Chapter 12
Aerodynamics

Drag Terminology Matrix

	Shear forces	Pressure forces		
		Separation	Shock	Circulation
Parasite drag	Skin friction	Viscous separation	Wave drag	
	Scrubbing drag	Shock-induced separation "drag rise"		
	Interference drag			
	Profile drag			
		Camber drag		
Drag due to lift [f(lift)]	Supervelocity effect on skin friction	Supervelocity effect on profile drag—e.g., landing gear		Induced drag
				Trim drag
			Wave drag due to lift	
Reference area	S_{wetted}	Max. cross-section	(Volume distribution)	S_{ref}

Zero-lift drag, C_{DO}

Drag due to lift, C_{Di} (usually, but erroneously, called "induced drag")

Wave drag

Remaining drag due to shear or separation

Inviscid drag

Wave drag

Topics in this Chapter

	Subsonic	Transonic	Supersonic
C_L vs α	12.4.1	12.4 Mach correction	12.4.2
C_{Lmax} (clean)	12.4.5		12.4.5
C_{Lmax} (high lift devices)	12.4.6		12.4.6
Parasite Drag	12.5	12.5.10 M_{DD} *	12.5.9 Area Rule
Drag due to lift	12.6.1 Oswald Span Efficiency	↓	12.6.2 Leading Edge Suction

* Includes transonic drag due to lift

High Lift Systems

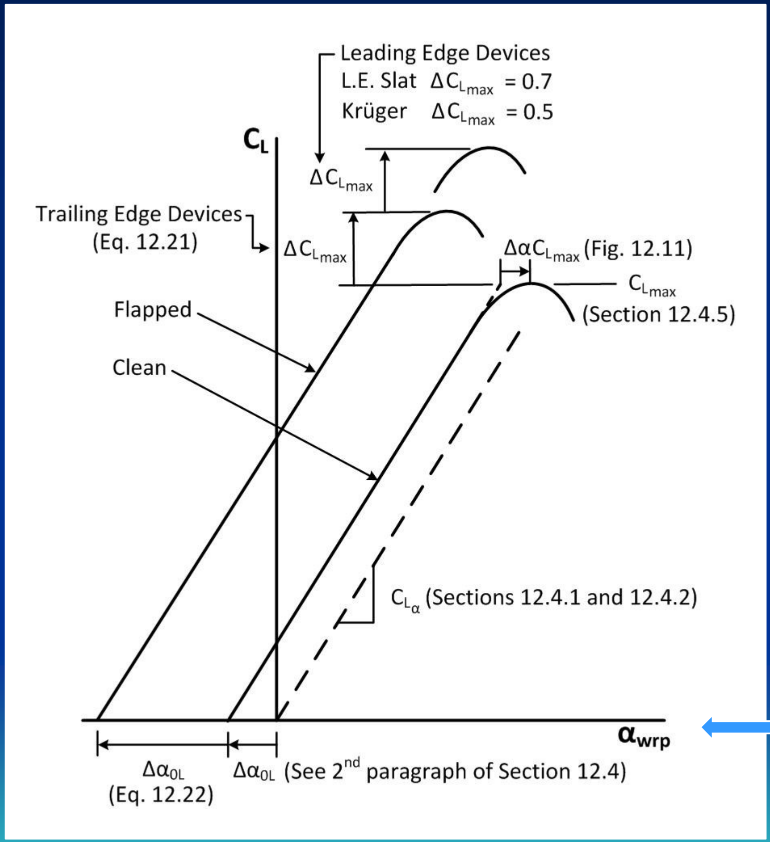
Zero-Lift Drag C_{D_0}

Drag due to Lift C_{D_i}

Wave Drag due to Volume $C_{D_{0_{supersonic}}}$

Wave Drag due to Lift C_{D_W}

Generation of C_L vs. α Plot



Values here are plotted wrt. wing reference plane

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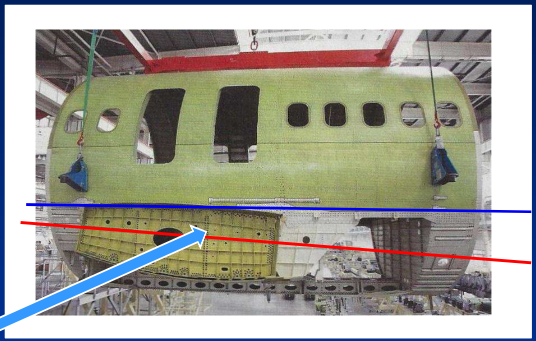
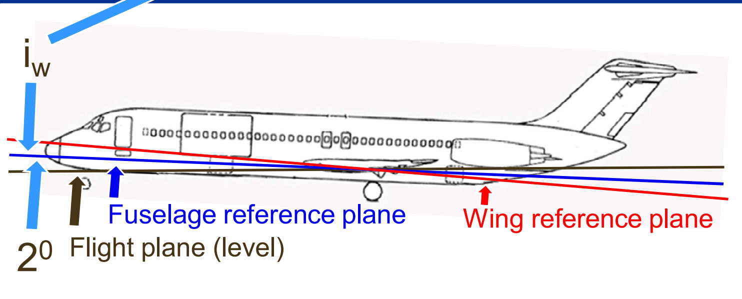
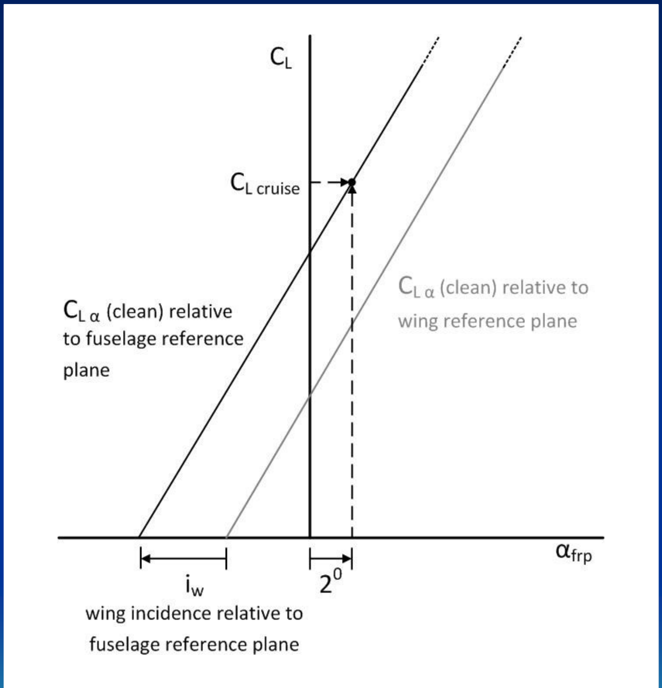
Translating C_L vs. α Plot to FRP

Set wing on fuselage for fuselage attitude of 2^0 at typical cruise C_L







C_L vs. α Gradient

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

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

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

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

In theory, $C_{l_\alpha} = 2\pi$ so this term reduces to A^2

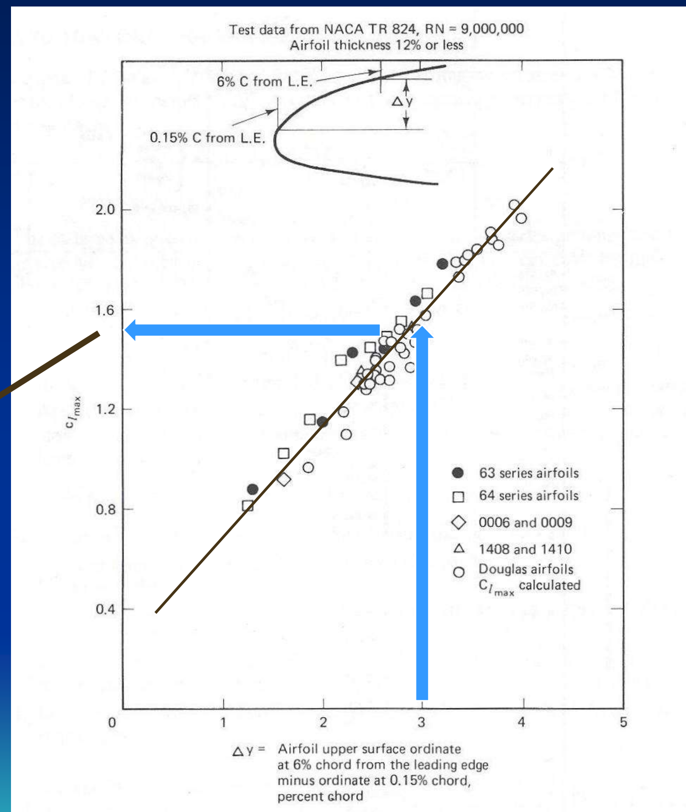
Estimation of Clean $C_{L_{max}}$ with Known Airfoil Section

For high AR wing

with moderate sweep

and $\frac{t}{c} \leq 12\%$

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



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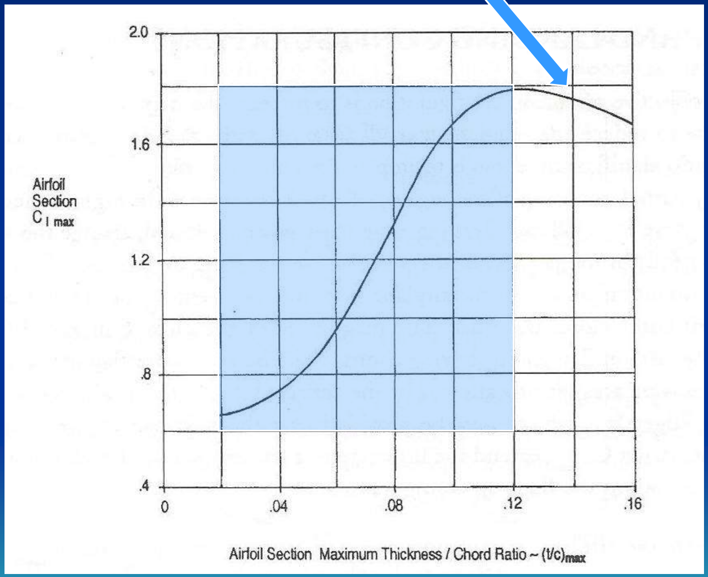
Estimation of $C_{L_{max}}$

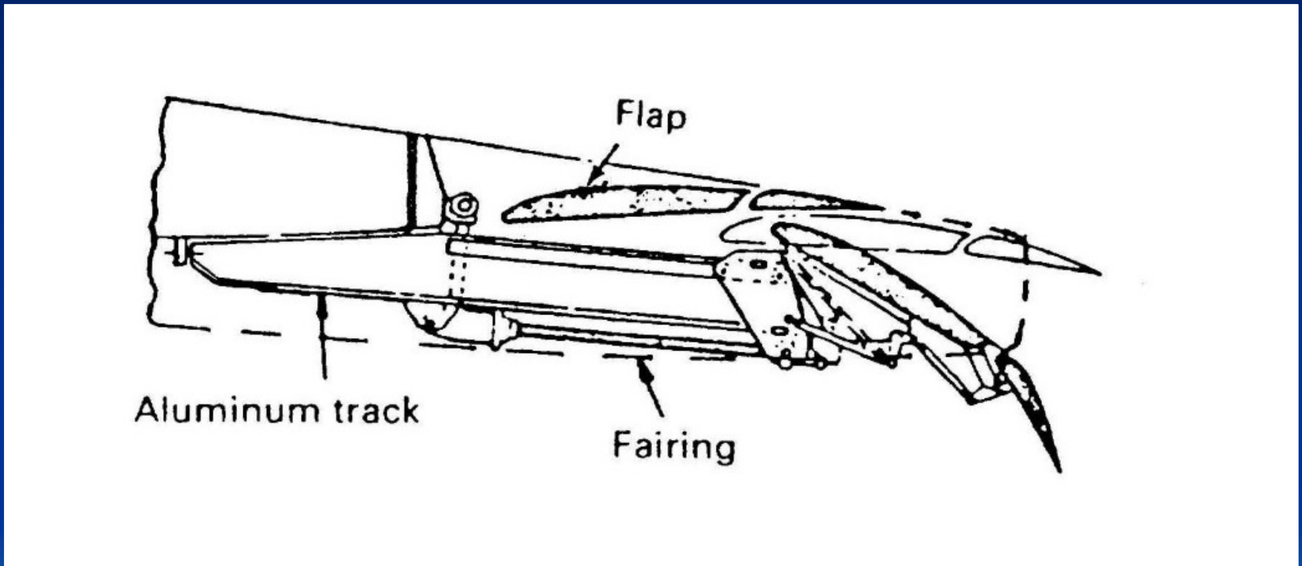
For high AR wing
with moderate sweep

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

For wing with t/c > 12%

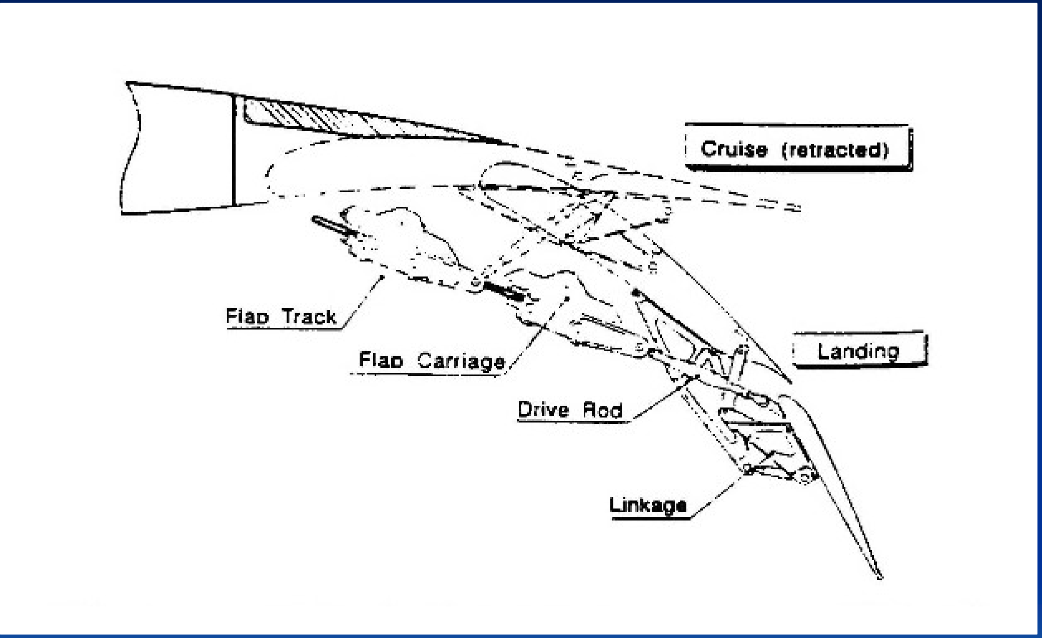


A300B Flap System



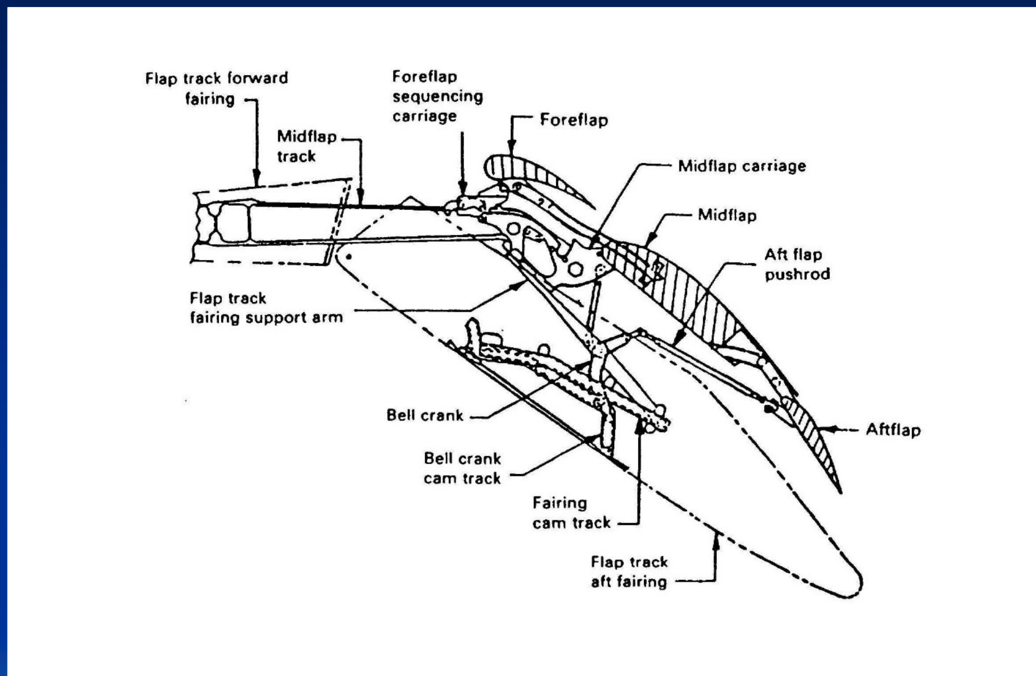
- Double-slotted
- Extends on flap tracks

