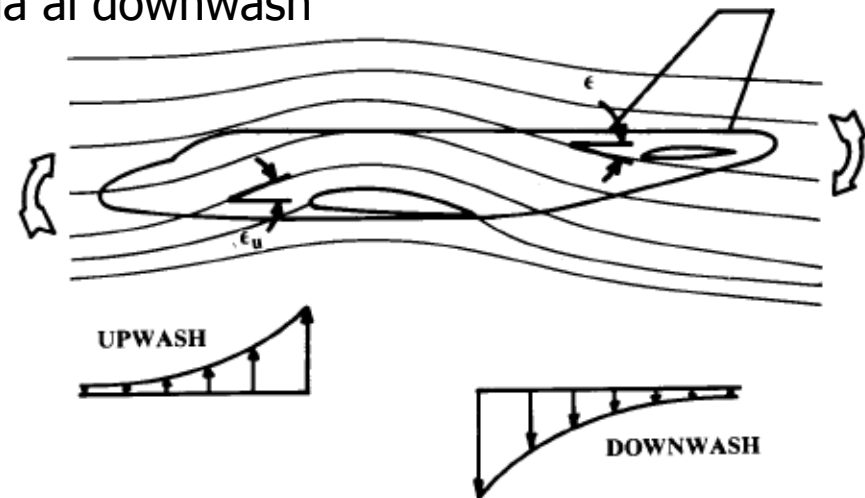


Fig. 16.10 Wing flowfield effect on pitching moment.

# Estimación del Downwash - 2

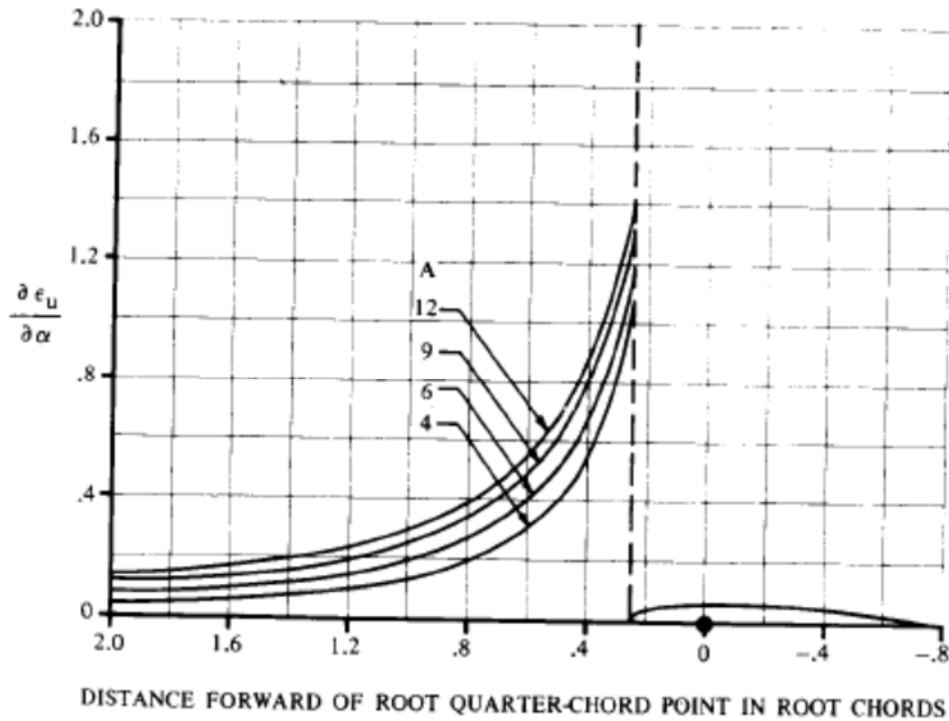
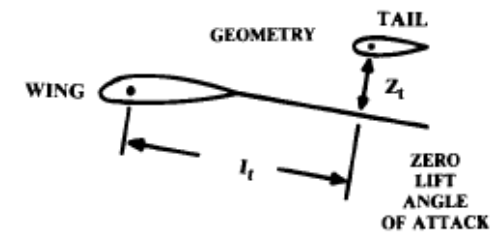
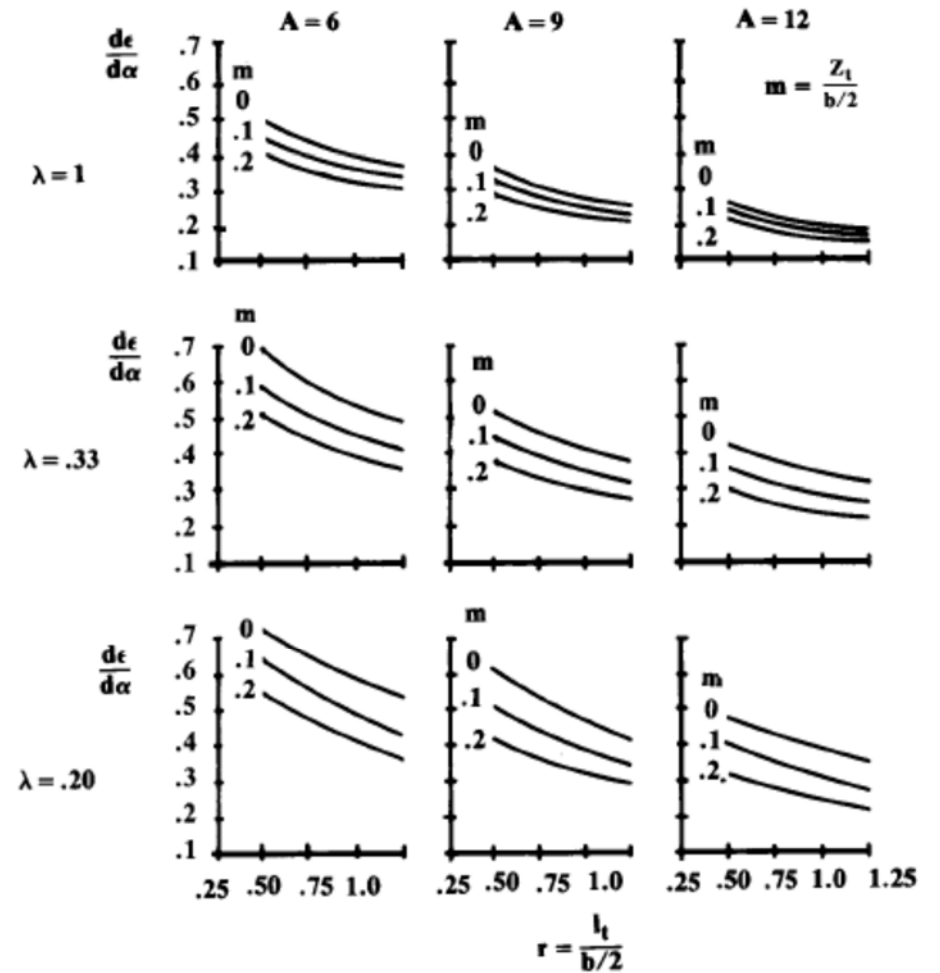


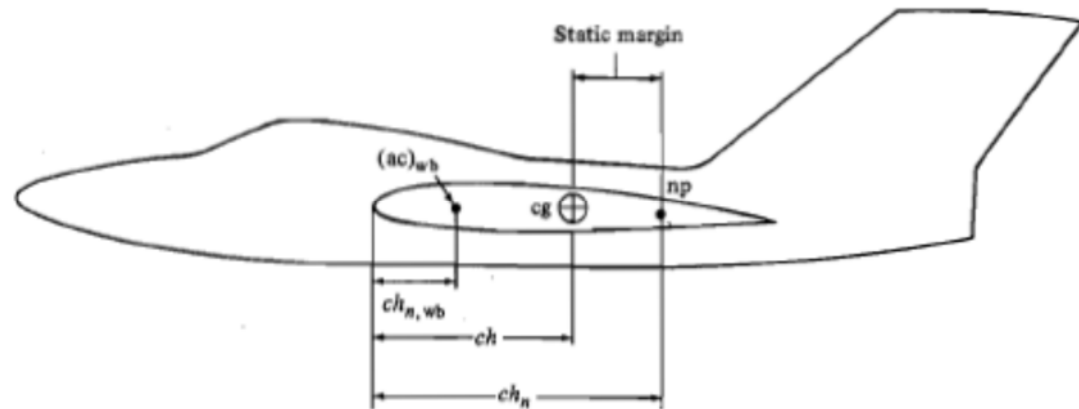
Fig. 16.11 Upwash estimation (subsonic only). (Ref. 37)



# Punto Neutral

- El punto neutral es el centro aerodinámico del avión, el cual corresponde al centro de acción de todas las fuerzas aerodinámicas.
- Al igual que el centro aerodinámico del ala, el centro aerodinámico del avión es el lugar donde el momento de cabeceo del perfil es constante para cualquier ángulo de ataque es decir:
- Es uno de los conceptos más importantes en la estabilidad de un avión.
- Una vez que el avión se ha construido, está fijo, y no puede ser cambiado, lo único que se puede variar es el C.G.
- Importancia del centro de gravedad del avión para obtener un avión estable

$$X_{NA} \longrightarrow C_{M_\alpha} = 0$$



$$\bar{X}_{NA} = \frac{\frac{q_c S_c}{q S} C_{L_{\alpha c}} \left(1 - \frac{\partial \epsilon_c}{\partial \alpha}\right) \bar{X}_{AC_c} + \frac{q_t S_t}{q S} C_{L_{\alpha t}} \left(1 - \frac{\partial \epsilon_t}{\partial \alpha}\right) \bar{X}_{AC_t} + C_{L_{\alpha WB}} \bar{X}_{AC_w}}{\frac{q_c S_c}{q S} C_{L_{\alpha c}} \left(1 - \frac{\partial \epsilon_c}{\partial \alpha}\right) + \frac{q_t S_t}{q S} C_{L_{\alpha t}} \left(1 - \frac{\partial \epsilon_t}{\partial \alpha}\right) + C_{L_{\alpha WB}}}$$

# Cálculo del punto neutro - 1

- Cálculo del punto neutro se realiza sin potencia:

$$\bar{X}_{np} = \frac{C_{L_\alpha} \bar{X}_{acw} - C_{m_{\alpha_{fus}}} + \eta_h \frac{S_h}{S_w} C_{L_{\alpha_h}} \frac{\partial \alpha_h}{\partial \alpha} \bar{X}_{ach} + \frac{F_{p_\alpha}}{qS_w} \frac{\partial \alpha_p}{\partial \alpha} \bar{X}_p}{C_{L_\alpha} + \eta_h \frac{S_h}{S_w} C_{L_{\alpha_h}} \frac{\partial \alpha_h}{\partial \alpha} + \frac{F_{p_\alpha}}{qS_w}} \quad \bar{X}_{np} = \frac{(4.85 \cdot 1.107) - 0.12 + 0.9 \cdot \frac{25.5}{118} (3.77)(0.62)(3.68)}{4.85 + 0.9 \cdot \frac{25.5}{118} (3.77)(0.62)} = 1.27$$

$$X_{np} = 1.27 \times 56 = 71.2 \quad \text{Static Margin } SM = \frac{71.2 - 64.5}{56} = 0.12 \rightarrow 12 \%$$

$$C_{m_\alpha} = -C_{L_\alpha}(\bar{X}_{np} - \bar{X}_{cg}) \quad C_{m_\alpha} = -0.58$$

- Stick Free análisis: para la mayoría de aviones actuales, los actuadores están de alguna manera fijados en el ala, eso quiere decir que ,los servomecanismos mantienen la posición de los actuadores teniendo que ejercer fuerzas para mantener los actuadores en su posición.
  - Para aviones pequeños, en los que no se dispone de un sistema totalmente mecanizado y automatizado para la gestión de las superficies de control, por lo que solo se ejerce fuerza en las superficies cuando es necesario, por lo que dichas superficies "flotan"
  - Para estos casos es necesario el analizar la estabilidad que se denomina como "stick free" en la que el hecho que los elevadores estén flotando reduce la sustentación efectiva de la deriva horizontal un % equivalente al 50% del % de la superficie de la cola que esta formada por elevador

Elevator area  $\cong$  40% of Tail area,

so  $C_{L_{\alpha_h}} \cong .8 C_{L_{\alpha_h}}(\text{fixed})$

# Cálculo del punto neutro - 2

$$\bar{X}_{np} = \frac{(4.85 \cdot 1.107) - 0.12 + 0.8 \left[ 0.9 \cdot \frac{25.5}{118} (3.77)(0.62)(3.68) \right]}{4.85 + 0.9 \frac{25.5}{118} (3.77)(0.62)} = 1.26$$

$$X_{np} = 1.26 \times 56 = 70.7 \text{ in} \quad \text{Static Margin } SM = \frac{70.7 - 64.5}{56} = 0.111 \rightarrow 11\%$$

$$C_{m_\alpha} = -C_{L_\alpha}(\bar{X}_{np} - \bar{X}_{cg}) \quad C_{m_\alpha} = -0.54$$