A321 Flap System



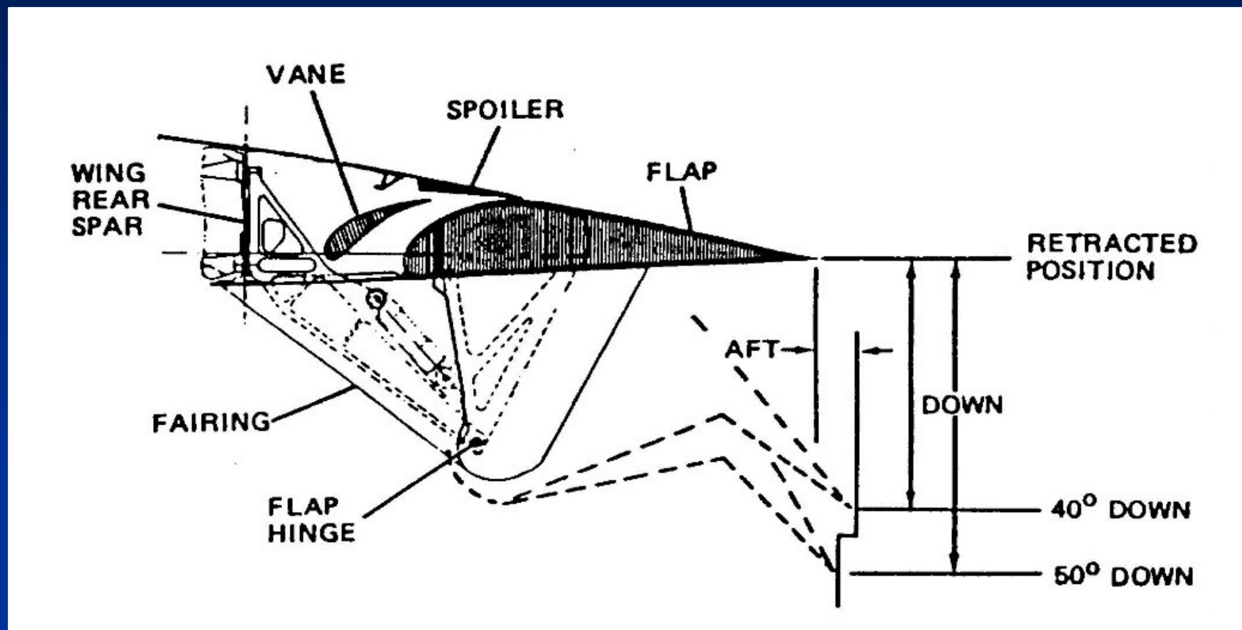
- Double-slotted
- Extends on flap tracks

737 Flap System



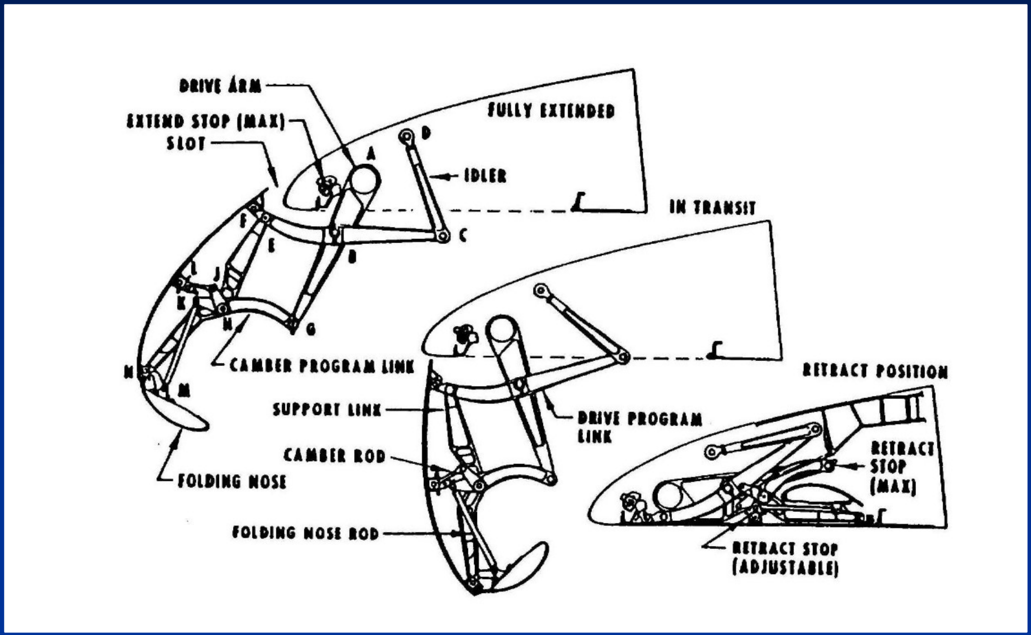
- Triple-slotted
- Extends on flap tracks

DC-9 Flap System



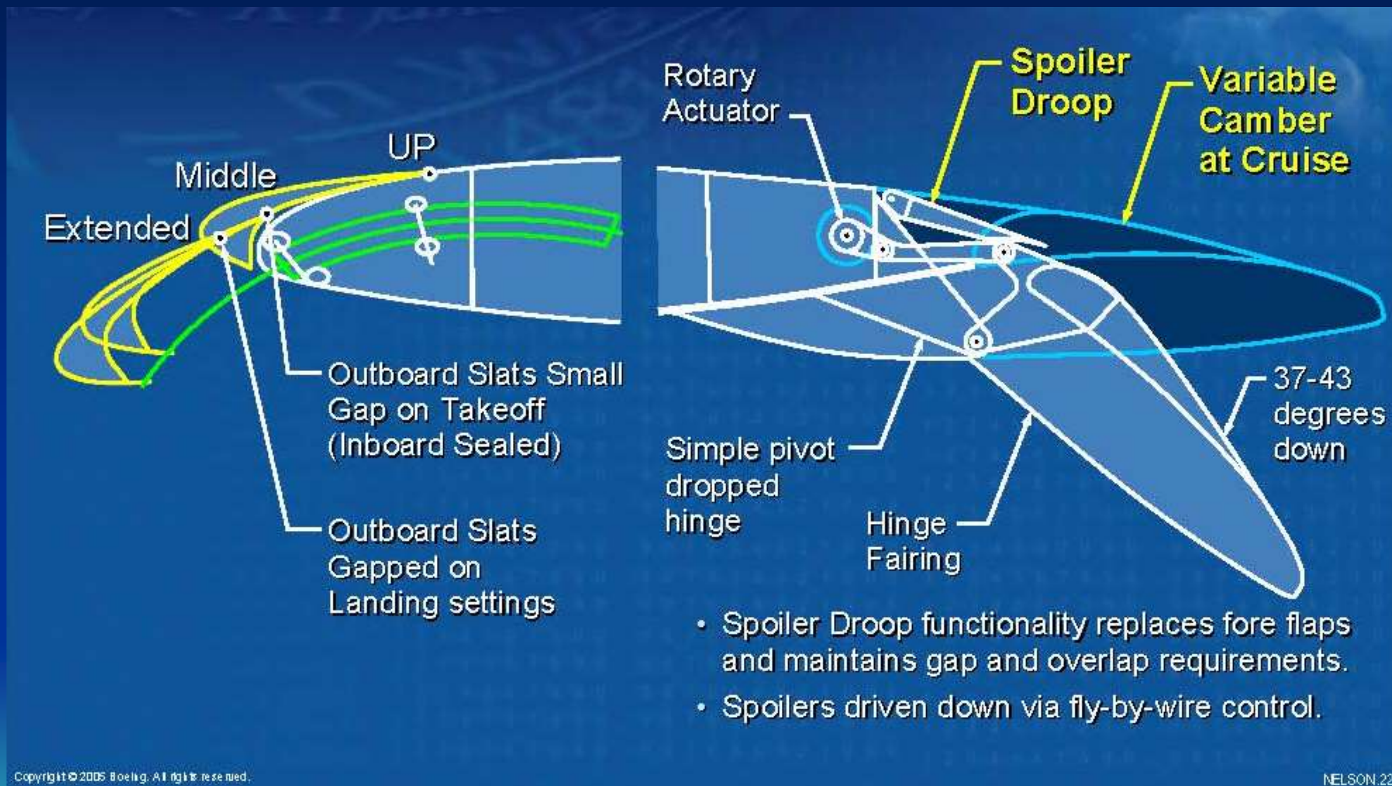
- Uses simple hinged flap with limited Fowler action
- Similar principle used on B787

747 Variable Camber Krüger Flap System

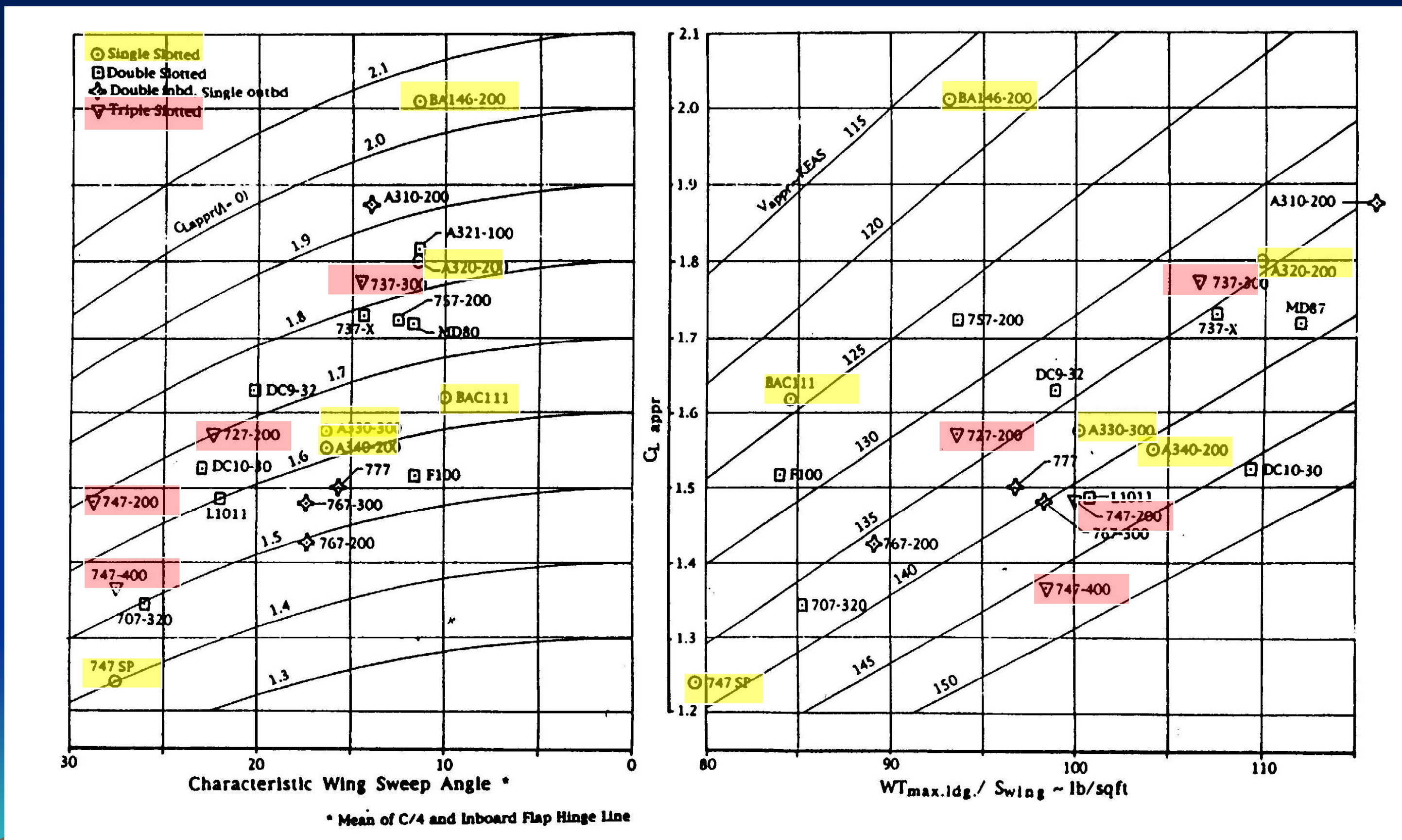


- Complex mechanical linkage

B787 Flap System



High-Lift System Performance



$$C_{L_{app}} = \frac{1}{\frac{1}{2}\rho V^2}\left(\frac{W_{\text{max landing}}}{S}\right)$$

High Lift Systems

Zero-Lift Drag C_{D_0}

Drag due to Lift C_{D_i}

Wave Drag due to Volume $C_{D_{0_{supersonic}}}$

Wave Drag due to Lift C_{D_W}

Quick Method for Estimating C_{D_o}

Equivalent Skin Friction Method:

For a flat plate with surface parallel to flow

$$D = C_f \, q \, S$$

where

C_f = skin friction coefficient

S = area

For an airplane

$$D_o = C_{f_e} \, q \, S_{wet}$$

where

C_{f_e} = equivalent skin friction coefficient

S_{wet} = airplane wetted area

$$C_{D_o} = \frac{D_o}{q S_{ref}} = C_{f_e} \frac{S_{wet}}{S_{ref}}$$

Aircraft type	C_{f_e}
Civil transport	0.0026
Bomber	0.0030
Military cargo	0.0035
Air Force fighter	0.0035
Navy fighter	0.0040
Supersonic cruise aircraft	0.0025
Light aircraft - single engine	0.0055
Light aircraft - twin engine	0.0045
Seaplane - propeller driven	0.0065
Seaplane - jet	0.0040

Source: Raymer (with modification)

Zero Lift Drag (C_{D_o}) Calculation (Incomp. Flow)

- Also called "parasite" drag (because you can't get rid of it)
- Defined as

$$C_{D_o} = C_{D_{streamlined}} + C_{D_{misc}} + C_{D_{L\&P}}$$

where

$C_{D_{streamlined}}$ = Zero lift drag coeff due to streamlined components

$C_{D_{misc}}$ = Zero lift drag coeff due to misc bluff assemblies

$C_{D_{L\&P}}$ = Zero lift drag coeff due to leakage and protuberances

Component Definitions

- Streamlined components are defined as objects for which skin friction drag dominates (e.g., wing, fuselage, horizontal and vertical tail, nacelles, pylons, etc.)
- Miscellaneous components are defined as bluff objects for which pressure drag dominates (e.g., wheels and struts, wire bracing, hemispherical protrusion on side, top, or bottom of fuselage, etc.)

Drag of Streamwise Flat Plate

Skin friction drag

$$D = C_f \left(\frac{1}{2}\right) \rho V^2 S_{wet} = C_f \, q \, S_{wet}$$

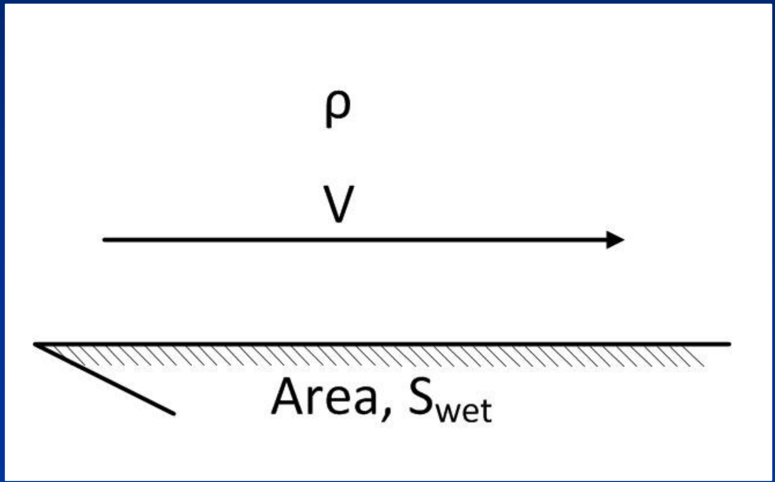
where

C_f = skin friction coefficient

Divide by q

$$\frac{D}{q} = C_f \, S_{wet}$$

$$\left(C_{D_0}\right)_{\text{flat plate}} = \frac{D}{q \, S_{ref}} = C_{f_{\text{flat plate}}} \frac{S_{wet}}{S_{ref}}$$



Summing Values of D/q

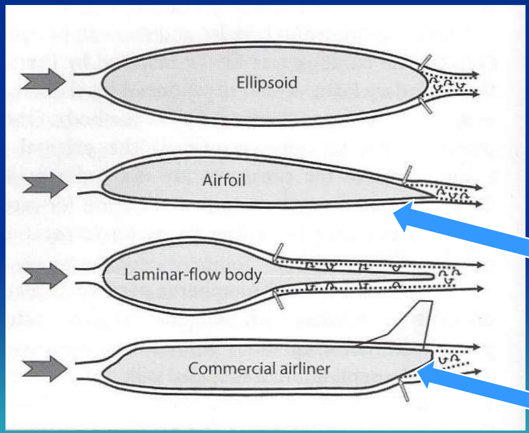
Considering skin friction only,

the sum of $\left(C_{D_0}\right)_c$ for all components

would be $\sum_{c=1}^{n} \frac{C_{f_c} S_{wet_c}}{S_{ref}}$

where c refers to an aircraft component

n = number of components



Source: Raymer

By including b.l. displacement effects,

we must deal with form drag and interference drag.

For each component, c, we factor the

value of $\left(C_{D_0}\right)_c$ by an empirical form factor, FF_c,

and (where appropriate) an empirical

interference factor Q_c

So

$$\sum \left(C_{D_0}\right)_{comp} = \sum_{c=1}^{n} \frac{\left(C_{f_c} S_{wet_c} FF_c Q_c\right)}{S_{ref}}$$

Boundary layer growth: pressure distribution is that of a body that is <u>not</u> closed (i.e. resolving D'Alembert's Paradox).

Aggravated if separation occurs

Form Factors

For wing, tail, strut and pylon

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

where

$\left(\frac{x}{c}\right)_m$ = chordwise location of the airfoil maximum thickness point

$\left(\frac{t}{c}\right) = \text{average}\ \frac{\text{thickness}}{\text{chord}}\ \text{ratio}$

Λ_m = sweep of the maximum thickness line

For fuselage and smooth canopy

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

For nacelle and smooth external store

$$FF = 1 + \frac{0.35}{f}$$

where

f = fineness ratio, defined as

$$f = \frac{l}{d} = \frac{l}{\sqrt{\frac{4}{\pi}A_{max}}}$$

where

l = component length

d = component diameter

For a nacelle $A_{max} = \frac{\pi}{4}\left(D_{nac}^2 - D_h^2\right)$

D_{nac} = nacelle max diameter

D_h = nacelle highlight diameter

Interference Factors

Condition	Q
Nacelle or external store mounted directly on fuselage or wing	1.5
Nacelle or external store less than one diameter from fuselage or wing	1.3
Nacelle or external store more than one diameter from fuselage or wing	1.0
Wingtip-mounted missiles	1.25
High wing, mid wing or well-filleted low wing	1.0
Unfilleted low wing	1.1-1.4
Conventional tail	1.04-1.05
V-tail	1.03
H-tail	1.08

Source: Raymer

For more information see Hoerner Chapter VIII Interference Drag

Flat Plate Skin Friction Coefficient

For laminar flow

$$C_f = \frac{1.328}{\sqrt{R_n}}$$

For turbulent flow

$$C_f = \frac{0.455}{\left(\log_{10} R_n\right)^{2.58}\left(1 + 0.144M^2\right)^{0.65}}$$

where

$$R_n = \frac{\rho V l}{\mu}$$

l = characteristic length

i.e.

- mac of lifting surface,
- length of fuselage
- average chord of pylon

For large airplanes, flow is nearly always turbulent

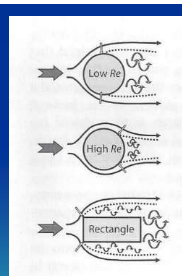
Miscellaneous Components

Calculate component $\frac{D}{q}$ based on frontal area

Sum the values of $\frac{D}{q}$ and divide by airplane reference area

$$C_{D_{msic}} = \sum_{c=1}^{n} \left(\frac{D}{q}\right)_c \frac{1}{S_{ref}}$$

Component	D/q per unit frontal area
Wheel and tire	0.25
Second wheel in tandem	0.15
Streamlined wheel and tire	0.18
Wheel and tire with fairing	0.13
Streamlined strut (0.17<t/c<0.33)	0.05
Round strut or wire	0.30 *
Flat spring gear leg	1.40
Fork, bogey, irregular fitting	1.0-1.4

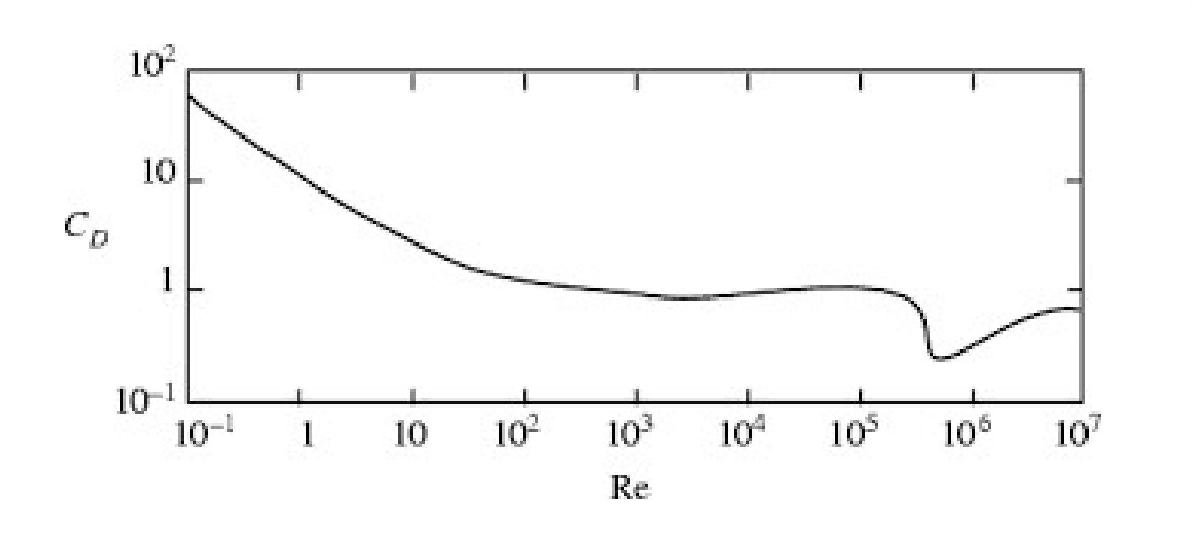


Source: Raymer

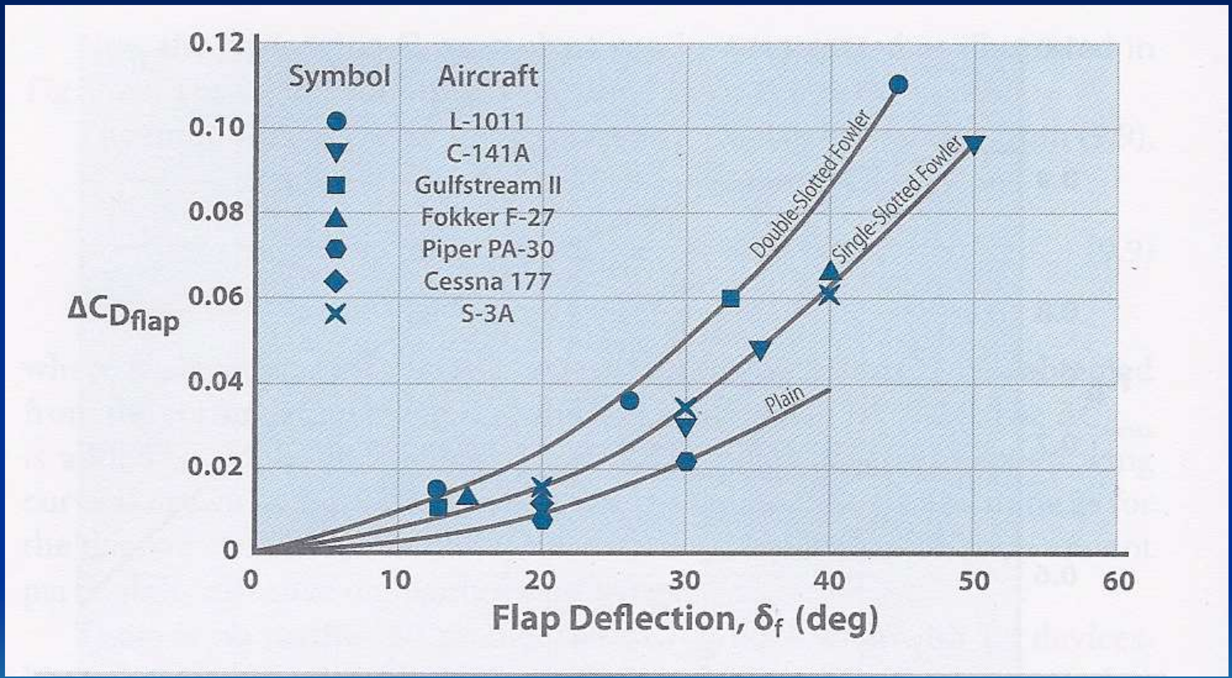
* If subcritical, use D/q = 1.2

For more information see Hoerner Chapter XIII Aircraft Components

Cylinder Drag is R_e - dependent



Approximate Flap Drag



Source: Nicolai/Carichner

$\Delta C_{D_{flap}}$ referenced to wing area

Detailed Flap Drag

Two components

– due to separated flow

– due to change in span loading

Flap drag due to separated flow

$$\Delta C_{D_{flaps}} = F_{flap}\left(\frac{c_{flap}}{c}\right)\left(\frac{S_{flapped}}{S_{ref}}\right)\left(\delta_{flap} - 10\right)$$

where

δ_{flap} = flap deflection in degrees

F_{flap} = 0.0144 for plain flaps

F_{flap} = 0.0074 for slotted flaps

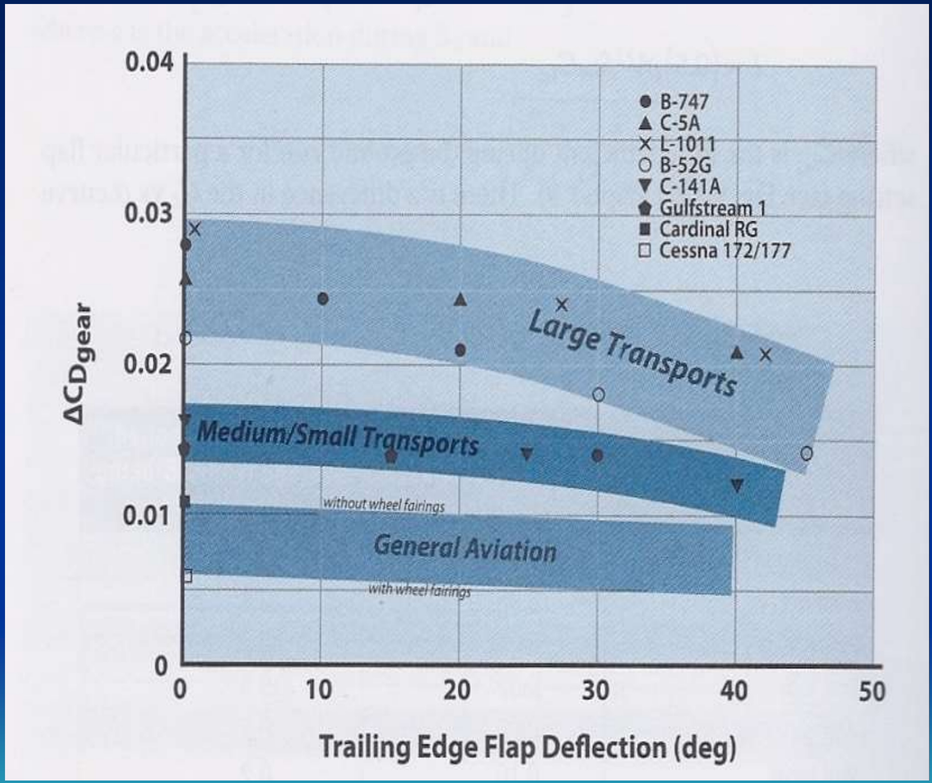
c_{flap} = chord length of flap

Boeing 727 flaps

Approximate Landing Gear Drag

Usually calculate landing gear drag by component, and verify with wind tunnel tests

Use this figure for ball-park check ($\Delta C_{D_{gear}}$ referenced to wing area)