- Quizás demasiado estable para competición

# Trim Analysis - 1

$$C_{m_{cg}} = C_L(\overline{X_{cg}} - \overline{X_{acw}}) + C_{m_w} + C_{m_w \delta_f} \delta_f + C_{m_{fus}} - \eta_h \frac{S_h}{S_w} C_{L_h} (\overline{X_{ach}} - \overline{X_{cg}}) - \frac{T}{qS_w} \overline{Z_t} + \frac{F_p}{qS_w} (\overline{X_{cg}} - \overline{X_p})$$

$$C_{m_{cg}} = 0 = 4.85\alpha(1.15 - 1.107) + 0 + 0 + 0.12\alpha - \left[ 0.9 \left( \frac{11.6}{118} \right) C_{L_h} (3.68 - 1.15) \right] \quad C_{m_{cg}} = 0.329\alpha - 0.492C_{L_h}$$

$$C_{L_h} = C_{L_{\alpha_h}} \left[ (\alpha + i_w) \left( 1 - \frac{\partial \epsilon}{\partial \alpha} \right) + (i_h - i_w) - \alpha_{OL_h} \right]$$

$$C_{L_h} = C_{L_{\alpha_h}} \left[ 0.62\alpha + (0 - 0) - \Delta\alpha_{OL} \right]$$

↳ due to flap (ie, elevator)

$$\Delta\alpha_{OL} = -\frac{1}{C_{L_{\alpha}}} \frac{\partial C_L}{\partial \delta_f} \delta_f$$

$$\frac{\partial C_L}{\partial \delta_f} = 0.9 K_f \left( \frac{\partial C_{\ell}}{\partial \delta_f} \right) \frac{S_{flapped}}{S_{ref}} \cos \Lambda_{H.L.}$$

$$\Delta\alpha_{OL} = -\frac{0.9}{(0.95)2\pi} (5.3)(1)(1) K_f \delta_e = -0.8\delta_e$$

$$C_{m_{cg}} = 0.329\alpha - 1.48\delta_e$$

$$C_{L_h} = C_{L_{\alpha_h}} \left[ (\alpha + i_w) \left( 1 - \frac{\partial \epsilon}{\partial \alpha} \right) + (i_h - i_w) - \alpha_{OL_h} \right]$$

$$C_{L_{total}} = 4.85\alpha + 0.9 \frac{25.5}{118} (3.77) (0.62\alpha + 0.8\delta_e)$$

$$C_{L_{total}} = C_{L_{\alpha}} [\alpha + i_w] + \eta_h \frac{S_h}{S_w} C_{L_h}$$

# Trim Analysis -2

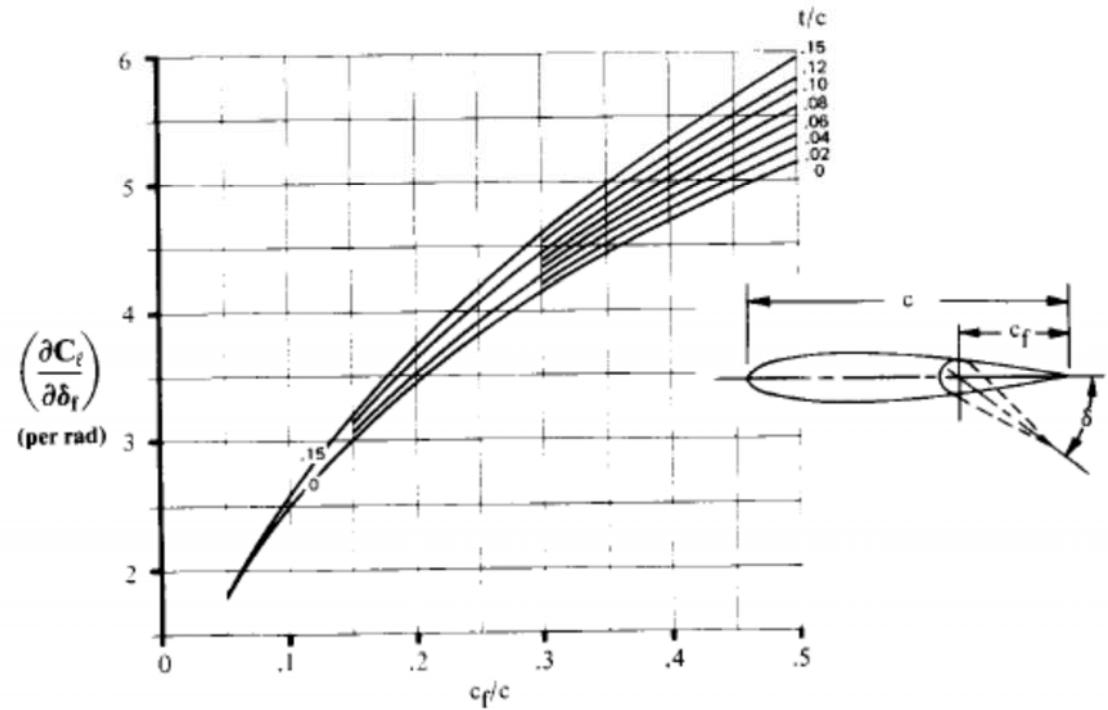
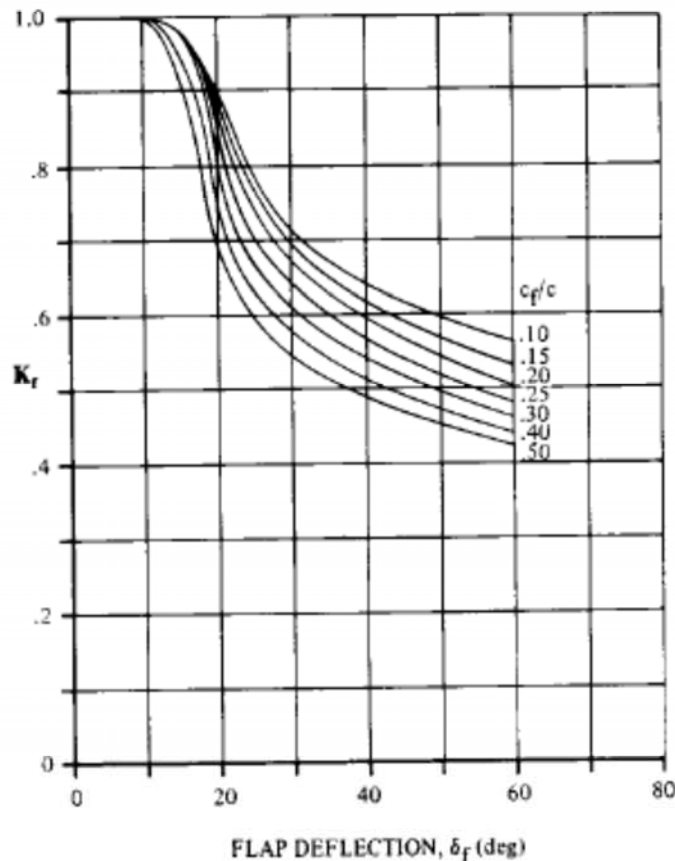


Fig. 16.6 Theoretical lift increment for plain flaps. (Ref. 37)

$$\Delta\alpha_{0L} = -\frac{1}{C_{L\alpha}} \frac{\partial C_L}{\partial \delta_f} \delta_f$$

$$\frac{\partial C_L}{\partial \delta_f} = 0.9 K_f \left(\frac{\partial C_l}{\partial \delta_f}\right)' \frac{S_{\text{flapped}}}{S_{\text{ref}}} \cos\Lambda_{H.L.}$$

Fig. 16.7 Empirical correction for plain lift increment. (Ref. 37)

# Trim Analysis -3

## TRIM PLOT

Vary  $\alpha$  and  $\delta_e$ , find  $C_m$  and  $C_L$

$\alpha$	$\delta_e$			
	$0^\circ$	$-2^\circ$	$-4^\circ$	$-6^\circ$
$0^\circ$	0; 0	.015; -.006	.030; -.01	.045; -.018
$2^\circ$	-.007; .18	.008; .179	.023; .170	.038; .16
$4^\circ$	-.014; .35	.001; .344	.016; .34	.031; .33
$6^\circ$	-.021; .53	-.006; .52	.009; .515	.024; .51
$8^\circ$	-.028; .71	-.012; .70	.002; .695	.017; .69
$10^\circ$	-.034; .89	-.019; .88	-.004; .875	.011; .87

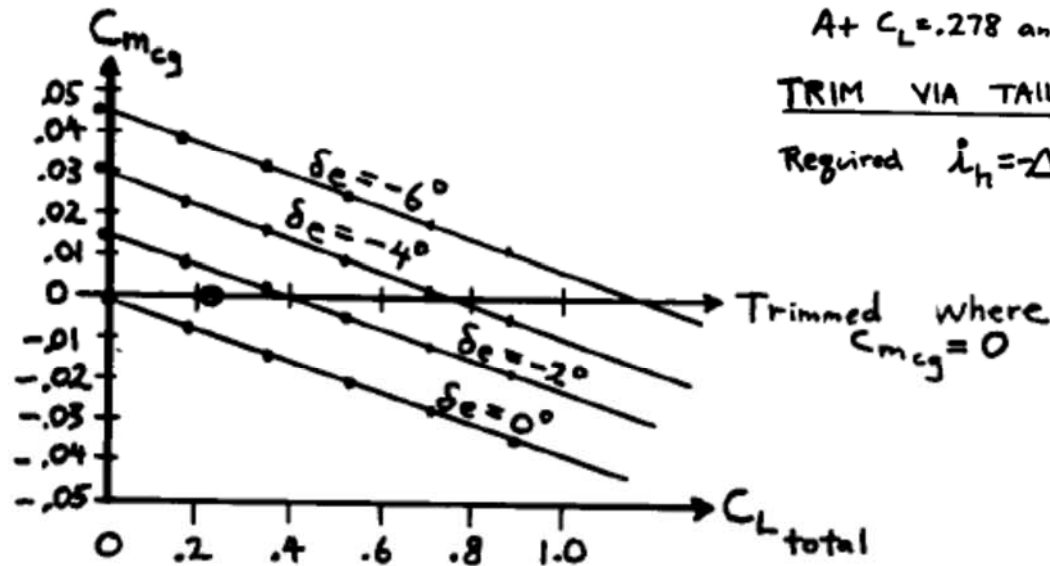
Values are  $\{C_{m_{cg}}; C_{L_{total}}\}$

CRUISE TRIM  $q = 35 \text{ lb/ft}^2$  so  $C_L = \frac{W/S}{q} = \frac{9.7}{35} = .278$

At  $C_L = .278$  and  $C_{m_{cg}} = 0$ ,  $\delta_e \approx -1.2^\circ$  (Circled point above)

TRIM VIA TAIL INCIDENCE ONLY

Required  $i_h = \Delta \alpha_{OL} = -(-.51 \delta_e) = -(-.51)(-1.2^\circ) = -.61^\circ$



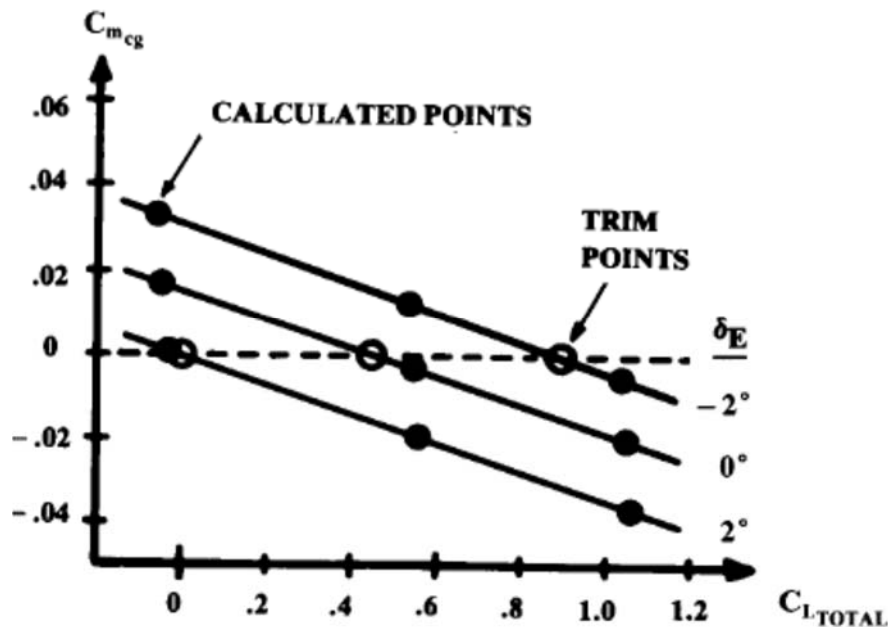


# Trim Analysis -4

Calculation table

	$\delta_E = -2^\circ$	$0^\circ$	$2^\circ$
$\alpha = 0^\circ$	$C_{m_{cg}} = 0.033$ $C_{L_{TOTAL}} = -0.07$	0.018 -0.05	0.002 -0.03
$\alpha = 5^\circ$	$C_{m_{cg}} = 0.012$ $C_{L_{TOTAL}} = 0.53$	-0.004 0.54	-0.02 0.56
$\alpha = 10^\circ$	$C_{m_{cg}} = -0.005$ $C_{L_{TOTAL}} = 1.03$	-0.021 1.04	-0.038 1.06

Trim crossplot



(Note: Positive  $\delta_E$  as defined produces an upload on the tail).

- Una vez calculada la configuración de trimado, y esta ser razonable, se puede pasar a calcular la componente de la resistencia inducida asociada a la configuración de trimado.
  - Se asume que la deriva horizontal se comporta como un ala, con lo que se estima la eficiencia aerodinámica de la deriva horizontal con métodos simplificados

$$C_{Di_{trimmed}} = K[C_{L_\alpha}(\alpha + i_w)]^2 + \eta_h \frac{S_h}{S_w} K_h [C_{L_h}]^2$$

# Análisis de Trimado – Alternativo - 1

- Para un vuelo trimado, que es equivalente a un vuelo estable, se requiere que la suma de fuerzas tiene que asegurar que no haya momentos alrededor del centro de gravedad del avión es decir que las fuerzas de sustentación compensen las fuerzas gravitatorias.
- Para cada segmento de vuelo debemos asegurarnos que las condiciones de vuelo (trimado) satisfacen estas condiciones.
- Si dichas condiciones no son satisfechas (diferente carga alar a medida que volamos) podemos satisfacer las ecuaciones mediante la variación de la sustentación de la cola, por lo que a su vez el ángulo de ataque del avión deberá cambiar.
- Esto implica un proceso iterativo para determinar cuales son las sustentaciones de la cola (deflexión de cola) y el ángulo de ataque del ala necesarias para satisfacer las ecuaciones de  $\Sigma M=0$

$$\Sigma F_x = W - L = \frac{W}{qS} - C_{L_0} - C_{L_\alpha}\alpha - C_{L_\delta}\delta$$
$$\Sigma M = 0 = C_{M_0} - C_{M_\alpha}\alpha - C_{M_\delta}$$

# Análisis de Trimado – Alternativo - 2

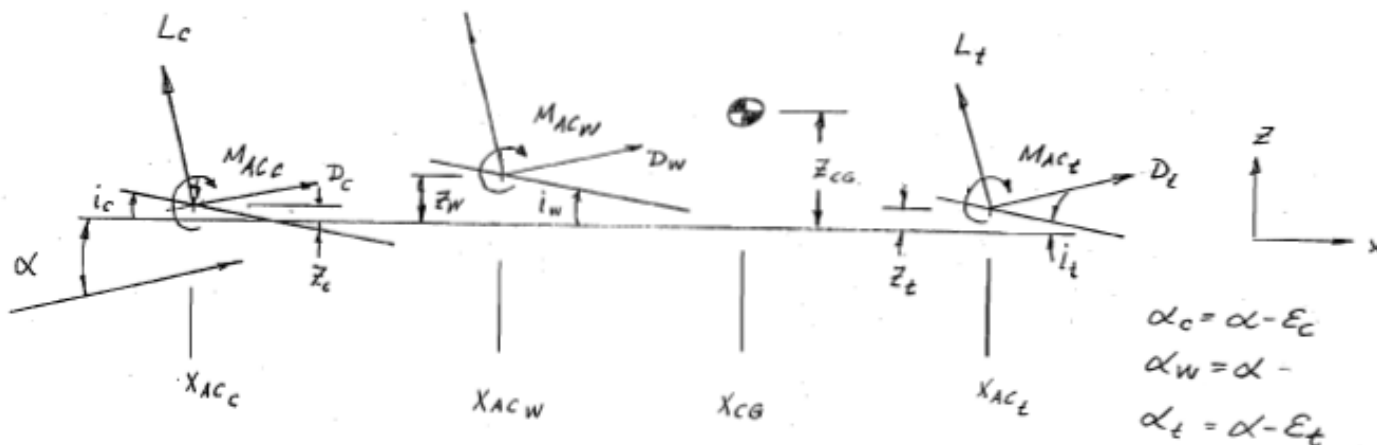
$$\Sigma F_x = W - L = \frac{W}{qS} - C_{L0} - C_{L\alpha}\alpha - C_{L\delta}\delta$$

$$\Sigma M = 0 = C_{M0} - C_{M\alpha}\alpha - C_{M\delta}$$

$$C_{L0} = \frac{q_c S_c}{q S} C_{L0c} + C_{L0WB} + \frac{q_t S_t}{q S} C_{L0t}$$

$$+ \frac{q_c S_c}{q S} C_{L\alpha c} (i_c - \epsilon_{0c}) + \frac{q_t S_t}{q S} C_{L\alpha t} (i_t - \epsilon_{0t}) + C_{L\alpha WB} i_w$$

$$C_{L\alpha} = \frac{q_c S_c}{q S} C_{L\alpha c} \left(1 - \frac{\partial \epsilon_c}{\partial \alpha}\right) + \frac{q_t S_t}{q S} C_{L\alpha t} \left(1 - \frac{\partial \epsilon_t}{\partial \alpha}\right) + C_{L\alpha WB}$$



$$C_{L\delta e} = C_{L\delta c} + C_{L\delta t}$$

$$C_{L\delta c} = \frac{q_c S_c}{q S} C_{Lc\delta e}$$

$$C_{L\delta t} = \frac{q_t S_t}{q S} C_{Lt\delta e}$$

# Análisis de Trimado – Alternativo - 3

$$\Sigma F_x = W - L = \frac{W}{qS} - C_{L0} - C_{L\alpha}\alpha - C_{L\delta}\delta$$

$$\Sigma M = 0 = C_{M0} - C_{M\alpha}\alpha - C_{M\delta}\delta$$

$$C_{MA} = C_{M0} + C_{M\alpha}\alpha$$

$$C_{M0} = \frac{q_c S_c \bar{c}_c}{q S \bar{c}} C_{MAC_c} + \frac{q_c S_c}{q S} (\bar{X}_{CG} - \bar{X}_{AC_c}) (C_{L0_c} + C_{L\alpha_c} (i_c - \epsilon_{oc}))$$

$$+ C_{M_{p0}} + C_{MAC_w} + (\bar{X}_{CG} - \bar{X}_{AC_w}) (C_{L0_w} + C_{L\alpha_w} i_w)$$

$$+ \frac{q_t S_t \bar{c}_t}{q S \bar{c}} C_{MAC_t} + \frac{q_t S_t}{q S} (\bar{X}_{CG} - \bar{X}_{AC_t}) (C_{L0_t} + C_{L\alpha_t} (i_t - \epsilon_{ot}))$$

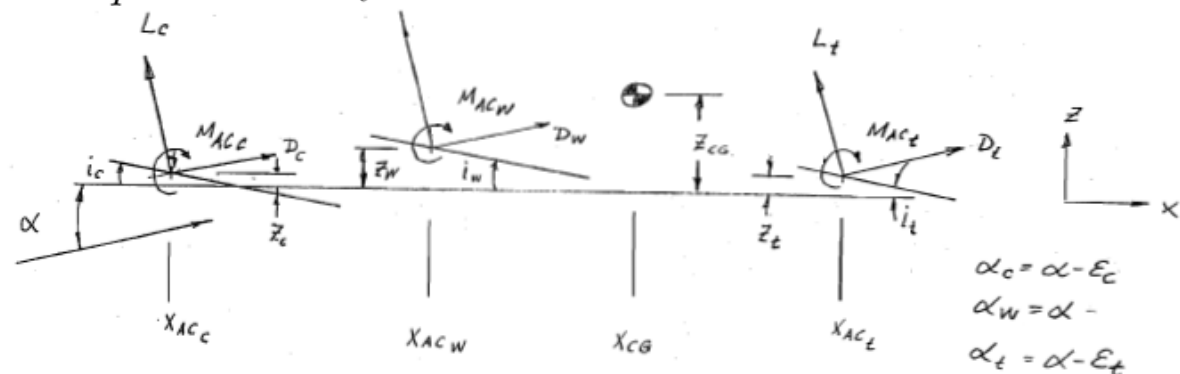
$$C_{M\alpha} = C_{L\alpha} (\bar{X}_{CG} - \bar{X}_{NA})$$

$$C_{L\alpha} = \frac{q_c S_c}{q S} C_{L\alpha_c} (1 - \frac{\partial \epsilon_c}{\partial \alpha}) + \frac{q_t S_t}{q S} C_{L\alpha_t} (1 - \frac{\partial \epsilon_t}{\partial \alpha}) + C_{L\alpha_{WB}}$$

$$C_{M\delta} = C_{M\delta_c} + C_{M\delta_t}$$

$$C_{M\delta_c} = \frac{q_c S_c}{q S} (\bar{X}_{CG} - \bar{X}_{AC_c}) C_{L\delta_c} + \frac{q_c S_c \bar{C}_c}{q S \bar{C}} C_{M\delta_{AC_c}}$$

$$C_{M\delta_t} = \frac{q_t S_t}{q S} (\bar{X}_{CG} - \bar{X}_{AC_t}) C_{L\delta_t} + \frac{q_t S_t \bar{C}_t}{q S \bar{C}} C_{M\delta_{AC_t}}$$



# Spin Recovery - 1

- Después de la entrada en pérdida, un tirabuzón (entrada en barrena) se producirá tanto en un buen diseño como en uno malo, la diferencia radica en la facilidad para recuperarse del tirabuzón.
- Si definimos el ala y el fuselaje como barrar con una masa al final de cada ellas, podemos caracterizar el efecto del tirabuzón sobre el comportamiento del avión.
  - Las fuerzas centrífugas que actúan en el fuselaje tienden a levantar el morro del avión, lo que incrementa aún más la entrada en pérdida del avión.
  - El tirabuzón está principalmente dirigido por la diferencia de sustentación existente entre el ala exterior (más rápida) y la interior (más lenta), siendo esta última, al ser la más lenta, en la que el efecto de la entrada en pérdida se magnifica.
  - Al tirabuzón se oponen las fuerzas amortiguadoras asociadas a partes del fuselaje y deriva vertical por debajo del fuselaje

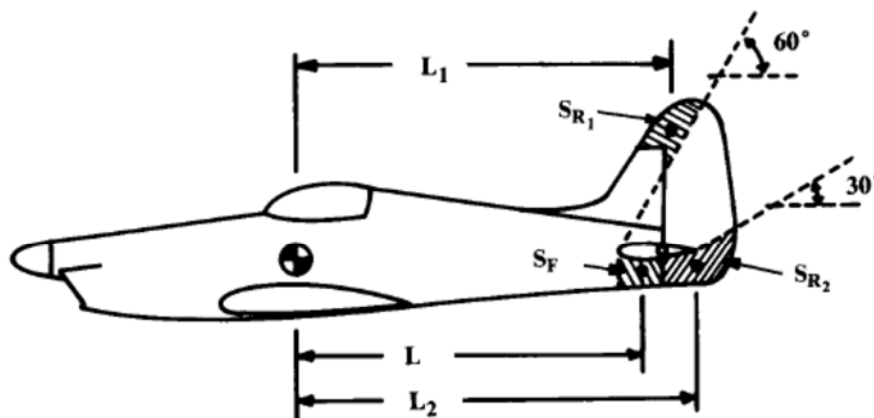


Fig. 16.30 Geometry for spin recovery estimation.

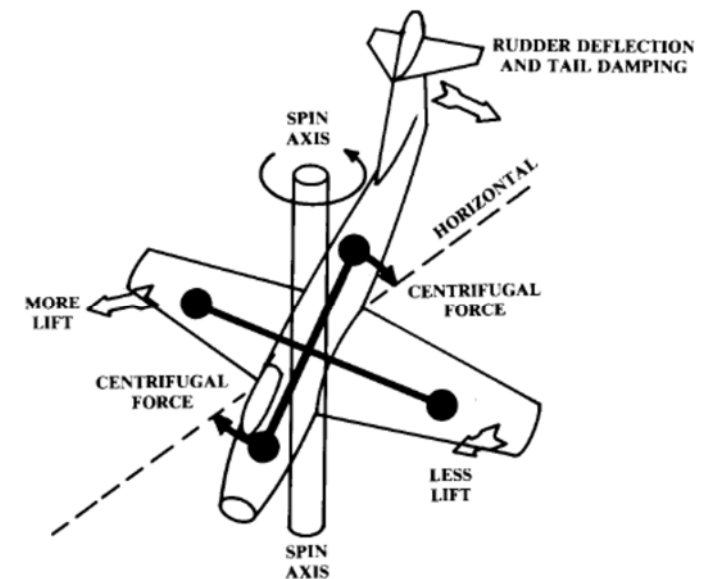


Fig. 16.29 Forces in spin.