Source: Nicolai /Carichner

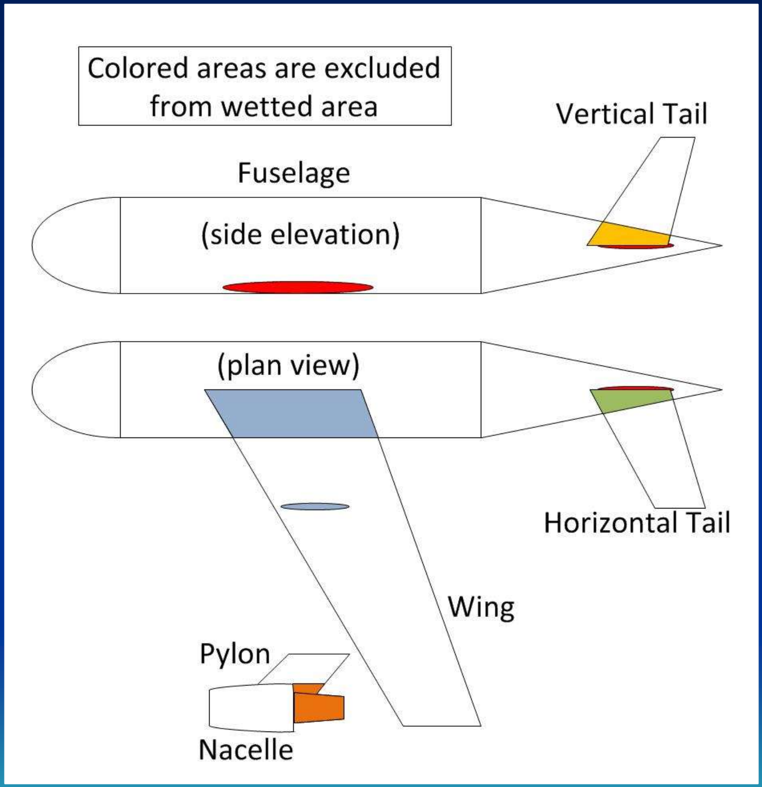
Leakage and Protuberance Drag

- Caused by
  - air entering airframe in high surface pressure areas (increased momentum drag)
  - air exiting airframe in low surface pressure areas (increased separation drag)

Category	$C_{D_{L\&P}}$
Bombers or jet transports	2-5%
Propeller-driven	5-10%
Current fighters	10-15%
Next-gen fighters	5-10%

Scaling Lifting Surfaces and Nacelles

- In mission sizing program some parts must be rescaled on every weight iteration
  - wing
  - horizontal tail
  - vertical tail and
  - nacelles



Spreadsheet Geometry Module

Wing		Horiz Tail		Vert Tail		Pylon		Fuselage		Nacelles	
AR_{wing}		AR_{ht}		AR_{vt}		l_{pylon}/d_{nac}		l_{fuse}		$l_{ref\text{-}nac}$	
Λ_{wing}		Λ_{ht}						d_{fuse}		$d_{ref\text{-}nac}$	
λ_{wing}		λ_{ht}						l_{taper}			
t/c_{wing}		t/c_{ht}		t/c_{vt}							
S_{wing}		S_{ht}		S_{vt}		l_{pylon}		$S_{wet\text{-}gross}$		l_{nac}	
mac_{wing}		mac_{ht}		mac		c		$S_{wet\text{-}net}$		d_{nac}	
$c_{wing\text{-}sob}$		$c_{ht\text{-}sob}$									
$t_{wing\text{-}sob}$		$t_{ht\text{-}sob}$		$t_{vt\text{-}sob}$							
$A_{wing\text{-}sob}$		$A_{ht\text{-}sob}$		$A_{vt\text{-}sob}$							
$S_{wing\text{-}wet}$		$S_{ht\text{-}wet}$		$S_{vt\text{-}wet}$							

Non-dimensional geometry (except fuselage)

Dimensions for input to drag buildup

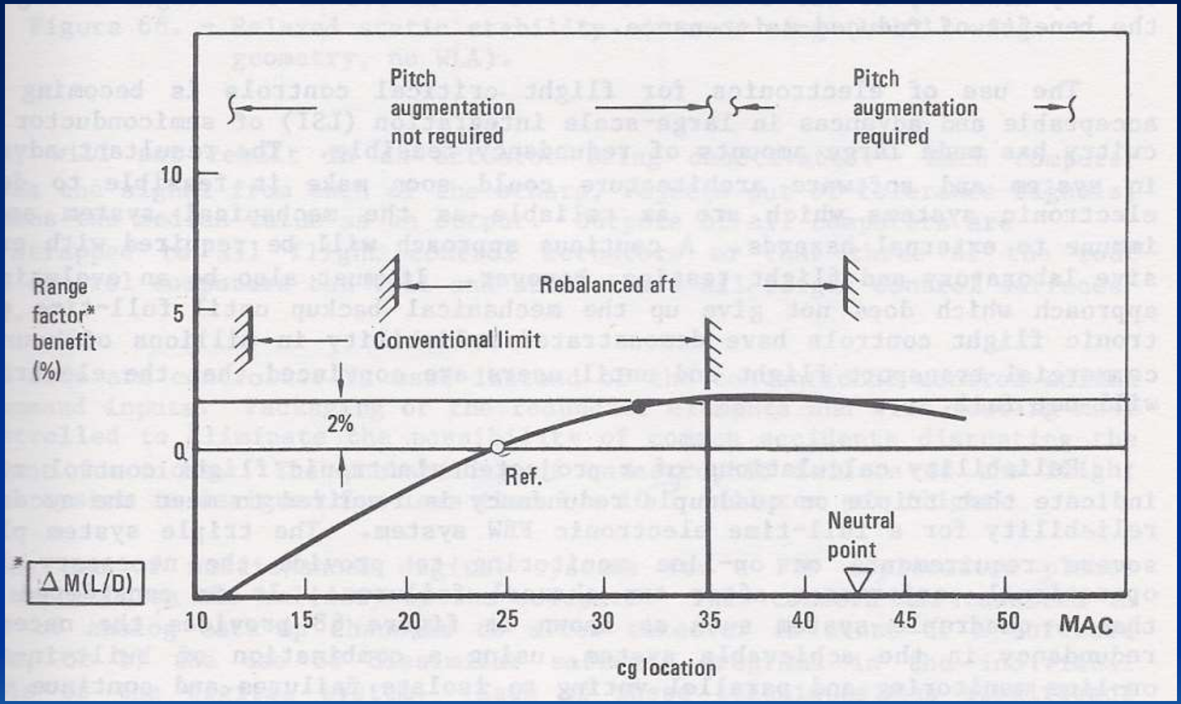
Zero-Lift Drag Module

Component	S_{wet}	S_{xs}	l_{ref}	R	C_f	FF	Q	$\frac{D/q}{S_{xs}}$	D/q	ΔC_{D_0}
Wing										
Horiz. Tail										
Vert Tail										
Pylons										
Fuselage										
Nacelles										
Landing gear										
Flaps+slats										
Total										$\Sigma\Delta C_{D_0}$

S_{wet} = wetted area S_{xs} = cross-section area l_{ref} = reference length R = Reynolds number
C_f = skin friction coeff Q = interference factor FF = form factor D/q = equivalent flat plate area
$\Delta C_{D_0} = (S_{wet}\ C_f\ Q\ FF)/S_{ref}$ or $\Delta C_{D_0} = D/q\ S_{ref}$

Trim Drag

- Often ignored in conceptual design
- Strong function of c.g. location
- Consists of
  - Drag of deflected elevator
  - Additional C_{D_i} due to additional wing lift



Effect of Relaxed Static Stability on L1011 Range Factor (NASA CR-3586)

High Lift Systems

Zero-Lift Drag C_{D_0}

Drag due to Lift C_{D_i}

Wave Drag due to Volume $C_{D_{0_{supersonic}}}$

Wave Drag due to Lift C_{D_W}

Drag due to Lift

Drag due to lift =
Incompressible drag due to lift
\+ Wave drag due to lift

K includes both subsonic and supersonic drag due to lift and is a function of Mach number

$$C_{D_{lift}} = C_{D_i} + \left(C_{D_w}\right)_{lift}$$

$$= \frac{1}{\pi A e} C_L^2 + \left(C_{D_w}\right)_{lift}$$

$$= K C_L^2$$

where K = Drag due to lift factor

Estimating Oswald Efficiency Factor, e

Estimate based on aspect ratio, A, and leading edge sweep, Λ_{le}

For straight wing aircraft:

$$e = 1.78\left(1 - 0.045A^{0.68}\right) - 0.64$$

For swept wing aircraft for which $\Lambda_{le} > 30$ deg:

$$e = 4.61\left(1 - 0.045A^{0.68}\right)\left(\cos\Lambda_{le}\right)^{0.15} - 3.1$$

For $0 < \Lambda_{le} < 30$ deg, use linear interpolation between values of both equations

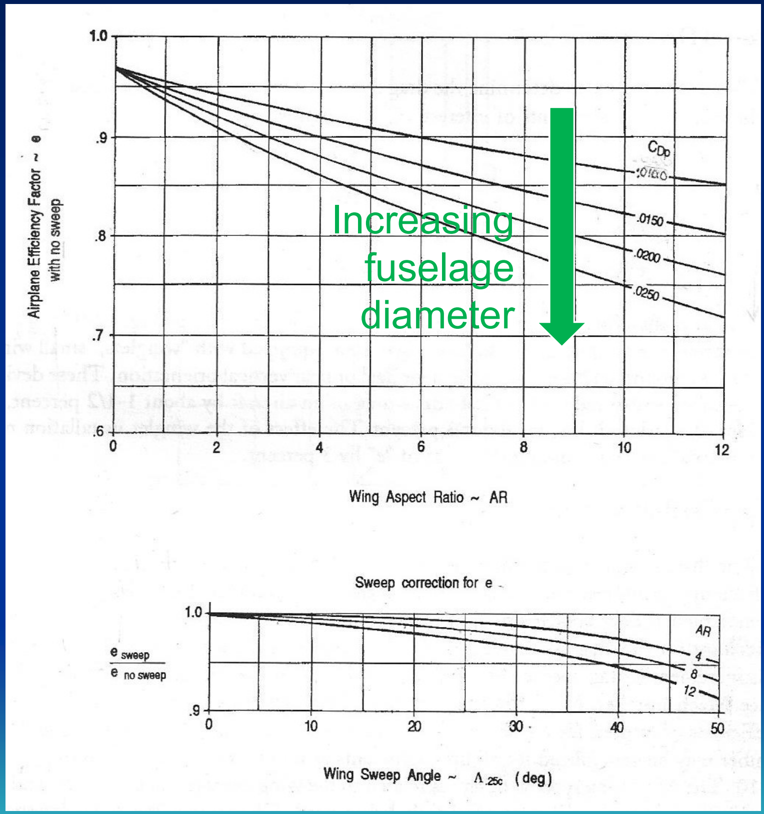
For high aspect ratio wings, use Shevell method (discussed later)

Cessna 172

Avro Vulcan

Oswald Efficiency Factor for Airliners

- Uses C_{DP} as a surrogate for d_{fuse}/b
- As d_{fuse}/b increases, spanwise lift distribution is less elliptical



Source: Schaufele

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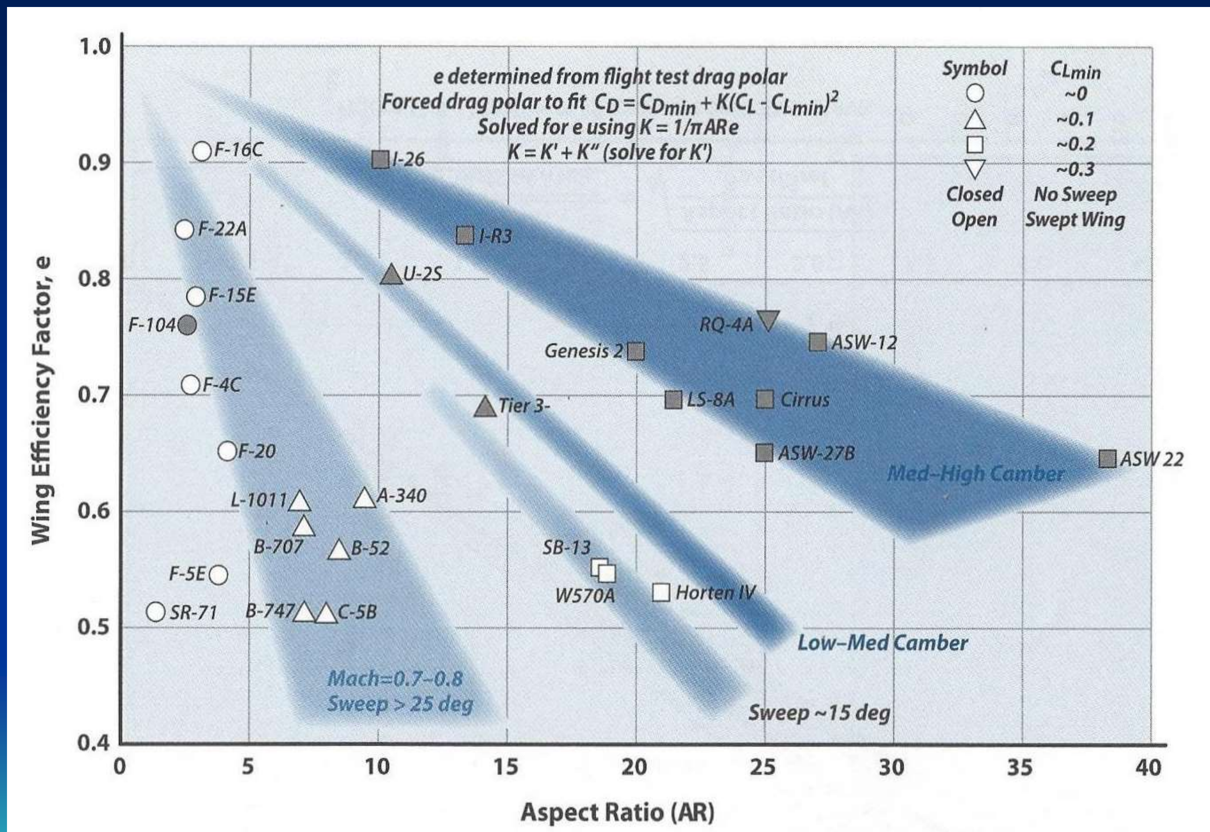
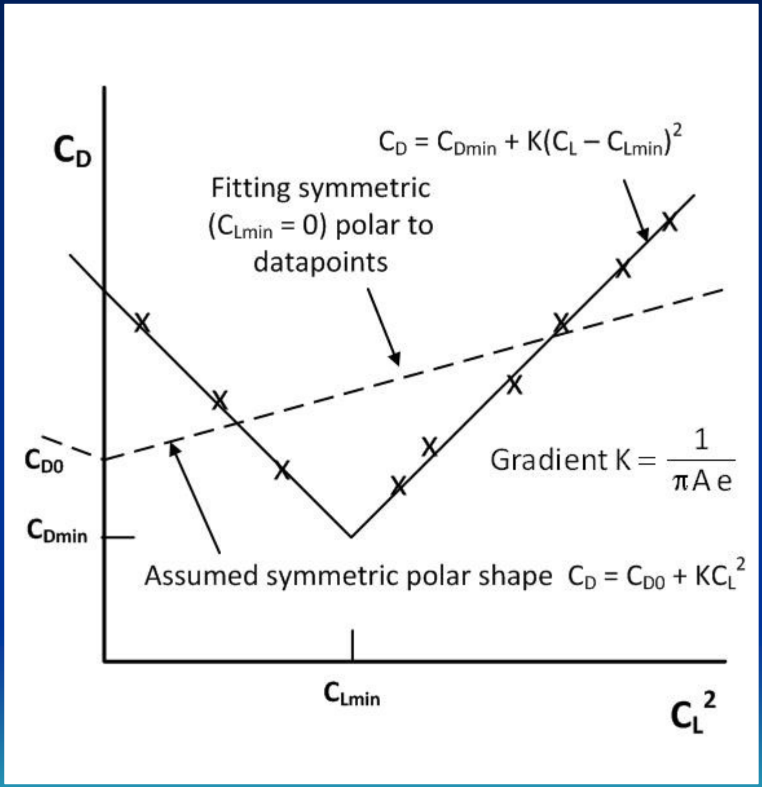
Estimation of Oswald Efficiency Factor