# Spin Recovery - 2

- Para recuperarse de un tirabuzón, el timón de cola se deflexiona en el sentido contrario del tirabuzón.
  - Todo y que tan solo una parte del timón es efectiva, ya que al haber entrado en pérdida, una porción grande del timón de cola está cubierta por el el flujo de aire de entrada en pérdida
  - Se requiere un determinado amortiguamiento de la cola, el cual se puede cuantificar con las superficies de la cola que no está cubiertas y de esta manera el asegurar que la eficiencia del timón de cola será suficiente para contrarrestar el movimiento del tirabuzón.
    - De igual forma se puede retrasar el efecto adverso del tirabuzón, o aumentar el efecto de recuperación de este mediante el rediseño del borde de entrada del ala para minimizar el desequilibrio de la sustentación en las alas 3D en punta del ala mediante un borde de entrada caído en punta del ala, pero conlleva aumento en la resistencia global
- Existe una estimación empírica de los valores de amortiguamiento recomendados (NACA Report TN 1042) que establece unos valores aconsejables para una buena recuperación del tirabuzón:
  - Requiere estimación de los momentos de inercia, que son muy importantes para poder determinar la tendencia que tiene el avión a resistirse a las aceleraciones

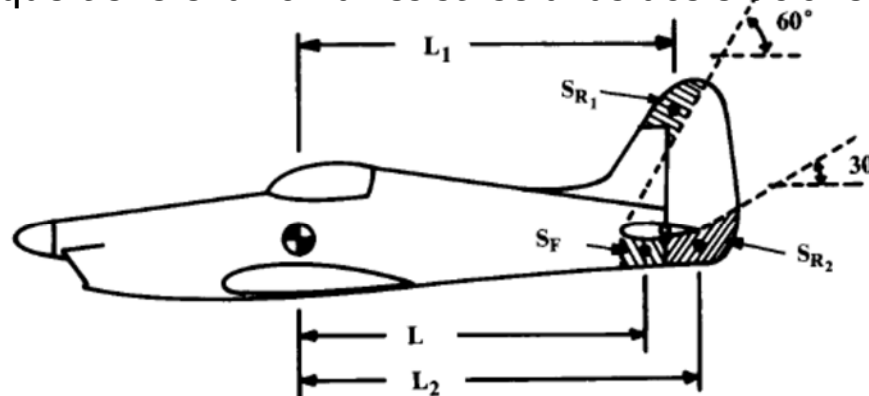


Fig. 16.30 Geometry for spin recovery estimation.

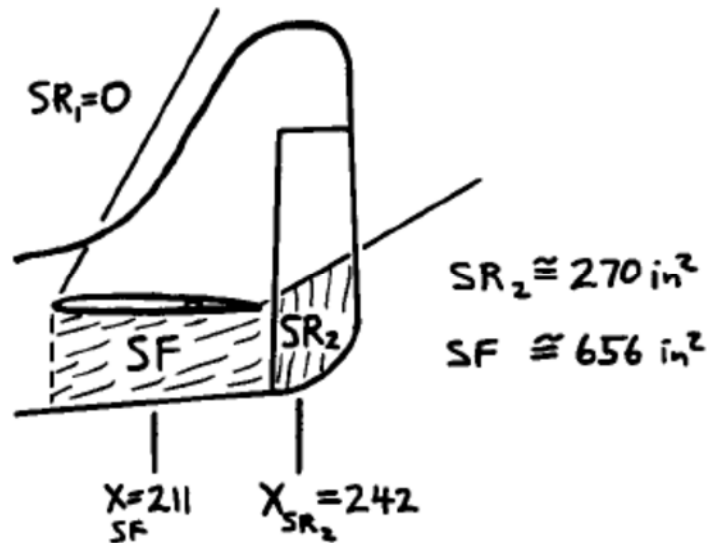
# Estimación de los momentos de Inercia

## SPIN RECOVERY

(Fig 16.30)

Wing span  
 $b = 3193 \text{ in} = 266 \text{ ft}$

Fuselage length  
 $L = 230 \text{ in} = 19.2 \text{ ft}$



$$I_{xx} = \frac{b^2 W \bar{R}_x^2}{4g}$$

$$I_{yy} = \frac{L^2 W \bar{R}_y^2}{4g}$$

$$I_{zz} = \left( \frac{b + L}{2} \right)^2 \frac{W \bar{R}_z^2}{4g}$$

$$I_{xx} \cong (26.6)^2 (1200)(.25)^2 / 4g = 412 \text{ slug-ft}^2$$

$$I_{yy} \cong (19.2)^2 (1200)(.38)^2 / 4g = 496 \text{ slug-ft}^2$$

Table 16.1 Nondimensional radii of gyration<sup>a</sup>

Aircraft class	$\bar{R}_x$	$\bar{R}_y$	$\bar{R}_z$
Single-engine prop	0.25	0.38	0.39
Twin-engine prop	0.34	0.29	0.44
Business jet twin	0.30	0.30	0.43
Twin turboprop transport	0.22	0.34	0.38
Jet transport—Fuselage-mounted engines	0.24	0.36	0.44
—2 wing-mounted engines	0.25	0.38	0.46
—4 wing-mounted engines	0.31	0.33	0.45
Military jet trainer	0.22	0.14	0.25
Jet fighter	0.23	0.38	0.52
Jet heavy bomber	0.34	0.31	0.47
Flying wing (B-49 type)	0.32	0.32	0.51
Flying boat	0.25	0.32	0.41

<sup>a</sup>Typical values see Ref. 11 for examples.

# Spin Recovery - 3

$$TDPF = (TDR)(URVC)$$

$$TDR = \frac{S_F L^2}{S_w (b/2)^2}$$

$$URVC = \frac{S_{R1} L_1 + S_{R2} L_2}{S_w (b/2)}$$

$$\mu = \frac{W/S}{\rho g b}$$

$$TDR = \frac{656(146.5)^2}{(118 \times 12^2)(319.3/2)^2} = 0.0325$$

$$URVC = \frac{0 + 270(177.5)}{(118 \times 12^2)(319.3/2)} = 0.0177$$

$$.0325 \times .0177 = .00057 = \underline{5.7 \times 10^{-4}} \text{ (Actual)}$$

- TDPF = Tail Power Damping Factor
- TDR = Tail Damping Ration
- URVC = Unshielded Rudder Volume Coefficient
- $\mu$  = airplane relative density parameter

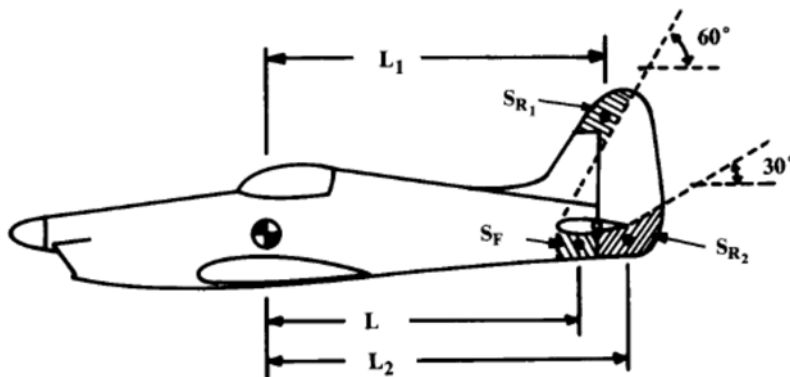


Fig. 16.30 Geometry for spin recovery estimation.

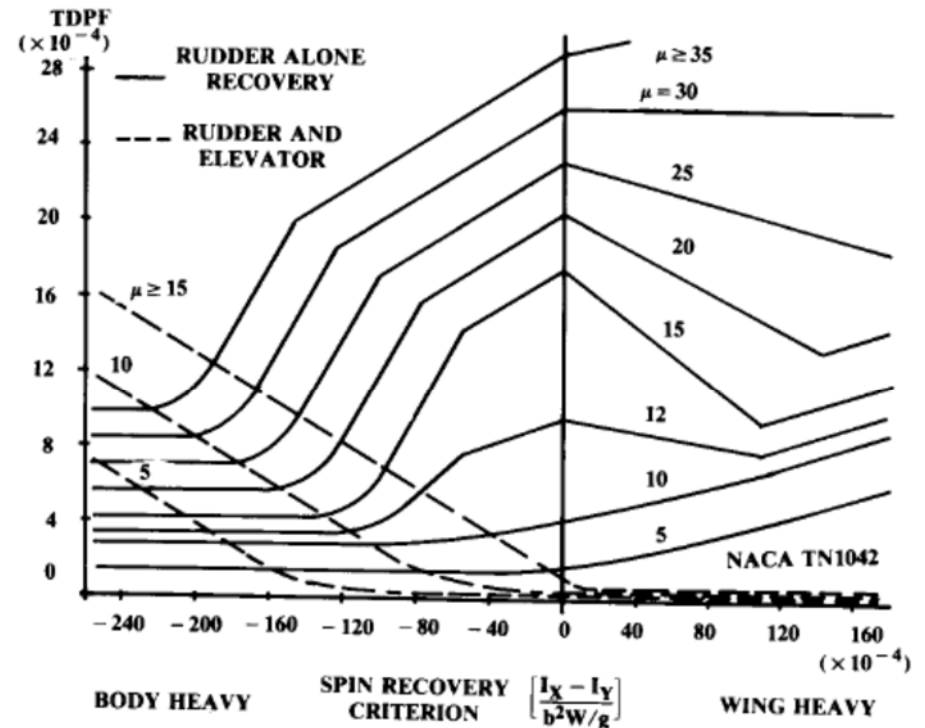


Fig. 16.31 Spin recovery criteria.



# Performance - 1

- Partiendo del diseño dibujado inicialmente, las actuaciones se definen:

$$\text{STALL: } V_{\text{Stall}} = \sqrt{\frac{2W/S}{\rho C_{L_{\text{max}}}}} = 84.5 \text{ ft/sec} = 50 \text{ KTS}$$

$$(W/S) = (\text{TOP}) \sigma C_{L_{\text{TO}}} (\text{hp}/W) \quad \text{TOP} = \frac{10.2}{(.99)(1/8)} = 82.4 \quad \text{Fig 5.4) } S_{\text{TO}} \cong 900 \text{ ft}$$

- Potencia y Empuje ajustado para altitud

$$\text{bhp} = \text{bhp}_{\text{SL}} \left( \frac{\rho}{\rho_0} - \frac{1 - \rho/\rho_0}{7.55} \right)$$

$$\text{bhp} = \text{bhp}_{\text{SL}} \left( \rho/\rho_0 - \frac{1 - \rho/\rho_0}{7.55} \right) = 0.76 \text{ bhp}_{\text{SL}} \quad (h=8000 \text{ ft})$$

$$C_L = \frac{W/S}{q} = \frac{10.2}{q}$$

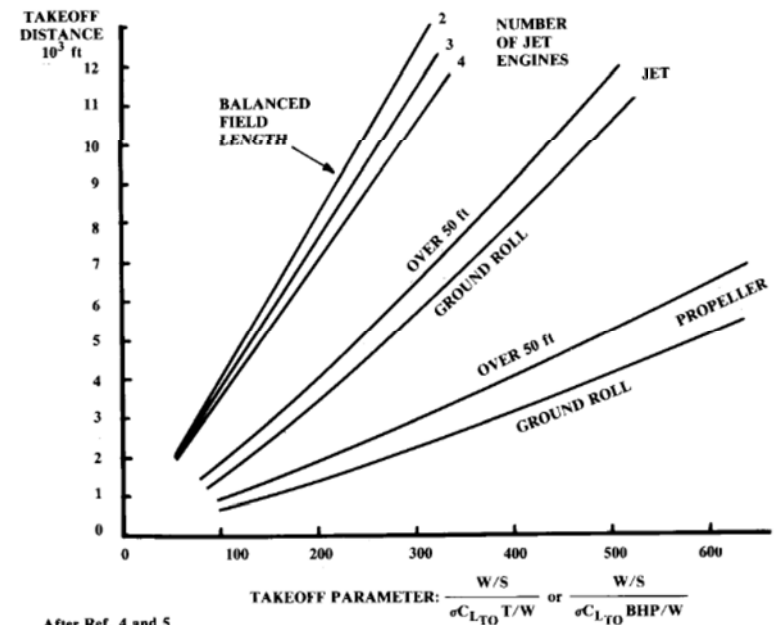
$$C_D = 0.0277 + 0.061 C_L^2$$

$$D = 118 q C_D$$

$$V_0 = V \sin \gamma = V \left( \frac{T-D}{W} \right) \cong V \left( \frac{T}{W} - \frac{1}{L/D} \right)$$

$$V_V = V \left( \frac{T-D}{W} \right)$$

- empuje corregido con altitud y velocidad



After Ref. 4 and 5

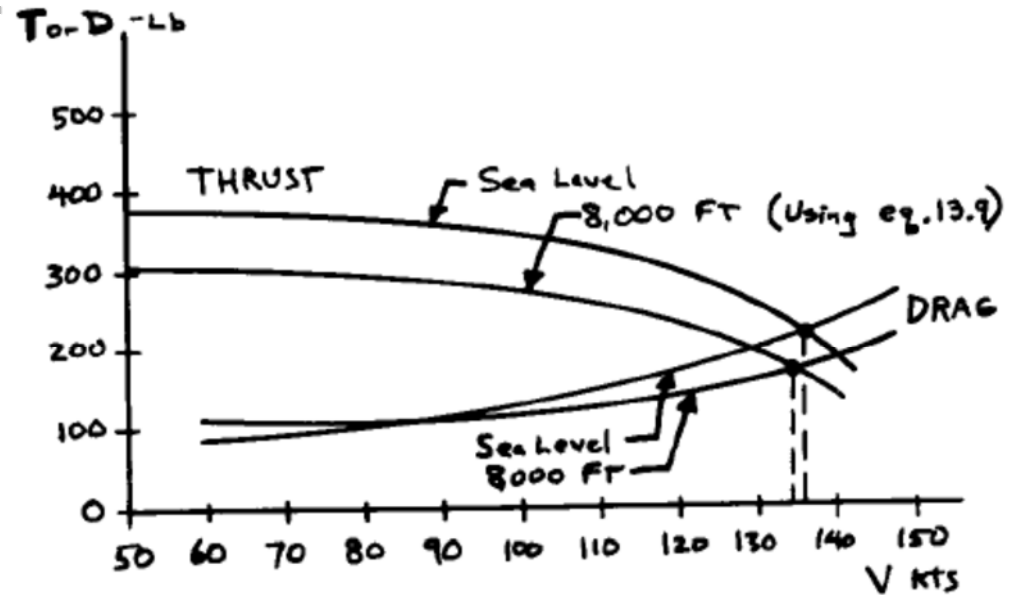
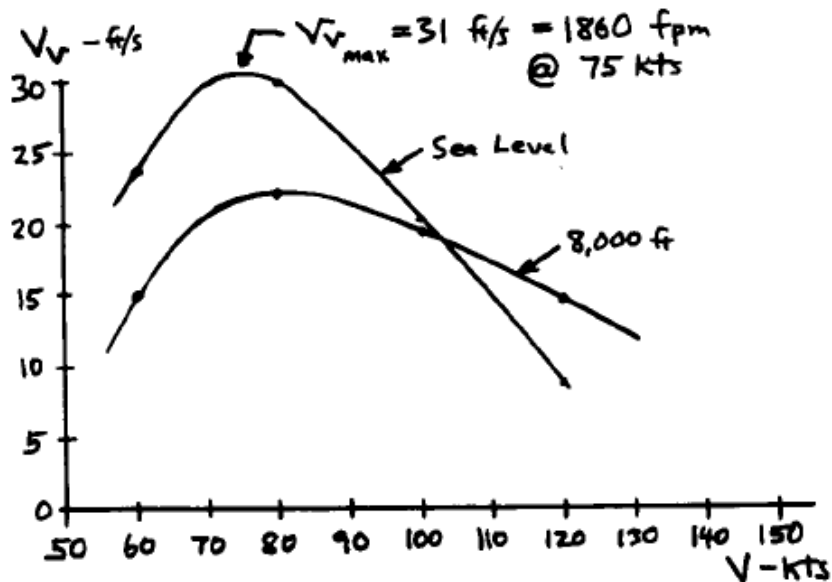
Fig. 5.4 Takeoff distance estimation.

# Performance - 2

$$bhp = bhp_{SL} \left( \frac{\rho}{\rho_0} - \frac{1 - \rho/\rho_0}{7.55} \right)$$

$$V_v = V \sin \gamma = V \left( \frac{T-D}{W} \right) \cong V \left( \frac{T}{W} - \frac{1}{L/D} \right)$$

## RATE OF CLIMB



So  $V_{max} = 136$  kts @ Sea Level  
 $= 134$  kts @ 8,000 ft

$V$ kts	$q - \text{Lb/ft}^2$		$C_L$		$C_D$		$D - \text{Lb}$		$V_v - \text{ft/s}$	
	SL	/8000 ft	SL	/8000	SL	/8000	SL	/8000	SL	/8000
60	12	9.4	.85	1.09	.07	.10	99	111	23	15
80	22	17	.46	.60	.04	.05	104	100	30	22
100	34	27	.30	.38	.033	.036	132	115	20	18
120	49	38	.21	.27	.030	.032	173	143	9	15
140	67	53	.15	.19	.029	.03	230	187	—	—

# Refined Sizing - 1

- Después de los cálculos previos, las características del avión son:

$$C_{D_0} = 0.0277$$

$$e = 0.87$$

$$c_{bhp} = 0.5 \text{ per hour}$$

$$T = 0.95T \text{ (sea level)}$$

$$W_0 = 1200 \text{ lbs}$$

$$W_e = 883 \text{ lbs}$$

$$W_{crew} = 220 \text{ lbs}$$

- Mission Segments Weight Fractions:

- Warmup and Takeoff:

$$\frac{W_1}{W_0} = 0.97$$

- Parece excesivo si se compara con la fracción de crucero

$$\frac{W_3}{W_2} = 0.953$$

- Se asume que son 5 minutos en máxima potencia:

$$W_f = c_{bhp}(8Hp)\left(\frac{5}{60}\right) = 0.5 \cdot 150 \cdot \frac{5}{60} = 6.25 \text{ lbs}$$

$$\frac{W_1}{W_0} = \frac{1200 - 6.25}{1200} = 0.995$$

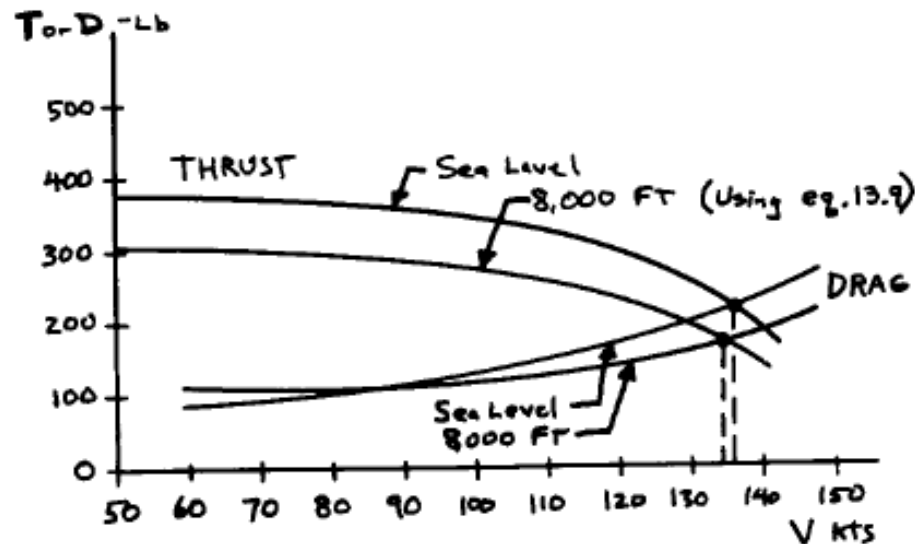
- Climb and Acceleration: basada en la gráfica de subida, subida a 80 knts hasta 8000 ft y despues acelerar hasta la velocidad de crucero (115 knts)

$$\Delta h_e = \Delta\left(h + \frac{1}{2g} V^2\right) \quad \Delta h_e = \left(8000 + \frac{1}{2g}(115 \times 1.689)^2\right) - \left(0 + \frac{1}{2g}(80 \times 1.689)^2\right) = 8302 \text{ ft}$$

$$C = C_{bhp} \frac{V}{550 \eta_p} \quad C = (0.5) \frac{80 \times 1.689}{550 (0.78 \times 0.95)} = 0.17 \text{ per hr}$$

# Refined Sizing - 2

- Mission Segments Weight Fractions (cont):
  - Warmup and Takeoff (cont):
    - A partir del gráfico generado previamente para el empuje y la resistencia para varias velocidades y altitudes de pueden determinar el empuje (T) y la resistencia (D) a nivel del mar y a la altitud de crucero, para las velocidades al principio del segmento de subida (60 knots) y la velocidad de crucero



$$V_{max} = 136 \text{ kts @ Sea Level} \\ = 134 \text{ kts @ 8,000 ft}$$

$$\frac{D}{T} = \frac{100}{360} = 0.278 \text{ a nivel del mar}$$

$$\frac{D}{T} = \frac{105}{290} = 0.362 \text{ a 8000 ft } \frac{W_1}{W_0} = 0.97$$

$$\frac{W_{i+1}}{W_i} = \exp \left[ \frac{-C \Delta h_e}{V(1 - D/T)} \right] \quad \frac{W_2}{W_1} = e^{\frac{-(.17/3600)(8302)}{(80 \times 1.689)(1 - .32)}} = \underline{.996}$$

# Refined Sizing - 3

## ■ Mission Segments Weight Fractions (cont):

- Crucero: Se requiere un alcance de 280 nm a una velocidad de 115 kts, y una altura de 8000 ft.

$$\left(\frac{W}{S}\right)_{cruise} = \left(\frac{W}{S}\right)_{stall} \times \frac{W_1}{W_0} \times \frac{W_2}{W_1} = 10.2 \times 0.995 \times 0.996$$

$$\left(\frac{L}{D}\right)_{cruise} = \frac{1}{\frac{qC_{D0}}{W/S_{cruise}} + \frac{1}{q\pi Ae} \left(\frac{W}{S}\right)_{cruise}} = \frac{1}{\frac{35 \times 0.0271}{10.1} + \frac{10.1}{35\pi \cdot 6 \cdot 0.87}} = 8.804$$

$$\text{for cruise } \frac{W_3}{W_2} = \exp\left[\frac{-R \cdot c}{V \left(\frac{L}{D}\right)_{cruise}}\right] \quad \frac{W_3}{W_2} = e\left[-\frac{280 \cdot 6076}{8.8 \times 550 \times (0.78 \cdot 0.965)}\right] = 0.93624$$

- Descenso y aterrizaje:

$$\frac{W_4}{W_3} = 0.995$$

- Fuel Fraction:

$$\frac{W_4}{W_0} = \frac{W_1}{W_0} \cdot \frac{W_2}{W_1} \cdot \frac{W_3}{W_2} \cdot \frac{W_4}{W_3} = 0.995 \times 0.996 \times 0.936 \times 0.995 = 0.92319 \quad \frac{W_f}{W_0} = 1.06 \left(1 - \frac{W_4}{W_0}\right) = 1.06 (1 - 0.923) = 0.08142$$

$$\text{con el peso inicial de } W_0 = 1200 \text{ lbs} \rightarrow W_{f_{required}} = 1200 \times 0.08142 = 97.7 \text{ lbs}$$

# Refined Sizing - 4

- Muy cercano al valor del peso dibujado, pero para ver como se realizaría el método de refinado:

$$W_0 = 1199 \text{ lbs.}$$

$$W_e = W_{e_{as\ dranw}} \left[ \frac{W_0}{W_{0_{as\ dranw}}} \right]^{1+c} = 882 \left[ \frac{1199}{1200} \right]^{1-0.1} = 881.4 \text{ lbs.}$$

$$W_0 = W_{crew} + W_{fuel} + W_e = 220 + 1199 \cdot 0.08142 + 881.4 = 1998.97 \text{ lbs.}$$

Pendiente de la línea que marca la tendencia del peso en vacío

Table 3.1 Empty weight fraction vs  $W_0$

$W_e/W_0 = A W_0^C K_{ws}$	A	C
Sailplane—unpowered	0.86	-0.05
Sailplane—powered	0.91	-0.05
Homebuilt—metal/wood	1.19	-0.09
Homebuilt—composite	0.99	-0.09
General aviation—single engine	2.36	-0.18
General aviation—twin engine	1.51	-0.10
Agricultural aircraft	0.74	-0.03
Twin turboprop	0.96	-0.05
Flying boat	1.09	-0.05
Jet trainer	1.59	-0.10
Jet fighter	2.34	-0.13
Military cargo/bomber	0.93	-0.07
Jet transport	1.02	-0.06