Source: Nicolai/Carichner

Caveat for Oswald Efficiency Factor Chart

- In Raymer's analysis, all polars are assumed symmetric ($C_D = C_{D_o} + K\,C_L^2$)
- Values of e using Raymer analysis are only valid for $C_{L_{min}} = 0$ (white circles on previous chart)



High Lift Systems

Zero-Lift Drag C_{D_0}

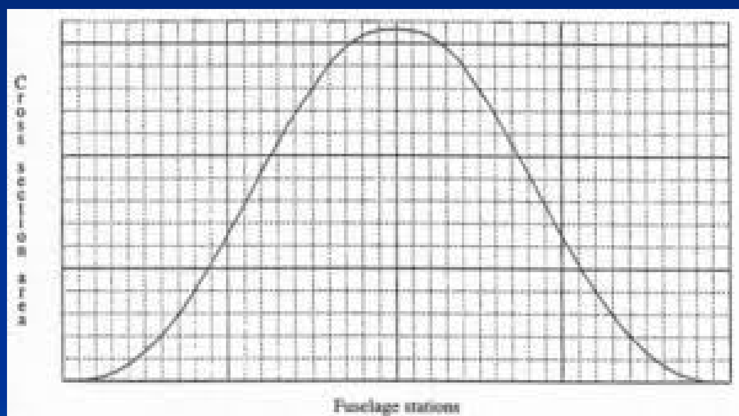
Drag due to Lift C_{D_i}

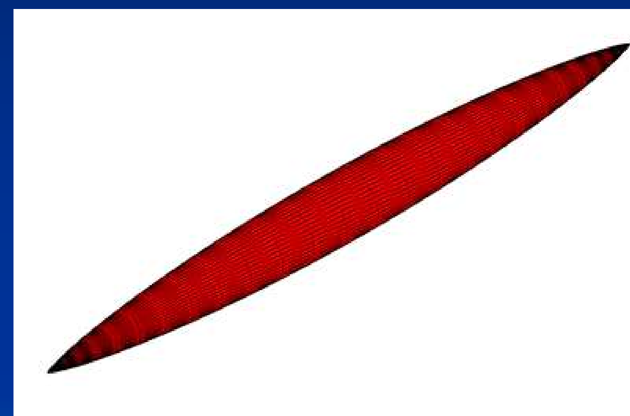
Wave Drag due to Volume $C_{D_{0_{supersonic}}}$

Wave Drag due to Lift C_{D_W}

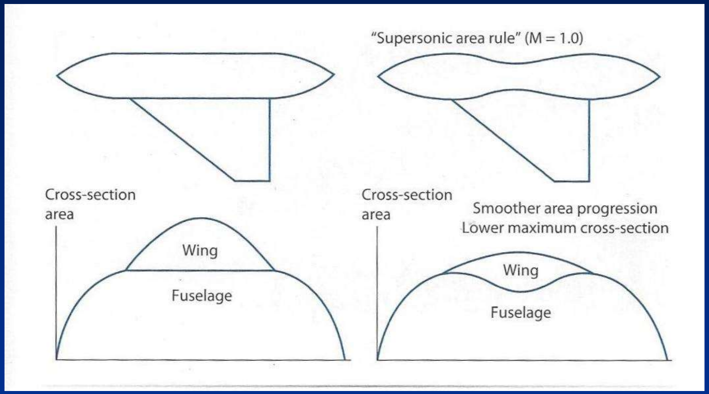
Sears-Haack Body

- Minimum transonic wave drag for given volume

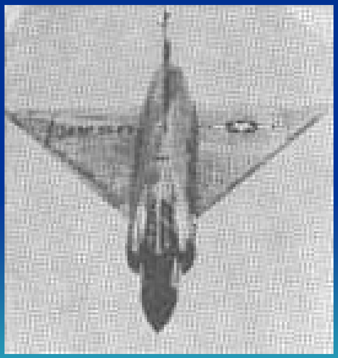




Area Ruling



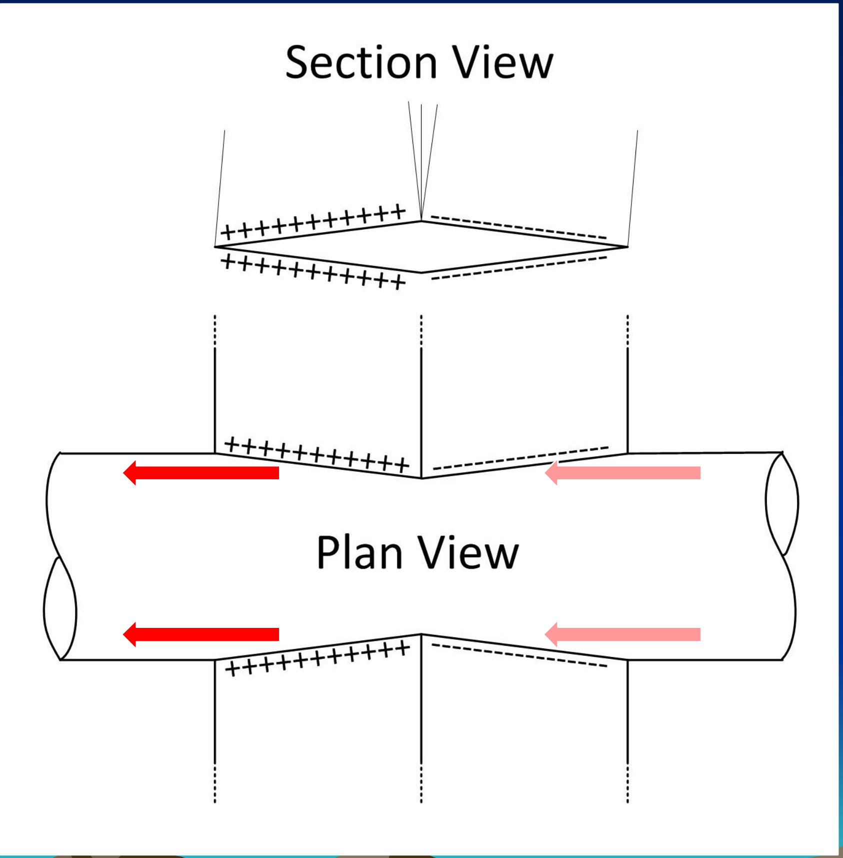
"Supersonic area rule" (M = 1.0)
Cross-section area
Wing
Fuselage
Cross-section area
Smoother area progression
Lower maximum cross-section
Wing
Fuselage



Transonic Area Ruling Simplified

Positive pressure on forward-facing wing surface increases drag

Positive pressure on aft-facing area of fuselage reduces drag



Negative pressure on aft-facing wing surface increases drag

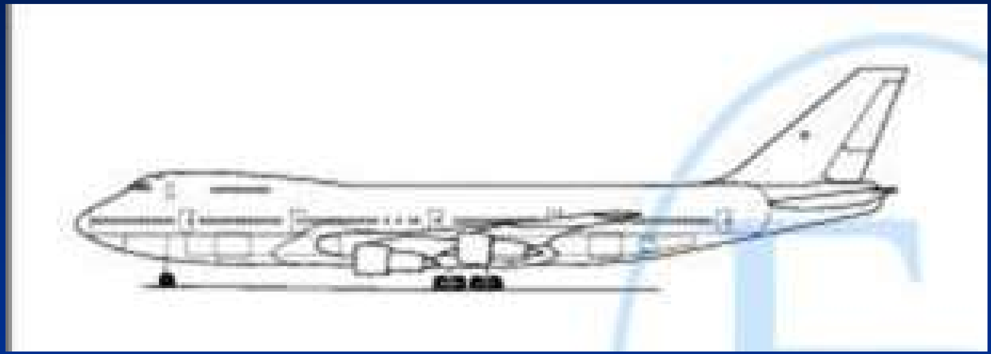
Negative pressure on forward-facing area of fuselage reduces drag

Boeing Transonic Airliner

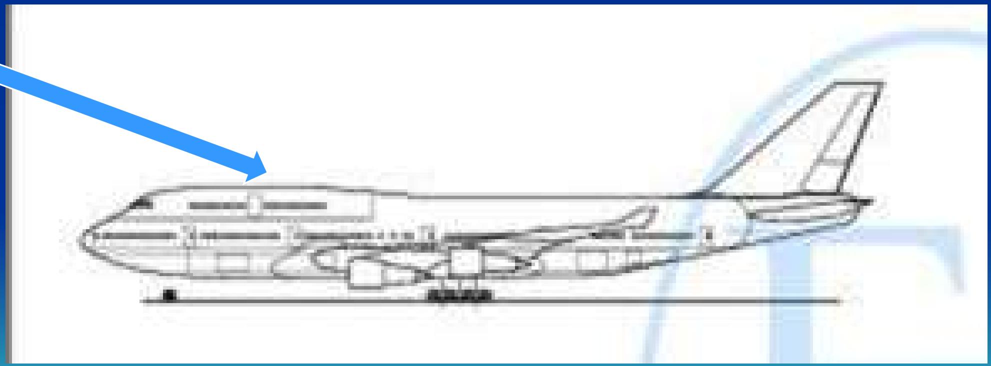
- Difficult and expensive to manufacture
- Inefficient seating
- Small reduction in flight time
- Small gain in aircraft and crew utilization
- Small gain in M L/D

A380 Underwing Fairing

Area Ruling 747-200 vs -400



OML of extended upper cabin smoothed out area distribution and reduced zero-lift transonic drag



High Lift Systems

Zero-Lift Drag C_{D_0}

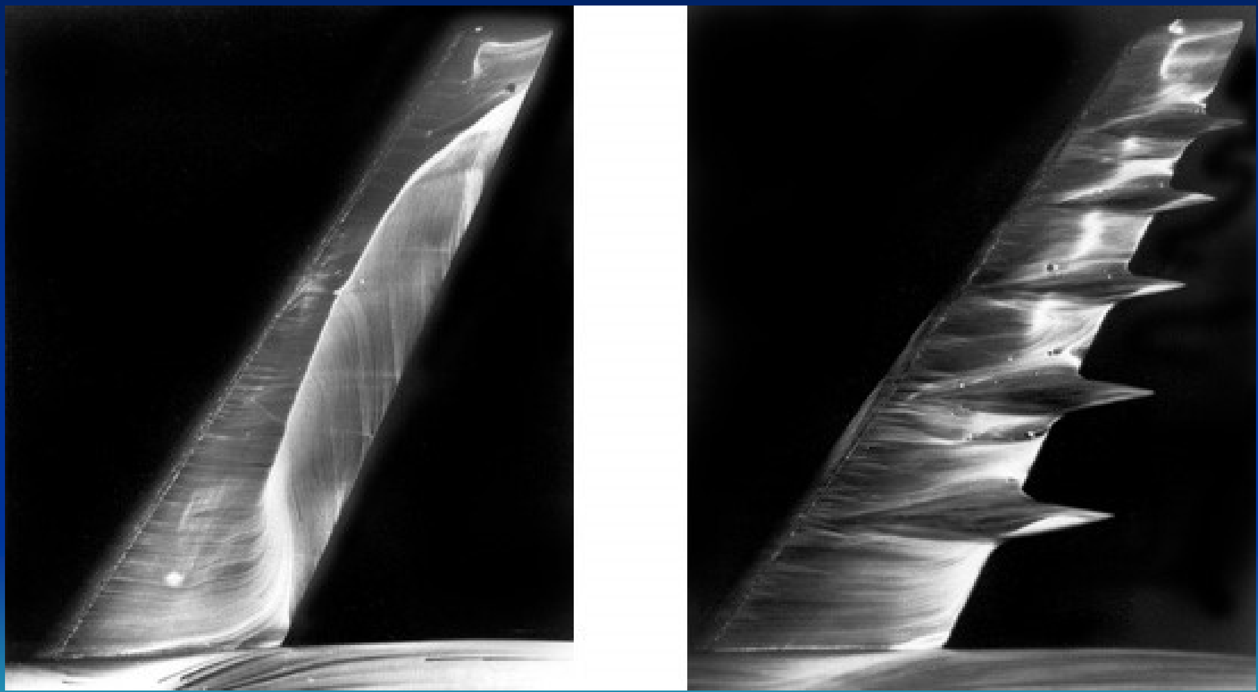
Drag due to Lift C_{D_i}

Wave Drag due to Volume $C_{D_{0_{supersonic}}}$

Wave Drag due to Lift C_{D_W}

Anti-shock Bodies Eliminate Wing Shock

- Also called Whitcomb fairings or Küchemann carrots
- Led to development of supercritical airfoil sections

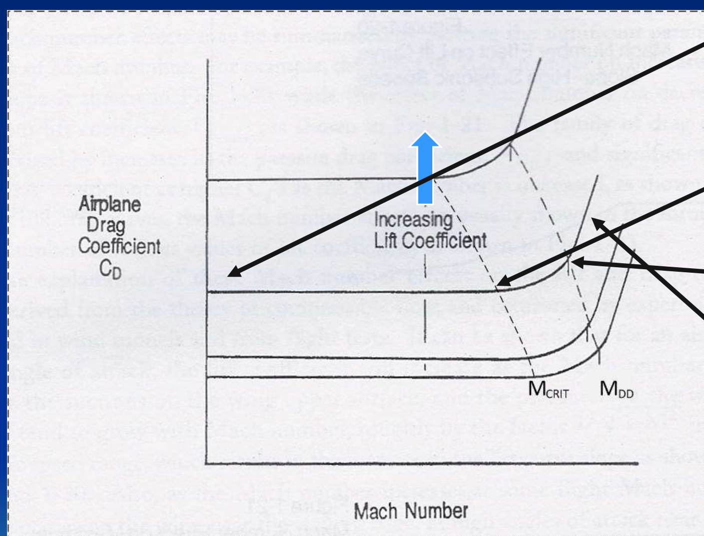


Küchemann Carrots on Convair 990

- Competed with B707 and DC-8
- First flight: January 1961
- Production run: 37

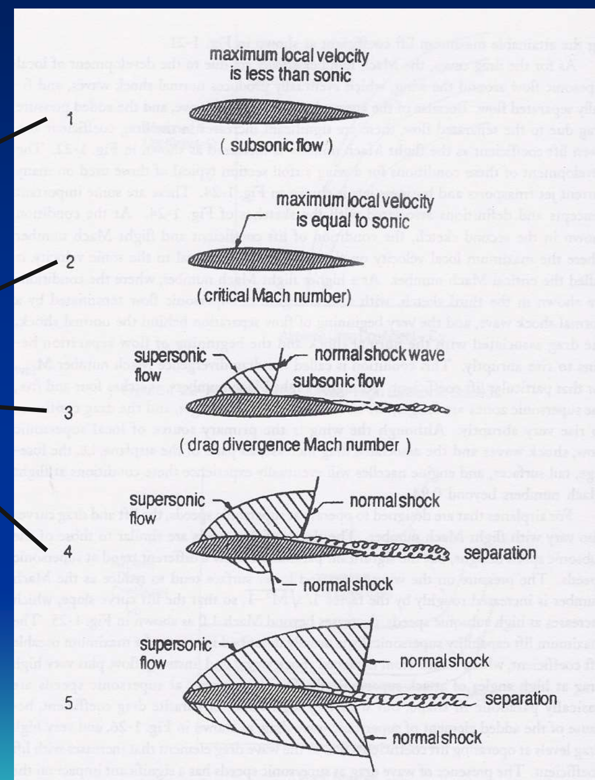
Flow Over Wing At Increasing Mach Number