$K_{ws}$  = variable sweep constant = 1.04 if variable sweep  
= 1.00 if fixed sweep

# Optimización W/S y Alargamiento - 1

- Una vez determinado las características generales del avión, tenemos que realizar un proceso de optimización para determinar si podemos mejorar algunas de las características del diseño, ya que las pautas que hemos seguido no son las que defines el diseño perfecto, sino que requiere iteración:
  - Variamos la carga alar (W/S) en un  $\pm 20\%$ : W/S= 8.16, 10.2, 12.24
  - Variamos el alargamiento en un  $\pm 33\%$ : 4, 6, 8
  - Lo que genera 9 aviones diferentes:

	8.16	10.2	12.24	
A: 4	①	②	③	⑤ IS THE AS-DRAWN BASELINE
6	④	⑤	⑥	
8	⑦	⑧	⑨	

- A primera vista, se observan las siguientes relaciones:

$$W_w \propto \frac{1}{W/S} \quad W_{tails} \propto \left(\frac{1}{W/S}\right)^{\frac{3}{2}}$$

# Optimización W/S y Alargamiento - 2

- Dependencias de los pesos:

$$W_{\text{wing}} = 0.036 S_w^{0.758} W_{fw}^{0.0035} \left( \frac{A}{\cos^2 \Lambda} \right)^{0.6} q^{0.006} \lambda^{0.04} \left( \frac{100 t/c}{\cos \Lambda} \right)^{-0.3} (N_z W_{dg})^{0.49} \rightarrow W_w \propto S_w^{0.758}$$

$$\rightarrow W_w \propto A^{0.6}$$

$$W_{\text{horizontal tail}} = 0.016 (N_z W_{dg})^{0.414} q^{0.168} S_{ht}^{0.896} \left( \frac{100 t/c}{\cos \Lambda} \right)^{-0.12} \rightarrow W_{ht} \propto S_{ht}^{0.896}$$

$$\times \left( \frac{A}{\cos^2 \Lambda_{ht}} \right)^{0.043} \lambda_h^{-0.02} \rightarrow W_{ht} \propto A^{0.043}$$

$$W_{\text{vertical tail}} = 0.073 \left( 1 + 0.2 \frac{H_t}{H_v} \right) (N_z W_{dg})^{0.376} q^{0.122} S_{vt}^{0.873} \left( \frac{100 t/c}{\cos \Lambda_{vt}} \right)^{-0.49} \rightarrow W_{vt} \propto S_{vt}^{0.873}$$

$$\times \left( \frac{A}{\cos^2 \Lambda_{vt}} \right)^{0.357} \lambda_{vt}^{0.039} \rightarrow W_{vt} \propto A^{0.357}$$

- Aerodinámica:

- Las áreas mojadas varían con las áreas del ala/colas
- Hay que normalizar el nuevo  $C_{D0}$  con las nuevas áreas de referencia.

$$\text{Straight-Wing Aircraft: } e = 1.78(1 - 0.045A^{0.68}) - 0.64$$



# Optimización W/S y Alargamiento - 3

$$W_w \propto \frac{1}{W/S} \quad W_{tails} \propto \left(\frac{1}{W/S}\right)^{3/2}$$

- Caso 1:  $\frac{W}{S} = 10.2 \cdot 0.8 = 8.16$   
 $\frac{A}{S} = 0.66 \cdot 6 = 4$

Reducción del área en un 20 %

- Teniendo en cuenta las relaciones de pesos para la reducción de alas y colas:

$\rightarrow W_w \propto S_w^{0.758}$   
 $\rightarrow W_w \propto A^{0.6}$

$S'_{tails} = \left(\frac{1}{.8}\right)^{3/2} S_{tails} = \frac{1}{.72} S_{tails}$

$W_w = 130 \left(\frac{1}{.8}\right)^{0.758} (.666)^6 = 121 \text{ Lb} \quad \Delta = -9 \text{ Lb}$   
 $W_{ht} = 40 \left(\frac{1}{.72}\right)^{0.896} = 54 \text{ Lb} \quad \Delta = 14 \text{ Lb}$   
 $W_{vt} = 15 \left(\frac{1}{.72}\right)^{0.873} = 20 \text{ Lb} \quad \Delta = 5 \text{ Lb}$   
 $\Delta = 10 \text{ Lb}$   
 so  $W_e = 882 + 10 = 892 \text{ Lb}$

COMPONENT	FUDGE FACTOR	ADJUSTED WEIGHT: Ch15/Other Methods	SELECTED WEIGHT	DISTANCE TO DATUM*
FUSELAGE	.90	104 / 128	130 Lb	115 in
WING	.85	143 / 175	160	70
HOR. TAIL	.83	17 / 45	40	210
VERT. TAIL	.83	12 / 13	15	225
ENGINE	-	452 / 380	380	16
GEAR	.95	66 / 57	60	45
FUEL SYS.	-	22	22	50
FL. CONTROLS	-	5	5	80
ELECTRICAL	-	73 / 40	40	40
AVIONICS	-	9.5	10	60
FURNISHINGS	-	20	20	100
$\Sigma W_e = 882 \text{ Lb} @ 59.5 \text{ in}$				
PILOT & CHUTE			220	85
FUEL (Available, if $W_0 = 1200 \text{ Lb}$ )			98	50
$\Sigma W_0 = 1200 \text{ Lb} @ 63.3 \text{ in}$				

wing:  $C_{D_{0w}} = \left(\frac{1}{.8}\right) \cdot 0.0092 = .0115 \quad \Delta = .0023$   
 tails:  $C_{D_{0t}} = \left(\frac{1}{.72}\right) \cdot 0.0033 = .0045 \quad \Delta = .0012$   
 so  $C_{D_0} = .0277 + .0023 + .0012 = .0312$  (ref. to old  $S_{ref}$ )  
 $C_{D_0} = .0312 (.8) = .0250$  (ref. to new  $S_{ref}$ )  
 $e = .93$  for  $A = 4$

# Optimización W/S y Alargamiento - 4

MISSION SEGMENT WEIGHT FRACTIONS:

Cruise:  $W/S = 8.16 (.995)(.996) = 8.09$

$$\frac{L}{D} = \frac{1}{\frac{qC_{D0}}{W/S} + \frac{W}{S} \frac{1}{q\pi Ae}}$$

$$L/D = \frac{1}{\frac{35(.0250)}{8.09} + \frac{8.09}{35\pi(4)(.93)}} = 7.8$$

Se corrige el weight fraction para crucero

Prop:  $\frac{W_i}{W_{i-1}} = \exp\left[\frac{-RC_{bhp}}{550 \eta_p (L/D)}\right]$

$$\frac{W_3}{W_2} = e^{\frac{-(280 \times 6076)(.5/3600)}{(7.8)(550)(.78)(.95)}} = .928$$

All OTHER MISSION SEGMENT WEIGHT FRACTIONS ARE ESSENTIALLY UNCHANGED

SIZING : USING SAME METHOD BUT WITH

$$W_{e\text{-drawn}} = 892 \quad \text{and} \quad \frac{W_3}{W_2} = .928$$

INPUT W0drawn, WEdrawn, We/Wo Exponent: 1200, 892, -.11  
INPUT CREW + PAYLOAD WEIGHT 220

MISSION SEG TYPES            1   1   1   1  
MISSION SEG WT FRACTS    .995   .996   .928   .995

\*\*\*\*\* SIZING ITERATIONS \*\*\*\*\*

WOG	WF	WE	WOCALC
1200.0	108.0	892.0	1220.0
1216.0	109.5	902.7	1232.2
1229.0	110.6	911.3	1242.0
1239.4	111.6	918.3	1249.9
1247.8	112.3	923.9	1256.2
1254.5	112.9	928.4	1261.4
1260.0	113.4	932.0	1265.5
1264.4	113.8	935.0	1268.8
1267.9	114.1	937.3	1271.4
1270.7	114.4	939.2	1273.6
1273.0	114.6	940.7	1275.3
1274.9	114.8	941.9	1276.7
1276.3	114.9	942.9	1277.8
1277.5	115.0	943.7	1278.7

SO THE MODIFIED AIRCRAFT WITH  $(W/S)=8.16$  AND ASPECT RATIO = 4 MUST BE SIZED UP TO  $W_0=1278$  LB TO PERFORM THE DESIGN MISSION

# Optimización W/S y Alargamiento - 5

②  $W/S = 10.2$        $A = .66 \times 6 = 4$   
 $W_w = 130 (.666)^6 = 102$        $\Delta = -28 \text{ Lb}$   
 No change in tail sizes, so  $W_e = 882 - 28 = \underline{854 \text{ Lb}}$   
 No change in wing/tail areas, so  $C_{D_0} = .0277$   
 $e = .93$  for  $A = 4$

Cruise:  $W/S = 10.1$  (same as baseline)

$$L/D = \frac{1}{\frac{35(.0277)}{10.1} + \frac{10.1}{35\pi(4)(.93)}} = 8.29$$

$$\frac{W_3}{W_2} = e^{\frac{-(280 \times 6076) \times (.5/3600)}{(8.29 \times 550) \times (.78 \times .98)}} = \underline{.932}$$

INPUT W0drawn, WEdrawn, We/W0 Exponent:1200,854,-.1  
 INPUT CREW + PAYLOAD WEIGHT 220

MISSION SEG TYPES            1 1 1 1  
 MISSION SEG WT FRACTS    .995 .996 .932 .995

\*\*\*\*\* SIZING ITERATIONS \*\*\*\*\*

W0B	WF	WE	WOCALC
1200.0	103.0	854.0	1177.0
1181.6	101.4	842.2	1163.7
1167.2	100.2	833.0	1153.2
1156.0	99.2	825.8	1145.0
1147.2	98.5	820.1	1138.6
1140.3	97.9	815.7	1133.6
1134.9	97.4	812.2	1129.6
1130.7	97.1	809.5	1126.5
1127.4	96.8	807.3	1124.1
1124.8	96.6	805.7	1122.2
1122.7	96.4	804.3	1120.7
1121.1	96.2	803.3	1119.6
1119.9	96.1	802.5	1118.6
1118.9	96.1	801.9	1117.9

# Optimización W/S y Alargamiento – 6

③  $w/s = 1.2 \times 10.2 = 12.24$        $A = .666 \times 6 = 4$

$$S'_{tails} = \left(\frac{1}{1.2}\right)^{3/2} S_{tails} = \left(\frac{1}{1.31}\right) S_{tails}$$

$$W_w = 130 \left(\frac{1}{1.2}\right)^{758} (.666)^6 = 89 \text{ Lb} \quad \Delta = -41 \text{ Lb}$$

$$W_{ht} = 40 \left(\frac{1}{1.31}\right)^{896} = 31 \quad \Delta = -9$$

$$W_{vt} = 15 \left(\frac{1}{1.31}\right)^{893} = 12 \quad \Delta = -3$$

$$\Delta = -53 \text{ Lb}$$

$$\text{so } W_e = 882 - 53 = \underline{829 \text{ Lb}}$$

$$\text{Wing: } C_{D_{ow}} = \left(\frac{1}{1.2}\right) \cdot .0092 = .0077 \quad \Delta = -.0015$$

$$\text{tails: } C_{D_{ox}} = \left(\frac{1}{1.31}\right) \cdot .0033 = .0025 \quad \Delta = -.0008$$

$$\text{so } C_{D_o} = .0277 - .0015 - .0008 = .0254 \text{ (old } S_{ref})$$

$$C_{D_o} = .0254 (1.2) = \underline{.0305} \text{ (new } S_{ref})$$

$$e = .93 \text{ for } A = 4$$

$$\text{Cruise: } w/s = 12.24 (.995)(.996) = 12.13 \quad L/D = 8.5 \quad \frac{W_3}{W_2} = .934$$

INPUT W0drawn, WEdrawn, We/Wo Exponent: 1200, 829, -.1  
INPUT CREW + PAYLOAD WEIGHT 220

MISSION SEG TYPES            1   1   1   1  
MISSION SEG WT FRACTS    .995   .996   .934   .995

\*\*\*\*\* SIZING ITERATIONS \*\*\*\*\*

WOG	WF	WE	WOCALC
1200.0	100.5	829.0	1149.5
1159.6	97.1	803.8	1121.0
1128.7	94.5	784.5	1099.1
1105.0	92.5	769.7	1082.2
1086.8	91.0	758.3	1069.3
1072.8	89.9	749.5	1059.3
1062.0	89.0	742.7	1051.6
1053.7	88.3	737.5	1045.7
1047.3	87.7	733.4	1041.2
1042.4	87.3	730.3	1037.6
1038.6	87.0	727.9	1034.9
1035.7	86.7	726.1	1032.8
1033.4	86.6	724.7	1031.2
1031.6	86.4	723.5	1030.0
1030.3	86.3	722.7	1029.0

# Optimización W/S y Alargamiento - 7

$$\textcircled{4} \quad w/s = .8 \times 10.2 = 8.16 \quad A=6$$

$$S'_{tails} = \left(\frac{1}{.72}\right) S_{tails}$$

$$W_w = 130 \left(\frac{1}{.8}\right)^{.758} = 154 \text{ Lb} \quad \Delta = 24 \text{ Lb}$$

$$W_{ht} = 40 \left(\frac{1}{.72}\right)^{.896} = 54 \quad \Delta = 14$$

$$W_{vt} = 15 \left(\frac{1}{.72}\right)^{.873} = 20 \quad \frac{\Delta = 5}{\Delta = 45}$$

So  $W_e = 882 + 45 = 927 \text{ Lb}$

$$C_{D0} = .0250 \text{ (same as } \textcircled{1})$$

$$e = .87 \text{ for } A=6$$

Cruise:  $w/s = 8.09 \text{ (same as } \textcircled{1}) \quad L/D = 8.18 \quad \frac{w_3}{w_2} = .9316$

```
INPUT W0drawn, WEdrawn, We/Wo Exponent: 1200, 927, -.1
INPUT CREW + PAYLOAD WEIGHT 220
```

```
MISSION SEG TYPES      1  1  1  1
MISSION SEG WT FRACTS  .995 .996 .9316 .995
```

```
***** SIZING ITERATIONS *****
```

W0G	WF	WE	W0CALC
1200.0	103.5	927.0	1250.5
1240.4	107.0	955.1	1282.1
1273.7	109.9	978.1	1308.0
1301.1	112.2	997.0	1329.3
1323.6	114.2	1012.5	1346.7
1342.1	115.8	1025.2	1361.0
1357.2	117.1	1035.6	1372.7
1369.6	118.1	1044.1	1382.3
1379.7	119.0	1051.1	1390.1
1388.0	119.7	1056.8	1396.5
1394.8	120.3	1061.4	1401.7
1400.3	120.8	1065.2	1406.0
1404.9	121.2	1068.3	1409.5
1408.5	121.5	1070.8	1412.3
1411.6	121.8	1072.9	1414.6
1414.0	122.0	1074.6	1416.5
1416.0	122.2	1075.9	1418.1
1417.7	122.3	1077.0	1419.3
1419.0	122.4	1078.0	1420.4

# Optimización W/S y Alargamiento - 8

⑤ IS THE BASELINE, SO  $W_0 = 1200$  Lb.

⑥  $W/S = 12 \times 10.2 = 12.24$        $A = 6$

$S'_{tails} = (\frac{1}{1.31}) S_{tails}$  (same as ③)

$W_W = 130 (\frac{1}{1.2})^{758} = 113$  Lb       $\Delta = -17$  Lb

$W_{ht} =$  (same as ③) = 31       $\Delta = -9$

$W_{ut} =$  " " " = 12       $\Delta = -3$

$\Delta = -29$  Lb

$W_e = 853$  Lb

Cruise:

$C_{D_0} = .0305$  } (same as ③)

$W/S = 12.13$  }

$L/D = 9.16$        $\frac{W_S}{W_E} = .939$

```
INPUT W0drawn, WEdrawn, We/W0 Exponent:1200,853,-.1
INPUT CREW + PAYLOAD WEIGHT 220
```

```
MISSION SEG TYPES      1  1  1  1
MISSION SEG WT FRACTS  .995 .996 .939 .995
```

\*\*\*\*\* SIZING ITERATIONS \*\*\*\*\*

WOG	WF	WE	WOCALC
1200.0	94.2	853.0	1167.2
1173.8	92.2	836.2	1148.4
1153.5	90.6	823.2	1133.8
1137.7	89.3	813.0	1122.4
1125.4	88.4	805.2	1113.5
1115.9	87.6	799.0	1106.6
1108.5	87.1	794.2	1101.3
1102.7	86.6	790.5	1097.1
1098.2	86.2	787.6	1093.9
1094.7	86.0	785.3	1091.3
1092.0	85.8	783.6	1089.3
1089.9	85.6	782.2	1087.8
1088.2	85.5	781.1	1086.6
1086.9	85.4	780.3	1085.7
1085.9	85.3	779.7	1084.9

# Optimización W/S y Alargamiento - 9

$$\begin{aligned} \textcircled{7} \quad W/S &= .8 \times 10.2 = 8.16 \\ S_{\text{tails}} &= \left(\frac{1}{12}\right) S_{\text{tails}} \end{aligned} \left. \vphantom{\begin{aligned} W/S \\ S_{\text{tails}} \end{aligned}} \right\} \text{Same as } \textcircled{1} \quad A = 1.333 \times 6 = 8$$

$$\begin{aligned} W_w &= 130 \left(\frac{1}{8}\right)^{758} (1.333)^6 = 183 \text{ Lb} & \Delta &= 53 \text{ Lb} \\ W_{ht} &= (\text{same as } \textcircled{1}) = 54 & \Delta &= 14 \\ W_{vt} &= \text{ " " " } = 20 & \Delta &= 5 \\ & & \Delta &= 72 \text{ Lb} \\ & & W_e &= 954 \text{ Lb} \end{aligned}$$

Cruise:  $C_{D_0} = .0250$  (same as  $\textcircled{1}$ )  
 $W/S = 8.09$      $L/D = 8.37$      $\frac{W_3}{W_e} = .933$

INPUT W0drawn, WEdrawn, We/Wc Exponent:1200,954,-.1  
 INPUT CREW + PAYLOAD WEIGHT 220

MISSION SEG TYPES            1 1 1 1  
 MISSION SEG WT FRACTS    .995 .996 .933 .995

\*\*\*\*\* SIZING ITERATIONS \*\*\*\*\*

W0G	WF	WE	WOCALC
1200.0	101.8	954.0	1275.8
1260.6	106.9	997.3	1324.2
1311.4	111.2	1033.4	1364.6
1354.0	114.8	1063.5	1398.3
1389.4	117.8	1088.5	1426.4
1419.0	120.3	1109.3	1449.7
1443.5	122.4	1126.6	1469.0
1463.9	124.1	1140.9	1485.0
1480.8	125.6	1152.8	1498.3
1494.8	126.8	1162.6	1509.3
1506.4	127.7	1170.7	1518.4
1516.0	128.6	1177.4	1526.0
1524.0	129.2	1182.9	1532.2
1530.5	129.8	1187.5	1537.3
1536.0	130.3	1191.3	1541.6
1540.5	130.6	1194.5	1545.1
1544.2	130.9	1197.0	1548.0
1547.2	131.2	1199.2	1550.4
1549.7	131.4	1200.9	1552.4
1551.8	131.6	1202.4	1554.0
1553.6	131.7	1203.6	1555.3
1555.0	131.9	1204.6	1556.5

# Optimización W/S y Alargamiento - 10

⑧  $w/s = 10.2$        $A = 1.33 \times 6 = 8$   
 $W_w = 130(1.333)^6 = 155$        $\Delta = 25$   
Cruise:       $W_e = 907$   
 $C_{D0} = .0277$  } (Same as ⑤)  
 $w/s = 10.1$  }  
 $e = .81$  for  $A = 8$   
 $L/D = 9.08$        $\frac{W_3}{W_2} = .938$

INPUT W0drawn, WEdrawn, We/Wo Exponent:1200,907,-.1  
 INPUT CREW + PAYLOAD WEIGHT 220

MISSION SEG TYPES            1 1 1 1  
 MISSION SEG WT FRACTS    .995 .996 .938 .995

\*\*\*\*\* SIZING ITERATIONS \*\*\*\*\*

WOG	WF	WE	WOCALC
1200.0	95.5	907.0	1222.5
1218.0	96.9	919.2	1236.2
1232.5	98.1	929.1	1247.2
1244.2	99.0	937.0	1256.1
1253.7	99.8	943.4	1263.2
1261.3	100.4	948.6	1269.0
1267.4	100.9	952.7	1273.6
1272.4	101.2	956.1	1277.3
1276.3	101.6	958.8	1280.3
1279.5	101.8	960.9	1282.7
1282.1	102.0	962.7	1284.7
1284.2	102.2	964.1	1286.2
1285.8	102.3	965.2	1287.5
1287.2	102.4	966.1	1288.5
1288.2	102.5	966.8	1289.3



# Optimización W/S y Alargamiento - 11

$$\begin{aligned} \textcircled{9} \quad W/S &= 1.2 \times 10.2 = 12.24 & A &= 1.333 \times 6 = 8 \\ S'_{\text{tail}} &= \frac{1}{1.31} S_{\text{tail}} \\ W_w &= 130 \left(\frac{1}{1.2}\right)^{758} (1.333)^{-6} = 135 \text{ lb} & \Delta &= 5 \text{ lb} \\ W_{ht} &= (\text{same as } \textcircled{3}) = 31 & \Delta &= -9 \\ W_{vt} &= \text{" " " " } = 12 & \Delta &= -3 \\ & & \Delta &= -7 \\ & & W_e &= 875 \\ \text{Cruise: } \left. \begin{array}{l} C_{D_0} = .0305 \\ W/S = 12.13 \\ e = .81 \end{array} \right\} & \text{(same as } \textcircled{3}) \\ L/D &= 9.52 & \frac{W_3}{W_2} &= .941 \end{aligned}$$

```
INPUT W0drawn, WEdrawn, We/Wo Exponent: 1200, 875, -.1
INPUT CREW + PAYLOAD WEIGHT 220
```

```
MISSION SEG TYPES      1  1  1  1
MISSION SEG WT FRACTS  .995 .996 .941 .995
```

```
***** SIZING ITERATIONS *****
```

WOG	WF	WE	WOCALC
1200.0	91.7	875.0	1186.7
1189.4	90.9	868.0	1178.9
1181.0	90.3	862.5	1172.8
1174.5	89.8	858.2	1168.0
1169.3	89.4	854.8	1164.2
1165.2	89.1	852.1	1161.2
1162.0	88.8	850.0	1158.9
1159.5	88.6	848.4	1157.0
1157.5	88.5	847.1	1155.5
1155.9	88.4	846.0	1154.4
1154.7	88.3	845.2	1153.5
1153.7	88.2	844.6	1152.8

# Optimización W/S y Alargamiento - 12

## SIZING MATRIX RESULTS ( $W_0$ )

		W/S:		
		8.16	10.2	12.24
A:	4	① 1278 Lb	② 1117 Lb	③ 1030 Lb
	6	④ 1420 Lb	⑤ 1200 Lb	⑥ 1085 Lb
	8	⑦ 1556 Lb	⑧ 1289 Lb	⑨ 1153 Lb

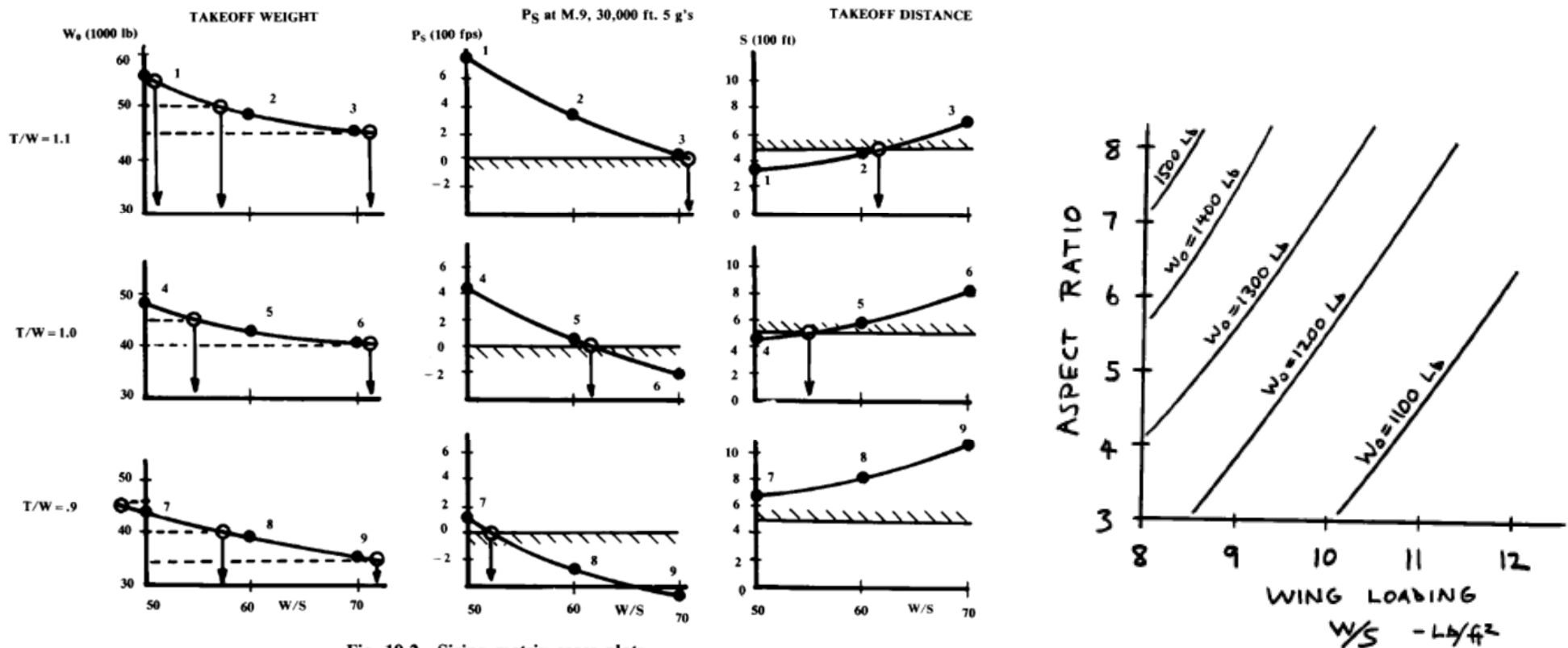


Fig. 19.2 Sizing matrix cross plots.