M_{CRIT} and M_{DD} are a function of C_L (shown here), Λ and t/c



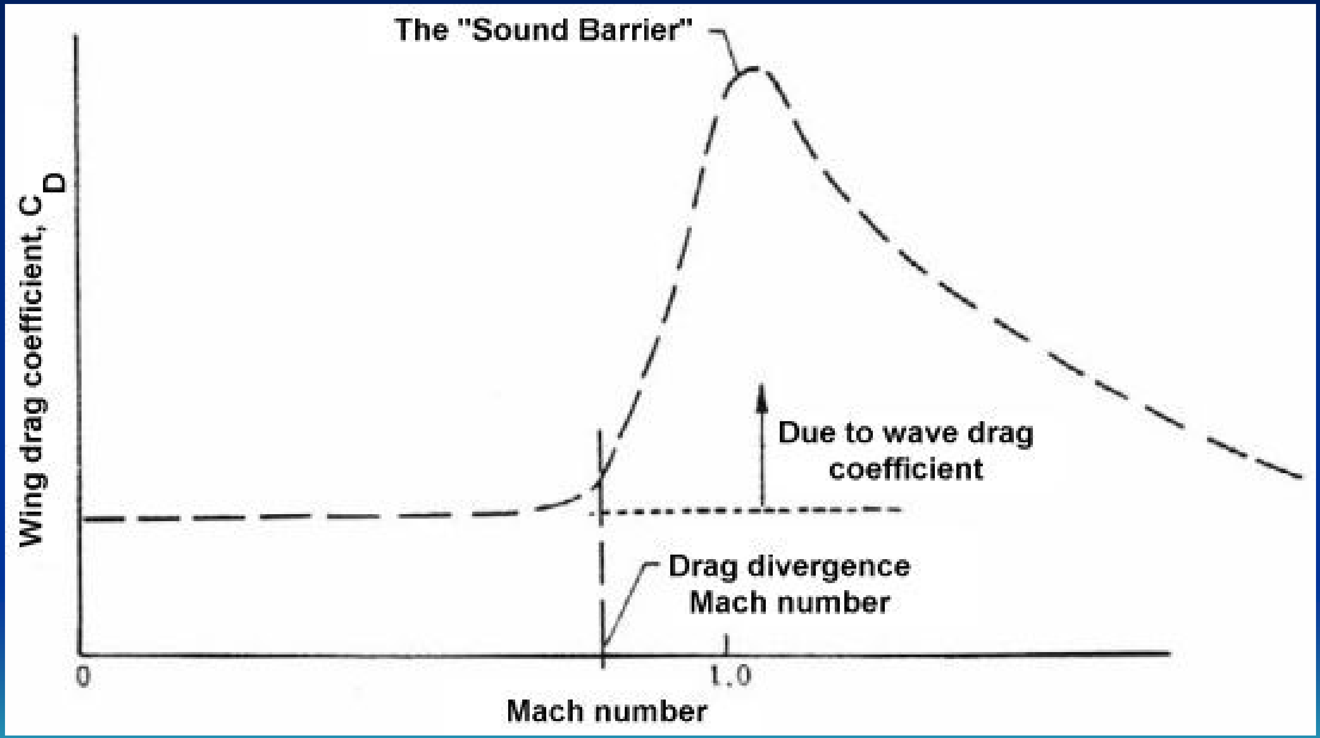
Source: Schaufele (modified)

Note: this is not a supercritical airfoil section



Source: Schaufele

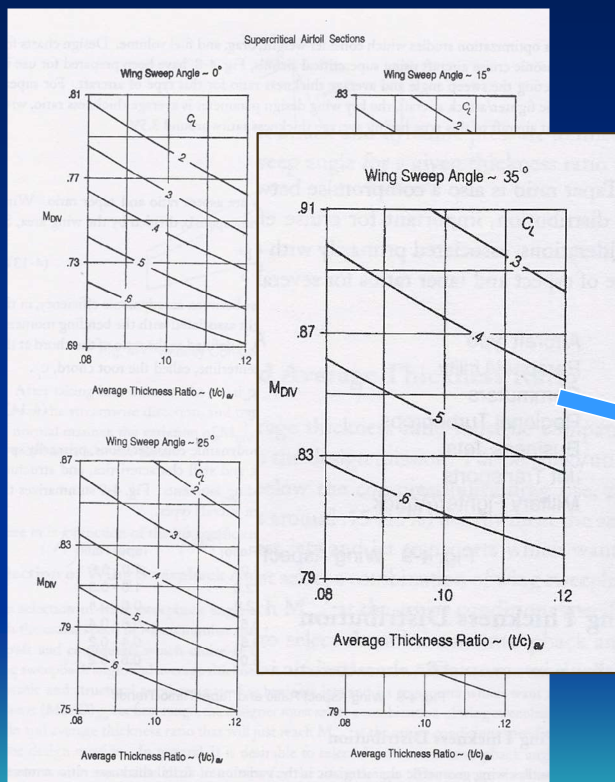
C_D vs Mach No. at Fixed C_L

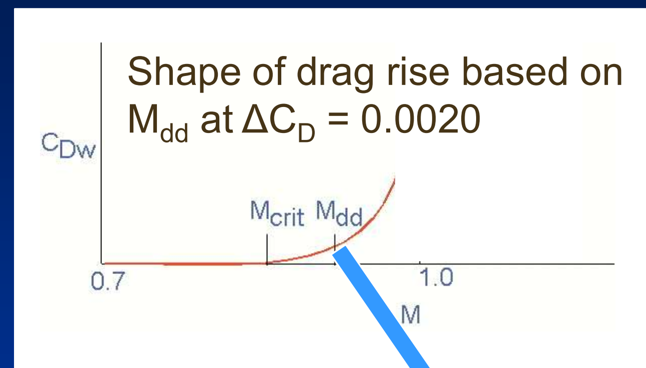


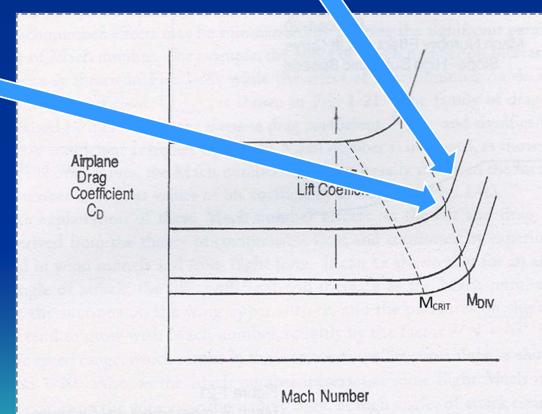
Generation of Drag Map

Mach number at which drag rise occurs (based on $\Delta C_D = 0.0014$)

Assume
$M_{DD} = M_{DIV} + 0.02$
where
M_{DD} is defined at $\Delta C_D = 0.0020$



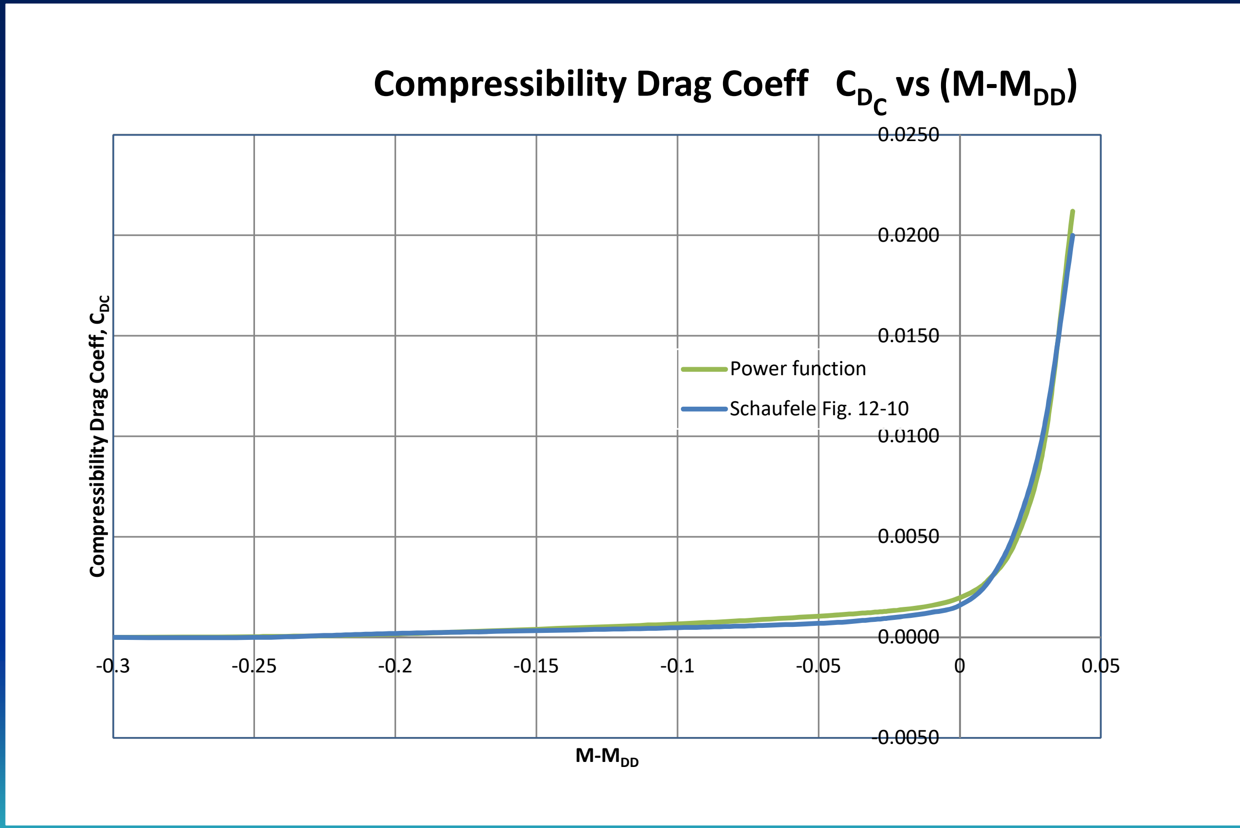




Empirical Estimate of Drag Rise

- Power function
  - Meets Boeing definition of MDD when $\Delta C_{D_C} = 0.0020$

$(M_{DD})_{Douglas} - (M_{DD})_{Boeing} = 0.1$



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Alternative Method of M_{DD} Estimation

Empirical Korn Equation applied to airfoil section

$$M_{DD} = \frac{k_a}{\cos\left(\Lambda_{\frac{c}{2}}\right)} - \frac{\frac{t}{c}}{\cos^2\left(\Lambda_{\frac{c}{2}}\right)} - \frac{C_l}{10\cos^3\left(\Lambda_{\frac{c}{2}}\right)} - 0.01$$

where

k_a = technology factor

$(= 0.87$ for NACA 6-series$)$

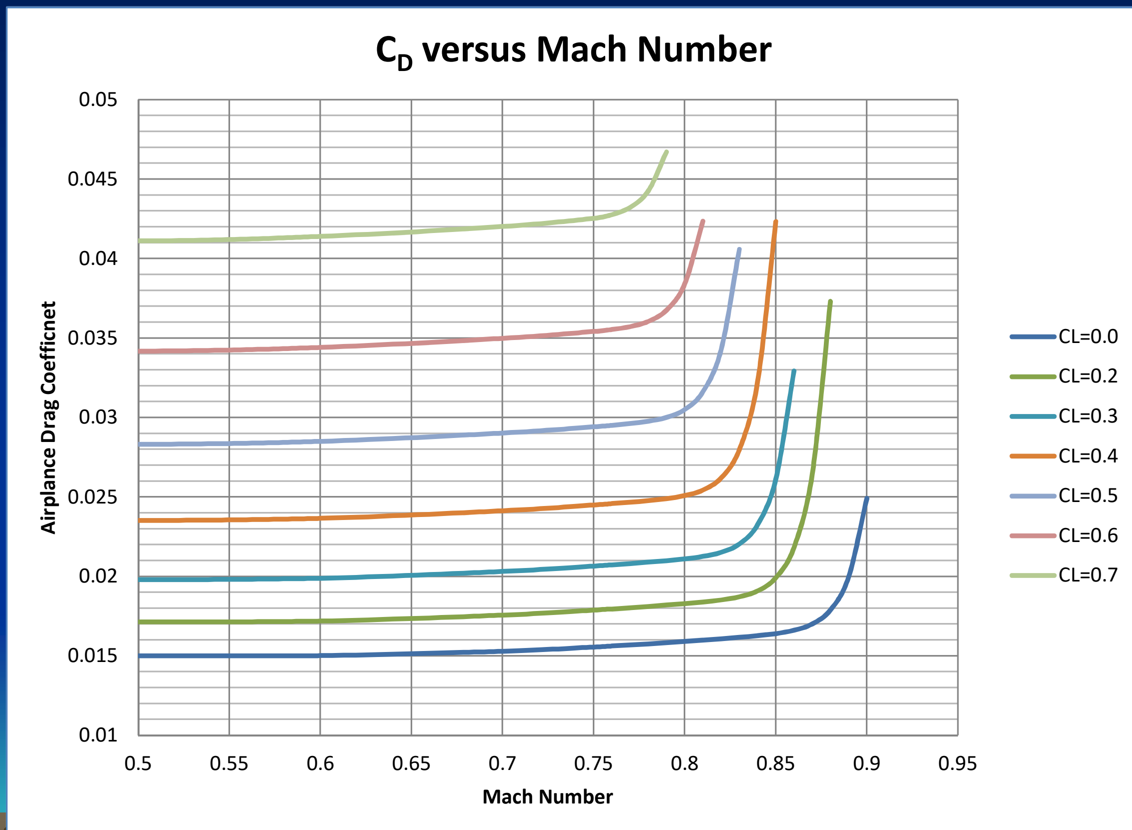
$(= 0.95$ for supercritical airfoil$)$

For wing, divide into sections and average results

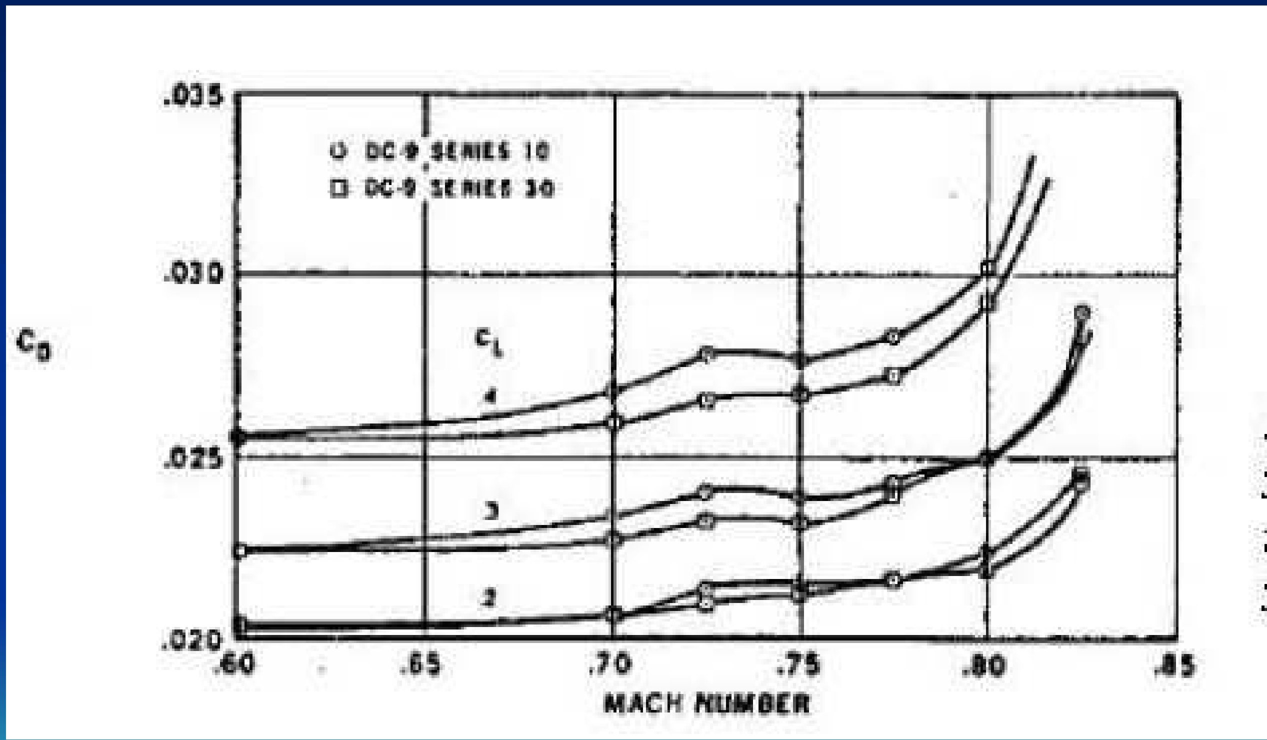
Modified from Douglas definition of $dC_D/dM = 0.10$ to Boeing definition of $\Delta C_D = 0.0020$ for this drag rise curve

For this approximation, use average values for whole wing

Empirical Estimate of DC-10 Drag Map

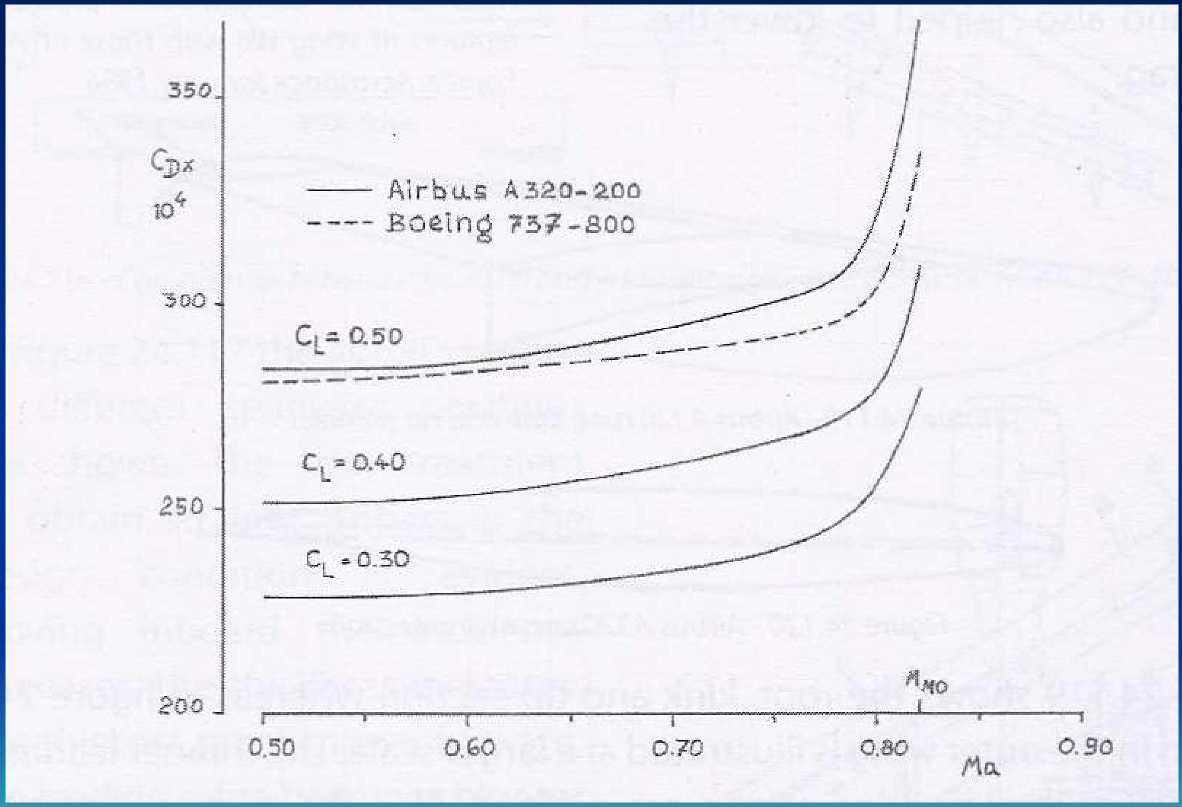


DC-9 Drag Plot

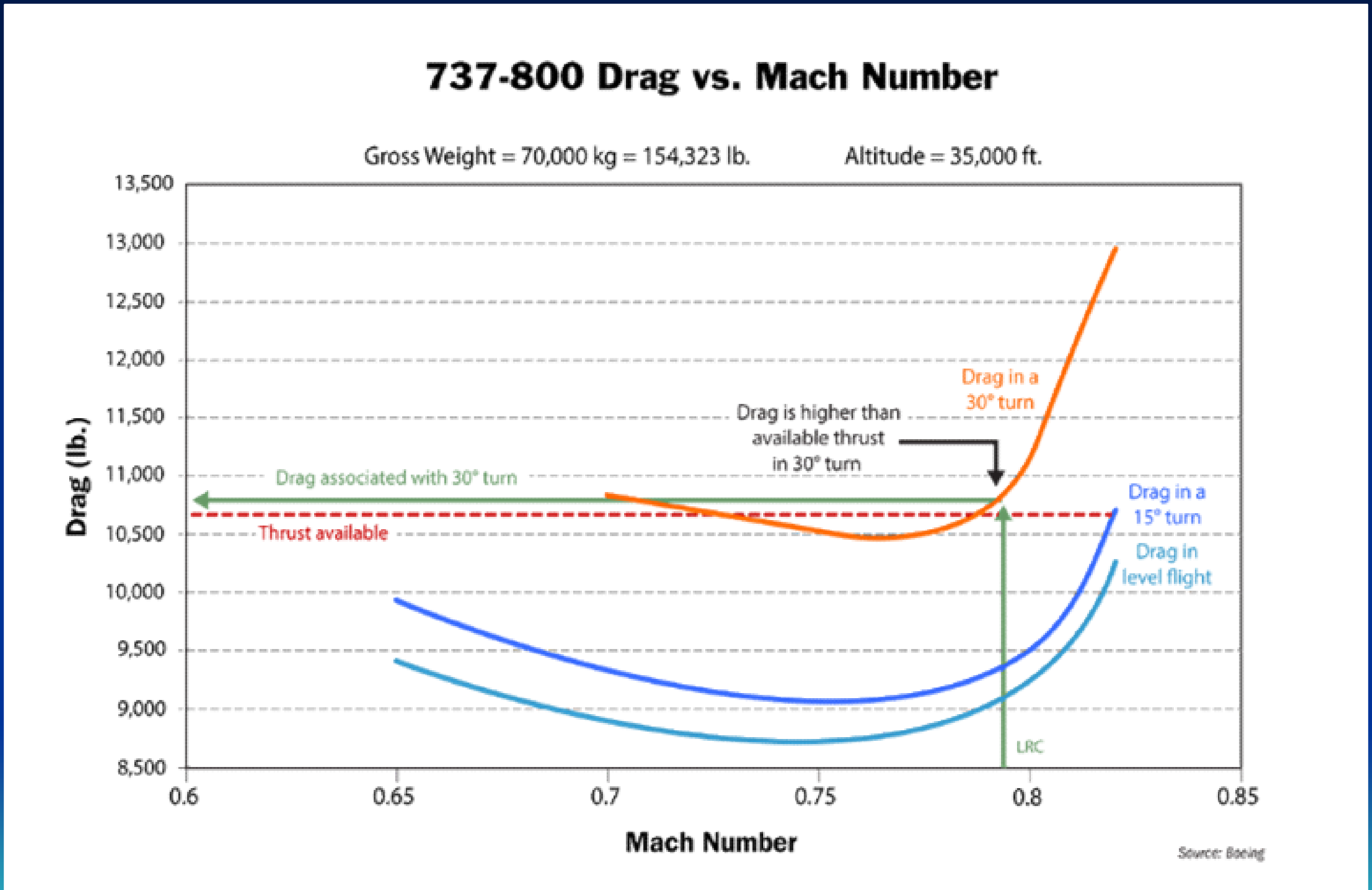


Source: Obert

Comparative Drag Plots



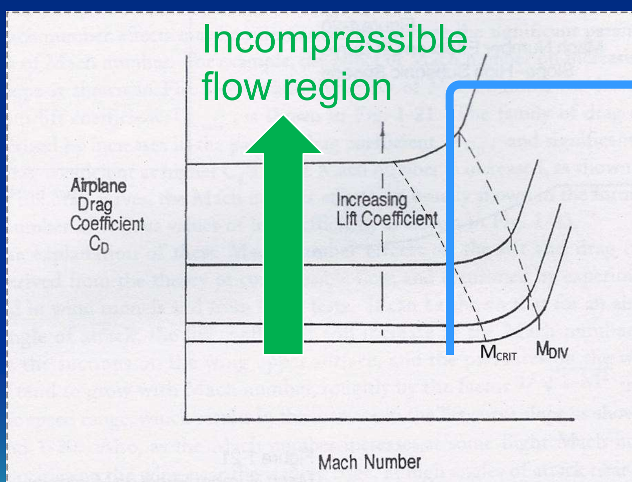
Source: Obert

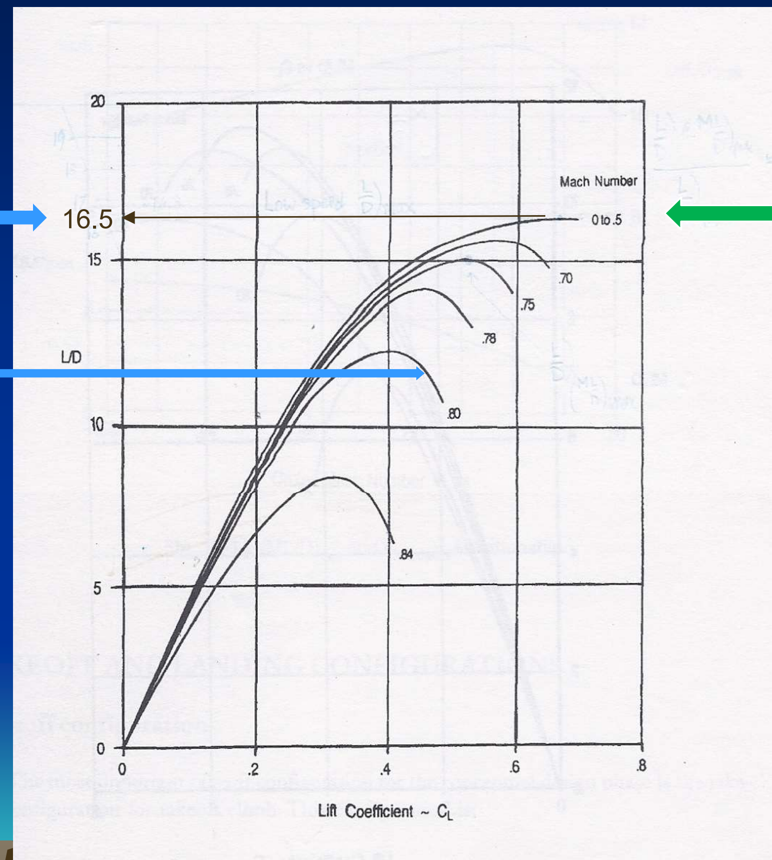

737-800 Drag vs. Mach Number
Gross Weight = 70,000 kg = 154,323 lb.
Altitude = 35,000 ft.
Drag (lb.)
13,500
13,000
12,500
12,000
11,500
11,000
10,500
10,000
9,500
9,000
8,500
0.6
0.65
0.7
0.75
0.8
0.85
Mach Number
Drag in a 30° turn
Drag is higher than available thrust in 30° turn
Drag associated with 30° turn
Thrust available
Drag in a 15° turn
Drag in level flight
LRC
Source: Boeing

DC-9 Lift/Drag Ratio vs C_L

- Max low speed L/D =16.5

Take vertical slice through drag map

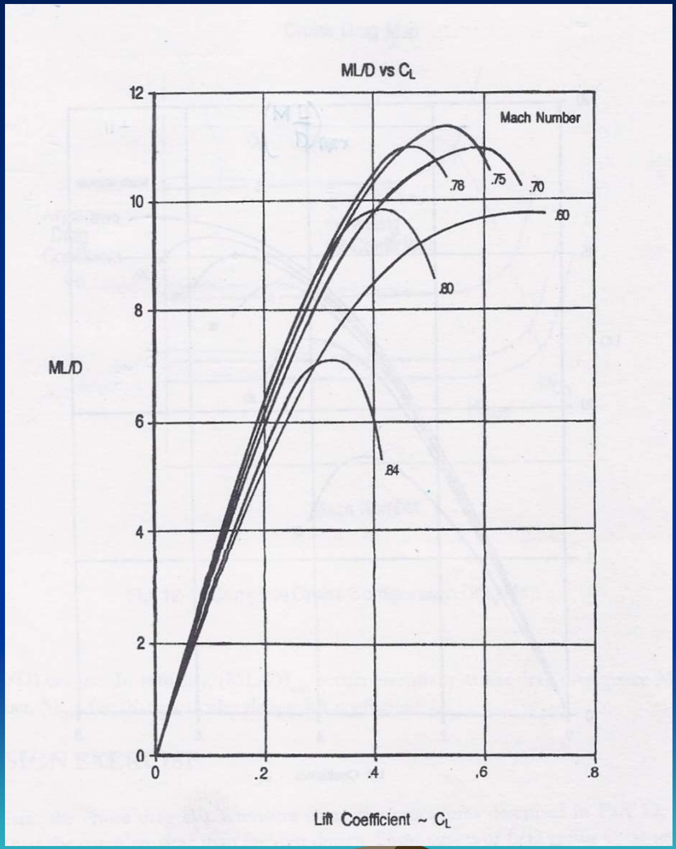




Incompressible flow region

DC-9 ML/D vs C_L

- DC-9 airfoil is not supercritical
- $(M\ L/D)_{max}$ occurs at about M = 0.75
- $(M\ L/D)_{max}$ = 11.5