# Refinamiento Actuaciones - 1

## ■ Límites de actuaciones

STALL: As before,  $V_{\text{stall}} = 50$  requires  $W/S \leq 10.2$

RATE OF CLIMB:  $V_v \geq 1500$  ft/min at Sea Level

Assume  $V = 75$  kts (best R.O.C. for baseline)

so  $q = 19.1$   $T = 360$  Lb (from graph)

$$D = q S \left( C_{D_0} + \frac{(W/S)^2}{\pi A e} \right) \quad V_v = \left( \frac{T-D}{W} \right) V$$

①  $W = 1278$  Lb  $S = 1278/8.16 = 156.6$  ft<sup>2</sup>

$$D = 156.6 \times 19.1 \left( .0250 + \frac{(8.16/19.1)^2}{\pi \cdot 4 \cdot (.93)} \right) = 122$$
 Lb
$$V_v = 75 \times 1.689 \left( \frac{360 - 122}{1278} \right) = 23.6$$
 ft/sec = 1415 ft/min

②  $W = 1117$  Lb  $S = 1117/10.2 = 109$  ft<sup>2</sup>

$$D = 109 \times 19.1 \left( .0277 + \frac{(10.2/19.1)^2}{\pi \cdot 4 \cdot (.93)} \right) = 108.5$$
 Lb
$$V_v = 75 \times 1.689 \left( \frac{360 - 108.5}{1117} \right) = 28.5$$
 ft/sec = 1711 ft/min

③  $W = 1030$  Lb  $S = 1030/12.24 = 84.1$

$$D = 84.1 \times 19.1 \left( .0305 + \frac{(12.24/19.1)^2}{\pi \cdot 4 \cdot (.93)} \right) = 105.4$$
 Lb
$$V_v = 75 \times 1.689 \left( \frac{360 - 105.4}{1030} \right) = 31.3$$
 ft/sec = 1878 ft/min

SIMILARLY:

④  $V_v = 1284$  ft/min

⑤  $V_v = 1636$  ft/min

⑥  $V_v = 1863$  ft/min

⑦  $V_v = 1154$  ft/min

⑧  $V_v = 1529$  ft/min

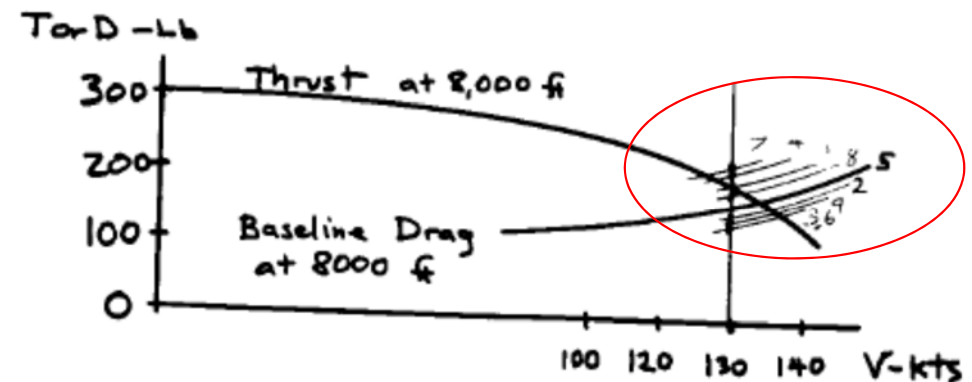
⑨  $V_v = 1774$  ft/min

# Refinamiento Actuaciones - 2

MAXIMUM SPEED:  $V_{max} \geq 130$  kts at 8,000 ft

• QUICK METHOD: CALCULATE DRAG AT 130 KTS, USE TO SHIFT PREVIOUS DRAG CURVE UP OR DOWN, THEN FIND INTERSECTION WITH THRUST CURVE.

① D=196 Lb	$V_{max} = 130$ kts
② D=157 Lb	$V_{max} = 136$ kts
③ D=139 Lb	$V_{max} = 138$ kts
④ D=211 Lb	$V_{max} = 127$ kts
⑤ D=163 Lb	$V_{max} = 134$ kts
⑥ D=139 Lb	$V_{max} = 138$ kts
⑦ D=228 Lb	$V_{max} = 125$ kts
⑧ D=172 Lb	$V_{max} = 133$ kts
⑨ D=145 Lb	$V_{max} = 137$ kts



# Sizing Matrix - 1

- Representamos gráficamente todos los requisitos en la matriz de dimensionado:
  - Velocidad de entrada en pérdida
  - Rate of Climb.
  - Velocidad Máxima.
- No hay límite inferior en el alargamiento, por lo que tenemos que buscar un requisito adicional.
  - Dado que la resistencia inducida será excesiva durante los giros cuando la planta del ala presente alargamientos muy bajos:

$\dot{\psi} \geq 30^\circ/\text{sec} \rightarrow$  mantenida a 100 kts y nivel del mar

$$n = (T/W)(L/D)$$

$$n = \sqrt{\frac{q}{K(W/S)} \left( \frac{T}{W} - \frac{qC_{D0}}{W/S} \right)}$$

$$\dot{\psi} = 30^\circ/\text{sec} = .5236 \text{ rad/sec} = \frac{g\sqrt{n^2-1}}{100 \times 1.689}$$

$$\text{so } n \geq 2.92$$

$$\text{so } n \geq 2.92$$

$T = 345 \text{ Lb}$ , from graph

$$\text{eg 17.52) } n = \sqrt{\frac{34\pi A c (345 - \frac{34 C_{D0}}{W/S})}{W/S}}$$

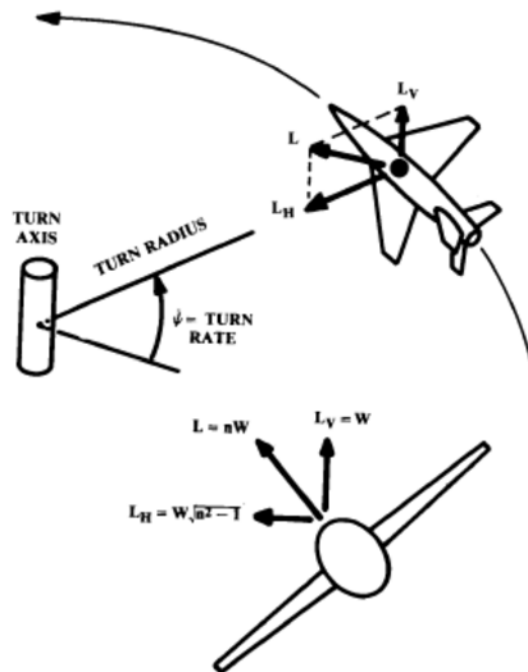
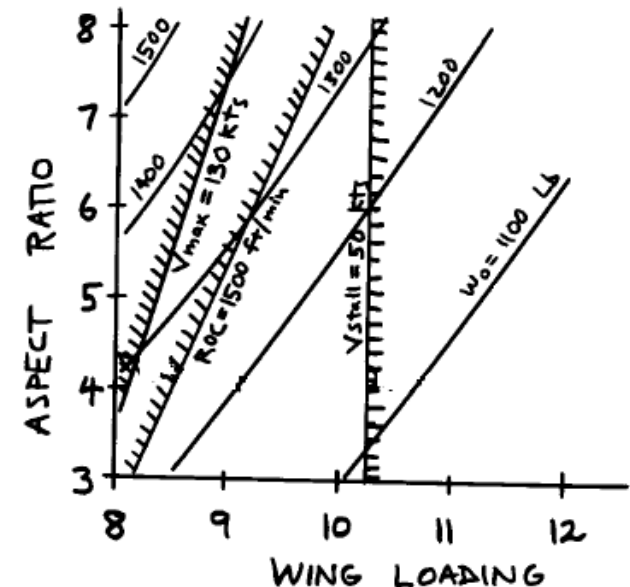


Fig. 17.4 Level turn geometry.



# Sizing Matrix - 2

Check if we are close to stall (which reduces  $e$ ):

$$C_L = \frac{nW/S}{q} = .88 \text{ for baseline } \left. \begin{array}{l} \text{Prior } e \\ \text{estimates} \\ \text{should be} \\ \text{approximately correct.} \end{array} \right\}$$

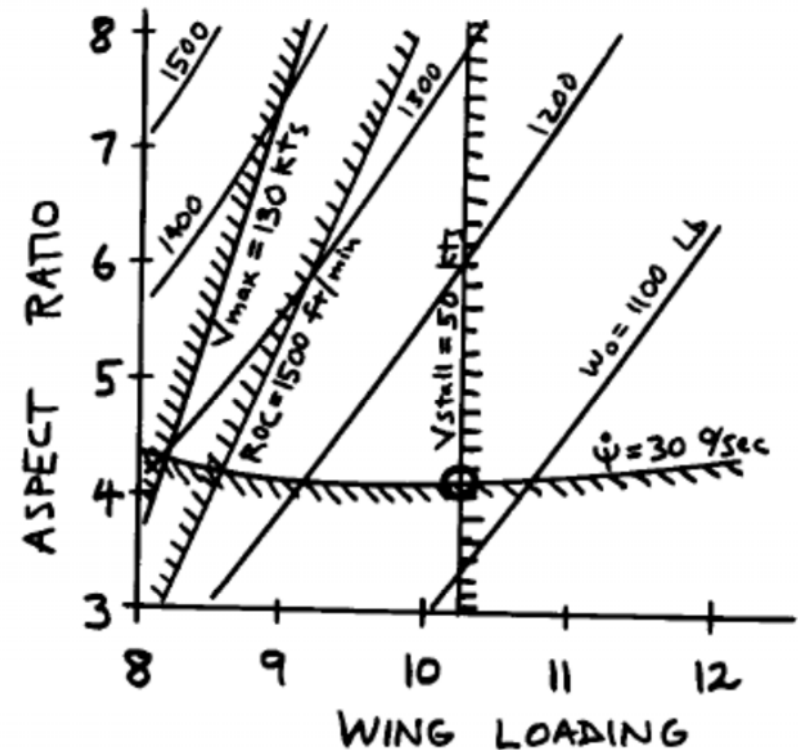
$$= 1.056 \text{ for } W/S = 12.24$$

USING PRIOR DATA:

- |             |             |
|-------------|-------------|
| ① $n = 2.8$ | ⑥ $n = 3.3$ |
| ② $n = 2.9$ | ⑦ $n = 3.2$ |
| ③ $n = 2.8$ | ⑧ $n = 3.4$ |
| ④ $n = 3.1$ | ⑨ $n = 3.5$ |
| ⑤ $n = 3.3$ |             |

NOTE: Large values of  $n$  are incorrect because they imply  $C_L > C_{L\text{stall}}$ . However, we can use those values to crossplot for  $n = 2.92$  which is below stall.

SO THE OPTIMAL AIRCRAFT FOR THE GIVEN REQUIREMENTS OCCURS AT  $\{W/S = 10.2\}$  AND  $\{A = 4.2\}$  AND HAS  $W_0 = 1130$  LB. THE NEXT STEP IN THE DESIGN PROCESS IS TO REDRAW THE AIRCRAFT AND ANALYZE IT IN DETAIL FOR STRUCTURES.



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