Source: Schaufele

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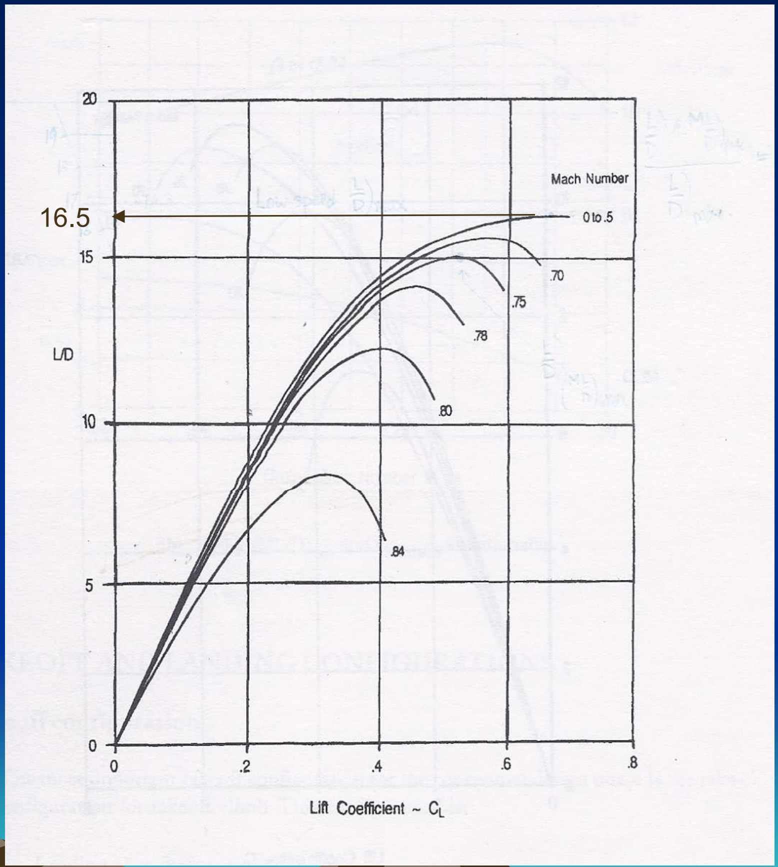
DC-9 L/D at $(M\ L/D)_{max}$

$(L/D)/(L/D)_{max} = 15/16.5$

$= 0.91$

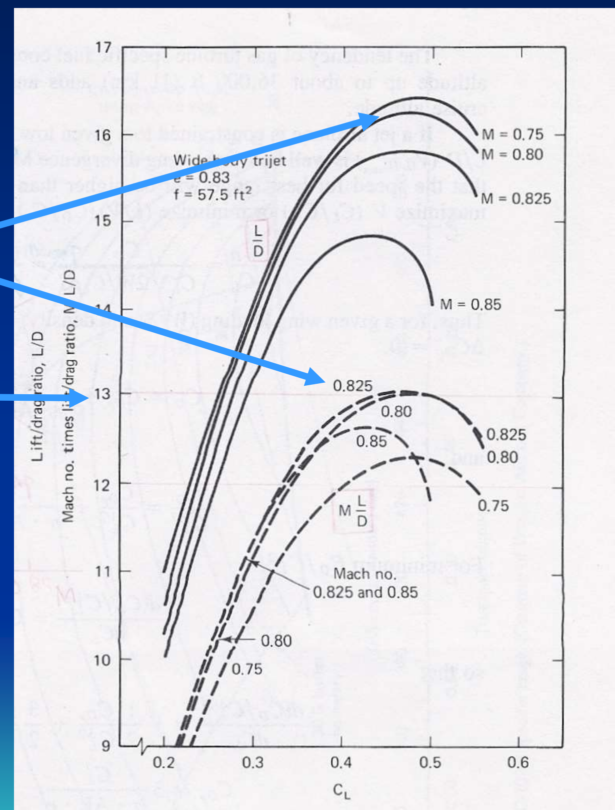


Source: Schaufele

DC-10 L/D and (M L/D)

- $M_{(ML/D)max} = 0.825$
- $((L/D)_{max})_{M=0.8} = 16.5$
- $(M\ L/D)_{max} = 13$
- $(L/D)/(L/D)_{max} = 0.96$

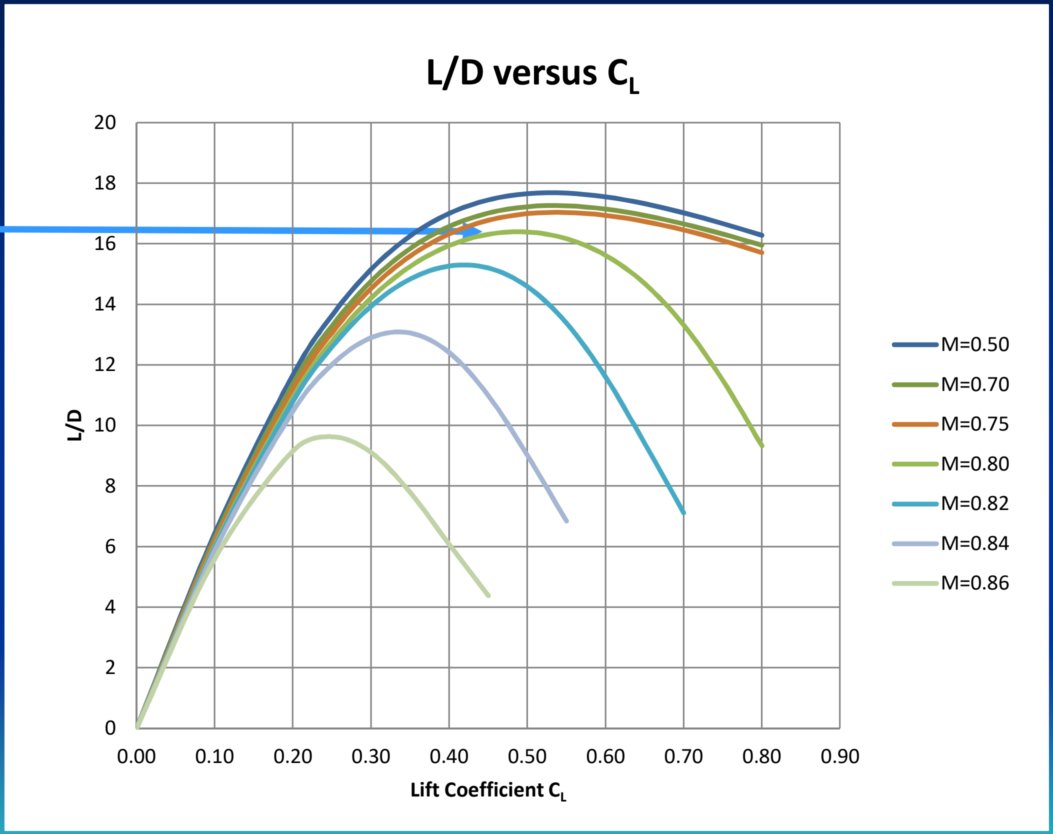
16.3



Source: Shevell

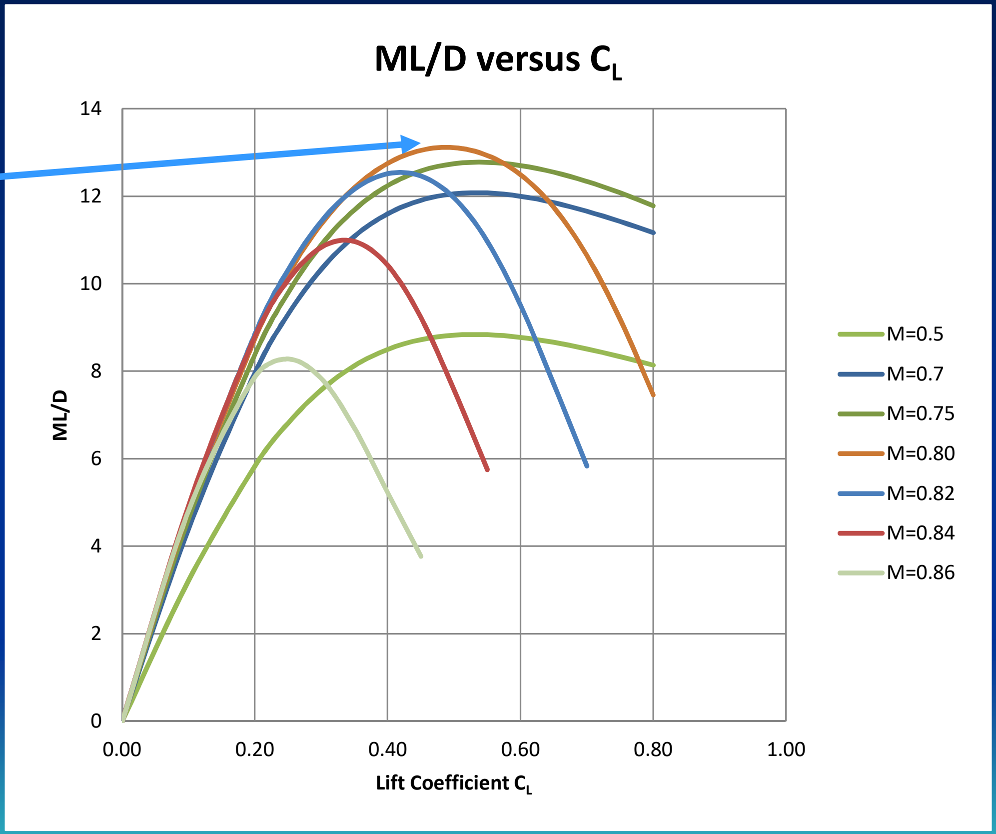
Spreadsheet Prediction for DC-10

$((L/D)_{max})_{M=0.8} = 16.4$



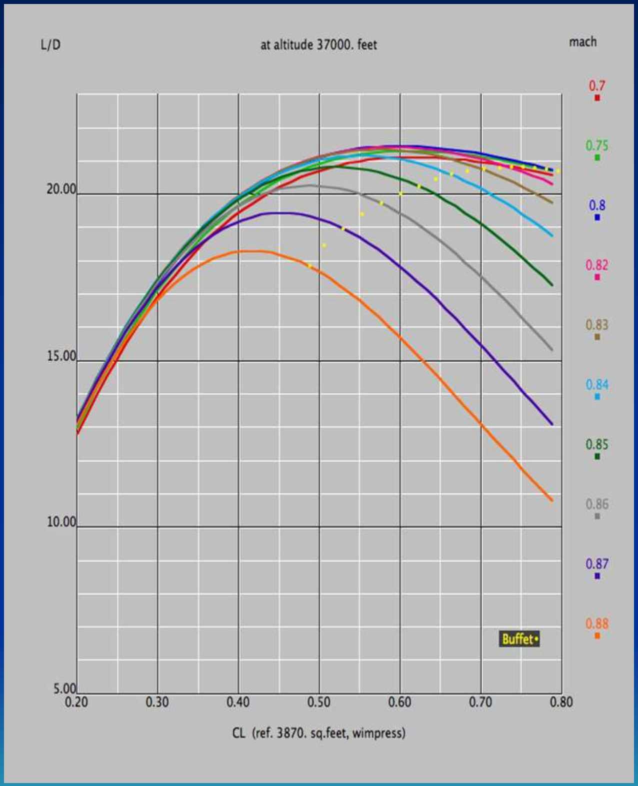
Spreadsheet Prediction for DC-10

- $M_{(ML/D)max} = 0.80$
- $(M\ L/D)_{max} = 13$



Piano Prediction for 787

Piano is European industrial-grade sizing and performance program



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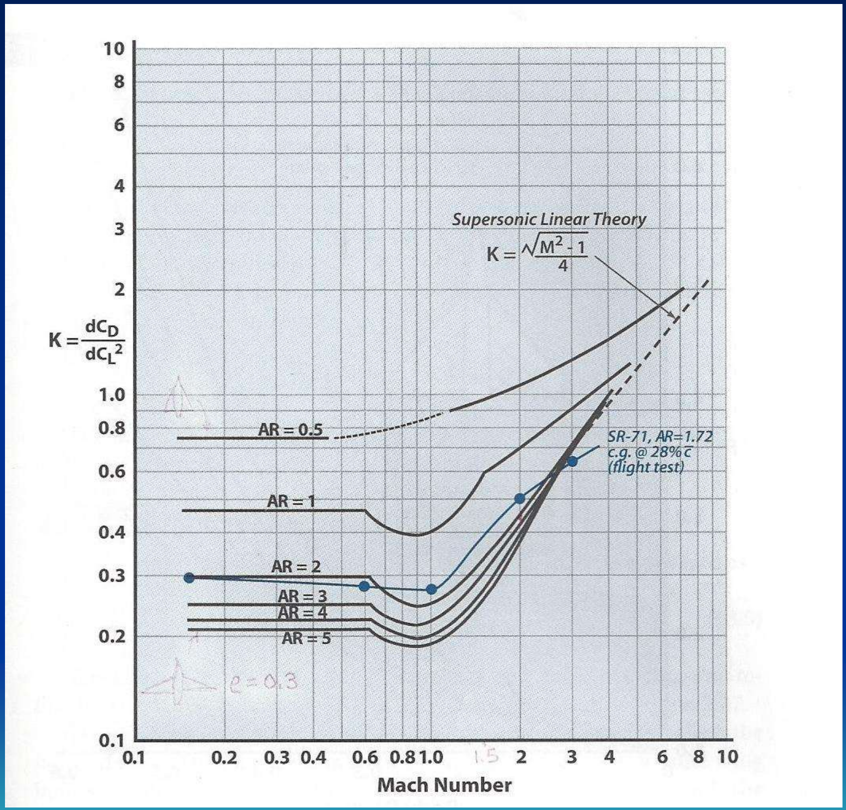
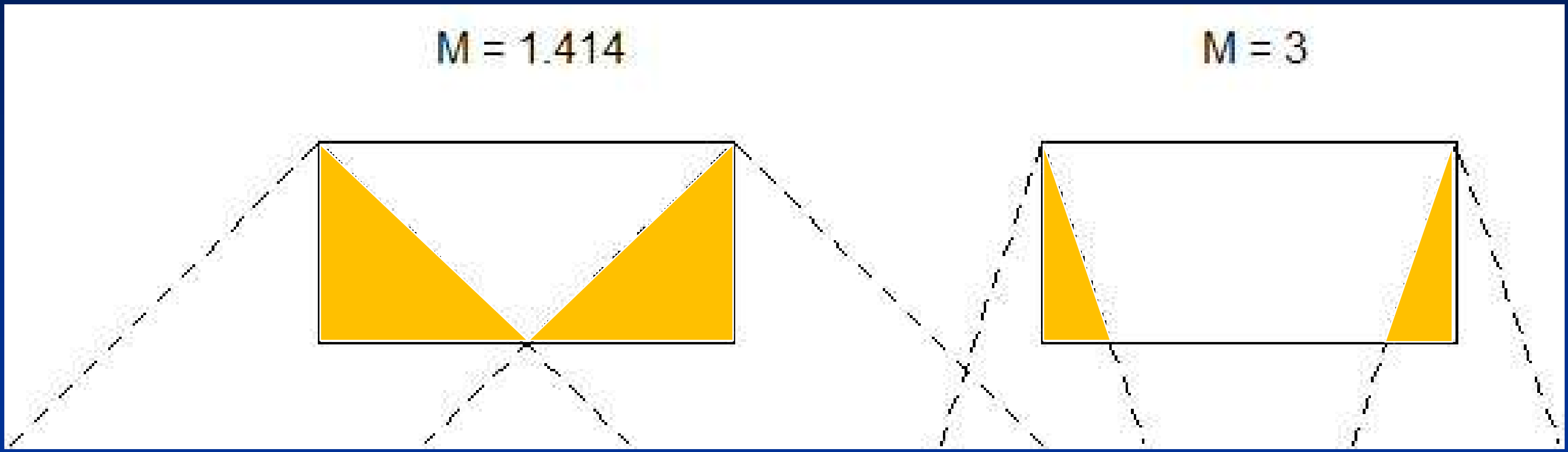
Estimation of K for Delta Wing Config.

- Chart based on wing with l.e. radius = 0.045%
- Curves for different AR are asymptotic to linear theory



Source: Shevell

Cones of Influence for AR=2 Wing



- As M increases, area of wing influenced by wingtips decreases and linear theory dominates

Aerodynamics
The End